Table 4.3a Class II Weight Estimates for the Eris

Component	Methods: Class II Page 14	GD	Torenbeek	Use as Class II Estimate
Structure weight, W_{struct}:				
Wing	6,762	9,490		8,126
Adjustment for Fowler flaps, 2 percent:				163
Horiz.Tail		720		707
Vert.Tail		938		921
Empennage	1,597	1,658		1,628
Fuselage	7,347 incl.booms	5,044		5,967*
Booms		458		458
Engine Section	160			160
Nose Gear	554		267	443
Main Gear	2,214		1,603	1,768
Landing gear	2,768	1,996	1,870	2,211
W_{struct}	18,634	20,304		18,713
Powerplant weight, W_{pwr}:				
Engines	4,000	4,000	4,000	4,000
Afterburners	2,000	2,000	2,000	2,000
Air ind. syst.	in propuls.	445		445
Fuel system	in propuls.	777		777
Propulsion inst.	2,834**	78		845***
W_{pwr}	8,834	7,632		8,067

*1/2(7,347 - 458 + 5,044) = 5967

**includes air induction and fuel system

***1/2(2,834 + 78 - 445 - 777) = 845

Table 4.3b Weight Fraction Estimates for the Eris:

Average Weight Fractions for Fixed Equipment Breakdown

Note: these data were used in Table 4.3c.

Component	Similar Airplane Type: Republic F105B	Chance Vought F8U	Grumman A2F(A6)	Use as Class II Estimate
Fixed equipment weight item:				
W_{fc}*	0.0561	0.0515	0.0317	0.0464
W_{els}	0.0223	0.0144	0.0200	0.0189
W_{iae}	0.0307	0.0337	0.0800	0.0481
W_{api}	0.0054	0.0108	0.0047	0.0070
W_{arm}	0.0229	0.0123	0.0093	0.0148
W_{fur}	0.0077	0.0069	0.0137	0.0094
W_{aux}	0.0029	0.0060		0.0045

* includes hydraulic and pneumatic system

Note: Specific airplane type data from Tables A9.2a, A9.3a and A9.4a in Appendix A.

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Table 4.3c Class II Weight Estimates for the Eris

Component	Methods: Table 4.3b x61,660	GD	Torenbeek	Use as Class II Estimate
Fixed equipment weight, W_{feq}:				
W_{fc}	3,459	1,513		2,486
$W_{fc_{cg}}$		102		102
W_{hps}: this is included in W_{fc}				
W_{els}	1,165	703		934
W_{iae}	1,893		1,033	1,463
W_{api}	431	347		389
W_{arm}	913			913
W_{fur}	580	214		397
W_{ox}	in W_{fur}	17		in W_{fur}
W_{aux}	277			277
GAU-8A Gun	2,014	Part II, Table 10.6:		2,014
W_{feq}	10,732*			8,975

* This disagrees significantly with W_{feq} in Table 10.6 of Part II.

<u>Summary:</u>

Class II empty weight, W_E follows from Eqn.(2.1):

$$W_E = 18,713 + 8,067 + 8,975 = 35,755 \text{ lbs}$$

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5. CLASS II METHOD FOR ESTIMATING STRUCTURE WEIGHT

The airplane structure weight, W_{struct} will be assumed to consist of the following components:

5.1 Wing, W_w 5.2 Empennage, W_{emp}

5.3 Fuselage, W_f 5.4 Nacelles, W_n

5.5 Landing gear, W_g Therefore:

$$W_{struct} = W_w + W_{emp} + W_f + W_n + W_g \quad (5.1)$$

Equations for structure weight estimation are presented for the following types of airplanes:

1. General Aviation Airplanes
2. Commercial Transport Airplanes
3. Military Patrol, Bomb and Transport Airplanes
4. Fighter and Attack Airplanes

5.1 WING WEIGHT ESTIMATION

5.1.1 General Aviation Airplanes

5.1.1.1 Cessna method

The following equations should be applied only to small, relatively low performance type airplanes with maximum speeds below 200 kts. The equations apply to wings of two types:

Cantilever wings: Eqn.(5.2)
Strut braced wings: Eqn.(5.3)

Both equations <u>include:</u> weight of wing tip fairing
wing control surfaces

Both equations <u>exclude:</u> fuel tanks
wing/fuselage spar carry-through structure
effect of sweep angle

<u>For cantilever wings:</u>

$$W_w = 0.04674(W_{TO})^{0.397}(S)^{0.360}(n_{ult})^{0.397}(A)^{1.712} \quad (5.2)$$

<u>For strut braced wings:</u>

$$W_w = 0.002933(S)^{1.018}(A)^{2.473}(n_{ult})^{0.611} \quad (5.3)$$

Definition of terms:

W_{TO} = take-off weight in lbs.

S = wing area in ft^2.

n_{ult} = design ultimate load factor

A = wing aspect ratio

Note that Eqn.(5.3) does not account for W_{TO}. It should therefore be used with caution. The reader should also realize that wings in this category have maximum thickness ratios of around 18 percent.

5.1.1.2 USAF Method

The following equation applies to light and utility type airplanes with performance up to about 300 kts:

$$W_w = 96.948[(W_{TO}n_{ult}/10^5)^{0.65}(A/\cos\Lambda_{1/4})^{0.57}(S/100)^{0.61} x$$
$$x\{(1+\lambda)/2(t/c)_m\}^{0.36}(1 + V_H/500)^{0.5}]^{0.993} \qquad (5.4)$$

Definition of new terms:

$\Lambda_{1/4}$ = wing quarter chord sweep angle

λ = wing taper ratio

$(t/c)_m$ = maximum wing thickness ratio

V_H = maximum level speed at sealevel in kts

5.1.1.3 Torenbeek Method

The following equation applies to light transport airplanes with take-off weights below 12,500 lbs:

$$W_w =$$
$$= 0.00125W_{TO}(b/\cos\Lambda_{1/2})^{0.75}[1 + \{6.3\cos(\Lambda_{1/2})/b\}^{1/2}]x$$
$$x(n_{ult})^{0.55}(bS/t_rW_{TO}\cos\Lambda_{1/2})^{0.30} \qquad (5.5)$$

See special notes in Section 5.2.2.

Definition of new terms:

b = wing span in ft

$\Lambda_{1/2}$ = wing semi-chord sweep angle

t_r = maximum thickness of wing root chord in ft

5.1.2 Commercial Transport Airplanes

5.1.2.1 GD Method

$$W_w = \frac{[0.00428(S^{0.48})(A)(M_H)^{0.43}(W_{TO}n_{ult})^{0.84}(\lambda)^{0.14}]}{[(100(t/c)_m)^{0.76}(\cos\Lambda_{1/2})^{1.54}]} \quad (5.6)$$

Note: This equation is valid only in the following parameter ranges:

M_H from 0.4 to 0.8, $(t/c)_m$ from 0.08 to 0.15, and A from 4 to 12.

Definition of new term:

M_H = maximum Mach number at sealevel

5.1.2.2 Torenbeek Method

The following equation applies to transport airplanes with take-off weights above 12,500 lbs:

$$W_w = 0.0017W_{MZF}(b/\cos\Lambda_{1/2})^{0.75}[1 + (6.3\cos(\Lambda_{1/2})/b)^{1/2}]x$$
$$x(n_{ult})^{0.55}(bS/t_rW_{MZF}\cos\Lambda_{1/2})^{0.30} \quad (5.7)$$

Definition of new term:

W_{MZF} = maximum zero fuel weight = $W_{TO} - W_F$ (5.8)

Special notes:

1. Eqns.(5.6) and (5.7) include the weight of normal high lift devices as well as ailerons.
2. For spoilers and speed brakes 2 percent should be added.

3. If the airplane has 2 wing mounted engines reduce the wing weight by 5 percent.
4. If the airplane has 4 wing mounted engines reduce the wing weight by 10 percent.
5. If the landing gear is not mounted under the wing reduce the wing weight by 5 percent.
6. For braced wings reduce the wing weight by 30 percent. The resulting wing weight estimate does include the weight of the strut. The latter is roughly 10 percent of the wing weight.
7. For Fowler flaps add 2 percent to wing weight.

5.1.3 Military Patrol, Bomb and Transport Airplanes

For predicting wing weight it is suggested to use Eqns.(5.6) and (5.7) but with the appropriate value for n_{ult}. For this type of military airplane the usual value for n_{ult} is 4.5. Refer to Table 4.1 for a listing of military limit load factors.

Note: wing weight in military airplanes is often based on the flight design gross weight, GW, rather than W_{TO}. Check the mission specification and/or the applicable military specifications to determine which weight value to use in Eqns.(5.6) and (5.7).

5.1.4 Fighter and Attack Airplanes

5.1.4.1 GD Method

For USAF fighter and attack airplanes:

$$W_w = 3.08[((K_w n_{ult} W_{TO})/(t/c)_m)((\tan\Lambda_{LE} - 2(1-\lambda)/A(1+\lambda))^2 + 1.0)\times10^{-6}]^{0.593}(A(1+\lambda))^{0.89}(S)^{0.741} \quad (5.9)$$

For USN fighter and attack airplanes:

$$W_w = 19.29[((K_w n_{ult} W_{TO})/(t/c)_m)((\tan\Lambda_{LE} - 2(1-\lambda)/A(1+\lambda))^2 + 1.0)\times10^{-6}]^{0.464}((1+\lambda)A)^{0.70}(S)^{0.58} \quad (5.10)$$

Definition of new terms:

K_w = 1.00 for fixed wing airplanes and
= 1.175 for variable sweep wing airplanes

Λ_{LE} = leading edge sweep angle of the wing

Note: wing weight in military airplanes is often based on the flight design gross weight, GW, rather than W_{TO}. Check the mission specification and/or the applicable military specifications to determine which weight to use in Eqns.(5.9) and (5.10).

5.2 EMPENNAGE WEIGHT ESTIMATION

Empennage weight, W_{emp} will be expressed as follows:

$$W_{emp} = W_h + W_v + W_c, \tag{5.11}$$

where: W_h = horizontal tail weight in lbs

W_v = vertical tail weight in lbs

W_c = canard weight in lbs

Equations for empennage weight components are presented in the remainder of this section.

5.2.1 General Aviation Airplanes

5.2.1.1 Cessna method

The following equations should be applied only to small, relatively low performance type airplanes with maximum speeds below 200 kts.

Horizontal tail:

$$W_h = \frac{3.184(W_{TO})^{0.887}(S_h)^{0.101}(A_h)^{0.138}}{57.5(t_{r_h})^{0.223}} \tag{5.12}$$

Note that no factor for horizontal tail sweep is included.

Vertical tail:

$$W_v = \frac{1.68(W_{TO})^{0.567}(S_v)^{1.249}(A_v)^{0.482}}{15.6(t_{r_v})^{0.747}(\cos\Lambda_{1/4_v})^{0.882}} \tag{5.13}$$

Canard: For a lightly loaded canard, Eqn.(5.12) may be used. For a significantly loaded canard (such as on the GP180 and the Starship I) it is suggested to use the appropriate wing weight equation.

Definition of terms:

W_{TO} = take-off weight ib lbs

S_h = horizontal tail area in ft^2

A_h = horizontal tail aspect ratio

t_{r_h} = horizontal tail maximum root thickness in ft

S_v = vertical tail area in ft^2

A_v = vertical tail aspect ratio

t_{r_v} = vertical tail maximum root thickness in ft

$1/4_v$ = vertical tail quarter chord sweep angle

5.2.1.2 USAF Method

The following equation applies to light and utility type airplanes with performance up to about 300 kts:

Horizontal tail:

$$W_h = 127[(W_{TO}n_{ult}/10^5)^{0.87}(S_h/100)^{1.2}x$$
$$x0.289(l_h/10)^{0.483}(b_h/t_{r_h})^{0.5}]^{0.458} \qquad (5.14)$$

Note that sweep angle is not a factor in this equation.

Vertical tail:

$$W_v = 98.5[(W_{TO}n_{ult}/10^5)^{0.87}(S_v/100)^{1.2}x$$
$$x0.289(b_v/t_{r_v})^{0.5}]^{0.458} \qquad (5.15)$$

Again, sweep angle is not a factor in this equation.

Canard:

The comments made under 5.2.1.2 also apply.

Definition of new terms:

l_h = distance from wing $\bar{c}/4$ to hor. tail $\bar{c}_h/4$ in ft

b_h = horizontal tail span in ft

b_v = vertical tail span in ft

5.2.1.3 Torenbeek Method

The following equation applies to light transport airplanes with design dive speeds up to 250 kts and with conventional tail configurations:

$$W_{emp} = 0.04(n_{ult}(S_v + S_h)^2)^{0.75}, \qquad (5.16)$$

If the airplane also has a canard, the comments made under 'canard' in 5.2.1.2 also apply here.

5.2.2 Commercial Transport Airplanes

5.2.2.1 GD Method

Horizontal tail:

$$W_h = 0.0034\{(W_{TO}n_{ult})^{0.813}(S_h)^{0.584} x (b_h/t_{r_h})^{0.033}(\bar{c}/l_h)^{0.28}\}^{0.915} \qquad (5.17)$$

Note: sweep angle is not a factor in this equation.

Vertical tail:

$$W_v = 0.19\{(1 + z_h/b_v)^{0.5}(W_{TO}n_{ult})^{0.363}(S_v)^{1.089}(M_H)^{0.601} x (l_v)^{-0.726}(1 + S_r/S_v)^{0.217}(A_v)^{0.337}(1+\lambda_v)^{0.363} x (\cos\Lambda_{1/4_v})^{-0.484}\}^{1.014} \qquad (5.18)$$

Canard: Comments made under 5.2.2.2 also apply here.

Definition of new terms:

z_h = distance from the vertical tail root to where the horizontal tail is mounted on the vertical tail, in ft. Warning: for fuselage mounted horizontal tails, set $z_h = 0$.

l_v = dist. from wing $\bar{c}/4$ to vert. tail $\bar{c}_v/4$ in ft

S_r = rudder area in ft^2

λ_v = vertical tail taper ratio

5.2.2.2 Torenbeek Method

The following equation applies to transport airplanes and to business jets with design dive speeds above 250 kts.

Horizontal tail:

$$W_h = \tag{5.19}$$

$$= K_h S_h [3.81\{(S_h)^{0.2} V_D\}/\{1,000(\cos \Lambda_{1/2_h})^{1/2}\} - 0.287]$$

where K_h takes on the following values:

K_h = 1.0 for fixed incidence stabilizers

K_h = 1.1 for variable incidence stabilizers

Vertical tail:

$$W_v = \tag{5.20}$$

$$= K_v S_v [3.81\{(S_v)^{0.2} V_D/1,000(\cos \Lambda_{1/2_v})^{1/2}\} - 0.287]$$

where K_v takes on the following values:

K_v = 1.0 for fuselage mounted horizontal tails

for fin mounted horizontal tails:

$$K_v = \{1 + 0.15(S_h z_h/S_v b_v)\} \tag{5.21}$$

Definition of new terms:

V_D = design dive speed in KEAS

$\Lambda_{1/2_h}$ horizontal tail semi-chord sweep angle

$\Lambda_{1/2_v}$ vertical tail semi-chord sweep angle

Canard: The comments made under 5.2.2.2 also apply here.

5.2.3 Military Patrol, Bomb and Transport Airplanes

See Sub-section 5.2.4.

5.2.4 Fighter and Attack airplanes

For estimation of empennage weight of airplanes in this category, use the methods of sub-section 5.2.2. Be sure to use the proper values for ultimate load factor. See Table 4.1.

Note: empennage weights of military airplanes are often based on the flight design gross weight, GW, rather than W_{TO}. Check the mission specification and/or the applicable military specifications to determine which weight to use.

5.3 FUSELAGE WEIGHT ESTIMATION

The equations presented for fuselage weight estimation are valid for land-based airplanes only. For flying boats and amphibious airplanes it is suggested to multiply the fuselage weight by 1.65:

$$W_{f_{fl.boat}} = 1.65W_f \tag{5.22}$$

For float equipped airplanes the weight due to the floats may be found with Eqn.(5.27), by substituting float wetted area for S_{fgs}.

For estimation of tailboom weight it is suggested to use Eqn.(5.27) applied to each tailboom individually, but with $K_f = 1$.

5.3.1 General Aviation airplanes

5.3.1.1 Cessna method

The following equations should be applied only to small, relatively low performance type airplanes with maximum speeds below 200 kts.

For low wing airplanes:

$$W_f = \tag{5.23}$$

$$= 0.04682(W_{TO})^{0.692}(N_{pax})^{0.374}(l_{f-n})^{0.590}/100$$

For high wing airplanes:

$$W_f = 14.86(W_{TO})^{0.144}(l_{f-n}/p_{max})^{0.778}(l_{f-n})^{0.383}(N_{pax})^{0.455} \quad (5.24)$$

Definition of terms:

W_{TO} = take-off weight in lbs

N_{pax} = number of passengers

l_{f-n} = fuselage length, not including nose mounted nacelle length in ft

Notes: 1. These equations do not account for pressurized fuselages.
2. There is no explanation for why the fuselage weight of low wing airplanes does not depend on the number of passengers.
3. For this type airplane the crew is counted in the number of passengers.

5.3.1.2 USAF Method

The following equation applies to light and utility type airplanes with performance up to about 300 kts:

$$W_f = 200[(W_{TO}n_{ult}/10^5)^{0.286}(l_f/10)^{0.857} x$$
$$x((w_f + h_f)/10)(V_C/100)^{0.338}]^{1.1} \quad (5.25)$$

Definition of new terms:

n_{ult} = ultimate load factor

l_f = fuselage length in ft

w_f = maximum fuselage width in ft

h_f = maximum fuselage height in ft

V_C = design cruise speed in KEAS

5.3.2 Commercial Transport Airplanes

5.3.2.1 GD Method

$$W_f = 2x10.43(K_{inl})^{1.42}(\bar{q}_D/100)^{0.283}(W_{TO}/1000)^{0.95}(l_f/h_f)^{0.71} \quad (5.26)$$

The factor K_{inl} takes on the following values:

K_{inl} = 1.25 for airplanes with inlets in or on the fuselage for a buried engine installation

K_{inl} = 1.0 for inlets located elsewhere

Definition of new term:

$\bar{q}_D$ = design dive dynamic pressure in psf

5.3.2.2 Torenbeek Method

The following equation applies to transport airplanes and to business jets with design dive speeds above 250 kts.

$$W_f = 0.021K_f\{(V_D l_h/(w_f + h_f)\}^{1/2}(S_{fgs})^{1.2} \qquad (5.27)$$

The constant K_f takes on the following values:

K_f = 1.08 for a pressurized fuselage

= 1.07 for a main gear attached to the fuselage.

= 1.10 for a cargo airplane with a cargo floor

These effects are multiplicative for airplanes equipped with all of the above.

Definition of new terms:

V_D = design dive speed in KEAS

l_h = distance from wing $\bar{c}/4$ to hor. tail $\bar{c}_h/4$ in ft

S_{fgs} = fuselage gross shell area in ft^2

5.3.3 Military Patrol, Bomb and Transport Airplanes

5.3.3.1 GD Method

For USAF airplanes, Eqn.(5.26) may be used.

For USN airplanes the following equation should be used:

$$W_f = \qquad (5.28)$$

$$= 11.03(K_{inl})^{1.23})(\bar{q}_L/100)^{0.245}(W_{TO}/1000)^{0.98}(l_f/h_f)^{0.61}$$

Values for K_{inl} are as given in 5.3.2.1.

Definition of new term:

$\bar{q}_L$ = design dive dynamic pressure in psf

5.3.4 Fighter and Attack Airplanes

For estimation of fuselage weights Equations (5.26) or (5.28) may be used.

Warning: In using Eqn. (5.26) for fighters, leave off the factor 2 at the beginning of the equation.

5.4 NACELLE WEIGHT ESTIMATION

The nacelle weight is assumed to consist of the following components:

1. For podded engines: the structural weight associated with the engine external ducts and or cowls. Any pylon weight is included.

2. For propeller driven airplanes: the structural weight associated with the engine external ducts and or cowls plus the weight due to the engine mounting trusses.

3. For buried engines: the structural weight associated with special cowling and or ducting provisions (other than the inlet duct which is included in the air induction system under powerplant weight, Section 6.2) and any special engine mounting provisions.

5.4.1 General Aviation Airplanes

5.4.1.1 Cessna Method

The following equations should be applied only to small, relatively low performance type airplanes with maximum speeds below 200 kts.

$$W_n = K_n W_{TO} \tag{5.29}$$

The constant K_n takes on the following values:

K_n = 0.37 lbs/hp for radial engines

K_n = 0.24 lbs/hp for horizontally opposed engines

Definition of term:

W_{TO} = take-off weight in lbs

These data should not be applied to turbopropeller nacelles.

5.4.1.2 USAF Method

In this method, the nacelle weight is included in the powerplant weight: refer to Chapter 6.

5.4.1.3 Torenbeek Method

For single engine propeller driven airplanes with the nacelle in the fuselage nose:

$$W_n = 2.5(P_{TO})^{1/2} \quad (5.30)$$

This weight includes the entire engine section forward of the firewall.

For multi-engine airplane with piston engines:

$$W_n = 0.32P_{TO} \text{ for horizontally opposed engines} \quad (5.31)$$

$$W_n = 0.045(P_{TO})^{5/4} \text{ for radial engines} \quad (5.32)$$

$$W_n = 0.14(P_{TO}) \text{ for turboprop engines} \quad (5.33)$$

Notes: 1. Since P_{TO} is the total required take-off horsepower, these weight estimates include the weights of all nacelles.

2. If the main landing gear retracts into the nacelles, add 0.04 lbs/hp to the nacelle weight

3. If the engine exhausts over the wing, as in the Lockheed Electra, add 0.11 lbs/hp to the nacelle weight.

5.4.2 Commercial Transport Airplanes

5.4.2.1 GD Method

For turbojet engines:

$$W_n = 3.0(N_{inl})\{(A_{inl})^{0.5}(l_n)(P_2)\}^{0.731} \quad (5.34)$$

For turbofan engines:

$$W_n = 7.435(N_{inl})\{(A_{inl})^{0.5}(l_n)(P_2)\}^{0.731} \qquad (5.35)$$

Definition of terms:

N_{inl} = number of inlets

A_{inl} = capture area per inlet $inft^2$

l_n = nacelle length from inlet lip to compressor face in ft

P_2 = maximum static pressure at engine compressor face in psi. Typical values range from 15 to 50 psi.

5.4.2.2 Torenbeek Method

For turbojet or low bypass ratio turbofan engines:

$$W_n = 0.055T_{TO} \qquad (5.36)$$

For high bypass ratio turbofan engines:

$$W_n = 0.065T_{TO} \qquad (5.37)$$

Since T_{TO} is the total required take-off thrust, these equations account for the weight of all nacelles.

5.4.3 Military Patrol Bomb and Transport Airplanes

For all airplanes in this category Eqns.(5.34) and (5.35) may be used.

5.4.4 Fighter and Attack Airplanes

For all airplanes in this category Eqns.(5.34) and (5.35) may be used.

5.5 LANDING GEAR WEIGHT ESTIMATION

5.5.1 General Aviation Airplanes

5.5.1.1 Cessna method

The following equations should be applied only to small, relatively low performance type airplanes with maximum speeds below 200 kts.

For non-retractable landing gears:

$$W_g = \tag{5.38}$$

$$\underbrace{0.013W_{TO}}_{\text{wheels + tires m.g.}} + \underbrace{0.146(W_L)^{0.417}n_{ult.l})^{0.950}(l_{s_m})^{0.183}}_{\text{strut assembly m.g.}} +$$

$$+ \underbrace{6.2 + 0.0013W_{TO}}_{\text{wheels + tires n.g.}} + \underbrace{0.000143(W_L)^{0.749}(n_{ult.l})(l_{s_n})^{0.788}}_{\text{strut assembly n.g.}}$$

For retractable landing gears:

$$W_g = W_{g_{Eqn.(5.38)}} + 0.014W_{TO} \tag{5.39}$$

Definition of terms:

W_{TO} = take-off weight in lbs

W_L = design landing weight in lbs (See Table 9.3, Part I for data relating W_L to W_{TO})

$n_{ult.l}$ = ultimate load factor for landing, may be taken as 5.7

l_{s_m} = shock strut length for main gear in ft

l_{s_n} = shock strut length for nose gear in ft

5.5.1.2 USAF Method

The following equation applies to light and utility type airplanes with performance up to about 300 kts:

$$W_g = 0.054(l_{s_m})^{0.501}(W_L n_{ult.l})^{0.684} \tag{5.40}$$

Notes: 1) This equation includes nose gear weight.

2) $N_{ult.l}$ may be taken as 5.7.

5.5.2 Commercial Transport Airplanes

5.5.2.1 GD Method

$$W_g = 62.61(W_{TO}/1,000)^{0.84} \tag{5.41}$$

5.5.2.2 Torenbeek Method

The following equation applies to transport airplanes and to business jets with the main gear mounted

on the wing and the nose gear mounted on the fuselage:

$$W_g = K_{g_r}\{A_g + B_g(W_{TO})^{3/4} + C_g W_{TO} + D_g(W_{TO})^{3/2}\} \qquad (5.42)$$

The factor K_{g_r} takes on the following values:

K_{g_r} = 1.0 for low wing airplanes

K_{g_r} = 1.08 for high wing airplanes

The constants A_g through D_g are defined in Table 5.1 which is taken from Reference 14.

Table 5.1 Constants in Landing Gear Weight Eqn.(5.42)

Airplane Type	Gear Type	Gear Comp.	A_g	B_g	C_g	D_g
Jet Trainers and Business Jets	Retr.	Main	33.0	0.04	0.021	0.0
		Nose	12.0	0.06	0.0	0.0
Other civil airplanes	Fixed	Main	20.0	0.10	0.019	0.0
		Nose	25.0	0.0	0.0024	0.0
		Tail	9	0.0	0.0024	0.0
	Retr.	Main	40.0	0.16	0.019	1.5×10^{-5}
		Nose	20.0	0.10	0.0	2.0×10^{-6}
		Tail	5.0	0.0	0.0031	0.0

5.5.3 Military Patrol, Bomb and Transport Airplanes

For USAF airplanes, Eqns.(5.41) and (5.42) may be used.

For USN airplanes the following equation should be used:

$$W_g = 129.1(W_{TO}/1,000)^{0.66} \qquad (5.43)$$

5.5.4 Fighter and Attack Airplanes

For USAF airplanes, Eqns.(5.41) and (5.42) may be used.

For USN airplanes, Eqn.(5.43) should be used.

6. CLASS II METHOD FOR ESTIMATING POWERPLANT WEIGHT

The airplane powerplant weight, W_{pwr} will be assumed to consist of the following components:

6.1 Engines, W_e: this includes engine, exhaust, cooling, supercharger and lubrication systems.

Note: afterburners and thrust reversers are not always included under engines. They are often treated as a separate powerplant component.

6.2 Air induction system, W_{ai}: this includes inlet ducts other than nacelles, ramps, spikes and associated controls.

6.3 Propellers, W_{prop}

6.4 Fuel System, W_{fs}

6.5 Propulsion System, W_p, this includes:

*engine controls
*starting systems
*propeller controls
*provisions for engine installation

Note: instead of the words 'propulsion system', the words 'propulsion installation' or even 'engine installation' are sometimes used.

Therefore:

$$W_{pwr} = W_e + W_{ai} + W_{prop} + W_{fs} + W_p \qquad (6.1)$$

General Note: for powerplant weight predictions it is highly recommended to obtain actual weight data from engine manufacturers.

Equations for powerplant weight prediction are presented for the following types of airplanes:

1. General Aviation Airplanes
2. Commercial Transport Airplanes
3. Military Patrol, Bomb and Transport Airplanes
4. Fighter and Attack Airplanes

6.1 ENGINE WEIGHT ESTIMATION

6.1.1 General Aviation Airplanes

6.1.1.1 Cessna method

The following equations should be applied only to small, relatively low performance type airplanes with maximum speeds below 200 kts.

$$W_e = K_p P_{TO} \tag{6.2}$$

The factor K_p takes on the following values:

For piston engines: K_p = 1.1 to 1.8, depending on whether or not supercharging is used.

For turbopropeller engines: K_p = 0.35 to 0.55.

These weights represent the so-called engine dry weight. Normal engine accessories are included in this weight but engine oil is not.

Definition of terms:

W_e = weight of all engines in lbs

P_{TO} = required take-off power in hp

6.1.1.2 USAF Method

$$W_e + W_{ai} + W_{prop} + W_p = 2.575(W_{eng})^{0.922}N_e \tag{6.3}$$

Use engine manufacturers data to obtain W_{eng} or use Eqn.(6.2).

Definition of new terms:

W_{eng} = weight per engine in lbs

N_e = number of engines

6.1.1.3 Torenbeek Method

For propeller driven airplanes:

$$W_{pwr} = K_{pg}(W_e + 0.24P_{TO}) \tag{6.4}$$

The constant K_{pg} takes on the following values:

K_{pg} = 1.16 for single engine tractor installations

K_{pg} 1.35 for multi-engine installations

For superchargers the following additional weight is incurred:

$$W_{sprch} = 0.455(W_e)^{0.943} \quad (6.5)$$

<u>For jet airplanes:</u>

$$W_{pwr} = K_{pg}K_{thr}W_e \quad (6.6)$$

The constant K_{pg} takes on the following values:

K_{pg} = 1.40 for airplanes with buried engines

The constant K_{thr} takes on the following values:

K_{thr} = 1.00 for airplanes without thrust reversers

K_{thr} = 1.18 for airplanes with thrust reversers

6.1.2 Commercial Transport Airplanes

Use of actual engine manufactures data is highly recommended. Figure 6.1 provides a graphical summary of engine dry weights versus take-off thrust. Figure 6.2 gives a graphical summary of engine dry weights versus take-off shaft horsepower.

When using Figures (6.1) or (6.2), keep in mind that:

$$W_e = N_eW_{eng}, \quad (6.7)$$

where W_{eng} is the weight per engine.

Equations (6.5) and (6.6) may also be used to obtain an initial estimate.

6.1.3 Military Patrol, Bomb and Transport Airplanes

See Sub-Section 6.1.2.

6.1.4 Fighter and Attack Airplanes

See Sub-Section 6.1.2.

Figure 6.1 Turbojets and Turbofans: Take-off Thrust and Dry Weight Trends

Figure 6.2 Turboprops: Take-off Shaft Horse Power and Dry Weight Trends

6.2 AIR INDUCTION SYSTEM WEIGHT ESTIMATION

6.2.1 General Aviation Airplanes

6.2.1.1 Cessna method

W_{ai} is included in the propulsion system weight, W_p.

6.2.1.2 USAF Method

See 6.2.1.1.

6.2.1.3 Torenbeek Method

$$W_{ai} + W_p = 1.03(N_e)^{0.3}(P_{TO}/N_e)^{0.7} \qquad (6.8)$$

6.2.2 Commercial Transport Airplanes

6.2.2.1 GD Method

For buried engine installations:

The air induction system weight is split into two items: the first one for duct support structure, the second one for the subsonic duct leading from the inlet lip to the engine compressor face.

$$W_{ai} = 0.32(N_{inl})(L_d)(A_{inl})^{0.65}(P_2)^{0.6} + \qquad (6.9)$$

(duct support structure)

$$1.735[(L_d)(N_{inl})(A_{inl})^{0.5}(P_2)(K_d)(K_m)]^{0.7331}$$

(subsonic part of duct)

The factors K_d and K_m are defined as follows:

K_d = 1.33 for ducts with flat cross sections

1.0 for ducts with curved cross sections

K_m = 1.0 for M_D below 1.4

= 1.5 for M_D above 1.4

Definition of terms:

L_d = duct length in ft

N_{inl} = number of inlets

A_{inl} = capture area per inlet in ft^2

$\bar{P}_2$ = maximum static pressure at engine compressor face in psi. Typical values range from 15 to 50 psi.

For podded engine installations:

The air induction system weight is included in the nacelle weight, W_n.

6.2.2.2 Torenbeek Method

For buried engine installations:

$$W_{ai} = 11.45[(L_d)(N_{inl})(A_{inl})^{0.5}(K_d)]^{0.7331} \qquad (6.10)$$

The constant K_d takes on the following values:

K_d = 1.0 for ducts with curved cross sections

1.33 for ducts with flat cross sections

For podded engine installations:

The air induction system weight is included in the nacelle weight, W_n.

Note: For supersonic installations additional weight items due to the special inlet requirements are needed. See Sub-section 6.2.4.

6.2.3. Military Patrol, Bomb and Transport Airplanes

See Section 6.2.2.

6.2.4 Fighter and Attack Airplanes

6.2.4.1 GD Method

For prediction of the duct support structure weight and the duct weight, Eqn.(6.9) may be used.

Particularly in supersonic applications the following additional weight items due to inlet provisions may be incurred:

For variable geometry ramps, actuators and controls:

$$W_{ramp} = 4.079[(L_r)(N_{inl})(A_{inl})^{0.5}(K_r)]^{1.201} \qquad (6.11)$$

The factor K_r takes on the following values:

K_r = 1.0 for M_D below 3.0

= $(M_D + 2)/5$ for M_D above 3.0

Definition of new term:

L_r is the ramp length forward of the inlet throat in ft

For inlet spikes:

$$W_{sp} = K_s(N_{inl})(A_{inl}) \quad (6.12)$$

The constant K_s takes on the following values:

K_s = 12.53 for half round fixed spikes
= 15.65 for full round translating spikes
= 51.80 for translating and expanding spikes

Note: these weights also apply to supersonic commercial installations.

6.3 PROPELLER WEIGHT ESTIMATION

6.3.1 General Aviation Airplanes

It is recommended to use propeller manufacturer data whereever possible. Lacking actual data the equation of Sub-Section 6.3.2 may be used.

Appendix A contains propeller installation data for a number of airplanes. Propeller installation weights usually include the propeller controls.

6.3.2 Commercial Transport Airplanes

6.3.2.1 GD Method

$$W_{prop} = K_{prop1}(N_p)(N_{bl})^{0.391}\{(D_p)(P_{TO}/N_e)/1,000\}^{0.782} \quad (6.13)$$

The constant K_{prop1} takes on the following values:

K_{prop1} = 24.0 for turboprops above 1,500 shp

= 31.92 for piston engines and for turboprops below 1,500 shp

Definition of terms:

N_p is the number of propellers

N_{bl} is the number of blades per propeller

D_p is the propeller diameter in ft

P_{TO} is the required take-off power in hp

N_e is the number of engines

6.3.2.2 Torenbeek Method

$$W_{prop} = K_{prop2}(N_p)^{0.218}\{D_p P_{TO}(N_{bl})^{1/2}\}^{0.782} \quad (6.14)$$

The factor K_{prop2} takes on the following values:

K_{prop2} = 0.108 for turboprops

K_{prop2} = 0.144 for piston engines

The reader is asked to show that equations (6.13) and (6.14) are in fact the same.

6.3.3 Military Patrol,Bomb and Transport Airplanes

See Sub-Section 6.3.2.

6.3.4 Fighter and Attack Airplanes

See Sub-Section 6.3.2.

6.4 FUEL SYSTEM WEIGHT ESTIMATION

Note: In some airplanes the fuel system is used to control the center of gravity location. Airplanes with relaxed static stability and/or supersonic cruise airplanes frequently require such a system. The weight increment incurred due to such a feature is included in the weight estimation of the flight control system, Section 7.1.

6.4.1 General Aviation Airplanes

6.4.1.1 Cessna method

For airplanes with internal fuel systems (no tiptanks):

$$W_{fs} = 0.40W_F/K_{fsp} \quad (6.15)$$

For airplanes with external fuel systems (with tiptanks):

$$W_{fs} = 0.70W_F/K_{fsp} \tag{6.16}$$

The constant K_{fsp} takes on the following values:

K_{fsp} = 5.87 lbs/gal for aviation gasoline

= 6.55 for lbs/gal for JP-4

Definition of term:

W_F = mission fuel weight (includes reserves) in lbs

6.4.1.2 USAF Method

$$W_{fs} = \tag{6.17}$$

$$= 2.49[(W_F/K_{fsp})^{0.6}(1/(1+int))^{0.3}(N_t)^{0.20}(N_e)^{0.13}]^{1.21}$$

The factor K_f is defined in 6.4.1.1.

Definition of new terms:

int = fraction of fuel tanks which are integral

N_t = number of separate fuel tanks

N_e = number of engines

6.4.1.3 Torenbeek Method

For turbine engines, see Sub-Section 6.4.2.

For single piston engine installations:

$$W_{fs} = 2(W_F/5.87)^{0.667} \tag{6.18}$$

For multi piston engine installations:

$$W_{fs} = 4.5(W_F/5.87)^{0.60} \tag{6.19}$$

6.4.2 Commercial Transport Airplanes

6.4.2.1 GD Method

For a fuel system with integral tanks see 6.4.2.2.

For a fuel system with self-sealing bladder cells:

$$W_{fs} = 41.6\{(W_F/K_{fsp})/100\}^{0.818} + W_{supp} \quad (6.20)$$

For a fuel system with non-self-sealing bladder cells:

$$W_{fs} = 23.1\{(W_F/K_{fsp})/100\}^{0.758} + W_{supp} \quad (6.21)$$

The factor K_{fsp} is defined in 6.4.1.1.

W_{supp} is the weight of the bladder support structure and is given by:

$$W_{supp} = 7.91\{(W_F/K_{fsp})/100\}^{0.854} \quad (6.22)$$

6.4.2.2 Torenbeek Method

For airplanes equipped with non-self-sealing bladder tanks:

$$W_{fs} = 3.2(W_F/K_{fsp})^{0.727} \quad (6.23)$$

For airplanes equipped with integral fuel tanks (wet wing):

$$W_{fs} = 80(N_e + N_t - 1) + 15(N_t)^{0.5}(W_F/K_{fsp})^{0.333} \quad (6.24)$$

6.4.3 Military Patrol, Bomb and Transport Airplanes

For basic fuel system weights, see Sub-Section 6.4.2.

Many military airplanes carry in flight refuelling systems. In addition, many are equipped with fuel dumping systems. The weights of these systems may be estimated from:

For in-flight refuelling:

$$W_{inflref} = 13.64\{(W_F/K_{fsp})/100\}^{0.392} \quad (6.25)$$

For fuel dumping:

$$W_{fd} = 7.38\{(W_F/K_{fsp})/100\}^{0.458} \quad (6.26)$$

6.4.4 Fighter and Attack Airplanes

See Sub-Sections 6.4.2 and 6.4.3.

6.5 PROPULSION SYSTEM WEIGHT ESTIMATION

Depending on airplane type, the propulsion system weight, W_p is either given as a function of total engine weight and/or mission fuel or by:

$$W_p = W_{ec} + W_{ess} + W_{pc} + W_{osc}, \text{ where:} \quad (6.22)$$

W_{ec} = weight of engine controls in lbs

W_{ess} = weight of engine starting system in lbs

W_{pc} = weight of propeller controls in lbs

W_{osc} = weight of oil system and oil cooler in lbs

6.5.1 General Aviation Airplanes

6.5.1.1 Cessna method

Use actual data.

6.5.1.2 USAF Method

W_p is included in Eqn.(6.3).

6.5.1.3 Torenbeek Method

W_p is included in Eqn.(6.3).

6.5.2 Commercial Transport Airplanes

6.5.2.1 GD Method

Engine controls:

For fuselage/wing-root mounted jet engines:

$$W_{ec} = K_{ec}(l_f N_e)^{0.792} \quad (6.23)$$

The factor K_{ec} takes on the following values:

K_{ec} = 0.686 for non-afterburning engines
= 1.080 for afterburning engines

For wing mounted jet engines:

$$W_{ec} = 88.46((l_f + b)N_e/100)^{0.294} \quad (6.24)$$

For wing mounted turboprops:

$$W_{ec} = 56.84\{(l_f + b)N_e/100\}^{0.514} \quad (6.25)$$

For wing mounted piston engines:

$$W_{ec} = 60.27\{(l_f + b)N_e/100\}^{0.724} \quad (6.26)$$

Definition of terms:

N_e = number of engines

l_f = fuselage length in ft

b = wing span in ft

Engine starting systems:

For airplanes with one or two jet engines using cartridge or pneumatic starting systems:

$$W_{ess} = 9.33(W_e/1,000)^{1.078} \quad (6.27)$$

For airplanes with four or more jet engines using pneumatic starting systems:

$$W_{ess} = 49.19(W_e/1,000)^{0.541} \quad (6.28)$$

For airplanes with jet engines using electric starting systems:

$$W_{ess} = 38.93(W_e/1,000)^{0.918} \quad (6.29)$$

For airplanes with turboprop engines using pneumatic starting systems:

$$W_{ess} = 12.05(W_e/1,000)^{1.458} \quad (6.30)$$

For airplanes with piston engines using electric starting systems:

$$W_{ess} = 50.38(W_e/1,000)^{0.459} \quad (6.31)$$

Propeller controls:

For turboprop engines:

$$W_{pc} = 0.322(N_{bl})^{0.589}\{(N_p D_p P_{TO}/N_e)/1,000\}^{1.178} \quad (6.32)$$

For piston engines:

$$W_{pc} = 4.552(N_{bl})^{0.379}\{(N_p D_p P_{TO}/N_e)/1,000\}^{0.759} \quad (6.33)$$

Definition of term:

W_e = total weight of all engines in lbs

6.5.2.2 Torenbeek Method

For airplanes with turbojet or turbofan engines using cartridge or pneumatic starting systems, the weight for accessory drives, powerplant controls, starting and ignition systems is:

$$W_{apsi} = 36N_e(dW_F/dt)_{TO} \quad (6.34a)$$

The take-off fuel flow rate, $(dW_F/dt)_{TO}$ has the dimension of lbs/sec.

For airplanes with turboprop engines this weight is:

$$W_{apsi} = 0.4K_b(N_e)^{0.2}(P_{TO}/N_e)^{0.8} \quad (6.34b)$$

The factor K_b takes on the following values:

K_b = 1.0 without beta controls
= 1.3 with beta controls

It is usually acceptable to assume that:

$$W_{api} = W_p - W_{osc} \quad (6.35)$$

Definition of new terms:

$(dW_F/dt)_{TO}$ = fuel flow at take-off in lbs/sec

P_{TO} = required take-off power in hp

Thrust reversers for jet engines:

The weight of thrust reversers was already included in the engine weight estimate of Eq.(6.6). To obtain a better estimate of the c.g. effect due to thrust reversers a separate weight estimate is needed:

$$W_{tr} = 0.18W_e \quad (6.36)$$

Water injection system:

Water injection systems are used to increase take-off performance of all types of engines. The installation of such a system is optional.

$$W_{wi} = 8.586W_{wtr}/8.35 \qquad (6.37)$$

W_{wtr} = weight of water carried in lbs

Oil system and oil cooler:

$$W_{osc} = K_{osc}W_e \qquad (6.38)$$

The factor K_{osc} takes on the following values:

K_{osc} = 0.00 for jet engines (weight incl. in W_e)
= 0.07 for turboprop engines
= 0.08 for radial piston engines
= 0.03 for horizontally opposed piston engines

6.5.3 Military Patrol, Bomb and Transport Airplanes

See Section 6.5.2.

6.5.4 Fighter and Attack Airplanes

See Section 6.5.2.

7. CLASS II METHOD FOR ESTIMATING FIXED EQUIPMENT WEIGHT

The list of fixed equipment carried on board airplanes varies significantly with airplane type and airplane mission. In this chapter it will be assumed that the following items are to be included in the fixed equipment category:

7.1. Flight control system, W_{fc}

7.2. Hydraulic and pneumatic System, W_{hps}

7.3. Electrical system, W_{els}

7.4. Instrumentation, avionics and electronics, W_{iae}

7.5. Air-conditioning, pressurization, anti- and de-icing system, W_{api}

7.6. Oxygen system, W_{ox}

7.7. Auxiliary power unit (APU), W_{apu}

7.8. Furnishings, W_{fur}

7.9. Baggage and cargo handling equipment, W_{bc}

7.10 Operational items, W_{ops}

7.11. Armament, W_{arm}

7.12. Guns, launchers and weapons provisions, W_{glw}

7.13. Flight test instrumentation, W_{fti}

7.14. Auxiliary gear, W_{aux}

7.15. Ballast, W_{bal}

7.16. Paint, W_{pt}

7.17. W_{etc}

Therefore:

$$W_{feq} = W_{fc} + W_{hps} + W_{els} + W_{iae} + W_{api} + W_{ox} + W_{apu} + W_{fur} + W_{bc} + W_{ops} + W_{arm} + W_{glw} + W_{fti} + W_{aux} + W_{bal} + W_{pt} + W_{etc} \quad (7.1)$$

The exact definition of which item belongs in a particular fixed equipment category is hard to find. The category W_{etc} was added to cover any items not specifically listed.

Methods for predicting weights of typical fixed equipment items are presented for the following types of airplanes:

1. General Aviation Airplanes
2. Commercial Transport Airplanes
3. Military Patrol, Bomb and Transport Airplanes
4. Fighters and Attack Airplanes

The reader should always consult actual fixed equipment weight data for similar airplanes. Appendix A presents this information for a large number of airplanes.

7.1 FLIGHT CONTROL SYSTEM WEIGHT ESTIMATION

7.1.1 General Aviation Airplanes

7.1.1.1 Cessna Method

$$W_{fc} = 0.0168W_{TO}, \tag{7.1}$$

where: W_{TO} = take-off weight in lbs

This equation applies only to airplanes under 8,000 lbs take-off weight with mechanical flight controls. The equation includes all flight control system hardware: cables, pulleys, pushrods, cockpit controls plus any required back-up structure.

Airplanes in this category all tend to have two sets of flight controls in the cockpit.

7.1.1.2 USAF Method

For airplanes with un-powered flight controls:

$$W_{fc} = 1.066(W_{TO})^{0.626} \tag{7.2}$$

For airplanes with powered flight controls:

$$W_{fc} = 1.08(W_{TO})^{0.7} \tag{7.3}$$

7.1.1.3 Torenbeek Method

For airplanes with un-powered, unduplicated flight controls:

$$W_{fc} = 0.23(W_{TO})^{2/3} \tag{7.4}$$

7.1.2 Commercial Transport Airplanes

7.1.2.1 GD Method

The following equation applies to business jets as well as to commercial transport airplanes:

$$W_{fc} = 56.01\ \{(W_{TO})(\bar{q}_D)/100,000\}^{0.576} \tag{7.5}$$

where: $\bar{q}_D$ is the design dive dynamic pressure in psf

7.1.2.2 Torenbeek Method

$$W_{fc} = K_{fc}(W_{TO})^{2/3} \tag{7.6}$$

The constant K_{fc} takes on the following values:

K_{fc} = 0.44 for airplanes with un-powered flight controls
= 0.64 for airplanes with powered flight controls

If leading edge devices are employed, these estimates should be multiplied by a factor 1.2. If lift dumpers are employed, a factor 1.15 should be used.

7.1.3 Military Patrol, Bomb and Transport Airplanes

7.1.3.1 GD Method

For transport airplanes:

$$W_{fc} = 15.96\{(W_{TO})(\bar{q}_L)/100,000\}^{0.815}, \tag{7.7}$$

where: $\bar{q}_L$ is the design dive dynamic pressure in psf

For Bombers:

$$W_{fc} = 1.049\{(S_{cs})(\bar{q})/1,000\}^{1.21}, \tag{7.8}$$

where: S_{sc} is the total control surface area in ft^2

Note: these estimates include the weight of all associated hydraulic and/or pneumatic systems!

7.1.4 Fighters and Attack Airplanes

7.1.4.1 GD Method

For USAF fighters:

$$W_{fc} = K_{fcf}(W_{TO}/1,000)^{0.581} \qquad (7.9)$$

The constant K_{fcf} takes on the following values:

K_{fcf} = 106 for airplanes with elevon control and no horizontal tail

= 138 for airplanes with a horizontal tail

= 168 for airplanes with a variable sweep wing

For USN fighters and attack airplanes:

$$W_{fc} = 23.77(W_{TO}/1,000)^{1.1} \qquad (7.10)$$

Note: these estimates include the weight of all associated hydraulic and/or pneumatic systems.

Certain airplanes require a center of gravity control system. This is normally implemented using a fuel transfer system. The extra weight due to a c.g. control system may be estimated from:

$$W_{fc_{cg}} = 23.38[(W_F/K_{fsp})/100]^{0.442} \qquad (7.11)$$

where: W_F is the mission fuel weight in lbs

K_{fsp} = 6.55 lbs /gal for JP-4

7.2 HYDRAULIC AND/OR PNEUMATIC SYSTEM WEIGHT ESTIMATION

As seen in Section 7.1 the weight of the hydraulic and/or pneumatic system needed for powered flight controls is usually included in the flight control system weight prediction.

The following weight ratios may be used to determine the hydraulic system weight separately:

For business jets: 0.0070 - 0.0150 of W_{TO}

For regional turboprops: 0.0060 - 0.0120 of W_{TO}

For commercial transports: 0.0060 - 0.0120 of W_{TO}

For military patrol, transport and bombers:

0.0060 - 0.0120 of W_{TO}

For fighters and attack airplanes:

0.0050 - 0.0180 of W_{TO}

The reader should consult the detailed weight data in Appendix A for more precise information.

7.3 ELECTRICAL SYSTEM WEIGHT ESTIMATION

The reader should consult the detailed weight data in Appendix A for electrical system weights of specific airplanes.

7.3.1 General Aviation Airplanes

7.3.1.1 Cessna Method

$$W_{els} = 0.0268W_{TO} \tag{7.12}$$

7.3.1.2 USAF Method

$$W_{els} = 426\{(W_{fs} + W_{iae})/1,000\}^{0.51} \tag{7.13}$$

Note that the electrical system weight in this case is given as a function of the weight of the fuel system plus the weight of instrumentation, avionics and electronics.

7.3.1.3 Torenbeek Method

$$W_{hps} + W_{els} = 0.0078(W_E)^{1.2}, \quad (7.14)$$

where: W_E is the empty weight in lbs

7.3.2 Commercial Transport Airplanes

7.3.2.1 GD Method

$$W_{els} = 1.163\{(W_{fs} + W_{iae})/1,000\}^{0.506} \quad (7.15)$$

7.3.2.2 Torenbeek Method

For propeller driven transports:

$$W_{hps} + W_{els} = 0.325(W_E)^{0.8} \quad (7.16)$$

For jet transports:

$$W_{els} = 10.8(V_{pax})^{0.7}\{1 - 0.018(V_{pax})^{0.35}\}, \quad (7.17)$$

where: V_{pax} is the passenger cabin volume in ft^3

7.3.3 Military Patrol, Bomb and Transport Airplanes

7.3.3.1 GD Method

For transport airplanes:

Use Eqn.(7.15)

For Bombers:

$$W_{els} = 185\{(W_{fs} + W_{iae})/1,000\}^{1.268} \quad (7.18)$$

7.3.4 Fighters and Attack Airplanes

7.3.4.1 GD Method

For USAF fighters:

$$W_{els} = 426\{(W_{fs} + W_{iae})/1,000\}^{0.51} \quad (7.19)$$

For USN fighters and attack airplanes:

$$W_{els} = 347\{(W_{fs} + W_{iae})/1,000\}^{0.509} \quad (7.20)$$

7.4 WEIGHT ESTIMATION FOR INSTRUMENTATION, AVIONICS AND ELECTRONICS

The reader should consult the detailed weight data in Appendix A for weights of instrumentation, avionics and electronics for specific airplanes. Another important source of weight data on actual avionics and electronics systems for civil airplanes is Reference 18. For data on military avionics systems the reader should consult Reference 13, Tables 8-1 and 8-2.

Important comment: The weight equations given in this section are obsolete for modern EFIS type cockpit installations and for modern computer based flight management and navigation systems. The equations provided are probably conservative.

7.4.1 General Aviation Airplanes

7.4.1.1 Torenbeek Method

For single engine propeller driven airplanes:

$$W_{iae} = 33N_{pax}, \tag{7.21}$$

where: N_{pax} is the number of passengers, including the crew

For multi-engine propeller driven airplanes:

$$W_{iae} = 40 + 0.008W_{TO} \tag{7.22}$$

7.4.2 Commercial Transport Airplanes

7.4.2.1 GD Method (Modified)

For the weight of instruments:

$$W_i = \underbrace{N_{pil}\{15 + 0.032(W_{TO}/1{,}000)\}}_{\text{flight instruments}} + \underbrace{N_e\{5 + 0.006(W_{TO}/1{,}000)\}}_{\text{engine instruments}} + \underbrace{0.15(W_{TO}/1{,}000) + 0.012W_{TO}}_{\text{other instruments}} \tag{7.23}$$

where: N_{pil} is the number of pilots

N_e is the number of engines

7.4.2.2 Torenbeek Method

For regional transports:

$$W_{iae} = 120 + 20N_e + 0.006W_{TO} \tag{7.24}$$

For jet transports:

$$W_{iae} = 0.575(W_E)^{0.556}(R)^{0.25}, \tag{7.25}$$

where: W_E is the empty weight in lbs

R is the maximum range in nautical miles

7.4.3 Military Patrol, Bomb and Transport Airplanes

Use Sub-section 7.4.2.

7.4.4 Fighter and Attack Airplanes

Use Sub-section 7.4.2.

7.5 WEIGHT ESTIMATION FOR AIR-CONDITIONING, PRESSURIZATION, ANTI- AND-DEICING SYSTEMS

7.5.1 General Aviation Airplanes

7.5.1.1 USAF Method

$$W_{api} = 0.265(W_{TO})^{0.52}(N_{pax})^{0.68} x$$
$$x(W_{iae})^{0.17}(M_D)^{0.08}, \tag{7.26}$$

where: N_{pax} is the number of passengers, including the crew

M_D is the design dive Mach number

7.5.1.2 Torenbeek Method

For single engine, unpressurized airplanes:

$$W_{api} = 2.5N_{pax} \tag{7.27}$$

For multi-engine, unpressurized airplanes:

$$W_{api} = 0.018W_E \tag{7.28}$$

7.5.2 Commercial Transport Airplanes

7.5.2.1 GD Method

For pressurized airplanes:

$$W_{api} = 469[V_{pax}(N_{cr} + N_{pax})/10,000]^{0.419} \qquad (7.29)$$

7.5.2.2 Torenbeek Method

For pressurized airplanes:

$$W_{api} = 6.75(l_{pax})^{1.28} \qquad (7.30)$$

where l_{pax} = length of the passenger cabin in ft

7.5.3 Military Patrol, Bomb and Transport Airplanes

7.5.3.1 GD Method

$$W_{api} = K_{api}(V_{pr}/100)^{0.242} \qquad (7.31)$$

The constant K_{api} takes on the following values:

K_{api} = 887 for subsonic airplanes with wing and tail anti-icing

= 610 for subsonic airplanes without anti-icing

= 748 for supersonic airplanes without anti-icing

7.5.4 Fighters and Attack airplanes

7.5.4.1 GD Method

For low subsonic airplanes:

$$W_{api} = K_{api}[(W_{iae} + 200N_{cr})/1,000]^{0.538} \qquad (7.32)$$

The constant K_{api} takes on the following values:

K_{api} = 212 for airplanes with wing and tail anti-icing

= 109 for airplanes without anti-icing

For high subsonic and for supersonic airplanes:

$$W_{api} = 202[(W_{iae} + 200N_{cr})/1,000]^{0.735} \qquad (7.33)$$

7.6 WEIGHT ESTIMATION FOR THE OXYGEN SYSTEM

7.6.1 General Aviation Airplanes

Use Sub-section 7.6.2.

Note: Equation number (7.34) has been intentionally deleted.

7.6.2 Commercial Transport Airplanes

7.6.2.1 GD Method

$$W_{ox} = 7(N_{cr} + N_{pax})^{0.702} \tag{7.35}$$

7.6.2.2 Torenbeek Method

For commercial transport airplanes and for business type airplanes:

For flights below 25,000 ft:

$$W_{ox} = 20 + 0.5N_{pax} \tag{7.36}$$

For short flights above 25,000 ft:

$$W_{ox} = 30 + 1.2N_{pax} \tag{7.37}$$

For extended overwater flights:

$$W_{ox} = 40 + 2.4N_{pax} \tag{7.38}$$

7.6.3 Military Patrol, Bomb and Transport airplanes

Use Sub-section 7.6.2.

7.6.4 Fighters and Attack airplanes

7.6.4.1 GD Method

$$W_{ox} = 16.9(N_{cr})^{1.494} \tag{7.39}$$

7.7 AUXILIARY POWER UNIT WEIGHT ESTIMATION

Auxiliary power units are often used in transport or patrol type airplanes, commercial as well as military.

Actual APU manufacturer data should be used, where possible. Reference 8 contains data on APU systems, under 'Engines'.

From the detailed weight statements in Appendix A it is possible to derive weight fractions for these systems as a function of the take-off weight, W_{TO}. The following ranges are typical of these weight fractions:

$$W_{apu} = (0.004 \text{ to } 0.013)W_{TO} \tag{7.40}$$

7.8 FURNISHINGS WEIGHT ESTIMATION

The furnishings category normally includes the following items:

1. seats, insulation, trim panels, sound proofing, instrument panels, control stands, lighting and wiring
2. Galley (pantry) structure and provisions
3. Lavatory (toilet) and associated systems
4. Overhead luggage containers, hatracks, wardrobes
5. Escape provisions, fire fighting equipment

Note: the associated consumable items such as potable water, food, beverages and toilet chemicals and papers are normally included in a weight category referred to as: Operational Items: W_{ops}, see Section 7.10.

The reader is referred to the detail weight statements in Appendix A for actual furnishings weight data on specific airplanes.

7.8.1 General Aviation airplanes

7.8.1.1 Cessna Method

$$W_{fur} = 0.412(N_{pax})^{1.145}(W_{TO})^{0.489}, \tag{7.41}$$

where: N_{pax} is the number of passengers including the crew

7.8.1.2 Torenbeek Method:

For single engine airplanes:

$$W_{fur} = 5 + 13N_{pax} + 25N_{row}, \quad (7.42)$$

where: N_{row} is the number of seat rows

For multi engine airplanes:

$$W_{fur} = 15N_{pax} + 1.0V_{pax+cargo}, \quad (7.43)$$

where: $V_{pax+cargo}$ is the volume of the passenger cabin plus the cargo volume in ft^3

7.8.2 Commercial Transport Airplanes

The weight of furnishings varies considerably with airplane type and with airplane mission. This weight item is a considerable fraction of the take-off weight of most airplanes, as the data in Appendix A illustrate.

Reference 14 contains a very detailed method for estimating the furnishings weight for commercial transport airplanes.

7.8.2.1 GD Method

$$W_{fur} = \quad (7.44)$$

$$\underbrace{55N_{fdc}}_{\text{fdc sts}} + \underbrace{32N_{pax}}_{\text{pax sts}} + \underbrace{15N_{cc}}_{\text{cc sts}} + \underbrace{K_{lav}(N_{pax})^{1.33}}_{\text{lavs + water}} + \underbrace{K_{buf}(N_{pax})^{1.12}}_{\text{food prov.}}$$

$$+ \underbrace{109((N_{pax}(1 + P_c)/100)^{0.505}}_{\text{cabin windows}} + \underbrace{0.771(W_{TO}/1,000)}_{\text{miscellaneous}}$$

The factor K_{lav} takes on the following values:

K_{lav} = 3.90 for business airplanes
= 0.31 for short range airplanes
= 1.11 for long range airplanes

The factor K_{buf} takes on the following values:

K_{buf} = 1.02 for short ranges
= 5.68 for very long ranges

The term P_c is the design ultimate cabin pressure in psi. The value of P_c depends on the design altitude for the pressure cabin.

7.8.2.2 Torenbeek Method

$$W_{fur} = 0.211(W_{TO} - W_F)^{0.91} \qquad (7.45)$$

In commercial transports it is usually desirable to make more detailed estimates than possible with Eqn.(7.45). Particularly if a more accurate location of the c.g. of items which contribute to the furnishings weight is needed, a more detailed method may be needed. Reference 14 contains the necessary detailed information.

7.8.3 Military Patrol, Bomb and Transport Airplanes

7.8.3.1 GD Method

W_{fur} = Sum ↓ in the tabulation below. (7.46)

Type	Patrol	Bomb	Transport
Crew Ej. Seats	$K_{st}(N_{cr})^{1.2}$	$K_{st}(N_{cr})^{1.2}$	
	K_{st} = 149 with survival kit = 100 without survival kit		
Crew Seats	$83(N_{cr})^{0.726}$	same	same
Passenger Seats			$32(N_{pax})$
Troop Seats			$11.2(N_{troop})$
Lav. and Water			$1.11(N_{pax})^{1.33}$
Misc.	$0.0019(W_{TO})^{0.839}$		$0.771(W_{TO}/1,000)$

7.8.4 Fighters and Attack Airplanes

$$W_{fur} = \qquad (7.47)$$

$$= 22.9(N_{cr}\bar{q}_D/100)^{0.743} + 107(N_{cr}W_{TO}/100,000)^{0.585}$$

ejection seats Misc. and emergency eqpmt

7.9 WEIGHT ESTIMATION OF BAGGAGE AND CARGO HANDLING EQUIPMENT

The GD method gives for military passenger transports:

$$W_{bc} = K_{bc}(N_{pax})^{1.456} \tag{7.48}$$

The constant K_{bc} takes on the following values:

K_{bc} = 0.0646 without preload provisions
= 0.316 with preload provisions

The Torenbeek method gives for commercial cargo airplanes:

$$W_{bc} = 3S_{ff}, \tag{7.49}$$

where: S_{ff} is the freight floor area in ft^2.

For baggage and for cargo containers, the following weight estimates may be used:

freight pallets:	88x108 in	225 lbs
(including nets)	88x125 in	262 lbs
	96x125 in	285 lbs

containers: 1.6 lbs/ft^3 (For container dimensions, see Part III.)

7.10 WEIGHT ESTIMATION OF OPERATIONAL ITEMS

Typical weights counted in operational items are:

*Food *Potable water *Drinks

*China *Lavatory supplies

Observe that Eqn. (7.44) includes these operational items. For more detailed information on operational items the reader should consult Reference 14, p.292.

7.11 ARMAMENT WEIGHT ESTIMATION

The category armament can contain a wide variety of weapons related items as well as protective shielding for the crew. Typical armament items are:

*Firing systems *Fire control systems

*Bomb bay or missile doors *Armor plating

*Weapons ejection systems

Note that the weapons themselves as well as any ammunition are not normally included in this item.

Appendix A contains data on 'armament' weight for several types of military airplanes.

7.12 WEIGHT ESTIMATION FOR GUNS, LAUNCHERS AND WEAPONS PROVISIONS

For detailed data on guns, lauchers and other military weapons provisions the reader is referred to Part III, Chapter 7.

Note: Ammunition, bombs, missiles, and most types of external stores are normally counted as part of the payload weight, W_{PL} in military airplanes.

7.13 WEIGHT ESTIMATION OF FLIGHT TEST INSTRUMENTATION

During the certification phase of most airplanes a significant amount of flight test instrumentation and associated hardware is carried on board. The magnitude of W_{fti} depends on the type of airplane and the types of flight tests to be performed. Appendix A contains weight data for flight test instrumentation carried on a number of NASA experimental airplanes (Tables A13.1-A13.4).

7.14 WEIGHT ESTIMATION FOR AUXILIARY GEAR

This item encompasses such equipment as:

*fire axes *sextants *unaccounted items

An item referred to as 'manufacturers variation' is sometimes included in this category as well. A safe assumption is to set:

$$W_{aux} = 0.01W_E \tag{7.50}$$

7.15 BALLAST WEIGHT ESTIMATION

When looking over the weight statements for various airplanes in Appendix A, the reader will make the startling discovery that some airplanes carry a

significant amount of ballast. This can have detrimental effects on speed, payload and range performance.

The following reasons can be given for the need to include ballast in an airplane:

1. The designer 'goofed' in the weight and balance calculations

2. To achieve certain aerodynamic advantages it was judged necessary to locate the wing or to size the empennage so that the static margin became insufficient. This problem can be solved with ballast. In this case, carrying ballast may in fact turn out to be advantageous.

3. To achieve flutter stability within the flight envelope ballast weights are sometimes attached to the wing and/or to the empennage.

Note: balance weights associated with flight control surfaces are not counted as ballast weight.

The amount of ballast weight required is determined with the help of the X-plot. Construction and use of the X-plot is discussed in Part II, Chapter 11. The Class II weight and balance method discussed in Chapter 9 of this part may also be helpful in determining the amount of ballast weight required to achieve a certain amount of static margin.

7.16 ESTIMATING WEIGHT OF PAINT

Transport jets and camouflaged military airplanes carry a considerable amount of paint. The amount of paint weight is obviously a function of the extent of surface coverage. For a well painted airplane a reasonable estimate for the weight of paint is:

$$W_{pt} = 0.003 - 0.006W_{TO} \tag{7.51}$$

7.17 ESTIMATING WEIGHT OF W_{etc}

This weight item has been included to cover any items which do not normally fit in any of the previous weight categories.

8. LOCATING COMPONENT CENTERS OF GRAVITY

The purpose of this chapter is to provide guidelines for the determination of the location of centers of gravity for individual airplane components. Knowledge of component c.g. locations is essential in both Class I and Class II weight and balance analyses as discussed in Chapter 10 of Part II and Chapter 4 of this book.

In Part II, Chapter 10, Table 10.2 provides a summary of c.g. locations for the major structural components of the airplane only. In this chapter a slightly more extensive data base is provided. The presentation of component c.g. locations follows the weight breakdowns of Chapters 5-7:

8.1 C.G. Locations of Structural Components
8.2 C.G. Locations of Powerplant Components
8.3 C.G. Locations for Fixed Equipment

8.1 C.G. LOCATIONS OF STRUCTURAL COMPONENTS

Table 8.1 lists the most likely c.g. locations for major structural components. There is no substitute for common sense: if the preliminary structural arrangement of Part III (Step 19 of p.d. sequence 2, Part II) suggests that a given structural component has a different mass distribution than is commonly the case, an 'educated guess' must be made as to the effect on the c.g. of that component.

Example: Looking at the threeview of the GP-180 of Figure 3.47, p.86, Part II it is obvious that there is a concentration of primary structure at the aft end of the fuselage. The fuselage c.g. should therefore not be placed at 38-40 percent of the fuselage length, but probably at 55 to 60 percent.

8.2 C.G. LOCATIONS OF POWERPLANT COMPONENTS

Table 8.2 lists the most likely c.g. locations for powerplant components. Note that for engine c.g. locations manufacturers data should be used. 'Guessing' at engine c.g. locations is not recommended!

8.3 C.G. LOCATIONS OF FIXED EQUIPMENT

Table 8.3 lists guidelines for locating centers of gravity of fixed equipment components.

Table 8.1 Center of Gravity Location of Structural Components

Component:	Center of gravity location:
Wing (half):	<u>Unswept wing:</u> 38-42 percent chord from the L.E. at 40 percent of the semi-span. <u>Swept wing:</u> 70 percent of the distance between the front and rear spar behind the front spar at 35 percent of the semi-span
Horizontal Tail: (half)	Regardless of sweep angle: 42 percent chord from the L.E. at 38 percent of the semi-span.
Vertical Tail: (low tail)	Regardless of sweep angle: 42 percent chord from the L.E. at 38 percent vertical tail span from the root chord.
Vertical Tail: (T-tail) root	Regardless of sweep angle: 42 percent chord from the L.E. at 55 percent vertical tail span from the root chord.
Vertical Tail: (cruciform)	Regardless of sweep angle: 42 percent chord from the L.E. at between 38 and 55 percent vertical tail span from the root chord. Interpolate according to z_h/b_v.
Fuselage: <u>Caution:</u> Do not count the propeller spinner in fuselage or nacelle length!	<u>Distances are given as a fraction of the fuselage length:</u> Single engine tractors: 0.32-0.35 Single engine pusher: 0.45-0.48 Propeller driven twins: 0.38-0.40 (tractors on wing) Propeller driven twins: 0.50-0.53 (pushers on wing) Jet transports: 0.42-0.45 (wing mounted engines) Jet transports: 0.47-0.50 (rear fuselage mounted engines) Fighters: 0.45 (engines buried in the fuselage)
Tail booms:	0.40-0.45 of boom length starting from most forward structural attachment of the boom.
Nacelles:	0.40 of nacelle length from nacelle nose
Landing gear:	at 0.50 of strutlength for gears with mostly vertical struts

Table 8.2 Center of Gravity Location of Powerplant

Components

Component:	Center of Gravity Location:
Engine(s)	Use manufacturers data
Air induction system	Use the c.g. of the gross shell area of the inlets
Propellers	On the spin axis, in the propeller spin plane
Fuel system	Refer to the fuel system layout diagram required as part of Step 17 in p.d. sequence II, Part II, p.18.
Filled fuel tank	Assuming a prismoidal shape (See figure left), the c.g. is located relative to plane S_1 at:

$$l_{cg} = (1/4)(S_1 + 3S_2 + 2(S_1S_2)^{1/2})/(S_1 + S_2 + (S_1S_2)^{1/2}) \quad (8.1)$$

Trapped fuel and oil	Trapped fuel is normally located at the bottom of fuel tanks and fuel lines. Trapped oil is normally located close to the engine case.
Propulsion system	Make a list of which items contribute to the propulsion system weight and 'guestimate' their c.g. location by referring to the powerplant installation drawing required in Step 5.10, pages 133 and 134 in Part II.

Table 8.3 Center of Gravity Location of Fixed Equipment

Component:	Center of Gravity Location:
Flight Control System Hydraulic and Pneumatic System Electrical System Instrumentation, Avionics and Electronics Air-conditioning, Pressurization, Anti-icing and de-icing System Oxygen System	Note: for all systems, the c.g. location can be most closely 'guestimated' by referring to the system lay-out diagrams described in Part IV of this text. These system lay-outs were required as part of Step 17 in p.d. sequence 2, Part II, p.18.
Auxiliary Power Unit	See engine manufacturer data.
Furnishings Baggage and Cargo Handling Equipment	Refer to the fuselage internal arrangement drawing required by Steps 4.1 and 4.2 in Part II, pp 107 and 108.
Operational items	See furnishings
Armament	This item is normally close to the cockpit
Guns, launchers and weapons provisions	From manufacturer data.
Flight test instrumentation	A sketch depicting the locations of sensors, recorders operating systems should help in locating the overall c.g. of this item.
Auxiliary gear	Make a list of items in this category and 'guestimate' their c.g. locations.
Ballast	Ballast weights are normally made from lead. Ballast c.g. is thus easily located.
Paint	Centroid of painted areas.

9. CLASS II WEIGHT AND BALANCE ANALYSIS

The basic method used in performing a Class II weight and balance analysis is identical to that used for the Class I weight and balance analysis. The latter was discussed in detail in Part II, Chapter 10. The only difference is, that a more detailed weight statement is used: the Class II weight prediction method of Chapters 4-7 is used.

During this stage of the preliminary design frequent questions which are raised, are:

1. How much does the overall airplane c.g. move as a result of moving some component?

2. How much does the airplane static margin change as a result of moving the wing?

These questions are answered in Sections 9.1 and 9.2 respectively.

9.1 EFFECT OF MOVING COMPONENTS ON OVERALL AIRPLANE CENTER OF GRAVITY

Figure 9.1 illustrates an airplane, its c.g. location and the c.g. location of component i. The overall center of gravity of the airplane is found from:

$$x_{cg} = (\text{Sum}_{i=1}^{i=n} W_i x_i)/(\text{Sum } W_i) \qquad (9.1)$$

Evidently:

$$\text{Sum}_{i-1}^{i=n} W_i = W \qquad (9.2)$$

The rate at which overall airplane c.g. moves, when a component i is moved, can be found by differentiation of Eqn.(9.1):

$$\partial x_{cg}/\partial x_i = (W_i)/(\text{Sum}_{i=1}^{i=n} W_i) \qquad (9.3)$$

If component i is moved over a distance Δx_i, the

Figure 9.1 Definition of Overall C.G. Location and of Component C.G. Location

Figure 9.2 Definition of Mean Geometric Chord ($\bar{c}$) Location and Overall C.G. Location

overall airplane c.g. moves over a distance given by:

$$\Delta x_{cg} = (\Delta x_i)(W_i)/(\text{Sum}_{i=1}^{i=n} W_i) \tag{9.4}$$

Equation (9.4) suggests that to move the overall c.g. of the airplane significantly, either a heavy weight component can be moved a small distance or a light weight component can be moved a large distance.

Items which are frequently moved about to achieve satisfactory weight and balance results are: batteries, air-conditioner units, certain 'black' boxes and sometimes just plain ballast. The reader will note from the detailed weight statements in Appendix A that several airplanes carry a relatively large amount of ballast.

9.2 EFFECT OF MOVING THE WING ON OVERALL AIRPLANE CENTER OF GRAVITY AND ON OVERALL AIRPLANE AERODYNAMIC CENTER

Figure 9.2 illustrates an airplane, its mean geometric chord location and its overall c.g. location.

If the leading edge of the mgc of the wing is at fuselage station (FS) x_{LE}, the airplane c.g. in terms of the wing mgc can be written as:

$$\bar{x}_{cg} = (x_{cg} - x_{LE})/\bar{c} \tag{9.5}$$

When a component i is moved over a distance Δx_i, the overall airplane c.g. moves relative to the wing mgc as:

$$\Delta\bar{x}_{cg} = (\Delta x_i)(W_i)/\bar{c}(\text{Sum}_{i=1}^{i=n} W_i) \tag{9.6}$$

For a conventional, tail-aft airplane, its aerodynamic center location can be written as:

$$\bar{x}_{ac} = \{C_1 + C_2(\bar{x}_{ac_h})\}/(1 + C_2) \tag{9.7}$$

$$\text{where: } C_1 = \bar{x}_{ac_w} + \Delta\bar{x}_{ac_{wb}} \tag{9.8}$$

$$C_2 = (C_{L_{\alpha_h}}/C_{L_{\alpha_{wb}}})(1 - d\epsilon/d\alpha)(S_h/S) \tag{9.9}$$

A detailed derivation of Eqn. (9.7) may be found in Reference 19, p.133.

Part VI contains methods for computing the liftcurve slopes and aerodynamic centers which appear in C_1 and in C_2.

<u>Warning:</u> the wing-body aerodynamic center shift, $\Delta\bar{x}_{ac_{wb}}$ in Eqn.(9.8) is always a negative number: it shifts the a.c. forward!

If the wing is moved aft over a distance Δx_w, the overall airplane c.g. is:

$$\bar{x}_{cg_{new}} = \bar{x}_{cg_{old}} + (\Delta x_w W_w)/(\bar{c}(\text{Sum}_{i=1}^{i=n} W_i) \qquad (9.10)$$

The new a.c. location can now be written as:

$$\bar{x}_{ac_{new}} = \{C_1 + C_2(\bar{x}_{ac_h} - \Delta x_w/\bar{c})\}/(1 + C_2) \qquad (9.11)$$

Equations (9.10) and (9.11) can be used to 'redo' the X-plot of Part II, Chapter 11. This 'redone' X-plot in turn is used to:

1. determine how much the horizontal tail area must be changed as a result of moving the wing

or:

2. determine which other weight components need to be moved and by how much. to maintain some desired level of stability (or instability as the case may be).

For a canard airplane and/or for a three surface airplane similar equations are easily derived. The reader should consult Equations (11.1) and (11.2) in Part II for guidance.

10. CLASS II METHOD FOR ESTIMATING AIRPLANE INERTIAS

The purpose of this chapter is to provide an outline for a Class II method for estimating moments and products of inertia. It will be assumed that the Class II weight estimating method of Chapter 4 has been applied: a rather detailed weight and c.g. breakdown for the airplane is therefore presumed to be available.

The following equations are a slight modification of the general inertia equations 2.22a through 2.22c in Reference 19.

$$I_{xx} = \sum_{i=1}^{i=n} m_i\{(y_i - y_{cg})^2 + (z_i - z_{cg})^2\} \qquad (10.1)$$

$$I_{yy} = \sum_{i=1}^{i=n} m_i\{(z_i - z_{cg})^2 + (x_i - x_{cg})^2\} \qquad (10.2)$$

$$I_{zz} = \sum_{i=1}^{i=n} m_i\{(x_i - x_{cg})^2 + (y_i - y_{cg})^2\} \qquad (10.3)$$

$$I_{xy} = \sum_{i=1}^{i=n} m_i(x_i - x_{cg})(y_i - y_{cg}) \qquad (10.4)$$

$$I_{yz} = \sum_{i=1}^{i=n} m_i(y_i - y_{cg})(z_i - z_{cg}) \qquad (10.5)$$

$$I_{zx} = \sum_{i=1}^{i=n} m_i(z_i - z_{cg})(x_i - x_{cg}) \qquad (10.6)$$

Figure 10.1 defines the coordinates used in these equations.

The reader should recall that for a symmetrical airplane the inertia products I_{xy} and I_{yz} are zero.

Equations (10.1) through (10.6) are valid whenever the weight breakdown contains a 'sufficiently' large number of parts so that the inertia moment and/or product of each part about its own c.g. location is negligible.

Whenever the latter assumption is not satisfied,

equations (10.1) through (10.6) should be all modified as follows:

$$I_{xx} = \sum_{i=1}^{i=n} I_{xx_i} + \sum_{i=1}^{i=n} m_i((y_i - y_{cg})^2 + (z_i - z_{cg})^2) \quad (10.7)$$

The first term in Eqn.(10.7) represents the moment (or product) of inertia of component i about its own center of gravity.

Moments (and products) of inertia of airplane components about their own center of gravity can be computed in a relatively straightforward manner by assuming uniform mass distributions for structural components and by using the 'lumped mass' assumption for distributed systems. An example of the latter would be the airplane fuel system. Major fuel system components such as pumps, bladders and the like can be considered to be concentrated masses distributed around the fuel system c.g.. Equations (10.1) through (10.2) are then used to compute the moments of inertia of the fuel system about its own c.g.

Figure 10.1 Definition of Coordinates

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APPENDIX A: DATA SOURCE FOR AIRPLANE COMPONENT WEIGHTS AND FOR WEIGHT FRACTIONS

Tables A1 through A13 present component weight data for the following types of airplanes:

1. Homebuilt propeller driven airplanes: Tables A1.1.

2. Single engine propeller driven airplanes: Tables A2.1 and A2.2.

3. Twin engine propeller driven airplanes: Tables A3.1 and A3.2.

4. Agricultural airplanes: Tables A4.1. At the time of printing no data were available. The reader should use Tables A2 and add spray equipment weights.

5. Business Jets: Tables A5.1 and A5.2.

6. Regional turbopropeller airplanes: Tables A6.1 through A6.3.
   Regional piston/propeller airplanes: Tables A6.4.

7. Jet transports: Tables A7.1 through A7.5.
   Turbopropeller driven transports: Tables A7.6.

8. Military trainers: Tables A8.1.

9. Fighters: Tables A9.1 through A9.5.

10. Military jet transports: Tables A.10.1.
    Military turbopropeller driven transports: Tables A10.2.
    Military piston/propeller driven transports: Table A10.3 and A10.4.
    Military patrol airplanes: Tables A10.5.

11. Flying boats, amphibious and float airplanes: Tables A11.1. At the time of printing no data were available. The reader should use suitable tables in categories 1-10 and account for hull weight with the method of Chapter 4.

12. Supersonic cruise airplanes: Tables A12.1.

13. NACA and NASA X (experimental) airplanes: Tables A13.1 through A13.4.
    <u>CAUTION:</u> most of the X airplanes were built for experimental purposes only. They should not be regarded as 'optimized' for a given mission.

Table A1.1a Group Weight Data for Homebuilt Propeller Driven Airplanes

Type	Bede BD5B	At the time of printing no other data were available
Weight Item, lbs		
Wing Group	87	
Empennage Group	17	
Fuselage Group	89	
Nacelle Group	0	
Landing Gear Group	32	
Nose Gear	10	
Main Gear	22	
Structure Total	225	
Engine	146	
Air Induct. System	0	
Fuel System	25	
Propeller Install.	5	
Thrust Attenuator	3	
Engine Install.	10	
Power Plant Total	189	
Avionics + Instrum.	15	
Surface Controls	0	
Electrical System	10	
Electronics	0	
Ballast	30	
Parachute	20	
Furnishings + paint	50	
Auxiliary Gear	0	
Fixed Equipm't Total	125	
$W_{oil} + W_{tof}$	2	
Fuel	340	
Payload(pilot)	170	

Table A1.1b Group Weight Data for Homebuilt Propeller Driven Airplanes

Type	Bede BD5B	At the time of printing no other data were available
Flight Design Gross Weight, GW, lbs	1,051	
Structure/GW	0.214	
Power Plant/GW	0.180	
Fixed Equipm't/GW	0.119	
Empty Weight/GW	0.513	
Wing Group/GW	0.083	
Empenn. Group/GW	0.016	
Fuselage Group/GW	0.085	
Nacelle Group/GW	0.000	
Land. Gear Group/GW	0.030	
Take-off Gross Wht, W_{TO}, lbs	1,051	
Empty Weight, W_E, lbs	539	
Wing Group/S, psf	1.8	
Emp. Grp/S_{emp}, psf	1.1	
Ultimate Load Factor, g's	5.7 assumed	
Surface Areas, ft^2		
Wing, S	47.4	
Horiz. Tail, S_h	10.5	
Vert. Tail, S_v	5.0	
Empenn. Area, S_{emp}	15.5	

Table A2.1a Group Weight Data for Single Engine Propeller Driven Airplanes

Type	Cessna 150	172	175	180 **	182	L-19A* **
Weight Item, lbs						
Wing Group	216	226	227	235	235	238
Empennage Group	36	57	57	62	62	64
Fuselage Group	231	353	351	404	400	216
Nacelle Group	22	27	30	32	34	33
Landing Gear Group	104	111	111	112	132	135
Nose Gear						
Main Gear						
Structure Total	609	774	776	845	863	686
Engine	197	254	318	417	417	399
Air Induct. System	2	1	3	1	1	4
Fuel System	17	21	26	26	26	39
Propeller Install.	22	33	33	64	64	46
Engine Install.	28	36	36	37	37	62
Power Plant Total	267	345	416	545	545	550
Avionics + Instrum.	3	4	4	6	6	36
Surface Controls	31	31	31	36	36	47
Electrical System	34	38	38	43	43	86
Electronics	0	0	0	0	0	39
Air Cond. System	1	1	1	1	1	9
Anti-icing System	33	85	85	87	87	65
Furnishings	0	0	0	0	0	3
Auxiliary Gear						
Fixed Equipm't Total	102	159	159	173	173	285
$W_{oil} + W_{tof}$	11	15	19	22	22	19
Fuel	156	252	312	390	390	252
Payload	398	702	719	734	715	321

*Military observation airplane
**Taildragger

Table A2.1b Group Weight Data for Single Engine Propeller Driven Airplanes

Type	Cessna 150	172	175	180 **	182	L-19A* **
Flight Design Gross Weight, GW, lbs	1,500	2,200	2,350	2,650	2,650	2,100
Structure/GW	0.406	0.352	0.330	0.319	0.326	0.327
Power Plant/GW	0.178	0.157	0.177	0.206	0.206	0.262
Fixed Equipm't/GW	0.068	0.072	0.068	0.065	0.065	0.136
Empty Weight/GW	0.631	0.565	0.561	0.576	0.583	0.727
Wing Group/GW	0.144	0.103	0.097	0.089	0.089	0.113
Empenn. Group/GW	0.024	0.026	0.024	0.023	0.023	0.030
Fuselage Group/GW	0.154	0.160	0.149	0.152	0.151	0.103
Nacelle Group/GW	0.015	0.012	0.013	0.012	0.013	0.016
Land. Gear Group/GW	0.069	0.050	0.047	0.042	0.050	0.064
Take-off Gross Wht, W_{TO}, lbs	1,500	2,200	2,350	2,650	2,650	2,100
Empty Weight, W_E, lbs	946	1,243	1,319	1,526	1,545	1,527
Wing Group/S, psf	1.4	1.4	1.3	1.3	1.3	1.4
Emp. Grp/S_{emp}, psf	0.85	1.1	1.1	1.2	1.2	1.2
Ultimate Load Factor, g's	5.7	5.7	5.7	5.7	5.7	5.7
Surface Areas, ft^2						
Wing, S	160	175	175	175	175	174
Horiz. Tail, S_h	28.5	34.6	34.6	34.6	34.1	35.2
Vert. Tail, S_v	14.1	18.4	18.4	18.4	18.4	18.4
Empenn. Area, S_{emp}	42.6	53.0	53.0	53.0	52.5	53.6

*Military observation airplane
**Taildragger

Table A2.2a Group Weight Data for Single Engine Propeller Driven Airplanes

Type	Cessna 210A	Beech J-35	Saab Safir	Rockwell 112TCA	Cessna 210J
Weight Item, lbs.					
Wing Group	261	379	276	334	335
Empennage Group	71	58	60	98	86
Fuselage Group	316*	200	386	358	408*
Nacelle Group	31	62	in fus.	61	28
Landing Gear Group	207	205	119	161	191
Nose Gear				35	50
Main Gear				126	141
Structure Total	886	904	841	1,082	1,048
Engine	390	432		475	450
Air Induct. System		3			7
Fuel System		30		17	24
Propeller Install.		73		in eng.	64
Engine Install.		45		65	36
Power Plant Total	577	583		557	581
Avionics + Instrum.	16	16		64	18
Surface Controls	44	56	in fus.	44	48
Hydraulic System	4			10	51
Electrical System	60	72		81	57
Air Cond. System	12	12		in misc.	10
Anti-icing System					
Furnishings	116	174		179	130
Oxygen System	0	0	0	20	0
Ballast	0	0	0	21	0
Auxiliary Gear	0	4	0	2	0
Misc. Equipment	20	0	0	24	0
Paint					21
Fixed Equipm't Total	272	334		445	335
$W_{oil} + W_{tof}$		11		31	24
Fuel (max. payload)		234		230	464**
Payload		845		740	693

*Includes wing-fuselage carry-through spars
**Maximum fuel

Table A2.2b Group Weight Data for Single Engine Propeller Driven Airplanes

Type	Cessna 210A	Beech J-35	Saab Safir	Rockwell 112TCA	Cessna 210J
Flight Design Gross Weight, GW, lbs	2,900	2,900	2,660	2,954	3,400
Structure/GW	0.306	0.312	0.316	0.366	0.308
Power Plant/GW	0.199	0.201		0.189	0.171
Fixed Equipm't/GW	0.094	0.115		0.151	0.099
Empty Weight/GW	0.598	0.628	0.620	0.705	0.578
Wing Group/GW	0.090	0.131	0.104	0.113	0.099
Empenn. Group/GW	0.024	0.020	0.023	0.033	0.025
Fuselage Group/GW	0.109	0.069	0.145	0.121	0.120
Nacelle Group/GW	0.011	0.021		0.021	0.008
Land. Gear Group/GW	0.071	0.071	0.045	0.055	0.056
Take-off Gross Wht, W_{TO}, lbs	2,900	2,900	2,660	2,954	3,400
Empty Weight, W_E, lbs	1,735	1,821	1,650	2,084	1,964
Wing Group/S, psf	1.5	2.1	1.9	2.2	1.9
Emp. Grp/S_{emp}, psf	1.3	1.6	1.4	2.0	1.5
Ultimate Load Factor, g's	5.7				5.7
Surface Areas, ft^2					
Wing, S	176	178	146	152	176
Horiz. Tail, S_h	38.6	*	27.6**	32.0	38.6
Vert. Tail, S_v	17.2	*	14.3**	17.0	17.2
Empenn. Area, S_{emp}	55.8	35.8	41.9	49.0	55.8

*V-tail
**Estimated

Table A3.1a Group Weight Data for Twin Engine Propeller Driven Airplanes

Type	Beech 65 QA*	E-18S	G-50 TB*	95 TA*	Cessna 310C
Number of engines:	2	2	2	2	2
Weight Item, lbs					
Wing Group	670	874	656	458	453
Empennage Group	153	180	156	79	118
Fuselage Group	601	768	495	276	319
Nacelle Group	285	331	261	180	129
Landing Gear Group	444	585**	447	218	263
Nose Gear					
Main Gear					
Structure Total	2,153	2,738	2,015	1,211	1,282
Engines	1,008	1,352	1,008	519	852
Air Induct. System	27	149	27	8	7
Fuel System	137	274	137	83	76
Propeller Install.	258	334	258	162	162
Engine Install.	180	172	172	101	153
Power Plant Total	1,610	2,281	1,610	873	1,250
Avionics + Instrum.	70	100	80	49	46
Surface Controls	132	115	120	73	66
Electrical System	166	295	184	96	121
Electronics	2	63	9	26	0
Air Cond. System	90	144	81	48	46
Anti-icing System					
Furnishings	438	524	333	194	154
Auxiliary Gear	5	0	7	0	65
Fixed Equipm't Total	903	1,241	814	486	498
$W_{oil} + W_{tfo}$	60	128	60	30	45
Fuel	1,380	1,908	1,380	672	612
Payload	1,287	1,474	1,311	733	1,186

*QA = Queen Air, TB = Twin Bonanza, TA = Travel Air
**Taildragger

Table A3.1b Group Weight Data for Twin Engine Propeller Driven Airplanes

Type	Beech 65 QA*	E-18S**	G-50 TB*	95 TA*	Cessna 310C
Flight Design Gross Weight, GW, lbs	7,368	9,700	7,150	4,000	4,830
Structure/GW	0.292	0.282	0.282	0.303	0.265
Power Plant/GW	0.219	0.235	0.225	0.218	0.259
Fixed Equipm't/GW	0.123	0.128	0.114	0.122	0.103
Empty Weight/GW	0.638	0.651	0.624	0.649	0.628
Wing Group/GW	0.091	0.090	0.092	0.115	0.094
Empenn. Group/GW	0.021	0.019	0.022	0.020	0.024
Fuselage Group/GW	0.082	0.079	0.069	0.069	0.066
Nacelle Group/GW	0.039	0.034	0.037	0.045	0.027
Land. Gear Group/GW	0.060	0.060	0.063	0.055	0.054
Take-off Gross Wht, W_{TO}, lbs	7,368	9,700	7,150	4,000	4,830
Empty Weight, W_E, lbs	4,701	6,318	4,459	2,595	3,032
Wing Group/S, psf	2.4	2.4	2.4	2.4	2.6
Emp. Grp/S_{emp}, psf	1.4	1.7	1.4	1.2	1.5
Ultimate Load Factor, g's	6.6		7.1		5.7
Surface Areas, ft^2					
Wing, S	277	361	277	194	175
Horiz. Tail, S_h	79.3	71.6	79.3	42.4	54.3
Vert. Tail, S_v	30.8	33.6	30.8	23.3	25.9
Empenn. Area, S_{emp}	110	105	110	65.7	80.2

*QA = Queen Air, TB = Twin Bonanza, TA = Travel Air
**Taildragger

Table A3.2a Group Weight Data for Twin Engine Propeller Driven Airplanes

Type*	Cessna 404-3	414A	TP-441	Rockwell 690B
Number of engines:	2	2	2	2
Weight Item, lbs	(PP)	(PP)	(PP)	(TBP)
Wing Group	860	638	873	1,001
Empennage Group	181	160	233	207
Fuselage Group	610	678	873	1,377
Nacelle Group	284	200	258	in prop/eng
Land. Gear Group	316	303	346	437
Nose Gear	67	75	69	53
Main Gear	249	228	277	384
Structure Total	2,251	1,979	2,583	3,022
Engines	1,000	862	745	720
Air Induct. System	23	36	0	17
Fuel System	107	96	93	180
Propeller Install.	215	165	302	758
Engine Install.	281	240	130	
Power Plant Total	1,626	1,399	1,270	1,675
Avionics + Instrum.	311	334	250	344
Hydraulic System	52	14	49	99
Surface Controls	113	107	223	81
Electrical System	169	157	403	379
Electronics	1	1	150	0
Oxygen System	0	0	0	23
Air Cond. System	49	130*	182*	205*
Anti-icing System	11	3	78	84
Furnishings	370	342	538	612
Auxiliary Gear	5	3	7	41**
Paint	48	42	48	40
Fixed Equipm't Total	1,129	1,133	1,928	1,908
$W_{oil} + W_{tfo}$	116	113	98	65
Fuel	1,379	961	2,446	1,575
Payload***	1,900	1,200	1,600	1,960

*Includes pressurization system
**This is all ballast in this model
***Includes a crew of two

Table A3.2b Group Weight Data for Twin Engine Propeller Driven Airplanes

Type	Cessna 404-3 (PP)	414A (PP)	TP-441 (TBP)	Rockwell 690B (TBP)
Flight Design Gross Weight, GW, lbs	8,400	6,785	9,925	10,205
Structure/GW	0.268	0.292	0.260	0.296
Power Plant/GW	0.194	0.206	0.128	0.164
Fixed Equipm't/GW	0.134	0.167	0.194	0.187
Empty Weight/GW	0.596	0.665	0.582	0.647
Wing Group/GW	0.102	0.094	0.088	0.098
Empenn. Group/GW	0.022	0.024	0.023	0.020
Fuselage Group/GW	0.073	0.100	0.088	0.135
Nacelle Group/GW	0.034	0.029	0.026	
Land. Gear Group/GW	0.038	0.045	0.035	0.043
Take-off Gross Wht, W_{TO}, lbs	8,400	6,785	9,925	10,205
Empty Weight, W_E, lbs	5,006	4,511	5,781	6,605
Wing Group/S, psf	3.6	2.8	3.4	3.8
Emp. Grp/S_{emp}, psf	1.7	1.6	2.2	2.0
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	242	226	254	266
Horiz. Tail, S_h	63.4	60.7	63.4	58.4
Vert. Tail, S_v	43.5	41.2	43.5	44.8
Empenn. Area, S_{emp}	107	102	107	103

*Assumed

Table A4.1a Group Weight Data for Agricultural Airplanes

Type	At the time of printing no data were available
Number of engines:	
Weight Item, lbs	
Wing Group	
Empennage Group	
Fuselage Group	
Nacelle Group	
Landing Gear Group	
Nose Gear	
Main Gear	
Structure Total	
Engines	
Air Induct. System	
Fuel System	
Propulsion System.	
Power Plant Total	
Spray equipment	
Avionics + Instrum.	
Surface Controls	
Hydraulic System	
Pneumatic System	
Electrical System	
Electronics	
Oxygen System	
Air Cond. System**	
Furnishings	
Auxiliary Gear	
Miscellaneous	
Fixed Equipm't Total	
$W_{oil} + W_{tfo}$	
Max. Fuel Capacity	
Max. Payload	

Table A4.1b Group Weight Data for Agricultural Airplanes

Type	At the time of printing no data were available
Flight Design Gross Weight, GW, lbs	
Structure/GW	
Power Plant/GW	
Fixed Equipm't/GW	
Empty Weight/GW	
Wing Group/GW	
Empenn. Group/GW	
Fuselage Group/GW	
Nacelle Group/GW	
Land. Gear Group/GW	
Take-off Gross Wht, W_{TO}, lbs	
Empty Weight, W_E, lbs	
Wing Group/S, psf	
Emp. Grp/S_{emp}, psf	
Ultimate Load Factor, g's	
Surface Areas, ft^2	
Wing, S	
Horiz. Tail, S_h	
Vert. Tail, S_v	
Empenn. Area, S_{emp}	

Table A5.1a Group Weight Data for Business Jets

Type	MS-760 Paris	Lockheed Jetstar	Gates-Learjet 25D	28
Number of engines:	2	2	2	2
Weight Item, lbs				
Wing Group	897	2,827	1,467	1,939
Empennage Group	176	879	361	361
Fuselage Group	912	3,491	1,575	1,624
Nacelle Group	49*	792	241	214
Landing Gear Group	307	1,061	584	584
Nose Gear			102	102
Main Gear			482	482
Structure Total	2,341	9,050	4,228	4,722
Engines	609	1,750	792	792
Air Induct. System	31	135	0	0
Fuel System	240	360	179	237
Propulsion System.	136	230	255	255
Power Plant Total	1,016	2,475	1,226	1,284
Avionics + Instrum.	70	153	383	383
Surface Controls	188	768	291	275
Hydraulic System		262	119	114
Pneumatic System				
Electrical System	284	973	620	603
Electronics	158	868	0	0
Oxygen System			28	26
Air Cond. System**	48	510	293	285
Anti-icing System			82	162
Furnishings	169	1,521	720	768
Auxiliary Gear	0	10	0	0
Miscellaneous	0	0	-40	-11
Fixed Equipm't Total	917	5,065	2,496	2,605
$W_{oil} + W_{tfo}$	28	204	177	
Max. Fuel Capacity	2,460	11,229	6,098	4,684
Max. Payload	884	2,100	2,980	1,962

*Engines buried inside the fuselage
**Includes pressurization system

Table A5.1b Group Weight Data for Business Jets

Type	MS-760 Paris	Lockheed Jetstar	Gates-Learjet 25D	28
Flight Design Gross Weight, GW, lbs	7,650	30,680	15,000	15,000
Structure/GW	0.306	0.295	0.282	0.315
Power Plant/GW	0.133	0.081	0.082	0.086
Fixed Equipm't/GW	0.120	0.165	0.166	0.174
Empty Weight/GW*	0.563	0.541	0.530	0.574
Wing Group/GW	0.117	0.092	0.098	0.129
Empenn. Group/GW	0.023	0.029	0.024	0.024
Fuselage Group/GW	0.119	0.114	0.105	0.108
Nacelle Group/GW	0.006*	0.026	0.016	0.014
Land. Gear Group/GW	0.040	0.035	0.039	0.039
Take-off Gross Wht, W_{TO}, lbs	7,650	30,680	15,000	15,000
Empty Weight, W_E, lbs	4,306	16,590	7,950	8,611
Wing Group/S, psf	4.6	5.4	6.3	7.3
Emp. Grp/S_{emp}, psf	3.5	3.4	3.9	3.9
Ultimate Load Factor, g's	3.75**	5.25	3.75**	3.75**
Surface Areas, ft^2				
Wing, S	194	521	232	265
Horiz. Tail, S_h	31.8	149	54.0	54.0
Vert. Tail, S_v	18.4	110	37.4	37.4
Empenn. Area, S_{emp}	50.2	259	91.4	91.4

*Engines buried inside the fuselage
**Assumed

Table A5.2a Group Weight Data for Business Jets

Type	Cessna Citation II	Rockwell JC-1121	Hawker-Siddeley 125	Gulfstr. American GII
Number of engines:	2	2	2	2
Weight Item, lbs				
Wing Group	1,288	1,322	1,968	6,372
Empenn. Group	295	425	608	1,965
Fuselage Group	1,069	1,622	1,628	5,944
Nacelle Group	220	350	in fusel.	1,239
Land. Gear Group	465	443	659	2,011
Nose Gear	87			321
Main Gear	378			1,690
Structure Total	3,337	4,162	4,863	17,531
Engine(s)	1,100			6,570
Air Induct. System	26			
Exhaust System	15			
Fuel System	189			316
Propulsion System	105			
Power Plant Total	1,435			6,886
Avionics + Instrum.	87			1,715
Surface Controls	203	223	217	1,021
Hydraulic System	96			959
Electrical System	340			1,682
Electronics	313			
Oxygen System				140
Air Cond. System*	264			927
Anti-icing System	98			
Furnishings	800			4,501
Auxiliary Gear	3			
Auxiliary power unit				258
Paint	47			
Fixed Equipm't Total	2,251			11,203
$W_{oil} + W_{tfo}$	143			
Max. Fuel Capacity	5,009	8,964	9,193	23,300
Max. Payload			1,905	5,380

*Includes pressurization system

Table A5.2b Group Weight Data for Business Jets

Type	Cessna Citation II	Rockwell JC-1121	Hawker Siddeley 125	Gulfstr. American GII
Flight Design Gross Weight, GW, lbs	13,500	20,500	23,300	64,800
Structure/GW	0.247	0.203	0.209	0.271
Power Plant/GW	0.106			0.106
Fixed Equipm't/GW	0.167			0.173
Empty Weight/GW	0.520	0.540	0.526	0.550*
Wing Group/GW	0.095	0.064	0.084	0.098
Empenn. Group/GW	0.022	0.021	0.026	0.030
Fuselage Group/GW	0.079	0.079	0.070	0.092
Nacelle Group/GW	0.016	0.017	in fusel.	0.019
Land. Gear Group/GW	0.034	0.022	0.028	0.031
Take-off Gross Wht, W_{TO}, lbs	13,500	20,500	23,300	64,800
Empty Weight, W_E, lbs	7,023	11,070	12,260	35,620
Wing Group/S, psf	4.6	4.4		8.0
Emp. Grp/S_{emp}, psf	2.4	3.3		5.8
Ultimate Load Factor, g's	3.75**			
Surface Areas, ft^2				
Wing, S	279	303	353	794
Horiz. Tail, S_h	70.6	70	100	182
Vert. Tail, S_v	50.9	59.3	51.6	155
Empenn. Area, S_{emp}	122	129	152	337

*Typical. Individual airplanes will vary. **Assumed

Table A6.1a Group Weight Data for Regional Turbopropeller Driven Airplanes

Type	Grumman G-I	Fokker F-27-100	Nord 262	Embraer 110-P2
Number of engines:	2	2	2	2
Weight Item, lbs				
Wing Group	3,735	4,408	2,698	1,502
Empennage Group	874	977	805	454
Fuselage Group	3,718	4,122	3,675	1,354
Nacelle Group	1,136	628	236	198
Land. Gear Group	1,207	1,940	1,085	538
Nose Gear	219			
Main Gear	988			
Structure Total	10,670	12,075	8,499	4,046
Engines	2,688	2,427		622
Air Induct. System				
Fuel System	133	390		86
Propeller Install.	1,002	918		1,140
Propulsion System	698	612		in prop.
Power Plant Total	4,521	4,347		1,848
Avionics + Instrum.	97	81	133	364
Surface Controls	461	613	408	342
Hydraulic System	235	242	(incl.in electr.)	176
Pneumatic System				
Electrical System	966	835	765	452
Electronics	99	386	238	in avion.
APU	355	0	0	0
Air Cond. System	755*	1,225*	527*	192
Anti-icing System				73
Furnishings	415	2,291	1,324	882
Auxiliary Gear	6		33	
Fixed Equipm't Total	3,389	5,673	3,428	2,481
$W_{oil} + W_{tfo}$	329			
Max. Fuel Capacity	10,447	9,198	3,559	3,062
Maximum Payload	4,270	12,500	6,175	3,706

*Includes pressurization system

Table A6.1b Group Weight Data for Regional Turbopropeller Driven Airplanes

Type	Grumman G-I	Fokker F-27-100	Nord 262	Embraer 110-P2
Flight Design Gross Weight, GW, lbs	35,100	37,500	22,930	12,500
Structure/GW	0.304	0.322	0.371	0.324
Power Plant/GW	0.129	0.116		0.148
Fixed Equipm't/GW	0.097	0.151	0.149	0.198
Empty Weight/GW	0.624	0.615	0.663	0.670
Wing Group/GW	0.106	0.118	0.118	0.120
Empenn. Group/GW	0.025	0.026	0.035	0.036
Fuselage Group/GW	0.106	0.110	0.160	0.108
Nacelle Group/GW	0.032	0.017	0.010	0.016
Land. Gear Group/GW	0.034	0.052	0.047	0.043
Take-off Gross Wht, W_{TO}, lbs	35,100	37,500	22,930	12,500
Empty Weight, W_E, lbs	21,900	23,054	15,200	8,375
Wing Group/S, psf	6.1	5.8	4.6	4.8
Emp. Grp/S_{emp}, psf	3.6	3.0	2.9	2.8
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	610	754	592	313
Horiz. Tail, S_h	127	172	169	105
Vert. Tail, S_v	117	153	109	59
Empenn. Area, S_{emp}	244	325	278	164

*Assumed

Table A6.2a Group Weight Data for Regional Turbopropeller Driven Airplanes

Type	Fokker F-27-200	F-27-500	Short* Skyvan
Number of engines:	2	2	2
Weight Item, lbs			
Wing Group	4,505	4,510	1,220
Empennage Group	1,053	1,060	374
Fuselage Group	4,303	5,142	2,154
Nacelle Group	667	668	254
Land. Gear Group	1,825	1,865	466
Nose Gear			
Main Gear			
Structure Total	12,353	13,245	4,468
Engines			714
Air Induct. System			
Fuel System			373
Propeller Install.			368
Propulsion System			87
Power Plant Total			1,542
Avionics + Instrum.		126	74
Surface Controls	620	626	265
Hydraulic System		256	64
Pneumatic System			
Electrical System		840	320
Electronics		329	12
APU		0	
Air Cond. System		1,257**	85
Anti-icing System			
Furnishings		3,035	135***
Auxiliary Gear		0	43
Paint			75
Fixed Equipm't Total		6,469	1,073
$W_{oil} + W_{tfo}$			44
Max. Fuel Capacity	9,146	9,146	4,924
Payload (Max.)	12,615	12,383	

*Strutbraced wing **Includes pressurization
***Cockpit furnishings only

Table A6.2b Group Weight Data for Regional Turbopropeller Driven Airplanes

Type	Fokker F-27-200	F-27-500	Short* Skyvan
Flight Design Gross Weight, GW, lbs	43,500	45,000	12,500
Structure/GW	0.284	0.294	0.357
Power Plant/GW			0.123
Fixed Equipm't/GW		0.144	0.086
Empty Weight/GW	0.537	0.548	0.570
Wing Group/GW	0.104	0.100	0.098
Empenn. Group/GW	0.024	0.024	0.030
Fuselage Group/GW	0.099	0.114	0.172
Nacelle Group/GW	0.015	0.015	0.020
Land. Gear Group/GW	0.042	0.041	0.037
Take-off Gross Wht, W_{TO}, lbs	43,500	45,000	12,500
Empty Weight, W_E, lbs	23,350	24,650	7,125
Wing Group/S, psf	6.0	6.0	3.3
Emp. Grp/S_{emp}, psf	3.2	3.3	2.2
Ultimate Load Factor, g's	3.75**	3.75**	3.75**
Surface Areas, ft^2			
Wing, S	754	754	373
Horiz. Tail, S_h	172	172	85
Vert. Tail, S_v	153	153	83
Empenn. Area, S_{emp}	325	325	168

*Strutbraced wing
**Assumed

Table A6.3a Group Weight Data for Regional Turbopropeller Driven Airplanes

Type	De Havilland Canada DHC7-102	DHC6-300
Number of engines:	2	2
Weight Item, lbs		
Wing Group	4,888	1,263*
Empennage Group	1,318	303
Fuselage Group	4,680	1,705
Nacelle Group	1,841	221
Land. Gear Group	1,732	613
Nose Gear		
Main Gear		
Structure Total	14,459	4,105
Engines		
Air Induct. System		
Fuel System		
Propeller Install.		
Propulsion System		
Power Plant Total	4,701	1,248
Avionics + Instrum.	850	371
Surface Controls	710	145
Hydraulic System	493	43
Pneumatic System		
Electrical System	1,651	356
Electronics		
Air Cond. System	550	103**
Pressurization System		
Anti-icing System	176	
Furnishings	2,862	732
Paint	150	64
Fixed Equipm't Total	7,442	1,814
$W_{oil} + W_{tfo}$	150	35
Full oil	130	54
Max. Fuel Capacity	6,968	1,114
Water and supplies	130	
Payload (Max.)	9,500	3,610

*Strutbraced wing **Heating system only

Table A6.3b Group Weight Data for Regional Turbopropeller Driven Airplanes

Type	De Havilland Canada DHC7-102	DHC6-300
Flight Design Gross Weight, GW, lbs	44,000	12,500
Structure/GW	0.329	0.328
Power Plant/GW	0.107	0.100
Fixed Equipm't/GW	0.169	0.145
Empty Weight/GW	0.605	0.573
Wing Group/GW	0.111	0.101*
Empenn. Group/GW	0.030	0.024
Fuselage Group/GW	0.106	0.136
Nacelle Group/GW	0.042	0.018
Land. Gear Group/GW	0.039	0.049**
Take-off Gross Wht, W_{TO}, lbs	44,000	12,500
Empty Weight, W_E, lbs	26,602	7,167
Wing Group/S, psf	5.7	3.0
Emp. Grp/S_{emp}, psf	3.4	2.1
Ultimate Load Factor, g's	3.75***	3.75***
Surface Areas, ft^2		
Wing, S	860	420
Horiz. Tail, S_h	217	100
Vert. Tail, S_v	170	48
Empenn. Area, S_{emp}	387	148

*Strutbraced wing **Fixed gear
***Assumed

Table A6.4a Group Weight Data for Regional Piston/ Propeller Driven Airplanes

Type	SAAB Scandia	Handley Page Herald	Scottish* Aviation Twin Pion.	Convair 240
Number of engines:	2	4	2	2
Weight Item, lbs				
Wing Group	4,195	4,365	2,121	3,943
Empennage Group	584	987	576	922
Fuselage Group	2,773	2,986	1,381	4,227
Nacelle Group	1,479	830	230	1,215
Land. Gear Group	1,841	1,625	703	1,530
Nose Gear				
Main Gear				
Structure Total	10,872	10,793	5,011	11,837
Engines				
Air Induct. System				
Fuel System				
Propeller Install.				
Propulsion System				
Power Plant Total				7,299
Avionics + Instrum.				
Surface Controls	369	364	300	
Hydraulic System				
Pneumatic System				
Electrical System				
Electronics				
APU				
Oxygen System				
Air Cond. System				
Anti-icing System				
Furnishings				
Auxiliary Gear				
Fixed Equipm't Total				4,444
$W_{oil} + W_{tfo}$	not--------	--------	--------	known
Max. Fuel Capacity	11,080		1,740	6,700
Payload (Max.)			2,950	16,000

*Strutbraced wing

Table A6.4b Group Weight Data for Regional Piston/ Propeller Driven Airplanes

Type	SAAB Scandia	Handley Page Herald	Scottish Aviation Twin Pion.	Convair 240
Flight Design Gross Weight, GW, lbs	30,860	37,500	14,600	43,500
Structure/GW	0.352	0.288	0.343	0.272
Power Plant/GW				0.168
Fixed Equipm't/GW				0.102
Empty Weight/GW	0.641	0.673	0.683	0.542
Wing Group/GW	0.136	0.116	0.145	0.091
Empenn. Group/GW	0.019	0.026	0.039	0.021
Fuselage Group/GW	0.090	0.080	0.095	0.097
Nacelle Group/GW	0.048	0.022	0.016	0.028
Land. Gear Group/GW	0.060	0.043	0.048	0.035
Take-off Gross Wht, W_{TO}, lbs	30,860	37,500	14,600	43,500
Empty Weight, W_E, lbs	19,780	25,240	9,969	23,580
Wing Group/S, psf	4.5	4.9	3.2	4.8
Emp. Grp/S_{emp}, psf	2.0	2.2	1.7	
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	922	886	670	817
Horiz. Tail, S_h	215	252	167	
Vert. Tail, S_v	82	193	167	
Empenn. Area, S_{emp}	297	445	334	

*Assumed

Table A7.1a Group Weight Data for Jet Transports

Type	McDonnell Douglas DC-9-30	MD-80	DC-10-10	DC-10-30
Number of engines:	2	2	3	3
Weight Item, lbs				
Wing Group	11,400	15,560	48,990	58,859
Empennage Group	2,780	3,320	13,660	14,676
Fuselage Group	11,160	16,150	44,790	47,270
Nacelle Group	1,430	2,120	8,490	9,127
Land. Gear Group	4,170	5,340	19,820	25,761
Nose Gear	470	550	1,520	1,832
Main Gear	3,700	4,790	18,300	23,929*
Structure Total	30,940	42,490	135,750	155,693
Engine(s)	6,410	8,820	23,688	26,163
Exhaust and Thrust Reverser System	1,240	1,540	7,232	6,916
Air Induct. System	0	0	0	0
Fuel System	600	640	2,040	4,308
Propulsion Install.	0	0	0	0
Power Plant Total	8,250	11,000	32,960	37,387
Avionics + Instrum.	1,450	2,130	3,410	4,274
Surface Controls	1,620	2,540	5,880	6,010
Hydraulic System	480	540	2,330	2,587
Pneumatic System	280	290	1,790	1,920
Electrical System	1,330	1,720	5,370	5,912
Electronics	Included in Avionics and Instrum.			
APU	820	840	1,590	1,643
Oxygen System	150	220	210	256
Air Cond. System**	1,120	1,580	2,390	2,723
Anti-icing System	480	550	420	471
Furnishings	8,450	11,400	35,810	34,124
Operating Items	2,700	3,650	13,340	16,274
Fixed Equipm't Total	18,880	25,460	72,540	76,194
W_{tfo}	Not ----------			known
Max. Fuel Capacity	28,746	39,362	146,683	247,034
Max. Payload	28,930	43,050	93,750	98,726

*Includes 3,590 lbs for centerline gear
**Includes pressurization system

Table A7.1b Group Weight Data for Jet Transports

Type	Mc Donnell Douglas DC-9-30	MD-80	DC-10-10	DC-10-30
Flight Design Gross Weight, GW, lbs	108,000	140,000	430,000	555,000
Structure/GW	0.286	0.304	0.316	0.281
Power Plant/GW	0.076	0.079	0.077	0.067
Fixed Equipm't/GW	0.175	0.182	0.169	0.137
Empty Weight/GW	0.538	0.564	0.561	0.485
Wing Group/GW	0.106	0.111	0.114	0.106
Empenn. Group/GW	0.026	0.024	0.032	0.026
Fuselage Group/GW	0.103	0.115	0.104	0.085
Nacelle Group/GW	0.013	0.015	0.020	0.016
Land. Gear Group/GW	0.039	0.038	0.046	0.046
Take-off Gross Wht, W_{TO}, lbs	108,000	140,000	430,000	555,000
Empty Weight, W_E, lbs	58,070	78,950	241,250	269,274
Wing Group/S, psf	11.4	12.3	12.7	14.9
Emp. Grp/S_{emp}, psf	6.4	5.7	7.0	7.6
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	1,001	1,270	3,861	3,958
Horiz. Tail, S_h	276	314	1,338	1,338
Vert. Tail, S_v	161	168	605	605
Empenn. Area, S_{emp}	437	582	1,943	1,943

*Assumed

Table A7.2a Group Weight Data for Jet Transports

Type	Boeing 737-200	727-100	747-100	Airbus A-300 B2
Number of engines:	2	3	4	2
Weight Item, lbs				
Wing Group	10,613	17,764	86,402	44,131
Empennage Group	2,718	4,133	11,850	5,941
Fuselage Group	12,108	17,681	71,845	35,820
Nacelle Group	1,392	3,870	10,031	7,039
Land. Gear Group	4,354	7,211	31,427	13,611
Nose Gear				
Main Gear				
Structure Total	31,185	50,659	211,555	106,542
Engines	6,217	9,325	34,120	16,825
Exhaust and Thrust-Reverser System	1,007	1,744	6,452	4,001
Air Induct. System	0	0	0	0
Fuel System	575	1,143	2,322	1,257
Propulsion Install.	378	250	802	814
Power Plant Total	8,177	12,462	43,696	22,897
Avionics + Instrum.	625	756	1,909	377
Surface Controls	2,348	2,996	6,982	5,808
Hydraulic System	873	1,418	4,471	3,701
Pneumatic System				
Electrical System	1,066	2,142	3,348	4,923
Electronics	956	1,591	4,429	1,726
APU	836	60	1,130	983
Air Cond. System*	1,416	1,976	3,969	3,642
Anti-icing System				
Furnishings	6,643	10,257	37,245	13,161
Miscellanous	124	85	-421	732
Fixed Equipm't Total	14,887	21,281	63,062	35,053
$W_{oil} + W_{tfo}$	Not			known
Max. Fuel Capacity	34,718	48,353	331,675	76,512
Max. Payload	34,790	29,700	140,000	69,865

*Includes pressurization system

Table A7.2b Group Weight Data for Jet Transports

Type	Boeing 737-200	727-100	747-100	Airbus A300-B2
Flight Design Gross Weight, GW, lbs	115,500	160,000	710,000	302,000
Structure/GW	0.270	0.317	0.298	0.353
Power Plant/GW	0.071	0.078	0.062	0.076
Fixed Equipm't/GW	0.129	0.133	0.089	0.116
Empty Weight/GW	0.521	0.552	0.498	0.559
Wing Group/GW	0.092	0.111	0.122	0.146
Empenn. Group/GW	0.024	0.026	0.017	0.020
Fuselage Group/GW	0.105	0.111	0.101	0.119
Nacelle Group/GW	0.012	0.024	0.014	0.023
Land. Gear Group/GW	0.038	0.045	0.044	0.045
Take-off Gross Wht. W_{TO}, lbs	115,500	160,000	710,000	302,000
Empty Weight, W_E, lbs	60,210	88,300	353,398	168,805
Wing Group/S, psf	10.8	10.4	15.7	15.8
Emp. Grp/S_{emp}, psf	4.9	5.6	5.2	4.8
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	980	1,700	5,500	2,799
Horiz. Tail, S_h	321	376	1,470	748
Vert. Tail, S_v	233	356	830	487
Empenn. Area, S_{emp}	554	732	2,300	1,235

*Assumed

Table A7.3a Group Weight Data for Jet Transports

Type	Boeing 707-121	707-320	707-320C	720-022
Number of engines:	4	4	4	4
Weight Item, lbs				
Wing Group	24,024	29,762	32,255	22,850
Empennage Group	5,151	5,511	6,165	5,230
Fuselage Group	20,061	21,650	26,937	19,035
Nacelle Group	4,639	4,497	4,183	4,510
Land. Gear Group	9,763	12,700	12,737	8,110
Nose Gear				
Main Gear				
Structure Total	63,638	74,120	82,277	59,735
Engines	16,458	20,200	17,368	13,770
Exhaust and Thrust-Reverser System			3,492	
Air Induct. System	0	0	0	0
Fuel System	1,808		2,418	1,240
Propulsion Install.	1,738		798	885
Power Plant Total	20,004		24,076	15,895
Avionics + Instrum.	505		515	555
Surface Controls	2,159	2,400	3,052	2,450
Hydraulic System	484		1,086	505
Pneumatic System				
Electrical System	3,772		4,179	4,070
Electronics	1,708		2,338	1,200
APU			151	
Air Cond. System*	3,110		3,608	2,890
Anti-icing System				
Furnishings	13,651		9,527	13,055
Auxiliary Gear	0	0	0	0
Miscellanous	0	0	-389	0
Fixed Equipm't Total	25,389		24,456	24,725
W_{tfo}	704			
Max. Fuel Capacity	90,842	160,783	160,783	99,954
Max. Payload	42,600	55,000	84,000	28,200

*Includes pressurization system

Table A7.3b Group Weight Data for Jet Transports

Type	Boeing 707-121	707-320	707-320C	720-022
Flight Design Gross Weight, GW, lbs	246,000	311,000	330,000	203,000
Structure/GW	0.259	0.238	0.249	0.294
Power Plant/GW	0.081		0.073	0.078
Fixed Equipm't/GW	0.103		0.074	0.122
Empty Weight/GW	0.444	0.434	0.396	0.494
Wing Group/GW	0.098	0.096	0.098	0.113
Empenn. Group/GW	0.021	0.018	0.019	0.026
Fuselage Group/GW	0.082	0.070	0.082	0.094
Nacelle Group/GW	0.019	0.014	0.013	0.022
Land. Gear Group/GW	0.040	0.041	0.039	0.040
Take-off Gross Wht. W_{TO}, lbs	246,000	311,000	330,000	203,000
Empty Weight, W_E, lbs	109,111	135,000	130,809	100,355
Wing Group/S, psf	9.9	10.3	10.6	9.4
Emp. Grp/S_{emp}, psf	6.2	5.8	6.5	6.3
Ultimate Load Factor, g's	3.75	3.75	3.75	3.75
Surface Areas, ft^2				
Wing, S	2,433	2,892	3,050	2,433
Horiz. Tail, S_h	500	625	625	500
Vert. Tail, S_v	328	328	328	328
Empenn. Area, S_{emp}	828	953	953	828

Table A7.4a Group Weight Data for Jet Transports

Type	Boeing 707-321	McDonnell Douglas DC-8	McDonnell Douglas DC-9-10	Hawker-Siddeley 121-IC
Number of engines:	4	4	2	3
Weight Item, lbs				
Wing Group	28,647	27,556	9,470	12,600
Empennage Group	6,004	4,840	2,630	3,225
Fuselage Group	22,129	19,910	11,206	12,469
Nacelle Group	5,119	3,534	1,417	in fusel.
Land. Gear Group	11,122	10,910	3,660	4,413
Nose Gear				
Main Gear				
Structure Total	73,021	66,750	28,383	32,707
Engine(s)	19,192		6,160	
Exhaust and Thrust Reverser System			658	
Air Induct. System				
Fuel System	1,956		510	
Propulsive Install.	1,113		409	
Power Plant Total	22,261	27,677	7,737	
Avionics + Instrum.	561		719	
Surface Controls	2,408		1,264	1,792
Hydraulic System	498		714	
Pneumatic System				
Electrical System	3,959		1,663	
Electronics	1,716		914	
APU			818	
Air Cond. System*	3,290		1,476	
Anti-icing System				
Furnishings	14,854		7,408	
Auxiliary Gear	0		24	
Fixed Equipm't Total	27,286	25,650	15,000	
W_{tfo}	1,089			
Max. Fuel Capacity		30,256	18,778	31,060
Max. Payload			18,050	22,000

*Includes pressurization system

Table A7.4b Group Weight Data for Jet Transports

Type	Boeing 707-321	McDonnell Douglas DC-8	McDonnell Douglas DC-9-10	Hawker Siddeley 121-IC
Flight Design Gross Weight, GW, lbs	302,000	215,000	91,500	115,000
Structure/GW	0.242	0.310	0.310	0.284
Power Plant/GW	0.074	0.129	0.085	
Fixed Equipm't/GW	0.090	0.119	0.164	
Empty Weight/GW	0.406	0.562	0.495	0.587
Wing Group/GW	0.095	0.128	0.103	0.110
Empenn. Group/GW	0.020	0.023	0.029	0.028
Fuselage Group/GW	0.073	0.093	0.122	0.108
Nacelle Group/GW	0.017	0.016	0.015	in fusel.
Land. Gear Group/GW	0.037	0.051	0.040	0.038
Take-off Gross Wht, W_{TO}, lbs	301,000	215,000	91,500	115,000
Empty Weight, W_E, lbs	122,509	120,877	45,300	67,500
Wing Group/S, psf	9.9	9.9	10.1	9.3
Emp. Grp/S_{emp}, psf	6.4	5.6	5.5	5.7
Ultimate Load Factor, g's	3.75	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	2,892	2,773	934	1,358
Horiz. Tail, S_h	625	607**	275	310
Vert. Tail, S_v	312	263**	200**	259
Empenn. Area, S_{emp}	937	870	475	569

*Assumed
**Estimated from threeview

Table A7.5a Group Weight Data for Jet Transports

Type	VFW-Fokker 614	Fokker F28-1000	BAC 1-11/300	Sud/Aerospatiale Caravelle
Number of engines:	2	2	2	2
Weight Item, lbs				
Wing Group	5,767	7,330	9,643	14,735
Empennage Group	1,121	1,632	2,369	1,957
Fuselage Group	5,233	7,043	9,713	11,570
Nacelle Group	971	834	in fusel.	1,581
Land. Gear Group	1,620	2,759	2,856	5,110
Nose Gear				
Main Gear				
Structure Total	14,712	19,598	24,581	34,953
Engines	3,413	4,495		7,055
Exhaust and Thrust-Reverser System	119	127		975
Air Induct. System				
Fuel System	162	545		518
Propulsive Install	690	215		179
Power Plant Total	4,384	5,382		8,727
Avionics + Instrum.	215	302	182	236
Surface Controls	745	1,387	1,481	2,063
Hydraulic System	403	364	997	1,976
Pneumatic System				
Electrical System	1,054	1,023	2,317	2,846
Electronics	436	869	1,005	1,187
APU	305	346	457	0
Air Cond. System*	719	1,074	1,579	1,752
Anti-icing System				
Furnishings	2,655	4,030	4,933	6,481
Auxiliary Gear	49	0	0	0
Operating Items				
Fixed Equipm't Total	6,581	9,395	12,951	15,941
W_{tfo}	not			known
Max. Fuel Capacity	10,142	17,331	24,954	33,808
Max. Payload	8,201	14,380	22,278	29,100

*Includes pressurization system

Table A7.5b Group Weight Data for Jet Transports

Type	VFW Fokker 614	Fokker F28-1000	BAC 1-11/300	Sud-Aero spatiale Caravelle
Flight Design Gross Weight, GW, lbs	40,981	65,000	87,000	110,230
Structure/GW	0.359	0.302	0.283	0.317
Power Plant/GW	0.107	0.083		0.079
Fixed Equipm't/GW	0.161	0.145	0.149	0.145
Empty Weight/GW*	0.586	0.480	0.560	0.590
Wing Group/GW	0.141	0.113	0.111	0.134
Empenn. Group/GW	0.027	0.025	0.027	0.018
Fuselage Group/GW	0.128	0.108	0.112	0.105
Nacelle Group/GW	0.024	0.013	in fusel.	0.014
Land. Gear Group/GW	0.040	0.042	0.033	0.046
Take-off Gross Wht, W_{TO}, lbs	40,981	65,000	87,000	110,230
Empty Weight, W_E, lbs	24,000	31,219	48,722	65,050
Wing Group/S, psf	8.4	8.9	9.6	9.3
Emp. Grp/S_{emp}, psf	3.8	4.8	6.3	4.2
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	689	822	1,003	1,579
Horiz. Tail, S_h	193	210	257	301
Vert. Tail, S_v	102	132	117	167
Empenn. Area, S_{emp}	295	342	374	468

*Assumed

Table A7.6a Group Weight Data for Turboprop. Transports

Type→	Bristol Britannia 300	Canadair CL-44C	Vickers Viscount 810	Lockheed Electra
Number of engines:	4	4	4	4
Weight Item, lbs				
Wing Group	13,433	15,710	6,250	7,670
Empennage Group	3,202	3,749	1,245	1,924
Fuselage Group	11,100	20,524	6,900	9,954
Nacelle Group	4,930	6,834	1,810	4,417
Land. Gear Group	5,785	7,083	2,469	3,817
Nose Gear				
Main Gear				
Structure Total	38,450	53,900	18,674	27,782
Engines	11,192	12,800		
Air Induct. System				
Fuel System	1,329	1,755		
Propeller Inst.	3,557	5,006		
Propulsion System	3,820	3,134		
Power Plant Total	19,898	22,695		13,733
Avionics + Instrum.	505	858	213	
Surface Controls	1,221	2,146	824	
Hydraulic System	650	630	457	
Pneumatic System				
Electrical System	1,800	3,040	2,826	
Electronics	1,040	1,229	617	
APU	0	0	0	
Air Cond. System*	3,000	2,536	2,092	
Anti-icing System				
Furnishings	6,866	12,349	3,476	
Auxiliary Gear	0	0	0	
Fixed Equipm't Total	15,082	22,788	10,505	14,469
$W_{oil} + W_{tfo}$				
Max. Fuel Capacity	69,395	82,170	13,897	37,205
Payload (Max.)	30,000	37,630	15,054	18,907

*Includes pressurization system

Table A7.6b Group Weight Data for Turboprop. Transports

Type	Bristol Britannia 300	Canadair CL-44C	Vickers Viscount 810	Lockheed Electra
Flight Design Gross Weight, GW, lbs	155,000	205,000	72,500	116,000
Structure/GW	0.248	0.263	0.258	0.240
Power Plant/GW	0.128	0.111		0.118
Fixed Equipm't/GW	0.097	0.111	0.145	0.125
Empty Weight/GW	0.587	0.516	0.569	0.491
Wing Group/GW	0.087	0.077	0.086	0.066
Empenn. Group/GW	0.021	0.018	0.017	0.017
Fuselage Group/GW	0.072	0.100	0.095	0.086
Nacelle Group/GW	0.032	0.033	0.025	0.038
Land. Gear Group/GW	0.037	0.035	0.034	0.033
Take-off Gross Wht, W_{TO}, lbs	155,000	205,000	72,500	116,000
Empty Weight, W_E, lbs	91,000	105,785	41,276	57,000
Wing Group/S, psf	6.5	7.6	6.5	5.9
Emp. Grp/S_{emp}, psf	3.4	4.0	2.9	3.1
Ultimate Load Factor, g's	3.75*	3.75*	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	2,075	2,075	963	1,300
Horiz. Tail, S_h	588	588	307	399
Vert. Tail, S_v	356	356	124	212
Empenn. Area, S_{emp}	944	944	431	611

*Assumed

Table A8.1a Group Weight Data for Military Trainers

Type	Northrop T-38A	Rockwell NAA T-39A	Cessna T-37A	Fouga Magis-ter	Cana-dair CL-41
Number of engines:	2	2	2	2	2
Weight Item, lbs					
Wing Group	765	1,753	531	1,089	892
Empennage Group	305	297	128	165	201
Fuselage Group	1,985	2,014	839	743	955
Engine Section	147	315*		in fuse.	40
Land. Gear Group	457	728	330	459	318
Nose Gear					
Main Gear					
Structure Total	3,659	5,107	1,828	2,456	2,406
Engine(s)	1,038	959	751		
Air Induct. System	136	12	14		
Fuel System	285	190	225		
Propulsion System	171	140	205		
Power Plant Total	1,630	1,301	1,195		
Avionics + Instrum.	211	122	132		
Surface Controls	425	344	154		172
Hydraulic System	154	116	56		
Pneumatic System					
Electrical System	296	720	194		
Electronics	246	407	86		
Air Cond. System**	142	333	69		
Anti-icing System					
Furnishings	460	857	256		
Auxiliary Gear	24		3		
Fixed Equipm't Total	1,958	2,899	950		
$W_{oil} + W_{tfo}$	62	89	104		
Max. Fuel Capacity	3,916	5,805	1,959	1,299	2,082
Payload (Max. Fuel)***	426	1,500	400	400	400

*Nacelle group for T-39A
**Includes pressurization system
***Includes crew

Table A8.1b Group Weight Data for Military Trainers

Type	Northrop T-38A	Rockwell NAA T-39A	Cessna T-37A	Fouga Magis-ter	Cana-dair CL-41
Flight Design Gross Weight, GW, lbs	11,651	16,316	6,228	6,280	11,288
Structure/GW	0.314	0.313	0.294	0.391	0.213
Power Plant/GW	0.140	0.080	0.192		
Fixed Equipm't/GW	0.168	0.178	0.152	N.A.	N.A.
Empty Wt/GW	0.622	0.570	0.638	0.755	0.576
Wing Group/GW	0.066	0.107	0.085	0.173	0.079
Emp. Group/GW	0.026	0.018	0.021	0.026	0.018
Fusel.Group/GW	0.170	0.123	0.135	0.118	0.085
Engine Section/GW	0.013	0.019*		in fus.	0.004
Land. Gear Group/GW	0.039	0.045	0.053	0.073	0.028
Take-off Gross Wht, W_{TO}, lbs	11,651	16,701	6,436	6,280	11,288
Empty Weight, W_E, lbs	7,247	9,307	3,973	4,740	5,296
Wing Group/S, psf	4.5	5.1	3.9	5.9	4.1
Emp. Grp/S_{emp}, psf	2.9	2.5	1.7	3.4	3.4
Ultimate Load Factor, g's	10.0**	6.0	10.0		
Surface Areas, ft^2					
Wing, S	170	342	135	186	220
Horiz. Tail, S_h	59	77	54	***	41.3
Vert. Tail, S_v	47.8	41.6	20.4	***	17.5
Empenn. Area, S_{emp}	107	119	74.4	48.8	58.8

*Nacelle group for T-39A
**Assumed
***V-tail

Table A9.1a Group Weight Data for Fighters (USAF)

Type	NAA F-100F	McDonnell F-101B	RF-101C	Gen.Dyn. F-102A*
Number of engines:	1	2	2	1
Weight Item, lbs				
Wing Group	3,896	3,507	3,680	3,000
Empennage Group	979	812	837	535
Fuselage Group	4,032	3,901	3,955	3,409
Engine Section	104	99	103	39
Land. Gear Group	1,509	1,592	1,596	1,056
Nose Gear				
Main Gear				
Structure Total	10,520	9,911	10,171	8,039
Engine(s)	5,121	10,800	9,676	4,993
Air Induct. System	504	729	638	693
Fuel System	761	1,226	1,412	394
Propulsion System	414	892	599	278
Power Plant Total	6,800	13,647	12,325	6,358
Avionics + Instrum.	303	318	204	141
Surface Controls	1,076	772	780	413
Hydraulic System	157	433	359	318
Pneumatic System				
Electrical System	568	825	819	594
Electronics	496	2,222	629	2,001
Armament	794	228	36	589
Air Cond. System**	435	270	362	259
Anti-icing System				
Furnishings	427	480	242	227
Auxiliary Gear	77	84	91	78
Fixed Equipm't Total	4,333	5,632	3,522	4,620
$W_{oil} + W_{tfo}$	166	223	223	216
Max. Fuel Capacity	7,729	8,892	9,782	7,053
Payload (Max. Fuel)	250	1,881	704	1,241

*This airplane is a delta wing configuration
**Includes pressurization system

Table A9.1b Group Weight Data for Fighters (USAF)

Type	NAA F-100F	McDonnell F-101B	RF-101C	Gen.Dyn. F-102A*
Flight Design Gross Weight, GW, lbs	29,391	39,800	37,000	25,500
Structure/GW	0.358	0.249	0.275	0.315
Power Plant/GW	0.231	0.343	0.333	0.249
Fixed Equipm't/GW	0.147	0.142	0.095	0.181
Empty Weight/GW	0.737	0.733	0.724	0.750
Wing Group/GW	0.133	0.088	0.099	0.118
Empenn. Group/GW	0.033	0.020	0.023	0.021
Fuselage Group/GW	0.137	0.098	0.107	0.134
Engine Section/GW	0.004	0.002	0.003	0.002
Land. Gear Group/GW	0.051	0.040	0.043	0.041
Take-off Gross Wht, W_{TO}, lbs	30,638	41,288	37,723	28,137
Empty Weight, W_E, lbs	21,653	29,190	26,774	19,130
Wing Group/S, psf	9.7	9.5	10.0	4.3
Emp. Grp/S_{emp}, psf	6.3	5.1	5.2	5.6
Ultimate Load Factor, g's	7.33	10.2	11.0	10.5
Surface Areas, ft^2				
Wing, S	400	368	368	698
Horiz. Tail, S_h	98.9	75.1	75.1	0
Vert. Tail, S_v	55.6	84.9	84.9	95.1
Empenn. Area, S_{emp}	155	160	160	95.1

* This airplane is a delta wing configuration

Table A9.2a Group Weight Data for Fighters (USAF)

Type	Republic F-105B	Gen.Dyn. F-106A*	North American F-107A	North American F-86H
Number of engines:	1	1	1	1
Weight Item. lbs				
Wing Group	3,409	3,302	3,787	2,702
Empennage Group	965	693	1,130	329
Fuselage Group	5,870	4,401	4,792	2,035
Engine Section	106	39	260	42
Land. Gear Group	1,848	1,232	1,410	989
Nose Gear				
Main Gear				
Structure Total	12,198	9,667	11,379	6,097
Engine	6,187	5,816	6,100	3,646
Air Induct. System	524	975	833	167
Fuel System	608	777	983	845
Propulsion System	406	503	368	340
Power Plant Total	7,725	8,071	8,284	4,998
Avionics + Instrum.	227	190	288	111
Surface Controls	1,311	445	1,454	358
Hydraulic System				
Pneumatic System	449	431	150	339
Electrical System	700	606	447	476
Electronics	737	2,743	382	230
Armament	719	626	1,006	828
Air Cond. System**	168	407	390	205
Anti-icing System				
Furnishings	243	290	282	182
Auxiliary Gear	92	69	42	12
APU	224	0	0	0
Fixed Equipm't Total	4,870	5,807	4,441	2,741
$W_{oil} + W_{tfo}$	198	303	143	57
Max. Fuel Capacity	7,540	8,476	11,050	3,660
Payload (Max. Fuel)	757	1,374	2,560	420

*This airplane is a delta wing configuration
**Includes pressurization system

Table A9.2b Group Weight Data for Fighters (USAF)

Type	Republic F-105B	Gen.Dyn. F-106A*	North American F-107A	F-86H
Flight Design Gross Weight, GW, lbs	31,392	30,590	29,524	19,012
Structure/GW	0.389	0.316	0.385	0.321
Power Plant/GW	0.246	0.264	0.281	0.263
Fixed Equipm't/GW	0.155	0.190	0.150	0.144
Empty Weight/GW	0.797	0.766	0.816	0.728
Wing Group/GW	0.109	0.108	0.128	0.142
Empenn. Group/GW	0.031	0.023	0.038	0.017
Fuselage Group/GW	0.187	0.144	0.162	0.107
Engine Section/GW	0.003	0.001	0.009	0.002
Land. Gear Group/GW	0.059	0.040	0.048	0.052
Take-off Gross Wht, W_{TO}, lbs	34,081	33,888	39,405	18,908
Empty Weight, W_E, lbs	25,022	23,448	24,104	13,836
Wing Group/S, psf	8.9	4.7	9.6	8.6
Emp. Grp/S_{emp}, psf	5.2	6.6	6.4	4.1
Ultimate Load Factor, g's	13.0	10.5	13.0	11.0
Surface Areas, ft^2				
Wing, S	385	698	395	313
Horiz. Tail, S_h	96.5	0	93.3	47.2
Vert. Tail, S_v	88.1	105	83.8	32.2
Empenn. Area, S_{emp}	185	105	177	79.4

* This airplane is a delta wing configuration

Table A9.3a Group Weight Data for Fighters (USN)

Type	Vought F8U-3	McDonnell F4H	Grumman F11F	F9F-5
Number of engines:	1	2	1	1
Weight Item, lbs				
Wing Group	4,128	4,343	2,180	2,294
Empennage Group	1,045	853	669	404
Fuselage Group	3,850	4,042	3,269	1,779
Engine Section	92	125	47	0
Land. Gear Group	949	1,735	907	728
Nose Gear				
Main Gear				
Structure Total	10,064	11,098	7,072	5,205
Engine(s)	6,010	6,940	3,489	2,008
Air Induct. System	673	1,037	159	225
Fuel System	849	953	463	529
Propulsion System.	338	106	192	116
Power Plant Total	7,870	9,036	4,303	2,878
Avionics + Instrum.	191	166	118	82
Surface Controls	1,425	919	760	345
Hydraulic System	150	441	166	267
Pneumatic System				
Electrical System	439	502	459	458
Electronics	840	1,386	439	292
Armament	376	446	358	416
Air Cond. System*	329	341	76	85
Anti-icing System				
Furnishings	210	321	166	144
Auxiliary Gear	183	0	131	51
Fixed Equipm't Total	4,143	4,522	2,673	2,140
$W_{oil} + W_{tfo}$	196	131	72	
Max. Fuel Capacity	14,306	13,410	6,663	7,160
Payload (Max. Fuel)	1,197	1,500	340	

*Includes pressurization system

Table A9.3b Group Weight Data for Fighters (USN)

Type	Vought F8U-3	McDonnell F4H	Grumman F11F	F9F-5
Flight Design Gross Weight, GW, lbs	30,578	34,851	17,500	14,900
Structure/GW	0.329	0.318	0.404	0.349
Power Plant/GW	0.257	0.259	0.246	0.193
Fixed Equipm't/GW	0.135	0.130	0.153	0.144
Empty Weight/GW	0.722	0.707	0.771	0.686
Wing Group/GW	0.135	0.125	0.125	0.154
Empenn. Group/GW	0.034	0.024	0.038	0.027
Fuselage Group/GW	0.126	0.116	0.187	0.119
Engine Section/GW	0.003	0.004	0.003	0
Land. Gear Group/GW	0.031	0.050	0.052	0.049
Take-off Gross Wht, W_{TO}, lbs	38,528	40,217	21,233	17,500
Empty Weight, W_E, lbs	22,092	24,656	13,485	10,223
Wing Group/S, psf	8.9	8.2	8.5	9.2
Emp. Grp/S_{emp}, psf	7.2	5.2	5.8	3.5
Ultimate Load Factor, g's	9.6		9.8	11.3
Surface Areas, ft^2				
Wing, S	462	530	255	250
Horiz. Tail, S_h	67.2	96.2	65.5	48
Vert. Tail, S_v	78.6	67.5	50.3	66
Empenn. Area, S_{emp}	146	164	116	114

Table A9.4a Group Weight Data for Fighters (USN)

Type	Grumman A2F(A6)	McDonnell F3H-2	NAA A3J	Vought F7U-1
Number of engines:	2	2	2	1
Weight Item, lbs				
Wing Group	4,733	4,314	5,072	3,583
Empennage Group	819	576	1,358	726
Fuselage Group	3,538	3,551	6,851	937
Engine Section	64	93	80	
Land. Gear Group	2,343	1,458	2,173	1,181
Nose Gear				
Main Gear				
Structure Total	11,497	9,992	15,534	6,427
Engine(s)	4,010	4,960	7,260	2,790
Air Induct. System	61	614	767	690
Fuel System	936	1,262	979	1,080
Propulsion System.	632	70	353	937
Power Plant Total	5,639	6,906	9,359	5,497
Avionics + Instrum.	133	145	210	108
Surface Controls	932	1,067	1,845	482
Hydraulic System	170	474	275	317
Pneumatic System				
Electrical System	695	535	821	371
Electronics	2,652	984	2,239	328
Armament	323	662	45	367
Air Cond. System*	164	101	424	79
Anti-icing System				
Furnishings	476	218	676	279
Auxiliary Gear		253		128
Fixed Equipm't Total	5,545	4,439	6,535	2,459
$W_{oil} + W_{tfo}$	195	147	320	97
Max. Fuel Capacity	8,764	9,789	19,074	5,826
Payload (Max. Fuel)	2,000	216	1,885	502

*Includes pressurization system

Table A9.4b Group Weight Data for Fighters (USN)

Type	Grumman A2F(A6)	McDonnell F3H-2	NAA A3J	Vought F7U-1*
Flight Design Gross Weight, GW, lbs	34,815	26,000	46,028	19,310
Structure/GW	0.330	0.384	0.337	0.333
Power Plant/GW	0.162	0.266	0.203	0.285
Fixed Equipm't/GW	0.159	0.171	0.142	0.127
Empty Weight/GW	0.651	0.818	0.679	0.746
Wing Group/GW	0.136	0.166	0.110	0.186
Empenn. Group/GW	0.024	0.022	0.030	0.038
Fuselage Group/GW	0.102	0.137	0.149	0.048
Engine Section/GW	0.002	0.004	0.002	
Land. Gear Group/GW	0.067	0.056	0.047	0.061
Take-off Gross Wht, W_{TO}, lbs	34,815	32,037	53,658	21,638
Empty Weight, W_E, lbs	22,680	21,272	31,246	14,397
Wing Group/S, psf	9.1	8.4	7.2	7.1
Emp. Grp/S_{emp}, psf	4.4	4.5	3.4	8.3
Ultimate Load Factor, g's		11.25		
Surface Areas, ft^2				
Wing, S	520	516	700	507
Horiz. Tail, S_h	120	82.5	304	0*
Vert. Tail, S_v	68.4	45.4	101	88
Empenn. Area, S_{emp}	188	128	405	88

*This airplane is essentially a flying wing with two vertical tails

Table A9.5a Group Weight Data for Fighters (USAF and USN)

Type	McDonnell Douglas F-4E (USAF)	F-15C (USAF)	F/A-18A (USN)	AV-8B* (USN)
Number of engines:	2	2	2	1
Weight Item, lbs				
Wing Group	5,226	3,642	3,798	1,443
Empennage Group	969	1,104	945	372
Fuselage Group	5,050	6,245	4,685	2,060
Engine Section	166	102	143	141
Land. Gear Group	1,944	1,393	1,992	1,011
Nose Gear	377	264	626	334
Main Gear	1,567	1,129	1,366	400
Outrigger Gear				277
Structure Total	13,355	12,486	11,563	5,027
Engine(s)	7,697	6,091	4,294	3,815
Air Induct. System	1,318	1,464	423	236
Fuel System	1,932	1,128	1,002	542
Propulsion System.	312	522	558	444
Power Plant Total	11,259	9,205	6,277	5,037
Instrument group	270	151	94	80
Surface Controls	1,167	810	1,067	698
Hydraulic System	543	433	364	176
Pneumatic System				
Electrical System	542	607	547	424
Electronics	2,227	1,787	1,538	697
Armament	641	627	387	152
Air Cond. System	406	685	593	218
Pressurization Syst.				
Anti-icing System			21	
Furnishings	611	294	317	298
Auxiliary Gear	412	119	189	
Photographic System		24		
Ballast		318	36	
Manuf. Variation	57	-97	-19	-16
Fixed Equipm't Total	6,900	5,734	5,134	2,727
Max. Fuel Capacity	12,058	13,455	17,592**	7,759
Expendable Payload	2,193	2,571	5,453	4,271
Fixed Payload***	incl. in armament		2,231	832

*V/STOL fighter **Incl. 6,732 lbs ext. fuel ***Pylons, racks, launchers, FLIR and camera pods

Table A9.5b Group Weight Data for Fighters (USAF and USN)

Type	McDonnell Douglas F-4E	F-15C	F/A-18A	AV-8B*
Flight Design Gross Weight, GW, lbs	37,500	37,400	32,357	22,950
Structure/GW	0.356	0.334	0.357	0.219
Power Plant/GW	0.300	0.246	0.194	0.219
Fixed Equipm't/GW	0.182	0.147	0.158	0.120
Empty Weight/GW	0.840	0.733	0.710	0.557
Wing Group/GW	0.139	0.097	0.117	0.063
Empenn. Group/GW	0.026	0.030	0.029	0.016
Fuselage Group/GW	0.135	0.167	0.145	0.090
Engine Section/GW	0.004	0.003	0.004	0.006
Land. Gear Group/GW	0.052	0.037	0.062	0.044
Take-off Gross Wht. W_{TO}, lbs	58,000	68,000	51,900	29,750
Empty Weight, W_E, lbs	31,514	27,425	22,974	12,791
Wing Group/S, psf	9.5	6.1	9.5	6.3
Emp. Grp/S_{emp}, psf	5.8	4.7	4.9	5.0
Ultimate Load Factor, g's	9.75	11.0	11.25	10.5
Surface Areas, ft^2				
Wing, S	548	599	400	230
Horiz. Tail, S_h	100	111	88.1	48.5
Vert. Tail, S_v	67.5	125	104	26.6
Empenn. Area, S_{emp}	168	236	192	75.1

*V/STOL Fighter

Table A10.1a Group Weight Data for Military Jet Transports

Type	Boeing KC135*	Lockheed C-141B	C-5A
Number of engines:	4	4	4
Weight Item, lbs			
Wing Group	25,251	35,272	100,015
Empennage Group	5,074	5,907	12,461
Fuselage Group	18,867	36,857	118,193
Nacelle Group	2,575	5,168	9,528
Land. Gear Group	10,180	10,850	38,353
Nose Gear		1,234	4,455
Main Gear		9,616	33,898
Structure Total	61,947	94,054	278,550
Engine(s)	16,687	23,665	38,035
Air Induct. System	172		
Fuel System	4,052	1,802	2,540
Propulsion System	591		
Power Plant Total	21,502	25,467	40,575
Avionics + Instrum.	553	3,078	3,823
Surface Controls	2,044	3,701	7,404
Hydraulic System	858	1,604	4,086
Pneumatic System			
Electrical System	2,470	2,826	3,503
Electronics	2,096	1,163	992
APU	0	534	987
Oxygen System		479	308
Air Cond. System**	1,464	2,283	3,416
Anti-icing System		453	229
Furnishings	1,518	5,210	19,272
Auxiliary Gear	1,899	103	39
Fixed Equipm't Total	12,902	21,434	44,059
$W_{oil} + W_{tfo}$	1,407	1,327	826
Max. Fuel Capacity	158,997	153,352	332,500
Payload (Max.)		73,873	200,000

*This is a tanker airplane
**Includes pressurization system

Table A10.1b Group Weight Data for Military Jet Transports

Type	Boeing KC135*	Lockheed C-141B	C-5A
Flight Design Gross Weight, GW, lbs	297,000	314,200	769,000
Structure/GW	0.209	0.299	0.362
Power Plant/GW	0.072	0.081	0.053
Fixed Equipm't/GW	0.043	0.068	0.057
Empty Weight/GW	0.323	0.449	0.472
Wing Group/GW	0.085	0.112	0.130
Empenn. Group/GW	0.017	0.019	0.016
Fuselage Group/GW	0.064	0.117	0.154
Nacelle Group/GW	0.009	0.016	0.012
Land. Gear Group/GW	0.034	0.035	0.050
Take-off Gross Wht, W_{TO}, lbs	297,000	314,000	769,000
Empty Weight, W_E, lbs	95,938	140,955	363,184
Wing Group/S, psf	10.4	10.9	16.1
Emp. Grp/S_{emp}, psf	6.2	6.6	6.5
Ultimate Load Factor, g's	3.75	3.75	3.75**
Surface Areas, ft^2			
Wing, S	2,435	3,228	6,200
Horiz. Tail, S_h	500	483	966
Vert. Tail, S_v	312	416	961
Empenn. Area, S_{emp}	812	899	1,927

*This is a tanker airplane
**after 100,000 lbs of fuel has been used.

Table A10.2a Group Weight Data for Turbo/Propeller Driven Military Transports

	A.W.(HS) Argosy	Douglas C-133A	Lockheed C-130H	Breguet 941*
Number of engines:	4	4	4	4
Weight Item, lbs				
Wing Group	10,800	27,403	13,950	4,096
Empennage Group	1,300	6,011	3,480	1,387
Fuselage Group	11,100**	30,940	14,695	6,481
Nacelle Group	1,200	3,512	2,756	in wing
Land. Gear Group	3,180	10,635	5,309	2,626
Nose Gear			730	
Main Gear			4,579	
Structure Total	27,580	78,501	40,190	14,590
Engines		10,470	13,746	
Air Induct. System				
Fuel System		1,338	3,105	
Propeller Inst.		5,403	in eng.	
Propulsion System		2,081	in eng.	
Power Plant Total		19,292	16,851	
Avionics + Instrum.		578	3,582	
Surface Controls	in struct.	1,804	1,673	1,056
Hydraulic System		2,678	664	
Pneumatic System				
Electrical System		2,004	2,459	
Electronics		2,047	in avionics	
APU		188	651	
Oxygen System			231	
Air Cond. System***		2,973	1,684	
Anti-icing System			797	
Furnishings		3,632	4,472	
Auxiliary Gear		117	6	
Operating items			532	
Fixed Equipm't Total		16,021	16,219	
$W_{oil} + W_{tfo}$		1,693	1,089	
Max. Fuel Capacity		60,000	45,240	
Payload (Max.)		97,162	33,461	

*This is a STOL airplane **Tailbooms at 2,360 lbs are included ***Includes pressurization system

Table A10.2b Group Weight Data for Turbo/Propeller Driven Military Transports

Type	A.W.(BS) Argosy	Douglas C-133A	Lockheed C-130E	Breguet 941
Flight Design Gross Weight, GW, lbs	82,000	275,000	155,000	58,421
Structure/GW	0.336	0.285	0.259	0.250
Power Plant/GW		0.070	0.109	
Fixed Equipm't/GW		0.058	0.105	
Empty Weight/GW	0.561	0.414	0.473	0.508
Wing Group/GW	0.132	0.100	0.090	0.070
Empenn. Group/GW	0.016	0.022	0.022	0.024
Fuselage Group/GW	0.135	0.113	0.095	0.111
Nacelle Group/GW	0.015	0.013	0.018	in wing
Land. Gear Group/GW	0.039	0.039	0.034	0.045
Take-off Gross Wht, W_{TO}, lbs	82,000	275,000	155,000	58,421
Empty Weight, W_E, lbs	46,000	113,814	73,260	29,675
Wing Group/S, psf	7.4	10.3	8.0	4.5
Emp. Grp/S_{emp}, psf	2.3	4.2	4.2	2.6
Ultimate Load Factor, g's	3.75*	2.50	3.75*	3.75*
Surface Areas, ft^2				
Wing, S	1,458	2,673	1,745	902
Horiz. Tail, S_h	327	801	536	320
Vert. Tail, S_v	250	641	300	223
Empenn. Area, S_{emp}	577	1,442	836	543

*Assumed

Table A10.3a Group Weight Data for Piston/Propeller Driven Military Transports

Type	Beech L-23F*	Chase C-123B	DeHavill. Caribou	Fairchild C-119B
Number of engines:	2	2	2	2
Weight Item, lbs				
Wing Group	692	6,153	2,925	7,226
Empennage Group	156	1,103	790	1,193
Fuselage Group	679	7,763	2,849	7,157
Nacelle Group	279	1,633	781	2,538**
Land. Gear Group	453	2,081	1,230	4,197
Nose Gear				
Main Gear				
Structure Total	2,259	18,733	8,575	22,311
Engines	1,015	4,810	3,170	6,500
Propellers	260	1,430	871	
Fuel System	127	671	221	
Propulsion System	215	1,103	453	
Power Plant Total	1,617	8,014	4,715	11,979
Avionics + Instrum.	91	161	121	
Surface Controls	129	490	326	
Hydraulic System	0	148	85	
Pneumatic System	0			
Electrical System	190	904	436	
Electronics	80	452	273	
APU	0	136	0	
Air Cond. System	104	642	82	
Anti-icing System				
Furnishings	459	428	271	
Auxiliary Gear	7	0	16	
Fixed Equipm't Total	1,060	3,361	1,610	6,834
$W_{oil} + W_{tfo}$	92	420	406	
Water	0	225	0	
Max. Fuel Capacity	1,080	5,452		15,540
Payload (Max.)	1,090	16,000	7,344	

*Military version of Twin Bonanza
**Tailbooms included

Table A10.3b Group Weight Data for Piston/Propeller Driven Military Transports

Type	Beech L-23F*	Chase C-123B	DeHavill. Caribou	Fairchild C-119B
Flight Design Gross Weight, GW, lbs	7,368	54,000	26,000	64,000
Structure/GW	0.307	0.347	0.330	0.349
Power Plant/GW	0.219	0.148	0.181	0.187
Fixed Equipm't/GW	0.144	0.062	0.062	0.107
Empty Weight/GW*	0.670	0.558	0.635	0.641
Wing Group/GW	0.094	0.114	0.113	0.113
Empenn. Group/GW	0.021	0.020	0.030	0.019
Fuselage Group/GW	0.092	0.144	0.110	0.112
Nacelle Group/GW	0.038	0.030	0.030	0.040**
Land. Gear Group/GW	0.061	0.039	0.047	0.066
Take-off Gross Wht, W_{TO}, lbs	7,368	52,802	26,000	64,000
Empty Weight, W_E, lbs	4,936	30,108	16,500	41,017
Wing Group/S, psf	2.5	5.0	3.2	5.0
Emp. Grp/S_{emp}, psf	1.7	2.1	1.9	2.7
Ultimate Load Factor, g's	6.6	4.5	5.1	
Surface Areas, ft^2				
Wing, S	277	1,223	912	1,447
Horiz. Tail, S_h	29.3		206	297
Vert. Tail, S_v	65.1		211	151
Empenn. Area, S_{emp}	94.4	520	417	448

*This is a military version of the Twin Bonanza
**Tailbooms included

Table A10.4a Group Weight Data for Piston/Propeller Driven Military Transports

Type	Douglas C-124C	Boeing C-97C	Lockheed C-69	C-121A
Number of engines:	4	4	4	4
Weight Item, lbs				
Wing Group	18,135	15,389	9,466	11,184
Empennage Group	3,025	2,078	2,026	2,094
Fuselage Group	18,073	13,572	6,794	8,520
Nacelle Group	6,119	4,753	2,505	3,970
Land. Gear Group	11,701	7,112	4,481	4,771
Nose Gear		963	1,019	1,077
Main Gear		6,149	3,462	3,694
Structure Total	57,053	42,904	25,272	30,539
Engine(s)	15,551	13,844	10,568	11,536
Air Induct. System	4,046			
Fuel System	4,059			
Propeller Inst.	4,363			
Propulsion System	in prop.			
Power Plant Total	28,019	23,051	15,633	15,676
Avionics + Instrum.	769			
Surface Controls	1,493			
Hydraulic System	582			
Pneumatic System				
Electrical System	1,952			
Electronics	1,886			
APU	410			
Air Cond. System*	3,294			
Anti-icing System				
Furnishings	7,539			
Auxiliary Gear	104			
Fixed Equipm't Total	18,029	9,997	7,625	13,710
$W_{oil} + W_{tfo}$	3,389			
Water and Alcohol	522			
Max. Fuel Capacity	22,000	23,094	33,470	41,496
Payload (Max.)	55,262	46,500	12,330	12,550

*Includes pressurization system

Table A10.4b Group Weight Data for Piston/Propeller Driven Military Transports

Type	Douglas C-124C	Boeing C-97C	Lockheed C-69	C-121A
Flight Design Gross Weight, GW, lbs	185,000	150,000	82,000	132,800
Structure/GW	0.308	0.286	0.308	0.230
Power Plant/GW	0.151	0.154	0.191	0.118
Fixed Equipm't/GW	0.097	0.067	0.093	0.103
Empty Weight/GW	0.552	0.506	0.592	0.450
Wing Group/GW	0.098	0.103	0.115	0.084
Empenn. Group/GW	0.016	0.014	0.025	0.016
Fuselage Group/GW	0.098	0.090	0.083	0.064
Nacelle Group/GW	0.033	0.032	0.031	0.030
Land. Gear Group/GW	0.063	0.047	0.055	0.036
Take-off Gross Wht, W_{TO}, lbs	185,000	150,000	82,000	132,800
Empty Weight, W_E, lbs	102,181	75,974	48,530	59,715
Wing Group/S, psf	7.2	8.7	5.7	6.8
Emp. Grp/S_{emp}, psf	2.6	3.3	2.9	3.0
Ultimate Load Factor, g's	3.75	3.75	3.75	3.75
Surface Areas, ft^2				
Wing, S	2,506	1,769	1,650	1,650
Horiz. Tail, S_h	681	333	464	464
Vert. Tail, S_v	465	306	262	242
Empenn. Area, S_{emp}	1,146	639	706	706

Table A10.5a Group Weight Data for Military Patrol Airplanes

Type →	Grumman S2F-1	Lockheed P2V-4	Lockheed U2
Number of engines:	2	2	1
Weight Item, lbs	Piston/Propeller		Jet
Wing Group	2,902	7,498	2,034
Empennage Group	681	1,589	320
Fuselage Group	1,701	5,155	1,410
Nacelle Group	965	2,303	0
Land. Gear Group	1,396	3,715	269
Nose Gear		tail gear	60
Main Gear			203
Structure Total	7,645	20,260	4,027
Engines	2,953	5,726	4,076
Propellers	866	1,137	
Fuel System	215	2,827	311
Propulsion System	390	1,633	479**
Power Plant Total	4,424	11,323	4,866
Avionics + Instrum.	147	194	57
Surface Controls	714	960	362
Hydraulic System	208	284	66
Pneumatic System			
Electrical System	988	1,503	290
Electronics	2,310	2,903	166
Armament	256	1,705	
Air Cond. System*	356	496	135
Anti-icing System			
Furnishings	657	1,327	82
Auxiliary Gear	281	0	193
Fixed Equipm't Total	5,917	9,372	1,351
$W_{oil} + W_{tfo}$	332	1,640	97
Engine oil			120
Armament Provisions	18	5,951	
Water	0	1,480	
Max. Fuel Capacity	3,126	14,006	5,810
Payload (Max.)	1,938		518
Crew	N.A.	N.A.	285

*Incl. press. system **Incl. air induction and exhausts

Table A10.5b Group Weight Data for Military Patrol Airplanes

Type	Grumman S2F-1	Lockheed P2V-4	Lockheed U2
	Piston/Propeller		Jet
Flight Design Gross Weight, GW, lbs	23,180	67,500	17,000
Structure/GW	0.330	0.300	0.237
Power Plant/GW	0.191	0.168	0.286
Fixed Equipm't/GW	0.255	0.139	0.079
Empty Weight/GW	0.775	0.607	0.603
Wing Group/GW	0.125	0.111	0.120
Empenn. Group/GW	0.029	0.024	0.019
Fuselage Group/GW	0.073	0.076	0.083
Nacelle Group/GW	0.042	0.034	
Land. Gear Group/GW	0.060	0.055	0.015
Take-off Gross Wht, W_{TO}, lbs	24,167	67,500	19,913
Empty Weight, W_E, lbs	17,953	40,955	10,244
Wing Group/S, psf	6.0	7.5	3.4
Emp. Grp/S_{emp}, psf	3.5	3.9	2.3
Ultimate Load Factor, g's	4.5	4.0	3.75
Surface Areas, ft^2			
Wing, S	485	1,000	600
Horiz. Tail, S_h	103	241	90
Vert. Tail, S_v	90.2	170	49
Empenn. Area, S_{emp}	193	411	139

Table A11.1a Group Weight Data for Flying Boats,

Amphibious and Float Airplanes

Type	At the time of printing no data were available
Number of engines:	
Weight Item, lbs	
Wing Group	
Empennage Group	
Fuselage Group	
Nacelle Group	
Land. Gear Group	
Nose Gear	
Main Gear	

Structure Total	

Engines	
Propellers	
Fuel System	
Propulsion System	

Power Plant Total	

Avionics + Instrum.	
Surface Controls	
Hydraulic System	
Pneumatic System	
Electrical System	
Electronics	
Armament	
Air Cond. System	
Anti-icing System	
Furnishings	
Auxiliary Gear	

Fixed Equipm't Total	

$W_{oil} + W_{tfo}$	
Armament Provisions	
Water	
Max. Fuel Capacity	
Payload (Max.)	

Table A11.1b Group Weight Data for Flying Boats,

Amphibious and Float Airplanes

Type	At the time of printing no data were available
Flight Design Gross Weight, GW, lbs	
Structure/GW	
Power Plant/GW	
Fixed Equipm't/GW	
Empty Weight/GW	
Wing Group/GW	
Empenn. Group/GW	
Fuselage Group/GW	
Nacelle Group/GW	
Land. Gear Group/GW	
Take-off Gross Wht, W_{TO}, lbs	
Empty Weight, W_E, lbs	
Wing Group/S, psf	
Emp. Grp/S_{emp}, psf	
Ultimate Load Factor, g's	
Surface Areas, ft^2	
Wing, S	
Horiz. Tail, S_h	
Vert. Tail, S_v	
Empenn. Area, S_{emp}	

Table A12.1a Group Weight Data for Supersonic Cruise Airplanes

Type	AST-100 *	SSXJET **	Super-cruiser ***
Number of engines:	4	2	2
Weight Item, lbs			
Wing Group	85,914	3,599	3,962
Empennage Group	10,655	481	225
Fuselage Group	52,410	3,494	2,195
Nacelle Group	16,803	505	700
Land. Gear Group	27,293	1,391	1,300
Nose Gear			
Main Gear			
Structure Total	193,075	9,470	8,382
Engines	52,000	3,016	4,781
Air Induct. System	incl. in propulsion system		
Fuel System	5,781	626	560
Propulsion System	1,780	59	165
Power Plant Total	59,561	3,701	6,756
Avionics + Instrum.	6,090	660	1,484
Surface Controls	9,405	564	1,207
Hydraulic System	5,600	266	302
Pneumatic System			
Electrical System	5,050	445	357
Electronics	incl. in avionics + instrum.		
Armament	0	0	600
Air Cond. System****	8,200	352	250
Anti-icing System	210	95	
Furnishings	25,111	417	242
Auxiliary Gear	0	0	40
Fixed Equipm't Total	59,666	2,799	4,482
$W_{oil} + W_{tfo}$	3,050	133	
Mission Fuel Reqd.	327,493	18,674	12,523
Payload	61,028	725	5,000

*NASA TM X-73936, M=2.2 large passenger transport
**NASA TM 74055, M=2.2 executive (business) jet
***NASA TM 78811, M=2.6 military missile carrying super cruiser
****Includes pressurization system

Table A12.1b Group Weight Data for Supersonic Cruise Airplanes

Type	AST-100 *	SSXJET **	Super-cruiser ***
Flight Design Gross Weight, GW, lbs	718,000	35,720	37,144
Structure/GW	0.269	0.265	0.226
Power Plant/GW	0.083	0.104	0.182
Fixed Equipm't/GW	0.083	0.078	0.121
Empty Weight/GW	0.435	0.447	0.528
Wing Group/GW	0.120	0.101	0.107
Empenn. Group/GW	0.015	0.013	0.006
Fuselage Group/GW	0.073	0.098	0.059
Nacelle Group/GW	0.023	0.014	0.019
Land. Gear Group/GW	0.038	0.039	0.035
Take-off Gross Wht, W_{TO}, lbs	718,000	35,720	47,900
Empty Weight, W_E, lbs	312,302	15,970	19,620
Wing Group/S, psf	8.6	3.7	10.7
Emp. Grp/S_{emp}, psf	11.0	3.9	3.1
Ultimate Load Factor, g's	3.75	3.75	6.0
Surface Areas, ft^2			
Wing, S	9,969	965	371
Horiz. Tail, S_h	579	62	0
Vert. Tail, S_v	386	62	73
Empenn. Area, S_{emp}	965	124	73

*NASA TM X-73936, M=2.2 large passenger transport
**NASA TM 74055, M=2.2 executive (business) jet
***NASA TM 78811, M=2.6 military missile carrying super cruiser

Table A13.1a Group Weight Data for NASA X Airplanes

Type	Ryan X-13*	North American X-15**	Hiller X-18***	Bell XV-15
Number of engines:	1	1	2	2
Weight Item, lbs	Jet	Rocket	TBP	TBP Tiltrotor
Wing Group	515	1,144	3,483	946
Empennage Group	146	1,267	928	259
Fuselage Group	415	3,806	4,694	1,589
Engine Section	69	187	728	nac. 369
Land. Gear Group	300	389	1,289	524
Nose Gear				
Main Gear				
Structure Total	1,445	6,793	11,122	3,687
Engine(s)	2,766	680	5,460	1,052
Fuel System	100	1,354	623	226
Propeller Inst.	0	0	3,679	863
Propulsion System	227	148	548	222
Drive system				1,340
Propeller Controls	0	0	2,023	****629
Power Plant Total	3,093	2,182	12,333	4,332
Avionics + Instrum.	41	172	141	231
Surface Controls	416	1,182	896	****777
Hydraulic System	214	240	863	util. 86
Electrical System	311	142	931	418
Electronics	29	175	63	
Test Instrumentation	139	1,328		1,160
Ballast				106
Air Cond. System	10	192		119
Furnishings	199	446	813	434
Auxiliary Gear	0	11	0	10
Fixed Equipm't Total	1,359	3,888	3,707	3,341
$W_{oil} + W_{tfo}$	43	0	353	32
Liquid Nitrogen		150	engine oil:	53
Max. Fuel Capacity	1,400	314	823	1,401
Payload	crew and test equipment only			

*Delta configuration, took off from vertical position
**Air launched by B-52
***Turbo/propeller driven, wing incidence variable over more than 90 degrees
****hydraulic system incl. in rotor and surface ctrls

Table A13.1b Group Weight Data for NASA X Airplanes

Type	Ryan X-13*	North American X-15**	Hiller X-18***	**** Bell XV-15
Flight Design Gross Weight, GW, lbs	7,000	13,592	33,000	13,226
Structure/GW	0.206	0.500	0.337	0.279
Power Plant/GW	0.442	0.161	0.374	0.328
Fixed Equipm't/GW	0.194	0.286	0.112	0.253
Empty Weight/GW	0.822	0.949	0.826	0.859
Wing Group/GW	0.074	0.084	0.106	0.072
Empenn. Group/GW	0.021	0.093	0.028	0.020
Fuselage Group/GW	0.059	0.280	0.142	0.120
Engine Section/GW	0.010	0.014	0.022	0.028
Land. Gear Group/GW	0.043	0.029	0.039	0.040
Take-off Gross Wht, W_{TO}, lbs	7,149	13,592	33,000	13,226
Empty Weight, W_E, lbs	5,755	12,901	27,272	11,360
Wing Group/S, psf	2.7	10.9	6.6	5.6
Emp. Grp/S_{emp}, psf	2.3	10.0	2.8	2.6
Ultimate Load Factor, g's	6.0	11.0		
Surface Areas, ft^2				
Wing, S	191	105	528	169
Horiz. Tail, S_h	0	52	201	50.3
Vert. Tail, S_v	62.8	74.9	133	50.5
Empenn. Area, S_{emp}	62.8	127	334	101

*Delta configuration, took off from vertical position
**Air launched by B-52
***Turbo/propeller driven, wing incidence variable over more than 90 degrees
****Tiltrotor research airplane

Table A13.2a Group Weight Data for NASA X Airplanes

Type	Bell X-2*	Bell X-5**	Northrop YP-61***	**** Bell XP-77
Number of engines:	1	1	2	1
Weight Item, lbs	Rocket	Jet	Piston/ Prop.	Piston/ Prop.
Wing Group	2,856	1,683	3,969	463
Empennage Group	445	198	629	59
Fuselage Group	4,108	1,064	1,557	218
Engine Section	30	274	1,817 incl.booms	123
Land. Gear Group	421	532	1,803	344
Nose Gear	108	464	303	123
Main Gear (skids)	313	68	1,500	221
Structure Total	8,281	3,751	9,775	1,207
Engine(s)	607	2,223	4,974	738
Air Induction System		31		
Fuel System	898	108	914	91
Propulsion System	13	77	1,081	101
Propellers			1,111	206
Power Plant Total	1,518	2,439	8,080	1,136
Avionics + Instrum.	65	35	119	37
Surface Controls	364	195	400	42
Hydraulic System	442	139	240	0
Electrical System	604	127	668	92
Electronics	63	86	721	99
Anti-icing System			100	
Test Instrumentation	708	155		
Armament(incl. guns and ammo)			3,364	391
Ballast		98		
Air Cond. System	102	69		
Furnishings	158	84	252	56
Auxiliary Gear	0	90	352	15
Fixed Equipm't Total	2,506	1,078	6,216	732
$W_{oil} + W_{tfo}$	484		152	30
Liquid Oxygen	7,180			
Liquid Nitrogen	26	oil: 23	oil: 270	28
Max. Fuel Capacity	5,716	1,200	3,168	312
Payload	crew and test equipment only			

*Air launched by B50 **Variable sweep wing
***Twin boom fighter ****Wood built lightweight fighter

Table A13.2b Group Weight Data for NASA X Airplanes

Type	Bell X-2*	Bell X-5**	Northrop YP-61***	**** Bell XP-77
Flight Design Gross Weight, GW, lbs	25,627	8,737	27,813	3,632
Structure/GW	0.323	0.429	0.351	0.332
Power Plant/GW	0.059	0.279	0.291	0.313
Fixed Equipm't/GW	0.098	0.123	0.223	0.202
Empty Weight/GW	0.480	0.832	0.865	0.847
Wing Group/GW	0.111	0.193	0.143	0.127
Empenn. Group/GW	0.017	0.023	0.023	0.016
Fuselage Group/GW	0.160	0.122	0.056	0.060
Engine Section/GW	0.001	0.091	0.065	0.034
			incl.booms	
Land. Gear Group/GW	0.016	0.061	0.065	0.095
Take-off Gross Wht, W_{TO}, lbs	25,627	8,737	27,813	3,632
Empty Weight, W_E, lbs	12,305	7,268	24,071	3,075
Wing Group/S, psf	11.0	9.6	6.0	4.6
Emp. Grp/S_{emp}, psf	4.1	3.4	3.0	2.1
Ultimate Load Factor, g's	not available			
Surface Areas, ft^2				
Wing, S	260	175	664	100
Horiz. Tail, S_h	49.5	31.8	120	18.7
Vert. Tail, S_v	58	25.8	92	9.0
Empenn. Area, S_{emp}	108	57.6	212	27.7

*Air launched by B50 **Variable sweep wing
***Twin boom fighter ****Wood built lightweight fighter

Table A13.3a Group Weight Data for NASA X Airplanes

Type	Mc Donnell XF-88A	Convair XF-92A**	NAA YF-93A***	Convair XFY-1
Number of engines:	2	1	1	1
Weight Item, lbs	Jet	Jet	Jet	TBP****
Wing Group	2,048	1,694	2,640	1,877
Empennage Group	472	590	444	623
Fuselage Group	3,267	2,149	2,850	1,084
Engine Section	29	0	44	157
Land. Gear Group	986	764	1,382	466
Nose Gear	193	155	254	gears on
Main Gear	793	609	1,128	four fins
Structure Total	6,802	5,197	7,360	4,207
Engine(s)	2,942*	2,254	2,787	2,935
Fuel System	920	362	1,520	185
Propeller Inst.	0	0		1,937
Propulsion System	261	198	396	470
Power Plant Total	4,123	2,814	4,703	5,527
Avionics + Instrum.	54	29	155	64
Surface Controls	509	672	686	364
Hydraulic System	307	406	210	214
Electrical System	662	408	488	377
Electronics	218	75	286	120
Test instrumentation				428
Ballast	337			230
Armament	479		1,008	
Guns or cannons	750		639	97
Air Cond. System	87	65	170	76
Furnishings	184	108	207	196
Miscellanous		30(paint)		121
Fixed Equipm't Total	3,587	1,793	3,849	2,287
$W_{oil} + W_{tfo}$	44	N.A.	50	102
Engine oil	75	23	18	94
Max. Fuel Capacity	4,404	4,440	10,593	1,839
Payload	829(ammo)	N.A.	863(ammo)	0
Crew	230	230	230	200

*Includes afterburners **Delta wing configuration
***F-86 modified with NACA flush side inlets
****Counter-rotating propeller driven tailsitter (VTOL)

Table A13.3b Group Weight Data for NASA X Airplanes

Type	Mc Donnell XF-88A	Convair XF-92A	NAA YF-93A	Convair XFY-1
Flight Design Gross Weight, GW, lbs	20,098	11,600	21,846	14,250
Structure/GW	0.338	0.448	0.337	0.295
Power Plant/GW	0.205	0.243	0.215	0.388
Fixed Equipm't/GW	0.178	0.155	0.176	0.160
Empty Weight/GW	0.722	0.845	0.728	0.844
Wing Group/GW	0.102	0.146	0.121	0.132
Empenn. Group/GW	0.023	0.051	0.020	0.044
Fuselage Group/GW	0.163	0.185	0.130	0.076
Engine Section/GW	0.001	0.000	0.002	0.011
Land. Gear Group/GW	0.049	0.066	0.064	0.033
Take-off Gross Wht, W_{TO}, lbs	20,098	11,600	27,788	15,185
Empty Weight, W_E, lbs	14,512	9,804	15,912	12,021
Wing Group/S, psf	5.9	4.0	8.6	5.3
Emp. Grp/S_{emp}, psf	4.3	7.8	5.6	3.5
Ultimate Load Factor, g's	11.0	11.0	11.0	11.3
Surface Areas, ft^2				
Wing, S	350	425	306	355
Horiz. Tail, S_h	N.A.	0	N.A.	N.A.
Vert. Tail, S_v	N.A.	76.0	N.A.	N.A.
Empenn. Area, S_{emp}	109	76.0	79.6	176

Table A13.4a Group Weight Data for NASA X Airplanes

Type	Lockheed XV-4A*	Lockheed XV-4B**	Ryan XV-5A***	**** Bell X-22A
Number of engines:	2	6	4	4
Weight Item, lbs	Jet	Jet	Jet + liftfan	Turbo-shaft
Wing Group	350	395	1,059	789
Empennage Group	170	167	267	131
Fuselage Group	1,207	1,274	1,341	1,324
Engine Section	245	333	45	610
Land. Gear Group	291	389	482	432
Nose Gear	57	77	82	94
Main Gear	234	312	400	338
Structure Total	2,263	2,558	3,194	3,286
Engine(s) (main)	872	758	913	1,191
Engine(s) (lift)	0	1,500	1,855	
Exhaust system	520	608	304	8
Fuel System	155	142	124	175
Propeller Inst. including drives:				2,147
Ducts and supports: fwd 695, aft 686, total:				1,381
Propulsion System	101	88	80	246
Power Plant Total	1,648	3,096	3,276	5,148
Avionics + Instrum.	73	133	73	121
Surface Controls	486	655	440	1,256
Hydraulic System	62	116	115	162
Electrical System	376	394	196	376
Electronics	29	35	40	237
Test instrumentation	583	200	515	1,520
Auxiliary gear	52	27	158	10
Air Cond. System	32	58	34	45
Furnishings	209	391	235	376
Fixed Equipm't Total	1,902	2,009	1,806	4,103
$W_{oil} + W_{tfo}$	40	30	29	72
Engine oil	20	62	12	22
Max. Fuel Capacity	1,147	3,815	2,430	2,031
Crew	180	430	180	360
Payload				1,200

*Ejector type VTOL **Lift engine type VTOL Liftfan research airplane ****Tiltrotor research airplane

Table A13.4b Group Weight Data for NASA X Airplanes

Type	Lockheed XV-4A	Lockheed XV-4B	Ryan XV-5A	Bell X-22A
Flight Design Gross Weight, GW, lbs	7,200	12,000	9,200	14,700
Structure/GW	0.314	0.213	0.347	0.224
Power Plant/GW	0.229	0.258	0.356	0.350
Fixed Equipm't/GW	0.264	0.167	0.196	0.279
Empty Weight/GW	0.807	0.639	0.900	0.853
Wing Group/GW	0.049	0.033	0.115	0.054
Empenn. Group/GW	0.024	0.014	0.029	0.009
Fuselage Group/GW	0.168	0.106	0.146	0.090
Engine Section/GW	0.034	0.028	0.005	0.041
Land. Gear Group/GW	0.040	0.032	0.052	0.029
Take-off Gross Wht, W_{TO}, lbs	7,200	12,000	9,972	14,700
Empty Weight, W_E, lbs	5,813	7,663	8,276	12,537
Wing Group/S, psf	3.4	3.8	4.1	4.9
Emp. Grp/S_{emp}, psf	3.2	3.1	2.6	1.5
Ultimate Load Factor, g's	7.5	4.5	6.0	4.5
Surface Areas, ft^2				
Wing, S	104	104	260	160
Horiz. Tail, S_h	26.4	26.4	52.9	20
Vert. Tail, S_v	27.5	27.5	51.0	68.5
Empenn. Area, S_{emp}	53.9	53.9	104	88.5

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APPENDIX B: DATA SOURCE FOR NON-DIMENSIONAL RADII OF GYRATION FOR AIRPLANES

The purpose of this appendix is to present tabulated data for non-dimensional radii of gyration of airplanes. Actual moments of inertia can be estimated from these non-dimensional radii of gyration with the help of Equations 3.7 through 3.8.

The tables are organized as follows:

Table B1:	Homebuilt propeller driven airplanes
Table B2:	Single engine propeller driven airplanes
Table B3:	Twin engine propeller driven airplanes
Table B4:	Agricultural airplanes
Table B5:	Business jets
Table B6:	Regional turbopropeller driven airplanes
Table B7a:	Jet transports
Table B7b:	Piston-propeller driven transports
Table B7c:	Turbopropeller driven transports
Table B8:	Military trainers
Table B9a:	Fighters (Jet)
Table B9b:	Fighters (Propeller)
Table B10a:	Bombers (Piston-Propeller)
Table B10b:	Bombers (Jet)
Table B10c:	Military patrol airplanes (Propeller)
Table B10d:	Military transports (Propeller)
Table B11:	Flying boats
Table B12:	Supersonic cruise airplanes

The data in all these table were derived from manufacturers data and/or from Ref.11.

Table B1 Non-dimensional Radii of Gyration for Homebuilt Propeller Driven Airplanes

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition

At the time of printing, no data were available for this type airplane

Table B2 Non-dimensional Radii of Gyration for Single Engine Propeller Driven Airplanes

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Beech N-35*	3,125	32.8	25.1	29.0	0.248	0.338	0.393	1 in fusel.
Cessna 150M**	1,127	33.5	21.5	27.5	0.254	0.405	0.418	1 in fusel.
Cessna 172M**	1,477	36.2	26.5	31.4	0.242	0.386	0.403	1 in fusel.
Cessna 177A**	1,761	35.6	27.0	31.3	0.212	0.362	0.394	1 in fusel.
Cessna R182**	1,885	36.2	28.0	32.1	0.342	0.397	0.393	1 in fusel.
Cessna 210K***	2,700	36.8	28.3	32.6	0.222	0.356	0.379	1 in fusel.

*at W_{TO} **at W_{OE} ***at W_{OE} plus 25 percent fuel

Note: one pilot included in all data

Table B3 Non-dimensional Radii of Gyration for Twin Engine Propeller Driven Airplanes

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Beech 55	4,880	37.8	25.7	31.8	0.260	0.329	0.399	2 on wing
Beech 95	4,000	37.8	25.3	31.6	0.251	0.327	0.391	2 on wing
Beech D-50	6,500	45.9	31.5	38.7	0.240	0.313	0.384	2 on wing
Beech D18S	9,000	47.7	34.2	41.0	0.232	0.360	0.396	2 on wing
Cessna 402*	5,000	39.9	36.3	38.1	0.414	0.278	0.502	2 on wing
Cessna 402	6,200	39.9	36.3	38.1	0.373	0.269	0.461	2 on wing
Cessna 404*	4,851	46.7	39.5	43.1	0.324	0.318	0.446	2 on wing
Cessna 404	8,400	46.7	39.5	43.1	0.340	0.284	0.445	2 on wing
Cessna 441*	5,642	49.3	39.0	44.2	0.285	0.345	0.429	2 on wing
Cessna 441	9,925	49.3	39.0	44.2	0.256	0.212	0.336	2 on wing

*at W_E

Table B4 Non-dimensional Radii of Gyration for Agricultural Airplanes

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition

At the time of printing, no data were available for this type of airplane

Table B5 Non-dimensional Radii of Gyration for Business Jets

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Morane/S 760	7,066	33.3	32.9	33.1	0.374	0.328	0.486	2 in W/F
Lockh. Jetstar	39,288	53.7	58.8	56.3	0.370	0.356	0.503	4 on fusel.
Cessna 500*	6,505	47.1	43.5	45.3	0.236	0.384	0.430	2 on fusel.
Cessna 500**	12,000	47.1	43.5	45.3	0.306	0.303	0.423	2 on fusel.
Cessna 550*	7,036	51.7	47.2	49.5	0.243	0.400	0.447	2 on fusel.
Cessna 550**	13,500	51.7	47.2	49.5	0.293	0.312	0.420	2 on fusel.

*at W_e **at W_{TO}

Table B6 Non-dimensional Radii of Gyration for Regional Turbopropeller Driven Airplanes

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Fokker F-27A	38,500	95.2	77.2	86.2	0.235	0.363	0.416	2 on wing
DHC6 Twin Otter	12,500	65.0	51.8	58.4	0.203	0.326	0.350	2 on wing

Table B7a Non-dimensional Radii of Gyration for Jet Transports

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Convair 880	185,000	120.0	124.2	122.1	0.320	0.342	0.465	4 on wing
Convair 880	191,500	120.0	124.2	122.1	0.322	0.339	0.464	4 on wing
Convair 990	240,000	120.0	134.8	127.4	0.335	0.338	0.473	4 on wing
Convair 990	245,000	120.0	134.8	127.4	0.305	0.334	0.472	4 on wing
Boeing 727-100	165,000	108.0	133.2	120.6	0.249	0.375	0.452	3 on fusel.
Boeing 727-100*	89,000	108.0	133.2	120.6	0.247	0.442	0.518	3 on fusel.
Boeing 727-200	180,000	108.0	153.2	130.6	0.248	0.394	0.502	3 on fusel.
Boeing 727-200*	100,000	108.0	153.2	130.6	0.240	0.451	0.550	3 on fusel.
Boeing 737-200	113,000	93.0	100.0	96.5	0.246	0.382	0.456	2 on wing
Boeing 737-200*	62,000	93.0	100.0	96.5	0.264	0.456	0.517	2 on wing
Boeing 747-100B	800,000	195.7	231.3	213.5	0.290	0.329	0.445	4 on wing
Boeing 747-100B*	350,000	195.7	231.3	213.5	0.332	0.380	0.508	4 on wing
McDD DC9-10	74,000	89.4	104.3	96.9	0.242	0.360	0.435	2 on fusel.
McDD DC8	210,000	142.4	150.5	146.5	0.301	0.349	0.434	4 on wing

*at W_{OE}

Table B7b Non-dimensional Radii of Gyration for Piston-Propeller Driven Transports

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Lockheed L-749A	107,000	123.0	95.2	109.1	0.300	0.298	0.426	4 on wing
Lockheed L-1049	120,000	123.0	113.6	118.3	0.316	0.336	0.448	4 on wing
Lockheed L-1649	146,500	150.0	116.2	133.1	0.371	0.278	0.473	4 on wing
Douglas DC-4	60,360	138.3	97.6	118.0	0.250	0.320	0.388	4 on wing
Douglas DC-6	97,200	117.5	100.5	109.0	0.322	0.324	0.456	4 on wing
Airspeed Ambass.	49,500	115.0	80.4	97.7	0.278	0.314	0.400	2 on wing
Martin 404	45,000	93.3	74.6	84.2	0.272	0.378	0.444	2 on wing
Convair T-240	41,800	91.7	74.7	83.2	0.286	0.351	0.443	2 on wing
Convair T-340	44,500	105.7	79.2	92.4	0.308	0.345	0.457	2 on wing
Beech Twin Quad	20,000	70.0	52.7	61.4	0.225	0.303	0.346	4 in wing

Table B7c Non-dimensional Radii of Gyration for Turbo-Propeller Driven Transports

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Bristol 175(k)*	103,000	130.0	110.0	120.0	0.317	0.356	0.455	4 on wing
Bristol 167(1)**	187,000	230.0	177.0	203.5	0.330	0.356	0.478	4 on wing
Lockh. Electra	116,000	99.0	104.7	101.9	0.394	0.341	0.497	4 on wing

*Britannia **Brabazon

Table B8 Non-dimensional Radii of Gyration for Military Trainers

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2 ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Cessna T-37A	6,300	38.4	30.0	34.2	0.220	0.142	0.245	2 in fusel.

Table B9a Non-dimensional Radii of Gyration for Fighters (Jet)

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
McD F2H-1	14,413	41.6	40.2	40.9	0.230	0.359	0.465	2 in W/F
McD F3H-2N	26,878	35.3	58.8	47.1	0.252	0.107	0.449	1 in fusel.
McD F-101A	36,969	39.7	67.4	53.6	0.209	0.329	0.428	2 in fusel.
VS Attacker	10,450	36.9	37.3	37.1	0.244	0.328	0.400	1 in fusel.
DH Vampire 20	10,891	40.0	30.1	35.1	0.286	0.318	0.409	1 in fusel.
Gl. Meteor II	11,100	43.0	41.4	42.2	0.286	0.330	0.404	2 in wing
Lockheed F-80A	11,940	38.9	34.3	36.6	0.286	0.356	0.444	1 in fusel.
Lockheed F-94B	13,650	37.5	40.1	38.8	0.284	0.396	0.488	1 in fusel.
Lockheed F-104G	20,900	21.9	54.8	38.4	0.224	0.392	0.563	1 in fusel.
NAA F-86A	13,900	37.1	37.5	37.3	0.266	0.346	0.400	1 in fusel.
NAA FJ-3	16,883	37.0	37.6	37.3	0.281	0.352	0.438	1 in fusel.
NAA F-100D	29,800	38.0	47.0	42.5	0.252	0.376	0.462	1 in fusel.
Vought XF8U-1	21,300	35.7	54.4	45.1	0.225	0.404	0.507	1 in fusel.
Vought F8U-3	30,600	40.0	58.9	49.5	0.225	0.375	0.467	1 in fusel.
GD XF-91	18,600	31.3	43.3	37.3	0.323	0.424	0.548	1 in fusel.
GD TF-102A	32,859	38.1	63.2	50.7	0.295	0.386	0.520	1 in fusel.
GD F-106B	36,834	38.3	70.7	54.5	0.247	0.379	0.516	1 in fusel.
Northrop F-89D	38,000	58.0	54.0	56.0	0.440	0.304	0.532	2 in fusel.
Republic RF-84F	19,000	33.6	47.5	40.6	0.310	0.308	0.432	1 in fusel.
Republic F-105D	34,058	35.0	64.4	49.7	0.231	0.425	0.567	1 in fusel.
Grumman F9F-8	16,744	34.5	41.9	38.2	0.248	0.374	0.454	1 in fusel.
Grumman XF10F-1	26,160	36.8	57.8	46.9	0.251	0.323	0.414	1 in fusel.
Grumman F11F-1	16,500	31.6	40.8	36.2	0.221	0.404	0.484	1 in fusel.

Table B9b Non-dimensional Radii of Gyration for Fighters (Propeller)

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Brewster Buffalo	5,066	35.0	26.0	30.5	0.208	0.358	0.374	1 in fusel.
Seversky P35	5,788	36.0	26.8	31.4	0.198	0.367	0.360	1 in fusel.
VS Spitfire-I	6,250	36.8	29.9	33.4	0.240	0.334	0.384	1 in fusel.
BP Defiant	6,410	39.4	35.0	37.2	0.234	0.360	0.404	1 in fusel.
Curtiss P36	6,825	37.3	31.7	34.5	0.172	0.356	0.370	1 in fusel.
Bell P39	7,533	34.0	30.0	32.0	0.276	0.340	0.425	1 in fusel.
Grumman F6F	10,560	42.8	33.5	38.2	0.242	0.346	0.404	1 in fusel.
Hawker Typhoon	11,017	41.5	31.7	36.6	0.277	0.300	0.394	1 in fusel.
Republic P47	12,500	40.7	36.0	38.4	0.296	0.322	0.428	1 in fusel.
Vought F4U	12,850	41.0	34.5	37.8	0.268	0.360	0.420	1 in fusel.
Bl.Firebr'd-III	13,660	49.8	38.2	44.0	0.250	0.300	0.397	1 in fusel.
Westland Welkin	18,340	70.0	42.0	56.0	0.270	0.304	0.408	2 in wing
Bristol Beauftr	22,635	57.8	42.5	50.2	0.330	0.299	0.447	2 in wing
Bristol Brigand	39,000	71.7	46.4	59.1	0.299	0.338	0.438	2 in wing

Table B10a Non-dimensional Radii of Gyration for Bombers (Piston-Propeller)

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Martin B-26	26,600	65.0	57.6	61.3	0.270	0.320	0.410	2 on wing
HP Halifax	55,000	99.0	71.6	85.3	0.346	0.306	0.395	4 on wing
Shorts Stirling	64,000	99.0	87.3	93.2	0.360	0.330	0.488	4 on wing
Boeing B-29	105,000	141.0	99.0	120.0	0.316	0.320	0.376	4 on wing
Boeing B-50	120,000	141.0	99.0	120.0	0.304	0.332	0.450	4 on wing
GD B-36	357,500	230.0	162.0	196.0	0.316	0.262	0.428	6 in wing

Table B10b Non-dimensional Radii of Gyration for Bombers (Jet)

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Martin XB-51	53,785	53.0	81.0	67.0	0.194	0.404	0.498	3 in/on fus.
Martin B57A	48,554	64.0	66.0	65.0	0.312	0.278	0.412	2 in wing
Boeing XB-47	125,000	116.0	107.0	111.5	0.346	0.320	0.474	4 in wing
Boeing B-52A	390,000	185.0	156.5	170.8	0.346	0.306	0.466	8 on wing
Northrop RB-49A	213,500	172.0	53.0*	112.5	0.316	0.316	0.510	6 in wing
NAA B-45A	82,600	89.0	75.3	82.2	0.325	0.290	0.438	4 on wing
NAA B-45C	82,600	89.0	75.3	82.2	0.340	0.299	0.456	4 on wing

*Flying wing

Table B10c Non-dimensional Radii of Gyration for Military Patrol Airplanes

(Propeller Driven)

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Piston-Propeller Driven								
Lockheed P2V-4	67,500	100.0	81.6	90.8	0.368	0.300	0.484	2 on wing
Lockheed P2V-7*	67,500	100.0	91.7	95.9	0.372	0.266	0.462	4 on wing
Grumman S2F-3	26,147	69.7	43.5	56.6	0.240	0.347	0.387	2 on wing
Grumman W2F-1	41,549	80.6	56.3	68.4	0.235	0.366	0.387	2 on wing
Turbo-propeller Driven								
Lockheed P3V-1	127,200	99.7	116.8	108.3	0.357	0.249	0.421	4 on wing

Table B10d Non-dimensional Radii of Gyration for Military Transports

(Propeller Driven)

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
Piston-Propeller Driven								
Douglas C-54	61,840	117.5	94.0	105.8	0.286	0.294	0.406	4 on wing
Fairchild C-119B	64,000	109.3	88.5	98.9	0.287	0.282	0.390	2 On wing
in wing								
Boeing C-97	128,340	141.2	110.3	125.8	0.276	0.325	0.424	4 on wing
GD XC-99	265,000	230.0	182.5	206.3	0.276	0.346	0.432	6 on wing
Turbo-propeller Driven								
Lockheed C-130B	135,000	132.6	97.8	115.2	0.363	0.319	0.489	4 on wing
Lockheed C-130E	155,000	132.6	97.8	115.2	0.375	o.290	0.486	4 on wing

*Has two jet engines outboard of the piston engines

Table B11 Non-dimensional Radii of Gyration for Flying Boats

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	$\bar{R}_x$	$\bar{R}_y$	$\bar{R}_z$	Number of engines and disposition
XBTM-1	20,009	50.0	41.2	45.6	0.230	0.340	0.380	1 in fusel.
XP4M-1	80,000	114.0	84.0	99.0	0.248	0.320	0.414	2 on wing
VS Seagull	14,230	52.5	44.0	48.3	0.297	0.364	0.402	1 in WF

Table B12 Non-dimensional Radii of Gyration for Supersonic Cruise Airplanes

Airplane Type	GW lbs	Wing Span, b, ft	Total Length, L, ft	e = (b+L)/2, ft	R_x	R_y	R_z	Number of engines and disposition
NAA A3J-1	44,305	53.0	72.5	62.8	0.240	0.372	0.472	2 in fusel.

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AIRPLANE DESIGN

PART VI: PRELIMINARY CALCULATION OF AERODYNAMIC, THRUST AND POWER CHARACTERISTICS

by

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1. GENERAL SYMBOLS

Symbol	Definition	Dimension
a	speed of sound	fps
a.c.	aerodynamic center	-----
$A = b^2/S$	Wing aspect ratio	-----
$A_c = b_c^2/S_c$	Canard aspect ratio	-----
$A_v = b_h^2/S_h$	Vert.tail aspect ratio	-----
A_∞	Inlet capture area	ft^2
A_c	Inlet area	ft^2
$A_{\bar{c}}$	Cowl cross section area at $\bar{d}_c$	ft^2
A_e	Nozzle (exit) area	ft^2
A_f	Internal area	ft^2
A_m	Cowl cross section area at d_m	ft^2
A_t	Nozzle throat area	ft^2
b	wing span	ft
b_c	canard span	ft
b_{f_i}	inboard flap span, p.89	ft

b_{f_o}	outboard flap span, p.89	ft
b_t	tire width	ft
b_v	vertical tail span, see p.387	ft
B	Compressible sweep correction factor, see Eqn.(10.64)	-----
c	chord	ft
c'	chord with flap extended	ft
$\bar{c}$	mean geometric chord	ft
c_b	control surface overhang, see p.471	ft
c_{d_c}	crossflow drag coeff	-----
c_e	elevator chord	ft
c_{δ_c}	canardvator chord	ft
c_f	flap chord	ft
c_h	two-dim. hingemoment coeff. about control surf. h.l.	-----
$c^t{}_h$	two-dim. hingemoment coeff. about tab h.l.	-----
$c^*{}_l$	airfoil lift coefficient at α^*	-----
$c_{m_{ac}}$	airfoil pitching moment coefficient about a.c.	-----
c_{m_o}	airfoil zero-lift pitching moment coeff.	-----
c_p	wing chord at wing pivot also: engine sfs	ft lbs/shp/hr
c_t, c_T	tip chord	ft
	c_t in Ch.10 also stands for tab chord, see p.473	ft
c_r, c_R	root chord	ft
$\bar{c}_{w_e}$	mean geometric chord of exposed wing	ft
C_D	Airplane drag coeff.	-----
C_{D_A}, $C_{D_{N_2}}$	wave drag coefficients see p.49 and 52	-----

$C_{D_{A(NC)}}$	interference drag coeff. see p.52	-----
C_{D_b}	base drag coefficient	-----
$C_{D_{L_i}}$	Drag due to lift coeff	-----
$C_{D_{L_v}}$	Viscous drag due to lift coefficient	-----
$C_{D_{min}}$	Drag coefficient at $C_{L_{min}}$	-----
$C_{D_{min_w}}$	Wing minimum drag coeff.	-----
$C_{D_{i_w}}$	Wing induced drag coeff.	-----
C_{D_o}	Zero-lift drag coeff.	-----
C_{D_p}	Profile drag coefficient	-----
C_f	turbulent flat plate friction coefficient	-----
C_{f_w}	turbulent flat plate friction coefficient of the wing	-----
C_h	Three-dim. hingemoment coeff. about contr. surf. h.l.	-----
$C^t{}_h$	Three-dim. hingemoment coeff. about tab h.l.	-----
c_l	airfoil lift coefficient	-----
c_{l_α}	airfoil lift curve slope	-----
$(c_{l_\alpha})_\delta$	section lift curve slope with the flaps down	1/deg. 1/rad
c_{l_δ}	derivative of airfoil lift coeff. with flap deflect.	-----
C_l	aerodynamic rolling moment coefficient	-----
$C_L = W/\bar{q}S$	Airplane lift coefficient	-----
C_{L_B}	Lift coefficient where drag rise due to separation begins	-----
C_{L_c}	Canard lift coefficient	-----
C_{L_h}	Hor. tail lift coeff.	-----

C_{L_o}	Lift coefficient for zero angle of attack	-----
$C_{L_{min}}$	Lift coefficient at minimum viscous drag due to lift	-----
C_{L_w}	Wing lift coefficient	-----
C_{L_α}	Airplane lift curve slope	1/deg, 1/rad
$C_{L_{\alpha_w}}$	Wing lift-curve slope	1/deg, 1/rad
C_m	Aerodynamic pitching moment coefficient	-----
c_{m_o}	Airfoil pitching moment coefficient at zero lift coefficient	-----
C_n	Aerodynamic yawing moment coefficient	-----
C_{p_b}	base pressure coeff., p.52	-----
C_R	Ram recovery factor	-----
C_y	Aerodynamic side force coefficient	-----
d_b	equivalent base diameter	-----
$\bar{d}_c$	cowl diameter at inlet area	ft
d_{exhnoz}	diam. of exhaust nozzle	in
d_{exhst}	exhaust stack diameter	in
d_f	equivalent fuselage diameter	--
d_{inl}	max. inlet diameter	ft
d_m	max. cowl diameter	ft
d_n	maximum nacelle (equivalent) diameter	ft
d_T	distance of thrustline to center of gravity, see Figure 8.126	ft
D	Drag	lbs
D_p	Propeller diameter	ft
D_t	Max. tire diameter	ft
e	span efficiency factor	-----

	(Also called Oswald's eff. factor)	
f	equivalent parasite area	ft^2
F_t	Factor in Eqn.(6.51)	-----
Δf	incremental eq. par. area	ft^2
F_{A_y}	Aerodynamic side force	lbs
i	incidence angle	deg
I_{xx}	Rolling moment of inertia in body axes	$slugft^2$
I_{xz}	XZ Product of inertia in body axes	$slugft^2$
I_{yy}	Piching moment of inertia in body axes	$slugft^2$
I_{zz}	Yawing moment of inertia in body axes	$slugft^2$
k	ratio of airfoil lift-curve slope to 2π, also:	-----
	equivalent sand roughness	-----
k'	correction factor for non-linear effects	-----
k_{cw}, k_{wc} and k_{wh}	interference factors, see pages 277, 278	-----
K	empirical constant, see p.86	--
K'	factor in Eqn.(3.6)	-----
K''	factor in Eqn.(3.6)	-----
K_B	factor in Eqn.(3.8)	-----
K_Λ	flap span factor for swept wings, p.313	-----
K_p	flap span factor, p.313	-----
K_t	constant in Eqn.(6.50)	-----
K_{wf}	wing-fuselage interference factor: see Eqn.(8.44)	
l	reference length	ft
l_f	fuselage length	ft
l_p	distance in Fig.10.34	ft
l_v	distance in Fig.10.27	ft
L	Lift	lbs
L'	Airfoil thickness location parameter	-----
L/D	Lift-to-drag ratio	-----

L_A	Aerodynamic rolling mom.	lbs
$\dot{m}_a$	engine massflow	slugs/sec
$\dot{m}_{bleed}$	bleed air massflow	slugs/sec
$\dot{m}_{comb}$	combustion massflow	slugs/sec
$\dot{m}_{cool}$	cooling air massflow	slugs/sec
M	Free stream Mach number	-----
M_A	Aerodynamic pitching moment	ftlbs
M_c	Crossflow Mach number	-----
M_{CR}, M_{crit}	Critical Mach number	-----
M_{DD}	Drag divergence Mach number	---
$n=W/\bar{q}S$	airplane load factor	-----
n_p	no. of blades per prop.	-----
N_A	Aerodynamic yawing moment	ftlbs
p	local static pressure	psf
	also: planform shape parameter, p.40	-----
p_{tot}	total pressure	psf
p_t	Factor in Eqn.(6.51)	-----
P_{av}	Power available	hp
P_{el}	Extracted electr. power	hp
P_{extr}	Extracted power	
P_{fp}	Extr. fuel pump power	hp
P_{hydr}	Extr. hydr. pump power	hp
P_{mech}	Extr. mechanical power	hp
P_{other}	Extr. other power	hp
P_{pneum}	Extr. pneumatic power	hp
P_{reqd}	Power required	hp

$\bar{q}$	free stream dyn. pressure	psf
R_N, R_l	Reynold's number	-----
R_{LS}	Lifting surface correction factor	-----
R_{wf}	Wing/fuselage interference factor	-----
S	Wing planform (ref) area	ft^2
$S_{b_{fus}}$	Fuselage base area	ft^2
S_c	Canard area	ft^2
S_{can}	Max. frontal area of canopy	ft^2
S_{cf}	Flapped canard area	ft^2
S_{ef}	Flapped hor.tail area	ft^2
S_{f_s}	Fuselage side area as defined in Fig.10.28	ft^2
S_{wf}	Flapped wing area	ft^2
S_h	Horiz.tail area	ft^2
S_n	Max. frontal area of nacelle	ft^2
S_v	Vertical tail area, see p.387	ft^2
S_{noz}	Nozzle cross section area	ft^2
S_{wet}	Total wetted area	ft^2
$S_{wet_{fus}}$	Wetted area of fuselage	ft^2
$S_{wet_{lam}}$	Wetted area with laminar flow	ft^2
$S_{wet_{turb}}$	Wetted area with turbulent flow	ft^2
S_{wet_w}	Wetted area of wing	ft^2

t/c	thickness ratio (at $\bar{c}$)	-----
T	Temperature of air	deg F
T_{av}	Thrust available	lbs
T_r	Remnant thrust	lbs
T_{reqd}	Thrust required	lbs
U_1 or U	Steady state airspeed	ft/sec
v	induced drag factor due to twist	-----
V_∞	Velocity at sta. A_∞	fps
V_c	Velocity at sta. A_c	fps
$\bar{V}_c$	Canard volume coefficient, see Eqn.(10.74)	-----
$\bar{V}_h$	Hor. tail volume coefficient, see Eqn.(10.23)	-----
VA_{plp}	Required electrical power	hp
$\dot{V}_{hydr}$	Hydr. fluid flow rate	gpm
V_{noz}	Average nozzle flow velocity	fps
w	zero-lift drag factor due to twist	-----
w_f	max. fuselage width	ft
W	Airplane weight	lbs
x_{ac}	position of a.c. on wing mgc,	ft
$\bar{x}_{ac}=x_{ac}/\bar{c}$	position of aerodynamic center on wing mgc, fr.$\bar{c}$	-----
x_{ac_h}	position of hor. tail a.c. on wing mgc	ft
$\bar{x}_{ac_h}=x_{ac}/\bar{c}$	position of hor. tail a.c. on wing mgc, fr.$\bar{c}$	-----
x_{cp}	position of wing center of pressure on wing mgc	ft
x_{ref}	position of reference point on wing mgc	ft

$\bar{x}_{ref}=x_{ref}/\bar{c}$	position of reference point on wing mgc, fr.$\bar{c}$	-----

NOTE: see Figure 8.114 for illustration!

x_w	distance of wing quarter chord mgc to c.g., see Fig.10.39	----
y	spanwise coordinate	ft
z_f	vertical height of fuselage at wing root chord	ft
z_p	distance in Fig.10.34	ft
z_v	distance in Fig.10.27	ft
z_w	wing distance to fuselage centerline, see Fig.(10.9)	ft

2. STABILITY, CONTROL AND HINGEMOMENT DERIVATIVES

NOTE: All derivatives are presented in the airplane stability axes system. See pages 371 and 372 for a definition of axes.

STABILITY DERIVATIVES

Symbol	Definition	Dimension
C_{D_u}	$\partial C_D/\partial(u/U_1)$	-----
C_{L_u}	$\partial C_L/\partial(u/U_1)$	-----
C_{m_u}	$\partial C_m/\partial(u/U_1)$	-----
$C_{T_{x_u}}$	$\partial C_{T_x}/\partial(u/U_1)$	-----
$C_{m_{T_u}}$	$\partial C_{m_T}/\partial(u/U_1)$	-----
C_{D_α}	$\partial C_D/\partial\alpha$	rad^{-1}
C_{L_α}	$\partial C_L/\partial\alpha$	rad^{-1}
C_{m_α}	$\partial C_m/\partial\alpha$	rad^{-1}
$C_{m_{T_\alpha}}$	$\partial C_{m_T}/\partial\alpha$	rad^{-1}
$C_{D_{\dot\alpha}}$	$\partial C_D/\partial(\dot\alpha\bar{c}/2U_1)$	rad^{-1}
$C_{L_{\dot\alpha}}$	$\partial C_L/\partial(\dot\alpha\bar{c}/2U_1)$	rad^{-1}
$C_{m_{\dot\alpha}}$	$\partial C_m/\partial(\dot\alpha\bar{c}/2U_1)$	rad^{-1}

C_{y_β}	$\partial C_y/\partial\beta$	rad^{-1}
C_{l_β}	$\partial C_l/\partial\beta$	rad^{-1}
C_{n_β}	$\partial C_n/\partial\beta$	rad^{-1}
$C_{n_{T_\beta}}$	$\partial C_{n_T}/\partial\beta$	rad^{-1}
$C_{y_{\dot\beta}}$	$\partial C_y/\partial(\dot\beta b/2U_1)$	rad^{-1}
$C_{l_{\dot\beta}}$	$\partial C_l/\partial(\dot\beta b/2U_1)$	rad^{-1}
$C_{n_{\dot\beta}}$	$\partial C_n/\partial(\dot\beta b/2U_1)$	rad^{-1}
C_{y_p}	$\partial C_y/\partial(pb/2U_1)$	rad^{-1}
C_{l_p}	$\partial C_l/\partial(pb/2U_1)$	rad^{-1}
C_{n_p}	$\partial C_n/\partial(pb/2U_1)$	rad^{-1}
C_{D_q}	$\partial C_D/\partial(q\bar{c}/2U_1)$	rad^{-1}
C_{L_q}	$\partial C_L/\partial(q\bar{c}/2U_1)$	rad^{-1}
C_{m_q}	$\partial C_m/\partial(q\bar{c}/2U_1)$	rad^{-1}
C_{y_r}	$\partial C_y/\partial(rb/2U_1)$	rad^{-1}
C_{l_r}	$\partial C_l/\partial(rb/2U_1)$	rad^{-1}
C_{n_r}	$\partial C_n/\partial(rb/2U_1)$	rad^{-1}

<u>CONTROL DERIVATIVES</u>

$C_{D_{i_h}}$	$\partial C_D/\partial i_h$	rad^{-1}
$C_{L_{i_h}}$	$\partial C_l/\partial i_h$	rad^{-1}
$C_{m_{i_h}}$	$\partial C_m/\partial i_h$	rad^{-1}
$C_{D_{\delta_e}}$	$\partial C_D/\partial\delta_e$	rad^{-1}

$C_{L_{\delta_e}}$	$\partial C_L/\partial \delta_e$	rad^{-1}
$C_{m_{\delta_e}}$	$\partial C_m/\partial \delta_e$	rad^{-1}
$C_{D_{i_c}}$	$\partial C_D/\partial i_c$	rad^{-1}
$C_{L_{i_c}}$	$\partial C_L/\partial i_c$	rad^{-1}
$C_{m_{i_c}}$	$\partial C_m/\partial i_c$	rad^{-1}
$C_{D_{\delta_c}}$	$\partial C_D/\partial \delta_c$	rad^{-1}
$C_{L_{\delta_c}}$	$\partial C_L/\partial \delta_c$	rad^{-1}
$C_{m_{\delta_c}}$	$\partial C_m/\partial \delta_c$	rad^{-1}
$C_{y_{\delta_a}}$	$\partial C_y/\partial \delta_a$	rad^{-1}
$C_{l_{\delta_a}}$	$\partial C_l/\partial \delta_a$	rad^{-1}
$C_{n_{\delta_a}}$	$\partial C_n/\partial \delta_a$	rad^{-1}
$C_{y_{\delta_s}}$	$\partial C_y/\partial \delta_s$	rad^{-1}
$C_{l_{\delta_s}}$	$\partial C_l/\partial \delta_s$	rad^{-1}
$C_{n_{\delta_s}}$	$\partial C_n/\partial \delta_s$	rad^{-1}
$C_{y_{i_h}}$	$\partial C_y/\partial i_h$	rad^{-1}
$C_{l_{i_h}}$	$\partial C_l/\partial i_h$	rad^{-1}

$C_{n_{i_h}}$	$\partial C_n/\partial i_h$	rad^{-1}
$C_{y_{\delta_r}}$	$\partial C_y/\partial \delta_r$	rad^{-1}
$C_{l_{\delta_r}}$	$\partial C_l/\partial \delta_r$	rad^{-1}
$C_{n_{\delta_r}}$	$\partial C_n/\partial \delta_r$	rad^{-1}

HINGEMOMENT DERIVATIVES

Two-Dimensional Hingemoment Derivatives

Symbol	Definition	Dimension
c_{h_o}	zero-angle-of-attack, zero-control-surface-deflection, zero-tab-angle-deflection hingemoment coefficient	-----
c_{h_α}	$\partial c_h/\partial \alpha$	rad^{-1}
$c_{h_\delta} = c_{h_{\alpha_{bal}}}$	$\partial c_h/\partial \delta$	rad^{-1}
$c_{h_{\delta_t}}$	$\partial c_{h_t}/\partial \delta_t$	rad^{-1}

Three-Dimensional Hingemoment Derivatives

Symbol	Definition	Dimension
C_{h_o}	zero-angle-of-attack, zero-control-surface-deflection, zero-tab-angle-deflection hingemoment coefficient	-----
C_{h_α}	$\partial C_h/\partial \alpha$	rad^{-1}
C_{h_δ}	$\partial C_h/\partial \delta$	rad^{-1}
$C_{h_{\delta_t}}$	$\partial C_h/\partial \delta_t$	rad^{-1}

NOTE: Control surface hingemoment derivatives defined sofar, are taken about the control surface hingeline. Tab hingemoment derivatives taken about the tab hingeline are defined in a similar manner but carry the superscript 't' as in $c^t{}_{h_x}$ and $C^t{}_{h_x}$

3. GREEK SYMBOLS

Symbol	Definition	Dimension
α	airfoil angle of attack	deg, rad
α	airplane angle of attack with subscript w, h or c, the symbol indicates the angle of attack of that component	deg, rad
α^*	linear range of α	deg, rad
α_{o_l}	airfoil zero-lift angle of attack	deg, rad
α_{o_L}	airplane zero-lift angle of attack	deg, rad
α_δ	derivative of angle of attack w.r.t. surface deflection angle	-----
β	$(1 - M^2)^{1/2}$	-----
	also: sideslip angle	deg, rad
γ	flight path angle	deg, rad
Γ	dihedral angle	deg, rad
δ	pressure ratio,	-----
	also: flap or control surface deflection	deg, rad
δ_f	flap defl. angle	deg, rad
Δ	increment	depends
$\Delta\alpha_g$	change in angle of attack due to ground effect	deg, rad
$\Delta\alpha_{w/c}$	difference in stall angle of attack between wing and canard, see p.280	deg, rad
Δc_l	incremental airfoil lift coefficient due to flaps	-----
ΔC_L	incremental airplane lift coefficient due to flaps	-----
ΔC_{L_c}	incremental canard lift coefficient to trim, based on wing area, S	-----
ΔC_{L_h}	incremental tail lift coefficient to trim, based on wing area, S	-----
Δc_m	incremental airfoil pitching moment coefficient due to flaps	-----
ΔC_m	incremental airplane piching moment coefficient due to flaps	-----
$\Delta\varepsilon_f$	change in tail downwash angle due to flaps	deg, rad
Δp_{hydr}	hydraulic pressure differential	psi

$\Delta\bar{x}_{ac_f}$	shift in aerodynamic center due to the fuselage, fr.$\bar{c}$	-----
Δy	leading edge shape parameter	-----
ϵ	downwash angle at the horizontal tail	deg, rad
ϵ_c	upwash angle at the canard	deg, rad
ϵ_t	wing twist angle	deg, rad
η	span fraction, or see p.46	-----
η_c	ratio of canard to wing dynamic pressure	-----
η_{diff}	pressure recovery through diffuser	-----
η_{fp}	fuel pump efficiency	-----
η_{gear}	gearbox efficiency	-----
η_{gen}	generator efficiency	-----
η_h	ratio of hor.tail to wing dynamic pressure	-----
η_{hp}	hydr. pump efficiency	-----
η_{inl}	inlet pressure recovery	-----
$\eta_{inl/com}$	inlet pressure recovery, compressible	-----
$\eta_{inl/inc}$	inlet pressure recovery, incompressible	-----
η_p	propeller efficiency	-----
η_{shock}	pressure recovery through shock	-----
$\Lambda_{c/2}$	semi-chord sweep angle	deg, rad
$\Lambda_{c/4}$	quarter chord sweep angle	deg, rad
Λ_{LE}	leading edge sweep angle	deg, rad
λ	taper ratio	-----
μ	coefficient of viscosity for air	-----
μ_{inl}	area ratio A_c/A_∞	-----
π	3.14	-----
ρ	air density	slug/ft^3

σ	sidewash angle	deg, rad
Φ	effective turning angle	deg, rad
Φ_{te}	trailing edge angle	deg, rad

4. SUBSCRIPTS

1	steady state
a	aileron
av	available
A	Airplane
b	base
bal	balance
bw	basic wing
c	canard
can	canopy
cg	center of gravity
co	cut-off
cool	cooling
cw	canopy/windshield
e	elevator
eff	effective
emp	empennage
f	friction, sometimes: flap
flap	flap
fus	fuselage
g	ground effect
gear	landing gear
h	horizontal tail
int	interference
kf	krueger flap
lam	laminar
lef	leading edge flap
L	landing
LE, le	leading edge
max	maximum
misc	miscellaneous
M	(at a given) Mach Number
n	nacelle
np	nacelle/pylon
p	pylon
plf	planform
prof	profile
r	rudder
rated	rated, usually SHP
ref	reference, usually the wing, or a point on the wing
reqd	required
s	slat(ted)
sf	split flap
sp	spoiler (speedbrake)

std	standard
store	store(s)
T	thrust or power effect
TE, te	trailing edge
TL	thrustline offset effect
TS	propeller slipstream effect
turb	turbulent
TO	take-off
trim	trim
w	wing
wave	wave (drag)
wb	wing-body (same as wing-fuselage)
wet	wetted
wf	wing-fuselage (same as wing-body)
wing	wing
wmprop	windmilling propeller
ws	windshield

5. ACRONYMS

AEO	All engines operating
APU	Auxiliary power unit
AWACS	Airborne warning and control system
b.l.	boundary layer
BPR	Bypass ratio
CBR	California Bearing Ratio
c.g.	center of gravity
FRP	Fuselage Reference Plane
hl or h.l.	hingeline
i.e.	id est (that is)
l.e.	leading edge
mgc or m.g.c.	mean geometric chord
OEI	one engine inoperative
RAT	Ram air turbine
SHP	Shaft Horsepower
t.e.	trailing edge
W.R.P.	Wing Reference Plane
w.r.t	with respect to

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Interavia (Swiss, monthly)
Flight International (British, weekly)
Business and Commercial Aviation (USA, monthly)
Aviation Week and Space Technology (USA, weekly)
Journal of Aircraft (USA, AIAA, monthly)

The author wishes to acknowledge the important role played by these magazines in his own development as an aeronautical engineer. Aeronautical engineering students and graduates should read these magazines regularly.

This part of the Airplane Design series presents rapid methods for estimating drag, lift, pitching moment, installed thrust and power data as well as stability, control and hingemoment derivatives. In preparing this part much use has been made of the USAF DATCOM, a most impressive and useful document which should be available in all aeronautical engineering design offices and libraries.

FOKKER D.XXI
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1. INTRODUCTION

The purpose of this series of books on Airplane Design is to familiarize aerospace engineering students with the design methodology and design decision making involved in the process of designing airplanes.

The series of books is organized as follows:

PART I:	PRELIMINARY SIZING OF AIRPLANES
PART II:	PRELIMINARY CONFIGURATION DESIGN AND INTEGRATION OF THE PROPULSION SYSTEM
PART III:	LAYOUT DESIGN OF COCKPIT, FUSELAGE, WING AND EMPENNAGE: CUTAWAYS AND INBOARD PROFILES
PART IV:	LAYOUT DESIGN OF LANDING GEAR AND SYSTEMS
PART V:	COMPONENT WEIGHT ESTIMATION
PART VI:	PRELIMINARY CALCULATION OF AERODYNAMIC, THRUST AND POWER CHARACTERISTICS
PART VII:	DETERMINATION OF STABILITY, CONTROL AND PERFORMANCE CHARACTERISTICS: FAR AND MILITARY REQUIREMENTS
PART VIII:	AIRPLANE COST ESTIMATION: DESIGN, DEVELOPMENT, MANUFACTURING AND OPERATING

The purpose of PART VI is to present a systematic approach to the prediction of drag, installed power and thrust, lift, pitching moment and other important stability and control data needed in preliminary design.

The methods presented in this volume are meant to be used in conjunction with Preliminary Design Sequence II as outlined in Chapter 2 of Part II of this series. For that reason these methods are sometimes referred to as Class II methods. The preceeding Class I methods are covered in Part I and Part II of this series and are meant to be used with Preliminary Design Sequence I as outlined also in Chapter 2 of Part II.

In Chapter 2 some important definitions relating to flight regime and reference geometries are discussed.

Chapter 3 presents different ways by which airplane drag polars can be represented by simple mathematical models. Several examples are given.

Chapter 4 presents a method to predict the component drag breakdown and the total drag of airplanes. Example drag data are given in Chapter 5.

Methods for predicting installed thrust and power characteristics are presented in Chapter 6. Example thrust and power data are given in Chapter 7.

Chapter 8 contains methods for predicting lift and pitching moment characteristics of airplanes with and without (mechanical) flaps. Example data are given in Chapter 9.

Prediction methods for stability, control and hinge moment derivatives are presented in Chapter 10 with example data provided in Chapter 11.

To make the use of this book easy on students, a USER's GUIDE is presented in Chapter 12. Aeronautical engineering students should use this guide to prevent wasting a lot of time chasing non-existing problems.

Appendix A contains data on the atmosphere.

Appendix B contains data needed in the estimation of areas and volumes.

2. IMPORTANT DEFINITIONS

In this text frequent use is made of:

2.1 Flow regimes defined in terms of Mach number
2.2 Reference geometries defined in terms of areas and lengths

The purpose of this chapter is to define the physical and the mathematical meaning of these terms.

2.1 FLOW REGIME DEFINITIONS

In this text the following flow regimes are used:

2.1.1 Subsonic Flow Regime
2.1.2 Transonic Flow Regime
2.1.3 Supersonic Flow Regime

2.1.1 Subsonic Flow Regime

In terms of Mach number the subsonic flow regime is defined as:

SUBSONIC: $0 < M < 0.60$

It is assumed that in this flow regime all compressibility effects are negligible. The reader should keep in mind that whether or not compressibility effects are negligible depends not just on the free stream Mach number but also on thickness and on angle of attack.

Where the subsonic flow regime ends depends on the values of critical and drag divergence Mach numbers:

The critical Mach number is that free stream Mach number for which a condition of $M=1$ is first reached somewhere on the airplane.

Figure 2.1 shows how the critical Mach number varies with airfoil shape and with airfoil lift coefficient (or angle of attack).

The drag divergence Mach number is that free stream Mach number for which:

> Boeing definition: the drag coefficient first reaches a value of 0.0020 above that in the subsonic flow regime.

Figure 2.1 Effect of Airfoil Shape and Lift Coefficient on Drag Divergence Mach Number

Figure 2.2 Boeing and Douglas Definitions for Drag Divergence Mach Number

Douglas definition: the slope of the drag coefficient versus Mach number first reaches the value of 0.10.

Figure 2.2 depicts these definitions graphically. Experience shows that there is no significant difference between these definitions.

The following equations may be used to estimate the drag divergence Mach number for airfoils:

For modern (aft-loaded) airfoils:

$$M_{DD} = 0.95 - (t/c)_{max} - c_l/10 \qquad (2.1)$$

For NACA airfoils:

$$M_{DD} = 0.90 - (t/c)_{max} - c_l/10 \qquad (2.2)$$

Figures 2.3 may be used to estimate the drag divergence Mach number of uncambered wings. For wings with camber, determine the lift coefficient for zero angle of attack, C_{L_o}. Then use Figures 2.3 with:

$$C_L = C_{L_{actual}} + C_{L_o} \qquad (2.3)$$

Reference 8 contains a wealth of information on the effect of the geometry of fuselages and wings on critical and on drag divergence Mach number.

2.1.2 Transonic Flow Regime

In terms of Mach number the transonic flow regime is defined as:

TRANSONIC: $0.60 < M < 1.2$

Mathematically speaking, the transonic flow regime starts at the critical Mach number.

Physically speaking, the transonic flow regime starts at the drag divergence Mach number.

In the transonic flow regime compressibility effects cannot be neglected. The reader should keep in mind that compressibility effects in this flow regime are a strong function of thickness ratio (for lifting surfaces such as wings, pylons and tails) and of cross sectional area distributions. The sweep angle of lifting surfaces is also an important parameter.

Figure 2.3 Effect of Thickness Ratio, Sweep Angle, Aspect Ratio and Lift Coefficient on the Drag Divergence Mach Number of Wings

Figure 2.3 (Cont'd) Effect of Thickness Ratio, Sweep Angle, Aspect Ratio and Lift Coefficient on the Drag Divergence Mach Number of Wings

To reduce drag in the transonic flow regime the concept of area ruling plays an important role.

Reference 8 contains information on the variation of drag with Mach number in the transonic speed regime below M=1. Reference 9 should be used for the estimation of drag in the transonic flow regime above M=1.

For applications of the area rule concept, ref.10 should be consulted.

2.1.3 Supersonic Flow Regime

In terms of Mach number the supersonic flow regime is defined as:

SUPERSONIC: $1.2 < M < 3.0$

The aerodynamic behavior of airplanes in this flow regime depends strongly on the location of Mach lines relative to the airplane geometry. In particular whether or not the wing has a subsonic, supersonic or mixed flow leading edge has important consequences.

Figure 2.4 illustrates what is meant by subsonic/supersonic leading edges.

Reference 9 contains methods for estimating drag in this flow regime.

Note: Reference 11 is strongly recommended as a general text on the effect of Mach number and of angle of attack on the drag and lift characteristics of airplanes.

2.2 IMPORTANT GEOMETRIC DEFINITIONS

2.2.1 Wing Planform Geometries

In this text, all aerodynamic characteristics of airplanes are referred to so-called reference geometries. This is accomplished through the introduction of dimensionless coefficients for drag, lift, pitching moment, side force, rolling moment and yawing moment in the following manner:

For drag: $D = C_D \bar{q} S$ (2.4)

For lift: $L = C_L \bar{q} S$ (2.5)

For pitching moment: $M_A = C_m \bar{q} S \bar{c}$ (2.6)

Figure 2.4 Subsonic and Supersonic Leading Edges

Figure 2.5 Reference Geometry for Straight, Tapered Wings

For side force: $F_{A_y} = C_y \bar{q} S$ (2.7)

For rolling moment: $L_A = C_l \bar{q} S b$ (2.8)

For yawing moment: $N_A = C_n \bar{q} S b$ (2.9)

The reference geometries in Equations (2.4) through (2.9) are S, $\bar{c}$ and b respectively. These reference geometries are normally based on the wing planform. For a straight, tapered wing these parameters are defined in Figure 2.5.

Most wings cannot be classified as straight, tapered wings. Leading edges and trailing edges are often broken or curved to achieve certain aerodynamic, structural or configurational objectives. Figure 2.6 shows how the parameters S, $\bar{c}$ and b are defined for non straight, tapered wings. Note that a so-called equivalent wing is defined for these planforms. This equivalent wing definition is found by 'averaging' areas.

In some airplanes the wings have very rapid changes in leading edge shape and/or trailing edge shape. Examples are shown in Figure 2.7. The definitions of S, $\bar{c}$ and b for such wings are also given in Figure 2.7.

Although the definition of airplane reference geometries is in principle arbitrary, the reader should not deviate too far from the definitions used here. The reason is that in the aerodynamic literature cited in this text the reference geometries are those defined here.

In addition to the parameters S, $\bar{c}$ and b, quantities such as taper ratio and sweep angle play an important role. Figures 2.5 and 2.6 also define these quantities.

2.2.2 Empennage Planform Geometries

To determine the contributions of empennage surfaces (horizontal tail, canard and/or vertical tail) to the aerodynamic characteristics of airplanes it is necessary to define their planform geometries. Figure 2.8 does this for horizontal tails and for canards. Figure 2.9 does this for vertical tails.

Figure 2.6 Reference Geometry and Equivalent Planform for Non Straight, Tapered Wings

$$y_{\bar{c}} = \frac{2}{S}\int_0^{b/2} y\,c(y)\,dy \qquad x_{LE_{\bar{c}}} = \frac{1}{S}\int_{-b/2}^{+b/2} x_{LE}(y)\,c(y)\,dy$$

$$S = \int_{-b/2}^{b/2} c(y)\,dy \qquad \bar{c} = \frac{1}{S}\int_{-b/2}^{+b/2} c^2(y)\,dy$$

Figure 2.7 Reference Geometry for Arbitrary Wings

Figure 2.8 Reference Geometries for Horizontal Tails and for Canards

Figure 2.9 Reference Geometries for Vertical Tails

3. SUMMARY OF DRAG CAUSES AND DRAG MODELLING

The purpose of this chapter is to summarize the physical causes of drag, to present ways of breaking down drag and to discuss frequently used methods of modelling drag for use in airplane performance calculations.

For a more detailed study of drag, drag causes and drag computation methods the reader should consult references 8-15.

3.1 PHYSICAL CAUSES OF DRAG

Total drag is defined as the sum of zero-lift drag and induced drag (or drag due to lift):

Total Drag = Zero-lift Drag + Drag-due-to-lift (3.1)

This basic breakdown for will be used in all three speed ranges used in this text:

1. Subsonic speed range: $0 < M < 0.6$
2. Transonic speed range: $0.6 < M < 1.2$
3. Supersonic speed range: $1.2 < M < 3.0$

The zero-lift drag of an airplane is considered to be the sum of skin friction drag and pressure drag:

Zero-lift Drag = Skin-friction Drag + Pressure Drag (3.2)

Skin-friction drag is caused by the shearing stresses within the thin layer of air adjacent to the skin. This layer of air is called the boundary layer. It arises as a result of the viscosity of the air which resists a body passing through it.

The magnitude of this viscous resistance depends on whether the flow in the boundary layer is laminar or turbulent. Whether the boundary layer is laminar or turbulent depends on the Reynolds Number, on the pressure distribution and on the roughness of the skin (surface).

Pressure drag is caused by the displacement thickness of the boundary layer, which prevents full pressure recovery at the trailing edge. As long as the boundary layer remains attached, pressure drag tends to be small in subsonic flight. However, in transonic and in supersonic flight pressure drag is identified with wave drag and then becomes very significant.

When the boundary layer becomes separated and/or when bluntness is significant, pressure drag can be large in any speed regime.

Drag-due-to-lift is considered to be the sum of induced drag and viscous drag-due-to-lift:

$$\text{Drag-due-to-lift} = \text{Induced Drag} + \text{Viscous Drag-due-to-lift} \qquad (3.3)$$

The induced drag, also called trailing edge vortex drag (or simply vortex drag) depends on the spanwise distribution of lift. It is proportional to the square of the lift coefficient.

The viscous drag due to lift results from the change in the boundary layer development as a result of lift. The upper surface boundary layer thickness increases with increasing angle of attack. This in turn results in an increase in the so-called profile drag which itself is the sum of skin-friction drag and pressure drag.

Figure 3.1 illustrates a drag breakdown according to these physical causes. Intermediate drag types such as induced drag, form drag and profile drag are indicated also in Figure 3.1.

A major problem is that the basic drag causes in Equations (3.1)-(3.3) are interdependent. The flow regime in which the airplane is operating: Mach number, Reynold's number and angle of attack all influence the basic drag causes.

In predicting total airplane drag some 'book-keeping' procedures must be adopted. One 'book-keeping' decision which needs to be made is that which splits drag from thrust. In this text the position is taken that flow phenomena outside the airplane are associated with the generation of drag. Flow phenomena inside the airplane which are not associated with the production of thrust, are also considered to produce drag. An example of this is cooling air flow around a piston engine.

On the other hand, flow phenomena inside engine(s) (and/or nacelles) are associated with the production of thrust (or power).

Chapters 4 and 5 deal with airplane drag. Chapters 6 and 7 deal with airplane thrust (or power).

Figure 3.1 Drag Breakdown According to Physical Causes

Figure 3.2 Typical Drag Breakdown for a Transport Jet

3.2 DRAG BREAKDOWN METHOD

It is useful to break down the drag of an airplane into that caused by its components. Figure 3.2 shows a typical drag breakdown for a transport operating at a high subsonic Mach number.

The drag breakdown method used in this text considers total airplane drag to be the sum of component drag contributions according to:

Airplane Drag = Component drag of (Wing + Fuselage + + Empennage + Nacelle/Pylon + Flap + + Landing Gear + Canopy/Windshield + + Store + Trim + Interference + + Miscellaneous) (3.4)

Chapter 4 presents a method for computing component drag for subsonic, transonic and supersonic flight.

3.3 DRAG MODELLING FOR PERFORMANCE CALCULATIONS

To enable the rapid calculation of airplane performance (discussed in Part VII), it is useful to represent airplane drag polars by simple mathematical models.

In addition, it is necessary to assume that the airplane can be held in a 'trimmed' state: no moments are acting on the airplane and the pilot does not have to exert a force on the cockpit controls. Requirements for trim are discussed in detail in Ref.16. Part VII contains methods for predicting trimmability.

To trim an airplane usually causes extra drag: trim drag. Eqn.(3.4) includes a term which accounts for trim drag. All drag polars used in performance calculations are assumed to be 'trimmed' drag polars.

For essentially uncambered airplanes in subsonic flight, the drag polar may be modelled as:

$$C_D = C_{D_o} + (C_L)^2/\pi Ae \qquad (3.5)$$

Note that the minimum drag occurs at $C_L = 0$.

The F4C fighter airplane is an example of an essentially uncambered airplane. Figure 3.3 shows its drag polar. It is very well represented by Eqn.(3.5).

For cambered airplanes, the minimum value of drag

Figure 3.3 Drag Polar for the McDonnell-Douglas F4C

Figure 3.4 Drag Polar for the Lockheed C-141

does not occur at $C_L = 0$ and it becomes necessary to represent the drag polar as:

$$C_D = C_{D_{min}} + K''(C_L - C_{L_{min}})^2 + K'C_L^2 \qquad (3.6)$$

Note that minimum drag in this case occurs at a lift coefficient of $C_L = [K''/(K''+K')]C_{L_{min}}$.

The Lockheed C-141 Starlifter is a cambered airplane despite the fact that its wing uses symmetrical airfoil sections (average section is NACA 0011). Since the wing is installed at an incidence angle of 3.2 degrees the airplane acts like it is cambered. Figure 3.4 shows the C-141 low speed drag polar. Note that: $C_{D_{min}} = 0.0140$, which occurs at $C_{L_{min}} = 0.27$.

Note, that for an uncambered airplane Eqn.(3.6) reduces to Eqn.(3.5) since in that case: $C_{L_{min}} = 0$ and:

$$K'' + K' = 1/\pi Ae \text{ (no camber only!)} \qquad (3.7)$$

For airplanes flying at relatively high angles of attack, the drag polars deviate significantly from the parabolic shape. This is caused by flow separation resulting in a steep rise in pressure drag. In that case the polar is modelled as:

$$C_D = C_{D_{min}} + K''(C_L - C_{L_{min}})^2 + K'C_L^2 + K_B(C_L - C_{L_B})^2 \qquad (3.8)$$

Figure 3.5 illustrates a drag polar with this type of behavior.

C_{L_B} is the lift coefficient for which the steep drag rise due to separation begins. The factor K_B is called the break drag-due-to-lift factor and is defined as:

$$K_B = 0 \text{ for } C_L < C_{L_B}$$
$$K_B > 0 \text{ for } C_L > C_{L_B} \qquad (3.9)$$

Table 3.1 presents a summary of important lift-to-drag ratios based on the drag polar models of Eqns 3.5 and 3.6. These lift-to-drag ratios and the lift coefficients at which they occur are used in the performance calculations presented in Part VII.

Figure 3.5 Drag Polars for the Boeing 747

Table 3.1 Maximum Lift-to-Drag Ratios and Corresponding Lift Coefficients

Drag Polar According to Eqn.(3.5)

$$(L/D)_{max} = 0.5/(KC_{D_o})^{1/2} \quad (3.10)$$

$$C_{L_{(L/D)_{max}}} = (C_{D_o}/K)^{1/2} \quad (3.11)$$

where: $K = 1/\pi Ae$ (3.12)

Drag Polar According to Eqn.(3.6)

$$(L/D)_{max} = (P^{1/2})/[C_{D_{min}} + K''((P^{1/2}) - C_{L_{min}})^2 + K'P] \quad (3.13)$$

where: $P = (C_{D_{min}} + K''(C_{L_{min}})^2)/(K' + K'')$ (3.14)

$$C_{L_{(L/D)_{max}}} = (P)^{1/2} \quad (3.15)$$

Note that equations (3.13) and (3.15) reduce to equations (3.10) and (3.11) for the case of $K''=0$ and $C_{D_{min}} = C_{D_o}$. In that case, $K' = K$.

4. DRAG POLAR PREDICTION METHODS

A Class I method for drag polar prediction was presented in Part I (Ch. 3). The purpose of this chapter is to present a Class II method for predicting drag polars of airplanes during the preliminary design phase. The method is based on Reference 9 and applies to airplanes with essentially straight, tapered wings. For other wing planforms, see Reference 9.

4.1 DRAG BREAKDOWN PROCEDURE

Total airplane drag in lbs is written as:

$$D = C_D \bar{q} S \tag{4.1}$$

where: C_D = the total airplane drag coefficient

$$\bar{q} = 0.5\rho(U_1)^2 = 14.825M^2, \text{ also called} \tag{4.2}$$

free stream dynamic pressure, with:

ρ = air density: see Appendix A.

U_1 = steady state airspeed

δ = pressure ratio: see Appendix A.

$$M = U_1/a = \text{Mach number} \tag{4.3}$$

a = speed of sound: see Appendix A.

S = the wing planform or reference area. Figures 2.5-2.7 define how this area is normally defined for a range of wing designs.

The total airplane drag coefficient is normally broken down into the following components:

$$C_D = C_{D_{wing}} + C_{D_{fus}} + C_{D_{emp}} + C_{D_{np}} + C_{D_{flap}} + C_{D_{gear}} + C_{D_{cw}} + C_{D_{store}} + C_{D_{trim}} + C_{D_{int}} + C_{D_{misc}} \tag{4.4}$$

where: $C_{D_{wing}}$ = wing drag coefficient: Section 4.2

$C_{D_{fus}}$ = fuselage drag coefficient: Section 4.3

$C_{D_{emp}}$ = empennage drag coefficient: Section 4.4

$C_{D_{np}}$ = nacelle/pylon drag coefficient, including cooling drag: Section 4.5

$C_{D_{flap}}$ = leading/trailing edge flap drag coefficient: Section 4.6

$C_{D_{gear}}$ = landing gear drag coefficient: Section 4.7

$C_{D_{cw}}$ = canopy/windshield drag coefficient: Section 4.8

$C_{D_{store}}$ = store(s) drag coefficient: Section 4.9

$C_{D_{trim}}$ = trim drag coefficient: Section 4.10

$C_{D_{int}}$ = interference drag coefficient: Section 4.11

$C_{D_{misc}}$ = miscellaneous drag coefficient, typically caused by such things as: speed brakes, struts, inlet drag, antennas, gaps and surface roughnesses: Section 4.12

Sections 4.2 through 4.12 present methods for predicting the drag coefficient components in Eqn.(4.2). Numerical examples of drag breakdowns and examples of airplane drag polars are presented in Chapter 5.

<u>Important Notes:</u> 1.) The drag prediction methods of Sections 4.2 through 4.5 apply only to flight cases where the boundary layer is mostly turbulent. If extensive laminar flow runs are present (for example because of use of natural laminar flow airfoils or because of use of 'forced' laminar flow by suction), Section 4.13 may be used to obtain the necessary drag corrections.

2.) The drag prediction methods of Sections 4.2 through 4.5 apply only to 'smooth' surfaces. If surface roughness is present, the procedure of Sub-section 4.12.2 should be used in conjunction with the methods of Sections 4.2 through 4.5.

4.2 WING DRAG COEFFICIENT PREDICTION

4.2.1 Subsonic Wing Drag Coefficient

The subsonic wing drag coefficient is found from:

$$C_{D_{wing}} = C_{D_{o_w}} + C_{D_{L_w}} \tag{4.5}$$

where: $C_{D_{o_w}}$ = wing zero-lift drag coefficient, see 4.2.1.1.

$C_{D_{L_w}}$ = wing drag coefficient due to lift, see 4.2.1.2.

4.2.1.1 Wing zero-lift drag coefficient

The subsonic wing zero-lift drag coefficient may be computed from:

$$C_{D_{o_w}} = (R_{wf})(R_{LS})(C_{f_w})\{1 + L'(t/c) + 100(t/c)^4\}S_{wet_w}/S \tag{4.6}$$

where: R_{wf} = wing/fuselage interference factor: see Figure 4.1. For a flying wing: $R_{wf} = 1.0$.

R_{LS} = lifting surface correction factor found from Figure 4.2.

C_{f_w} = turbulent flat plate friction coefficient of the wing.

The general turbulent, flat plate friction coefficient, C_f is shown in Figure 4.3 as a function of Mach number and of Reynolds number, R_N.

For the wing, use: $R_{N_w} = \rho U_1 \bar{c}_{w_e}/\mu$ (4.7)

where: $\bar{c}_{w_e}$ = exposed wing m.g.c. (= wing m.g.c. for a flying wing)

μ = coefficient of viscosity of air: see Appendix A.

L' = airfoil thickness location parameter: see Figure 4.4.

Figure 4.1 Wing Fuselage Interference Factor

Figure 4.2 Lifting Surface Correction Factor

Figure 4.3 Turbulent Mean Skin-Friction Coefficient

Figure 4.4 Airfoil Thickness Location Parameter

Figure 4.5 Definition of Thickness Ratio for a Wing

Figure 4.6 Definition of Wing Wetted Area

t/c = thickness ratio defined at the mean geometric chord of the wing: see Figure 4.5.

S_{wet_w} = wetted area of the wing: see Figure 4.6 and Appendix B.

S = wing or reference area. This is normally the wing planform area: see Figs 2.5-2.7.

4.2.1.2 Wing drag coefficient due to lift

The wing drag coefficient due to lift is found from:

$$C_{D_{L_w}} = (C_{L_w})^2/\pi Ae + 2\pi C_{L_w} \epsilon_t v + 4\pi^2 (\epsilon_t)^2 w \qquad (4.8)$$

where: C_{L_w} = wing lift coefficient, defined as:

$$C_{L_w} = C_L - C_{L_c} S_c/S + C_{L_h} S_h/S \qquad (4.9)$$

where: $C_L = W/\bar{q}S$ (4.10)

The canard and horizontal tail lift coefficients, C_{L_c} and C_{L_h} respectively, are determined from trim considerations. Their values depend on the center of gravity location. To determine these empennage lift coefficients a trim calculation must be performed. This is done with the help of a trim diagram. Part VII contains a method for constructing a trim diagram.

In early preliminary design it is sufficiently accurate to set:

$$C_{L_w} = 1.05 C_L \qquad (4.11)$$

A = wing aspect ratio = b^2/S, where:
b = wing span: see Figures 2.5 - 2.7.

e = span efficiency factor, defined as:

$$e = 1.1(C_{L_{\alpha_w}}/A)/[R(C_{L_{\alpha_w}}/A) + (1 - R)\pi] \qquad (4.12)$$

where: $C_{L_{\alpha_w}}$ is the wing lift-curve slope which can be found with the help of Chapter 10.

R = leading edge suction parameter as defined in Figure 4.7.

ϵ_t = wing twist angle, positive for wash-in, negative for wash-out, see Figure 4.8.

v = induced drag factor due to linear twist, see Figures 4.9.

w = zero-lift drag factor due to linear twist, see Figures 4.10.

<u>Note:</u> this method for determining C_{L_w} applies only for $C_{L_w} < C_{L_B}$, the value of lift coefficient where the flow begins to separate.

4.2.2 Transonic Wing Drag Coefficient

The transonic wing drag coefficient is found from:

$$C_{D_{wing}} = C_{D_{o_w}} + C_{D_{L_w}} \tag{4.13}$$

where: $C_{D_{o_w}}$ = wing zero-lift drag coefficient, see 4.2.2.1.

$C_{D_{L_w}}$ = wing drag coefficient due to lift, see 4.2.2.2.

4.2.2.1 Wing zero-lift drag coefficient

In the transonic speed range the wing zero-lift drag coefficient is found from:

$$C_{D_{o_w}} = C_{D_{o_w}\,\text{at } M=0.6} + C_{D_{w_{wave}}} \tag{4.14}$$

where: $C_{D_{o_w}\,\text{at } M=0.6}$ = the drag coefficent due to friction. It is found from Eqn.(4.6) and is assumed to stay constant with Mach number in the entire transonic speed range.

$C_{D_{w_{wave}}}$ = the wing wave drag coefficient which depends on the wing sweep angle, $\Lambda_{c/4}$.

Figure 4.7 Leading Edge Suction Parameter

Figure 4.8 Definition of Wing Twist Angle

Figure 4.9a Induced Drag Factor due to Linear Twist

Figure 4.9b Induced Drag Factor due to Linear Twist

Figure 4.10a Zero-Lift Drag Factor due to Linear Twist

Figure 4.10b Zero-Lift Drag Factor due to Linear Twist

Unswept wings:

The wave drag coefficient, $C_{D_{w_{wave}}}$ for an unswept wing follows from Figure 4.11. This wave drag coefficient should be plotted against the Mach number in the transonic speed range.

Swept wings:

For a swept wing, proceed as follows:

From a plot of $C_{D_{w_{wave}}}$ versus M for the unswept wing of the same aspect ratio, and thickness ratio, read the following values:

M_{DD}, $C_{D_{w_{wave_{peak}}}}$ and $M_{at\ C_{D_{w_{wave_{peak}}}}}$.

Correct these values for sweep angle as follows:

$$M_{DD_{\Lambda_{c/4}}} = M_{DD}/(\cos\Lambda_{c/4})^{1/2} \quad (4.15)$$

$$C_{D_{wave_{peak_{\Lambda_{c/4}}}}} = C_{D_{wave_{peak}}} (\cos\Lambda_{c/4})^{2.5} \quad (4.16)$$

$$M_{at\ C_{D_{wave_{peak_{\Lambda_{c/4}}}}}} = (M_{at\ C_{D_{wave_{peak}}}})/(\cos\Lambda_{c/4})^{1/2} \quad (4.17)$$

Draw in the line of $C_{D_{w_{wave_{\Lambda_{c/4}}}}}$ versus Mach number in the transonic speed range. A typical result is illustrated in Figure 4.12.

4.2.2.2 Wing drag coefficient due to lift

The transonic wing drag coefficient due to lift is written as:

$$C_{D_{L_w}} = (C_{D_L}/C_L^{\ 2})C_L^{\ 2} \quad (4.18)$$

VALID ONLY AT TRANSONIC SPEEDS
VALID ONLY FOR UNSWEPT, ROUNDNOSE AIRFOILS

Figure 4.11 Zero-Lift Wave Drag Coefficient

Figure 4.12 Transonic Fairing of Zero-Lift Wave Drag

The induced drag parameter (C_{D_L}/C_L^2) may be found as a function of transonic similarity parameters which are defined in Figures 4.13.

4.2.3 Supersonic Wing Drag Coefficient

The supersonic wing drag coefficient is found from:

$$C_{D_{wing}} = C_{D_{o_w}} + C_{D_{L_w}} \tag{4.19}$$

where: $C_{D_{o_w}}$ = wing zero-lift drag coefficient, see 4.2.3.1.

$C_{D_{L_w}}$ = wing drag coefficient due to lift, see 4.2.3.2.

4.2.3.1 Wing zero-lift drag coefficient

The supersonic wing zero-lift drag coefficient follows from:

$$C_{D_{o_w}} = C_{D_{w_f}} + C_{D_{w_{wave}}} \tag{4.20}$$

where: $C_{D_{w_f}}$ = the supersonic skin-friction drag coefficient, found from:

$$C_{D_{w_f}} = C_{f_w} S_{wet}/S \tag{4.21}$$

where: C_{f_w} may be found from Figure 4.3.

$C_{D_{w_{wave}}}$ = the wing wave drag coefficient which depends on the leading edge radius and on whether the wing has a subsonic or a supersonic leading edge: see Fig.2.4 for definitions. For wings with sharp nose airfoils, see Ref.9. For wings with round nose airfoils:

For wings with a supersonic leading edge:

$$C_{D_{w_{wave}}} = C_{D_{LE}} + (16/3\beta)\{(t/c)_{eff}\}^2 S_{bw}/S \tag{4.22}$$

For wings with a subsonic leading edge:

$$C_{D_{w_{wave}}} = C_{D_{LE}} + (16/3)\cot\Lambda_{LE_{bw}}\{(t/c)_{eff}\}^2 S_{bw}/S \tag{4.23}$$

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NOTE: $A \tan \Lambda_{LE} = 0$

Figure 4.13a Transonic Drag due to Lift

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NOTE: $A \tan \Lambda_{LE} = 3$

Figure 4.13b Transonic Drag due to Lift

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NOTE: $\lambda = 0.5$

Figure 4.13c Transonic Drag due to Lift

The value of the pressure drag coefficient, $C_{D_{LE}}$ follows from Figures 4.14.

Figure 4.15 defines the meaning of basic wing area, S_{bw} for several types of planform. The basic planform leading edge sweep angle, $\Lambda_{LE_{bw}}$ is also defined. Observe, that for a straight tapered wing: $S_{bw} = S$.

The term $(t/c)_{eff}$ in Eqns (4.22) and (4.23) is defined as follows:

$$(t/c)_{eff} = \{\int_0^{b/2}(t/c)^2 c_{bw} dy\}^{1/2}/(S_{bw}/2)^{1/2} \qquad (4.24)$$

<u>4.2.3.2 Wing drag coefficient due to lift</u>

The wing drag coefficient due to lift is found from:

$$C_{D_{L_w}} = (C_{D_L}/C_L^2)C_L^2 \qquad (4.25)$$

To compute the slope (C_{D_L}/C_L^2), proceed as follows:

1.) calculate two wing planform parameters:

planform shape parameter, $p = S/bc_r$

planform slenderness parameter, $b/2c_r$ (4.26)

Figure 4.15 illustrates the meaning of the quantities in Eqns (4.26) for some wing planforms.

2.) at the appropriate value of $\beta b/2c_r$, obtain the value of $\{(\pi A)(C_{D_L}/C_L^2)p/(p+1)\}$ from Fig.4.16. The symbol β is defined as:

$$\beta = (1 - M^2)^{1/2} \qquad (4.27)$$

3.) calculate the value of (C_{D_L}/C_L^2) from:

$$(C_{D_L}/C_L^2) = \{(\pi A)(C_{D_L}/C_L^2)p/(p+1)\}(1/\pi A)(1+p)/p \qquad (4.28)$$

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$$C_{D_{LE}}\left[\frac{S_{ref}}{2r_{LE_{bw}}\left(\frac{b_{bw}}{\cos\Lambda_{LE_{bw}}}\right)}\right] = 1.28\,\frac{M^3\cos^4\Lambda_{LE_{bw}}}{1+M^3\cos^3\Lambda_{LE_{bw}}}$$

$r_{LE_{bw}}$ = BASIC WING L.E. RADIUS

Figure 4.14a Leading Edge Pressure Drag Coefficient

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$r_{LE_{bw}}$ = BASIC WING L.E. RADIUS

Figure 4.14b Leading Edge Pressure Drag Coefficient

Figure 4.15 Basic Wing and Actual Wing Geometry

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Figure 4.16 Supersonic Drag due to Lift

4.3 FUSELAGE DRAG COEFFICIENT PREDICTION

4.3.1 Subsonic Fuselage Drag Coefficient

The subsonic fuselage drag coefficient is found from:

$$C_{D_{fus}} = C_{D_{o_{fus}}} + C_{D_{L_{fus}}} \tag{4.29}$$

where: $C_{D_{o_{fus}}}$ = fuselage zero-lift drag coefficient, see 4.3.1.1.

$C_{D_{L_{fus}}}$ = fuselage drag coefficient due to lift, see 4.3.1.2.

4.3.1.1 Fuselage zero-lift drag coefficient

The subsonic fuselage zero-lift drag coefficient is found from:

$$C_{D_{o_{fus}}} = R_{wf} C_{f_{fus}} \{1 + 60/(l_f/d_f)^3 + 0.0025(l_f/d_f)\} S_{wet_{fus}}/S + C_{D_{b_{fus}}} \tag{4.30}$$

where: R_{wf} = wing/fuselage interference factor, see Figure 4.1. Note: for a fuselage alone, use $R_{wf} = 1.0$.

$C_{f_{fus}}$ = turbulent flat plate skin-friction coefficient of the fuselage.

The general, turbulent, flat plate friction coefficient C_f is shown in Fig.4.3

as a function of Mach number and of Reynolds number, R_N.

For the fuselage, use:

$$R_{N_{fus}} = \rho U_1 l_f/\mu \tag{4.31}$$

l_f = fuselage length as shown in Figure 4.17.

d_f = maximum fuselage diameter (equivalent diameter for fuselages with non-circular cross section, see Figure 4.17).

$S_{wet_{fus}}$ = wetted area of the fuselage. See Figure 4.17 and Appendix B.

Figure 4.17 Definition of Fuselage Parameters

Figure 4.18 Defition of Fuselage Angle of Attack

$C_{D_{b_{fus}}}$ = fuselage base-drag coefficient, given by:

$$C_{D_{b_{fus}}} = \{0.029(d_b/d_f)^3/(C_{D_{o_{fus-base}}}(S/S_{fus}))^{1/2}\}(S_{fus}/S) \quad (4.32)$$

where: d_b = fuselage base diameter, defined in Figure 4.17.

$C_{D_{o_{fus-base}}}$ = zero-lift drag coefficient of the fuselage exclusive of the base (see Figure 4.17) as determined from the first term on the right hand side in Eqn.(4.30).

S_{fus} = fuselage maximum frontal area as defined in Figure 4.17.

<u>4.3.1.2 Fuselage drag coefficient due to lift</u>

The fuselage drag coefficient due to lift is found from:

$$C_{D_{L_{fus}}} = 2\alpha^2 S_{b_{fus}}/S + \eta c_{d_c} \alpha^3 S_{plf_{fus}}/S \quad (4.33)$$

where: α is the fuselage angle of attack in radians, which is the same as the airplane angle of attack as seen in Figure 4.18.

The airplane angle of attack can be estimated as follows:

$$\alpha = \{(W/\bar{q}S) - C_{L_o}\}/C_{L_\alpha} \quad (4.34)$$

where: C_{L_o} = airplane zero-angle-of-attack lift coefficient, see Chapter 10.

C_{L_α} = airplane lift-curve slope, see Chapter 10.

$S_{b_{fus}}$ = fuselage base area defined in Fig.4.17.

η = ratio of the drag of a finite cylinder to the drag of an infinite cylinder, see Figure 4.19.

c_{d_c} = experimental steady state cross-flow drag coefficient of a circular cylinder, obtained from Figure 4.20.

$S_{plf_{fus}}$ = fuselage planform area, illustrated in Figure 4.17.

Figure 4.19 Ratio of the Drag Coefficient of a Circular Cylinder of Finite Length to that of a Cylinder of Infinite Length

Figure 4.20 Steady State Cross-Flow Drag Coefficient for Two-Dimensional Circular Cylinders

4.3.2 Transonic Fuselage Drag Coefficient

The transonic fuselage drag coefficient is found from:

$$C_{D_{fus}} = C_{D_{o_{fus}}} + C_{D_{L_{fus}}} \qquad (4.35)$$

where: $C_{D_{o_{fus}}}$ = fuselage zero-lift drag coefficient, see 4.3.2.1.

$C_{D_{L_{fus}}}$ = fuselage drag coefficient due to lift, see 4.3.2.2.

4.3.2.1 Fuselage zero-lift drag coefficient

The transonic fuselage zero-lift drag coefficient is found from:

$$C_{D_{o_{fus}}} = R_{wf}(C_{D_{f_{fus}}} + C_{D_{p_{fus}}}) + C_{D_{b_{fus}}} + (C_{D_{wave_{fus}}})S_{fus}/S \qquad (4.36)$$

where: R_{wf} may be obtained from Figure 4.1 up to M=0.9.

For the remainder of the transonic speed range, R_{wf} = 1.0 should be used.

$C_{D_{f_{fus}}} = C_{f_{fus}}(S_{wet_{fus}})/S,$ (4.37)

the fuselage skin-friction drag coefficient at M = 0.6 which is assumed to stay constant in the entire transonic range. The value of $C_{f_{fus}}$ follows from Fig.4.3.

$$C_{D_{p_{fus}}} = (C_{f_{fus_{at\ M=0.6}}})[60/(l_f/d_f)^3 + 0.0025(l_f/d_f)](S_{wet_{fus}})/S, \qquad (4.38)$$

the fuselage pressure drag coefficient. This drag component is assumed to stay constant from M=0.6 to M=1.0 and then decrease linearly to zero at M=1.2.

$C_{D_{b_{fus}}}$ = the fuselage base drag coefficient as given by Eqn.(4.32) at M=0.6. From 0.6 > M < 1.2 this drag coefficient

may be determined by extrapolation, using the 'fairing lines' shown in Figure 4.21.

$C_{D_{wave_{fus}}}$ = the fuselage wave drag coefficient which follows from Figure 4.22.

Figure 4.23 illustrates how the transonic fuselage zero-lift drag coefficient is built up in this manner.

NOTE WELL: The wave drag component of zero lift drag can be much higher than predicted here if the cross sectional area distribution of the wing/fuselage combination is not 'smooth'. Sub-section 4.3.4 defines what is meant by 'smooth' cross sectional area distributions.

4.3.2.2 Fuselage drag coefficient due to lift

The transonic fuselage drag coefficient due to lift is found from:

$$C_{D_{b_{fus}}} = \alpha^2 (S_{base_{fus}})/S, \qquad (4.39)$$

with the angle of attack in radians.

4.3.3 Supersonic Fuselage Drag Coefficient

The supersonic fuselage drag coefficient is given as:

$$C_{D_{fus}} = C_{D_{o_{fus}}} + C_{D_{L_{fus}}} \qquad (4.40)$$

where: $C_{D_{o_{fus}}}$ = fuselage zero-lift drag coefficient, see 4.3.3.1.

$C_{D_{L_{fus}}}$ = fuselage drag coefficient due to lift, see 4.3.3.2.

4.3.3.1 Fuselage zero-lift drag coefficient

The supersonic fuselage zero-lift drag coefficient follows from:

$$C_{D_{o_{fus}}} = C_{f_{fus}} (S_{wet_{fus}})/S_{fus} + \{C_{D_{N_2}} + C_{D_A} + C_{D_{A(NC)}} + C_{D_{b_{fus}}}\}(S_{fus})/S \qquad (4.41)$$

EXAMPLE:
ASSUME
$C_{D_b}/(d_b/d_f)^2 = .09$
THE PROPER FAIRING LINE TO BE USED IS ____

Figure 4.21 Transonic Fairing for Fuselage Base Drag Coefficient

Figure 4.22 Wave Drag Coefficient for Parabolic Fuselages

Figure 4.23 Build-up of Transonic Fuselage Drag

Figure 4.24 Drag of Slender Forebodies or Afterbodies of Parabolic Profile

where: $C_{f_{fus}}$ = turbulent flat plate skin-friction coefficient of the fuselage, determined as described under Eqn.(4.30).

$C_{D_{N_2}}$ and C_{D_A} are the wave drag coefficients of the fuselage nose and afterbody respectively.

These wave drag coefficients may be determined from Figures 4.24 and 4.25 respectively. If the fuselage does not have a circular cross section, the respective equivalent diameters should be used when using Figures 4.24 and 4.25.

<u>NOTE WELL:</u> The wave drag components of zero lift drag can be much higher than predicted here if the cross sectional area distribution of the wing/fuselage combination is not 'smooth'. Sub-section 4.3.4 defines what is meant by 'smooth' cross sectional area distributions.

$C_{D_{A(NC)}}$ = interference drag coefficient acting on the aft-fuselage due to the center-fuselage. This coefficient may be found in Figures 4.26 or 4.27.

$C_{D_{b_{fus}}}$ = fuselage base drag coefficient. For fuselages with an approximate circular cross section this coefficient may be read from Figure 4.28, provided the fuselage has no appreciable base area.

<u>Note:</u> if the fuselage has an appreciable base area (and this almost always is the case in fighter airplanes), the value for the fuselage base drag coefficient may be obtained from:

$$C_{D_b} = -C_{p_b}(d_b/d_f)^2 \qquad (4.42)$$

where the base pressure coefficient, C_{p_b} follows from Figures 4.29.

<u>4.3.3.2 Fuselage drag coefficient due to lift</u>

The supersonic fuselage drag coefficient due to lift depends strongly on the fuselage cross section:

$$C_{D_{L_{fus}}} = F[(2\alpha^2)S_{b_{fus}}/S + c_{d_c}(S_{plf_{fus}}/S)\alpha^3] \qquad (4.43)$$

Figure 4.25 Drag of Slender, Conical Forebodies and Afterbodies

Figure 4.26 Interference Drag for Parabolic Bodies

Figure 4.27 Interference Drag of Truncated Afterbodies Behind Parabolic Forebody with no Constant Centersection

Figure 4.28 Base Drag Coefficient for Bodies of Revolution with no Boattail

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Figure 4.29 Base Pressure Coefficient for Boattails of Approximate Conical Shape

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Figure 4.29 Base pressure Coefficient for Boattails of Approximate Conical Shape

For a fuselage with circular cross section: $F = 1.0$

For a fuselage with elliptical cross section:

$$F = \{(a/b)(\cos\omega)^2 + (b/a)(\sin\omega)^2\} \qquad (4.44)$$

where: a is the major axis of the elliptical cross section.

b is the minor axis of the elliptical cross section.

$\omega = 0$ with the major axis horizontal.
$\omega = 90$ deg. with the minor axis horizontal.

Values for c_{d_c} are found in Figure 4.20.

4.3.4 The Area Rule Concept

The methods for predicting wave drag in Sub-sections 4.3.2 and 4.3.3 are valid only if the wing/fuselage/empennage combination has a 'smooth' cross sectional area distribution. Figure 4.30 illustrates how a cross sectional area distribution of an airplane is obtained.

Whitcomb showed in Reference 17 that the wave drag of an airplane in the transonic speed range is roughly equal to the wave drag of its 'equivalent body of revolution. Figure 4.30 also shows the equivalent body of revolution for the same airplane. Note the irregularities in the equivalent body of revolution.

Minimum wave drag can be achieved by designing the wing/fuselage/empennage cross sectional area distribution in such a way that its equivalent body of revolution approximates the so-called Sears-Haack bodies. This design concept (due to Whitcomb) is referred to as area ruling.

Three types of Sears-Haack bodies are depicted in Figure 4.31:

Type I: used to minimize wave drag for a given length and volume.

Type II: used to minimize wave drag for a given length and diameter.

Type III: used to minimize wave drag for a given diameter and volume.

CONSTRUCT 'EQUIVALENT' BODY OF REVOLUTION SO THAT $S_{i_{AIRPLANE}} = S_{i_{EQ.}}$

Figure 4.30 Development of a Cross Sectional Area Distribution and its Equivalent Body of Revolution

MINIMUM PRESSURE DRAG			VOL = $C_{P_n} \times A_{max} \times L$
TYPE I	$C_D = \frac{3}{8}\pi^2\left(\frac{d}{L}\right)^2$	GIVEN LENGTH & VOL	$C_{P_I} = 0.59$
TYPE II	$C_D = \pi^2\left(\frac{d}{L}\right)^2$	LENGTH & DIA	$C_{P_{II}} = 0.519$
TYPE III	$C_D = 3/2\,\pi^2\left(\frac{d}{L}\right)^2$	DIA & VOL	$C_{P_{III}} = 0.392$

Figure 4.31 Sears Haack Bodies and Drag Coefficients

The wave drag coefficients and volumes for the three Sears-Haack bodies are also given in Figure 4.31.

Note: these wave drag coefficients apply to supersonic Mach numbers only! To a first order of approximation these supersonic wave drag coefficients are independent of Mach number.

It is clear that to minimize wave drag it is essential to distribute any given volume over as much length as possible. However, in airplane layout design several constraints are encountered. Typical constraints are:

a) the airplane may not exceed a given length: this constraint is nearly always encountered in carrier based airplanes. Supersonic transports may encounter length limitations for reasons of ground maneuvering, gate space restrictions and hangar size limits.

b) the airplane must have a minimum total volume: this constraint comes about because of the need to carry fuel, payload, fixed equipment, landing gear and engines. Even airplane structures need a minimum amount of volume.

c) the airplane must have a minimum outside diameter: this constraint arises because of the need for cabin volume, engine size and/or payload size.

Because of these constraints on length, volume and diameter it may be necessary to strike a compromise in the layout design of transonic and supersonic airplanes.

From the previous discussion it is obvious that early knowledge of volume requirements inside the airplane is essential. The airplane layout design procedures of Chapters 4-9 in Part II must be augmented with the following step-by-step procedure:

Step 1: Tabulate preliminary volume requirements as indicated in Table 4.1.

Tables 4.2 give data for the volumetric requirements of the elements listed in Table 4.1. It is essential that the 'critical' (= minimum required) length for all elements is identified. The reader should use judgement in deciding where element critical lengths may overlap.

Step 2: Prepare a preliminary airplane layout which includes all elements listed in Step 1. Do not forget to cross-check with the design layout procedures of Chapters 4-9 of Pt.II.

Table 4.1 Critical Volume and Length Requirements

Airplane Element	Critical Volume (ft^3)	Critical Length (ft)
Forebody: radome, avionics, cockpit, nose gear		
Mid section: gun or weapons bay, main gear, avionics, control runs, body fuel, fixed equipment, wing sweep mechanism, inlets		
Aft body: engine bay(s), tail-pipes, nozzles, afterburners, body fuel, empennage carry-through		
Wing: total wing volume outside the body.		
Refuelling system: if required, this normally resides in the forward part of the fuselage.		
Empennage: total empennage volume outside the body.		
Streamtube: that volume taken up by the inlet, engines, tail-pipes, afterburners and nozzles.		
Inlet stream tube: that volume taken up by the inlet up to the engine compressor faces.		

Note: This table can be completed only after a preliminary layout of the airplane has been prepared. Chapters 4-9 in Part II provide step-by-step procedures for preparing such a layout. Tables 4.2a and 4.2b should be consulted for preliminary data on volumetric requirements of airplane elements.

Table 4.2a Critical Volume, Cross Section and Length Data

Notes: 1. Data given are minimum data for fighters. For transports the volume needs follow from layout design results of Chapters 4-9 in Part II.
2. All data refer to 'installed' volumes, not to actual volumes. Data based on Reference 18.

Element		Volume (ft^3)	Cross Section (ft^2)	Length (ft)
Radome:		$0.22(d_{rad} + 0.33)^4$	$(\pi/4)(d_{rad})^2$	
Cockpit:	today	70.0	14.0 upright	
	future	50.0	11.0 upright	
			7.0 semi-supine	
Nose gear:		15.0	3.0 (fighters only!)	
Weapons:	see Part IV, Chapter 3 for dimensioned data on weapons and stores. Some statistical data are:			
conformal carriage:		1.5 ft^3/1,000 lbs		3.0
internal bay:				
	tube launch:	20 ft^3/1,000 lbs		weapon
	ejector launch:	33 ft^3/1,000 lbs		weapon
Guns:	Mk61 20 mm	14.0		gun + 2
	Oerlikon 30 mm	4.8		gun + 2
Ammunition:	20 mm	0.013 ft^3/round		
	30 mm	0.052 ft^3/round		

Main Gear: $(9 + 10^{-6}(2.56CBR - 4.86)W_{TO}(1.924CBR)^{-0.158}$

For CBR (California Bearing Ratio) definitions and its relation to airfield type, see Ref.19.

For fighters: CBR = 3.0 - 5.0 For transports: CBR = 12

Arresting gear: 4.0

Hydr./pneum. systems: $0.00046W_{TO}$

Electrical systems: 4.0 (for fighters only)

Armor: 1.0

Environmental system: 15.0 (for fighters only)

Table 4.2b Critical Volume, Cross Section and Length Data

Element	Volume (ft^3)	Cross Section (ft^2)	Length (ft)
Oxygen system:	6.0 per crew member for fighters		
Auxiliary gear:	$0.00005W_{TO}$		
Control runs:		1.0 ft^2 between the cockpit and empennage	
Fuselage structure:	$0.13(l_f/d_f)S_{wet_{fus}}$		

Wing: volume outside the body, computed from the wing layout drawing of Step 6.13 in Part II.

Wing sweep mechanism: $0.4(c_p)^2(t/c)_{max}w_f$, c_p= chord at pivot

Body fuel: use a volumetric efficiency of 85 percent for fuselage tanks.

Refuelling system: 2 ft^3, normally in the forebody

Empennage: volume outside the body, computed from the empennage layout drawing of Step 8.7, Part II.

Streamtube: that volume taken up by inlets, engines, tailpipes, afterburners and nozzles.

Wasted volume: it is not possible to package a fuselage without wasted volume. Many system components require accessibility: this adds volume. The following equation is suggested for computing wasted volume:

$$V_{waste} = 0.212\{3(Vol._{fus}/l_f) + 2.7(Vol_{fus}l_f)^{1/2}\}$$

where: Vol_{fus} = sum of all previous volume requirements.

l_f is the overall fuselage length

Total volume: = $Vol_{fus} + V_{waste}$.

Inlet stream tube: that volume taken up by the inlet up to the engine compressor faces. This volume is subtracted from the total volume in arriving at the net cross-sectional area distribution.

Step 3: Translate the preliminary airplane layout into a cross sectional area plot as shown in Figure 4.32. Make certain that all volumetric and critical length requirements of Step 1 are satisfied.

Step 4: Decide which of the Sears-Haack bodies of Figure 4.31 best fit the cross sectional area plot of Step 3. Overlay the selected Sears-Haack cross sectional area plots.

Step 5: Adjust the fuselage geometry until the airplane cross sectional area plot MINUS its inlet capture area conforms to the selected Sears-Haack shapes.

Figure 4.33 shows an example of a cross sectional area distribution of a fighter airplane. Figure 4.34 shows several critical cross sections.

Note that the 'bump' in the area distribution due to the canopy are eliminated by indenting the local fuselage cross section. The Dassault Rafale is an example of a fighter airplane where this has also been done.

Note that the 'bump' in the area distribution due to the wing has been eliminated also by indenting the local fuselage cross section. The result is a 'coke-bottle' shape. Further examples of this coke-bottling are shown in Figures 3.54 - 3.56 in Part II.

Airplanes which fly at high subsonic Mach numbers may require so-called 'local' area ruling. Examples of 'local' area ruling are found in the Boeing 737-300 wing/nacelle region, in the Northrop F-5 wing/tiptank region and in the DC-10 center-engine/vertical tail region.

In using 'local' area ruling, only local cross sectional area distributions are constructed. When these local cross sectional area distributions are 'smooth', no major problems in dragrise at high subsonic Mach numbers should be expected.

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FIRST STAGE AIRPLANE CROSS SECTION DEFINITION (TARGET)

Figure 4.32 Development of a Cross Sectional Area Distribution from Volumetric Requirements

Figure 4.33 Example Cross Sectional Area Distribution

Figure 4.34 Critical Cross Sections for Figure 4.33

4.4 EMPENNAGE DRAG COEFFICIENT PREDICTION

4.4.1 Subsonic Empennage Drag Coefficient

The subsonic empennage drag coefficient is found from:

$$C_{D_{emp}} = SUM_i\{(C_{D_{o_{emp}}})_i + (C_{D_{L_{emp}}})_i\} \qquad (4.45)$$

where: $(C_{D_{o_{emp}}})_i$ = empennage zero-lift drag coefficient of the number i empennage surface, see 4.4.1.1.

$(C_{D_{L_{emp}}})_i$ = empennage drag coefficient due to lift of the number i empennage surface, see 4.4.1.2.

4.4.1.1 Empennage zero-lift drag coefficient

The empennage zero-lift drag coefficient, $(C_{D_{o_{emp}}})_i$ may be computed with Equation (4.6) after replacing the appropriate wing parameters with those for the empennage.

It is assumed here that any airplane may have the following empennage surfaces:

* horizontal tail surface(s),
* canard surface(s) and
* vertical tail surface(s)

How many of these surfaces an airplane has and where they are located depends strongly on the configuration of the airplane. Chapter 3 of Part II presents more than 150 airplane configurations with different approaches to empennage layout design. Chapter 8 of Part II as well as Chapter V of Part III contain methods and examples for the sizing of empennage surfaces.

NOTE: For each of these empennage surfaces Eqn.(4.6) must be used with the following substitutions:

$S_{wet_w} = (S_{wet_{emp}})_i$

$R_{wf} = 1.0$

R_{LS} = as determined from Figure 4.2, but using the individual empennage surface sweep angle.

$C_{f_w} = C_{f_{emp}}$ which follows from Figure 4.3 at the appropriate empennage Reynolds number computed with as characteristic length, $\bar{c}_{emp_e}$, the exposed mean geometric chord for each empennage surface.

L' = as determined from Figure 4.4, using the maximum thickness location associated with each empennage airfoil.

(t/c) = maximum thickness ratio associated with each empennage airfoil, taken at its exposed mean geometric chord.

The empennage wetted area, $(S_{wet_{emp}})_i$ will normally be different for each empennage surface.

4.4.1.2 Empennage drag coefficient due to lift

The empennage drag coefficient due to lift, $(C_{D_{L_{emp}}})_i$ may be computed with the following method:

Horizontal Tail(s) and Canard(s)

Most canard and horizontal tail surfaces develop lift in a steady state flight condition. This lift causes induced drag.

In this text, the lift carried by empennage surfaces is split into two components:

1. Lift caused by the reference incidence setting of any empennage surface: the induced drag caused by this lift is accounted for in this paragraph.

2. Lift caused by the requirement to 'trim' the airplane at a particular center of gravity location: the induced drag caused by this lift is referred to as 'trim' drag: see Section 4.10.

If the empennage employs the so-called butterfly arrangement, the projection method of page 206 in Part II may be used to arrive at the 'equivalent horizontal' and the 'equivalent vertical' tail surface.

The amount of lift on a horizontal tail surface can

be computed from:

$$C_{L_h} = C_{L_{\alpha_h}}(\alpha_h - \alpha_{o_{L_h}}) \tag{4.46}$$

with: $\alpha_h = \alpha(1 - d\epsilon/d\alpha) + i_h$ (4.47)

The amount of lift on a canard surface can be computed from:

$$C_{L_c} = C_{L_{\alpha_c}}(\alpha_c - \alpha_{o_{L_c}}) \tag{4.48}$$

with: $\alpha_c = \alpha(1 + d\epsilon_c/d\alpha) + i_c$ (4.49)

Methods for predicting downwash ϵ, upwash ϵ_c and the zero lift angles of attack, $\alpha_{o_{L_h}}$ and $\alpha_{o_{L_c}}$, are presented in Chapter 10.

In airplanes with a fixed horizontal tailplane, i_h is usually zero. In airplanes with a fixed canard, i_c is normally set to assure that the canard stalls before the wing. In preliminary design it is acceptable to set:

$$i_c = \alpha_{stall_c} - \alpha_{stall_w} + i_w + 3 \tag{4.50}$$

where the values for α_{stall_c} and α_{stall_w} are taken as the airfoil stall angles of attack at the m.g.c. of their respective surfaces and where the wing incidence angle, i_w may be found from Tables 6.7 in Part II or from page 197 in Part III.

Since horizontal tails and canards are normally not twisted, their drag-due-to-lift may be computed from Eqn.(4.8) with ϵ_t=0, to yield:

$$C_{D_{L_{emp}}} = \{(C_{L_h})^2/\pi A_h e_h\}S_h/S + \{(C_{L_c})^2/\pi A_c e_c\}S_c/S \tag{4.51}$$

This equation applies to an airplane equipped with one horizontal tail and with one canard. If more empennage surfaces are present, they must be accounted for in a manner analogous to Eqn.(4.51).

Values for C_{L_h} and C_{L_c} follow from Eqns (4.46) and (4.48) respectively.

Values for A_h, A_c, S_h and S_c follow from airplane threeviews as developed in Chapter 13 of Part II.

For the Oswald efficiencies e_h and e_c the following values are suggested:

e_h = 0.5 for fuselage mounted tails

= 0.75 for T-tails

e_c = 0.5

Vertical Tail Surface(s)

Vertical tails are normally installed symmetrically so that the vertical tail drag contribution due to lift is usually zero. If the airplane is sideslipping over an angle β, this angle β should be considered as the angle of attack of the vertical tail. A modified version of Eqn.(4.51) may then be used to calculate the drag due to lift (in sideslip) of the vertical tail.

4.4.2 Transonic Empennage Drag Coefficient

The transonic empennage drag coefficient may be found from:

$$C_{D_{emp}} = SUM_i\{(C_{D_{o_{emp}}})_i + (C_{D_{L_{emp}}})_i\} \qquad (4.52)$$

where: $(C_{D_{o_{emp}}})_i$ = empennage zero-lift drag coefficient of the number i empennage surface, see 4.4.2.1.

$(C_{D_{L_{emp}}})_i$ = empennage drag coefficient due to lift of the number i empennage surface, see 4.4.2.2.

4.4.2.1 Empennage zero-lift drag coefficient

In the transonic speed range the empennage zero-lift drag coefficient is found from:

$$(C_{D_{o_{emp}}})_i = (C_{D_{o_{emp}}})_i + (C_{D_{emp_{wave}}})_i (S_{emp})_i / S \quad (4.53)$$

at M=0.6

where: $(C_{D_{o_{emp}}})_i$ at M=0.6 = empennage drag coefficient due to friction of the number i empennage surface at M=0.6. It is obtained from Eqn.(4.6) with the same substitutions as noted in 4.4.1.1.

$(C_{D_{emp_{wave}}})_i$ = empennage zero-lift wave drag coefficient obtained with the procedure of 4.2.2.2 but with the appropriate substitution of empennage (emp) paparameters for wing (w) parameters.

To achieve acceptable wave drag characteristics, the entire configuration, including its empennage surfaces must be 'area-ruled'. A procedure for 'area-ruling' any configuration is given in Sub-section 4.3.4.

4.4.2.2 Empennage drag coefficient due to lift

The transonic empennage drag coefficient due to lift, $(C_{D_{L_{emp}}})_i$ may be computed from Eqn.(4.18), as long as the following substitutions are used:

To find $(C_{D_L}/C_L^2)_{emp_i}$ from Figures 4.13, use the appropriate transonic similarity parameters for the number i empennage surface.

For C_{L_i} use either C_{L_h} or C_{L_c} as in Equations (4.46) or (4.48).

4.4.3 Supersonic Empennage Drag Coefficient

The supersonic empennage drag coefficient is found from:

$$C_{D_{emp}} = SUM_i \{(C_{D_{o_{emp}}})_i + (C_{D_{L_{emp}}})_i\} \quad (4.54)$$

where: $(C_{D_{o_{emp}}})_i$ = zero lift drag coefficient of the number i empennage surface, see 4.4.3.1.

$(C_{D_{L_{emp}}})_i$ = drag coefficient due to lift of the number i empennage surface, see 4.4.3.2.

4.4.3.1 Empennage zero-lift drag coefficient

The supersonic empennage zero lift drag coefficient is found from:

$$C_{D_{o_{emp}}})_i = (C_{D_{emp_f}})_i + (C_{D_{emp_{wave}}})_i \qquad (4.55)$$

where: $$(C_{D_{emp_f}})_i = (C_{f_{emp}})_i (S_{wet_{emp}})_i / S \qquad (4.56)$$

where: $(C_{f_{emp}})_i$ may be found from Fig.4.3 at the appropriate empennage Reynolds number computed with as characteristic length, $\bar{c}_{emp_e}$, the exposed mean geometric chord of each empennage surface.

$(C_{D_{emp_{wave}}})_i$ = wave drag coefficient of the number i empennage surface. This wave drag coefficient may be determined with the method of 4.2.3.1 as long as the appropriate empennage parameters are substituted for their wing counterparts.

4.4.3.2 Empennage drag coefficient due to lift

The supersonic empennage drag coefficient due to lift is found from:

$$(C_{D_{L_{emp}}})_i = (C_{D_L}/C_L^2)_{emp_i} (C_{L_{emp}}^2)_i \qquad (4.57)$$

Values for $(C_{D_L}/C_L^2)_{emp_i}$ may be obtained from the method of 4.2.3.2, as long as the appropriate parameter substitutions are made.

Values for $(C_{L_{emp}})_i$ follow from either Eqn.(4.46) or from Eqn.(4.48).

4.5 NACELLE/PYLON DRAG COEFFICIENT PREDICTION

Methods for predicting nacelle/pylon drag are presented as follows:

4.5.1 Isolated Nacelle/Pylon Drag Coefficient

This method is applicable in all speed regimes considered in this text.

4.5.2 Installed Nacelle/Pylon Drag Coefficient Increment

This method applies only in the subsonic speed regime and provides the 'interference' drag increment due to nacelle/pylon installations.

It is assumed that this 'interference' drag increment is independent of Mach number as long as the appropriate area ruling is used in the transonic and supersonic speed ranges. See Sub-section 4.3.4 for details on area ruling.

4.5.3 Windmilling Drag and Propeller Drag Coefficients

This method provides the drag increase caused by engine and/or propeller stoppage.

These methods should be applied as follows:

Step 1: Determine the subsonic installed drag increment due to the nacelle or the nacelle/pylon combination from Sub-section 4.5.2. Again: this drag increment represents the interference drag of the installation.

Step 2: Determine the drag increments due to transonic and supersonic operations from Sub-section 4.5.1 which assumes the nacelle/pylon to be isolated.

Step 3: Add the transonic and/or supersonic increments from Step 2 to the nacelle/pylon interference drag obtained in Step 1.

Step 4: For windmilling and/or propeller drag increments use Sub-section 4.5.3. Add this drag increment to the drag obtained in Step 3.

4.5.1 Isolated Nacelle/Pylon Drag Coefficient

The nacelle/pylon drag coefficient may be computed from:

$$C_{D_{np}} = C_{D_n} + C_{D_p} \tag{4.58}$$

where: C_{D_n} = nacelle drag coefficient, see 4.5.1.1.

C_{D_p} = pylon drag coefficient, see 4.5.1.2.

4.5.1.1 Nacelle drag coefficient

Engine operating:

The nacelle drag coefficient is found from:

$$C_{D_n} = SUM_i (C_{D_n})_i \tag{4.59}$$

The number i equals the number of nacelles on the airplane. Each nacelle drag coefficient, $(C_{D_n})_i$ is computed with the method of Section 4.3. This assumes that a nacelle can be treated as if it is a (small) fuselage.

Note: nacelles will generate lift when placed at an angle of attack. Local nacelle angle of attack should be expressed as:

$$\alpha_n = \alpha_w + i_n + \varepsilon_n \tag{4.60}$$

where: α_w = wing angle of attack

i_n = nacelle incidence angle, see Figure 4.35.

ε_n = nacelle upwash or downwash angle. This depends on where the nacelle is installed on the airplane. Figure 4.36 illustrates two possibilities.

The nacelle up- or downwash angle follows from:

$$\varepsilon_n = (d\varepsilon_n/d\alpha)\alpha \tag{4.61}$$

where: $d\varepsilon_n/d\alpha$ is computed with the methods described in Chapter 10.

Figure 4.35 Definition of Nacelle Incidence Angle

Figure 4.36 Examples for Nacelle Upwash and Downwash

Figure 4.37 Examples of Vertically Installed Pylons

Engine Inoperative:

In addition to the drag given by Eqn.(4.59) there will be extra drag caused by engine or propeller windmilling. If a propeller is stopped, there will be extra drag due to that.

Sub-section 4.5.3 deals with such cases.

4.5.1.2 Pylon drag coefficient

The pylon drag coefficient may be found from:

$$C_{D_p} = SUM_i \ (C_{D_p})_i \qquad (4.62)$$

It will be assumed here that the drag behavior of a pylon can be modelled as that of an empennage. For this reason, values for $(C_{D_p})_i$ may be obtained by using the method of Section 4.4.

For vertically installed pylons (see Figure 4.37), the pylon should be treated as a vertical tail: the drag prediction method of Section 4.4 may be used.

For horizontally installed pylons (see Figure 4.38) the pylon should be treated as a horizontal tail or as a canard: use the drag prediction method of Section 4.4.

NOTA BENE 1: Nacelles installed in the proximity of wings (see Figure 4.37a) must be installed so that they are at zero local sideslip angle in the local flowfield. This results in the need for 'toe-in' for nacelles such as in the Boeing 747 series airplanes: see Figure 4.39.

NOTA BENE 2: The methods of 4.5.1.1 and 4.5.1.2 apply to all speed regimes. However, in the transonic and supersonic speed regimes the method is valid only as long as the nacelle/pylon combination is included in the area-ruling process described in Sub-section 4.3.3. Note: the nacelle stream tube should not be included in this area ruling process.

4.5.2 Installed Nacelle/Pylon Drag Coefficient Increment

Sub-section 4.5.1 provides a method for predicting the drag coefficient, $C_{D_{np}}$ due to isolated nacelle/pylon combinations.

In the following, a method for computing the interference drag increment caused by a nacelle/pylon

Figure 4.38 Examples of Horizontally Installed Pylons

Figure 4.39 Example of Nacelles with Toe-in

Figure 4.40 Definition of Nacelle Parameters

installation is presented. The method is broken down into three parts:

4.5.2.1 Wing/nacelle interference drag coefficient
4.5.2.2 Fuselage/nacelle interference drag coefficient
4.5.2.3 Cooling drag coefficient increment

Which drag increment should be accounted for depends on the airplane configuration.

4.5.2.1 Wing/nacelle interference drag coefficient

For turboprop and for piston-prop nacelle/wing installations the interference drag coefficient is found from:

$$C_{D_{n_{int}}} = 0.036\,(c_n b_n/S)(\Delta c_{l_1} + \Delta c_{l_2})^2 \qquad (4.63)$$

where: c_n and b_n are defined in Figure 4.40.

Δc_{l_1} = 0.2 for a nacelle on top of the wing

= -0.3 for a nacelle below the wing

$\Delta c_{l_2} = -\,0.056(i_n)$

i_n = nacelle incidence angle defined in Figure 4.35.

For jet-driven airplanes the wing/nacelle interference drag coefficient is found from:

$$C_{D_{n_{int}}} = F_{a_1}(\Delta C_{D_n}/C_{D_n})C_{D_n} \qquad (4.64)$$

where: $(\Delta C_{D_n}/C_{D_n})$ follows from Figure 4.41 at the Mach numbers indicated.

C_{D_n} = nacelle drag coefficient as determined in 4.5.1.1.

F_{a_1} = 1.0 for $M_\infty < 0.5$

1.0 for $M > 0.5$ and no local area ruling of the nacelle/pylon/wing intersection.

0.5 for $M > 0.5$ and a locally area ruled nacelle/pylon/wing intersection.

Note: Valid for swept wings only

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Figure 4.41 Wing-Nacelle Drag Interference Factor

4.5.2.2 Fuselage/nacelle interference drag coefficient

The fuselage/nacelle interference drag coefficient may be found from:

$$C_{D_{n_{int}}} = F_{a_2}\{(C_{D_n})' - 0.05\}(S_n/S) \qquad (4.65)$$

where: $(C_{D_n})'$ is found from Figure 4.42

S_n = maximum frontal area of the nacelle, excluding the pylon: see Figure 4.38.

F_{a_2} = 1.0 for fuselage/nacelle intersections without local area ruling

= 0.5 for fuselage/nacelle intersections with local area ruling.

4.5.2.3 Cooling drag coefficient increment

The cooling drag coefficient increment for radial air-cooled engine installations may be estimated from:

$$C_{D_{n_{cool}}} = (\Delta C_{D_{cool}})(\pi(d_n)^2/4S) \qquad (4.66)$$

where: $\Delta C_{D_{cool}}$ follows from Figure 4.43.

d_n = the maximum nacelle diameter as shown in Figure 4.38.

4.5.3 Windmilling Drag and Propeller Drag Coefficients

The following cases should be considered:

1.) Windmilling drag coefficient due to jet engines: see 4.5.3.1.

2.) Windmilling drag coefficient due to propellers: see 4.5.3.2.

3.) Drag coefficient due to a stopped propeller: see 4.5.3.3.

4.5.3.1 Windmilling drag coefficient due to jet engines

The incremental drag coefficient due to a windmilling jet engine may be estimated from:

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Note: 1. C_{D_n} is based on maximum nacelle frontal area

2. drag coefficient of the free nacelle is 0.05

Figure 4.42 Fuselage-Nacelle Drag Interference Factor

①	②	③	④	⑤	⑥	⑦	⑧
$\frac{d_f}{\ell_f}=1,78$	1,78	1,62	1,5	1,63	1,63	1,63	3,42
$\frac{a}{\ell_f}=0,59$	0,59	0,59	0,63	0,66	0,66	0,66	0,16
$\Delta C_D^{*}=0,16$	0,075	0,10	0,07	0,06	0,088	0,212	0,152

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*) ΔC_D BASED ON: $\frac{\pi}{4} d_f^2$

Figure 4.43 Cooling Drag Coefficient Increment

$$\Delta C_{D_{wmj}} = 0.0785(d_{inl}^2)/S +$$

$$+ \{2/(1 + 0.16M^2)\}(V_{noz}/U_1)(1 - V_{noz}/U_1)S_{noz}/S \quad (4.67)$$

where: d_{inl} = engine inlet diameter

S_{noz} = nozzle cross section area

V_{noz}/U_1 = ratio of average flow velocity in the nozzle to the steady state flight speed The following values are suggested for this ratio:

V_{noz}/U_1 = 0.25 for turbojets and turboprops

= 0.42 for low by-pass ratio jet engines

= 0.12 for the primary airflow of high by-pass jet engines

= 0.92 for the fan airflow of high bypass jet engines

4.5.3.2 Windmilling drag coefficient due to propellers

Reference 20 provides a method for estimating the windmilling drag due to a propeller. In absence of that reference it is suggested to use:

$$\Delta C_{D_{wmprop}} = 33(1/\bar{q}S)SHP_{rated}/U_1 \quad (4.68)$$

where: SHP_{rated} = maximum rated shaft horsepower of the engine in the flight condition being considered. Note: use hp!

U_1 = steady state flight speed, in fps.

4.5.3.3 Drag coefficient due to a stopped propeller

The incremental drag coefficient due to a stopped propeller may be estimated from:

$$\Delta C_{D_{prop}} = 0.00125n_p(D_p)^2/S \quad (4.69)$$

where: n_p = number of blades per propeller

D_p = diameter of the propeller

4.6 FLAP DRAG PREDICTION

The assumption is made that flaps will be deployed only in the subsonic speed regime. For the effect of flap drag at high speed the reader should consult reference 9.

The drag coefficient due to flap deflection may be estimated from:

$$C_{D_{flap}} = \Delta C_{D_{prof_{flap}}} + \Delta C_{D_{i_{flap}}} + \Delta C_{D_{int_{flap}}} \qquad (4.70)$$

where: $\Delta C_{D_{prof_{flap}}}$ = the flap profile drag increment, see 4.6.1.

$\Delta C_{D_{i_{flap}}}$ = the induced drag increment due to the flap, see 4.6.2.

$\Delta C_{D_{int_{flap}}}$ = the interference drag increment due to the flap, see 4.6.3.

4.6.1 Flap Profile Drag Increment

The following method applies to wings with sweep angles up to 40 degrees. For higher sweep angles the reader should consult Reference 9.

The flap profile drag increment may be found from:

$$\Delta C_{D_{prof_{flap}}} = (\Delta C_{d_{p_{\Lambda_{c/4}=0}}})(\cos\Lambda_{c/4})S_{wf}/S \qquad (4.71)$$

where: $\Delta C_{d_{p_{\Lambda_{c/4}=0}}}$ = the two-dimensional profile drag increment due to flaps.

This increment depends on the type of flaps used:

For <u>plain</u> flaps: use Figure 4.44.

For <u>split</u> flaps, use Figure 4.45.

For <u>single slotted</u> flaps, use Figure 4.46.

For <u>double slotted</u> flaps: use Figure 4.47.

For <u>Fowler</u> flaps: use Figure 4.48.

Figure 4.45 Profile Drag Increment: Split Flaps

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Figure 4.44 Profile Drag Increment: Plain Flaps

VALID FOR $\Lambda_{c/4} = 0°$

Figure 4.46 Profile Drag Increment: Single Slotted Flaps

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VALID FOR $\Lambda_{c/4} = 0°$

Figure 4.47 Profile Drag Increment: Double Slotted Flaps

Figure 4.48 Profile Drag Increment: Fowler Flaps

Figure 4.49 Definition of Flap Chords

For <u>Krueger</u> flaps, use:

$$\Delta C_{d_{p_{\Lambda_{c/4}=0}}} = C_{D_{o_w}} (c_{kf}/c) \qquad (4.72)$$

where: $C_{D_{o_w}}$ is obtained from Eqn.(4.1).

c_{kf}/c = ratio of wing chord with Krueger flap extended to that of the wing with the Krueger flap retracted: see Figure 4.49.

For <u>slats</u>, use:

$$\Delta C_{d_{p_{\Lambda c/4}=0}} = C_{D_{o_w}} (c_s/c) \qquad (4.73)$$

where: c_s/c = ratio of wing chord with slats extended to that of the wing with the slats retracted: See Figure 4.49.

Figure 4.50 illustrates these various flap types.

$\Lambda_{c/4}$ = wing quarter chord sweep angle.

S_{wf} = the flapped wing area: see Figure 4.50.

4.6.2 Induced Drag Increment due to Flaps

The induced drag increment due to flaps may be estimated from:

$$\Delta C_{D_{i_{flap}}} = K^2 (\Delta C_{L_{flap}})^2 \cos\Lambda_{c/4} \qquad (4.74)$$

where: $\Delta C_{L_{flap}}$ = the incremental lift coefficient due to the flap. This quantity follows from comparing a flaps-up with a flaps-down C_L-versus-α curve: see Figure 4.51.

Chapter 9 presents a method for constructing Fig.4.51.

K = an empirical constant which follows from Figures 4.52 and 4.53.

4.6.3 Interference Drag Increment due to Flaps

The interference drag increment due to flaps may be estimated from:

Figure 4.50 Typical Flap Types for Figures 4.44 - 4.48

Figure 4.51 Example of Flaps-up and Flaps-down Lift Coefficient Versus Angle-of-Attack Behavior

$$\Delta C_{D_{int_{flap}}} = K_{int} \Delta C_{D_{prof_{flap}}} \qquad (4.75)$$

where: $\Delta C_{D_{prof_{flap}}}$ follows Sub-section 4.6.1.

K_{int} = - 0.15 for split flaps

= 0 for plain flaps

= + 0.40 for slotted flaps

= + 0.25 for Fowler flaps

= + 0.10 for slats and for Kruegers

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Figure 4.52 Induced Drag Factor for Uninterrupted Flaps

Figure 4.53 Induced Drag Factor for Interrupted Flaps

4.7 LANDING GEAR DRAG PREDICTION

The assumption is made that landing gear deployment will occur only in the subsonic speed regime. The method presented here applies only at low speed.

The landing gear drag coefficient for an airplane with i landing gears may be computed from:

$$C_{D_{gear}} = SUM_i[\{(C_{D_{gear_{C_L=0}}})_i + p_i C_L\}(S_{gear})_i/S] \qquad (4.76)$$

where: $C_{D_{gear_{C_L=0}}}$ = the zero-lift drag coefficient of the landing gear based on its own reference area, S_{gear}. Figures 4.54 through 4.60 provide data from which the zero-lift drag coefficient of the landing gear may be determined.

S_{gear} = reference area for the zero-lift gear drag coefficient. This reference area is defined in Figures 4.54-4.60. For most landing gear types: $S_{gear} = b_t x D_t$, where b_t is the tire width and D_t is the tire diameter.

p = a factor which accounts for the variation of gear drag with lift. Figure 4.61 gives a method for determining this factor p.

Airplanes can have a number of different types of landing gear. The summation over i landing gears in Eqn.(4.76) indicates this.

Figures 4.54 - 4.60 should be used as follows:

<u>For fixed landing gears:</u> Figures 4.54 and 4.55 apply to non-retractable (= fixed) landing gears attached to wing or fuselage. Note that these drag coefficients are based on: $S_{gear} = b_t x D_t$.

Figure 4.56 applies to non-retractable (= fixed) landing gears attached to nacelles. These drag coefficients are based on: $S_{gear} = b_t x D_t$.

<u>Important note:</u> The data in Figures 4.54 - 4.56 apply to the entire main gear which is assumed to consist of two of the legs shown in these figures.

Note: 1. All C_D values are referenced to $b_t x D_t$ and are valid for $C_L=0$

2. All landing gears shown in this figure are assumed to have streamlined tires such as shown in Figure 4.57

TYPE 1

Effect of streamlining

Non streamlined struts	: $C_D=2.56$
Streamlined struts	: $C_D=1.11$
Struts also streamlined at intersections A and B	: $C_D=0.93$
Struts also streamlined at intersections A, B and C	: $C_D=0.85$

Effect of wheel fairings as shown in Figure 4.57

Fairing type A	: $C_D=1.15$
Fairing type B	: $C_D=1.05$
Fairing type C	: $C_D=0.71$

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TYPE 2

Without wheel fairing	: $C_D=0.565$
With fairing type B	: $C_D=0.54$
With fairing type C	: $C_D=0.49$

as shown in Figure 4.57

Figure 4.54 Gear Drag Increment: Non-retractable Gears Attached to Wing or Fuselage, Types 1-2

Without wheel fairing : C_D=0.62

With fairing type A : C_D=0.46
as shown in Figure 4.57

Type 4, narrow strut, no fairing : C_D=0.52

Type 4a, Narrow strut, small fairing : C_D=0.34

Type 4b, strut and wheel faired completely : C_D=0.34

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With wheel fairing type C, according to Figure 4.57 : C_D= .68

Without wheel fairing : C_D=1.05

Figure 4.55 Gear Drag Increment: Non-retractable Gears Attached to Wing or Fuselage, Types 3-5

Note: All C_D values are referenced to $b_t \times D_t$ of one wheel, but apply to the entire landing gear, including interference drag

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C_D = 0,45 - 0,74

C_D = 0,45 - 0,60

C_D = 0,60 - 0,90

C_D = 1,85 - 2,10

Valid for C_L=0 only

Figure 4.56 Gear Drag Increment: Non-retractable Gears Attached to Nacelles

For landing gear wheels alone: Figure 4.57 applies to wheels with and without streamline caps (fairings). Note that these drag coefficients are based on: $S_{gear} = b_t x D_t$.

For nose gears: Figure 4.58 applies to nose gears with closed nosewheel doors. Note that the reference area is again: $S_{gear} = b_t x D_t$.

For retractable landing gears: Figure 4.59 provides the data. Nota bene: these data are based on the reference area: (axb) as defined in Figure 4.59. The drag coefficients again apply to the entire gear which is assumed to consist of two legs.

For landing gears with more than one wheel per bogey: Figure 4.61 shows a relationship between Δf_{gear} and W_{TO} for different types of landing gear. From these data:

$$C_{D_{gear}} = \Delta f_{gear}/S \qquad (4.77)$$

Many airplane configurations employ 'landing-gear-bulge-fairings'. Many examples of this are depicted in Chapter 3 of Part II. The extra drag created by these bulge fairings should be accounted for in the the estimation of fuselage drag: see Section 4.3. Note that a bulge fairing will add to the cross sectional area of the fuselage and therefore will reduce the 'effective' fuselage slenderness ratio parameter l_f/d_f. This causes an increase in fuselage drag as given by Eqn.(4.30).

Drag Coefficients of Wheels

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Drag Coefficients of Wheels With Fairings

Figure 4.57 Wheel Drag Increments

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NOTE:
GEAR CAVITY
CLOSED!

Figure 4.58 Nosegear Drag Increments

NOTE:
GEAR CAVITY
OPEN!

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Note: All C_D values are referenced to axb of one gear, but apply to the entire landing gear, including interference drag

Figure 4.59 Gear Drag Increments: Retractable Gears

Figure 4.60 Equivalent Parasite Area Increment for Gears with Multiple Wheel Bogies

Figure 4.61 Landing Gear Induced Drag Factor

4.8 CANOPY/WINDSHIELD DRAG PREDICTION

The following method applies in the subsonic speed range. In the transonic and supersonic speed ranges the wave drag generated by canopies and windshields can be significant. In these speed ranges it will be necessary to employ area-ruling to cut wave drag to a minimum. Sub-section 4.3.4 presents a discussion on area-ruling.

The drag coefficient due to a canopy and/or a windshield may be found from:

$$C_{D_{cw}} = C_{D_{can}} + C_{D_{ws}} \tag{4.78}$$

where: $C_{D_{can}}$ = drag coefficient due to a canopy, see 4.8.1.

$C_{D_{ws}}$ = drag coefficient due to a windshield, see 4.8.2.

4.8.1 Canopy Drag Prediction

The drag coefficient due to a canopy may be estimated from:

$$C_{D_{can}} = (\Delta C_{D_{can}})S_{can}/S \tag{4.79}$$

where: $\Delta C_{D_{can}}$ = the incremental drag coefficient due to the canopy. Figures 4.62 through 4.67 provide data from which this increment may be found. Note from Figure 4.67, that this increment varies with Mach number in the subsonic speed range!

S_{can} = the maximum frontal area of the canopy as shown in Figure 4.62.

4.8.2 Windshield Drag Prediction

The drag coefficient due to a windshield may be determined from:

$$C_{D_{ws}} = (\Delta C_{D_{ws}})S_{fus}/S \tag{4.80}$$

where: $\Delta C_{D_{ws}}$ = incremental drag coefficient due to a windshield: see Figure 4.68.

$\Delta C_{D_{can}}$ = Canopy Drag Coefficient Referenced to Frontal Area of Canopy, S_{can}

Figure 4.62 Effect of L_1 on Canopy Drag

$\Delta C_{D_{can}}$ = Canopy Drag Coefficient Referenced to Frontal Area of Canopy, S_{can}

Figure 4.63 Effect of 'r' on Canopy Drag

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$\Delta C_{D_{can}}$ = Canopy Drag Coefficient Referenced to Frontal Area of Canopy, S_{can}

Figure 4.64 Effect of L_3 on Canopy Drag

COPIED FROM: REF. 21

α = Fuselage Angle of Attack

$\Delta C_{D_{can}}$ = Canopy Drag Coefficient Referenced to Frontal Area of Canopy, S_{can}

Figure 4.65 Effect of Cross Section on Canopy Drag

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α = Fuselage Angle of Attack

$\Delta C_{D_{can}}$ = Canopy Drag Coefficient Referenced to Frontal Area of Canopy, S_{can}

Figure 4.66 Effect of Irregularities on Canopy Drag

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$\Delta C_{D_{can}}$ = Canopy Drag Coefficient Referenced to Frontal Area of Canopy, S_{can}

Figure 4.67 Effect of Mach Number on Canopy Drag

$C_{D_{fus}}$ = Fuselage Drag Coefficient Referenced to the Frontal Area of the Fuselage

$\Delta C_{D_{WS}}$ = Drag Coefficient of the Windshield Referenced to the Frontal Area of the Fuselage, S_{fus}

Figure 4.68 Effect of Windshield Configuration on Drag

4.9 STORE DRAG PREDICTION

Reference 9 provides detailed methods for estimating the drag due to many external store arrangements. Lacking that reference it is suggested to proceed as follows:

Step 1: Determine the isolated store drag coefficient, $(C_{D_{store}})_i$ for each store from the method of Section 4.3 by considering each isolated store to be like a fuselage.

Step 2: Compute the total store drag coefficient from:

$$C_{D_{store}} = SUM_i\{(K_{store})_i(C_{D_{store}})_i\} \qquad (4.81)$$

where: $(K_{store})_i$ = store interference factor:

For semi-submerged stores as shown in Figure 4.69: $K_{store} = 0.7$

For external stores as shown in Figure 4.70: $K_{store} = 1.3$

$(C_{D_{store}})_i$ = drag coefficient of the isolated store, see Step 1.

Additional data on the drag of various externally carried stores may be found in Reference 8.

Figure 4.69 Example of Semi-submerged (Conformal) Stores

Figure 4.70 Example of External Stores

4.10 TRIM DRAG PREDICTION

Trim drag is caused by lift generated on a horizontal tail and/or a canard as a result of the requirement to 'moment trim' an airplane. Figure 4.71 illustrates two types of trim forces to be considered here. These trim force requirements result in two types of drag:

1. Trim drag due-to-lift generated on the trimming surface
2. Profile drag generated on the trimming surface as due to a control surface or a flap deflection.

The total trim drag coefficient may be found from:

$$C_{D_{trim}} = \Delta C_{D_{trim_{lift}}} + \Delta C_{D_{trim_{prof}}} \qquad (4.82)$$

where: $\Delta C_{D_{trim_{lift}}}$ = the induced drag coefficient increment due to the need for generating trim lift: see 4.10.1.

$\Delta C_{D_{trim_{prof}}}$ = the profile drag coefficient increment due to the need for generating trim lift: see 4.10.2.

4.10.1 Trim Drag Due to Lift

The trim drag due-to-lift increment, $\Delta C_{D_{trim_{lift}}}$ may be estimated from:

$$\Delta C_{D_{trim_{lift}}} = \{(\Delta C_{L_h})^2/\pi A_h e_h\} S/S_h + \{(\Delta C_{L_c})^2/\pi A_c e_c\} S/S_c \qquad (4.83)$$

where: ΔC_{L_h} = the horizontal tail incremental lift coefficient required for trim: see Section 8.3.

ΔC_{L_c} = the canard incremental lift coefficient required for trim: see Section 8.3.

A_h and A_c follow from the airplane threeview as developed in Chapter 13 of Part II.

e_h and e_c are defined on page 68.

4.10.2 Trim Drag Due to Profile Drag

The trim drag increment due to profile drag, is caused by the need to deflect an elevator, a canard-vator or a canard flap for purposes of trimming an airplane. The following equation assumes that trim is accomplished by a combination of elevator deflection and canard flap deflection:

$$\Delta C_{D_{trim_{prof}}} = \Delta C_{D_{p_{\Lambda_{c/4}=0}}} \cos\Lambda_{c/4} (S_{ef}/S_h)(S_h/S) + \Delta C_{D_{p_{\Lambda_{c/4}=0}}} \cos\Lambda_{c/4} (S_{cf}/S_c)(S_c/S) \qquad (4.84)$$

where: $\Delta C_{D_{p_{\Lambda_{c/4}=0}}}$ = the profile drag coefficient due to an elevator and/or a canard control surface. It may be found with the method of Section 4.6, but with the following substitutions:

<u>Elevator and/or canard-vator:</u> these should be considered to be <u>plain</u> flaps. Elevator deflection, δ_e and/or canard-vator deflection, δ_{cv} are used instead of the flap deflection, δ_f.

<u>Canard flap:</u> if trimming is accomplished with a canard flap then it must be determined what type of flap the canard flap resembles most. The data of Sub-section 4.6.1 can be used to estimate the profile drag due to trim by a canard flap.

S_{ef} = flapped horizontal tail area, see Fig.4.72

S_{cf} = flapped canard area, see Fig.4.72

S_h and S_c are the canard and horizontal tail areas respectively.

<u>Note:</u> trim can be accomplished also with a variable incidence stabilizer or with an elevator trim tab. It is left up to the user to make the required modifications in Equations (4.83) and (4.84) to account for such cases.

Figure 4.71 Example of Trim Force Requirements

Figure 4.72 Definition of Flapped Areas

4.11 INTERFERENCE DRAG PREDICTION

Interference drag is a type of drag which is not yet fully understood. Its effect is that the total drag of two or more airplane components when integrated in a configuration is always larger than the sum of the individual component drags.

Several interference drag factors were already accounted for in Sections 4.2, 4.3, 4.5, 4.6 and 4.9. In addition to interference drag accounted for by these factors, extra interference drag may be caused by:

1. Wing struts meeting a wing and/or a fuselage at 'sharp' angles. Figure 4.73 shows such a case.

2. Empennage surfaces meeting each other or a fuselage at 'sharp' angles. Figure 4.74 illustrates several cases.

3. Low or high wing airplanes tend to have more interference drag than midwing airplanes: in the midwing configuration, the 'venturi' effect is absent. Figure 4.75 shows examples.

4. Wing/nacelle installations can have significantly greater interference drag than suggested by the the methods of 4.5.2.1 and 4.5.2.2.

In many instances it is possible to decrease interference drag by the use of so-called 'fairings'. Figures 4.73 - 4.75 show typical fairings.

The reader should carefully examine his configuration for additional causes of interference drag. To estimate their effect, the interference drag data base presented in Reference 8 should be used.

4.12 MISCELLANEOUS DRAG PREDICTION

The following drag items will be considered as miscellaneous drag contributions:

4.12.1 Drag due to spoilers (or speed brakes)
4.12.2 Drag due to surface roughness
4.12.3 Drag due to other causes

4.12.1 Drag Due to Spoilers (or Speed Brakes)

The incremental drag coefficient due to spoilers may be estimated from:

Figure 4.73 Wing Strut Arrangements With High and Low Interference Drag

Figure 4.74 Empennage Arrangements with High and Low Interference Drag

Figure 4.75 Wing/Fuselage Arrangements with High and Low Interference Drag

SOLID PLATES
$\Delta C_{D_{sp}} = 0.63$

SHOWN AT .60 C
NACA 2412
$\Delta C_{D_{sp}} = 1.60$ AT .60 C
1.25 AT 1.00 C

FLAT PLATE DEFLECTED PERPENDICULAR TO FUSELAGE $\Delta C_D = 1.0$

Note: All coefficients are based on the area of the speed brake or spoiler

Figure 4.76 Drag Due to Spoilers and Speed Brakes

$$\Delta C_{D_{sp}} = Sum_i[(\Delta C_{D_{sp_i}})(S_{sp_i}/S)] \quad (4.85)$$

where: $C_{D_{sp_i}}$ is found from Figure 4.76 for several spoiler locations.

S_{sp_i} is the flat plate area of each spoiler.

4.12.2 Drag Due to Surface Roughness

The methods for estimating friction and profile drag of wings and fuselages as discussed in Sections 4.2 and 4.3 apply only to 'smooth' surfaces. If a surface is 'rough', additional drag is created.

The actual level of turbulent boundary layer friction drag of 'rough' surfaces may be estimated with the following procedure:

Step 1: Determine the reference length l: this is $\bar{c}_{w_e}$ in the case of a wing and l_f in the case of a fuselage.

Step 2: Compute the parameter l/k, where k is determined from the following table:

Type of Surface	Equivalent Sand Roughness, k, in ft
Aerodynamically smooth	0.0
Polished metal or polished wood	0.00000167 to 0.00000667
Natural sheet metal	0.00001333
Smooth matte paint, carefully applied	0.00002083
Standard camouflage paint, average application	0.00003333
Camouflage paint, mass-production spray	0.0001
Dip-galvanized metal surface	0.0005
Natural surface of cast iron	0.00083

As a practical matter, use this information as follows:

For light airplanes with standard sheet metal manufacturing methods: use k = 0.00001333 ft.

For business jets and jet transports with sheet metal manufacturing methods but polished surfaces: use k = 0.000005 ft.

For airplanes made of composites and polished surfaces: use k = 0.00000167 ft.

For military airplanes with camouflage paint applied in the factory: use k = 0.00002083 ft.

For military airplanes with camouflage paint applied in the field: use k = 0.0001 ft.

<u>Step 3:</u> Determine the 'cut-off Reynolds number', $R_{N_{cut-off}}$ with the help of Figure 4.77.

<u>Step 4:</u> Determine the actual Reynolds number with Equation (4.7) or Equation (4.31).

<u>Step 5:</u> If $R_{N_{cut-off}} > R_N$ find C_f from Figure 4.3 at the actual Reynolds number.

If $R_{N_{cut-off}} < R_N$ find C_f from Figure 4.3 at the 'cut-off' Reynolds number.

<u>Step 6:</u> Substitute the appropriate value of C_f for C_{f_w} in Eqn.(4.6) or for $C_{f_{fus}}$ in Eqn.(4.30) and proceed with the determination of the desired zero-lift drag coefficients as indicated in Sections 4.2 and 4.3.

<u>4.12.3 Drag Due to Other Causes</u>

Other causes for drag may be items such as: struts, antennas, surface gaps, extra drag caused by inlet air spillage and by exhaust nozzle integration. Strut and antenna drag may be estimated by assuming these to be like small wings. For surface gap drag the reader should consult Ref.8. For estimation of extra inlet drag the method of Sub-section 6.2.4 may be used. For estimation of drag due to exhaust nozzles, see Section 6.3.

Figure 4.77 Effect of Mach Number on the Relation Between Cut-off Reynolds Number and Roughness

4.13 DRAG ADJUSTMENTS FOR LAMINAR FLOW

During the 70's and the 80's it has become increasingly evident that natural laminar boundary layer flow (instead of turbulent boundary layer flow) is practical in many instances. The resulting reduction in friction drag is very significant and should be accounted for in any realistic drag prediction procedure.

The methods presented in Sections 4.3 through 4.12 assume that the boundary layer is turbulent. To adjust the airplane zero-lift drag coefficient downward due to the existence of laminar flow, the following procedure is recommended:

If laminar flow is expected to occur naturally (natural laminar flow, also called NLF) start at Step 1.

If laminar flow is being forced, by suction or by blowing, start at Step 2.

Step 1: Determine which components of the airplane are likely to experience NLF.

In preliminary design the following criteria for the existence of NLF may be used:

$M < 0.65$ $\qquad$ $\Lambda_{LE} < 15$ degrees

$C_{L_{design}} < 0.65$ $\qquad$ $R_N < 10^7$

Note: by careful design it is possible to extend these criteria. Use of advanced airfoil design codes is recommended to verify that NLF is practical.

Step 2: Divide the surface area of all airplane components which according to Step 1 will have a certain amount of NLF in two parts: $S_{wet_{lam}}$ and $S_{wet_{turb}}$.

Figure 4.78 shows how these wetted areas are defined for a wing and for a fuselage respectively.

Step 3: For a wing or for an empennage surface, use Equation (4.6), but replace the term:

$[(C_{f_w})S_{wet_w}/S]$ by: (4.86)

$S_{WET_{W_{LAM}}}$ = WETTED AREA OF THE WING IN THE LAMINAR FLOW REGION

$S_{WET_{FUS_{LAM}}}$ = WETTED AREA OF THE FUSELAGE IN THE LAMINAR FLOW REGION

Figure 4.78 Definition of Laminar and Turbulent Reference Lengths and Reference Areas

$$\{(C_{f_{w_{lam}}} - C_{f_{w_{tur}}})S_{wet_{w_{lam}}} + (C_{f_w})S_{wet_w}\}/S$$

For a fuselage or for a body similar to a fuselage, use Equation (4.30), but replace the term:

$$\{(C_{f_{fus}})S_{wet_{fus}}/S\} \text{ by:} \qquad (4.87)$$

$$\{(C_{f_{fus_{lam}}} - C_{f_{fus_{tur}}})S_{wet_{fus_{lam}}} + (C_{f_{fus}})S_{wet_{fus}}\}/S$$

where: C_{f_w} is the turbulent wing skin friction coefficient as found on p.23 of this text.

$$C_{f_{w_{lam}}} = 1.33/R_{N_{w_{lam}}}^{1/2}, \qquad (4.88)$$

with: $$R_{N_{w_{lam}}} = \rho U_1 c_{w_{lam}}/\mu \qquad (4.89)$$

$c_{w_{lam}}$ is the wing reference length of the laminar part of the wing: see Figure 4.78.

$S_{wet_{w_{lam}}}$ = the wing wetted area part exposed to laminar flow, see Figure 4.78.

$C_{f_{fus}}$ is the turbulent fuselage skin friction coefficient as found on p.44 of this text.

$$C_{f_{fus_{lam}}} = 1.33/(R_{N_{fus_{lam}}})^{1/2}, \qquad (4.90)$$

with: $$R_{N_{fus_{lam}}} = \rho U_1 l_{f_{lam}}/\mu \qquad (4.91)$$

$l_{f_{lam}}$ is the fuselage reference length of the laminar part of the fuselage: see Figure 4.78.

$S_{wet_{fus_{lam}}}$ = the fuselage wetted area part exposed to laminar flow, see Figure 4.78.

NOTES: 1. This procedure applies in the subsonic speed range only!

2. This procedure applies only as long as any surface irregularities (steps, waviness and roughness) are within the limits defined in Reference 23.

By using laminar flow control (sucking and/or blowing) it is possible to achieve laminar flow under conditions where natural laminar flow cannot be maintained. References 24 - 26 provide some data on the types of systems needed to achieve controlled laminar flow and on a number of operational considerations.

Once such systems are in place, the procedure given above for the adjustment of friction drag would apply except for the need to account for compressibility on the equations for skin friction coefficient. Reference 27 should be consulted for replacement of Eqns (4.88) and (4.90) by equations which account for compressibility effects in the transonic speed range below M=1.0

This text does not provide a method for accounting for laminar flow in the transonic speed range above M=1.0 nor for the supersonic flow regimes. This does not mean that laminar flow in these speed ranges is not feasible. Research in progress at NASA Langley indicates that in particular in the supersonic flow range extensive laminar flow may be possible. The conditions for which this can be achieved have not yet been firmly established.

References 28-30 provide some data on the design of fuselage shapes which are conducive to NLF. Design details such as inspection covers, doors and windshields must receive very careful attention, if laminar flow capability is to be retained after the airplane has been in service for some time!

5. AIRPLANE DRAG DATA

The purpose of this chapter is to present a range of actual airplane drag data. These data are given in the form of:

5.1 Drag polars
5.2 Equivalent parasite areas
5.3 Oswald's efficiency factors
5.4 Wetted area breakdown examples

Finally, once drag data for a new airplane have been computed, they should always be 'verified' by comparison against known drag data for similar airplanes. A method for verifying drag polar predictions is given in:

5.5 Verification of realism of computed drag polars

5.1 DRAG POLARS

Figures 5.1 through 5.19 present examples of actual airplane drag polars. The information is organized in the following manner:

Figure 5.1 Cessna 177: includes flap drag
Figure 5.2 Cessna 310: includes gear and flap drag
Figure 5.3 Gulfstream I: includes gear drag, flap drag and drag due to a feathered engine
Figure 5.4 SAAB 340
Figure 5.5 Fokker F-27: includes gear and flap drag
Figure 5.6 Lockheed C-130H: includes ground effect and compressibility data
Figure 5.7 SIAI-M S-211: includes gear and flap drag
Figure 5.8 NAA T2C: includes compressibility data
Figure 5.9 Convair F-106: includes supersonic drag
Figure 5.10 McDD AV8B: includes compressibility data
Figure 5.11 Learjet M25: includes compressibility data
Figure 5.12 Boeing 727-100: includes compressibility data
Figure 5.13 Boeing 707-320B: incl. compressibility data
Figure 5.14 Boeing 747-200: includes compressibility data
Figure 5.15 Boeing B-47B: includes compressibility data
Figure 5.16 Boeing B-52A: includes compressibility data
Figure 5.17 Lockheed C-141B: incl. compressibility data
Figure 5.18 Lockheed C-5A: includes compressibility data
Figure 5.19 Boeing SST Design: includes supersonic data

The reader should note that the wing (or reference) area, S, upon which the drag and lift coefficient data are based, is indicated on all Figures 5.1 through 5.19.

Figure 5.1 Drag Polars: Cessna 177

Figure 5.2 Drag Polars: Cessna 310

Figure 5.3 Drag Polars: Gulfstream I

Figure 5.4 Drag Polar: SAAB 340

Figure 5.5 Drag Polars: Fokker F-27

Figure 5.6 Drag Polars: Lockheed C-130B

Figure 5.7 Drag Polars: SIAI-Marchetti S-211

Figure 5.8 Drag Polars: NAA Rockwell T2C

Figure 5.9 Drag Polars: Convair F-106

Figure 5.10 Drag Polars: McDonnell Douglas AV8B

Figure 5.11 Drag Polars: Learjet M25

Figure 5.12 Drag Polars: Boeing 727-100

Figure 5.13 Drag Polars: Boeing 707-320B

Figure 5.14 Drag Polars: Boeing 747-200

Figure 5.15 Drag Polars: Boeing B-47B

Figure 5.16 Drag Polars: Boeing B-52A

Figure 5.17 Drag Polars: Lockheed C-141B

Figure 5.18 Drag Polars: Lockheed C-5A

Figure 5.19 Drag Polars: Boeing SST Design

5.2 EQUIVALENT PARASITE AREAS

Figures 5.20 through 5.22 provide data on the relationship between total airplane equivalent parasite area, 'f', the equivalent skin friction coefficient, C_f and the total airplane wetted area, S_{wet}. The zero-lift airplane drag coefficient C_{D_o} is related to the equivalent parasite area, 'f' and to airplane wing area, S by:

$$C_{D_o} = f/S \qquad (5.1)$$

The airplanes for which data are included in Figures 5.20-5.22 are described in some detail in various issues of Reference 31. Wing areas, S, are also given in Reference 31.

5.3 OSWALD'S EFFICIENCY FACTORS

Table 5.1 provides data for Oswald's efficiency factor 'e' in the simplified airplane drag polar equation:

$$C_D = C_{D_o} + (C_L^2)/\pi Ae \qquad (5.2)$$

Note that high values of 'e' are rare, but they do occur. The reader may 'reconstruct' a value for 'e' from any drag polar by matching the polar to Equation 5.2.

5.4 EXAMPLES OF WETTED AREA BREAKDOWNS

Tables 5.2 and 5.3 present example data for wetted area breakdowns of fighters, commuters, transports and a business jet. Table 5.3 also includes data on the breakdown of equivalent parasite area with the corresponding value of average skin friction coefficient.

The reader should always verify any computed wetted area breakdown with known breakdowns for similar airplanes. Any significant differences should be explained!

The reader is also reminded of the correlations between total airplane wetted area and airplane take-off weight, provided in Chapter 3 of Part I. These data should be regarded as a source of comparative information.

Figure 5.20 Relation Between Equivalent Parasite Area, Equivalent Skin Friction and Wetted Area for Single Engine Propeller Driven Airplanes

Figure 5.21 Relation Between Equivalent Parasite Area, Equivalent Skin Friction and Wetted Area for Multi-engine Propeller Driven Airplanes

Figure 5.22 Relation between Equivalent Parasite Area, Equivalent Skin Friction and Wetted Area for Jets: Fighters, Bombers and Transports

Table 5.1 Oswald's Efficiency Factor 'e' for Several Airplanes

Single Engine Propeller

	'e'
PT-18	0.75
AT-7	0.74
AT-8	0.61
Cessna L-5	1.02
O-46A	0.80
OE-2	0.70
Cessna 180	0.75
Beech 35	0.82

	'e'
Cessna 150	0.77
Cessna 172	0.77
Cessna 182	0.84
Cessna 185	0.86
Cessna 177	0.57

Four Engine Prop.

	'e'
C-54A	0.81
C-60A	0.63
C-64	0.97
C-69	0.82
XB-19	0.76
B-24D	0.78
B-24G	0.84
B-29A.B	0.94

Prop. Fighters

	'e'
P-38J	0.76
P-40F	0.70
P-47D	1.02
P-49	0.80
P-51B	0.86
P-51F	0.80
XP-60C	0.66
P-61A	0.86

Jet Fighter/Trainers

	'e'
P-63A	0.86
T-37	0.78
NAA T2C	0.816

Jet Bombers

	'e'
Boeing B-47B	0.84
Boeing B-52A	0.924

Twin Engine Prop.

	'e'
A-26B	0.79
C-46A	0.88
C-47B	0.89
B-25D	0.78
B-26F	0.76
Cessna 310	0.73
Gulfstream GI	0.78
SAAB SF 340	0.80

Jet Transports

	'e'
Learjet M 25	0.721
Gulfstr.GII	0.950
F-28-2000	0.818
B 707-320B	0.983
L C-141B	1.067
L C-5A	1.091

Table 5.2 Wetted Area Breakdowns for Fighters, Commuters and Transports

Type:	AV-8B	F-4E	F-15C	F/A-18A	SDS330	SC7	F-27
Wing area, S, ft^2	230	530	608	400	453	373	754
Wing aspect ratio, A	4.0	2.82	3.0	3.5	12.3	11.0	12.0
Wetted Areas (ft^2:							
Wing	379	730	691	562	788	477	1,507
Wing strake/LEX				210			
Fuselage	597	1,124	1,468	890	1,543	1,034	1,614
Horizontal tail	86	154	216	176	98	98	344
Vertical tail	56	116	257	208	173	155	387
Nacelles					163	115	in fus.
Ventral	11						
Miscellaneous	45 (outriggers)				114	81 (wing struts)	
					45	48 (stub wings)	
					126 (gear fairing)		
TOTALS	1,174	2,123	2,632	2,046	3,049	2,008	3,852

Type:	B-757	F-28 1000	F-28 4000	VFW 614	GL M26
Wing area, S, ft^2	1,951	822	850	689	253
Wing aspect ratio, A	7.95	7.3	7.96	7.21	5.74
Wetted areas (ft^2)					
Wing	3,358	1,334	1,399	1,077	402
Fuselage	5,601	2,217	2,453	1,682	520
Horizontal tail	893	430	430	342	108
Vertical tail	762	269	269	202	75
Nacelles	662	323	323	77	140
Pylons	127	75	75	74	25
Flap tracks	225			87	11 (ventral)
Miscellaneous			(antennas) 5		132 (tiptanks)
TOTALS	11,588	4,648	4,950	3,547	1,413

Table 5.3 Wetted Area and Parasite Area Breakdowns for Jet Transports (ft^2)

Type:	B-47B			B-52A			KC-135		
Component:	S_{wet}	c_f	f	S_{wet}	c_f	f	S_{wet}	c_f	f
Fuselage	2,611	.00167	5.552	4,260	.00168	8.489	3,830	.00169	7.976
Wing	2,580	.00229	7.953	7,710	.00224	23.594	4,332	.00224	13.819
Horiz. tail	508	.00243	1.617	1,650	.00232	4.831	875	.00236	2.655
Vert. tail	450	.00229	1.356	930	.00235	2.755	622	.00230	1.842
Trim			.170			1.000			.380
Nacelles	782	.00223	4.510	1,320	.00235	8.437	632	.00231	5.640
Nac. interf.			.541						
Pylons	139	.00232	.376	310	.00234	.843	304	.00227	.803
Aircon. noz.									.650
Roughness			.438			1.029			.658
Miscellaneous							refuel boom:		1.250
TOTALS	7,070		22.513	16,180		50.978	10,595		33.673

Type:	720			707-320B			727-100		
Component	S_{wet}	c_f	f	S_{wet}	c_f	f	S_{wet}	c_f	f
Fuselage	3,971	.00171	8.453	4,590	.00165	9.400	3,838	.00167	8.080
Wing	4,451	.00226	14.693	5,463	.00222	17.246	2,800	.00223	9.490
Horiz. tail	875	.00239	2.685	1,050	.00236	3.186	720	.00233	2.170
Vert. tail	684	.00235	2.066	684	.00234	2.057	685	.00210	1.860
Trim			.380			.550			.200
Nacelles	660	.00234	4.301	368	.00265	2.696	432	.00219	2.270
						center:	130	.00226	.700
Nacelle interf.			.308			.572			.150
Noise suppr.	140	.00311	.435						
Thrust rev.			1.063						
Pylons	304	.00230	.813	304	.00230	.813	108	.00219	.290
Aircond. nozzle			1.300			.350			.250
Roughness			.700			.765			.530
Miscellaneous				scrubbing drag:		4.238	fl.track fair.:		1.000
TOTALS	11,085		37.197	12,459		41.873	8,713		26.990

5.5 VERIFICATION OF REALISM OF COMPUTED DRAG POLARS

After estimating the drag polar of a new airplane with the methods of Chapter 4, the following procedure should always be followed to verify that the computed drag data are indeed 'in-the-ball-park':

Step 1: Determine for which flight condition and for which external configuration the drag polars need to be determined.

The reader should recognize the fact that airplane drag polars depend upon the following factors:

A) Flight condition:

Mach Number, altitude, lift coefficient (or angle of attack) and Reynold's number which also depends on a characteristic length.

B) External configuration:

The external airplane configuration may be influenced by:

1) flap deflection (take-off, climb, cruise, landing or maneuvering
2) landing gear position: up or down (gear well doors open or not?)
3) speed brake position: open or closed
4) external store disposition
5) loading door position: open or closed
6) weapons door position: open or closed
7) cooling flap position: open for take-off, open for climb, or closed
8) propeller control position: feathered, windmilling or normal
9) jet engine condition: windmilling or normal
10) inlet configuration: blow-down doors, ramps, spikes, bleed- and bypass doors
11) control surface deflection for trim (trim drag), with AEO or OEI

Drag polars can differ significantly depending on how these factors are selected for any given airplane in any given flight condition.

Step 2: Plot the C_D versus α data from the calculations performed with Chapter 4.

Step 3: Plot the C_L versus α data for the airplane with the method of Chapter 10.

Step 4: From steps 2 and 3 crossplot the C_D versus C_L polar.

Figure 5.23 shows an example of how Steps 2, 3 and 4 are carried out.

Step 5: A) From step 4, determine C_{D_o} and compute:

$$f = C_{D_o} S \qquad (5.3)$$

B) Determine the wetted area of your airplane. The data for doing this are already available from the drag polar calculations performed in Chapter 4.

'Verify' the computed wetted area breakdown with the data of Tables 5.2 or 5.3. Also check the total wetted area against the trend data of Figures 3.22 in Part I.

C) Plot the 'f' and S_{wet} data on one of the graphs in Figures 5.20-5.22, whichever is applicable. Determine the C_f value for the airplane and judge whether or not the value of C_f is a reasonable one from an 'aerodynamics technology' viewpoint. This is done by comparison to other airplanes in the same figure.

Step 6: Plot $(C_L)^2$ versus C_D, and determine $dC_D/d(C_L)^2$ as indicated in Figure 5.23.

Compute: $e = [\pi A(dC_D/dC_L)^2$ (5.4)

Compare this value of 'e' with that for similar airplanes in Table 5.1. If there is a large difference, check the calculations or explain why the airplane might be different!

Figure 5.23 Drag Polar Construction from Lift and Drag Data with Angle of Attack

AV-8B Harrier II

MCDONNELL DOUGLAS

The United States Marine Corps AV-8B Harrier II V/STOL Light Attack aircraft built by the McDonnell Douglas Corporation is a vastly improved derivative of the British Aerospace AV-8A Harrier currently in service with the USMC. The Marines have, with ten years of operational experience in the AV-8A, proven the effectiveness advantages of V/STOL for the close air support mission. With the AV-8B Harrier II, the Marines will have a V/STOL aircraft with payload/radius capability comparable to any modern, conventional, light attack aircraft.

The AV-8B will provide:

- Double the payload/radius,
- Accurate, first pass weapon delivery,
- Improved VTO and STO capability,
- Greatly reduced pilot workload, and will be
- More reliable and more maintainable

The key to the V/STOL capability of the AV-8B is found in the vectored, 21,180 lb thrust Rolls-Royce Pegasus 11 engine. The pilot can control the direction of engine thrust by positioning the four exhaust nozzles: aft for wingborne flight, down for vertical flight and at intermediate positions for short takeoff and short landing. Aircraft attitude control during V/STOL and hover is accomplished through the reaction control system which directs engine bleed air through reaction nozzles located at the wing tips, the nose and the tail.

Performance Tropical Day (94°F)

- STO distance at max TOGW ..... 1,200 ft
- VTO Weight ..... 19,185 lb
- VL Weight ..... 17,500 lb
- Vmax at altitude ..... 0.91 M
- Vmax at sea level ..... 585 Kts
- Load factor ..... +7 g, −3 g

Interdiction Mission Hi-Lo-Hi

Ferry Ranges

- Clean ..... 1500+ NM
- Two tanks retained ..... 1800+ NM
- Four tanks retained ..... 2000+ NM
- Four tanks dropped ..... 2400+ NM

Weights

- Max design TOGW ..... 29,750 lb
- Operating weight empty ..... 12,750 lb
- Internal fuel ..... 7,500 lb
- Internal and external fuel ..... 15,829 lb
- Max external load ..... 9,200 lb

Major AV-8B Changes from the AV-8A

- Supercritical/composite material wing
- Lift improvement devices
- Large, positive circulation flaps
- Narrowed outrigger mod
- High recovery engine inlets
- Raised, high visibility cockpit
- Stability augmentation and attitude hold system
- Angle rate bombing set
- Inertial navigation set
- Integrated crew station
- Onboard oxygen generation system
- Zero scarf front engine nozzles
- Lengthened aft fuselage

6. INSTALLED POWER AND THRUST PREDICTION METHODS

The purpose of this chapter is to present rapid methods for the prediction of installed power and/or thrust in airplanes. The assumption will be made that the following characteristics of the engine are known:

1. For piston engines: manufacturers shaft horsepower data for a range of altitudes and throttle settings.

2. For gas generators: manufacturers shaft horsepower and thrust data for a range of altitudes, Mach numbers and throttle settings.

3. For jet engines: manufacturers thrust data for a range of altitudes, Mach numbers and throttle settings.

These engine manufacturers data (also called uninstalled data) are based on ideal (teststand) conditions and do <u>not</u> normally include the following effects:

A. Effect of the inlet (air induction system) on pressure recovery, on drag and therefore on engine performance.

   <u>Note:</u> A bellmouth inlet system, assuring very high inlet pressure recovery, is frequently used to determine engine manufacturers data. Fig.6.1 shows a typical teststand arrangement.

B. Effect of power extraction (needed to run essential airplane services) on engine performance.

   <u>Note:</u> Power extraction to run essential engine services ARE normally included in engine manufacturers data.

C. Effect of the exhaust or nozzle configuration on drag and on engine performance.

   <u>Note:</u> The engine manufacturer uses a nozzle configuration which is usually different from that preferred by the airframer. See Figure 6.1.

The methods presented in this chapter are organized as follows:

6.1 Power extraction requirements

6.2 Inlet sizing and integration

6.3 Exhaust or nozzle sizing and integration

6.4 Prediction of installed power and thrust

For a detailed and concise methodology of inlet design and analysis the reader should consult Reference 32. For considerations of design and analysis of inlet and exhaust systems both, Reference 33 is recommended.

Figure 6.1 Engine Test Stand with Bellmouth Inlet

6.1 POWER EXTRACTION REQUIREMENTS

To operate an airplane in any phase of its mission a certain amount of electrical, mechanical and pneumatic power may be required. These power requirements are normally satisfied by the engines. For that reason they are referred to as power extraction requirements. In some cases power sources other than the propulsion system (for example an APU) are used.

The magnitude of power extraction requirements differs from one airplane to another and from one mission phase to another.

In this section a rapid method for estimating power extraction requirements is presented for:

6.1.1 Piston-propeller driven airplanes

6.1.2 Turbopropeller and jet driven airplanes

6.1.1 Piston-propeller Driven Airplanes

In this type of airplane the following power extraction requirements may be present:

$$P_{extr} = P_{el} + P_{mech} \qquad (6.1)$$

where: P_{el} = the electrical power extraction in shp

P_{mech} = the mechanical power extraction in shp

Electrical power extraction requirements, P_{el} follow from the essential electrical services which are required during any given mission phase. To determine electrical power requirements, an 'electric power load profile' must be prepared. Using p.320 of Part IV as an example, the reader should prepare an electric power load profile for his airplane.

From this electric power load profile the electrical power extraction requirement, P_{el} is determined as:

$$P_{el} = .00134(VA_{plp})/\eta_{gen} \text{ in shp} \qquad (6.2)$$

where: VA_{plp} = the maximum required electrical power in Volt-amperes as obtained from the electric power load profile.

η_{gen} = 0.9 is the efficiency of the electric power generator(s) which are assumed to be driven off the engine accessory drive pad. Modern generators can achieve efficiencies of 0.90 to 0.95.

Lacking a detailed electric power load profile, the P_{el} values suggested in Table 6.1 may be used. The reader should also consult Chapter 6 in Part IV for more information on electrical system power capabilities.

<u>Mechanical power extraction requirements</u>. P_{mech} depend on the systems which are required for the operation of the airplane in a given mission phase. Examples of such system may be:

fuel pumps, hydraulic pumps, cooling fans, heating/airconditioning system, pressurization system, spray system (in agricultural airplanes).

The reader should prepare a list of those systems which require the generation of mechanical power. The total mechanical power required can be written as:

$$P_{mech} = P_{fp} + P_{hydr} + P_{other} \tag{6.3}$$

where: P_{fp} = the mechanical power required to drive the fuel pumps. This may be found from:

$$P_{fp} = 0.00014(c_p)(SHP)/\eta_{fp} \tag{6.4}$$

where: c_p is the engine sfc in lbs/shp/hr

SHP is the engine shaft horsepower required in the flight condition being analyzed

η_{fp} = 0.65 is the fuel pump efficiency.

The assumption has been made here that the fuel pumps are operating on a pressure differential of 50 psi. Information on pumps may be found in Ref.34. Data on fuel systems is provided in Chapter 5 of Part IV.

Note: power required to drive electric fuel pumps should be included in P_{el}.

P_{hydr} = the mechanical power required to drive

the hydraulic pumps. The magnitude of P_{hydr} depends on the hydraulic flow and pressure differential needs of the airplane hydraulic system. Section 6.2 of Part IV contains a discussion on the sizing of hydraulic systems. As indicated in Part IV, a hydraulic system load analysis must be performed to find the total hydraulic fluid flow which is required. Having also selected the hydraulic system operating pressure (Part IV, Ch.6), the hydraulic system shaft horsepower requirements are found from:

$$P_{hydr} = 0.0006(\Delta p_{hydr})(\dot{V}_{hydr})/\eta_{hp} \qquad (6.5)$$

where: Δp_{hydr} = the pressure differential (psi), over which the hydraulic system operates. This is roughly equal to the system operating pressure which ranges from 1,500 to 5000 psi. See Chapter 6 in Part IV.

$\dot{V}_{hydr}$ = the hydraulic fluid flow rate in gallons/min (gpm). Page 308 in Part IV provides some guidance for estimating hydraulic fluid flow rates.

η_{hp} = the hydraulic pump(s) operating efficiency. This may be taken to be 0.75 in modern systems.

Note: the power required for electrically driven hydraulic pumps should be included in P_{el}.

P_{other} = the sum of all 'other' mechanical power extraction requirements. Determination of these 'other' mechanical power extraction needs is left to the reader. The best way to proceed is to make a complete list of all 'other' mechanical power requirements for on-board systems. By using appropriate efficiency values, the required value of 'to be extracted' engine shaft horsepower, P_{other} can then be estimated. Ref.34 is a good source for general mechanical system data. Part IV contains discussions of various 'other' types of

system which may be required in airplanes.

Lacking a detailed listing of mechanical power extraction requirements, the values suggested for P_{mech} in Table 6.1 may be used.

The effect of P_{extr} as determined from Eqn. 6.1 on installed engine performance is discussed in Section 6.4.

Table 6.1 Summary of Power Extraction Requirements

Power Extraction Type:	Electrical shp, P_{el}	Mechanical shp, P_{mech}	Pneumatic Bleed slugs/sec
Airplane Type:			
Piston Propeller Driven:			
Single engine, light airplanes	1-2	1-2	0
Single engine, military trainers	2-4	2-4	0
Twin engine, light airplanes	4-6	5-10	0
Multi-engine transports	20-40	30-50	0
Turboprop and Jet Airplanes:			
Single engine, light airplanes	2-4	3-5	$0.01\dot{m}_a$
Single engine, military trainers	5-7	6-10	$0.015\dot{m}_a$
Twin engine turboprops	6-8	7-9	$0.015\dot{m}_a$
Twin engine turbojets or fans	8-10	9-11	$0.025\dot{m}_a$
Twin jet military trainers	10-15	15-20	$0.03\dot{m}_a$
Jet Fighters, air-superiority	50-100	50-100	$0.03\dot{m}_a$
Jet Fighters, attack	100-200	100-200	$0.04\dot{m}_a$
Jet Transports, civil	$0.00070W_{TO}$	$0.00060W_{TO}$	$0.03\dot{m}_a$
Jet Transports, military	$0.00100W_{TO}$	$0.00080W_{TO}$	$0.04\dot{m}_a$

6.1.2 Turbopropeller and Jet Driven Airplanes

For this class of airplanes the power extraction requirements may be determined from:

$$P_{extr} = P_{el} + P_{mech} + P_{pneum} \qquad (6.6)$$

where: P_{el} = the electrical power extraction in shp

P_{mech} = the mechanical power extraction in shp

P_{pneum} = the pneumatic (also called bleed air) power extraction in shp

Electrical power extraction requirements. P_{el} may be determined with the method of Sub-section 6.1.1, via preparation of an electric power load profile as shown on p.320 of Part IV. Lacking such detailed information the P_{el} values suggested in Table 6.1 may be used.

Some airplanes may require extensive radar and electronic warfare equipment. Examples are the Boeing E-3A, the Boeing E-6, the Grumman E-2C and the Lockheed P3V. In those cases large electrical power requirements may exist: 600 kVA in the case of the Boeing E-6!

Normally, these power requirements are satisfied by the installation of directly driven generators on the propulsion installation. If these power requirements cannot be satisfied by power extraction from the regular propulsion system separate power sources may have to be installed.

Mechanical power extraction requirements. P_{mech} may be determined with the method of Sub-section 6.1.1, via a listing of all required mechanical services. Lacking a detailed power extraction calculation, the values suggested in Table 6.1 may be used.

Pneumatic power extraction requirements. P_{pneum} are determined by those systems which are driven by bleed air form the main engines. Typical of such services are:

de-icing and anti-icing systems, heating and airconditioning systems, engine starting systems (for ground start and for air restart), pressurization of fuel tanks, for flap deployment (For example, the B-747 leading edge devices) and for water system pressurization.

A list of airplane services which require bleed air as the source of power must be prepared. Next, the required bleed airflows, $\dot{m}_{bleed}$ (in slugs/sec) must be estimated for each flight situation. From this the total engine bleed airflow can be computed. As a general rule, engine bleed airflow should not exceed 5 percent of the total engine massflow requirement in any given flight condition, $\dot{m}_a$, or major degradation of thrust will occur:

$$\dot{m}_{bleed} < 0.05\dot{m}_a \text{ in slugs/sec} \qquad (6.7)$$

Lacking a detailed evaluation of bleedair requirements, the data of Table 6.1 may be used.

As long as Eqn.(6.7) is satisfied, a first order estimate for P_{pneum} is:

$$P_{pneum} = (\dot{m}_{bleed}/\dot{m}_a)P_{reqd} \text{ for turboprops} \qquad (6.8)$$

and:

$$P_{pneum} = (\dot{m}_{bleed}/\dot{m}_a)(T_{reqd}U_1/550) \text{ for jets} \qquad (6.9)$$

where: P_{reqd} = power required in some flight condition

T_{reqd} = thrust required in some flight condition.

Methods for estimating the power required, P_{reqd} or the thrust required, T_{reqd} are discussed in Part VII.

The effect of P_{extr} as determined form Eqn.(6.6) on installed power or thrust is discussed in Section 6.4.

6.2 INLET SIZING AND INTEGRATION

In preliminary design the process of inlet sizing consists of the determination of the inlet area and the shaping of the duct leading from the inlet area to the engine compressor face.

The inlet must be sized in such a way that it is 'matched' to the airflow requirements of the engine. Figure 6.2 shows a generalized inlet flow situation. Note the following important areas:

A_∞: streamtube cross section at infinity, also called the inlet capture area

A_c: streamtube cross section at the inlet, also called inlet area or cowl capture area

A_f: streamtube cross section at the engine station, also called internal area (note that this cross section is determined by the maximum cross section of the engine plus tolerances for cooling and for installation)

A_e: streamtube cross section at the exit or exhaust, also called the nozzle area

Inlet operation is often characterized by the inlet flow ratio, A_∞/A_c.

During static ground operation, the inlet flow ratio is infinite. Inlet lip flow separation is a major problem in such a case. Figure 6.3a shows such a situation. Frequently an auxiliary inlet is required to allow enough air into the inlet. This is done with a variable inlet geometry feature. Figure 6.3a shows several options.

In the design cruise condition, the inlet is normally matched so that the flow ratio is in the range of 0.5 to 0.8. The inlet operates at its peak performance (high pressure recovery). Figure 6.3b shows this situation.

In some inlets, external surface area is present at points ahead of the inlet: see Figure 6.3c. If external surface area exists ahead of the inlet, the flow ratio is selected to be closer to 1.0 to prevent inlet separation. If no wetted surface area exists ahead of the inlet, the flow ratio is in the 0.5 - 0.8 range.

In a climb, the inlet delivers more air than the engine requires, excess air will be spilled, resulting in

Figure 6.2 Generalized Engine Flow Stations

Figure 6.3 Flow Conditions for Subsonic Inlets

extra drag. The flow ratio is in excess of 1.0. Such a situation is shown in Figure 6.3d. If spillage drag becomes too high, bleedair doors may be required: Fig.6.3d shows one such option.

At very high speeds, such as occur in a high speed dive, the flow ratio is much smaller than 1.0. Such low flow ratios, particularly in the presence of large wetted areas in front of the inlet can lead to inlet flow separation resulting in compressor surge. Figure 6.3e shows such a situation.

If an inlet is undersized (such that it does not deliver enough air to the engine), unsatisfactory engine operation may result, causing deficiencies in thrust and/or power.

The objective of an inlet is to deliver air to the engine such that:

1. The correct amount of airflow is delivered to the engine

2. Pressure losses are minimized: pressure losses reduce thrust

3. Inlet flow distortion is minimized (i.e. as much as possible uniform flow is realized)

4. Inlet flow swirl is minimized or matched to the compressor requirements

Fundamentally, two types of inlets exist:

1. Straight through inlets: see Figure 6.4a
   This inlet type is used for engines with axial flow compressors, one-sided centrifugal flow compressors and supercharged piston engines.

2. Plenum chamber inlets: see Figure 6.4b
   This inlet type is used for engines with two-sided centrifugal flow compressors and with normally aspirated piston engines.

The detailed design of inlets is a strong function of how the engine (s) is (are) being integrated into the airframe.

The following subjects will be discussed:

Figure 6.4a Examples of Straight Through Inlets

6.2.1 General inlet arrangements: presents a discussion of different types of inlet arrangement

6.2.2 Inlet sizing: presents a rapid method for estimating the required inlet area, A_c

6.2.3 Inlet pressure loss estimation: presents a rapid method for estimating inlet pressure losses

6.2.4 Inlet drag estimation: presents a rapid method for estimating inlet drag

Selection of the correct type of inlet and the associated inlet geometry has important consequences to the realism of any proposed airplane design. For that reason, inlet design should receive considerable attention in the early design phases of an airplane.

Figure 6.4b Examples of Plenum Inlets

6.2.1 General Inlet Arrangements

The purpose of this sub-section is to present a number of example inlet arrangements as well as comments regarding their applications.

The information is organized as follows:

6.2.1.1 Piston engine inlets
6.2.1.2 Turbopropeller inlets
6.2.1.3 Jet engine inlets: subsonic
6.2.1.4 Jet engine inlets: supersonic

6.2.1.1 Piston engine inlets

Figures 6.5a and 6.5b show several examples of piston engine inlet arrangements.

Figures 6.5a show that inlets for normally aspirated piston engines are of the plenum type. Most plenum installations do not completely seal off all air: cooling air must be routed to those engine components which need cooling for proper operation. After taking care of the cooling function the air is dumped overboard, preferably such that drag is not increased.

Engines with turbochargers tend to have straight through type inlets as shown in Figures 6.5b.

6.2.1.2 Turbopropeller inlets

Figure 6.6 shows several examples of turbopropeller engine (gas generator) inlets. Those shown are all of the straight through type. An example of a plenum type inlet for a turbopropeller installation was shown in Figure 6.4b.

The pressure recovery of the so-called concentric inlet (Figure 6.6d) can be strongly influenced by the design of the propeller airfoils which are in front of the inlet. Icing of such inlets is a major problem and requires detailed attention in the development of inlet anti- and de-icing systems. See Part IV, Chapter 10 for a general discussion of the icing problem.

6.2.1.3 Jet engine inlets: subsonic

Figures 6.7 show several example inlets for subsonic jet engine installations. Note the following types:

Figure 6.5a Inlets for Normally Aspirated Piston Engines

Figure 6.5b Inlets for Supercharged Piston Engines

Figure 6.6 Inlets for Turbopropeller Engines

Figure 6.7a Subsonic Plenum Inlet

Figure 6.7b Subsonic Bifurcated Inlet

Figure 6.7c Subsonic Podded Nacelle Inlet

Figure 6.7d Subsonic Pitot Inlet

Figure 6.7e Subsonic NACA Submerged Inlet

Figure 6.7a: plenum inlet
Figure 6.7b: bifurcated, straight through inlet
Figure 6.7c: podded nacelle inlet
Figure 6.7d: pitot inlet
Figure 6.7e: NACA submerged inlet

Plenum inlets (Figure 6.7a) are used mainly in combination with double-sided centrifugal flow compressors.

Bifurcated inlets (Figure 6.7b) are used primarily in single engine installations with side inlets. Flow characteristics of bifurcated inlets are complicated, especially at high sideslip angles: inlet buzz and reversed flow on one side are phenomena which need to be 'designed out' of such inlets. Ref.32 contains more information.

Podded nacelle type inlets (Figure 6.7c) have become popular because of easy engine access. When using wing mounted engine pods there is an additional wing weight advantage due to inertial relief.

Pitot type inlets (Figure 6.7d) have been applied to many fighter airplanes. They are not influenced by the flowfield of other airplane components. However, they require very long ducts which causes extra weight and loss in pressure recovery.

The NACA submerged type inlet shown in Figure 6.7e is not very efficient for use with propulsion installations. However, when used for inlets of auxiliary systems (APU, heating and avionics bay cooling) they are quite acceptable and frequently used.

<u>Important Note:</u> Except for pitot and podded nacelle type inlets, all jet engine inlets must be equipped with so-called boundary layer diverters (or b.l. splitters). Figure 6.8 shows two b.l. diverter installations in some detail. If such boundary layer diverters are not used, large pressure recovery losses (thus losses in thrust) will be incurred.

A major consideration in jet fighter inlet design is the behavior of the inlet at very high angles of attack and sideslip. Compressor stall and engine surging are easily induced in such conditions. Ref.32 should be consulted for data on high angle of attack operation.

In subsonic installations it is usually best to keep the inlet as short as possible: long ducts translate into weight and pressure recovery losses. In jet fighters and in jet trainers long ducts cannot always be avoided.

SEE REF. 32 FOR DETAILS ON B.L. SPLITTERS (DIVERTERS)

Figure 6.8 Boundary Layer Diverter Examples

6.2.1.4 Jet engine inlets: supersonic

Figures 6.9 show various types of supersonic inlets. Note the boundary layer splitters!

Figure 6.10 shows three fundamentally different types of supersonic inlet:

Fig.6.10a Pitot inlet
Fig.6.10b External compression inlet
Fig.6.10c Mixed (or external/internal) compression inlet

Pressure recovery in supersonic inlets is a strong function of the number and types of shock employed. The theoretical pressure recovery attainable with oblique and with conical shocks are shown in Figure 6.11.

Proper inlet design is extremely critical to supersonic installations as illustrated in Figure 6.12. Note that the inlet is responsible for 75 percent of the total installed thrust! A long inlet duct in supersonic engine installations is often needed to assure smooth flow deceleration (to around M=0.4 at the compressor face) and to assure full use of the favorable pressure distribution in the inlet duct. Figure 6.13 shows the effect of Mach number on thrust distribution. Note that at subsonic speeds, the engine itself (including a convergent nozzle) produces virtually all the thrust. Note also, that at supersonic Mach numbers the engine contribution itself can become negative!

Supersonic inlets frequently require a considerable amount of variable geometry devices. Examples are shown in Figures 6.13. For more information on the operation of such inlets, see References 12 and 32.

Figure 6.9 Supersonic Inlet Examples

Figure 6.10a Supersonic Pitot Inlet

COPIED FROM REF. 12, COURTESY: L. NICOLAI

Figure 6.10b Supersonic External Compression Inlet

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Figure 6.10c Supersonic Mixed Compression Inlet

COPIED FROM REF. 12, COURTESY: L. NICOLAI

Figure 6.11 Ideal Pressure Recoveries for Oblique and for Conical Shocks

Figure 6.12 Example of the Thrust Distribution over a Supersonic Engine Installation at M=2.2

Figure 6.13 Effect of Mach Number on Thrust Distribution of a Supersonic Engine Installation

Mach 2.3 Two-Dimensional External Compression Inlet

Supersonic Mixed Compression Axisymmetric Inlet

Figure 6.14 Examples of Variable Geometry in Supersonic Engine Installations

6.2.2 Inlet Sizing

The purpose of this sub-section is to present inlet sizing methods for:

6.2.2.1 Piston engine installations
6.2.2.2 Turbopropeller installations
6.2.2.3 Jet engine installations: subsonic
6.2.2.4 Jet engine installations: supersonic

For each of these inlet installations, the general flow picture of Figure 6.2 will be used to identify the required inlet area, A_c.

6.2.2.1 Piston engine installations

Figure 6.5 shows typical inlet configurations for piston engine installations. To determine the required inlet area, A_c it is first required that the engine air flow requirements be determined.

Inlet air for piston engines is required for:

1. Combustion 2. Cooling

The mass flow rate required for a piston engine may be estimated as follows:

$$\dot{m}_a = \dot{m}_{comb} + \dot{m}_{cool} \quad \text{(in slugs/sec)} \qquad (6.10)$$

where: $\dot{m}_{comb}$ = the mass flow rate required for combustion. It may be estimated from:

$$\dot{m}_{comb} = (0.000062)(SHP)_{reqd} \qquad (6.11)$$

$$\text{where: } (SHP)_{reqd} = P_{reqd}/\eta_p \qquad (6.12)$$

where: P_{reqd} is the horsepower required for the flight condition being analyzed. Part VII contains methods for determining P_{reqd} for a variety of flight conditions.

η_p is the propeller efficiency.

Section 6.4 and Ref.15 show how to obtain data for

propeller efficiency. In preliminary design, first approximations are:

η_p = 0.85 for cruise
0.80 for climb
0.70 for take-off

In Equation (6.11) it is assumed that:

1. that the engine efficiency in converting chemical energy in the fuel to shafthorsepower (SHP) is 30 percent.
2. that the fuel/air ratio used is the stochiometrically required value.

$\dot{m}_{cool}$ = the mass flow rate required for cooling.

Its magnitude depends on the type of engine:

<u>For aircooled</u> airplane engines the mass flow rate required for engine cooling may be estimated from:

$$\dot{m}_{cool} = 0.00056(SHP)_{reqd} \quad \text{in slugs/sec} \qquad (6.13)$$

where: $(SHP)_{reqd}$ follows from Eqn.(6.12).

In preliminary design, SHP_{TO} may be used for $(SHP)_{reqd}$.

Detailed methods for estimating $\dot{m}_{cool}$ for aircooled engines are given in References 35 and 36.

<u>For liquid cooled</u> airplane engines the required mass flow rate for the radiator depends strongly on the type of liquid cooling used. Chapter 9 in Reference 8 and Reference 37 should be consulted.

For preliminary design purposes it is suggested to use Eqn.(6.10) for liquid cooled engines also.

Considerable experience with liquid cooled engine installations was obtained in WWII. In many cases it proved possible to design the radiator system in such a way that negligible losses were incurred. However, this was achieved at a considerable increase in weight and complexity.

Knowing the total air mass flow rate, the size of the required inlet area may be estimated from:

$$A_c = \dot{m}_a/\rho U_1 \qquad (6.14)$$

where: $\dot{m}_a$ follows from Eqn.(6.10),

ρ is the air density in slugs/ft^3 and

U_1 is the steady state airspeed in fps.

<u>Note:</u> Eqn.(6.14) is valid for incompressible flow!

The reader must determine which conditions for air density, ρ and airspeed, U_1, yield the largest value for inlet area, A_c. Since zero airspeed represents an anomalie, lift-off speed or climbout speeds are normally used. Extra inlet doors (or cowl flaps) may be required for prolonged static and taxi operations.

To minimize the drag caused by momentum loss in the cooling air, careful design of the ducting leading from the inlet to the overboard dumping point is required. References 8 and 38 should be consulted for more details. Reference 39 contains example calculations for inlet area sizing for piston engines.

<u>6.2.2.2 Turbopropeller installations</u>

Figure 6.6 shows typical inlet configurations for turbopropeller engines.

To determine the required inlet area, A_c it is first necessary to determine the engine air flow requirements.

Inlet air for turbopropeller engines (also called: gas generators) is required for:

1. combustion and mass flow 2. cooling

The mass flow rate for turbopropeller engines may be estimated from:

$$\dot{m}_a = \dot{m}_{gas} + \dot{m}_{cool} \qquad (6.15)$$

where: $\dot{m}_{gas}$ = the air flow rate required by the engine for combustion and for mass flow. In a gas generator the mass flow rate is much larger than the air flow required for combustion. Manufacturers engine data normally include the maximum mass flow rate. Reference 31 also lists required mass flow rates for gas generators. Tables 6.6 and 6.7 in Part III list take-off mass flow rates in lbs/sec for a range of turbopropeller engines.

In preliminary design, a good approximation is:

$$\dot{m}_{gas} = (0.00028)(SHP)_{reqd} \quad \text{(in slugs/sec)} \qquad (6.16)$$

where: $(SHP)_{reqd}$ is given by Eqn.(6.12).

$\dot{m}_{cool}$ = the mass flow rate needed for cooling.

For gasgenerators this may be taken as 5 percent of $\dot{m}_{gas}$ for most turboprop installations:

$$\dot{m}_{cool} = 0.05\dot{m}_{gas} \qquad (6.17)$$

The required inlet area may be estimated from:

$$A_c = \dot{m}_a/\rho U_1 \qquad (6.18)$$

where: $\dot{m}_a$ is found from Eqn.(6.15).

Note: Eqn.(6.18) is valid for incompressible flow!

The reader must determine which conditions for air density, ρ and airspeed, U_1 yield the largest value for inlet area A_c.

Since zero airspeed represents an anomalie, lift-off speed or climbout speeds are normally used. Extra inlet doors (variable inlet geometry) may be needed for prolonged static and taxi operations.

6.2.2.3 Jet engine instr'lations: subsonic

Figure 6.7 shows examples of subsonic jet engine inlets.

To determine the required inlet area, A_c it is first

necessary to determine the engine airflow requirements.

Inlet air for jet engines is required for:

1. combustion and mass flow 2. cooling

The mass flow rate required for a subsonic jet engine installation may be found from:

$$\dot{m}_a = \dot{m}_{gas} + \dot{m}_{cool} \qquad (6.19)$$

where: $\dot{m}_{gas}$ = the air flow rate required for the engine.

In jet engines the flow rate required for combustion is much less than that needed for mass flow. Manufacturers engine data normally include the maximum mass flow $\dot{m}_{gas}$ required by the engine.

Reference 31 also lists those flow rates. Tables 6.8 - 6.11 in Part III list take-off mass flows for a range of jet engines. In preliminary design the following approximation may be used:

$$\dot{m}_{gas} = k_{gas} T_{TO} \qquad (6.20)$$

where: k_{gas} = 0.0003 for BPR values of 0 to 1.0

= 0.0007 for BPR values of 1.0 to 2.0

= 0.0009 for BPR values of 2.0 to 4.0

= 0.0011 for BPR values of 4.0 to 6.0

$\dot{m}_{cool}$ = the air flow rate needed for cooling.

In preliminary design it may be assumed that:

$$\dot{m}_{cool} = 0.06\dot{m}_{gas} \qquad (6.21)$$

The required inlet area, A_c may be estimated from:

$$A_c = \dot{m}_a/\rho U_1 \qquad (6.22)$$

where: $\dot{m}_a$ is found from Eqn.(6.19)

ρ and U_1 are determined by the flight condition.

<u>Note:</u> Eqn.(6.22) is valid for compressible flow.

The inlet size for most subsonic airplanes is dictated by low speed requirements. For that reason, Eqn.(6.22) is applicable to preliminary inlet sizing for subsonic jets.

6.2.2.4 Jet engine installations: supersonic

Figures 6.9 and 6.10 show examples of supersonic jet engine inlet installations.

Inlet air for jet engines in supersonic installations is required for:

1. combustion and mass flow
2. cooling
3. inlet boundary layer bleed

The mass flow rate required for a jet engine in a supersonic installation may be estimated from:

$$\dot{m}_a = \dot{m}_{gas} + \dot{m}_{cool} + \dot{m}_{blb} \qquad (6.23)$$

where: $\dot{m}_{gas}$ = the air flow rate required for the engine.

In jet engines the air flow rate required for combustion is much less than that for mass flow. Manufacturers engine data normally include the maximum mass flow, $\dot{m}_{gas}$ required by the engine.

Reference 31 also lists those flow rates. Tables 6.8 - 6.11 list mass flow rate values for a range of engines. Since most supersonic inlets must also operate at subsonic speeds, the take-off mass flow rates normally size the inlet area. For this reason, Eqn.(6.20) may be used for computing $\dot{m}_{gas}$.

$\dot{m}_{cool}$ = the air flow rate needed for cooling.

In preliminary design it is acceptable to use:

$$\dot{m}_{cool} = 0.08\dot{m}_{gas} \qquad (6.24)$$

$\dot{m}_{blb}$ = the air flow rate needed for boundary layer bleed in the inlet. Because of the fact that supersonic installations require fairly long inlets, the resulting build-up of boundary layer air must be bled away before it reaches the compressor. Figure 6.14 shows such a boundary layer bleed system. The need for boundary layer bleed

translates into a need for extra inlet capture area as shown in Eqn.(6.25).

The inlet area, A_c for a supersonic inlet is as defined in Figure 6.15. The required inlet area, A_c for a supersonic jet engine installation may be estimated from:

$$A_c = (1.08\dot{m}_{gas}/\rho U_1)(1 + k_{bl}(M_1 - 0.8)) \qquad (6.25)$$

where: $\dot{m}_{gas}$ follows from Eqn.(6.20)

ρ and U_1 are determined by the flight condition which in the case of a supersonic installation is taken to be the design supersonic cruise condition or the take-off condition. If the latter results in a larger inlet area than the former, the possibility of using extra inlet doors for subsonic operation must be weighed against 'oversizing' the inlet for supersonic conditions.

$$M_1 = U_1/a \qquad (6.26)$$

where: a is the speed of sound in the design cruise flight condition

k_{bl} is a constant which depends on the type of inlet used:

$k_{bl} = 0$ for pitot inlets and for $M_1 < 0.8$

$k_{bl} = 0.028$ for external compression inlets

$k_{bl} = 0.041$ for mixed external/internal compression inlets

The inlet throat area, A_{throat} in Figure 6.15 must be carefully sized to assure that it can handle the required engine mass flow rate. References 12 and 32 contain methods for determining the throat size.

Figure 6.15 Geometry Definition for a Supersonic Inlet

Figure 6.16 Definition of Flow Parameters for Figure 2

Figure 6.17 Pressure Recovery for Turbopropeller Inlets

6.2.3 Inlet Pressure Loss Estimation

The generalized flow situation shown in Figure 6.2 is repeated in Figure 6.16 with the addition of speed, U, Mach number, M and static pressure, p at each station.

The total pressure at each station is defined as:

$$P_{tot} = p + 0.5\rho U^2 = p + \bar{q} \qquad (6.27)$$

where: p is the local static pressure,

ρ is the local air density and

U is the local air velocity.

The inlet pressure loss is defined as:

$$\Delta p_{inl} = P_{tot_\infty} - P_{tot_f} \qquad (6.28)$$

Inlet pressure loss is frequently compared to free stream dynamic pressure or to free stream total pressure, in which cases it is referred to as the inlet efficiency or inlet pressure recovery, η_{inl}:

For incompressible flow:

$$\eta_{inl/inc} = (P_{tot_f} - p_\infty)/\bar{q}_\infty \qquad (6.29)$$

which can be written as:

$$\eta_{inl/inc} = 1 - \Delta p_{inl}/\bar{q}_\infty \qquad (6.30)$$

For compressible flow:

$$\eta_{inl/com} = P_{tot_f}/P_{tot_\infty} \qquad (6.31)$$

which can be written as:

$$\eta_{inl/com} = 1 - \Delta p_{inl}/P_{tot_\infty} \qquad (6.32)$$

The purpose of this sub-section is to present rapid methods for the calculation of inlet pressure losses in terms of either Δp_{inl} or η_{inl}.

Inlet pressure losses in turn have a detrimental effect on installed engine power or thrust. Methods for determining the effect of inlet pressure losses on installed power or thrust are presented in Section 6.4.

The information is organized as follows:

6.2.3.1 Piston engine inlets
6.2.3.2 Turbopropeller inlets
6.2.3.3 Jet engine inlets: subsonic
6.2.3.4 Jet engine inlets: supersonic

6.2.3.1 Piston engine inlets

For plenum inlets:

Figure 6.5a shows plenum inlets for piston engines.

In well designed piston engine plenum installations, the inlet losses may be held to less than 2 percent. For preliminary design purposes it is acceptable to use:

$$\eta_{inl/inc} = 0.98 \qquad (6.33)$$

For 'straight through' inlets:

Figure 6.5b shows 'straight through' inlets for piston engine applications. The method of 6.2.3.3 for subsonic jet inlets may be used to estimate pressure losses.

6.2.3.2 Turbopropeller inlets

For plenum inlets:

For turbopropellers with plenum inlet installations (See Figure 6.6), the pressure losses may be estimated with the plenum inlet method of 6.2.3.3.

For 'straight through' inlets:

As shown in Figure 6.6, many turbopropeller installations utilize 'straight through' inlets. For such inlets the pressure recovery may be estimated from:

$$\eta_{inl/inc} = f(\mu_{inl}) \qquad (6.34)$$

where: $f(\mu_{inl})$ is established in Figure 6.17 for inlets with unducted spinners as well as ducted spinners.

$$\mu_{inl} = A_c/A_\infty \qquad (6.35)$$

Note that μ_{inl} is the inverse of the inlet flow ratio as defined on page 147.

6.2.3.3 Jet engine inlets: subsonic

For plenum inlets:

Figure 6.7a shows a plenum inlet for a subsonic jet. Pressure losses may be estimated from:

$$\Delta p_{inl}/\bar{q}_\infty = [1/(\mu_{inl}^2)](A_c/A_d)^2(1 - A_d/A_f)^2 \quad (6.36)$$

where: A_c, A_d and A_f for a plenum installation are as defined in Figure 6.18. Ref. 32 should be consulted for more details on jet engine plenum inlets.

For straight through inlets:

Figures 6.7b-d show subsonic, 'straight through' jet inlet examples. The pressure loss in such inlets is a strong function of the absence or presence of boundary layer diverters. Figure 6.8 shows examples of boundary layer diverters.

Without a boundary layer diverter, the pressure loss of a subsonic jet engine inlet may be estimated from:

$$\Delta p_{tot}/\bar{q}_\infty = IC_{Fd}/(\mu_{inl})^2 + BC_{Fa}\mu_{inl} \quad (6.37)$$

where: I is the so-called duct integral, defined as:

$$I = \int_{l_c}^{l_f}(A_c/A)^2(per_A/A)dl \quad (6.38)$$

where: A is the duct area at station l,

per_A is the duct perimeter at station l

l_c is the duct station at the inlet area, A_c: see Figure 6.19

l_f is the duct station at the engine compressor face: see Figure 6.19.

$$C_{Fd} = fC_f \quad (6.39)$$

where: the factor f is given in Figure 6.20

C_f is the equivalent flat plate friction coefficient based on the duct Reynold's Number with the characteristic length taken as $(l_f - l_c)$. Figure 4.3 may

Figure 6.18 Geometry Definition for a Plenum Inlet

Figure 6.19 Duct Geometry with External Wetted Area

Figure 6.20 Friction Coefficient Factor for Inlet Ducts

be used to find C_f.

μ_{inl} is defined in Eqn.(6.35)

H is the corrected position ratio, defined as:

$$H = (0.8S_{wet\ appr})/A_c \qquad (6.40)$$

where: $S_{wet\ appr}$ is the wetted area of the inlet 'approach' as indicated by the cross-hatched area in Figure 6.21. Observe that for podded nacelle inlets and for pitot inlets: $S_{wet\ appr} = 0$!

C_{Fa} is the overall approach friction coefficient.

It may be set equal to the flat plate friction coefficient corresponding to a Reynolds Number based on l_{appr} as the characteristic length.

Figure 6.21 shows how l_{appr} is defined. Note that $l_{appr} = 0$ for podded nacelle inlets and for pitot inlets!

With a boundary layer diverter, the pressure loss of a subsonic jet engine inlet may be estimated from Eqn.(6.37) by setting H = 0.

The reader should consult Reference 32 for further details.

6.1.3.4 Jet engine inlets: supersonic

Figures 6.9, 6.10 and 6.14 give examples of supersonic inlet configurations. Note that all are of the so-called 'straight through' type. The pressure recovery of a supersonic inlet system is defined as:

$$\eta_{inl/com} = (\eta_{shock})(\eta_{diff}) \qquad (6.41)$$

where: η_{shock} is the pressure recovery through the inlet shock system. It is found from Fig.6.11 depending on the type on inlet.

η_{diff} is the pressure recovery through the subsonic diffuser which follows the system of shocks in the inlet. It may be found from:

Figure 6.21 Definition of Approach Length

Figure 6.22a Effect of Diffuser Geometry on Diffuser Loss Coefficient (Shock Ahead of Entrance)

$$\eta_{diff} = 1 - \epsilon_D\{1 - (1 + 0.2M_T^2)^{-3.5}\} \qquad (6.42)$$

where: ϵ_D is the diffuser loss coefficient found from Figure 6.22. It depends on the subsonic diffuser geometry!

M_T is the diffuser entry Mach number. This Mach number is equal to the Mach number of the flow through the last shock in the inlet shock system. It can range from 0.4 to 0.9 in practical applications. Usually a number of 0.6 to 0.7 is found in such inlets. In preliminary design it is suggested to 'pick' a suitable number in this range.

M_T must be computed from the system of shocks. A method for doing this may be found in Appendix D of Ref.12.

Methods for determining the effect of inlet pressure losses on installed engine power or thrust are presented in Section 6.4.

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SEE FIG. 6.15
FOR L_D AND H_t

Figure 6.22b Effect of Diffuser Geometry on Diffuser Loss Coefficient (Subsonic Entrance)

6.2.4 Inlet Extra Drag Estimation

The purpose of this sub-section is to present rapid methods for estimation of the so-called inlet extra drag, $C_{D_{inlextra}}$, which was considered a part of $C_{D_{misc}}$ in Sub-section 4.12.3. The inlet extra drag considered here is that contribution of inlet drag not accounted for in Ch.4 as a result of wetted areas or as a result of cross sectional area distribution.

Inlet pressure recovery and any associated loss in inlet pressure have been discussed in Sub-section 6.2.3. The effect of these losses on installed power and thrust is discussed in Section 6.4.

The extra inlet drag discussion is organized as in the following manner:

6.2.4.1 Piston engine inlets
6.2.4.2 Turbopropeller inlets
6.2.4.3 Jet engine inlets: subsonic
6.2.4.4 Jet engine inlets: supersonic

6.2.4.1 Piston engine inlet extra drag

In well designed piston engine installations, the inlet extra drag should be negligible. If the inlet is undersized, spillage drag may result. Estimation of inlet spillage drag is discussed in 6.2.4.3.

6.2.4.2 Turbopropeller inlet extra drag

In properly designed turbopropeller inlets the inlet extra drag should be negligible. If the inlet is undersized, spillage drag may result. Estimation of inlet spillage drag is discussed in 6.2.4.3.

6.2.4.3 Jet engine inlet extra drag: subsonic

For properly designed subsonic jet inlets the inlet extra drag should be negligible. However, if the inlet is undersized for some flight condition, spillage drag may result. The inlet extra drag due to spillage may be estimated from:

$$C_{D_{inlextra}} = C_f[1 + 0.33\{(d_m - \bar{d}_c)/l_{mc}\}F_{inl}]^{1.667} \quad (6.43)$$

with:

$$F_{inl} = [1 + 1.75(\mu_{inl} - 1)/\mu_{inl}\{(A_m/\bar{A}_c) - 1\}]\bar{A}_c/S \quad (6.44)$$

where: C_f is the equivalent flat plate friction coefficient at a Reynold's Number based on a characteristic length equal to l_{mc}: see Figure 6.23.

d_m is the maximum cowl diameter for the inlet

$\bar{d}_c$ is the cowl diameter at the inlet area position

A_m is the cowl cross section area at d_m

$\bar{A}_c$ is the cowl cross section area at $\bar{d}_c$

μ_{inl} is defined in Eqn.(6.35).

Equation (6.43) applies up to the critical Mach number. To reduce inlet drag above M_{crit}, it is necessary to shape the inlet lips so that they in fact have properties similar to high speed airfoils. Reference 32 contains discussions on inlet lip design at high subsonic Mach numbers.

6.2.4.4 Jet engine inlet extra drag: supersonic

In the supersonic case there are three sources for extra inlet drag:

1. Spillage drag (called additive drag)
2. Bypass drag
3. Boundary layer diverter drag
4. Boundary layer bleed drag

Figure 6.24 shows where these drag types are caused in a supersonic inlet. All three drag types depend on the state of the shock systems in the inlet. Detailed presentations of methods for estimating these drag contributions are beyond the scope of this text. Ref.12 contains methods for estimating these drag increments. Since Ref.12 uses A_c as the inlet drag reference area, all results must be multiplied by A_c/S.

Figure 6.23 Duct Geometry for Inlet Extra Drag Evaluation

Figure 6.24 Visualization of Extra Drag Types in a Supersonic Inlet

6.3 EXHAUST OR NOZZLE SIZING AND INTEGRATION

In preliminary design the process of exhaust or nozzle sizing and integration consists of:

1. the determination of exhaust and/or nozzle areas

2. the integration of the exhausts and/or nozzles into the airplane

Figure 6.2 defines the nozzle exit area, A_e. Note from Fig.6.77 in Part III that in turbofan engines the nozzle area consists of two concentric areas, located at different longitudinal stations.

Integration of exhausts and/or nozzles into the airplane configuration must be done with care! Without such care, major increases in drag can be the result.

The information in this section is presented as:

6.3.1 General exhaust/nozzle arrangements

6.3.2 Exhaust/nozzle sizing

6.3.3 Estimation of exhaust/nozzle extra drag

The reader should also refer to Part III, Chapter 6 for examples of engine exhaust/nozzle installations.

6.3.1 General Exhaust/Nozzle Arrangements

The purpose of this sub-section is to present examples of exhaust/nozzle configurations which are being used in a number of airplanes.

The information is organized as follows:

6.3.1.1 Piston engine exhausts
6.3.1.2 Turbopropeller exhausts
6.3.1.3 Jet engine nozzles: subsonic
6.3.1.4 Jet engine nozzles: supersonic

6.3.1.1 Piston engine exhausts

Figure 6.5 shows several examples of piston engine exhaust configurations. In many older installations the engine exhaust is 'dumped' overboard in the manner shown in the 'upper' Figure 6.5a: this is very inefficient and causes extra drag. A lower drag installation is the one

shown in the 'lower' Figure 6.5a. Figure 6.25 shows the difference between a 'low drag' and 'high drag' case.

The reader must keep in mind that lowering drag is not always good. If the lower drag is obtained at the cost of increased weight and complexity, a 'design trade study must provide the information on basis of which the decision is made to proceed one way or the other.

The installations of Figures 6.5b represent modern concepts which take maximum advantage of the momentum exchange between inlet, combustion process, cooling requirements and exhaust.

6.3.1.2 Turbopropeller nozzles

Figure 6.6 shows several examples of turbopropeller engine (gas generator) exhaust installations. Note the trend toward exhausting 'parallel' to the local stream.

In many contemporary installations the exhaust pipes are arranged as shown in Figure 6.26. This creates much extra drag. The 'design trade study' comments made under 6.3.1.1 also apply here!

6.3.1.3 Jet engine nozzles: subsonic

Figure 6.7 shows several example of nozzle installations for subsonic jet engines. Note that all nozzles are of the convergent type. In subsonic flow, convergent nozzles are the only efficient nozzle configuration.

6.3.1.4 Jet engine nozzles: supersonic

Figure 6.27 shows a typical supersonic exhaust configuration. Depending on the flight Mach number of the airplane the nozzle should have a different geometry. Figure 6.28 illustrates the effect of flight condition on desired nozzle configuration. Note the convergent/divergent shape of nozzles at supersonic speeds.

The integration of nozzles into the after body of a supersonic airplane is particularly critical to drag. Figure 6.29 shows a number of nozzle/airframe integration concepts with commentary about the effect on drag. Here also, the consequence of weight, cost and complexity must be weighed against aerodynamic efficiency.

6.3.2 Exhaust/Nozzle Sizing

The purpose of this sub-section is to present rapid

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Figure 6.25 Example of High Drag and Lower Drag Exhaust Stack Installation

Figure 6.26 Example of a High Drag Exhaust Installation as Seen on Contemporary Turboprops

Figure 6.27 Examples of Supersonic Exhaust Configurations

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SEE FIG. 6.30 FOR A_e AND A_t DEFINITION

Figure 6.28 Variation of Nozzle Geometry with Mach Number

Figure 6.29 Examples of Nozzle/Airframe Integration

Figure 6.30 Definition of Nozzle Exit and Throat Areas

Figure 6.31 Effect of Exhaust Stack Configuration on Drag

methods for the 'sizing' of exhausts and/or nozzle exit areas. Figure 6.30 shows the definition of important nozzle exit and throat areas.

The material in this sub-section is presented as:

6.3.2.1 Piston engine exhausts
6.3.2.2 Turbopropeller nozzles
6.3.2.3 Jet engine nozzles: subsonic
6.3.2.4 Jet engine nozzles: supersonic

6.3.2.1 Piston engine exhausts

The exit size of piston engine exhausts is normally defined by the engine manufacturer. The exit area of the exhaust(s) are dependent on the cross sectional areas of the engine exhaust valve ports. Reference 37 contains detailed descriptions of this aspect of engine design.

The airframer may wish to use the exhaust manifold for purposes of heat extraction (by routing cabin air through a heat exchanger which is wrapped around the exhaust manifold). The airframer may also add noise suppressors and/or ejectors to the exhaust stack(s).

Refs 38 and 40 as well as Section 6.9 in Part III should be consulted for more details on piston engine exhaust configurations.

As a first approximation to the sizing of the total exit area, it is suggested to use an exhaust stack diameter given by:

$$d_{exhst} = (0.0038)SHP_{TO} \text{ in inches} \qquad (6.45)$$

If this diameter becomes too large, it is best to split the exhaust manifold into two or more small stacks.

6.3.2.2 Turbopropeller nozzles

Because turbopropellers are used only in subsonic flight conditions, the nozzles are always of the convergent type as seen in Figure 6.28.

As a first approximation to the sizing of the nozzle exit area, it is suggested to use:

$$d_{exhnoz} = (0.016)SHP_{TO} \text{ in inches} \qquad (6.46)$$

In many turboprop engines the nozzle area is divided over two exhausts. Engine manufacturers specify the ex-

haust area on the basis of their performance guarantees.

6.3.2.3 Jet engine nozzles: subsonic

In subsonic applications the exhaust nozzle is nearly always of the convergent type as shown in Figure 6.28.

The exhaust nozzle size and its configuration depend strongly on the following factors:

1. maximum take-off thrust
2. by-pass ratio
3. need for afterburning
4. design Mach number

The reader should use engine manufacturers data for exhaust nozzle areas. References 12, 33 and 41 provide more details on the subject of nozzle design.

6.3.2.4 Jet engine nozzles: supersonic

The exhaust nozzle size and its configuration depend strongly on the following factors:

1. maximum take-off thrust
2. by-pass ratio
3. need for afterburning
4. design Mach number

The usual nozzle configuration employs variable geometry: convergent for subsonic flight and convergent/divergent for supersonic flight: see Fig.6.28.

Engine manufacturers data should be used to find the size of the subsonic nozzle exit area. Figure 6.28 can be used to estimate the required exit area of the divergent part of the nozzle in its supersonic position.

Sizing of the length of the convergent part of the nozzle is not critical as long as it is not too short: a length of twice the entry diameter should be sufficient.

Sizing of the length of the diverging part of the nozzle is critical if separation is to be prevented. In preliminary design, a length of at least three times the nozzle throat diameter should be adequate.

References 12, 33 and 41 provide more details on the subject of nozzle design.

6.3.3 Estimation of Exhaust/Nozzle Extra Drag

In this sub-section rapid methods for estimating the extra drag caused by exhaust and/or nozzle installations. The material is organized as follows:

6.3.3.1 Piston engines
6.3.3.2 Turbopropeller engines
6.3.3.3 Jet engines: subsonic
6.3.3.4 Jet engines: supersonic

6.3.3.1 Piston engines

Figure 6.31 illustrates the drag increments associated with two piston engine exhaust configurations. In preliminary design it is suggested to use drag increments on the basis of similarity of the proposed installation with those of Figure 6.31.

6.3.3.2 Turbopropeller engines

If the exhaust configuration of a turbopropeller is similar to the one shown in Figure 6.26, the drag increments of 6.3.3.1 may be used as a guide.

If the exhaust configuration is similar to those of Figure 6.6, the additional drag penalty caused by the exhaust may be negligible.

6.3.3.3 Jet engines: subsonic

For engines mounted in nacelle pods, no additional drag increment is incurred: the nacelle drag is accounted for separately.

For engines mounted in a fuselage and exhausting in the rear, extra drag may be incurred. Figure 6.32 shows several nozzle concepts used for subsonic applications. Each concept has a drag penalty associated with it. The drag penalty may be found with the help of Table 6.2.

The airplane drag increment due to the nozzle may be found from:

$$\Delta C_{D_{noz}} = (\Delta C_{D_{noztype}}) S_{fus}/S \qquad (6.47)$$

<u>Note:</u> Eqn.(6.47) assumes that only ONE nozzle is at the rear of the fuselage. If more are present, the drag increases accordingly.

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Figure 6.32 Typical Subsonic Nozzle Concepts

Table 6.2 Drag Penalties for Nozzles Mounted in the Rear of a Fuselage, for Subsonic Flight

Nozzle Type	Drag Increment $\Delta C_{D_{noztype}}$
Short convergent	0.036 - 0.042
Blow-in-door ejector	0.025 - 0.035
Plug	0.015 - 0.020
Fully variable ejector	0.010 - 0.020
Iris	0.010 - 0.020
Ramp	0.010

Notes: 1. these data apply in the M = 0.8 - 0.95 range
2. these data apply to nozzle pressure ratios of 2.5 to 3.0

If more than one engine exhausts at the rear of the fuselage, a difficult 'aerodynamic fairing' problem between the engines can arise. Figure 6.33 shows a number of possibilities. The drag increments due to such 'faired' nozzle arrangements may be estimated from Fig.6.33. Note that longer fairings reduce the drag of the installation. However: a longer fairing also implies a weight increase: as usual, a trade study must be performed to arrive at the correct decision.

6.3.3.4 Jet engines: supersonic

The supersonic drag increment of nozzles is strongly dependent on the method used to 'fair' the rear end of the airplane, especially in the area of the variable geometry nozzle. No simple method can be given for the estimation of nozzle drag increment. References 9, 12, 33 and 41 should be consulted for details.

Figure 6.33 Effect of Nozzle Fairing Concept on Drag

6.4 PREDICTION OF INSTALLED POWER AND THRUST

The purpose of this section is to present rapid methods for estimating the installed performance of engines in airplanes. The material is presented as follows:

6.4.1 Propeller Driven Airplanes
6.4.2 Jet Driven Airplanes

6.4.1 Propeller Driven Airplanes

Propeller driven airplanes can use different methods for driving the propeller. In this text the following possibilities will be presented:

6.4.1.1 Piston propeller driven airplanes
6.4.1.2 Turbopropeller driven airplanes

6.4.1.1 Piston propeller driven airplanes

The installed performance of piston engines is normally stated in terms of available, installed power, P_{av}.

In some applications, the static thrust obtainable from a propeller may be important. Methods for finding static propeller thrust for a given amount of available shaft-horse-power are given in Reference 15.

The following step-by-step procedure is suggested for finding available, installed power, P_{av}.

Step 1: Determine the flight conditions for which the installed power available calculation is to be made. This consists of the selection of altitude, temperature and airspeed.

Step 2: From engine manufacturers data determine the available shaft horse power, SHP_{av} for each flight condition. Figure 6.34 shows a typical example of such data.

Step 3: Find the installed, available power, P_{av} from:

$$P_{av} = (SHP_{av} - P_{extr})(\eta_{inl/inc})\eta_p\eta_{gear} \qquad (6.48)$$

where: SHP_{av} follows from Step 2,

NOTE:

To find the actual horsepower from altitude, rpm, manifold pressure and air inlet temperature:

1. Locate A on full throttle altitude curve for given rpm and manifold pressure
2. Locate B on sea level curve for rpm and manifold pressure and transfer to C
3. Connect A and C by a straight line and read horsepower at given altitude: D
4. Modify horsepower at D for variation of air inlet temperature T from standard altitude temperature T_s by the formula:

Actual hp = hp at D x $\sqrt{\frac{T_s}{T}}$

where T and T_s are absolute temperatures

NORMALLY ASPIRATED

SUPERCHARGED

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Figure 6.34 Example of Manufacturers Uninstalled Engine Performance Data for a Piston Engine

P_{extr} follows from sub-section 6.1.1.

η_p, the propeller efficiency, may be found from propeller data. Methods to compute propeller efficiency for conventional propellers are provided in References 15, 42, 43, 44 and 45.

NOTE: a rapid method for the determination of the diameter of propellers was given in Chapter 5 of Part II. External noise considerations were not a part of that method. To assure that a propeller meets FAR 36 noise requirements, the method of References 46, 47 and 48 may be used. For preliminary design purposes, as long as the propeller tip speed is kept below a Mach number of 0.85, noise certification should not be a problem.

η_{gear} is the gearbox efficiency.

For direct drive installations use:

$\eta_{gear} = 1.0$

For geared installations with a well designed gearbox, use:

$\eta_{gear} = 0.98$

<u>Step 4:</u> Plot P_{av} versus speed and altitude. Refer to Figure 7.1 in Chapter 7 for an example.

<u>6.4.1.2 Turbopropeller driven airplanes</u>

Installed turbopropeller performance is presented in terms of installed, available power P_{av}.

Most turboprops also deliver a remnant thrust, T_r which varies with the flight condition. For performance calculations this remnant thrust is usually converted to power. This will be done in this text also.

For some applications it is necessary to know the static thrust available from a turboprop installation. For such cases the static remnant thrust is added to the static propeller thrust. Methods for determining static propeller thrust are given in Reference 15.

The following step-by-step procedure is suggested to find

installed, available power, P_{av}:

<u>Step 1:</u> Determine the flight conditions for which the calculation of available installed power and thrust is to be made. This consists of the selection of altitude, temperature and airspeed.

<u>Step 2:</u> From engine manufacturers data determine the available shaft horsepower, SHP_{av} as well as the available remnant thrust, T_r for each flight condition. An example of such data is shown in Figure 6.35.

<u>Step 3:</u> Find the available, installed power, P_{av} from:

$$P_{av} = (\eta_{inl/inc} SHP_{av} - P_{extr})\eta_p + (\eta_{inl/inc}) T_r U_1 / 550 \qquad (6.49)$$

where: $\eta_{inl/inc}$ follows from 6.2.3.2

SHP_{av} and T_r follow from Step 2

P_{extr} follows from sub-section 6.1.2

η_p, the propeller efficiency follows from Step 3 in 6.4.1.1.

<u>Note 1:</u> Eqn.(6.49) assumes that T_r is not affected by power extraction. Actually, this is not correct. Because the contribution of T_r to total available power is usually very small, the error made by this assumption is negligible.

<u>Note 2:</u> Eqn.(6.49) is not valid for $U_1 = 0$.

<u>Note 3:</u> Most turbopropeller engines already have a gearbox installed. The engine manufacturers data include the gearbox losses.

<u>Step 4:</u> Plot P_{av} versus speed and altitude. Refer to Figure 7.2 in Chapter 7 for an example.

Figure 6.33 Example of Manufacturers Uninstalled Engine Performance Data for a Turboprop Engine

6.4.2 Jet Driven Airplanes

For jet driven airplanes, engine performance is given in terms of available, installed thrust, T_{av}.

Step-by-step procedures for determining T_{av} are presented as follows:

6.4.2.1 Subsonic operations
6.4.2.2 Supersonic operations

6.4.2.1 Subsonic operations:

Step 1: Determine the flight condition for which the installed thrust must be determined. This consists of the selection of altitude, temperature and airspeed.

Step 2: From engine manufacturers data, determine the available uninstalled thrust, $T_{tst/av}$.

Figure 6.36 shows an example of such data.

Step 3: Find the available installed thrust from:

$$T_{av} = \tag{6.50}$$

$$[(T_{tst/av})(1 - 0.35K_tM_1(1 - \eta_{inl/inc})) - 550(P_{extr}/U_1)]$$

where: $T_{tst/av}$ follows from Step 2

M_1 is the flight Mach number

$\eta_{inl/inc}$ may be found from 6.2.3.3

P_{extr} follows from Sub-section 6.1.2

K_t is determined from Figure 6.37

Step 4: Plot T_{av} versus speed and altitude. Refer to Figure 7.3 in Chapter 7 for an example.

6.4.2.2 Supersonic operations:

Step 1: Determine the flight conditions for which the available thrust, T_{av} must be found.

This consists of the selection of altitude, temperature and airspeed.

Figure 6.36 Example of Manufacturers Uninstalled Engine Performance Data for a Subsonic Turbofan

Figure 6.37 Effect of Mach Number on K_t

Figure 6.38 Example of Manufacturers Uninstalled Engine Performance Data for a Supersonic Turbofan

Step 2: From engine manufacturers data find the uninstalled, available thrust, $T_{tst/av}$.

Figure 6.38 gives an example of such data.

Note: These data are normally given for an assumed pressure recovery schedule with free stream Mach number: $\eta_{ref/inl}$. Figure 6.38 shows a typical pressure recovery schedule used by engine manufacturers for military engines.

Step 3: Compute the installed, available thrust, T_{av} from:

$$T_{av} = T_{tst/av}(1 - F_t - P_t) - 550P_{extr}/U_1 \qquad (6.51)$$

where: F_t accounts for actual inlet pressure recoveries and is found from:

$$F_t = C_R(\eta_{ref/inl} - \eta_{inl/com}) \qquad (6.52)$$

with: C_R is the ram recovery correction factor found from Figure 6.39.

$\eta_{inl/com}$ is the actual inlet pressure recovery which may be determined from Eqn.(6.41).

$\eta_{ref/inl}$ is found either from Fig.6.38 or from engine manufacturers data

$$P_t = 2\dot{m}_{bleed}/\dot{m}_a \qquad (6.53)$$

with: $\dot{m}_{bleed}$ found from Eqn.(6.7)

$\dot{m}_a$ determined from engine manufacturers data

P_{extr} follows from Sub-section 6.1.2

NOTE: The installed, available thrust, T_{av} is the

total thrust available from the entire installation. In other words, the effect of the thrust distribution over the various installation components as shown in Fig.6.12 has already been accounted for in the uninstalled engine manufacturers data albeit based on a reference pressure recovery and a reference nozzle.

Step 4: Plot T_{av} versus speed and altitude. Refer to Figure 7.5 in Chapter 7 for an example.

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Figure 6.39 Effect of Mach Number on the Ram Recovery Factor of Several Jet Engines

7. INSTALLED POWER AND THRUST DATA

The purpose of this chapter is to present example data for installed power and thrust. The information is presented as follows:

7.1 Propeller driven airplanes
7.2 Jet driven airplanes

7.1 PROPELLER DRIVEN AIRPLANES

In this section, two examples will be given for the determination of installed power data for propeller driven airplanes:

7.1.1 Piston propeller driven airplanes
7.1.2 Turbopropeller driven airplanes

7.1.1 Piston Propeller Driven Airplanes

The step-by-step procedure of 6.4.1.1 will be used.

Step 1: It will be assumed that installed, available power data must be provided for the following flight conditions:

altitude:	0 - 20,000 ft in increments of 5,000 ft
speed:	from 0 - 200 kts
temperature:	standard atmosphere

Note: the flight condition range must be compatible with the stated mission objectives of the airplane. Such mission objectives are normally defined in the airplane mission specification. Examples of airplane mission specifications are discussed in Part I.

The airplane used in this example is assumed to be a single engine airplane. It has an engine with the uninstalled characteristics of Figure 6.34.

Step 2: Figure 6.34 provides typical engine manufacturers data for a small piston engine. Notice that the data are given in terms of SHP_{av} for various altitudes, engine rpm (throttle) and inlet manifold pressure.

Step 3: The following input information is required before Eqn.(6.48) can be used:

P_{extr}, $\eta_{inl/inc}$, η_{gear} and η_p

This input information must be determined for each individual installation. In preliminary design the methods suggested in 6.4.1.1 may be used.

For the current example the following data will be assumed:

$P_{extr} = 4$ hp $\qquad \eta_{inl/inc} = 0.98$

$\eta_{gear} = 1.0$ (direct drive) $\qquad \eta_p = 0.88$

Warning: propeller efficiency for a fixed pitch propeller can be this high for only one flight condition. In this example a variable pitch propeller has been used.

Step 4: The determination of P_{av} now proceeds as follows:

For a given altitude and engine rpm, Figure 6.34 is used to find the SHP_{av}. Using Equation (6.48) it is then possible to compute P_{av} for each flight condition.

Figure 7.1 shows the results of these calculations for the required range of altitudes and flight speeds.

Note: for the effect of atmospheric temperature on engine performance, engine manufacturers data should be consulted. In the absence of such data, the following approximation may be used:

$$SHP_{av_{at\ T}} = SHP_{av_{at\ std\ T}} (T_{std}/T)^{1/2} \qquad (7.1)$$

7.1.2 Turbopropeller Driven Airplanes

The step-by-step procedure of 6.4.1.2 will be used.

Step 1: It will be assumed that installed, available power data must be provided for the following flight conditions:

altitude: 0 - 40,000 ft in increments of 10,000 ft
speed: 0 - 400 kts
temperature: standard sealevel

Note: the flight condition range must be compatible

The data in this Figure were arrived at with the assumptions listed in Sub-section 7.1.1.

Eqn.(6.48):

$$P_{av} = (SHP_{av} - P_{extr})\eta_{inl/inc}\eta_p\eta_{gear}$$

$$= (SHP_{av} \qquad - 4)(0.98)(0.88)(1.0)$$

<From Fig.6.34> <from Step 3, p.203/204>

3,400 RPM, max. allow.	Sea	5K	10K	15K	20K
SHP_{av}	285	248	206	172	140
$- P_{extr}$	281	244	202	168	136
P_{av}	242	210	174	145	117

Note that P_{av} is not a function of speed in these calculations. The reason is: propeller efficiency was assumed to be constant. This is not always the case!

Figure 7.1 Installed Available Horsepower Data for a Single Engine Piston Propeller Airplane

with the stated mission objectives of the airplane. Such mission objectives are normally defined in the airplane mission specification. Examples of airplane mission specifications are discussed in Part I.

The airplane used in this example is assumed to be a twin engine turbopropeller driven airplane. The engines have the uninstalled characteristics of Figure 6.35.

Step 2: Figure 6.35 defines the engine manufacturers data for this example calculation. Note that these data are given in terms of SHP_{av} and T_r.

Step 3: The following input information is required before Equation (6.49) can be used:

P_{extr}, $\eta_{inl/inc}$, U_1 and η_p

This input information must be determined for each individual installation. In preliminary design the methods suggested in 6.4.1.2 may be used.

For the current example the following data will be assumed:

P_{extr} = 10 shp

Plenum inlet with bypass duct: $\eta_{inl/inc}$ = 0.89

η_p = 0.92

U_1 is selected in increments of kts airspeed

Warning: propeller efficiency varies considerably with Mach number. The assumption has been made here that the propeller blades have supercritical airfoil sections and that a propeller pitch angle schedule is used which allows the propeller efficiency to be optimized at all speeds.

Step 4: The determination of installed, available power, P_{av} proceeds as follows:

For any given altitude, Figure 6.35 is used to find SHP_{av} and T_r. Using Equation (6.49) it is now possible to compute P_{av} for each flight condition. Figure 7.2 shows the installed engine performance for the airplane with both engines operating.

Note: for the effect of atmospheric temperature on engine performance, engine manufacturers data should be consulted. In the absence of such data, the following approximation may be used for sealevel performance:

$$SHP_{av_{at\ T}} = SHP_{av_{at\ std\ T}} - 2.33(T - T_{std}) \qquad (7.2)$$

where: T is the actual atmospheric temperature in degrees F.

T_{std} is the atmospheric temperature in the standard atmosphere in degrees F.

Observe that Eqn.(7.2) implies that for each degree of temperature increase above standard, 2.33 hp will be lost!

Figure 7.2 Installed Available Horsepower Data for a Twin Engine Turbopropeller Airplane

7.2 JET DRIVEN AIRPLANES

In this section, two examples will be given for the determination of installed thrust for jet driven airplanes:

7.2.1 Subsonic operations

7.2.2 Supersonic operations

7.2.1 Subsonic Operations

The step-by-step procedure of 6.4.2.1 will be used.

Step 1: It will be assumed that installed, available thrust data must be provided for the following flight conditions:

altitude:	0 - 50,000 ft in increments of 10,000 ft
speed:	0 - 500 kts
temperature:	standard atmosphere

Note: the flight condition range must be compatible with the stated mission objectives of the airplane. Such mission objectives are normally defined in the airplane mission specification. Examples of airplane mission specifications are discussed in Part I.

The airplane used in this example is assumed to be a twin engine airplane. It has engines with the uninstalled characteristics of Figure 6.36.

The airplane will be assumed to have a straight-through inlet.

Step 2: Figure 6.36 defines the engine manufacturers data for this example calculation. Note that these data are in terms of $T_{tst/av}$ for a range of flight conditions.

Step 3: The following input information is required before Eqn.(6.50) can be used:

K_t, M_1, $\eta_{inl/inc}$, P_{extr} and U_1

This input information must be determined for each individual installation. In preliminary design the methods suggested in 6.4.2.1 may be used.

For the current example, the following data will be assumed:

K_t from Figure 6.37 at each value of M_1

M_1 follows from the speed and altitude for which the calculations are made.

$\eta_{inl/inc}$ follows from Eqns (6.30) and (6.37).

Note: the inlet geometry must be available before the inlet pressure recovery can be computed! In this example it will be assumed that $\eta_{inl/inc} = 0.95$.

P_{extr} follows from Sub-section 6.1.2. In this example it will be assumed that $P_{extr} = 60$ hp.

U_1 is selected in increments of kts airspeed.

Step 4: The determination of T_{av} now proceeds as follows:

For each speed and altitude combination, Eqn.(6.50) is now used to determine T_{av}. This information is then plotted as in Figure 7.3.

Note: for the effect of atmospheric temperature on engine performance, engine manufacturers data should be consulted.

7.2.2 Supersonic Operations

The step-by-step procedure of 6.4.2.2 will be used.

Step 1: It will be assumed that installed, available thrust must be provided for the following range of flight conditions:

altitude:	0 - 60,000 ft in increments of 10,000 ft
speed:	M = 0 to 2.5
temperature:	standard atmosphere

Note: the flight condition range must be compatible with the stated mission objectives of the airplane. Such mission objectives are normally defined in the airplane mission specification. Examples of airplane mission specifications are discussed in Part I.

The airplane used in this example is assumed to be a

The data in this Figure were arrived at with assumptions listed in Sub-section 7.1.3.

Figure 7.3 Installed Available Thrust Data for a Subsonic Twin Engine Jet Transport

twin engined, supersonic business jet. The inlets are assumed to be of the mixed compression type (Fig.6.10c) with four shocks.

The uninstalled engine performance characteristics are assumed to be those of Figure 6.38.

Step 2: Figure 6.38 defines the engine manufacturers data for this installation in terms of $T_{tst/av}$. Note the reference pressure recovery schedule.

Step 3: The following input information is required before Eqn.(6.51) can be used:

F_t: which depends on C_R, $\eta_{inl/com}$ and $\eta_{ref/inl}$,

P_t: which depends on $\dot{m}_{bleed}$ and $\dot{m}_a$,

P_{extr} and U_1

For the current example it will be assumed that F_t has been determined to vary with Mach number as shown in Figure 7.4, that P_t = 0.07 and that P_{extr} = 20 hp

Step 4: The determination of T_{av} now proceeds as follows:

For any given speed and altitude, Eqn.(6.51) is used to compute T_{av}. The result is plotted in Figure 7.5.

Note: for the effect of atmospheric temperature on engine performance, engine manufacturers data should be consulted.

Figure 7.4 Effect of Mach Number on F_t for an Example Supersonic Twin Engine Business Jet

The data in this Figure were arrived at with assumptions listed in Sub-section 7.1.4 and the F_t data of Fig.7.4.

Figure 7.5 Installed Available Thrust Data for a Supersonic Twin Engine Business Jet

8. LIFT AND PITCHING MOMENT PREDICTION METHODS

The purpose of this chapter is to present rapid methods for the prediction of lift and pitching moment characteristics of airplanes. These characteristics are expressed in terms of the following relationships:

*Lift coefficient versus angle of attack

*Pitching moment coefficient versus lift coefficient

The reason for combining these relationships is the fact that unless an airplane can be trimmed (i.e. flown at zero pitching moment coefficient), it is not useful.

Figure 8.1 shows the fundamental C_L-α and C_m-C_L relationships which must be determined for any new design. Note the following maximum lift coefficients:

$$\left.\begin{array}{l} \text{Clean airplane (flaps up): } C_{L_{max}} \\ \text{Flaps down: take-off: } C_{L_{max_{TO}}} \\ \qquad\qquad\qquad \text{landing: } C_{L_{max_L}} \end{array}\right\} \begin{array}{l} \text{Trimmed with:} \\ \bar{x}_{cg} = \bar{x}_{ref} \end{array}$$

Methods for determining the magnitude of these maximum lift coefficients required to satisfy performance objectives are discussed in Part I as part of the overall sizing process of airplanes.

Class I methods for determining whether or not any given airplane design can achieve certain required maximum lift coefficients are given in Chapter 7 of Part II. The reader must take note of the fact that the effect of trim requirements was accounted for in Ch.7 of Pt.II by subtracting a 'token' increment from the untrimmed maximum lift coefficients. In <u>this</u> chapter Class II methods for estimating trimmed lift coefficients and trimmed maximum lift coefficients are presented. The material is organized as follows:

8.1 Prediction of lift coefficient versus angle of attack
8.2 Prediction of pitching moment versus lift coefficient
8.3 Prediction of trimmed lift and trimmed maximum lift coefficient

The methods are presented for the subsonic flow range. The reader is referred to specific sections of Ref.9 for those transonic and supersonic effects which are considered beyond the scope of this text.

8.1 PREDICTION OF LIFT COEFFICIENT VERSUS ANGLE OF ATTACK

In this section methods for predicting the variation of lift coefficient with angle of attack will be presented as follows:

8.1.1 Airfoil lift and maximum lift: flaps up
8.1.2 Airfoil lift and maximum lift: flaps down
8.1.3 Wing lift and maximum lift: flaps up
8.1.4 Wing lift and maximum lift: flaps down
8.1.5 Airplane lift and maximum lift: flaps up
8.1.6 Airplane lift and maximum lift: flaps down
8.1.7 Airplane lift in ground effect
8.1.8 Power effects on airplane lift

Figure 8.1 Fundamental Airplane Lift and Pitching Moment Characteristics

8.1.1 Airfoil Lift and Maximum Lift: Flaps Up

Figure 8.2 shows the relationship between airfoil lift coefficient and airfoil angle of attack which must be be determined with the methods to be presented in this Section. Key quantities needed in the construction of the airfoil c_l versus α curve are listed, with an indication of where methods for their estimation may be found.

8.1.1.1 Airfoil zero-lift angle of attack: α_{0_l}

Table 8.1 provides a summary of basic airfoil data from which α_{0_l} may be determined. Whenever possible, actual airfoil data should be used.

The airfoil zero-lift angle of attack may be estimated from Ref.9 (4.1.1.1) for arbitrary airfoils.

Experience shows that the airfoil zero-lift angle of attack is NOT a function of Reynold's number or Mach number in the subsonic speed range.

8.1.1.2 Airfoil lift curve slope: c_{l_α}

Table 8.1 provides a summary of basic airfoil data from which c_{l_α} may be determined. Whenever possible, actual airfoil data should be used.

The airfoil lift curve slope may be estimated from Ref.9 (4.1.1.2) for arbitrary airfoils.

Experience shows that the airfoil lift curve slope is dependent on Mach number in accordance with the Prandt-Glauert transformation:

For subsonic speeds:

$$\underset{\text{at } M}{c_{l_\alpha}} = \underset{\text{at } M=0}{(c_{l_\alpha})}/(1 - M^2)^{1/2} \qquad (8.1)$$

For supersonic speeds:

$$\underset{\text{at } M}{c_{l_\alpha}} = 4/(M^2 - 1)^{1/2} \qquad (8.2)$$

For transonic speeds: use tunnel data.

Figure 8.2 Airfoil Lift Coefficient Versus Angle of Attack Curve

Table 8.1a Experimental Low Speed Data for 4- and 5- Digit NACA Airfoils with a Smooth Leading Edge and for $R_N = 9 \times 10^6$ (Ref.16)

Airfoil	α_{o_ℓ} (deg)	c_{m_o}	c_{l_α} (deg^{-1})	a.c. (tenths c)	$\alpha_{c_{l_{max}}}$ (deg)	$c_{l_{max}}$	α^* (deg)
0006	0	0	.108	.250	9.0	.92	9.0
0009	0	0	.109	.250	13.4	1.32	11.4
1408	0.8	-.023	.109	.250	14.0	1.35	10.0
1410	-1.0	-.020	.108	.247	14.3	1.50	11.0
1412	-1.1	-.025	.108	.252	15.2	1.58	12.0
2412	-2.0	-.047	.105	.247	16.8	1.68	9.5
2415	-2.0	-.049	.106	.246	16.4	1.63	10.0
2418	-2.3	-.050	.103	.241	14.0	1.47	10.0
2421	-1.8	-.040	.103	.241	16.0	1.47	8.0
2424	-1.8	-.040	.098	.231	16.0	1.29	8.4
4412	-3.8	-.093	.105	.247	14.0	1.67	7.5
4415	-4.3	-.093	.105	.245	15.0	1.64	8.0
4418	-3.8	-.088	.105	.242	14.0	1.53	7.2
4421	-3.8	-.085	.103	.238	16.0	1.47	6.0
4424	-3.8	-.082	.100	.239	16.0	1.38	4.8
23012	-1.4	-.014	.107	.247	18.0	1.79	12.0
23015	-1.0	-.007	.107	.243	18.0	1.72	10.0
23018	-1.2	-.005	.104	.243	16.0	1.60	11.8
23021	-1.2	0	.103	.238	15.0	1.50	10.3
23024	-0.8	0	.097	.231	15.0	1.40	9.7

Table 8.1b Experimental Low Speed Data for 6- Digit NACA Airfoils with a Smooth Leading Edge and for $R_N = 9 \times 10^6$ (Ref.16)

Airfoil	α_{0_ℓ} (deg)	c_{m_0}	c_{l_α} (deg^{-1})	a.c. (tenths c)	$\alpha_{c_{l_{max}}}$ (deg)	$c_{l_{max}}$	α^* (deg)
63-006	0	.005	.112	.258	10.0	.87	7.7
63-009	0	0	.111	.258	11.0	1.15	10.7
63-206	-1.9	-.037	.112	.254	10.5	1.06	6.0
63-209	-1.4	-.032	.110	.262	12.0	1.40	10.8
63-210	-1.2	-.035	.113	.261	14.5	1.56	9.6
63_1-012	0	0	.116	.265	14.0	1.45	12.8
63_1-212	-2.0	-.035	.114	.263	14.5	1.63	11.4
63_1-412	-2.8	-.075	.117	.271	15.0	1.77	9.6
64-006	0	0	.109	.256	9.0	.8	7.2
64-009	0	0	.110	.262	11.0	1.17	10.0
64-206	-1.0	-.040	.110	.253	12.0	1.03	8.0
64-209	-1.5	-.040	.107	.261	13.0	1.40	8.9
64-210	-1.6	-.040	.110	.258	14.0	1.45	10.8
64_1-012	0	0	.111	.262	14.5	1.45	11.0
64_1-212	-1.3	-.027	.113	.262	15.0	1.55	11.0
64_1-412	-2.6	-.065	.112	.267	15.0	1.67	8.0

Table 8.1c Experimental Low Speed Data for 6- Digit NACA Airfoils with a Smooth Leading Edge and for $R_N = 9 \times 10^6$ (Ref.16)

Airfoil	α_{0_ℓ} (deg)	c_{m_0}	c_{l_α} (deg^{-1})	a.c. (tenths c)	$\alpha_{c_{l_{max}}}$ (deg)	$c_{l_{max}}$	α^* (deg)
65-006	0	0	.105	.258	12.0	.92	7.6
65-009	0	0	.107	.264	11.0	1.08	9.8
65-206	-1.6	-.031	.105	.257	12.0	1.03	6.0
65-209	-1.2	-.031	.106	.259	12.0	1.30	10.0
65-210	-1.6	-.034	.108	.262	13.0	1.40	9.6
65_1-012	0	0	.110	.261	14.0	1.36	10.0
65_1-212	-1.0	-.032	.108	.261	14.0	1.47	9.4
65_1-412	-3.0	-.070	.111	.265	15.5	1.66	10.5
63A010	0	.005	.105	.254	13.0	1.20	10.0
63A210	-1.5	-.040	.103	.257	14.0	1.43	10.0
64A010	0	0	.110	.253	12.0	1.23	10.0
64A210	-1.5	-.040	.105	.251	13.0	1.44	10.0
64A410	-3.0	-.080	.100	.254	15.0	1.61	10.0
64_1A212	-2.0	-.040	.100	.252	14.0	1.54	11.0
64_2A215	-2.0	-.040	.093	.252	15.0	1.50	12.0

8.1.1.3 Airfoil linear range angle of attack: α^*

Table 8.1 provides a summary of basic airfoil data from which α^* may be determined. Whenever possible, actual airfoil data should be used.

The airfoil linear range angle of attack may also be estimated from Ref.9 (4.1.1.3) for existing airfoils.

8.1.1.4 Airfoil angle of attack for maximum lift: $\alpha_{c_{l_{max}}}$

Table 8.1 provides a summary of basic airfoil data from which $\alpha_{c_{l_{max}}}$ may be determined. Whenever possible, actual airfoil data should be used.

The airfoil angle of attack for maximum lift may be estimated from Ref.9 (4.1.1.4) for existing airfoils.

8.1.1.5 Airfoil maximum lift coefficient: $c_{l_{max}}$

It has been found that airfoil maximum lift coefficient, $c_{l_{max}}$ depends upon the following parameters:

1. Leading edge shape as quantified by the so-called Δy parameter.
2. Maximum thickness and position of maximum thickness.
3. Maximum camber and position of maximum camber.
4. Reynold's number: $\rho c U_1/\mu$
5. Mach number.

Figure 8.3 defines these parameters for an arbitrary airfoil.

The method is valid for a 'base Reynold's number' of 9×10^6 and for Mach numbers below 0.2. The effect of variations in Reynold's number and Mach number relative to this 'base' is accounted for in the method.

Figure 8.3 Airfoil Geometric Characteristics

Figure 8.4 Basic Airfoil Maximum Lift Coefficient for Uncambered Airfoils

The maximum lift coefficient of an airfoil may be estimated from:

$$c_{l_{max}} = (c_{l_{max}})_{base} + \Delta_1 c_{l_{max}} + \Delta_2 c_{l_{max}} + \Delta_3 c_{l_{max}} + \Delta_3 c_{l_{max}} + \Delta_4 c_{l_{max}} + \Delta_5 c_{l_{max}} \qquad (8.3)$$

where: $(c_{l_{max}})_{base}$ is the basic airfoil maximum lift coefficient, found from Figure 8.4 as a function of airfoil geometry. The parameter Δy in Figure 8.4 is defined in Figure 8.3. Table 8.2 lists example data for Δy.

$\Delta_1 c_{l_{max}}$ is the airfoil maximum lift increment due to camber and due to position of maximum camber. See Figure 8.5.

$\Delta_2 c_{l_{max}}$ is the airfoil maximum lift increment due position of maximum thickness. This increment is obtained from Figure 8.6. NOTE: when the position of maximum thickness is at 30 percent chord, this increment is ZERO.

$\Delta_3 c_{l_{max}}$ is the airfoil maximum lift increment due to Reynold's number: see Figure 8.7.

$\Delta_4 c_{l_{max}}$ is the airfoil maximum lift increment due to airfoil roughness. The basic roughness used is the so-called standard NACA roughness of 0.011 inch grit applied over the first 8 percent chord. Actual airplane roughness is much less severe. For airplanes with 'smooth' and 'clean' leading edges this increment is ZERO. Figure 8.8 provides data for finding this increment.

$\Delta_5 c_{l_{max}}$ is the section maximum lift increment due to Mach number (compressibility) Fig. 8.9 gives data for finding this increment

Whenever possible, actual data should be used. References 49, 50 and 51 should be consulted for specific data on airfoil maximum lift coefficients.

Table 8.1 contains $c_{l_{max}}$ data for a range of airfoils at the base Reynold's number of 9×10^6.

Table 8.2 Example Data for the Leading Edge Ay Parameter

Airfoil NACA	Position of max t/c perc. of c	Camber perc. of c	Position of max camber perc. of c	Ay perc. of c
0009	30	0	not appl.	2.35
1410	30	1.0	40	2.60
2415	30	2.0	40	3.80
4412	30	4.0	40	3.08
4315	30	4.0	30	3.30
4321	30	4.0	30	5.24
23012	30	1.8	13	3.03
23021	30	1.8	15	5.24
43012	30	3.7	15	3.08
63012	30	5.5	15	3.08
63-009	35	0	not appl.	2.00
63-209	35	1.1	50	2.00
63_1-012	35	0	not appl.	2.65
63_1-212	35	1.1	50	2.65
63_1-412	35	2.2	50	2.65
63_1-018	35	0	not appl.	3.95
63_1-618	35	3.3	40	3.95
64-108	40	0.55	50	1.70
64_1-112	40	0.55	50	2.52
64_3-013	40	0	not appl.	3.75
64_3-412	40	2.2	50	3.75
66-009	45	0	not appl.	1.65
66-205	45	1.1	50	1.65
66_2-018	45	0	not appl.	3.30
66_2-416	45	3.2	50	3.30

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Figure 8.5 Effect of Camber on Maximum lift

Figure 8.6 Effect of Maximum Thickness Position on Maximum Lift

Figure 8.7 Effect of Reynold's Number on Maximum Lift

Figure 8.8 Effect of Surface Roughness on Maximum Lift

Figure 8.9 Effect of Mach Number on Maximum Lift

8.1.1.6 Construction of airfoil lift curve: flaps up

All ingredients needed to construct the flaps-up airfoil c_l versus α curve are now available. The flaps-up curve in Figure 8.2 can therefore be constructed. Figure 8.10 shows how this is done in a step-by-step manner. The fairing between α^* and $\alpha_{c_{l_{max}}}$ is done with the help of a french curve. The behavior of the lift curve just before and just after stall depends on Reynold's number. Reference 49 and Ref.9 (4.1.1.3) contain more information.

Figure 8.10 Construction of Airfoil Lift Versus α Curve

8.1.2 Airfoil Lift and Maximum Lift: Flaps Down

Figure 8.11 shows the comparison between flaps-up and flaps-down airfoil lift characteristics. Key quantities which are required to determine the flaps-down airfoil lift characteristics are listed with an indication of where methods for their estimation may be found.

8.1.2.1 Airfoil lift increment due to flaps: Δc_l

The airfoil incremental lift coefficient increment due to flaps, Δc_l depends on the type of flaps used.

Methods are presented for the following flap types:

A) Trailing Edge Flaps

a) Plain flaps
b) Single-slotted flaps
c) Fowler flaps
d) Double slotted flaps
e) Split flaps

B) Leading Edge Flaps

a) Nose flaps
b) Krueger flaps
c) Leading edge slats
d) Spoilers

Note: If a wing is equipped with any combination of flaps as defined under A) or B) it is acceptable in preliminary design to add the increments due to each individual flap type. The reader should be aware that this can result in over-prediction of high lift capability!

A) Trailing Edge Flaps

a) Plain flaps

Figure 8.12 defines the geometry of a plain flap. The airfoil incremental lift coefficient due to a plain flap deflection with a sealed gap is given by:

$$\Delta c_l = (\delta_f)[c_{l_\delta}/(c_{l_\delta})_{theory}](c_{l_\delta})_{theory} k' \qquad (8.4)$$

where: k' is a correction factor which accounts for non-linearities at high flap deflections: see Figure 8.13.

$(c_{l_\delta})_{theory}$ is found from Figure 8.14. It accounts for flap size and for thickness ratio.

Figure 8.11 Fundamental Airfoil Lift Versus α Curve with the Flaps Down

Figure 8.12 Plain Flap Geometry

Figure 8.13 Correction Factor for Nonlinear Lift Behavior of Plain Flaps

Figure 8.14 Lift Effectiveness of a Plain Flap

$(c_{l_\delta}/(c_{l_\delta})_{theory}$ is found from Figure 8.15.

δ_f is the flap deflection in rad.

<u>Note:</u> for plain flaps with open gaps (not sealed) Reference 49 should be consulted.

<u>b) Single-slotted flaps</u>

Figure 8.16 defines the geometry of a single-slotted flap. The airfoil incremental lift coefficient due to a single-slotted flap deflection is given by:

$$\Delta c_l = (c_{l_\alpha})(\alpha_\delta)(\delta_f) \qquad (8.5)$$

where: c_{l_α} is the airfoil lift-curve-slope with the flaps up. It is found from 8.1.1.2.

α_δ is the airfoil lift effectiveness parameter found from Figure 8.17.

<u>c) Fowler flaps</u>

Figure 8.18 defines the geometry of a Fowler flap. The airfoil incremental lift coefficient due to a Fowler flap deflection is given by:

$$\Delta c_l = (c_{l_\alpha})(\alpha_\delta)(c'/c)(\delta_f) \qquad (8.6)$$

where: (c'/c) is defined in Figure 8.18. All other quantities in Eqn.(8.6) are those listed under b) Single-slotted flaps.

<u>d) Double slotted flaps</u>

Figure 8.19 defines the geometry of double slotted flaps: Types I and II.

<u>Type I:</u>

The airfoil incremental lift coefficient due to a Type I double slotted flap deflection is given by:

$$\Delta c_l = \eta_1(c_{l_{\delta_{f_1}}})(\delta_1)[(c + c_1)/c] + \eta_2(c_{l_{\delta_{f_2}}})(\delta_{f_1} + \delta_{f_2})(c'/c) \qquad (8.7)$$

where: η_1 and η_2 are found from Figure 8.20 using c_1/c

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FOR C_{l_α} SEE: 8.1.1.2

FOR $(C_{l_\alpha})_{theory}$ SEE: REF 9 SECTION 4.1.1.2 OR USE 2π

Figure 8.15 Correction Factor for Plain Flap Lift

Figure 8.16 Single Slotted Flap Geometry

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Figure 8.17 Lift Effectiveness of a Single Slotted Flap

Figure 8.18 Fowler Flap Geometry

Figure 8.19 Double Slotted Flap Geometry

Figure 8.20 Lift Effectiveness for Slotted Flaps

Figure 8.21 Lift Effectiveness for Trailing Edge Flaps

and c_2/c respectively. The effective turning angle, $\bar{\Phi}$ is defined as:

for vane: $$\bar{\Phi} = \delta_{f_1} + \bar{\Phi}_{TE_{upper}} \quad (8.8)$$

for flap segment: $$\bar{\Phi} = \delta_{f_1} + \delta_{f_2} + \bar{\Phi}_{TE_{upper}} \quad (8.9)$$

with: $$\bar{\Phi}_{TE_{upper}} = \arctan\{(10)(y_{90} - y_{100})\} \quad (8.10)$$

where: y_{90} and y_{100} are also defined in Figure 8.19.

$c_{l_{\delta_{f_1}}}$ and $c_{l_{\delta_{f_2}}}$ are found from Figure 8.21 for c_1/c and c_2/c respectively.

Type II:

The incremental lift coefficient due to a Type II double-slotted flap is given by:

$$\Delta c_l = \eta_1 (c_{l_{\delta_{f_1}}})(\delta_{f_1})(c_a'/c) + \eta_2 \eta_t (c_{l_{\delta_{f_2}}})(\delta_{f_2})\{1 + (c'-c_a')/c\} \quad (8.11)$$

where: η_t is found from Figure 8.22.

c' and c_a' are defined in Figure 8.19.

all other quantities are defined under Type I.

e) Split flaps

Figure 8.23 defines the geometry for a split flap. The incremental lift coefficient due to a split flap is given by:

$$\Delta c_l = (c_{l_\alpha})(\alpha_{\delta_{sf}})(\delta_f) \quad (8.12)$$

where: c_{l_α} is found from 8.1.1.2.

$\alpha_{\delta_{sf}}$ is found from Figure 8.24.

Figure 8.22 Correction Factor for Aft Flap

Figure 8.23 Split Flap Geometry

Figure 8.24 Lift Effectiveness of a Split Flap

B) Leading Edge Flaps

a) Nose flaps

Figure 8.25 defines the geometry for a nose flap. The airfoil incremental lift coefficient due to a nose flap deflection is given by:

$$\Delta c_l = c_{l_\delta} \delta_f \qquad (8.13)$$

where: c_{l_δ} is the leading edge flap effectiveness parameter for a nose flap: see Figure 8.26. Use c_f/c as the flap-chord to wing-chord ratio.

δ_f is the nose flap deflection in degrees as defined in Figure 8.25.

b) Krueger Flaps

Figure 8.27 defines the geometry for a Krueger flap. The airfoil incremental lift coefficient due to a Krueger flap is given by:

$$\Delta c_l = c_{l_\delta} \delta_f (c'/c) \qquad (8.14)$$

where: c_{l_δ} is the leading edge flap effectiveness parameter for a Krueger flap: see Figure 8.26. Use c_f/c' as the flap-chord to wing-chord ratio.

δ_f is the Krueger flap deflection in degrees as defined in Figure 8.27.

c'/c is as defined in Figure 8.27.

c) Leading Edge Slats

Figure 8.28 defines the geometry of a leading edge slat. The airfoil incremental lift coefficient due to a leading edge slat is given by:

$$\Delta c_l = c_{l_\delta} \delta_f (c'/c) \qquad (8.15)$$

where: c_{l_δ} is the leading edge flap effectiveness parameter for a leading edge slat as found from Figure 8.26 by using c_f/c' as the leading-edge-slat-chord to wing-chord ratio.

δ_f is the slat deflection in deg.: see Fig. 8.28.

c'/c is as defined in Figure 8.28.

Figure 8.25 Nose Flap Geometry

Figure 8.26 Lift Effectiveness of a Leading Edge Flap

Figure 8.27 Krueger Flap Geometry

Figure 8.28 Leading Edge Slat Geometry

Figure 8.29 Spoiler Geometry

Figure 8.30 Lift Effectiveness of a Spoiler

d) Spoilers

Figure 8.29 defines the geometry of a spoiler. Observe that the spoiler is defined by chord position and by vertical position of the spoiler trailing edge. The location of the spoiler hinge line is not important.

The airfoil incremental lift coefficient due to a spoiler deflection is given by:

$$\Delta c_l = -c_{l_\alpha} \Delta\alpha_s' \qquad (8.16)$$

where: c_{l_α} is the unflapped airfoil lift curve slope as found from 8.1.1.2

$\Delta\alpha_s'$ is the spoiler lift effectiveness parameter as found from Figure 8.30.

8.1.2.2 Airfoil lift curve slope due to flaps: $(c_{l_\alpha})_\delta$

As long as the chord length of a flapped airfoil does not change, its lift curve slope does not change either. However, as seen from Figures 8.12, 8.16, 8.19, 8.23, 8.27, 8.28 and 8.29 the flapped airfoil chord is normally larger than the unflapped airfoil chord. For that reason, the flapped airfoil lift curve slope is given by:

$$(c_{l_\alpha})_\delta = (c'/c)c_{l_\alpha} \qquad (8.17)$$

where: c is the chord of the unflapped airfoil

c' is the chord of the flapped airfoil as defined in Figs 8.12, 8.16, 8.23, 8.27, 8.28 or 8.30.

c_{l_α} is the unflapped airfoil lift curve slope as found from 8.1.1.2.

Note: spoiler deflections will be assumed not to alter the lift curve slope of an airfoil.

8.1.2.3 Airfoil maximum lift increment due to flaps: $\Delta c_{l_{max}}$

The magnitude of the maximum lift increment due to flaps depends on the type of flaps used. Methods are presented for trailing edge flaps as well as for leading edge flaps.

A. Trailing Edge Flaps

The airfoil incremental, maximum lift coefficient due to trailing edge flaps is given by:

$$\Delta c_{l_{max}} = k_1 k_2 k_3 (\Delta c_{l_{max}})_{base} \qquad (8.18)$$

where: $(\Delta c_{l_{max}})_{base}$ is the airfoil incremental, maximum lift coefficient due to flaps as determined from Figure 8.31. Note that the data in Figure 8.31 are based on a 25 percent reference flap-chord to airfoil chord ratio AND on a reference flap deflection angle defined in Figure 8.33.

k_1 is a factor which accounts for flap-chord to airfoil chord ratios different from 25 percent. It can be found from Figure 8.32.

k_2 is a factor which accounts for flap angles different from the reference flap angle. It can be found from Figure 8.33.

k_3 is a factor which accounts for flap motion as a function of flap deflection. It can be found from Figure 8.34.

B) Leading Edge Flaps

The airfoil incremental, maximum lift coefficient due to leading edge flaps is given by:

$$\Delta c_{l_{max}} = (c_{l_{\delta_{max}}}) \eta_{max} \delta_f (c'/c) \qquad (8.19)$$

where: $c_{l_{\delta_{max}}}$ is the theoretical maximum lifting effectiveness as found from Figure 8.35.

η_{max} is an empirical factor which accounts for the ratio of airfoil leading edge radius to airfoil thickness of the unflapped airfoil. It is obtained from Figure 8.36.

η_δ is an empirical factor which accounts for the difference between the actual leading edge flap deflection and the reference deflection. It is found from Figure 8.37.

δ_f is the leading edge flap deflection angle in

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Figure 8.31 Basic Airfoil Maximum Lift Increment due to Trailing Edge Flaps

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Figure 8.32 Flap Chord Correction Factor

Figure 8.33 Flap Angle Correction Factor

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$$\frac{\text{ACTUAL FLAP ANGLE}}{\text{REFERENCE FLAP ANGLE}} = \frac{\delta_f}{\text{SEE FIG. 8.33}}$$

Figure 8.34 Flap Motion Correction Factor

Figure 8.35 Maximum Lift Effectiveness for Leading Edge Flaps

Figure 8.36 Effect of Leading Edge Radius and Thickness Ratio on Maximum Lift of Leading Edge Flaps

radians as shown in Figure 8.25, 8.27 or 8.28.

c'/c is the ratio of the airfoil chord with leading edge flaps extended to the basic airfoil chord.

8.1.2.4 Construction of airfoil lift curve: flaps down

All ingredients needed to construct the flaps-down airfoil c_l versus α curve are now available. The flaps-down curve in Figure 8.2 can therefore be constructed. Figure 8.38 shows how this is done in a step-by-step manner. Since no empirical method for the determination of α^* has been presented, the fairing between the linear range and the maximum lift point must be done by guestimation.

The following groundrules are useful:

Trailing edge flaps: the angle of attack for maximum lift with trailing edge flaps down is normally below that for the basic airfoil.

Leading edge flaps: the angle of attack for maximum lift with leading edge flaps down is normally above that for the basic airfoil.

All flaps: the flap deflection angle beyond which the flap lift increment Δc_l ceases to be linear depends on detailed attention paid to flap contour and flap gap design. Table 8.3 provides some guidelines for flap deflection ranges for which linear behavior is expected.

Table 8.3 Linear Flap Effectiveness Range at Low Angles of Attack for Trailing Edge Flaps

Flap Type	δ_f (deg)	
	Poor Design	Good Design
Plain	0 to 10	0 to 20
Single Slotted and Fowler	0 to 20	0 to 30
Double Slotted	0 to 30	0 to 50
Split	0 to 30 or 45	

Figure 8.37 Effect of Leading Edge Flap Deflection on Maximum Lift

Figure 8.38 Construction of Flaps Down Airfoil Lift Versus α Curve

8.1.3 Wing Lift and Maximum Lift: Flaps Up

Figure 8.39 shows the relationship between wing lift coefficient and wing angle of attack which must be determined with the methods to be presented in this section. Key quantities needed in the construction of the wing C_{L_W} versus α_W curve are listed, with an indication of where methods for their estimation may be found.

8.1.3.1 Wing zero-lift angle of attack: $\alpha_{o_{L_W}}$

The wing angle of attack, α_W is defined as follows:

$$\alpha_W = \alpha - i_W \tag{8.20}$$

where: α is the airplane angle of attack and

i_W is the wing incidence angle

Figure 8.40 shows how these quantities relate to each other on an airplane. Criteria for the selection of the wing incidence angle, i_W are discussed in Part III, p.195.

For subsonic speeds:

For wings with constant airfoil sections and linear twist distributions the wing zero-lift angle of attack may be estimated from:

$$\alpha_{o_{L_W}} = \{\alpha_{o_1} + (\Delta\alpha_o/\epsilon_t)\epsilon_t\}\{(\alpha_{o_1})_{\text{at } M}\}/\{(\alpha_{o_1})_{\text{at } M=0.3}\} \tag{8.21}$$

where: α_{o_1} is found from 8.1.1.1

$(\Delta\alpha_o/\epsilon_t)$ is the change in wing zero-lift angle of attack per degree of linear wing twist: See Figure 8.41.

ϵ_t is the wing twist angle: see Figure 8.41. Criteria for the selection of wing twist angle are presented in Part III, p.193.

$\{(\alpha_{o_1})_{\text{at } M}\}\{(\alpha_{o_1})_{\text{at } M=0.3}\}$ is found from Fig.8.42.

Figure 8.39 Wing Lift Coefficient Versus Angle of Attack Curve

Figure 8.40 Relationship Between Wing Incidence, Wing Angle of Attack and Airplane Angle of Attack

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Figure 8.41 Effect of Linear Twist on Wing Angle of Attack for Zero Lift

Figure 8.42 Mach Number Correction for Zero-Lift Angle of Attack of Cambered Airfoils

For transonic and supersonic speeds:

In the transonic and supersonic speed ranges experimental data should be used.

8.1.3.2 Wing lift curve slope: $C_{L_{\alpha_W}}$

For subsonic speeds:

For conventional, straight tapered wings, with 'moderate' sweep angles, the lift curve slope may be estimated from Figures 8.43 or from:

$$C_{L_{\alpha_W}} = 2\pi A/[2 + \{A^2\beta^2/k^2(1 + \tan^2\Lambda_{c/2}/\beta^2) + 4\}^{1/2}] \quad (8.22)$$

where: $A = b^2/S$ is the wing aspect ratio - (8.23)

Wing aspect ratio may have to be computed for the so-called 'equivalent' wing planform: See Figure 8.44.

$$\beta = (1 - M^2)^{1/2} \quad (8.24)$$

$$k = (c_{l_\alpha})_{\text{at } M}/2\pi \quad (8.25)$$

where $(c_{l_\alpha})_{\text{at } M}$ follows from Eqn.(8.1).

$\Lambda_{c/2}$ is the semi-chord sweep angle. Relations between semi-chord, quarter-chord and leading edge sweep angles are defined in Figures 8.45 and 8.46.

For transonic speeds: see Reference 9.

For supersonic speeds: See Figures 8.47. Note: these figures apply only for low angles of attack.

Methods for estimating wing lift curve slopes for arbitrary wing planforms are presented in Reference 9.

8.1.3.3 Wing linear range of angle of attack: $\alpha_W{}^*$

In preliminary design it is acceptable to use:

$\alpha_W{}^* = \alpha^*$ as found from 8.1.1.3.

Figure 8.43 Wing Lift Curve Slope at Low Mach Numbers

Figure 8.44 Example of an Equivalent Wing Planform

Figure 8.45 Leading Edge Sweep Angle Versus Quarter Chord Sweep Angle for Straight Tapered Wings

Figure 8.45 (Cont'd) Leading Edge Sweep Angle Versus Quarter Chord Sweep Angle for Straight Tapered Wings

Figure 8.46 Leading Edge Sweep Angle Versus Semi-chord Sweep Angle for Straight Tapered Wings

Figure 8.46 (Cont'd) Leading Edge Sweep Angle Versus Semi-chord Sweep Angle for Straight Tapered Wings

Figure 8.47a Wing Lift Curve Slope for Supersonic Mach Numbers and Low Angle of Attack

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Figure 8.47b Wing Lift Curve Slope for Supersonic Mach Numbers and Low Angle of Attack

HINTS FOR USING FIGURES 8.47 a, b, AND c:

Note: for low angles of attack:

$$C_{L_\alpha} = C_{N_\alpha}$$

Note: $C_{N_\alpha} = (C_{N_\alpha})_{theory}[(C_{N_\alpha})/(C_{N_\alpha})_{theory})]$,

where: $(C_{N_\alpha})_{theory}$ follows from Figures 8.47a or 8.47b

$[(C_{N_\alpha})/(C_{N_\alpha})_{theory})]$ follows from Figure 8.47c

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Figure 8.47c Wing Lift Curve Slope Correction Factor for Wings with Sonic Leading Edge Regions

8.1.3.4 Wing maximum lift coefficient: $C_{L_{max_w}}$ and wing angle of attack for maximum lift: $(\alpha_{C_{L_{max}}})_w$

In this section, the assumption is made that the reader has available a computer program for the calculation of the spanwise lift coefficient distribution for a given wing angle of attack, α_w as defined by Eqn.(8.20).

Methods for finding spanwise distributions of lift coefficient with angle of attack can be found in:

Reference 49: Chapter 1: this method applies to unswept wings at subsonic Mach numbers. However, by using the cosine rule, this method will work for sweep angles up to 35 degrees.

Reference 52: this method applies to arbitrary wing planforms at subsonic speeds.

To estimate the maximum wing (flaps-up) lift coefficient, the following step-by-step procedure is suggested:

Step 1: Determine the spanwise distribution of section maximum lift coefficient. Plot this as shown in Figure 8.48. Note: the section maximum lift coefficient at each spanwise station is found from Section 8.1.1.5 or from airfoil data.

NOTE: at each spanwise station the Reynold's number will be different!

Step 2: With the help of either Ref.49, Ref.52 or with the help of a spanwise-lift computer program, plot the spanwise variation in local section lift coefficient for increasing values of wing angle of attack. Note that at some value of α_w the spanwise plot of

of c_l is tangent to the spanwise plot of

$c_{l_{max}}$ as obtained from Step 1. That value of wing angle of attack is: $(\alpha_{C_{L_{max}}})_w$.

Fig.8.48 also shows the resulting plot(s).

Step 3: Calculate the wing maximum lift coefficient from:

$$C_{L_{max_w}} = (1/S)\int_0^{1.0} bc(c_{l_{w_{stall}}})d\eta \qquad (8.26)$$

where $c_{l_{w_{stall}}}$ is the spanwise value of the local section lift coefficient at:

$$\alpha_w = (\alpha_{c_{L_{max}}})_w$$

Important note: the magnitude of $C_{L_{max_w}}$ is always below that of the magnitude of $c_{l_{max}}$ of any wing airfoil.

8.1.3.5 Construction of wing lift curve: flaps up

All ingredients for constructing the wing lift curve are now available. Figure 8.49 shows how this can be done in a stepwise manner.

Figure 8.48 Section Maximum Lift Coefficient and Section Lift Coefficient at a Given Angle of Attack Versus Spanwise Position on a Wing

Figure 8.49 Construction of Flaps Up Wing Lift Versus Angle of Attack Curve

Figure 8.50 Fundamental Wing Lift Versus α Curve with the Flaps Down

8.1.4 Wing Lift and Maximum Lift: Flaps Down

Figure 8.50 shows the relationship between flaps up and flaps down wing lift characteristics. Key quantities needed in the construction of the flaps down wing C_{L_w} versus α_w curve are listed, with an indication of where methods for their estimation may be found.

8.1.4.1 Wing lift increment due to flaps: ΔC_{L_w}

The wing lift increment due to trailing edge and/or leading edge flaps may be estimated from:

$$\Delta C_{L_w} = K_b(\Delta c_l)(C_{L_{\alpha_w}}/c_{l_\alpha})\,[((\alpha_\delta)_{C_L})/((\alpha_\delta)_{c_l})] \qquad (8.27)$$

where: K_b is the flap-span factor as obtained from the procedure suggested in Figure 8.51 but with the data from Figure 8.52.

Δc_l is the airfoil lift increment due to flaps as obtained from 8.1.2.1.

$C_{L_{\alpha_w}}$ is the wing lift curve slope as obtained from 8.1.3.2.

c_{l_α} is the wing airfoil lift curve slope as obtained from 8.1.1.2.

$\dfrac{(\alpha_\delta)_{C_L}}{(\alpha_\delta)_{c_l}}$ is the ratio of the three-dimensional flap-effectiveness parameter to the two-dimensional flap-effectiveness parameter as found in Figure 8.53.

Note: If a mechanical high lift system consists of a combination of leading and trailing edge high lift devices, the method should be applied to each type of device separately. The resulting increments in lift coefficients can then be added.

8.1.4.2 Wing lift curve slope due to flaps: $(C_{L_{\alpha_w}})_\delta$

Wings with NON-TRANSLATING flap systems:

For wings with NON-TRANSLATING flap systems (such as split flaps, plain flaps and nose flaps), the wing lift curve slope flaps-down is considered to be the same as

Figure 8.51 Procedure for Estimating K_b

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Figure 8.52 Effect of Taper Ratio and Flap Span on K_b

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Figure 8.53 Effect of Aspect Ratio and Flap-Chord Ratio on the Three-Dimensional Flap Effectiviness

Figure 8.54 Definition of 'Flapped' Wing Area

that for the flaps-up: the method of 8.1.4.1 may be used.

Wings with TRANSLATING flap systems:

For wings with translating flaps (such as Fowler flaps, most slotted flaps, slats and Krueger flaps), the flaps-down lift curve slope may be estimated from:

$$(C_{L_{\alpha_w}})_\delta = C_{L_{\alpha_w}} \{1 + (c'/c - 1)(S_{w_f}/S)\} \qquad (8.28)$$

where: $C_{L_{\alpha_w}}$ is found from 8.1.4.1

c'/c is the ratio of extended wing chord to the chord of the flaps-up wing. Figures 8.18, 8.19 and 8.27 show definitions for these chords depending on the type of flap used.

S_{w_f}/S is the ratio of the 'flapped' wing area to that of the area of the flaps-up wing. Figure 8.54 shows how this is defined.

8.1.4.3 Wing maximum lift increment due to flaps: $\Delta C_{L_{max_w}}$

A) Trailing Edge Flaps

The maximum wing incremental lift coefficient due to trailing edge flaps is found from:

$$\Delta C_{L_{max_w}} = (\Delta c_{l_{max}})(S_{w_f}/S)K_\Delta \qquad (8.29)$$

where: $(\Delta c_{l_{max}})$ is the airfoil incremental lift coefficient due to trailing edge flaps as found from 8.1.2.3.

S_{w_f}/S is defined in Figure 8.54

K_Δ is a planform correction factor found from Figure 8.55.

B) Leading Edge Flaps

The maximum wing incremental lift coefficient due to leading edge flaps (slats and/or Kruegers) is found from:

$$\Delta C_{L_{max_w}} = 7.11(c_{lef}/c)(b_{lef}/b_e)^2\cos^2\Lambda_{c/4} \qquad (8.30)$$

Figure 8.55 Effect of Sweep on Planform Correction Factor

Figure 8.56 Definition of Leading Edge Flap Chord Ratio

Figure 8.57 Definition of Leading Edge Flap Span Ratio

where: c_{lef}/c is defined in Figure 8.56.

b_{lef}/b_e is defined in Figure 8.57.

8.1.4.4 Construction of the wing lift curve: flaps down

All ingredients necessary to construct the flaps down wing lift curve are now available. Figure 8.58 shows the step-by-step manner in which this can be done.

Figure 8.58 Construction of Flaps Down Wing Lift Versus Angle of Attack Curve

8.1.5 Airplane Lift and Maximum Lift: Flaps Up

Figure 8.59 shows the relationship between airplane lift coefficient and airplane angle of attack which must be determined with the methods to be presented in this section. Key quantities needed in the construction of airplane C_L versus α curve are listed, with an indication of where methods for their estimation may be found.

The assumption will be made, that an airplane can be considered to consist of three components:

a) wing + fuselage b) horizontal tail c) canard

Figure 8.60 shows the relative arrangement of these three major components. If an airplane is equipped with pylon mounted nacelles (such as the B-727 and the DC-9), the pylon + nacelle combination should be 'counted' as an additional horizontal tail. Figure 8.61 indicates the 'equivalent' geometries which should then be used.

Figure 8.62 shows a number of important geometric parameters which are used in the calculation of overall airplane lift characteristics.

IMPORTANT CONSIDERATIONS:

In this chapter the following incidence angles are accounted for:

Canard: i_c Wing: i_w Horizontal Tail: i_h

These incidence angle are defined in Figure 8.60. In this sub-section, these incidence angles are assumed to be CONSTANT. The effect of trim requirements is considered in Section 8.3.

A horizontal tail and a canard may be equipped with a trailing edge control surface:

Horizontal tail with elevator, deflection: δ_e

Canard with canardvator, deflection: δ_c

In this sub-section, the assumption will be made that all control surface deflection angles are ZERO. The effect of trim requirements which would cause these control surface deflections to be non-zero is considered in Section 8.3.

Figure 8.59 Airplane Lift Coefficient Versus Angle of Attack Curve

Figure 8.60 Relative Arrangement of Canard, Wing and Horizontal Tail

Figure 8.61 Equivalent Geometries for Pylon Engine Mounts

Figure 8.62 Geometric Parameters Required for Computing Overall Airplane Lift

8.1.5.1 Airplane zero-lift angle of attack: α_{o_L}

The airplane zero-lift angle of attack may be estimated from:

$$\alpha_{o_L} = (-C_{L_o})/C_{L_\alpha} \tag{8.31}$$

where: C_{L_o} is the zero-angle-of-attack lift coefficient which is found from 8.1.5.2.

C_{L_α} is the airplane lift curve slope which may be found from 8.1.5.3.

8.1.5.2 Airplane zero-angle-of-attack lift coefficient: C_{L_o}

The airplane zero-angle-of-attack lift coefficient may be estimated from:

$$C_{L_o} = C_{L_{o_{wf}}} + C_{L_{\alpha_h}} \eta_h (S_h/S)(i_h - \varepsilon_{o_h}) + C_{L_{\alpha_c}} \eta_c (S_c/S)(i_c + \varepsilon_{o_c}) \tag{8.32}$$

where: $C_{L_{o_{wf}}}$ is the zero-angle-of-attack lift coefficient of the wing-fuselage combination. Unless the fuselage is severely cambered:

$$C_{L_{o_{wf}}} = (i_w - \alpha_{o_{L_w}}) C_{L_{\alpha_{wf}}} \tag{8.33}$$

where: $\alpha_{o_{L_w}}$ is found from 8.1.3.1.

$C_{L_{\alpha_{wf}}}$ is the wing-fuselage lift curve slope as found from 8.1.5.3.

Note: the subscripts wf (wing-fuselage) and wb (wing-body) are interchangeable.

$C_{L_{\alpha_h}}$ is the horizontal tail lift curve slope as found from 8.1.3.2 with appropriate substitution of tail parameters for wing parameters.

$C_{L_{\alpha_c}}$ is the canard lift curve slope as found from 8.1.3.2 with appropriate substitution of canard parameters for wing parameters.

NOTE: Figure 8.62 defines the necessary geometric parameters needed to compute horizontal tail and canard lift curve slopes.

$$\eta_h = \bar{q}_h/\bar{q} \tag{8.34}$$

$$\eta_c = \bar{q}_c/\bar{q} \tag{8.35}$$

The dynamic pressures $\bar{q}_h$ and $\bar{q}_c$ seen by a horizontal tail and a canard differ from the free stream dynamic pressure $\bar{q}$ for the following reasons:

a) as the air passes over a wing-fuselage combination it gradually looses some of its kinetic energy. This energy loss is proportional to the friction drag of the wing-fuselage.

b) when a horizontal tail or a canard surface are placed in the slipstream of a propeller, the local dynamic pressure depends on the power absorbed by the propeller and on the distance of the surface from the propeller.

In preliminary design it is acceptable to use:

For jet airplanes:

$$\eta_c = 1.0 \tag{8.36}$$

$$\eta_h = \tag{8.37}$$

$$[1 - \{\cos^2(\pi z_h/2z_w)\}\{2.42(C_{D_{o_w}})^{1/2}\}/(x_h/\bar{c} + 0.30)]$$

where: $z_h = x_h \tan(\gamma_h + \varepsilon_{cl} - \alpha_w)$ (8.38)

with x_h, γ_h, ε_{cl} and α_w shown in Fig. 8.63.

$$\varepsilon_{cl} = 1.62C_{L_w}/\pi A \tag{8.39}$$

$$z_w = 0.68\bar{c}\{C_{D_{o_w}}(x_h/\bar{c} + 0.15)\}^{1/2} \tag{8.40}$$

$C_{D_{o_w}}$ is the wing zero-lift drag coefficient as found from 4.2.1.1.

Figure 8.63 Geometric Parameters Required for Computing Dynamic Pressure Ratios

Figure 8.64 Definition of Canard and Tail Areas which are Submerged in Propeller Slipstream

For propeller driven airplanes:

The dynamic pressure ratios depend on where the canard and/or the horizontal tail are located relative to the propeller. The following approximation is suggested:

$$\eta_{h \text{ or } c} = 1 + [(S_{h \text{ or } c})_{slip}/(S_{h \text{ or } c})] \times$$

$$\times [(2200P_{av})/(\bar{q}U_1\pi(D_p)^2)] \qquad (8.41)$$

where: $(S_{h \text{ or } c})_{slip}$ is the area of the tail or canard which is submerged in the propeller slipstream: see Figure 8.64.

U_1 is the steady state speed of the airplane

P_{av} is the available horsepower, see Section 6.4.

D_p is the propeller diameter in ft

S_h is the horizontal tail area: see Figure 8.62.

S_c is the canard area: see Figure 8.62.

ϵ_{o_h} is the horizontal tail downwash angle for zero airplane angle of attack

ϵ_{o_c} is the canard upwash angle for zero airplane angle of attack

NOTE: the magnitudes of tail downwash angle and canard upwash angle at zero airplane angle of attack are dependent on the wing lift distribution at zero airplane angle of attack. For most airplanes (flaps up) it is acceptable to use: $\epsilon_{o_h} = \epsilon_{o_c} = 0$.

In Section 8.1.6 it is seen that with the flaps down this approximation is not valid. In that case, an incremental downwash angle, $\Delta\epsilon_f$ must be accounted for.

8.1.5.3 Airplane lift curve slope: C_{L_α}

The airplane lift curve slope may be estimated from:

$$C_{L_\alpha} = C_{L_{\alpha_{wf}}} + C_{L_{\alpha_h}} \eta_h (S_h/S)(1 - d\epsilon/d\alpha) + C_{L_{\alpha_c}} \eta_c (S_c/S)(1 + d\epsilon_c/d\alpha) \quad (8.42)$$

where: $C_{L_{\alpha_{wf}}}$ is the wing-fuselage (wing-body) lift curve slope, given by:

$$C_{L_{\alpha_{wf}}} = K_{wf} C_{L_{\alpha_w}} \quad (8.43)$$

where: K_{wf} is the wing-fuselage interference factor given by:

$$K_{wf} = 1 + 0.025(d_f/b) - 0.25(d_f/b)^2 \quad (8.44)$$

$C_{L_{\alpha_w}}$ is found from 8.1.3.2.

$d\epsilon/d\alpha$ = downwash gradient at the horizontal tail =

$$4.44[(K_A K_\lambda K_h (\cos\Lambda_{c/4})^{1/2})^{1.19}] \times \{(C_{L_{\alpha_w}})_{at\ M}]/[(C_{L_{\alpha_w}})_{at\ M=0}]\} \quad (8.45)$$

where: $K_A = (1/A) - 1/(1 + A^{1.7})$ (8.46)

as shown in Figure 8.65a.

$K_\lambda = (10 - 3\lambda)/7$ (8.47)

as shown in Figure 8.65b.

$K_h = (1 - h_h/b)/(2l_h/b)^{1/3}$ (8.48)

as shown in Figure 8.65c with the parameters h_h, l_h defined in Fig. 8.66.

$d\epsilon_c/d\alpha$ is the upwash gradient at the canard. It may be found from Figure 8.67 for wings of quarter chord sweep angles up to 35 deg.

All other quantities in Eqn. (8.42) are defined in 8.1.5.2.

Figure 8.65 Factors for Computing Downwash

Figure 8.66 Geometric Parameters for Horizontal Tail Location

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Figure 8.67 Wing Upwash Gradient

8.1.5.4 Airplane linear range of angle of attack: α_A^*

In preliminary design it is acceptable to use:

$$\alpha_A^* = \alpha_w^* - i_w \tag{8.49}$$

where: α_w^* is found from 8.1.3.3.

8.1.5.5 Airplane maximum lift coefficient: $C_{L_{max}}$ and airplane angle of attack for maximum lift: $\alpha_{C_{L_{max}}}$

The airplane maximum lift coefficient is found from:

$$C_{L_{max}} = C_{L_{max_w}} - (C_{L_{\alpha_{wf}}})\Delta\alpha_{w/c} + \tag{8.50}$$

$$+ (C_{L_{\alpha_h}})(S_h/S)\{(\alpha_{C_{L_{max}}})(1 - d\varepsilon/d\alpha - \varepsilon_{o_h}) + i_h\} +$$

$$+ (C_{L_{\alpha_c}})(S_c/S)\{(\alpha_{C_{L_{max}}})(1 + d\varepsilon_c/d\alpha) + \varepsilon_{o_c} + i_c\}$$

where: $C_{L_{max_w}}$ is found from 8.1.3.4.

$C_{L_{\alpha_{wf}}}$ is found from Eqn.(8.43)

$\Delta\alpha_{w/c}$ is the difference between the airplane angles of attack for canard stall and for wing stall. In a canard airplane, the canard must always stall before the wing! In preliminary design it is suggested to use: $\Delta\alpha_{w/c} = 3$ deg.

$$\alpha_{C_{L_{max}}} = \alpha_{C_{L_{max_w}}} - i_w - \Delta\alpha_{w/c} \tag{8.51}$$

where: $\alpha_{C_{L_{max_w}}}$ is found from 8.1.3.4.

All other quantities are defined in 8.1.5.2.

8.1.5.6 Construction of airplane lift curve: flaps-up

All ingredients necessary to construct the airplane C_L versus α curve are now available. Figure 8.68 shows how this can be done in a step-by-step manner.

Figure 8.68 Construction of Airplane Lift Versus α Curve

Figure 8.69 Fundamental Airplane Lift Versus α Curve with the Flaps Down

8.1.6 Airplane Lift and Maximum Lift: Flaps Down

Figure 8.69 shows the relationship between airplane lift coefficient in the flaps down configuration and airplane angle of attack which must be determined with the methods to be presented in this section. Key quantities needed in the construction of the flaps down airplane C_L versus α curve are listed, with an indication of where methods for their estimation may be found.

8.1.6.1 Airplane lift increment due to flaps: ΔC_L

The airplane lift increment due to flaps may be estimated from:

$$\Delta C_L = k_{cw}\Delta C_{L_w} + k_{wc}(S_c/S)\Delta C_{L_c} + \tag{8.52}$$

$$+ k_{wh}(S_h/S)\Delta C_{L_h} - C_{L_{\alpha_h}}\eta_h(S_h/S)\Delta\epsilon_f$$

where: ΔC_{L_w} is found from 8.1.4.1

k_{cw} is a canard-on-wing interference factor. If the canard is small relative to the wing and if the canard is far enough forward of the wing, $k_{cw} = 1.0$ may be used in early design.

Whenever one or both of these conditions are not met, a finite element aerodynamic analysis of the canard-on-wing effect is needed. Such methods are beyond the scope of this text.

k_{wc} is the wing-on-canard interference factor. It is similar to (albeit not the same as) k_{cw}.

In early design it is acceptable to use: $k_{wc} = 1.0$.

S_c is the canard area: see Figure 8.62.

ΔC_{L_c} is found from 8.1.4.1 with appropriate substitution of canard parameters for wing parameters. If a canardvator deflection is present, it should be treated as a plain flap and its lift increment must be added.

Note: In airplanes such as the Piaggio P 180 Avanti, the canard is equipped with a Fowler

flap which is geared to the wing flap. The effect of such a flap must be accounted for! A detailed treatment of such a system is beyond the scope of this text.

k_{wh} is the wing-on-horizontal-tail interference factor. It is similar to (albeit not the same as) k_{cw}. In early design it is acceptable to use: $k_{wh} = 1.0$.

S_h is the horizontal tail area: see Figure 8.62.

ΔC_{L_h} is found from 8.1.4.1 with appropriate substitution of horizontal tail parameters for wing parameters. If an elevator is present, its deflection can be accounted for by considering the elevator to be a plain flap.

$C_{L_{\alpha_h}}$ is the horizontal tail lift-curve-slope which may be found from Eqn. (8.22) with appropriate substitution of horizontal tail parameters for wing parameters.

$\Delta\epsilon_f$ is the increase in tail downwash angle due to wing flap deflection. It may be estimated from Figure 8.70.

<u>8.1.6.2 Airplane lift curve slope due to flaps:</u> $(C_{L_\alpha})_\delta$

The airplane lift curve slope with the flaps down may be estimated from:

$$(C_{L_\alpha})_\delta = K_{wf}(C_{L_{\alpha_w}})_\delta + C_{L_{\alpha_h}}\eta_h(S_h/S)[1 - (d\epsilon/d\alpha)_\delta] + C_{L_{\alpha_c}}\eta_c(S_c/S)[1 - (d\epsilon_c/d\alpha)_\delta] \qquad (8.53)$$

where: $(C_{L_{\alpha_w}})_\delta$ is the wing lift-curve-slope with the flaps down. It is found from Eqn. (8.28).

where: $(d\epsilon/d\alpha)_\delta$ is the flaps-down downwash gradient at the horizontal tail.

$(d\epsilon_c/d\alpha)_\delta$ is the flaps-down upwash gradient at the canard.

<u>Note:</u> in preliminary design it is acceptable to set these flaps-down downwash and upwash gradients equal to those with the flaps up. The latter may be determined

Figure 8.70 Incremental Downwash Angle at the Horizontal Tail due to Flaps

Figure 8.71 Construction of Airplane Lift Versus α Curve with the Flaps Down

from 5.3.

1 other quantities were defined in 8.1.5.2 and 8.1.5

8.1.6 irplane maximum lift increment due to flaps: $C_{L_{max}}$

Tairplane maximum lift increment due to flaps may be estted from:

$$\Delta_{max} = k_{cw}\Delta C_{L_{max_w}} - (C_{L_{\alpha_w}})_\delta \Delta\alpha_{w/c} + (S_c/S)\Delta C_{L_{max_c}} + (S_h/S)C_{L_{\alpha_h}}\{(1 - d\epsilon/d\alpha) + i_h - \Delta\epsilon_f\} \quad (8.54)$$

where: $_c$ is the canard-on-wing interference factor. If the canard is small relative to the wing and if the canard is far enough forward of the wing, $k_{cw} = 1.0$ may be used in early design.

Whenever one or both of these conditions are not met, a finite element aerodynamic analysis of the canard-on-wing effect is needed. Such methods are beyond the scope of this text.

$\Delta\alpha_{wc}$ is the difference between the airplane angles of attack for canard stall and for wing stall. In a canard airplane, the canard must always stall before the wing! In In preliminary design, use: $\Delta\alpha_{w/c} = 3$ deg.

$\Delta\epsilon_f$ is the increase in tail downwash angle due to flaps. It may be found from Figure 8.70.

All other quantities in Eqn.(8.51) are defined in 8.1.6.1 and 8.1.6.2.

8.1.6.4 Construction of airplane lift curve: flaps down

All ingredients necessary to construct the flaps down airplane lift curve are now available. Figure 8.71 shows the step-by-step manner in which this can be done.

8.1.7 Airplane Lift in Ground Effect

When an airplane operates close to the ground, the downwash and the upwash patterns around its lifting surfaces change. References 9 and 14 contain physical explanations for these effects. In this section, the position is taken that the effect of the ground on the C_L versus α behavior of an airplane is a change in angle of attack at constant lift coefficient (or, vice versa, a change in lift coefficient at constant angle of attack). Figure 8.72 shows this in terms of the ground induced change in angle of attack at constant lift coefficient: $\Delta\alpha_g$. Note the difference in ground effect on high and on low aspect ratio configurations. This difference is accounted for as follows:

8.1.7.1 High aspect ratio configurations: transports
8.1.7.2 Low aspect ratio configurations: fighters

NOTE: it will be assumed that the effect of the ground on tail downwash and canard upwash can be neglected as far as overall airplane lift is concerned. The effect of the ground on tail downwash does alter tail lift and because of that the airplane pitching moment is affected: this is accounted for in Section 8.2.

8.1.7.1 High aspect ratio configurations: transports

Figure 8.72a shows the ground induced change in angle of attack at constant lift coefficient: $\Delta\alpha_g$.

For high aspect ratio configurations the change in angle of attack at constant lift coefficient in ground effect may be found from:

$$\Delta\alpha_g = -F_{tv}\{(9.12/A) + 7.16(c_r/b)\}(C_{L_{wf}}) + \qquad (8.55)$$

$$- \{A/(2C_{L_{\alpha_{wf}}})\}(c_r/b)\{(L/L_o) - 1\}(C_{L_{wf}})r +$$

$$- \{(\delta_f/50)^2/(C_{L_{\alpha_{wf}}})\}A(\Delta C_L)_f$$

where: F_{tv} is a factor which accounts for the effect on lift due to the image trailing vortex. It is found from Figure 8.73.

A is the wing aspect ratio

Figure 8.72 Ground Effect on Lift Curves

Figure 8.73 Factor Due to Image Trailing Vortex

c_r/b is the ratio of wing root chord to wing

$C_{L_{wf}}$ is the wing-fuselage lift coefficient (ground effect. This is found from Sec 8.1.5 or from Section 8.1.6 (depending e flap deflection) by setting $S_h = S_c =$ (

$C_{L_{\alpha_{wf}}}$ is the wing-fuselage lift-curve-slope determined from Eqn.(8.43), in 1/deg.

$(L/L_o - 1)$ accounts for the effect on lift o the image bound vortex. It is f from Figure 8.74.

r_g is a factor which accounts for the effec finite span. It may be found from Figur 6.

δ_f is the flap deflection in degrees

$\Delta(\Delta C_L)_f$ is a factor which accounts for the c of flaps in ground effect. It is f from Figure 8.76.

Equation (8.55) is used to estimate $\Delta\alpha_g$ for t values of $C_{L_{wf}}$ after which the airplane C_L versus re is modified as indicated in Figure 8.72a. This pre neglects the effect of the horizontal tail and the d on airplane lift in ground effect. In Section 8.2 will be seen that the effect of the horizontal tai the canard on airplane pitching moment in ground e is NOT neglected.

<u>8.1.7.2 Low aspect ratio configurations: fighters</u>

Figure 8.72b shows the ground induced change gle of attack at constant lift coefficient: $\Delta\alpha_g$. w aspect ratio configurations, this quantity follows

$$\Delta\alpha_g = \qquad (5)$$

$$-18.24(C_{L_{wf}})\sigma_g/A + r_g T_g (C_{L_{wf}})^2/57.3(C_{L_\alpha})_{wf} -$$

where: $C_{L_{wf}}$ is the wing-fuselage lift coefficient free air. This follows from Section or from Section 8.1.6 (depending on tip state) by setting $S_h = S_c = 0$.

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Figure 8.74 Factor Due to Image Bound Vortex

Figure 8.75 Factor Due to Finite Span

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Figure 8.76 Ground Effect on Lift Due to Flaps

Figure 8.77 Factor Due to Vertical Induced Velocity

σ_g is found from Figure 8.77.

A is the wing aspect ratio.

r_g is found from Figure 8.75.

T_g is found from Figure 8.78.

$(C_{L_\alpha})_{wf}$ is found from Eqn.(8.43), in 1/deg.

B_g is found from Figure 8.79.

Equation (8.56) neglects the effect of wing thickness which is small except for very thick wings close to the ground.

Equation (8.56) is used to estimate $\Delta\alpha_g$ for two values of $C_{L_{wf}}$ after which the airplane C_L versus α curve is modified as indicated in Figure 8.72b. This procedure neglects the effect of the horizontal tail and the canard on airplane lift in ground effect. In Section 8.2.7 it will be seen that the effect of the horizontal tail and the canard on airplane pitching moment in ground effect is NOT neglected.

8.1.8 Power Effects on Airplane Lift

The only power effect on lift which will be considered here is that due to propeller slipstream acting on a wing. Figure 8.80 shows the basic propeller-on-wing geometry. The effect of the propeller is to increase the dynamic pressure over the wing behind the propeller. In turn, this causes an increase in wing lift (and thus in airplane lift) which may be estimated from:

$$\Delta C_{L_w} = \sum_{i=1}^{i=n} [(S_{p_i}/S)(C_{L_w})[(2200P_{av_i})/(\bar{q}U_1\pi(D_{p_i})^2)]] \qquad (8.57)$$

where: S_{p_i} is the wing area affected by the slipstream, see Figure 8.80.

C_{L_w} is the lift coefficient at which the wing is operating. It follows from Sub-sections 8.1.3 or 8.1.4.

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Figure 8.78 Factor Due to Horizontal Induced Velocity

Figure 8.79 Factor Due to Circulation Change

P_{av_i} is the available horsepower, see Section 6.4.

U_i is the steady state speed of the airplane

D_{p_i} is the propeller diameter in ft.

A more detailed treatment of power effects on lift is considered beyond the scope of this text. Reference 9 addresses this topic.

Figure 8.80 Wing Area Affected by Propeller Slipstream

8.2. PREDICTION OF PITCHING MOMENT COEFFICIENT VERSUS LIFT COEFFICIENT

In this section methods for predicting the variation of pitching moment coefficient with lift coefficient will be presented as follows:

8.2.1 Airfoil pitching moment: flaps up
8.2.2 Airfoil pitching moment: flaps down
8.2.3 Wing pitching moment: flaps up
8.2.4 Wing pitching moment: flaps down
8.2.5 Airplane pitching moment: flaps up
8.2.6 Airplane pitching moment: flaps down
8.2.7 Airplane pitching moment in ground effect
8.2.8 Power effects on airplane pitching moment

8.2.1 Airfoil Pitching Moment: Flaps Up

Figure 8.81 shows the relationship between airfoil pitching moment coefficient and airfoil lift coefficient which must be determined with the methods presented in this Section. Key quantities needed in the construction of the airfoil c_m versus c_l curve are listed, with an indication of where methods for their estimation are found. Note that c_m has meaning only when taken about a known reference point. The location of this reference point is defined in 8.2.1.3.

8.2.1.1 Airfoil zero-lift pitching moment coefficient: c_{m_o}

IMPORTANT NOTES:

1. In Chapter 4 the subscript 'o' indicates a drag coefficient at zero lift coefficient.

2. In Section 8.1 the subscript 'o' indicates a lift coefficient at zero angle of attack.

3. In this section (8.2) the subscript 'o' indicates a pitching moment coefficient at zero lift coefficient.

Reference 49 contains a large data base for finding c_{m_o} for many airfoils. This data base applies to the so-called NACA airfoils.

Table 8.1 (p.216-217) provides a summary of airfoil

Figure 8.81 Airfoil Pitching Moment Coefficient Versus Lift Coefficient

Figure 8.82 Location of Airfoil Aerodynamic Center

Figure 8.83 Airfoil Center of Pressure: Origin and Location

data from which c_{m_o} may be determined. Where possible, actual airfoil data should be used. Such data may be obtained either from windtunnel data or from theoretical predictions obtained with a modern airfoil code.

Note from Table 8.1 that these c_{m_o} values are zero for symmetrical sections and negative (i.e. nose-down!) for positively cambered airfoils. This will have important consequences to airplane trim!

8.2.1.2 Airfoil aerodynamic center: x_{ac} and airfoil center of pressure: x_{cp}

Definition: The aerodynamic center is that point about which the variation of pitching moment coefficient with angle of attack is zero.

Figure 8.82 shows how the aerodynamic center for an airfoil is located.

Note that this concept implies also that the variation of pitching moment coefficient with lift coefficient is zero, when:

$$\alpha < \alpha^* \tag{8.58}$$

Definition: The center of pressure is that point where the resultant force caused by the pressure distribution acts.

Figure 8.83 shows how the center of pressure arises from the airfoil pressure distribution and how the center of pressure is located.

Evidently, there are two methods to represent forces and moments acting on an airfoil. Figure 8.84 shows both methods. The following relationship exists between the two methods:

$$x_{cp} = x_{ac} - (c_{m_{ac}})c/c_l \tag{8.59}$$

This can also be written as:

$$c_{m_{ac}} = -c_l(x_{cp} - x_{ac})/c \tag{8.60}$$

Note from this that:

$$c_{m_{ac}} = c_{m_o} \tag{8.61}$$

a. Forces at the Center of Pressure

b. Forces at the Aerodynamic Center

Figure 8.84 Methods for Resolving Airfoil Forces and Moments

Figure 8.85 Typical Airfoil Lift and Pitching Moment Behavior at Subsonic Speeds

Subsonic:

Figure 8.85 shows typical variations of airfoil pitching moment coefficient with airfoil lift coefficient and with airfoil angle of attack in the subsonic flow range. Such data may be obtained from Reference 49.

Modern airfoil theory is capable of estimating airfoil lift, drag and pitching moment data very accurately. In the absence of experimental data, such theoretically obtained airfoil data may be used.

Table 8.1 (p.216-217) lists values for x_{ac} for several airfoils. Fig. 8.86 shows how airfoil a.c. location changes with the thickness ratio, t/c and with the trailing edge angle, ϕ_{te}.

Transonic:

Airfoil pitching moment behavior changes markedly above the so-called critical Mach number. Experimental data or data obtained from modern airfoil codes should be used to predict the airfoil pitching moment behavior above M_{crit}. The critical Mach number of an airfoil at any lift coefficient may be estimated from:

For NACA airfoils:

$$M_{crit} = 0.86 - 0.1c_l - t/c \qquad (8.62)$$

For supercritical airfoils:

$$M_{crit} = 0.91 - 0.1c_l - t/c \qquad (8.63)$$

Supersonic:

Theoretically, for very thin airfoils at low lift coefficients, the center of pressure moves toward the 50 percent chord point. Figure 8.87 indicates how the centers of pressure vary with thickness ratio, t/c and with angle of attack for symmetrical airfoils. For symmetrical airfoils it is noted that: $x_{cp} = x_{ac}$!

Note: this aft shift of the center of pressure in the supersonic speed range has very significant implications to airplane trim and drag in this speed range!

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Figure 8.86 Effect of Trailing Edge Angle and Thickness Ratio on Airfoil Aerodynamic Center Location at Subsonic Speeds

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Figure 8.87 Effect of Thickness Ratio and Angle of Attack on Airfoil Center of Pressure Location at Supersonic Speeds

8.2.1.3 Airfoil pitching moment variation with lift coefficient: dc_m/dc_l

To determine the variation of airfoil pitching moment coefficient with lift coefficient it is necessary to first select a so-called reference center: Figure 8.88 shows such a reference center. The pitching moment coefficient is defined relative to this reference center:

$$c_m = c_{m_0} + c_l(x_{ref} - x_{ac})/c \qquad (8.64)$$

The variation of pitching moment coefficient with lift coefficient follows by differentiation:

$$dc_m/dc_l = (x_{ref} - x_{ac})/c = \bar{x}_{ref} - \bar{x}_{ac} \qquad (8.65)$$

8.2.1.4 Airfoil linear range for pitching moment: c_l^*

The range of lift coefficients for which the variation of airfoil pitching moment coefficient with lift coefficient is linear follows from:

$$c_l^* = (c_{l_\alpha})(\alpha^* - \alpha_{o_l}) \qquad (8.66)$$

where all quantities are defined in 8.1.1.

8.2.1.5 Construction of airfoil pitching moment curve: flaps up

All ingredients necessary to construct the airfoil pitching moment curve about a reference center x_{ref} are now available. Figure 8.89 shows how this may be done in a step-by-step manner. Whether or not the pitching moment coefficient breaks in a stable or in an unstable manner depends on the separation behavior of the airfoil. Reference 49 contains experimental data which defines the pitching moment 'break' behavior of airfoils. Many (in fact, most) airfoils have stable pitching moment breaks.

Figure 8.88 Example Location of Reference Center

Figure 8.89 Construction of Airfoil Pitching Moment Versus Lift Curve

8.2.2 Airfoil Pitching Moment: Flaps Down

Figure 8.90 shows the comparison between flaps-up and flaps-down pitching moment characteristics for an airfoil. Key quantities which are required in the estimation of flap-down behavior are listed with an indication of where methods for their estimation may be found.

Note that the assumption is made that the dc_m/dc_l slope of the airfoil with the flaps down is the same as that for the airfoil with the flaps up. This is reasonable because in the linear angle of attack range, a change in airfoil camber (read flap deflection) has no effect on the c_m-c_l slope.

8.2.2.1 Airfoil pitching moment increment due to flaps: Δc_m

The airfoil incremental pitching moment coefficient due to flaps, Δc_m depends on the type of flaps used.

Methods are presented for trailing edge and for leading edge flaps.

A. Trailing Edge Flaps

For Split, Slotted and Fowler Flaps:

The following method applies to all types of trailing edge flaps, except for plain flaps:

$$\Delta c_m = \Delta c_l \{x_{ref}/c - (x_{cp}/c')(c'/c)\} \qquad (8.67)$$

where: Δc_l is the lift increment due to flaps as found from 8.1.2.

x_{ref} is defined in Figures 8.88 and 8.91.

x_{cp}/c' is found from Figure 8.91.

c' is defined in Figures 8.18 and 8.19. For a split flap and for a plain flap: $c' = c$.

Note: since Δc_l due to flaps as used in Eqn. (8.64) depends on the flap type and on the flap deflection, the latter is inherently accounted for in this method.

Figure 8.90 Airfoil Pitching Moment Coefficient Versus Lift Coefficient with the Flaps Down

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Figure 8.91 Location of Center of Pressure due to Incremental Flap Load

For Plain Flaps:

Δc_m is found from Figure 8.92. Observe that this is a pure moment coefficient: its lift dependence is expressed through the plain flap deflection.

B. Leading Edge Flaps:

$$\Delta c_{m_{le}} = (c_{m_{\delta le}})'(c'/c)^2 \delta_{f_{le}} + \\ + [(x_{ref}/c) + (c' - c)/c]\Delta c_{l_{le}} + \\ + c_m[(c'/c)^2 - 1] + 0.75 c_l (c'/c)[(c'/c) - 1] \quad (8.68)$$

where: $(c_{m_{\delta le}})'$ is found from Figure 8.93.

c'/c is defined in Figures 8.27, 8.28 and 8.93.

δ_{fle} is defined in Figures 8.25, 8.27 and 8.28, where it is called δ_f (in deg).

$\Delta c_{l_{le}}$ is the lift coefficient increment due to leading edge flaps as found from 8.1.2.

c_m is the airfoil pitching moment coefficient with the flaps-up, taken about x_{ref} and as found from 8.2.1.

c_l is the airfoil lift coefficient with the flaps-up as determined from the flight condition.

8.2.2.2 Construction of the flaps-down airfoil pitching moment curve

Figure 8.94 shows how the flaps-down pitching moment curve is obtained from the corresponding flaps-up curve.

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Figure 8.92 Incremental Pitching Moment Coefficient due to Plain Flaps

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Figure 8.93 Pitching Moment Effectiveness due to Leading Edge Flaps

Figure 8.94 Construction of Flaps Down Airfoil Pitching Moment Versus Lift Curve

Figure 8.95 Wing Pitching Moment Coefficient Versus Lift Coefficient Curve

8.2.3 Wing Pitching Moment: Flaps-Up

Figure 8.95 shows the flaps-up pitching moment behavior which needs to be predicted with the methods of this section. Key quantities which are required in the estimation of flaps-up behavior are listed with an indication of where methods for their estimation may be found.

A key parameter in estimating wing pitching moment behavior is the mean geometric chord of the wing (mgc), $\bar{c}$ and its location on the planform. Figure 8.96 defines these quantities for two types of straight tapered wing as well as for a more general wing.

Since pitching moments are defined relative to a reference point, it is mandatory to locate such a reference point (also called moment reference center) on the wing planform. Figures 8.97 shows two methods used do this.

Observe that:

$$n_{ref} = x_{ref} + n_{mgc} \tag{8.69}$$

The method labelled as 2 is the preferred method.

8.2.3.1 Wing zero-lift pitching moment coefficient: $C_{m_{o_w}}$

Subsonic:

$$C_{m_{o_w}} = [(A\cos^2\Lambda_{c/4})/(A + 2\cos\Lambda_{c/4})](c_{m_{o_r}} + c_{m_{o_t}})/2 + (\Delta C_{m_o}/\epsilon_t)\epsilon_t \tag{8.70}$$

where: $c_{m_{o_r}}$ and $c_{m_{o_t}}$ are the zero-lift pitching moment coefficients of the root and tip airfoils respectively. These follow from 8.2.1.1.

$\Delta C_{m_o}/\epsilon_t$ is found from Figure 8.98.

Note: This method applies to conventional straight tapered wings with sweep angles below 45 degrees and aspect ratios above 2.5. For other wing types, Reference 9 should be consulted.

Transonic:

Up to the critical Mach number, use:

Figure 8.96 Methods for Determining the Wing Mean Geometric Chord

Figure 8.97 Methods for Locating a Wing Reference Point and the Wing Aerodynamic Center

COPIED FROM:
REF. 9

Figure 8.98 Effect of Linear Twist on Wing Zero-lift Pitching Moment Coefficient

$$C_{m_{o_w}}\Big|_{\text{at } M} = C_{m_{o_w}}\Big|_{\text{at } M=0} \{(C_{m_o})_M/(C_{m_o})_{M=0}\} \qquad (8.71)$$

where: $\{(C_{m_o})_M/(C_{m_o})_{M=0}\}$ is given in Figure 8.99.

$C_{m_{o_w}}\Big|_{\text{at } M=0}$ is found from Eqn.(8.70).

Supersonic:

For supersonic Mach numbers it is suggested to use experimental data. Since supersonic wings frequently have little camber, their $C_{m_{o_w}}$ values tend to be small.

8.2.3.2 Slope of the wing pitching moment curve: $(dC_m/dC_L)_w$

The method which follows applies only to straight, tapered planforms. For very highly swept wings and for cranked wings the method of Reference 9 should be used.

Subsonic:

The subsonic slope of the wing pitching moment curve may be found from:

$$(dC_m/dC_L)_w = \{(n_{ref} - n_{ac})/c_r\}(c_r/\bar{c}) \qquad (8.72)$$

where: n_{ref} is the location of the moment reference center relative to the wing apex: see Fig.8.97a.

n_{ac} is the location of the wing aerodynamic center relative to the wing apex: see Figure 8.97a.

For wings of aspect ratios above 5 and sweep angles below 35 degrees it is usually acceptable to use:

$$n_{ac} = n_{mgc} + 0.\bar{2}5c \qquad (8.73)$$

For other wings, Figure 8.100 should be used.

Transonic:

For transonic Mach numbers, see Reference 9.

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Figure 8.99 Effect of Mach Number on Wing Zero-lift Pitching Moment Coefficient

Figure 8.100 Effect of Aspect Ratio, Sweep Angle and Taper Ratio on Wing Aerodynamic Center

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Figure 8.100 (Cont'd) Effect of Aspect ratio, Sweep Angle and Taper Ratio on Wing Aerodynamic Center

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Figure 8.100 (Cont'd) Effect of Aspect Ratio, Sweep Angle and Taper Ratio on Wing Aerodynamic Center

Figure 8.101 Wing Alone Pitch Break Stability Boundary

Supersonic:

For supersonic Mach numbers, see Reference 9.

8.2.3.3 Prediction of stable or unstable pitch break

Whether a wing has stable or unstable pitch break behavior depends on the type of airfoil(s) used as well as on the wing planform. Figure 8.101 may be used as an initial guide in the prediction of pitch break behavior. The reader should refer to Part III, p.266-269 for a more detailed discussion of the pitch break phenomenon. As pointed out in Part IV (p.269) wings with highly swept strakes should be expected to have severely unstable pitch breaks.

8.2.3.4 Construction of the wing pitching moment curve: flaps-up

All ingredients needed for the construction of the flaps-up wing pitching moment curve are now available. Figure 8.102 shows how this can be done in a stepwise manner.

Figure 8.102 Construction of Flaps Up Wing Pitching Moment Versus Lift Curve

8.2.4 Wing Pitching Moment: Flaps-Down

Figure 8.103 shows the flaps-down pitching moment behavior which must be predicted with the methods of this section. Key quantities which are required in the estimation of flaps-down behavior are listed with an indication of where methods for their estimation may be found.

The method given here does not apply to wings with sweep angles in excess of 45 degrees. The method is also not applicable to wings with aspect ratios: $A < 2.5$.

8.2.4.1 Wing pitching moment increment due to flaps: ΔC_{m_w}

The wing incremental pitching moment coefficient due to flaps depends on the type of flaps used. Methods are presented for trailing edge flaps and for leading edge flaps.

A. Trailing Edge Flaps

In the low angle of attack range, the incremental wing pitching moment coefficient due to trailing edge flaps (taken relative to x_{ref}) may be found from:

$$\Delta C_{m_w} = \tag{8.74}$$

$$(\bar{x}_{ref} - 0.25)(C_{L_w})_\delta + K_\Lambda(A/1.5)(\Delta C_{L_{ref_w}})\tan\Lambda_{c/4} +$$

$$+ K_p[(\Delta C'_m/\Delta C_{L_{ref_w}})\Delta C_{L_{ref_w}}(c'/c)^2] +$$

$$- K_p[0.25C_{L_w}\{(c'/c)^2 - (c'/c)\}] + K_p C_{m_w}\{(c'/c)^2 - 1\}$$

where: x_{ref} is defined in Figure 8.97.

$(C_{L_w})_\delta$ is the wing lift coefficient with the flaps down. It follows from 8.1.4.

C_{L_w} is the flaps-up wing lift coefficient as determined from 8.1.3.

Figure 8.103 Wing Pitching Moment Versus Lift Curve with the Flaps Down

Figure 8.104 Method to Account for Partial Span Flaps

$\Delta C_{L_{ref_w}}$ is the lift increment due to flaps for a reference wing defined by the following geometry:

*full span flap *A = 6 *$\Lambda_{c/2}$ = 0

This quantity can be determined with the methods of 8.1.4.

K_p is the flap span factor which is obtained as shown in Fig.8.104 with the data of Fig.8.105.

c'/c is the ratio of the flaps extended wing chord to that with the flaps retracted. Fig.8.106 shows typical definitions.

$(\Delta C'_m/\Delta C_{L_{ref_w}})$ is obtained from Figure 8.106.

K_Λ is a conversion factor which accounts for a partial-span flap on a swept wing. It is obtained from Fig.8.107 using the procedure of Figure 8.104.

A is the wing aspect ratio

$\Lambda_{c/4}$ is the wing quarter chord sweep angle.

C_{m_w} is the wing pitching moment coefficient with the flaps retracted as obtained from 8.2.3.

B. Leading Edge Flaps

In the low angle of attack range, the incremental wing pitching moment coefficient due to leading edge flaps (taken relative to $\bar{m}_{ref}$) may be found from:

$$\Delta C_{m_w} = \tag{8.75}$$

$$\{(c_{m_{\delta le}})'(c'/c) + (\bar{m}_{ref} - \bar{m}_{le})c_{l_\delta}\}(S_{w_f}/S)\delta_f +$$

$$+ [C_{m_w}\{(\bar{c}'/c)^2 - 1\} + 0.75C_{L_w}\{(\bar{c}'/c)(\bar{c}'-c)/c)\}](b_{lef}/b)$$

where: $(c_{m_{\delta le}})'$ is found from Figure 8.93.

$\bar{c}'$ is the mgc of that wing segment affected by the leading edge devices as shown in Figure 8.108.

Figure 8.105 Partial Flap Span Factor

Figure 8.106 Effect of Lift on Pitching Moment for the Reference Wing

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Figure 8.107 Conversion Factor which Accounts for Partial Span Flaps on a Swept Wing

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Figure 8.107 (Cont'd) Conversion Factor which Accounts for Partial Span Flaps on a Swept Wing

Figure 8.108 Definition of $\bar{c}'$

Figure 8.109 Construction of the Flaps Down Wing Pitching Moment Versus Lift Curve

c is the wing chord at $\bar{c}'$, see Figure 8.108.

$\bar{n}_{ref}$ and $\bar{n}_{le}$ are defined in Figure 8.108.

b_{lef} is defined in Figure 8.108.

S_{wf} is the flapped wing area: see Figure 8.108.

C_{m_w} is the wing pitching moment coefficient with the flaps retracted as obtained from 8.2.3.

C_{L_w} is the wing lift coefficient with flaps-up.

8.2.4.2 Slope of the wing pitching moment curve, flaps-down: $(dC_m/dC_L)_{w\delta}$

It may be assumed, that the slope of the wing pitching moment curve with the flaps-down is identical to that with the flaps-up: see 8.2.3.2.

8.2.4.3 Prediction of stable or unstable pitch break: flaps-down

It will be assumed that the pitch break behavior of the wing with the flaps down is similar to that with the flaps up: see 8.2.3.3.

8.2.4.4 Construction of the wing pitching moment curve: flaps-down

All ingredients necessary to construct the flaps-down wing pitching moment curve are now available. Figure 8.109 shows how this may be done in a step-by-step manner.

8.2.5 Airplane Pitching Moment: Flaps Up

Figure 8.110 shows the relationship between airplane pitching moment coefficient and airplane lift coefficient which is to be determined with the methods presented in this section. Key quantities needed in the construction of airplane C_m versus C_L curve are listed, with an indication of where methods for their estimation are found.

The assumption will be made, that an airplane can be considered to consist of three components:

a) wing + fuselage b) horizontal tail c) canard

Figure 8.60 shows the relative arrangement of these three major components. If an airplane is equipped with pylon mounted nacelles (such as the B-727 and the DC-9), the pylon + nacelle combination should be 'counted' as an additional horizontal tail. Figure 8.61 indicates the 'equivalent' geometries which should then be used.

Figure 8.62 shows a number of important geometric parameters which are used in the calculation of overall airplane pitching moment characteristics.

IMPORTANT CONSIDERATIONS:

In this chapter the following incidence angles are accounted for:

Canard: i_c Wing: i_w Horizontal Tail: i_h

These incidence angle are defined in Figure 8.60. In this sub-section, these incidence angles are assumed to be CONSTANT. The effect of trim requirements is considered in Section 8.3.

A horizontal tail and a canard may be equipped with a trailing edge control surface:

Horizontal tail with elevator, deflection: δ_e

Canard with canardvator, deflection: δ_c

In this sub-section, the assumption will be made that all control surface deflection angles are ZERO. The effect of trim requirements which would cause these control surface deflections to be non-zero is considered in Section 8.3.

Figure 8.110 Airplane Pitching Moment Versus Lift Curve with the Flaps Up

Figure 8.111 Effect of Fuselage Slenderness on the Apparent Mass Factor

8.2.5.1 Airplane zero-lift pitching moment coefficient: C_{m_o}

Subsonic:

The airplane zero-lift pitching moment coefficient may be estimated from:

$$C_{m_o} = C_{m_{o_{wf}}} + C_{m_{o_c}} + C_{m_{o_h}} \tag{8.76}$$

where: $C_{m_{o_{wf}}}$ is the zero-lift pitching moment coefficient of the wing-fuselage combination. It may be computed from:

$$C_{m_{o_{wf}}} = \{(C_{m_{o_w}}) + (C_{m_{o_f}})\}\{(C_{m_o})_M/(C_{m_o})_{M=0}\} \tag{8.77}$$

where: $C_{m_{o_w}}$ is found from Eqn.(8.70)

$$C_{m_{o_f}} = \tag{8.78}$$

$$\{(k_2 - k_1)/36.5\bar{Sc}\}[\mathrm{Sum}_{i=1}^{i=13}\{(w_{f_i}{}^2)(i_w + \alpha_{o_{L_w}} + i_{cl_f})\Delta x_i\}]$$

where: $(k_2 - k_1)$ is found from Figure 8.111.

w_{f_i} is the average width of a fuselage segment as shown in Figure 8.112.

Δx_i is the length of a fuselage segment as shown in Figure 8.112.

i_w is the wing incidence angle as defined in Figure 8.60.

$\alpha_{o_{L_w}}$ is found from Eqn.(8.21)

i_{cl_f} is the incidence angle of the fuselage camber line RELATIVE to the fuselage reference plane (FRP) at the center of each fuselage increment. The sign convention for this angle is illustrated in Figure 8.112. Note that this angle is NEGATIVE for down-camber at the fuselage nose as well as for up-camber at the fuselage tailcone.

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Figure 8.112 Fuselage Segmentation and Fuselage Camber

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Figure 8.113 Definition of Maximum Fuselage Width for Several Fuselage Cross Sections

The fuselage camber line is determined in the fuselage sideview by the so-called maximum width locus which in turn arises from the fuselage cross section distribution. It is shown in Figure 8.113 how the maximum fuselage width is defined for three types of fuselage cross sections.

$C_{m_{o_c}}$ and $C_{m_{o_h}}$ are the zero-lift pitching moment coefficient due to the canard and due to the horizontal tail respectively. Their sum may be computed from:

$$C_{m_{o_c}} + C_{m_{o_h}} = \tag{8.79}$$

$$(\bar{x}_{ac_c} + \bar{x}_{ref})C_{L_{o_c}} - (\bar{x}_{ac_h} - \bar{x}_{ref})C_{L_{o_h}}$$

where: x_{ac_c} and x_{ac_h} are defined in Figure 8.114. POSITIVE as shown in that Figure! Note that this is an inconsistent sign convention!!

x_{ref} is the reference (or c.g.) location about which the airplane pitching moment is to be determined: see Figures 8.114 and 8.97b. <u>NOTE:</u> this quantity is POSITIVE when behind the leading edge of the wing m.g.c. and NEGATIVE when forward of the leading edge of the wing m.g.c.!

Observe that:

$$\bar{x}_{ref} = (x_{ref})/c \tag{8.80a}$$

$$\bar{x}_{ac_c} = (x_{ac_c})/c \tag{8.80b}$$

$$\bar{x}_{ac_h} = (x_{ac_h})/c \tag{8.80c}$$

<u>NOTE:</u> for the entire airplane, x_{ref} is normally selected to be a c.g. location in between the most forward and the most aft center of gravity location as found from Chapter 10 in Part II.

$C_{L_{o_c}}$ and $C_{L_{o_h}}$ are the lift coefficients at

zero angle of attack of the canard and of the horizontal tail respectively. These quantities are the third and second term respectively in Eqn.(8.32).

Transonic:

For Mach numbers between 0.6 and 0.9 the subsonic method may be used. Above M=0.9 experimental data should be employed.

Supersonic:

Experimental methods should be used. For slender configurations such as fighter aircraft, Ref.9 presents a reasonable procedure for estimating C_{m_o}.

Figure 8.114 Definition of Aerodynamic Center Location Parameters and Moment Reference Center

8.2.5.2 Airplane pitching moment variation with lift coefficient: dC_m/dC_L

Subsonic:

The variation of airplane pitching moment coefficient with lift coefficient may be estimated from:

$$dC_m/dC_L = \bar{x}_{ref} - \bar{x}_{ac_A} \tag{8.81}$$

where:

$\bar{x}_{ref}$ is the location of the moment reference center in fractions of the m.g.c. It is defined in Figures 8.97b and 8.114.

NOTE: for the entire airplane, x_{ref} is normally selected to be a c.g. location in between the most forward and the most aft center of gravity location as determined from Chapter 10 in Part II.

$\bar{x}_{ac_A}$ is the location of the airplane aerodynamic center in fractions of the m.g.c. Fig. 8.114 defines its location on the m.g.c. The following equation may be used to compute it:

$$\bar{x}_{ac_A} = \tag{8.82}$$

$$[(\bar{x}_{ac_{wf}})C_{L_{\alpha_{wf}}} + (\eta_h C_{L_{\alpha_h}}(1 - d\epsilon/d\alpha)(S_h/S)\bar{x}_{ac_h} + \eta_c C_{L_{\alpha_c}}(1 + d\epsilon_c/d\alpha)(S_c/S)\bar{x}_{ac_c})]/C_{L_\alpha}$$

where: $\bar{x}_{ac_{wf}} = \bar{x}_{ac_w} + \Delta\bar{x}_{ac_f}$, (8.83)

both being given as fractions of the m.g.c. Note that in terms of their actual lengths:

$$x_{ac_w} = n_{ac} - n_{mgc}, \tag{8.84}$$

both of which are defined in Figure 8.97a.

n_{ac} may be determined from Figure 8.100 or from Eqn. (8.73), depending on the wing type used.

$\Delta\bar{x}_{ac_f}$ is the shift in aerodynamic center caused by adding the fuselage to the wing: it is determined with the method of 8.2.5.3.

η_h and η_c may be found from 8.1.5.2.

$C_{L_{\alpha_{wf}}}$ is found from Eqn.(8.43)

$C_{L_{\alpha_h}}$ and $C_{L_{\alpha_c}}$ are found from Eqn.(8.22) with appriate substitution of parameters.

$d\epsilon/d\alpha$ and $d\epsilon_c/d\alpha$ are found from 8.1.5.3.

C_{L_α} is the total airplane liftcurve slope which follows from Eqn.(8.42).

Transonic:

In the transonic speed range, the airplane undergoes an aft shift in aerodynamic center location. In preliminary design it may be assumed that this aft shift is the same as that due to the wing alone. The latter may be determined from Figure 8.100. Note that this figure applies only to straight tapered wings. For other wings, use Ref.9.

Supersonic:

In the supersonic speed range, the airplane undergoes an aft shift in aerodynamic center location. In preliminary design it may be assumed that this aft shift is the same as that due to the wing alone. The latter can be found from Figure 8.100. Note that this figure applies only to straight tapered wings. For other wings, use Ref.9.

8.2.5.3 Calculation of the aerodynamic center shift due to the fuselage: $\Delta\bar{x}_{ac_f}$

The following method may be used up to a Mach number of 0.9. Beyond this Mach number, the fuselage induced shift in a.c. location is assumed constant throughout the transonic and supersonic speed range.

The method presented here is due to Munk, as reported in Reference 54:

$$\Delta x_{ac_f} = -(dM/d\alpha)/(\bar{q}S\bar{c}C_{L_{\alpha_w}}) \qquad (8.85)$$

where: $C_{L_{\alpha_w}}$ is found from Eqn.(8.22) in 1/deg.

$$(dM/d\alpha) = \qquad (8.86)$$

$$(\bar{q}/36.5)(C_{L_{\alpha_w}}/0.08)\left[\mathrm{Sum}_{i=1}^{i=13}((w_{f_i})^2(d\bar{\epsilon}/d\alpha)_i\Delta x_i)\right]$$

where: w_{f_i} and Δx_i are defined in 8.2.5.1.

$C_{L_{\alpha_w}}$ is found from Eqn.(8.22) in 1/deg.

$(d\bar{\epsilon}/d\alpha)_i$ takes on values which depend on the fuselage segment under consideration:

for i=1,2,3,4, use curve 1 in Figure 8.115

for i=5 use curve 2 in Figure 8.115.

for i=6,7,8,9,10,11,12,13, use:

$$(d\bar{\epsilon}/d\alpha)_i = (x_i/x_h)(1 - d\epsilon/d\alpha) \qquad (8.87)$$

where: x_i and x_h are defined in Fig.8.116,

and $d\epsilon/d\alpha$ is found from Eqn.(8.45), but with: $h_h = 0$.

<u>IMPORTANT NOTE:</u> if the airplane has wing mounted nacelles forward of the wing or wing mounted stores forward of the wing, this method should be used to compute the additional a.c. shift. In such cases only segments 1-5 need to be accounted for.

8.2.5.4 Prediction of stable or unstable pitch break

Whether or not an entire airplane has an unstable or a stable pitch break depends on the overall configuration as well as on the sequence with which separated wakes arrive from the wing at aft mounted pylon/nacelle combinations and/or at horizontal tails. The reader should con-

Figure 8.115 Effect of Fuselage (or Nacelle) Segment Location on Upwash Gradient

Figure 8.116 Layout for Computing Fuselage and (or) Nacelle Contribution to Airplane Aerodynamic Center Location

sult the discussion of pitch break behavior found in Subsection 5.1.4 of Part III, pages 263-272. As a result of applying this material to a given design, a determination of stable or unstable pitch break behavior can be made.

8.2.5.5 Construction of airplane pitching moment coefficient versus lift coefficient

All ingredients needed to construct the overall airplane C_m versus C_L curve are now available. A step-by-step procedure for doing this is given in Fig. 8.117.

Figure 8.117 Construction of the Flaps Up Airplane Pitching Moment Versus Lift Curve

8.2.6 Airplane Pitching Moment: Flaps Down

Figure 8.118 shows the relationship between airplane pitching moment coefficient and airplane lift coefficient in the flaps down configuration. This relationship will be determined with the methods presented in this section. Key quantities needed in the construction of the airplane C_m versus C_L curve are listed, with an indication of where methods for their estimation are found.

8.2.6.1 Airplane pitching moment coefficient increment due to flaps: ΔC_m

The airplane pitching moment increment due to flaps may be estimated from:

$$\Delta C_m = \Delta C_{m_w} + \Delta C_{m_c} \eta_c (S_c \bar{c}_c / S\bar{c}) + \Delta C_{L_c} (\bar{x}_{ac_c} + \bar{x}_{ref}) + C_{L_{\alpha_h}} \eta_h (S_h/S)(\bar{x}_{ac_h} - \bar{x}_{ref}) \Delta\epsilon_f \qquad (8.88)$$

where: ΔC_{m_w} accounts for the pitching moment change due to wing flaps (leading edge and/or trailing edge) and is found from 8.2.4.1

ΔC_{m_c} accounts for the pitching moment change due to canard flaps (usually trailing edge only) and is found from 8.2.4.1 with appropriate substitution of canard parameters for wing parameters. If the canard has no flaps, this quantity is equal to zero.

ΔC_{L_c} is found from 8.1.4.1 with appropriate substitution of canard parameters for wing parameters. The comments made on p.277 apply!

x_{ref}, x_{ac_c} and x_{ac_h} are defined in Figure 8.114.

NOTE: the reader should carefully observe the inconsistent sign convention used for these quantities!

$C_{L_{\alpha_h}}$ is found from 8.1.3.2 with appropriate substitution of horizontal tail parameters for wing parameters.

η_c and η_h are found from 8.1.5.2.

$\Delta\epsilon_f$ is found from Figure 8.70.

8.2.6.2 Slope of the airplane pitching moment curve flaps down: $(dC_m/dC_L)_\delta$

It may be assumed that the slope of the flaps down pitching moment curve is the same as that of the flaps up pitching moment curve:

$$(dC_m/dC_L)_\delta = dC_m/dC_L \qquad (8.89)$$

8.2.6.3 Prediction of stable or unstable pitch break: flaps down

It will be assumed that the pitch break behavior of the airplane with the flaps down is similar to that with the flaps up. The reader is referred to 8.2.5.4.

8.2.6.4 Construction of the airplane flaps down pitching moment curve

All items needed in the construction of the total airplane pitching moment curve with the flaps down are now available. Figure 8.119 shows a step-by-step procedure to accomplish this.

Figure 8.118 Airplane Pitching Moment Versus Lift Curve with the Flaps Down

Figure 8.119 Construction of the Flaps Down Airplane Pitching Moment Versus Lift Curve

Figure 8.120 Ground Effect on Lift Curves: Revisited

8.2.7 Airplane Pitching Moment in Ground Effect

Figure 8.120 shows a re-interpretation of the airplane lift versus angle of attack curve in and out of ground attack. Figure 8.120 is a different way of looking at Figure 8.72. In Figure 8.72 the assumption was made that the tail lift (and/or the canard lift) do not contribute significantly to airplane lift change due to ground effect. Although this is usually true to a first order of approximation, that does not mean that the pitching moments due to ground induced lift changes on these surfaces are also negligible.

It will be assumed that the aerodynamic center of the airplane does not change due to ground effect. In that case, the pitching moment increment due to ground effect on the entire airplane may be estimated from:

$$(\Delta C_m)_g = \tag{8.90}$$

$$(\Delta C_{m_{wf}})_g + (\Delta C_{m_c})_g + (\Delta C_{m_h})_g$$

where: $$(\Delta C_{m_{wf}})_g = (\bar{x}_{ref} - \bar{x}_{ac_A})(\Delta C_{L_{wf}})_g \tag{8.91}$$

with: x_{ref} and x_{ac_A} defined in Section 8.2.5: observe the inconsistent sign convention!

$(\Delta C_{L_{wf}})_g = (\Delta C_L)_g$ as shown in Figure 8.120.

$(\Delta C_L)_g$ in turn is found at two values of angle of attack, α, from Section 8.1.7 and plotted as shown in Figure 8.120. Note from Section 8.1.7 the difference between high and low aspect ratio configurations!

$$(\Delta C_{m_c})_g = (\Delta C_{L_c})_g \eta_c (\bar{x}_{ac_c} + \bar{x}_{ref}) \tag{8.92}$$

with: x_{ac_c} and x_{ref} defined in Section 8.2.5: observe the inconsistent sign convention!

η_c is found from 8.1.5.2 and:

$$(\Delta C_{L_c})_g = -C_{L_{\alpha_c}} (S_c/S)(\Delta\epsilon_c)_g \tag{8.93}$$

where: $C_{L_{\alpha_c}}$ is found from 8.1.3.2 with appropriate substitution of canard parameters for wing parameters.

$(\Delta\epsilon_c)_g$, the ground induced change in upwash angle at the canard is found from 8.2.7.1.

$$(\Delta C_{m_h})_g = -(\Delta C_{L_h})_g \eta_h (\bar{x}_{ac_h} - \bar{x}_{ref}) \qquad (8.94)$$

with: x_{ac_h} and x_{ref} defined in Section 8.2.5: observe the inconsistent sign convention!

η_h is found from 8.1.5.2 and:

$$(\Delta C_{L_h})_g = -C_{L_{\alpha_h}} (S_h/S)(\Delta\epsilon)_g \qquad (8.95)$$

where: $C_{L_{\alpha_h}}$ is found from 8.1.3.2 with appropriate substitution of horizontal tail parameters for wing parameters.

$(\Delta\epsilon)_g$ the ground induced change in tail downwash angle is found from 8.2.7.1.

Equation (8.90) is used to find ΔC_m for two angles of attack. The C_m-C_L curve of the airplane in ground effect can now be replotted as shown in Figure 8.121. Note that this in effect causes a slight change in airplane stability due to ground effect despite the assumption of constant a.c. location which was made in Eqn.(8.91).

8.2.7.1 Ground effect on downwash and on upwash

Ground Effect on Downwash

The decrease in tail downwash due to ground effect may be computed from:

$$(\Delta\epsilon)_g = \qquad (8.96)$$
$$\epsilon[(b_{eff}^2 + 4(H_h - H_w)^2)/(b_{eff}^2 + 4(H_h + H_w)^2)]$$

$$\text{where: } \epsilon = \alpha(d\epsilon/d\alpha) \qquad (8.97)$$

Figure 8.121 Construction of Airplane Pitching Moment Versus Lift Curve in Ground Effect

Figure 8.122 Definition of Canard, Wing and Tail Height Above the Ground

with $d\epsilon/d\alpha$ found from Eqn. (8.45).

H_h is the height of the horizontal tail $\bar{c}_h/4$ above the ground: see Figure 8.122.

H_w is the height of the wing $\bar{c}/4$ above the ground: see Figure 8.122.

$$b_{eff} = \tag{8.98a}$$

$$(C_{L_{wf}} + \Delta C_L)/\{(C_{L_{wf}}/b'_w) + (\Delta C_L)/b'_f\}$$

where: $C_{L_{wf}}$ is the wing-fuselage lift coefficient flaps up, out of ground effect as found from 8.1.5.

ΔC_L is the lift increment due to flaps out of ground effect as found from 8.1.6.

$$b'_w = b(b'_w/b) \tag{8.98b}$$

with: (b'_w/b) given in Figure 8.123.

$$b'_f = b(b'_f/b'_w)(b'_w/b) \tag{8.98c}$$

with: (b'_f/b'_w) given in Figure 8.124.

Ground Effect on Upwash

The decrease in canard upwash due to ground effect may be computed from:

$$(\Delta\epsilon_c)_g = \epsilon_c\{(H_c - H_w)^2/(H_c + H_w)^2\} \tag{8.99}$$

where: $\epsilon_c = (\alpha + i_c)(d\epsilon_c/d\alpha)$ (8.100)

with: α being the airplane angle of attack and

i_c being the canard incidence angle, while

$d\epsilon_c/d\alpha$ is found from Figure 8.67

H_c is the height of the canard $\bar{c}_c/4$ above the ground: see Figure 8.122.

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Figure 8.123 Effective Wing Span Close to the Ground

Figure 8.124 Effective Flap Span Close to the Ground

Figure 8.125 Power Effects on Airplane Pitching Moment

8.2.8 Power Effects on Airplane Pitching Moment

For a detailed treatment of power effects on pitching moment the reader should consult Reference 9.

Figure 8.125 shows that two types of power effect on the airplane C_m-C_L curve must be accounted for:

1. A shift in pitching moment at zero lift coefficient: ΔC_{m_T}

   This type of power effect is caused by thrustline offset and/or by the effect of propeller slipstream.

2. A change in slope: $\Delta(dC_m/dC_L)_T$

   The change in slope must be interpreted as a change in apparent static longitudinal stability. This type of power effect is caused by thrustline offset and/or by propeller/inlet normal forces.

These effects are accounted for as follows:

8.2.8.1 Power effect on pitching moment at zero lift coefficient

8.2.8.2 Power effect on longitudinal stability

8.2.8.1 Power effect on pitching moment at zero lift coefficient: ΔC_{m_T}

The effect of power on the pitching moment coefficient at zero lift coefficient may be estimated from:

$$\Delta C_{m_T} = \Delta C_{m_{TL}} + \Delta C_{m_{TS}} \tag{8.101}$$

where: $\Delta C_{m_{TL}}$ is the pitching moment increment due to thrustline offset

$\Delta C_{m_{TS}}$ is the pitching moment increment due to propeller slipstream

Effect of Thrustline Offset: $\Delta C_{m_{TL}}$

Figure 8.126 illustrates the geometry of thrustline offset as it affects the pitching moment of an airplane.

Since multi-engine airplanes can have the engines

Figure 8.126 Geometry of Thrustline Offset

Figure 8.127 Geometric Parameters Affecting Slipstream Effect on a Wing

installed in relationsship to the moment reference point, x_{ref} in a variety of ways, a summation of the effect of individual engines will be required. The incremental pitching moment due to thrustline offset is found from:

$$\Delta C_{m_{TL}} = \underset{i=1}{\overset{i=n}{\text{Sum}}}(T_{av_i} d_{T_i} / \bar{q}S\bar{c}) \qquad (8.102)$$

where: T_{av_i} is the available installed thrust from a propeller or from a jet engine

d_{T_i} is the thrustline offset relative to x_{ref}

Note from Figure 8.126, that d_{T_i} is counted as positive if the thrust line is located beneath the reference point.

Effect of Propeller Slipstream:

There are two types of slipstream effect which can alter the pitching moment of an airplane:

1. Effect of slipstream on a canard or on a horizontal tail

2. Effect of slipstream on a wing

1. Effect of slipstream on a canard or on a horizontal tail

The effect of (propeller) slipstream on the local dynamic pressure at the canard or at the horizontal tail has already been accounted for through the dynamic pressure ratios, η_c and η_h as found from 8.1.5.2.

In all previous calculations for pitching moment, the effect of power can be left out by using $\eta_{h \text{ or } c} = 0$.

2. Effect of slipstream on a wing

The effect of propeller slipstream on a wing may be accounted for from:

$$\Delta C_{m_{TS}} = (\bar{x}_{ac_{TS}} - \bar{x}_{ref})\Delta C_{L_w} \qquad (8.103)$$

where: $x_{ac_{TS}}$ and x_{ref} are defined in Figure 8.127.

ΔC_{L_w} is found from Eqn.(8.57)

8.2.8.2 Power effect on longitudinal stability: $\Delta(dC_m/dC_L)_T$

The following method to account for power effect on stability is adapted from Reference 54:

$$\Delta(dC_m/dC_L)_T = (dC_m/dC_L)_{TL} + (dC_m/dC_L)_N \qquad (8.104)$$

where: $(dC_m/dC_L)_{TL}$ is the effect of thrustline offset on longitudinal stability

$(dC_m/dC_L)_N$ is the effect of propeller or inlet normal force on longitudinal stability

Methods for estimating these stability effects are now given for:

1. Propeller driven airplanes

2. Jet driven airplanes

1. Propeller driven airplanes:

The effect of thrustline offset on longitudinal stability for propeller driven airplanes is:

$$(dC_m/dC_L)_{TL} = \mathrm{Sum}_{i=1}^{i=n}[(dT_{c_i}/dC_L)\{2(D_{p_i})^2 d_{T_i}/S\bar{c}\}] \qquad (8.105)$$

where: D_{p_i} is the diameter of propeller i.

d_{T_i} is the thrustline offset for propeller i, see Figure 8.126.

$$dT_{c_i}/dC_L = (3/2)K_{T_i}\eta_{p_i}(C_L)^{1/2} \qquad (8.106)$$

with: η_{p_i} is the efficiency of propeller i, and:

$$K_{T_i} = \{550(SHP_{av_i})(\rho)^{1/2}\}/\{(2W/S)^{3/2}(D_{p_i})^2 \qquad (8.107)$$

Note: The effect of thrustline offset on longitudinal stability can be used by the designer to obtain minor changes in static margin by tilting the thrustline relative to the c.g. or relative to x_{ref}. Figure 8.128 shows

an exa

For Propeller Driven Airplanes:

$$C_{n_{T_\beta}} = \qquad (10.44)$$

$$\mathrm{Sum}_{i=1}^{i=n}[\{(dC_N/d\alpha)_{p_i})(0.79)(D_{p_i})^2(l_{p_i})\}/Sb]$$

where: all terms are as defined on page 342.

light air-

Figure 8.128 Example of Thrustline Tilt

Figure 8.129 Moment Arms for Normal Forces

plane to achieve such an effect.

The effect of propeller normal force on longitudinal stability may be found from:

$$(dC_m/dC_L)_N = \qquad (8.108)$$

$$\sum_{i=1}^{i=n}[\{(dC_N/d\alpha)_{p_i}(1 + d\bar{\epsilon}_{p_i}/d\alpha)(l_{p_i})(0.79)(D_{p_i})^2\}/S\bar{c}C_{L_{\alpha_W}}]$$

where: l_{p_i} is the moment arm of the propeller normal force to the reference point: see Fig. 8.129.

$d\bar{\epsilon}_{p_i}/d\alpha$ is found from Figure 8.115.

$(dC_N/d\alpha)_{p_i}$ is the change in propeller normal force coefficient with angle of attack. It may be computed from:

$$(dC_N/d\alpha)_{p_i} = \qquad (8.109)$$

$$[\{(C_{N_\alpha})_{p_i}\}_{K_{N_i}=80.7}][1 + 0.8\{(K_{N_i}/80.7) - 1\}]$$

with: $\{(C_{N_\alpha})_{p_i}\}_{K_{N_i}=80.7}$ found from Figure 8.130,

and with:

$$K_{N_i} = 262\{(w_{p_i}/R_{p_i})_{0.3R_{p_i}}\} + \qquad (8.110)$$

$$+ 262(w_{p_i}/R_{p_i})_{0.6R_{p_i}} + 135(w_{p_i}/R_{p_i})_{0.9R_{p_i}}$$

where: w_{p_i} is the propeller blade width at the radius station indicated in the subscript

R_{p_i} is the propeller blade radius $(0.5D_{p_i})$

2. Jet driven airplanes:

The effect of thrustline offset on the longitudinal stability of most jet driven airplanes is negligible. The reason is the relatively small change in thrust with small changes in speed.

The effect of inlet normal force on longitudinal stability of a jet airplane may be estimated from:

$$(dC_m/dC_L)_N = \sum_{i=1}^{i=n} \left[\frac{(0.035(\dot{m}_{a_i})(d\bar{\epsilon}/d\alpha)_i(l_{n_i}))}{(Sc\rho U_i(C_{L_{\alpha_w}}))}\right] \qquad (8.111)$$

where: $\dot{m}_{a_i}$ is the mass flow rate through engine i
This mass flow rate follows from Chapter 6.

$(d\bar{\epsilon}/d\alpha)_i$ is obtained from Figure 8.115 for inlet i

l_{n_i} is the moment arm of the inlet lip as illustrated in Figure 8.129.

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Figure 8.130 Propeller Normal Force Parameter

8.3 PREDICTION OF TRIMMED LIFT AND TRIMMED MAXIMUM LIFT COEFFICIENT

In Sections 8.1 and 8.2 methods are presented for the construction of airplane lift and pitching moment curves. The position was taken that all longitudinal control surface deflections were set at zero.

In any equilibrium flight condition the airplane must also be in moment equilibrium. The effect of pitching moment equilibrium (also known as longitudinal moment trim, or simply: trim) on airplane lift at different center of gravity locations will be accounted for in this Section. The following condition must be satisfied in a trimmed flight condition:

$$C_m = 0 \qquad (8.112)$$

The pitching moment coefficient C_m takes on values which depend on:

1. the lift coefficient at which the airplane is flying,
2. the location of the center of gravity and
3. the power setting.

Methods for computing C_m are given in Section 8.2.

The trim condition as expressed by Eqn.(8.112) is achieved by deflection of one or more control surfaces. In this Section, the following control surfaces will be accounted for:

1. Stabilizer incidence, i_h
2. Elevator deflection, δ_e
3. Canard incidence, i_c
4. Canardvator deflection, δ_c

Figure 8.131 shows typical examples of these control surfaces and their typical arrangement on an airplane. Different types of longitudinal controls may be required on certain airplanes. Examples are: elevons on delta wings (as on the F-106) or symmetrically deflected ailerons (as on the Starship). In this section only types 1-4 will be considered.

Figure 8.131 Example of Control Surface Arrangements

Figure 8.132 Effect of Control Surface Deflection on Airplane Lift

Figure 8.133 Effect of Control Surface Deflection on Airplane Pitching Moment

Deflection of a control surface (which effectively can be thought of as a plain flap) causes changes in airplane lift and in airplane pitching moment in the following ways:

1. Effect of control surface deflection on lift

2. Effect of control surface deflection on pitching moment

1. Effect of Control Surface Deflection on Lift:

$$\Delta C_{L_{ctl}} = \tag{8.113}$$

$$(C_{L_{i_h}})i_h + (C_{L_{\delta_e}})\delta_e + (C_{L_{i_c}})i_c + (C_{L_{\delta_c}})\delta_c$$

where: $C_{L_{i_h}}$, $C_{L_{\delta_e}}$, $C_{L_{i_c}}$ and $C_{L_{\delta_c}}$ are the control surface lift derivatives. They are computed with the methods given in Chapter 10.

Figure 8.132 shows how a control surface deflection affects the airplane lift versus angle of attack curve. Note the control surface sign conventions in Fig. 8.131!

2. Effect of Control Surface Deflection on Pitching Moment:

$$\Delta C_{m_{ctl}} = \tag{8.114}$$

$$(C_{m_{i_h}})i_h + (C_{m_{\delta_e}})\delta_e + (C_{m_{i_c}})i_c + (C_{m_{\delta_c}})\delta_c$$

where: $C_{m_{i_h}}$, $C_{m_{\delta_e}}$, $C_{m_{i_c}}$ and $C_{m_{\delta_c}}$ are the control power derivatives. They may be computed with the methods given in Chapter 10.

Figure 8.133 shows how a control surface deflection affects the airplane pitching moment versus lift curve. Note the control surface sign conventions in Fig. 8.131!

Next, methods are presented for determining the trimmability of airplanes. This is accomplished using so-called trim diagrams.

A method for determining trimmed lift and trimmed maximum lift capability of an airplane is presented for the following cases:

8.3.1 Stable airplane with stable pitchbreak
8.3.2 Unstable airplane with stable pitchbreak
8.3.3 Stable airplane with unstable pitchbreak
8.3.4 Unstable airplane with unstable pitchbreak

In the case of military airplanes with external stores and/or with deployed speedbrakes, significant additional pitching moments may be introduced due to these devices. Such 'drag-induced' pitching moments can be accounted for through a shift of the origin of the C_m-C_L plot. This shift is computed from:

$$\Delta C_{m_{store/speedbrake}} = (\Delta C_{D_{store/speedbrake}}) d_D/\bar{c} \quad (8.115)$$

where: $\Delta C_{D_{store/speedbrake}}$ is the drag increment due to the store or speedbrake. This increment follows from Section 4.9 or from 4.12.

d_D is the moment arm, positive if the drag increment acts ABOVE the c.g.

8.3.1 Stable Airplane with Stable Pitch Break

Figures 8.134a,b present trim diagrams for a stable airplane with a stable pitch break:

Fig. 8.134a is for a conventional (tail-aft) airplane
Fig. 8.134b is for a canard airplane.

The C_L-α and C_L-C_m curves for ZERO control surface deflections are obtained with the methods of Sections 8.1 and 8.2 respectively. The effect of control surface deflections on these curves is determined with the method indicated in Figures 8.132 and 8.133.

Note that the C_m axis is labelled: $C_{m_{x_{ref}}}$. The location of the moment reference center. x_{ref} is normally selected somewhere between the most forward and the most aft locations of the airplane center of gravity. Methods for determining the center of gravity location of any airplane are discussed in Part II, Chapter 10.

Figure 8.134a Trim Diagram: Stable, Conventional with Stable Pitch Break

Figure 8.134b Trim Diagram: Stable, Canard with Stable Pitch Break

Observe in Figures 8.134a,b that C_m=0 lines at different centers of gravity have slopes different from that of the C_m=0 line with x_{cg}=x_{ref}. These different slopes may be determined with the help of Procedure 1.

PROCEDURE 1: Determination of Slopes of C_m=0 Lines at Different C.G. Locations

1. Determine the most forward and the most aft c.g. locations of the airplane. This is done with the method of Chapter 10 in Part II.

2. Compute $\bar{x}_{cg} - \bar{x}_{ref}$

3. At C_L=1.0 determine:

$$\Delta C_{m_{cg-ref}} = -1.0(\bar{x}_{cg} - \bar{x}_{ref}) \qquad (8.116)$$

4. Plot $\Delta C_{m_{cg-ref}}$ as shown in Figures 8.134a,b and connect that point with the origin. The line so obtained is the C_m = 0 locus for the new c.g. location.

Note: the proof for this procedure is found in Chapter 5 of Reference 16.

The determination of trimmed lift and maximum lift now proceeds according to Procedure 2.

PROCEDURE 2: Determination of Trimmed Lift and Trimmed Maxiximum Lift

1. Construct the trim diagram as shown in Figures 8.134a or 8.134b. This includes a number of α = constant lines up to α_{stall}

2. Construct the C_m=0 lines for the most aft and for the most forward c.g. locations, with the help of Procedure 1.

3. Identify the trim-triangle: OAB.

4. Draw in the tail or canard stall locus with the

following considerations. For a stalled tail:

$$\alpha_h = \alpha_{h_{stall}} = \qquad (8.117)$$

$$\alpha + i_h - \epsilon_{o_h} - \alpha(d\epsilon/d\alpha)$$

where: ϵ_{o_h} follows from 8.1.5.2

$d\epsilon/d\alpha$ follows from 8.1.5.3

A value for the tail stall angle, $\alpha_{h_{stall}}$ is found from 8.1.3.4 by substitution of appropriate tail parameters for wing parameters.

For any value of airplane angle of attack, α, Eqn.(8.116) can be used to solve for the corresponding value of i_h which will cause the tail to stall. By repeating this for a range of angles of attack, a tail stall locus can be drawn into Figure 8.134a.

<u>Note:</u> for a canard airplane, the canard stall locus is found in a similar manner by using the following equation:

$$\alpha_c = \alpha_{c_{stall}} = \qquad (8.118)$$

$$\alpha + i_c + \epsilon_{o_c} + \alpha_c(d\epsilon_c/d\alpha)$$

where: ϵ_{o_c} follows from 8.1.5.2

$d\epsilon_c/d\alpha$ follows from 8.1.5.3

<u>IMPORTANT OBSERVATION:</u> the tail stall and/or the canard stall loci MUST be outside the so-called trim triangle identified in Step 3. If this condition is not satisfied, severe restrictions on the performance of the airplane may result.

5. Compute $C_L = W/qS$ for significant points in the airplane flight envelope. Figure 8.135 shows examples of significant trim points in the flight envelope of low and high speed airplanes. These points depend on the mission requirements

Figure 8.135 Significant Trim Points in Flight Envelope

Figure 8.136a Trim Diagram: Unstable, Conventional with Stable Pitch Break

Figure 8.136b Trim Diagram: Unstable, Canard with Stable Pitch Break

placed on the airplane.

6. Determine the c.g. location(s) at which the trim needs to be investigated. Combinations of weight and c.g. locations follow from the weight-c.g. excursion diagram discussed in detail in Ch.10 of Part II.

7. Plot the C_L points on the appropriate C_m=0 line (corresponding to its c.g. location) in Figures 8.134a,b and determine the amount of control surface deflection required to trim. From these control deflections required for trim it is posble to compute the tail or canard lift increments to trim, needed in the trim drag calculations on p.104:

$$\Delta C_{L_{h \text{ or } c}} = (C_{L_{i_{h \text{ or } c}}})i_{h \text{ or } c} \qquad (8.119)$$

or:

$$\Delta C_{L_{h \text{ or } c}} = (C_{L_{\delta_{e \text{ or } c}}})\delta_{e \text{ or } c} \qquad (8.120)$$

8. Points A and B in Figures 8.134a,b represent the trimmed maximum lift coefficients at the corresponding c.g. locations of the airplane.

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SPECIAL CASE: Three-surface Airplane Trim

In the case of a three surface configuration the designer can choose how to deflect control surfaces on the canard, the wing and the tail in conjunction with each other. Several possible optimization schemes are now possible:

1. Minimize trimmed drag
2. Maximize trimmed maximum lift
3. Maximize maneuvering capability

The construction of a trim diagram for a three surface airplane is left to the reader. The Grumman X-29 and the Piaggio Avanti are examples of three surface airplanes.

8.3.2 Unstable Airplane with Stable Pitch Break

Figures 8.136a,b present example trim diagrams for an unstable airplane with a stable pitch break. The pro-

cedures outlined in 8.3.1 for determining trimmed lift and trimmed maximum lift also apply in this case.

8.3.3 Stable Airplane with Unstable Pitch Break

Figures 8.137a,b present example trim diagrams for a stable airplane with an unstable pitch break. The method of 8.3.1 for determining trimmed lift can be used without modification.

However, to determine the allowable trimmed maximum lift coefficient now is not very straightforward. Depending on the number of pitch curve 'wiggles' and depending on the change in control power at high angle of attack, the trimmed maximum lift coefficient may be limited for reasons other than aerodynamic stall. If handling quality problems or severity of pitch divergence so dictate, the airplane may have to be equipped with stick-shakers and/or with stick-pushers. In such cases the maximum trimmable lift coefficient which can be used in the normal performance envelope of the airplane is predicated on the lift coefficient corresponding to stick-pusher operation. Figures 8.137a,b indicate such an artificial limit.

In the case of airplanes equipped with digital flight control systems a control-limiting scheme may be added to the flight control laws. Such a control limiting scheme may limit the angle of attack which actually can be reached by the airplane. The limiting value of trimmed maximum lift coefficient must then be the one which corresponds to the limit set by the flight control law and not by any inherent aerodynamic limit.

8.3.4 Unstable Airplane with Unstable Pitch Break

Figures 8.138a,b show examples of trim diagrams for an unstable airplane with an unstable pitch break. The comments made under 8.3.2 apply to this case.

Figure 8.137a Trim Diagram: Stable, Conventional with Unstable Pitch Break

Figure 8.137b Trim Diagram: Stable, Canard with Unstable Pitch Break

Figure 8.138a Trim Diagram: Unstable, Conventional with Unstable Pitch Break

Figure 8.138b Trim Diagram: Unstable, Canard with Unstable Pitch Break

9. AIRPLANE HIGH LIFT DATA

The purpose of this chapter is to present a range of high lift (and some pitching moment) data. These data are presented in the following form:

9.1. Airfoil high lift data: flaps up and down

9.2. Airplane high lift data: flaps up and down

9.3. Mach number effects on high lift

9.1 AIRFOIL HIGH LIFT DATA: FLAPS UP AND FLAPS DOWN

A summary of NACA airfoil high lift and pitching moment data is provided in Table 9.1 for flaps up.

Tables 9.2, 9.3 and 9.4 provide data on airfoil high lift and pitching moment for flaps down.

The reader should carefully note the effect of Reynold's Number on the maximum lift coefficient. This effect must not be neglected in preliminary design. A summary of the effect of Reynold's Number on airfoil maximum lift is provided in Figures 9.1 and 9.2.

References 49 and 51 provide much additional data on airfoil maximum lift and pitching moment as well as on the effect of Reynold's Number.

9.2 AIRPLANE HIGH LIFT DATA: FLAPS UP AND FLAPS DOWN

In this section some examples of actual airplane high lift data are presented.

Tables 9.5 present a summary of airplane trimmed maximum lift coefficient data, flaps up anf flaps down.

Actual airplane high lift versus angle of attack data are presented as follows:

Figure 9.3 N.A.Rockwell T2C
Figure 9.4 N.A.Rockwell S-60 Sabreliner
Figure 9.5 Raisbeck Modification, S-65 Sabreliner
Figure 9.6 Canadair Challenger
Figure 9.7 Learjet M55
Figure 9.8 Boeing 767-200
Figure 9.9 Boeing 737-300

9.3 MACH NUMBER EFFECTS ON HIGH LIFT

At high subsonic Mach numbers the magnitude of maximum lift coefficient tends to decrease as a result of shock induced separations. This decrease in maximum lift manifests itself as buffet (below maximum lift) and as a reduction in maneuvering capability of the airplane. This effect is strongly influenced by:

1. wing sweep angle
2. airfoil thickness
3. airfoil camber

Figure 9.10 shows some early data indicating the effect of planform and Mach number on the maximum lift of wings with a 6 percent thick airfoil.

With modern airfoil design it is possible to achieve significant improvements in high lift at all Mach numbers and do so at much higher airfoil thickness ratios. (This also results in lower structural weights!) Figure 9.11 illustrates this on the Rockwell T2C: the basic airfoil has t/c = 0.12, the modified airfoil has t/c = 0.17.

A penalty which accompanies almost any increase in high lift is an increase in pitching moment. Since this needs to be trimmed out, the net gain in lift is not as high as untrimmed data would indicate. Figure 9.12 shows the effect of airfoil change on the T2C on its pitching moment behavior: note the rather large negative increase in pitching moment.

An example of the buffet boundary ($C_{L_{buffet}} < C_{L_{max}}$) for a modern business jet airplane with an advanced airfoil is given in Figure 9.13.

Most high lift data are obtained for steady state conditions. In flight test such conditions are simulated by accepting high lift data only when the rate of reduction of airspeed is not greater than 1 kt/sec.

In addition to the effect of speed changes on high lift, there are strong effects of rate of change of angle of attack on high lift. The data in Figs 9.14 and 9.15 indicate this! In determining the performance of fighter aircraft these unsteady effects are very important.

Table 9.1 Experimental Low Speed Data for 4- and 5- Digit NACA Airfoils with a Smooth Leading edge and for $R_N = 9x10^6$ (Ref.49)

Airfoil	α_{0_L} (deg)	c_{m_0}	c_{l_α} (deg^{-1})	a.c. (tenths c)	$\alpha_{c_{l_{max}}}$ (deg)	$c_{l_{max}}$	α^* (deg)
0006	0	0	.108	.250	9.0	.92	9.0
0009	0	0	.109	.250	13.4	1.32	11.4
1408	0.8	-.023	.109	.250	14.0	1.35	10.0
1410	-1.0	-.020	.108	.247	14.3	1.50	11.0
1412	-1.1	-.025	.108	.252	15.2	1.58	12.0
2412	-2.0	-.047	.105	.247	16.8	1.66	9.5
2415	-2.0	-.049	.106	.246	16.4	1.63	10.0
2418	-2.3	-.050	.103	.241	14.0	1.47	10.0
2421	-1.8	-.040	.103	.241	16.0	1.47	8.0
2424	-1.8	-.040	.098	.231	16.0	1.29	8.4
4412	-3.8	-.093	.105	.247	16.0	1.67	7.5
4415	-4.3	-.093	.105	.245	15.0	1.64	8.0
4418	-3.8	-.088	.105	.242	14.0	1.53	7.2
4421	-3.8	-.085	.103	.238	16.0	1.47	6.0
4424	-3.8	-.082	.100	.239	16.0	1.38	4.8
23012	-1.4	-.014	.107	.247	18.0	1.79	12.0
23015	-1.0	-.007	.107	.243	18.0	1.72	10.0
23018	-1.2	-.005	.104	.243	16.0	1.60	11.8
23021	-1.2	0	.103	.238	15.0	1.50	10.3
23024	-0.8	0	.097	.233	15.0	1.40	9.7

Airfoil	α_{0_L} (deg)	c_{m_0}	c_{l_α} (deg^{-1})	a.c. (tenths c)	$\alpha_{c_{l_{max}}}$ (deg)	$c_{l_{max}}$	α^* (deg)
63-006	0	.005	.112	.258	10.0	.87	7.7
63-009	0	0	.111	.258	11.0	1.15	10.7
63-206	-1.8	-.037	.112	.254	10.5	1.06	6.0
63-209	-1.4	-.032	.110	.262	12.0	1.40	10.8
63-210	-1.2	-.035	.113	.261	14.5	1.54	9.6
63_1-012	0	0	.116	.265	14.0	1.45	12.8
63_1-212	-2.0	-.035	.114	.263	14.5	1.63	11.4
63_1-412	-2.8	-.075	.117	.272	15.0	1.77	9.6
64-006	0	0	.109	.256	9.0	.8	7.1
64-009	0	0	.110	.262	11.0	1.17	10.0
64-206	-1.0	-.040	.110	.253	12.0	1.03	6.0
64-209	-1.5	-.040	.107	.261	13.0	1.40	8.9
64-210	-1.6	-.040	.110	.258	14.0	1.45	10.8
64_1-012	0	0	.111	.262	14.5	1.45	11.0
64_1-212	-1.3	-.027	.113	.262	15.0	1.55	11.0
64_1-412	-2.6	-.065	.112	.267	15.0	1.67	8.0

Airfoil	α_{0_L} (deg)	c_{m_0}	c_{l_α} (deg^{-1})	a.c. (tenths c)	$\alpha_{c_{l_{max}}}$ (deg)	$c_{l_{max}}$	α^* (deg)
65-006	0	0	.105	.258	12.0	.92	7.6
65-009	0	0	.107	.264	11.0	1.08	9.6
65-206	-1.6	-.031	.105	.257	12.0	1.03	6.0
65-209	-1.2	-.031	.106	.259	12.0	1.30	10.0
65-210	-1.6	-.034	.108	.262	13.0	1.40	9.6
65_1-012	0	0	.110	.261	14.0	1.36	10.0
65_1-212	-1.0	-.032	.108	.261	14.0	1.47	9.4
65_1-412	-3.0	-.070	.111	.265	15.5	1.66	10.5
63A010	0	.005	.105	.254	13.0	1.20	10.0
63A210	-1.5	-.040	.103	.257	14.0	1.45	10.0
64A010	0	0	.110	.253	12.0	1.23	10.0
64A210	-1.5	-.040	.105	.251	13.0	1.44	10.0
64A410	-3.0	-.080	.100	.254	15.0	1.62	10.0
64_1A212	-2.0	-.040	.100	.252	14.0	1.54	11.0
64_2A215	-2.0	-.040	.095	.252	13.0	1.50	12.0

Table 9.2 Effect of Airfoil, Flap Type and Flap Deflection on Drag and on High Lift

ADAPTED FROM REF. 51

FLAP TYPE	FLAP GEOMETRY	NACA AIRFOIL	FLAP CHORD %C	R_N	$C_{l_{max}}$ t.o.	δ_f (DEG) t.o.	C_{d_0} for $C_{l_{t.o.max}}$	C_{d_0} for δ_f at t.o. with C_l = 1,5	2,0	2,5	$C_{l_{max}}$ LAND.	δ_f (DEG) LAND.	C_{d_0} for $\delta_f = 0$	SOURCE:
DOUBLE SLOTTED FLAP		23012	0,1467 0,2200	$3,8 \cdot 10^6$	2,08	28,68	0,20	—	0,072	0,086	3,88	26,70	0,0088	NACA ARR 3L10
✓		"	0,2970 0,2564	"	2,15	30,40	0,17	—	0,055	0,076	3,18	30,70	"	NACA Rep. 723
✓		23021	0,1467 0,2200	"	2,00	20,00	0,18	—	0,148	0,178	3,38	"	0,0188	NACA ARR L4J08
✓		"	0,2845 0,2560	"	3,00	20,30	0,18	0,048	0,066	0,095	3,56	30,00	"	NACA Rep. 723
✓		23020	0,2200 0,2800	"	3,30	"	0,22	0,108	0,110	0,128	3,90	40,00	0,0178	" "
✓		65,3-118	0,309	$6,8 \cdot 10^6$	2,86	35	0,035	0,009	0,017	0,026	3,40	45	0,0047	NACA ACR 3120
✓		66_3-418	0,35	$1,5 \cdot 10^6$	2,82	40	—	0,0240	—	—	3,81	"	0,0061	NACA TN 1071
SINGLE SLOTTED FLAP		23012	0,40	$3,5 \cdot 10^6$	2,60	40	0,16	0,016	0,043	0,082	2,51	50	0,0100	NACA TN 715
✓		23021	"	"	2,62	20	0,13	0,43	0,054	0,078	2,07	"	0,0140	NACA TN 728
✓		23030	0,40	"	2,48	"	0,14	0,052	0,047	—	2,88	"	0,0230	NACA TN 753
✓		63,4-420	0,28	$6,0 \cdot 10^6$	2,83	20	0,0297	0,0118	0,0135	0,0238	3,00	40	0,0068	NACA ACR 3121
VANED SLOTTED FLAP		23012	0,20	$3,6 \cdot 10^6$	2,70	50	0,13	0,074	0,095	0,110	2,30	60	0,0150	NACA TN 688
SLOTTED FLAP + .05C SPLIT FLAP		"	0,2546	"	2,74	50,60	0,12	0,056	0,080	0,074	2,81	40,70	0,0260	NACA Rep. 673
SLOTTED FLAP + .10C SL. FLAP		"	"	"	2,94	40,60	0,155	—	0,085	0,116	3,18	"	0,0188	" "
PLAIN FLAP		"	0,20	"	2,38	60	0,18	0,104	0,154	—	2,60	75	0,0050	NACA Rep. 664
JUNKERS FLAP		"	0,2067	"	2,28	50,3	0,099	0,026	0,046	—	2,87	40,5	0,0100	" "
SPLIT FLAP		"	0,20	"	2,29	30	0,115	0,080	0,083	—	2,04	75	0,0050	" "
FOWLER FLAP		"	0,30	"	3,30	40	0,17	"	0,097	0,113	3,30	40	"	NACA TN 808
QUAD. SLOTTED FOWLER FLAP	.10C EACH	"	0,40	"	3,28	30	0,14	0,028	0,031	0,060	3,60	40,50 50,70	"	NACA Rep. 688
QUAD SLOTTED FOWLER FLAP	.075C EACH	"	0,30	"	3,40	30,50 40,50	0,15	—	0,058	0,090	3,40	30,50 40,50	"	NACA Rep. 748

Table 9.3 Effect of Flap Type and Flap Deflection on High Lift

ADAPTED FROM REF. 51

FLAP TYPE	FLAP GEOMETRY	AIRFOIL	R_N	FLAP CHORD %C	FLAP ANGLE (DEG)	A	$C_{L_{MAX}}$	$\Delta C_{L_{MAX}}$	(DEG) α AT $C_{L_{MAX}}$	SOURCE:
FIXED SLOT		clark y	$1{,}03.10^6$	—	—	6	1,77	0,48	24°	TR 427, TR 554
✓		"	"	—	—	"	1,44	0,15	17°	" "
FIXED SLAT		"	"	0,145	0°	"	1,70	0,41	24°	TR 428, TR 554
TRIPLE SLOTTED WING		"	"	—	—	"	1,93	0,64	25°	TR 427, TR 554
HANDLEY PAGE SLAT		"	"	0,147	—	"	1,84	0,55	29°	TR 400, TR 554 TN 433, TN 443
MAXWELL SLAT	closed open	"	$0{,}85.10^6$	—	—	"	2,07	0,82	29°	TN 598, TR 544
PLAIN FLAP + FIXED SLOT		"	$1{,}03.10^6$	0,30	45°	"	2,18	0,89	19°	TR 427, TR 554
✓ (S)		"	"	"	"	"	2,24	0,95	20°	" "
SPLIT FLAP + MAXWELL SLAT		"	$0{,}85.10^6$	0,211	60°	"	2,53	1,28	21,6°	TR 598, TR 544
SLOTTED FLAP + FIXED SLOT		"	$1{,}03.10^6$	0,30	45°	"	2,26	0,97	19°	TR 427, TR 554, TN 702
✓		"	"	"	"	"	2,44	1,15	16°	TR 427, TR 554, TN 702
SLOTTED FLAP + 2 FIXED SLOTS		"	"	"	"	"	2,60	1,31	20°	TR 427, TR 554

Table 9.4 Effect of Flap Type on High Lift, on Pitching Moment and on Drag

HIGH-LIFT CONFIGURATION	HIGH-LIFT DEVICE TYPE	δ_f	$\Delta C_{L_{max}}$	$\Delta C_{m_{0_{c/4}}}/\Delta C_L$	$\Delta C_m/\Delta T$	$\Delta C_{D_{min}}$	Reyn. No. of Test	References
1	BASIC WING	-	$C_{L_{max}} = 1.0$	$C_{m_0} = 0$	-	$C_{D_{min}} = .007$	5×10^6	NACA TN 3007 NACA TN 3224
2	SPLIT FLAP	60	.8	-.275	-.0017	.23	6×10^6	NACA TN 3007 NACA TN 4040
3	ZAP FLAP	45	1.15	-.30		.21		N.S. DIRE, NASA ARM NACA TN 4[illegible] AERE TRANSACTIONS [illegible]
4	PLAIN FLAP	60	.9	-.275	-.0025	.12	6×10^6	NACA TN [illegible] NACA TN 4040 ARC R&M 2[illegible]
5	SINGLE SLOTTED FLAP	40	1.18	-.32	-.04	.15	3.5×10^6	NACA TN [illegible] NACA TN 4040 ARC R&M 2[illegible]
6	DOUBLE SLOTTED FLAP	30/55	1.4	-.41	-.025	.23	6×10^6	NACA TN 3007 NACA TN 4040
7	SINGLE SLOTTED FLAP	30/40/55	1.6	-.44		.25	6×10^6	NACA WR L-441
8	FOWLER FLAP	30	1.67	-.42		.10	-3.5×10^6	1st AIL-TRS. INT'L. CONF. 1947, [illegible] NACA [illegible] ARC R&M 2[illegible] F.D. Fowler, The Fowler Flap
9	DOUBLE SLOTTED FOWLER FLAP	15/30	1.85	-.44		-.15	-3.5×10^6	1st AIL-TRS. INT'L. CONF. 1947. [illegible]
10	LEADING EDGE SLAT	25 [illegible] 30 [illegible] [illegible]	-.13	+.11		~0	6×10^6	NACA TN 3007 Journal of Aeron. Aero. Sci. [illegible]
11	FOWLER FLAP	[illegible] 40 (Double slotted)	-.5	-.10		~0	[illegible]	NACA TN 3[illegible] Journal of Aeron. Aero. Sci. [illegible]
12	LEADING EDGE FLAP (Leading edge)	30	[illegible]	[illegible]		~0	6×10^6	NACA TN 3007 NACA TN [illegible]

NOTE: ALL DATA FOR:

$A = 12 \quad \lambda = 1.0$

$t/c = 0.10$

$\Lambda_{c/4} = 0°$

$b_f/b = 1.0$

$c_f/c = 0.30$ T.E.

0.15 L.E.

COPIED FROM: REF. 56

Table 9.5a Experimental Trimmed Maximum Lift Coefficients for Several Airplanes

COPIED FROM REF. 56

MODEL	A	$\Lambda_{c/4}$	b_f/b	HLD TYPE	TEST TYPE	$C_{L_{max}}$* CLEAN	$C_{L_{max}}$* ALL HLD	REFERENCE
Grumman A6A	6.0	26	−0.5	Fowler Flaps + LE Flaps	F	1.26	2.00	Flight Manual Data
North Am. A4A	4.0	27.5	.704	Plain Flaps + LE Flaps	F	1.05	1.5	Flight Manual Data
North Am. RA-5C	4.0	27.5	.7	Plain Flaps + Blown Outbd. LE Flaps	F		1.8	AIAA Paper 66-751
Douglas A4D	2.91	39	.57	Split Flaps + LE Automatic	F	.901	1.45	Flight Manual Data
McDonnell F4	2.70	45	.54	Plain Blown Flaps + Inbd. LE Flaps + Outbd. Blown LE Flaps	F	1.05	1.40	Canad. Aeron & Space Journ. March 1968
Fiat G91	4.6	37	.52	Single Slotted Flaps	F	1.0	1.10	Calculated
Republic F105D	3.10	43	.701	Single Slotted Flaps + LE Flaps	F	1.1	1.26	Pilot Handbk. Data
Lockheed F104G	2.45	18.4	.65	Plain Flaps, Droop Ailerons, LE Flaps	F	.75	1.12	Pilot Handbk. Data
Northrop F5	3.7	24	.55	Single Slotted Flaps + LE Flaps	F	1.0	1.4	Pilot Handbk. Data
Hawker Siddeley Buccaneer	3.58	34	1.00	Blown Plain Flaps + Blown LE	F	.90	3.2	Aviation Week 7 Aug 1961 Flight Int'l 3 Nov 1966
Gen. Dynamics F-111	6.0	13	.665	Blown Plain Flaps + Blown Center & Outer LE Flaps	F	1.15	2.45	NASA 1964 SST Conference
North Am. F-100 A (Exp.)	3.73	45	.0	Blown Plain Flaps + Blown LE Flaps	F	1.1	1.5	NASA TN D-221
Douglas F5D-1 (Trailing, NASA OGEE Wing Mod.)	1.7	48	1.0	Plain Elevons + Partial LE Slats	F	1.0	−1.0	NASA TN D-2071
Lockheed XV-4A	5.0	0	.75	Plain Flaps	Full Scale WT	1.15	1.5	NASA TN D-3734
Boeing 727	7.1	20	.75	Triple Slotted Fowler Flaps + LE Slats Outbd. + Kruger Flaps Inbd.	F		2.02	Journ of Aircr. March 65, The Aeroplane Feb. 21, 1963
Boeing 707-320C	7.0	35	.665	Triple Slotted Fowler Flaps + Split Fillet Flaps + LE Flaps	F		1.9	Journ. of Aircr. Mar - Apr. 1965.
Boeing 707-120	7.0	36	.663	Double Slotted Fowler Flaps + Split Fillet Flaps + LE Flaps	F	−1.37 (WT)	1.75	Journ. of Aircr. Mar-Apr. 1964. NACA RM A53K12
Boeing 707-120 (Exp)	7.0	36	.665	Blown Plain Flaps + LE Flaps	F		2.34	NASA TN D-6004

F = Flight; WT = Wind Tunnel; HLD = High Lift Device(s) Down * TRIMMED, GEAR-UP

Table 9.5b Experimental Trimmed Maximum Lift Coefficients for Several Airplanes

COPIED FROM: REF. 56

MODEL	A	$\Lambda_{c/4}$	$b_{f/b}$	HLD TYPE	TEST TYPE	$C_{L_{max}}$ * CLEAN	$C_{L_{max}}$ * ALL HLD	REFERENCE
Dragon 541 **	4.32	0	-1.4	Double & Triple-Slotted Flaps (Droop Ailerons)	F	1.19	2.8	Journ. of Aircr. May-June 1966 NASA TN D-3231
Lockheed Electra **	7.3	0	.66	Fowler Flaps	F	1.1	2.6	Journ. of Aircr. May-June 1966 NASA TN D-3231
Bad. Caravelle	8.2	30	.63	Double Slotted Flaps	F	1.6	2.32	Journ. of Aircr. May-June 1966 NASA TN D-3231
Lockheed C6A	8	35	.716	Single Slotted Fowler & LE Slats	WT F	1.2 1.45	2.66 2.60	Aircr. Engr. June 1966
			.74	Double Slotted Flaps + LE Kruger Flaps + External Jet Blowing	WT		3.4	NASA TN D-1924
Mitsubishi/Mooney MU-60 **	6.6	-6	.63	Double Slotted Flaps + Droop Ailerons	F	1.77	3.13	Vertiz. World July 1966 Flight Magaz. Febr. 1964.
Cessna 177 **	7.4	0	.66	Plain Flaps	F	.66	1.38	SAE Paper 660188
Ryan Model 143 (XV-5A) **	2.42	30/37	1.0	Single-Slotted Flaps + Droop Ailerons	F	1.06	1.83	Ryan DP-196 (1965) (Ryan Rep. 18466-4 Vol. II)
Ryan Model 68 (VZ-8RY) **	4.4	0	1.0	Double-Slotted Flaps (Bidirectional Deflection) + Fixed LE Slats	WT	1.15	1.6	NASA TN D-68
Douglas A3D	6.75	35.0	.575	Single-Slotted Flaps + LE Slats	Full Scale WT	1.37	1.9	NACA RM A57A14
Douglas A3D	6.75	35.0	.575	Slotted, Single-Slotted Flaps + LE Slats	Full Scale WT	1.37	2.7	NACA RM A57D11
North Am. F-86	4.79	35.0	.60	Plain Flaps + LE Autoslats	Full Scale WT	1.09	1.64	NACA RM A53E09
Dornier Do-27 **	7.4	0	.71	Double-Slotted Flaps (Partial Droop Ailerons) + LE Fixed Slats	F	1.3	2.16	Luftfahrtforsch. No. 2, 1962
Butler-Blaud DHC-4 Caribou **	8.8	-0	1.0	Double-Slotted Flaps (Droop Ailerons)	F	1.37	2.60	Calcul. from Pilot Hdbk. Data
Lockheed 49 Constellation **	8.1	0	.62	Fowler Flaps + Split-Fillet Flap	F		2.88	The Fowler Flap
Boeing B-17 **	7.3	0	.67	Zap Flaps	F		2.96	Neville, Aircr. Designers Data Book
Douglas DC-8	5.6	34	.84	Double-slotted Flaps + LE Slat	F	1.3	3.0	SAE Paper 670846
Piper PA-28-180 **	7.3	0	.63	Plain Flaps	Full Scale WT	1.34	1.6	NASA TN D-4943

F = FLIGHT WT = WINDTUNNEL HLD = HIGH LIFT DEVICES DOWN
* TRIMMED, GEAR UP ** PROPELLER(S) OFF

Figure 9.1 Effect of Thickness Ratio and Reynold's Number on NACA Airfoil Maximum Lift Coefficient

Figure 9.2 Effect of Thickness Ratio and Reynold's Number on Modern Airfoil Maximum Lift Coefficient

Figure 9.3 Airplane Lift Curve: N.A.Rockwell T2C

Figure 9.4 Airplane Lift Curve: N.A.Rockwell S-60

COURTESY: RAISBECK

Figure 9.5 Airplane Lift Curve: Raisbeck/Rockwell S-65

Figure 9.6 Airplane Lift Curve: Canadair Challenger

Figure 9.7 Airplane Lift Curve: Gates-Learjet M55

Figure 9.8 Airplane Lift Curve: Boeing 767-300

Figure 9.9 Airplane Lift Curve: Boeing 737-300

Figure 9.10 Effect of Wing Planform on the Variation of Maximum Lift Coefficient with Mach Number

Figure 9.11 Effect of Mach Number on Buffet Onset for a NACA and for and Advanced Airfoil

Figure 9.12 Effect of Airfoil Change on Pitching Moment

Figure 9.13 Buffet Boundary for the Cessna Citation III

Figure 9.14 Effect of Angle of Attack Rate on the Variation of Maximum Lift with Mach Number

Figure 9.15 Effect of Angle of Attack Rate and Mach Number on Maximum Lift Coefficient

10. STABILITY, CONTROL AND HINGEMOMENT DERIVATIVES

The purpose of this chapter is to present preliminary design methods for the calculation of stability, control and hingemoment derivatives. These derivatives are required as input to the determination of static and dynamic stability and control behavior (handling qualities) of airplanes. Methods for computing static and dynamic stability and control behavior of airplanes are given in Part VII of this text.

Table 10.1 presents the mathematical model used in representing aerodynamic and thrust forces and moments as well as cockpit control forces: Eqns 10.1 through 10.5.

The equations of Table 10.1 are discussed and justified in Chapter 4 of Reference 16. The coefficients and derivatives shown in Table 10.1 may be estimated with the methods of this chapter. These methods are presented in the following manner:

10.1 Steady state coefficients
10.2 Stability Derivatives
10.3 Control derivatives
10.4 Hingemoment Derivatives.

IMPORTANT NOTES:

1.) The methods presented in this chapter apply to rigid airplanes only. Methods which account for aeroelastic effects are discussed in Chapter 8 of Ref.16.

2.) The methods presented in this chapter do not always apply to all speed regimes. Restrictions on applicability to certain speed regimes are given in the text.

3.) All derivatives are in rad^{-1}.

4.) All coefficients and derivatives are defined in the stability axes system. see Table 10.1.

10.1 STEADY STATE COEFFICIENTS

In Table 10.1 there appear a number of steady state coefficients: C_{D_1}, C_{L_1}, C_{m_1}, $C_{T_{x_1}}$ and $C_{m_{T_1}}$.

In the following, definitions are given from which these coefficients may be determined.

Table 10.1a Longitudinal Aerodynamic and Thrust Forces and Moments and Cockpit Control Forces

Forces and Moments

AERODYNAMIC

$$\begin{Bmatrix} \dfrac{f_{A_x}}{\bar{q}S} \\ \dfrac{f_{A_z}}{\bar{q}S} \\ \dfrac{m_A}{\bar{q}S} \end{Bmatrix} = \begin{bmatrix} -(C_{D_u}+2C_{D_1}) & (-C_{D_\alpha}+C_{L_1}) & -C_{D_{\dot\alpha}} & -C_{D_q} & -C_{D_{\delta_E}} \\ -(C_{L_u}+2C_{L_1}) & (-C_{L_\alpha}-C_{D_1}) & -C_{L_{\dot\alpha}} & -C_{L_q} & -C_{L_{\delta_E}} \\ (C_{m_u}+2C_{m_1}) & C_{m_\alpha} & C_{m_{\dot\alpha}} & C_{m_q} & C_{m_{\delta_E}} \end{bmatrix} \begin{Bmatrix} \dfrac{u}{U_1} \\ \alpha \\ \dfrac{\dot{\bar{\alpha}}\bar{c}}{2U_1} \\ \dfrac{q\bar{c}}{2U_1} \\ \delta_E \end{Bmatrix} \quad (10.1)$$

(3x1) (3x5) (5x1)

THRUST

$$\begin{Bmatrix} \dfrac{f_{T_x}}{\bar{q}_1 S} \\ \dfrac{f_{T_z}}{\bar{q}_1 S} \\ \dfrac{m_T}{\bar{q}_1 S\bar{c}} \end{Bmatrix} = \begin{bmatrix} (C_{T_{x_u}} + 2C_{T_{x_1}}) & 0 \\ 0 & 0 \\ (C_{m_{T_u}} + 2C_{m_{T_1}}) & C_{m_{T_\alpha}} \end{bmatrix} \begin{Bmatrix} \dfrac{u}{U_1} \\ \alpha \end{Bmatrix} \quad (10.2)$$

3x1 3x2 2x1

$$F_{S_e} = G_e \cdot HM_e = G_e \cdot \bar{q}_h \cdot S_e \bar{c}_e C_{h_e} \quad (10.3a)$$

$$C_{h_e} = C_{h_0} + C_{h_\alpha}\alpha_h + C_{h_{\delta_e}}\delta_e + C_{h_{\delta_t}}\delta_t \quad (10.3b)$$

B = BODY

S = STABILITY

Table 10.1b Lateral-Directional Aerodynamic and Thrust Forces and Moments and Cockpit Control Forces

Forces and Moments

AERODYNAMIC

$$\underset{(3\times1)}{\begin{Bmatrix} \dfrac{f_{A_Y}}{qS} \\ \dfrac{\ell_A}{qSb} \\ \dfrac{n_A}{qSb} \end{Bmatrix}} = \underset{(3\times6)}{\begin{bmatrix} C_{y_\beta} & C_{y_{\dot\beta}} & C_{y_p} & C_{y_r} & C_{y_{\delta_A}} & C_{y_{\delta_R}} \\ C_{\ell_\beta} & C_{\ell_{\dot\beta}} & C_{\ell_p} & C_{\ell_r} & C_{\ell_{\delta_A}} & C_{\ell_{\delta_R}} \\ C_{n_\beta} & C_{n_{\dot\beta}} & C_{n_p} & C_{n_r} & C_{n_{\delta_A}} & C_{n_{\delta_R}} \end{bmatrix}} \underset{(6\times1)}{\begin{Bmatrix} \beta \\ \dfrac{\dot\beta b}{2U_1} \\ \dfrac{pb}{2U_1} \\ \dfrac{rb}{2U_1} \\ \delta_A \\ \delta_R \end{Bmatrix}} \quad (10.4)$$

THRUST

$$\underset{(3\times1)}{\begin{Bmatrix} \dfrac{f_{T_Y}}{\bar q_1 S} \\ \dfrac{\ell_T}{\bar q_1 Sb} \\ \dfrac{n_T}{\bar q_1 Sb} \end{Bmatrix}} = \underset{(3\times1)}{\begin{Bmatrix} 0 \\ 0 \\ C_{n_{T_\beta}} \end{Bmatrix}} \underset{(1\times1)}{\beta} \quad (10.5)$$

AILERON

$$F_{s_a} = G_a \cdot HM_a = G_a \cdot \bar q \cdot S_a \bar c_a C_{h_a} \quad (10.3c)$$

$$C_{h_a} = C_{h_{\delta_a}} \delta_a + C_{h_{\delta_{a_t}}} \delta_{a_t} \quad (10.3d)$$

RUDDER

$$F_{s_r} = G_r \cdot HM_r = G \cdot \bar q_v \cdot S_r \bar c_r C_{h_r} \quad (10.3e)$$

$$C_{h_r} = C_{h_\beta} \beta + C_{h_{\delta_r}} \delta_r + C_{h_{\delta_{r_t}}} \delta_{r_t} \quad (10.3f)$$

C_{D_1} is the airplane steady state drag coefficient. It is found with the methods of Chapter 4 and is tied uniquely to a specific value of the steady state lift coefficient, C_{L_1}. Figure 10.1 illustrates this.

C_{L_1} is the airplane steady state lift coefficient:

$$C_{L_1} = nW/\bar{q}S \tag{10.6}$$

where: n is the airplane load factor. Note that for 1g flight, n=1.

C_{m_1} is the airplane steady state pitching moment coefficient. It is determined with the methods of Chapter 8 (8.2.5 and 8.2.6) and is tied uniquely to a specific value of the steady state lift coefficient, C_{L_1}. Figure 10.2 shows this.

$C_{T_{x_1}}$ is the airplane steady state thrust coefficient component in the direction of the stability X-axis. It is determined from:

$$C_{T_{x_1}} = T_{reqd}/\bar{q}S \tag{10.7}$$

where: T_{reqd} is the thrust required in the steady state. The condition:

$$C_{T_{x_1}} = -C_{D_1} \tag{10.8}$$

is normally satisfied in the steady state.

$C_{m_{T_1}}$ is the airplane steady state pitching moment coefficient due to thrust. It is found from:

$$C_{m_{T_1}} = \Delta C_{m_{TL}}, \text{ see: Eqn.(8.102)}$$

The trim condition:

$$C_{m_1} = -C_{m_{T_1}} \tag{10.9}$$

is normally satisfied in the steady state.

Figure 10.1 Determination of Drag Coefficient from a Known Value of the Lift Coefficient

Figure 10.2 Determination of the Value of the Pitching Moment Coefficient

Figure 10.3 Determination of: $\partial C_D/\partial C_m$

10.2 STABILITY DERIVATIVES

In this section, methods for computing the following stability derivatives will be presented:

10.2.1 Speed derivatives
10.2.2 Angle-of-attack derivatives
10.2.3 Rate of angle-of-attack derivatives
10.2.4 Angle of sideslip derivatives
10.2.5 Rate of angle of sideslip derivatives
10.2.6 Roll rate derivatives
10.2.7 Pitch rate derivatives
10.2.8 Yaw rate derivatives

10.2.1 Speed Derivatives: C_{D_u}, C_{L_u}, C_{m_u}, $C_{T_{x_u}}$ and $C_{m_{T_u}}$

Table 10.1 lists the required speed derivatives: note the existence of aerodynamic as well as thrust derivatives.

10.2.1.1 Aerodynamic speed derivatives: C_{D_u}, C_{L_u} and C_{m_u}

1.) C_{D_u}

The drag-due-to-speed derivative, C_{D_u} may be determined in all speed regimes from:

$$C_{D_u} = M_1(\partial C_D/\partial M) \qquad (10.10)$$

where: M_1 is the Mach number in steady state flight

$\partial C_D/\partial M$ is the derivative of the airplane drag coefficient with respect to Mach number. This derivative is found from Figure 10.3.

2.) C_{L_u}

The lift-due-to-speed derivative, C_{L_u} may be determined as follows:

Subsonic:

$$C_{L_u} = [M_1^2(\cos\Lambda_{c/4})^2 C_{L_1}]/[1 - M_1^2(\cos\Lambda_{c/4})^2] \qquad (10.11)$$

Transonic: Fair from M=0.8 to M=1.2 as in Fig.10.4

Supersonic: use Eqn.(10.11) also!

3.) C_{m_u}

The pitching-moment-due-to-speed derivative, C_{m_u} may be determined in all speed regimes from:

$$C_{m_u} = - C_{L_1} (\partial \bar{x}_{ac_A} / \partial M) \qquad (10.12)$$

where: C_{L_1} is the airplane lift coefficient in the steady state as found from Eqn. (10.6).

$\bar{x}_{ac_A}$ is the location of the airplane aerodynamic center in fractions of the wing m.g.c. as determined from 8.2.5.2.

<u>10.2.1.2 Thrust versus speed derivatives:</u> $C_{T_{x_u}}$ and $C_{m_{T_u}}$

These derivatives are equal to zero in all speed regimes for gliders and for airplanes equipped with rocket powerplants only.

For propeller-driven airplanes and for jet-driven airplanes these derivatives depend on the characteristics of the propulsion installation.

1.) $C_{T_{x_u}}$

The thrust-due-to-speed derivative, $C_{T_{x_u}}$ may be computed in all speed regimes from:

<u>For Propeller Driven Airplanes:</u>

<u>Fixed pitch propellers:</u>

$$C_{T_{x_u}} = (1/\bar{q}S)(\partial P_{reqd}/\partial u) - 2C_{T_{x_1}} \qquad (10.13)$$

where: P_{reqd} is the installed power required for the flight condition being studied: see Ch. 6.

<u>Variable pitch (constant speed) propellers:</u>

$$C_{T_{x_u}} = - 3C_{T_{x_1}} \qquad (10.14)$$

<u>For Jet Driven Airplanes:</u>

$$C_{T_{x_u}} = (M_1/\bar{q}S)(\partial T_{reqd}/\partial M) - 2C_{T_{x_1}} \qquad (10.15)$$

where: M_1 is the steady state Mach number

$\partial T_{reqd}/\partial M$ is the derivative of installed thrust required with respect to Mach number. It is determined from plots of installed thrust versus Mach number: see Ch.6.

2.) $C_{m_{T_u}}$

The thrust-moment-due-to-speed derivative, $C_{m_{T_u}}$ may be computed in all speed regimes from:

$$C_{m_{T_u}} = (d_T/\bar{c})C_{T_{x_u}} \qquad (10.16)$$

where: d_T is the thrust moment arm relative to the center of gravity as indicated in Figure 8.126. Note: the sign of d_T depends on the location of the thrust line relative to the c.g.

Figure 10.4 Example of Transonic Fairing for: C_{L_u}

10.2.2 Angle-of-Attack Derivatives: C_{D_α}, C_{L_α}, C_{m_α} and $C_{m_{T_\alpha}}$

Table 10.1 identifies the required angle-of-attack derivatives: note that there are aerodynamic as well as thrust derivatives.

10.2.2.1 Aerodynamic angle-of-attack derivatives: C_{D_α}, C_{L_α} and C_{m_α}

1.) C_{D_α}

The drag-due-to-angle-of-attack derivative, C_{D_α} may be found in all speed regimes from:

$$C_{D_\alpha} = (\partial C_D/\partial C_L)C_{L_\alpha} \qquad (10.17)$$

where: $(\partial C_D/\partial C_L)$ is determined as shown in Figure 10.5. Note that this slope must be obtained at the appropriate value of the lift coefficient, C_{L_1} AND at the appropriate value of Mach number, M_1.

C_{L_α} is the overall airplane lift curve slope as found from Eqn.(8.42): see the note of caution under 2.) C_{L_α}.

For airplanes with parabolic drag polars it is often more convenient to use:

$$C_{D_\alpha} = (2C_{L_1}/\pi Ae)C_{L_\alpha} \qquad (10.18)$$

2.) C_{L_α}

The lift-due-to-angle-of-attack derivative (also called lift curve slope), C_{L_α} is found from Eqn.(8.42).

NOTE OF CAUTION: Equation (8.42) applies to all speed regimes, provided the values for upwash and for downwash are valid for the appropriate speed regimes. The method of page 272 for computing downwash is valid only below M=0.9. For the transonic and for the supersonic speed regime Ref.[must be consulted.

3.) C_{m_α}

The pitching-moment-due-to-angle-of-attack derivati-

Figure 10.5 Determining $\partial C_D/\partial C_L$ From the Drag Polar

Figure 10.6 Definition of Geometric Parameters for Volume Coefficient

ve (also called static longitudinal stability). C_{m_α} is computed in all speed regimes from:

$$C_{m_\alpha} = (dC_m/dC_L)C_{L_\alpha} \tag{10.19}$$

where: dC_m/dC_L is found from Eqn.(8.81) with possible corrections for power effects as given in 8.2.8.

C_{L_α} is determined from Eqn.(8.42). The note of caution given before, applies here also!

<u>10.2.2.2 Thrust versus angle-of-attack derivative:</u> $C_{m_{T_\alpha}}$

The only thrust versus angle-of-attack derivative which appears in Table 10.1 is $C_{m_{T_\alpha}}$. This derivative may be found from:

$$C_{m_{T_\alpha}} = \{\Delta(dC_m/dC_L)_T\}C_{L_\alpha} \tag{10.20}$$

where: $\{\Delta(dC_m/dC_L)_T\}$ is found from Eqn.(8.104).

C_{L_α} is found from Eqn.(8.42). The note of caution given before, applies here also!

<u>10.2.3 Rate of Angle-of-attack Derivatives:</u> $C_{D_{\dot\alpha}}$, $C_{L_{\dot\alpha}}$ <u>and</u> $C_{m_{\dot\alpha}}$

Except for $C_{m_{\dot\alpha}}$, these derivatives are usually not important. The reader should consult Ref.9 for detailed methods covering all speed regimes. The methods given here apply only in the subsonic speed range.

1.) $C_{D_{\dot\alpha}}$

The drag-due-to-rate-of-angle-of-attack derivative, $C_{D_{\dot\alpha}}$ is normally neglected:

$$C_{D_{\dot\alpha}} = 0 \tag{10.21}$$

2.) $C_{L_{\dot\alpha}}$

The lift-due-to-rate-of-angle-of-attack derivative, $C_{L_{\dot\alpha}}$ may be computed from:

$$C_{L_{\dot\alpha}} = 2(C_{L_{\alpha_h}})\eta_h(\bar{V}_h)(d\epsilon/d\alpha) \tag{10.22}$$

where: $C_{L_{\alpha_h}}$ is found from 8.1.3.2 with appropriate substitution of horizontal tail parameters for wing parameters.

η_h is found from 8.1.5.2

$d\epsilon/d\alpha$ is found from Eqn.(8.45)

$\bar{V}_h$ is the horizontal tail volume coefficient:

$$\bar{V}_h = (\bar{x}_{ac_h} - \bar{x}_{cg})(S_h/S) \qquad (10.23)$$

where: $\bar{x}_{ac_h}$ and $\bar{x}_{cg}$ (or $\bar{x}_{ref}$) are defined in Figure 10.6.

3.) $C_{m_{\dot{\alpha}}}$

The pitching-moment-due-to-rate-of-angle-of-attack, $C_{m_{\dot{\alpha}}}$ may be computed from:

$$C_{m_{\dot{\alpha}}} = -2(C_{L_{\alpha_h}})\eta_h(\bar{V}_h)(\bar{x}_{ac_h} - \bar{x}_{cg})(d\epsilon/d\alpha \qquad (10.24)$$

where: all quantities are defined before.

<u>NOTE:</u> the reader will observe that Equations (10.23) and (10.24) are based on the assumption that the contribution of the horizontal tail is the only important contribution to these derivatives. This assumption is frequently satisfied. If not, Ref.9 should be used.

Figure 10.7 Definition of Wing Geometric Dihedral

10.2.4 Angle-of-Sideslip Derivatives: C_{y_β}, C_{l_β}, C_{n_β} and $C_{n_{T_\beta}}$

Table 10.1 identifies the required angle-of-sideslip derivatives: note that there are aerodynamic as well as thrust derivatives.

NOTE: All methods presented for the sideslip derivatives apply to the subsonic speed regime only. For other speed ranges, Ref.9 should be used.

10.2.4.1 Aerodynamic angle-of-sideslip derivatives: C_{y_β}, C_{l_β} and C_{n_β}

1.) C_{y_β}

The sideforce-due-to-sideslip derivative, C_{y_β} may be found from:

$$C_{y_\beta} = C_{y_{\beta_w}} + C_{y_{\beta_f}} + C_{y_{\beta_v}} \tag{10.25}$$

where: the wing contribution is given by:

$$C_{y_{\beta_w}} = -0.00573(|\Gamma|) \tag{10.26}$$

with: Γ the wing geometric dihedral angle in deg. Figure 10.7 defines this angle.

the fuselage contribution is given by:

$$C_{y_{\beta_f}} = -2K_1(S_o/S) \tag{10.27}$$

with: K_1 defined in Figure 10.8. The quantities z_w and d_f are defined in Figure 10.9

S_o is the cross-sectional area of the fuselage at station x_o, where the flow ceases to be potential. The distance x_o depends on the distance x_1. The latter is equal to that fuselage station where the derivative dS_x/dx first reaches its maximum negative value: see Figure 10.10. The

Figure 10.8 Wing-Fuselage Interference Factor, K_i

Figure 10.9 Definition of Wing-Fuselage Parameters: z_w and d_f

Figure 10.10 Determination of S_o

Figure 10.11 Fuselage Station Where Flow Becomes Viscous

(For definition of $2r_1$ see Figure 10.17)

Figure 10.12 Empirical Factor for Estimating Side-force due to Sideslip of a Single Vertical Tail

distances x_o and x_1 are correlated in

Figure 10.11. The fuselage contribution is virtually independent of Mach number.

<u>the vertical tail contribution is given by:</u>

<u>1. For Single Vertical Tails:</u>

$$C_{y_{\beta_v}} = -k_v(C_{L_{\alpha_v}})(1 + d\sigma/d\beta)\eta_v(S_v/S) \qquad (10.28)$$

with: k_v given by Figure 10.12.

$C_{L_{\alpha_v}}$ found from 8.1.3.2 with appropriate substitution of vertical tail parameters for wing parameters. The vertical tail aspect ratio. $A_{v_{eff}}$ must be substituted for A in the method of 8.1.3.2. The effective aspect ratio of the vertical tail may be estimated from:

$$A_{v_{eff}} = \qquad (10.29)$$

$$(A_{v(f)}/A_v)A_v\{1 + K_{vh}[(A_{v(hf)}/A_{v(f)}) - 1]$$

with: $A_v = (b_v)^2/S_v$ (10.30)

where: b_v and S_v depend on the vertical tail configuration as shown in Figure 10.13 for a range of vertical tail configurations.

$(A_{v(f)}/A_v)$ is the ratio of the vertical tail aspect ratio in the presence of the fuselage to that of an isolated vertical tail. This ratio can be determined from Figure 10.14.

$(A_{v(hf)}/A_{v(f)}$ is the ratio of the vertical tail aspect ratio in the presence of the horizontal tail and the fuselage to that in the presence of the fuselage alone. This ratio can be determined from Figure 10.15.

Figure 10.13 Examples of Effective Vertical Tail Geometry

Figure 10.14 Ratio of Vertical Tail Aspect Ratio in Presence of Fuselage to that of Isolated Tail

Figure 10.15 Ratio of Vertical Tail Aspect Ratio in Presence of Fuselage and Horizontal Tail to that in Presence of Fuselage Alone

K_{vh} is a factor which accounts for the relative size of the horizontal and the vertical tail. It may be determined from Figure 10.16.

$$(1 + d\sigma/d\beta)\eta_v = \tag{10.31}$$

$$0.724 + 3.06\{(S_v/S)/(1 + \cos\Lambda_{c/4})\} + 0.42z_w/z_f + 0.009A$$

S_v is the effective vertical tail area as defined in Figure 10.13 for several example configurations.

$\Lambda_{c/4}$ is the wing quarter chord sweep angle

A is the wing aspect ratio

<u>2. For Twin Vertical Tails:</u>

$$C_{y_{\beta_v}} = \tag{10.32}$$

$$-2\{C_{y_{\beta_{v(wfh)}}}/C_{y_{\beta_{v_{eff}}}}\}(C_{y_{\beta_{v_{eff}}}})(S_v/S)$$

with: $\{C_{y_{\beta_{v(wfh)}}}/C_{y_{\beta_{v_{eff}}}}\}$ determined from Figure 10.17.

$C_{y_{\beta_{v_{eff}}}}$ determined from Figure 10.18 with the corresponding value of $A_{v_{eff}}$ found from Figure 10.19.

S_v is the area of a single vertical tail panel as defined in Figure 10.13.

2.) C_{l_β}

The rolling-moment-due-to-sideslip derivative, C_{l_β} (also called dihedral effect) may be found from:

$$C_{l_\beta} = C_{l_{\beta_{wf}}} + C_{l_{\beta_h}} + C_{l_{\beta_v}} \tag{10.33}$$

where: <u>the wing-fuselage contribution is given by:</u>

Figure 10.16 Factor which Accounts for Relative Size of Horizontal and Vertical Tail

Reproduced from Reference 9

Figure 10.17 Wing-Fuselage-Horizontal-Tail Interference on Side-force due to Sideslip of Twin Vertical Tails

Figure 10.18 Effective Value of the Side-force due to Sideslip Derivative for Twin Vertical Tails

For definition of b'_v see Figure 10.18

Figure 10.19 Effective Value of Vertical Tail Aspect Ratio Used with Figure 10.18

$$C_{l_{\beta_{wf}}} = 57.3[C_{L_{wf}}\{(C_{l_\beta}/C_L)_{\Lambda_{c/2}}(K_{M_\Lambda})(K_f) + (C_{l_\beta}/C_L)_A\} +$$

$$+ \Gamma\{(C_{l_\beta}/\Gamma)K_{M_\Gamma} + (\Delta C_{l_\beta}/\Gamma)\} + (\Delta C_{l_\beta})_{z_w} +$$

$$+ (\epsilon_t \tan\Lambda_{c/4})((\Delta C_{l_\beta})/\epsilon_t \tan\Lambda_{c/4})] \qquad (10.34)$$

with: $C_{L_{wf}}$ is the lift coefficient of the wing-fuselage combination. For any given value of airplane lift coefficient, this may be computed by subtracting the tail and/or the canard lift coefficients (but based on wing area!). For preliminary design purposes it seems acceptable to set: $C_{L_{wf}} = C_{L_1}$.

$(C_{l_\beta}/C_L)_{\Lambda_{c/2}}$ is the wing sweep contribution which may be found from Figure 10.20.

K_{M_Λ} is the compressibility correction to sweep. It is found from Figure 10.21.

K_f is a fuselage correction factor obtained from Figure 10.22.

$\Lambda_{c/2}$ is the wing semi-chord sweep angle

$\Lambda_{c/4}$ is the wing quarter-chord sweep angle

$(C_{l_\beta}/C_L)_A$ is the aspect ratio contribution obtained from Figure 10.23.

Γ is the geometric dihedral angle of the wing as defined in Figure 10.7.

(C_{l_β}/Γ) is the wing dihedral effect found from Figure 10.24.

K_{M_Γ} is the compressibility correction to dihedral as obtained from Figure 10.25.

$(\Delta C_{l_\beta}/\Gamma)$ is the fuselage induced effect on the wing height and is found from:

$$(\Delta C_{l_\beta}/\Gamma) = -0.0005A(d_{f_{ave}}/b)^2 \qquad (10.35)$$

where: A is the wing aspect ratio

Figure 10.20 Wing Sweep Contribution to Rolling Moment due to Sideslip

Figure 10.21 Compressibility Correction to Wing Sweep

Figure 10.22 Fuselage Correction Factor

Figure 10.23 Wing Aspect Ratio Contribution to Rolling Moment due to Sideslip

Figure 10.24 Wing Geometric Dihedral Contribution to Rolling Moment due to Sideslip

Figure 10.25 Compressibility Correction to Wing Dihedral

Figure 10.26 Contribution of Wing Twist to Rolling Moment due to Sideslip

Figure 10.27 Geometry for Locating Vertical Tail(s)

b is the wing span

$$d_{f_{ave}} = \{(\text{ave. fusel. cross-section area})/0.7854\}^{1/2} \quad (10.36)$$

$$(\Delta C_{l_\beta})_{z_w} = 0.042(A)^{1/2}(z_w/b)(d_{f_{ave}}/b) \quad (10.37)$$

where: z_w is defined in Figure 10.9

$d_{f_{ave}}$ is given by Eqn.(10.36)

$\{(\Delta C_{l_\beta})/s_t \tan\Lambda_{c/4}\}$ is a wing twist correction factor which is obtained from Fig.10.26.

<u>the horizontal tail contribution is given by:</u>

$$C_{l_{\beta_h}} = (C_{l_{\beta_{hf}}})(S_h b_h/Sb) \quad (10.38)$$

with: $C_{l_{\beta_{hf}}}$ the horizontal tail dihedral effect as computed from Equation (10.34) with appropriate substitution of tail-fuselage for wing-fuselage parameters.

S_h is the horizontal tail area

b_h is the horizontal tail span

<u>the vertical tail contribution is given by:</u>

$$C_{l_{\beta_v}} = (C_{y_\beta})\{z_v \cos\alpha - l_v \sin\alpha)/b\} \quad (10.39)$$

with: C_{y_β} given by Eqn.(10.28)

z_v and l_v defined in Figure (10.27)

3.) C_{n_β}

The yawing-moment-due-to-sideslip derivative, C_{n_β} (also called static directional stability) may be computed from:

$$C_{n_\beta} = C_{n_{\beta_w}} + C_{n_{\beta_f}} + C_{n_{\beta_v}} \quad (10.40)$$

where: <u>the wing contribution</u> is important only at high angles of attack. Ref.9 contains a method for estimating this contribution in such cases. For preliminary design purposes:

$$C_{n_{\beta_w}} = 0 \qquad (10.41)$$

<u>the fuselage contribution is found from:</u>

$$C_{n_{\beta_f}} = -57.3K_N K_{R_l}(S_{f_s} l_f/Sb) \qquad (10.42)$$

with: K_N an empirical factor determined from Figure 10.28

K_{R_l} a factor dependent on Reynold's Number and obtained from Figure 10.29

S_{f_s} and l_f are defined in Figure 10.28.

<u>the vertical tail contribution is found from:</u>

$$C_{n_{\beta_v}} = -(C_{y_{\beta_v}})(l_v\cos\alpha + z_v\sin\alpha)/b \qquad (10.43)$$

where: $C_{y_{\beta_v}}$ is found from Eqn.(10.28)

l_v and z_v are given in Figure 10.27.

<u>10.2.4.2 Thrust versus sideslip derivative:</u> $C_{n_{T_\beta}}$

As suggested in Table 10.1, the only contribution which is of some significance is the yawing-moment-due-to-thrust-in-sideslip derivative. $C_{n_{T_\beta}}$.

<u>For Propeller Driven Airplanes:</u>

$$C_{n_{T_\beta}} = \qquad (10.44)$$

$$\sum_{i=1}^{i=n}[((dC_N/d\alpha)_{p_i})(0.79)(D_{p_i})^2(l_{p_i})]/Sb]$$

where: all terms are as defined on page 342.

Figure 10.28 Factor Accounting for Wing-Fuselage Interference with Directional Stability

For Jet Driven Airplanes:

$$C_{n_{T_\beta}} = \text{Sum}_{i=1}^{i=n}[(0.035(m_{a_i})(l_{n_i}))/Sb\rho U_1] \qquad (10.45)$$

where: all terms are as defined on page 343.

Figure 10.29 Effect of Fuselage Reynold's Number on Wing-Fuselage Directional Stability

10.2.5. Rate of Angle-of-Sideslip Derivatives: $C_{y_{\dot\beta}}$, $C_{l_{\dot\beta}}$ and $C_{n_{\dot\beta}}$

Table 10.1 identifies the required rate-of-angle-of-sideslip derivatives. According to Reference 9, only the vertical tail contributes significantly to these rate derivatives.

NOTE: All methods presented for the rate-of-sideslip derivatives are valid only for the subsonic speed regime. For other speed ranges the reader should consult Ref.9.

1. $C_{y_{\dot\beta}}$

The sideforce-due-to-rate-of-sideslip derivative, $C_{y_{\dot\beta}}$ may be estimated from:

$$C_{y_{\dot\beta}} = \tag{10.46}$$

$$2(C_{L_{\alpha_v}})(d\sigma/d\beta)(S_v/S)(l_p\cos\alpha_f + z_p\sin\alpha_f)/b$$

where: $C_{L_{\alpha_v}}$ is found with the method indicated under Equation (10.28)

$$d\sigma/d\beta = \tag{10.47}$$

$$(\sigma_{\beta_\alpha})\alpha_f + (\sigma_{\beta_\Gamma})(\Gamma/57.3) - (\sigma_{\beta_{\epsilon_t}})\epsilon_t + (\sigma_{\beta_{wf}})$$

where: σ_{β_α} is the sidewash contribution due to angle of attack. It is found from Figures 10.30.

α_f is the angle of attack of the fuselage

σ_{β_Γ} is the sidewash contribution due to wing dihedral. It is found from Figures 10.31.

Γ is the wing dihedral angle as defined in Figure 10.7.

$\sigma_{\beta_{\epsilon_t}}$ is the sidewash contribution due to wing twist as obtained from Figures 10.32.

ϵ_t is the wing twist angle as shown in Figure 10.26.

$\sigma_{\beta_{wf}}$ is the sidewash effect due the fuselage as

Figure 10.30a Sidewash Contribution due to Angle of Attack

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Figure 10.30b Sidewash Contribution due to Angle of Attack

Figure 10.30c Sidewash Contribution due to Angle of Attack

Figure 10.30d Sidewash Contribution due to Angle of Attack

Figure 10.31 Sidewash Contribution due to Wing Dihedral

Figure 10.32a Sidewash Contribution due to Wing Twist

Figure 10.32b Sidewash Contribution due to Wing Twist

Figure 10.32c Sidewash Contribution due to Wing Twist

Figure 10.32d Sidewash Contribution due to Wing Twist

Figure 10.33a Sidewash Contribution due to the Fuselage

Figure 10.33b Sidewash Contribution due to the Fuselage

Figure 10.33c Sidewash Contribution due to the Fuselage

Figure 10.33d Sidewash Contribution due to the Fuselage

Figure 10.33e Sidewash Contribution due to the Fuselage

obtained from Figures 10.33. Note that the data are presented for a low wing as shown in Figure 10.9a. For a high wing (Fig.10.9b) this term becomes negative because z_w changes sign.

z_p and l_p are defined in Figure 10.34.

2.) $C_{l_{\dot{\beta}}}$

The rolling-moment-due-to-rate-of-sideslip derivative, $C_{l_{\dot{\beta}}}$ may be estimated from:

$$C_{l_{\dot{\beta}}} = C_{y_{\dot{\beta}}}(z_p \cos\alpha_f - l_p \sin\alpha_f)/b \qquad (10.48)$$

where: $C_{y_{\dot{\beta}}}$ is found from Eqn.(10.46)

3.) $C_{n_{\dot{\beta}}}$

The yawing-moment-due-to-rate-of-sideslip derivative, $C_{n_{\dot{\beta}}}$ may be estimated from:

$$C_{n_{\dot{\beta}}} = C_{y_{\dot{\beta}}}(l_p \cos\alpha_f + z_p \sin\alpha_f)/b \qquad (10.49)$$

where: $C_{y_{\dot{\beta}}}$ is found from Eqn.(10.46)

Figure 10.34 Definition of z_p and l_p in Eqn.(10.46)

10.2.6 Roll Rate Derivatives: C_{y_p}, C_{l_p} and C_{n_p}

Table 10.1 identifies the required roll rate derivatives.

NOTE: All methods presented for estimation of the roll rate derivatives apply only in the subsonic speed range. For other speed ranges Ref.9 should be used.

1.) C_{y_p}

The side-force-due-to-roll-rate derivative, C_{y_p} is primarily influenced by the vertical tail and may be determined from:

$$C_{y_p} = 2(C_{y_{\beta_v}})(z_v \cos\alpha - l_v \sin\alpha)/b \qquad (10.50)$$

where: $C_{y_{\beta_v}}$ is found from Eqns (10.28) or (10.32).

z_v and l_v are defined in Figure 10.27.

2.) C_{l_p}

The rolling-moment-due-to-roll-rate derivative, C_{l_p} (also called the roll damping derivative) may be found from:

$$C_{l_p} = C_{l_{p_w}} + C_{l_{p_h}} + C_{l_{p_v}} \qquad (10.51)$$

where: the wing contribution is given by:

$$C_{l_{p_w}} = (\beta C_{l_p}/k)_{C_L=0}(k/\beta) * \qquad (10.52)$$

$$* \{(C_{L_{\alpha_w}})_{C_L}/(C_{L_{\alpha_w}})_{C_L=0}\} * \{(C_{l_p})_{\Gamma}/(C_{l_p})_{\Gamma=0}\} + (\Delta C_{l_p})_{drag}$$

where: $(\beta C_{l_p}/k)_{C_L=0}$ is the roll damping parameter at zero lift which is obtained from Figures 10.35

Note: the parameter β in Eqn.(10.52) is NOT sideslip! In this case:

$$\beta = (1 - M^2)^{1/2} \qquad (10.53)$$

Figure 10.35 Roll Damping Parameter

$$k = (C_{l_\alpha})_M \beta/2\pi \qquad (10.54)$$

$(C_{L_{\alpha_w}})_{C_L=0}$ is the wing lift-curve slope at zero lift as obtained from Equation (8.22).

$(C_{L_\alpha})_{C_L}$ is the wing lift-curve slope at any lift coefficient. It is obtained as the local slope of the wing C_L versus α curve as obtained from 8.1.3.5 or from 8.1.4.4.

$\{(C_{l_p})_\Gamma/(C_{l_p})_{\Gamma=0}\}$ is the dihedral effect parameter which may be obtained from:

$$\{(C_{l_p})_\Gamma/(C_{l_p})_{\Gamma=0}\} =$$

$$\{1 - 4z_w/b\sin\Gamma + 12(z_w/b)^2(\sin\Gamma)^2 \qquad (10.55)$$

with: Γ defined in Figure 10.7

z_w defined in Figure 10.9

The wing drag contribution to roll damping is given by:

$$(\Delta C_{l_p})_{drag} = \qquad (10.56)$$

$$\{(C_{l_p})_{C_{D_L}}/(C_{L_w})^2\}(C_{L_w})^2 - 0.125C_{D_{o_w}}$$

with: $\{(C_{l_p})_{C_{D_L}}/(C_L)^2\}$, the drag-due-to-lift roll damping parameter as found from Figure 10.36

C_{L_w} is the wing lift coefficient as obtained from:

$$C_{L_w} = C_{L_1} - C_{L_h} - C_{L_c} \qquad (10.57)$$

In preliminary design it is acceptable to set:

$$C_{L_1} = C_{L_w} \qquad (10.58)$$

$C_{D_{o_w}}$ is the wing zero-lift drag coefficient as obtained from Chapter 4.

Figure 10.36 Drag-due-to-lift Roll Damping Parameter

Figure 10.37 Effect of Wing Twist on C_{n_p}

The fuselage contribution to C_{l_p} tends to be negligible for airplanes for which $d_f/b<0.3$. Most airplane configurations satisfy this criterion.

<u>the horizontal tail contribution is given by:</u>

$$C_{l_{p_h}} = 0.5(C_{l_p})_h(S_h/S)(b_h/b)^2 \qquad (10.59)$$

where: $(C_{l_p})_h$ is the roll-damping derivative of the horizontal tail based on its own reference geometry. It is obtained from Eqn.(10.52) with appropriate substitution of horizontal tail parameters for wing parameters.

<u>the vertical tail contribution is given by:</u>

$$C_{l_{p_v}} = 2(z_v/b)^2 C_{y_{\beta_v}} \qquad (10.60)$$

where: z_v is defined in Figure 10.27

$C_{y_{\beta_v}}$ is found from Eqn.(10.28) or from Eqn.(10.32)

3.) C_{n_p}

The yawing-moment-due-to-roll-rate derivative, C_{n_p} may be determined from:

$$C_{n_p} = C_{n_{p_w}} + C_{n_{p_v}} \qquad (10.61)$$

where: <u>the wing contribution is given by:</u>

$$C_{n_{p_w}} = -\{(C_{n_p}/C_L)_{\substack{C_L=0\\M}}\}C_L \qquad (10.62)$$

$$+ (C_{n_p}/\epsilon_t)\epsilon_t + [(\Delta C_{n_p}/\alpha_{\delta_f})(\delta_f)](\alpha_{\delta_f})\delta_f$$

with:

$$\left(\frac{C_{n_p}}{C_L}\right)_{\substack{C_L=0\\M}} = \left(\frac{A+4\cos\Lambda_{c/4}}{AB+4\cos\Lambda_{c/4}}\right)\left[\frac{AB+\frac{1}{2}(AB+\cos\Lambda_{c/4})\tan^2\Lambda_{c/4}}{A+\frac{1}{2}(A+\cos\Lambda_{c/4})\tan^2\Lambda_{c/4}}\right]\left(\frac{C_{n_p}}{C_L}\right)_{\substack{C_L=0\\M=0}} \qquad (10.63)$$

where: $B = (1 - M^2(\cos\Lambda_{c/4})^2)^{1/2}$ (10.64)

and with:

$$\left(\frac{C_{n_p}}{C_L}\right)_{\substack{C_L=0\\ M=0}} = -\frac{1}{6}\,\frac{A+6(A+\cos\Lambda_{c/4})\left(\frac{\bar{x}}{c}\,\frac{\tan\Lambda_{c/4}}{A} + \frac{\tan^2\Lambda_{c/4}}{12}\right)}{A+4\cos\Lambda_{c/4}}\ (\mathrm{rad}^{-1}) \qquad (10.65)$$

C_{n_p}/ϵ_t is the wing twist contribution as given by Figures 10.37.

ϵ_t is the wing twist angle defined in Fig.10.37.

$(\Delta C_{n_p}/\alpha_{\delta_f}\delta_f)$ is the contribution due to symmetrical flap deflection as found from Figures 10.38.

$$\alpha_{\delta_f} = \Delta c_l / c_{l_\alpha}\delta_f \qquad (10.66)$$

with: Δc_l determined from 8.1.2.1 for the type of flap used

c_{l_α} is the airfoil (flaps-up) lift-curve-slope as found from 8.1.1.2

δ_f is the flap deflection employed

<u>the vertical tail contribution</u> is given by:

$$C_{n_{p_v}} = \qquad (10.67)$$

$$-(2/b^2)(l_v\cos\alpha + z_v\sin\alpha)(z_v\cos\alpha - l_v\sin\alpha - z_v)C_{y_{\beta_v}}$$

where: z_v and l_v are defined in Figure 10.27

$C_{y_{\beta_v}}$ is found from Eqns (10.28) or (10.32)

Figure 10.38 Effect of Symmetrical Flap Deflection on C_{n_p}

10.2.7 Pitch Rate Derivatives: C_{D_q}, C_{L_q} and C_{m_q}

Table 10.1 identifies the required pitch rate derivatives.

NOTE: All methods presented for the pitch rate derivatives are valid only for the subsonic speed regime. For other speed ranges the reader should consult Ref.9.

1.) C_{D_q}

The drag-due-to-pitch-rate derivative, C_{D_q} is negligible for almost all airplanes:

$$C_{D_q} = 0 \tag{10.68}$$

2.) C_{L_q}

The lift-due-to-pitch-rate derivative, C_{L_q} may be estimated from:

$$C_{L_q} = C_{L_{q_w}} + C_{L_{q_h}} + C_{L_{q_c}} \tag{10.69}$$

where: the wing contribution is given by:

$$C_{L_{q_w}} = \tag{10.70}$$

$$[(A + 2\cos\Lambda_{c/4})/(AB + 2\cos\Lambda_{c/4})](C_{L_{q_w}})_{M=0}$$

where: B is given by Eqn.(10.64)

$$(C_{L_{q_w}})_{M=0} = (0.5 + 2x_w/\bar{c})c_{l_{\alpha_w}} \tag{10.71}$$

with: x_w defined in Figure 10.39

$c_{l_{\alpha_w}}$ is the average wing airfoil lift-curve-slope. The wing airfoil lift curve slope at the $\bar{c}$ may be used.

the horizontal tail contribution is given by:

$$C_{L_{q_h}} = 2(C_{L_{\alpha_h}})\eta_h\bar{V}_h \qquad (10.72)$$

where: $C_{L_{\alpha_h}}$ is found from Eqn.(8.22) with appropriate substitution of horizontal tail parameters for wing parameters.

η_h is found from 8.1.5.2

$\bar{V}_h$ is found from Eqn.(10.23)

the canard contribution is found from:

$$C_{L_{q_c}} = -2(C_{L_{\alpha_c}})\eta_c\bar{V}_c \qquad (10.73)$$

where: $C_{L_{\alpha_c}}$ is found from Eqn.(8.22) with appropriate substitution of canard parameters for wing parameters.

η_c is found from 8.1.5.2

$$\bar{V}_c = (\bar{x}_{ac_c} + \bar{x}_{cg})S_c/S \qquad (10.74)$$

where: x_{ac_c} and x_{cg} are shown in Fig.10.39. Note that x_{ac_c} is positive, while x_{cg} is negative as shown!!!

3.) C_{m_q}

The pitching-moment-due-to-pitch-rate derivative, C_{m_q} (also called pitch damping derivative) may be estimated from:

$$C_{m_q} = C_{m_{q_w}} + C_{m_{q_h}} + C_{m_{q_c}} \qquad (10.75)$$

where: the wing contribution is found from:

$$C_{m_{q_w}} = \qquad (10.76)$$

with:

$$(C_{m_{q_w}})_{M=0} = \qquad (10.77)$$

where: K_w is found from Figure 10.40.

<u>the horizontal tail contribution is given by:</u>

$$C_{m_{q_h}} = -2(C_{L_{\alpha_h}})\eta_h\bar{V}_h(\bar{x}_{ac_h} - \bar{x}_{cg}) \qquad (10.78)$$

where: x_{ac_h} is defined in Figure 10.40

All other quantities were defined before!

<u>the canard contribution is given by:</u>

$$C_{m_{q_c}} = -2(C_{L_{\alpha_c}})\eta_c\bar{V}_c(\bar{x}_{ac_c} + \bar{x}_{cg}) \qquad (10.79)$$

where: all quantities have been defined before!

<u>NOTE:</u> many airplanes have pylon mounted nacelles. These pylons contribute to the pitch rate derivatives in the same manner as any empennage surface.

<u>Figure 10.39 Definition of Geometric Parameters</u>

Figure 10.40 Correction Constant for Wing Contribution to Pitch Damping

10.2.8 Yaw Rate Derivatives: C_{y_r}, C_{l_r} and C_{n_r}

Table 10.1 identifies the required yaw rate derivatives.

NOTE: All methods presented for the yaw rate derivatives are valid only for the subsonic speed regime. For other speed ranges the reader should consult Ref.9.

1.) C_{y_r}

The side-force-due-to-yaw-rate derivative, C_{y_r} is primarily influenced by the vertical tail and may be determined from:

$$C_{y_r} = -2(C_{y_{\beta_v}})(l_v\cos\alpha + z_v\sin\alpha)/b \qquad (10.80)$$

where: $C_{y_{\beta_v}}$ is found from Eqns (10.28) or (10.32).

l_v and z_v are defined in Figure 10.27.

2.) C_{l_r}

The rolling-moment-due-to-yaw-rate derivative, C_{l_r} may be estimated from:

$$C_{l_r} = C_{l_{r_w}} + C_{l_{r_v}} \qquad (10.81)$$

where: the wing contribution is found from:

$$C_{l_{r_w}} = (C_{L_w})(C_{l_r}/C_L)_{\substack{C_L=0\\M}} + (\Delta C_{l_r}/\Gamma)\Gamma +$$

$$+ (\Delta C_{l_r}/\epsilon_t)\epsilon_t + (\Delta C_{l_r}/\alpha_{\delta_f})(\alpha_{\delta_f})\delta_f \qquad (10.82)$$

where: $(C_{l_r})_{\substack{C_L=0\\M}}$, the slope of the rolling moment due to roll rate at zero lift is found from:

$$\left(\frac{C_{l_r}}{C_L}\right)_{\substack{C_L=0\\M}} = \frac{1 + \frac{A(1-B^2)}{2B(AB+2\cos\Lambda_{c/4})} + \frac{AB+2\cos\Lambda_{c/4}}{AB+4\cos\Lambda_{c/4}}\,\frac{\tan^2\Lambda_{c/4}}{8}}{1 + \frac{A+2\cos\Lambda_{c/4}}{A+4\cos\Lambda_{c/4}}\,\frac{\tan^2\Lambda_{c/4}}{8}}\left(\frac{C_{l_r}}{C_L}\right)_{\substack{C_L=0\\M=0}} \qquad (10.83)$$

with B determined by Eqn.(10.64)

and $(C_{l_r}/C_L)_{C_L=0}$ is the slope of the low-speed rolling moment due to yaw rate at zero lift found from Figure 10.41.
$M=0$

C_{L_w} is the wing lift coefficient as in Eqn.(10.57) or (10.58)

$$(\Delta C_{l_r}/\Gamma) = 0.083(\pi A \sin\Lambda_{c/4})/(A + 4\cos\Lambda_{c/4}) \quad (10.84)$$

Γ is defined in Figure 10.7

$\Delta C_{l_r}/\epsilon_t$ is the increment in C_{l_r} due to twist and may be determined from Figure 10.42

ϵ_t is the wing twist angle defined in Figure 10.42

$\Delta C_{l_r}/\alpha_{\delta_f}\delta_f$ is the effect of symmetric flap deflection on the rolling moment due to roll rate and is found from Figure 10.43.

α_{δ_f} is found from Eqn.(10.66)

δ_f is the flap deflection used

<u>the vertical tail contribution is found from:</u>

$$C_{l_{r_v}} = \quad (10.85)$$

$$-(2/b^2)(l_v\cos\alpha + z_v\sin\alpha)(z_v\cos\alpha - l_v\sin\alpha)C_{y_{\beta_v}}$$

where: z_v and l_v are defined in Figure 10.27

$C_{y_{\beta_v}}$ is found from Eqn.(10.28) or (10.32)

Reproduced from Reference 9

$$\left(\frac{C_{l_r}}{C_L}\right)_{\substack{C_L=0\\ \mu=0}} \text{ (per rad)}$$

Figure 10.41 Wing Rolling Moment due to Yaw Rate Derivative: Lifting Effect

Figure 10.42 Effect of Wing Twist on C_{l_r}

NOTE:
ΔC_{l_r} per radian
δ_f in degrees

NOTE: $$\frac{\Delta C_{l_r}}{(\alpha_{\delta_f})\,\delta_f} = \left[\frac{\Delta C_{l_r}}{(\alpha_{\delta_f})\,\delta_f}\right]_{\text{outboard}} - \left[\frac{\Delta C_{l_r}}{(\alpha_{\delta_f})\,\delta_f}\right]_{\text{inboard}}$$

Figure 10.43 Effect of Symmetric Flap Deflection on C_{l_r}

3.) C_{n_r}

The yawing-moment-due-to-yaw-rate derivative, C_{n_r} (also called yaw-damping derivative) follows from:

$$C_{n_r} = C_{n_{r_w}} + C_{n_{r_v}} \tag{10.86}$$

where: <u>the wing contribution is found from:</u>

$$C_{n_{r_w}} = \tag{10.87}$$

$$(C_{n_r}/C_L{}^2)(C_{L_w})^2 + (C_{n_r}/C_{D_o})C_{D_{o_w}}$$

where: $(C_{n_r}/C_L{}^2)$ follows from Figure 10.44

C_{L_w} is given by Eqn.(10.57) or (10.58)

(C_{n_r}/C_{D_o}) is found from Figure 10.45

$C_{D_{o_w}}$ is the zero-lift drag coefficient of the wing as found from Chapter 4.

<u>the vertical tail contribution is found from:</u>

$$C_{n_{r_v}} = (2/b^2)(l_v\cos\alpha + z_v\sin\alpha)^2 C_{y_{\beta_v}} \tag{10.88}$$

where: l_v and z_v are defined in Figure 10.27

$C_{y_{\beta_v}}$ is found from Eqnn.(10.20) or (10.32)

Figure 10.44 Wing Yaw Damping Derivative: Lifting Effect

NOTE: $\bar{x}$ is the distance from the c.g. to the a.c., positive for the a.c. aft of the c.g.
$\bar{c}$ is the wing mean aerodynamic chord.

Figure 10.45 Wing Yaw Damping Derivative: Drag Effect

10.3 CONTROL DERIVATIVES

In Table 10.1 the following control derivatives are identified: elevator, aileron and rudder. Because in most airplanes the horizontal stabilizer itself is used as a control device, its derivatives must also be considered. In addition, many recent fighter airplanes and several small commercial airplanes sprout canards, with or without a separate control surface. Many airplanes use spoilers and differential stabilizers as well as ailerons for lateral control.

For these reasons, the following control (power) derivatives are considered in this Section:

10.3.1 Stabilizer Control Derivatives
10.3.2 Elevator Control Derivatives
10.3.3 Canard Control Derivatives
10.3.4 Canardvator Control Derivatives
10.3.5 Aileron Control Derivatives
10.3.6 Spoiler Control Derivatives
10.3.7 Differential Stabilizer Control Derivatives
10.3.8 Rudder Control Derivatives

The methods presented in this section apply only to the subsonic speed regime. For other speed regimes Ref.9 should be consulted.

10.3.1 Stabilizer Control Derivatives: $C_{D_{i_h}}$, $C_{L_{i_h}}$ and $C_{m_{i_h}}$

For a discussion of preliminary sizing of the horizontal stabilizer the reader is referred to Chapters 8 and 11 of Part II.

The methods to be presented apply to cases where the ratio of stabilizer span to local fuselage width is larger than 4.0.

1.) $C_{D_{i_h}}$

The drag-due-to-stabilizer-incidence derivative, $C_{D_{i_h}}$ may be estimated from:

$$C_{D_{i_h}} = 2\{(C_{L_h})/\pi A_h e_h\}(C_{L_{\alpha_h}})\eta_h \qquad (10.89)$$

where: C_{L_h} is the lift coefficient carried by the hori-

zontal stabilizer, based on wing area, S. In preliminary design it is acceptable to set:

$$C_{L_h} = \Delta C_{L_h}, \tag{10.90}$$

the tail lift increment (based on S) required for trim as defined on p.104. It may be determined with Eqn.(8.119)

A_h is the horizontal tail aspect ratio

e_h is found from p.69.

$C_{L_{\alpha_h}}$ is found from Eqn.(8.22) with appropriate substitution of horizontal tail parameters for wing parameters

η_h is found from p.269 or p.271.

2.) $C_{L_{i_h}}$

The lift-due-to-stabilizer-incidence derivative may be estimated from:

$$C_{L_{i_h}} = \eta_h (S_h/S) C_{L_{\alpha_h}} \tag{10.91}$$

where: all quantities have been defined before.

3.) $C_{m_{i_h}}$

The pitching-moment-due-to-stabilizer-incidence derivative, $C_{m_{i_h}}$ (also called stabilizer control power) may be found from:

$$C_{m_{i_h}} = -(C_{L_{\alpha_h}})\eta_h \bar{V}_h \tag{10.92}$$

where: $\bar{V}_h$ is defined by Eqn.(10.23)

all other quantities were defined before.

10.3.2 Elevator Control Derivatives: $C_{D_{\delta_e}}$, $C_{L_{\delta_e}}$ and $C_{m_{\delta_e}}$

For a discussion of preliminary sizing of the elevator the reader is referred to Chapter 8 of Part II.

Although many airplanes carry approximately full span elevators on the horizontal stabilizer, exceptions do occur. For that reason the elevator should be thought of as a partial span, plain flap.

1.) $C_{D_{\delta_e}}$

The drag-due-to-elevator derivative, $C_{D_{\delta_e}}$ may be estimated from:

$$C_{D_{\delta_e}} = (\alpha_{\delta_e})C_{D_{i_h}} \tag{10.93}$$

where: $C_{D_{i_h}}$ follows from Eqn.(10.89)

$$(\alpha_{\delta_e}) = \tag{10.94}$$

$$K_b\{c_{l_\delta}/(c_{l_\delta})_{theory}\}(c_{l_\delta})_{theory}*$$

$$*(k'/c_{l_{\alpha_h}})[\{(\alpha_\delta)_{C_L}\}/\{(\alpha_\delta)_{c_l}\}]$$

where: K_b is the elevator (= plain flap) span factor as obtained from page 259

$\{c_{l_\delta}/(c_{l_\delta})_{theory}\}$ is found from Figure 8.15

$(c_{l_\delta})_{theory}$ is found from Figure 8.14

k' is a correction factor which accounts for nonlinearities at high elevator deflection angles. It is found from Figure 8.13

$[\{(\alpha_\delta)_{C_L}\}/\{(\alpha_\delta)_{c_l}\}]$ is found from Figure 8.53.

Note: in Figs 8.13, 8.14, 8.15 and 8.53 use c_e/c_h for c_f/c

$c_{l_{\alpha_h}}$ is the airfoil lift curve slope of the horizontal stabilizer as obtained from 8.1.1.2.

2.) $C_{L_{\delta_e}}$

The lift-due-to-elevator derivative, $C_{L_{\delta_e}}$ may be found from:

$$C_{L_{\delta_e}} = (\alpha_{\delta_e})C_{L_{i_h}} \tag{10.95}$$

where: (α_{δ_e}) is found from Eqn.(10.94)

$C_{L_{i_h}}$ is found from Eqn.(10.91)

3.) $C_{m_{\delta_e}}$

The pitching-moment-due-to-elevator derivative, $C_{m_{\delta_e}}$ (also called elevator control power) may be found from:

$$C_{m_{\delta_e}} = (\alpha_{\delta_e})C_{m_{i_h}} \tag{10.96}$$

where: (α_{δ_e}) is found from Eqn.(10.94)

$C_{m_{i_h}}$ is found from Eqn.(10.92)

10.3.3 Canard Control Derivatives: $C_{D_{i_c}}$, $C_{L_{i_c}}$ and $C_{m_{i_c}}$

For a discussion of preliminary canard sizing the reader is referred to Chapter 11 of Part II.

The methods to be presented apply to cases where the ratio of canard span to local fuselage width is larger than 3.0.

1.) $C_{D_{i_c}}$

The drag-due-to-canard-incidence derivative, $C_{D_{i_c}}$ may be estimated from:

$$C_{D_{i_c}} = 2[(C_{L_c})/\pi A_c e_c](C_{L_{\alpha_c}})\eta_c \qquad (10.97)$$

where: C_{L_c} is the lift coefficient carried by the canard, based on wing area, S. In preliminary design it acceptable to set:

$$C_{L_c} = \Delta C_{L_c}, \qquad (10.98)$$

the canard lift increment (based on S) required for trim as defined on p.104. It may be determined with Eqn.(8.120)

A_c is the canard aspect ratio

e_c is found from p.69.

$C_{L_{\alpha_c}}$ is found from Eqn.(8.22) with appropriate substitution of canard parameters for wing parameters

η_c is found from p.269 or p.271.

2.) $C_{L_{i_c}}$

The lift-due-to-canard-incidence derivative may be estimated from:

$$C_{L_{i_c}} = \eta_c(S_c/S)C_{L_{\alpha_c}} \qquad (10.99)$$

where: all quantities have been defined before.

3.) $C_{m_{i_c}}$

The pitching-moment-due-to-canard-incidence derivative, $C_{m_{i_c}}$ (also called canard control power) may be found from:

$$C_{m_{i_c}} = -(C_{L_{\alpha_c}})\eta_c\bar{V}_c \qquad (10.100)$$

where: $\bar{V}_c$ is defined by Eqn.(10.74)

all other quantities were defined before.

10.3.4 Canardvator Control Derivatives: $C_{D_{\delta_c}}$, $C_{L_{\delta_c}}$ and $C_{m_{\delta_c}}$

Because of a lack of statistical data on canard equipped airplanes, no simple, preliminary design methods are available for the sizing of a canardvator. If the canard is used for trim and control purposes, the trim considerations of Section 8.3 should be used.

Although several canard equipped airplanes carry full span canardvators on the canard, exceptions do occur. For that reason the canardvator should be thought of as a partial span, plain flap.

1.) $C_{D_{\delta_c}}$

The drag-due-to-canardvator derivative, $C_{D_{\delta_c}}$ may be estimated from:

$$C_{D_{\delta_c}} = (\alpha_{\delta_c})C_{D_{i_c}} \tag{10.101}$$

where: $C_{D_{i_c}}$ follows from Eqn.(10.97)

$$(\alpha_{\delta_c}) = \tag{10.102}$$

$$K_b\{c_{l_\delta}/(c_{l_\delta})_{theory}\}(c_{l_\delta})_{theory}*$$

$$*(k'/c_{l_{\alpha_h}})[\{(\alpha_\delta)_{C_L}\}/\{(\alpha_\delta)_{c_l}\}]$$

where: K_b is the canardvator (= plain flap) span factor as obtained from page 259

$\{c_{l_\delta}/(c_{l_\delta})_{theory}\}$ is found from Figure 8.15

$(c_{l_\delta})_{theory}$ is found from Figure 8.14

k' is a correction factor which accounts for nonlinearities at high canardvator deflection angles. It is found from Figure 8.13

$[\{(\alpha_\delta)_{C_L}\}/\{(\alpha_\delta)_{c_l}\}]$ is found from Figure 8.53

<u>Note:</u> in Figs 8.13, 8.14, 8.15 and 8.53 use c_{δ_c}/c_c for c_f/c

$c_{l_{\alpha_c}}$ is the airfoil lift curve slope of the canard as obtained from 8.1.1.2.

2.) $C_{L_{\delta_c}}$

The lift-due-to-canardvator derivative, $C_{L_{\delta_c}}$ may be found from:

$$C_{L_{\delta_c}} = (\alpha_{\delta_c})C_{L_{i_c}} \tag{10.103}$$

where: (α_{δ_c}) is found from Eqn.(10.102)

$C_{L_{i_c}}$ is found from Eqn.(10.99)

3.) $C_{m_{\delta_c}}$

The pitching-moment-due-to-canardvator derivative, $C_{m_{\delta_c}}$ (also called canardvator control power) may be found from:

$$C_{m_{\delta_c}} = (\alpha_{\delta_c})C_{m_{i_c}} \tag{10.104}$$

where: (α_{δ_c}) is found from Eqn.(10.102)

$C_{m_{i_c}}$ is found from Eqn.(10.100)

10.3.5 Aileron Control Derivatives: $C_{y_{\delta_a}}$, $C_{l_{\delta_a}}$ and $C_{n_{\delta_a}}$

For a discussion of preliminary aileron sizing the reader is referred to Chapter 6 of Part II.

1.) $C_{y_{\delta_a}}$

The side-force-due-to-aileron derivative, $C_{y_{\delta_a}}$ is negligible for most conventional aileron arrangements:

$$C_{y_{\delta_a}} = 0 \qquad (10.105)$$

If ailerons are located in close proximity to a vertical tail (F-106) a significant side-force due to aileron deflection may arise. Windtunnel data are recommended to determine such aileron induced side forecs.

2.) $C_{l_{\delta_a}}$

The rolling-moment-due-to-aileron derivative, $C_{l_{\delta_a}}$ (also called roll control power) can be estimated with the following procedure:

Step 1: Determine the inboard span location η_i and the outboard span location η_o for the aileron(s) as fractions of the semi-span of the wing. These data follow from the Class I threeview obtained from Chapter 13, Part II.

Step 2: For full chord ailerons ($c_a/c = 1.0$), anti-symmetrically deflected, and running from: $\eta=0$ to η_i and to η_o respectively, determine the rolling moment effectiveness parameter: $\beta C'_{l_\delta}/k$ from Figures 10.46, with:

β found from Eqn.(10.53)

k found from Eqn.(10.54), where $(c_{l_\alpha})_M$ is

Figure 10.46a Aileron Rolling Moment Parameter

Figure 10.46b Aileron Rolling Moment Parameter

Figure 10.46c Aileron Rolling moment Parameter

determined for the airfoil at the m.g.c of the part of the wing covered by the aileron(s)

A is the wing aspect ratio

$$\Lambda_\beta = \arctan(\tan\Lambda_{c/4}/\beta) \qquad (10.106)$$

Figures 10.46 give the control effectiveness parameter for control spans measured from the plane of symmetry (η=0) outboard. For partial span controls running from η_i to η_o the actual effectiveness parameter is obtained as the difference between the two full span controls. This is illustrated in Figure 10.47.

<u>Step 3:</u> Determine the rolling effectiveness of two full-chord controls anti-symmetrically deflected by:

$$C'_{l_\delta} = (k/\beta)(\beta C'_{l_\delta}/k) \qquad (10.107)$$

<u>Step 4:</u> Determine the rolling effectiveness of the partial-chord controls ($c_a/c < 1.0$), anti-symmetrically deflected from:

$$C_{l_\delta} = (\alpha_{\delta_a})C'_{l_\delta} \qquad (10.108)$$

where:

$$\alpha_{\delta_a} = c_{l_\delta}/(c_{l_\alpha})_a \qquad (10.109)$$

with: c_{l_δ} =

$$\{c_{l_\delta}/(c_{l_\delta})_{theory}\}(c_{l_\delta})_{theory} \qquad (10.110)$$

where: $\{c_{l_\delta}/(c_{l_\delta})_{theory}\}$ is found from Figure 8.15

$(c_{l_\delta})_{theory}$ is found from Figure 8.14

Note: in Figures 8.14 and 8.15, use c_a/c for c_f/c

and where: $(c_{l_\alpha})_a$ is the average airfoil lift-curve-slope of that part of the wing covered by the ailerons. The airfoil at the wing span station corresponding to that of the aileron m.g.c. may be used to compute this.

Step 5: The effect of differential aileron control deflection (for an example, see Part IV, pgs 217 and 220) is taken into account by considering C_{l_δ} of each control as ONE-HALF the anti-symmetric value determined with the aid of Eqn(10.108). The total rolling moment coefficient for differential aileron control deflection is the found from:

$$C_{l_a} = \tag{10.111}$$

$$((C_{l_\delta}/2)_{left} + (C_{l_\delta}/2)_{right})(\delta_{a_{left}} - \delta_{a_{right}})$$

where it must be kept in mind, that a trailing edge down deflection is a positive control deflection!

THE aileron delfection, δ_a of the airplane is defined as:

$$\delta_a = 0.5(\delta_{a_{left}} - \delta_{a_{right}}) \tag{10.112}$$

Step 6: The aileron roll control power derivative now follows from:

$$C_{l_{\delta_a}} = (C_{l_\delta})_{left} + (C_{l_\delta})_{right} \tag{10.113}$$

3.) $C_{n_{\delta_a}}$

The yawing-moment-due-to-aileron derivative, $C_{n_{\delta_a}}$ (also called adverse aileron yaw) may be computed from:

$$C_{n_{\delta_a}} = K_a C_{L_w} C_{l_{\delta_a}} \qquad (10.114)$$

where: K_a is found from Figure 10.48

C_{L_w} is found from Eqns (10.57) or (10.58)

$C_{l_{\delta_a}}$ is given by Eqn.(10.113)

Reproduced from Reference 9

Figure 10.47 Finding the Partial Span Aileron Parameter

Reproduced from Reference 9

Figure 10.48 Correlation Constant for Yawing Moment due to Aileron Deflection

10.3.6 Spoiler Control Derivatives: $C_{Y_{\delta_s}}$, $C_{l_{\delta_s}}$ and $C_{n_{\delta_s}}$

For a discussion of preliminary spoiler sizing the reader is referred to Chapter 6 of Part II.

In this sub-section the following two types of spoilers will be considered:

A) Plug or Flap-Type Spoilers

B) Spoiler-Slot-Deflector Arrangements

Figures 10.49A and B show typical layouts for these spoiler types. Dimensions needed in the determination of spoiler derivatives are also given.

Definition: a positive spoiler deflection is one which results in a positive rolling moment i.e. a roll to the right.

1.) $C_{Y_{\delta_s}}$

The side-force-due-to-spoiler derivative, $C_{Y_{\delta_s}}$ is negligible for most conventional spoiler arrangements, regardless of spoiler type:

$$C_{Y_{\delta_s}} = 0 \qquad (10.115)$$

If spoilers are located in close proximity to a vertical surface a significant side-force due to spoiler deflection may arise. Windtunnel data are recommended to determine such spoiler induced side forces.

2.) $C_{l_{\delta_s}}$

The rolling-moment-due-to-spoiler derivative $C_{l_{\delta_s}}$ (also called spoiler roll control power) depends on the type of spoiler used.

Figure 10.49 General Arrangement for Two Types of Spoiler

Figure 10.50 Spoiler Lift Effectiveness

Figure 10.51 Effect of Spoiler Slot and Deflector on Spoiler Rolling Moment Effectiveness

A) Plug or Flap-Type Spoilers

The rolling-moment-due-to-spoiler derivative, $C_{l_{\delta_s}}$ for a plug or flap-type spoiler can be estimated with the following procedure:

Step 1: Determine the inboard span location η_i and the outboard span location η_o for the spoiler panels as fractions of the semi-span and as fractions of the chord of the wing. These data follow from the Class I three-view obtained from Chapter 13, Part II.

Step 2: Assuming a maximum spoiler plate deflection of 60 degrees, translate the spoiler geometry of Step 1 into one consistent with the spoiler geometry of Figure 10.49A: in other words, determine values for h_s/c and x_s/c.

These quantities may be averaged over the span of the spoiler.

Step 3: From Figure 10.50 find the corresponding value of $\Delta\alpha'_s$.

Step 4: Find the spoiler roll control derivative from:

$$C_{l_{\delta_s}} = (1/120)(C'_{l_\delta})\Delta\alpha'_s \cos\Lambda_{c/4} \qquad (10.116)$$

where: C'_{l_δ} is the rolling-moment effectiveness parameter for full chord, antisymmetrically deflected controls as obtained from Eqn.(10.107). The inboard and outboard ends of these controls are taken to be the same as the inboard and outboard ends of the hinge line of the spoiler.

$\Lambda_{c/4}$ is the wing quarter chord sweep angle.

NOTE: this method is valid only for sweep angles up to 40 degrees. For higher sweep angles, see the method of Ref.9.

B) Spoiler-Slot Deflector Arrangements

For a spoiler-slot-deflector arrangement the roll control power derivative may be determined from:

$$C_{l_{\delta_{ssd}}} = K_{ssd} C_{l_{\delta_s}} \qquad (10.117)$$

where: K_{ssd} follows from Figure 10.51 for a given ratio of spoiler angle to deflector angle, δ_s/δ_d

$C_{l_{\delta_s}}$ is determined from Eqn.(10.116).

3.) $C_{n_{\delta_s}}$

The yawing-moment-due-to-spoiler derivative, $C_{n_{\delta_s}}$ (also called proverse yawing moment due to spoiler) also depends on the type of spoiler used.

A) Plug and Flap Type Spoilers

$$C_{n_{\delta_s}} = (1/60) C_{n_s} \qquad (10.118)$$

where: C_{n_s} is found from Figures 10.52 and 10.53 for straight and for swept wings respectively.

B) Spoiler-Slot-Deflector Arrangements

$$C_{n_{\delta_{ssd}}} = K_{ssd} C_{n_{\delta_s}} \qquad (10.119)$$

where: K_{ssd} is found from Figure 10.54

$C_{n_{\delta_s}}$ follows from Eqn.(10.118).

Figure 10.52 Yawing Moment due to Spoiler for Straight Wings at Low Angles of Attack

Figure 10.53 Yawing Moment due to Spoiler for Swept Wings at Low Angles of Attack

Figure 10.54 Effect of Slot and Deflector on Spoiler Yawing Moment

Figure 10.55 Tail Effectiveness Parameter for Fuselage Mounted Horizontal Stabilizers

10.3.7 Differential Stabilizer Control Derivatives: $C_{y_{i_h}}$, $C_{l_{i_h}}$ and $C_{n_{i_h}}$

Many fighter airplanes require so much lateral control power to meet combat roll requirements that differentially controlled horizontal stabilizers are used in addition to wing mounted lateral control devices. The sizing of the horizontal tail (stabilizer) is normally based on requirements for longitudinal stability and control. The reader should refer to Chapter 8 of Part II for preliminary horizontal tail sizing methods.

In this sub-section it will be assumed that the geometry of the horizontal tail surfaces is known.

Definition: a positive differential stabilizer deflection is one resulting in a positive rolling moment: i.e. left stabilizer up AND i.e. right stabilizer down).

1.) $C_{y_{i_h}}$

The side-force-due-to-differential stabilizer derivative, $C_{y_{i_h}}$ is negligible for many airplanes:

$$C_{y_{i_h}} = 0 \tag{10.120}$$

If the stabilizers are located in close proximity to to a vertical surface a significant side-force due to differential stabilizer deflection may arise. Windtunnel data must be used to determine these side forces.

2.) $C_{l_{i_h}}$

The rolling-moment-due-to-differential stabilizer derivative $C_{l_{i_h}}$ (also called differential stabilizer roll control power) may be found from:

$$C_{l_{i_h}} = 0.5\{1 - (\pi A/57.3)(d\epsilon/d\alpha)\}(\eta_h \bar{q}_h/\bar{q}) \cdot$$

$$\cdot\{(y_{h_e} S_{h_e})/Sb\}(C_{L_{\alpha_h}})_e \tag{10.121}$$

where: $d\epsilon/d\alpha$ is found from Eqn.(8.45)

$(\eta_h \bar{q}_h/\bar{q})$ is found from Figure 10.55)

y_{h_e} is the distance from the exposed stabilizer center of pressure to the airplane centerline, and may be determined from:

$$y_{h_e} = (\eta_{cp} b_{h_e}/2) + r_{fh} \qquad (10.122)$$

where: η_{cp} is found from Figure 10.56. In Figure 10.16, A_{h_e} is the aspect ratio of the exposed horizontal tail

b_{h_e} is the semispan of the exposed horizontal stabilizer

r_{fh} is the radius or one half of the equivalent fuselage width at the point of stabilizer attachment

S_{h_e} is the exposed stabilizer area

$(C_{L_{\alpha_h}})_e$ is the lift-curve slope of the exposed horizontal tail. It is obtained by using the exposed horizontal tail aspect ratio, A_{h_e} in Eqn.(8.22) and substituting other appropriate horizontal tail parameters for wing parameters.

<u>NOTE:</u> This method is valid only for angles of attack below roughly six degrees. At higher angles of attack body shed vortices will interfere with the flow over the horizontal tail A method to account for these vortices is given in Ref.9.

3.) $C_{n_{i_h}}$

No reliable preliminary design methods are available for the estimation of this derivative. The qualitative discussion which follows has been adapted from Ref.9.

Figure 10.57 shows typical relations between aileron and differential tail induced yawing moments. The magnitude of the differential tail induced yawing moment is a strong function of tail height (relative to the vertical tail and relative to the wing): Figure 10.58 illustrates

Figure 10.56 Exposed Stabilizer Center of Pressure Location as Affected by Aspect Ratio, Sweep Angle and Taper Ratio

Figure 10.56 (Cont'd) Exposed Stabilizer Center of Pressure Location as Affected by Aspect Ratio, Sweep Angle and Taper Ratio

Figure 10.57 Effect of Mach Number on the Yaw-to-Roll Ratio of Differential Stabilizers

Figure 10.58 Effect of Tail Location on the Yaw-to-Roll Ratio of Differential Stabilizers

typical trends. In addition, the dihedral angle of the horizontal tail can be significant as shown in Fig.10.59.

Finally, the deflection of wing flaps can have significant influence as well. This is shown in Fig.10.60.

For purposes of preliminary design the reader may wish to 'guestimate' a value of $C_{n_{i_h}}/C_{l_{i_h}}$ from Figs 10.57 through 10.60. Where possible windtunnel data should be used.

Figure 10.59 Effect of Tail Dihedral on the Yaw-to-Roll Ratio of Differential Stabilizers

Figure 10.60 Effect of Flaps on the Yaw-to-Roll Ratio of Differential Stabilizers

10.3.8 Rudder Control Derivatives: $C_{y_{\delta_r}}$, $C_{l_{\delta_r}}$ and $C_{n_{\delta_r}}$

For a discussion of vertical tail sizing and rudder sizing the reader is referred to Chapters 8 and 11 in Part II.

Definition: a positive rudder deflection is one resulting in a negative yawing moment (t.e. rudder to the left)

1.) $C_{y_{\delta_r}}$

The side-force-due-to-rudder derivative, $C_{y_{\delta_r}}$ may be computed from:

$$C_{y_{\delta_r}} = (C_{L_{\alpha_v}})(k'K_b)* \qquad (10.123)$$

$$*(c_{l_\delta}/(c_{l_\delta})_{theory})(c_{l_\delta})_{theory}(S_v/S)$$

where: $C_{L_{\alpha_v}}$ is found from p.386

k' is found from Figure 8.53

K_b is found from page 259

$(c_{l_\delta}/(c_{l_\delta})_{theory})$ is found from Figure 8.15

$(c_{l_\delta})_{theory}$ is found from Figure 8.14

S_v is the effective vertical tail area as defined in Figure 10.13.

2. $C_{l_{\delta_r}}$

The rolling-moment-due-to-rudder derivative, $C_{l_{\delta_r}}$ is found from:

$$C_{l_{\delta_r}} = ((z_v\cos\alpha - l_v\sin\alpha)/b)C_{y_{\delta_r}} \qquad (10.124)$$

where: z_v and l_v are defined in Figure 10.27

$C_{y_{\delta_r}}$ is found from Eqn.(10.122)

3. $C_{n_{\delta_r}}$

The yawing-moment-due-to-rudder derivative, $C_{n_{\delta_r}}$ (also called rudder control power) may be estimated from:

$$C_{n_{\delta_r}} = -C_{y_{\delta_r}} (l_v \cos\alpha + z_v \sin\alpha)/b \qquad (10.125)$$

where: l_v and z_v are defined in Figure 10.27

$C_{y_{\delta_r}}$ is given by Eqn.(10.122).

10.4 HINGEMOMENT DERIVATIVES OF CONTROL SURFACES

The purpose of this section is to present rapid methods for the estimation of hingemoment derivatives of control surfaces and tabs in the subsonic speed range. For methods which apply in the transonic and supersonic speed ranges the reader should consult Reference 9.

The methods to be presented apply only in the linear range of control surface deflections (< 20 degrees at best) and in the linear range of angles of attack (roughly 12 degrees). For nonlinear effects the reader should consult Ref.9 and use windtunnel data wherever possible.

Hingemoment derivatives are used for two purposes:

1. Computing stick, wheel and pedal cockpit control forces so they can be checked against airworthiness requirements.

   Part VII contains methods for computing the cockpit control forces as well as methods for checking with airworthiness standards.

2. Computing actuator force levels so that hydraulic or electro-mechanical actuators can be properly sized.

   Chapter 4 in Part IV contains a discussion of actuator sizing criteria.

Figure 10.61 shows a basic control surface and tab arrangement with the necessary geometric parameters identified. Note the use of the following nomenclature: main surface, control surface and tab, shown in Figure 10.61. This nomenclature applies to the following combinations:

1. horizontal tail + elevator + elevator tab

2. canard + canardvator + canardvator tab

3. wing + aileron + aileron tab

4. vertical tail + rudder + rudder tab

For a detailed discussion of the purpose of various combinations of tabs and control surfaces, the reader should consult Chapter 4 of Part IV.

In most cases, hingemoments are taken about the control surface hingeline: corresponding hingemoment coeffi-

cients and derivatives are denoted:

c_h and c_{h_x} respectively.

In some cases, hingemoments need to be considered about the tab hinge line: for example in servo-tab systems. The corresponding hingemoment coefficients and derivatives in that case are denoted:

$c^t{}_h$ and $c^t{}_{h_y}$ respectively.

The material in this sub-section is organized as follows:

10.4.1 Two-Dimensional Control Surface and Tab Hingemoment Derivatives about the Control Surface Hingeline

10.4.2 Three-Dimensional Control Surface and Tab Hingemoment Derivatives about the Control Surface Hingeline

10.4.3 Two-Dimensional Tab Hingemoment Derivatives about the Tab Hingeline

10.4.4 Three-Dimensional Tab Hingemoment Derivatives about the Tab Hingeline

The methods apply as long as the airflow over the control surface is attached.

<u>NOTES:</u> 1.) All Two-D hingemoment derivatives are based on $c_f{}^2$, where c_f is the plain flap chord defined in Figure 10.61. The plain flap is either the control surface or the tab, depending on the point about which the hingemoments are taken.

2.) All Three-D hingemoment derivatives are based on $(S_{surface})*(\bar{c}_{surface})$, where the area and chord depend on the type of control surface considered: rudder, aileron, elevator, tab, etc.

Figure 10.61 Geometry and Nomenclature Used with Control Surfaces and Tabs

Figure 10.62 Definitions of Trailing Edge Angles

10.4.1 Two-Dimensional Control Surface and Tab Hingemoment Derivatives about the Control Surface Hingeline

The two-dimensional (=airfoil) hingemoment coefficient for a control surface is estimated from:

$$c_h = c_{h_o} + c_{h_\alpha}\alpha + c_{h_\delta}\delta + c_{h_{\delta_t}}\delta_t \qquad (10.126)$$

where: c_{h_o} is the zero-angle-of-attack, zero-control-surface-deflection, zero-tab-angle-deflection hingemoment coefficient. For main surfaces with symmetrical airfoils:

$$c_{h_o} = 0 \qquad (10.127)$$

For main surfaces with cambered airfoils, experimental data should be used.

c_{h_α} is the control surface hingemoment derivative due to angle of attack. It is estimated from 10.4.1.1, where it is called: $c_{h_{\alpha_{bal}}}$

c_{h_δ} is the control surface hingemoment derivative due to control surface deflection. It is obtained from 10.4.1.2

$c_{h_{\delta_t}}$ is the control surface hingemoment derivative due to a tab deflection. It is estimated from 10.4.1.3.

NOTE: The reader should recognize, that depending on the application, the following substitutions must be made in Equation (10.126):

for a wing: $\alpha = \alpha_w$ and $\delta = \delta_a$ or δ_{flap}

for a horizontal tail: $\alpha = \alpha_h$ and $\delta = \delta_e$

for a canard: $\alpha = \alpha_c$ and $\delta = \delta_c$

for a vertical tail: $\alpha = \beta$ and $\delta = \delta_r$

10.4.1.1 Two-D control surface hingemoment derivative due to angle of attack: c_{h_α}

The Two-D control surface hingemoment due to angle of attack derivative, c_{h_α} is determined with the following procedure:

Step 1: Check whether or not the following trailing edge angle condition is satisfied:

$$\tan(\Phi'_{te}/2) = \tan(\Phi''_{te}/2 = \tan(\Phi_{te}/2) = t/c \qquad (10.128)$$

where: Φ'_{te} is the trailing-edge angle defined as the angle between straight lines passing through points at 90 and 99 percent of the chord on the upper and lower airfoil surfaces

Φ''_{te} is the trailing-edge angle defined as the angle between straight lines passing through points at 95 and 99 percent of the chord on the upper and lower airfoil surfaces

Φ_{te} is the trailing-edge angle defined as the angle between tangents to the upper and lower airfoil surfaces at the trailing-edge

Figure 10.62 illustrates these angles.

Condition (10.128) is satisfied whenever the upper and lower surface lines of the control surface are straight.

Step 2: Determine c'_{h_a} from:

$$c'_{h_a} = \qquad (10.129)$$

$$\{c'_{h_a}/(c_{h_a})_{theory}\}(c_{h_a})_{theory}$$

where: $\{c'_{h_a}/(c_{h_a})_{theory}\}$ is found from Fig.10.63a which applies only to radius nose, sealed gap, plain flap type control surfaces

$(c_{h_a})_{theory}$ is found from Figure 10.63b. The parameter $\{c_{l_a}/(c_{l_a})_{theory}\}$ in Fig.10.63a is itself found from Figure 10.64a.

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Figure 10.63 Two-Dimensional Control Surface Hingemoment Derivative due to Angle of Attack

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Figure 10.64 Effect of Airfoil Thickness and Trailing Edge Angle on Lift Curve Slope

Step 3: If condition (10.126) is not satisfied, compute c''_{h_α} from:

$$c''_{h_\alpha} = (c'_{h_\alpha}) + \tag{10.130}$$

$$+ 2(c_{l_\alpha})_{theory}[1 - \{c_{l_\alpha}/(c_{l_\alpha})_{theory}\}] *$$

$$*\{\tan(\Phi''_{te}/2) - (t/c)\}$$

where: (c'_{h_α}) is obtained from Eqn.(10.129)

$(c_{l_\alpha})_{theory}$ is obtained from Figure 10.64b at the appropriate thickness ratio, t/c

$\{c_{l_\alpha}/(c_{l_\alpha})_{theory}\}$ is obtained from Figure 10.64a

Φ''_{te} is defined under Step 1.

Note that if condition (10.128) is satisfied, and only in that case:

$$c''_{h_\alpha} = c'_{h_\alpha} \tag{10.131}$$

Step 4: Since the value for c'_{h_α} or c''_{h_α} as found from either Step 2 or Step 3 applies only to round-nose control surfaces, corrections must be made which account for different nose shapes and for aerodynamic balance.

The corrected value for c_{h_α} is found from:

$$(c_{h_\alpha})_{bal} = \tag{10.132}$$

$$(c''_{h_\alpha})\{(c_{h_\alpha})_{bal}/c''_{h_\alpha}\}$$

where: c''_{h_α} is obtained from Step 3.

$\{(c_{h_\alpha})_{bal}/c''_{h_\alpha}\}$ is found from Figure 10.65a for various nose shapes and

Figure 10.65a Effect of Nose Shape and Balance on the Two-Dimensional Hingemoment Derivative due to Angle of Attack

Figure 10.65b Nose Shape Examples for 35 Percent Balance

at the proper balance ratio. Nose shapes and balance ratio are illustrated and defined in Figure 10.65b.

Step 5: Correct the hingemoment derivative for the effect of Mach Number:

$$(c_{h_\alpha})_M = (c_{h_\alpha})_{bal}/(1 - M^2)^{1/2} \qquad (10.133)$$

Step 6: Hingemoments also depend on whether or not the control surface and/or tab gaps are closed (i.e sealed). Figure 10.66 illustrates the difference between open and closed gaps.

Figure 10.67 may be used to introduce corrections for open gaps, depending on gap-size for the control surface.

Figure 10.68a may be used to correct for the effect of gap-size in the case of an unsealed tab.

NOTE: the Step 6 corrections are applied to $(c_{h_\alpha})_M$ by using ratios obtained from Figures 10.67 or 10.68a as appropriate.

Figure 10.66 Examples of Closed (Sealed) and Open (Unsealed) Control Surface and Tab Gaps

Figure 10.67 Effect of Gap Size and Overhang on Two-Dimensional Hingemoment Derivatives

Figure 10.68 Effect of Tab Gap Size and Tab Overhang On Two-Dimensional Hingemoment Derivatives

10.4.1.2 Two-D control surface hingemoment derivative due to control surface deflection: c_{h_δ}

The Two-D control surface hingemoment due to control surface deflection derivative, c_{h_δ} is determined with the following procedure:

Step 1: This is a repeat of Step 1 in 10.4.1.1.

Step 2: Determine c'_{h_δ} from:

$$c'_{h_\delta} = \{c'_{h_\delta}/(c_{h_\delta})_{theory}\}(c_{h_\delta})_{theory} \qquad (10.134)$$

where: $\{c'_{h_\delta}/(c_{h_\delta})_{theory}\}$ is found from Fig.10.69a which applies only to radius nose, sealed gap, plain flap type control surfaces

$(c_{h_\delta})_{theory}$ is found from Figure 10.69b. The parameter $\{c_{l_\alpha}/(c_{l_\alpha})_{theory}\}$ in Fig.10.69a is itself found from Figure 10.64a.

Step 3: If condition (10.128) is not satisfied, compute c''_{h_δ} from:

$$c''_{h_\delta} = (c'_{h_\delta}) + 2(c_{l_\delta})_{theory} * \qquad (10.135)$$

$$*[1 - \{c_{l_\delta}/(c_{l_\delta})_{theory}\}]\{\tan(\Phi''_{te}/2) - (t/c)\}$$

where: (c'_{h_δ}) is obtained from Eqn.(10.134)

$(c_{l_\delta})_{theory}$ is obtained from Figure 8.14 at the appropriate thickness ratio, t/c

$\{c_{l_\delta}/(c_{l_\delta})_{theory}\}$ is obtained from Figure 8.15

Figure 10.69 Two-Dimensional Control Surface Hingemoment Derivative due to Control Surface Deflection

Φ''_{te} is defined under Step 1 in 10.4.1.1.

NOTE: If the control surface trailing edge is beveled as in Fig.10.70, the value of Φ''_{te} should be set equal to the angle of the bevel!

Note that if condition (10.128) is satisfied, any in that case:

$$c''_{h_\delta} = c'_{h_\delta} \qquad (10.136)$$

ep 4: Since the value for c_{h_δ} obtained under either Step 2 or Step 3 applies only to round-nose control surfaces, corrections must be made which account for different nose shapes and for aerodynamic balance.

The corrected value for c_{h_δ} is found from:

$$(c_{h_\delta})_{bal} = \qquad (10.137)$$

$$(c''_{h_\delta})\{(c_{h_\delta})_{bal}/c''_{h_\delta}\}$$

ere: c''_{h_δ} is obtained from Step 3.

$\{(c_{h_\delta})_{bal}/c''_{h_\delta}\}$ is found from Figure 10.71 for various nose shapes and at the proper balance ratio. Nose shapes and balance ratio are illustrated and defined in Figure 10.65b.

ep 5: Correct the hingemoment derivative for the effect of Mach Number:

$$(c_{h_\delta})_M = (c_{h_\delta})_{bal}/(1 - M^2)^{1/2} \qquad (10.138)$$

ep 6: Hingemoments also depend on whether or not the control surface gaps are closed (i.e. i.e sealed). Figure 10.66 illustrates the the difference between open and closed gaps.

Figures 10.67 and 10.68b may be used to introduce corrections for open gaps, depending on gap size for the control surface and for the tab.

Figure 10.70 Example of Control Surface Beveling

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Figure 10.71 Effect of Nose Shape and Balance on the Two-Dimensional Control Surface Hingemoment Derivative due to Control Surface Deflection

10.4.1.3 Two-D control surface hingemoment derivative due to tab deflection: $c_{h_{\delta_t}}$

The Two-D control surface hingemoment derivative due to tab deflection, $c_{h_{\delta_t}}$, measured at constant angle of attack and at constant control surface deflection is found from:

$$c_{h_{\delta_t}} = (c_{h_{\delta_t}})_{c_l,\delta} + - \{(c_{h_{c_l}})_{\delta_t,\delta}\} * \{(c_{l_\alpha})_{\delta_t,\delta}\} * \{(\alpha_{\delta_t})_{c_l,\delta}\} \qquad (10.139)$$

where: $(c_{h_{\delta_t}})_{c_l,\delta}$ is the change in control surface hingemoment coefficient due to tab deflection at constant lift and at constant control surface deflection. It is obtained from Figure 10.72.

$(c_{h_{c_l}})_{\delta_t,\delta}$ is the change in control surface hingemoment due to lift at constant tab deflection and at constant control surface deflection. It is obtained from Figure 10.73.

$(c_{l_\alpha})_{\delta_t,\delta}$ is the airfoil lift curve slope of the main surface to which the control surface is attached. It is found in 8.1.1.2.

$(\alpha_{\delta_t})_{c_l,\delta}$ is the change in angle of attack due to a change in tab deflection. It is found from Figure 10.74.

The derivatives in Equation (10.139) do not account for the effects of tab nose shape and tab gaps: they apply to round nose tabs with sealed gaps only! To account for the effect of differing tab nose shapes and for unsealed tab gaps, the reader should correct $c_{h_{\delta_t}}$ by using appropriate ratios obtained from Figs 10.75 and 10.76.

Figure 10.72 Effect of Control Surface Size and Tab Size on the Change in Control Surface Hingemoment Coefficient due to Tab Deflection at Constant Lift Coefficient and at Constant Control Surface Deflection

Figure 10.73 Effect of Control Surface Size on the Change in Control Surface Hingemoment due to Lift at Constant Tab Deflection and at Constant Control Surface Deflection

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Figure 10.74 Change in Angle of Attack due to a Change in Tab Deflection

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TAB GAP	TRANSITION STRIPS
.004c	AT .01c
SEALED	AT .01c
.004c	OFF
SEALED	OFF

Figure 10.75 Effect of Tab Nose Shape and Tab Overhang on the Control Surface Hingemoment Derivative due to Tab Deflection

BLUNT
ROUND
ELLIPTICAL
SHARP

COPIED FROM: REF. 8

Figure 10.76 Effect of Tab Gap Size and Tab Overhang on the Control Surface Hingemoment Derivative due to Tab Deflection

10.4.2 Three-Dimensional Control Surface and Tab Hingemoment Derivatives

The three-dimensional hingemoment coefficient for a control surface is estimated from:

$$C_h = C_{h_o} + C_{h_\alpha}\alpha + C_{h_\delta}\delta + C_{h_{\delta_t}}\delta_t \qquad (10.140)$$

where: C_{h_o} is the zero-angle-of-attack, zero-control-surface-deflection and zero-tab-deflection hingemoment coefficient. For main surfaces with symmetrical airfoils:

$$C_{h_o} = 0 \text{ (symmetrical airfoils only)} \qquad (10.141)$$

For main surfaces with cambered airfoils, experimental data should be used to determine this quantity.

C_{h_α} is the Three-D control surface hingemoment derivative due to angle of attack. It is found from 10.4.2.1

C_{h_δ} is the Three-D control surface hingemoment derivative due to control surface deflection. It is estimated from 10.4.2.2

$C_{h_{\delta_t}}$ is the Three-D control surface hingemoment derivative due to a tab deflection. It is estimated from 10.4.2.3.

10.4.2.1 Three-D control surface hingemoment derivative due to angle of attack: C_{h_α}

The Three-D control surface hingemoment due to angle of attack derivative, C_{h_α} is determined from:

$$C_{h_\alpha} = \qquad (10.142)$$

$$\{(A\cos\Lambda_{c/4})/(A + 2\cos\Lambda_{c/4})\}(c_{h_\alpha})_M + \Delta C_{h_\alpha}$$

where: $(c_{h_\alpha})_M$ is obtained from Eqn.(10.133)

$$\Delta C_{h_\alpha} = \qquad (10.143)$$

$$\{\Delta C_{h_\alpha}/(c_{l_\alpha}B_2K_\alpha\cos\Lambda_{c/4})\} * (c_{l_\alpha}B_2K_\alpha\cos\Lambda_{c/4})$$

with: $(\Delta C_{h_\alpha} / (c_{l_\alpha} B_2 K_\alpha \cos\Lambda_{c/4}))$ a factor obtained from Figure 10.77a

c_{l_α} is the airfoil lift-curve-slope of the surface to which the control surface is attached. The main surface airfoil at the m.g.c. of the control surface may be used. The method of 8.1.1.2 can be employed to estimate this quantity.

B_2 accounts for control surface and balance chord ratios. It may be determined from Figure 10.77c. The primed values of control surface and balance chord ratios in Figure 10.77c refer to measurements normal to the quarter chord line of the main surface.

K_α accounts for the effect of control surface span. It is found from:

$$K_\alpha = \{(K_\alpha)_{\eta_i}(1 - \eta_i) - (K_\alpha)_{\eta_o}(1 - \eta_o)\}/(\eta_o - \eta_i) \qquad (10.144)$$

where: η_i is the inboard span station of the control surface as a fraction of the main surface semi-span

$(K_\alpha)_{\eta_i}$ is found from Figure 10.77b

η_o is the outboard span station of the control surface as a fraction of the main surface semi-span

$(K_\alpha)_{\eta_o}$ is found from Figure 10.77b

<u>NOTE:</u> Control surface hingemoments are also affected by items such as horns, internal balance plates and various types of tab configurations. A discussion of these items is found in Chapter 3 of Part IV and in Chapter 5 of Reference 16. A detailed treatment of methods for estimating hingemoment derivatives due to these effects is beyond the scope of this text.

Figure 10.77 Three-Dimensional Correction Factors for the Control Surface Hingemoment Derivative due to Angle of Attack

10.4.2.2 Three-D control surface hingemoment derivative due to control surface deflection: C_{h_δ}

The Three-D control surface hingemoment due to control surface deflection derivative, C_{h_δ} is found from:

$$C_{h_\delta} = (\cos\Lambda_{c/4})(\cos\Lambda_{hl})* \qquad (10.145)$$

$$*[(c_{h_\delta})_M + \alpha_\delta(c_{h_\alpha})_M\{(2\cos\Lambda_{c/4})/(A + 2\cos\Lambda_{c/4})\} + \Delta C_{h_\delta}$$

where: $(c_{h_\alpha})_M$ is found from Equation (10.133)

$(c_{h_\delta})_M$ is found from Equation (10.138)

α_δ is obtained from Figure 8.17

$$\Delta C_{h_\delta} = \qquad (10.146)$$

$$\{\Delta C_{h_\delta}/(c_{l_\delta}B_2K_\delta\cos\Lambda_{c/4}\cos\Lambda_{hl})\}* (c_{l_\delta}B_2K_\delta\cos\Lambda_{c/4}\cos\Lambda_{hl})$$

with: $\{\Delta C_{h_\delta}/(c_{l_\delta}B_2K_\delta\cos\Lambda_{c/4}\cos\Lambda_{hl})\}$ a factor obtained from Fig.10.78a

c_{l_δ} is found from Figure 8.14

B_2 is found from Figure 10.77c, where the primed values of the control-surface and the balance-chord ratios refer to measurements normal to the quarter chord line

$$K_\delta = \qquad (10.147)$$

$$\{(K_\delta)_{\eta_i}(1 - \eta_i) - (K_\delta)_{\eta_o}(1 - \eta_o)\}/(\eta_o - \eta_i)$$

where: η_i is defined in 10.4.2.1

$(K_\delta)_{\eta_i}$ is found from Figure 10.78b

η_o is defined in 10.4.2.1

$(K_\delta)_{\eta_o}$ is found from Figure 10.78b

The note at the end of 10.4.2.1 applies here also!

10.4.2.3 Three-D control surface hingemoment derivative due to tab deflection: $C_{h_{\delta_t}}$

The Three-D control surface hingemoment due to tab deflection derivative, $C_{h_{\delta_t}}$ is estimated by employing Equation (10.145) and its sequel with appropriate substitution of tab parameters for control surface parameters.

Figure 10.78 Three Dimensional Correction Factors for the Control Surface Hingemoment Derivative due to Control Surface Deflection

10.4.3 Two-Dimensional Tab Hingemoment Derivatives about the Tab Hingeline

The two-dimensional tab hingemoment coefficient about the tab hingeline can be written as:

$$c^t_h = c^t_{h_o} + c^t_{h_\alpha}\alpha + (c^t_{h_\delta})_{\alpha,\delta_t}\delta + c^t_{h_{\delta_t}}\delta_t \quad (10.148)$$

where: $c^t_{h_o}$ is the zero-angle-of-attack, zero-control-surface-deflection and zero-tab-deflection hingemoment coefficient of the tab about its own hingeline. This quantity is zero for symmetrical airfoils. For cambered airfoils experimental data should be used.

$c^t_{h\alpha}$ is found from 10.4.1.1 by substitution of tab parameters for control surface parameters

$c^t_{h_{\delta_t}}$ is found from 10.4.1.2 by substitution of tab parameters for control surface parameters

$$(c^t_{h_\delta})_{\alpha,\alpha_t} = \quad (10.149)$$

$$(c^t_{h_\delta})_{c_l,\delta_t} - (c^t_{h_{c_l}})_{\delta,\delta_t} * (c_{l_\alpha})_{\delta,\delta_t} * (\alpha_\alpha)_{c_l,\delta_t}$$

with: $(c^t_{h_\delta})_{c_l,\delta_t}$ being the change in tab hingemoment coefficient due to control surface deflection at constant lift and at constant tab deflection. It is found from Figure 10.79a.

$(c^t_{h_{c_l}})_{\delta,\delta_t}$ being the change in tab hingemoment coefficient due to lift at constant control surface deflection and at constant tab deflection. It is found from Figure 10.79b.

$(c_{l_\alpha})_{\delta,\delta_t}$ is the airfoil lift curve slope of the main surface to which the tab (via the control surface) is attached. It is found in 8.1.1.2.

$(\alpha_\delta)_{c_l,\delta_t}$ is the change in angle of attack due to a change in control surface deflection. It is obtained from Figure 10.80 by using the tab-chord to main surface chord ratio.

The derivatives in Equation (10.148) do not account for the effects of tab nose shape and tab gaps: they apply to round nose tabs with sealed gaps only! To account for the effect of differing tab nose shapes and for unsealed tab gaps, the reader should correct $c^t{}_{h_y}$ by using appropriate ratios obtained from Figs 10.81a,b,c and d.

10.4.4 Three-Dimensional Tab Hingemoment Derivatives about the Tab Hingeline

The three-dimensional tab hingemoment coefficient about the tab hingeline can be determined from:

$$C^t{}_h = C^t{}_{h_o} + C^t{}_{h_\alpha}\alpha + (C^t{}_{h_\delta})_{\alpha,\delta_t}\delta + C^t{}_{h_{\delta_t}}\delta_t \qquad (10.150)$$

where all coefficients may be determined from their two-dimensional counterparts by using the methods of 10.4.2.1 and 10.4.2.2.

Figure 10.79a Effect of Tab and Control Surface Size on the Change in Tab Hingemoment Coefficient due to Control Surface Deflection at Constant Lift and at Constant Tab Deflection

Figure 10.79b Effect of Tab Size on the Change in Tab Hingemoment Coefficient due to Lift at Constant Control Surface Deflection and at Constant Tab Deflection

Figure 10.80 Change in Angle of Attack due to a Change in Control Surface Deflection

Figure 10.81a Effect of Tab Nose Shape on the Tab Hingemoment Derivative due to Control Surface Deflection

Figure 10.81b Effect of Tab Gap and Tab Size on the Tab Hingemoment Derivative due to Angle of Attack

Figure 10.81c Effect of Tab Gap and Tab Size on the Tab Hingemoment Derivative due to Control Surface Deflection

Figure 10.81d Effect of Tab Gap and Tab Size on the Tab Hingemoment Derivative due to Tab Deflection

11. STABILITY AND CONTROL DERIVATIVE DATA

The purpose of this chapter is to provide example data of stability and control derivatives for the following airplanes:

Tables 11.1 Airplane A (representative of a Cessna Model 172 type of airplane)

Tables 11.2 Airplane B (representative of a Beech Model 99 type of airplane)

Tables 11.3 Airplane C (representative of a SIAI-Marchetti S-211 type of airplane)

Tables 11.4 Airplane D (representative of a Gates Learjet Model 24 type of airplane)

Tables 11.5 Airplane E (representative of a McDonnell Douglas F4C type of airplane)

Tables 11.6 Airplane F (representative of a Boeing 747-100 type of airplane)

Except for airplane A, no data were available for hingemoment derivatives. All data are dimensionless.

Table 11.1a Geometry and Derivative Data for Airplane A

Flight Condition	1
	Cruise
Altitude (ft)	5,000
Air Density (slugs/ft^2)	.002050
Speed (fps)	219
Center of Gravity ($\bar{h}_{cg}$)	.25
Initial Attitude (θ_1 in rad)	0
Geometry and Inertias	
Wing Area (ft^2)	174
Wing Span (ft)	35.8
Wing Mean Geometric Chord (ft)	4.9
Weight (lbs)	2,665
I_{xx} (slug ft^2)	948
I_{yy} (slug ft^2)	1,346
I_{zz} (slug ft^2)	1,967
I_{xz} (slug ft^2)	0
Steady State Coefficients	
C_{L_1}	.31
C_{D_1}	.031
$C_{T_{x_1}}$	.031
C_{m_1}	0
$C_{m_{T_1}}$	0

Table 11.1b Geometry and Derivative Data for Airplane A

Longitudinal Derivatives		Lateral Directional Derivatives	
C_{m_u}	0	C_{ℓ_β}	-.089
C_{m_α}	-.89	C_{ℓ_p}	-.47
$C_{m_{\dot\alpha}}$	-5.2	C_{ℓ_r}	.096
C_{m_q}	-12.4	$C_{\ell_{\delta_A}}$	.178
$C_{m_{T_u}}$	0	$C_{\ell_{\delta_R}}$	.0147
$C_{m_{T_\alpha}}$	0	C_{n_β}	.065
C_{L_u}	0	C_{n_p}	-.03
C_{L_α}	4.6	C_{n_r}	-.099
$C_{L_{\dot\alpha}}$	1.7	$C_{n_{\delta_A}}$	-.053
C_{L_q}	3.9	$C_{n_{\delta_R}}$	-.0657
C_{D_α}	.13	C_{y_β}	-.31
C_{D_u}	0	C_{y_p}	-.037
$C_{T_{X_u}}$	-.093	C_{y_r}	.21
$C_{L_{\delta_E}}$	.43	$C_{y_{\delta_A}}$	0
$C_{D_{\delta_E}}$	.06	$C_{y_{\delta_R}}$	.187
$C_{m_{\delta_E}}$	-1.28		
C_{h_α}	-.050	$C_{h_{\delta_a}}$	-.010
$C_{h_{\delta_e}}$	-.590	$C_{h_{\delta_r}}$	-.570
		C_{h_β}	+.082

Table 11.2a Geometry and Derivative Data for Airplane B

Flight Condition	1	2	3
	Power Approach	Cruise (Low)	Cruise (Normal)
Altitude (ft)	Sealevel	5,000	20,000
Air Density (slugs/ft^3)	.002378	.00205	.001268
Speed (fps)	170	340	450
Center of Gravity ($\bar{x}_{cg}$)	.16	.16	.16
Initial Attitude (θ_1 in rad)	0	0	0
Geometry and Inertias			
Wing Area (ft^2)	280	280	280
Wing Span (ft)	46	46	46
Wing Mean Geometric Chord (ft)	6.5	6.5	6.5
Weight (lbs)	11,000	7,000	11,000
I_{xx} (slug ft^2)	15,189	10,085	15,189
I_{yy} (slug ft^2)	20,250	15,148	20,250
I_{zz} (slug ft^2)	34,141	23,046	34,141
I_{xz} (slug ft^2)	4,371	1,600	4,371
Steady State Coefficients			
C_{L_1}	1.15	.191	.30
C_{D_1}	.162	.0298	.0298
$C_{T_{X_1}}$	.162	.0298	.0298
C_{m_1}	0	0	0
$C_{m_{T_1}}$	0	0	0

Table 11.2b Geometry and Derivative Data for Airplane B

Longitudinal Derivatives	1	2	3
C_{m_u}	0	0	0
C_{m_α}	-2.08	-1.89	-1.89
$C_{m_{\dot\alpha}}$	-9.1	-9.1	-9.1
C_{m_q}	-34.0	-34.0	-34.0
$C_{m_{T_u}}$	0	0	0
$C_{m_{T_\alpha}}$	0	0	0
C_{L_u}	.027	.020	.020
C_{L_α}	6.24	5.48	5.48
$C_{L_{\dot\alpha}}$	2.7	2.5	2.5
C_{L_q}	8.1	8.1	8.1
C_{D_α}	.933	.131	.131
C_{D_u}	0	0	0
$C_{T_{X_u}}$	-.324	-.0596	-.0596
$C_{L_{\delta_E}}$	.58	.4	.6
$C_{D_{\delta_E}}$	0	0	0
$C_{m_{\delta_E}}$	-1.9	-2.0	-2.0

Lateral-Directional Derivatives	1	2	3
C_{l_β}	-.13	-.13	-.13
C_{l_p}	-.50	-.50	-.50
C_{l_r}	.06	.14	.14
$C_{l_{\delta_A}}$	.156	.156	.156
$C_{l_{\delta_R}}$	.0087	.0109	.0106
C_{n_β}	.120	.080	.080
C_{n_p}	-.005	.019	.019
C_{n_r}	-.204	-.197	-.197
$C_{n_{\delta_A}}$	-.0012	-.0012	-.0012
$C_{n_{\delta_R}}$	-.0763	-.0772	-.0758
C_{y_β}	-.59	-.59	-.59
C_{y_p}	-.21	-.19	-.19
C_{y_r}	.39	.39	.39
$C_{y_{\delta_A}}$	0	0	0
$C_{y_{\delta_R}}$	.144	.148	.144

Table 11.3a Geometry and Derivative Data for Airplane C

Flight Condition	1	2	3
	Power Approach	Cruise (Normal)	Cruise (High)
Altitude (ft)	Sealevel	25,000	35,000
Air Density (slugs/ft^3)	.002378	.001066	.000739
Speed (fps)	124 (1.2 $V_{S_{TA}}$)	610 (M = .6)	584 (M = .6)
Center of Gravity ($\bar{x}_{cg}$)	.25 (mid)	.25 (mid)	.25 (mid)
Initial Attitude (θ_1 in rad)	0	0	0
Geometry and Inertias			
Wing Area (ft^2)	136	136	136
Wing Span (ft)	26.3	26.3	26.3
Wing Mean Geometric Chord (ft)	5.4	5.4	5.4
Weight (lbs)	3,500	4,000	4,000
I_{xx} (slug ft^2)	750	800	800
I_{yy} (slug ft^2)	4,800	4,800	4,800
I_{zz} (slug ft^2)	5,000	5,200	5,200
I_{xz} (slug ft^2)	200	200	200
Steady State Coefficients			
C_{L_1}	1.4	.15	.23
C_{D_1}	.21	.022	.025
$C_{T_{x_1}}$	.21	.022	.025
C_{m_1}	0	0	0
$C_{m_{T_1}}$	0	0	0

Table 11.3b Geometry and Derivative Data for Airplane C

Longitudinal Derivatives	1	2	3
C_{m_u}	0	0	0
C_{m_α}	-.6	-.24	-.24
$C_{m_{\dot\alpha}}$	-7.0	-9.6	-9.6
C_{m_q}	-15.7	-17.7	-17.7
$C_{m_{T_u}}$	0	0	0
$C_{m_{T_\alpha}}$	0	0	0
C_{L_u}	.071	.084	.132
C_{L_α}	5.0	5.5	5.5
$C_{L_{\dot\alpha}}$	3.0	4.2	4.2
C_{L_q}	9.0	10.0	10.0
C_{D_0}	1.14	.12	.17
C_{D_u}	0	0	0
$C_{T_{X_u}}$	0	0	0
$C_{L_{\delta_E}}$	.39	.38	.35
$C_{D_{\delta_E}}$	0	0	0
$C_{m_{\delta_E}}$	-.90	-.88	-.82

Lateral-Directional Derivatives	1	2	3
C_{l_β}	-.140	-.110	-.110
C_{l_p}	-.350	-.390	-.390
C_{l_r}	.540	.280	.310
$C_{l_{\delta_A}}$	.110	.100	.100
$C_{l_{\delta_R}}$	.030	.050	.050
C_{n_β}	.160	.170	.170
C_{n_p}	-.030	+.090	+.080
C_{n_r}	-.310	-.260	-.260
$C_{n_{\delta_A}}$	-.030	-.003	-.005
$C_{n_{\delta_R}}$	-.110	-.120	-.120
C_{y_β}	-.94	-1.00	-1.00
C_{y_p}	-.010	-.14	-.12
C_{y_r}	.590	.610	.620
$C_{y_{\delta_A}}$	0	0	0
$C_{y_{\delta_R}}$	.260	.280	.280

Table 11.4a Geometry and Derivative Data for Airplane D

Flight Condition	1	2	3
	Power Approach	Cruise Max. Wht.	Cruise Low Wht.
Altitude (ft)	Sealevel	40,000	40,000
Air Density (slugs/ft^3)	.002378	.000588	.000588
Speed (fps)	170	677 (M = .7)	677 (M = .7)
Center of Gravity ($\bar{x}_{cg}$)	.32 (aft)	.32 (aft)	.32 (aft)
Initial Attitude (deg)	1.8	2.7	1.5
Geometry and Inertias			
Wing Area (ft^2)	230	230	230
Wing Span (ft)	34	34	34
Wing Mean Geometric Chord (ft)	7	7	7
Weight (lbs)	13,000	13,000	9,000
I_{xx_B} (slug ft^2)	26,000	26,000	6,000
I_{yy_B} (slug ft^2)	17,800	18,800	17,800
I_{zz_B} (slug ft^2)	47,000	47,000	25,000
I_{xz_B} (slug ft^2)	1,300	1,300	1,400
Steady State Coefficients			
C_{L_1}	1.64	.41	.28
C_{D_1}	.256	.0335	.0279
$C_{T_{X_1}}$	.256	.0335	.0279
C_{m_1}	0	0	0
$C_{m_{T_1}}$	0	0	0

Table 11.4b Geometry and Derivative Data for Airplane D

Longitudinal Derivatives	1	2	3
C_{x_u}	-.01	.05	.07
C_{x_α}	-.66	-.66	-.64
$C_{x_{\delta_e}}$	-5.0	-6.7	-6.7
C_{x_q}	-13.5	-15.5	-15.5
$C_{m_{T_u}}$	.006	-.003	-.003
$C_{m_{T_\alpha}}$	0	0	0
C_{L_u}	.04	.40	.28
C_{L_α}	5.04	5.84	5.84
$C_{L_{\dot\alpha}}$	1.6	2.2	2.2
C_{L_q}	4.1	4.7	4.7
C_{D_α}	1.06	.30	.22
C_{D_u}	0	.104	.104
$C_{T_{X_u}}$	0	0	0
$C_{L_{\delta_E}}$	.40	.46	.46
$C_{D_{\delta_E}}$	0	0	0
$C_{m_{\delta_E}}$	-.98	-1.24	-1.24

Lateral-Directional Derivatives	1	2	3
C_{l_β}	-.173	-.110	-.100
C_{l_p}	-.39	-.45	-.45
C_{l_r}	.45	.16	.14
$C_{l_{\delta_A}}$	.149	.178	.178
$C_{l_{\delta_R}}$	.014	.019	.021
C_{n_β}	.150	.127	.124
C_{n_p}	-.13	-.008	-.022
C_{n_r}	-.26	-.20	-.20
$C_{n_{\delta_A}}$	-.05	-.02	-.02
$C_{n_{\delta_R}}$	-.074	-.074	-.074
C_{Y_β}	-.73	-.73	-.73
C_{Y_p}	0	0	0
C_{Y_r}	.4	.4	.4
$C_{Y_{\delta_A}}$	0	0	0
$C_{Y_{\delta_R}}$	.140	.140	.140

Table 11.5a Geometry and Derivative Data for Airplane E

Flight Condition	1	2	3
	Power Approach	Subsonic Cruise	Supersonic Cruise
Altitude (ft)	Sealevel	35,000	55,000
Air Density (slugs/ft^3)	.002378	.000738	.000287
Speed (fps)	230	876	1742
Center of Gravity ($\bar{x}_{cg}$)	.29	.29	.29
Initial Attitude (deg)	11.7	2.6	3.3
Geometry and Inertias			
Wing Area (ft^2)	530	530	530
Wing Span (ft)	38.7	38.7	38.7
Wing Mean Geometric Chord (ft)	16.0	16.0	16.0
Weight (lbs)	33,200	39,000	39,000
I_{xx_B} (slug ft^2)	23,700	25,000	25,000
I_{yy_B} (slug ft^2)	117,500	122,200	122,200
I_{zz_B} (slug ft^2)	133,700	139,800	139,800
I_{xz_B} (slug ft^2)	1,600	2,200	2,200
Steady State Coefficients			
C_{L_1}	1.0	.26	.17
C_{D_1}	.2	.03	.048
$C_{Z_{X_1}}$	.2	.03	.048
C_{m_1}	0	0	0
$C_{m_{T_1}}$	0	0	0

Table 11.5b Geometry and Derivative Data for Airplane E

Longitudinal Derivatives	1	2	3
C_{m_u}	0	-.117	+.054
C_{m_α}	-.098	-.40	-.78
$C_{m_{\dot\alpha}}$	-.95	-1.3	-.25
C_{m_q}	-2.0	-2.7	-2.0
$C_{m_{T_u}}$	0	0	0
$C_{m_{T_\alpha}}$	0	0	0
C_{L_u}	0	+.27	-.18
C_{L_α}	2.8	3.75	2.6
$C_{L_{\dot\alpha}}$	0	0	0
C_{L_q}	0	0	0
C_{D_α}	.555	.3	.4
C_{D_u}	0	+.027	-.054
$C_{T_{X_u}}$	0	0	0
$C_{L_{\delta_E}}$	.24	.40	.25
$C_{D_{\delta_E}}$	-.14	-.10	-.15
$C_{m_{\delta_E}}$	-.322	-.58	-.38

(Note: longitudinal control through stabilizer only)

Lateral-Directional Derivatives	1	2	3
C_{l_β}	-.156	-.080	-.025
C_{l_p}	-.272	-.240	-.200
C_{l_r}	.205	.070	.040
$C_{l_{\delta_A}}$	.057	.042	.015
$C_{l_{\delta_R}}$	.0009	.0060	.0030
C_{n_β}	.199	.125	.090
C_{n_p}	.013	-.036	0
C_{n_r}	-.320	-.270	-.260
$C_{n_{\delta_A}}$	+.0041	-.0010	-.0009
$C_{n_{\delta_R}}$	-.072	-.068	-.025
C_{y_β}	-.655	-.68	-.70
C_{y_p}	0	0	0
C_{y_r}	0	0	0
$C_{y_{\delta_A}}$	-.0055	-.016	-.010
$C_{y_{\delta_R}}$	.124	.095	.050

Table 11.6a Geometry and Derivative Data for Airplane F

Flight Condition	1	2	3
	Power Approach	Cruise (High)	Cruise (Low)
Altitude (ft)	Sealevel	40,000	20,000
Air Density (slugs/ft^3)	.002309	.000588	.001268
Speed (fps)	221	871	673
Center of Gravity ($\bar{x}_{cg}$)	.25	.25	.25
Initial Attitude (deg)	8.5	2.4	2.5
Geometry and Inertias			
Wing Area (ft^2)	5,500	5,500	5,500
Wing Span (ft)	196	196	196
Wing Mean Geometric Chord (ft)	27.3	27.3	27.3
Weight (lbs)	564,000	636,636	636,636
I_{xx_B} (slug ft^2)	13.7×10^6	18.2×10^6	18.2×10^6
I_{yy_B} (slug ft^2)	30.5×10^6	33.1×10^6	33.1×10^6
I_{zz_B} (slug ft^2)	43.1×10^6	49.7×10^6	49.7×10^6
I_{xz_B} (slug ft^2)	$.83 \times 10^6$	$.97 \times 10^6$	$.97 \times 10^6$
Steady State Coefficients			
C_{L_1}	1.76	.52	.40
C_{D_1}	.263	.045	.025
$C_{T_{X_1}}$	.263	.045	.025
C_{m_1}	0	0	0
$C_{m_{T_1}}$	0	0	0

Table 11.6b Geometry and Derivative Data for Airplane F

Longitudinal Derivatives	1	2	3
C_{m_u}	.071	-.09	+.013
C_{m_θ}	-1.45	-1.60	-1.00
$C_{m_{\dot\alpha}}$	-3.3	-9.0	-4.0
C_{m_q}	-21.4	-25.5	-20.5
$C_{m_{T_u}}$	0	0	0
$C_{m_{T_\alpha}}$	0	0	0
C_{L_u}	-.22	-.23	+.13
C_{L_α}	5.67	5.5	4.4
$C_{L_{\dot\alpha}}$	6.7	8.0	7.0
C_{L_q}	5.65	7.8	6.6
C_{D_0}	1.13	.50	.20
C_{D_u}	0	.22	0
$C_{T_{X_u}}$	0	0	0
$C_{L_{\delta_E}}$	.36	.30	.32
$C_{D_{\delta_E}}$	0	0	0
$C_{m_{\delta_E}}$	-1.40	-1.20	-1.30

Lateral-Directional Derivatives	1	2	3
C_{l_β}	-.281	-.095	-.160
C_{l_p}	-.502	-.320	-.340
C_{l_r}	.195	.200	.130
$C_{l_{\delta_A}}$	.053	.014	.013
$C_{l_{\delta_R}}$	0	.005	.008
C_{n_β}	.184	.210	.160
C_{n_p}	-.222	+.020	-.026
C_{n_r}	-.36	-.33	-.28
$C_{n_{\delta_A}}$	+.0063	-.0028	+.0018
$C_{n_{\delta_R}}$	-.113	-.095	-.100
C_{y_β}	-1.08	-.90	-.90
C_{y_p}	0	0	0
C_{y_r}	0	0	0
$C_{y_{\delta_A}}$	0	0	0
$C_{y_{\delta_R}}$	.179	.060	.120

SAAB SF-340
COURTESY: SAAB

12. USER'S GUIDE

The purpose of this chapter is to present a User's Guide to the prediction methods presented in this part of the eight part series on Airplane Design.

It is assumed that the Preliminary Design Sequence I (see p.11, Part II) has been completed and that the results of the accompanying Class I work have been properly documented. The stage is now set for a more 'in depth' (Class II) analysis of the capabilities of the design. To accomplish this, the following characteristics need to be determined:

1.) Drag polars
2.) Installed thrust or power data
3.) Lift versus angle of attack
4.) Pitching moment versus angle of attack
5.) Stability, control and hingemoment derivatives

The data which are the result of determining items 1.) through 5.) are themselves input data to the calculation and the evaluation of performance, stability, control, handling qualities and airworthiness capabilities of the airplane as outlined in Part VII.

NOTE: The reader should not attempt to perform any of the calculations represented by 1.) through 5.) without a complete geometric definition of the airplane: a dimensioned, Class I threeview MUST be available. This threeview should be of a size large enough so that dimensions needed in predicting items 1.) through 5.) can be 'scaled' directly from this threeview drawing. If a CAD (Computer Aided Design) data base is available for the airplane, so much the better.

12.1 USER'S GUIDE FOR DRAG POLAR DETERMINATION

Step 1: The reader should determine for which conditions drag polars are needed:

Airplane Configuration:
- *Clean
- *Flap position (up, take-off or landing)
- *Landing Gear (up or down)
- *External stores
- *Engine and inlet status

Flight Condition: *Mach Number
*Altitude
*Reynold's Number
*Weight
*Center of Gravity Location

The conditions selected depend on the type of performance calculations which need to be performed with the drag polars. Example of performance capabilities which typically must be determined are:

A) Mission oriented performance such as:

* take-off	* climb	* cruise
* loiter	* descent	* landing
* maneuvering	* high speed dash	* accelerate

B) Airworthiness oriented performance such as:

* climb rate or gradient with failed engine(s)
* balanced fieldlength
* performance following major damage (for example combat damage)

<u>Step 2:</u> Proceed to Chapter 4 to compute the required drag data. Refer to Equation 4.4 and identify which drag components must be determined. Prepare a list of input data needed for the calculation of each drag component.

Plot the results in the form of C_D versus α.

<u>Step 3:</u> Proceed to Section 12.3 and use the resulting C_L versus α plot together with the C_D versus α plot obtained from Step 2 to construct a C_L versus C_D plot (=drag polar).

<u>Step 4:</u> Verify the drag polars with the procedure of Section 5.5, p.135. Also compare the drag polar data with data on similar airplanes: see Section 5.1, p.117.

12.2 USER'S GUIDE FOR DETERMINATION OF INSTALLED THRUST OR POWER

It will be assumed that the Class I powerplant integration has been carried out as indicated in Chapter 5 of Part II. The powerplant installation has therefore been temporarily 'frozen' which means that the following information should be available:

* engine type
* inlet type and inlet size
* number of engines
* exhaust type and nozzle size

Manufacturers uninstalled thrust or power data must also be available at this point.

Finally, it will be assumed that a list of systems required by the airplane including a preliminary determination of the power requirements of these systems, is avalable.

Step 1: Obtain engine manufacturers uninstalled thrust or power data. These data should define uninstalled thrust (or power) as a function of altitude, temperature, airspeed and fuel flow.

If such detailed data are not available, the data in Chapter 6 of Part III may be used to 'guestimate' a set of uninstalled data.

Step 2: Refer to pages 139 and 140 (Chapter 6) and proceed with the preparation of the input data necessary in the calculation of installed thrust or power data.

Step 3: Using the methods of Sections 6.1 through 6.4 estimate the installed power or thrust capabilities of the airplane.

Example data are given in Chapter 7.

12.3 USER'S GUIDE FOR DETERMINATION OF LIFT VERSUS ANGLE OF ATTACK

Step 1: The reader should determine for which flight conditions the lift versus angle of attack data need to be prepared. Refer to Step 1 in Section 12.1 for some guidelines.

Step 2: Proceed to Section 8.1 for the calculation of the required lift versus angle of attack characteristics. Plot the results in the form of C_L versus α.

Step 3: Check the results of Step 2 by comparing with example data on similar airplanes such as given in Chapter 9.

12.4 USER'S GUIDE FOR THE DETERMINATION OF PITCHING MOMENT VERSUS ANGLE OF ATTACK AND THE TRIM DIAGRAM

Step 1: The reader should determine for which flight conditions the pitching moment versus angle of attack data need to be prepared. Refer to Step 1 of Section 12.1 for guidelines.

Step 2: Refer to the weight versus center of gravity diagram (See Chapter 10, Part II) and determine the most forward and aft c.g. location which applies to each flight condition selected in Step 1. Select a suitable c.g. location for use as the reference point in the pitching moment calculations.

Step 3: Proceed to Section 8.2 and determine the pitching moment versus angle of attack characteristics. Plot the results in the form of C_m versus α for the reference point selected in Step 2.

Step 4: Proceed to Section 8.3 and determine the airplane trim diagram in the form of a plot of C_m versus C_L for the reference point selected in Step 2.

Step 5: Verify the results of Step 4 by comparing with data on similar airplanes. Some data are given in Chapter 9.

12.5 USER'S GUIDE FOR THE DETERMINATION OF STABILITY, CONTROL AND HINGEMOMENT DERIVATIVES

Step 1: Determine the flight conditions for which the derivatives need to be determined. Use Step 1 in Section 12.1 for guidelines.

Step 2: Proceed to Chapter 10 and compute the required derivatives.

Step 3: Verify the results of Step 2 by comparing with data on similar airplanes. Example derivative data are given in Chapter 11.

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APPENDIX A: STANDARD ATMOSPHERE, SPECIFIC WEIGHTS AND CONVERSION FACTORS

This appendix presents tabulated data for: Page

A1. U.S. Standard Atmosphere, 1962 519
A2. Specific weights of liquids and gases 521
A3. Conversion Factors 522

The data have been copied from Reference 57.

A1. U.S. Standard Atmosphere, 1962

Definition of Standard Atmosphere

A standard atmosphere is a hypothetical vertical distribution of atmospheric temperature, pressure and density which by international or national agreement is taken to be representative of the atmosphere for the purpose of altimeter calibrations, aircraft design, performance calculations, etc. The internationally accepted standard atmosphere is called the International Civil Aeronautical Organization (ICAO) Standard Atmosphere or the International Standard Atmosphere (ISA). The U.S. Standard Atmosphere, 1962 is in agreement with the ICAO Standard Atmosphere up to 65,000 feet altitude. It is ideal air devoid of moisture, water vapor, and dust, and obeys the perfect gas law. It is based upon accepted standard values of sea level air density, temperature and pressure. Other standard atmosphere models, such as MIL-STD-210A, which represents hot and cold ambient temperature extremes, have also been established.

ICAO and U.S. Standard Atmospheres

Standard Values at Sea Level

	British Units	Metric Units
Pressure, P_o	2116.22 lb/ft² 29.92 in. Hg	1.013250×10^5 N/m² 760 mm Hg
Temperature, T_o	518.67°R 59.0°F	288.15°K 15.0°C
Acceleration due to gravity, g_o	32.1741 ft/sec²	9.80665 m/sec²
Specific weight, $g\rho_o$	0.076474 lb/ft³	1.2250 kg/m³
Density, ρ_o	0.0023769 lb-sec²/ft⁴	0.12492 kg sec²/m⁴
Kinematic viscosity, ν_o	1.5723×10^{-4} ft²/sec	1.4607×10^{-5} m²/sec
Absolute viscosity, μ_o	1.2024×10^{-7} lb/ft sec	1.7894×10^{-5} kg/m sec

Standard Values at Altitude

Isothermal altitude, Z_i	36089 ft	11000 m
Isothermal temperature, t	−69.7°F	−56.5°C
Temperature lapse rate (sea level to isothermal)	−3.57°F/1000 ft	−6.5°C/km

Temperature Conversion Formulas

$$t\,(°C) = T(°K) - 273.15$$
$$t\,(°C) = [T(°R) - 491.67]/1.8$$
$$t\,(°C) = [t(°F) - 32]/1.8$$

$$T\,(°R) = 1.8\,T(°K)$$
$$t\,(°F) = 1.8[T(°K) - 273.15] + 32$$
$$t\,(°F) = T\,(°R) - 459.67$$
$$t\,(°F) = 1.8\,t(°C) + 32$$

U.S. Standard Atmosphere, 1962
(Geopotential Altitude)
British Units

Altitude feet	Temperature °F	°R	°C	Pressure psia	in. Hg	θ	$\sqrt{\theta}$	δ	σ	$\bar{g}/M^*$ lb/ft²	Sonic Velocity ft/sec	kts
−2000	66.1	525.8	19.0	15.79	32.15	1.0138	1.0069	1.074	1.060	1592.	1124.1	666.0
−1000	62.5	522.2	17.0	15.23	31.02	1.0069	1.0034	1.037	1.030	1536.	1120.2	663.7
0	59.0	518.7	15.0	14.70	29.92	1.0000	1.0000	1.000	1.000	1481.	1116.4	661.5
1000	55.4	515.1	13.0	14.17	28.86	.9932	.9966	.9644	.9710	1429.	1112.6	659.2
2000	51.9	511.6	11.0	13.66	27.82	.9863	.9931	.9298	.9427	1377.	1108.7	656.9
3000	48.3	508.0	9.1	13.17	26.82	.9794	.9897	.8962	.9151	1328.	1104.9	654.6
4000	44.7	504.4	7.1	12.69	25.84	.9725	.9862	.8637	.8881	1279.	1101.0	652.3
5000	41.2	500.9	5.1	12.23	24.90	.9657	.9827	.8321	.8616	1233.	1097.1	650.0
6000	37.6	497.3	3.1	11.78	23.98	.9588	.9792	.8014	.8358	1187.	1093.2	647.7
7000	34.0	493.7	1.1	11.34	23.09	.9519	.9757	.7716	.8106	1143.	1089.2	645.4
8000	30.5	490.2	−0.8	10.92	22.23	.9450	.9721	.7428	.7860	1100.	1085.3	643.0
9000	26.9	486.6	−2.8	10.50	21.39	.9382	.9686	.7148	.7619	1059.	1081.3	640.7
10000	23.3	483.0	−4.8	10.11	20.58	.9313	.9650	.6877	.7385	1019.	1077.4	638.3
11000	19.8	479.5	−6.8	9.720	19.79	.9244	.9615	.6614	.7155	979.8	1073.4	636.0
12000	16.2	475.9	−8.8	9.346	19.03	.9175	.9579	.6360	.6932	942.1	1069.4	633.6
13000	12.6	472.3	−10.7	8.984	18.29	.9107	.9543	.6113	.6713	905.6	1065.4	631.2
14000	9.1	468.8	−12.7	8.633	17.58	.9038	.9507	.5875	.6500	870.2	1061.3	628.8
15000	5.5	465.2	−14.7	8.294	16.89	.8969	.9470	.5644	.6292	836.0	1057.3	626.4
16000	1.9	461.6	−16.7	7.965	16.22	.8900	.9434	.5420	.6089	802.9	1053.2	624.0
17000	−1.6	458.1	−18.7	7.647	15.57	.8831	.9398	.5203	.5892	770.8	1049.2	621.6
18000	−5.2	454.5	−20.7	7.339	14.94	.8763	.9361	.4994	.5699	739.8	1045.1	619.2
19000	−8.8	450.9	−22.6	7.041	14.34	.8694	.9324	.4791	.5511	709.8	1041.0	616.7
20000	−12.3	447.4	−24.6	6.754	13.75	.8625	.9287	.4596	.5328	680.8	1036.8	614.3
21000	−15.9	443.8	−26.6	6.475	13.18	.8556	.9250	.4406	.5150	652.7	1032.7	611.9
22000	−19.5	440.2	−28.6	6.207	12.64	.8488	.9213	.4223	.4976	625.6	1028.5	609.4
23000	−23.0	436.7	−30.6	5.947	12.11	.8419	.9175	.4047	.4806	599.4	1024.4	606.9
24000	−26.6	433.1	−32.5	5.696	11.60	.8350	.9138	.3876	.4642	574.1	1020.2	604.4
25000	−30.2	429.5	−34.5	5.454	11.10	.8281	.9100	.3711	.4481	549.7	1016.0	601.9
26000	−33.7	426.0	−36.5	5.220	10.63	.8213	.9062	.3552	.4325	526.2	1011.7	599.4
27000	−37.3	422.4	−38.5	4.994	10.17	.8144	.9024	.3398	.4173	503.4	1007.5	596.9
28000	−40.9	418.8	−40.5	4.777	9.725	.8075	.8986	.3250	.4025	481.5	1003.2	594.4
29000	−44.4	415.3	−42.4	4.567	9.298	.8006	.8948	.3107	.3881	460.3	988.9	591.9
30000	−48.0	411.7	−44.4	4.364	8.886	.7938	.8909	.2970	.3741	439.9	994.6	589.3
31000	−51.6	408.1	−46.4	4.169	8.489	.7869	.8871	.2837	.3605	420.3	990.3	586.8
32000	−55.1	404.6	−48.4	3.981	8.106	.7800	.8832	.2709	.3473	401.3	986.0	584.2
33000	−58.7	401.0	−50.4	3.800	7.737	.7731	.8793	.2586	.3345	383.1	981.6	581.6
34000	−62.3	397.4	−52.4	3.626	7.383	.7663	.8754	.2467	.3220	365.5	977.3	579.0
35000	−65.8	393.9	−54.3	3.458	7.041	.7594	.8714	.2353	.3099	348.6	972.9	576.4
36000	−69.4	390.3	−56.3	3.297	6.712	.7525	.8675	.2243	.2981	332.3	968.5	573.8
*36089	−69.7	390.0	−56.5	3.282	6.683	.7519	.8671	.2234	.2971	330.9	968.1	573.6
37000	−69.7	390.0	−56.5	3.142	6.397	.7519	.8671	.2138	.2843	316.7	968.1	573.6
38000	−69.7	390.0	−56.5	2.994	6.097	.7519	.8671	.2038	.2710	301.8	968.1	573.6
39000	−69.7	390.0	−56.5	2.854	5.811	.7519	.8671	.1942	.2583	287.7	968.1	573.6
40000	−69.7	390.0	−56.5	2.720	5.538	.7519	.8671	.1851	.2462	274.2	968.1	573.6
41000	−69.7	390.0	−56.5	2.592	5.278	.7519	.8671	.1764	.2346	261.3	968.1	573.6
42000	−69.7	390.0	−56.5	2.471	5.030	.7519	.8671	.1681	.2236	249.0	968.1	573.6
43000	−69.7	390.0	−56.5	2.355	4.794	.7519	.8671	.1602	.2131	237.4	968.1	573.6
44000	−69.7	390.0	−56.5	2.244	4.569	.7519	.8671	.1527	.2031	226.2	968.1	573.6
45000	−69.7	390.0	−56.5	2.139	4.355	.7519	.8671	.1455	.1936	215.6	968.1	573.6
46000	−69.7	390.0	−56.5	2.039	4.151	.7519	.8671	.1387	.1845	205.5	968.1	573.6
47000	−69.7	390.0	−56.5	1.943	3.956	.7519	.8671	.1322	.1758	195.8	968.1	573.6
48000	−69.7	390.0	−56.5	1.852	3.770	.7519	.8671	.1260	.1676	186.7	968.1	573.6
49000	−69.7	390.0	−56.5	1.765	3.593	.7519	.8671	.1201	.1597	177.9	968.1	573.6
50000	−69.7	390.0	−56.5	1.682	3.425	.7519	.8671	.1145	.1522	169.5	968.1	573.6
51000	−69.7	390.0	−56.5	1.603	3.264	.7519	.8671	.1091	.1451	161.6	968.1	573.6
52000	−69.7	390.0	−56.5	1.528	3.111	.7519	.8671	.1040	.1383	154.0	968.1	573.6
53000	−69.7	390.0	−56.5	1.456	2.965	.7519	.8671	.09909	.1318	146.8	968.1	573.6
54000	−69.7	390.0	−56.5	1.388	2.826	.7519	.8671	.09444	.1256	139.9	968.1	573.6
55000	−69.7	390.0	−56.5	1.323	2.693	.7519	.8671	.09000	.1197	133.3	968.1	573.6
56000	−69.7	390.0	−56.5	1.261	2.567	.7519	.8671	.08578	.1141	127.1	968.1	573.6
57000	−69.7	390.0	−56.5	1.201	2.446	.7519	.8671	.08175	.1087	121.1	968.1	573.6
58000	−69.7	390.0	−56.5	1.145	2.321	.7519	.8671	.07792	.1036	115.4	968.1	573.6
59000	−69.7	390.0	−56.5	1.091	2.222	.7519	.8671	.07426	.09877	110.0	968.1	573.6
60000	−69.7	390.0	−56.5	1.040	2.118	.7519	.8671	.07078	.09413	104.8	968.1	573.6
61000	−69.7	390.0	−56.5	.9913	2.018	.7519	.8671	.06746	.08971	99.93	968.1	573.6
62000	−69.7	390.0	−56.5	.9448	1.924	.7519	.8671	.06429	.08550	95.24	968.1	573.6
63000	−69.7	390.0	−56.5	.9005	1.833	.7519	.8671	.06127	.08149	90.77	968.1	573.6
64000	−69.7	390.0	−56.5	.8582	1.747	.7519	.8671	.05840	.07767	86.51	968.1	573.6
65000	−69.7	390.0	−56.5	.8179	1.665	.7519	.8671	.05566	.07402	82.45	968.1	573.6

U.S. Standard Atmosphere, 1962
(Geopotential Altitude)
British Units

Altitude feet	Temperature °F	°R	°C	Pressure psia	in. Hg	θ	$\sqrt{\theta}$	δ	σ	$\bar{\rho}/\bar{\rho}^{0}$ lb/ft³	Sonic Velocity ft/sec	kts
* 65617	−69.7	390.0	−56.5	.7941	1.617	.7519	.8671	.05403	.07186	80.04	968.1	573.6
70000	−67.3	392.4	−55.2	.6437	1.311	.7565	.8698	.04380	.05789	64.88	971.0	575.3
75000	−64.6	395.1	−53.6	.5073	1.0333	.7618	.8728	.03452	.04532	51.14	974.4	577.3
80000	−61.8	397.9	−52.1	.4005	.8155	.7671	.8759	.02726	.03553	40.37	977.8	579.3
85000	−59.1	400.6	−50.6	.3167	.6449	.7724	.8789	.02155	.02790	31.93	981.2	581.3
90000	−56.3	403.4	−49.1	.2509	.5108	.7777	.8819	.01707	.02195	25.29	984.5	583.3
95000	−53.6	406.1	−47.5	.1990	.4052	.7830	.8849	.01354	.01730	20.06	987.9	585.3
100000	−50.8	408.9	−46.0	.1581	.3220	.7883	.8878	.01076	.01365	15.94	991.2	587.3
*104987	−48.1	411.6	−44.5	.1259	.2563	.7935	.8908	.008567	.010800	12.69	994.5	589.2
150000	21.0	480.7	−6.1	.01893	.03854	.9269	.9627	.001288	.001390	1.908	1074.8	636.8
*154199	27.5	487.2	−2.5	.01609	.03275	.9393	.9692	.0001095	.001165	1.622	1082.0	641.1
*170604	27.5	487.2	−2.5	.00557	.01742	.9393	.9692	.0005823	.0006199	.8626	1082.0	641.1
200000	−4.8	454.9	−20.4	.02655	.005406	.8771	.9365	.0001807	.0002060	.2677	1045.5	619.5
*200131	−4.9	454.8	−20.5	.02641	.005377	.8768	.9364	.0001797	.0002050	.2662	1045.4	619.4

*Boundary between atmosphere layers of constant thermal gradient.

Note: The ICAO atmosphere is identical to the U.S. Standard Atmosphere for altitudes below 65,617 ft.

A2. Specific Weights of Liquids and Gases

Weights of Liquids

Liquid	Specific Gravity	at °C	Specific Wt lb/U.S. gal	lb/cu ft
Acetylene tetrabromide ($AcBr_4$)	2.99	25	24.95	186.67
Alcohol (methyl)	0.810	0	6.75	50.5
Benzine	0.899	0	7.5	56.1
Carbon tetrachloride	1.595	20	13.32	99.6
Ethylene glycol	1.12		9.3	69.6
Gasoline	0.72		5.87	44.9
Glycerine	1.261	20	10.52	78.71
JP1 (MIL-L-5616)	0.80		6.65	49.7
JP3 (MIL-J-5624D)	0.775		6.45	48.2
JP4 (MIL-J-5624D)	0.785		6.55	49.0
JP5 (MIL-J-5624D)	0.817	15	6.82	51.1
Kerosene	0.82		6.7	51.2
Mercury	13.546	20	113.0	845.6
Oil (MIL-6082B Grade 1100)	0.89	15	7.4	55.3
Sea water	1.025	15	8.55	63.99
Synthetic oil (MIL-L-7808C-1)	0.928	15	7.74	57.9
Water	1.000	4	8.345	62.43
Jet A fuel (ASTM-D-1655)	.808	15	6.74	50.4

Weights of Gases

Gas	Specific Wt * lb/cu ft
Air	.07651 (at 59.0°F)
Air	.08071
Carbon dioxide	.12341
Carbon monoxide	.07806
Helium	.01114
Hydrogen	.005611
Nitrogen	.07807
Oxygen	.089212

*At atmospheric pressure and 0°C.

A3. Conversion Factors

Multiply	By			To obtain
acre	4.3560	X	10^4	square feet
	4.0469	X	10^{-1}	hectares
	4.0469	X	10^3	square meters
	1.5625	X	10^{-3}	square miles
	4.8400	X	10^3	square yards
atmosphere (atm)(1962)	7.6000	X	10	centimeters of mercury
	2.9921	X	10	inches of mercury
	1.0332	X	10^4	kilograms/square meter
	1.0133	X	10^5	newtons/square meter
	1.4696	X	10	pounds/square inch
bar	9.8692	X	10^{-1}	atmospheres
	1.0000	X	10^6	dynes/square centimeter
	7.5006	X	10^2	millimeters of mercury
	1.0000	X	10^5	newtons/square meter
	1.4504	X	10	pounds/square inch
barn	1.0000	X	10^{-24}	square centimeters (nuclear cross-section)
barrel, liquid (U.S.)	3.1500	X	10	gallons
	1.1924	X	10^{-1}	cubic meters
British thermal unit (Btu)	2.5180	X	10^2	calories(post-1956 IST)
	7.7817	X	10^2	foot-pounds
	1.0551	X	10^{10}	ergs
	3.9301	X	10^{-4}	horsepower-hours
	1.0551	X	10^3	joules
	1.0551	X	10^3	newton-meters
	2.9302	X	10^{-4}	kilowatt-hours
	1.0551	X	10^3	watt-seconds
British thermal unit/minute (Btu/min)	4.1999			calories/second
	1.7548	X	10^8	ergs/second
	1.2970	X	10	foot-pounds/second
	2.3581	X	10^{-2}	horsepower
	1.7548	X	10	joules/second
	1.7931			kilogram-meters/second
	1.7548	X	10	watts
calorie (cal)	3.9683	X	10^{-3}	British thermal units
	3.0880			foot-pounds
	4.1868	X	10^7	ergs
	4.1868			joules
	1.1630	X	10^{-6}	kilowatt-hours
	4.1868			watt-seconds
calories/second (cal/sec)	2.3810	X	10^{-1}	British thermal units/minute
	4.1868	X	10^7	ergs/second
	3.0880			foot-pounds/second
	4.1868			joules/second
centimeter (cm)	3.2808	X	10^{-2}	feet
	3.9370	X	10^{-1}	inches
	1.0000	X	10^{-5}	kilometers
	1.0000	X	10^{-2}	meters
	1.0936	X	10^{-2}	yards
centimeter/second (cm/sec)	3.2808	X	10^{-2}	feet/second
	3.9370	X	10^{-1}	inches/second
	1.0000	X	10^{-2}	meters/second
centipoise	6.7197	X	10^{-4}	pounds(mass)/second-foot
	3.6000			kilograms/hour-meter
cubic centimeter (cm^3)	1.0000	X	10^{-3}	cubic decimeters
	3.5315	X	10^{-5}	cubic feet
	6.1024	X	10^{-2}	cubic inches
	1.0000	X	10^{-6}	cubic meters
	1.3080	X	10^{-6}	cubic yards
cubic decimeter (liter) (dm^3)	1.0000	X	10^3	cubic centimeters
	3.5315	X	10^{-2}	cubic feet
	6.1024	X	10	cubic inches
	1.0000	X	10^{-3}	cubic meters
	1.3080	X	10^{-3}	cubic yards
cubic foot (ft^3)	2.8317	X	10^4	cubic centimeters
	2.8317	X	10	cubic decimeters
	1.7280	X	10^3	cubic inches
	2.8317	X	10^{-2}	cubic meters
	3.7037	X	10^{-2}	cubic yards
cubic inch (in^3)	1.6387	X	10	cubic centimeters
	1.6387	X	10^{-2}	cubic decimeters
	5.7870	X	10^{-4}	cubic feet
	1.6387	X	10^{-5}	cubic meters
	2.1433	X	10^{-5}	cubic yards
cubic meter (m^3)	1.0000	X	10^6	cubic centimeters
	1.0000	X	10^3	cubic decimeters
	3.5315	X	10	cubic feet
	6.1024	X	10^4	cubic inches
	1.3080			cubic yards
curie	3.7000	X	10^{10}	disintegrations/second
degree (deg)	6.0000	X	10	minutes
	1.7453	X	10^{-2}	radians
	2.7778	X	10^{-3}	revolutions
	3.6000	X	10^3	seconds
dyne	1.0197	X	10^{-3}	grams
	1.0197	X	10^{-6}	kilograms
	1.0000	X	10^{-5}	newtons
	3.5970	X	10^{-3}	ounces
	2.2481	X	10^{-6}	pounds
dyne/square centimeter	2.9530	X	10^{-6}	inches of mercury
	1.0197	X	10^{-7}	kilograms/square meter
	7.5006	X	10^{-4}	millimeters of mercury
	1.0000	X	10	newtons/square meter
	1.4504	X	10^{-5}	pounds/square inch
electron volt (eV)	3.8268	X	10^{-20}	calories
	1.6022	X	10^{-12}	ergs
	1.0000	X	10^{-6}	MeV(mega electron volts)
erg	9.4782	X	10^{-11}	British thermal units
	2.3885	X	10^{-8}	calories
	1.0000			dyne-centimeters
	7.3756	X	10^{-8}	foot-pounds
	1.0000	X	10^{-7}	joules
erg/second	5.6869	X	10^{-9}	British thermal units/minute
	2.3885	X	10^{-8}	calories/second
	7.3756	X	10^{-8}	foot-pounds/second
	1.0000	X	10^{-7}	joules/second
	1.0000	X	10^{-7}	watts
flow rate, fuel (lb/hr)	4.5359	X	10^{-1}	kilograms/hour
foot (ft)	3.0480	X	10	centimeters
	1.2000	X	10	inches
	3.0480	X	10^{-4}	kilometers
	3.0480	X	10^{-1}	meters
	1.8939	X	10^{-4}	miles
	3.3333	X	10^{-1}	yards
foot-pound (ft-lb)	1.2851	X	10^{-3}	British thermal units
	1.3558	X	10^7	ergs
	5.0505	X	10^{-7}	horsepower-hours
	1.3558			joules
	3.7662	X	10^{-7}	kilowatt-hours
	1.3558			newton-meters
foot-pound/second (ft-lb/sec)	7.7104	X	10^{-2}	British thermal units/minute
	3.2383	X	10^{-1}	calories/second
	1.8182	X	10^{-3}	horsepower
	1.3558			joules/second
	1.3826	X	10^{-1}	kilogram-meters/second
	1.3558			watts
foot/second (fps)	3.0480	X	10	centimeters/second
	1.0973			kilometers/hour
	5.9248	X	10^{-7}	knots
	3.0480	X	10^{-1}	meters/second
	6.8182	X	10^{-1}	miles/hour

Conversion Factors (continued)

Multiply	By			To obtain
gallon (U.S.) (gal)	1.3368	X	10^{-1}	cubic feet
	3.7854			liters
	3.7854	X	10^{-3}	cubic meters
	8.0000			pints
	4.0000			quarts
gram (gm)	1.0000	X	10^{-3}	kilograms
	3.5274	X	10^{-2}	ounces
	2.2046	X	10^{-3}	pounds
	9.8067	X	10^{2}	dynes
	9.8067	X	10^{-3}	newtons
hectare	2.4711			acres
	1.0000	X	10^{2}	ares
	1.0000	X	10^{4}	square meters
	3.8610	X	10^{-3}	square miles
horsepower (hp)	4.2436	X	10	British thermal units/minute
	5.5000	X	10^{3}	foot-pounds/second
	3.3000	X	10^{4}	foot-pounds/minute
	7.4570	X	10^{2}	joules/second
	7.6040	X	10	kilogram-meters/second
	7.4570	X	10^{2}	watts
horsepower-hour (hp-hr)	2.5461	X	10^{3}	British thermal units
	1.9800	X	10^{6}	foot-pounds
	2.6845	X	10^{6}	joules
	7.4570	X	10^{-1}	kilowatt-hours
hour (hr)	6.0000	X	10	minutes
	3.6000	X	10^{3}	seconds
	4.1781	X	10^{-2}	sidereal days
	4.1667	X	10^{-2}	solar days
	1.1416	X	10^{-4}	solar years
imperial gallon	2.7742	X	10^{2}	cubic inches
	1.2009			gallons (U.S.)
	4.5460			liters
inch (in)	2.5400			centimeters
	8.3333	X	10^{-2}	feet
	2.5400	X	10^{-2}	meters
	2.7778	X	10^{-2}	yards
inch of mercury at 0°C (in Hg)	3.3421	X	10^{-2}	atmospheres
	3.3864	X	10^{-2}	bars
	3.3864	X	10^{4}	dynes/square centimeter
	1.3596	X	10	inches of water
	2.5400	X	10	millimeters of mercury
	3.3864	X	10^{3}	newtons/square meter
	7.0727	X	10	pounds/square foot
	4.9116	X	10^{-1}	pounds/square inch
inch/second (ips)	8.3333	X	10^{-2}	feet/second
	2.5400			centimeters/second
	2.5400	X	10^{-2}	meters/second
inch of water at 4°C (in H_2O)	2.4584	X	10^{-3}	atmospheres
	7.3556	X	10^{-2}	inches of mercury
	1.8683			millimeters of mercury
	2.4910	X	10^{2}	newtons/square meter
	3.6128	X	10^{-2}	pounds/square inch
joule (J)	9.4771	X	10^{-4}	British thermal units
	2.3889	X	10^{-1}	calories
	1.0000	X	10^{7}	dyne-centimeters
	1.0000	X	10^{7}	ergs
	7.3756	X	10^{-1}	foot-pounds
	1.0000			newton-meters
	1.0000			watt-seconds
kilogram (kg)	1.0000	X	10^{3}	grams
	3.5274	X	10	ounces
	2.2046			pounds
	6.8521	X	10^{-2}	slugs
	9.8067			newtons
	7.2330	X	10	poundals
kilogram/square meter (kg/m²)	9.6783	X	10^{-5}	atmospheres
	9.8067	X	10^{-5}	bars
	2.8959	X	10^{-3}	inches of mercury
	9.8067			newtons/square meter
kilogram/square meter (kg/m²)	6.5895			poundals/square foot
	2.0482	X	10^{-1}	pounds/square foot
	1.4223	X	10^{-3}	pounds/square inch
kilogram-meter (kgm)	9.2998	X	10^{-3}	British thermal units
	7.2330			foot-pounds
	9.8067			joules
	9.8067			newton-meters
	2.7222	X	10^{-6}	kilowatt-hours
kilogram-meter/second (kgm/sec)	3.3458	X	10	British thermal units/hour
	2.3423			calories/second
	7.2330			foot-pounds/second
	9.8067			joules/second
	1.3151	X	10^{-2}	horsepower
	9.8067	X	10^{-3}	kilowatts
kilometer (km)	3.2808	X	10^{3}	feet
	3.9370	X	10^{4}	inches
	1.0000	X	10^{3}	meters
	6.2137	X	10^{-1}	miles
	1.0936	X	10^{3}	yards
kilometer/hour (km/hr)	9.1130	X	10^{-1}	feet/second
	5.3960	X	10^{-1}	knots
	6.2137	X	10^{-1}	miles/hour
	2.7778	X	10^{-1}	meters/second
kilonewton (kN)	2.2481	X	10^{2}	pounds
kilowatt hour (kWh)	3.4128	X	10^{3}	British thermal units
	2.6560	X	10^{6}	foot-pounds
	1.3414			horsepower-hours
	3.6000	X	10^{6}	joules
	3.6721	X	10^{5}	kilogram-meters
	3.6000	X	10^{6}	watt-seconds
knot (kt)	1.6890			feet/second
	1.1516			miles/hour
	1.8532			kilometers/hour
	5.1480	X	10^{-1}	meters/second
light year	3.1040	X	10^{16}	feet
	5.8786	X	10^{12}	miles
	9.4608	X	10^{15}	meters
liter (l)	6.1024	X	10	cubic inches
	3.5315	X	10^{-2}	cubic feet
	2.6417	X	10^{-1}	gallons (U.S. liquid)
	1.0000	X	10^{-3}	cubic meters
	2.1134			pints (U.S. liquid)
	1.0567			quarts (U.S. liquid)
meter (m)	1.0000	X	10^{2}	centimeters
	3.2808			feet
	3.9370	X	10	inches
	1.0000	X	10^{-3}	kilometers
	6.2137	X	10^{-4}	miles
	1.0936			yards
meter/second (m/sec)	3.2808			feet/second
	3.6000			kilometers/hour
	1.9438			knots
	2.2369			miles/hour
metric horsepower	9.8632	X	10^{-1}	horsepower
	7.3550	X	10^{-1}	kilowatts
mile (mi)	5.2800	X	10^{3}	feet
	6.3360	X	10^{4}	inches
	1.6093			kilometers
	1.6093	X	10^{3}	meters
	3.2000	X	10^{2}	rods
	1.7600	X	10^{3}	yards
miles/hour (mph)	1.4667			feet/second
	1.6093			kilometers/hour
	8.6898	X	10^{-1}	knots
	4.4704	X	10^{-1}	meters/second
millimeter of mercury at 0°C (torr) (mm Hg)	1.3332	X	10^{3}	dynes/square centimeter
	3.9370	X	10^{-2}	inches of mercury (0°C)
	5.3526	X	10^{-1}	inches of water (4°C)
	1.3332	X	10^{3}	newtons/square meter
	1.9337	X	10^{-2}	pounds/square inch

Conversion Factors (continued)

Multiply	By			To obtain
minute (angle) (min)	1.6667	X	10^{-2}	degrees
	2.9089	X	10^{-4}	radians
	4.6296	X	10^{-5}	revolutions
	6.0000	X	10	seconds
minute (time) (min)	1.6667	X	10^{-2}	hours
	6.0000	X	10	seconds
	6.9444	X	10^{-4}	solar days
	1.9026	X	10^{-6}	solar years
nautical mile (international) (n mi)	6.0761	X	10^{3}	feet
	1.8520	X	10^{3}	meters
newton (N)	1.0000	X	10^{5}	dynes
	1.0197	X	10^{2}	grams
	1.0197	X	10^{-1}	kilograms
	2.2481	X	10^{-1}	pounds
	7.2330			poundals
newton/ square meter (pascal (Pa)) (N/m²)	9.8692	X	10^{-6}	atmospheres
	1.0000	X	10	dynes/square centimeter
	2.9530	X	10^{-4}	inches of mercury (0°C)
	1.0197	X	10^{-1}	kilograms/square meter
	6.7200	X	10^{-1}	poundals/square foot
	2.0885	X	10^{-2}	pounds/square foot
	1.4504	X	10^{-4}	pounds/square inch
ounce (oz)	2.8349	X	10	grams
	2.8349	X	10^{-2}	kilograms
	6.2500	X	10^{-2}	pounds
	1.9428	X	10^{-3}	slugs
	2.7801	X	10^{4}	dynes
parsec	1.9163	X	10^{13}	miles
	3.0857	X	10^{16}	meters
pieze	1.0000	X	10^{3}	newtons/square meter
poundmass (lb)	4.5359	X	10^{2}	grams
	4.5359	X	10^{-1}	kilograms
	1.6000	X	10	ounces
pound (force) (lbf)	3.1081	X	10^{-2}	slugs
	4.4482			newtons
	4.4482	X	10^{-1}	dekanewtons
	4.4482	X	10^{-3}	kilonewtons
	3.2174	X	10	poundals
pound/ square foot (psf)	4.7254	X	10^{-4}	atmospheres
	4.7880	X	10^{-4}	bars
	4.7880	X	10^{2}	dynes/square centimeter
	1.4139	X	10^{-2}	inches of mercury (0°C)
	4.8824			kilograms/square meter
	4.7880	X	10	newtons/square meter
	3.2174	X	10	poundals/square foot
	6.9444	X	10^{-3}	pounds/square inch
pound/ square inch (psi)	6.8046	X	10^{-2}	atmospheres
	6.8948	X	10^{4}	dynes/square centimeter
	2.0360			inches of mercury (0°C)
	2.7681	X	10	inches of water (4°C)
	7.0307	X	10^{2}	kilograms/square meter
	6.8948	X	10^{3}	newtons/square meter
	4.6333	X	10^{3}	poundals/square foot
	1.4400	X	10^{2}	pounds/square foot
poundal	1.4098	X	10^{-2}	kilograms
	1.3825	X	10^{-1}	newtons
	3.1081	X	10^{-2}	pounds
poundal/ square foot	1.5174	X	10^{-1}	kilograms/square meter
	1.4882	X	10^{-1}	newtons/square meter
	3.1081	X	10^{-2}	pounds/square foot
	2.1583	X	10^{-4}	pounds/square inch
radian (rad)	5.7296	X	10	degrees
	3.4378	X	10^{3}	minutes
	1.5916	X	10^{-1}	revolutions
	2.0626	X	10^{5}	seconds

Multiply	By			To obtain
revolution (rev)	3.6000	X	10^{2}	degrees
	2.1600	X	10^{4}	minutes
	6.2832			radians
	1.2960	X	10^{6}	seconds
second (angle) (sec)	2.7778	X	10^{-4}	degrees
	1.6667	X	10^{-2}	minutes
	4.8481	X	10^{-6}	radians
	7.7160	X	10^{-7}	revolutions
second (time) (sec)	2.7778	X	10^{-4}	hours
	1.6667	X	10^{-2}	minutes
	1.1574	X	10^{-5}	solar days
slug	1.4594	X	10^{4}	grams
	1.4594	X	10	kilograms
	5.1478	X	10^{2}	ounces
	3.2174	X	10	pounds
square centimeter (cm²)	1.0764	X	10^{-3}	square feet
	1.5500	X	10^{-1}	square inches
	1.0000	X	10^{-4}	square meters
	1.0000	X	10^{2}	square millimeters
square foot (ft²)	2.2957	X	10^{-5}	acres
	9.2903	X	10^{2}	square centimeters
	1.4400	X	10^{2}	square inches
	9.2903	X	10^{-2}	square meters
	3.5870	X	10^{-8}	square miles
	1.1111	X	10^{-1}	square yards
square inch (in²)	1.2732	X	10^{6}	circular mils
	6.4516			square centimeters
	6.9444	X	10^{-3}	square feet
	6.4516	X	10^{-4}	square meters
	6.4516	X	10^{2}	square millimeters
	1.0000	X	10^{6}	square mils
	7.7160	X	10^{-4}	square yards
square kilometers (km²)	2.4711	X	10^{2}	acres
	1.0764	X	10^{7}	square feet
	1.0000	X	10^{6}	square meters
	3.8610	X	10^{-1}	square miles
square meter (m²)	2.4711	X	10^{-4}	acres
	1.0000	X	10^{-4}	hectares
	1.0000	X	10^{4}	square centimeters
	1.0764	X	10	square feet
	1.5500	X	10^{3}	square inches
	3.8610	X	10^{-7}	square miles
	1.1960			square yards
square mile (mi²)	6.4000	X	10^{2}	acres
	2.5900	X	10^{2}	hectares
	2.7878	X	10^{7}	square feet
	2.5900			square kilometers
	2.5900	X	10^{6}	square meters
	3.0976	X	10^{6}	square yards
therme	4.1868	X	10^{8}	joules
thrust specific fuel consumption (TSFC) (lb/hr/lb Fn)	1.0197	X	10^{2}	kilograms/hour/ dekanewton
watt (joule/ second) (W)	3.4121			British thermal units/hour
	5.6869	X	10^{-2}	British thermal units/minute
	2.3900	X	10^{-1}	calories/second
	1.0000	X	10^{7}	ergs/second
	7.3756	X	10^{-1}	foot-pounds/second
	1.3410	X	10^{-3}	horsepower
	1.0197	X	10^{-1}	kilogram-meters/second
watt second (Wsec)	9.4782	X	10^{-4}	British thermal units
	7.3756	X	10^{-1}	foot-pounds
	1.0000			joule
	2.7778	X	10^{-7}	kilowatt-hours

APPENDIX B: METHODS FOR COMPUTING CIRCUMFERENCES, AREAS AND VOLUMES

In the process of airplane project design it is frequently necessary to compute the cross-section circumference, the surface area and/or the volume of components of the airplane such as: fuselages, fuselage boattails, wings, fuel tanks and nacelles.

Reference 58, by Professor E.Torenbeek of Delft University of Technology in Delft, The Netherlands is an excellent source of methods for computing such items. For that reason, Reference 58 has been copied and adapted for incorporation as Appendix B in this book.

Nomenclature

Most symbols are used only occasionaly. Therefore, the nomenclature is mentioned in the text or on the relevant figure or diagram.

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§1. Introduction

In the calculation of mass properties and aerodynamic coefficients like profile drag coefficients and stability derivatives, reference is made to frontal areas, sideview areas, wetted areas and volumes. Once the detailed shape of the major aircraft components is accurately established by means of lofting processes (e.g. ref.B3), these characteristic areas and volumes can be computed numerically or by means of graphical methods. The present subject, however, is associated with project design studies, where the detailed shape and dimensions are not always accurately known or, in some cases, may have been chosen provisionally. An estimation, accurate within a few percent of the exact value, may be acceptable when emphasis is laid on time saving.

In the present memorandum methods are presented for the calculation of circumferences, areas and volumes, having different degrees of accuracy. The most exact procedure is based on a representation of the surface contour lines by a suitable polynomial with fractional exponents. Such a two-parameter method is explained in ch.B2 and is very simple to use when a three-view drawing of the component is available. Generalized results of the various integrations are presented in diagrams (ch.B3). These are useful not only for accurate computations, but they have been used for the derivation of simplified methods as well.

The reader who is interested in practical results only may omit the reading of chapters B2 and B3. Chapters B4, B5 and B6 deal with applications for fuselages, wings, tailplanes, fuel tanks and engine nacelles. In general, both accurate and simplified methods are presented. For fuselages a review of the various available methods is given on fig. 5.

§2. Approximation of contours by analytical functions

The rigid requirements in aeronautical design have led to the adoption of streamline shapes with smooth, continuous external lines, resulting in a smooth rate of change of curvature along the entire length of the body. For fuselages, the application of a pressurized cabin makes a circular cross section or a section built up from circular sections a very desirable, if not mandatory feature. Extensive areas of double curvature, like saddle surfaces, should be avoided because of the associated costly manufacturing processes. The design criteria for wings are entirely different as the external shape is dictated by aerodynamic requirements, viz. low drag, high maximum lift, favourable characteristics at high incidences and at Mach numbers where compressibility effects are dominant. For subsonic aircraft, leading and trailing edges of wings and tail surfaces are frequently straight lines. Aerofoils have neatly

rounded noses, while sharp trailing edges are used for good lifting properties. For ease of production, intermediate sections are often constructed from tip and root sections by assuming straight lines between corresponding points ("linear lofting").

In general, it will be necessary to subdivide the body, for which the characteristic areas and volumes must be calculated, into several sections. For example, the pressurized fuselage of a transport aircraft is logically split up into a nose section, a cylindrical mid section and a tail section. The more sections are used in non-cylindrical parts, the higher will be the accuracy of prediction. For project-design purposes, however, a method is required, resulting in an error of probably not more than a few percent, using a very limited amount of subdivisions to minimize the computational time.

The general shape of many curves and external lines can be represented in a most satisfactory way by the following two-dimensional convex polynomial:

$$\left(\frac{x}{a}\right)^n + \left(\frac{y}{b}\right)^m = 1 \qquad ; \text{ for } n, m \geq 1 \qquad (1)$$

For typical fuselage nose and tail sections examples of this function are plotted in fig. 1 for two combinations of the exponents n and m. Several special cases of eq. 1 are plotted in fig. 2 and discussed below.

The straight line ($n = m = 1$), used for cones.
Parabola 1 ($n = 2$ and $m = 1$) is representative to some degree for fuselage tail sections.
Parabola 2 ($n = 1$ and $m = 2$). Provided that the base area is not too large, this parabola may occasionally be used for blunt fuselage ends.
The ellipse ($n = m = 2$), appearing as a circle in fig. 2. Many fuselage and wing sections and nacelle noses have near-elliptic shapes.
Quartic 1 ($n = 4$; $m = 2$) and quartic 2 ($n = 2$; $m = 4$), representative for blunt noses and tails.
Lamé's quartic ($n = m = 4$). This curve is of particular interest for the representation of cross-sectional shapes of small aircraft with unpressurized cabins.

It is not suggested that for a particular body the sectional contour should be approximated by any of these special functions; errors of 5 - 10% in volumes and wetted areas may be introduced by doing so. However, several of these special cases can be treated analytically and are therefore useful to check the general solution, in which intermediate (fractional) values for n and m are treated as well.

For a specified contour shape, n and m can be determined readily from the geometric parameters φ and γ, defined in fig. 3. To this end, a suitable choice of the X- and Y-axes must be made.
For fuselages, the X-axis will usually be located in the plane of symmetry, the Y-axis may be on the intersection of the cylindrical part with the nose or tail section.
For wing sections, the affect of camber on the circumferential length of a section is negligible. On the corresponding symmetrical section, the X-axis is the chordline and the Y-axis will be taken at the point of maximum section thickness.

Point S in fig. 3 is determined graphically in the drawing as the intersection of the contourline and the diagonal of the rectangle which encloses the contour. The X-coordinate of S as a fraction of the projected contourlength (a) defines the parameter φ. The intersection of the tangent at S and the X-axis defines the parameter γ. The projected height of the contour is b. Although the graphical construction of a tangent is not always a very accurate procedure, it will be shown later on that the results are not affected to a great deal by errors in γ.

The equations relating combinations (n,m) and (φ,γ) to each other are derived in Ref. 58 ; the results are plotted in diagram 1. Once φ and γ are measured in the drawing, n and m can be found and, if desired, a check on the accuracy of the approximation by eq. 1 can be made by calculation of the contour. However, in ch.§3 it will be demonstrated that for the purpose of calculating volumes and areas, considerable deviations from the actual shape are acceptable, provided that the value of φ is accurate. For example, the parabolae 1 and 2 in fig. 2 have the same value $\varphi = \frac{1}{2}(\sqrt{5}-1)$. The sectional areas for both curves are exactly equal, the circumferential lengths are equal within a few tenths of a percent, while the volumes and wetted areas of corresponding bodies of revolution differ by one or two percent only.

§3. Diagrams for areas, circumferences and volumes

The choice of the polynomial given by eq. 1 provides the tool to calculate areas, circumferences and volumes of body sections. The suitability of the function is confirmed by the fact that all results can be plotted in two-dimensional diagrams, although 4 geometric parameters (φ, γ, a and b) are involved. The diagrams can be used for a very quick calculation, as no integrations are necessary.
For cross sections of pressure cabins a separate diagram is included.

B3.1 Sectional (projected) area - diagram 2

The area enclosed by the contour, the X-axis and the Y-axis is:

$$\text{SECTIONAL AREA} = \int_0^a y\,dx \qquad (2)$$

By substitution of y from eq. 1 and integration, the final result as derived in Ref. 58 is found and plotted in diagram 2. In terms of n and m, the area is determined by Γ-functions, which are symmetrical with respect to n and m. The simple representation in diagram 2 is possible, as the area is not affected by γ.

B3.2 Circumference of a section contour - diagram 3

The expression for the circumferential length is:

$$\text{CIRCUMFERENCE} = \int\sqrt{(dx)^2 + (dy)^2} = \int_0^a \sqrt{1 + \left(\frac{dy}{dx}\right)^2}\,dx \qquad (3)$$

For several special combinations of n and m, this integral can be solved analytically Ref. 58 , resulting in expressions with n, m and the ratio a/b as parameters. As in the case of the sectional area, it appears convenient to use φ and γ instead, but the effect of γ appears to be small. Hence, in diagram 3 only φ and a/b are used as parameters. Although the result is now no longer exact, the error introduced is quite small.
For example:

a) the difference between the exact results for the two parabolas 1 and 2 is indistinguishable in the diagram,
b) the two quartics (n = 4; m = 2 and n = 2; m = 4) are both represented by the dotted line $\varphi = \sqrt{(\sqrt{5}-1)/2} = 0.786$. For a given value of a/b, the error with respect to the exact result, as shown in the diagram, is less than one percent.

In some cases, the calculation of the circumferential length of a cross section is an intermediate step in the computation of wetted area. In those cases, the height/width ratio is always between 0.5 and 2 and, within the corresponding range $0.5 < a/b < 1.0$, the circumference is hardly affected by a/b as well. This property will be used for the simplified calculations of par. 4.2.

B3.3 Volume of bodies of revolution - diagram 4

By rotation of the contour, given by eq. 1, about the X-axis, a body of revolution is formed, with a volume equal to:

$$\text{VOLUME} = \int_0^a \pi y^2\,dx \qquad (4)$$

The complete solution is derived in Ref. 58 in terms of n and m and plotted in diagram 4. The parameters φ and γ may be used instead. The volume, expressed as a fraction of the cylinder volume (length a, radius b), is not affected by a/b. As in the previous cases, the parameter φ is most prominent, γ is of secondary importance. It can be shown that formulae for the sectional shape different from eq. 1 yield essentially the same results, provided that the same value of φ is used.

§3.4 Wetted area of bodies of revolution - diagram 5

The body of revolution as defined in the previous paragraph has the following wetted area:

$$\text{EXTERNAL AREA} = \int 2\pi y\sqrt{(dx)^2 + (dy)^2} = 2\pi \int_0^b y\sqrt{1 + \left(\frac{dy}{dx}\right)^2}\, dx \qquad (5)$$

For special combinations of n and m, analytical expressions for this integral can be derived (see Ref. 56). The wetted area, plotted in diagram 5 as a fraction of the cylinder external area ($2\pi ab$) plus one side area (πb^2), is affected primarily by φ and a/b. In analogy to the case for the circumference, the omission of γ results in errors of not more than 1 or 2%. This is illustrated in the diagram for the two parabolas, both represented by $\varphi = \frac{1}{2}(\sqrt{5} - 1)$ and the two quartics, with $\varphi = \sqrt{(\sqrt{5} - 1)/2}$.

§3.5 Correction factors for double bubble and flattened cross sections - diagram 6

For fuselages with "double bubble" cross sectional shapes, like the DC-8, DC-9, VC-10 and others, or a flattened belly below the cabin floor (e.g. F-27, C-5A), correction factors can be derived, relating the circumference and sectional area to those of a circle. The circle diameter is assumed equal to the max. width of the actual fuselage. The relevant equations are presented in fig. 4 and the results plotted in diagram 6. All parameters used can be measured on a drawing of the cross section or front view.

§4. Fuselages

The methods presented in this chapter refer to the gross wetted area of the streamline body by which most fuselage shapes can be approximated. Cockpit canopies, fillets, wing-fuselage attachments, air scoops, etc., are ignored and must be accounted for seperately.
A choice can be made between several methods. The most accurate of these is described in par.§4.1 and can be used when a complete three-view drawing is available. For substantially non-cylindrical fuselages, several cross sections must be available as well. The simplified methods discussed in par.§4.2 are useful when the fuselage shape or some dimensions are not completely known. A survey of the applicability of the various methods is given in fig. 5.

§4.1 "Accurate" calculations

a. Fuselages with circular cross sections

The nose and tail sections are usually nearly bodies of revolution as well. The planview is used to subdivide the fuselage into a nose section, a cylindrical mid section (if present) and a tail section and the parameters ρ and γ are determined (fig. 3). Projected areas in sideview or planview can be derived from diagram 2. Volumes and wetted areas are computed with diagrams 4 and 5.

b. Fuselages with a blunt base or a beaver tail

The approximation for blunt and beaver tails, as presented in diagrams 7 and 8, can be used on the condition that the contour lines are not too different from a parabolic shape. Formulae for ogives can be found in ref.B6.

c. Fuselages with double-bubble or flattened cross sections

The volume and wetted area are computed, assuming that the fuselage is a body of revolution with diameter equal to the maximum width of the fuselage (fig. 4). Similarly to case a., the planview is used. The correction factors in diagram 6 are then applied to the frontal area, the volume and the wetted area.

d. Fuselages with non-circular cross sections

The following procedure is suggested:

1. A subdivision into a suitable number of segments is made.

2. The parallel end faces of the segments are local cross sections; their areas and circumference are calculated with diagrams 2 and 3.
3. The segments are approximated by bodies of type A or B in fig. 7. Their volumes are computed and added.
4. The circumferential lengths of the cross sections are plotted on the longitudinal axis and integrated numerically (e.g. Simpson's rule, ref.86) or with diagram 3.
5. The integral is corrected by addition of an extra percentage:

$$\frac{\Delta(\text{wetted area})}{\text{wetted area}} = 75 \frac{\text{frontal area}}{(\text{fuselage length})^2} \%$$ (6)

This approximate correction accounts for the fact that integration of the circumferential length should take place along the external contour instead of the longitudinal axis.

§4.2 Simplified methods

In many cases the fuselage contour is not completely defined and a certain amount of help from statistical data will be acceptable. The designer can make a choice between the following approximate methods:

a. Three-view drawing is available

In chapters 2 and 3 the parameters φ and γ were introduced to find the fractional exponents n and m. It was found that

. the effect of γ is very small or completely absent,

. for practical cross-sectional shapes the effect of the height/width ratio on the circumference is negligible.

Moreover, by application of the method to practical fuselages, it was concluded that for the most common nose and tail sections the wetted area as a fraction of the cilinder area ($2\pi ab$ in diagram 5) is approximately a function of φ only. Therefore, the procedure of ch.§4.1 can be simplified and the results are summarised in diagram 9.

b. Fuselage length, cross-sectional shape and length of mid section are known

Diagram 10 is composed from the results of calculations of volumes and wetted areas for several actual fuselages with near-circular cross sections, using diagrams 2-6. The dotted line is taken as the mean value and corrections are derived from diagram 2 and 3. This simple diagram may be of use when the actual shape of nose and tail sections are not specified.

c. Axisymmetric streamline t dies of given length and diameter

In fig. 6 the ratio:

$$\frac{\text{wetted area}}{2\sqrt[3]{2\pi\lambda}(\text{volume})^{2/3}}, \text{ where } \lambda = \frac{\text{length}}{\text{diameter}},$$

is plotted for cylinders, ellipsoids, paraboloids, double cones and typical streamline shapes with and without a cylindrical center section. This ratio is approximated by a simple function of λ for streamline bodies. Once the volume is known, the wetted area can be calculated in a straight forward manner. The expressions for the volume are based on typical nose and tail section shapes, according to fig. 1.

d. Volume and wetted area based on cabin dimensions

When observing actual fuselage shapes, it is noted that the fuselage length is considerably affected by the detailed shape of the nose and the tail extremities. On the contrary, the volume and wetted area are much less affected. In preliminary design the fuselage tail length may be subject to optimization at a later stage, when accurate data are available on structure weight and afterbody drag, in order to find the most favourable tailplane moment arm.

The dimensions of the passenger cabin and the freighthold volume, however, are established at a very early phase of the design, these being derived mainly from the design specifications. Hence, diagram 11 gives a statistical correlation of gross wetted areas and volumes with the principal dimensions of the pressurized section and with cabin plus freighthold volume. The diagram may be used in different ways:

- for given pressure cabin outside dimensions, the volume and wetted area are found immediately;
- if only the passenger cabin plus freighthold volume are known, the fuselage volume is read from the diagram and the wetted area is computed with the presented formula (based on fig. 6).

For payloads in excess of 30 passengers roughly, the correlation is remarkable in view of the simplicity of the method.

B5. Wings, tailplanes and fuel tanks

B5.1 Wetted area of wings and tailplanes

The wetted area is computed by spanwise integration of the sectional circumference. The net (exposed) area (S_{net}) to be used is the horizontal projection of the area exposed to the airflow.

The circumference of an arbitrary section can be computed from diagram 3, using a suitable subdivision of the contour. For most subsonic sections, the following simple expression is reasonably accurate:

$$\text{circumference} = 2 \times \text{chordlength}\ (1 + 0.25 \cdot t/c) \quad (7)$$

where t/c = thickness/chord ratio (fig. 8)

The wetted area of a linear lofted lifting surface is derived by integration:

$$\text{wetted area} = 2S_{net}\left\{1 + 0.25(t/c)_r \frac{1+r\lambda}{1+\lambda}\right\} \qquad (8)$$

λ = tipchord/root chord (taper ratio)
$r = (t/c)_t \,/\, (t/c)_r$
r = root; t = tip

In this particular case, the root and tip sections refer to the exposed part of the wing or tailplane.

B5.2 Fuel tank volume

a. Most internal bag-type tanks can be represented by a geometric body with parallel end faces of type A or B in fig. 7.
b. For integral tanks the external shape of the structure will be used to apply fig. 7. In this case, the volume of the structure * should be subtracted. In a completely filled tank roughly 5% of the volume must be available for expansion of the fuel (subsonic aircraft).
c. External fuel tanks can be treated in a similar fashion as the fuselage, e.g. with fig. 6 or diagram 10.
d. In pr ject design a check on the available tank capacity must be made. A first estimate for the integral tank volume is given in fig. 8. Although statistical data were used to derive the constant 0.54, the method is not very accurate. Hence, when the available tank volume appears to be critical, a more precise calculation is necessary, to account for the actual section variation and the location of the spars.

B6. Engine nacelles and air ducts

In the most general case, the engine nacelle group may consist of a fan cowling, a gasgenerator cowling and a plug in the hot flow (fig. 9). The wetted areas of these components can be computed with the data on fig. 9 and diagram 7.

---

* to be derived from the structural weight.

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FIG. 1 : REPRESENTATION OF TYPICAL NOSE AND TAIL SECTIONS OF STREAMLINE BODIES BY MATHEMATICAL FUNCTIONS

FIG. 2 : SPECIAL CASES OF THE FUNCTION $\left(\frac{x}{a}\right)^n + \left(\frac{y}{b}\right)^m = 1$

FIG. 3 : GRAPHICAL DETERMINATION OF THE PARAMETERS φ AND γ FROM THE DESIGN DRAWING

$$\cos\alpha = 2\frac{h_2}{b} - 1$$

$$\cos\beta = \frac{1 - b/h_2 + (h_1/h_2 - 1)^2}{1 - b/h_2 - (h_1/h_2 - 1)^2}$$

NOTE : CIRCUMFERENCE AND AREA ARE PLOTTED IN DIAGRAM 6 IN TERMS OF h_2/b AND h_2/h_1

$$\frac{\text{CIRCUMFERENCE}}{\pi b} = 1 - \frac{\alpha}{\pi} + \frac{\beta}{\pi}\,\frac{\sin\alpha}{\sin\beta}$$

$$\frac{\text{AREA}}{\frac{\pi}{4}b^2} = 1 - \left(\frac{\alpha}{\pi} - \frac{\sin 2\alpha}{2\pi}\right) + \left(\frac{\beta}{\pi} - \frac{\sin 2\beta}{2\pi}\right)\left(\frac{\sin\alpha}{\sin\beta}\right)^2$$

FIG. 4 : GEOMETRY OF CROSS SECTIONS COMPOSED OF TWO CIRCLE SECTORS

FIG. 5 : CHOICE OF THE METHOD TO COMPUTE FUSELAGE VOLUME AND CHARACTERISTIC AREAS

STREAMLINE BODY WITH CYLINDRICAL CENTER SECTION ($\lambda > 4.5$)

FOR FUSELAGES WITH CYLINDRICAL CENTER SECTION, NOSE AND TAIL SECTION SHAPES ACCORDING TO FIG. 1 AND LENGTH/DIAMETER RATIOS EQUAL TO 1.75 AND 2.75 RESP. :

$$\left.\begin{aligned} \text{VOLUME} &= \frac{\pi}{4} D^2 l_f \left(1 - \frac{2}{\lambda}\right) \\ \text{WETTED AREA} &= \pi D l_f \left(1 - \frac{2}{\lambda}\right)^{2/3} \left(1 + \frac{1}{\lambda^2}\right) \end{aligned}\right\} \lambda \geq 4.5$$

FOR STREAMLINE BODIES WITHOUT CYLINDRICAL CENTER SECTION, NOSE AND TAIL SECTION SHAPES ACCORDING TO FIG. 1 :

$$\text{VOLUME} = \frac{\pi}{4} D^2 l_f \left(0.50 + 0.135 \frac{l_n}{l_f}\right) \qquad (l_n = \text{LENGTH OF BODY NOSE})$$

$$\text{WETTED AREA} = \pi D l_f \left(0.50 + 0.135 \frac{l_n}{l_f}\right)^{2/3} \left(1.018 + \frac{0.3}{\lambda^{3/2}}\right)$$

FIG. 6 : VOLUME AND WETTED AREA OF AXISYMMETRIC STREAMLINE BODIES.

FIG. 7 : VOLUME OF GEOMETRIC BODIES WITH PARALLEL END FACES

FIG. 8.: APPROXIMATION FOR INTEGRAL FUEL TANK VOLUME, AVAILABLE IN A LINEAR LOFTED WING

COMPONENTS OF AN ENGINE NACELLE GROUP

EXTERNAL WETTED AREA OF FAN COWLING:

$$l_n D_n \left\{ 2 + 0.35\beta + 0.40\beta \frac{D_b}{D_n} + 1.15(1-\beta)\frac{D_s}{D_n} \right\}$$

WETTED AREA OF GASGENERATOR COWLING: DIAGRAM 7

WETTED AREA OF PLUG:

2 * PLUG LENGTH * MAX. DIAMETER

NO. 8: WETTED AREA OF AN ENGINE NACELLE

GENERAL EQUATION OF THE CONTOUR : $\left(\frac{x}{a}\right)^n + \left(\frac{y}{b}\right)^m = 1$

$\varphi = \frac{OP}{OA}$; $\gamma = \frac{OP}{PS}$

RELATION BETWEEN (φ, γ) AND (n, m) :

$\varphi^n + \varphi^m = 1$; $\gamma = \frac{n}{m}\varphi^{n-m}$

DIAGRAM 1: DETERMINATION OF n AND m FROM KNOWN VALUES OF φ AND γ

DIAGRAM 2: CROSS-SECTIONAL AREA

DIAGRAM 3: CIRCUMFERENCE OF A CONTOURLINE

DIAGRAM 4. VOLUME OF A BODY OF REVOLUTION

DIAGRAM 5: EXTERNAL AREA OF A BODY OF REVOLUTION.

DIAGRAM 6: AREA AND CIRCUMFERENCE OF A CROSS SECTION BUILT UP FROM TWO CIRCLE SECTORS

$$\frac{\text{VOLUME}}{\frac{\pi}{4} D^2 L} = \frac{1}{15}\left\{8 + 4\frac{d}{D} + 3\left(\frac{d}{D}\right)^2\right\}$$

DIAGRAM 7: GEOMETRIC PROPERTIES OF A FRUSTRUM OF A PARABOLOID

$$\frac{\text{VOLUME}}{\frac{\pi}{4} D^2 L} = \frac{1}{15}\left\{8 + 2\frac{b}{D}\right\}$$

DIAGRAM 8: APROXIMATE WETTED AREA AND VOLUME OF A "BEAVER TAIL"

FRONT VIEW : AREA $S_f = k_A \cdot b_{f_{max}} \cdot h_{f_{max}}$

CIRCUMFERENCE $C_f = 2\, k_c\, (b_{f_{max}} + h_{f_{max}})$

(IF NECESSARY , THE FRONT VIEW IS SUBDIVIDED INTO SEVERAL PARTS)

VOLUME : $V_f = S_f\,(l_c + k_{v_n}\, l_n + k_{v_t}\, l_t)$

k_v IS DETERMINED FROM THE PLANVIEW FOR THE NOSE SECTION (k_{v_n}) AND THE TAIL SECTION (k_{v_t})

WETTED AREA : $S_{WET} = C_f\,(l_c + k_{w_n}\, l_n + k_{w_t}\, l_t)$

k_w - VALUES ARE DETERMINED FROM THE PLAN VIEW

PLANVIEW AND SIDEVIEW AREAS CAN BE COMPUTED IN SIMILAR FASHION

DIAGRAM 10: APPROXIMATE CALCULATION OF FUSELAGE VOLUME AND WETTED AREA (GIVEN: LENGTH OF CYL. SECTION)

NOTE:
CABINE + FREIGHT HOLD VOLUME ≈ 60 % OF FUSELAGE VOLUME

ACCORDING TO FIG.6:
FUSELAGE WETTED AREA = $2\sqrt[3]{2\pi\lambda}\left(1+\frac{1}{\lambda^2}\right)(\text{VOLUME})^{2/3}$, $\lambda = \frac{\text{FUSELAGE LENGTH}}{\text{DIAMETER}}$

DIAGRAM II : STATISTICAL CORRELATION OF VOLUME AND WETTED AREA BASED ON PRESSURE CABIN DIMENSIONS (VALID FOR FUSELAGES WITH CILINDRICAL SECTION)

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AIRPLANE DESIGN

PART VII: DETERMINATION OF STABILITY, CONTROL AND PERFORMANCE CHARACTERISTICS: FAR AND MILITARY REQUIREMENTS

by

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IMPORTANT NOTES:

1. All aerodynamic coefficients and derivatives are defined in the stability axis system, unless stated to the contrary!

2. All rate derivatives are with respect to:

   (rate)$\bar{c}/2U_1$ or (rate)$b/2U_1$ depending on

   whether they are longitudinal or lateral-directional derivatives.

3. All speed derivatives are with respect to: u/U_1

4. Appendices contain separate symbol listings.

1. GENERAL SYMBOLS

Symbol	Definition	Dimension
$\bar{a}$	average deceleration during the groundrun	ft/sec^2
a_{lat}	lateral acceleration	ft/sec^2
a_{vert}	vertical acceleration	ft/sec^2
A	Wing aspect ratio	-----
A_h	Horizontal tail aspect ratio	-----
b	wing span	ft
b_f	flapped wing span	ft

$\bar{c}$	wing mean geometric chord, m.g.c.	ft
c_j	specific fuel consumption	lbs/lbs/hr
c_p	specific fuel consumption	lbs/hp/hr
$\bar{c}_{subscript}$	mean geometric chord of the surface corresponding to subscript area	ft
c_β	gear cornering coefficient, see page 47	lbs/deg
C_{D_o}	Zero lift drag coefficient	-----
C_{D_1}	Airplane steady state drag coefficient	-----
C_{D_u}	Airplane drag-due-to-speed derivative	-----
CGR	Climb gradient	rad
C_h	Hingemoment coefficient	-----
C_{h_o}	Hingemoment coefficient at zero angle of attack	-----
C_{h_α}	Hingemoment derivative due to angle of attack	1/rad
C_{h_β}	Hingemoment derivative due to sideslip angle	1/rad
C_{h_δ}	Hingemoment derivative due to control surface defl.	1/rad
C_{l_p}	Rolling moment due to roll rate derivative	1/rad
C_{l_r}	Rolling moment due to yaw rate derivative	1/rad
C_{l_β}	Rolling moment due to sideslip derivative	1/rad
$C_{l_{\beta_B}}$	Rolling moment due to sideslip derivative in the body axis system	1/rad
$C_{l_{\delta_a}}$	Rolling moment due to aileron derivative	1/rad
$C_{l_{\delta_{cpt}}}$	Rolling moment due to cockpit controller derivative, see Eqn.(2.63)	1/rad
$C_{l_{\delta_r}}$	Rolling moment due to rudder derivative	1/rad
$C_{l_{\delta_s}}$	Rolling moment due to spoiler derivative	1/rad

$C_L = W/\bar{q}S$	Airplane lift coefficient level flight	-----
C_{L_o}	Airplane lift coefficient at zero angle of attack	-----
C_{L_1}	Airplane steady state lift coefficient	-----
C_{L_q}	Lift due to pitchrate derivative	1/rad
C_{L_α}	Airplane lift curve slope	1/rad
$C_{L_{\delta_{ctrl}}}$	Airplane lift due to control surface deflection deriv.	1/rad
$C_{L_{\delta_e}}$	Airplane lift due to elevator deflection deriv.	1/rad
$C_{L_{wf}}$	Lift coefficient of the wing + fuselage	-----
$c_{m_{ac_f}}$	section pitching moment coefficient including flaps. about the section a.c.	-----
$C_{m_{ac_{wf}}}$	Pitching moment coefficent about the wing/fuselage aerodynamic center	-----
C_m	Airplane pitching moment coefficient	-----
C_{m_o}	Airplane pitching moment coefficient at zero angle of attack	-----
C_{m_α}	Airplane static longitudinal stability deriv.	1/rad
$C_{m_{\delta_{ctrl}}}$	Airplane pitching moment due to control surface deflection derivative	1/rad
$C_{m_{\delta_e}}$	Airplane pitching moment due to elevator deriv.	1/rad
C_{m_q}	Airplane pitching moment due to pitchrate deriv.	1/rad
C_{n_p}	Yawing moment due to roll-rate derivative	1/rad
C_{n_r}	Yawing moment due to yaw-rate derivative	1/rad
C_{n_β}	Yawing moment due to sideslip derivative	1/rad
$C_{n_{\beta_B}}$	Yawing moment due to sideslip in the body fixed axis system	1/rad
$C_{n_{\delta_a}}$	Yawing moment due to aileron derivative	1/rad

$C_{n_{\delta_r}}$	Yawing moment due to rudder derivative	1/rad
C_{ride}	Ride comfort index	-----
$C_{T_{x_1}}$	Airplane steady state thrust coefficient	-----
$C_{T_{x_u}}$	Airplane thrust-due-to-speed derivative	-----
C_{y_p}	Sideforce due to rollrate derivative	1/rad
C_{y_r}	Sideforce due to yawrate derivative	1/rad
C_{y_β}	Sideforce due to sideslip derivative	1/rad
$C_{y_{\delta_a}}$	Sideforce due to aileron deflection deriv.	1/rad
$C_{y_{\delta_r}}$	Sideforce due to rudder deflection deriv.	1/rad
d_T	thrust moment arm in pitch about the airplane c.g.	ft
$D = C_D\bar{q}S$	Airplane drag	lbs
e	Oswald's efficiency factor	-----
E	Endurance	hrs
f_{mp}	range factor, page 134	-----
f_{TO}	obstacle height factor, see page 119	-----
F_a	Lateral cockpit control force	lbs
F_r	Directional cockpit control force	lbs
F_s	Longitudinal cockpit control force	lbs
F_{T_y}	Sideforce component due to thrust	lbs
g=32.2	acceleration of gravity	ft/sec^2
G	gearing ratio	rad/ft
h	altitude	ft
h_L	obstacle height for landing	ft

h_{TO}	obstacle height for takeoff	ft
HM	Hingemoment	ftlbs
I_{xx}	Rolling moment of inertia about the c.g.	$slugft^2$
I_{yy}	Pitching moment of inertia about the c.g.	$slugft^2$
$I_{yy_{mg}}$	Pitching moment of inertia about the main gear to runway contact point	$slugft^2$
I_{zz}	Yawing moment of inertia about the c.g.	$slugft^2$
K	See Eqn.(4.7), p.109	-----
K_1	Gust parameter, see p.107	-----
K_2	Gust parameter, see pages 107 and 108	-----
K_a, K_r	Proportionality constant for cockpit control force coupling of aileron to rudder and vice versa	lbs/rad
l_h	distance of the horizontal tail aerodynamic center to the airplane c.g.	ft
$L = C_L\bar{q}S$	Airplane lift	lbs
L.E.	Leading Edge	-----
L_h	Lift acting on the horizontal tail at the hor. tail aerodynamic center	lbs
L_v	Gust length in y-dir.	ft
L_w	Gust length in z-dir.	ft
L_{wf}	Lift acting on wing + fuselage at the wing/fuselage aerodynamic center	lbs
L_p	$C_{l_p}\bar{q}_1Sb^2/2I_{xx}U_1$	1/sec
L_r	$C_{l_r}\bar{q}_1Sb^2/2I_{xx}U_1$	1/sec
L_T	Rolling moment due to thrust	ftlbs
L_β	$C_{l_\beta}\bar{q}_1Sb/I_{xx}$	1/sec

$m=W/g$	airplane mass	slugs
$M_{ac_{wf}}$	Pitching moment about the wing/fuselage aerodynamic center	ftlbs
M_q	$C_{m_q}\bar{q}_1 S\bar{c}^2/2I_{yy}U_1$	1/sec
M_α	$C_{m_\alpha}\bar{q}_1 S\bar{c}/I_{yy}$	$1/sec^2$
$M_{\dot\alpha}$	$C_{m_{\dot\alpha}}\bar{q}_1 S\bar{c}^2/2I_{yy}U_1$	1/sec
n	load factor (L/W)	-----
$\bar{n}$	load factor function, see page 32	-----
n_α	load factor derivative with respect to angle of attack	1/rad
$N_{1,2,3}$	Numerator determinants defined in Table 2.1	
N_D	Yawing moment due to asymmetric drag	ftlbs
N_r	$C_{n_r}\bar{q}_1 Sb^2/2I_{zz}U_1$	1/sec
N_T	Yawing moment due to asymmetric thrust	ftlbs
N_β	$C_{n_\beta}\bar{q}_1 Sb/I_{zz}$	$1/sec^2$
p	roll rate	deg/sec
P_{mg}	Vertical reaction force on the main gear	lbs
P_{ng}	Vertical reaction force on the nose gear	lbs
P_{reqd}	$T_{reqd}U_1/550$, power reqd	hp
P_s	Specific excess power	ft/sec
$\bar{q}$	dynamic pressure	psf

Q_1	Steady state pitch rate	rad/sec
R	Range	nm or sm
RC	Rate of climb	ft/min or ft/sec
RD	Rate of descent	ft/min or ft/sec
R_e/R_r	Rigid-to-elastic ratio	-----
R_L	Loop radius	ft
R_t	Turn radius	ft
s_{AIR}	Air distance in landing	ft
s_{FL}	Landing fieldlength	ft
s_L	Landing distance	ft
s_{LG}	Landing ground distance	ft
s_{TO}	Takeoff distance	ft
s_{TOG}	Takeoff ground distance	ft
S	wing area	ft^2
$S.M._{free}$	Static margin, stick free	-----
$S_{subscript}$	Surface area corresponding to subscript area	ft^2
t	time	sec
T	installed thrust	lbs
$\bar{T}$	Average thrust (p.117)	lbs
T_{reqd}	W/(L/D). Thrust reqd	lbs
T_R	Roll time constant	sec
T_2	Time to double amplitude	sec
$\dot{u}$	airplane forward acceleration, also:	ft/sec^2
$\dot{u}$	bleed rate	ft/sec^2
U	forward speed	ft/sec
$\dot{U}$	forward acceleration	ft/sec^2
U_1	Steady state speed	fps
V	Speed	fps
V_A	Approach speed	fps

V_s	Stall speed	fps
$V_{s_{PA}}$	Stall speed in power approach	fps
V_{TD}	Touchdown speed	fps
W	Airplane weight	lbs
$\dot{W}_F$	Fuel flow rate	lbs/sec
$\bar{x}_{ac_A}$	airplane aerodynamic center location in fractions of wing m.g.c.	-----
x_{ac_h}	horiz. tail aerodynamic center location relative to leading edge of the wing m.g.c.	ft
$x_{ac_{wf}}$	location of wing/fuselage aerodynamic center relative to leading edge of wing m.g.c.	ft
$\bar{x}_{cg}$	airplane center of gravity location in fractions of wing m.g.c.	-----
x_{ng}	distance from nose gear to leading edge of wing m.g.c.	ft
x_{mg}	distance from main gear to leading edge of wing m.g.c.	ft
y_{mg}	lateral distance from the main gear to the c.g.	ft
Y_{ng}	Lateral ground friction force on nose gear	lbs
Y_{mg}	Lateral ground friction force on main gear	lbs
Y_r	$C_{Y_r}\bar{q}_1 Sb/2mU_1$	ft/sec
Y_β	$C_{Y_\beta}\bar{q}_1 S/m$	ft/sec^2
z_D	vertical distance from drag force to the c.g.	ft
z_{mg}	vertical distance from the c.g. to the main gear to ground contact point	ft

z_T	vertical distance from the thrustline to the c.g.	ft
Z_α	$-\bar{q}_1 S(C_{L_\alpha} + C_{D_1})/m$	ft/sec^2

2. GREEK SYMBOLS

α	angle of attack	deg or rad
α_g	angle of attack on the ground	deg or rad
α_δ	angle of attack effectiveness due to control surface deflection	-----
β	sideslip angle	deg or rad
β_{cross}	crosswind induced sideslip	deg or rad
$\beta_{ng \text{ or } mg}$	gear slip angle	deg or rad
γ	flight path angle	deg or rad
$\dot{\gamma}$	rate of change of flight path angle	rad/sec
γ_2	angle defined on p.121	deg or rad
$\gamma_{2_{min}}$	angle defined on p.122	deg or rad
Λ_{le}	leading edge sweep angle	deg or rad
δ	control surface deflection angle	deg or rad
δ_{cpt}	cockpit control deflection	deg or rad
Δ	Denominator determinant defined in Table 2.1, also: incremental value of ...	
ϵ	downwash angle at the horizontal tail	deg or rad
η_h	ratio of dynamic pressure at hor. tail to free stream	-----
θ	pitch attitude angle	deg or rad
λ	taper ratio, also engine bypass ratio	-----
μ_g	tire-to-ground friction coefficient	-----
μ'	see p.121	-----
ρ	air density	$slug/ft^3$

σ_v	rms lateral gust velocity	fps
σ_w	rms vert. gust velocity	fps
ζ	damping ratio	-----
τ_R	roll time constant, also: T_R	sec
ϕ	bank angle	deg or rad
ϕ_T	thrust inclination angle	deg or rad
$\dot{\Psi}$	Airplane turnrate	deg/sec or rad/sec
Ψ_A	Airplane crab angle	deg or rad
Ψ_{steer}	Nose gear steering angle	deg or rad
ω_n	undamped natural freq.	rad/sec

3. SUBSCRIPTS

1	Steady state
2	takeoff safety speed
3	speed at obstacle
a	aileron
abs	absolute
cl	climb
cr	cruise
cross	crosswind
e	elevator
end	end
fg	front gear
h	horizontal tail
in	initial
lat	lateral
mc	minimum control
mg	main gear
man	maneuver
max	maximum
min	minimum
ng	nose gear

o	sealevel
r	rudder
reqd	required
rg	rear gear
s	stall, also: stability axes
sb	speedbrake
t	tab
tg	tail gear
v	vertical tail
w	wing
wf	wing + fuselage
A	Approach
B	Body axes
D	Dutch Roll
E	Empty
F	Fuel
GL	Glide
L	Landing, also: Left
P	Phugoid
PL	Payload
R	Roll Also: Right
S	Spiral
SP	Short Period
T	Throttle
TO	Takeoff

4. Acronyms

AEO	All engines operating
ASW	Anti submarine warfare
BFL	Balanced fieldlength
CGR	Climb gradient
LOF	Liftoff
NADC	Naval Air Development Center
OEI	One engine inoperative
P.D.	Preliminary design
rms	root mean square
SHP	Shaft horsepower

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1. INTRODUCTION

The purpose of this series of books on Airplane Design is to familiarize aerospace engineering students with the design methodology and design decision making involved in the process of designing airplanes.

The series of books is organized as follows:

Part I:	PRELIMINARY SIZING OF AIRPLANES
Part II:	PRELIMINARY CONFIGURATION DESIGN AND INTEGRATION OF THE PROPULSION SYSTEM
PART III:	LAYOUT DESIGN OF COCKPIT, FUSELAGE, WING AND EMPENNAGE: CUTAWAYS AND INBOARD PROFILES
PART IV:	LAYOUT DESIGN OF LANDING GEAR AND SYSTEMS
PART V:	COMPONENT WEIGHT ESTIMATION
PART VI:	PRELIMINARY CALCULATION OF AERODYNAMIC, THRUST AND POWER CHARACTERISTICS
PART VII:	DETERMINATION OF STABILITY, CONTROL AND PERFORMANCE CHARACTERISTICS: FAR AND MILITARY REQUIREMENTS
PART VIII:	AIRPLANE COST ESTIMATION: DESIGN, DEVELOPMENT, MANUFACTURING AND OPERATING

In this part (Part VII) frequent use is made of most other parts in this series on Airplane Design: Refs 1-7.

In Parts I and II so-called Class I methods for sizing airplanes to performance, stability and control requirements were presented. These methods are compatible with Preliminary Design Sequence I as outlined in Steps 1-16 in Part II. A flowchart which relates Class I methods and P.D. Sequence I to Class II methods and P.D. Sequence II is presented in Table 1.1.

The purpose of Part VII is to present Class II preliminary design methods for determining stability, control and performance characteristics of airplanes. These methods are compatible with Steps 17-36 of Preliminary Design Sequence II as outlined also in Part II.

The objective of these methods is to assure that:

1. the mission requirements of the airplane (as stated in the mission specification) are satisfied

2. the airworthiness regulations (as stated in References 8, 9 and 10) are met

Table 1.1 Flowchart of the Preliminary Design Process

Airplanes must be controllable, maneuverable and trimmable to be safe as well as useful. Chapter 2 presents methods for assessing the controllability, the maneuverability and the trimmability capabilities of new airplane designs.

Stability is a requirement for all airplanes*. Methods for predicting the static and dynamic stability characteristics of airplanes are found in Chapter 3.

To be useful most airplanes must possess ride qualities such that the crew can carry out its functions and such that passengers are not made uncomfortable. The subject of ride qualities is taken up in Chapter 4.

All airplanes must satisfy specific performance objectives which are normally defined in a statement of mission requirements. For an airplane to be airworthy, it must also meet certain safety related performance requirements. Methods for computing the performance capabilities of airplanes are discussed in Chapter 5.

Airworthiness regulations have a major impact on the design and development of airplanes. Therefore, the most important aspects of civil and military stability, control and performance airworthiness requirements (regulations) are presented in Appendices A and B.

Part VII is not intended as a textbook on airplane performance, stability and control. It is designed as a reference manual for students to ascertain that their designs will meet mission performance as well as stability and control requirements. For a detailed presentation of airplane performance, stability and control theory and methods References 11 and 12 are recommended.

Frequently, the mission specification of an airplane will identify which regulations form the certification base for an airplane. If that is not the case it is recommended that Table 1.2 be used to identify the type of airplane which is being designed. With this information, Table 1.3 can then be used to identify which regulations apply to the design.

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* Stability can be satisfied in an open or closed loop manner. If stability is satisfied 'open loop' it is referred to as 'inherent stability'. If stability is satisfied 'closed loop' it is referred to as 'de facto stability'.

A most important aspect of airplane design is to ensure that the airplane has not only satisfactory performance characteristics but also acceptable flying qualities following one or more failures of flight crucial components. To analyze the effect of failures of flight crucial components, the regulations must be interpreted in terms of the so-called airworthiness code as well as in terms of minimum acceptable levels of flying qualities. Appendix C contains a brief outline of the methodology used in determining the required levels of flying qualities in case of flight crucial failures.

TABLE 1.2 AIRPLANE TYPES

1. Homebuilt Propeller Driven Airplanes
2. Single Engine Propeller Driven Airplanes
3. Twin Engine Propeller Driven Airplanes
4. Agricultural Airplanes
5. Business Jets
6. Regional Turbopropeller Driven Airplanes
7. Transport Jets
8. Military Trainers
9. Fighters
10. Military Patrol, Bomb and Transport Airplanes
11. Flying Boats, Amphibious and Float Airplanes
12. Supersonic Cruise Airplanes

TABLE 1.3 RELATION BETWEEN AIRPLANE TYPE AND APPLICABLE AIRWORTHINESS REQUIREMENTS (REGULATIONS)

Airplane Type (See Table 1.2)	Passenger Limit	Weight Limit	Regulations
1	none	none	Experimental: FAR 21
2,3,4,5,11,12	<9	12,500	Normal Category: FAR 23, Appendix A
3,6,7,12	<19	<19,000	Commuter Category: FAR 23, Appendix A, see page 207
5,6,7,11,12	>19	none	FAR 25: Appendix A
8,9,10	none	none	Military: Appendix B

2. CONTROLLABILITY, MANEUVERABILITY AND TRIM

The purpose of this chapter is to present Class II methods for analyzing the controllability, maneuverability and trimmability of a newly designed airplane. These methods are compatible with Step 24 in Preliminary Design Sequence II as outlined in Chapter 2 of Part II.

The objectives of the methods in this chapter are to assure that:

1. the airplane has sufficient control power to maintain steady state, straight line flight
2. the airplane can be safely maneuvered from one steady state flight condition to another
3. cockpit control force levels are acceptable under all expected conditions, including those caused by configuration changes
4. the airplane can be trimmed in certain flight conditions

The qualitative statements 1-4 are only one component of what constitutes good flying qualities (handling qualities) of airplanes. These <u>qualitative</u> statements are translated in terms of <u>quantitative</u> requirements in the airworthiness requirements of References 8 and 9. These airworthiness requirements will be referred to as the REGULATIONS.

Which regulations apply to a given new airplane design depends on the projected use of that airplane. Certification according to more than one regulation is possible at the option of the designer. Broadly speaking, the regulations are applied as indicated in Table 1.3 of Chapter 1.

The material in this chapter is organized as follows:

2.1 Longitudinal controllability and trim
2.2 Directional and lateral controllability and trim
2.3 Minimum control speed
2.4 Maneuvering flight (acrobatic maneuvers)
2.5 Control during the takeoff groundrun
2.6 Control during the landing groundrun
2.7 Roll performance
2.8 High speed characteristics
2.9 Aeroelastic characteristics

It is essential to gain insight into how the various regulatory requirements for control, maneuvering and trim fit into the airplane design process. For that reason, the material in most sections is presented in the following general sequence:

1. identification of the applicable airworthiness regulation(s)

2. relationship to the stability and control checks performed as part of the preliminary layout design steps described in Chapter 2 of Part II: P.D. Sequences I and II (mostly Steps 11 and 24)

3. presentation of a model for analyzing controllability, trimmability and/or maneuverability

4. a step-by-step procedure for determining whether or not the airplane meets the control, trim and/or maneuvering requirements of the regulations

In this chapter, frequent use is made of the following terms:

* control surface
* trim surface
*trim
*trim diagram

Definitions of these terms are as follows:

<u>Definition:</u> A control surface is a surface used by the pilot to attain airplane moment equilibrium and/or to maneuver the airplane from one flight condition to another flight condition.

<u>Definition:</u> A trim surface is a surface used by the pilot to attain zero required cockpit control force while retaining moment equilibrium.

<u>Definition:</u> When an airplane is in moment equilibrium while the cockpit control force is zero, the airplane is trimmed.

<u>Definition:</u> A trim diagram is used to determine the conditions for which moment equilibrium (not necessarily with zero cockpit control forces) can be attained.

For detailed discussion(s) of airplane stability and control characteristics Reference 12 should be consulted.

2.1 LONGITUDINAL CONTROLLABILITY AND TRIM

2.1.1 Applicable Regulations

Civil: FAR 23.143, 23.145, 23.161, FAR 25.143, FAR 25.145, 25.161, see: Appendix A.

Military: Mil-F-8785C, Par.3.2, see Appendix B.

2.1.2 Relationship to Preliminary Design

See Part II, Chapter 2 (in particular Step 24), Chapter 8 and Chapter 11.

2.1.3 Mathematical Model for Analyzing Longitudinal Controllability and Trim

The regulations essentially require that any airplane, when trimmed in a given flight condition at a given airspeed:

A. can be maneuvered safely and easily to another speed

B. can be controlled safely and easily when a configuration change is made

The words 'safely' and 'easily' as used here are defined in the regulations in terms of control forces which may not be exceeded and/or in terms of control power levels which must be available.

The words 'configuration change' are meant to imply any of the following events:

1. Change in center of gravity location
2. Change in thrust (or power) setting (voluntary or failure)
3. Hard-over failure of any control or trim surface
4. Change in landing gear position (up or down)
5. Change in flap position (up, take-off, approach or landing)
6. Change in speed brake position (retracted or deployed)
7. Store release (i.e. bomb or payload drop): symmetrical and asymmetrical
8. Weapons firing (i.e. moment due to recoil force)
9. Combat damage

Whether or not these events must be assumed to occur

simultaneously is stated in the applicable regulations!

For purposes of Class II analyses, longitudinal controllability is considered satisfied if:

1. Sufficient control power is available to cope with all required configuration changes

2. Sufficient control power is available to allow the airplane to be maneuvered from one flight speed to another

3. Cockpit control forces required in the use of control power to satisfy controllability are within the limits prescribed in the regulations

Figure 2.1 depicts the forces and moments which act on an airplane in a symmetrical, wings level, steady state, straight line flight condition.

The following three equations of motion apply to this case:

<u>1. The Drag Equation:</u>

$$C_{L_1}\sin\gamma = C_{D_1} + T\cos(\phi_T + \alpha) \qquad (2.1)$$

<u>2. The (One-g) Lift Equation:</u>

$$C_{L_1}\cos\gamma = (C_{L_o} + C_{L_\alpha}\alpha + \sum_{i=n}^{i=1}(C_{L_{\delta_{ctrl_i}}}\delta_{ctrl_i}) + (T/\bar{q}S)\sin(\phi_T + \alpha) \qquad (2.2)$$

<u>3. The Pitching Moment Equation:</u>

$$0 = (C_{m_o} + C_{m_\alpha}\alpha + \sum_{i=n}^{i=1}(C_{m_{\delta_{ctrl_i}}}\delta_{ctrl_i}) + (Td_T/\bar{q}S\bar{c}) \qquad (2.3)$$

For any given flight condition (i.e. known $\bar{q}$) at any given thrust setting (i.e. known T) the usual variables in these equations are:

α, γ and one of the i control surface deflections, δ_{ctrl_i}.

Figure 2.1 Forces and Moments Acting on an Airplane in Symmetrical, Wings Level, Steady State, Straight Line Flight

Figure 2.2 Example of a Weight/C.G. Envelope

The numerical solutions for these variables must be 'tested' against acceptability criteria. These acceptability criteria normally consist of allowable bounds on the numerical values of α, γ and δ_{ctrl_i}. Acceptability criteria are presented in Subsection 2.1.4.

Methods for computing airplane drag coefficient, C_D and installed thrust, T in Eqn.(2.1) are given in Chapters 4 and 6 of Part VI. The reader is reminded of the fact that the drag coefficient in Eqn.(2.1) is the so-called 'trimmed' drag coefficient! That means: C_D includes the drag effect of all control surface deflections necessary to achieve moment equilibrium and trim.

Methods for computing the stability and control derivatives in Equations (2.2) and (2.3) are provided in Chapter 10 of Part VI.

The reason for using the 'Sum' terms in Equations (2.2) and (2.3) explicitly is to recognize the fact that many airplanes use two (or more) different types of longitudinal controls. Frequently, one of these controls is used for primary longitudinal control while the other is used for longitudinal trim.

A detailed justification for Equations (2.1) through (2.3) is given in Reference 12 (Chapter 5).

In preliminary design, the following assumptions are made to enable a rapid resolution of the controllability issue:

1. The drag equation (2.1) is inherently satisfied.

   For any given desired flight path angle, γ this means that the throttles are set to balance the drag equation.

2. The flight path angle, γ is small enough to use: $\cos\gamma = 1.0$.

The consequence of these assumptions is that only Equations (2.2) and (2.3) are needed to determine controllability! This can be done with the help of the airplane trim diagram. The trim diagram allows a graphical

solution of Equations (2.2) and (2.3) while accounting for all eight configuration changes mentioned before. A method for constructing airplane trim diagrams is presented in Part VI, Chapter 8.

Class II analysis of longitudinal trim is usually accomplished by writing an equation for the cockpit control force required to hold the aerodynamic controls in a position such that Eqns (2.2) and (2.3) are satisfied. Such an equation is called the stick- (or wheel-) force equation and it takes the following form:

$$F_s = G(HM) + F_{s_{artificial}} \tag{2.4}$$

where: G is the control system gearing ratio. For a discussion and definition of control system gearing ratios, see Part IV, Chapter 4 and Reference 12, Chapter 5. Typical numerical values for gearing ratios are given in Table 4.1 of Part VI.

HM is the control surface hingemoment, normally expressed in ftlbs and computed from:

$$HM = C_h \bar{q} (S\bar{c})_{control\ surface} \tag{2.5}$$

where: C_h is the control surface hingemoment coefficient. This hingemoment coefficient depends on the detail design of the control surfaces of the airplane. Section 10.4 of Part VI contains methods for estimating hingemoment coefficients and their derivatives.

$F_{s_{artificial}}$ is the incremental cockpit control force due to such items as: downsprings, bobweights and other 'feel'-systems. For a discussion of such devices, see Reference 12, Chapter 5 and Part IV, Chapter 4.

In these Class II methods the mathematical coupling which exists between Equations (2.1) through (2.4) is neglected. For an exact approach to the question of controllability and trim the reader is referred to the generalized matrix approach to trim as described in Section 5.6 of Reference 12.

2.1.4 Step-by-Step Procedure for Analyzing Longitudinal Controllability and Trim

The following step-by-step procedure is suggested to determine the longitudinal controllability and trim characteristics of an airplane during preliminary design:

Step 1: Determine which regulations apply to the design: see Table 1.3 and Subsection 2.1.1.

Step 2: The regulations define pairs of flight conditions so that the airplane must be controllable and maneuverable within each pair. Tabulate these flight conditions.

Step 3: The regulations define for which airplane weights the various controllability and trim requirements must be met. Tabulate these weights.

Note: It is important to also determine the airplane configuration and/or change in configuration associated with each pair of flight conditions: see items 1-9 on page 7. Tabulate the airplane configurations and/or the change in configuration as stipulated by the regulations.

Step 4: For the flight conditions and configurations defined in Steps 2 and 3 determine and tabulate: Mach number, M, dynamic pressure,

$\bar{q}$ and center of gravity location(s), $\bar{x}_{cg}$.

Note: the center of gravity location associated with any given configuration follows from the weight/c.g. envelope developed with the method of Chapter 10 in Part II. A typical example of such a weight/c.g. envelope is shown in Figure 2.2.

Step 5: For the conditions defined in Steps 2, 3 and 4 construct the airplane trim diagram. This can be done with the help of Section 8.3 in Part VI. Chapter 12 of Part VI should also be consulted. Figure 2.3 shows an example trim diagram for a conventional airplane.

Notes: 1.) Before the trim diagram can be constructed, a decision must have been made relative to the type and size of longitudi-

Figure 2.3 Example Trim Diagram for a Conventional (i.e. Tail Aft) Airplane

nal control surfaces to be employed!

2.) The method for constructing airplane trim diagrams in Part VI applies to conventional, pure canard as well as three-surface airplanes.

3.) The c.g. location associated with the vertical $C_m=0$ line in Figure 2.3 is normally selected to be the one for a typical 'design' mission of the airplane.

4.) The boundaries of the trim-triangle as shown in the example of Fig.2.3 are:

a) the $C_m=0$ lines for the most forward and for the most aft c.g. locations which in turn follow from the weight/c.g. envelope of Figure 2.2.

b) the α_{max} locus. This locus follows from either the airplane stall angle of attack or from the airplane angle of attack for which the stick pusher is set to trigger.

5.) Make sure that the effect of configuration changes 2-9 of page 7 are taken into account. This is particularly important in the case of thrust-induced-pitching-moment coefficient $Td_T/\bar{q}S\bar{c}$ and flaps.

Configuration events 2-9 (page 7) can be interpreted as a change in C_{m_o} and/or C_{L_o}.

<u>Step 6:</u> From the trim diagram of Step 5 determine whether or not equilibrium flight is possible within each pair of flight conditions.

This is accomplished by first computing the airplane lift coefficient for the initial and for the end flight condition associated with each pair of flight conditions from:

$$C_{L_{in}} = W/\bar{q}_{in}S \qquad (2.6)$$

and from:

$$C_{L_{end}} = W/\bar{q}_{end}S \qquad (2.7)$$

Second, in Figure 2.3 enter these lift coefficients into the trim diagram at the appropriate center of gravity line.

Step 7: Determine whether or not the results of Step 6 are acceptable. Criteria for acceptability can be:

1. the control surface deflections are within the control power capabilities designed into the airplane. Typical control deflection ranges may be found in various issues of Reference 13.

2. the angle of attack (or the lift coefficient) is below airplane stall.

3. the tail and/or the canard are not stalled inside the 'trim-triangle'.

NOTE: if one or more of these acceptability criteria are violated, the design must be adjusted until they are satisfied. Such design adjustments may consist of increases in control surface sizes, a change in airfoil(s) and/or rebalancing of the configuration.

In marginal cases a decision to scrap the design and start from scratch may be the only good choice!!

Step 8: Determine whether the flight control system of the airplane is of the reversible or irreversible type.

If reversible, proceed to Step 9. If irreversible, proceed to Step 12.

Note: airplanes with irreversible flight control systems usually require artificial control force feel systems. The design and analysis of such systems is beyond the scope of this text. Chapter 5 of Reference 12 may be consulted for further information on this subject.

Step 9: Using Eqns (2.4) and (2.5), compute the trim surface deflection required to set the cockpit control force equal to zero for the initial flight condition.

To do this requires that a preliminary decision as to the sizing and location of trim control surfaces has been made. If not, now is the time to make this decision.

An example of how the trim control surface deflection for $F_s=0$ may be computed will

be given for the case of an airplane with an elevator as the primary control surface and an elevator tab as the trim control surface. The reader will have to adjust this analysis to suit his own control system.

For an airplane with an elevator as the primary control surface and an elevator tab as the trim control surface the stick force equation (2.4) may be specialized to:

$$F_s = F_{s_{artificial}} + G\bar{q}\eta_h S_e \bar{c}_e [C_{h_o} + C_{h_\alpha}(\alpha(1 - d\epsilon/d\alpha) + i_h - \epsilon_o) + C_{h_{\delta_e}}\delta_e + C_{h_{\delta_t}}\delta_t] \qquad (2.8)$$

Setting F_s equal to zero the required tab deflection to trim the airplane is found from:

$$\delta_t = [(-F_{s_{artificial}}/(G\bar{q}_{in}\eta_h S_e \bar{c}_e)) - [C_{h_o} + C_{h_\alpha}(\alpha(1 - d\epsilon/d\alpha) + i_h - \epsilon_o) + C_{h_{\delta_e}}\delta_e]]/C_{h_{\delta_t}} \qquad (2.9)$$

where: α and δ_e both follow from the trim diagram, Figure 2.3 at the appropriate value of $C_{L_{in}}$ and at the appropriate combination of c.g. location and airplane configuration!

G, the gearing ratio depends on the detail design of the flight control system. Ty-

pical values are found in Table 4.1 of Part IV.

i_h is the stabilizer incidence setting, fixed or vaiable, depending on the design of the control system.

all hingemoment coefficients in Eqns (2.8) and (2.9) can be computed with the methods Section 10.4, Part VI.

the value of $F_{s_{artificial}}$ depends on the detail design of the flight control system. In airplanes without artificial feel gadgetry such as a bobweight, a downspring, a springtab or a feel system, this force should be set equal to zero.

<u>Note:</u> The tab deflection magnitude as determined from Eqn.(2.9) should not exceed 25 degrees!

If the airplane is equipped with a different type of flight control system than that assumed in this example, the reader must redevelop Equation (2.8) accordingly. A detailed treatment of the many possible types of flight control systems is beyond the scope of this text. The reader is referred to Part IV, Chapter 4 for many practical examples of different primary and trim control systems. A general analytical treatment of the cockpit control force behavior of various types of systems is given in Reference 12, Chapter 5.

<u>Step 10:</u> For the final (end) flight condition find the cockpit control force with Eqn.(2.8),

<u>PROVIDED:</u> $C_{L_{end}}$ and $\bar{q}_{end}$ are used instead of $C_{L_{in}}$ and $\bar{q}_{in}$, to find α and δ_e!

<u>NOTES:</u> 1.) The value for δ_t in this case is that obtained exactly from Eqn.(2.8) if the regulations do not allow retrimming of the airplane between the two flight conditions!

2.) The cockpit control force required to cope with one or more configuration changes (listed as 2-9

on page 7) may also be computed from Eqn.(2.8).

Step 11: Check the cockpit control force against the magnitude allowed by the regulation(s). If the cockpit control force exceeds the value allowed, a redesign of the flight control system may be required. Methods for 'tailoring control forces by tabs, balances and control surface design are discussed in Part III, Chapter 3 and in Reference 12, Chapter 5.

NOTE: The method presented here neglects coupling between Equations (2.1) through (2.4). For purposes of preliminary design this method is usually adequate. For a more complete approach to the solution of airplane trim problems the reader should consult Section 5.5 of Ref.12.

Step 12: Document the results obtained, including any design changes made as a result of not meeting (or exceeding!) any of the airworthiness regulations.

2.2 DIRECTIONAL AND LATERAL CONTROLLABILITY AND TRIM

2.2.1 Applicable Regulations

Civil: FAR 23.143, 23.147, 23.161, FAR 25.143, FAR 25.147, 25.161, see: Appendix A.

Military: Mil-F-8785C, Par.3.3, see Appendix B.

2.2.2 Relationship to Preliminary Design

See Part II, Chapter 2 (in particular Step 24), Chapter 8 and Chapter 11.

2.2.3 Mathematical Model for Analyzing Directional and Lateral Controllability and Trim

The regulations essentially require that any airplane, when trimmed in a given flight condition at a given airspeed:

A. can make turns at a specified bank angle into and away from one or more inoperative engines

B. can make sudden changes in heading while keeping the wings approximately level

The magnitudes of required bank angles and changes in heading angle are specified differently in each regulation, depending on the type of airplane.

In addition, the regulations specify the airplane configuration for which directional and lateral controllability must be satisfied. Configuration definitions usually include such items as:

1. Center of gravity location
2. Thrust or power setting (voluntary or failure)
3. Hard-over failure of any control or trim surface
4. Landing gear position (up or down)
5. Flap position (up, take-off, approach or landing)
6. Speed brake position (retracted or deployed)
7. Weapons loading (mostly asymmetrical loadings)
8. Weapons firing (mostly asymmetrical firing)
9. Combat damage

The regulations determine the various combinations of these configuration items which must be considered.

For purposes of Class II analyses, directional and lateral controllability is considered satisfied if:

1. Sufficient control power is present to perform the required turns and heading changes in all required configurations

2. Cockpit control forces required in the use of control power to satisfy controllability are within the limits prescribed in the regulations

The making of turns and changes of heading is assumed to place at such a slow rate that controllability can be verified by examining:

a. the numerical values of the lateral-directional motion variables: β, δ_a and δ_r

and:

b. the numerical values of the lateral-directional cockpit control forces

for 'before' and 'after' flight conditions (initial and end).

The following three equations of motion apply to this case:

$$-C_L \sin\phi \cos\gamma = (F_{T_Y})/\bar{q}S + C_{Y_\beta}\beta + C_{Y_{\delta_a}}\delta_a + C_{Y_{\delta_r}}\delta_r \qquad (2.10)$$

$$0 = (L_T)/\bar{q}Sb + C_{l_\beta}\beta + C_{l_{\delta_a}}\delta_a + C_{l_{\delta_r}}\delta_r \qquad (2.11)$$

$$0 = (N_T + N_D)/\bar{q}Sb + C_{n_\beta}\beta + C_{n_{\delta_a}}\delta_a + C_{n_{\delta_r}}\delta_r \qquad (2.12)$$

The numerical solutions to these equations must be 'tested' against acceptability criteria. These acceptability criteria normally consist of allowable bounds on the numerical values of the motion variables in these equations. These acceptability criteria are given in Subsection 2.2.4.

For any given flight condition (i.e. known $\bar{q}$) at any given thrust setting (i.e. known F_{Y_T}, L_T, N_T and N_D) the usual variables in these equations are:

β, δ_a and δ_r, which assumes that the bank angle, ϕ is determined by the 'initial' and 'end' conditions of the turns and/or the associated heading changes if the latter are to be made within a specified time.

The numerical solutions to these equations must be 'tested' against acceptability criteria. These acceptability criteria normally consist of allowable bounds on the numerical values of β, δ_a and δ_r. Acceptability criteria are presented in Subsection 2.2.4.

Methods for computing the stability and control derivatives which appear in Equations (2.10) - (2.12) are presented in Chapter 10 of Part VI.

Methods for computing installed thrust (from which F_{y_T}, L_T and N_T may be found) are presented in Chapter 6 of Part VI.

The yawing moment due to drag, N_D is found from the drag due to windmilling powerplants. Drag coefficients for windmilling jet engines and propellers may be computed with the method of Chapter 4, Part VI.

The lateral-directional cockpit control forces may be found from the following equations:

$$F_a = G_a \bar{q} S_a \bar{c}_a (C_{h_{\delta_a}} \delta_a + C_{h_{\delta_{a_t}}} \delta_{a_t}) + K_a \delta_r \tag{2.13}$$

$$F_r = G_r \bar{q} S_r \bar{c}_r (C_{h_{\delta_r}} \delta_r + C_{h_{\delta_{r_t}}} \delta_{r_t} + C_{h_{r_\beta}} \beta) + K_r \delta_a \tag{2.14}$$

The hingemoment derivatives in Equations (2.13) and (2.14) may be found with the methods of Ch.10, Part VI.

The reader should assume that the β-hingemoment derivative in Eqn.(2.14) is similar to the α-hingemoment derivatives of Chapter 10, Part VI. Likewise, the δ_a- and δ_r-hingemoment derivatives in Eqns (2.13) and (2.14) are similar to the δ-hingemoment derivatives of Chapter 10, Part VI.

Typical values for the gearing ratios, G_a and G_r may be found in Table 4.1 of Part IV.

The rudder-aileron interconnect terms, K_a and K_r are normally present only in airplanes with unsatisfactory control behavior. Ideally these terms should be zero.

2.2.4 Step-by-Step Procedure for Analyzing Directional and Lateral Controllability and Trim

Directional and lateral controllability can be verified with the following step-by-step procedure:

Step 1: Determine which regulations apply to the design: see Table 1.3 and Subsection 2.2.1.

Step 2: The regulations define flight conditions for which it must be possible to carry out turning and sudden heading change maneuvers. Identify these flight conditions.

When changes of bank angle are involved from positive to negative (right to left) values, identify one as the 'initial' and the other as the 'end' flight condition.

Note: It is important to also determine the airplane configuration and/or change in configuration associated with each flight condition: see items 1-9 on page 19.

Tabulate the flight condition and airplane configuration data as stipulated by the regulations.

Step 3: For the flight conditions defined in Step 2, determine: Mach number, M, dynamic pressure, $\bar{q}$ and center of gravity location, $\bar{x}_{cg}$. See the note under Step 4 in Subsection 2.1.4.

Step 4: For the conditions defined in Steps 2 and 3 determine the stability, control and other airplane characteristics required to solve the lateral-directional equations of motion: (2.10) - (2.12).

Notes: 1) methods for calculating stability and control derivatives are given in Ch.10 of Part VI. Methods for determining installed thrust (or power) are given in Ch.6 of Part VI.

2) which engines are to be considered inoperative is defined in the regulations. The magnitude and sign of F_{y_T}, L_T, N_T and N_D

depend on which engines are inoperative!!

Step 5: From solutions to Eqns (2.10) - (2.12) as defined by Eqns (2.15) - (2.17) in Table 2.1 determine whether or not turning maneuvers can be made which satisfy the regulations.

Note: The regulations may be considered to be satisfied if the magnitudes of the following variables are within acceptable bounds:

1. lateral control surface deflection(s), δ_a

2. directional control surface deflection(s), δ_r

3. sideslip angle, β

For a control surface deflection to be 'acceptable' means that it must be within the stall limits of that surface. Typical acceptability criteria are:

Lateral: $|\delta_a| < 25$ deg.

Directional: $|\delta_r| < 25$ deg. for single hinge rudder
< 35 deg. for double hinge rudder

For a sideslip angle to be 'acceptable' it must be sufficiently small so that the airplane drag is not significantly increased and so that the directional stability of the airplane has not significantly deteriorated due to vertical tail stall. Typical acceptability criteria are:

Sideslip angle for drag: $|\beta| < 5$ deg.

Sideslip angle for directional stability: $|\beta| < 12$ deg.

NOTE: If one or more of these acceptability criteria are violated, the design must be adjusted until they are satisfied. Such design adjustments may consist of increases in control surface sizes, addition of other types of control surfaces and/or a change (size and/or location) of the vertical tail.

In marginal cases a decision to scrap the design and start from scratch may be the only good choice!

Step 6: Determine whether the flight control system of the airplane is of the reversible or irreversible type.

Table 2.1 Solutions for Eqns.(2.10) - (2.12) of Page 20

$\beta = N_1/\Delta$ (2.15) $\qquad \delta_a = N_2/\Delta$ (2.16) $\qquad \delta_r = N_3/\Delta$ (2.17)

with:

$$N_1 = \begin{vmatrix} -(C_{L_1}\sin\phi\cos\gamma + F_{T_{Y_1}}/\bar{q}_1 S) & C_{Y_{\delta_a}} & C_{Y_{\delta_r}} \\ -L_{T_1}/\bar{q}_1 Sb & C_{l_{\delta_a}} & C_{l_{\delta_r}} \\ (-N_{T_1} - N_{D_1})/\bar{q}_1 Sb & C_{n_{\delta_a}} & C_{n_{\delta_r}} \end{vmatrix} \qquad (2.18)$$

$$N_2 = \begin{vmatrix} C_{Y_\beta} & -(C_{L_1}\sin\phi\cos\gamma + F_{T_{Y_1}}/\bar{q}_1 S) & C_{Y_{\delta_r}} \\ C_{l_\beta} & -L_{T_1}/\bar{q}_1 Sb & C_{l_{\delta_r}} \\ C_{n_\beta} & -(N_{T_1} + N_{D_1})/\bar{q}_1 Sb & C_{n_{\delta_r}} \end{vmatrix} \qquad (2.19)$$

$$N_3 = \begin{vmatrix} C_{Y_\beta} & C_{Y_{\delta_a}} & -(C_{L_1}\sin\phi\cos\gamma + F_{T_{Y_1}}/\bar{q}_1 S) \\ C_{l_\beta} & C_{l_{\delta_a}} & -L_{T_1}/\bar{q}_1 Sb \\ C_{n_\beta} & C_{n_{\delta_a}} & -(N_{T_1} + N_{D_1})/\bar{q}_1 Sb \end{vmatrix} \qquad (2.20)$$

$$\Delta = \begin{vmatrix} C_{Y_\beta} & C_{Y_{\delta_a}} & C_{Y_{\delta_r}} \\ C_{l_\beta} & C_{l_{\delta_a}} & C_{l_{\delta_r}} \\ C_{n_\beta} & C_{n_{\delta_a}} & C_{n_{\delta_r}} \end{vmatrix} \qquad (2.21)$$

NOTE: The subscript 1 indicates steady state flight!

If reversible, proceed to Step 7. If irreversible, proceed to Step 10.

Note: airplanes with irreversible flight control systems usually require artificial control force feel systems. The design and analysis of such systems is beyond the scope of this text. Chapter 5 of Reference 12 may be consulted for further information on this subject.

Step 7: Using Eqns (2.13) and (2.14), compute the trim surface deflections, δ_{a_t} and δ_{r_t} required to set both cockpit control forces, F_a and F_r equal to zero for the 'initial' flight condition.

Acceptable values for the trim-tab deflections are:

$$|\delta_{(a \text{ or } r)_t}| < 20 \text{ degrees}$$

Step 8: For the 'end' flight condition(s), use Eqns (2.13) and (2.14) to compute the cockpit control forces, F_a and F_r.

Note: be sure to use the trim surface values obtained from Step 7!

Step 9: Check the cockpit control force values against the magnitudes allowed by the regulation.

If any cockpit control force exceeds the value allowed, a redesign of the flight control system may be required. Methods for tailoring control forces by tabs, balances and control surface design are discussed in Part III, Chapter 3 and in Reference 12, Chapter 5.

Step 10: Document the results obtained, including any design changes made as a result of not meeting (or exceeding!) any of the airworthiness regulations.

NOTE: The method presented here neglects coupling between Equations (2.10) through (2.14). A generalized approach to the solution of airplane trim problems is presented in Chapter 5 of Reference 12.

2.3 MINIMUM CONTROL SPEED

2.3.1 Applicable Regulations

Civil: FAR 23.149, FAR 25.149, see Appendix A.

Military: Mil-F-8785C, Par. 3.3.9.2, see Appendix B.

2.3.2 Relationship to Preliminary Design

See Part II, Chapter 2 (in particular Step 24), Chapter 8 and Chapter 11.

2.3.3 Mathematical Model for Analyzing Minimum Control Speed

The regulations essentially require that any airplane, when experiencing failure of its most critical engine:

1. can be brought under control in such a manner that straight line flight is possible with zero sideslip or with a bank angle of no more than five degrees toward the operating engine

2. does not require more than:

   civil airplanes: 150 lbs of rudder pedal force

   military airplanes: a) 180 lbs of rudder pedal force
   b) the roll control force specified in Table X of Par.3.3.4.3, Appendix B

3. can do so at a speed called the minimum control speed, V_{mc} which must satisfy the following conditions:

   For take-off with one engine inoperative

   civil airplanes: $V_{mc} < 1.2V_s$

   military airplanes: $V_{mc} < V_{min_{TO}}$, where in

   preliminary design: $V_{min_{TO}} = 1.1V_{s_{TO}}$

   For landing with one or two engines inoperative

   civil airplanes: $V_{mc} > V_A$

military airplanes: $V_{mc} > 1.4V_{min}$, where V_{min} is defined in Section 6.2 of Appendix B

In addition, the regulations specify the airplane configuration for which the minimum control speed capability must be satisfied. As a general rule, configuration changes and/or power (thrust) changes are not allowed.

For purposes of Class II analyses, controllability is considered satisfied if:

1. Sufficient control power is present to cope with the stated requirements

2. Cockpit control forces required in the use of control power to satisfy controllability are within the limits prescribed in the regulations

Equations (2.15) through (2.17), see Table 2.1, are normally used to determine the magnitude of sideslip angle, rudder angle and aileron angle needed to keep the airplane in straight line flight and with a favorable bank angle of 5 degrees.

Equations (2.13) and (2.14) are used next to determine the cockpit control forces needed to keep the rudder and aileron deflections at the values required by Equations (2.15) through (2.17).

2.3.4 Step-by-Step Procedure for Determining the Minimum Control Speed

In preliminary design the minimum control speed capability of an airplane can be verified with the following step-by-step procedure:

Step 1: Determine which regulations apply to the design: see Table 1.3 and Subsection 2.3.1.

Step 2: The regulations define flight conditions and airplane configurations for which the regulations apply. Identify these flight conditions and configurations and determine the corresponding airplane weight, W.

Step 3: For the flight conditions and configurations of Step 2, determine: Mach Number, M, dynamic pressure, $\bar{q}$, lift coefficient, C_L

and center of gravity location, $\bar{x}_{cg}$.

Step 4: For the conditions defined in Steps 2 and 3, determine the required stability and control derivatives as well as the asymmetric engine side force, rolling moment and yawing moment (including any drag induced effects): see Equations (2.10) - (2.12).

Step 5: Using Equations (2.15) through (2.17), find the values of sideslip angle, aileron angle and rudder angle required at V_{mc}. The regulations allow the use of a bank angle of no more than 5 degrees toward the operating engine(s).

Step 6: Determine whether or not the results of Step 5 are acceptable.

Criteria for acceptability can be:

1. The sideslip angle, β must not be so large that the added drag due to sideslip would cause the airplane to violate the minimum climb requirements of Section 5.3

2. The aileron and/or rudder deflection angles are less than those values for which wing and/or vertical tail stall might result. Typical maximum acceptable angles are:

   Ailerons: < 25 degrees or whatever corresponds to no more than 75 percent of available lateral control power in the flight condition and airplane configuration being considered

   Rudder: < 25 degrees for single hinge-line rudders
   < 35 degrees for double hinge-line rudders

If the airplane violates any of these acceptability criteria, the design must be adjusted until the regulatory requirements are satisfied. This may involve changing control surface sizes, changing vertical tail size and location and/or re-arranging the most critical engines.

In marginal cases the decision to scrap the design and start from scratch may be the appropriate choice!!

If the airplane is equipped with a reversible flight control system in the roll and yaw axes proceed to Step 7. If not, proceed to Step 10.

Step 7: Determine the hingemoment coefficients and derivatives contained in Equations (2.13) and (2.14).

Step 8: Calculate the lateral and directional cockpit control forces needed to hold the aileron and rudder surfaces at deflection values corresponding to those obtained in Step 5. Use Equations (2.13) and (2.14).

Note: the values for the trim-tab deflections must be those corresponding to the trimmed flight condition prior to engine failure.

Step 9: Determine whether or not the roll control and rudder control forces of Step 8 are acceptable. The rudder control forces are normally the most critical ones!

Criteria for acceptability are given in the applicable regulations. A summary of the allowable rudder pedal forces is given under Subsection 2.3.3.

Step 10: Document the results obtained, including any design changes made as a result of not meeting (or exceeding!) any of the airworthiness regulations.

NOTE: The method presented here neglects coupling between Equations (2.10) through (2.14). A generalized approach to the solution of the lateral-directional trim problem is given in Chapter 5 of Reference 12.

2.4 MANEUVERING FLIGHT

2.4.1 Applicable Regulations

Civil: FAR 23.151 (acrobatic and utility category only), FAR 23.155, see Appendix A.
Note: there are no sustained maneuvering requirements in FAR 25!

Military: Mil-F-8785C, Par. 3.2.2.2, see Appendix B.

2.4.2 Relationship to Preliminary Design

Maneuvering flight requirements are accounted for in the initial sizing process to assure that:

1. The powerplant installation can overcome the extra drag incurred during maneuvers: see Part I, Chapter 3.

2. The maximum trimmed lift capability of the airplane is consistent with the maneuvering demands. See Part I, Chapter 3 and Part II, Chapter 7.

Note: these maneuvering requirements are of great importance to the design of fighters and acrobatic and utility category airplanes.

2.4.3 Mathematical Model for Analyzing Maneuvering Flight

The regulations essentially require that:

1. the airplane must be able to attain certain specified load factors: instantaneous or sustained.

2. the longitudinal cockpit control force per unit load factor increase (stick-force per 'g') shall be bounded by maximum as well as by minimum allowable values. Reversal in the stick-force per 'g' gradient is not allowed.

To satisfy the first item requires that the airplane be capable of achieving a value of maximum lift coefficient which is consistent with the required load factor. There are two types of required load factors:

A) Instantaneous load factors

and

B) Sustained load factors.

A) Instantaneous load factors

Three conditions must be satisfied to achieve a specified instantaneous load factor, n_{reqd}:

A1) The available lift condition,
A2) The available control power condition

and

A3) The cockpit control force condition.

A1) The available lift condition

The available lift condition is:

$$n_{reqd} < C_{L_{max}} \bar{q}S/W \qquad (2.22)$$

where: n_{reqd} is the load factor which must be attained according to the mission specification and/ or according to the applicable regulation.

$C_{L_{max}}$ must be interpreted as the maximum trimmed lift coefficient of which the airplane is capable in the given flight condition. The magnitude of $C_{L_{max}}$ depends on the wing design and on the type and size of flaps used. on the type of longitudinal control used and on the center of gravity location. Methods to estimate maximum trimmed lift coefficients are given in Section 8.3 of Part VI.

A2) The available control power condition

The available control power condition is:

$$\delta_{e_{reqd}} = \delta_{e_{trim}} + (\partial\delta_e/\partial n)n_{reqd} \qquad (2.23)$$

where: $\delta_{e_{trim}}$ is the elevator deflection required to trim the airplane in the steady state flight condition before the maneuver was initiated. It is computed with the procedure of sub-section 2.1.4.

$\partial\delta_e/\partial n$ is the elevator versus load-factor gradient. It may be computed from:

$$\partial\delta_e/\partial n = \qquad (2.24)$$

$$\{-C_{L_\alpha} C_{m_q} (\bar{c}g/2U_1^2)\bar{n} - C_{m_\alpha} C_{L_1}\}/(C_{L_\alpha} C_{m_{\delta_e}} - C_{L_{\delta_e}} C_{m_\alpha})$$

with: $\bar{n} = n$ for symmetrical pull-up maneuvers

$\bar{n} = (1 + 1/n^2)$ for turning maneuvers

The stability and control derivatives in Eqn.(2.24) may be computed with the methods of Chapter 10, Part VI.

n_{reqd} is defined under Eqn.(2.22)

<u>Note:</u> the condition: $\delta_{e_{reqd}} < \delta_{e_{available}}$ must always be satisfied. where: $\delta_{e_{available}}$ is that value of elevator deflection which is geometrically available: also called the maximum available elevator throw.

<u>A3) The cockpit control force condition</u>

This condition demands that the level of required cockpit control force needed to perform an instantaneous maneuver must be less than values specified in the regulations:

$$F_s = F_{s_1} + (\partial F_s/\partial n)n_{reqd} \qquad (2.25)$$

where: F_{s_1} is the cockpit control force required to 'hold' the steady state. If the airplane is in complete trim in the steady state, $F_{s_1} = 0$.

n_{reqd} is defined under Eqn.(2.22)

The 'stick-force-per-g', $\partial F_s/\partial n$ is found from

an equation. the terms in which depend strongly on the type of control system employed. For an airplane which uses an elevator for primary control, and no bobweight, Ref.12 shows that the following equation determines the stick-force per 'g':

$$\partial F_{s_e}/\partial n = \eta_h \bar{q} S_e \bar{c}_e G_e * \qquad (2.26)$$

$$*\{C_{L_1} (C_{h_{\delta_e}}/C_{m_{\delta_e}})(SM_{free}) + (gl_h/U_1^2)(C_{h_\alpha} - C_{h_{\delta_e}}/\alpha_{\delta_e})\}$$

where: l_h is the distance of the horizontal tail aerody-

namic center to the airplane center of gravity

SM_{free} is the static margin stick free which may be determined from:

$$SM_{free} = \bar{x}_{ac_A} - \bar{x}_{cg} + \tag{2.27}$$

$$+ \{(C_{m_{\delta_e}} C_{h_\alpha})/(C_{L_{\alpha_A}} C_{h_{\delta_e}})\}(1 - d\epsilon/d\alpha)$$

where: all terms are defined in the list of symbols. They may be determined with the help of Part VI.

Note: For an airplane which uses another type of primary longitudinal controller (other than an elevator), the reader should re-develop the appropriate equations in this Section to reflect the actual control system used.

B) Sustained load factors

Four conditions must be satisfied to achieve a specified magnitude of sustained load factor:

B1) The available lift condition,
B2) The available control power condition,
B3) The cockpit control force condition
and
B4) The available thrust condition.

Conditions B1 through B3 are identical to conditions A1 through A3 which were discussed before.

B4) The available thrust condition

This condition is discussed in Section 5.7.

2.4.4 Step-by-step Procedure for Determining Maneuvering Flight Ability

The following step-by-step procedure is suggested to check maneuvering flight ability in preliminary design.

Step 1: Read the regulations and determine which regulations apply to the design.

Step 2: The regulations define flight conditions and airplane configurations for which the regulations apply. Identify these flight conditions and configurations and determine the corresponding airplane weight, W.

Step 3: For the flight conditions and configurations of Step 2, determine: Mach Number, M, dynamic pressure, $\bar{q}$, lift coefficient, C_L and center of gravity location, $\bar{x}_{cg}$.

Step 4: This step concerns the available lift condition as given by A1) or B1) in sub-section 2.4.3.

For the flight conditions and for the configurations of Steps 2 and 3 determine the maximum achievable load factor from:

$$n_{max} = C_{L_{max}} \bar{q} S/W \qquad (2.28)$$

The maximum lift coefficient, $C_{L_{max}}$ in Eqn.(2.28) is that 'trimmed' value which applies to the airplane configuration (flaps up or down) as well as to the flight condition (particularly the Mach number) being analyzed. Methods for determining trimmed maximum lift coefficients are presented in Section 8.3 of Part VI.

Compare n_{max} from Eqn.(2.28) with n_{reqd} from Eqn.(2.22) and determine whether or not the airplane can satisfy its maneuvering requirements from a maximum lift viewpoint.

Step 5: This step concerns the available control power condition, as given in A2) or B2) in sub-section 2.4.3.

For the conditions defined in Steps 2 and 3, determine the required stability, control and hingemoment derivatives as well as other terms identified in Eqns (2.23) and (2.24).

Part VI contains methods for estimating all required parameters.

Compute the amount of elevator required to attain the specified load factor from Eqns (2.23) and (2.24).

The amount of elevator deflection required to attain the specified load factor should not exceed the available elevator travel, nor should it exceed that amount which would cause the tail to stall.

Step 6: This step concerns the required cockpit control force level as given in A3) or B3) in sub-section 2.4.3.

For the conditions defined in Steps 2 and 3, determine the required stability, control, hingemoment derivatives as well as other terms identified in Eqns (2.25) and (2.26).

Compute the stick-force required to pull the specified load factor from Eqns (2.25) and (2.26). Compare the computed magnitude of stick-force with those specified in the regulations.

Step 7: This step concerns the available thrust condition of B4) in sub-section 2.4.3.

Using the methods of Section 5.7, determine whether sufficient thrust is available to maintain (sustain) the required load factor.

Step 8: Determine whether or not the results of Steps 4, 5, 6 and 7 are acceptable. If not, inspect the applicable equations for 'clues' as to what design changes may be needed to 'fix' the problem. It may very well be that major changes in the design of the wing and/or the flight control system are required!

If the available thrust condition of Step 7 is not met, an increase in thrust may be required. This could cause a major change in the design of the airplane!

Step 9: Document the results obtained, including any design changes made as a result of not meeting (or exceeding!) any of the mission specifications and/or regulations.

2.5 CONTROL DURING THE TAKEOFF GROUNDRUN

2.5.1 Applicable Regulations

Civil: FAR 23.143, 23.51, 23.231, 23.233 and 23.235, see Appendix A.

FAR 25.107, 25.143, 25.149, 25.231, 25.233, 25.235 and 25.237, see Appendix A.

Military: MIL-F-8785C, Section 3.3 (in particular 3.3.7 and 3.2.3.3), see Appendix B.

2.5.2 Relationship to Preliminary Design

See Chapter 11, Part II.

2.5.3 Mathematical Model for Analyzing Control During the Takeoff Groundrun

The regulations essentially require that any airplane shall be safely controllable and maneuverable during take-off ground operations.

The words 'safely' and 'easily' as used here imply the following:

1. Sufficient control power must exist to effect take-off rotation: this applies to airplanes with tricycle as well well as to airplanes with tail-dragger landing gears.

2. Sufficient control power must exist to lift the tail early in the takeoff run for airplanes with tailwheels. Following this, there must be sufficient control power to effect lift-off.

3. Directional control and lateral control must be sufficient to allow for straight line taxiing as well as for straight line takeoff runs in cross winds as specified in the regulations.

4. The control forces required to accomplish 1,2 and 3 shall be sufficiently small. Also, nu unusual pilot control technique shall be required.

The regulations also imply that these characteristics be satisfied under the most adverse conditions of:

1. center of gravity location
2. power setting (voluntary or failure)

3. flap position as required for takeoff
4. weapons or payload disposition as required by the airplane mission

Class II analysis of control during takeoff is normally accomplished by uncoupling the longitudinal control problem from the directional/lateral control problem:

2.5.3.1 Mathematical model for analyzing longitudinal control during the takeoff groundrun

2.5.3.2 Mathematical model for analyzing directional and lateral control during the takeoff groundrun

2.5.3.1 Mathematical model for analyzing longitudinal control during the takeoff groundrun

Figure 2.4 shows the forces and moments which act on an airplane during the take-off run. Note the difference between:

a) Tricycle Gear Configurations
b) Taildragger Gear Configurations
c) Tandem Gear Configurations

The following mathematical model is presented for airplanes with conventional (that is tail-aft) configurations and with a tricycle landing gear.

NOTA BENE: The reader should modify this mathematical model to reflect the actual airplane configuration and landing gear layout of his airplane.

The following example applies to conventional airplanes equipped with a tricycle landing gear. Figure 2.5 defines all forces, moments and moment arms.

NOTA BENE: All aerodynamic quantities in Figure 2.5 must be determined in the presence of ground effect! See Part VI for methods to do this.

The longitudinal equations of motion which govern the controllability of the airplane on the ground are:

$$T\cos(\alpha_g + \phi_T) - \mu_g(P_{mg} + P_{ng}) - D = (W/g)\dot{u} \qquad (2.29)$$

$$T\sin(\alpha_g + \phi_T) + L_{wf} + L_h + P_{mg} + P_{ng} - W = 0 \qquad (2.30)$$

2.4a Tricycle Landing Gear

2.4b Taildragger Landing Gear

2.4c Tandem Landing Gear

Figure 2.4 Longitudinal Forces and Moments Acting on an Airplane During the Takeoff Groundrun

Figure 2.5 Forces, Moments and Moment Arms for Longitudinal Control During the Takeoff Groundrun

$$M_{ac_{wf}} - T(z_{mg} - z_T) - W(x_{mg} - x_{cg}) + P_{ng}(x_{ng} + x_{cg}) +$$

$$+ L_{wf}(x_{mg} - x_{ac_{wf}}) + D(z_{mg} - z_D) + (W/g)\dot{u}z_{mg} +$$

$$- L_h(x_{ac_h} - x_{mg}) = I_{yy_{mg}}\ddot{\theta} \qquad (2.31)$$

where: T is the installed thrust during the take-off run. It varies with speed during the take-off run and is determined with the method of Part VI, Ch.6.

α_g is the angle of attack during the groundrun. This angle is usually very small except at the instant before liftoff.

ϕ_T is the thrust inclination angle

μ_g is the wheel-ground rolling friction coefficient. During take-offs use:

for concrete or asphalt: $\mu_g = 0.02$

for hard turf: $\mu_g = 0.04$

for short grass: $\mu_g = 0.05$

for long grass: $\mu_g = 0.10$

for soft ground: $\mu_g = 0.10$ to 0.30

Note: these values are recommended for use on civil airplanes. For military airplanes consult Appendix B.

P_{mg} and P_{ng} are the ground reaction forces. It is assumed that the nosewheel is no longer in touch with the ground (i.e. $P_{ng}=0$), at the instant that the rotation has been initiated.

D is the airplane drag in ground effect:

$$D = (C_D)\bar{q}S \qquad (2.32)$$

with: C_D the drag coefficient on the ground. This drag coefficient follows from

the airplane drag polar on the ground as discussed in Part VI, Chapter 4.

$\bar{q}$ is the dynamic pressure. It varies during the take-off run.

W is the weight. This is not necessarily the airplane maximum take-off weight. It is up to the designer to determine the most critical combination of weight and center of gravity location, x_{cg}. The weight versus c.g. diagram of Ch.10 of Part II should be used.

$\dot{u}$ is the forward acceleration along the runway.

L_{wf} is the wing-fuselage lift in ground effect:

$$L_{wf} = C_{L_{wf}} \bar{q} S \tag{2.33}$$

with: $C_{L_{wf}}$ the airplane lift coefficient during the ground roll, in ground effect. Its value will be constant before initiation of the rotation. The methods of Pt VI, Chapter 8 can be used to determine this lift coefficent.

<u>NOTE:</u> The relative location of L_{wf} to the main landing gear is important. If it too far forward, the airplane will have a tendency to rapidly autorotate after the pilot has initiated the rotation by pulling the cockpit control aft. This autorotation is caused by the rapid increase in L_{wf} as the angle of attack begins to increase. The designer should attempt to locate L_{wf} as close as possible to but forward of the main gear!

L_h is the tail lift during the ground roll:

$$L_h = C_{L_h} \bar{q}_h S_h \tag{2.34}$$

with: C_{L_h} the horizontal tail lift coefficient during the ground roll. It must be evaluated in ground effect. It will be constant (and usually negative due to the preset negative incidence angle on the stabilizer) <u>before initiation</u> of rotation.

<u>Upon initiation</u> of rotation the tail lift coefficient becomes large and negative. The negative tail lift <u>in ground effect</u> may be computed with the methods of Part VI, Chapter 8.

$\bar{q}_h$ is the dynamic pressure at the horizontal tail. It varies during the take-off roll and is related to $\bar{q}$ by: $\bar{q}_h = \eta_h \bar{q}$ (2.35)

The dynamic pressure ratio at the tail, η_h is found from Part VI, p.269.

$M_{ac_{wf}}$ is the wing-fuselage pitching moment about the wing-fuselage aerodynamic center <u>in ground effect</u> and with the flaps in the take-off position:

$$M_{ac_{wf}} = C_{m_{ac_{wf}}} \bar{q} S \bar{c} \qquad (2.36)$$

with: $C_{m_{ac_{wf}}}$ the wing-fuselage pitching moment coefficient about the wing-fuselage aerodynamic center and with the flaps in the take-off position. This coefficient may be determined with the methods of Part VI, Chapter 8. It must be computed <u>in ground effect.</u>

If the reader does not wish to employ the rather laborious methods of Part VI, the following approximation may be employed:

$$C_{m_{ac_{wf}}} = 1.1(b_f/b)c_{m_{ac_f}} \qquad (2.37)$$

where: the factor 1.1 accounts for ground effect

b_f/b is the flap-span to wing-span ratio

$c_{m_{ac_f}}$ is the airfoil pitching moment coefficient with the flap down (in this case in the take-off position). Methods for determining the effect of flaps on airfoil $c_{m_{ac}}$ are given in subsection 8.2.2.1 of Part VI.

z_{mg} is defined in Figure 2.5.

z_T is defined in Figure 2.5.

x_{mg} is defined in Figure 2.5.

x_{cg} is defined in Figure 2.5.

$x_{ac_{wf}}$ is defined in Figure 2.5.

z_D is defined in Figure 2.5. It is usually taken at the wing-fuselage aerodynamic center.

x_{ac_h} is defined in Figure 2.5.

$I_{yy_{mg}}$ is the pitching moment of inertia about the main gear contact point. It may be computed from:

$$I_{yy_{mg}} = I_{yy} + (W/g)[z_{mg}^2 + (x_{mg} - x_{cg})^2] \quad (2.38)$$

with: I_{yy} the airplane moment of inertia as determined from Part V, Chapter 3 or, more appropriately for Class II methods from Part V, Chapter 10

$\ddot{\theta}$ is the pitch angular acceleration at the instant of initiation of rotation. In preliminary design it is suggested to use:

For large transports: $\ddot{\theta}$ = 6 to 8 deg/sec^2

For small transports: $\ddot{\theta}$ = 8 to 10 deg/sec^2

For light airplanes and for fighters: $\ddot{\theta}$ = 10 to 12 deg/sec^2

Equations (2.29) through (2.31) are normally solved for the amount of horizontal tail area required to achieve take-off rotation at the rotation speed, V_{rot}:

$$S_h = \{-z_T T + z_D D + W(x_{mg} - x_{cg} + \mu_g z_{mg}) - L_{wf}(x_{mg} + - x_{ac_{wf}} + \mu_g z_{mg}) - C_{m_{ac_{wf_g}}} \bar{q}_{rot} S\bar{c} - I_{yy_{mg}} \ddot{\theta}\} / \{\bar{q}_{rot}(x_{ac_h} - x_{mg} + \mu_g z_{mg}) C_{L_{h_{max}}}\} \qquad (2.39)$$

The rotation speed is defined relative to the stall speed in the regulations: Appendix A or Appendix B.

Values for maximum lift coefficient for the horizontal tail depend on the tail design and on the ground effect (ground proximity) of the tail during the rotation process. If the reader does not wish to employ the rather laborious methods of Part VI to estimate $C_{L_{h_{max}}}$ it is suggested to use:

$C_{L_{h_{max}}} = -0.35(A_h)^{1/3}$ for fixed stabilizer with 30 percent chord elevator

$= -0.8$ for a controllable stabilizer with a fixed elevator at zero deflection

$= -1.2$ for a controllable stabilizer with a 30 percent chord elevator deflected for maximum down lift

<u>Note:</u> higher values for maximum downlift on the tail are possible with negative camber, negative slots and/or blowing over the lower surface.

2.5.3.2 Mathematical model for analyzing lateral-directional control during the take-off groundrun

The regulations essentially require that controllability during the take-off run is such as to ensure that:

1) Positive steering is possible
2) Wing-over is impossible
3) The airplane does not slide off the runway

Wet and/or slick runway surfaces coupled with strong crosswinds can cause anyone of these problems. The regulations specify the magnitude of crosswinds which each airplane category must be able to cope with.

Figure 2.6 shows the forces and moments which are important to these ground operation problems. To satisfy items 1), 2) and 3) it is necessary for the designer to show that the following three runway constraints are satisfied:

Runway Constraint 1:

The lateral ground friction force on the nosegear should not reverse sign. For positive sideslip due to crosswind (as shown in Figure 2.6), it is required that:

$$Y_{ng} > 0 \tag{2.40}$$

If condition (2.40) is violated, nosewheel steering becomes ineffective.

Runway Constraint 2:

The normal force (= vertical ground reaction) on either main gear should not become zero. For positive sideslip due to crosswind it is required that:

$$P_{mg_L} > 0 \text{ and } P_{mg_R} > 0, \tag{2.41}$$

where:

$$P_{mg_L} + P_{mg_R} = P_{mg} \tag{2.42}$$

If either of the conditions (2.41) is violated, the airplane will start to 'wing-over'.

Runway Constraint 3:

The sum of all lateral ground friction forces should

Figure 2.6 Forces and Moments which Act on an Airplane During the Groundrun

not reverse sign. For positive sideslip due to crosswind it is required that:

$$Y_{ng} + Y_{mg_L} + Y_{mg_R} > 0 \qquad (2.43)$$

If condition (2.43) is violated, the airplane will begin to slide off the runway.

<u>Note:</u> As soon as the take-off rotation process as described in sub-section 2.5.3.1 is completed, the runway constraints loose their validity: the airplane is flying.

The magnitudes of the lateral gear forces, Y_{ng} and $Y_{mg_{(L\ or\ R)}}$, depend on the values of the vertical gear reactions: P_{ng} and $P_{mg_{(L\ or\ R)}}$ and on the so-called gear cornering coefficient.

The vertical gear reactions, P_{ng} and P_{mg} (and the airplane forward acceleration, $\dot{u}$) may be computed from from Equations (2.29) through (2.31) by setting:

a) Thrust equal to takeoff thrust

b) L_h to a value corresponding to the primary longitudinal control deflection used during the take-off roll (prior to initiation of rotation!).

c) $\ddot{\theta} = 0$

The relationship between the lateral gear forces, Y_{ng} and $Y_{mg_{(L\ or\ R)}}$, the so-called cornering coefficient, c_β and the vertical gear reactions, P_{ng} and $P_{mg_{(L\ or\ R)}}$ are normally writtent as follows:

$$Y_{ng} = c_\beta P_{ng} \qquad (2.44)$$

$$Y_{mg_{(L\ or\ R)}} = c_\beta P_{mg_{(L\ or\ R)}} \qquad (2.45)$$

The cornering coefficient itself is a function of the gear slip angles, $\beta_{ng\ or\ mg}$ which are defined in

Figure 2.7. They can be expressed as:

for the main gear: $\beta_{mg} = \psi_A$ (2.46)

for the nose gear: $\beta_{ng} = \psi_A + \psi_{steer}$ (2.47)

where: ψ_A is the airplane crab angle.

ψ_{steer} is the nosegear steering angle.

<u>Note:</u> These relations assume a steerable nosegear and a non-steerable main gear!

In preliminary design it is conservative to write:

<u>For dry surfaces:</u>

for the main gear: $c_\beta = 0.025\psi_A$ (2.48)

for the nose gear: $c_\beta = 0.025(\psi_A + \psi_{steer})$ (2.49)

<u>For slick surfaces:</u>

for the main gear: $c_\beta = 0.005\psi_A$ (2.50)

for the nose gear: $c_\beta = 0.005(\psi_A + \psi_{steer})$ (2.51)

For purposes of preliminary design it is acceptable to assume that the <u>maximum possible</u> value for the cornering coefficients are:

$c_\beta = 0.70$ for dry surfaces

$= 0.15$ for slick surfaces

In checking the runway constraint conditions (2.40) and (2.43) these maximum allowable values should be kept in mind!

Figure 2.7 illustrates the relationships between the cornering coefficients and the gear slip angle.

To find the <u>actual values</u> of the lateral gear forces, the lateral-directional equations of motion are needed. The steady state lateral-directional equations of motion for a straight line track along the runway are:

$$[(C_{Y_\beta})(\beta_{cross} - \psi_A) + C_{Y_{\delta_a}}\delta_a + C_{Y_{\delta_r}}\delta_r]\bar{q}S + Y_{ng} +$$

$$+ Y_{mg_L} + Y_{mg_R} = 0 \qquad (2.52)$$

$$[(C_{l_\beta})(\beta_{cross} - \Psi_A) + C_{l_{\delta_a}}\delta_a + C_{l_{\delta_r}}\delta_r]\bar{q}Sb + P_{mg_L}y_{mg} +$$

$$- P_{mg_R}y_{mg} - Y_{ng}z_{ng} - (Y_{mg_L} + Y_{mg_R})z_{mg} = 0 \qquad (2.53)$$

$$[(C_{n_\beta})(\beta_{cross} - \Psi_A) + C_{n_{\delta_a}}\delta_a + C_{n_{\delta_r}}\delta_r]\bar{q}Sb +$$

$$+ Y_{ng}(x_{ng} - x_{cg}) - (Y_{mg_L} + Y_{mg_R})(x_{mg} - x_{cg}) = 0 \qquad (2.54)$$

The x- and z- distances in these equations are defined in Figure 2.6. They are consistent with Figure 2.5.

Figure 2.7 Gear Slip Angle, Gear Steering Angle, and Cornering Coefficients

Assumptions made in setting up these equations are:

1. The airplane does not bank (gear is rigid in the vertical direction).

2. All gears are rigid in torsion.

3. The tire induced torque on the gears can be neglected.

4. The engine induced torque in roll and in yaw can be neglected.

5. The dynamic pressure is a constant value between zero and that value corresponding to liftoff.

   The change in dynamic pressure during the takeoff groundrun is accounted for by solving Equations (2.52) - (2.54) for a range of values of dynamic pressure.

By using Equations (2.42), and (2.44) through (2.49) it is possible to cast Equations (2.52) - (2.54) in the following, more tractable form:

$$\{(C_{y_\beta})(\beta_{cross} - \psi_A) + C_{y_{\delta_A}}\delta_a + C_{y_{\delta_r}}\delta_r\}\bar{q}S +$$

$$+ 0.025(\psi_A + \psi_{steer})P_{ng} + 0.025\psi_A P_{mg} = 0 \qquad (2.55)$$

$$\{(C_{l_\beta})(\beta_{cross} - \psi_A) + C_{l_{\delta_a}}\delta_a + C_{l_{\delta_r}}\delta_r\}\bar{q}Sb +$$

$$+ 2P_{mg_L}y_{mg} - P_{mg}y_{mg} - 0.025(\psi_A + \psi_{steer})P_{ng}z_{ng} +$$

$$- 0.025\psi_A P_{mg}z_{mg} = 0 \qquad (2.56)$$

$$\{(C_{n_\beta})(\beta_{cross} - \psi_A) + C_{n_{\delta_a}}\delta_a + C_{n_{\delta_r}}\delta_r\}\bar{q}Sb +$$

$$+ 0.025(\psi_A + \psi_{steer})P_{ng}(x_{ng} - x_{cg}) +$$

$$- 0.025\psi_A P_{mg}(x_{mg} - x_{cg}) = 0 \qquad (2.57)$$

The following observations are in order:

A. The crosswind, V_{cross}, is specified by the mission specification and/or by the regulations. From the crosswind and the airplane speed along the runway, U_1:

$$\beta_{cross} = \arctan(V_{cross}/U_1) \qquad (2.58)$$

B. The variables in Equations (2.55) - (2.57) are:

1. The runway crab angle, ψ_A
2. The left vertical main gear reaction, P_{mg_L}
3. The lateral control deflection angle, δ_a
4. The rudder deflection angle, δ_r
5. The nosegear steering angle, ψ_{steer}

There are five variables and three equations. Thus, two variables must be specified. Usually the assumption is made that the pilot will select the nosegear steering angle, ψ_{steer} and the aileron deflection angle, δ_a to obtain the desired straight line track along the runway <u>without winging over.</u> The other variables are then completely determined by Eqns (2.55)-(2.57). Based on assumed values for ψ_{steer} and δ_a, and the solved variables ψ_A, P_{mg_L} and δ_r the designer must then make sure that the constraints 1 - 3 are not violated. Subsection 2.5.4 contains a procedure for these calculations.

2.5.4 Step-by-Step Procedure for Analyzing Control During the Take-off Groundrun

2.5.4.1 Step-by-step procedure for analyzing longitudinal control during the take-off groundrun

The following step-by-step procedure is suggested to verify the longitudinal controllability of an airplane during take-off:

<u>Step 1:</u> Read the regulations and determine which regulations apply to the design.

Step 2: The regulations define the speeds at which take-off rotation must be possible. Determine these speeds.

Step 3: The mission of the airplane defines the most severe field conditions in terms of altitude, temperature and surface conditions. This determines the atmospheric density to be used in the analysis. The field condition also deter mines the wheel-ground rolling friction coefficient, μ_g which should be used: see page 40 for guidelines.

Step 4: Determine the most critical weight versus c.g. locations for which take-off rotation must be possible.

Step 5: Determine all forces and coefficients needed to find the horizontal tail area required to rotate, from Eqn.(2.39). Make sure that ground effect is accounted for!

NOTE: If the airplane is not a conventional tricycle gear airplane it will be necessary to rewrite Equations (2.29) through (2.31)!

Step 6: Using Eqn.(2.39), find S_h as a function of speed, plot this relationship and mark the desired take-off rotation speed. Next, perform a sensitivity analysis to determine the sensitivity of the solution to such parameters as:

z_T, the vertical thrustline location.

$C_{m_{ac_{wf}}}$, the pitching moment coefficient of wing-fuselage with flaps in the take-off position. Determine the effect of flap type and flap size on the solution for S_h.

$C_{L_{h_{max}}}$, the maximum available lift coefficient of the horizontal tail.

Compare the value of S_h required for takeoff rotation with the value required from other considerations such as: control, trim and

stability). Decide on a course of action.

IMPORTANT NOTE: It turns out that the horizontal tail of many airplanes is determined by the take-off rotation requirement rather than by stability or by trim requirements. If that is the case a sensitivity analysis will allow for rapid decision making relative to changes in the design to bring this tail sizing requirement in better balance with the other tail sizing requirements.

Step 7: Document the results obtained, including any design changes made.

2.5.4.2 Step-by-step procedure for analyzing lateral-directional control during the take-off groundrun

Step 1: Read the regulations and determine which regulations apply to the design.

Step 2: The regulations define the crosswind magnitudes which the airplane must be able to cope with. Note these!

Step 3: Compute the stability and control derivatives identified in Eqns (2.55) - (2.57). The methods of Part VI can be used to do this.

Step 4: For a range of speeds between zero and V_{rot} determine the vertical gear reactions, P_{ng} and P_{mg} from Eqns (2.29) - (2.31) but using conditions a,b and c as stated on page 47.

Note: The nosegear reaction depends strongly on the longitudinal control deflections used during the take-off roll. The longitudinal trim control (usually the stabilizer incidence angle) must be set at an angle consistent with the requirement for take-off rotation: this usually means a negative angle!! The primary longitudinal control deflection may be assumed to be that giving the most favorable steering (usually full nose down).

Step 5: With the help of the specified crosswind determine the crosswind angle, β_{cross} from Equation (2.58). This must be done for a range of speeds consistent with Step 4.

Step 6: Select values for the nosegear steering angle, Ψ_{steer} and for the aileron angle, δ_a.

With the help of Eqns (2.55) - (2.57) determine the airplane crab angle, Ψ_A, the left vertical gear reaction, P_{mg_L} and the rudder angle, δ_r.

Step 7: With the results of Step 6, compute all gear reactions which appear in the runway constraint conditions 1, 2 and 3. See whether or not these constraints are satisfied. If not, different initial assumptions for Ψ_A and δ_a may be tried as long as these new values are feasible. It may be, that the airplane cannot cope with the required cross wind conditions. In that case, possible design solutions are:

1. A larger vertical tail: more directional stability and perhaps more rudder control power.

2. Larger values for y_{mg} (larger distance between the main gears. This can have major redesign implications!

3. Incorporate a crosswind main gear into the airplane. See Part IV, Chapter 2 for implications!

4. Sometimes a smaller crosswind capability has to be accepted despite the fact that this will hurt the operational flexibility of the airplane.

Step 8: Document the results obtained, including any design changes made.

2.6 CONTROL DURING THE LANDING GROUNDRUN

2.6.1 Applicable Regulations:

Civil: FAR 23.231, see Appendix A.

FAR 25.231, 233, 237, see Appendix A.

Military: MIL-F-8785C, Section 3.3 (inparticular 3.2.3.3), see Appendix B.

2.6.2 Relationship to Preliminary Design

See Chapter 11, Part II.

2.6.3 Mathematical Model for Analyzing Control During the Landing Groundrun

The mathematical model to be used in this case is essentially that of Section 2.5.3 with the following additions:

1. The thrust, T should be investigated for flight idle as well as for reversed thrust if the airplane is equipped with a thrust reverser system.

2. Application of brakes should now be considered. The rolling friction coefficient, μ_g encountered during braking depends on brake design. For typical values, consult Part IV, pages 60 and 61.

3. Application of wing mounted spoilers can reduce the wing-fuselage lift very considerably thus changing L_{wf} in Eqns. (2.30) and (2.31).

4. The airplane may touch down in a 'crabbed' position. The crab-angle, ψ_A depends on the value of crosswind and on the touchdown speed, V_L:

$$\psi_A = \arctan(V_{cross}/V_L) \qquad (2.59)$$

5. The longitudinal control deflections are those most favorable for positive nosegear steering.

2.6.4 Step-by-Step Procedure for Analyzing Control During the Landing Groundrun

Use the procedures of Section 2.5.4 with the exception, that airplane speed ranges from V_L down to zero.

2.7 ROLL PERFORMANCE

2.7.1 Applicable Regulations

Civil: FAR 23.157 and FAR 25.147, see Appendix A.

Military: MIL-F-8785C, Section 3.3, see Appendix B.

2.7.2 Relationship to Preliminary Design

Part II, Chapter 2 (see Step 24), and Chapter 6.

2.7.3 Mathematical Model for Analyzing Roll Performance

The regulations essentially require that any airplane, when trimmed in a given flight condition at a given airspeed and in a given configuration:

1. can be maneuvered safely and easily from one bank angle to another (rolled) within some specified time period (Except FAR 25.147)

2. has roll time constant values which are within certain limits (Military only)

The words 'safely' and 'easily' as used here are defined in the regulations in terms of lateral cockpit control forces which may not be exceeded while achieving the bank angles and roll rates which must be obtained within some specified amount of time. In addition, the regulations imply that reversals in bank angle and roll rate may not occur outside well defined boundaries.

The words 'airplane configuration' imply one or more of the following as defined in the regulations:

1. center of gravity location
2. power setting (voluntary or failure)
3. landing gear position (up or down)
4. flap position (up, take-off, approach or landing)
   Note: in some instances asymmetric flap deployment has to be considered.
5. asymmetric fuel distribution
6. asymmetric weapons or payload disposition

When in doubt about the specific meaning of civil roll performance requirements, the author suggests to use the corresponding military specification.

Class II analysis of roll performance is normally accomplished with solutions of the so-called single-de-

gree-of-freedom differential rolling equation:

$$\ddot{\phi} = L_p\dot{\phi} + L_{\delta_{cpt}}\delta_{cpt} \tag{2.60}$$

where: ϕ is the bank angle in rad

$\dot{\phi} = p$ is the roll rate in rad/sec

L_p is the dimensional roll damping derivative, which is defined as:

$$L_p = (C_{l_p}\bar{q}Sb^2)/(2I_{xx_s}U_1) \tag{2.61}$$

in which: C_{l_p} is the dimensionless roll damping derivative. Part VI, Chapter 10 gives a method for estimating C_{l_p}

S is the wing area in ft^2

b is the wing span in ft

U_1 is the steady state speed in fps

I_{xx_s} is the rolling moment of inertia in $slugft^2$ in the stability axis system. Methods for computing the moments of inertia in an arbitrary body axis system are provided in Chapters 9 and 10 of Part V. Transformation of airplane moments of inertia to the stability axes system can be accomplished with the help of Appendix D.

$L_{\delta_{cpt}}$ is the dimensional roll control power derivative. It is defined as:

$$L_{\delta_{cpt}} = (C_{l_{\delta_{cpt}}}\bar{q}Sb)/I_{xx_s} \tag{2.62}$$

$C_{l_{\delta_{cpt}}}$ is the lateral control power derivative. Read on to see how it may be computed.

δ_{cpt} is the lateral cockpit control deflection in degrees, in radians or in inches depending on the type of cockpit controller in use. The cockpit controller is normally geared to the

lateral control surfaces. Many airplanes employ more than one type of lateral control surface. If an airplane uses both ailerons and spoilers and if a linear gearing is assumed between the cockpit controller and the lateral control surface it is possible to write:

$$C_{l_{\delta_{cpt}}} \delta_{cpt_{max}} = C_{l_{\delta_a}} \delta_{a_{max}} + C_{l_{\delta_s}} \delta_{s_{max}} \qquad (2.63)$$

where: $C_{l_{\delta_a}}$ is the aileron control power derivative in 1/deg or in 1/rad. It may be computed with the method of Chapter 10 in Part VI.

$\delta_{a_{max}}$ is the maximum available aileron deflection in degrees or in radians

$C_{l_{\delta_s}}$ is the spoiler control power derivative in 1/deg or in 1/rad. It may be computed with the method of Chapter 10 in Part VI.

δ_s is the maximum available spoiler deflection in degrees or in radians.

$\delta_{cpt_{max}}$ represents the maximum available deflection of the lateral cockpit controller in inches or in radians. This quantity follows from the detail design of the cockpit as discussed in Chapter 2 of Part III. Maximum allowable deflections for sticks and wheels are defined in Chapter 2, Part III. Once $\delta_{cpt_{max}}$ has been determined, Eqn.(2.63) can be used to solve for $C_{l_{\delta_{cpt}}}$.

It must be possible for the pilot to move the lateral cockpit controller (stick, wheel or other) to a position consistent with the required roll performance. This means that the accompanying cockpit control force should not exceed the capabilities of the pilot. The magnitudes of allowable lateral control forces are defined in the regulations. The lateral cockpit control force may be estimated from:

$$F_{s_{lat}} = \text{Sum}\,(G_i HM_i) + F_{s_{lat_{artificial}}} \qquad (2.64)$$

where: G_i is the gearing ratio in rad/in or in rad/ft associated with the ith lateral control surface

HM_i is the hingemoment in ftlbs associated with the ith lateral control surface. Such hingemoments may be estimated from:

$$HM_i = C_{h_i} \bar{q} S_i \bar{c}_i \qquad (2.65)$$

where: C_{h_i} is the hingemoment coefficient associated with the ith lateral control surface. Methods for estimating hingemoment coefficients are given in Chapter 10 of Part VI.

S_i is the reference area of the ith lateral control surface

$\bar{c}_i$ is the m.g.c. of the ith lateral control surface

$F_{s_{lat_{artificial}}}$ is the lateral cockpit controller force produced by the lateral force-feel system. For a discussion of force-feel systems, see Ref.12, Chapter 5.

2.7.4 Step-by-Step Procedure for Analyzing Roll Performance

In preliminary design, the roll performance capabilities of an airplane can be verified with the following step-by-step procedure:

Step 1: Read the regulations and determine which regulations apply to the design.

Step 2: The regulations define flight conditions and airplane configurations for which certain roll performance standards are to be met. Identify these flight conditions.

Note: Also determine the airplane configu-

ration for which the regulations apply.

<u>Step 3:</u> For the flight conditions defined in Step 2, determine: Mach number, M, dynamic pressure, $\bar{q}$ and the rolling moment of inertia (in stability axes!).

<u>Step 4:</u> For the conditions defined in Steps 2 and 3 determine the roll damping derivative, C_{l_p}.

<u>NOTE:</u> for airplanes subject to aeroelastic deformations, read Section 2.9!

<u>Step 5:</u> For the conditions defined in Steps 2 and 3 determine the roll control power derivative due to lateral cockpit control, $C_{l_{\delta_{cpt}}}$.

This derivative depends on the type of roll control system used. For an airplane which uses ailerons and/or spoilers, Eqn.(2.63) may be used. If a differential stabilizer is used, the appropriate term must be added to Eqn.(2.63). Methods for estimating control power derivatives due to ailerons, spoilers and differential stabilizer are given in Chapter 10 of Part VI.

Acceptable values for control surface deflections depend on aerodynamic considerations. Suggested 'not-to-exceed' values for lateral control surface deflections are:

Ailerons: $\delta_{a_{max}} < 25$ degrees

Spoilers: $\delta_{s_{max}} < 60$ degrees

Differential stabilizer: $\delta_{i_h} < 15$ degrees

<u>NOTE:</u> for airplanes subject to aeroelastic deformations, read Section 2.9!

<u>Step 6:</u> Determine the roll angle performance capability of the airplane from:

$$\phi(t) = [(-L_{\delta_{cpt}}\delta_{cpt})t/L_p] + [(L_{\delta_{cpt}}\delta_{cpt})/((L_p)^2)](e^{L_p t} - 1) \qquad (2.66)$$

where: $\phi(t)$ is the bank angle reached at t seconds after full movement of the lateral cockpit controller, δ_{cpt}. Note that Eqn. (2.66) is based on the assumption that the control input can be thought of as a 'perfect' step input.

t is the time elapsed from full movement of the cockpit controller, δ_{cpt} to the time at which the bank angle is to be measured.

<u>IMPORTANT NOTE:</u> in most regulations, the values for $\phi(t)$ and the corresponding value of t are given. Therefore, Eqn. (2.66) can be used by the designer to solve for that value of the control power derivative, $L_{\delta_{cpt}}$ which is needed to satisfy the regulation. With that information the designer can decide how to change the lateral control surface sizes and locations until the regulation is satisfied.

<u>Step 7:</u> Calculate the airplane roll time constant, τ_R from:

$$\tau_R = -1/L_p \qquad (2.67)$$

where L_p is found from Equation (2.61).

If the airplane is to have acceptable roll performance, the roll time constant must be below some maximum allowable value. Maximum allowable values for the roll time constant are presented in the military regulations: Appendix B, Table VII, page 297.

<u>Important Note:</u> The civil regulations do not require specific values of the roll time constant. Accepted design practice in the U.S.A. is to use the appropriate military specifications (Airplane class, flight phase and handling quality level) whenever the civil regulations fail to provide specific design guidance.

<u>Step 8:</u> Document the results obtained, including any design changes made.

2.8 HIGH SPEED CHARACTERISTICS

2.8.1 Applicable Regulations

Civil: FAR 23.253 and FAR 25.253, see Appendix A.

Military: MIL-F-8785C, Section 3.2, in particular Sub-sections 3.2.3.6 and 3.2.3.7, see Appendix B.

2.8.2 Relationship to Preliminary Design

Part II, Chapter 8, in particular Step 8.4.

2.8.3 Mathematical Model for Analyzing High Speed Characteristics

The regulations essentially require that any airplane, when inadvertently placed in a flight condition between the maximum allowable operating speed and the design dive speed, must be able to be recovered to a normal operating flight condition without the use of exceptional pilot skills and without exceeding reasonable cockpit control force levels.

To satisfy these requirements it is usually sufficient in preliminary design to show that no reversals occur in control force versus speed gradients within the speed range prescribed in the regulations.

Calculation of the control force versus speed gradients can be done with the method of Section 3.1.3 with the proviso that all derivatives and coefficients must be evaluated at the high Mach numbers required by the regulations. This cannot normally be done without the availability of windtunnel data and/or advanced aerodynamic panel codes.

Note: Another way around the high speed handling problem is to prevent the airplane from becoming exposed to certain Mach number ranges. This can be done by the addition of 'flight envelope protection' systems, such as on the Airbus 320.

2.8.4 Step-by-Step Procedure for Analyzing High Speed Characteristics

The step-by-step procedure for analyzing high speed characteristics is similar to that of Section 3.1.4. All stability derivatives and coefficients must be evaluated at the high Mach numbers required by the regulations.

2.9 AEROELASTIC CONSIDERATIONS

The analysis methods and procedures presented in Sections 2.1 through 2.8 are based on the assumption that designer has available ALL values for aerodynamic coefficients, control power derivatives and stability derivatives which reflect the ACTUAL airplane behavior in a particular flight condition.

In Sections 2.1 through 2.8 frequent reference is made to the methods for estimating aerodynamic coefficients, control power derivatives and stability derivatives in Part VI. The reader MUST keep in mind that these methods are VALID ONLY for RIGID airplanes. Most high performance airplanes, including fighters and transports are often far from rigid: they exhibit significant aeroelastic behavior!

Therefore it becomes necessary to evaluate the aerodynamic coefficients, the control power derivatives and the stability derivatives not only as a function of Mach number, but also as a function of dynamic pressure. It is the dynamic pressure which (at any combination of angle of attack and Mach number) determines the severity of aeroelastic distortions.

Calculating aeroelastic effects requires rather detailed knowledge of the airplane structural properties: in particular the EI (bending stiffness) and GJ (torsional stiffness) distribution about the airplane elastic axes. In the early design phase such knowledge may not yet be available. Therefore, in the early preliminary design phase, use is made of so-called elastic-to-rigid ratios. These ratios are used to 'guestimate' the ELASTIC AIRPLANE value of a coefficient (or derivative) from a known value of that coefficient (or derivative) for the corresponding RIGID AIRPLANE. A typical example is:

$$C_{L_{\alpha_{elastic}}} = (R_e/R_r)C_{L_{\alpha_{rigid}}} \tag{2.68}$$

where: $C_{L_{\alpha_{rigid}}}$ is the lift-curve slope of the rigid airplane as computed with the methods of Part VI.

R_e/R_r is the elastic-to-rigid ratio which may be computed with the methods of Ch. 8, Reference 12, provided the required structural stiffness information and elastic axes locations are available.

In preliminary design, these ratios are frequently 'guestimated' on the basis of computed data on similar configurations. Chapter 8 of Reference 12 provides examples of elastic airplane versus rigid airplane derivatives. The reader may use these to arrive at an 'educated' guestimate of R_e/R_r for a newly designed airplane. Care must be exercised in doing this: the data in Reference 12 apply to transport airplanes only!

Figure 2.8 shows an example application of these ideas.

AT M = 0.6 AND 10,000 FT $R_e/R_r = -13.5/-18.2 = 0.74$

" " AND 35,000 FT $R_e/R_r = -16.1/-18.2 = 0.88$

Figure 2.8 Example of Extraction of R_e/R_r Values from Elastic Airplane Derivative Data

3. STABILITY: STATIC AND DYNAMIC

The purpose of this chapter is to present Class II methods for analyzing the static and dynamic stability characteristics of airplanes. These methods are compatible with Step 24 in Preliminary Design Sequence II as outlined in Chapter 2 of Part II.

The designer must decide at the outset whether his airplane should be designed with:

1. Inherent stability OR WITH: 2. De-facto stability.

Definition 1: Airplanes with 'inherent stability' do not require any form of 'closed loop' stability augmentation.

The Cessna 172 and the Fokker F-50 are examples of such airplanes.

As a general rule, low to moderate performance airplanes are designed with inherent stability.

Definition 2: Airplanes with 'de-facto stability' require static and/or dynamic stability augmentation.

The F-18 and the X-29 are examples of such airplanes. Both are inherently unstable and rely on a feedback system to achieve theirs de-facto stability: they cannot be controlled without feedback!

As a general rule very high performance airplanes are designed with de-facto stability.

Most moderate to high performance airplanes (B-727, DC-10, C- 141) have inherent static stability but are deficient in inherent dynamic stability. Such airplanes are typically equipped with feedback systems to assure that they have de-facto dynamic stability.

The B-727 and the DC-10 have yaw dampers to assure that the dutch roll damping ratio is sufficiently large. The C-141 also has a pitch damping system to assure that the short period damping ratio is sufficiently large.

QUESTION: What drives airplane designers toward de-facto (or artificial) stability?

ANSWERS: 1. For fighters: Enhanced maneuverability!

2. For transports: Savings in tail areas and weight as well as savings in drag! These savings must be traded against greater complexity of the flight control system and its sensors.

Reliability and maintainability issues also play an important role in the design decision making process between inherent and de-facto stability.

IMPORTANT COMMENT: The choice between inherent stability and de-facto stability is made by the designer (together with the customer) and NOT by the regulations!

The objective of the methods in this chapter is to assure that the airplane meets all the regulatory static and dynamic stability requirements which are placed on an airplane. These regulatory requirements are placed on an airplane to ensure that it is airworthy. Airworthiness in this case implies that the airplane possesses safe, 'minimum' handling characteristics.

To ensure safe, minimum handling characteristics the civil and the military regulations specify certain minimum allowable frequency, damping, time constant and response parameters.

A problem with many civil stability regulations (see (Appendix A) is that they are frequently cast in language which is not easily translated in numerical statements. This makes them difficult to use by designers!

However, the military regulations (see Appendix B) are written in terms of very specific numerical design objectives. For that reason, the military regulations are commonly used in the preliminary design process of BOTH civil and military airplanes.

The military regulations (Appendix B) define static, dynamic and control-response characteristics in three levels. These levels can be summarized as follows:

LEVEL 1: Flying qualities which are clearly adequate for any given mission flight phase.

LEVEL 2: Flying qualities which are adequate to complete the mission flight phase but some increase in pilot workload or degradation in mission effectiveness, or both, exists.

LEVEL 3: Flying qualities such that the airplane can

be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. HOWEVER: the airplane can be safely landed.

IMPORTANT DESIGN GUIDELINES FOR CIVIL AND FOR MILITARY AIRPLANES:

1. Inherently stable airplanes must be designed to LEVEL 1 requirements.

2. De-facto stable airplanes must be designed to LEVEL 1 requirements with the feedback augmentation system in the normal state.

3. After failures occur in the flight control system the degraded handling quality levels which are tolerated depend on the probability, P (per flight) with which the failures occur.

HANDLING QUALITY LEVEL which is tolerated:	Civil Airplanes	Military Airplanes
LEVEL 2	$P < 10^{-4}$	for $P < 10^{-2}$
LEVEL 3	$P < 10^{-6}$	for $P < 10^{-4}$

The airplane designer must be aware of the fact that a penalty paid for 'de-facto' stability is that stability augmentation systems use control power to achieve their objective. It is up to the designer to assure that the required levels of control power are indeed designed into the airplane during the preliminary design process.

The material in this chapter is organized as follows:

3.1 Static longitudinal stability
3.2 Static lateral and directional stability
3.3 Dynamic longitudinal stability
3.4 Dynamic lateral-directional stability
3.5 Dynamic coupling
3.6 Stall characteristics
3.7 Spinning
3.8 Aeroelastic considerations

Reference 12 can be used as a source for derivations and applications of the theory of airplane static and dynamic stability and control: open and closed loop.

3.1 STATIC LONGITUDINAL STABILITY

3.1.1 Applicable Regulations

Civil: FAR 23.171, 23.173, 23.175, and 23.253, see Appendix A.

FAR 25.171, 25.173, 25.175, 25.253, and 25.255, see Appendix A.

Military: Mil-F-8785C, Sub-section 3.2.1 and Paragraph 3.2.2.2, see Appendix B.

3.1.2 Relationship to Preliminary Design

See Part II, Chapter 11.

3.1.3 Mathematical Model for Analyzing Static Longitudinal Stability

The regulations essentially require that the airplane must possess the following characteristics:

1.) With the airplane trimmed at speeds which are specified in the regulations, a PULL must be required to obtain AND maintain speeds below the trim speed. Conversely, a PUSH must be required to obtain AND maintain speeds above the trim speed.

2.) The speed must return to within some specified percentage of the trim speed if the cockpit control is released from the push or pull condition(s) implied by 1.).

3.) The stick-force speed gradient must not be less than that specified in the regulations.

Figure 3.1 illustrates these concepts.

It is clear that a mathematical model for analysis of these characteristics must relate the stick force to speed relative to any given trim speed. Since the mathematical relationship between stick force and speed depends on the type of control system employed (as shown in Eqn.(2.8)) the designer must decide on the type of flight control system to be used in his airplane.

In the following it is assumed that the airplane is equipped with an elevator as the primary longitudinal controller. The stabilizer angle is assumed to have the

Figure 3.1 Stick-force Versus Speed and Return-to-Trim-Speed Characteristics

stick force trim function. In that case, Reference 12, Chapter 5 shows that the following relations hold:

1. For the stick force needed to maintain a speed U, relative to a given trimspeed, U_{trim}:

$$F_s = \eta_h G S_e \bar{c}_e (W/S) x \qquad (3.1)$$

$$x (C_{h_{\delta_e}} / C_{m_{\delta_e}}) (S.M._{free}) (1 - (U/U_{trim})^2$$

2. For the stick-force-speed-gradient at the trim-speed:

$$(\partial F_s / \partial U)_{trim} = -(2/U_{trim}) x \qquad (3.2)$$

$$x \eta_h G S_e \bar{c}_e (W/S) (C_{h_{\delta_e}} / C_{m_{\delta_e}}) (S.M._{free})$$

where: $S.M._{free}$ is the stick-free static margin of the airplane. This quantity may be found from the following approximation:

$$S.M._{free} = \bar{x}_{ac_A} - \bar{x}_{cg} + \qquad (3.3)$$

$$+ (C_{m_{\delta_e}} / C_{L_\alpha}) (C_{h_\alpha} / C_{h_{\delta_e}}) (1 - d\varepsilon/d\alpha)$$

with: $\bar{x}_{ac_A}$ the airplane aerodynamic center as found from Part VI, Sub-section 8.2.5.2

$\bar{x}_{cg}$ the airplane c.g. location in fractions of the wing m.g.c.

All other quantities have been previously defined.

<u>NOTE 1:</u> As long as the hingemoment derivative $C_{h_{\delta_e}}$ has the normal (negative) sign, the stick-force as found from Equation (3.1) and the stick-force-speed-gradient as found from Equation (3.2) have the correct sign, provided the airplane has a POSITIVE STATIC MARGIN!

<u>NOTE 2:</u> The reader should modify these equations if the flight control system of his airplane is different

than the one assumed here. For guidance on how to do this the reader should consult Chapter 5 of Reference 12.

3.1.4 Step-by-Step Procedure for Analyzing Static Longitudinal Stability

Step 1: Determine which regulations apply to the design: see Table 3.1 and Subsection 3.1.1.

Step 2: The regulations define the flight conditions and the airplane configurations for which the stability requirements must be met. Tabulate these flight conditions and these configurations.

Step 3: For the flight conditions and for the airplane configurations of Step 2 determine the Mach number, the dynamic pressure and the center of gravity locations.

Step 4: For the flight conditions and for the airplane configurations of Steps 2 and 3 determine all derivatives and coefficients used in Equations (3.1) through (3.3).

Step 5: Calculate the stick-force-speed gradient from Eqn.(3.2) and check its value against that allowed in the applicable regulation. If the required gradient is not achieved, determine which parameters need to be adjusted. Note: in many cases a satisfactory gradient can be achieved only by moving the aft allowable center of gravity forward! This has obvious implications to the utility of the airplane!

Step 6: Document the results obtained, including any design changes made.

3.2 STATIC LATERAL AND DIRECTIONAL STABILITY

3.2.1 Applicable Regulations

Civil: FAR 23.171 and FAR 23.177, see Appendix A.

FAR 25.171 and FAR 25.177, see Appendix A.

Military: Mil-F-8785C, Sub-section 3.3.6, see Appendix B.

3.2.2 Relationship to Preliminary Design

See Part II, Chapter 11.

3.2.3 Mathematical Model for Analyzing Static Lateral and Directional Stability

The regulations essentially require that the airplane must possess the following characteristics:

1.) When the airplane is put in a sideslip (skid) condition it shall have the tendency to return to the original (zero sideslip) condition.

2.) The rudder pedal force required to put the airplane in a sideslip condition shall be such that the pedal-force-gradient is does not reverse its sign.

3.) When the airplane is put in a positive sideslip (skid), it must have the tendency to raise the left wing.

Figure 3.2 illustrates these concepts. The conditions for which these requirements are satisfied are now presented.

1.) This condition is referred to as the condition for positive directional stability:

a) For irreversible control systems:

$$C_{n_\beta} > 0 \qquad (3.4)$$

The directional stability derivative, C_{n_β} may be computed with the method of Section 10.2.4 of Part VI. From those methods it will be clear that directional stability strongly de-

Figure 3.2 Requirements for Lateral and Directional Static Stability

pends on the size and shape of the fuselage and on the size, shape and location of the vertical tail.

b) For reversible control systems:

$$C_{n_{\beta_{free}}} = C_{n_{\beta_{fixed}}} + \tag{3.5}$$

$$+ C_{n_{\delta_r}} (C_{h_{\beta_v}} / C_{h_{\delta_r}})$$

where: $C_{n_{\beta_{fixed}}} = C_{n_\beta}$ of Eqn.(3.4)

$C_{n_{\delta_r}}$ is the rudder control power derivative. It may be computed with the method of Section 10.3.8 of Part VI.

$C_{h_{\beta_v}}$ is the rudder hingemoment derivative due to sideslip. It is physically equivalent to the elevator hingemoment derivative due to elevator deflection, $C_{h_{\delta_e}}$ and may be computed with the help of Sect.10.4 of Pt VI.

$C_{h_{\delta_r}}$ is the rudder hingemoment derivative due to rudder deflection. It is physically equivalent to the elevator hingemoment derivative due to elevator deflection, $C_{h_{\delta_e}}$ and may be computed with the method of Section 10.4 of Pt VI.

2.) This condition is satisfied if:

$$\partial F_r / \partial \beta > 0 \tag{3.6}$$

It is shown in Chapter 5 of reference 12 that this in turn is satisfied if:

$$C_{n_{\beta_{free}}} > 0 \tag{3.7}$$

Note therefore that for this type of airplane the detail design of the flight control system becomes (again) important!

3.) This condition is satisfied as long as:

$$C_{l_\beta} < 0 \qquad (3.8)$$

The lateral stability derivative, C_{l_β} may be determined with the method of Section 10.2.4 of Part VI. The method makes it clear that this derivative depends mainly on three factors:

a) wing geometric dihedral angle
b) wing position on the fuselage (high or low)
c) wing sweep angle and lift coefficient

3.2.4 Step-by-Step Procedure for Analyzing Static Lateral and Directional Stability

Step 1: Determine which regulations apply to the design: see Table 1.3 and Subsection 3.2.1.

Step 2: The regulations define the flight conditions and the airplane configurations for which the static lateral-directional stability requirements must be satisfied. Tabulate these flight conditions and these configurations.

Step 3: For the flight conditions and for the airplane configurations of Step 2 determine the Mach number, the dynamic pressure and the center of gravity locations.

Step 4: For the conditions and configurations of Steps 2 and 3 determine the derivatives used in Equations (3.4) through (3.8).

Step 5: Using Equations (3.4) through (3.8) determine whether or not these stability conditions are satisfied. If not, determine how the design must be modified to meet the requirements.

NOTE: It may very well be that the airplane needs a larger vertical tail or that the geometric dihedral angle of the wing will have to be changed to meet the regulations.

Step 6: Document the results obtained, including any design changes made.

3.3 DYNAMIC LONGITUDINAL STABILITY

3.3.1 Applicable Regulations

Civil: FAR 23.181 and FAR 25.181, see Appendix A.

Military: Mil-F-8785C, Par.3.2.2, see Appendix B.

3.3.2 Relationship to Preliminary Design

See Part II, Step 24 in Chapter 2. Also, read Chapter 11 in Part II.

3.3.3 Mathematical Model for Analyzing Dynamic Longitudinal Stability

The civil regulations in Appendix A are vague about specific requirements for frequency and damping characteristics. For that reason the military regulations of Appendix B are used to check whether or not a civil airplane design meets the intent of the civil regulations.

The civil and military requirements for dynamic stability must be met with the cockpit controls held fixed as well as with the cockpit controls kept free. The analysis of dynamic stability with the controls held free is beyond the scope of preliminary design since it requires detailed design data on the flight control system. For that reason, only the 'controls fixed' case is considered here. An analytical approach to the analysis of dynamic stability with controls free is given in Reference 12.

The military regulations (Appendix B) specify ranges of acceptable values for the following dynamic stability parameters of the airplane phugoid (P) and short-period (SP) modes of perturbed motion:

1. Undamped natural frequency: $\omega_{n_{SP}}$
2. Damping ratio: ζ_P and ζ_{SP}

Because phugoid frequency is primarily determined by airplane speed, there is no requirement for it.

The required numerical magnitudes for frequency and damping behavior of airplanes depends on:

1. Airplane Type: this is recognized by classifying airplanes in four classes:

   Class I, II, III or IV: see Appendix B.

2. Flight Phase: this is recognized by classifying flight conditions into three flight phases:

Phase: A, B, and C: see Appendix B.

The following characterization is typical of the modal breakdown for most airplanes in most flight conditions:

Mode	Undamped Natural Frequency	Damping Ratio	Most Important Motion Variables
Phugoid	Low	Low	Speed and Pitch Attitude Angle
Short Period	High	Moderately High	Angle of Attack, Pitch Attitude Angle

The damping characteristics of the phugoid mode of motion primarily affects the pilot's ability to control airspeed with the longitudinal controls. The phugoid damping ratio must not be too low.

The frequency and damping characteristics of the short period mode of motion primarily affects the pilot's ability to control pitch attitude angle. The short period frequency must not be too low, nor too high. The short period damping ratio must not be too low.

The phugoid and short period frequencies and damping ratios can be found from a solution of the three-degree-of-freedom, small perturbation equations of motion of an airplane. General methods (so-called Class III methods) for solving such equations are presented in Chapter 6 of Reference 12. In preliminary design it is acceptable to employ approximations for the phugoid and for the short-period behavior of an airplane. The Class II methods to be presented next are based on such approximate methods. The justification for these Class II methods is also discussed in Chapter 6 of Reference 12.

3.3.3.1 Class II method for analysis of phugoid characteristics

The following Class II method is suggested for determining phugoid undamped natural frequency and damping ratio:

$$\omega_{n_P} = (1.414g/U_1) \tag{3.9}$$

$$\zeta_P = \{g/(2C_{L_1}U_1\omega_{n_P})\}(C_{D_u} + 2C_{D_1} - C_{T_{x_u}} - 2C_{T_{x_1}}) \tag{3.10}$$

where: U_1 is the steady state speed for the flight condition for which the dynamic stability of the airplane is being analyzed.

ρ is the air density for the flight condition being analyzed.

C_{L_1} is the steady state lift coefficient in the flight condition being analyzed.

C_{D_u} and $C_{T_{x_u}}$ are the drag-due-to-speed and thrust-due-to-speed derivatives. These derivatives may be computed with the methods of Ch.10 of Part VI.

C_{D_1} and $C_{T_{x_1}}$ are the steady state drag and thrust coefficients. These may be computed with the methods of Ch.10 of Part VI.

Observe that the phugoid frequency depends only on airplane speed! The phugoid damping ratio depends on the steady state drag and thrust as well as on the variation of drag and thrust with speed (Mach number!). Also observe, that the phugoid damping ratio is inversely proportional to the airplane lift-to-drag ratio, C_{L_1}/C_{D_1}.

3.3.3.2 Class II method for analysis of short period characteristics

The following Class II method is suggested for determination of the short period undamped natural frequency and damping ratio:

$$\omega_{n_{SP}} = \{(Z_\alpha M_q/U_1) - M_\alpha\}^{1/2} \tag{3.11}$$

$$\zeta_{SP} = -\{M_q + (Z_\alpha/U_1) + M_{\dot\alpha}\}/2\omega_{n_{SP}} \tag{3.12}$$

where: the dimensional derivatives: Z_α, M_α, $M_{\dot\alpha}$ and M_q are all defined in the list of symbols. These de-

rivatives are functions of several dimensional derivatives which can be determined with the methods of Chapter 10, Part VI.

U_1 is the steady state speed corresponding to the flight condition for which dynamic stability is being analyzed.

It turns out that for many airplanes, in many flight conditions, the combination $(Z_\alpha M_q/U_1)$ in Eqn.(3.11) is small compared to M_α. The significance of this is that the short period undamped natural frequency is determined mostly by the magnitude of the derivative M_α.

IMPORTANT OBSERVATION ON EQN.(3.11):

The derivative M_α depends directly on the center of gravity location of the airplane, because:

$$C_{m_\alpha} = C_{L_\alpha}(\bar{x}_{cg} - \bar{x}_{ac_A}) \qquad (3.13)$$

Therefore: the short period undamped natural frequency of an airplane depends primarily on the distance between the center of gravity and the airplane aerodynamic center (stick-fixed neutral point!).

The reader is reminded of the fact that the c.g. location at any given airplane weight depends on the loading state of the airplane as reflected by the airplane weight-c.g. excursion diagram of Chapter 10 in Part II.

IMPORTANT OBSERVATION ON EQN.(3.12):

This equation predicts that the short period damping ratio of an airplane will always be positive! This is NOT correct. If this equation predicts the damping ratio to be less than 0.1, the reader is advised to use the Class III method of Chapter 6 of Reference 12.

3.3.4 Step-by-Step Procedure for Analyzing Dynamic Longitudinal Stability

Step 1: Determine which regulations apply to the design: see Table 1.3 and Subsection 3.3.1.

Step 2: Determine from the description of airplane Classes in Appendix B which airplane Class (I, II, III or IV) best 'fits' the design.

Step 3: Determine for which flight conditions and for which airplane configurations the dynamic stability must be verified. Determine which flight phase category (A, B or C) is associated with each flight condition. The definition of flight phase categories is given in Appendix B.

Tabulate this information.

For inexperienced readers the following guidelines may be useful:

Flight conditions:

For purposes of preliminary design, the most critical flight conditions may be determined from the 'design' flight envelope. Figure 3.3 gives examples of 'design' flight envelopes for a civil and for a military airplane. The 'design' flight envelope normally follows from the airplane mission specification and from the Class II performance calculations which have been performed as part of Step 28 as described in Chapter 2 of Part II. Methods for Class II performance calculations are presented in Chapter 5.

Figure 3.3 shows the most critical flight conditions for which the dynamic stability should be verified at the early design stage.

Airplane configurations:

For purposes of preliminary design, the most critical airplane configurations occur at the most forward and at the most aft center of gravity locations. These c.g. locations follow from the weight-c.g. excursion diagram of Chapter 10 in Part II.

To decide on other critical configuration effects, the reader should check items 1-9 on page 7 and determine which of these effects apply to his design.

Figure 3.3 Examples of Civil Transport and Military Fighter Flight Envelopes with Indication of Flight Conditions Requiring Static and Dynamic Stability Calculations in Preliminary Design

Step 4: For the flight conditions and for the airplane configurations defined in Step 3, calculate the derivatives and other parameters which appear in Eqns (3.9) through (3.12).

Since the allowable short-period frequency ranges in Appendix B depend on the load-factor-to-angle-of-attack derivative n_α, also compute:

$$n_\alpha = \bar{q}SC_{L_\alpha}/W \qquad (3.14)$$

These 'input' data should all be tabulated!

Step 5: Decide whether the airplane being designed is to be 'inherently stable' or 'de-facto' stable.

Definitions for these types of stability are given on page 65.

If the airplane is to be 'inherently stable' proceed to Step 6.

If the airplane is to be 'de-facto stable' proceed to Step 8.

The choice between 'inherent' and 'de-facto' stability is made by the designer and not by the regulations!

Step 6: Determine the phugoid damping ratios associated with the flight conditions and airplane configurations of Step 3. Use Eqn.(3.10).

Determine the LEVEL 1 phugoid damping ratio requirements from Paragraph 3.2.1.2, in Appendix B.

Decide whether or not the airplane meets the LEVEL 1 requirements. If not, determine from Eqn.(3.10) what (if any) design changes should be made to satisfy the LEVEL 1 requirements OR put the airplane in the 'de-facto' stable category and proceed to Step 8.

Note: in most cases it will be found to be impractical to modify airplane phugoid damping through airplane design changes. The reasons are that phugoid damping depends mostly on drag and on installed thrust characteris-

tics. These in turn are normally dictated by performance considerations! Therefore, airplanes with deficient inherent phugoid damping are usually equipped with speed-to-thrust-feedback, speed-to-speedbrake feedback and/or with pitch-attitude-feedback. Chapter 13 of Reference 12 contains detailed discussions of such systems.

<u>Step 7:</u> Determine the short period frequencies and damping ratios associated with the flight conditions and airplane configurations of Step 5. Use Eqns.(3.11) and (3.12).

Determine the LEVEL 1 short period frequency and damping ratio requirements with the help of Figures B1 through B3 and Table IV in Appendix B. Compute n_α with Eqn.(3.14).

Decide whether or not the airplane meets the LEVEL 1 requirements. If not, use Eqns (3.11) and (3.12) to determine what design changes should be made to the airplane.

<u>Note 1:</u> By performing a derivative sensitivity analysis on Equations (3.11) and (3.12) the reader will find that the short period frequency is very sensitive to C_{m_α} (thus, according to Eqn.(3.13) to c.g. location!). It is also found that for a given short period frequency, the short period damping ratio is very sensitive to C_{m_q} (in other words to tail moment arm as shown in Section 10.2.7 of Part VI!).

<u>Note 2:</u> From a practical viewpoint, the only derivative over which the designer has much influence at this stage of the design is C_{m_α}. Eqn.(3.13) shows that this derivative may be altered by changes in the center of gravity location. The designer may therefore wish to 'taylor' the c.g. range of the airplane such as to bring the short period frequency range 'in line' with the requirements. Such c.g. range 'tailoring' will result in modification of the weight and balance of the airplane as discussed in Chapter 10 of Part II.

Another option is to put the airplane in the 'de-facto stable' category and proceed to Step 9.

Step 8: Carry out the instructions in the first two paragraphs of Step 6: the result is the inherent phugoid damping ratio, ζ_p. Note from Appendix B, Par.3.2.1.2 that the Level 1, minimum required phugoid damping ratio is: $\zeta_p = 0.04$. If the inherent phugoid damping ratio is insufficient, augmentation can be accomplished with speedbrake or throttle (i.e. thrust) feedback to speed. The required feedback gains can be found as follows:

1. For speed feedback to a speedbrake:

Compute the required increment in the speed-damping derivative, C_{D_u} from:

$$\Delta C_{D_u} = [(0.08 C_{L_1} U_1 \omega_{n_p})/g] - C_{D_u} - 2C_{D_1} + C_{T_{x_u}} + 2C_{T_{x_1}} \qquad (3.15)$$

Compute the speedbrake feedback gain from:

$$k_{u/\delta_{sb}} = \Delta C_{D_u} / (U_1 C_{D_{\delta_{sb}}}) \qquad (3.16)$$

The magnitude of this gain should not exceed a value of 150 degrees of speedbrake deflection per unit of u/U_1.

The drag-due-to-speedbrake derivative $C_{D_{\delta_{sb}}}$ may be computed using the 'spoiler drag' method of Sub-section 4.12.1 in Part VI.

For speed feedback to throttle:

Compute the required increment in $C_{T_{x_u}}$ from:

$$\Delta C_{T_{x_u}} = -[(0.08 C_{L_1} U_1 \omega_{n_p})/g] - C_{D_u} - 2C_{D_1} + C_{T_{x_u}} + 2C_{T_{x_1}} \qquad (3.17)$$

Compute the throttle feedback gain from:

$$k_{u/\delta_T} = \Delta C_{T_{x_u}} / (U_1 C_{T_{\delta_T}}) \qquad (3.18)$$

The magnitude of this feedback gain should not exceed a value which corresponds to more than 15 percent increase in thrust for a sheargust of 25 fps.

The thrust-coefficient-due-to-throttle derivative, $C_{T_{\delta_T}}$ follows from taking the slope of thrust coefficient versus throttle position of the engine, remembering that:

$$C_T = T/\bar{q}S \qquad (3.19)$$

where: T is the installed thrust of the engine. It may be found with the method of Chapter 6 in Part VI.

For more detailed methods of synthesizing speed-damping feedback systems, refer to Chapter 13 of Reference 12.

<u>Step 9:</u> For the appropriate flight conditions and for the airplane configurations of Step 2 AND for the n_α values of Step 3, find the allowable short period frequency values from Figures B.1 through B.3 in Appendix B.

Decide which frequency levels the airplane should have for each flight condition. Such a frequency level is called: $\omega_{n_{SP_{de\text{-}facto}}}$.

Compute the 'de-facto' level of static stability at which the airplane should operate from:

$$(C_{m_\alpha})_{de\text{-}facto} = \qquad (3.20)$$

$$\{(Z_\alpha M_q/U_1) - (\omega_{n_{SP_{de\text{-}facto}}})^2\}(I_{yy}/\bar{q}S\bar{c})$$

Find the angle-of-attack feedback gain, k_α from:

$$k_\alpha = [(C_{m_\alpha})_{de\text{-}facto} - C_{m_\alpha}]/C_{m_{\delta_e}} \tag{3.21}$$

where: C_{m_α} is the inherent value of the static longitudinal stability derivative of the airplane. This value will normally be positive (unstable) for a 'de-facto' stable airplane!

The value for k_α should not be larger than 5 degrees of elevator deflection per degree of angle of attack. If k_α does not satisfy this condition, the longitudinal control power level, $C_{m_{\delta_e}}$ must be increased until it does.

<u>Note:</u> if the feedback is around the stabilizer instead of around the elevator, replace the elevator derivative by the stabilizer derivative in Equation (3.16).

<u>Step 10:</u> For the appropriate flight conditions and for the airplane configurations of Step 3, find the minimum allowable damping ratios at which the airplane should operate by using Table IV in Appendix B. This damping ratio level is called: $\xi_{SP_{de\text{-}facto}}$.

Compute the 'de-facto' level of C_{m_q} at which the airplane should operate from:

$$C_{m_{q_{de\text{-}facto}}} = -(2\omega_{n_{SP_{de\text{-}facto}}}\xi_{SP_{de\text{-}facto}}) + -(Z_\alpha/U_1) - M_\alpha \tag{3.22}$$

Compute the pitchrate feedback gain from:

$$k_q = [(C_{m_{q_{de\text{-}facto}}} - C_{m_q})(\bar{c}/2U_1)]/C_{m_{\delta_e}} \tag{3.23}$$

The magnitude of this gain should not exceed 2 deg/deg/sec. If it does, the airplane needs more control power.

<u>Step 11:</u> Document the results obtained, including any design changes made.

3.4 DYNAMIC LATERAL-DIRECTIONAL STABILITY

3.4.1 Applicable Regulations

Civil: FAR 23.181 and FAR 25.181, see Appendix A.

Military: Mil-F-8785C, Par.3.3.1, see Appendix B.

3.4.2 Relationship to Preliminary Design

See Part II, Step 24 in Chapter 2.

3.4.3 Mathematical Model for Analyzing Lateral-Directional Dynamic Stability

Because the civil regulations in Appendix A are vague about specific requirements of frequency, damping and time-constant characteristics it is customary to use the military regulations of Appendix B to verify whether or not a new civil airplane design meets the intent of the regulations.

The civil and military requirements for dynamic stability must be met with the cockpit controls held fixed as well as with the cockpit controls kept free. The analysis of dynamic stability with the controls held free is beyond the scope of preliminary design since it requires detailed design data on the flight control system. For that reason, only the 'controls fixed' case is considered here. An analytical approach to the analysis of dynamic stability with controls free is given in Reference 12.

The military regulations (Apppendix B) specify ranges of acceptable values for the following dynamic stability parameters:

For the spiral mode: minimum allowable time-to-double the amplitude, T_{2_S}

For the dutch roll mode: minimum allowable undamped natural frequency, ω_{n_D}

minimum allowable damping ratio, ζ_D

minimum allowable real root part value: $\omega_{n_D}\zeta_D$

For the roll mode: see Section 2.7.

The following characterization is typical of the

modal breakdown for most airplanes in most flight conditions:

Mode	Undamped Natural Frequency	Damping Ratio	Time Constant	Most Important Motion Variables
Spiral	N.A.	N.A.	Large, Unstable	Heading and bank angles
Dutch Roll	Moderately High	Low	N.A.	Sideslip and bank angles
Roll	N.A.	N.A.	Small	Bank angle

Note: Roll mode requirements are discussed in Section 2.7

The spiral behavior primarily affects the pilot's ability to maintain wings level flight, particularly when in IFR conditions.

The dutch roll behavior tends to make passengers sick if it is undamped. If the dutch roll mode has a high degree of bank angle participation, it can interfere with a pilot's ability to carry out rapid rolling maneuvers and/or bank angle tracking maneuvers.

The spiral and dutch roll mode behavior can be obtained from a solution of the three-degree-of-freedom, small perturbation equations of motion of an airplane. General methods (so-called Class III methods) for solving such equations are given in Chapter 6 of Reference 12. In preliminary design it is acceptable to employ approximations for the spiral and for the dutch roll behavior of an airplane. The Class II methods to be presented in Subsections 3.4.3.1 and 3.4.3.2 are based on such approximate methods. The justification for these Class II methods is also discussed in Chapter 6 of Reference 12.

3.4.3.1 Class II method for analysis of the spiral characteristics

There is no requirement for a stable spiral root. In fact, too much spiral stability is considered undesirable in most airplanes. An acceptable Class II condition for spiral stability is:

$$L_\beta N_r - N_\beta L_r > 0 \qquad (3.24)$$

If this spiral stability condition is violated (as

is often the case), an acceptable Class II method for predicting the time-to-double the amplitude in the spiral mode is:

$$T_{2_S} = (L_\beta \ln 2)/(N_\beta L_r - L_\beta N_r) \tag{3.25}$$

All dimensional stability derivatives in Equations (3.24) and (3.25) are defined in the list of symbols. The corresponding dimensionless derivatives may be computed with the methods of Chapter 10 in Part VI.

3.4.3.2 Class II method for analysis of the dutch roll characteristics

The following Class II method is suggested for the calculation of the dutch roll undamped natural frequency and damping ratio:

$$\omega_{n_D} = [\{Y_\beta N_r + N_\beta (U_1 - Y_r)\}/U_1]^{1/2} \tag{3.26}$$

$$\zeta_D = -\{N_r + (Y_\beta/U_1)\}/2\omega_{n_D} \tag{3.27}$$

All dimensional stability derivatives in Equations (3.26) and (3.27) are defined in the list of symbols. The corresponding dimensionless derivatives may be computed with the methods of Chapter 10, Part VI.

IMPORTANT OBSERVATION ON EQUATION (3.27):

This equation predicts that the dutch roll damping ratio will always be positive. This is NOT correct! The problem is that this dutch roll approximation ignores the effects of the derivatives C_{l_β}, C_{l_p} and C_{n_p}.

If this equation predicts the damping ratio to be less than 0.05, it is suggested that the reader use the Class III method of Chapter 6, Reference 12 to predict the actual dutch roll damping ratio.

3.4.4 Step-by-Step Procedure for Analyzing Dynamic Lateral-Directional Stability

Step 1: Determine which regulations apply to the design: see Table 1.3 and Subsection 3.4.1.

Step 2: Determine from the description of airplane Classes in Appendix B which airplane Class (I, II, III or IV) best 'fits' the design.

Step 3: Determine for which flight conditions and for which airplane configurations the dynamic stability must be verified. Determine which flight phase category (A, B or C) is associated with each flight condition. The definition of flight phase categories is given in Appendix B.

Tabulate this information.

NOTE: Be sure to read the guidelines stated under Step 3 in Subsection 3.3.4 (pages 80-81)!

Step 4: For the flight conditions and for the airplane configurations defined in Step 3, calculate the derivatives and other parameters which appear in Eqns. (3.24) through (3.27).

Step 5: Decide whether the airplane is to be 'inherently' stable or 'de-facto' stable.

Definitions for these types of stability are given on pages 65-67.

If the airplane is to be 'inherently stable' proceed to Step 6.

If the airplane is to be 'de-facto' stable, proceed to Step 8.

The choice between 'inherent' and 'de-facto' stability is made by the designer and not by the regulations!

Step 6: Determine whether or not the airplane is spirally stable with the help of Eqn.(3.24).

Case 1: Stable spiral

If the airplane is found to be spirally stable, it may in fact be too spirally stable!

Since specific levels of spiral stability are not required by the regulations, the designer should check how much of a change in the derivative L_β (i.e. C_{l_β}) is needed to obtain neutral spiral stability. The required change in L_β can normally be achieved through a change in the wing geometric dihedral angle. The required change in the wing geometric dihedral angle can be obtained with Eqn.(10.34) on p.392 of Part VI. Recalculate L_β as required!

<u>Case 2: Unstable spiral</u>

If the airplane is found to be unstable in the spiral mode, use Eqn.(3.25) to find the spiral 'time-to-double' the amplitude.

From the regulations determine whether or not the Level 1 requirements are met. If not, use Eqn.(3.25) to determine that value of L_β (and C_{l_β}) for which the Level 1 requirement is met. Use Eqn.(10.34) on p.392 of Part VI to determine the change in wing geometric dihedral angle needed to get the required Level 1 value of T_{2_S}. Recalculate L_β as required!

<u>IMPORTANT NOTE:</u> When it is decided to change the wing geometric dihedral angle, make sure that the lateral landing gear clearance criterion of Figure 9.1b, page 221 of Part II is not violated!

<u>Step 7:</u> Use Equations (3.26) and (3.27) to find the dutch roll characteristics of the airplane.

From the regulations determine whether or not Level 1 requirements are met. If not, use Eqns.(3.26) and (3.27) to find <u>'compatible'</u> values for N_r and N_β so that the Level 1 requirements are met.

<u>NOTE 1:</u> The reader will usually find that the undamped natural frequency of Eqn.(3.26) is 'driven' by N_β, while the damping ratio of Eqn. (3.27) is 'driven' by N_r.

NOTE 2: from Part VI, Equations (10.43) and (10.88) the reader will be able to verify that N_r and N_β BOTH depend on:

1. Vertical tail size
2. Vertical tail moment arm
3. Vertical tail lift-curve-slope and therefore on vertical tail planform geometry

'Compatible' values for N_r and N_β are required because of these relationships.

Make the necessary changes in the design and check that the spiral condition has not been significantly affected. If it has, an iteration between Steps 6 and 7 will be needed.

Step 8: Using Equations (3.25) - (3.27) determine the spiral and dutch roll characteristics of the airplane. Check these characteristics against the requirements of Appendix B, Tables VI and VIII and determine the level of handling qualities for the airplane.

It will now be necessary to determine what type of feedback is required to 'augment' the airplane such that it meets the Level 1 requirements. When it does, it is referred to as a 'de-facto' stable airplane. The following requirements for feedback may arise:

1) sideslip angle feedback: this changes C_{y_β}, C_{l_β} and C_{n_β}

2) yawrate feedback: this changes C_{y_r}, C_{l_r} and C_{n_r}

3) rollrate feedback: this changes C_{y_p}, C_{l_p} and C_{n_p}

The usual procedure is:

Step 8.1: Find those values of N_r, N_β and L_β which meet Level 1 requirements for the dutch roll and spiral characteristics as expressed by Equations (3.25) - (3.27).

NOTES: 1) Use Eqn.(3.26) to find the value of N_β which

meets Level 1 frequency requirement

2) Use Eqn.(3.27) to find the value of N_r which meets Level 1 damping requirements: using the 'new' frequency value!

3) Use Eqn.(3.25) to find the value of L_β which meets Level 1 time-to-double-spiral-amplitude.

<u>Step 8.2:</u> By subtraction, determine the 'increment', (Δ = de-facto - inherent) in each derivative which must be generated by the feedback system.

<u>Step 8.3:</u> Estimate the feedback gains and judge them by an acceptability criterion.

The feedback gains may be estimated from:

$$k_{\beta/\delta_r} = (C_{n_\beta}{}_{\text{de-facto}} - C_{n_\beta}{}_{\text{inherent}})/C_{n_{\delta_r}} \tag{3.28}$$

$$k_{r/\delta_r} = (C_{n_r}{}_{\text{de-facto}} - C_{n_r}{}_{\text{inherent}})b/(2U_1 C_{n_{\delta_r}}) \tag{3.29}$$

$$k_{\beta/\delta_a} = (C_{l_\beta}{}_{\text{de-facto}} - C_{l_\beta}{}_{\text{inherent}})/C_{l_{\delta_a}} \tag{3.30}$$

Gains estimated from Eqns.(3.28) - (3.30) are judged acceptable in preliminary design if they satisfy the following inequalities:

$$(k_{\beta/\delta_r})\beta_{\text{expected}} < 0.3\delta_{r_{\text{available}}} \tag{3.31}$$

$$(k_{r/\delta_r})r_{\text{expected}} < 0.3\delta_{r_{\text{available}}} \tag{3.32}$$

$$(k_{\beta/\delta_a})\beta_{\text{expected}} < 0.3\delta_{a_{\text{available}}} \tag{3.33}$$

The 'expected values of the sideslip and yawrate perturbations in Equations (3.29) - (3.32) depend on the type of airplane being designed. The following guidelines are suggested:

Airplane Type	$\beta_{expected}$	$r_{expected}$
Transports	5 deg	10 deg/sec
Fighters	10 deg	20 deg/sec
Light airplanes	10 deg	20 deg/sec

The reader is referred to Chapter 6 of Reference 12 for the development of the theory behind these gain estimations.

If the gain acceptability criteria are violated, the airplane does not have sufficient control power to be augmented to Level 1 requirements. The designer should incorporate the necessary design changes to increase the appropriate control power levels. Control power equations in Section 10.3 of Part VI should be used to determine which design changes are feasible. In some cases it will be found necessary to introduce new control surfaces and/or to scrap the design and start from scratch!

Step 9: Document the results obtained, including any design changes made.

3.5 DYNAMIC COUPLING

3.5.1 Applicable Regulations

Civil: There are no specific regulations governing dynamic coupling phenomena. However, the FAR's can be interpreted as requiring that no dynamic coupling phenomena may exist which make normal piloting difficult.

Note: the only type of civil airplanes which conceivably might encounter dynamic coupling phenomena are aerobatic type airplanes.

Military: Mil-F-8785C, Par.3.4.3, see Appendix B

Notes: 1) Roll-rate coupling into pitch and yaw degrees of freedom occurs in fighter type airplanes during combat roll rate maneuvers.

2) Pitch-rate coupling into the lateral-directional degrees of freedom occurs in fighter type airplanes during rolling pull-up or during rolling push-over maneuvers.

3.5.2 Relationship to Preliminary Design

Dynamic coupling was not addressed in Part II.

3.5.3 Mathematical Model for Analyzing Dynamic Coupling

Two types of dynamic coupling phenomena will be addressed:

3.5.3.1 Roll-rate coupling

3.5.3.2 Pitch-rate coupling

3.5.3.1 Roll-rate coupling

As shown in Chapter 7 of Reference 12, when roll-rate coupling is divergent, violent excursions in angle of attack and/or in sideslip angle may occur during combat roll-rate maneuvers. These excursions can be so violent, that structural failure occurs.

A theoretical model for determining whether or not an airplane will experience roll-rate coupling induced instabilities is discussed in Chapter 7 of Reference 12. Based on this mathematical model the following condition for stability can be derived:

$$[\{(M_q N_r/M_\alpha) + D_1 + (N_\beta/M_\alpha)C_1\}^2 +$$

$$- 4C_1 D_1 (N_\beta/M_\alpha)] < 0 \qquad (3.34)$$

where: all dimensional derivatives are defined in the list of symbols.

$$C_1 = (I_{xx} - I_{zz})/I_{yy} \qquad (3.35)$$

$$D_1 = (I_{yy} - I_{xx})/I_{zz} \qquad (3.36)$$

The moments of inertia in Eqns (3.35) and (3.36) must be determined in the airplane stability axes system!!

3.5.3.2 Pitch-rate coupling

No suitable Class II mathematical model is available for the analysis of pitch-rate coupling into the lateral-directional degrees of freedom. For a Class III method, the reader is referred to Chapter 7 of Reference 12.

3.5.4 Step-by-Step Procedure for Analyzing Roll-rate Coupling

Step 1: Read the applicable regulation, see 3.5.1.

Step 2: From the operational flight envelope of the airplane, determine the flight conditions and airplane configurations for which roll-rate coupling must be analyzed.

Note: flight conditions which are critical for roll coupling are usually those at moderate to high altitude.

Step 3: For the flight conditions and airplane configurations of Step 2 determine the dimensional derivatives which occur in Eqn.(3.34)

Note: for a highly augmented airplane, these derivatives must include the effect of the stability augmentation system.

Step 4: Using Eqn.(3.34) determine whether or not the airplane has roll-rate coupling instability. If not, proceed to Step 8, otherwise proceed to Step 5.

Step 5: Determine the minimum roll-rate for which roll-rate coupling instability instability occurs from the method of Chapter 7, Ref.12.

Step 6: Determine the maximum roll rate of which the airplane is capable from:

$$|p_{max}| = |-(L_{\delta_a} \delta_{a_{max}})/L_p| \qquad (3.37)$$

where: all dimensional stability derivatives are defined in the list of symbols.

If the airplane maximum roll rate from Eqn.(3.37) is less than that of Step 5, proceed to Step 8, if not, proceed to Step 7.

Step 7: Investigate with the help of Eqn.(3.34) what values the derivatives must have to assure that the airplane is stable against roll-rate coupling. These new values are the so-called 'augmented' (or 'de-facto') values which these derivatives must have to prevent roll coupling instability.

With the help of Equations (3.28) - (3.30), determine the required feedback gains.

Next, using the acceptability criteria as expressed by Eqns (3.31) - (3.33) determine whether or not sufficient control power is available in the airplane. If not, introduce the necessary design changes.

Step 8: Document the results obtained as well as any design changes made.

3.6 STALL CHARACTERISTICS

3.6.1 Applicable Regulations

Civil: FAR 23.201, 23.203, 23.305 and 23.207, see Appendix A.

FAR 25.201, 25.203, 25.205 and 25.207, see Appendix A.

Military: Mil-F-8785C, Par.3.4.2, see Appendix B

3.6.2 Relationship to Preliminary Design

See Step 6 in Preliminary Design Sequence I, p.12, Chapter 2 in Part II. Also see Chapter 7 in Part II and Subsection 5.1.4 in Part III.

3.6.3 Mathematical Model for Analyzing Stall Characteristics

Stall characteristics must be shown to satisfy the regulations for stalls starting from the following types of flight condition:

1. Wings level flight

2. Turning flight, including accelerated stalls

3. Engine-out stalls

The regulations essentially require that the airplane possess the following desirable tendencies when put in an approach to any of these stalls:

A. No sudden, uncontrollable pitch-up

B. Adequate roll control and no more wing drop than specified in the regulations

C. Adequate directional control

D. Adequate stall warning

E. Resistance to spin departure.

There are no closed form Class II methods for assuring satisfactory stall characteristics, with the exception of item E.

The following fundamental characteristics will nor-

mally lead to acceptable stall behavior relative to A-D.

1. Spanwise lift distributions which are conducive to wing root stall instead of to wing tip stall. Read the suggestions on wing design in Chapter 4 of Part III.

2. Wingtip mounted roll control devices when used to 25 percent of maximum available should not produce wing tip stall.

3. The location of the horizontal tail relative to the wing should satisfy the 'no-pitch-up' design criteria of Subsection 5.1.4 in Part III, or if not, the appropriate stick-shaker and stick pusher systems must be installed.

4. The rudder should not be blanked by a stalled wing wake emanating from the wing root or from any other sources of separated flow (such as from nacelles or from leading edge strakes). Read the suggestions on empennage design in Chapter 5 of Part III.

Methods for determining spanwise lift distributions are presented in References 14 and 15.

As for item E, an airplane has been found to be resistant to spin departure as long as its so-called $C_{n_{\beta_{dyn}}}$ is positive:

$$C_{n_{\beta_{dyn}}} = \{C_{n_{\beta_B}} + - (I_{zz_B}/I_{xx_B})C_{l_{\beta_B}} (\tan\alpha)\}\cos\alpha > 0 \qquad (3.38)$$

3.6.4 Step-by-Step Procedure for Analyzing Stall Characteristics

Step 1: Read the applicable regulations, see Table 1.3 and Section 3.6.1.

Step 2: From the regulations determine the flight conditions and airplane configurations for which satisfactory stall characteristics must be shown.

Step 3: For the flight conditions and airplane configurations of Step 2 determine the spanwise

lift distribution for a range of angles of attack. Also determine the spanwise variation of section (= airfoil) maximum lift coefficient. Be sure to account for the effect of Reynold's Number!!

This is readily done with the method of Sub-sub-section 8.1.3.4 (pages 256-257), Pt VI.

1. The span location where tangency occurs between the two spanwise distributions is the span location where the wing will begin to stall. This span location must be close to the wing root for satisfactory stall characteristics.

   If stall occurs close to the tip, the wing design should be changed. Possible design changes are:

   a) addition of twist (or wash-out)
   b) change to airfoils capable of higher maximum lift coefficient values at the appropriate Reynold's Numbers.
   c) a variation of b) is to add leading edge droop to the outboard wing.

2. The angle of attack where tangency of item one first occurs is the value of WING angle of attack for which the stall will occur. Remember that the airplane angle of attack differs from the wing angle of attack by the wing incidence angle!

3. If the decision is made to 'live' with outboard wing stall, a stick-shaker and/or a stick-pusher may have to be installed. In such an event, the value of lift coefficient corresponding to that at which the stick-pusher is triggered must be used as the effective value of wing maximum lift coefficient.

<u>Step 4:</u> Determine the airplane pitch-up characteristics with the guidelines of Subsection 5.1.4 of Part III. If a stick-pusher is required, see item 3.) under Step 3.

<u>Step 5:</u> Determine whether directional control at the stall is available by sketching in the location of the separated wing-root wake at the

tail location of the airplane: see Fig.5.13 in Part III.

<u>Step 6:</u> For the conditions of Step 2, compute the body axis directional and lateral stability derivatives as well as the moments of inertia in Eqn.(3.38) and verify whether the corresponding spin departure resistance criterion is met. If not, introduce the required design changes.

Note: these design changes can take the form of a larger or relocated vertical tail (or tails), a change in wing geometric dihedral and/or a change in mass distribution.

<u>Step 7:</u> Document the results obtained as well as any design changes made.

3.7 SPIN CHARACTERISTICS

3.7.1 Applicable Regulations

Civil: FAR 23.221, see Appendix A.

Military: Mil-F-8785C, Par.3.4.2.2, see Appendix B

3.7.2 Relationship to Preliminary Design

See Chapter 5 in Part III!

3.7.3 Mathematical Model for Analyzing Spin Characteristics

There is no suitable mathematical model for Class II design analysis purposes. A method for predicting aerodynamic forces and moments on a spinning airplane is given in Reference 16.

3.7.4 Step-by-Step Procedure for Analyzing Spin Characteristics

Step 1: Read the applicable regulations, see Table 1.3 and Section 3.7.1.

Step 2: From the regulations determine the flight conditions and airplane configurations for which satisfactory spin characteristics must be shown.

Step 3: Read the sections on aircraft spin and corresponding design guidelines in Stinton, Reference 17, pages 464 - 490. Check the design of the airplane in terms of the 'Stinton' approach and make design adjustments where required.

Step 4: Document the results obtained as well as any design changes made.

3.8 AEROELASTIC CONSIDERATIONS

All stability derivatives used in determining dynamic stability characteristics in this chapter are subject to aeroelastic effects. The method of Section 2.9 can be applied to all stability and control derivatives of Ch.3. Highly elastic airplanes, such as very large transport airplanes, are also subject to acceleration/mass induced aeroelastic effects. Acceleration/mass effects are discussed in detail in Chapter 8 of Reference 12.

4. RIDE AND COMFORT CHARACTERISTICS

The purpose of this chapter is to provide methods and guidelines for the analysis and selection of those design aspects which influence the ride and comfort characteristics of airplanes. These characteristics are important for the following reasons:

1. If ride and comfort are not acceptable to passengers, they are unlikely to return for another flight. This hurts the commercial viability of an airplane!

2. If ride and comfort are not acceptable to crewmembers, they may not be able to carry out their duties thus negatively affecting safety and/or mission effectiveness!

The following aspects of ride and comfort can be important, depending on such factors as time exposure and the requirement to perform duties:

1. Airplane response to atmospheric turbulence:

   High levels of vertical and lateral accelerations are uncomfortable to passengers and may make it difficult (sometimes impossible) for crew members to carry out their duties.

   Additional factors to be considered here are:

   a) distance to the center of gravity
   b) flexible (elastic) behavior of the structure

2. Cabin interior noise and temperature

3. Cabin (or cockpit) interior dimensions

4. Seat dimensions and seat comfort

References 18 through 23 should be consulted for detailed information (and further references) on the importance of these factors to ride and comfort.

<u>Neither</u> military <u>nor</u> civil regulations contain specific numerical design guidelines for the above factors. The design guidelines given in this chapter cover only turbulence induced accelerations as experienced by the crew and by the passengers. These design guidelines,

augmented with the use of 'common sense' and the use of 'standard' design practice in the other areas should suffice during the preliminary design evaluation of a new airplane.

The material in this chapter is organized as follows:

4.1 Relationship to preliminary design
4.2 Mathematical model for analyzing ride characteristics
4.3 Step-by-step procedure for analyzing ride characteristics

4.1 RELATIONSHIP TO PRELIMINARY DESIGN

Selection of wing loading (W/S), wing planform (A, Λ_{le}, λ), fuselage and cockpit interior dimensions and seating arrangement all affect the ride and comfort of passengers and crew members. Having said that, it is sobering to reflect on the following statements:

1. The airplane sizing process described in Part I results in a choice of wing-loading without even considering ride characteristics!

2. The decisions made in P.D. Sequence I, Steps 4 and 6 (Part II) came about without specific reference to comfort and ride!

As a result of the ride analysis and ride criteria presented in Section 4.2, the designer may wish to modify some of these early design decisions. <u>However</u>, the effect of such design changes on performance and/or on stability and control must be assessed!

For some airplane missions, flight through moderate to severe turbulence may be necessary. Examples are:

1. Low altitude attack missions
2. Pipeline spotting missions

In such cases, designing the basic airplane and its wing for reasonable ride quality levels at the crew station, turns out to be not feasible. In such cases it may become necessary to employ a so-called 'ride-control system'. The B-1 bomber has such a ride control system. A schematic of this type of system is shown in Figure 4.1.

Figure 4.3 Example of a Ride Control System for the B-1

Figure 4.2 Effect of Ride Comfort Rating on Percentage of Passengers Satisfied with the Ride

4.2 MATHEMATICAL MODEL FOR ANALYZING RIDE AND COMFORT CHARACTERISTICS

Two ingredients are required for analyzing the ride characteristics of a new design:

1. A ride characteristic rating scale (metric)

2. A formula which relates vehicle design parameters to that metric.

Two models will be presented:

4.2.1 A model for the prediction of ride comfort from a passenger viewpoint (civil and military)

4.2.2 A model for the prediction of ride comfort from a crew station viewpoint (civil or military)

4.2.1 A Model for the Prediction of Ride Comfort from a Passenger Viewpoint (Civil or Military)

The following model was distilled from Reference 18.

The relative ride-comfort will be measured in terms of a ride comfort index, C_{ride} which is related to passenger satisfaction in Figure 4.2. In this section, only the effect of vertical and lateral accelerations on this ride comfort index will be accounted for.

The effect of airplane vertical and lateral accelerations (in response to atmospheric turbulence) on the ride comfort index, C_{ride} may be estimated from:

$$C_{ride} = 2 + 18.9a_{vert} + 12.1a_{lat} \qquad (4.1a)$$

which applies for $a_{vert} > 1.6a_{lat}$, and:

$$C_{ride} = 2 + 1.62a_{vert} + 38.9a_{lat} \qquad (4.1b)$$

which applies for $a_{vert} < 1.6a_{lat}$

where: the accelerations a_{vert} and a_{lat} are measured in relative 'g'-level ('g'/fps of vertical or lateral rms gust). They are measured at the center of

gravity of the airplane and can be found from:

$$a_{vert} = (0.5\rho U_1)(C_{L_\alpha})(\sigma_w)/(W/S) \qquad (4.2)$$

and:

$$a_{lat} = (0.5\rho U_1)(C_{y_\beta})(\sigma_v)/(W/S) \qquad (4.3)$$

where: the derivatives C_{L_α} and C_{y_β} are for the entire airplane (in 1/rad). Part VI contains methods for their estimation. For elastic airplanes, the method of Section 3.8 should be used.

σ_w and σ_v are the rms of gust (or turbulence) velocities induced by the atmosphere. Their values depend on altitude, terrain and probability of occurrence. Figure 4.3 can be used to determine these gust values.

4.2.2 A Model for the Prediction of Ride Comfort from a Crew Station Viewpoint (Civil or Military)

In preliminary design it is acceptable to use the following criterion for the acceptability of turbulence response at the crew station:

$$\bar{A} < 0.005 \qquad (4.4)$$

where: $\bar{A}$ is the 'rms g-level' per fps gust input at the crew station. This gust (or turbulence) response may be computed from:

$$\bar{A} = (0.5\rho U_1 C_{L_\alpha})K_1K_2/(W/S) \qquad (4.5)$$

$$\text{with: } K_1 = 0.66 + (0.39/\bar{c})l_{crew} \qquad (4.6)$$

where: $\bar{c}$ is the wing m.g.c. in ft

l_{crew} is the distance from the crew station to the leading edge of the wing m.g.c., in ft

C_{L_α} is the airplane liftcurve slope in 1/rad. Aeroelastic effects on

FROM: FIG. B7, APPENDIX B

Figure 4.3 Root Mean Square (RMS) Gust or Turbulence Amplitudes as Related to Altitude and Probability of Exceedance

FROM: REF. 25

Figure 4.4 Effect of Gust Length and Mass Parameter on Gust Response Factor

this derivative may have to be accounted for: see Section 3.8.

K_2, the gust response factor, is found from: Figure 4.4.

Notes: 1.) In Figure 4.4, $L_{v \text{ or } w}$ is the so-called gust length. In preliminary design it is acceptable to use: $L_{v \text{ or } w} = 2{,}500$ ft

2. Also in Figure 4.4:

$$K = 4W/(32.2\pi\rho S\bar{c} \qquad (4.7)$$

To give the reader a 'feel' for the meaning of ride criterion (4.4), consider the following. Table 4.1 lists a subjective crew rating of how rms vertical accelerations are perceived by pilots.

If an airplane with an $\bar{A}$ value of 0.005 at sealevel, encounters a rms vertical gust at sealevel of 10 fps (P = 0.001), the crew would feel a 0.05 rms 'g' level.

Table 4.1 Subjective Ratings of Vertical Acceleration

Pilot's Description of Turbulence		RMS Vertical Acceleration in 'g'
Negligible	(Data from:	0.05
Slight	Ref.20)	0.10
Moderate		0.10 - 0.15
Moderately Heavy		0.20 - 0.30
Severe		0.30 - 0.60
Extreme		0.60

4.3 STEP-BY-STEP PROCEDURE FOR ANALYZING RIDE AND COMFORT CHARACTERISTICS

Step 1: For those flight conditions and configurations for which ride qualities are a concern, list:

Weight, Mach Number (and Speed), altitude and configuration (flaps, power, loading, etc.).

Step 2: For the flight conditions of Step 1, determine:

a) the derivatives C_{y_β} and C_{L_α}

Note: include the effect of aeroelasticity

when necessary!

b) the atmospheric density

c) the wing loading, W/S

d) the gust (turbulence) rms values from Figure 4.3. In preliminary design a probability level of 0.01 or 0.001 is usually appropriate.

Step 3: Using Eqns.(4.2) and (4.3) compute the relative accelerations and using Eqn.(4.1) determine the comfort index. Use Figure 4.1 to determine the percentage of satisfaction with the ride.

This percentage of satisfaction should be about 80. If it is less, a design change should be considered.

From Eqns.(4.1)-(4.3) it is evident that the wing loading, W/S and the vertical tail size and shape are the only available design parameters. The real problem is, that these parameters are probably already 'fixed' on the basis of previous design considerations. If that is the case, a ride control system should be considered. For the design of such systems, References 26 and 27 are suggested. Figure 4.1 shows an example of such a system.

Step 4: Determine the additional parameters needed to estimate $\bar{A}$ in Equation (4.4), then compute $\bar{A}$.

If $\bar{A}$ does not meet criterion (4.4) a design change is in order. Most likely, the only practical alternative is a ride control system, such as shown in Figure 4.1. For synthesis methods of such systems, see References 26 and 27.

Step 5: Document the results obtained, including any design changes made.

5. PERFORMANCE

The purpose of this chapter is to present Class II methods for predicting the performance characteristics of airplanes. These methods are compatible with Step 24 in Preliminary Design Sequence II as outlined in Chapter 2 of Part II.

The performance characteristics of airplanes must meet the following requirements:

1) Mission performance requirements

2) Airworthiness performance requirements (performance regulations)

The mission performance requirements are normally dictated by the customer, civil or military. The airworthiness performance requirements (referred to in this text as the REGULATIONS) are set by government agencies such as the FAA (civil) or one of the armed forces: USAF, USNavy, USMC or USArmy (military).

To provide the reader with insight into where a given performance characteristic fits into the airplane preliminary design process, the material in each section is organized to provide the following information:

1) identification of the applicable airworthiness regulation(s)

2) relationship to the performance sizing process of Part I of this series of books

3) presentation of a model for analyzing the performance

4) a step-by-step procedure for determining whether or not an airplane meets the mission and airworthiness performance requirements

An important assumption made in this chapter is that the airplane can be considered to be a so-called point-mass model. For the point-mass model to be valid, the stability and control characteristics of the airplane (as discussed in Chapters 3 and 4) must satisfy the corresponding regulations. In addition, the airplane is assumed to be in complete moment equilibrium: any drag polars used in this chapter MUST be the TRIMMED DRAG POLARS!

The material in this chapter is organized in the following manner:

5.1 Stall
5.2 Takeoff
5.3 Climb
5.4 Cruise, range and payload-range
5.5 Endurance and loiter
5.6 Dive
5.7 Maneuvering
5.8 Descent and glide
5.9 Landing
5.10 Mission profile analysis
5.11 Productivity
5.12 Presentation of airplane performance data

COURTESY: CANADAIR

5.1 STALL

5.1.1 Applicable Regulations

Civil: FAR 23.45, 23.49, FAR 25.101 and 25.103, see Appendix A.

Military: MIL-C-005011B, Par.3.4.2.3 or AS-5263, Par.3.5.2.3, see Appendix B.

5.1.2 Relationship to Preliminary Design

See Part I, Section 3.1 and Chapter 7, Part II.

5.1.3 Mathematical Model for Analyzing Stall

Figure 5.1 depicts the forces acting on the airplane in a 1-g stall in steady, level, symmetrical flight. The following equations of motion apply:

$$T\sin(\alpha + \phi_T) + L = W \tag{5.1}$$

$$T\cos(\alpha + \phi_T) = D \tag{5.2}$$

At the stall, the airplane is flying at:

$C_L = C_{L_{max}}$ and $\alpha = \alpha_{C_{L_{max}}}$. Therefore, the stall speed follows from:

$$V_s = [2\{W - T\sin(\alpha_{C_{L_{max}}} + \phi_T)\}/\{\rho C_{L_{max}} S\}]^{1/2} \tag{5.3}$$

where: W is the weight at which the stall speed is to be determined

T is the thrust setting used in the stall maneuver. This will normally be equal to zero (power-off stall). Since thrust does depend on speed, an iterative solution to Eqn.(5.3) is required if T is not zero.

Note: If the thrust is derived from a propeller instead of from a jet engine, it may be computed from:

$$T = (550\eta_p SHP)/V_s \tag{5.4}$$

where: η_p is the propeller efficiency

Figure 5.1 Forces Acting on an Airplane in a 1-g Stall In Steady, Level symmetrical Flight

Figure 5.2 Example Flight Envelope

SHP is the shaft horsepower delivered by the engine(s)

V_s is the stall speed as found from Equation (5.3). Note that an iterative solution of Eqn.(5.3) is required if SHP is not zero.

$\alpha_{C_{L_{max}}}$ is the angle of attack at the stall

ϕ_T is the thrustline inclination, see Fig.5.1

Therefore, Eqn.(5.3) applies to vectorable thrust airplanes. For most airplanes it is acceptable to assume $\phi_T = 0$.

ρ is the air density corresponding to the altitude at which the stall is to be done

$C_{L_{max}}$ is the trimmed maximum lift coefficient of the airplane. As explained in Ch.2 of this text as well as in Section 8.3 of Part VI, its value depends strongly on center of gravity location AND on controllability and pitch-up behavior.

<u>NOTE WELL:</u> For high speed airplanes the maximum lift coefficient is also dependent on Mach number. This dependency is discussed in Chapter 9 of Part VI.

The dependency of maximum lift coefficient on Mach number is very important in defining the stall boundary of the airplane flight envelope. This is discussed further in Section 5.11.

S is the (reference) wing area

If T = 0 and $\phi_T = 0$, Equation (5.3) reduces to its classical form:

$$V_s = \{(2W)/(\rho C_{L_{max}} S)\}^{1/2} \qquad (5.5)$$

<u>IMPORTANT COMMENT:</u> The 1-g stall speed as determined from this method is somewhat conservative in view of the manner in which the regulations define the stall speed as determined by certification flight tests. The conservatism usually amounts to about 6 percent. For a more dis-

cussion of this, see Reference 11, pages 482-486.

5.1.4 Step-by-Step Procedure for Analyzing Stall

The following step-by-step procedure is suggested to calculate the airplane stall speed.

Step 1: Determine which regulations apply to the design: see Table 1.3 and Sub-section 2.1.1. READ the regulations!

Step 2: Tabulate the flight conditions, configurations, loading conditions and c.g. locations for which the stall speed of the airplane must be determined. This information comes from the regulations AND from the mission performance requirements. The reader should consult the list of items 1-9 on page 7!

Step 3: For the flight conditions and configurations defined in Step 2, determine the value(s) of the maximum lift coefficient. Be sure to account for the effects of Mach number and c.g. location on $C_{L_{max}}$!

Step 4: For the conditions of Step 2 find the values of all other terms in Equation 5.3.

Step 5: Compute the stall speed(s) from Eqn.(5.3), and record in kts, fps and in Mach number.

Step 6: Check the stall speed(s) of Step 5 against the regulatory requirements and against the mission performance requirements. If a discrepancy is found, determine what (if any) redesign action needs to be taken.

Discrepancies occur usually as a result of not being able to generate the required magnitude of $C_{L_{max}}$ (TRIMMED!!). This in turn can be remedied usually by a change in wing and/or flap design!

Step 7: Plot the variation of Mach number at the stall with altitude. This constitutes the left side of the airplane operational flight envelope. Figure 5.2 shows an example.

Step 8: Document the results obtained, including any design changes made.

5.2 TAKE-OFF

5.2.1 Applicable Regulations

Civil: FAR 23.45, 23.51, FAR 25.101, and 25.107, see Appendix A.

Military: MIL-C-005011B, par. 3.4.2.4 and 3.4.5 or AS-5263, par. 3.5.2.4, 3.5.2.5 and 3.5.5, see Appendix B.

5.2.2 Relationship to Preliminary Design

See Part I, Section 3.2.

5.2.3 Mathematical Model for Analyzing Takeoff Performance

The definition of take-off distance and the associated reference speeds depends on which regulation is used to certify the airplane: FAR 23, FAR 25 or Military.

Figure 5.3 shows the differences in the definitions for takeoff distances and the associated reference speeds.

The following method for computing the takeoff distance, s_{TO} is due to Torenbeek (Reference 26). The take-off distance, s_{TO} equals the takeoff field length, s_{TOFL} when the landing distance, s_L or s_{FL} (See Section 5.9) is less than or equal to s_{TO}.

$$s_{TO} = f_{TO}h_{TO}\left[(1/\gamma_{LOF}) + \frac{(V_3/V_{s_{TO}})^2(W/S)_{TO}\{(\bar{T}/W)_{TO} - \mu')^{-1} + 1.414\}}{(h_{TO}\rho g C_{L_{max_{TO}}})(1 + 1.414\gamma_{LOF})}\right] \quad (5.6)$$

where: f_{TO} depends on the obstacle height: see Table 5.1.

h_{TO} is the obstacle height, see Table 5.1.

$(V_3/V_{s_{TO}})$ is the ratio of the speed at the obstacle height, V_3 to the stall speed in

Definition of FAR 23 Take-off Distance

Definition of FAR 25 Take-off Distance

Definition of Military Takeoff Distance

Figure 5.3 Definition of Takeoff Distances According to the Regulations

the takeoff configuration, $V_{s_{TO}}$. This ratio is defined in Table 5.1.

Table 5.1 Parameter Values for Equation (5.6)

Regulation	$V_3/V_{s_{TO}}$	f_{TO}	h_{TO}
FAR 23	1.3	1.0	50 ft
FAR 25	1.25 to 1.3 (no requirement)	1.15	35 ft
AS-5263	1.2	1.0	50
MIL-C-005011B	1.15	1.0	50 ft

$(W/S)_{TO}$ is the takeoff wing loading

$(\bar{T}/W)_{TO}$ is the mean thrust-to-weight ratio taken at a speed of $0.707V_{LOF}$. The value of T may be computed from:

For Jet Driven Airplanes:

$$\bar{T} = 0.75T_{TO}(5 + \lambda)/(4 + \lambda) \qquad (5.7)$$

where: $T_{TO_{max}}$ is the maximum static thrust at takeoff

λ is the engine bypass ratio

For Propeller Driven Airplanes:

$$\bar{T} = 5.75P_{TO}((\sigma ND_p^2)/P_{TO})^{1/3} \qquad (5.8)$$

where: P_{TO} is the maximum shaft horsepower at takeoff (static), all engines operating

$(P_{TO}/ND_p^2$ is the propeller diskloading at takeoff (static). Its Class I determined value (Step 5.6 in Chapter 5 of Part II) may be checked for acceptability by referring to Figure 5.4.

NOTE:
FACTOR 5.75 IN EQN. (5.8) APPLIES TO VARIABLE PITCH PROPELLERS ONLY! FOR FIXED PITCH USE 4.60.

Figure 5.4 Method for Selecting Propeller Disk Loading During Preliminary Design

COURTESY: TORENBEEK, REF. 28

Figure 5.5 Sketch of the Meaning of Balanced Field Length

N is the number of engines

$$\mu' = \mu_g + 0.72(C_{D_o}/C_{L_{max_{TO}}}) \quad (5.9)$$

where: μ_g is the friction coefficient as determined from page 40

C_{D_o} is the zero-lift drag coefficient in in the takeoff configuration

$C_{L_{max_{TO}}}$ is the maximum lift coefficient in the takeoff configuration. It is determined from Part VI, Section 8.3. Ground effect must be accounted for.

$$\gamma_{LOF} = [(T - D)/W]_{LOF} \quad (5.10)$$

but this may be approximated by:

$$\gamma_{LOF} = 0.9(\bar{T}/W)_{TO} - 0.3/(A^{1/2}) \quad (5.11)$$

In the design of commercial transports and certain military transports the concept of balanced fieldlength, (BFL) is often used. In the case of military airplanes, BFL is normally called the Critical Field Length.

The definition of balanced fieldlength is illustrated in Figure 5.5. Torenbeek (Reference 28) shows that:

$$BFL = [(655/(\sigma)^{1/2})] + \quad (5.12)$$

$$+ [0.863/(1 + 2.3(\gamma_2 - \gamma_{2_{min}}))]*$$

$$*[(W/S)_{TO}/(0.694\rho g C_{L_{max_{TO}}}) + h_{TO}][1/((\bar{T}/W_{TO}) - \mu') + 2.7]$$

$$\text{where: } \gamma_2 = (T/W)_{TO_{OEI}} - [(C_L/C_D)_{TO_{OEI}}]^{-1} \quad (5.13)$$

which is called the second-segment climb gradient with one engine inoperative.

with: $(T/W)_{TO_{OEI}}$ is the thrust-to-weight ratio in the takeoff configuration, but with one engine inoperative and at $1.2V_{s_{TO}}$.

$((C_L/C_D)_{TO_{OEI}})^{-1}$ is the lift-to-drag ratio in the takeoff configuration, with one engine inoperative (this does cause extra drag!!) and at $1.2V_{s_{TO}}$

γ_{2min} = 0.024 for N=2

0.027 for N=3

0.030 for N=4

All other parameters were previously defined.

In several instances a takeoff specification will call merely for a groundrun during takeoff to be less than some specified value. In such cases the following equation for the groundrun only is useful:

$$s_{TOG} = [(V_{LOF})^2/2g]/[(\bar{T}/W)_{TO} - \mu'] \qquad (5.14)$$

where: $V_{LOF} = 1.2V_{s_{TO}}$ for commercial airplanes, and

$1.1V_{s_{TO}}$ for military airplanes

all other parameters have been previously defined.

<u>Important Note:</u>

Airplanes which are also carrier based must be compatible with the performance restrictions inherent in each catapult system. A suitable mathematical model for determining airplane compatibility with USNavy catapult systems is given in Sub-sub-section 3.2.5.2, page 103, in Part I.

5.2.4 Step-by-Step Procedure for Analyzing Takeoff Performance

The following step-by-step procedure is suggested for determining the takeoff distance of a new airplane:

Step 1: Determine which regulation applies to the design and read that regulation. See Table 1.3 and Subsection 2.1.1.

Determine the takeoff distance requirements from the mission specification.

Step 2: Determine whether Eqn.(5.6), Eqn.(5.12) or Eqn.(5.14) govern the takeoff distance calculation and prepare the required input data.

If the airplane has a catapulting requirement, such as carrier based airplanes, the reader should use the method of Sub-sub-section 3.2.5.2 of page 103 in Part I to determine the compatibility of the airplane with catapult performance capability.

Step 3: Compute the takeoff distance with either Eqn. (5.6), (5.12) or (5.14). Compare the result with the mission fieldlength requirements. If there is more than a 5 percent discrepancy, a design adjustment to the airplane is in order.

Note: The parameters which affect the takeoff distance most strongly are:

1. The takeoff thrust-to-weight ratio $(T/W)_{TO}$
2. The takeoff wing loading, $(W/S)_{TO}$
3. The takeoff maximum lift coefficient, $C_{L_{max_{TO}}}$

Remember: there is little value in significantly exceeding a mission requirement AND:

not meeting a mission requirement may lead to rejection of the design

Step 4: Document the results obtained, including any design changes made.

5.3 CLIMB

5.3.1 Applicable Regulations

Civil: FAR 23.45, 23.65, 23.67, and 23.77, see Appendix A.

FAR 25.101, 25.111, 25.115, 25.117, 25.119, 25.121 and 25.123, see Appendix A.

Military: MIL-C-005011B, Par.3.4.2.4 and 3.4.2.5 or AS-5263, Par.3.5.2.4.1 and 3.5.2.6, see Appendix B.

5.3.2 Relationship to Preliminary Design

See Part I, Chapter 3, Section 3.4. In addition, see Part II, Chapter 2, Step 14 and Step 27.

5.3.3 Mathematical Model for Analyzing Climb Performance

Figure 5.6 depicts the forces which act on an airplane in an accelerated, symmetrical flight condition. The corresponding equations of motion are:

$$T\cos(\alpha + \phi_T) - C_D\bar{q}S - W\sin\gamma = (W/g)\dot{U} \qquad (5.15)$$

$$T\sin(\alpha + \phi_T) + C_L\bar{q}S - W\cos\gamma = (W/g)U\dot{\gamma} \qquad (5.16)$$

The reader is reminded of the fact, that these equations assume that moment equilibrium exists. This implies, that the drag polar, which defines the relation between C_L and C_D is the 'trimmed' drag polar. See Part VI, Section 4.10 for a discussion of trim drag.

Equations (5.15) and (5.16) are used to determine the rate of climb and the climb gradient of which an airplane is capable in a range of flight conditions.

In preliminary design, the effect of acceleration normal to the flight path as expressed by the $\dot{\gamma}$-term will be neglected. Furthermore, other than for fighters, the climb angle, γ is usually smaller than about 15 degrees so that $\cos\gamma=1.0$ and $\sin\gamma=\gamma$ can be used.

The rate-of-climb, RC (for any flight path angle) is

Figure 5.6 Forces Acting on an Airplane in Accelerated, Symmetrical Flight

ACTUAL

LINEARIZED

Figure 5.7 Variations of Rate-of-Climb with Altitude

defined as:

$$RC = dh/dt = U\sin\gamma \tag{5.17}$$

With the help of Eqns.(5.16) - (5.17) it is shown in Reference 11, page 384, that:

$$RC = \{(T - D)U/W\}/\{1 + (U/g)dU/dh\} \tag{5.18}$$

The term $(U/g)dU/dh$ is called the acceleration factor. It has a significant effect on fighter performance. It also has a significant effect on climb performance at high altitude. Since most regulations which deal with climb performance apply close to the ground, it is usually acceptable to consider only climb at constant true airspeed, in which case the acceleration factor is zero.

Therefore, for steady, symmetrical flight the rate-of-climb is:

$$RC = (T - D)U_1/W \tag{5.19}$$

From this, the climb gradient, CGR follows as:

$$CGR = RC/U_1 = (T - D)/W \tag{5.20}$$

The regulations specify either a minimum required rate-of-climb, RC or a minimum required climb gradient, CGR. The reader should read the applicable regulations to determine which values for CR and/or CGR apply to a given airplane.

For Class II climb performance analysis it is acceptable to use the equations of Chapter 3, Part I. These equations will be stated here in general. The reader must realize that the parameters in these equations are different for each particular flight condition and for each particular airplane configuration!

<u>1. For propeller-driven airplanes:</u>

$$RC\ (fpm) = 33{,}000[\eta_p/(W/P) + \tag{5.21}$$

$$- \{(W/S)^{1/2}/19(C_L^{3/2}/C_D)\sigma^{1/2}\}]$$

$$CGR = -(L/D)^{-1} + \tag{5.22}$$

$$+ \{(C_L)^{1/2}18.97\eta_p\sigma^{1/2}\}/\{(W/P)(W/S)^{1/2}\}$$

2. For jet-driven airplanes:

$$RC\ (fpm) = 60U_1[(T/W) - (L/D)^{-1}] \tag{5.23}$$

$$CGR = (T/W) - (L/D)^{-1} \tag{5.24}$$

Many airplane mission specifications require mission specific climb capabilities such as:

1. Minimum climb rate at some altitude (such as sea level)
2. Minimum time-to-climb to some altitude
3. Ceiling
4. Specific excess power

1. Minimum climb rate at some altitude

The climb rate at any altitude can be estimated with Equations (5.21) and (5.23).

2. Minimum time-to-climb to some altitude

The time to climb to any altitude can be estimated from:

$$t_{cl} = \int_0^h (1/RC)dh \tag{5.25}$$

This equation can be solved by integration over several increments of altitude. Increments of 5,000 ft provide sufficient accuracy in preliminary design. The average value of the rate-of-climb, RC in Eqn.(5.25) at some altitude increment can be estimated from Eqn.(5.21) or (5.23). Figure 5.7 shows typical variations of rate-of-climb with altitude.

If an estimate is available for the absolute ceiling of an airplane (see next item 3.), AND if the RC varies approximately linearly with altitude, then it is possible to find the time-to-climb to any altitude between sealevel and the absolute ceiling from:

$$t_{cl} = (h_{abs}/RC_o)\ln((1 - h/h_{abs})^{-1} \tag{5.26}$$

where: RC_o is the maximum rate-of-climb at sealevel. found from either Equation (5.21) or (5.23). Figure 5.7 also shows the linearized variation of rate-of-climb with altitude.

3. Ceiling

Equation (5.21) or (5.23) can be used to find that altitude for which the rate of climb corresponds to one of the ceiling definitions of Table 5.2. The ceiling is the upper side of the flight envelope: see Figure 5.2.

4. Specific excess power

The specific excess power of an airplane is computed from:

$$P_s = (T - D)U/W \qquad (5.27)$$

Note that this equation is similar to Eqn.(5.18).

More accurate and specialized mathematical models for the computation of climb performance are presented in Reference 11. They will not be repeated here.

To help the reader find specific models for the computation of climb performance, the following guide is presented:

1. For Propeller Driven Airplanes:

   See Reference 11, Section 9.2, pages 384-395.

2. For Jet Powered Airplanes:

   See Reference 11, Section 9.3, pages 395-400.

3. For Fighters (Steep Climbs):

   See Reference 11, Section 9.4, pages 400-401.

4. For Ceilings and Time-to-Climb:

   See Reference 11, pages 405-407

5. For Specific Excess Power:

   See Reference 11, pages 510-516.

Table 5.2 Definition of Airplane Ceilings

Ceiling Type	Minimum Required Climb Rate
Absolute ceiling	0 fpm
Service ceiling	
Commercial/Piston-propeller	100 fpm
Commercial/jet	500 fpm
Military at maximum power	100 fpm
Combat ceiling	
Military/Subsonic/maximum power	500 fpm at M<1
Military/Supersonic/maximum power	1,000 fpm at M>1
Cruise ceiling	
Military/Subsonic/max.cont. power	500 fpm at M<1
Military/Supersonic/max.cont. power	1,000 fpm at M>1

Table 5.3 Summary of Regulatory Climb Requirements for FAR 25 Transports

PHASE OF FLIGHT		AIRPLANE CONFIGURATION				FLIGHT SPEED	MINIMUM, CLIMB GRADIENT, %		
		FLAPS	U.C.	ENGINES			N=1	N=2	N=3
LIFTOFF	1ST SEGMENT	TAKE OFF	+	ONE ENGINE OUT	TAKE OFF	LIFT-OFF	0	.3	.5
TAKEOFF FLIGHT PATH	2ND SEGMENT	TAKE OFF	+	ONE ENGINE OUT	TAKE OFF	V_2	2.4	2.7	3.0
	FINAL TAKEOFF	EN ROUTE	+	ONE ENGINE OUT	MAX. CONT.	$\geq 1.25V_s$	1.2	1.5	1.7
APPROACH CLIMB		APPR.	+	ONE ENGINE OUT	TAKEOFF	$\leq 1.5V_s$	2.1	2.4	2.7
LANDING CLIMB		LAND.	+	ALL ENGINES TAKEOFF		$\leq 1.3V_s$	3.2	3.2	3.2

COURTESY: TORENBEEK, REF. 28

V_s = Stalling speed

V_2 = Takeoff safety speed ($\geq 1.2V_s$)

* Summary only

5.3.4 Step-by-Step Procedure for Determining Climb Performance

Step 1: Determine which regulations apply to the design and read that regulation.

Tabulate the regulatory climb requirements in a manner similar to that shown in Table 5.3.

Determine the climb performance requirements stated in the airplane mission requirements.

Tabulate all mission climb requirements in a manner similar to Table 5.3.

Note from Table 5.3 that the prescribed configuration of the airplane is carefully listed! This has significant consequences for the drag polars!

Step 2: Using the guidelines of Sub-section 5.3.3 determine which methods and/or equations apply to the analysis of the climb requirements of Step 1.

Prepare the necessary input data for all flight conditions and airplane configurations for which climb performance must be assessed.

Step 3: Carry out the climb performance calculations and compare the results against the regulations and/or the mission climb requirements. Note any discrepancies.

Step 4: Analyze what changes are required to eliminate the discrepancies in climb performance noted in Step 3.

Typical changes which may be contemplated are:

a) Increase or decrease in thrust or power

b) Change in aerodynamic design to lower the drag

Note that major changes in these areas may have repercussions for most other aspects of the design and may therefore require an iteration!

Step 5: Document the results, including any design changes made.

5.4 CRUISE, RANGE AND PAYLOAD-RANGE

5.4.1 Applicable Regulations

Civil: There are no airworthiness regulations which deal specifically with cruise performance. Fuel reserve regulations are specified in FAR 91. Most commercial operations are conducted with fuel reserve rules which are more conservative than those of FAR 91.

Military: MIL-C-005011C (USAF) and AS-5263 (USN and USMC), give definitions, see Appendix B.

5.4.2 Relationship to Preliminary Design

See Part I, Chapter 2. In addition, see Part II, Chapter 2, Step 14, Step 27 and Step 28.

5.4.3 Mathematical Model for Analyzing Cruise and Range Performance

Figure 5.8 depicts the forces which act on an airplane in a horizontal, steady, symmetrical, 1-g cruise flight condition. The equations of motion are:

$$T\cos(\alpha + \phi_T) - C_D\bar{q}S = 0 \qquad (5.28)$$

$$T\sin(\alpha + \phi_T) + C_L\bar{q}S = W \qquad (5.29)$$

The reader is reminded of the fact, that these equations assume that moment equilibrium exists. This implies, that the airplane drag polar, which defines the relationship between C_L and C_D is the 'trimmed' drag polar. Part VI, Section 4.10 presents a method for computing trim drag.

There also exists a unique relationship between the angle of attack, α and the lift coefficient, C_L. A method for determining C_L versus α is given in Part VI, Chapter 8.

The quantity T in Equations (5.28) and (5.29) is the so-called <u>installed thrust.</u> Methods for determining the installed thrust capability of an airplane are presented in Part VI, Chapter 6.

For most conventional airplanes, the angle $(\alpha + \phi_T)$

Figure 5.8 Forces Acting on an Airplane in Level, Steady, Symmetrical Flight

Figure 5.9 Example Mission Profiles for a Transport and for a Fighter

Figure 5.10 Determination of Cruise Speed and Maximum Level Flight Speed

is so small that it can be neglected. In that case, the cruise lift coefficient follows from:

$$C_{L_{cr(uise)}} = W/\bar{q}S \tag{5.30}$$

The required cruise thrust-to-weight ratio then is:

$$(T/W)_{cr} = 1/(L/D)_{cr} \tag{5.31}$$

If $(\alpha + \phi_T)$ is not negligibly small, Eqn.(5.30) is used to find a first approximation for the cruise lift coefficient. This value of C_L is then used to find C_D from the trimmed drag polar and also to find α from the C_L versus α curve. Since the thrust inclination angle, ϕ_T is assumed to be known, Eqn.(5.28) can now be used to find the required installed thrust. This thrust value must be less than the maximum installed thrust available in that flight condition! This thrust value is then substituted into Eqn.(5.29) and a new value for C_L is determined, etc., etc. until the process converges. Convergence may be assumed to have been reached when the lift-to-drag ratio is within 1 percent. This determines the cruise lift-to-drag ratio, $(L/D)_{cr}$.

IMPORTANT NOTE: The reader will find that in nearly all cruise flight conditions:

$$(L/D)_{cr} < (L/D)_{max} \tag{5.32}$$

In a 'cruise matched' airplane, the cruise lift-to-drag ratio is roughly 90 percent of the maximum lift-to-drag ratio.

Most airplane missions specify cruise requirements in the following manner:

1. Cruise speed and cruise altitude

2. Range and payload (Requirement for a specific Range-Payload Diagram)

3. Fuel reserves at the end of the design mission as a fraction of fuel used up to that point or in terms of a so-called reserve mission.

4. Maximum cruise speed, altitude and payload (some of which may be external)

5. Payload expended, as in military airplanes which expend ammunition and release weapons and stores.

It is always useful to translate airplane mission requirements into a mission profile. Figure 5.9 shows examples for a civil and for a military airplane.

For Class II analysis of cruise range performance it is acceptable to use the following Breguet equations, PROVIDED the procedure of Step 5 in Sub-section 5.4.4 is used:

<u>For Propeller Driven Airplanes:</u>

For constant altitude cruise:

$$R = f_{mp}(\eta_p/c_p)(L/D)\ln(W_{initial}/W_{end}) \qquad (5.33)$$

Note that: when f_{mp} = 326, R is in nautical miles!

when f_{mp} = 375, R is in statue miles!

For constant speed cruise:

$$R = f_{mp}(\eta_p/c_p)(L/D)\ln(W_{initial}/W_{end}) \qquad (5.34)$$

Note that this equation is identical to Eqn.(5.33)! This works only if the assumption is made that the airplane cruises roughly at the same values of η_p, c_p and L/D, regardless of the type of cruise. That assumption is usually valid.

<u>For Jet Driven Airplanes:</u>

For constant altitude cruise:

$$R = (f_{mj}/c_j)(\rho S)^{-1/2}\{(C_L)^{1/2}/C_D\}\{(W_{initial})^{1/2} - (W_{end})^{1/2}\} \qquad (5.35)$$

Note that: when f_{mj} = 1.677, R is in nautical miles!

when f_{mj} = 1.929, R is in statute miles!

For constant speed cruise:

$$R = (V/c_j)(L/D)\ln(W_{initial}/W_{end}) \quad (5.36)$$

Note that if V is expressed in sm/hr, R is in sm, but, if V is expressed in nm/hr, R is in nm!

In many missions (see Figure 5.91) range credit may be taken for the climb and/or for the descent part of a mission. If that is the case, the following equations may be used to determine these range increments:

$$R_{cl} = V_{cl}t_{cl} \quad (5.37)$$

where: V_{cl} is the speed at which the climb is conducted. For high performance airplanes, 250 kts is a good guess!

$$t_{cl} = h_{cl}/RC_{ave} \quad (5.38)$$

with: RC_{ave} being determined by one of the climb equations in Section 5.3.

For descent range, Equations (5.37) and (5.38) must be changed to reflect the appropriate descent terms.

The construction of the payload-range diagram for airplanes is discussed in Sub-section 5.4.4.

If part of a mission profile requires a given cruise speed or a given maximum speed, this can be determined graphically from Figure 5.10. At a given thrust (or power) setting the corresponding speed is found from the intersection of the thrust (or power) available and the thrust (or power) required curves. Maximum speed corresponds to the right hand side of the flight envelope as shown in Figure 5.2.

More detailed methods for determining cruise range and cruise speed are presented in Ref.11, Chapter 11. These will not be repeated here. To help the reader find specific models for the computation of range performance, the following guide is presented:

For Propeller Driven Airplanes:

See Reference 11, Section 11.1, pages 454-464.

For Jet Driven Airplanes:

See Reference 11, Section 11.2, pages 464-478.

5.4.4 Step-by-Step Procedure for Analyzing Cruise and Range Performance

Step 1: From the mission requirements of the airplane determine the cruise and range requirements for the airplane. This is most readily done with the help of a mission profile which can be constructed from the airplane mission specification.

Figure 5.9 shows examples of mission profiles for a transport and for a fighter, with the cruise portions indicated.

Tabulate all cruise altitude, cruise speed, range and payload requirements. Also note the configuration the airplane is supposed to be in! For military airplanes, external stores and weapons can have a major effect on the drag polar!

NOTE: Certain military airplanes are required to fly several cruise segments under different speed and altitude conditions!

Step 2: For all flight conditions and configurations defined in Step 1, determine the airplane trimmed drag polars. This can be done with the method of Section 4.10 in Part VI.

Step 3: Using installed engine data, obtain the numerical values for the engine efficiency parameters: c_j (for jets) and/or c_p and η_p (for props).

These parameters depend on speed and altitude which are normally prescribed in the mission requirements.

Chapter 6 in Part VI contains methods for computing installed thrust and/or power characteristics, including efficiencies and s.f.c.'s.

Step 4: Determine the range capability of the airplane in each of the range portions of the mission profile. This is done as follows:

Decide whether the range is to be at constant altitude or at constant speed. Identify which of the Breguet equations (5.33)-(5.36) apply.

For airplanes with short ranges (about 500 nm) it is reasonable to assume that all parameters in

the Breguet range equations are constant during the intended cruise operation. For airplanes with medium to long ranges this is not acceptable. For such airplanes it is suggested to break the cruise part of the mission into segments of approximately 500 nm each. The parameters in the Breguet range equations can be assumed constant but different for each cruise segment!

Now proceed as follows:

For the first range segment, estimate the average cruise lift coefficient from:

$$C_{L_{cr_1}} = (W_{begin} - 0.5W_{fuel_1})/\bar{q}_1 S \qquad (5.39)$$

where for propeller driven airplanes:

$$W_{fuel_1} = (R_1/V_{cr_1})\{(c_p/\eta_p)P_{reqd}\}_1 \qquad (5.40)$$

and where for jet driven airplanes:

$$W_{fuel_1} = (R_1/V_{cr_1})\{c_j W/(L/D)\}_1 \qquad (5.41)$$

Note: the subscript '1' here means that the particular quantity applies to cruise segment 1.

The average value of cruise lift coefficient (in Eqn.(5.39)) is entered into the trimmed drag polar to determine the corresponding value of the drag coefficient. This information in turn is used to compute the values of (L/D) and/or of $\{(C_L)^{1/2}/C_D\}$ which are needed in the Breguet range Eqns.(5.33)-(5.36).

This procedure is repeated as many times as needed until the required total range is met or until the fuel available is exhausted.

<u>Step 5:</u> Note any discrepancies between the computed range and the required range. Decide what design changes (if any) must be made.

Possible design changes which can be contemplated at this stage are:

1. Improve L/D by lowering wetted area or increasing the wing loading and/or the aspect ratio.

2. Switch to engines with lower fuel consumption.

3. Switch to a more efficient propeller.

4. Try to design a more efficient structure so that the fuel-to-weight ratio can be improved.

5. Carry more or less fuel, as required.

Any of these design changes may have repercussions for other areas of the design. Design iterations may have to be performed.

<u>Step 6:</u> Prepare a payload range diagram.

This step normally applies only to transport type airplanes. Figure (5.11) shows an example of a payload-range diagram. The important points A, B, C and D are indicated. These points can be determined in the following manner:

Point A: At this point the airplane weight is equal to the sum of the operating weight empty, (W_{OE}) and the maximum payload weight, W_{PL}. There is no fuel on board and thus the range is zero.

Point B: At this point the range is that for maximum payload. This is sometimes called the harmonic range of the airplane. The weight of the airplane at takeoff is equal to the sum of the operating weight empty, W_{OE}, the maximum payload weight, W_{PL} and that fuel weight which causes the airplane to be at its maximum allowable takeoff weight.

The range corresponding to point B is computed with the procedure of Step 4.

Point C: Between points B and C, payload has to be traded for fuel. This can be done until the maximum volumetric capacity (in terms of fuel) of the airplane has been reached. That is the case at point C. The takeoff weight of the airplane is still at its maximum allowable value.

The range corresponding to point C is computed

with the procedure of Step 4. Make sure that the payload weight is computed correctly!

Note: fuel volume limits are normally reached first due to wing fuel volume limitations. In many airplanes additional fuel can be stored in 'safe' parts of the fuselage, in horizontal and /or vertical tails and sometimes in external containers.

Point D: Beyond point C, the only way to get more range out of the airplane is to unload payload weight. At point D, the takeoff weight of the airplane is the sum of its operating weight empty, W_{OE}

and its maximum fuel capability (on basis of volumetric capacity). The range corresponding to point D is called the ferry range.

The range corresponding to point D is computed with the procedure of Step 4. Make sure that NO payload is included and that the takeoff weight is no more than the maximum allowable weight. If there is no fuel volume limit, then the maximum allowable takeoff weight is the limit.

<u>Step 7:</u> Document all the results, including any design changes made.

Figure 5.11 Example of a Payload-Range Diagram

5.5 ENDURANCE AND LOITER

5.5.1 Applicable Regulations

Civil: There are no airworthiness regulations which deal specifically with endurance and loiter. Fuel reserve regulations are specified in FAR 91. Most commercial operations are conducted with fuel reserve rules which are more conservative than those of FAR 91.

Military: MIL-C-005011C (USAF) and AS-5263 (USN and USMC), give definitions, see Appendix B.

5.5.2 Relationship to Preliminary Design

See Part I, Chapter 2. In addition, see Part II, Chapter 2, Step 14, Step 27 and Step 28.

5.5.3 Mathematical Model for Analyzing Endurance and Loiter

Figure 5.8 depicts the forces which act on an airplane in a horizontal, steady, symmetrical, 1-g loiter flight condition. The equations of motion are the same as those for range in Sub-section 5.4.3: Eqns.(5.28) and (5.29).

The reader is reminded of the fact, that these equations assume that moment equilibrium exists. This implies, that the airplane drag polar, which defines the relationship between C_L and C_D is the 'trimmed' drag polar. Part VI, Section 4.10 presents a method for computing trim drag.

There also exists a unique relationship between the angle of attack, α and the lift coefficient, C_L. A method for determining C_L versus α is given in Part VI, Chapter 8.

The quantity T in Equations (5.28) and (5.29) is the so-called <u>installed thrust.</u> Methods for determining the installed thrust capability of an airplane are presented in Part VI, Chapter 6.

For most conventional airplanes, the angle $(\alpha + \phi_T)$ is so small that it can be neglected. This, despite the fact that endurance or loitering flights are normally carried out at fairly low speeds and thus at relatively

high angles of attack. The lift coefficient for endurance or loiter can be computed from:

$$C_{L_{ltr}} = W/\bar{q}S \tag{5.42}$$

where: W is that weight value appropriate to the endurance/loiter flight condition being analyzed.

The required loiter thrust-to-weight ratio then is:

$$(T/W)_{ltr} = 1/(L/D)_{ltr} \tag{5.43}$$

If $(\alpha + \phi_T)$ is not negligibly small, Eqn.(5.42) is used to find a first approximation for the loiter lift coefficient. This value of C_L is then used to find C_D from the trimmed drag polar and also to find α from the C_L versus α curve. Since the thrust inclination angle, ϕ_T is assumed to be known, Eqn.(5.28) can now be used to find the required installed thrust. This thrust value <u>must be less than the maximum installed thrust available</u> in that flight condition! This thrust value is then substituted into Eqn.(5.29) and a new value for C_L is determined, etc., etc. until the process converges. Convergence may be assumed to have been reached when the lift-to-drag ratio is within 1 percent. This determines the loiter lift-to-drag ratio, $(L/D)_{ltr}$.

<u>IMPORTANT NOTE:</u> The reader will find that in nearly all loiter flight conditions:

$$(L/D)_{ltr} \approx (L/D)_{max} \tag{5.44}$$

Most airplane missions specify loiter or endurance requirements in the following manner:

1. Loiter speed and loiter altitude.

   In high altitude observation airplanes it is essential that the loiter speed be at least equal to that of prevailing winds at loiter altitude.

   In civil airplanes there normally is a requirement to 'hold' at the end of a cruise mission. This is in fact a requirement to loiter. The

loiter speed must be compatible with air traffic control requirements.

In military airplanes there often is a requirement to loiter over a given location to wait for attack or engagement instructions.

2. Payload to be carried while loitering.

It is always useful to translate airplane mission requirements and into a mission profile. Figure 5.9 shows examples for a civil and for a military airplane.

For Class II analysis of loiter/endurance performance it is acceptable to use the following Breguet equations PROVIDED the procedure of Step 4 in Sub-section 5.5.4 is used:

<u>For Propeller Driven Airplanes:</u>

For constant altitude endurance/loiter:

$$E = 778(\eta_p/c_p)(\rho S)^{1/2}\{(C_L)^{3/2}/C_D\}*$$

$$*[(W_{end})^{-1/2} - (W_{initial})^{-1/2}] \qquad (5.45)$$

Note that E is in hours!

For constant speed endurance/loiter:

$$E = 928(\eta_p/c_p)(1/V)(L/D)\ln(W_{initial}/W_{end}) \qquad (5.46)$$

Note that E is in hours and V is in kts!

<u>For Jet Driven Airplanes:</u>

For constant altitude endurance/loiter:

$$E = (1/c_j)(L/D)\ln(W_{initial}/W_{end}) \qquad (5.47)$$

Note that E is in hours!

For constant speed endurance/loiter:

$$E = (1/c_j)(L/D)\ln(W_{initial}/W_{end} \qquad (5.48)$$

Note that this equation is identical to Eqn.(5.47)!

If part of a mission profile requires a given loiter speed, this can be determined graphically from Figure 5.10. At a given thrust (or power) setting the corresponding speed is found from the intersection of the thrust (or power) available and the thrust (or power) required curves. Note that loiter usually represents the low speed intersection while cruise was represented by the high speed intersection.

More detailed methods for determining endurance/loiter performance are presented in Ref.11, Chapter 11. These will not be repeated here. To help the reader find specific models for the computation of range performance, the following guide is presented:

For Propeller Driven Airplanes:

See Reference 11, Section 11.1, pages 454-464.

For Jet Driven Airplanes:

See Reference 11, Section 11.2, pages 464-478.

5.5.4 Step-by-Step Procedure for Analyzing Endurance and Loiter

Step 1: From the mission requirements of the airplane determine the endurance and/or loiter requirements. This is most readily done with the help of a mission profile which can be constructed from the airplane mission specification.

Figure 5.9 shows examples of mission profiles for a transport and for a fighter. In Figure 5.9 the loiter requirements are items 6.

Tabulate all endurance/loiter altitude, speed and payload requirements. Also note the configuration the airplane is supposed to be in! For military airplanes, external stores and weapons can have a major effect on the drag polar!

Step 2: For all flight conditions and configurations defined in Step 1, determine the airplane trimmed drag polars. This may be done with the method of Section 4.10 in Part VI.

Step 3: Using installed engine data, obtain the numerical values for the engine efficiency parameters: c_j (for jets) and/or c_p and η_p (for props).

These parameters depend on speed and altitude which are normally prescribed in the mission specification.

Methods for determining installed engine characteristics are given in Chapter 6 of Part VI.

Step 4: Determine the endurance and/or loiter capability of the airplane for each of the endurance or loiter portions of the mission profile. This is done as follows:

Decide whether the endurance and/or loiter is to be performed at constant altitude or at constant speed. Identify which of the Breguet equations (5.45)-(5.48) apply.

For airplanes with short endurance or loiter segments (approximately 0.5 hours) it is reasonable to assume that all parameters in the Breguet equations are constant during the intended opera-

tion. For airplanes with longer endurance or loiter requirements this is not acceptable. For such airplanes it is suggested to break the endurance or loiter part of the mission into segments of approximately 0.5 hours each. The parameters in the Breguet endurance/loiter equations can be assumed constant but different for each segment!

Now proceed as follows:

For the first endurance/loiter segment, estimate the average loiter lift coefficient from:

$$C_{L_{ltr_1}} = (W_{begin} - 0.5W_{fuel_1})/\bar{q}_1 S \qquad (5.49)$$

where for propeller driven airplanes:

$$W_{fuel_1} = E_1\{(c_p/\eta_p)P_{reqd}\}_1 \qquad (5.50)$$

and where for jet airplanes:

$$W_{fuel_1} = E_1\{c_j W/(L/D)\}_1 \qquad (5.51)$$

Note: the subscript '1' here means that the particular quantity applies to endurance/loiter segment 1.

The average value of endurance/loiter lift coefficient (in Eqn.(5.49)) is entered into the trimmed drag polar to determine the corresponding value of the drag coefficient. This information in turn is used to compute the values of (L/D) or $\{(C_L)^{3/2}/C_D\}$ or $\{(C_L)^{1/2}/C_D\}$ which are needed in the Breguet endurance/loiter Eqns.(5.45)-(5.48).

This procedure is repeated as many times as needed until the required total endurance/loiter time is met or until the fuel is exhausted.

<u>Step 5:</u> Note any discrepancies between the computed endurance/loiter times and those required by the mission specification. Decide what design changes (if any) must be made.

Possible design changes are discussed in Step 5 in Sub-section 5.4.4.

<u>Step 6:</u> Document all the results, including any design changes made.

5.6 DIVE

5.6.1 Applicable Regulations

Civil: There are no airworthiness regulations which deal specifically with dives from a performance viewpoint. The following FAR's define the required load factors at the dive speed for civil airplanes:

FAR 23.333, 23.335, 25.333 and 25.335, see Appendix A and also Part V, pages 31-38.

Military: There are no performance related requirements placed on the dive speed. For load factor definitions, see Part V, pages 38 and 39.

5.6.2 Relationship to Preliminary Design

See Part II, Chapter 2, Step 20.

5.6.3 Mathematical Model for Analyzing Dives

Figure 5.12 depicts the forces which act on an airplane in a straight line dive. The equations of motion for a straight line dive at constant speed are:

$$T\cos(\alpha + \phi_T) - C_D\bar{q}S - W\sin\gamma = 0 \qquad (5.52)$$

$$T\sin(\alpha + \phi_T) + C_L\bar{q}S - W\cos\gamma = 0 \qquad (5.53)$$

These equations represent a 'snapshot' taken at constant altitude. The flight path (here: dive) angle, γ is negative in these equations and is negative as shown in Figure 5.12.

In these equations, which assume that moment equilibrium exists, the airplane drag coefficient comes from a 'trimmed' drag polar. Part VI, Section 4.10 presents a method for computing trim drag.

The stability and control characteristics of the airplane in dives (high speed) were already discussed in Chapter 4: Sections 2.1, 2.8 and 2.9.

There also exists a unique relationship between the angle of attack, α and the lift coefficient, C_L. A method for determining C_L versus α is given in Part VI, Chapter 8.

Regular Dive

Vertical Dive

Figure 5.12 Forces Acting on an Airplane in a Dive

The quantity T in Equations (5.52) and (5.53) is the so-called installed thrust. Methods for determining the installed thrust capability of an airplane are presented in Part VI, Chapter 6.

The following quantities in Eqns.(5.52) and (5.53) are to be considered as variables: speed or dynamic pressure, $\bar{q}$, angle of attack, α and flight path angle, γ. The thrust (or power) setting, T, the weight, W and the thrust inclination angle ϕ_T are assumed to be known. Obviously one of the variables will have to be preselected!

Usually a dive is considered at some known dive angle or at some known speed.

1. Dive for known speed.

In this case, the dynamic pressure, $\bar{q}$ is known. The solution process goes as follows:

Assume a value for angle of attack, α. This is used to find C_D and C_L. Equation (5.52) is then used to solve for the flight path angle, γ. Next, Equation (5.53) is used to solve for α. This value of α is compared with the first one and an iteration is performed until there is agreement to within 0.1 degrees.

2. Dive for known dive angle.

In this case, the flight path angle, γ is known. The solution process goes as follows:

Assume a value for angle of attack, α. This is used to find C_D and C_L. Equation (5.52) is then used to solve for the dynamic pressure, $\bar{q}$. Next, Equation (5.53) is used to solve for α. This value of α is compared with the first one and an iteration is performed until there is agreement to within 0.1 degrees.

NOTE: in certain applications a speed brake may be employed. The effect of speedbrakes on the drag polar is discussed in Section 4.12 in Part VI.

In extreme cases an airplane may be required to dive vertically. The equilibrium dive speed in such a case is called the 'terminal dive speed'.

Figure 5.12 also shows the forces which act on the airplane in that case. Since no lift is required in a vertical dive, the drag coefficient is equal to the zero-lift drag coefficient, C_{D_o} and the equation of motion is:

$$T - W = C_{D_o} \bar{q} S \tag{5.54}$$

Since T and C_{D_o} both depend on speed, this equation must be solved also with an iteration. Assume a terminal Mach number at some altitude. Find C_{D_o} and the installed thrust (for whatever thrust setting has been assumed) and see if the equation is satisfied. If not, iterate until the terminal dive speed is within 0.5 percent.

5.6.4 Step-by-Step Procedure for Analyzing Dives

Step 1: Determine the flight conditions and airplane configurations for which dives must be performed. This information is normally contained in the mission specification.

Step 2: For the flight conditions and configurations defined in Step 1, determine the trimmed drag polars of the airplane. Also determine the installed thrust (or power) characteristics.

Trimmed drag polars are determined with the method of Chapter 4, Part VI. Installed thrust (or power) characteristics are determined with the method of Chapter 6, Part VI.

Step 3: Use the iteration process defined in Sub-section 5.6.3 to determine the dive performance of the airplane. Compare the results with the requirements and decide whether or not design adjustments are in order.

Step 4: Document the results, including any design changes made.

5.7 MANEUVERING

5.7.1 Applicable Regulations

Civil: There are no airworthiness regulations which deal specifically with maneuvering performance. The load factors which an airplane must be able to withstand from a structural viewpoint are covered in:

FAR 23.333, 23.335, 23.337, 25.333, 25.335 and 25.337.

Military: MIL-F-8785C, Pars. 3.1.7 and 3.2.3.5-6.

5.7.2 Relationship to Preliminary Design

See Part I, Section 3.5 and Step 27, Part II.

5.7.3 Mathematical Model for Analyzing Maneuvering Flight

The following maneuvers will be considered:

1. Instantaneous maneuvers: pull-up (push-over) and level turns

2. Sustained maneuvers: pull-up (push-over) and level turns.

Figures 5.13 and 5.14 show the forces which act on an airplane in pull-ups and in level turns.

The equations of motion for the situation of Figure 5.13, which represents a steady, symmetrical pullup are:

$$T\cos(\alpha + \phi_T) - C_D\bar{q}S - W\sin\gamma = 0 \qquad (5.55)$$

$$T\sin(\alpha + \phi_T) + C_L\bar{q}S - (W/g)U_1Q_1 - W\cos\gamma = 0 \qquad (5.56)$$

Note that at the bottom of the pullup: $\gamma = 0$.

These equations represent a 'snapshot' taken at constant altitude. The flight path (here: dive) angle, γ is negative in these equations and as shown in Figure 5.13.

In these equations, which assume that moment equilibrium exists, the airplane drag coefficient comes from a 'trimmed' drag polar. Part VI, Section 4.10 presents a method for computing trim drag.

Figure 5.13 Forces Acting on an Airplane in a Pullup

Figure 5.14 Forces Acting on an Airplane in a Level Turn

The stability and control characteristics of the airplane in dives (high speed) were already discussed in Chapter 4: Sections 2.1, 2.8 and 2.9.

There also exists a unique relationship between the angle of attack, α and the lift coefficient, C_L. A method for determining C_L versus α is given in Part VI, Chapter 8. The lift coefficient in this case is also dependent on pitchrate, Q_1 through the stability derivative: C_{L_q}. The reader is referred to Chapter 5 of Reference 12 for a discussion of this effect.

The pitchrate, Q_1 in a pullup is:

$$Q_1 = (g/U_1)(n - 1) \tag{5.57}$$

where: n is the load factor in the maneuver.

The quantity T in Equations (5.55) and (5.56) is the so-called <u>installed thrust.</u> Methods for determining the installed thrust capability of an airplane are presented in Part VI, Chapter 6.

The equations of motion for the situation depicted in Figure 5.14 (steady level turn) are:

$$T\cos(\alpha + \phi_T) - C_D\bar{q}S = 0 \tag{5.58}$$

$$T\sin(\alpha + \phi_t) + C_L\bar{q}S - W\cos\phi - (W/g)U_1Q_1 = 0 \tag{5.59}$$

All comments made for Equations (5.55) and (5.56) apply also to these equations. The pitchrate, Q_1 in a level turn is:

$$Q_1 = (g/U_1)(n - 1/n) \tag{5.60}$$

The <u>turnrate</u> in a level turn is:

$$\dot{\psi}_1 = (g\tan\phi)/U_1 = (g/U_1)(n^2 - 1)^{1/2} \tag{5.61}$$

The <u>turn radius</u> in a level turn is:

$$R_t = \{(U_1)^2/g\}/\tan\phi = \{(U_1)^2/g\}/(n^2 - 1)^{1/2} \quad (5.62)$$

As a help to the reader, Figure 5.15 has been included. It allows rapid determination of turnrate and turn radius.

In the case of acrobatic airplanes and certain military airplanes the mission specification may contain specific numerical requirements for 'pulling g's or for specific turnrates and /or turn radii.

To meet these requirements on an <u>instantaneous</u> basis, all that is required is to show that the maximum trimmed lift capability of the airplane in a given flight condition and airplane configuration is not exceeded:

$$C_{L_{maneuver}} < C_{L_{max}} \text{ or } C_{L_{buffet}} \quad (5.63)$$

To meet these requirements on a <u>sustained</u> basis, condition (5.63) must still be satisfied. However, in addition, the following condition must be satisfied:

$$T_{reqd} < T_{max} \text{ or } P_{reqd} < P_{max} \quad (5.64)$$

At this stage in the preliminary design process it suffices to verify the sustained capability by showing that:

<u>for jet airplanes:</u>

$$T_{reqd} = \{C_{D_o} + (C_{L_{man}})^2/\pi Ae\} < T_{max} \quad (5.65)$$

<u>for propeller driven airplanes:</u>

$$P_{reqd} = \{C_{D_o} + (C_{L_{man}})^2/\pi Ae\}U_1/550 < P_{max} \quad (5.66)$$

The lift coefficient in Equations (5.65) and (5.66) is to be found from:

$$C_{L_{man}} = nC_{L_1} \quad (5.67)$$

where: n is the loadfactor in the maneuver and

$$C_{L_1} = W/\bar{q}S \quad (5.68)$$

Figure 5.15 Generalized Turning Performance in a Level Turn

Taken from: Hurt, H.H.Jr., Aerodynamics for Pilots, ATC Manual 51-3, 1963.

IMPORTANT NOTE:

Whenever sufficient thrust (or power) IS NOT available to keep the speed constant in a pullup or turn, the speed will begin to bleed off. The rate at which this occurs is called the 'Bleed Rate'. See Appendix B for further information on bleed rate. Modern fighters may have to be designed to have certain minimum bleed rate values in certain combat situations. The equations of motion which govern the airplane behavior in such cases is too complex for Class II methods. Reference 12 in Chapter 2, page 43 shows a development of the general equations of motion. The bleed rate can be determined from these equations by integration.

An approximation for the bleed rate is given in Subsection 5.7.4.

The reader is encouraged to read Reference 29 for some interesting views on fighter agility.

5.7.4 Step-by-Step Procedure for Analyzing Maneuvering Flight

Step 1: From the mission specification of the airplane determine the required maneuvering capabilities.

Typically these are stated in terms of:

1. Instantaneous g's
2. Sustained g's
3. Turn radius
4. Turnrate
5. Bleedrate

Tabulate the flight conditions and airplane configurations for which maneuvering requirements must be met.

Step 2: Match each maneuvering requirement with the appropriate equation in Sub-section 5.7.3 and determine the required input information.

This input information generally will consist of:

1. Trimmed drag polars for the appropriate Mach number and configuration.

2. Installed thrust (or power) data for the ap-

propriate Mach number and throttle setting.

3. Maximum trimmed lift capability and buffet boundary data.

NOTE: In sub-section 5.7.3 no equation was given for bleed rate. As a first and rough approximation it is suggested to use:

$$\text{Bleed rate} = \dot{u} = C_D \bar{q} S - T\cos(\alpha + \phi_T) \qquad (5.69)$$

$$\text{where: } C_D = C_{D_o} + (C_{L_{man}})^2/\pi Ae \qquad (5.70)$$

with: $C_{L_{man}}$ given by Eqn.(5.67).

<u>Step 3:</u> With the data and equations from Steps 1 and 2 determine whether the airplane meets the maneuvering requirements.

If discrepancies are evident, identify the reason(s). Typical reasons for not meeting maneuvering requirements are:

1. Deficiency in maximum lift capability: this requires adjustments in wing design

2. Deficiencies in thrust (or power) capability: this requires adjustments in powerplant choice or powerplant installation design.

Decide on any design adjustments required and see whether the design needs to be iterated.

Note: For fighter airplanes the maneuvering and specific excess power (specific energy) capabilities are often plotted as a function of speed and altitude. Figures 5.16 and 5.17 represent typical examples.

<u>Step 4:</u> Document the results, including any design changes made.

COURTESY: NADC

Figure 5.16 Maneuvering Performance: F-4J Versus A-4M

COURTESY: NADC

Figure 5.17 Flight Envelopes and Specific Energy: F-4J Versus A-4M

5.8 DESCENT AND GLIDE

5.8.1 Applicable Regulations

Civil: There are no regulations which deal with descents and glides from a performance viewpoint. The descent flight phase normally terminates in an approach to landing. During that phase the climb regulations are in effect: see Section 3.5.

Military: The only regulation in force is that which disallows range credit for descents by subsonic airplanes. For supersonic airplanes descent range credit may be taken in certain instances. See: MIL-C-005011B, par.3.5.3.4. The USNavy AS-5263 does not allow range credit for descent at all. See Appendix B.

5.8.2 Relationship to Preliminary Design

This performance item was not included in the airplane sizing process, except to account for fuel used: see Part I, Section 2.4.

5.8.3 Mathematical Model for Analyzing Descent and Glide

Descents and glides are closely related to climbs and dives. The differences are subtle. In a descent the flight path angle is normally shallow and thrust (or power) is reduced. In a glide the flight path angle is still shallow (the Space Shuttle is an exception!) but the thrust (or power) is at flight idle, zero or absent (such as in gliders!).

Figure 5.18 depicts the forces which act on an airplane in a descent, Figure 5.19 for a glide.

For a (partial) power descent, the equations of motion are:

$$T\cos(\alpha + \phi_T) - C_D\bar{q}S - W\sin\gamma = 0 \qquad (5.71)$$

$$T\sin(\alpha + \phi_T) + C_L\bar{q}S - W\cos\gamma = 0 \qquad (5.72)$$

These equations are identical to Eqns.(5.52) and (5.53), with the flight path angle, γ being NEGATIVE again. Solutions to these equations are discussed in Subsection 5.6.3 on pages 146-148.

Figure 5.18 Forces Acting on an Airplane in a Descent

Figure 5.19 Forces Acting on an Airplane in a Glide

Once the flight path angle is known, the rate of descent, RD is found from:

$$RD = -U_1 \sin\gamma, \text{ with } \gamma \text{ being negative} \qquad (5.73)$$

For a glide (power off), the equations of motion are:

$$C_D\bar{q}S + W\sin\gamma = 0 \qquad (5.74)$$

$$C_L\bar{q}S - W\cos\gamma = 0 \qquad (5.75)$$

In this case the flight path angle, γ follows from:

$$\tan\gamma = -(C_D/C_L) = -1/(C_L/C_D) \qquad (5.76)$$

The rate of descent follows from:

$$RD = \{(W/S)(2/\rho)(C_D{}^2/C_L{}^{3/2})(\cos\gamma)^3\}^{1/2} \qquad (5.77)$$

If the glide is conducted at constant lift-to-drag ratio, the glide range is:

$$R_{GL} = -h/\tan\gamma \qquad (5.78)$$

Similarly, the time-in-the-air follows from:

$$t_{GL} = h/RD \qquad (5.79)$$

5.8.4 Step-by-Step Procedure for Analyzing Descent and Glide

Step 1: Determine the descent and/or glide requirements for the airplane from the mission specification.

Note: Except for gliders, most airplane specifications do not contain requirements for descents and/or glides.

Step 2: Determine which of equations (5.71)-(5.79) apply to and obtain the required input information.

Step 3: Compute the descent and/or glide flight path angle, speed, rate-of-descent, range and time-in-the-air. Compare these data with the requirements (if any). Determine what (if any) design adjustments must be made.

If the deficiency is in glide range or time-in-the-air, the problem is usually too much drag.

Step 4: Document the results of all calculations, including any design adjustments made.

5.9 LANDING

5.9.1 Applicable Regulations

Civil: FAR 23.75 and FAR 25.125, see Appendix A.

Military: MIL-C-005011B, par. 3.4.2.11, 3.4.2.12 and 3.4.7 and:
AS-5263, par. 3.5.2.12-13, and 3.5.7, see Appendix B.

5.9.2 Relationship to Preliminary Design

See Part I, Section 3.3.

5.9.3 Mathematical Model for Analyzing Landing Performance

The definition of landing distance and the associated reference speeds depends on which regulation is used to certify the airplane: FAR 23, FAR 25 or Military.

Figure 5.20 shows the differences in the definitions for landing distances and the associated reference speeds.

The following method for computing the landing distance, s_L and the landing fieldlength, s_{FL} is due to Torenbeek (Reference 28):

For FAR 25:

$$s_{FL} = s_L/0.6 \tag{5.80}$$

For FAR 23 and for Military:

$$s_L = s_{AIR} + s_{LG} \tag{5.81}$$

where: s_{AIR} is the distance from the obstacle height, h_L to the point of touchdown:

$$s_{AIR} = (1/\bar{\gamma})\{(V_A^2 - V_{TD}^2)/2g + h_L\} \tag{5.82}$$

$$\text{with: } \bar{\gamma} = \{(D - T)/W\}_{ave}, \tag{5.83}$$

for which an average value of 0.10 is often used. This quantity actually ranges from about 0.05 (transports) at the obstacle, to a value of C_D/C_L in ground effect at touch-

Definition of FAR 23 Landing Distance

Definition of FAR 25 landing Distance

Definition of Military Landing Distance

Figure 5.20 Definition of Landing Distances According to the Regulations

down which assumes that T=0 at touchdown.

V_A is the approach speed at the obstacle:

<u>FAR 23 and 25:</u> $V_A = 1.3V_{s_L}$ (5.84)

<u>MIL-C-005011B:</u> $V_A = 1.2V_{s_L}$ (5.85)

<u>AS-5263:</u> $V_A = 1.1V_{s_{PA}}$ (5.86)

but with thrust required for level flight at $1.15V_{s_{PA}}$

$$V_{TD} = V_A[1 - [(\bar{\gamma})^2/\Delta n)]^{1/2} \quad (5.87)$$

with: Δn = 0.10 as a reasonable average: this quantity depends on pilot technique and on airplane handling qualities!

h_L = 50 feet in all regulations

s_{LG} is the landing ground run to zero speed on the runway:

$$s_{LG} = [(V_{TD})^2]/2\bar{a} \quad (5.88)$$

with: $\bar{a}$, the average deceleration for the ground-run. In preliminary design it is acceptable to use:

<u>For light airplanes</u> with simple brake systems:

$\bar{a}/g$ = 0.30 to 0.35

<u>For turboprops</u> without use of reversible propellers:

$\bar{a}/g$ = 0.35 to 0.45

For <u>turbojets</u> and <u>turbofans</u> with ground-spoilers, antiskid devices and speed brakes (but no reverse thrust):

$\bar{a}/g = 0.40$ to 0.50

For the latter, including nosewheel braking:

$\bar{a}/g = 0.50$ to 0.60

Important Note:

Airplanes which are carrier based must be compatible with the performance restrictions inherent in each arresting system. A suitable mathematical model for determining compatibility with USNavy arresting gear systems is given in Sub-sub-section 3.3.5.2, page 115, Part I.

5.9.4 Step-by-Step Procedure for Analyzing Landing Performance

The following step-by-step procedure is suggested for determining the landing distance(s) of an airplane:

Step 1: Determine which regulation applies to the design and read that regulation. See Table 1.3 and Subsection 2.1.1.

Determine the landing distance requirements from the mission specification.

Step 2: Determine whether Eqn.(5.80) or Eqn.(5.81) govern the landing distance calculation and prepare the necessary input data.

IMPORTANT NOTES:

1. Military airplanes may have to land with a variety of stores on board. Their effect on weight and on the drag polar must be accounted for.

2. The weight at landing is the so-called landing weight, W_L. The regulations require the designer to identify what that landing weight is in relationship to the takeoff weight, W_{TO}.

   Table 3.3, page 107 of Part I provides typical ranges of landing-weight-to-takeoff-weight ratios.

Step 3: Compute the landing distance(s) with the appropriate equation of Step 2 and compare the results

with the requirements of Step 1. If there is a discrepancy of more than 5 percent, design changes are warranted.

<u>Step 4:</u> Decide what (if any) design changes are warrented. Typical design changes which can be contemplated are:

1. Lower wing loading, W/S

2. Higher maximum landing lift coefficient, $C_{L_{max_L}}$. This can be achieved with a change in wing flap design. The effect on weight, complexity and lateral control space must be accounted for!

<u>Step 5:</u> Document the results obtained, including any design changes made.

5.10 MISSION PROFILE ANALYSIS

Particularly in the case of military airplanes, but also for civil airplanes it is useful to prepare mission profiles for all intended missions of the airplane.

Two examples mission profiles are given in Fig.5.21. These mission profiles were used in the early sizing process as outlined in Chapter 2 of Part I. The initial mission performance verification was done as part of P.D. Sequence I, see Step 14, page 16, Part II.

These mission profiles are used again in the final mission performance verification process as outlined in Sections 5.1-5.9 of this chapter. This corresponds to Step 24 in P.D. Sequence II, see page 20, Part II.

For military airplanes various military missions have been given 'standardized' mission profiles. Detailed examples of these may be found in Appendix B.

It will be seen that these mission profiles consist of mission segments which are typically labelled as:

Takeoff, Climb, Cruise, Loiter,

Descent, Landing, Combat, etc.

The methods of Sections 5.1 - 5.9 can be used to perform almost any mission profile analysis.

A typical mission profile analysis will produce the following results:

1. Verification of stall speed requirements: see Section 5.1 for the method.

2. Verification of takeoff fieldlength requirements: see Section 5.2 for the method.

3. Verification of climb requirements: see Section 5.3 for the method.

4. Verification of cruise, range and payload-range requirements: see Section 5.4 for the method.

5. Verification of endurance/loiter requirements: see Section 5.5 for the method.

6. Verification of dive requirements: see Section 5.6 for the method.

Transport

Fighter

Figure 5.21 Example Mission Profiles

7. Verification of maneuvering requirements: see Section 5.7 for the method.

8. Verification of descent and glide requirements: see Section 5.8 for the method.

9. Verification of landing fieldlength requirements: see Section 5.9 for the method.

10. Verification of fuel used during each segment of the intended mission profile:

    For each mission profile segment. i the fuel used can be estimated from:

$$(W_{F_{used}})_i = (\dot{W}_F)_i t_i \qquad (5.89)$$

where: $(\dot{W}_F)_i$ is the average fuel flow rate in pounds per second for segment i.

t_i is the time to complete segment i.

Notes: 1. for climb, cruise, loiter, and descent mission phases values for t_i are easily obtained. For other mission phases these times have to be based on the operational environment. For example, taxi-fuel will be more for Chicago O'Hare than for Kansas City International.

2. for military airplanes the method used to compute 'fuel'used' in specific mission segments is outlined in Appendix B!

3. lacking data on the operational environment of a civil airplane, use the military method of Appendix B.

11. Preparation of a payload-range diagram: see Section 5.4 for the methodology.

12. Mission performance critique: this consists of a list of performance shortcomings AND performance excesses PLUS a list of suggested 'fixes'

The effect of performance demands during each individual mission profile segment on the takeoff weight of an airplane is easily evaluated with the so-called 'Breguet-Partials' method of Section 2.7 in Part I.

5.11 PRODUCTIVITY

When the technical aspects of the design of a new airplane have been decided on and the configuration is frozen, the important question is:

> IS THE PRODUCTIVITY OF THE AIRPLANE SUFFICIENTLY HIGH TO WARRANT ITS FULL SCALE DEVELOPMENT?

To give an answer to this question, two tasks must be performed:

1. An analysis of the COST of the airplane.

   This corresponds to Step 36, page 23 of Part II. Methods for cost analysis are given in Part VIII.

2. A set of productivity evaluation criteria must be agreed upon so that the new design can be judged in a fair and understood manner.

   A number of possible evaluation criteria are given in this section.

The methods of Parts I through VII are aimed at designing an airplane to a set of stated mission requirements, while meeting all applicable airworthiness regulations. Along the way a large number of 'arbitrary' design decisions are made. The impact of these design decisions on the productivity of the airplane are not always clear. One example of such an 'arbitrary' decision was the selection of the overall configuration: Step 3 in Preliminary Design Sequence I, page 11, Part II.

That such an important decision as the selection of the overall configuration is labelled as 'arbitrary' may surprise the reader, but it should not. Refer to Figure 1.1, page 5 of Part II and to Note 1 at the bottom of page 3 of Part II!!!

In many instances, the red-white-and-blue team approach will have been used and as a result several airplane designs have evolved, all of which meet the mission and airworthiness requirements. Which is 'best'?

Again, it will be necessary to establish a set of evaluation criteria before this question can be answered.

Typical evaluation criteria which are used to judge the productivity of airplane designs are:

1. <u>Performance Related Criteria such as:</u>

1.1 Takeoff weight (mostly because of a historical relationship between acquisition cost and weight)

All else being the same, a low takeoff weight is judged to be 'good'.

1.2 Volume of the payload-range diagram. Examples of payload-range diagrams with different 'volumes' are given in Figures 5.22 and 5.23.

All else being the same, a large payload-range volume is judged to be 'good'.

1.3 Specific range, nm/lbs of fuel: U_1/c_jT_{reqd}

or U_1/c_pP_{reqd}.

All else being the same, a large specific range is judged to be 'good'.

1.4 Seat-miles per pound of fuel:

U_1N_{seats}/c_jT_{reqd} or U_1N_{seats}/c_pP_{reqd}

All else being the same a high value of seat-miles per pound of fuel is judged to be 'good'.

1.5 Available sortie rate: high sortie rates are judged 'good'.

2. <u>Cost Related Criteria such as:</u>

2.1 Acquisition Cost (Also referred to as cost of ownership.

All else being the same this cost must be small for the airplane to be judged 'good'.

2.2 Profit potential: Payload available MINUS payload to break-even of operating costs.

All else being the same, the larger the profit potential, the 'better' the airplane.

2.3 (Speed times payload)/(Acquisition Cost), used in judging general aviation airplanes.

FROM: FLIGHT INTERNATIONAL*, MAY 10 1986
*BRITISH AEROSPACE WEEKLY

Figure 5.22 Payload-Range Diagrams for Turboprops

FROM: INTERAVIA* MAY 1986

* SWISS MONTHLY

Figure 5.23 Payload-Range Diagrams for Jet Transports

All else being the same, when this parameter is larger, the airplane is judged to be better.

2.4 Direct Operating Cost (DOC).

All else being the same, low DOC is judged to be 'good'.

2.5 Return on Investment (ROI).

All else being the same, high ROI is judged to be 'good'.

2.6 Life Cycle Cost (LCC).

All else being the same, low LCC is judged to be 'good'.

2.7 Maintenance manhours required per flight hour or per sortie.

This quantity should be low for the airplane to be judged 'good'.

The 'performance' related criteria, 1.1-1.6 can be evaluated with the methods of Sections 5.1-5.10.

The 'cost' related criteria, 2.1-2.7 require a method to relate airplane design parameters to cost.

Part VIII specifically addresses the problem of estimating airplane cost. Once cost relationships are established it is possible to iterate certain influential (to cost) design parameters with as goal to 'minimize' certain costs. Reference 30 deals specifically with these design optimization problems.

5.12 PRESENTATION OF AIRPLANE PERFORMANCE DATA

Once a new airplane design is 'finished' it is advisable to present the performance capabilities of the airplane in a generally understandable format AND to include the basic aerodynamic and installed thrust (or power) data on which all performance is based.

Since this type of documentation is not regulated by the FAA, there are no guidelines for the preparation of such data for civil airplanes. Each airplane manufacturer has evolved his own method of doing so.

For military airplanes the situation is different. The USAF, but in particular the USNAVY has standardized the required documentation of aerodynamic, thrust and performance data in the following manner:

For Aerodynamic and Thrust data: See Appendix B1.2, pages 265-271.

For Performance data: See Appendix B1.2, pages 272-279.

The author suggests that, when lacking a given format for the documentation of aerodynamic, thrust and performance data, this USNAVY method be used.

COURTESY: FOKKER

FOKKER T. VIII-W

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APPENDIX A: CIVIL AIRWORTHINESS REGULATIONS FOR AIRPLANE PERFORMANCE, STABILITY AND CONTROL

Before a civil airplane can be sold to the public, to corporations or to airlines a type certificate must have been issued. This type certificate ensures that the airplane has been designed and built in accordance with federally enforced minimum airworthiness standards. These minimum airworthiness standards are incorporated in the Federal Aviation Regulations (FAR's) of Reference 8. These regulations have a major impact on the design of airplanes. For that reason it is essential that airplane designers are aware of content and meaning of these regulations.

To obtain a type certificate for a new airplane design the procedures laid down in FAR 21 (Reference 8) must be followed.

The purpose of this appendix is to present copies of those federal aviation regulations which pertain to the minimum required performance, stability and control characteristics of airplanes. In the regulations, the minimum required performance characteristics are listed under the heading: 'performance'. The minimum required stability and control characteristics are listed under the heading: 'flight characteristics'.

Complete versions of the federal aviation regulations may be found in Reference 8. This appendix is organized as follows:

A1. Definitions and Abbreviations: page 186

A2. Federal Aviation Regulation: FAR 23, with 1987 Amendments page 191

Note: with the 1987 amendments, FAR 23 now also applies to commuter airplanes seating up to 19 passengers.

If there is a question about applicability,
see: FAR 23.1 Applicability: page 191
OR: FAR 23 Amendment: 23.1 Applicability: page 207

A3. Federal Aviation Regulation: FAR 25 page 214

This regulation applies to transport category airplanes.

A1. Definitions and Abbreviations

In this Section a summary is given of those definitions which are needed for proper interpretation of those FAR 23 and 25 Parts which deal with performance and with flight characteristics.

PART 1—DEFINITIONS AND ABBREVIATIONS

Sec.
1.1 General definitions.
1.2 Abbreviations and symbols.
1.3 Rules of construction.

AUTHORITY: 49 U.S.C. 1347, 1348, 1354(a), 1357(d)(2), 1372, 1421 through 1430, 1432, 1442, 1443, 1472, 1510, 1522, 1652(e), 1655(c), 1657(f), 49 U.S.C. 106(g) (Revised Pub. L. 97-449, January 12, 1983).

§ 1.1 General definitions.

As used in Subchapters A through K of this chapter, unless the context requires otherwise:

"Aerodynamic coefficients" means non-dimensional coefficients for aerodynamic forces and moments.

"Aircraft" means a device that is used or intended to be used for flight in the air.

"Aircraft engine" means an engine that is used or intended to be used for propelling aircraft. It includes turbosuperchargers, appurtenances, and accessories necessary for its functioning, but does not include propellers.

"Airframe" means the fuselage, booms, nacelles, cowlings, fairings, airfoil surfaces (including rotors but excluding propellers and rotating airfoils of engines), and landing gear of an aircraft and their accessories and controls.

"Airplane" means an engine-driven fixed-wing aircraft heavier than air, that is supported in flight by the dynamic reaction of the air against its wings.

"Brake horsepower" means the power delivered at the propeller shaft (main drive or main output) of an aircraft engine.

"Calibrated airspeed" means the indicated airspeed of an aircraft, corrected for position and instrument error. Calibrated airspeed is equal to true airspeed in standard atmosphere at sea level.

"Category":

(1) As used with respect to the certification, ratings, privileges, and limitations of airmen, means a broad classification of aircraft. Examples include: airplane; rotorcraft; glider; and lighter-than-air; and

(2) As used with respect to the certification of aircraft, means a grouping of aircraft based upon intended use or operating limitations. Examples include: transport, normal, utility, acrobatic, limited, restricted, and provisional.

"Category A," with respect to transport category rotorcraft, means multiengine rotorcraft designed with engine and system isolation features specified in Part 29 and utilizing scheduled takeoff and landing operations under a critical engine failure concept which assures adequate designated surface area and adequate performance capability for continued safe flight in the event of engine failure.

"Category B," with respect to transport category rotorcraft, means single-engine or multiengine rotorcraft which do not fully meet all Category A standards. Category B rotorcraft have no guaranteed stay-up ability in the event of engine failure and unscheduled landing is assumed.

"Category II operations", with respect to the operation of aircraft, means a straight-in ILS approach to the runway of an airport under a Category II ILS instrument approach procedure issued by the Administrator or other appropriate authority.

"Category III operations," with respect to the operation of aircraft, means an ILS approach to, and landing on, the runway of an airport using a Category III ILS instrument approach procedure issued by the Administrator or other appropriate authority.

"Clearway" means:

(1) For turbine engine powered airplanes certificated after August 29, 1959, an area beyond the runway, not less than 500 feet wide, centrally located about the extended centerline of the runway, and under the control of the airport authorities. The clearway is expressed in terms of a clearway plane, extending from the end of the runway with an upward slope not exceeding 1.25 percent, above which no object nor any terrain protrudes. However, threshold lights may protrude above the plane if their height above the end of the runway is 26 inches or

less and if they are located to each side of the runway.

(2) For turbine engine powered airplanes certificated after September 30, 1958, but before August 30, 1959, an area beyond the takeoff runway extending no less than 300 feet on either side of the extended centerline of the runway, at an elevation no higher than the elevation of the end of the runway, clear of all fixed obstacles, and under the control of the airport authorities.

"Climbout speed," with respect to rotorcraft, means a referenced airspeed which results in a flight path clear of the height-velocity envelope during initial climbout.

"Critical altitude" means the maximum altitude at which, in standard atmosphere, it is possible to maintain, at a specified rotational speed, a specified power or a specified manifold pressure. Unless otherwise stated, the critical altitude is the maximum altitude at which it is possible to maintain, at the maximum continuous rotational speed, one of the following:

(1) The maximum continuous power, in the case of engines for which this power rating is the same at sea level and at the rated altitude.

(2) The maximum continuous rated manifold pressure, in the case of engines, the maximum continuous power of which is governed by a constant manifold pressure.

"Critical engine" means the engine whose failure would most adversely affect the performance or handling qualities of an aircraft.

"Decision height," with respect to the operation of aircraft, means the height at which a decision must be made, during an ILS or PAR instrument approach, to either continue the approach or to execute a missed approach.

"Equivalent airspeed" means the calibrated airspeed of an aircraft corrected for adiabatic compressible flow for the particular altitude. Equivalent airspeed is equal to calibrated airspeed in standard atmosphere at sea level.

"Flap extended speed" means the highest speed permissible with wing flaps in a prescribed extended position.

"Flightcrew member" means a pilot, flight engineer, or flight navigator assigned to duty in an aircraft during flight time.

"Flight level" means a level of constant atmospheric pressure related to a reference datum of 29.92 inches of mercury. Each is stated in three digits that represent hundreds of feet. For example, flight level 250 represents a barometric altimeter indication of 25,000 feet; flight level 255, an indication of 25,500 feet.

"Flight time" means the time from the moment the aircraft first moves under its own power for the purpose of flight until the moment it comes to rest at the next point of landing. ("Block-to-block" time.)

"Flight visibility" means the average forward horizontal distance, from the cockpit of an aircraft in flight, at which prominent unlighted objects may be seen and identified by day and prominent lighted objects may be seen and identified by night.

"Glider" means a heavier-than-air aircraft, that is supported in flight by the dynamic reaction of the air against its lifting surfaces and whose free flight does not depend principally on an engine.

"Ground visibility" means prevailing horizontal visibility near the earth's surface as reported by the United States National Weather Service or an accredited observer.

"Idle thrust" means the jet thrust obtained with the engine power control level set at the stop for the least thrust position at which it can be placed.

"IFR conditions" means weather conditions below the minimum for flight under visual flight rules.

"Indicated airspeed" means the speed of an aircraft as shown on its pitot static airspeed indicator calibrated to reflect standard atmosphere adiabatic compressible flow at sea level uncorrected for airspeed system errors.

"Landing gear extended speed" means the maximum speed at which an aircraft can be safely flown with the landing gear extended.

"Landing gear operating speed" means the maximum speed at which the landing gear can be safely extended or retracted.

"Large aircraft" means aircraft of more than 12,500 pounds, maximum certificated takeoff weight.

"Load factor" means the ratio of a specified load to the total weight of the aircraft. The specified load is expressed in terms of any of the following: aerodynamic forces, inertia forces, or ground or water reactions.

"Mach number" means the ratio of true airspeed to the speed of sound.

"Manifold pressure" means absolute pressure as measured at the appropriate point in the induction system and usually expressed in inches of mercury.

"Minimum descent altitude" means the lowest altitude, expressed in feet above mean sea level, to which descent is authorized on final approach or during circle-to-land maneuvering in execution of a standard instrument approach procedure, where no electronic glide slope is provided.

"Pitch setting" means the propeller blade setting as determined by the blade angle measured in a manner, and at a radius, specified by the instruction manual for the propeller.

"Propeller" means a device for propelling an aircraft that has blades on an engine-driven shaft and that, when rotated, produces by its action on the air, a thrust approximately perpendicular to its plane of rotation. It includes control components normally supplied by its manufacturer, but does not include main and auxiliary rotors or rotating airfoils of engines.

"Rated maximum continuous augmented thrust", with respect to turbojet engine type certification, means the approved jet thrust that is developed statically or in flight, in standard atmosphere at a specified altitude, with fluid injection or with the burning of fuel in a separate combustion chamber, within the engine operating limitations established under Part 33 of this chapter, and approved for unrestricted periods of use.

"Rated maximum continuous power," with respect to reciprocating, turbopropeller, and turboshaft engines, means the approved brake horsepower that is developed statically or in flight, in standard atmosphere at a specified altitude, within the engine operating limitations established under Part 33, and approved for unrestricted periods of use.

"Rated maximum continuous thrust", with respect to turbojet engine type certification, means the approved jet thrust that is developed statically or in flight, in standard atmosphere at a specified altitude, without fluid injection and without the burning of fuel in a separate combustion chamber, within the engine operating limitations established under Part 33 of this chapter, and approved for unrestricted periods of use.

"Rated takeoff augmented thrust", with respect to turbojet engine type certification, means the approved jet thrust that is developed statically under standard sea level conditions, with fluid injection or with the burning of fuel in a separate combustion chamber, within the engine operating limitations established under Part 33 of this chapter, and limited in use to periods of not over 5 minutes for takeoff operation.

"Rated takeoff power", with respect to reciprocating, turbopropeller, and turboshaft engine type certification, means the approved brake horsepower that is developed statically under standard sea level conditions, within the engine operating limitations established under Part 33, and limited in use to periods of not over 5 minutes for takeoff operation.

"Rated takeoff thrust", with respect to turbojet engine type certification, means the approved jet thrust that is developed statically under standard sea level conditions, without fluid injection and without the burning of fuel in a separate combustion chamber, within the engine operating limitations established under Part 33 of this chapter, and limited in use to periods of not over 5 minutes for takeoff operation.

Rated "30-minute power", with respect to helicopter turbine engines, means the maximum brake horsepower, developed under static conditions at specified altitudes and atmospheric temperatures, under the maximum conditions of rotor shaft rotational speed and gas temperature, and limited in use to periods of not over 30 minutes as shown on the engine data sheet.

Rated "2½ minute power", with respect to helicopter turbine engines, means the brake horsepower, developed statically in standard atmosphere at sea level, or at a specified altitude, for one-engine-out operation of multiengine helicopters for 2½ minutes at rotor shaft rotation speed and gas temperature established for this rating.

"Rating" means a statement that, as a part of a certificate, sets forth special conditions, privileges, or limitations.

"Small aircraft" means aircraft of 12,500 pounds or less, maximum certificated takeoff weight.

"Standard atmosphere" means the atmosphere defined in U.S. Standard Atmosphere, 1962 (Geopotential altitude tables).

"Stopway" means an area beyond the takeoff runway, no less wide than the runway and centered upon the extended centerline of the runway, able to support the airplane during an aborted takeoff, without causing structural damage to the airplane, and designated by the airport authorities for use in decelerating the airplane during an aborted takeoff.

"Takeoff power":

(1) With respect to reciprocating engines, means the brake horsepower that is developed under standard sea level conditions, and under the maximum conditions of crankshaft rota-

tional speed and engine manifold pressure approved for the normal takeoff, and limited in continuous use to the period of time shown in the approved engine specification; and

(2) With respect to turbine engines, means the brake horsepower that is developed under static conditions at a specified altitude and atmospheric temperature, and under the maximum conditions of rotor shaft rotational speed and gas temperature approved for the normal takeoff, and limited in continuous use to the period of time shown in the approved engine specification.

"Takeoff safety speed" means a referenced airspeed obtained after lift-off at which the required one-engine-inoperative climb performance can be achieved.

"Takeoff thrust", with respect to turbine engines, means the jet thrust that is developed under static conditions at a specific altitude and atmospheric temperature under the maximum conditions of rotorshaft rotational speed and gas temperature approved for the normal takeoff, and limited in continuous use to the period of time shown in the approved engine specification.

"Time in service", with respect to maintenance time records, means the time from the moment an aircraft leaves the surface of the earth until it touches it at the next point of landing.

"True airspeed" means the airspeed of an aircraft relative to undisturbed air. True airspeed is equal to equivalent airspeed multiplied by $(\rho 0/\rho)^{1/2}$.

§ 1.2 Abbreviations and symbols.

In Subchapters A through K of this chapter:

"AGL" means above ground level.

"ALS" means approach light system.

"ASR" means airport surveillance radar.

"ATC" means air traffic control.

"CAS" means calibrated airspeed.

"CAT II" means Category II.

"CONSOL or CONSOLAN" means a kind of low or medium frequency long range navigational aid.

"DH" means decision height.

"DME" means distance measuring equipment compatible with TACAN.

"EAS" means equivalent airspeed.

"FAA" means Federal Aviation Administration.

"FM" means fan marker.

"GS" means glide slope.

"HIRL" means high-intensity runway light system.

"IAS" means indicated airspeed.

"ICAO" means International Civil Aviation Organization.

"IFR" means instrument flight rules.

"ILS" means instrument landing system.

"IM" means ILS inner marker.

"INT" means intersection.

"LDA" means localizer-type directional aid.

"LFR" means low-frequency radio range.

"LMM" means compass locator at middle marker.

"LOC" means ILS localizer.

"LOM" means compass locator at outer marker.

"M" means mach number.

"MAA" means maximum authorized IFR altitude.

"MALS" means medium intensity approach light system.

"MALSR" means medium intensity approach light system with runway alignment indicator lights.

"MCA" means minimum crossing altitude.

"MDA" means minimum descent altitude.

"MEA" means minimum en route IFR altitude.

"MM" means ILS middle marker.

"MOCA" means minimum obstruction clearance altitude.

"MRA" means minimum reception altitude.

"MSL" means mean sea level.

"NDB(ADF)" means nondirectional beacon (automatic direction finder).

"NOPT" means no procedure turn required.

"OM" means ILS outer marker.

"PAR" means precision approach radar.

"RAIL" means runway alignment indicator light system.

"RBN" means radio beacon.

"RCLM" means runway centerline marking.

"RCLS" means runway centerline light system.

"REIL" means runway end identification lights.

"RR" means low or medium frequency radio range station.

"RVR" means runway visual range as measured in the touchdown zone area.

"SALS" means short approach light system.

"SSALS" means simplified short approach light system.

"SSALSR" means simplified short approach light system with runway alignment indicator lights.

"TACAN" means ultra-high frequency tactical air navigational aid.

"TAS" means true airspeed.

"TDZL" means touchdown zone lights.

"TVOR" means very high frequency terminal omnirange station.

V_a means design maneuvering speed.

V_b means design speed for maximum gust intensity.

V_c means design cruising speed.

V_d means design diving speed.

V_{df}/M_{df} means demonstrated flight diving speed.

V_f means design flap speed.

V_{fc}/M_{fc} means maximum speed for stability characteristics.

V_{fe} means maximum flap extended speed.

V_h means maximum speed in level flight with maximum continuous power.

V_{le} means maximum landing gear extended speed.

V_{lo} means maximum landing gear operating speed.

V_{lof} means lift-off speed.

V_{mc} means minimum control speed with the critical engine inoperative.

V_{mo}/M_{mo} means maximum operating limit speed.

V_{mu} means minimum unstick speed.

V_{ne} means never-exceed speed.

V_{no} means maximum structural cruising speed.

V_r means rotation speed.

V_s means the stalling speed or the minimum steady flight speed at which the airplane is controllable.

V_{s0} means the stalling speed or the minimum steady flight speed in the landing configuration.

V_{s1} means the stalling speed or the minimum steady flight speed obtained in a specific configuration.

V_{TOSS} means takeoff safety speed for Category A rotorcraft.

V_x means speed for best angle of climb.

V_y means speed for best rate of climb.

V_1 means takeoff decision speed (formerly denoted as critical engine failure speed).

V_2 means takeoff safety speed.

V_2 min means minimum takeoff safety speed.

"VFR" means visual flight rules.

"VHF" means very high frequency.

"VOR" means very high frequency omnirange station.

"VORTAC" means collocated VOR and TACAN.

[Doc. No. 1150, 27 FR 4500, May 15, 1962]

EDITORIAL NOTE: For FEDERAL REGISTER citations affecting § 1.2, see the List of CFR Sections Affected appearing in the Finding Aids section of this volume.

§ 1.3 Rules of construction.

(a) In Subchapters A through K of this chapter, unless the context requires otherwise:

(1) Words importing the singular include the plural;

(2) Words importing the plural include the singular; and

(3) Words importing the masculine gender include the feminine.

(b) In Subchapters A through K of this chapter, the word:

(1) "Shall" is used in an imperative sense;

(2) "May" is used in a permissive sense to state authority or permission to do the act prescribed, and the words "no person may * * *" or "a person may not * * *" mean that no person is required, authorized, or permitted to do the act prescribed; and

(3) "Includes" means "includes but is not limited to".

[Doc. No. 1150, 27 FR 4500, May 15, 1962, as amended by Amdt. 1-10, 31 FR 5055, Mar. 29, 1966]

A2. Federal Aviation Regulation: FAR 23 with 1987 Amendments

In this Section a summary is given of that part of FAR 23 which deals with performance and with flight characteristics.

Subpart A—General

§ 23.1 Applicability.

(a) This part prescribes airworthiness standards for the issue of type certificates, and changes to those certificates, for small airplanes in the normal, utility, and acrobatic categories that have a passenger seating configuration, excluding pilot seats, of nine seats or less.

(b) Each person who applies under Part 21 for such a certificate or change must show compliance with the applicable requirements of this part.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-10, 36 FR 2864, Feb. 11, 1971]

§ 23.2 Special retroactive requirements.

Notwithstanding §§ 21.17 and 21.101 of this chapter and irrespective of the type certification basis, each normal, utility, and acrobatic category airplane having a passenger seating configuration, excluding pilot seats, of nine or less, manufactured one year after December 12, 1985, or any such foreign manufactured airplane for entry into the U.S., must meet the requirements of § 23.785 (g) and (h). For the purpose of this paragraph, the date of manufacture is:

(a) The date the inspection acceptance records, or equivalent, reflect that the airplane is complete and meets the FAA Approved Type Design Data; or

(b) In the case of a foreign manufactured airplane, the date the foreign civil airworthiness authority certifies the airplane is complete and issues an original standard airworthiness certificate, or the equivalent in that country.

[Doc. No. 23-32, 50 FR 46877, Nov. 13, 1985]

§ 23.3 Airplane categories.

(a) The normal category is limited to airplanes intended for nonacrobatic operation. Nonacrobatic operation includes—

(1) Any maneuver incident to normal flying;

(2) Stalls (except whip stalls); and

(3) Lazy eights, chandelles, and steep turns, in which the angle of bank is not more than 60 degrees.

(b) The utility category is limited to airplanes intended for limited acrobatic operation. Airplanes certificated in the utility category may be used in any of the operations covered under paragraph (a) of this section and in limited acrobatic operations. Limited acrobatic operation includes—

(1) Spins (if approved for the particular type of airplane); and

(2) Lazy eights, chandelles, and steep turns, in which the angle of bank is more than 60 degrees.

(c) The acrobatic category is limited to airplanes intended for use without restrictions other than those shown to be necessary as a result of required flight tests.

(d) Small airplanes may be certificated in more than one category if the requirements of each requested category are met.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-4, 32 FR 5934, Apr. 14, 1967]

Subpart B—Flight

General

§ 23.21 Proof of compliance.

(a) Each requirement of this subpart must be met at each appropriate combination of weight and center of gravity within the range of loading conditions for which certification is requested. This must be shown—

(1) By tests upon an airplane of the type for which certification is requested, or by calculations based on, and equal in accuracy to, the results of testing; and

(2) By systematic investigation of each probable combination of weight and center of gravity, if compliance cannot be reasonably inferred from combinations investigated.

(b) The following general tolerances are allowed during flight testing. However, greater tolerances may be allowed in particular tests:

Item	Tolerance
Weight	+5%, -10%.
Critical items affected by weight	+5%, -1%.
C.G.	±7% total travel.

§ 23.23 Load distribution limits.

Ranges of weight and centers of gravity within which the airplane may be safely operated must be established and must include the range for lateral centers of gravity if possible loading conditions can result in significant variation of their positions. If low fuel adversely affects balance or stability, the airplane must be tested under conditions simulating those that would exist when the amount of usable fuel does not exceed one gallon for each 12 maximum continuous horsepower of the engine or engines.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-17, 41 FR 55463, Dec. 20, 1976]

§ 23.25 Weight limits.

(a) *Maximum weight.* The maximum weight is the highest weight at which compliance with each applicable requirement of this part (other than those complied with at the design landing weight) is shown. The maximum weight must be established so that it is—

(1) Not more than—

(i) The highest weight selected by the applicant;

(ii) The design maximum weight, which is the highest weight at which compliance with each applicable structural loading condition of this part (other than those complied with at the design landing weight) is shown; or

(iii) The highest weight at which compliance with each applicable flight requirement is shown, except for airplanes equipped with standby power rocket engines, in which case it is the highest weight established in accordance with Appendix E of this part; or

(2) Assuming a weight of 170 pounds for each occupant of each seat for normal category airplanes and 190 pounds (unless otherwise placarded) for utility and acrobatic category airplanes, not less than the weight with—

(i) Each seat occupied, oil at full tank capacity, and at least enough fuel for one-half hour of operation at rated maximum continuous power; or

(ii) The required minimum crew, and fuel and oil to full tank capacity.

(b) *Minimum weight.* The minimum weight (the lowest weight at which compliance with each applicable requirement of this part is shown) must be established so that it is not more than the sum of—

(1) The empty weight determined under § 23.29;

(2) The weight of the required minimum crew (assuming a weight of 170 pounds for each crewmember); and

(3) The weight of—

(i) For turbojet powered airplanes, 5 percent of the total fuel capacity of that particular fuel tank arrangement under investigation, and

(ii) For other airplanes, the fuel necessary for one-half hour of operation at maximum continuous power.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-7, 34 FR 13086, Aug. 13, 1969; Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.29 Empty weight and corresponding center of gravity.

(a) The empty weight and corresponding center of gravity must be determined by weighing the airplane with—

(1) Fixed ballast;

(2) Unusable fuel determined under § 23.959; and

(3) Full operating fluids, including—

(i) Oil;

(ii) Hydraulic fluid; and

(iii) Other fluids required for normal operation of airplane systems, except potable water, lavatory precharge water, and water intended for injection in the engines.

(b) The condition of the airplane at the time of determining empty weight must be one that is well defined and can be easily repeated.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964; 30 FR 258, Jan. 9, 1965, as amended by Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.31 Removable ballast.

Removable ballast may be used in showing compliance with the flight requirements of this subpart, if—

(a) The place for carrying ballast is properly designed and installed, and is marked under § 23.1557; and

(b) Instructions are included in the airplane flight manual, approved manual material, or markings and placards, for the proper placement of the removable ballast under each loading condition for which removable ballast is necessary.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964; 30 FR 258, Jan. 9, 1965, as amended by Amdt. 23-13, 37 FR 20023, Sept. 23, 1972]

§ 23.33 Propeller speed and pitch limits.

(a) *General.* The propeller speed and pitch must be limited to values that will assure safe operation under normal operating conditions.

(b) *Propellers not controllable in flight.* For each propeller whose pitch cannot be controlled in flight—

(1) During takeoff and initial climb at $V^{\prime}$, the propeller must limit the engine r.p.m., at full throttle or at maximum allowable takeoff manifold pressure, to a speed not greater than the maximum allowable takeoff r.p.m.; and

(2) During a closed throttle glide at the placarded "never-exceed speed", the propeller may not cause an engine speed above 110 percent of maximum continuous speed.

(c) *Controllable pitch propellers without constant speed controls.* Each propeller that can be controlled in flight, but that does not have constant speed controls, must have a means to limit the pitch range so that—

(1) The lowest possible pitch allows compliance with paragraph (b)(1) of this section; and

(2) The highest possible pitch allows compliance with paragraph (b)(2) of this section.

(d) *Controllable pitch propellers with constant speed controls.* Each controllable pitch propeller with constant speed controls must have—

(1) With the governor in operation, a means at the governor to limit the maximum engine speed to the maximum allowable takeoff r.p.m.; and

(2) With the governor inoperative, a means to limit the maximum engine speed to 103 percent of the maximum allowable takeoff r.p.m. with the pro-

peller blades at the lowest possible pitch and with takeoff manifold pressure, the airplane stationary, and no wind.

PERFORMANCE

§ 23.45 General.

(a) Unless otherwise prescribed, the performance requirements of this subpart must be met for still air and a standard atmosphere.

(b) The performance must correspond to the propulsive thrust available under the particular ambient atmospheric conditions, the particular flight condition, and the relative humidity specified in paragraph (d) or (e) of this section, as appropriate.

(c) The available propulsive thrust must correspond to engine power or thrust, not exceeding the approved power or thrust, less—

(1) Installation losses; and

(2) The power or equivalent thrust absorbed by the accessories and services appropriate to the particular ambient atmospheric conditions and the particular flight condition.

(d) For reciprocating engine-powered airplanes, the performance, as affected by engine power, must be based on a relative humidity of 80 percent in a standard atmosphere.

(e) For turbine engine-powered airplanes, the performance, as affected by engine power or thrust, must be based on a relative humidity of—

(1) 80 percent, at and below standard temperature; and

(2) 34 percent, at and above standard temperature plus 50 degrees F.

Between these two temperatures, the relative humidity must vary linearly.

[Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.49 Stalling speed.

(a) V_{s0} is the stalling speed, if obtainable, or the minimum steady speed, in knots (CAS), at which the airplane is controllable, with the—

(1) Applicable power or thrust condition set forth in paragraph (e) of this section;

(2) Propellers in the takeoff position;

(3) Landing gear extended;

(4) Wing flaps in the landing position;

(5) Cowl flaps closed;

(6) Center of gravity in the most unfavorable position within the allowable landing range; and

(7) Weight used when V_{s0} is being used as a factor to determine compliance with a required performance standard.

(b) V_{s0} at maximum weight may not exceed 61 knots for—

(1) Single-engine airplanes; and

(2) Multiengine airplanes of 6,000 pounds or less maximum weight that cannot meet the minimum rate of climb specified in § 23.67(b) with the critical engine inoperative.

(c) V_{s1} is the calibrated stalling speed, if obtainable, or the minimum steady speed, in knots, at which the airplane is controllable, with the—

(1) Applicable power or thrust condition set forth in paragraph (e) of this section;

(2) Propellers in the takeoff position;

(3) Airplane in the condition existing in the test in which V_{s1} is being used; and

(4) Weight used when V_{s1} is being used as a factor to determine compliance with a required performance standard.

(d) V_{s0} and V_{s1} must be determined by flight tests, using the procedure specified in § 23.201.

(e) The following power or thrust conditions must be used to meet the requirements of this section:

(1) For reciprocating engine-powered airplanes, engines idling, throttles closed or at not more than the power necessary for zero thrust at a speed not more than 110 percent of the stalling speed.

(2) For turbine engine-powered airplanes, the propulsive thrust may not be greater than zero at the stalling speed, or, if the resultant thrust has no appreciable effect on the stalling speed, with engines idling and throttles closed.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-7, 34 FR 13066, Aug. 13, 1969; Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.51 Takeoff.

(a) For each airplane (except a skiplane for which landplane takeoff data has been determined under this paragraph and furnished in the Airplane Flight Manual) the distance required to takeoff and climb over a 50-foot obstacle must be determined with—

(1) The engines operating within approved operating limitations; and

(2) The cowl flaps in the normal takeoff position.

(b) For multiengine airplanes, the lift-off speed, V_{LOF}, may not be less than V_{MC} determined in accordance with § 23.149.

(c) Upon reaching a height of 50 feet above the takeoff surface level, the airplane must have reached a speed of not less than the following:

(1) For multiengine airplanes, the higher of—

(i) 1.1 V_{MC}; or

(ii) 1.3 V_{S_1}, or any lesser speed, not less than Vx plus 4 knots, that is shown to be safe under all conditions, including turbulence and complete engine failure.

(2) For single engine airplanes—

(i) 1.3 V_{S_1}; or

(ii) Any lesser speed, not less than Vx plus 4 knots, that is shown to be safe under all conditions, including turbulence and complete engine failure.

(d) The starting point for measuring seaplane and amphibian takeoff distance may be the point at which a speed of not more than three knots is reached.

(e) Takeoffs made to determine the data required by this section may not require exceptional piloting skill or exceptionally favorable conditions.

[Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.65 Climb: All engines operating.

(a) Each airplane must have a steady rate of climb at sea level of at least 300 feet per minute and a steady angle of climb of at least 1:12 for landplanes or 1:15 for seaplanes and amphibians with—

(1) Not more than maximum continuous power on each engine;

(2) The landing gear retracted;

(3) The wing flaps in the takeoff position; and

(4) The cowl flaps or other means for controlling the engine cooling air supply in the position used in the cooling tests required by §§ 23.1041 through 23.1047.

(b) Each airplane with engines for which the takeoff and maximum continuous power ratings are identical and that has fixed-pitch, two-position, or similar propellers, may use a lower propeller pitch setting than that allowed by § 23.33 to obtain rated engine r.p.m. at Vx, if—

(1) The airplane shows marginal performance (such as when it can meet the rate of climb requirements of paragraph (a) of this section but has difficulty in meeting the angle of climb requirements of paragraph (a) of this section or of § 23.77); and

(2) Acceptable engine cooling is shown at the lower speed associated with the best angle of climb.

(c) Each turbine engine-powered airplane must be able to maintain a steady gradient of climb of at least 4 percent at a pressure altitude of 5,000 feet and a temperature of 81 degrees F (standard temperature plus 40 degree F) with the airplane in the configuration prescribed in paragraph (a) of this section.

[Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.67 Climb: one engine inoperative.

(a) Each reciprocating engine-powered multiengine airplane of more than 6,000 pounds maximum weight must be able to maintain a steady rate of climb of at least 0.027 $V_{S_0}^2$ (that is, the number of feet per minute is obtained by multiplying the square of the number of knots by 0.027 at an altitude of 5,000 feet with the—

(1) Critical engine inoperative, and its propeller in the minimum drag position;

(2) Remaining engines at not more than maximum continuous power;

(3) Landing gear retracted;

(4) Wing flaps in the most favorable position; and

(5) Cowl flaps in the position used in the cooling tests required by §§ 23.1041 through 23.1047.

(b) For reciprocating engine-powered multiengine airplanes of 6,000 pounds or less maximum weight, the following apply:

(1) Each airplane with a V_{S_0} of more (than 61 knots must be able to maintain a steady rate of climb of at least 0.027 $V_{S_0}^2$ (that is, the number of feet per minute is obtained by multiplying the square of the number of knots by 0.027), at an altitude of 5,000 feet with the—

(i) Critical engine inoperative and its propeller in the minimum drag position;

(ii) Remaining engines at not more than maximum continuous power;

(iii) Landing gear retracted;

(iv) Wing flaps in the most favorable position; and

(v) Cowl flaps in the position used in the cooling tests required by §§ 23.1041 through 23.1047.

(2) For each airplane with a stalling speed of 61 knots or less, the steady rate of climb at 5,000 feet must be determined with the—

(i) Critical engine inoperative and its propeller in the minimum drag position;

(ii) Remaining engines at not more than maximum continuous power;

(iii) Landing gear retracted;

(iv) Wing flaps in the most favorable position; and

(v) Cowl flaps in the position used in the cooling tests required by §§ 23.1041 through 23.1047.

(c) For turbine-powered multiengine airplanes the following apply:

(1) The steady gradient of climb must be determined at each weight, altitude, and ambient temperature within the operational limits established by the applicant, with the—

(i) Critical engine inoperative, and its propeller in the minimum drag position;

(ii) Remaining engines at not more than maximum continuous power or thrust;

(iii) Landing gear retracted;

(iv) Wing flaps in the most favorable position; and

(v) The means for controlling the engine cooling air supply in the position used in the engine cooling tests required by §§ 23.1041 through 23.1047.

(2) Each airplane must be able to maintain the following climb gradients with the airplane in the configuration prescribed in paragraph (c)(1) of this section:

(i) 1.2 percent (or, if greater, a gradient equivalent to a rate of climb of 0.027 Vs_0^2) at a pressure altitude of 5,000 feet and standard temperature (41 degrees F).

(ii) 0.6 percent (or, if greater, a gradient equivalent to a rate of climb of 0.014 Vs_0^2) at a pressure altitude of 5,000 feet and 81 degrees F (standard temperature plus 40 degrees F).

(3) The minimum climb gradient specified in paragraphs (c)(2) (i) and (ii) of this section must vary linearly between 41 degrees F and 81 degrees F and must change at the same rate up to the maximum operating termperature approved for the airplane.

(4) In paragraphs (c)(2) (i) and (ii) of this section, rate of climb is expressed in feet per minute and Vs_0 is expressed in knots.

(d) For all multiengine airplanes, the speed for best rate of climb with one engine inoperative must be determined.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-7, 34 FR 13088, Aug. 13, 1969; Amdt. 23-21, 43 FR 2317, Jan. 16, 1978]

§ 23.75 Landing.

For airplanes (except skiplanes for which landplane landing data have been determined under this section and furnished in the Airplane Flight Manual), the horizontal distance necessary to land and come to a complete stop (or to a speed of approximately 3 knots for water landings of seaplanes and amphibians) from a point 50 feet above the landing surface must be determined as follows:

(a) A steady gliding approach with a calibrated airspeed of at least 1.3 Vs_1 must be maintained down to the 50-foot height.

(b) The landing may not require exceptional piloting skill or exceptionally favorable conditions.

(c) The landing must be made without excessive vertical acceleration or tendency to bounce, nose over, ground loop, porpoise, or water loop.

(d) It must be shown that a safe transition to the balked landing conditions of § 23.77 can be made from the conditions that exist at the 50-foot height.

(e) The pressures on the wheel braking system may not exceed those specified by the brake manufacturer.

(f) Means other than wheel brakes may be used if that means—

(1) Is safe and reliable;

(2) Is used so that consistent results can be expected in service; and

(3) Is such that exceptional skill is not required to control the airplane.

[Amdt. 23-21, 43 FR 2318, Jan. 16, 1978]

§ 23.77 Balked landing.

(a) For balked landings, each airplane must be able to maintain a steady angle of climb at sea level of at least 1:30 with—

(1) Takeoff power on each engine;

(2) The landing gear extended; and

(3) The wing flaps in the landing position, except that if the flaps may safely be retracted in two seconds or less without loss of altitude and without sudden changes of angle of attack or exceptional piloting skill, they may be retracted.

(b) Each turbine engine-powered airplane must be able to maintain a steady rate of climb of at least zero at a pressure altitude of 5,000 feet at 81 degrees F (standard temperature plus 40 degrees F), with the airplane in the configuration prescribed in paragraph (a) of this section.

[Amdt. 23-21, 43 FR 2318, Jan. 16, 1978]

Flight Characteristics

§ 23.141 General.

The airplane must meet the requirements of §§ 23.143 through 23.253 at the normally expected operating altitudes without exceptional piloting skill, alertness, or strength.

[Amdt. 23-17, 41 FR 55464, Dec. 20, 1976]

Controllability and Maneuverability

§ 23.143 General.

(a) The airplane must be safely controllable and maneuverable during—

(1) Takeoff;

(2) Climb;

(3) Level flight;

(4) Dive; and

(5) Landing (power on and power off with the wing flaps extended and retracted).

(b) It must be possible to make a smooth transition from one flight condition to another (including turns and slips) without danger of exceeding the limit load factor, under any probable operating condition (including, for multiengine airplanes, those conditions normally encountered in the sudden failure of any engine).

(c) If marginal conditions exist with regard to required pilot strength, the "strength of pilots" limits must be shown by quantitative tests. In no case may the limits exceed those prescribed in the following table:

Values in pounds of force as applied to the control wheel or rudder pedals	Pitch	Roll	Yaw
(a) For temporary application:			
Stick	60	30	.....
Wheel (applied to rim)	75	80	.....
Rudder pedal	.....	.....	150
(b) For prolonged application	10	5	20

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-14, 38 FR 31819, Nov. 19, 1973; Amdt. 23-17, 41 FR 55464, Dec. 20, 1976]

§ 23.145 Longitudinal control.

(a) It must be possible, at speeds below the trim speed, to pitch the nose downward so that the rate of increase in airspeed allows prompt acceleration to the trim speed with—

(1) Maximum continuous power on each engine and the airplane trimmed at V_X;

(2) Power off and the airplane trimmed at a speed determined in accordance with § 23.161(c)(3) or (4) as appropriate or at the minimum trim speed, whichever is higher; and

(3) Wing flaps and landing gear (i) retracted, and (ii) extended.

(b) With the landing gear extended no change in trim or exertion of more control force than can be readily applied with one hand for a short period of time may be required for the following maneuvers:

(1) With power off, flaps retracted, and the airplane trimmed at $1.4V_{s1}$ or the minimum trim speed, whichever is higher, extend the flaps as rapidly as possible and allow the airspeed to transition from $1.4V_{s1}$ to $1.4V_{s0}$, or if appropriate from the minimum trim speed to a speed equal to V_{s0} increased by the same percentage that the minimum trim speed at the initial condition was greater than V_{s1}.

(2) With power off, flaps extended, and the airplane trimmed at $1.4V_{s0}$ or the minimum trim speed, whichever is higher, retract the flaps as rapidly as possible and allow the airspeed to transition from $1.4V_{s0}$ to $1.4V_{s1}$, or if appropriate, from the minimum trim speed to a speed equal to $1.4V_{s1}$ increased by the same percentage that the minimum trim speed at the initial condition was greater than V_{s1}.

(3) Repeat paragraph (b)(2) of this section except with maximum continuous power.

(4) With power off, flaps retracted, and the airplane trimmed at a speed determined in accordance with § 23.161 (c)(3) or (4), as appropriate or at the minimum trim speed, whichever is higher, apply takeoff power rapidly while maintaining the same airspeed.

(5) Repeat subparagraph (4) of this paragraph, except with the flaps extended.

(6) With power off, flaps extended, and the airplane trimmed at a speed determined in accordance with § 23.161 (c)(3) or (4), as appropriate or at the minimum trim speed, whichever is higher, obtain and maintain airspeeds between 1.1 V_{s1} and either 1.7 V_{s1} or V_F, whichever is lower.

(c) It must be possible to maintain approximately level flight when flap retraction from any position is made during steady horizontal flight at 1.1 V_{s1} with simultaneous application of not more than maximum continuous power.

(d) It must be possible, with a pilot control force of not more than 10 pounds, to maintain a speed of not more than the speed determined in accordance with § 23.161(c)(4), during a power-off glide with landing gear and wing flaps extended.

(e) By using normal flight and power controls, except as otherwise noted in paragraphs (e)(1) and (e)(2), it must be possible in the following airplanes to establish a zero rate of descent at an attitude suitable for a controlled landing without exceeding the operational and structural limitations of the airplane:

(1) For single engine and multiengine airplanes, without the use of the primary longitudinal control system.

(2) For multiengine airplanes—

(i) Without the use of the primary directional control; and

(ii) If a single failure of any one connecting or transmitting link would affect both the longitudinal and directional primary control system, without the primary longitudinal and directional control system.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-7, 34 FR 13086, Aug. 13, 1969; Amdt. 23-14, 38 FR 31819, Nov. 19, 1973; Amdt. 23-17, 41 FR 55464, Dec. 20, 1976]

§ 23.147 Directional and lateral control.

(a) For each multiengine airplane, it must be possible to make turns with 15 degrees of bank both towards and away from an inoperative engine, from a steady climb at 1.4 V_{s1} or V_Y with—

(1) One engine inoperative and its propeller in the minimum drag position;

(2) The remaining engines at not more than maximum continuous power;

(3) The rearmost allowable center of gravity;

(4) The landing gear (i) retracted, and (ii) extended;

(5) The flaps in the most favorable climb position; and

(6) Maximum weight.

(b) For each multiengine airplane, it must be possible, while holding the wings level within five degrees, to make sudden changes in heading safely in both directions. This must be shown at 1.4 V_{s1} or V_Y with heading changes up to 15 degrees (except that the heading change at which the rudder force corresponds to the limits specified in § 23.143 need not be exceeded), with the—

(1) Critical engine inoperative and its propeller in the minimum drag position;

(2) Remaining engines at maximum continuous power;

(3) Landing gear (i) retracted, and (ii) extended;

(4) Flaps in the most favorable climb position; and

(5) Center of gravity at its rearmost allowable position.

§ 23.149 Minimum control speed.

(a) V_{MC} is the calibrated airspeed, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than five degrees. The method used to simulate critical engine failure must represent the most critical mode of powerplant failure with respect to controllability expected in service.

(b) For reciprocating engine-powered airplanes, V_{MC} may not exceed 1.2 V_S, (where V_S, is determined at the maximum takeoff weight with—

(1) Takeoff or maximum available power on the engines;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for takeoff;

(4) The maximum sea level takeoff weight (or any lesser weight necessary to show V_{MC});

(5) Flaps in the takeoff position;

(6) Landing gear retracted;

(7) Cowl flaps in the normal takeoff position;

(8) The propeller of the inoperative engine—

(i) Windmilling;

(ii) In the most probable position for the specific design of the propeller control; or

(iii) Feathered, if the airplane has an automatic feathering device; and

(9) The airplane airborne and the ground effect negligible.

(c) For turbine engine-powered airplanes, V_{MC} may not exceed 1.2 V_S, (where V_S, is determined at the maximum takeoff weight) with—

(1) Maximum available takeoff power or thrust on the engines;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for takeoff;

(4) The maximum sea level takeoff weight for any lesser weight necessary to show V_{MC});

(5) The airplane in the most critical takeoff configuration except with the landing gear retracted; and

(6) The airplane airborne and the ground effect negligible.

(d) At V_{MC}, the rudder pedal force required to maintain control may not exceed 150 pounds, and it may not be necessary to reduce power or thrust of the operative engines. During recovery, the airplane may not assume any dangerous attitude and it must be possible to prevent a heading change of more than 20 degrees.

[Amdt. 23-21, 43 FR 2318, Jan. 16, 1978]

§ 23.151 Acrobatic maneuvers.

Each acrobatic and utility category airplane must be able to perform safely the acrobatic maneuvers for which certification is requested. Safe entry speeds for these maneuvers must be determined.

§ 23.153 Control during landings.

For an airplane that has a maximum weight of more than 6,000 pounds, it must be possible, while in the landing configuration, to safely complete a landing without encountering forces in excess of those prescribed in § 23.143(c) following an approach to land:

(a) At a speed 5 knots less than the speed used in complying with § 23.75 and with the airplane in trim or as nearly as possible in trim;

(b) With neither the trimming control being moved throughout the maneuver nor the power being increased during the landing flare; and

(c) With the thrust settings used in demonstrating compliance with § 23.75.

[Amdt. 23-14, 38 FR 31819, Nov. 19, 1973]

§ 23.155 Elevator control force in maneuvers.

(a) The elevator control force needed to achieve the positive limit maneuvering load factor may not be less than:

(1) For wheel controls, W/100 (where W is the maximum weight) or 20 pounds, whichever is greater, except that it need not be greater than 50 pounds; or

(2) For stick controls, W/140 (where W is the maximum weight) or 15 pounds, whichever is greater, except that it need not be greater than 35 pounds.

(b) The requirement of paragraph (a) of this section must be met with wing flaps and landing gear retracted under each of the following conditions:

(1) At 75 percent of maximum continuous power for reciprocating engines, or the maximum power or thrust selected by the applicant as an operating limitation for use during cruise for reciprocating or turbine engines.

(2) In a turn, after the airplane is trimmed with wings level at the minimum speed at which the required normal acceleration can be achieved without stalling, and at the maximum level flight trim speed except that the speed may not exceed V_{MO} or V_{MO}/M_{MO}, whichever is appropriate.

(c) Compliance with the requirements of this section may be demonstrated by measuring the normal acceleration that is achieved with the limiting stick force or by establishing the stick force per g gradient and extrapolating to the appropriate limit.

[Amdt. 23-14, 38 FR 31819, Nov. 19, 1973; 38 FR 32784, Nov. 28, 1973]

§ 23.157 Rate of roll.

(a) *Takeoff.* It must be possible, using a favorable combination of controls, to roll the airplane from a steady 30-degree banked turn through an angle of 60 degrees, so as to reverse the direction of the turn within:

(1) For an airplane of 6,000 pounds or less maximum weight, 5 seconds from initiation of roll; and

(2) For an airplane of over 6,000 pounds maximum weight,

$$(W+500)/1,300$$

seconds, where W is the weight in pounds.

(b) The requirement of paragraph (a) must be met when rolling the airplane in either direction in the following condition:

(1) Flaps in the takeoff position;

(2) Landing gear retracted;

(3) For a single engine airplane, at maximum takeoff power or thrust; and for a multiengine airplane, with the critical engine inoperative, the propeller in the minimum drag position, and the other engines at maximum continuous power or thrust; and

(4) The airplane trimmed at $1.2V_{s_1}$, or as nearly as possible in trim for straight flight.

(c) *Approach.* It must be possible, using a favorable combination of controls, to roll the airplane from a steady 30-degree banked turn through an angle of 60 degrees, so as to reverse the direction of the turn within:

(1) For an airplane of 6,000 pounds or less maximum weight, 4 seconds from initiation of roll; and

(2) For an airplane of over 6,000 pounds maximum weight,

$$(W+2,800)/2,200$$

seconds, where W is the weight in pounds.

(d) The requirement of paragraph (c) must be met when rolling the airplane in either direction in the following conditions:

(1) Flaps extended;

(2) Landing gear extended;

(3) All engines operating at idle power or thrust and with all engines operating at the power or thrust for level flight; and

(4) The airplane trimmed at the speed that is used in determining compliance with § 23.75.

[Amdt. 23-14, 38 FR 31819, Nov. 19, 1973]

TRIM

§ 23.161 Trim.

(a) *General.* Each airplane must meet the trim requirements of this section after being trimmed, and without further pressure upon, or movement of, the primary controls or their corresponding trim controls by the pilot or the automatic pilot.

(b) *Lateral and directional trim.* The airplane must maintain lateral and directional trim in level flight at 0.9 V_H or V_C, whichever is lower, with

the landing gear and wing flaps retracted.

(c) *Longitudinal trim.* The airplane must maintain longitudinal trim under each of the following conditions:

(1) A climb with maximum continuous power at a speed between Vx and 1.4 V_{S_1}, with—

(i) The landing gear and wing flaps retracted; and

(ii) The landing gear retracted and the wing flaps in the takeoff position.

(2) A power approach with a 3 degree angle of descent, the landing gear extended, and with—

(i) The wing flaps retracted and at a speed of 1.4 V_{S_1}; and

(ii) The applicable airspeed and flap position used in showing compliance with § 23.75.

(3) Level flight at any speed from 0.9 V_H to either Vx or 1.4 V_{S_1}, with the landing gear and wing flaps retracted.

(d) In addition, each multiengine airplane must maintain longitudinal and directional trim at a speed between V_Y and 1.4 V_{S_1}, with—

(1) The critical engine inoperative;

(2) The remaining engines at maximum continuous power;

(3) The landing gear retracted;

(4) The wing flaps retracted; and

(5) An angle of bank of not more than five degrees.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-21, 43 FR 2318, Jan. 16, 1978]

STABILITY

§ 23.171 General.

The airplane must be longitudinally, directionally, and laterally stable under §§ 23.173 through 23.181. In addition, the airplane must show suitable stability and control "feel" (static stability) in any condition normally encountered in service, if flight tests show it is necessary for safe operation.

§ 23.173 Static longitudinal stability.

Under the conditions specified in § 23.175 and with the airplane trimmed as indicated, the characteristics of the elevator control forces and the friction within the control system must be as follows:

(a) A pull must be required to obtain and maintain speeds below the specified trim speed and a push required to obtain and maintain speeds above the specified trim speed. This must be shown at any speed that can be obtained, except that speeds requiring a control force in excess of 40 pounds or speeds above the maximum allowable speed or below the minimum speed for steady unstalled flight, need not be considered.

(b) The airspeed must return to within plus or minus 10 percent of the original trim speed when the control force is slowly released at any speed within the speed range specified in paragraph (a) of this section.

(c) The stick force must vary with speed so that any substantial speed change results in a stick force clearly perceptible to the pilot.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-14, 38 FR 31820 Nov. 19, 1973]

§ 23.175 Demonstration of static longitudinal stability.

Static longitudinal stability must be shown as follows:

(a) *Climb.* The stick force curve must have a stable slope, at speeds between 85 and 115 percent of the trim speed, with—

(1) Flaps in the climb position;

(2) Landing gear retracted;

(3) 75 percent of maximum continuous power for reciprocating engines or the maximum power or thrust selected by the applicant as an operating limitation for use during a climb for turbine engines; and

(4) The airplane trimmed for V_Y, except that the speed need not be less than 1.4 V_{S_1}.

(b) *Cruise—Landing gear retracted (or fixed gear).* (1) For the cruise conditions specified in paragraphs (b) (2) and (3) of this section, the following apply:

(i) The speed need not be less than 1.3 V_{S_1}.

(ii) For airplanes with V_{NE} established under § 23.1505(a), the speed need not be greater than V_{NE}.

(iii) For airplanes with V_{MO}/M_{MO} established under § 23.1505(c), the speed need not be greater than a speed midway between V_{MO}/M_{MO} and the lesser of V_D/M_D or the speed demon-

strated under § 23.251, except that for altitudes where Mach number in the limiting factor, the speed need not exceed that corresponding to the Mach number at which effective speed warning occurs.

(2) *High speed cruise.* The stick force curve must have a stable slope at all speeds within a range that is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 40 knots plus the resulting free return speed range, above and below the trim speed, with—

(i) Flaps retracted.

(ii) Seventy-five percent of maximum continuous power for reciprocating engines or, for turbine engines, the maximum cruising power or thrust selected by the applicant as an operating limitation, except that the power need not exceed that required at V_{NE} for airplanes with V_{NE} established under § 23.1505(a), or that required at V_{MO}/M_{MO} for airplanes with V_{MO}/M_{MO} established under § 23.1505(c).

(iii) The airplane trimmed for level flight.

(3) *Low speed cruise.* The stick force curve must have a stable slope under all the conditions prescribed in paragraph (b)(2) of this section, except that the power is that required for level flight at a speed midway between 1.3 V_{S1} and the trim speed obtained in the high speed cruise condition under paragraph (b)(2) of this section.

(c) *Landing gear extended (airplanes with retractable gear).* The stick force curve must have a stable slope at all speeds within a range from 15 percent of the trim speed plus the resulting free return speed range below the trim speed, to the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1} nor speeds greater than V_{LE}, with—

(1) Landing gear extended;

(2) Flaps retracted;

(3) 75 percent of maximum continuous power for reciprocating engines, or for turbine engines, the maximum cruising power or thrust selected by the applicant as an operating limitation, except that the power need not exceed that required for level flight at V_{LE}; and

(4) The airplane trimmed for level flight.

(d) *Approach and landing.* The stick force curve must have a stable slope at speeds between 1.1 V_{S1} and 1.8 V_{S1} with—

(1) Wing flaps in the landing position;

(2) Landing gear extended;

(3) The airplane trimmed at a speed in compliance with § 23.161(c)(4).

(4) Both power off and enough power to maintain a 3° angle of descent.

[Amdt. 23-7, 34 FR 12087, Aug. 13, 1969, as amended by Amdt. 23-14, 38 FR 31820, Nov. 19, 1973; Amdt. 23-17, 41 FR 55464, Dec. 20, 1976]

§ 23.177 Static directional and lateral stability.

(a) *Three-control airplanes.* The stability requirements for three-control airplanes are as follows:

(1) The static directional stability, as shown by the tendency to recover from a skid with the rudder free, must be positive for any landing gear and flap position appropriate to the takeoff, climb, cruise, and approach configurations. This must be shown with symmetrical power up to maximum continuous power, and at speeds from 1.2 V_{S1} up to the maximum allowable speed for the condition being investigated. The angle of skid for these tests must be appropriate to the type of airplane. At larger angles of skid up to that at which full rudder is used or a control force limit in § 23.143 is reached, whichever occurs first, and at speeds from 1.2 V_{S1} to V_A, the rudder pedal force must not reverse.

(2) The static lateral stability, as shown by the tendency to raise the low wing in a slip, must be positive for any landing gear and flap positions. This must be shown with symmetrical power up to 75 percent of maximum continuous power at speeds above 1.2 V_{S1}, up to the maximum allowable speed for the configuration being investigated. The static lateral stability may not be negative at 1.2 V_{S1}. The angle of slip for these tests must be appropriate to the type of airplane, but in no case may the slip angle be less than that obtainable with 10 degrees of bank.

(3) In straight, steady slips at 1.2 V_{s1} for any landing gear and flap positions, and for any symmetrical power conditions up to 50 percent of maximum continuous power, the aileron and rudder control movements and forces must increase steadily (but not necessarily in constant proportion) as the angle of slip is increased up to the maximum appropriate to the type of airplane. At larger slip angles up to the angle at which the full rudder or aileron control is used or a control force limit contained in § 23.143 is obtained, the rudder pedal force may not reverse. Enough bank must accompany slipping to hold a constant heading. Rapid entry into, or recovery from, a maximum slip may not result in uncontrollable flight characteristics.

(b) *Two-control (or simplified control) airplanes.* The stability requirements for two-control airplanes are as follows:

(1) The directional stability of the airplane must be shown by showing that, in each configuration, it can be rapidly rolled from a 45 degree bank in one direction to a 45 degree bank in the opposite direction without showing dangerous skid characteristics.

(2) The lateral stability of the airplane must be shown by showing that it will not assume a dangerous attitude or speed when the controls are abandoned for two minutes. This must be done in moderately smooth air with the airplane trimmed for straight level flight at 0.9 V_H or V_C, whichever is lower, with flaps and landing gear retracted, and with a rearward center of gravity.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964; 30 FR 258, Jan. 9, 1965, as amended by Amdt. 23-21, 43 FR 2318, Jan. 16, 1978]

§ 23.179 Instrumented stick force measurements.

Instrumented stick force measurements must be made unless—

(a) Changes in speed are clearly reflected by changes in stick forces; and

(b) The maximum forces obtained under §§ 23.173 and 23.175 are not excessive.

§ 23.181 Dynamic stability.

(a) Any short period oscillation not including combined lateral-directional oscillations occurring between the stalling speed and the maximum allowable speed appropriate to the configuration of the airplane must be heavily damped with the primary controls—

(1) Free; and

(2) In a fixed position.

(b) Any combined lateral-directional oscillations ("Dutch roll") occurring between the stalling speed and the maximum allowable speed appropriate to the configuration of the airplane must be damped to 1/10 amplitude in 7 cycles with the primary controls—

(1) Free; and

(2) In a fixed position.

[Amdt. 23-21, 43 FR 2318, Jan. 16, 1978]

STALLS

§ 23.201 Wings level stall.

(a) For an airplane with independently controlled roll and directional controls, it must be possible to produce and to correct roll by unreversed use of the rolling control and to produce and to correct yaw by unreversed use of the directional control, up to the time the airplane pitches.

(b) For an airplane with interconnected lateral and directional controls (2 controls) and for an airplane with only one of these controls, it must be possible to produce and correct roll by unreversed use of the rolling control without producing excessive yaw, up to the time the airplane pitches.

(c) The wing level stall characteristics of the airplane must be demonstrated in flight as follows: The airplane speed must be reduced with the elevator control until the speed is slightly above the stalling speed, then the elevator control must be pulled back so that the rate of speed reduction will not exceed one knot per second until a stall is produced, as shown by an uncontrollable downward pitching motion of the airplane, or until the control reaches the stop. Normal use of the elevator control for recovery is allowed after the pitching motion has unmistakably developed.

(d) Except where made inapplicable by the special features of a particular type of airplane, the following apply

to the measurement of loss of altitude during a stall:

(1) The loss of altitude encountered in the stall (power on or power off) is the change in altitude (as observed on the sensitive altimeter testing installation) between the altitude at which the airplane pitches and the altitude at which horizontal flight is regained.

(2) If power or thrust is required during stall recovery the power or thrust used must be that which would be used under the normal operating procedures selected by the applicant for this maneuver. However, the power used to regain level flight may not be applied until flying control is regained.

(e) During the recovery part of the maneuver, it must be possible to prevent more than 15 degrees of roll or yaw by the normal use of controls.

(f) Compliance with the requirements of this section must be shown under the following conditions:

(1) *Wing flaps:* Full up, full down, and intermediate, if appropriate.

(2) *Landing gear:* Retracted and extended.

(3) *Cowl flaps:* Appropriate to configuration.

(4) *Power:* Power or thrust off, and 75 percent maximum continuous power or thrust.

(5) *Trim:* 1.5 V_{s1} or at the minimum trim speed, whichever is higher.

(6) *Propeller:* Full increase rpm position for the power off condition.

[Amdt. 23-14, 38 FR 31820, Nov. 19, 1973]

§ 23.203 Turning flight and accelerated stalls.

Turning flight and accelerated stalls must be demonstrated in flight tests as follows:

(a) Establish and maintain a coordinated turn in a 30 degree bank. Reduce speed by steadily and progressively tightening the turn with the elevator until the airplane is stalled or until the elevator has reached its stop. The rate of speed reduction must be constant, and:

(1) For a turning flight stall, may not exceed one knot per second; and

(2) For an accelerated stall, be 3 to 5 knots per second with steadily increasing normal acceleration.

(b) When the stall has fully developed or the elevator has reached its stop, it must be possible to regain level flight without:

(1) Excessive loss of altitude;

(2) Undue pitchup;

(3) Uncontrollable tendency to spin;

(4) Exceeding 60 degree of roll in either direction from the established 30 degree bank; and

(5) For accelerated entry stalls, without exceeding the maximum permissible speed or the allowable limit load factor.

(c) Compliance with the requirements of this section must be shown with:

(1) *Wing flaps:* Retracted and fully extended for turning flight and accelerated entry stalls, and intermediate, if appropriate, for accelerated entry stalls;

(2) *Landing gear:* Retracted and extended;

(3) *Cowl flaps:* Appropriate to configuration;

(4) *Power:* 75 percent maximum continuous power; and

(5) *Trim:* 1.5 V_{s1}, or minimum trim speed, whichever is higher.

[Amdt. 23-14, 38 FR 31820, Nov. 19, 1973]

§ 23.205 Critical engine inoperative stalls.

(a) A multiengine airplane may not display any undue spinning tendency and must be safely recoverable without applying power to the inoperative engine when stalled. The operating engines may be throttled back during the recovery from stall.

(b) Compliance with paragraph (a) of the section must be shown with:

(1) *Wing flaps:* Retracted.

(2) *Landing gear:* Retracted.

(3) *Cowl flaps:* Appropriate to level flight critical engine inoperative.

(4) *Power:* Critical engine inoperative and the remaining engine(s) at 75 percent maximum continuous power or thrust or the power or thrust at which the use of maximum control travel just holds the wings laterally level in the approach to stall, whichever is lesser.

(5) *Propeller:* Normal inoperative position for the inoperative engine.

(6) *Trim:* Level flight, critical engine inoperative, except that for an air-

plane of 6,000 pounds or less maximum weight that has a stalling speed of 61 knots or less and cannot maintain level flight with the critical engine inoperative, the airplane must be trimmed for straight flight, critical engine inoperative, at a speed not greater than $1.5V_{s1}$.

[Amdt. 23-14, 36 FR 21820, Nov. 19, 1973]

§ 23.207 Stall warning.

(a) There must be a clear and distinctive stall warning, with the flaps and landing gear in any normal position, in straight and turning flight.

(b) The stall warning may be furnished either through the inherent aerodynamic qualities of the airplane or by a device that will give clearly distinguishable indications under expected conditions of flight. However, a visual stall warning device that requires the attention of the crew within the cockpit is not acceptable by itself.

(c) The stall warning must begin at a speed exceeding the stalling speed by a margin of not less than 5 knots, but not more than the greater of 10 knots or 15 percent of the stalling speed, and must continue until the stall occurs.

[Amdt. 23-7, 34 FR 13087, Aug. 13, 1969]

SPINNING

§ 23.221 Spinning.

(a) *Normal category.* A single-engine, normal category airplane must be able to recover from a one-turn spin or a 3-second spin, whichever takes longer, in not more than one additional turn, with the controls used in the manner normally used for recovery. In addition—

(1) For both the flaps-retracted and flaps-extended conditions, the applicable airspeed limit and positive limit maneuvering load factor may not be exceeded;

(2) There may be no excessive back pressure during the spin or recovery; and

(3) It must be impossible to obtain uncontrollable spins with any use of the controls.

For the flaps-extended condition, the flaps may be retracted during recovery.

(b) *Utility category.* A utility category airplane must meet the requirements of paragraph (a) of this section or the requirements of paragraph (c) of this section.

(c) *Acrobatic category.* An acrobatic category airplane must meet the following requirements:

(1) The airplane must recover from any point in a spin, in not more than one and one-half additional turns after normal recovery application of the controls. Prior to normal recovery application of the controls, the spin test must proceed for six turns or 3 seconds, whichever takes longer, with flaps retracted, and one turn or 3 seconds, whichever takes longer, with flaps extended. However, beyond 3 seconds, the spin may be discontinued when spiral characteristics appear with flaps retracted.

(2) For both the flaps-retracted and flaps-extended conditions, the applicable airspeed limit and positive limit maneuvering load factor may not be exceeded. For the flaps-extended condition, the flaps may be retracted during recovery, if a placard is installed prohibiting intentional spins with flaps extended.

(3) It must be impossible to obtain uncontrollable spins with any use of the controls.

(d) *Airplanes "characteristically incapable of spinning".* If it is desired to designate an airplane as "characteristically incapable of spinning", this characteristic must be shown with—

(1) A weight five percent more than the highest weight for which approval is requested;

(2) A center of gravity at least three percent aft of the rearmost position for which approval is requested;

(3) An available elevator up-travel four degrees in excess of that to which the elevator travel is to be limited for approval; and

(4) An available rudder travel seven degrees, in both directions, in excess of that to which the rudder travel is to be limited for approval.

[Doc. No. 4080, 29 FR 17955, Dec. 18, 1964, as amended by Amdt. 23-7, 34 FR 13087, Aug. 13, 1969]

Ground and Water Handling Characteristics

§ 23.231 Longitudinal stability and control.

(a) A landplane may have no uncontrollable tendency to nose over in any reasonably expected operating condition, including rebound during landing or takeoff. Wheel brakes must operate smoothly and may not induce any undue tendency to nose over.

(b) A seaplane or amphibian may not have dangerous or uncontrollable porpoising characteristics at any normal operating speed on the water.

§ 23.233 Directional stability and control.

(a) There may be no uncontrollable ground or water looping tendency in 90 degree cross winds, up to a wind velocity of 0.2 V_{so}, at any speed at which the airplane may be expected to be operated on the ground or water.

(b) A landplane must be satisfactorily controllable, without exceptional piloting skill or alertness, in power-off landings at normal landing speed, without using brakes or engine power to maintain a straight path.

(c) The airplane must have adequate directional control during taxiing.

§ 23.235 Taxiing condition.

The shock-absorbing mechanism may not damage the structure of the airplane when the airplane is taxied on the roughest ground that may reasonably be expected in normal operation.

§ 23.239 Spray characteristics.

Spray may not dangerously obscure the vision of the pilots or damage the propellers or other parts of a seaplane or amphibian at any time during taxiing, takeoff, and landing.

Miscellaneous Flight Requirements

§ 23.251 Vibration and buffeting.

Each part of the airplane must be free from excessive vibration under any appropriate speed and power conditions up to at least the minimum value of V_D allowed in § 23.335. In addition, there may be no buffeting, in any normal flight condition, severe enough to interfere with the satisfactory control of the airplane, cause excessive fatigue to the crew, or result in structural damage. Stall warning buffeting within these limits is allowable.

§ 23.253 High speed characteristics.

If a maximum operating speed V_{MO}/M_{MO} is established under § 23.1505(c), the following speed increase and recovery characteristics must be met:

(a) Operating conditions and characteristics likely to cause inadvertent speed increases (including upsets in pitch and roll) must be simulated with the airplane trimmed at any likely cruise speed up to V_{MO}/M_{MO}. These conditions and characteristics include gust upsets, inadvertent control movements, low stick force gradient in relation to control friction, passenger movement, leveling off from climb, and descent from Mach to airspeed limit altitude.

(b) Allowing for pilot reaction time after effective inherent or artificial speed warning occurs, it must be shown that the airplane can be recovered to a normal attitude and its speed reduced to V_{MO}/M_{MO}, without—

(1) Exceptional piloting strength or skill;

(2) Exceeding V_D/M_D, the maximum speed shown under § 23.251, or the structural limitations; or

(3) Buffeting that would impair the pilot's ability to read the instruments or to control the airplane for recovery.

(c) There may be no control reversal about any axis at any speed up to the maximum speed shown under § 23.251. Any reversal of elevator control force or tendency of the airplane to pitch, roll, or yaw must be mild and readily controllable, using normal piloting techniques.

[Amdt. 23-7, 34 FR 13087, Aug. 13, 1969; as amended by Amdt. 23-26, 45 FR 60170, Sept. 11, 1980]

Adoption of the Amendments

Accordingly, Parts 21, 23, 36, 91, and 135 of the Federal Aviation Regulations (14 CFR Parts 21, 23, 36, 91, and 135) are amended, as follows:

1. By amending Part 23 by revising the title to read as follows:

PART 23—AIRWORTHINESS STANDARDS: NORMAL, UTILITY, ACROBATIC, AND COMMUTER CATEGORY AIRPLANES

2. The authority citation for Part 23 continues to read as follows:

Authority: 49 U.S.C. 1344, 1354(a), 1355, 1421, 1423, 1425, 1428, 1429, 1430, and 1502; and 49 U.S.C. 106(g) (Revised, Public L. 97-449, January 12, 1983).

3. By amending § 23.1 by revising paragraph (a) to read as follows:

§ 23.1 Applicability.

(a) This part prescribes airworthiness standards for the issue of type certificates, and changes to those certificates, for airplanes in the normal, utility, acrobatic, and commuter categories.

* * * * *

4. By amending § 23.3 by revising paragraphs (a) introductory text, (b) introductory text, and (c); by revising and redesignating paragraph (d) as (e), and by adding a new paragraph (d) to read as follows:

§ 23.3 Airplane categories.

(a) The normal category is limited to airplanes that have a seating configuration, excluding pilot seats, of nine or less, a maximum certificate takeoff weight of 12,500 pounds or less, and intended for nonacrobatic operation. Nonacrobatic operation includes:

* * * * *

(b) The utility category is limited to airplanes that have a seating configuration, excluding pilot seats, of nine or less, a maximum certificated takeoff weight of 12,500 pounds or less, and intended for limited acrobatic operation. Airplanes certificated in the utility category may be used in any of the operations covered under paragraph (a) of this section and in limited acrobatic operations. Limited acrobatic operation includes:

* * * * *

(c) The acrobatic category is limited to airplanes that have a seating configuration, excluding pilot seats, of nine or less, a maximum certificated takeoff weight of 12,500 pounds or less, and intended for use without restrictions, other than those shown to be necessary as a result of required flight tests.

(d) The commuter category is limited to propeller-driven, multiengine airplanes that have a seating configuration excluding pilot seats, of 19 or less, and a maximum certificated takeoff weight of 19,000 pounds or less, intended for nonacrobatic operation as described in paragraph (a) of this section.

(e) Airplanes may be type certificated in more than one category of this part if the requirements of each requested category are met.

5. By amending § 23.25(a)(2) by inserting the words "and commuter" after the word "normal"; and by revising paragraph (a) introductory text to read as follows:

§ 23.25 Weight limits.

(a) *Maximum weight.* The maximum weight is the highest weight at which compliance with each applicable requirement of this Part (other than those complied with at the design landing weight) is shown. In addition, for commuter category airplanes, the applicant must establish a maximum zero fuel weight. The maximum weight must be established so that it is—

* * * * *

6. By amending § 23.45 by revising paragraph (a) and by adding a new paragraph (f) to read as follows:

§ 23.45 General.

(a) Unless otherwise prescribed, the performance requirements of this subpart must be met for still air; and

(1) Standard atmospheric conditions for normal, utility, and acrobatic category airplanes; or

(2) Ambient atmospheric conditions for commuter category airplanes.

* * * * *

(f) For commuter category airplanes, the following also apply:

(1) Unless otherwise prescribed, the applicant must select the takeoff, en route, approach, and landing configurations for the airplane;

(2) The airplane configuration may vary with weight, altitude, and temperature, to the extent they are compatible with the operating procedures required by paragraph (f)(3) of this section;

(3) Unless otherwise prescribed, in determining the critical-engine-inoperative takeoff performance, takeoff flight path, the accelerate-stop distance, takeoff distance, and landing distance, changes in the airplane's configuration, speed, power, and thrust must be made in accordance with procedures established by the applicant for operation in service;

(4) Procedures for the execution of missed approaches and balked landings associated with the conditions prescribed in §§ 23.67(e)(3) and 23.77(c) must be established; and

(5) The procedures established under paragraphs (f)(3) and (f)(4) of this section must—

(i) Be able to be consistently executed by a crew of average skill;

(ii) Use methods or devices that are safe and reliable; and

(iii) Include allowance for any reasonably expected time delays in the execution of the procedures.

7. By amending § 23.51 by removing paragraphs (b) and (c); by redesignating paragraphs (d) and (e) as (b) and (c) respectively; and by adding a new paragraph (d) to read as follows:

§ 23.51 Takeoff.

* * * * *

(d) For commuter category airplanes, takeoff performance and data as required by §§ 23.53 through 23.59 must be determined and included in the Airplane Flight Manual—

(1) For each weight, altitude, and ambient temperature within the operational limits selected by the applicant;

(2) For the selected configuration for takeoff;

(3) For the most unfavorable center of gravity position;

(4) With the operating engine within approved operating limitations;

(5) On a smooth, dry, hard surface runway; and

(6) Corrected for the following operational correction factors:

(i) Not more than 50 percent of nominal wind components along the takeoff path opposite to the direction of takeoff and not less than 150 percent of nominal wind components along the takeoff path in the direction of takeoff; and

(ii) Effective runway gradients.

8. By adding a new § 23.53 to read as follows:

§ 23.53 Takeoff speeds.

(a) For multiengine airplanes, the lift-off speed, V_{LOF}, may not be less than V_{MC} determined in accordance with § 23.149.

(b) Each normal, utility, and acrobatic category airplane, upon reaching a height of 50 feet above the takeoff surface level, must have reached a speed of not less than the following:

(1) For multiengine airplanes, the higher of—

(i) 1.1 V_{MC}; or

(ii) 1.3 V_{S1}, or any lesser speed, not less than V_X plus 4 knots, that is shown to be safe under all conditions, including turbulence and complete engine failure.

(2) For single engine airplanes—

(i) 1.3 V_{S1}; or

(ii) Any lesser speed, not less than V_X plus 4 knots, that is shown to be safer

under all conditions, including turbulence and complete engine failure.

(c) For commuter category airplanes, the following apply:

(1) The takeoff decision speed, V_1, is the calibrated airspeed on the ground at which, as a result of engine failure or other reasons, the pilot is assumed to have made a decision to continue or discontinue the takeoff. The takeoff decision speed, V_1, must be selected by the applicant but may not be less than the greater of the following:

(i) 1.10 V_{s1};

(ii) 1.10 V_{mc} established in accordance with § 23.149;

(iii) A speed at which the airplane can be rotated for takeoff and shown to be adequate to safely continue the takeoff, using normal piloting skill, when the critical engine is suddenly made inoperative; or

(iv) V_{EF} plus the speed gained with the critcial engine inoperative during the time interval between the instant that the critical engine is failed and the instant at which the pilot recognizes and reacts to the engine failure as indicated by the pilot's application of the first retarding means during the accelerate-stop determination of § 23.55.

(2) The takeoff safety speed, V_2, in terms of calibrated airspeed, must be selected by the applicant so as to allow the gradient of climb required in § 23.67 but must not be less than V_1 or less than $1.2VV_{s1}$.

(3) The critical engine failure speed, V_{EF}, is the calibrated airspeed at which the critical engine is assumed to fail. V_{EF} must be selected by the applicant but not less than V_{mc} determined in accordance with § 23.149.

(4) The rotation speed, V_R in terms of calibrated airspeed, must be selected by the applicant and may not be less than the greater of the following:

(i) V_1; or

(ii) The speed determined in accordance with § 23.57(c) that allows attaining the initial climb out speed, V_2, before reaching a height of 35 feet above the takeoff surface.

(5) For any given set of conditions, such as weight, altitude, configuration, and temperature, a single value of V_R must be used to show compliance with both the one-engine-inoperative takeoff and all-engines-operating takeoff requirements:

(i) One-engine-inoperative takeoff determined in accordance with § 23.57; and

(ii) All-engines-operating takeoff determined in accordance with § 23.59.

(6) The one-engine-inoperative takeoff distance, using a normal rotation rate at a speed of 5 knots less than V_R established in accordance with paragraphs (c)(4) and (5) of this section, must be shown not to exceed the corresponding one-engine-inoperative takeoff distance determined in accordance with §§ 23.57 and 23.59 using the established V_R. The take off distance determined in accordance with § 23.59 and the takeoff must be safely continued from the point at which the airplane is 35 feet above the takeoff surface at a speed not less than 5 knots less than the established V_2 speed.

(7) The applicant must show, with all engines operating, that marked increases in the scheduled takeoff distances determined in accordance with § 23.59 do not result from over-rotation of the airplane and out-of-trim conditions.

9. By adding a new § 23.55 to read as follows:

§ 23.55 Accelerate-stop distance.

For each commuter category airplane, the accelerate-stop distance must be determined as follows:

(a) The accelerate-stop distance is the sum of the distances necessary to—

(1) Accelerate the airplane from a standing start to V_1; and

(2) Come to a full stop from the point at which V_1 is reached assuming that in the case of engine failure, the pilot has decided to stop as indicated by application of the first retarding means at the speed V_1.

(b) Means other than wheel brakes may be used to determine the accelerate-stop distance if that means is available with the critical engine inoperative and if that means—

(1) Is safe and reliable;

(2) Is used so that consistent results can be expected under normal operating conditions; and

(3) Is such that exceptional skill is not required to control the airplane.

10. By adding a new § 23.57 to read as follows:

§ 23.57 Takeoff path.

For each commuter category airplane, the takeoff path is as follows:

(a) The takeoff path extends from a standing start to a point in the takeoff at which the airplane is 1,500 feet above the takeoff surface or at which the transition from the takeoff to the en route configuration is completed, whichever point is higher; and

(1) The takeoff path must be based on the procedures prescribed in § 23.45;

(2) The airplane must be accelerated on the ground to V_{EF} at which point the critical engine must be made inoperative and remain inoperative for the rest of the takeoff; and

(3) After reaching V_{EF}, the airplane must be accelerated to V_2.

(b) During the acceleration to speed V_2, the nose gear may be raised off the ground at a speed not less than V_R. However, landing gear retraction may not be initiated until the airplane is airborne.

(c) During the takeoff path determination, in accordance with paragraphs (a) and (b) of this section—

(1) The slope of the airborne part of the takeoff path must be positive at each point;

(2) The airplane must reach V_2 before it is 35 feet above the takeoff surface, and must continue at a speed as close as practical to, but not less than V_2, until it is 400 feet above the takeoff surface;

(3) At each point along the takeoff path, starting at the point at which the airplane reaches 400 feet above the takeoff surface, the available gradient of climb may not be less than—

(i) 1.2 percent for two-engine airplanes;

(ii) 1.5 percent for three-engine airplanes;

(iii) 1.7 percent for four-engine airplanes; and

(4) Except for gear retraction and automatic propeller feathering, the airplane configuration may not be changed, and no change in power or thrust that requires action by the pilot may be made, until the airplane is 400 feet above the takeoff surface.

(d) The takeoff path must be determined by a continuous demonstrated takeoff or by synthesis from segments. If the takeoff path is determined by the segmental method—

(1) The segments must be clearly defined and must be related to the distinct changes in the configuration, power or thrust, and speed;

(2) The weight of the airplane, the configuration, and the power or thrust must be constant throughout each segment and must correspond to the most critical condition prevailing in the segment;

(3) The flight path must be based on the airplane's performance without ground effect;

(4) The takeoff path data must be checked by continuous demonstrated takeoffs up to the point at which the airplane is out of ground effect and its speed is stabilized to ensure that the path is conservative relative to the continuous path; and

(5) The airplane is considered to be out of the ground effect when it reaches a height equal to its wing span.

11. By adding a new § 23.59 to read as follows:

§ 23.59 Takeoff distance and takeoff run.

For each commuter category airplane—

(a) Takeoff distance is the greater of—

(1) The horizontal distance along the takeoff path from the start of the takeoff to the point at which the airplane is 35 feet above the takeoff surface as determined under § 23.57; or

(2) With all engines operating, 115 percent of the horizontal distance along the takeoff path, with all engines operating, from the start of the takeoff to the point at which the airplane is 35 feet above the takeoff surface, as determined by a procedure consistent with § 23.57.

(b) If the takeoff distance includes a clearway, the takeoff run is the greater of—

(1) The horizontal distance along the takeoff path from the start of the takeoff to a point equidistant between the point at which V_{LOF} is reached and the point at which the airplane is 35 feet above the takeoff surface as determined under § 23.57; or

(2) With all engines operating, 115 percent of the horizontal distance along the takeoff path, with all engines operating, from the start of the takeoff to a point equidistant between the point at which V_{LOF} is reached and the point at which the airplane is 35 feet above the takeoff surface determined by a procedure consistent with § 23.57.

12. By adding a new § 23.61 to read as follows:

§ 23.61 Takeoff flight path.

For each commuter category airplane, the takeoff flight path must be determined as follows:

(a) The takeoff flight path begins 35 feet above the takeoff surface at the end of the takeoff distance determined in accordance with § 23.59.

(b) The net takeoff flight path data must be determined so that they represent the actual takeoff flight paths, as determined in accordance with § 23.57 and with paragraph (a) of this section, reduced at each point by a gradient of climb equal to—

(1) 0.8 percent for two-engine airplanes;

(2) 0.9 percent for three-engine airplanes; and

(3) 1.0 percent for four-engine airplanes.

(c) The prescribed reduction in climb gradient may be applied as an equivalent reduction in acceleration along that part of the takeoff flight path at which the airplane is accelerated in level flight.

13. By amending § 23.65 by adding a new paragraph (d) to read as follows:

§ 23.65 Climb: All engines operating.

* * * * *

(d) In addition for commuter category airplanes, performance data must be determined for variations in weight, altitude, and temperatures at the most critical center of gravity for which approval is requested.

14. By amending § 23.67 by inserting the words "normal, utility, and acrobatic category" before the word "reciprocating" in both paragraphs (a) and (b) and before the word "turbine" in paragraph (c); and by adding a new paragraph (e) to read as follows:

§ 23.67 Climb: One engine inoperative.

* * * * *

(e) For commuter category airplanes, the following apply:

(1) *Takeoff climb:* The maximum weight at which the airplane meets the minimum climb performance specified in paragraphs (i) and (ii) must be determined for each altitude and ambient temperature within the operating limitations established for the airplane, out of ground effect in free air, with the airplane in the takeoff configuration, with the most critical center of gravity, the critical engine inoperative, the remaining engines at the maximum takeoff power or thrust, and the propeller of the inoperative engine windmilling with the propeller controls in the normal position, except that, if an approved automatic propeller feathering system is installed, the propeller may be in the feathered position:

(i) *Takeoff, landing gear extended.* The minimum steady gradient of climb between the lift-off speed, V_{LOF}, and until the landing gear is retracted must be measurably positive for two-engine airplanes, not less than 0.3 percent for three-engine airplanes, or 0.5 percent for four-engine airplanes at all points along the flight path; and

(ii) *Takeoff, landing gear retracted.* The minimum steady gradient of climb must not be less than 2 percent for two-engine airplanes, 2.3 percent for three-engine airplanes, and 2.6 percent for four-engine airplanes at the speed V_2, until the airplane is 400 feet above the takeoff surface. For airplanes with fixed landing gear, this requirement must be met with the landing gear extended.

(2) *En route climb:* The maximum weight must be determined for each altitude and ambient temperature within the operational limits established for the airplane, at which the steady gradient of climb is not less than 1.2 percent for two-engine airplanes, 1.5 percent for three-engine airplanes, and 1.7 percent for four-engine airplanes at an altitude of 1,500 feet above the takeoff surface, with the airplane in the en route configuration, the critical engine inoperative, the remaining engine at the maximum continuous power or thrust, and the most unfavorable center of gravity.

(3) *Approach:* In the approach configuration corresponding to the normal all-engines-operating procedure in which V_{s_1} for this configuration does not exceed 110 percent of the V_{s_1} for the related landing configuration, the steady gradient of climb may not be less than 2.1 percent for two-engine airplanes, 2.4 percent for three-engine airplanes, and 2.7 percent for four-engine airplanes, with—

(i) The critical engine inoperative and the remaining engines at the available takeoff power or thrust;

(ii) The maximum landing weight; and

(iii) A climb speed established in connection with the normal landing procedures but not exceeding 1.5 V_{s_1}.

15. By amending § 23.75 by adding a new paragraph (g) to read as follows:

§ 23.75 Landing.

* * * * *

(g) In addition, for commuter category airplanes, the following apply:

(1) The landing distance must be determined for standard temperatures at each weight, altitude, and wind condition within the operational limits established by the applicant;

(2) A steady gliding approach, or a steady approach at a gradient of descent not greater than 5.2 percent (3°), at a calibrated airspeed not less than $1.3V_{s_1}$, must be maintained down to the 50-foot height; and

(3) The landing distance data must include correction factors for not more than 50 percent of the nominal wind components along the landing path opposite to the direction of landing and not less than 150 percent of the nominal wind components along the landing path in the direction of landing.

16. By amending § 23.77 by inserting the words "normal, utility, and acrobatic category" before the word "airplane"; and by adding an "s" to the word "airplane" in paragraph (a); by inserting the words "normal, utility, and acrobatic category" before the word "turbine"; by adding an "s" to the word "airplane" in the first part of the sentence in paragraph (b); and by adding a new paragraph (c) to read as follows:

§ 23.77 Balked landing.

* * * * *

(c) For each commuter category airplane, with all engines operating, the maximum weight must be determined with the airplane in the landing configuration for each altitude and ambient temperature within the operational limits established for the airplane, with the most unfavorable center of gravity and out-of-ground effect in free air, at which the steady gradient of climb will not be less than 3.3 percent with—

(1) The engines at the power or thrust that is available 8 seconds after initiation of movement of the power or thrust controls from the minimum flight-idle position to the takeoff position.

(2) A climb speed not greater than the approach speed established under § 23.75 and not less than the greater of 1.05 V_{mc} or $1.10V_{s_1}$.

17. By amending § 23.161 by revising paragraphs (b), (c) introductory test, and (c)(3) to read as follows:

§ 23.161 Trim.

* * * * *

(b) *Lateral and directional trim.* The airplane must maintain lateral and directional trim in level flight with the landing gear and wing flaps retracted as follows:

(1) For normal, utility, and acrobatic category airplanes, at a speed of $0.9V_H$ or V_C, whichever is lower; and

(2) For commuter category airplanes, at a speed of V_H or V_{MO}/M_{MO}, whichever is lower.

(c) *Longitudinal trim.* The airplane must maintain longitudinal trim under each of the following conditions, except that it need not maintain trim at a speed greater than V_{MO}/M_{MO}:

* * * * *

(3) Level flight at any speed with the landing gear and wing flaps retracted as follows:

(i) For normal, utility, and acrobatic category airplanes, at any speed from $0.9V_H$ to either V_X or $1.4V_{S_1}$; and

(ii) For commuter category airplanes, at a speed of V_H or V_{MO}/M_{MO}, whichever is lower, to either V_X or $1.4V_{S_1}$.

* * * * *

18. By amending § 23.173 by revising paragraph (b) to read as follows:

§ 23.173 Static longitudinal stability.

* * * * *

(b) The airspeed must return to within the tolerances specified for applicable categories of airplanes when the control force is slowly released at any speed within the speed range specified in paragraph (a) of this section. The applicable tolerances are—

(1) The airspeed must return to within plus or minus 10 percent of the original trim airspeed; and

(2) For commuter category airplanes, the airspeed must return to within plus or minus 7.5 percent of the original trim airspeed for the cruising condition specified in § 23.175(b).

* * * * *

19. By amending § 23.175 by revising paragraph (b)(2) introductory text to read as follows:

§ 23.175 Demonstration of static longitudinal stability.

* * * * *

(b) *Cruise—Landing gear retracted (or fixed gear).*

* * * * *

(2) *High speed cruise.* The stick force curve must have a stable slope at all speeds within a range that is the greater of 15 percent of the trim speed plus the resulting free return speed range or 40 knots plus the resulting free return speed range for normal, utility, and acrobatic category airplanes, above and below the trim speed. For commuter category airplanes, the stick force curve must have a stable slope for a speed range of 50 knots from the trim speed, except that the speeds need not exceed V_{FC}/M_{FC} or be less than $1.4\ V_{S_1}$ and this speed range is considered to begin at the outer extremes of the friction band with a stick force not to exceed 50 pounds. In addition, for commuter category airplanes, V_{FC}/M_{FC} may not be less than a speed midway between V_{MO}/M_{MO} and V_{DF}/M_{DF}, except that, for altitudes where Mach number is the limiting factor, M_{FC} need not exceed the Mach number at which effective speed warning occurs. These requirements for all categories of airplane must be met with—

* * * * *

A3. Federal Aviation Regulation: FAR 25

In this Section a summary is given of that part of FAR 25 which deals with performance and with flight characteristics.

Subpart A—General

§ 25.1 Applicability.

(a) This part prescribes airworthiness standards for the issue of type certificates, and changes to those certificates, for transport category airplanes.

(b) Each person who applies under Part 21 for such a certificate or change must show compliance with the applicable requirements in this part.

§ 25.2 Special retroactive requirements.

Notwithstanding §§ 21.17 and 21.101 of this chapter and irrespective of the date of application, each applicant for a type certificate and each applicant for a supplemental-type certificate (or an amendment to a type certificate) involving an increase in passenger seating capacity to a total greater than that for which the airplane has been type certificated, must show:

(a) After October 23, 1967, that the airplane concerned meets the requirements of §§ 25.783(g), 25.803(c) (2) through (9), 25.803(d), 25.807 (a), (c), and (d), 25.809 (f), and (h), 25.811 (a), (b), (d), (e), (f), and (g), 25.812(a)(1), (b), (c), (d), (e), (h), (i), (j), and (k) (1) and (2), 25.813 (a), (b), and (c), 25.815, 25.817, 25.853 (a) and (b), 25.855(a), 25.993(f), and 25.1359(c), in effect on October 24, 1967, or June 20, 1968, and

(b) After April 24, 1969, that the airplane concerned meets the requirements of §§ 25.721(d), 25.803(e), 25.811(c), 25.812 (a)(2), (f), (g), and (k)(3) in effect on October 24, 1967; and

(c) After April 23, 1969, that the airplane concerned meets the requirements of § 25.785(c) in effect either prior to or on that date; and

(d) After April 23, 1969, that the airplane concerned meets the requirements of §§ 25.803(b) and 25.803(c)(1) in effect on that date.

(Sec. 604, 72 Stat. 778; (49 U.S.C. 1424)

[Amdt. 25-15, 32 FR 13262, Sept. 20, 1967; 32 FR 13635, Sept. 29, 1967; Amdt. 25-17, 33 FR 9066, June 20, 1968; Amdt. 25-20, 34 FR 5544, Mar. 22, 1969]

Subpart B—Flight

General

§ 25.21 Proof of compliance.

(a) Each requirement of this subpart must be met at each appropriate combination of weight and center of gravity within the range of loading conditions for which certification is requested. This must be shown—

(1) By tests upon an airplane of the type for which certification is requested, or by calculations based on, and equal in accuracy to, the results of testing; and

(2) By systematic investigation of each probable combination of weight and center of gravity, if compliance cannot be reasonably inferred from combinations investigated.

(b) If there is less than a 2 knot difference in the forward and rearward c.g. stalling speeds, the flying qualities may be based upon the forward c.g. stalling speeds.

(c) The controllability, stability, trim, and stalling characteristics of the airplane must be shown for each altitude up to the maximum expected in operation.

(d) The following general tolerances from specified values are allowed during flight testing. However, greater tolerances may be allowed in particular tests. These tolerances are plus or minus variations unless otherwise noted in the particular test:

Item	Tolerance
Weight	+5%, −10%.
Critical items affected by weight.	+5%, −1%.
C.G.	7% total travel.
Airspeed	3 knots or 3%, whichever is higher
Power	5%.
Wind (takeoff and landing tests).	As low as possible but not to exceed approximately 12% V_{s_1} or 10.0 knots, whichever is lower, along the runway surface.

(e) If compliance with the flight characteristics requirements is dependent upon a stability augmentation system or upon any other automatic or power-operated system, compliance must be shown with §§ 25.671 and 25.672.

(f) In meeting the requirements of §§ 25.105(d), 25.125, 25.233, and 25.237, the wind velocity must be measured at a height of 10 meters above the surface, or corrected for the difference between the height at which the wind velocity is measured and the 10-meter height.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-42, 43 FR 2320, Jan. 16, 1978]

§ 25.23 Load distribution limits.

(a) Ranges of weights and centers of gravity within which the airplane may be safely operated must be established. If a weight and center of gravity combination is allowable only within certain load distribution limits (such as spanwise) that could be inadvertently exceeded, these limits and the corresponding weight and center of gravity combinations must be established.

(b) The load distribution limits may not exceed—

(1) The selected limits;

(2) The limits at which the structure is proven; or

(3) The limits at which compliance with each applicable flight requirement of this subpart is shown.

§ 25.25 Weight limits.

(a) *Maximum weights.* Maximum weights corresponding to the airplane operating conditions (such as ramp, ground or water taxi, takeoff, en route, and landing), environmental conditions (such as altitude and temperature), and loading conditions (such as zero fuel weight, center of gravity position and weight distribution) must be established so that they are not more than—

(1) The highest weight selected by the applicant for the particular conditions; or

(2) The highest weight at which compliance with each applicable structural loading and flight requirement is shown, except that for airplanes equipped with standby power rocket engines the maximum weight must not be more than the highest weight established in accordance with Appendix E of this part.

(b) *Minimum weight.* The minimum weight (the lowest weight at which compliance with each applicable requirement of this part is shown) must be established so that it is not less than—

(1) The lowest weight selected by the applicant;

(2) The design minimum weight (the lowest weight at which compliance with each structural loading condition of this part is shown); or

(3) The lowest weight at which compliance with each applicable flight requirement is shown.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970]

§ 25.27 Center of gravity limits.

The extreme forward and the extreme aft center of gravity limitations must be established for each practicably separable operating condition. No such limit may lie beyond—

(a) The extremes selected by the applicant;

(b) The extremes within which the structure is proven; or

(c) The extremes within which compliance with each applicable flight requirement is shown.

§ 25.29 Empty weight and corresponding center of gravity.

(a) The empty weight and corresponding center of gravity must be determined by weighing the airplane with—

(1) Fixed ballast;

(2) Unusable fuel determined under § 25.959; and

(3) Full operating fluids, including—

(i) Oil;

(ii) Hydraulic fluid; and

(iii) Other fluids required for normal operation of airplane systems, except potable water, lavatory precharge water, and water intended for injection in the engines.

(b) The condition of the airplane at the time of determining empty weight must be one that is well defined and can be easily repeated.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2320, Jan. 16, 1978]

§ 25.31 Removable ballast.

Removable ballast may be used on showing compliance with the flight requirements of this subpart.

§ 25.33 Propeller speed and pitch limits.

(a) The propeller speed and pitch must be limited to values that will ensure—

(1) Safe operation under normal operating conditions; and

(2) Compliance with the performance requirements of §§ 25.101 through 25.125.

(b) There must be a propeller speed limiting means at the governor. It must limit the maximum possible governed engine speed to a value not exceeding the maximum allowable r.p.m.

(c) The low pitch blade stop, or other means used to limit the low pitch position of the propeller blades, must be set so that the engine speed does not exceed 103 percent of the maximum allowable engine r.p.m. with—

(1) The propeller blades at the low pitch limit and governor inoperative; and

(2) Takeoff manifold pressure with the airplane stationary under standard atmospheric conditions.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-57, 49 FR 6848, Feb. 23, 1984]

PERFORMANCE

§ 25.101 General.

(a) Unless otherwise prescribed, airplanes must meet the applicable performance requirements of this subpart for ambient atmospheric conditions and still air.

(b) The performance, as affected by engine power or thrust, must be based on the following relative humidities:

(1) For turbine engine powered airplanes, a relative humidity of—

(i) 80 percent, at and below standard temperatures; and

(ii) 34 percent, at and above standard temperatures plus 50° F.

Between these two temperatures, the relative humidity must vary linearly.

(2) For reciprocating engine powered airplanes, a relative humidity of 80 percent in a standard atmosphere. Engine power corrections for vapor pressure must be made in accordance with the following table:

Altitude H (ft.)	Vapor pressure e (in. Hg.)	Specific humidity w (Lb. moisture per lb. dry air)	Density ratio $\rho/\sigma = 0.0023769$
0	0.403	0.00849	0.99506
1,000	.354	.00773	.96672
2,000	.311	.00703	.93885
3,000	.272	.00638	.91175
4,000	.238	.00578	.88514
5,000	.207	.00523	.85810
6,000	.1806	.00472	.83361
7,000	.1566	.00425	.80870
8,000	.1354	.00382	.78434
9,000	.1172	.00343	.76063
10,000	.1010	.00307	.73722
15,000	.0463	.001710	.62668
20,000	.01978	.000896	.53283
25,000	.00778	.000436	.44806

(c) The performance must correspond to the propulsive thrust available under the particular ambient atmospheric conditions, the particular flight condition, and the relative humidity specified in paragraph (b) of this section. The available propulsive thrust must correspond to engine

power or thrust, not exceeding the approved power or thrust less—

(1) Installation losses; and

(2) The power or equivalent thrust absorbed by the accessories and services appropriate to the particular ambient atmospheric conditions and the particular flight condition.

(d) Unless otherwise prescribed, the applicant must select the takeoff, en route, approach, and landing configurations for the airplane.

(e) The airplane configurations may vary with weight, altitude, and temperature, to the extent they are compatible with the operating procedures required by paragraph (f) of this section.

(f) Unless otherwise prescribed, in determining the accelerate-stop distances, takeoff flight paths, takeoff distances, and landing distances, changes in the airplane's configuration, speed, power, and thrust, must be made in accordance with procedures established by the applicant for operation in service.

(g) Procedures for the execution of balked landings and missed approaches associated with the conditions prescribed in §§ 25.119 and 25.121(d) must be established.

(h) The procedures established under paragraphs (f) and (g) of this section must—

(1) Be able to be consistently executed in service by crews of average skill;

(2) Use methods or devices that are safe and reliable; and

(3) Include allowance for any time delays, in the execution of the procedures, that may reasonably be expected in service.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-38, 41 FR 55466, Dec. 20, 1976]

§ 25.103 Stalling speed.

(a) V_S is the calibrated stalling speed, or the minimum steady flight speed, in knots, at which the airplane is controllable, with—

(1) Zero thrust at the stalling speed, or, if the resultant thrust has no appreciable effect on the stalling speed, with engines idling and throttles closed;

(2) Propeller pitch controls (if applicable) in the position necessary for compliance with paragraph (a)(1) of this section and the airplane in other respects (such as flaps and landing gear) in the condition existing in the test in which V_S is being used;

(3) The weight used when V_S is being used as a factor to determine compliance with a required performance standard; and

(4) The most unfavorable center of gravity allowable.

(b) The stalling speed V_S is the minimum speed obtained as follows:

(1) Trim the airplane for straight flight at any speed not less than 1.2 V_S or more than 1.4 V_S At a speed sufficiently above the stall speed to ensure steady conditions, apply the elevator control at a rate so that the airplane speed reduction does not exceed one knot per second.

(2) Meet the flight characteristics provisions of § 25.203.

§ 25.105 Takeoff.

(a) The takeoff speeds described in § 25.107, the accelerate-stop distance described in § 25.109, the takeoff path described in § 25.111, and the takeoff distance and takeoff run described in § 25.113, must be determined—

(1) At each weight, altitude, and ambient temperature within the operational limits selected by the applicant; and

(2) In the selected configuration for takeoff.

(b) No takeoff made to determine the data required by this section may require exceptional piloting skill or alertness.

(c) The takeoff data must be based on—

(1) A smooth, dry, hard-surfaced runway, in the case of land planes and amphibians;

(2) Smooth water, in the case of seaplanes and amphibians; and

(3) Smooth, dry snow, in the case of skiplanes.

(d) The takeoff data must include, within the established operational limits of the airplane, the following operational correction factors:

(1) Not more than 50 percent of nominal wind components along the takeoff path opposite to the direction of takeoff, and not less than 150 per-

cent of nominal wind components along the takeoff path in the direction of takeoff.

(2) Effective runway gradients.

§ 25.107 Takeoff speeds.

(a) V_1 must be established in relation to V_{EF} as follows:

(1) V_{EF} is the calibrated airspeed at which the critical engine is assumed to fail. V_{EF} must be selected by the applicant, but may not be less than VMCG determined under § 25.149(e).

(2) V_1, in terms of calibrated airspeed, is the takeoff decision speed selected by the applicant; however, V_1 may not be less than V_{EF} plus the speed gained with the critical engine inoperative during the time interval between the instant at which the critical engine is failed, and the instant at which the pilot recognizes and reacts to the engine failure, as indicated by the pilot's application of the first retarding means during accelerate-stop tests.

(b) V_{MU}, in terms of calibrated airspeed, may not be less than—

(1) 1.2 V_2 for—

(i) Two-engine and three-engine turbopropeller and reciprocating engine powered airplanes; and

(ii) Turbojet powered airplanes without provisions for obtaining a significant reduction in the one-engine-inoperative power-on stalling speed;

(2) 1.15 V_2 for—

(i) Turbopropeller and reciprocating engine powered airplanes with more than three engines; and

(ii) Turbojet powered airplanes with provisions for obtaining a significant reduction in the one-engine-inoperative power-on stalling speed; and

(3) 1.10 times V_{MC} established under § 25.149.

(c) V_2, in terms of calibrated airspeed, must be selected by the applicant to provide at least the gradient of climb required by § 25.121(b) but may not be less than—

(1) V_{MU}, and

(2) V_R plus the speed increment attained (in accordance with § 25.111 (c)(2)) before reaching a height of 35 feet above the takeoff surface.

(d) V_{MU} is the calibrated airspeed at and above which the airplane can safely lift off the ground, and continue the takeoff. V_{MU} speeds must be selected by the applicant throughout the range of thrust-to-weight ratios to be certificated. These speeds may be established from free air data if these data are verified by ground takeoff tests.

(e) *VR*, in terms of calibrated airspeed, must be selected in accordance with the conditions of paragraphs (e)(1) through (4) of this section:

(1) V_R may not be less than—

(i) V_1;

(ii) 105 percent of V_{MC};

(iii) The speed (determined in accordance with § 25.111(c)(2)) that allows reaching V_2 before reaching a height of 35 feet above the takeoff surface; or

(iv) A speed that, if the airplane is rotated at its maximum practicable rate, will result in a V_{LOF} of not less than 110 percent of V_{MU} in the all-engines-operating condition and not less than 105 percent of V_{MU} determined at the thrust-to-weight ratio corresponding to the one-engine-inoperative condition.

(2) For any given set of conditions (such as weight, configuration, and temperature), a single value of V_R, obtained in accordance with this paragraph, must be used to show compliance with both the one-engine-inoperative and the all-engines-operating takeoff provisions.

(3) It must be shown that the one-engine-inoperative takeoff distance, using a rotation speed of 5 knots less than V_R established in accordance with paragraphs (e)(1) and (2) of this section, does not exceed the corresponding one-engine-inoperative takeoff distance using the established V_R. The takeoff distances must be determined in accordance with § 25.113(a)(1).

(4) Reasonably expected variations in service from the established takeoff procedures for the operation of the airplane (such as over-rotation of the airplane and out-of-trim conditions) may not result in unsafe flight characteristics or in marked increases in the scheduled takeoff distances established in accordance with § 25.113(a).

(f) V_{LOF} is the calibrated airspeed at which the airplane first becomes airborne.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-38, 41 FR 55466, Dec. 20, 1976; Amdt. 25-42, 43 FR 2320, Jan. 16, 1978]

§ 25.109 Accelerate-stop distance.

(a) The accelerate-stop distance is the greater of the following distances:

(1) The sum of the distances necessary to—

(i) Accelerate the airplane from a standing start to V_{EF} with all engines operating;

(ii) Accelerate the airplane from V_{EF} to V_1 and continue the acceleration for 2.0 seconds after V_1 is reached, assuming the critical engine fails at V_{EF}; and

(iii) Come to a full stop from the point reached at the end of the acceleration period prescribed in paragraph (a)(1)(ii) of this section, assuming that the pilot does not apply any means of retarding the airplane until that point is reached and that the critical engine is still inoperative.

(2) The sum of the distances necessary to—

(i) Accelerate the airplane from a standing start to V_1 and continue the acceleration for 2.0 seconds after V_1 is reached with all engines operating; and

(ii) Come to a full stop from the point reached at the end of the acceleration period prescribed in paragraph (a)(2)(i) of this section, assuming that the pilot does not apply any means of retarding the airplane until that point is reached and that all engines are still operating.

(b) Means other than wheel brakes may be used to determine the accelerate-stop distance if that means—

(1) Is safe and reliable;

(2) Is used so that consistent results can be expected under normal operating conditions; and

(3) Is such that exceptional skill is not required to control the airplane.

(c) The landing gear must remain extended throughout the accelerate-stop distance.

(d) If the accelerate-stop distance includes a stopway with surface characteristics substantially different from those of a smooth hard-surfaced runway, the takeoff data must include operational correction factors for the accelerate-stop distance. The correction factors must account for the particular surface characteristics of the stopway and the variations in these characteristics with seasonal weather conditions (such as temperature, rain, snow, and ice) within the established operational limits.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2321, Jan. 16, 1978]

§ 25.111 Takeoff path.

(a) The takeoff path extends from a standing start to a point in the takeoff at which the airplane is 1,500 feet above the takeoff surface, or at which the transition from the takeoff to the en route configuration is completed and a speed is reached at which compliance with § 25.121(c) is shown, whichever point is higher. In addition—

(1) The takeoff path must be based on the procedures prescribed in § 25.101(c);

(2) The airplane must be accelerated on the ground to V_{EF}, at which point the critical engine must be made inoperative and remain inoperative for the rest of the takeoff; and

(3) After reaching V_{EF}, the airplane must be accelerated to V_2.

(b) During the acceleration to speed V_2, the nose gear may be raised off the ground at a speed not less than V_R. However, landing gear retraction may not be begun until the airplane is airborne.

(c) During the takeoff path determination in accordance with paragraphs (a) and (b) of this section—

(1) The slope of the airborne part of the takeoff path must be positive at each point;

(2) The airplane must reach V_2 before it is 35 feet above the takeoff surface and must continue at a speed as close as practical to, but not less than V_2, until it is 400 feet above the takeoff surface;

(3) At each point along the takeoff path, starting at the point at which the airplane reaches 400 feet above the takeoff surface, the available gradient of climb may not be less than—

(i) 1.2 percent for two-engine airplanes;

(ii) 1.5 percent for three-engine airplanes; and

(iii) 1.7 percent for four-engine airplanes; and

(4) Except for gear retraction and propeller feathering, the airplane configuration may not be changed, and no change in power or thrust that requires action by the pilot may be made, until the airplane is 400 feet above the takeoff surface.

(d) The takeoff path must be determined by a continuous demonstrated takeoff or by synthesis from segments. If the takeoff path is determined by the segmental method—

(1) The segments must be clearly defined and must be related to the distinct changes in the configuration, power or thrust, and speed;

(2) The weight of the airplane, the configuration, and the power or thrust must be constant throughout each segment and must correspond to the most critical condition prevailing in the segment;

(3) The flight path must be based on the airplane's performance without ground effect; and

(4) The takeoff path data must be checked by continuous demonstrated takeoffs up to the point at which the airplane is out of ground effect and its speed is stabilized, to ensure that the path is conservative relative to the continous path.

The airplane is considered to be out of the ground effect when it reaches a height equal to its wing span.

(e) For airplanes equipped with standby power rocket engines, the takeoff path may be determined in accordance with section II of Appendix E.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-6, 30 FR 8468, July 2, 1965; Amdt. 25-42, 43 FR 2321, Jan. 16, 1978; Amdt. 25-54, 45 FR 60172, Sept. 11, 1980]

§ 25.113 Takeoff distance and takeoff run.

(a) Takeoff distance is the greater of—

(1) The horizontal distance along the takeoff path from the start of the takeoff to the point at which the airplane is 35 feet above the takeoff surface, determined under § 25.111; or

(2) 115 percent of the horizontal distance along the takeoff path, with all engines operating, from the start of the takeoff to the point at which the airplane is 35 feet above the takeoff surface, as determined by a procedure consistent with § 25.111.

(b) If the takeoff distance includes a clearway, the takeoff run is the greater of—

(1) The horizontal distance along the takeoff path from the start of the takeoff to a point equidistant between the point at which V_{LOF} is reached and the point at which the airplane is 35 feet above the takeoff surface, as determined under § 25.111; or

(2) 115 percent of the horizontal distance along the takeoff path, with all engines operating, from the start of the takeoff to a point equidistant between the point at which V_{LOF} is reached and the point at which the airplane is 35 feet above the takeoff surface, determined by a procedure consistent with § 25.111.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970]

§ 25.115 Takeoff flight path.

(a) The takeoff flight path begins 35 feet above the takeoff surface at the end of the takeoff distance determined in accordance with § 25.113(a).

(b) The net takeoff flight path data must be determined so that they represent the actual takeoff flight paths (determined in accordance with § 25.111 and with paragraph (a) of this section) reduced at each point by a gradient of climb equal to—

(1) 0.8 percent for two-engine airplanes;

(2) 0.9 percent for three-engine airplanes; and

(3) 1.0 percent for four-engine airplanes.

(c) The prescribed reduction in climb gradient may be applied as an equivalent reduction in acceleration along that part of the takeoff flight path at which the airplane is accelerated in level flight.

§ 25.117 Climb: general.

Compliance with the requirements of §§ 25.119 and 25.121 must be shown at each weight, altitude, and ambient temperature within the operational limits established for the airplane and with the most unfavorable center of gravity for each configuration.

§ 25.119 Landing climb: All-engine-operating.

In the landing configuration, the steady gradient of climb may not be less than 3.2 percent, with—

(a) The engines at the power or thrust that is available eight seconds after initiation of movement of the power or thrust controls from the minimum flight idle to the takeoff position; and

(b) A climb speed of not more than 1.3 V_s.

§ 25.121 Climb: One-engine-inoperative.

(a) *Takeoff; landing gear extended.* In the critical takeoff configuration existing along the flight path (between the points at which the airplane reaches V_{LOF} and at which the landing gear is fully retracted) and in the configuration used in § 25.111 but without ground effect, the steady gradient of climb must be positive for two-engine airplanes, and not less than 0.3 percent for three-engine airplanes or 0.5 percent for four-engine airplanes, at V_{LOF} and with—

(1) The critical engine inoperative and the remaining engines at the power or thrust available when retraction of the landing gear is begun in accordance with § 25.111 unless there is a more critical power operating condition existing later along the flight path but before the point at which the landing gear is fully retracted; and

(2) The weight equal to the weight existing when retraction of the landing gear is begun, determined under § 25.111.

(b) *Takeoff; landing gear retracted.* In the takeoff configuration existing at the point of the flight path at which the landing gear is fully retracted, and in the configuration used in § 25.111 but without ground effect, the steady gradient of climb may not be less than 2.4 percent for two-engine airplanes, 2.7 percent for three-engine airplanes, and 3.0 percent for four-engine airplanes, at V_s and with—

(1) The critical engine inoperative, the remaining engines at the takeoff power or thrust available at the time the landing gear is fully retracted, determined under § 25.111, unless there is a more critical power operating condition existing later along the flight path but before the point where the airplane reaches a height of 400 feet above the takeoff surface; and

(2) The weight equal to the weight existing when the airplane's landing gear is fully retracted, determined under § 25.111.

(c) *Final takeoff.* In the en route configuration at the end of the takeoff path determined in accordance with § 25.111, the steady gradient of climb may not be less than 1.2 percent for two-engine airplanes, 1.5 percent for three-engine airplanes, and 1.7 percent for four-engine airplanes, at not less than 1.25 V_s and with—

(1) The critical engine inoperative and the remaining engines at the available maximum continuous power or thrust; and

(2) The weight equal to the weight existing at the end of the takeoff path, determined under § 25.111.

(d) *Approach.* In the approach configuration corresponding to the normal all-engines-operating procedure in which V_s for this configuration does not exceed 110 percent of the V_s for the related landing configuration, the steady gradient of climb may not be less than 2.1 percent for two-engine airplanes, 2.4 percent for three-engine airplanes, and 2.7 percent for four-engine airplanes, with—

(1) The critical engine inoperative, the remaining engines at the available takeoff power or thrust;

(2) The maximum landing weight; and

(3) A climb speed established in connection with normal landing procedures, but not exceeding 1.5 V_S.

§ 25.123 En route flight paths.

(a) For the en route configuration, the flight paths prescribed in paragraphs (b) and (c) of this section must be determined at each weight, altitude, and ambient temperature, within the operating limits established for the airplane. The variation of weight along the flight path, accounting for the progressive consumption of fuel and oil by the operating engines, may be included in the computation. The flight paths must be determined at any selected speed, with—

(1) The most unfavorable center of gravity;

(2) The critical engines inoperative;

(3) The remaining engines at the available maximum continuous power or thrust; and

(4) The means for controlling the engine-cooling air supply in the position that provides adequate cooling in the hot-day condition.

(b) The one-engine-inoperative net flight path data must represent the actual climb performance diminished by a gradient of climb of 1.1 percent for two-engine airplanes, 1.4 percent for three-engine airplanes, and 1.6 percent for four-engine airplanes.

(c) For three- or four-engine airplanes, the two-engine-inoperative net flight path data must represent the actual climb performance diminished by a gradient of climb of 0.3 percent for three-engine airplanes and 0.5 percent for four-engine airplanes.

§ 25.125 Landing.

(a) The horizontal distance necessary to land and to come to a complete stop (or to a speed of approximately 3 knots for water landings) from a point 50 feet above the landing surface must be determined (for standard temperatures, at each weight, altitude, and wind within the operational limits established by the applicant for the airplane) as follows:

(1) The airplane must be in the landing configuration.

(2) A steady gliding approach, with a calibrated airspeed of not less than 1.3 V_S, must be maintained down to the 50 foot height.

(3) Changes in configuration, power or thrust, and speed, must be made in accordance with the established procedures for service operation.

(4) The landing must be made without excessive vertical acceleration, tendency to bounce, nose over, ground loop, porpoise, or water loop.

(5) The landings may not require exceptional piloting skill or alertness.

(b) For landplanes and amphibians, the landing distance on land must be determined on a level, smooth, dry, hard-surfaced runway. In addition—

(1) The pressures on the wheel braking systems may not exceed those specified by the brake manufacturer;

(2) The brakes may not be used so as to cause excessive wear of brakes or tires; and

(3) Means other than wheel brakes may be used if that means—

(i) Is safe and reliable;

(ii) Is used so that consistent results can be expected in service; and

(iii) Is such that exceptional skill is not required to control the airplane.

(c) For seaplanes and amphibians, the landing distance on water must be determined on smooth water.

(d) For skiplanes, the landing distance on snow must be determined on smooth, dry, snow.

(e) The landing distance data must include correction factors for not more than 50 percent of the nominal wind components along the landing path opposite to the direction of landing, and not less than 150 percent of the nominal wind components along the landing path in the direction of landing.

(f) If any device is used that depends on the operation of any engine, and if the landing distance would be noticeably increased when a landing is made with that engine inoperative, the landing distance must be determined with that engine inoperative unless the use of compensating means will result in a landing distance not more than that with each engine operating.

CONTROLLABILITY AND MANEUVERABILITY

§ 25.143 General.

(a) The airplane must be safely controllable and maneuverable during—

(1) Takeoff;

(2) Climb;

(3) Level flight;

(4) Descent; and

(5) Landing.

(b) It must be possible to make a smooth transition from one flight condition to any other flight condition without exceptional piloting skill, alertness, or strength, and without danger of exceeding the airplane limit-load factor under any probable operating conditions, including—

(1) The sudden failure of the critical engine;

(2) For airplanes with three or more engines, the sudden failure of the second critical engine when the airplane is in the en route, approach, or landing configuration and is trimmed with the critical engine inoperative; and

(3) Configuration changes, including deployment or retraction of deceleration devices.

(c) If, during the testing required by paragraphs (a) and (b) of this section, marginal conditions exist with regard to required pilot strength, the "strength of pilots" limits may not exceed the limits prescribed in the following table:

Values in pound of force as applied to the control wheel or rudder pedals	Pitch	Roll	Yaw
For temporary application	75	40	150
For prolonged application	10	5	20

(d) In showing the temporary control force limitations of paragraph (c) of this section, approved operating procedures or conventional operating practices must be followed (including being as nearly trimmed as possible at the next preceding steady flight condition, except that, in the case of takeoff, the airplane must be trimmed in accordance with approved operating procedures).

(e) For the purpose of complying with the prolonged control force limitations of paragraph (c) of this section, the airplane must be as nearly trimmed as possible.

(Secs. 313(a), 601 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655 (c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2321, Jan. 16, 1978]

§ 25.145 Longitudinal control.

(a) It must be possible at any speed between the trim speed prescribed in § 25.49(c)(2)(i) and V_{S1}, (for reciprocating engine powered airplanes), or at any speed between the trim speed prescribed in § 25.103(b)(1) and V_S (for turbine engine powered airplanes), to pitch the nose downward so that the acceleration to this selected trim speed is prompt with—

(1) The airplane trimmed at the trim speed prescribed in § 25.49(c)(2)(i) (for reciprocating engine powered airplanes), or in § 25.103(b)(1) (for turbine engine powered airplanes);

(2) The landing gear extended;

(3) The wing flaps (i) retracted and (ii) extended; and

(4) Power (i) off and (ii) at maximum continuous power on the engines.

(b) With the landing gear extended, no change in trim control, or exertion of more than 50 pounds control force (representative of the maximum temporary force that readily can be applied by one hand) may be required for the following maneuvers:

(1) With power off, flaps retracted, and the airplane trimmed at 1.4 V_{S1}, extend the flaps as rapidly as possible while maintaining the airspeed at approximately 40 percent above the stalling speed existing at each instant throughout the maneuver.

(2) Repeat paragraph (b)(1) except initially extend the flaps and then retract them as rapidly as possible.

(3) Repeat paragraph (b)(2) except with takeoff power.

(4) With power off, flaps retracted, and the airplane trimmed at 1.4 V_{S1}, apply takeoff power rapidly while maintaining the same airspeed.

(5) Repeat paragraph (b)(4) except with flaps extended.

(6) With power off, flaps extended, and the airplane trimmed at 1.4 V_{s1}, obtain and maintain airspeeds between 1.1 V_{s1}, and either 1.7 V_{s1}, or V_{fe}, whichever is lower.

(c) It must be possible, without exceptional piloting skill, to prevent loss of altitude when complete retraction of the high lift devices from any position is begun during steady, straight, level flight at 1.1 V_{s1} for propeller powered airplanes, or 1.2 V_{s1} for turbojet powered airplanes, with—

(1) Simultaneous application of not more than takeoff power taking into account the critical engine operating conditions;

(2) The landing gear extended; and

(3) The critical combinations of landing weights and altitudes.

If gated high-lift device control positions are provided, retraction must be shown from any position from the maximum landing position to the first gated position, between gated positions, and from the last gated position to the full retraction position. In addition, the first gated control position from the landing position must correspond with the high-lift devices configuration used to establish the go-around procedure from the landing configuration. Each gated control position must require a separate and distinct motion of the control to pass through the gated position and must have features to prevent inadvertent movement of the control through the gated position.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970]

§ 25.147 Directional and lateral control.

(a) *Directional control; general.* It must be possible, while holding the wings approximately level, to safely make reasonably sudden changes in heading in both directions. This must be shown at 1.4 V_{s1} for heading changes up to 15° (except that the heading change at which the rudder pedal force is 150 pounds need not be exceeded), and with—

(1) The critical engine inoperative and its propeller in the minimum drag position;

(2) The power required for level flight at 1.4 V_{s1}, but not more than maximum continuous power;

(3) The most unfavorable center of gravity;

(4) Landing gear retracted;

(5) Flaps in the approach position; and

(6) Maximum landing weight.

(b) *Directional control; airplanes with four or more engines.* Airplanes with four or more engines must meet the requirements of paragraph (a) of this section except that—

(1) The two critical engines must be inoperative with their propellers (if applicable) in the minimum drag position;

(2) The center of gravity must be in the most forward position; and

(3) The flaps must be in the most favorable climb position.

(c) *Lateral control; general.* It must be possible to make 20° banked turns, with and against the inoperative engine, from steady flight at a speed equal to 1.4 V_{s1}, with—

(1) The critical engine inoperative and its propeller (if applicable) in the minimum drag position;

(2) The remaining engines at maximum continuous power;

(3) The most unfavorable center of gravity;

(4) Landing gear (i) retracted and (ii) extended;

(5) Flaps in the most favorable climb position; and

(6) Maximum takeoff weight.

(d) *Lateral control; airplanes with four or more engines.* Airplanes with four or more engines must be able to make 20° banked turns, with and against the inoperative engines, from steady flight at a speed equal to 1.4 V_{s1}, with maximum continuous power, and with the airplane in the configuration prescribed by paragraph (b) of this section.

(e) *Lateral control; all engines operating.* With the engines operating, roll response must allow normal maneuvers (such as recovery from upsets produced by gusts and the initiation of evasive maneuvers). There must be enough excess lateral control in sideslips (up to sideslip angles that might be required in normal operation), to allow a limited amount of maneuver-

ing and to correct for gusts. Lateral control must be enough at any speed up to V_{FC}/M_{FC} to provide a peak roll rate necessary for safety, without excessive control forces or travel.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655 (c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2321, Jan. 16, 1978]

§ 25.149 Minimum control speed.

(a) In establishing the minimum control speeds required by this section, the method used to simulate critical engine failure must represent the most critical mode of powerplant failure with respect to controllability expected in service.

(b) V_{MC} is the calibrated airspeed, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than five degrees.

(c) V_{MC} may not exceed 1.2 V_S with—

(1) Maximum available takeoff power or thrust on the engines;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for takeoff;

(4) The maximum sea level takeoff weight (or any lesser weight necessary to show V_{MC});

(5) The airplane in the most critical takeoff configuration existing along the flight path after the airplane becomes airborne, except with the landing gear retracted;

(6) The airplane airborne and the ground effect negligible; and

(7) If applicable, the propeller of the inoperative engine—

(i) Windmilling;

(ii) In the most probable position for the specific design of the propeller control; or

(iii) Feathered, if the airplane has an automatic feathering device acceptable for showing compliance with the climb requirements of § 25.121.

(d) The rudder forces required to maintain control at V_{MC} may not exceed 150 pounds nor may it be necessary to reduce power or thrust of the operative engines. During recovery, the airplane may not assume any dangerous attitude or require exceptional piloting skill, alertness, or strength to prevent a heading change of more than 20 degrees.

(e) V_{MCG}, the minimum control speed on the ground, is the calibrated airspeed during the takeoff run, at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with the use of primary aerodynamic controls alone (without the use of nosewheel steering) to enable the takeoff to be safely continued using normal piloting skill and rudder control forces not exceeding 150 pounds. In the determination of V_{MCG}, assuming that the path of the airplane accelerating with all engines operating is along the centerline of the runway, its path from the point at which the critical engine is made inoperative to the point at which recovery to a direction parallel to the centerline is completed may not deviate more than 30 feet laterally from the centerline at any point. V_{MCG} must be established with—

(1) The airplane in each takeoff configuration or, at the option of the applicant, in the most critical takeoff configuration;

(2) Maximum available takeoff power or thrust on the operating engines;

(3) The most unfavorable center of gravity;

(4) The airplane trimmed for takeoff; and

(5) The most unfavorable weight in the range of takeoff weights.

(f) V_{MCL}, the minimum control speed during landing approach with all engines operating, is the calibrated airspeed at which, when the critical engine is suddenly made inoperative, it is possible to recover control of the airplane with that engine still inoperative, and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL} must be established with—

(1) The airplane in the most critical configuration for approach with all engines operating;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for approach with all engines operating;

(4) The maximum sea level landing weight (or any lesser weight necessary to show V_{MCL}); and

(5) Maximum available takeoff power or thrust on the operating engines.

(g) For airplanes with three or more engines, V_{MCL-2}, the minimum control speed during landing approach with one critical engine inoperative, is the calibrated airspeed at which, when a second critical engine is suddenly made inoperative, it is possible to recover control of the airplane with both engines still inoperative and maintain straight flight either with zero yaw or, at the option of the applicant, with an angle of bank of not more than 5 degrees. V_{MCL-2} must be established with—

(1) The airplane in the most critical configuration for approach with the critical engine inoperative;

(2) The most unfavorable center of gravity;

(3) The airplane trimmed for approach with the critical engine inoperative;

(4) The maximum sea level landing weight (or any lesser weight necessary to show V_{MCL-2});

(5) The power or thrust on the operating engines required to maintain an approach path angle of 3 degrees when one critical engine is inoperative; and

(6) The power or thrust on the operating engines rapidly changed, immediately after the second critical engine is made inoperative, from the power or thrust prescribed in paragraph (g)(5) of this section to—

(i) Minimum available power or thrust; and

(ii) Maximum available takeoff power or thrust.

(h) The rudder control forces required to maintain control at V_{MCL} and V_{MCL-2} may not exceed 150 pounds, nor may it be necessary to reduce the power or thrust of the operating engines. In addition, the airplane may not assume any dangerous attitudes or require exceptional piloting skill, alertness, or strength to prevent a divergence in the approach flight path that would jeopardize continued safe approach when—

(1) The critical engine is suddenly made inoperative; and

(2) For the determination of V_{MCL-2}, the power or thrust on the operating engines is changed in accordance with paragraph (g)(6) of this section.

(Secs. 313(a), 601, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2331, Jan. 16, 1978]

TRIM

§ 25.161 Trim.

(a) *General.* Each airplane must meet the trim requirements of this section after being trimmed, and without further pressure upon, or movement of, either the primary controls or their corresponding trim controls by the pilot or the automatic pilot.

(b) *Lateral and directional trim.* The airplane must maintain lateral and directional trim with the most adverse lateral displacement of the center of gravity within the relevant operating limitations, during normally expected conditions of operation (including operation at any speed from 1.4 V_{S1} to V_{MO}/M_{MO}).

(c) *Longitudinal trim.* The airplane must maintain longitudinal trim during—

(1) A climb with maximum continuous power at a speed not more than 1.4 V_{S1}, with the landing gear retracted, and the flaps (i) retracted and (ii) in the takeoff position;

(2) A glide with power off at a speed not more than 1.4 V_{S1}, with the landing gear extended, the wing flaps (i) retracted and (ii) extended, the most unfavorable center of gravity position approved for landing with the maximum landing weight, and with the most unfavorable center of gravity position approved for landing regardless of weight; and

(3) Level flight at any speed from 1.4 V_{S1}, to V_{MO}/M_{MO}, with the landing gear and flaps retracted, and from 1.4 V_{S1} to V_{LE} with the landing gear extended.

(d) *Longitudinal, directional, and lateral trim.* The airplane must maintain longitudinal, directional, and lateral trim (and for the lateral trim, the angle of bank may not exceed five degrees) at 1.4 V_{S1} during climbing flight with—

(1) The critical engine inoperative;

(2) The remaining engines at maximum continuous power; and

(3) The landing gear and flaps retracted.

(e) *Airplanes with four or more engines.* Each airplane with four or more engines must maintain trim in rectilinear flight—

(1) At the climb speed, configuration, and power required by § 25.123(a) for the purpose of establishing the rate of climb;

(2) With the most unfavorable center of gravity position; and

(3) At the weight at which the two-engine-inoperative climb is equal to at least 0.013 V_{SR1} at an altitude of 5,000 feet.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-38, 41 FR 55466, Dec. 20, 1976]

STABILITY

§ 25.171 General.

The airplane must be longitudinally, directionally, and laterally stable in accordance with the provisions of §§ 25.173 through 25.177. In addition, suitable stability and control feel (static stability) is required in any condition normally encountered in service, if flight tests show it is necessary for safe operation.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-7, 30 FR 13117, Oct. 15, 1965]

§ 25.173 Static longitudinal stability.

Under the conditions specified in § 25.175, the characteristics of the elevator control forces (including friction) must be as follows:

(a) A pull must be required to obtain and maintain speeds below the specified trim speed, and a push must be required to obtain and maintain speeds above the specified trim speed. This must be shown at any speed that can be obtained except speeds higher than the landing gear or wing flap operating limit speeds or V_{FC}/M_{FC}, whichever is appropriate, or lower than the minimum speed for steady unstalled flight.

(b) The airspeed must return to within 10 percent of the original trim speed for the climb, approach, and landing conditions specified in § 25.175 (a), (c), and (d), and must return to within 7.5 percent of the original trim speed for the cruising condition specified in § 25.175(b), when the control force is slowly released from any speed within the range specified in paragraph (a) of this section.

(c) The average gradient of the stable slope of the stick force versus speed curve may not be less than 1 pound for each 6 knots.

(d) Within the free return speed range specified in paragraph (b) of this section, it is permissible for the airplane, without control forces, to stabilize on speeds above or below the desired trim speeds if exceptional attention on the part of the pilot is not required to return to and maintain the desired trim speed and altitude.

[Amdt. 25-7, 30 FR 13117, Oct. 15, 1965]

§ 25.175 Demonstration of static longitudinal stability.

Static longitudinal stability must be shown as follows:

(a) *Climb.* The stick force curve must have a stable slope at speeds between 85 and 115 percent of the speed at which the airplane—

(1) Is trimmed, with—

(i) Wing flaps retracted;

(ii) Landing gear retracted;

(iii) Maximum takeoff weight; and

(iv) 75 percent of maximum continuous power for reciprocating engines or the maximum power or thrust selected by the applicant as an operating limitation for use during climb for turbine engines; and

(2) Is trimmed at the speed for best rate-of-climb except that the speed need not be less than 1.4 V_{S1}.

(b) *Cruise.* Static longitudinal stability must be shown in the cruise condition as follows:

(1) With the landing gear retracted at high speed, the stick force curve must have a stable slope at all speeds within a range which is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 50 knots plus the resulting free return speed range, above and below the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1}, nor speeds greater than V_{FC}/M_{FC}, nor speeds that require a stick force of more than 50 pounds), with—

(i) The wing flaps retracted;

(ii) The center of gravity in the most adverse position (see § 25.27);

(iii) The most critical weight between the maximum takeoff and maximum landing weights;

(iv) 75 percent of maximum continuous power for reciprocating engines or for turbine engines, the maximum cruising power selected by the applicant as an operating limitation (see § 25.1521), except that the power need not exceed that required at V_{MO}/M_{MO}; and

(v) The airplane trimmed for level flight with the power required in paragraph (b)(1)(iv) of this section.

(2) With the landing gear retracted at low speed, the stick force curve must have a stable slope at all speeds within a range which is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 50 knots plus the resulting free return speed range, above and below the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1}, nor speeds greater than the minimum speed of the applicable speed range prescribed in paragraph (b)(1), nor speeds that require a stick force of more than 50 pounds), with—

(i) Wing flaps, center of gravity position, and weight as specified in paragraph (b)(1) of this section;

(ii) Power required for level flight at a speed equal to V_{MO} + 1.4 $V_{S1}/2$; and

(iii) The airplane trimmed for level flight with the power required in paragraph (b)(2)(ii) of this section.

(3) With the landing gear extended, the stick force curve must have a stable slope at all speeds within a range which is the greater of 15 percent of the trim speed plus the resulting free return speed range, or 50 knots plus the resulting free return speed range, above and below the trim speed (except that the speed range need not include speeds less than 1.4 V_{S1}, nor speeds greater than V_{LE}, nor speeds that require a stick force of more than 50 pounds), with—

(i) Wing flap, center of gravity position, and weight as specified in paragraph (b)(1) of this section;

(ii) 75 percent of maximum continuous power for reciprocating engines or, for turbine engines, the maximum cruising power selected by the applicant as an operating limitation, except that the power need not exceed that required for level flight at V_{LE}; and

(iii) The aircraft trimmed for level flight with the power required in paragraph (b)(3)(ii) of this section.

(c) *Approach.* The stick force curve must have a stable slope at speeds between 1.1 V_{S1} and 1.8 V_{S1}, with—

(1) Wing flaps in the approach position;

(2) Landing gear retracted;

(3) Maximum landing weight; and

(4) The airplane trimmed at 1.4 V_{S1} with enough power to maintain level flight at this speed.

(d) *Landing.* The stick force curve must have a stable slope, and the stick force may not exceed 80 pounds, at speeds between 1.1 V_{S0} and 1.3 V_{S0} with—

(1) Wing flaps in the landing position;

(2) Landing gear extended;

(3) Maximum landing weight;

(4) Power or thrust off on the engines; and

(5) The airplane trimmed at 1.4 V_{S0} with power or thrust off.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-7, 30 FR 13117, Oct. 15, 1965]

§ 25.177 Static directional and lateral stability.

(a) The static directional stability (as shown by the tendency to recover from a skid with the rudder free) must be positive for any landing gear and flap position and symmetrical power condition, at speeds from 1.2 V_{S1} up to V_{FE}, V_{LE}, or V_{FC}/M_{FC} (as appropriate).

(b) The static lateral stability (as shown by the tendency to raise the

low wing in a sideslip with the aileron controls free and for any landing gear and flap position and symmetrical power condition) may not be negative at any airspeed (except speeds higher than V_{FE} or V_{LE}, when appropriate) in the following airspeed ranges:

(1) From 1.2 V_{s1} to V_{MO}/M_{MO}.

(2) From V_{MO}/M_{MO} to V_{FC}/M_{FC} unless the Administrator finds that the divergence is—

(i) Gradual;

(ii) Easily recognizable by the pilot; and

(iii) Easily controllable by the pilot.

(c) In straight, steady, sideslips (unaccelerated forward slips) the aileron and rudder control movements and forces must be substantially proportional to the angle of sideslip, and the factor of proportionality must lie between limits found necessary for safe operation throughout the range of sideslip angles appropriate to the operation of the airplane. At greater angles, up to the angle at which full rudder control is used or a rudder pedal force of 180 pounds is obtained, the rudder pedal forces may not reverse and increased rudder deflection must produce increased angles of sideslip. Unless the airplane has a yaw indicator, there must be enough bank accompanying sideslipping to clearly indicate any departure from steady unyawed flight.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.181 Dynamic stability.

(a) Any short period oscillation, not including combined lateral-directional oscillations, occurring between stalling speed and maximum allowable speed appropriate to the configuration of the airplane must be heavily damped with the primary controls—

(1) Free; and

(2) In a fixed position.

(b) Any combined lateral-directional oscillations ("Dutch roll") occurring between stalling speed and maximum allowable speed appropriate to the configuration of the airplane must be positively damped with controls free, and must be controllable with normal use of the primary controls without requiring exceptional pilot skill.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

STALLS

§ 25.201 Stall demonstration.

(a) Stalls must be shown in straight flight and in 30 degree banked turns with—

(1) Power off; and

(2) The power necessary to maintain level flight at 1.6 V_{s1} (where V_{s1} corresponds to the stalling speed with flaps in the approach position, the landing gear retracted, and maximum landing weight).

(b) In either condition required by paragraph (a) of this section, it must be possible to meet the applicable requirements of § 25.203 with—

(1) Flaps and landing gear in any likely combination of positions;

(2) Representative weights within the range for which certification is requested; and

(3) The most adverse center of gravity for recovery.

(c) The following procedure must be used to show compliance with § 25.203:

(1) With the airplane trimmed for straight flight at the speed prescribed in § 25.103(b)(1), reduce the speed with the elevator control until it is steady at slightly above stalling speed. Apply elevator control so that the speed reduction does not exceed one knot per second until (i) the airplane is stalled, or (ii) the control reaches the stop.

(2) As soon as the airplane is stalled, recover by normal recovery techniques.

(d) Occurrence of stall is defined as follows:

(1) The airplane may be considered stalled when, at an angle of attack measurably greater than that for maximum lift, the inherent flight characteristics give a clear and distinctive indication to the pilot that the airplane is stalled. Typical indications of a stall,

occurring either individually or in combination, are—

(i) A nose-down pitch that cannot be readily arrested;

(ii) A roll that cannot be readily arrested; or

(iii) If clear enough, a loss of control effectiveness, an abrupt change in control force or motion, or a distinctive shaking of the pilot's controls.

(2) For any configuration in which the airplane demonstrates an unmistakable inherent aerodynamic warning of a magnitude and severity that is a strong and effective deterrent to further speed reduction, the airplane may be considered stalled when it reaches the speed at which the effective deterrent is clearly manifested.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-38, 41 FR 55466, Dec. 20, 1976; Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.203 Stall characteristics.

(a) It must be possible to produce and to correct roll and yaw by unreversed use of the aileron and rudder controls, up to the time the airplane is stalled. No abnormal nose-up pitching may occur. The longitudinal control force must be positive up to and throughout the stall. In addition, it must be possible to promptly prevent stalling and to recover from a stall by normal use of the controls.

(b) For level wing stalls, the roll occurring between the stall and the completion of the recovery may not exceed approximately 20 degrees.

(c) For turning flight stalls, the action of the airplane after the stall may not be so violent or extreme as to make it difficult, with normal piloting skill, to effect a prompt recovery and to regain control of the airplane.

§ 25.205 Stalls: Critical engine inoperative.

(a) It must be possible to safely recover from a stall with the critical engine inoperative—

(1) Without applying power to the inoperative engine;

(2) With flaps and landing gear retracted; and

(3) With the remaining engines at up to 75 percent of maximum continuous power, or up to the power at which the wings can be held level with the use of maximum control travel, whichever is less.

(b) The operating engines may be throttled back during stall recovery from stalls with the critical engine inoperative.

§ 25.207 Stall warning.

(a) Stall warning with sufficient margin to prevent inadvertent stalling with the flaps and landing gear in any normal position must be clear and distinctive to the pilot in straight and turning flight.

(b) The warning may be furnished either through the inherent aerodynamic qualities of the airplane or by a device that will give clearly distinguishable indications under expected conditions of flight. However, a visual stall warning device that requires the attention of the crew within the cockpit is not acceptable by itself. If a warning device is used, it must provide a warning in each of the airplane configuations prescribed in paragraph (a) of this section at the speed prescribed in paragraph (c) of this section.

(c) The stall warning must begin at a speed exceeding the stalling speed (i.e., the speed at which the airplane stalls or the minimum speed demonstrated, whichever is applicable under the provisions of § 25.201(d)) by seven percent or at any lesser margin if the stall warning has enough clarity, duration, distinctiveness, or similar properties.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-7, 30 FR 13118, Oct. 15, 1965; Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

Ground and Water Handling Characteristics

§ 25.231 Longitudinal stability and control.

(a) Landplanes may have no uncontrollable tendency to nose over in any reasonably expected operating condition or when rebound occurs during landing or takeoff. In addition—

(1) Wheel brakes must operate smoothly and may not cause any undue tendency to nose over; and

(2) If a tail-wheel landing gear is used, it must be possible, during the takeoff ground run on concrete, to maintain any altitude up to thrust line level, at 80 percent of V_{s1}.

(b) For seaplanes and amphibians, the most adverse water conditions safe for takeoff, taxiing, and landing, must be established.

§ 25.233 Directional stability and control.

(a) There may be no uncontrollable ground-looping tendency in 90° cross winds, up to a wind velocity of 20 knots or 0.2 V_{s1}, whichever is greater, except that the wind velocity need not exceed 25 knots. At any speed at which the airplane may be expected to be operated on the ground. This may be shown while establishing the 90° cross component of wind velocity required by § 25.237.

(b) Landplanes must be satisfactorily controllable, without exceptional piloting skill or alertness, in power-off landings at normal landing speed, without using brakes or engine power to maintain a straight path. This may be shown during power-off landings made in conjunction with other tests.

(c) The airplane must have adequate directional control during taxiing. This may be shown during taxiing prior to takeoffs made in conjunction with other tests.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.235 Taxiing condition.

The shock absorbing mechanism may not damage the structure of the airplane when the airplane is taxied on the roughest ground that may reasonably be expected in normal operation.

§ 25.237 Wind velocities.

(a) For landplanes and amphibians, a 90-degree cross component of wind velocity, demonstrated to be safe for takeoff and landing, must be established for dry runways and must be at least 20 knots or 0.2 V_{s0}, whichever is greater, except that it need not exceed 25 knots.

(b) For seaplanes and amphibians, the following applies:

(1) A 90-degree cross component of wind velocity, up to which takeoff and landing is safe under all water conditions that may reasonably be expected in normal operation, must be established and must be at least 20 knots or 0.2 V_{s0}, whichever is greater, except that it need not exceed 25 knots.

(2) A wind velocity, for which taxiing is safe in any direction under all water conditions that may reasonably be expected in normal operation, must be established and must be at least 20 knots or 0.2 V_{s0}, whichever is greater, except that it need not exceed 25 knots.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Amdt. 25-42, 43 FR 2322, Jan. 16, 1978]

§ 25.239 Spray characteristics, control, and stability on water.

(a) For seaplanes and amphibians, during takeoff, taxiing, and landing, and in the conditions set forth in paragraph (b) of this section, there may be no—

(1) Spray characteristics that would impair the pilot's view, cause damage, or result in the taking in of an undue quantity of water;

(2) Dangerously uncontrollable porpoising, bounding, or swinging tendency; or

(3) Immersion of auxiliary floats or sponsons, wing tips, propeller blades,

or other parts not designed to withstand the resulting water loads.

(b) Compliance with the requirements of paragraph (a) of this section must be shown—

(1) In water conditions, from smooth to the most adverse condition established in accordance with § 25.231;

(2) In wind and cross-wind velocities, water currents, and associated waves and swells that may reasonably be expected in operation on water;

(3) At speeds that may reasonably be expected in operation on water;

(4) With sudden failure of the critical engine at any time while on water; and

(5) At each weight and center of gravity position, relevant to each operating condition, within the range of loading conditions for which certification is requested.

(c) In the water conditions of paragraph (b) of this section, and in the corresponding wind conditions, the seaplane or amphibian must be able to drift for five minutes with engines inoperative, aided, if necessary, by a sea anchor.

Miscellaneous Flight Requirements

§ 25.251 Vibration and buffeting.

(a) The airplane must be designed to withstand any vibration and buffeting that might occur in any likely operating condition. This must be shown by calculations, resonance tests, or other tests found necessary by the Administrator.

(b) Each part of the airplane must be shown in flight to be free from excessive vibration, under any appropriate speed and power conditions up to at least the minimum value of V_D allowed in § 25.335. The maximum speeds shown must be used in establishing the operating limitations of the airplane in accordance with § 25.1505. In addition, it must be shown by analysis or tests, that the airplane is free from such vibration that would prevent safe flight under the conditions in § 25.629(d).

(c) Except as provided in paragraph (d) of this section, there may be no buffeting condition, in normal flight, including configuration changes during cruise, severe enough to interfere with the control of the airplane, to cause excessive fatigue to the crew, or to cause structural damage. Stall warning buffeting within these limits is allowable.

(d) There may be no perceptible buffeting condition in the cruise configuration in straight flight at any speed up to V_{MO}/M_{MO}, except that stall warning buffeting is allowable.

(e) With the airplane in the cruise configuration, the positive maneuvering load factors at which the onset of perceptible buffeting occurs must be determined for the ranges of airspeed or Mach Number, weight, and altitude for which the airplane is to be certificated. The envelopes of load factor, speed, altitude, and weight must provide a sufficient range of speeds and load factors for normal operations. Probable inadvertent excursions beyond the boundaries of the buffet onset envelopes may not result in unsafe conditions.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970]

§ 25.253 High-speed characteristics.

(a) *Speed increase and recovery characteristics.* The following speed increase and recovery characteristics must be met:

(1) Operating conditions and characteristics likely to cause inadvertent speed increases (including upsets in pitch and roll) must be simulated with the airplane trimmed at any likely cruise speed up to V_{MO}/M_{MO}. These conditions and characteristics include gust upsets, inadvertent control movements, low stick force gradient in relation to control friction, passenger movement, leveling off from climb, and descent from Mach to airspeed limit altitudes.

(2) Allowing for pilot reaction time after effective inherent or artificial speed warning occurs, it must be shown that the airplane can be recovered to a normal attitude and its speed reduced to V_{MO}/M_{MO}, without–

(i) Exceptional piloting strength or skill;

(ii) Exceeding V_D/M_D, V_{DF}/M_{DF}, or the structural limitations; and

(iii) Buffeting that would impair the pilot's ability to read the instruments or control the airplane for recovery.

(3) There may be no control reversal about any axis at any speed up to V_{DF}/M_{DF}. Any reversal of elevator control force or tendency of the airplane to pitch, roll, or yaw must be mild and readily controllable, using normal piloting techniques.

(b) *Maximum speed for stability characteristics, V_{FC}/M_{FC}.* V_{FC}/M_{FC} is the maximum speed at which the requirements of §§ 25.147(e), 25.175(b)(1), 25.177, and 25.181 must be met with flaps and landing gear retracted. It may not be less than a speed midway between V_{MO}/M_{MO} and V_{DF}/M_{DF}, except that, for altitudes where Mach number is the limiting factor, M_{FC} need not exceed the Mach number at which effective speed warning occurs.

[Doc. No. 5066, 29 FR 18291, Dec. 24, 1964, as amended by Amdt. 25-23, 35 FR 5671, Apr. 8, 1970; Amdt. 25-54, 45 FR 60172, Sept. 11, 1980]

§ 25.255 Out-of-trim characteristics.

(a) From an initial condition with the airplane trimmed at cruise speeds up to V_{MO}/M_{MO}, the airplane must have satisfactory maneuvering stability and controllability with the degree of out-of-trim in both the airplane nose-up and nose-down directions, which results from the greater of—

(1) A three-second movement of the longitudinal trim system at its normal rate for the particular flight condition with no aerodynamic load (or an equivalent degree of trim for airplanes that do not have a power-operated trim system), except as limited by stops in the trim system, including those required by § 25.655(b) for adjustable stabilizers; or

(2) The maximum mistrim that can be sustained by the autopilot while maintaining level flight in the high speed cruising condition.

(b) In the out-of-trim condition specified in paragraph (a) of this section, when the normal acceleration is varied from +1 g to the positive and negative values specified in paragraph (c) of this section—

(1) The stick force vs. g curve must have a positive slope at any speed up to and including V_{FC}/M_{FC}; and

(2) At speeds between V_{FC}/M_{FC} and V_{DF}/M_{DF} the direction of the primary longitudinal control force may not reverse.

(c) Except as provided in paragraphs (d) and (e) of this section, compliance with the provisions of paragraph (a) of this section must be demonstrated in flight over the acceleration range—

(1) −1 g to +2.5 g; or

(2) 0 g to 2.0 g, and extrapolating by an acceptable method to −1 g and +2.5 g.

(d) If the procedure set forth in paragraph (c)(2) of this section is used to demonstrate compliance and marginal conditions exist during flight test with regard to reversal of primary longitudinal control force, flight tests must be accomplished from the normal acceleration at which a marginal condition is found to exist to the applicable limit specified in paragraph (b)(1) of this section.

(e) During flight tests required by paragraph (a) of this section, the limit maneuvering load factors prescribed in §§ 25.333(b) and 25.337, and the maneuvering load factors associated with probable inadvertent excursions beyond the boundaries of the buffet onset envelopes determined under § 25.251(e), need not be exceeded. In addition, the entry speeds for flight test demonstrations at normal acceleration values less than 1 g must be limited to the extent necessary to accomplish a recovery without exceeding V_{DF}/M_{DF}.

(f) In the out-of-trim condition specified in paragraph (a) of this section, it must be possible from an overspeed condition at V_{DF}/M_{DF} to produce at least 1.5 g for recovery by applying not more than 125 pounds of longitudinal control force using either the primary longitudinal control alone or the primary longitudinal control and the longitudinal trim system. If the longitudinal trim is used to assist in producing the required load factor, it must be shown at V_{DF}/M_{DF} that the longitudinal trim can be actuated in the airplane nose-up direction with the primary surface loaded to correspond to the least of the following airplane nose-up control forces:

(1) The maximum control forces expected in service as specified in §§ 25.301 and 25.397.

(2) The control force required to produce 1.5 g.

(3) The control force corresponding to buffeting or other phenomena of such intensity that it is a strong deterrent to further application of primary longitudinal control force.

(Secs. 313(a), 601, 603, 604, and 605 of the Federal Aviation Act of 1958 (49 U.S.C. 1354(a), 1421, 1423, 1424, and 1425); and sec. 6(c) of the Dept. of Transportation Act (49 U.S.C. 1655(c)))

[Amdt. No. 25-42, 43 FR 2322, Jan. 16, 1978]

APPENDIX B: MILITARY AIRWORTHINESS REGULATIONS FOR AIRPLANE PERFORMANCE, STABILITY AND CONTROL

Before a military airplane is accepted for service in the US Air Force, the US Navy, the US Marine Corps or the US Army it must meet a number of safety and airworthiness standards which are spelled out in so-called Military Specifications. For purposes of airplane preliminary design, the following specifications are important:

B1. Airplane Performance

B1.1 MIL-C-005011B(USAF): Military Specification, Charts: Standard Aircraft Characteristics and Performance, Piloted Aircraft (Fixed Wing), June 1977.

B1.2 AS-5263(USNAVY): Naval Air Systems Command Specification, Guidelines for the Preparation of Standard Aircraft Characteristics Charts and Performance Data, Piloted Aircraft (Fixed Wing), October 1986.

B2. Airplane Stability, Control and Flying Qualities

B2.1 MIL-F-8785C: Military Specification, Flying Qualities of Piloted Airplanes, November 1980.

IMPORTANT NOTE: Only those parts of the specifications which are of importance to the preliminary designer are reproduced in this Appendix and organized as follows:

B1.	Airplane Performance	page 236
B1.1	Performance: MIL-C-005011B(USAF)	page 236
	Requirements	page 237
	Standard Missions	page 244
B1.1	Performance: AS-5263(USNAVY)	page 249
	Requirements	page 251
	Basic Aerodynamic Data Report	page 265
	Substantiating Data Report	page 272
	Mission Profiles	page 280
B2.	Airplane Stability, Control and Flying Qualities	page 285
B2.1	Stability, Control and Flying Qualities: MIL-F-8785C	page 285
	Note: a listing of contents by paragraph is given on pages 341-343	

B1. Airplane Performance

B1.1 MIL-C-005011B(USAF): Military Specification, Charts: Standard Aircraft Characteristics and Performance, Piloted Aircraft (Fixed Wing), June 1977.

This limited coordination Military Specification has been prepared by the Air Force based upon currently available technical information, but it has not been approved for promulgation as a coordinated revision of Military Specification MIL-C-5011A. It is subject to modification. However, pending its promulgation as a coordinated Military Specification, it may be used in procurement.

1. SCOPE

1.1 Scope. This specification governs the definitions of requirements for, and methods of presenting characteristics and performances for military piloted airplanes. This specification while primarily oriented to conventional take-off and landing (CTOL) aircraft may be applied to STOL and VTOL airplanes if the design criteria established by requirements for specific designs are substituted for the CTOL criteria established herein.

1.2 Application. This specification is applicable to the preparation and presentation of characteristics and performance data. It is also applicable, when appropriate, as an outline of design requirements and mission rules for use in contractual documents and specifications.

1.3 Classification. Characteristics and performance data shall be presented on the following types of charts as required by the procuring agency, and utilizing format as provided. Unauthorized reproduction of such charts bearing the (by authority of the Secretary of the Air Force) statement is prohibited; however no restriction is placed upon use of the format.

1.3.1 Standard aircraft characteristics charts. The standard aircraft characteristics charts are intended to provide a concise, accurate compilation of physical characteristics and performance capabilities of a weapon system.

1.3.1.1 Arrangement. The standard aircraft characteristics chart is basically composed of 10 pages with provisions for supplemental pages as required by the procuring agency. Pages 1, 2, 3, 4, 6 and 9 are mandatory. Pages 5, 7 and 8 are optional as determined by the procuring agency. If certain pages are not required for a specific weapon system, and are omitted, the remaining pages shall be numbered consecutively.

1.3.1.1.1 Basic. The arrangement of the standard aircraft characteristics chart shall be as follows:

a. Page 1 -- Cover sheet which shall include a photograph or perspective drawing of the aircraft model in flight.

b. Page 2 -- Drawings showing descriptive details of the aircraft, such as: Three-view, fuel and oil tankage, armament, inboard profile, etc.

c. Page 3 -- Mission, description, and principal characteristics of the aircraft.

d. Page 4 -- Performance data for the aircraft in tabulated form.

e. Page 5 -- Supplemental tabulated performance data.

f. Page 6 -- Performance graphs.

g. Page 7 -- Supplemental performance graphs.

h. Page 8 -- External store loadings.

i. Page 9 -- Notes. Mission profiles, applicable allowances, and explanatory notes.

Specific requirements are detailed in 3.6.3.

1.3.1.1.2 Supplemental pages. Aircraft characteristics and performance data not coming within the scope of the standard aircraft characteristics charts shall be presented on supplemental pages. Reasons for preparing supplemental pages may be as follows:

a. Possible special loadings or conditions which may:

(1) Be used in restricted tactical operations

(2) Involve non-standard procedures and special operating techniques

(3) Show the maximum potential use of certain aircraft in special missions.

b. Special loadings that may involve equipment, which, for security reasons are only suitable for limited distribution.

c. Theater operations involving non-standard atmospheric conditions.

d. To show additional drawings, illustrations, and graphs. The supplemental page format should be the same as the standard aircraft characteristics chart but may consist of a special design suitable for binding along with the corresponding pages.

1.3.2 Characteristics summary. The characteristics summary is intended to present a summary of performance capabilities on the design mission and principal features in an abbreviated format. Data shown on the characteristics summary shall be in agreement with similar data shown on the standard aircraft characteristics chart. The standard format for the characteristics summary of each model, shall consist basically of a two-page, single-sheet, and shall be 8 by 10-1/2 inches in size after reproduction.

1.4 Categories. The foregoing charts shall be identified by categories to show the development status of the aircraft or data involved.

1.4.1 Development. Charts in this category provide information on new designs during the detail design development after the design becomes stabilized and only minor configuration changes are anticipated.

1.4.2 Service. Charts in this category provide information of aircraft during production and operational use.

1.5 Markings. Each of the foregoing chart types shall be marked as follows:

1.5.1 Designation. The military model designation shall be shown on the lower outer corner.

1.5.2 Category. The chart category as defined in 1.4 shall be shown on the upper outer corner.

1.5.3 Date. The date of publication will be inserted by the procuring agency on the lower inner corner.

1.5.4 Security. The security classification shall be marked as specified by the current DOD Security Regulations. For security purposes each block or graph shall be considered as a paragraph.

1.5.5 Reserved. The upper inner corner is reserved for the use of the procuring agency.

1.6 Submittal. Initial submittal of charts in the development category shall be accomplished after the design has stabilized as determined by the procuring agency. Charts in the service category shall be submitted after the system has been approved for production.

2. APPLICABLE DOCUMENTS

2.1 Issues of documents. The following documents, (effective on the date of invitation for request for proposal) form a part of this specification to the extend specified herein:

SPECIFICATIONS

MILITARY

MIL-G-5572	Gasoline, Aviation, Grades 80/87, 100/130, 115/145
MIL-T-5624	Turbine Fuel, Aviation, Grades JP-4 and JP-5
MIL-M-7700	Manuals, Flight
MIL-F-8785	Flying Qualities of Piloted Airplanes
MIL-A-008860	Airplane Strength and Rigidity, General Specification for
MIL-W-25140	Weight and Balance Control System (For Airplanes and Rotorcraft)
MIL-T-83133	Turbine Fuel, Aviation, Kerosene Type, Grade JP-8

STANDARDS

MILITARY

MIL-STD-210	Climatic Extremes for Military Equipment
MIL-STD-1374	Weight and Balance Data Reporting Forms for Aircraft (Including Rotorcraft)

MANUAL

AFM 60-16	General Flight Rules

(Copies of specifications, standards, drawings, and publications required by contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer.)

3. REQUIREMENTS

3.1 General. Unless otherwise specified by the procuring agency, preparation by contractors of charts (and revision thereto) for each model shall include the preparation of photographically reproducible copy in the required types and categories. Substantiating reports containing supporting characteristics and performance data are required.

3.1.1 Revisions. Revisions to the charts, shall be prepared and submitted by the contractor. Unless otherwise specified by the procuring agency, revisions are required whenever significant changes in vehicle configuration or data occur, as for:

a. A change in vehicle dimensions
b. An accumulation of changes resulting in a significant performance change. (See 3.1.3.1).

c. A change in propulsion system designation, augmentation, or rating.
d. The addition of external stores.
e. The availability of test data showing significant performance change. (See 3.1.1.1).
f. When specifically directed by the procuring agency.

3.1.1.1 Criteria. The following criteria will be used in forming a judgment as to whether a significant change in performance exists:

a. A change of 5 percent or more in drag.
b. A change of 5 percent or more in installed thrust.
c. A change of 5 percent or more in specific fuel consumption.
d. A change in weight which in itself results in a 5 percent or greater change in mission radius or range.
e. Any combination of two or more of the above resulting in a change of 5 percent or more in any performance parameter.

3.1.1.2 Number. Each chart shall cover only one aircraft model. The probable number of charts and revisions which are required throughout the life of the aircraft model will depend on the number of aircraft changes experienced.

3.2 Substantiating report. All data presented on the charts shall be substantiated by reports submitted with the charts. The reports may be legible rough draft copies of the contractor's work sheets. They shall be complete and shall present in detail the contractor's build up of aerodynamic and propulsive data and shall contain a listing of adequate references, authority, and justification for all data used. Contractors are free to use calculation methods of their own selection, but such methods shall be explained in detail so as to permit a ready understanding of aerodynamic, propulsive and weights bookkeeping methods. Calculations shall be presented in sufficient detail as to permit ready review and check of conclusions.

3.2.1 Basic aerodynamic data. Prior to proceeding with the initial performance calculations for the Standard Aircraft Characteristics charts mutual agreement shall be established between the contractor and the procuring agency relative to the aerodynamic and propulsive and weight data to be used for performance data. This agreement shall be accomplished through mutual review and reporting procedures.

3.2.2 Report. The basic aerodynamic propulsive and weight data (see 3.2.1) shall, after review and acceptance by the procuring agency, form the basis for the detailed preparation of the substantiating data report. These data shall be expanded as necessary and used to prepare the detailed performance data required to substantiate the Standard Aircraft Characteristics.

3.2.3 Revisions. The substantiating data report shall be revised under the same criteria as the charts (see 3.1.1).

3.2.4 Text. The required data and the arrangement of the substantiating data report is contained in Appendix IA.

3.3 Standards. Characteristics and performance data shall be based on engineering analysis which produce results consistent with flight test results.

3.3.1 Basis for data. All characteristics and performance data shall be based on the latest reliable aerodynamic, propulsion system, and weight information. The information given shall include the effects on weight and performance of all authorized contract and service changes, together with important changes assured of authorization but pending at the date of chart issue.

3.3.1.1 Changes in characteristics. Changes in aircraft characteristics which do not result in a significant performance change (see 3.1.1.1) do not require a revision by the contractor. However, the procuring agency shall be notified by correspondence so that proper notes on may be appended to the published chart.

3.3.1.2 Flight test. The latest flight test data approved by the procuring agency shall be used as a basis for performance.

3.3.2 Limitations. Performance data shall fall within all established limitations on the vehicle and its components.

3.3.3 Aircraft condition. Performance shall be presented in such a manner as to show clearly the applicable aerodynamic configuration, propulsion system, and loading information. Aircraft configurations shall include the installation of complete service equipment applicable to that particular aircraft model for the mission concerned. Flight performance shall be presented with guns, retable enclosures, bomb bay doors, etc., in position of least drag, and external bombs or other armament in position for each loading condition, as noted.

3.3.4 Atmosphere. Performance shall be based on the latest approved standard atmospheric tables as specified by the procuring agency.

3.3.4.1 Standard day. Unless otherwise specified, performance shall be based on the standard atmosphere as tabulated and described in Appendix IC.

3.3.4.2 Non-standard day. Tropical and Hot day properties must conform to Appendix IC.

3.4 Definitions. The following definitions are used for the various data on the charts and shall be strictly adhered to.

3.4.1 Weights. Weights given on the charts shall comply with the following definitions derived from, and consistent with, MIL-STD-1374, MIL-M-25140 and T.O.AN 01-1B-50 (see 6.2).

3.4.1.1 Empty weight. The weight of the structure, propulsion system, equipment, etc., in the configuration defined in current system specification.

3.4.1.2 Basic weight. The empty weight adjusted for non-expendable operational items. (Weight empty plus unusable fuel and oil and all fixed armament and equipment for normal operation.)

3.4.1.3 Operating weight. Mission take-off weight less payload and usable fuel. (Basic weight plus usable oil, crew, crew baggage, steward equipment, emergency equipment, special mission fixed equipment, pylon and racks not in basic weight, and other nonexpendable items not in basic weight.)

3.4.1.4 Design weight. Weight at which specified structural design requirements are met or are required to be met.

3.4.1.5 Take-off weight. Take-off weight is the total weight of the aircraft with the fuel and payload (see 3.4.1.10) for the mission presented. The take-off weight normally shall be determined prior to start of engines except in specially approved cases when weight expended during taxi and take-off are excluded (see 3.4.1.5.1.b).

3.4.1.5.1 Maximum. Maximum take-off weight is the greatest weight for take-off established by Technical Orders, design requirements, or other specific recommendations of the procuring agency and is the least weight determined by the following criteria:

a. The weight of the vehicle fully loaded with fuel, oil, armament or cargo to the capacity for which space or tonnage is normally provided. The bearing load for the floor and supporting structure shall not be exceeded.
b. The aircraft and its components (wings, landing gear, supporting structures for ordnance, cargo, etc.) shall be capable of sustaining the authorized load factor and shall not violate the minimum criteria of applicable specifications for taxi and ground handling. When ground handling criteria permits a higher weight than does the flight limit, those items expended during take-off, (water, ATO, etc.) may be added to the quoted maximum take-off weight for mission computations, and proper notation thereof will be carried in qualifying note on the performance charts.'
c. Throughout the mission profile the center of gravity shall remain within design limits.
d. The maximum tow force, shall not be exceeded.

3.4.1.5.2 Typical design criteria. For design purposes consideration may be made of alternate definitions of maximum take-off weight such as: Specifying the critical field length and ground run associated with the operational concept of the design, including the effects of such items as runway surface (hard, sod, etc.), ambient runway temperature and pressure altitudes.

3.4.1.6 Maximum inflight weight. The maximum weight at which the aircraft is authorized to be airborne. This weight may be greater than maximum take-off weight if in-flight refueling is utilized.

3.4.1.7 Maximum ramp weight. Maximum in-flight weight unreduced plus fuel, water, etc., used during engine start, taxi, and take-off, shall not exceed other limits such as those for towing, ground handling, wheel jacking, etc., as specified in MIL-A-[illegible].

3.4.1.8 Combat weight. Weight over the target for the mission presented with fuel and oil but without bombs, missiles, mines, cargo or droppable tanks unless otherwise noted.

3.4.1.8.1 Fuel load is determined as follows:

a. Bomber, Fighter Bomber, Missile Carrier (for ground attack) - Immediately after dropping the offensive ordnance, but prior to escape.

b. Fighter (interceptor, air superiority) - Immediately prior to combat.

c. Tanker - Immediately after completion of fuel transfer.

d. Reconnaissance - Immediately after arrival at target (after drop of photo flash bombs if carried).

e. Others (cargo-trainers) - Prior to start of return flight for resupply (radius mission) and prior to landing for range missions.

f. Ferry Mission (all vehicle types) - Reserve fuel only.

3.4.1.8.1.1 Typical design criteria. For design purposes, consideration of alternate definitions of combat fuel load such as: with 50 percent of combat fuel allowance consumed, with 100 percent of combat fuel allowance consumed, or any other criteria selected to optimize the aircraft design.

3.4.1.9 Landing weight

3.4.1.9.1 Maximum. Maximum landing weight is the greatest weight established for landing by structural criteria.

3.4.1.9.2 Mission. The weight at the end of the mission as determined by the mission ground rules. It shall include the fuel reserve as specified by the mission.

3.4.1.10 Payload. The load which justifies the mission. Payload includes cargo, personnel other than crew, bombs, chaff, missiles, reconnaissance cameras, electronic countermeasures pods, photo flash flares, fuel carried for transfer by tankers, and ammunition.

Special equipment required for the mission such as winterization, rescue equipment, except that carried for drop by (H) type vehicles (search-rescue), cargo handling, etc., shall not be included in payload.

3.4.1.11 Fuel. Standard fuel weight of fuel in pounds per U. S. gallon shall be as follows:

a. MIL-F-5572 (Gasoline in all grades) — 6.0 lbs/gal.
b. MIL-F-5624 (JP-4) Jet fuel — 6.5 lbs/gal.
c. MIL-F-5624C (JP-5) Jet fuel — 6.8 lbs/gal.
d. MIL-T-83133 (JP-8) Jet fuel — 6.7 lbs/gal.
e. If design requires special fuels of specified densities or BTU content, such shall be used and specified in the chart notes.

3.4.2 Speeds. All speeds shall be level flight true airspeeds in knots and mach number as applicable.

3.4.2.1 Maximum speed. The highest speed obtainable for configuration and weight in level flight. The altitude at which this speed occurs shall be stated. It shall be the lesser of the speeds determined by the intersection of the thrust (power) available and required curves or the speed limit imposed through structural or heating consideration.

3.4.2.2 Penetration speed. A specified speed ( ) at which the aircraft shall conduct the final run in to the target at a specified altitude. This speed shall be specified by design requirements.

3.4.2.3.1 Combat speed. Maximum speed at combat weight and combat altitude with maximum power.

3.4.2.3 Stall speed. The stall speed shall be computed on the basis of 1.0g flight with the maximum trimmed lift coefficient established by computation or wind tunnel testing. Upon availability of flight test results, stall speed shall be changed to the highest of the speeds for steady straight 1.0g flight at $C_{L\,max.}$, the speed at which abrupt loss of control occurs about any of the pitch, roll or yaw axes, the speed at which intolerable buffet or structural vibration is encountered, or other minimum permissible speed as defined in MIL-F-8785.

3.4.2.3.1 Power-off. The stall speed without power.

3.4.2.4 Take-off speed

3.4.2.4.1 Take-off speed shall be the highest of the speeds specified below:

a. A speed corresponding to 110 percent of power off stall speed in the take-off configuration.

b. A speed determined by the lift coefficient, in ground effect, for the maximum angle of attack attainable with the main landing gear oleo in the static position with aircraft on ground.
c. Minimum speed at which the aircraft has a climb gradient potential of 1/2 percent (0.005), with maximum power, in the take-off configuration, out of ground effect. For multi-engine aircraft this potential shall be obtainable with the most critical engine inoperative. Where:

$$\text{gradient} = \frac{[\text{Vertical Height in climb (ft)}]}{[\text{Horizontal distance in climb (ft)}]} \qquad (1)$$

d. Air minimum control speed as specified in 3.4.2.8.2.
e. The speed which permits attaining obstacle climb-out speed, as defined in 3.4.2.5, at or before reaching 50 ft. height above the runway.

3.4.2.4.1.1 Typical design criteria. For design purposes, consideration may be made of alternate definitions of take-off speed such as: a higher or lower percentage of stall speed, a higher climb gradient potential or other criteria which optimizes the design.

3.4.2.5 Obstacle climb-out speed. The climb speed at the 50-foot obstacle shall not be less than the highest of the speeds specified below.

a. One hundred fifteen percent of power off, 1.0g, stall speed. 1.15 Vs
b. Air minimum control speed.
c. Speed at which the aircraft has a climb gradient of 2.5 percent (0.025) with gear up, flaps in take-off position, with maximum power, out-of-ground effect. For multi-engine aircraft this potential shall be obtainable with the most critical engine inoperative.
d. If gear retraction results in a transient drag increase over that for gear down, the speed at which the aircraft has a 3/2 percent (0.005) climb gradient potential with flaps in take-off setting, gear in transit, with maximum thrust out-of-ground effect. For multi-engine aircraft, the most critical engine shall be inoperative.

3.4.2.5.1 Typical design criteria. For design purposes, consideration may be made of alternative limitations to the obstacle climb-out speed such as: a higher or lower percentage of the stall speed, an increase in climb gradient potential or any criteria which is in keeping with the operational concept of the design.

3.4.2.6 Climb speed. The climb speed shall be the airspeed at which the maximum rate-of-climb is attained for the given configuration, weight, altitude, and power. Consideration shall be made for kinetic energy corrections in optimizing the climb speed schedule. When authorized in the applicable flight manual, a simplified, non-optimum speed schedule may be used.

3.4.2.7 Critical engine failure speed. The critical engine failure speed shall be the speed at which the most critical engine can fail and the same distance be required to either continue the take-off or abort. See 3.4.5.4.

3.4.2.8 Minimum engine-out control speed

3.4.2.8.1 Ground. The minimum control speed, ground, shall be the minimum speed during the take-off run where the engine, most critical to directional control, can fail and directional control can be maintained as defined in MIL-F-8785.

3.4.2.8.2 Air. The minimum control speed, air, shall be the minimum airborne speed with maximum thrust where the engine, most critical to control, can fail and directional control can be maintained.

3.4.2.9 Cruise speed

3.4.2.9.1 Maximum range cruise speed. The speed for maximum range operation shall be the speed at which maximum nautical miles per pound of fuel are attainable at the momentary weight and altitude.

3.4.2.9.2 Long range cruise speed. The higher of the two airspeeds which give nautical miles per pound of fuel equal to 99 percent of the maximum nautical miles per pound of fuel for momentary weight and altitude. This speed may be used to decrease mission time without severe penalty to range.

3.4.2.9.3 Maximum cruise speed. The highest speed that can be maintained with maximum continuous power at stated altitude, weight and configuration.

3.4.2.9.4 Average cruise speed. Total distance covered in cruise divided by the time for cruise (distance and time for climb, acceleration to combat speed, combat time, loiter time etc. are not included).

3.4.2.10 Maximum endurance (loiter) speed. The airspeed for maximum endurance shall correspond to the speed for minimum fuel flow attainable at momentary weight and altitude except as limited by acceptable flying qualities.

3.4.2.10.1 Combat loiter speed. The airspeed for maximum endurance shall correspond to the speed for minimum fuel flow attained at momentary weight and attitude except that the airspeed must be adequate to allow an instantaneous load factor of a specified value.

3.4.2.11 Approach speed. The approach speed down to the 50-foot obstacle shall be the higher of:

a. Air minimum control speed, gear down, flaps in approach configuration and lift augmentation operable.
b. A speed of 120 percent of 1.0g power-off stall speed, out-of-ground effect, gear down, flaps in approach configuration.

c. A speed at which the aircraft has a climb gradient potential of 2.5 percent (0.025) with gear retracted, flaps in approach configuration, and with maximum dry take-off power. For multi-engine aircraft the most critical engine shall be inoperative.

Note: If other than landing flap is selected for approach, it's characteristics shall be specified i.e., approach flap develops a stall -- percent higher than does landing flap.

3.4.2.11.1 Typical design criteria. For design purposes, consideration may be made of alternate definitions of approach speed such as: higher or lower percentage of stall speed, a higher climb gradient potential, alternate go-around power settings or any other criteria which might optimize the design.

3.4.2.12 Landing speed. The landing speed shall be the greater of:

a. A speed determined by the lift coefficient, in ground effect, for the maximum angle attainable with the main landing gear also in the static compressed position with aircraft on ground.
b. One hundred fifteen percent (115 percent) of 1.0g power-off stall speed in the landing configuration.

3.4.2.12.1 Typical design criteria. For design purposes, consideration may be given to alternate definitions of landing speed such as: geometry-limited with flaps in full or partial extended position, changes in percentage of stall speed and climb gradient potential or any other criteria which would optimize the design.

3.4.3 Ceiling

3.4.3.1 Service ceiling. The altitude at which the maximum rate of climb at subsonic speed is 100 ft/min at stated weight and engine power.

3.4.3.2 Combat ceiling

3.4.3.2.1 Subsonic. The altitude at which the max subsonic rate of climb is 500 ft/min at stated weight and power.

3.4.3.2.2 Supersonic. The highest altitude at which the vehicle can fly supersonically and have a climb potential of 1000 fpm at stated power and weight.

3.4.3.3 Cruise ceiling

3.4.3.3.1 Subsonic. The altitude at which the maximum rate-of-climb potential is 300 ft/min at maximum continuous engine rating at momentary weight.

3.4.3.3.2 Supersonic. The highest altitude at which the vehicle can fly supersonic at maximum continuous power with a climb potential of 1000 fpm at momentary weight.

3.4.4 Altitude

3.4.4.1 Cruise altitude. The altitude at which the cruise portion of the mission is computed. Depending on the mission ground rules, the cruise altitude may be assigned or it may be otherwise governed by limitations such as terrain clearance, mission length, ceilings, oxygen or other crew/aircraft constrictions. In no case shall cruise altitude exceed cruise ceiling.

3.4.4.2 Optimum cruise altitude. The altitude at which the aircraft attains the maximum nautical miles per pound of fuel for the momentary weight and configuration. If this altitude exceeds cruise ceiling the latter shall be used for cruise.

3.4.4.3 Combat altitude. The altitude at the target for the specific mission shown.

3.4.5 Take-off. Criteria for conventional take-off aircraft shall comply with the following: (for STOL aircraft the criteria shall be determined by design criteria.) (Vertical components of thrust may be used in take-off computation.)

3.4.5.1 Ground run distance (i.e. take-off distance). Take-off ground run distance shall be that normally obtainable in service operation at Sea Level with standard atmospheric conditions, zero wind, no runway slope on hard (concrete or asphalt) surfaced runways. The take-off speed criteria of 3.4.2.4 shall be used.

3.4.5.1.1 Typical design criteria. For design purposes, consideration may be made of alternate definitions of take-off ground run such as: non-standard atmospheric conditions, higher pressure altitudes, alternate runway surfaces (hard, sod etc.), head or tail-wind or other criteria in keeping with the operational concept of the design.

3.4.5.2 Distance to 50 ft. The distance to clear a 50-foot obstacle shall be the sum of take-off ground run distance of 3.4.5.1 plus the airborne distance needed to accelerate and climb to arrive at the 50-foot height at the speed specified in 3.4.2.5.

3.4.5.3 Take-off time. The take-off time shall be that normally obtainable in service operation at sea level under standard day atmospheric conditions with no wind. The time is measured from start of take-off (brake release) to start of enroute climb (attainment of climb speed).

3.4.5.4 Critical field length. Critical field length is the sum of the distance required to accelerate with all engines operative to critical engine failure speed (3.4.3.7) plus the distance to accelerate with the critical engine inoperative to take-off or to decelerate to a stop from critical engine failure speed in the same distance.

3.4.5.4.1 Data basis. The data basis for the computation of the stopping distance for the chart for critical field length shall be as follows:

a. At engine failure speed the aircraft continues to accelerate for 3 seconds pilot reaction time with remaining engines at maximum power and zero thrust on the inoperative engine.
b. At the end of the 3-second acceleration time, power on all engines is instantaneously reduced to idle, brakes applied, and deceleration devices deployed.
c. Sufficient time, after, b, above, shall be allowed for deployment of the deceleration device(s) or for reverse thrust to reach maximum before including its effect on deceleration.

3.4.5.5 Coefficient of friction. The coefficient μ, as used in this document is defined as the ratio of the total retardation force attributable to the braking system to the momentary gross weight of the aircraft. The following values will be used unless ground or flight test data are available.

3.4.5.5.1 Rolling μ. The rolling (unbraked) coefficient of friction for a dry, hard surface runway shall be assumed equal to 0.025.

3.4.5.5.2 Braking μ. The total braking coefficient of friction for a dry hard surface runway shall be assumed equal to 0.30.

3.4.5.5.3 Test data. Test μ values may be either the results of tests conducted on the specific aircraft or similar types, i.e. commercial aircraft.

3.4.5.5.4 Typical design criteria. For design purposes, consideration may be given to the effects of new and improved methods of increasing the total retardation force such as by anti-skid devices.

3.4.6 Climb. Climb after take-off may be divided into two segments as specified by the procuring agency: initial climb-out and enroute climb.

3.4.6.1 Initial climb-out. Climb-out shall be at a speed which shall not be less than that limited by the criteria of 3.4.2.5. Gear retraction shall be initiated as soon as an adequate positive climb gradient (3.4.2.5c and d), using applicable power, has been established and maintained while accelerating to climb-out speed. Flaps shall be in the take-off position.

3.4.6.1.1 All engines operating. Initial climb-out with all engines operating shall be based on all engines operating from brake release to take-off. Acceleration to climb-out speed and climb-out shall be based on the thrust (power) available with all available engines.

3.4.6.1.2 One engine inoperative. Initial climb-out with one engine inoperative shall be based on all engines operating from brake release to critical engine failure speed and with the critical engine inoperative from critical engine failure speed to take-off. Acceleration to climb-out speed and climb-out shall be based upon the thrust (power) available with the remaining engines at take-off thrust and the drag of the inoperative engine. If means of reducing drag of the inoperative engine are a design feature, such drag reduction shall be utilized with a time allowance for activation.

3.4.6.2 Climb path angle. The climb flight path angle shall be expressed in terms of a gradient (vertical feet per 100 horizontal feet). This path shall be determined from the 50-foot height point and at the 50-foot height climb-out speed as determined in 3.4.2.5. Conditions shall be with gear up, flaps in take-off position, out of ground effect and with appropriate configuration, power and weight. For multi-engine aircraft the climb flight path with the critical engine inoperative shall be included.

3.4.6.3 Enroute climb. Enroute climb data shall be based on the appropriate configuration, power and weight. The aircraft shall have the landing gear and flaps retracted and have attained the airspeed for best climb for the applicable condition.

3.4.6.4 Time to climb. The time to climb to specified altitude(s) shall be expressed in minutes from start of enroute climb. Weight reduction as a result of fuel consumption shall be applied to the calculations.

3.4.6.5 Combat climb. Combat climb is the instantaneous maximum vertical speed capability in feet per minute at combat conditions, such as, weight, configuration, altitude, and power.

3.4.7 Landing distance. The following criteria are for conventional aircraft. (For STOL aircraft the criteria shall be as established by design requirements.) Landing distance includes: (a) landing ground roll and (b) distance over a 50 foot height. Distances shall be for the landing configuration and weight and shall be based on the landing speeds defined in 3.4.2.12. Unless otherwise specified, ground roll deceleration shall be based on operation at Sea Level, standard day, zero wind, no runway slope, idle power and a braking coefficient as defined in 3.4.5.5.2.

3.4.7.1 Typical design criteria. For design purposes, consideration may be given to alternate definitions of landing distance such as: reverse thrust, atmospheric conditions, runway slopes and winds of a non-standard nature, a rigid computer analysis of the air distance and other similar criteria selected to optimize the design of the airplane.

3.4.8 Power. The term (power) is used to mean brake horsepower or thrust as applicable with due consideration for installation effects and limitations. Engine and airplane takeoff ratings as defined in 3.4.2.1.1 c and 3.4.3.1.4 shall be those which appear in the approved engine model specification without regard to installation effects or limitations.

3.4.8.1 Maximum power. Maximum engine power output. This condition of operation may have an incremental duration time limit. This term is used for both augmented and non-augmented engines.

3.4.8.2 Intermediate power. Maximum engine power output without augmentation. This condition may be time limited. This term is used only for augmented engines.

3.4.8.3 Maximum continuous power. Maximum engine power output which may be used continuously, no time limit is imposed.

3.4.8.4 Cruise power. The power required to fly the aircraft at cruise speed for the configuration, altitude and weight designated.

3.4.8.5 Minimum augmented. Lowest power at which the engine will operate with augmentation at any point specified within the augmented operating envelope.

3.4.9 Fuel. Unless otherwise specified, fuel for gas turbine engines shall be JP-4. Weights shall be obtained from 3.4.1.11.

3.4.9.1 Fuel consumption corrections. Corrections or allowances to engine fuel flow shall be made for all propulsion system installation losses such as accessory drives, BLC bleed, environmental system bleed, nozzle losses, pressure recovery, etc.

3.4.10 Mission types. Representative operational missions for various types of aircraft are specified in table I. Typical maximum effort missions are shown in Appendix II to this specification. These maximum effort missions specify the exact fuel allowances for take-off and climb, combat, and landing reserves and are included since they are often used to compare USAF aircraft and foreign aircraft performance capabilities on a common basis.

3.4.10.1 Design mission. The design mission is defined as the primary mission for which the aircraft was specifically procured. This mission will normally be defined in procurement documents such as the statement of work and will include the flight profile, allowances, fuel (clean or external tanks) and payload. Ground rules and allowances for the design mission are dictated by the specific operational requirements and will be used in describing the mission capabilities in the Standard Aircraft Characteristics charts. Some useful alternate design criteria are discussed in 3.5.2 and in other applicable parts of this specification.

3.4.10.2 Ferry mission (ferry range). The greatest distance attainable on a practical one-way mission with maximum authorized fuel and no payload.

3.4.10.3 Typical missions. Any missions, preferably from table I, which would present the additional capability of the aircraft. Normally these will include at least one mission at the minimum take-off weight (3.4.1.3.1) with the ground rules corresponding to the design mission.

3.4.10.4 Inflight refueled mission. For aircraft capable of inflight refueling, a refueled mission is the distance (radius or range) attainable through receipt of replacement fuel during flight. A single refueling operation is required although multiple refueling operations may be added if considered to be feasible. Basic profiles from table I shall apply with special allowance from table II considered.

3.4.10.5 Combat range. Combat range is the distance (including distance covered in climb) attainable on a one-way flight carrying payload (bombs, cargo, personnel) the entire distance. Droppable fuel tanks are dropped when empty. Allowances for take-off, climb, cruise are taken from the design mission. Combat range for bomber, fighter, and attack aircraft should be computed without loading reserves. Loading reserves should be included for aircraft.

3.4.10.6 Combat radius. Combat radius is the distance (including distance covered in climb(s)) to the mid-point of a equal legged mission from base to target and return. Specific mission profile sections, allowances and reserves shall be as set forth in table I and in the mission being considered.

3.4.10.6.1 Typical design criteria. For design purposes, consideration may be given to the requirement for missions containing unequal legs (offset). Significant design impact could result from a mission where the aircraft is recovered at a remote base without the requirement of returning to home base.

3.5 Mission detailed requirements

3.5.1 General mission requirements. Unless otherwise specified, the following general ground rules shall apply:

3.5.1.1 Standard atmosphere. Data shall be presented for standard day atmosphere.

3.5.1.2 Wind. Data shall be for a no-wind condition.

3.5.1.3 Formation flight. Data shall be for a single aircraft only.

3.5.1.4 Ordnance expenditure. All ordnance shall be expended at the start of combat unless otherwise specified.

3.5.1.5 Off-loading fuel. Fuel may be off-loaded to avoid exceeding the maximum allowable take-off weight.

3.5.1.6 External fuel tanks. External fuel tanks on combat aircraft, shall be dropped when empty or prior to combat unless such tanks are designed to be carried during combat. Unless otherwise restricted (Center of Gravity etc.), dropping of external tanks shall be sequenced to provide maximum range. Cargo and tanker aircraft shall not drop empty external tanks.

3.5.1.7 Pylons/racks. In the computation of range/radius performance, pylons and racks shall be retained unless required to be dropped by design requirements

3.5.1.8 Reduced engine operation. When applicable, a minimum number of engines may be used to increase range if such operation would represent normal service usage. However, such action shall conform to 3.5.1.9.

3.5.1.9 Authorized operation. No operational technique, see 3.5.1.8, shall be utilized that is not included, or is not intended to be included as recommended procedure in the applicable flight manual.

3.5.1.10 Trainer aircraft. The trainer mission as defined by table I is applicable to basic and advanced trainer airplanes. Combat and tactical trainer airplanes fly the design mission for the appropriate parent-type airplane.

3.5.1.11 Variable geometry wing (VGW) aircraft. Normally VGW aircraft will have wings in unsweep position for take-off and subsonic flight and swept for supersonic dash and chase profile segment unless footnoted otherwise.

3.5.2 Mission loading requirements. In order to facilitate and expedite the submittal of the charts, the contractor should contact the procuring agency to discuss the various mission loadings prior to submittal. In the absence of special instructions, the following shall apply.

3.5.2.1 Design mission loading. The fuel and payload loading for the design mission shall be the primary loading condition as defined in the system specification for the aircraft.

3.5.2.2 Typical missions loading. Loadings shall be selected from those included in the system specification or other approved loadings which depict a particular capability of the aircraft. At least one mission shall conform to the maximum gross weight specified in 3.4.3.5.1.

3.5.2.3 Ferry missions loading. Loading shall consist of the maximum authorized fuel and no payload.

3.5.2.4 Inflight refueled mission loading. One mission shall be for the same loading as the design mission. Other loadings may be selected from the typical missions.

3.5.2.5 Combat range mission loading. Identical to the loading of the associated combat radius mission.

3.5.3 Mission segments. Rationale for mission segments is presented below.

3.5.3.1 Take-off. Fuel allowances for ground operation including starting engines warm-up, taxi, take-off and acceleration to climb speed, are as defined in the requirements for the design mission.

3.5.3.1.1 Typical design criteria. For design purposes, consideration may be given to defining the take-off allowance to fully utilize the state-of-the-art. Some typical examples are:

a. Specify engine operation for specific time periods at specified powers. Such as, fuel used during 5 minutes of maximum continuous power operation at Sea Level on a standard day plus 1 minute of maximum power operation if afterburner is used during take-off.
b. Estimate fuel required to start the engine(s), run-up, taxi a specified distance at a specified power setting and to accelerate from brake release to climb speed at a specified power.
c. Estimate fuel for a specified time at a specified thrust/weight ratio to account for starting and taxi plus fuel for take-off and acceleration to climb speed computed from the following:

$$W_{f_{TO}} = \frac{V_c W_{TO}}{2g} + \frac{(\dot{W}_o + \dot{W}_c)}{T-D}$$

Where: $W_{f_{TO}}$ = Take-off and acceleration fuel, lbs

V_c = initial climb speed, ft/sec

W_{TO} = take-off weight, lbs

$\dot{W}_o$ = static fuel flow at take-off power, lbs/sec

$\dot{W}_c$ = fuel flow at initial climb speed at take-off power, lbs/sec

T-D = thrust minus drag at V_c, lbs

g = acceleration of gravity, 3.1., ft/sec^2 (2)

Note: If power is to be varied between lift-off and climb speed this equation can be so modified.

d. Other specific criteria may be selected to more accurately portray the operational characteristics of the specific design.

3.5.3.2 Climb. Except for point intercept missions, all climbs shall be accurate with power and speed schedules optimized to maximize mission range. Point intercept missions shall be optimized to obtain minimum time to combat altitude.

3.5.3.2.1 Typical design criteria. For design purposes, consideration may be given to alternate climb schedules to more adequately portray the desired operational capability of the design. For example the following schedules could apply: minimum time, minimum fuel, maximum range, specified power or speed, accelerate during climb, etc.

3.5.3.3 Cruise. Unless specifically assigned, aircraft shall cruise at the speed and altitude for maximum or long range for the applicable configuration, power and weight. Except where the altitude is specified, the aircraft may utilize a cruise climb to optimize cruise distance. This altitude shall not exceed cruise ceiling.

3.5.3.3.1 Typical design criteria. For design purposes, consideration may be given to specifying a cruise technique selected to optimize the desired characteristics of the design. Techniques to be considered include: constant altitude cruise, constant speed cruise, cruise climb profile, step climb, cruise at specified power, cruise with reduced number of engines, cruise altitude in excess of cruise ceiling, fixed distance segment, headwinds or tail winds, non-standard temperatures, etc.

3.5.3.4 Combat. Combat shall be considered by setting aside a quantity of fuel based upon a specified measure of combat performance. For task-oriented fuel allowances, computation shall be based upon weight at start of combat period with benefit due to weight reduction credited; change in speed due to weight reduction shall be ignored.

3.5.3.4.1 Escape and evasion. Escape and evasion shall be considered by setting aside a quantity of fuel based upon a specified measure of performance.

3.5.3.4.2 Typical design criteria. For design purposes, consideration may be given to various methods of accounting for the fuel to be used during combat or escape and evasion action. Some examples of methods are:

a. Fuel required for a specified time with a specified power at a specified speed and a specified altitude.
b. Fuel consumed in expending a specified quantity of energy. For example:

$$\text{Combat fuel} = \frac{E_s \dot{W}_f}{P_s}$$

Where: E_s = Specific energy, ft

$\dot{W}_f$ = fuel flow at combat speed, power, and altitude, lbs/sec

P_s = excess energy or $\frac{(T-D)V_c}{W_{T_c}}$, ft/sec

V_c = combat true airspeed, ft/sec

W_{T_c} = combat weight, lbs

(T-D) = thrust minus drag, lbs (3)

c. The quantity of fuel determined as the sum of the fuel required to accelerate from cruise speed to a specified speed, plus fuel required to make a specified number of sustainable turns at a specified speed or speeds. These operations shall be performed at a selected power(s) and altitude(s).
d. All or a portion of the armament may be expended.
e. Other specific criteria selected to more accurately portray the operational characteristics of the specific design.

3.5.3.4 Descent. For vehicles whose best cruise is subsonic, no time, fuel or distance shall be credited for descent. For supersonic cruise vehicles, credit shall be taken for descent and deceleration to a specified altitude and speed. Vehicles which conduct a supersonic run out from the target may, if the cruise altitude and speed are lower than the run out altitude and speed, account for distance in descent and deceleration to cruise.

3.5.3.5 Typical design criteria. For design purposes, consideration may be given to alternate definitions of descent. For example: Time, fuel and distance could be credited, descent could be a long range (airline) approach, use of power could vary from none to full, speeds could vary from near stall to redline, altitudes could be reduced in step increments, etc.

3.5.3.6 Landing reserve. Since the mission profiles of table I are generalized, no compliance with the alternate landing destination of AFM 60-16 is possible. Instead, a landing reserve is required which would be typical of operational use.

3.5.3.6.1 Typical design criteria. For design purposes, consideration may be given to defining the landing reserves to fully utilize the state-of-the-art. Some examples are:

a. The fuel required for a ground controlled approach; a wave-off, go-around and a second, successful landing. This could be approximated by using the equivalent of fuel consumed during a specified time at maximum endurance at Sea level with all engines operating.
b. A specified percentage of initial fuel load.
c. Fuel consumed during a specified time of operation at a specified power at a specified altitude.
d. The greater of the fuel required for 10 percent of mission time or 20 minutes at maximum endurance speed at 10,000 feet (AFM 60-16).
e. Fuel required to fly to an alternate field (specify distance) plus a specified time at a specified speed at a specified altitude to account for landing.
f. Combinations of the above or other criteria selected to optimize the design.

3.5.4 Mission time. Time in air excluding the time before the start of initial climb and reserve unless otherwise specified and noted. For interceptors only: Includes actual time required for take-off and acceleration to climb speed.

3.5.5 Cycle time. The time of flight from the start of initial climb (omitting take-off time) to the time when the engines are stopped after landing.

3.5.6 Block time. The total time of flight from engine start to engine stop after landing.

3.5.7 Intercept time. The time from engine start until initiation of combat at the intercept altitude. This time includes the period required for take-off and acceleration to climb speed.

TABLE I

STANDARD MISSIONS

ATTACK

- * A-1 Hi-Hi-Hi
- A-2 Hi-Lo-Hi
- * A-3 Hi-Lo-Lo-Hi
- A-4 Lo-Lo-Lo-Hi
- * A-5 Lo-Lo-Lo-Lo
- * A-6 CAP

BOMBER

- * B-1 Hi-Hi-Hi-Hi
- B-2 Hi-Lo-Lo-Hi

CARGO

- C-1 Supply
- * C-2 Assault

FIGHTER

- * F-1 Air Superiority
- F-2 Point Intercept
- F-3 Area Intercept
- F-4 CAP
- F-5 Hi-Hi-Hi
- F-6 Hi-Lo-Hi
- F-7 Hi-Lo-Lo-Hi
- F-8 Lo-Lo-Lo-Hi
- F-9 Lo-Lo-Lo

TANKER

- T-1 Buddy Refuel
- * T-2 Rendezvous Refuel

GENERAL

- G-1 Airborne Warning and Control
- G-2 Rescue
- G-3 Forward Air Controller
- G-4 Trainer
- G-5 ASW Search
- * G-6 Ferry Mission

* MISSION PROFILES INCLUDED IN THIS TEXT. FOR OTHERS SEE MIL-C-005011B

TABLE II. STANDARD MISSIONS (Continued) MISSION A-1

HI-HI-HI - ATTACK

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to target ( )	3. Speed and altitude for maximum range
4. Drop Stores ( )	4. Weight reduction equal to store weight
5. Escape and evasion ( )	5. See 3.5.3.4.1
6. Cruise to base ( )	6. Same as 3
7. Arrive over base with reserve fuel ( )	7. See 3.5.3.6

NOTES

1. For segment details and rationale - see 3.5.3
2. For each segment, enter incremental values: (Time; hours; Fuel; pounds; Distance; n. miles)
3. For tanker missions with a specified receiver - See table III

TABLE 1. STANDARD MISSIONS (Continued) MISSION A-3

HI-LO-LO-HI - ATTACK

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to start of penetration ( )	3. Speed and altitude for maximum range
4. Descend to Sea Level	4. No time, fuel or distance credited except where specified
5. Run-in specified distance at Sea Level to target ( )	5. At penetration speed at Sea Level with power as required
6. Drop Stores ( )	6. Weight reduction equal to store weight
7. Attack target ( )	7. See 3.5.3.4
8. Run-out specified distance at Sea Level from target ( )	8. Same as 5
9. Climb on course to cruise	9. Same as 2
10. Cruise to base ( )	10. Same as 3
11. Arrive over base with reserve fuel ( )	11. See 3.5.3.6

TABLE 1. STANDARD MISSIONS (Continued) MISSION A-5

LO-LO-LO-LO - ATTACK

SEGMENT	ALLOWANCE
1. Take-off and accelerate to cruise speed ( )	1. See 3.5.3.1
2. Cruise at Sea Level to target ( )	2. Speed and power for maximum range
3. Drop stores ( )	3. Weight reduction equal to store weight
4. Attack target ( )	4. See 3.5.3.4
5. Cruise to base at Sea Level	5. Same as 2
6. Arrive over base with reserve fuel ( )	6. See 3.5.3.6

TABLE I. STANDARD MISSIONS (Continued)

MISSION A-6

COMBAT AIR PATROL - ATTACK

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to target area specified distance from base ( )	3. Speed and altitude for maximum range
4. Loiter at specified altitude awaiting target assignment ( )	4. Speed and power for combat loiter at specified altitude
5. Drop stores ( )	5. Weight reduction equal to store weight
6. Attack target, escape and evade ( )	6. See 3.5.3.4
7. Cruise to base ( )	7. Same as 3
8. Arrive over base with reserve fuel ( )	8. See 3.5.3.6

TABLE I. STANDARD MISSIONS (Continued)

MISSION B-1

HI-HI-HI-HI - BOMBER

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Power and speed to maximize range
3. Cruise ( )	3. Speed and altitude for max range
4. Climb to combat altitude	4. Power and speed to maximize range
5. Run-in specified distance at speed for maximum continuous power at combat ceiling ( )	5. Maximum continuous power. If this results in supersonic speeds, include time fuel and distance to accelerate as part of the penetration.
6. Drop stores and conduct evasive and turn action ( )	6. Weight reduction includes store weight plus combat fuel allowance. See 3.5.3.4.
7. Run-out specified distance at speed for maximum continuous power at combat ceiling ( )	7. Same as 5
8. Descend to cruise altitude ( )	8. No time, fuel or distance is credited for subsonic vehicles. If segment 7 is supersonic, credit may be taken for time, fuel and distance to decelerate and descent to cruise altitude
9. Cruise back to base ( )	9. Same as 3
10. Arrive over base with reserve fuel ( )	10. See 3.5.3.6

TABLE I. STANDARD MISSIONS (Continued)

MISSION F-1

AIR SUPERIORITY - FIGHTER

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to combat area ( )	3. Speed and altitude for maximum range
4. Engage enemy aircraft and expend ordnance at start of combat ( )	4. See 3.5.3.4
5. Cruise to base ( )	5. Same as 3. The aircraft is assumed to be at cruise altitude, speed and heading at the completion of combat
6. Arrive over base with reserve fuel ( )	6. See 3.5.3.6

TABLE I. STANDARD MISSIONS (Continued)

MISSION C-2

CARGO-ASSAULT

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to combat area ( )	3. Speed and altitude for maximum range
4. Descend to Sea Level	4. No time, fuel or distance credited except where specified
5. Land off-load payload and reload one half payload of casualties etc ( )	5. No time, fuel or distance credited. Payload is reduced by one-half.
6. Take-off and accelerate to climb speed ( )	6. See 3.5.3.1
7. Climb on course to cruise altitude ( )	7. Same as 2
8. Cruise to base ( )	8. Same as 3
9. Arrive over base with fuel reserve ( )	9. See 3.5.3.6

TABLE I. STANDARD MISSIONS (Continued)

MISSION T-2

TANKER - RENDEZVOUS

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to refuel point ( )	3. Speed and altitude for maximum range
4. Descend to specified altitude	4. No fuel, time or distance credited except where specified
5. Loiter 1 hour for rendezvous ( )	5. Fuel for 1 hour maximum endurance. No distance credited
6. Transfer fuel at speed for maximum continuous power ( )	6. Credit time, fuel and distance while transferring fuel at maximum rate. Segment performed at maximum continuous power
7. Climb on course to return cruise altitude ( )	7. Same as 2
8. Cruise to base	8. Same as 3
9. Arrive over base with reserve fuel ( )	9. See 3.5.3.6

Note: This mission is to present tanker capability without consideration of a specific receiver.

TABLE I. STANDARD MISSIONS (Continued)

MISSION G-6

FERRY MISSION

SEGMENT	ALLOWANCE
1. Take-off and accelerate to climb speed ( )	1. See 3.5.3.1
2. Climb on course to cruise altitude ( )	2. Speed and power for maximum range
3. Cruise to remote base	3. Speed and altitude for maximum range. Normally external fuel tanks shall be retained if carried.
4. Descent to sea level	4. No time, fuel or distance credited unless otherwise specified
5. Arrive over remote base with reserve fuel ( )	5. See 3.5.3.6

B1.2 AS-5263(USNAVY): Naval Air Systems Command Specification, Guidelines for the Preparation of Standard Aircraft Characteristics Charts and Performance Data, Piloted Aircraft (Fixed Wing), October 1986.

THIS SPECIFICATION HAS BEEN APPROVED BY THE
NAVAL AIR SYSTEMS COMMAND, DEPARTMENT OF THE NAVY

1. SCOPE.

1.1 SCOPE. This specification governs the definition of requirements for and methods of presenting characteristics and performance for Navy piloted fixed wing aircraft.

1.2 APPLICATION. For all piloted fixed wing aircraft proposed or contracted for subsequent to the effective date of this specification, characteristics and performance data shall be prepared and presented in accordance with the provisions of this specification and submitted to the Naval Air Systems Command for acceptance, unless specifically exempted by the Navy. Deviations from the provisions of this specification to portray more adequately the capability of certain aircraft are permissible, but shall in all cases be approved by the Naval Air Systems Command. Authorized deviations shall be fully explained through proper annotations on the data charts.

1.3 TYPES OF CHARTS. Characteristics and performance data shall be presented on the following types of charts as required by the Navy and utilizing formats as provided. Unauthorized reproduction of such charts is prohibited.

1.3.1 Standard Aircraft Characteristics Charts. The Standard Aircraft Characteristics (SAC) Charts are intended to provide Navy technical and staff personnel with a concise, accurate compilation of physical characteristics and performance capabilities of a weapon system. Standardization is required for convenience and to allow direct comparison with other weapon systems intended for a similar mission.

1.3.1.1 Arrangement. The Standard Aircraft Characteristics Chart is basically composed of ten pages with provisions for supplemental pages as required by the Naval Air Systems Command. If certain pages are not required for a specific weapon system, they shall be omitted and the remaining pages renumbered consecutively.

1.3.1.1.1 Basic. The normal arrangement of the Standard Aircraft Characteristics Chart shall be as follows (Refer to Sample SAC Chart in Appendix I):

a. Page 1. Cover sheet, which shall include a photograph or perspective drawing of the aircraft model in flight.

b. Page 2. A drawing showing a descriptive arrangement of the aircraft and a drawing showing the armament installations, the tankage installation, cargo space or interior arrangements as required by the Navy.

c. Page 3. Mission, description and principal characteristics of the aircraft.

d. Page 4. Performance data for the aircraft in tabulated form, with applicable notes.

e. Page 5. Auxiliary performance page, giving alternate loadings and the respective radii and associated performance, where applicable.

f. Page 6. Performance graphs. Speed, takeoff and landing distance, rates of climb, maneuverability, etc.

g. Page 7. Performance graphs. Trade-off curves as applicable.

h. Page 8. Carrier suitability graphs. Performance curves for catapulting, arresting, single engine climb, accelerations, approach speeds, and required Wind over Deck.

i. Page 9. External store loadings. Tabulated stores and stations on which they may be carried.

j. Page 10. Notes. Mission profiles, applicable allowances and explanatory notes.

1.3.1.1.2 Supplemental Pages. Aircraft characteristics and performance data not coming within the scope of the Standard Aircraft Characteristics Charts shall be presented on supplemental pages. Reasons for preparing supplemental pages may be as follows:

a. Possible special loadings or extreme overload conditions which may:

1. Be used in restricted tactical operations.

2. Involve non-standard procedures and special operating techniques.

3. Show the maximum potential use of certain aircraft in special missions.

b. Such loadings that may involve equipment which for security reasons are only suitable for limited distribution.

c. Theater operations involving non-standard atmospheric conditions.

d. To show inboard profiles, additional drawings, illustrations and graphs.

The supplemental page format should be the same as the Standard Aircraft Characteristics Chart or may consist of a special design suitable for binding along with the corresponding basic Standard Aircraft Characteristics Charts.

1.4 CATEGORIES. The foregoing charts shall be identified by categories to show the development status of the aircraft or data involved. All chart format shall be completed in full detail.

1.4.1 Proposal. Proposal data charts are intended to provide information during the evaluation of new designs, design studies and proposed modifications of existing designs and are primarily for limited distribution within the Navy.

1.4.2 Mock-Up. Mock-up charts are intended to provide information on new designs during the initial period from source selection to completion of mock-up. The initial mock-up chart need not include the effect of all design changes recommended by the Navy, but should contain under notes a complete list of major design changes.

1.4.3 Pre-Service. Pre-service charts are intended to provide information on new designs from the period of time from completion of mock-up to roll out of flight test article. Initial pre-service chart will normally be issued as soon after mock-up as the configuration and weight have stabilized to define the initial test article to be fabricated.

1.4.4 Service. Service data charts are intended to provide information on service models. Preparation of the initial issue of a chart on a service model shall normally be initiated not later than when the configuration and weight have stabilized following mock-up inspection. Thus, it may take the place of pre-service charts or may be delayed until after initial flight test if design changes are anticipated therefrom.

1.5 MARKINGS. Each of the foregoing chart types shall be marked as follows (Refer to Sample SAC Chart in Appendix I):

1.5.1 Designation. The military model designation or the contractor's model designation (in the case of charts in the proposal category) shall be shown on the lower outer corner.

1.5.2 Category. The chart category as defined in Paragraph 1.4 shall be shown on the upper outer corner.

1.5.3 Date. The date of publication will be inserted by the Navy on the lower inner corner.

1.5.4 Security. The security classification shall be specified by the Navy and shall be shown on center at top and bottom.

1.5.5 Chart Identification. The upper inner corner is to be used for chart identification. Identification number shall be obtained from the Navy prior to submission of the charts for approval.

2. APPLICABLE SPECIFICATIONS AND OTHER PUBLICATIONS.

The following publications (effective on the date of invitation for request for proposal) shall form a part of this specification to the extent specified herein:

2.1 SPECIFICATIONS AND PRIMARY PUBLICATIONS.

MILITARY

MIL-D-7822	DRAWINGS: FOR STANDARD AIRCRAFT CHARACTERISTICS AND PERFORMANCE CHARTS, PILOTED AIRCRAFT.
MIL-A-08860	AIRPLANE STRENGTH AND RIGIDITY; GENERAL SPECIFICATION FOR.
MIL-G-5572	FUEL; AVIATION GRADES 80/87, 100/130, 115/145 - 6.0 LBS/GAL.
MIL-T-5624	FUEL; AIRCRAFT TURBINE AND JET ENGINE, GRADES JP-4 - 6.5 LBS/GAL. AND JP-5 - 6.8 LBS/GAL.
MIL-T-83133	TURBINE FUEL, AVIATION, KEROSENE TYPE, GRADE JP-8 - 6.7 LBS/GAL.

MILITARY

MIL-F-8785C	FLYING QUALITIES OF PILOTED AIRPLANES.
MIL-M-85025(AS)	MANUALS, NATOPS FLIGHT; REQUIREMENTS FOR PREPARATION OF.
MIL-STD-210	CLIMATIC EXTREMES FOR MILITARY EQUIPMENT.
MIL-P-22203(AER)	PERFORMANCE DATA REPORT FOR STANDARD AIRCRAFT CHARACTERISTICS CHARTS FOR PILOTED AIRCRAFT.
MIL-STD-1374	WEIGHT AND BALANCE DATA REPORTING FORMS FOR AIRCRAFT (INCLUDING ROTORCRAFT).
MIL-W-25140	WEIGHT AND BALANCE CONTROL SYSTEM. (FOR AIRPLANES AND ROTORCRAFT).
OPNAVINST 3710.7	NATOPS GENERAL FLIGHT AND OPERATING INSTRUCTIONS.
NAVAIR 01-1B-40	NAVY TECHNICAL MANUAL OF WEIGHT AND BALANCE DATA.
NAVAIR 01-1B-50	NAVY TECHNICAL MANUAL, USN AIRCRAFT WEIGHT AND BALANCE CONTROL.
TECHNICAL ORDER 01-1B-40	AIR FORCE TECHNICAL MANUAL OF WEIGHT AND BALANCE DATA.

2.2 OTHER PUBLICATIONS. AS 2694. Engines, Aircraft, Turboshaft and Turboprop, General Specification for.

(Copies of specifications, standards and drawings required by contractors in connection with specific procurement functions should be obtained from the procuring agency or as directed by the contracting officer).

3. REQUIREMENTS.

3.1 GENERAL. Unless otherwise specified by the Navy, preparation by contractors of charts (and revision thereto) for each model shall include the preparation of photographically reproducible copy in the required types and categories, and, in addition, satisfactory reports containing supporting characteristics and performance data.

3.1.1 Revisions. Revisions to the charts shall be prepared and submitted by the contractor throughout the life of the contract unless specified otherwise by the Navy. Revisions are required whenever significant changes in vehicle configuration or data occur, as for example:

1. A change in vehicle dimensions.
2. An accumulation of weight changes resulting in a significant performance change (Paragraph 3.1.1.1).
3. A change in power plant designation, augmentation, or power plant rating.
4. The addition of external stores.
5. The availability of test data or new test data showing significant performance change (Paragraph 3.1.1.1).
6. When specifically directed by the Navy.

3.1.1.1 Criteria. The following criteria will be used in forming a judgement as to whether a significant change in performance exists:

1. A change of 5 percent or more in drag.
2. A change of 5 percent or more in installed thrust (power).
3. A change of 5 percent or more in specific fuel consumption.
4. A change in weight which in itself results in a 5 percent or greater change in mission radius or range.
5. Any combination of two or more of the above resulting in a change of 5 percent or more in mission radius or range.

3.1.1.2 Number. Each chart shall cover only one aircraft model. For the information of the contractor, the following guide is given regarding the probable number of charts and revisions thereto which are required throughout the life of the aircraft model. The exact number of revisions required will depend on the number of aircraft changes experienced.

CATEGORY OF CHART	REASON	BASIS FOR DATA
PROPOSAL	NEW DESIGN	ESTIMATED
MOCK-UP	CONTRACT FOR NEW AIRCRAFT	ESTIMATED
PRE-SERVICE	BETWEEN MOCK-UP AND FIRST FLIGHT	ESTIMATED
SERVICE	FLIGHT TEST	FLIGHT TEST
SERVICE (REVISION)	OPERATIONAL OR FLEET INTRODUCTION	FLIGHT TEST

3.2 STANDARD AIRCRAFT CHARACTERISTICS CHARTS.

3.2.1 Required Characteristic Data (Including Descriptive Detail). A sample Standard Aircraft Characteristics Chart is provided for reference in Appendix I.

3.2.1.1 Page 1. Cover Sheet. The cover sheet shall include a picture of the aircraft. In order of preference: A photograph of the aircraft in flight, a photograph of the aircraft on the ground, a photograph of a model, or an artist's conceptual drawing of the aircraft in flight. The photograph or drawing shall be of good contrast or permit satisfactory reproduction and should portray the distinguishing features of the aircraft. The photograph shall be glossy black and white and have dimensions not less than 5" x 8" and not greater than 7" x 11 1/2", not including the border. The aircraft model designation and the approved popular name shall be typeset using 24 point Futura Demibold or equivalent, centered below the title leaving a 1/2 inch space. One-half inch below the aircraft designation, center the contractor's name using 18 point Futura Demibold or equivalent.

3.2.1.2 Page 2. Drawings. The three-view drawings shall be drawn in ink on suitable drawing material and may be made oversize at whatever scale the manufacturer deems suitable. This over-size ink drawing shall then be photographically reduced and inserted on the appropriate block within the format sheet. Full advantage shall be taken of the space allotted so as to provide the largest three-view arrangement attainable within the 7 1/8" x 10 7/8" block in keeping with the positioning guidelines of paragraph 3.2.1.2.a and 3.2.1.2.b. The line weights used on the three-view drawings must be suitable to provide reproductionof the format page when reduced to 9 1/2" wide. All dimensions and text entered on this format page shall be typeset using 10 point Futura Medium or equivalent.

a. Descriptive arrangement drawing. In top of block give top view drawing; in middle of block give front view drawing; in bottom of block give side view drawings. Folded wings and extended tail hook shall be shown with dotted lines. Dimension in feet and fractions: wing span extended and folded, horizontal tail span, length, height, wheel base, wheel tread. A scale should be included. Describe wing airfoil shape at wing root and wing tip. Give wing planform area, mean aerodynamic chord (M.A.C.) and aspect ratio.

b. Armament and tankage drawing. In top of block give top view drawing; in middle of block give front view drawing; in bottom of block give side view drawing. Drawings shall be shown with external fuel tanks mounted. Internal fuel tank locations shall be shown by means of cross hatching. Tanks with survivability enhancements, i.e. foam, should be designated. Tank capacities shall be given. A scale should be included.

3.2.1.3 Page 3. Mission, Description and Principal Characteristics. The mission and description page shall include the information given below.

a. Mission and Description. The first paragraph in this block shall describe the primary and secondary missions of the aircraft. The second paragraph shall describe general design features, such as configuration, type of structure (use of composites, etc.), powered wing fold mechanism, etc. The third paragraph shall describe features of the propulsion and fuel systems, such as turbofan engine bypass ratio, number and capacity of external fuel tanks, self sealing fuel tanks, retractable refueling probe, etc. The fourth paragraph shall describe armament features, such as number of external store stations, gun caliber, forward looking IR, chaff dispensers, etc. The fifth paragraph shall describe control system features, such as type of high lift system, digital fly-by-wire, mechanical backup to primary control surfaces, rudder aileron interconnect, etc. The sixth paragraph shall describe avionic suite features, such as multi-mode radar, central digital computer, inertial navigation, etc. The seventh paragraph shall describe crew system features, such as ejection seat, anti-G system, etc.; if cabin is pressurized, the airplane altitude at which a 10,000 foot cabin altitude is reached shall be stated. Following the above descriptions, a sub-heading DEVELOPMENT shall give important dates: contract date, first flight, initial carrier sea trials, initial service date, etc.

b. Power Plant. Data to be listed shall include: number and model of engines, manufacturer, engine specification number, type, augmentation, length with afterburner, inlet diameter, dry weight, etc. If propellers are used, give the following data: manufacturer, propeller specification number, diameter, number of blades, gear ratio.

c. Ratings. Engine ratings shall include thrust or power, rpm, altitude(s) and time limits or deviations, as applicable. Engine ratings and auxiliary thrust device (ATD) ratings shall conform to those established in the officially approved engine specifications. Ratings with an augmentation shall be identified by note. If performance items are based on thrusts (powers) which differ appreciably from the listed specification ratings due to flight or engine laboratory test results or restrictions, such thrusts (powers) with explanations will be listed under notes. Reference to source of such thrust (power) shall be clearly stated in the performance data report.

d. Weight and Load Factors. The gross weights and the corresponding allowable load factors shall not exceed the limits established by the latest applicable technical orders, design requirements, or other specific recommendation of the Naval Air Systems Command. Maximum weights for which a mission is shown on the Standard Aircraft Characteristics Charts to illustrate maximum combat capabilities, but which may involve non-standard operating procedure and/or special operating techniques associated with such weight may be given, provided such weights are clearly identified with a note defining the limitations on usage. The following weights with corresponding load factors for both land based and carrier based aircraft as applicable, shall be given:

LOADING	POUNDS	LOAD FACTOR	REFERENCE
EMPTY			3.5.1.1
BASIC			3.5.1.2
DESIGN			3.5.1.4
COMBAT (BASIC MISSION)			3.5.1.6
MAXIMUM TAKEOFF			3.5.1.7.1
OVERLOAD MAXIMUM TAKEOFF			3.5.1.7.2
MAXIMUM IN-FLIGHT			3.5.1.8
MAXIMUM LANDING			3.5.1.9.2

NOTE:

Basis of Weight Data. The weights given shall correspond to the definitions of Paragraph 3.5.1. Weight empty shall be identified by the symbols "E": (estimated), "C": (calculated), or "A": (actual). As applicable, notation shall be made immediately below the takeoff weight of the immediate factor(s) limiting takeoff weight.

e. Fuel and Oil. The number of fuel and oil tanks, their usable capacities and locations, extent of self sealing provisions, together with grade and specification of fuel and oil used, shall be listed. Fuel tanks shall be grouped by fuel system.

f. Electronics. Sub-headings for airborne weapons control, electronic warfare, navigation and flight aids, communication and identification, control and display, and flight control shall be given. Under each sub-heading, list item and military model number in tabular form.

g. Ordnance. Data concerning the standard size and number of each type of droppable ordnance items such as bombs, torpedoes, mines, rockets, missiles and the maximum bomb load which may be accommodated by the aircraft. Ordnance carried externally shall be identified. The number and caliber of guns, the number of turrets, rounds of ammunition per gun, and the gun stations shall be listed.

h. Cargo. Maximum cargo load, clear space dimensions, limit floor loads, door size and location, usable cubage, etc. are to be given as applicable. Additional cargo information may be entered on a supplemental "NOTES" page.

i. Dimensions. Overall dimensions, in agreement with the general arrangement drawings of the basic aircraft in the three point position, such as length, height, width, maximum tread, and propeller ground clearance shall be given. Dimensions should be given in the wings folded and flight condition. Also include wing area and Aspect Ratio. Dimensions shall be given in feet and tenths of a foot.

3.2.1.4 Page 4. Performance Summary. Tabulated performance for the clean mission, basic mission, ferry mission, other typical missions (Paragraphs 3.5.10 to 3.5.10.4) shall include applicable loading and performance items. (See Page 4 of sample chart in Appendix I). Columns 1 and 2 of Page 4 are restricted to the clean and basic missions, as defined in Paragraph 3.5.10.1 and 3.5.10.2, respectively. Other columns, except the last column are restricted for the contractor's use in presenting performance data depicting the mission for which the vehicle was designed; requirement for compliance with ground rules outlined in this specification is waived for presentation of data on these typical missions (See Paragraph 3.5.10.5). Criteria (ground rules) for the typical missions shall be presented on Page 9. The last column used is restricted for use in depicting data for the Ferry Mission (See Paragraph 3.5.10.4).

a. The format of the performance summary page is shown on page I-4 in Appendix I. Show tabulated performance data for no more than six missions on each page. Data for all missions for which Navy requests are submitted shall be placed on the performance summary. A second performance summary page shall be used if the Navy requests data for more than 6 missions.

b. For each mission, divide the takeoff loading condition block into upper and lower sections; in the upper section give the name of the mission, in the lower section on the left side give the loading number in a circle, and to the right of that, show the missile, bomb and external tank loadings with the number of each item in parentheses. In the combat loading condition block on the left side, show the loading number in a circle, and to the right of that, describe the armament and external tank configuration (tanks off, missiles, retained, etc.). The takeoff loading condition for mission 1 is loading number 1; the combat loading condition for mission 1 is loading number 2; the takeoff loading condition for mission 2 is loading number 3; the combat loading condition for mission 2 is loading number 4; etc.

c. The space at the bottom of the page shall be used for footnotes lettered (A), (B), (C) etc. The applicability of the notes shall be shown by placing the appropriate letter behind the wording of the item affected; for example, the letter (A) placed behind the words "Takeoff Run at S.L." could be used along with the footnote, "(A) Intermediate Thrust (Power), Standard Day," to denote the thrust (power) setting for takeoff data. The last footnote shall be, "( ) Performance Basis: wind tunnel or flight test (as appropriate) followed by contractor's report number for the substantiating performance data report."

3.2.1.5 Page 5. Mission Summary - Alternate Loadings. For the basic mission and several secondary missions, show external store loading, takeoff gross weight, combat radius and mission time in tabular form. Data for various combinations of missiles, bombs, rocket packages and external fuel tanks shall be shown. Data should be given to demonstrate the capabilities of fighter, attack, patrol and anti-submarine warfare aircraft across a broad spectrum of loadings. Data may be given for missions not covered in the Performance Summary (Paragraph 3.2.1.4).

3.2.1.6 Page 6. Performance Graphs. Performance data shall be shown graphically on the appropriate grids provided. Curves shall not extend beyond any applicable limit.

a. Speed. As a function of altitude, plot maximum speed at basic mission combat weight with maximum, intermediate, and normal thrust (power), as applicable. Show maximum speed for several alternate mission loadings at the same power settings to show the effects of drag of significant external stores and/or important weight changes.

b. Climb. As a function of altitude, plot rate of climb at basic mission combat weight with maximum, intermediate, or normal thrust (power), as applicable. Show rates of climb for alternate loadings in order to show the effects of drag changes with various external stores and/or important weight changes. The effects of weight reduction during climb shall not be considered.

c. Takeoff. Plot gross weight versus takeoff distance for sea level, zero wind, standard day and tropical day. Show lines for ground run distance, total distance over 50-foot obstacle and critical field length.

d. Fourth Block. Data shown here should depend on type of aircraft. For fighter aircraft, show maneuverability; i.e. Mach number versus load factor in a constant altitude turn for the basic mission at a typical combat altitude; show load factor at various rates of longitudinal acceleration. For attack aircraft, patrol aircraft, anti-submarine aircraft, early warning aircraft and mission-dedicated electronic aircraft, show search time; i.e., radius versus time on station; show lines for basic mission and for several typical missions. For trainer aircraft, plot range versus cruise altitude; show lines for basic mission and for several typical missions. For cargo aircraft, plot range versus cargo weight; show lines for normal and overload takeoff gross weights. For tanker aircraft, plot radius versus pounds of fuel transferred; show data for loiter times of 0.5, 1.0, 1.5 hours, etc.

3.2.1.7 Page 7. Trade-Off Graphs.

a. This page shall contain up to four graphs, showing performance trade-offs for fighter, attack, patrol and anti-submarine warfare aircraft (See page I-7 of Appendix I).

b. The trade-offs shown may be selected from the following or be special graphs designed by the contractor to more aptly display the capabilities of the aircraft: combat radius versus combat time, combat radius versus combat patrol time, combat radius versus dash radius, combat radius versus combat loiter time. On each graph, lines for several loadings from the Performance Summary (Paragraph 3.2.1.4) shall be given.

3.2.1.8 Page 8. Carrier Suitability Graphs.

a. For carrier based aircraft, performance graphs are required as follows: gross weight versus minimum wind over deck required for catapulting shown for catapult gear selected by the Navy, gross weight versus minimum wind over deck required for arrestment shown for arresting gear selected by the Navy, gross weight versus single engine rate of climb at V_{PA} approach speed, gross weight versus V_{PA} and versus V_{SPA} stall speed at approach power. Both standard and tropic day values shall be shown.

3.2.1.9 Page 9. External Store Loadings.

a. This page shall contain a simplified front view drawing of the aircraft with landing gear retracted showing external store stations. Under the drawing, a table shall be given with columns numbered corresponding to external store stations. The left hand column of the table shall contain a list of all possible external stores. Within the table, the number of each external store that can be carried at each external store station shall be given with the number 1, 2, etc.

3.2.1.10 Page 10. Notes.

a. For each mission, a sketch shall be given showing combat radius versus altitude with lines drawn showing climb, cruise to target, action over target, return cruise and descent. Altitude at significant points shall be called out in feet.

b. Above each sketch, the name of the mission shall be given with notes underneath. Notes for a typical mission are as follows:

1. For Taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. CLIMB: On course at best climb speed at intermediate thrust to best cruise altitude (not to exceed cruise ceiling).

3. CRUISE OUT: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. DESCENT: Descend to sea level (no fuel used, no distance gained).

5. COMBAT: Fuel allowance equal to 5 minutes at maximum speed with intermediate thrust at sea level. No distance is credited (drop bombs, retain mounting hardware and missiles after combat).

6. CLIMB: On course at best climb speed at intermediate thrust from sea level to best cruise altitude (not to exceed cruise ceiling).

7. CRUISE BACK: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

8. DESCENT: Descend to sea level (no fuel used, no distance gained).

9. RESERVE: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

a. Mission Time: Items 2 through 8.
b. Cycle Time: Items 2 through 9.

3.3 SUBSTANTIATING DATA. All data presented on the charts shall be substantiated by reports which shall be submitted with the charts. The reports may be in legible rough draft form utilizing the contractor's worksheet copy, but they shall be complete and shall contain a list of adequate references, authority and justification for all data used. Contractors are free to use calculation methods of their own selection, but such methods shall be fully explained and sample calculations shall be given. Calculations shall be presented in sufficient detail to permit ready review and check of conclusions and to enable additional calculations to be made by the Navy as required.

3.3.1 Basic Aerodynamic Data Report. Prior to preparation of the formal Substantiating Data Report, the approval of the Navy shall be obtained for the data which will form the basis for the Standard Aircraft Characteristics Charts. These basic data, including adequate calculations and material for verification shall include those data described in appendix II of this report. Data not accepted by the Navy shall be replaced after conference with the contractor by similar data to be designated by the Navy.

3.3.1.1 Revisions. The Basic Aerodynamic Data Report shall be revised under the same criteria as the charts (Paragraph 3.1.1).

3.3.2 Substantiating Performance Data Report. The Basic Aerodynamic Data (Paragraph 3.3.1) shall, after verification and approval by the Navy, form the basis for the detailed preparation of the formal Substantiating Data Report. These data shall be expanded as necessary and used to prepare the detailed performance data required to substantiate the validity of the Standard Aircraft Characteristics Charts. The Substantiating Data Report also serves as the data base for the Naval Air Training and Operating Procedures Standardization (NATOPS) Flight Manual.

3.3.2.1 Revisions. The substantiating data report shall be revised under the same criteria as the charts (Paragraph 3.1.1).

3.3.2.2 Text. The arrangement of the substantiating data report shall be arranged as shown in Appendix III.

3.4 STANDARDS. Characteristics and performance data shall be based on practical engineering analysis which produce results consistent with flight test results of vehicles of like types using standard operating procedures.

3.4.1 Basis for Data. All characteristics and performance data shall be based on the latest reliable aerodynamic, power plant and weight information available. The information given shall include the effect on weight and/or performance of all authorized contract and service changes, together with important changes assured of authorization but pending at the date of chart issue.

3.4.1.1 Changes in Characteristics. Changes in aircraft characteristics which do not result in a significant performance change (Paragraph 3.1.1) need not be justification for a revision by the contractor. However, the Navy shall be notified by correspondence so that proper notation may be appended to the published chart.

3.4.1.2 Flight Tests. Latest approved flight test data shall be used, as soon as available, as a basis for performance. While official military flight test results are to be preferred, contractor flight test results shall be considered, provided:

1. The contractor submits his method of flight test, instrumentation used and analysis leading to reduction of test results to standard conditions for review and approval by the Navy.

2. Specific "raw" test data and data reduced to standard conditions are provided to the Navy for approval prior to use in chart preparation.

3.4.1.3 Guarantees. The data quoted need not necessarily reflect contractor's aircraft performance guarantees.

3.4.2 Limitations. Performance data shall fall within all established limitations on the vehicle and its components, except as specifically provided herein.

3.4.3 Aircraft Condition. Performance shall be presented in such a manner as to show clearly the applicable aerodynamic configuration, power plant and loading information. Aircraft configurations shall include the installation of complete service equipment applicable to that particular aircraft model for the mission concerned. No special sealing of doors or cracks, filling of seams, waxing, or polishing shall be allowed, unless this is standard practice and is so stated on the charts. Flight performance shall be presented with guns, rotatable enclosures, bomb bay doors, etc., in position of least drag, retractable enclosures and wheels in retracted or closed position and external bombs or other armament in position for each condition, as noted. Fuel loadings shall comprise only those for which service approval has been obtained.

3.4.4 Atmosphere. Performance shall be based on the latest approved standard atmospheric tables as specified by the Navy.

3.4.4.1 Standard Day. Unless otherwise specified, performance shall be based on the latest approved ICAO standard atmosphere (59°F @ S.L.) as tabulated and described in Appendix IV.

3.4.4.2 Non-Standard Day. Unless otherwise specified, non-standard day performance shall be based on MIL-STD-210 Tropical (89.8°F @ S.L.) and Hot (103°F @ S.L.) conditions tabulated and described in Appendix IV.

3.5 DEFINITIONS. The following definitions are used for the various data on the charts and shall be strictly adhered to.

3.5.1 Weights. Weights used in preparation of and presented on the charts shall conform with the following definitions and be consistent with MIL-STD-1374, MIL-W-25140 and Technical Manual 01-1B-40.

3.5.1.1 Weight Empty. The weight empty condition shall be as defined in the latest model detail specification (does not include crew, fuel, armament, cargo, bombs, disposable or special equipment). The empty weight to be used in preparation of the SAC Chart shall be, in order of preference, the actual empty weight, the latest available calculated or estimated empty weight.

3.5.1.2 Basic Weight. Configuration for operating purposes, as defined in Technical Manual of Weight and Balance Data, NAVAIR 01-1B-40 and AIR FORCE Technical Order 01-1B-40 (empty weight plus trapped fuel and all fixed armament and equipment for normal operation).

3.5.1.3 Operating Weight. Zero fuel and zero payload weight - a convenience weight to which operators need add only fuel and payload for gross weight. (Basic weight plus crew and any special equipment that may be required. Does not include usable fuel, ammo, bombs, or auxiliary fuel tanks if such tanks are to be dropped in flight).

3.5.1.4 Design Weight. Weight at which specified flight structural design requirements are met or are required to be met.

3.5.1.5 Ramp Weight. Maximum in-flight weight unrefueled plus fuel, water, etc., used during takeoff. Shall not exceed other limits, such as maximum taxi weight, ground handling, wheel jacking, etc.; as specified in MIL-A-008860.

3.5.1.6 Combat Weight. Weight over the target for the mission presented with fuel, ammunition (including missile ordnance) used for air-to-air combat but without bombs, missiles (used for attack of surface targets), torpedoes, mines, cargo or droppable tanks unless otherwise noted.

1. For aircraft without external tanks, fuel load shall be 60 percent of initial usable fuel.

2. For aircraft with external drop tanks, fuel load shall be 60 percent of initial usable fuel load, or full internal fuel, whichever is less.

3.5.1.7 Takeoff Weight. Takeoff weight is the total weight of the aircraft with the fuel and payload for the mission presented. The takeoff weight normally shall be determined prior to start of engines, except in specially approved cases when weight expended during taxi and takeoff are excluded (Paragraph 3.5.1.5). Takeoff weight shall not exceed maximum takeoff weight.

3.5.1.7.1 Maximum Takeoff Weight. Maximum takeoff weight is the greatest weight for takeoff established by Technical Orders, design requirements, or other specific recommendations of the Navy.

3.5.1.7.2 Overload. Unless otherwise specified by the Navy, the maximum (overload) takeoff weight shall not exceed the least determined by the following:

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a. The weight of the aircraft fully loaded with fuel, bombs and cargo to capacity for which space and/or tankage is normally provided. Bearing capacity for the floor and/or supporting structure shall not be exceeded. The expendable weight of ATO and water used for takeoff may be added to the quoted maximum takeoff weight provided the criteria of Paragraph 3.5.1.7.2.b is satisfied and a qualifying note appears on the chart.

b. The aircraft and its components (wing, landing gear, supporting structure for ordnance, cargo, etc.) shall sustain at least a 2.0g normal load factor for each phase of operation and shall meet the minimum criteria of the applicable specifications for taxi and ground handling (MIL-A-008860).

c. The maximum rate of climb at Sea Level altitude under standard atmospheric conditions shall not be less than 500 ft/min with all engines operating at normal (maximum continuous) rating.

d. For multi-engine aircraft without ejection seats, the rate of climb with one engine inoperative at Sea Level altitude under standard atmospheric conditions shall not be less than 100 ft/min at the 50 feet obstacle, out of ground effect, in takeoff configuration and at maximum takeoff engine rating minus easily jettisonable items (i.e., external fuel tanks and bombs). For multi-engine aircraft with ejection seats, the required rate of climb will depend on the mission and will be specified by the Naval Air Systems Command.

e. Throughout the flight, the center of gravity shall remain within the limits for satisfactory ground handling and flight.

f. Such other criteria as may be specified by the Navy for the specific aircraft model presented.

g. The following takeoff criteria will be based on Sea Level altitude and standard atmospheric conditions:

1. For all vehicles (single and multi-engine), critical field length shall not exceed 8,000 feet.
2. The maximum tow force for carrier operation and maximum acceptable wind-over-deck, as defined by the Naval Air Systems Command, shall not be exceeded.

3.5.1.7.3 Normal. The maximum (normal takeoff weight) shall not exceed the least weight determined by criteria of Paragraph 3.5.1.7.2 and the following additional criteria:

a. For multi-engine aircraft without ejection seats, the maximum thrust (power) rate of climb with one engine inoperative (propeller feathered or rotor windmilling) shall not be less than 100 ft/min at takeoff speed, in takeoff configuration at sea level under a 103°F atmospheric temperature minus easily jettisonable items (i.e., external fuel tanks and bombs). For multi-engine aircraft with ejection seats, the required rate of climb will depend on the mission and will be specified by the Naval Air Systems Command.

b. The aircraft shall be capable of cruising at airspeeds of maximum range at 5,000 feet altitude (pressure) on a 103°F day with power not exceeding 70 percent of normal (maximum continuous) rating for reciprocating engines and 85 percent for gas turbine engines.

c. The takeoff distance over 50 feet height at sea level on a 103°F day shall not exceed 10,000 feet.

3.5.1.8 Maximum In-Flight Weight. Weight at which the aircraft is authorized to be airborne. It is possible to be greater than maximum takeoff weight if in-flight refueling is utilized.

3.5.1.9 Landing Weight.

3.5.1.9.1 Normal. The weight as determined by the computation of the mission ground rules. It shall include the fuel reserve as specified in Paragraph 3.6.4.

3.5.1.9.2 Maximum. Maximum landing weight is the greatest weight established for landing by flight restrictions, detailed specifications, or specific recommendations of the Navy.

3.5.1.10 Payload. The load which justifies the mission. Payload includes cargo, personnel other than crew (passengers), bombs, chaff, missiles (offensive and decoy), reconnaissance cameras, photo flash flares, bombs, fuel carried for transfer by tankers, ammunition and air-to-air missiles carried by fighter aircraft and gunnery trainers. Special equipment required for the mission, such as winterization, rescue equipment (except that carried for drop by "H" type aircraft [search-rescue]), cargo handling, etc., shall not be included in payload. The maximum zero fuel weight limitations must be observed when selecting payload.

3.5.1.11 Weight Definition Guide. For the information of the contractor, the following guide (ref. MIL-W-25140) is given to the above weight definitions:

3.5.1.12 Fuel. Standard weight of fuel in pounds per U.S. gallon shall be as follows:

a. MIL-G-5572, Gasoline in all grades - 6.0 lbs/gal.

b. MIL-T-5624, JP-4, Jet Fuel - 6.5 lbs/gal.

c. MIL-T-5624, JP-5, Jet Fuel - 6.8 lbs/gal.

d. MIL-T-83133, JP-8, Jet Fuel - 6.7 lbs/gal.

3.5.2 Speed. All speeds shall be true airspeeds in knots and Mach number as applicable (Mach number for jet aircraft, true airspeed for propeller aircraft).

3.5.2.1 Maximum Speed. Highest speed obtainable in level flight. State the weight, altitude and engine-power rating. Such maximum speed shall be within all operating restrictions (i.e., thrust, structural, heating limitations), with the limiting restriction noted. For the Performance Summary, maximum speed shall be at specified weight, maximum thrust (power) and the altitude for best speed (or Navy designated conditions).

3.5.2.1.1 Level Flight Maximum Speed (V_M). The maximum speed attainable at the basic flight design gross weight in the basic configuration in level flight with maximum available thrust (power) including use of afterburners, rocket thrust augmentation considering engine limitations, or afterburners and rocket thrust augmentation considering engine limitations, whichever is applicable.

3.5.2.1.2 Limit Speed (V_L). For the basic and high drag configurations, the maximum attainable speed commensurate with the operational use of the airplane considering shallow and steep dive angles, thrust, operation and nonoperation of speed brakes, and inadvertant upsets from gusts.

3.5.2.2 Combat Speed. Highest speed obtainable in level flight at combat weight with maximum thrust (power) at combat altitude.

3.5.2.3 Stall Speed. The stall speed shall be computed on the basis of 1.0g flight with the maximum trimmed lift coefficient established by computation or wind tunnel testing. Upon availability of flight test results, stall speed shall be changed to the highest of the speeds for steady straight 1.0g flight at C_L maximum, the speed at which abrupt loss of control occurs about any of the pitch, roll or yaw axes, the speed at which intolerable buffet or structural vibration is encountered, or other minimum permissible speed as defined in MIL-F-8785.

3.5.2.3.1 Power-Off. The stall speed without thrust (power) or, if significantly affected, with flight idle thrust (power).

3.5.2.3.2 Power-On. The stall speed using approach power which is defined as the thrust (power) required for level flight at 1.15 times the power-off stall speed in the landing configuration (Reference MIL-P-8785C).

3.5.2.4 Takeoff Speed

3.5.2.4.1 Takeoff Speed Criteria. Takeoff speed shall be the highest of the speeds specified by the following:

a. 1.1 times the speed represented by 90 percent maximum lift coefficient, power-on, including ground effects in the takeoff configuration.

b. A speed determined by the lift coefficient, in ground effect, for the maximum angle of attack attainable with the main landing gear also in the static position with aircraft on ground.

c. Minimum speed at which the aircraft has a climb gradient potential of 1/2 percent (0.005), with takeoff thrust (power), in the takeoff configuration, out-of-ground effect. For multi-engine aircraft, this potential shall be obtainable with the most critical engine inoperative (with rotor windmilling or propeller feathered). Where;

$$\text{gradient} = \frac{[\text{Vertical Height in Climb (ft)}]}{[\text{Horizontal Distance in Climb (ft)}]} \qquad (1)$$

d. 1.05 times air minimum control speed as specified in Paragraph 3.5.2.8.2.

e. The speed which permits attaining obstacle climb-out speed, as defined in Paragraph 3.5.2.6, at or before reaching 50 feet height above the runway.

3.5.2.4.1.1 Typical Design Criteria. For design purposes, consideration may be made of alternate definitions of takeoff speed such as: a higher or lower percentage of stall speed, a higher climb gradient potential or other criteria which optimizes the design.

3.5.2.4.2 Refusal Speed. The maximum speed during takeoff which will allow the aircraft to stop within the available remaining runway length.

3.5.2.5 Catapult End Airspeed (V_C). The airspeed required at the end of the catapult stroke to support the airplane under the conditions of attitude, lift, and longitudinal acceleration specified for catapulting. An example would be where the c.g. position of the aircraft shall sink no more than 10 feet from its position at the end of the power stroke, with a deck run not to exceed 32 feet (distance from the end of the power stroke to round-down), without exceeding the angle of attack for 0.9 $C_{L_{max}}$, power off, and with cockpit control position fixed for a specified loading, weight and thrust (power) setting. The aircraft is also required to have a longitudinal acceleration of a/g > .065 at zero flight path angle at the catapult end airspeed.

3.5.2.6 Obstacle Climb-Out Speed. The climb speed at the 50-foot obstacle shall not be less than the highest of the speeds specified below.

a. 1.2 times power-off stall speed with flaps in takeoff position, landing gear retracted.

b. 1.1 times air minimum control speed.

c. Speed at which the aircraft has a climb gradient of 2.5 percent (0.025) with gear up, flaps in takeoff position, with maximum thrust (power), out-of-ground effect. For multi-engine aircraft, this potential shall be obtainable with the most critical engine inoperative (rotor windmilling or propeller feathered).

d. If gear retraction results in a transient drag increase over that for gear down, the speed at which the aircraft has a 1/2 percent (0.005) climb gradient potential with flaps in takeoff setting, gear in transit, with maximum thrust (power) out-of-ground effect. For multi-engine aircraft, the most critical engine shall be inoperative (with propeller feathered or rotor windmilling).

3.5.2.7 Climb Speed. The climb speed shall be the airspeed at which the optimum rate-of-climb is attained for the given configuration, weight, altitude and power. Consideration shall be made for kinetic energy corrections in optimizing the climb speed schedule. When authorized in the applicable Flight Manual, a simplified, non-optimum speed schedule may be used.

3.5.2.8 Critical Engine Failure Speed. The critical engine failure speed shall be the speed occurring during the takeoff run at which an engine can fail and the same distance is required to either continue the takeoff to lift off or to stop the aircraft.

3.5.2.9 Minimum Engine-Out Control Speed.

3.5.2.9.1 Ground. The minimum control speed, ground, shall be the minimum speed during the takeoff run where the engine most critical to directional stability can fail and directional control can be maintained.

3.5.2.9.2 Air. The minimum control speed, air, shall be the minimum airborne speed with maximum thrust (power) where the engine most critical to stability can fail and control can be maintained under the conditions specified in MIL-P-8785.

3.5.2.10 Cruise Speed.

3.5.2.10.1 Maximum Range Cruise Speed. The speed for maximum range operation shall be the speed at which maximum nautical miles per pound of fuel are attainable at the momentary weight and altitude conditions.

3.5.2.10.2 Long Range Cruise Speed. The higher of the two airspeeds which give nautical miles per pound of fuel equal to 99 percent of the maximum nautical miles per pound of fuel for momentary weight and altitude unless otherwise limited by handling characteristics. For propeller driven aircraft, long range cruise speed is used in lieu of maximum range cruise speed.

3.5.2.10.3 Maximum Cruise Speed. The highest speed that can be maintained with stated thrust (power), altitude, weight and configuration.

3.5.2.10.4 Average Cruise Speed. Total distance covered in cruise divided by time for cruise (distance and time for climb, acceleration to combat speed, combat time and loiter time are not included).

3.5.2.11 Loiter Speed

3.5.2.11.1 Maximum Endurance (Loiter) Speed. The airspeed for maximum endurance shall correspond to the speed for minimum fuel flow attainable at momentary weight and altitude, except as limited by acceptable handling characteristics of the aircraft.

3.5.2.11.2 Combat Loiter Speed. The airspeed for combat loiter shall correspond to the speed for minimum fuel flow attained at momentary weight and altitude, except that the airspeed must be adequate to allow an instantaneous load factor of a specified value.

3.5.2.11.3 Corner Speed. The lowest airspeed at which the lift and structural lines intersect. This is the speed at which the maximum turn rate and minimum turn radius exists for the specified altitude.

3.5.2.12 Landing Speed. The landing speed shall be determined by the maximum angle of attack attainable with the main landing gear oleo strut positioned for the static condition. The landing speed shall not be less than 110 percent of power-off stall speed for the landing configuration. The speed over a 50-foot height shall be at least 120 percent of power-off stall speed in the landing configuration.

3.5.2.13 Landing Touchdown Speed. For design purposes landing touchdown speed is that speed equal to 1.05 times approach speed. For operational aircraft, landing touchdown speed will be determined using fleet survey data.

3.5.2.13.1 Approach Speed. With the aircraft in the landing configuration and on a 4° glide slope on a 89.8°F day, the minimum usable approach speed $V_{PA_{MIN}}$, shall be the highest of the airspeeds defined by the following:

a. The lowest speed at which it is possible to achieve a level flight longitudinal acceleration of 5 ft/sec² within 2.5 seconds after initiation of throttle movement and speed brake retraction.

b. V_{SPA} x 1.1 where V_{SPA} is the power-on stall speed using the thrust (power) required for level flight at 1.15 V_{S_L}, the power-off stall speed.

c. The lowest level flight speed at which the pilot, at the design eye position, can see the stern of the carrier at the waterline when intercepting a 4° glide slope at an altitude of 600 feet. The origin of the glide slope is 500 feet forward of the stern and 63 feet above the waterline.

d. The lowest speed at which all stability and control requirements are satisfied (MIL-F-8785).

e. The lowest speed at which the aircraft is capable of making a glide path correction from stabilized flight at $V_{PA_{MIN}}$ to a new glide path 50 feet above the original glide path within five (5) seconds after initiation of the maneuver. The maneuver shall be performed without change in thrust settings, and the aircraft angle of attack during the maneuver shall not exceed that necessary to achieve 50 percent of the maximum positive delta load factor available, based on static lift coefficient, at the initiation of the maneuver. Control rate input for simulation of $V_{PA_{MIN}}$ shall not exceed control system limits. The maneuver shall be considered complete when a glide path correction of 50 feet has been reached. After completion of this maneuver, the aircraft shall be capable of maintaining a new glide path at least 50 feet above and parallel to the initial glide path, with the pilot permitted to change thrust setting as required.

f. To insure rapid aircraft response to step throttle commands corresponding to + 3.86 ft/sec² longitudinal acceleration, such throttle inputs shall result in achieving 90 percent of the commanded acceleration within 1.2 seconds. This requirement shall apply in the approach configuration throughout the range of all throttle settings required for operations over the usable approach configuration weight/drag levels while trimmed on a 4° glide slope.

Note: Control rate input for simulation of $V_{PA_{MIN}}$ shall not exceed control system limits. Calculation of $V_{PA_{MIN}}$ shall be based on static lift coefficient.

3.5.3 Ceiling

3.5.3.1 Service Ceiling. Service ceiling is that altitude at which the rate of climb at subsonic speed is 100 ft/min at stated loading, weight and engine thrust (power).

3.5.3.2 Combat Ceiling.

3.5.3.2.1 Subsonic Vehicles. Combat ceiling for subsonic vehicles is that altitude at which the rate of climb is 500 ft/min at the stated loading, weight and thrust (power).

3.5.3.2.2 Supersonic Vehicles. Combat ceiling for supersonic vehicles is the highest altitude at which the vehicle can fly supersonically and have a 500 ft/min rate of climb at the stated loading, weight and thrust (power).

3.5.3.3 Cruise Ceiling

3.5.3.3.1 Subsonic Vehicles. Cruise ceiling for subsonic cruise vehicles is that altitude at which the rate of climb is 300 ft/min at normal (maximum continuous) engine rating at stated weight and loading.

3.5.3.3.2 Supersonic Vehicles. Cruise ceiling for supersonic cruise vehicles is that altitude at which the rate of climb is 300 ft/min at normal (maximum continuous) engine rating at stated weight and loading.

3.5.4 Altitude.

3.5.4.1 Cruise Altitude. The cruise altitude is the altitude at which the cruise portion of the missions is computed. Depending on the mission ground rules, the cruise altitude may be

assigned. Otherwise, it is governed by the following limitations: for pressurized aircraft, cruise ceiling shall not exceed the altitude where cabin altitude is 10,000 feet. For unpressurized aircraft with oxygen masks, cruise ceiling shall not exceed 20,000 feet (Reference OPNAVINST 3710.7K Paragraph 714). In no case shall cruise altitude exceed cruise ceiling.

3.5.4.2 Optimum Cruise Altitude. The altitude at which the aircraft attains the maximum nautical miles per pound of fuel for the momentary weight and configuration. If this altitude exceeds cruise ceiling, the latter shall be used for cruise.

3.5.4.3 Combat Altitude. Combat altitude is the altitude at the target for the specific mission shown.

3.5.5 Takeoff. Criteria for conventional takeoff aircraft shall comply with the following: (For STOL aircraft the criteria shall be determined by design criteria.) (Vertical components of thrust may be used in takeoff computation.)

3.5.5.1 Ground Run Distance (i.e., Takeoff Distance). Takeoff ground run distance shall be that normally obtainable in service operation at Sea Level with standard atmospheric conditions, zero wind, no runway slope on hard (concrete or asphalt) surfaced runways. For estimated data, the takeoff speed criteria of Paragraph 3.5.2.4 shall be used.

3.5.5.1.1 Typical Design Criteria. For design purposes, consideration may be made of alternate definitions of takeoff ground run such as: non-standard atmospheric conditions, higher pressure altitudes, alternate runway surfaces (hard, sod, etc.), head or tail-wind or other criteria in keeping with the operational concept of the design.

3.5.5.2 Distance To 50 Feet. The distance to clear a 50-foot obstacle shall be the sum of takeoff ground run distance of 3.5.5.1 plus the airborne distance required to accelerate and climb to arrive at the 50-foot height at the speed specified in 3.5.2.6.

3.5.5.3 Takeoff Time. The takeoff time shall be that normally obtainable in service operation at Sea Level under standard day atmospheric conditions with no wind. The time is measured from start of takeoff (brake release) to start of enroute climb (attainment of climb speed).

3.5.5.4 Critical Field Length. Critical field length is defined as the total length of runway required to accelerate on all engines to the critical engine failure speed, experience an engine failure, and either continue to takeoff or stop.

3.5.5.4.1 Data Basis. The data basis for the computation of the stopping distance for critical field length shall be as follows:

a. At engine failure speed, the aircraft continues to accelerate for 3 seconds with remaining engine(s) operating at maximum thrust (power) and with zero thrust on the inoperative engine.

b. At the end of the 3-second acceleration time, thrust (power) on all engines is instantaneously reduced to idle, brakes applied and decelaration devices deployed.

c. Sufficient time after b, above, shall be allowed for deployment of the deceleration device(s) or for reverse thrust to reach maximum before including its effect on deceleration.

3.5.5.5 Coefficient of Friction. The coefficient, μ, as used in this document is defined as the ratio of the total retardation force attributable to the braking system to the momentary gross weight of the aircraft (momentary gross weight defined as weight on wheels which is weight minus lift). The following values will be used unless ground or flight test data are available.

3.5.5.5.1 Rolling. The rolling (unbraked) coefficient of friction for a dry, hard runway shall be equal to 0.025.

3.5.5.5.2 Braking. The braking coefficient of friction for a dry, hard runway shall be equal to 0.3. Application of anti-skid criteria will be specified by the Navy.

3.5.5.5.3 Test Data. Test μ values may be either the results of tests conducted on the specific aircraft or similar types, i.e. commerical aircraft.

3.5.5.5.4 Typical Design Criteria. For design purposes, the following coefficient of friction values should be used for the conditions specified on a hard surfaced runway:

RUNWAY CONDITION	ROLLING UNBRAKED	BRAKES	ANTI-SKID BRAKES
Dry	.025	.30	.38
Wet (Rain)	.05	.14	.20
Snow	.09	.10	.15
Ice	.05	.07	.09

3.5.6 Climb. Climb after takeoff may be divided into two segments: initial climbout and enroute climb.

3.5.6.1 Initial Climb-Out. Climb-out shall be at a speed which shall not be less than that limited by the criteria of 3.5.2.6. Gear retraction shall be initiated as soon as an adequate positive climb gradient (3.5.2.6.c and d), using applicable power, has been established and maintained while accelerating to climb-out speed. Flaps shall be in the takeoff position. Enroute climb speed shall be reached at a height no greater than 1,000 feet above ground.

3.5.6.1.1 All Engines Operating. Initial climb-out with all engines operating shall be based on all engines operating from brake release to takeoff. Acceleration to climb-out speed and climb-out shall be based on the thrust (power) available with all available engines.

3.5.6.1.2 One Engine Inoperative. Initial climb-out with one engine inoperative shall be based on all engines operating from brake release to critical engine failure speed and with the

critical engine inoperative from critical engine failure speed to takeoff. Acceleration to climb-out speed and climb-out shall be based upon the thrust (power) available with the remaining engines at takeoff thrust (power) and the drag of the inoperative engine. If means of reducing drag of the inoperative engine are a design feature, such drag reduction shall be utilized with a time allowance for activation.

3.5.6.2 Enroute Climb. Except for point intercept missions, all climbs shall be enroute with thrust (power) and speed schedules optimized to maximize mission range. Point intercept missions shall be optimized to obtain minimum time to combat altitude.

3.5.6.2.1 Enroute Climb Data. Enroute climb data shall be based on the appropriate configuration, thrust (power) and weight. The aircraft shall have the landing gear and flaps retracted and have attained the airspeed for best climb for the applicable condition.

3.5.6.2.2 Enroute Climb Power. For jet (fighter, attack, trainers, etc.) aircraft enroute climb to cruise altitude shall be at intermediate (military) thrust. For propeller (patrol, transport, etc.) aircraft use maximum continuous power.

3.5.6.2.3 Typical Design Criteria. For design purposes, consideration may be given to alternate climb schedules to more adequately portray the desired operational capability of the design. For example, the following schedules could apply: minimum time, minimum fuel, maximum range, specified thrust (power) or speed, accelerate during climb, etc.

3.5.6.3 Time to Climb. The time to climb to a specified altitude(s) shall be expressed in minutes from start of enroute climb. Weight reduction as a result of fuel consumption shall be applied to the calculations.

3.5.6.4 Combat Climb. Combat climb is the instantaneous maximum vertical speed capability in feet per minute at combat conditions, such as, weight, configuration, altitude, and thrust (power).

3.5.7 Landing Distance. The following criteria are for conventional aircraft. (For STOL aircraft, the criteria shall be as established by design requirements). Landing distance includes: (a) landing ground roll and (b) distance over a 50-foot height. Distances shall be for the landing configuration and weight and shall be based on the landing speeds defined in 3.5.2.12. Unless otherwise specified, ground roll deceleration shall be based on operation at Sea Level, standard day, zero wind, no runway slope, on hard (concrete or asphalt) surfaced runways, idle thrust (power) and a braking coefficient as defined in 3.5.5.5.2. Factors that should be considered are pilot reaction time, thrust decay, aerodynamic and mechanical braking, and maximum brake capacity.

3.5.7.1 Typical Design Criteria. For design purposes, consideration may be given to alternate definitions of landing distance such as: reverse thrust, atmospheric conditions alternate runway surfaces, runway slopes and winds of a non-standard nature, a rigid computer analysis of the air distance and other similar criteria selected to optimize the design of the airplane.

3.5.8 Thrust (Power). The term thrust (power) is used to mean thrust (jet engine) and/or brake horsepower (shaft engines) as applicable with due consideration for installation effects and limitations. Engine and ATD ratings as defined in Paragraph 3.2.1.3.c shall be those which appear in the approved engine model specification without regard to installation effects or limitations.

3.5.8.1 Maximum Thrust (Power). Maximum thrust (power) is the highest thrust (power) which the engine will consistently deliver at specific ground or flight conditions for the durations (incremental and total) specified in the model specification for demonstration during the qualification or preliminary flight rating tests.

3.5.8.2 Intermediate Thrust (Power). Intermediate thrust (power) is the highest thrust (power) which the engine will consistently deliver at specific ground or flight conditions for an incremental duration of at least 30 minutes, and a total duration as specified in the engine model specification for demonstration during qualification or preliminary flight rating tests. Intermediate thrust is equivalent to the old term military thrust.

3.5.8.3 Maximum Continuous Thrust (Power). Maximum continuous thrust (power) is the highest thrust (power) which the engine will consistently deliver at specific ground or flight conditions for an unlimited time period.

3.5.8.4 Cruise Thrust (Power). The thrust (power) required to fly the aircraft at cruise speed for the configuration, altitude and weight designated.

3.5.8.5 Idle Thrust (Power). Idle thrust (power) is the lowest thrust (power) which the engine will consistently deliver at specific ground or flight conditions for an unlimited duration or as defined in the engine model specification for demonstration during qualification or preliminary flight rating tests.

3.5.9 Fuel Consumption Service Tolerance. Unless authorized otherwise, for proposal aircraft, all fuel consumption data, regardless of source, shall be increased by 5 percent for all engine thrust (power) conditions as a service tolerance to allow for practical operation. In addition, corrections or allowances to engine fuel flow shall be made for all power plant installation losses such as accessory drives, ducts, fans, cabin pressure bleed, etc. Fuel consumption data will not be increased by 5 percent for service aircraft, if verified by Navy approved flight test.

3.5.10 Mission Types. For pre-service and service SAC Charts (1.4.3 and 1.4.4), the tabulated performance data of SAC Chart page 4 shall show performance data for missions designated by the Navy. Typical missions for various types of aircraft are

shown in Appendix V for use in Proposal SAC Charts (1.4.1). Unless otherwise specified, reserve fuel shall conform to the allowances shown in Appendix V.

3.5.10.1 Clean Mission. The first mission to be described in the Standard Aircraft Characteristics Charts will be the Clean Mission. This mission is intended to show the maximum capabilities of the aircraft (usually a high-high-high profile).

3.5.10.2 Basic Mission. The basic mission is the mission profile detailed in Appendix V which most nearly depicts the primary intended operational use of the aircraft. To maintain the capability of presenting a direct comparison between similar type aircraft, no deviation from the ground rules of Appendix V can be allowed.

3.5.10.3 Design Mission. The design mission is defined as the primary mission for which the aircraft was specifically procured. This mission will normally be defined in procurement documents such as the statement of work and will include the flight profile, allowances, fuel (clean or external tanks) and payload. Ground rules and allowances for the design mission are dictated by the specific operational requirements and will be used in describing the mission capabilities in the Standard Aircraft Characteristics Charts.

3.5.10.4 Ferry Mission. Ferry range is the greatest distance attainable on a practicable one-way mission with maximum authorized fuel and no pay load according to a specified sequence of operations, allowances and reserves. External fuel tanks may be carried and must be retained for the duration of the flight.

3.5.10.5 Typical Missions. Any missions, preferably from Appendix V, which would present the additional capabilities of the aircraft. If different from Appendix V, the mission definitions should be coordinated with the Navy.

3.5.10.6 In-Flight Refueled Mission. For aircraft capable of in-flight refueling, a refueled mission is the greatest distance (radius or range) attainable through receipt of replacement fuel during flight. A single refueling operation is required although multiple refueling operations may be used if considered to be feasible. Basic ground rules from Appendix V shall apply.

3.5.11 Combat Radius. Combat radius is the distance (including distance covered in climb) attainable on a practicable flight to the target and return a distance equal to that flown out, carrying a specific load (bombs, cargo, personnel, etc.), to or from the target according to a sequence of operations specified under "Mission Types" (Paragraph 3.5.10). Droppable fuel tanks are discussed in Paragraph 3.6.1.6.

3.5.12 Combat Range. Combat range is the distance (including distance covered in climb) attainable on a practicable one-way flight carrying payload (bombs, cargo, personnel) the entire distance. Droppable fuel tanks are not dropped when empty.

3.6 MISSION DETAILED REQUIREMENTS.

3.6.1 General Mission Requirements. Unless otherwise specified, the following general ground rules shall apply:

3.6.1.1 Standard Atmosphere. Data shall be presented for standard day atmosphere.

3.6.1.2 Wind. Data shall be for a no-wind condition.

3.6.1.3 Formation Flight. Data shall be for a single aircraft only.

3.6.1.4 Ordnance Expenditure. Ammunition and air-to-air missiles shall not be expended during the mission.

3.6.1.5 Off-Loading Fuel. Fuel may be off-loaded to avoid exceeding the maximum allowable takeoff weight.

3.6.1.6 External Fuel Tanks. For combat radius missions only, external fuel tanks shall be dropped when empty or prior to combat unless such tanks are designed to be carried during combat. Unless otherwise restricted (e.g. CG, etc.), dropping of external tanks shall be sequenced to provide maximum range. Cargo, tanker and training missions for Attack and Fighter aircraft shall not drop external tanks.

3.6.1.7 Pylons/Racks. Bomb racks, etc. shall not be jettisoned with the external stores. Pylons shall be retained during return to base.

3.6.1.8 Reduced Engine Operation. When applicable, a minimum number of engines may be used to increase range or loiter time if such operation would represent normal service usage. However, such action shall conform to Paragraph 3.6.1.9.

3.6.1.9 Authorized Operation. No operational technique shall be utilized that is not, or is not intended to be included as a recommended procedure in the applicable flight manual.

3.6.1.10 Trainer Aircraft. The trainer basic missions, as defined in Appendix V is applicable to basic and advanced trainer airplanes. Combat and tactical trainer airplanes fly the basic mission for the appropriate parent-type airplane.

3.6.1.11 Variable Geometry Wing (VGW) Aircraft. For VGW aircraft, the automatic sweep program will be clearly defined and used unless otherwise noted. If not automatic, the VGW aircraft will be assumed to have wings in the unswept position for takeoff and subsonic flight and swept for supersonic dash unless footnoted otherwise.

3.6.2 Mission Loading Requirements. In order to facilitate and expedite the make-up and delivery of the charts, it is suggested that the contractor contact the procuring service to discuss the various mission loadings prior to submission. In the absence of special instructions, the following shall apply:

3.6.2.1 Basic Mission Loading. The fuel and payload loading for the basic mission shall be the basic loading condition as defined by the first load condition given in the detail specification weight statement for the aircraft.

3.6.2.2 Typical Mission Loadings. Loadings shall be selected from those included in the detail specification or other approved loadings which depict a particular capability of the

aircraft. At least one mission shall conform to the maximum (overload) gross weight (Paragraph 3.5.1.7.2).

3.6.2.3 Ferry Mission Loadings. Loading shall consist of maximum authorized fuel and no payload.

3.6.2.4 In-Flight Refueling Mission Loading. One mission shall be for the same loading as the basic mission. Other loadings may be selected by the typical missions.

3.6.2.5 Combat Range Mission Loading. Identical to the loading of the associated combat radius mission.

3.6.3 Mission Segments. Rationale for mission segments is presented in the following:

3.6.3.1 Takeoff. Ground operation, including starting engines, warm-up, taxi, takeoff and acceleration to climb speed are variable. An arbitrary fuel allowance, based on statistical analysis, must be used. The takeoff fuel allowance used for gas turbine engine and turboprop powered aircraft is a quantity of fuel equal to the fuel used during 4.6 minutes of intermediate thrust operation at Sea Level Standard Day (10 minutes maximum continuous power for reciprocating engines). If afterburners are required for use during takeoff, a fuel allowance equal to [illegible] seconds of operation at maximum thrust at Sea Level Standard Day must be used along with fuel for 4.6 minutes operation at intermediate thrust.

3.6.3.2 Climb. See Paragraph 3.5.6.

3.6.3.2.1 Typical Design Criteria. For design purposes, consideration may be given to defining the takeoff allowance to fully utilize the state-of-the-art. Some typical examples are:

a. Specify engine operation for specific time periods at specified powers. Such as, fuel used during 4.6 minutes of intermediate thrust operation at Sea Level, on a standard day plus 30 seconds of maximum thrust operation if afterburner is used during takeoff.

b. Estimate fuel required to start the engine(s), run-up, taxi a specified distance at a specified thrust (power) setting and to accelerate from brake release to climb speed at a specified power.

c. Estimate fuel for a specified time at a specified thrust (power)/weight ratio to account for starting and taxi plus fuel for takeoff and acceleration to climb speed computed from the following:

$$W_{F_{TO}} = \frac{V_c W_{TO}}{2g} \cdot \frac{(\dot{W}_o + \dot{W}_c)}{T-D} \qquad (2)$$

Where:

$W_{F_{TO}}$ = takeoff and acceleration fuel, lbs.

V_c = initial climb speed, ft/sec.

W_{TO} = takeoff weight, lbs.

$\dot{W}_o$ = static fuel flow at takeoff power, lbs/sec.

$\dot{W}_c$ = fuel flow at initial climb speed at takeoff power, lbs/sec.

T-D = thrust minus drag at V_c, lbs.

g = acceleration of gravity, Sea Level, ft/sec^2

NOTE:

If thrust (power) is to be varied between lift-off and climb speed, this equation can be so modified.

d. Other specific criteria may be selected to more accurately portray the operational characteristics of the specific design.

3.6.3.3 Cruise. Unless specifically assigned, aircraft shall cruise at the speed and altitude for maximum specific range (optimum cruise altitude) for the applicable configuration, thrust (power) and weight. This altitude shall not exceed cruise ceiling. For aircraft having a low optimum altitude (e.g.: Reciprocating Engine Aircraft), the cruise altitude shall not be less than 5,000 feet for terrain clearance over land or 1,500 feet over water. Except where the altitude is specified, the aircraft may utilize a cruise climb to optimize cruise distance. Turbojet and turbofan driven aircraft shall cruise at maximum range cruise speed (Paragraph 3.5.2.10.1) and propeller driven aircraft shall cruise at long range cruise speed (Paragraph 3.5.2.10.2).

3.6.3.3.1 Typical Design Criteria. For design purposes, consideration may be given to specifying a cruise technique selected to optimize the desired characteristics of the design. Techniques to be considered include: constant altitude cruise, constant speed cruise, cruise climb profile, step climb, cruise at specified power, cruise with reduced number of engines, cruise altitude in excess of cruise ceiling, fixed distance segment, headwinds or tail winds, non-standard temperatures, etc.

3.6.3.4 Combat. Combat shall be considered by setting aside a quantity of fuel to be used for that purpose if required. Normally, fuel flow for this allowance shall be based on the level flight stabilized speeds for the altitude and thrust (power) stated in Appendix V. The change in speed due to weight reduction during the combat period shall be ignored. When more than one thrust (power) setting is used, the lesser thrust (power) will be used first and each treated independently. For task-oriented fuel allowances, computation shall be based upon weight at start of combat period with benefit due to weight reduction credited; change in speed due to weight reduction shall be ignored.

3.6.3.4.1 Escape and Evasion. Escape and evasion shall be considered by setting aside a quantity of fuel based upon a specified measure of performance.

3.6.3.4.2 Typical Design Criteria. For design purposes, consideration may be given to various methods of accounting for the fuel to be used during combat or escape and evasion action. Some examples of methods are:

a. Fuel required for a specified time with a specified thrust (power) at a specified speed and a specified altitude.

b. Fuel consumed in expending a specified quantity of energy. For example:

$$\text{Combat Fuel} = \frac{E_s \dot{W}_f}{P_s} \qquad (3)$$

When:

E_s = specific energy, feet.

$\dot{W}_f$ = fuel flow at combat speed, power, and altitude, lbs/sec.

P_s = excess energy or $\frac{(T-D)V_c}{W_{T_c}}$, ft/sec.

V_c = combat true airspeed, ft/sec.

W_{T_c} = combat weight, lbs.

(T-D) = thrust minus drag, lbs.

c. The quantity of fuel determined as the sum of the fuel required to accelerate from cruise speed to a specified speed, plus fuel required to make a specified number of sustainable turns at a specified speed or speeds. These operations shall be performed at a selected thrust (power)(s) and altitude(s).

d. All or a portion of the armament may be expended.

e. Other specific criteria selected to more accurately portray the operational characteristics of the specific design.

3.6.3.5 Search and Loiter. Speed shall be as given in Paragraph 3.5.2.11.

3.6.3.6 Descent. For vehicles whose best cruise is subsonic, no time, fuel or distance will be credited for descent. For supersonic cruise vehicles, credit may be taken for descent and deceleration to an altitude of 25,000 feet and a specified speed. Aircraft which conduct a supersonic runout from the target may, if the cruise altitude and speed are lower than the runout altitude and speed, account for distance in descent and deceleration to cruise.

3.6.3.6.1 Typical Design Criteria. For design purposes, consideration may be given to alternate definitions of descent. For example: time, fuel and distance could be credited, descent could be a long range (airline) approach, use of thrust (power) could vary from none to full, speeds could vary from near stall to redline, altitudes could be reduced in step increments, etc. For gas turbine powered aircraft, the criteria will usually be idle thrust (power) descent to a given altitude at a given speed, i.e., M=0.8 to Sea Level (outbound leg to target area), 250 KCAS to 20,000 feet (return leg to land).

3.6.4 Landing Reserve. Fuel onboard at landing shall be the greater of the following:

a. Fuel allowance equal to 10 percent of initial usable fuel as required by OPNAVINST 3710.7K Paragraph 326.

b. Fuel allowance equal to 20 minutes loiter (30 minutes for cargo and transport aircraft) at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total usable fuel (internal plus external).

3.6.4.1 Typical Design Criteria. For design purposes, consideration may be given to defining the landing reserves to fully utilize the state-of-the-art. Some examples are:

a. The fuel required for a ground controlled approach; a wave-off, go-around and a second, successful landing. This could be approximated by using the equivalent of fuel consumed during a specified time at maximum endurance at Sea Level with all engines operating.

b. A specified percentage of initial fuel load.

c. Fuel consumed during a specified time of operation at a specified power at a specified altitude.

d. The greater of the fuel required for 10 percent of mission time or 20 minutes at maximum endurance speed at 10,000 feet.

e. Fuel required to fly to an alternate field (specify distance) plus a specified time at a specified speed at a specified altitude to account for landing.

f. Combinations of the above or other criteria selected to optimize the design.

3.6.5 Mission Time. Time in air (excludes time before start of initial climb and reserve).

3.6.6 Cycle Time. The time of flight from the start of enroute climb (omitting takeoff time) to stopping engines after landing.

3.6.7 Block Time. The total time of flight from start engines to stop engines after landing.

3.6.8 Intercept Time. The time from engine start until initiation of combat at the intercept altitude. This time includes the period required for takeoff and acceleration to climb speed.

SECTION 1

BASIC AERODYNAMIC DATA REPORT

INTRODUCTION

SECTION 1. INTRODUCTION
2. THRUST/LIFT/DRAG BOOKKEEPING PROCEDURE FOR PERFORMANCE CALCULATIONS
3. EXAMPLE TABLE OF CONTENTS
4. EXAMPLE PLOTS

INTRODUCTION

Prior to proceeding with the initial performance calculations for the Standard Aircraft Characteristics Charts, the following basic data, including all calculations and materials necessary to substantiate these data, shall be submitted to NAVAIR (AIR-53012) for acceptance [normally not later than sixty (60) days following receipt of Authority to Proceed unless specifically extended by NAVAIR (AIR-53012)]. The aerodynamic and propulsion system thrust/drag bookkeeping procedures for performance calculations are outlined in Section 2 of this Appendix and shall be utilized, where applicable to the particular design, in the presentation of basic data and for performance calculations. Prior to subsequent performance calculations, these basic data shall be submitted.

1. Low speed drag analysis itemized according to various aircraft components (wing, fuselage, drag devices, etc.).

2. Plots of parasitic drag coefficient, $C_{D_{MIN}}$; incremental skin friction drag coefficient due to variation of flat plate skin friction drag with Reynolds number (fully turbulent boundary layer case adjusted for the effects of compressibility should be utilized); lift coefficient for minimum drag, C_{L_D}, as a function of Mach number; and trimmed airplane efficiency factor "e" versus Mach number for various values of lift coefficient. Trimmed airplane drag polars at various values of Mach number may be submitted in lieu of the "e" and C_{L_D} curves. Incremental drag coefficient, ΔC_D, versus Mach number for drag devices and for each required external store and combination of stores shall be presented.

3. Standard Day net thrust available and fuel flow variation with altitude and Mach number for maximum afterburner, intermediate and normal engine operation (including idle power) with all losses (induction system, nozzle, compressor bleed and accessory drive power extraction) indicated. The variation of fuel flow with net thrust as a function of Mach number and altitude for partial thrust operation shall be presented. Also, intermediate and maximum (afterburner) thrust available at Sea Level, 89.6°F, between Mach equal 0 and 0.3. Engine ram drag shall also be presented for all of the above engine conditions. When applicable, one engine inoperative data and partial afterburner net thrust versus fuel flow should be presented at specific altitudes/conditions.

4. Untrimmed lift coefficient plotted against pitching moment, angle-of-attack and drag coefficient for three horizontal tail positions, without thrust effect, in the takeoff and landing configurations, and both in and out of ground effect. Trimmed lift coefficient versus angle-of-attack and drag coefficient, without thrust effect, for the takeoff and approach configuration (in and out of ground effect) and for the clean configuration.

5. Breakout of wetted areas and aircraft dimensional data.

6. Area distribution for MN = 1.0 (Total and broken down by components).

7. Catapult:

   a. Load-stroke data for main and nose gear.

   b. Load-tire deflection data for main and nose gear. For 7.a and 7.b, specify if load is total load, load on one side, or load on one wheel.

   c. Moment of inertia about Y-Y axis (pitch).

   d. CM ξ and CM s.

8. Drawings:

   a. General Arrangement, 1/20 scale dimensioned, including CG and thrust line locations.

   b. Landing gear (side view) showing main and nose gear hub locations (station and waterline) in extended and compressed position; tow bar attachment point and tow bar length.

NOTE: Data not accepted by NAVAIR shall be replaced, after conference with the contractor, by similar data designated by NAVAIR.

Section 3 presents an example "Table of Contents" that illustrates the format for the Basic Aerodynamic Data Report. It is required that this format be used unless exempted by NAVAIR (AIR-53012).

Section 4 presents example plot formats to be used in the graphical presentation of data for the Basic Aerodynamic Data Report. The aircraft and configuration should be noted where applicable. This includes maneuver devices and schedules (Mach and angle-of-attack) for these devices. For maximum lift coefficient, include maximum defensive and offensive limits, maximum tracking and buffet onset. On all external stores show effect of C_L on drag coefficient. Also, effect of speed brakes should be shown, where applicable.

SECTION 2

BASIC AERODYNAMIC DATA REPORT

THRUST/LIFT/DRAG BOOKKEEPING PROCEDURES
FOR PERFORMANCE CALCULATIONS

1. This Appendix discusses the requirements for the presentation and calculation of aircraft performance. These procedures outlined in Figure II-1, are intended for use during design, development and test of the aircraft (whether performance calculations are based on analytical, wind tunnel or flight analysis) and shall be utilized, where applicable to the particular design, (a) in the presentation of basic data, (b) for aircraft performance calculations, and (c) to enhance generalization of aerodynamic data obtained from flight test.

2. The basis for the thrust/lift/drag bookkeeping procedures and attendant list of symbols and definitions is:

a. Realism in the allocation of aerodynamic and propulsion system force components to thrust and drag.

b. The inclusion of all force components independent of engine power setting in the aircraft lift/drag characteristics.

c. The inclusion of all force components which are functions of engine throttle setting and/or induction and exhaust system geometry in installed propulsion system characteristics.

d. The availability of aerodynamic/propulsion system interaction effects and Reynolds number effects to generalize flight-measured lift/drag data for correlation of flight-measured and wind-tunnel-derived full-scale lift/drag characteristics.

3. The "reference" configuration between aerodynamic force and moment and propulsion wind tunnel tests shall be as follows:

Inlet: Inlet(s) operating at the critical mass flow ratio or the Mach number being tested. Inlet drag for this condition will be included in aircraft drag.

Nozzle/afterbody: The nozzle(s) should be in the full-open position, with $P_{EXIT}/P_{\infty} = 1.0$. Nozzle/afterbody drag for this condition is included in aircraft drag.

FIGURE II-1. THRUST/LIFT/DRAG BOOKKEEPING PROCEDURES FOR PERFORMANCE CALCULATIONS

LIST OF SYMBOLS AND DEFINITION OF TERMS

Alt	ALTITUDE
A_{EXIT}	NOZZLE EXIT AREA
C_D	DRAG COEFFICIENT (CLEAN CONFIGURATION)
C_{Di}	INDUCED DRAG COEFFICIENT
$\Delta C_{D_{AB}}$	INCREMENTAL NOZZLE/AFTERBODY DRAG COEFFICIENT THAT IS A FUNCTION OF ENGINE THROTTLE POSITION
$C_{D_{min}}$	MINIMUM DRAG COEFFICIENT (INCLUDING SEA LEVEL SKIN FRICTION DRAG, ROUGHNESS AND EXCRESCENCE, RAM COOLING AND VENTILATION, INLET BLEED DRAG, SUPERCRITICAL INLET SPILLAGE DRAG AND BASELINE NOZZLE/AFTERBODY DRAG)
$\Delta C_{D_{RN}}$	INCREMENTAL SKIN FRICTION DRAG COEFFICIENT DUE TO VARIATION OF FLAT PLATE SKIN FRICTION DRAG WITH REYNOLDS NUMBER
$\Delta C_{D_{SB}}$	INCREMENTAL DRAG COEFFICIENT DUE TO SPEEDBRAKE DEFLECTION
$\Delta C_{D_{ST}}$	INCREMENTAL DRAG COEFFICIENT DUE TO CARRIAGE OF EXTERNAL STORES
$\Delta C_{D_{SUB}}$	INCREMENTAL SUBCRITICAL INLET SPILLAGE DRAG COEFFICIENT
$C_{D_{TOT}}$	TOTAL DRAG COEFFICIENT (C_D AND $C_{D_{RN}}$ AND $C_{D_{ST}}$ AND $C_{D_{SB}}$)
C_L	LIFT COEFFICIENT
CG	CENTER OF GRAVITY POSITION
D	DRAG
F_g	INSTALLED ENGINE GROSS THRUST
F_R	ENGINE RAM DRAG
L	LIFT
M	MACH NUMBER
m	INLET MASS FLOW OF OPERATING ENGINE CONDITION

LIST OF SYMBOLS AND DEFINITION OF TERMS

m_o	THEORETICAL MAXIMUM MASS FLOW FOR REFERENCE INLET CAPTURE AREA
$(m/m_o)_{oper}$	INLET OPERATING MASS FLOW RATIO
NPF	NET PROPULSIVE FORCE ($F_g \cos(\alpha + \tau) - F_R - qS_{REF}(C_{D_{SUB}} + C_{D_{AB}})$)
NPR	NOZZLE PRESSURE RATIO (P_{EXIT}/P_a)
N_z	AIRCRAFT ACCELERATION ALONG VERTICAL AXIS
P_a	AMBIENT PRESSURE
P_{EXIT}	NOZZLE EXIT STATIC PRESSURE
$P_{T_{EXIT}}$	NOZZLE EXIT TOTAL PRESSURE
P_{T_o}	FREESTREAM TOTAL PRESSURE
P_{T_2}	TOTAL PRESSURE AT ENGINE COMPRESSOR FACE
P_{T_2}/P_{T_o}	INLET PRESSURE RECOVERY
q	DYNAMIC PRESSURE
RN	REYNOLDS NUMBER
S.L.	SEA LEVEL
S_{REF}	REFERENCE WING AREA FOR AERODYNAMIC COEFFICIENTS
W	AIRPLANE GROSS WEIGHT
W_a	INLET AIRFLOW
W_{BL}	ENGINE COMPRESSOR BLEED AIRFLOW
α	ANGLE-OF-ATTACK
δ	RATIO OF AMBIENT TO STANDARD SEA LEVEL STATIC PRESSURE
θ	RATIO OF AMBIENT TO STANDARD SEA LEVEL STATIC TEMPERATURE
τ	LONGITUDINAL THRUST LINE INCLINATION

SECTION 3

BASIC AERODYNAMIC DATA REPORT

EXAMPLE TABLE OF CONTENTS

1.0 INTRODUCTION

2.0 AIRCRAFT DESCRIPTION AND TABULATED DATA

- 2.1 Three View Drawing and Configuration Schematic Drawings
- 2.2 Tabulated Aircraft Dimensional Data
- 2.3 Catapulting Geometry
- 2.4 Wetted Area
- 2.5 Gross Sectional Area
- 2.6 Weight, Center of Gravity and Inertia Data Summary
- 2.7 Powerplant and Propeller (if required) Data

3.0 AERODYNAMIC DATA

- 3.1 Source of Data
- 3.2 Thrust Drag Bookkeeping
- 3.3 High Speed and Maneuvering Configuration (Flaps and Gear Up)
  - 3.3.1 Drag Substantiation
  - 3.3.2 Lift Data
- 3.4 Delta Drag Coefficients for Pylons and External Stores
- 3.5 Miscellaneous Drag, i.e., Speed Brake, etc.
- 3.6 High Lift Configuration Characteristics
  - 3.6.1 General
  - 3.6.2 Catapult and Approach Configuration In and Out of Ground Effects
  - 3.6.3 Landing Gear Drag Coefficients
  - 3.6.4 Delta Drag for Single Engine Trim

4.0 PROPULSION DATA

- 4.1 Source of Data
- 4.2 Installation Effects
- 4.3 Installed Performance
  - 4.3.1 Standard Day
  - 4.3.2 Tropical Day Takeoff Data (Mach Number $\leq$ 0.4)

5.0 AERODYNAMIC ENVELOPE

- 5.1 Structural Limits (V_H and Load Factor)
- 5.2 Engine Limitations
- 5.3 Alpha Restrictions

6.0 ANGLE-OF-ATTACK AND AIRSPEED CORRECTIONS

- 6.1 Angle-of-Attack (Degrees to Units)
- 6.2 Position Error (Mach Number and Airspeed)

7.0 ABBREVIATIONS AND SYMBOLS

8.0 REFERENCES

SECTION 4

BASIC AERODYNAMIC DATA REPORT

EXAMPLE PLOTS

HIGH SPEED LIFT & DRAG CHARACTERISTICS

FIGURE - 1

FIGURE - 2

HIGH SPEED LIFT & DRAG CHARACTERISTICS (CONT'D)

FIGURE - 3

FIGURE - 4

FIGURE - 9

FIGURE - 10

FIGURE - 5

FIGURE - 6

FIGURE - 11

FIGURE - 12

FIGURE - 7

FIGURE - 8

LOW SPEED HIGH LIFT CONFIGURATIONS

FIGURE - 13

FIGURE - 14

HIGH SPEED LIFT & DRAG CHARACTERISTICS (CONT'D)

FIGURE - 15

FIGURE - 16

FIGURE - 17

PROPULSION DATA

FIGURE - 18

FIGURE - 19

FIGURE - 20

FIGURE - 21

FIGURE - 22

SPECIALIZED PROPELLER AIRCRAFT

FIGURE - 23

FIGURE - 24

FIGURE - 27

FIGURE - 28

FIGURE - 25

FIGURE - 26

FIGURE - 29

FIGURE - 30

SECTION 1

SUBSTANTIATING DATA REPORT

INTRODUCTION

SECTION 1. INTRODUCTION
2. EXAMPLE TABLE OF CONTENTS
3. EXAMPLE PLOTS

INTRODUCTION

The performance Substantiating Data Report shall be prepared using the data contained in the Basic Aerodynamic Data Report which has been accepted by NAVAIR (AIR-53012). It shall include data to substantiate the information given in the Standard Aircraft Characteristics Charts and must be sufficiently complete to expedite a prompt investigation and review of the derived performance and to permit additional calculations to be made by NAVAIR (AIR-53012). The report also provides the general basis for and should be consistent with the Flight Manual Performance Data.

Section 2 presents an example "Table of Contents" that illustrates the format for the performance Substantiating Data Report. This format outlines the pertinent data to be provided and should be used unless exempted by NAVAIR (AIR-53012).

Section 3 presents example plot formats to be used in the graphical presentation of data for the performance Substantiating Data Report. Where applicable, one engine inoperative data should be provided in addition to the all engines operating condition. In the high lift configuration, data should be provided for all normal and emergency conditions. These include the various flap settings, lift devices (slats, etc.), and drag devices (speed brakes, etc.) available for pilot use. Field stopping/landing distance should be determined with and without reverse thrust, if appropriate.

The graphical presentation should be clear and concise with scales easy to read. Any questions concerning scales or format should be directed to NAVAIR (AIR-53012).

SECTION 2

SUBSTANTIATING DATA REPORT

EXAMPLE TABLE OF CONTENTS

1.0 SUMMARY (INCLUDING SAC CHARTS)

2.0 INTRODUCTION (INCLUDES DATA SOURCE)

3.0 TABULATED DATA

3.1 Aircraft General Arrangement

3.2 Aircraft Dimensional Data

3.3 Weight and Configuration Summary
(Includes tabulation of weight used in SAC Chart)

3.4 Aircraft Basic Useful Load

3.5 Fuel and Oil

3.6 Propulsion System
(Reference Engine Specification) Dimensions, Capture Area, Uninstalled Thrust, RPM (N_2), SFC, Airflow, Time Limit for Various Power Settings

4.0 MISSION AND TRADE-OFF SUMMARIES

5.0 PERFORMANCE DATA PRESENTATION METHOD

6.0 MAXIMUM MACH NUMBER
(Includes V_L, V_H and other limits)

7.0 CEILINGS
(Absolute, Service, Cruise and Combat Ceilings for Applicable Power Settings)

8.0 CLIMB PERFORMANCE

8.1 Climb - Speed Schedule

8.2 Rate-of-Climb

8.3 Time, Distance and Fuel to Climb
(Maximum Range and Minimum Time)

8.3.1 Intermediate Power

8.3.2 Maximum Power

9.0 CRUISE PERFORMANCE

9.1 Optimum Cruise

9.2 Cruise at Constant Mach and Altitude

10.0 ENDURANCE AND LOITER

10.1 Optimum Loiter

10.2 Specific Endurance at Constant Altitude

11.0 SPECIALIZED PERFORMANCE

11.1 Buddy Refueling

11.2 Search Performance (Flat Turning and Orbiting)

11.3 Loiter at Corner Speeds

11.4 Bingo

11.5 Single Engine Cruise

12.0 DESCENT PERFORMANCE

12.1 Rate of Descent

12.2 Fuel, Distance and Time to Descend (Optimum and Tactical)

13.0 ACCELERATION AND DECELERATION

13.1 Acceleration to Climb Speed

13.2 Level Acceleration at Applicable Power Settings (Intermediate and Maximum Power)

13.3 Deceleration Performance

14.0 MANEUVERABILITY

14.1 Specific Excess Power

14.2 Maneuverability Envelopes

14.2.1 Sustained

14.2.2 Instantaneous

15.0 LONGITUDINAL ACCELERATION FOR CATAPULT TAKEOFF

16.0 CATAPULT PERFORMANCE

17.0 CARRIER APPROACH SPEED AND ARRESTING WIND-OVER-DECK

18.0 STALL SPEEDS

19.0 FIELD TAKEOFF

20.0 FIELD LANDING

21.0 SINGLE ENGINE RATE-OF-CLIMB

22.0 ABBREVIATIONS AND SYMBOLS

23.0 REFERENCES

SECTION 3

SUBSTANTIATING DATA REPORT

EXAMPLE PLOTS

PLOTS ARE ARRANGED TO CONFORM WITH SECTION 2:
ITEMS 5, 22 AND 23 OMITTED

4.0 MISSION TRADEOFF SUMMARIES

FIGURE - 1

FIGURE - 2

FIGURE - 3

FIGURE - 4

FIGURE - 5

6.0 MAXIMUM MACH NUMBER

FIGURE-6

FIGURE - 7

7.0 CEILINGS

FIGURE - 8

FIGURE - 9

FIGURE - 10

8.0 CLIMB PERFORMANCE

FIGURE - 11

FIGURE - 12

FIGURE - 13

FIGURE - 14

FIGURE - 15

FIGURE - 16

9.0 CRUISE PERFORMANCE

FIGURE - 17

FIGURE - 18

FIGURE - 19

FIGURE - 20

FIGURE - 21

FIGURE - 22

9.0 CRUISE PERFORMANCE CONT'D

FIGURE - 23

FIGURE - 24

FIGURE - 25

10. ENDURANCE & LOITER

FIGURE - 26

FIGURE - 27

FIGURE - 28

FIGURE - 29

FIGURE - 30

12.0 DESCENT PERFORMANCE

FIGURE - 31

FIGURE - 32

13.0 ACCELERATION PERFORMANCE

FIGURE - 33

FIGURE - 34

FIGURE - 35

14.0 MANEUVERABILITY COMBAT PERFORMANCE

FIGURE - 36

FIGURE - 37

FIGURE - 38

OPERATIONAL ENVELOPES

FIGURE - 39

FIGURE - 40

15.0 LONGITUDINAL ACCELERATION

FIGURE - 41

FIGURE - 42

16.0 CATAPULT PERFORMANCE

FIGURE - 43

FIGURE - 44

17.0 CARRIER APPROACH SPEED & ARRESTING WIND-OVER-DECK

FIGURE - 45

FIGURE - 46

18.0 STALL SPEEDS

FIGURE - 47

FIGURE - 48

19.0 FIELD TAKE-OFF

FIGURE - 49

FIGURE - 50

20.0 FIELD LANDING

FIGURE - 51

FIGURE - 52

21.0 SINGLE ENGINE RATE-OF-CLIMB

FIGURE - 53

FIGURE - 54

MISCELLANEOUS PROPULSION DATA

FIGURE - 55

FIGURE - 56

FIGURE - 57

FIGURE - 58

APPENDIX V

MISSION PROFILES

GENERAL MISSIONS

* 1. HI-HI-HI

2. Fighter Escort

3. Fighter Escort (Alternate)

* 4. Deck Launched Intercept

5. Deck Launched Intercept (Alternate)

* 6. Combat Air Patrol

* 7. Close Support

8. Ferry/Cross Country Navigation

* 9. Interdiction (HI-LO-LO-HI)

10. Interdiction (Alternate)

11. HI-LO-HI

12. LO-LO-LO

13. LO-LO-LO-HI

* 14. ASW SEARCH

* 15. ASW

* 16. MINELAYING

* 17. WEAPONS DELIVERY/ GUNNERY (TRAINING)

* MISSION PROFILES INCLUDED IN THIS TEXT. FOR OTHERS SEE AS-5263

HI-HI-HI
(High Altitude Subsonic)

1. For taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate power to best cruise altitude (not to exceed cruise ceiling).

3. Cruise Out: To target at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. Combat: Fuel allowance equal to 5 minutes at maximum speed with intermediate thrust at best cruise altitude. No distance is credited (drop bombs, retain mounting hardware and missiles after combat).

5. Cruise Back: To base at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

6. Descent: Descend to sea level (no fuel used, no distance gained).

7. Reserve: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 6.
(b) Cycle Time: Items 2 through 7.

DECK LAUNCHED INTERCEPT

1. For taxi, warmup, takeoff and acceleration to Mach 0.3: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Acceleration: Maximum power acceleration from Mach 0.3 to Mach 0.9 at sea level.

3. Climb: On course at Mach 0.9 at maximum power to 35,000 feet.

4. Acceleration: Maximum power acceleration from Mach 0.9 to Mach 1.35 at 35,000 feet.

5. Dash Out: Mach 1.35 dash at 35,000 feet.

6. Combat: Fuel allowance equal to 1 minute at maximum power, Mach 1.35 at 35,000 feet (no distance is credited, missiles are retained).

7. Climb: On course at best climb speed at intermediate power to best cruise altitude (not to exceed cruise ceiling).

8. Cruise Back: To base at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

9. Descent: Descend to sea level (no fuel used, no distance gained).

10. Reserve: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 9.
(b) Cycle Time: Items 2 through 10.

NOTE: Dash Mach and Altitude variations should be considered for this mission.

COMBAT AIR PATROL

1. For taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust to best cruise altitudes (not to exceed cruise ceiling).

3. Cruise Out: To 150 nautical miles at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. Descent: Descend to 35,000 feet (no fuel used, no distance gained).

5. Loiter: Loiter at speed for maximum endurance at 35,000 feet (no distance is credited).

6. Combat: Fuel allowance equals that used to accelerate from loiter speed at 35,000 feet to Mach 1.2 plus 2 minutes at maximum power, Mach 1.2 at 35,000 feet (no distance is credited, missiles are retained).

7. Climb: On course at best climb speed at intermediate thrust to best cruise altitude (not to exceed cruise ceiling).

8. Cruise Back: To base at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

9. Descent: Descend to sea level (no fuel used, no distance gained).

10. Reserve: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 9.
(b) Cycle Time: Items 2 through 10.

NOTE: Loiter altitude and combat variations should be considered for this mission.

CLOSE SUPPORT

1. For taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust to best cruise altitude (not to exceed cruise ceiling).

3. Cruise Out: To target at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. Descent: Descend to 5,000 feet (no fuel used, no distance gained).

5. Loiter: Loiter for 1 hour at speed for maximum endurance at 5,000 feet (no distance is credited, drop bombs after loiter, retain mounting hardware and missiles).

6. Climb: On course at best climb speed at intermediate thrust from 5,000 feet to best cruise altitude (not to exceed cruise ceiling).

7. Cruise Back: To base at speed and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

8. Descent: Descend to sea level (no fuel used, no distance gained).

9. Reserve: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 8.
(b) Cycle Time: Items 2 through 9.

INTERDICTION

1. For taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust to best cruise altitude (not to exceed cruise ceiling).

3. Cruise Out: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. Descent: Descend to sea level (no fuel used, no distance gained).

5. Run-in to Target: Sea level dash for 50 nautical miles at Mach 0.8 (or maximum speed at intermediate thrust if less than Mach 0.8).

6. Combat: Fuel allowance equal to 5 minutes at intermediate thrust, Mach 0.8 (or maximum speed if less than Mach 0.8) at sea level. No distance is credited (drop bombs, retain mounting hardware and missiles after combat).

7. Run-out from Target: Sea level dash for 50 nautical miles at Mach 0.8 (or maximum speed at intermediate thrust if less than Mach 0.8).

8. Climb: On course at best climb speed at intermediate thrust from sea level to best cruise altitude (not to exceed cruise ceiling).

9. Cruise Back: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

10. Descent: Descend to sea level (no fuel used, no distance gained).

11. Reserve: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 10.
(b) Cycle Time: Items 2 through 11.

ASW SEARCH

1. For Taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust (10 minutes for propeller engine aircraft at normal power) plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust (normal power for props) to search altitude (not to exceed cruise ceiling).

3. Cruise Out: At search altitude at speed for maximum endurance (unless otherwise limited by handling qualities).

4. Cruise Back: At search altitude at speed for maximum endurance (unless otherwise limited by handling qualities).

5. Descent: Descend to sea level (no fuel used, no distance gained).

6. Reserve: Fuel allowance equal to 20 minutes (30 minutes for props) loiter at sea level at speeds for maximum endurance (maximum range for props) with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 5.
(b) Cycle Time: Items 2 through 6.

ASW

1. For Taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust (10 minutes for propeller engine aircraft at normal power) plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust (normal power for props) to best cruise altitude (not to exceed cruise ceiling).

3. Cruise Out: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. Descent: Descend to 20,000 feet (no fuel used, no distance gained).

5. Search: Search for 3 hours at speed for maximum endurance at 20,000 feet.

6. Descent: Descend to 200 feet (no fuel used, no distance gained).

7. Search: Search for 1 hour at speed for maximum endurance at 200 feet.

8. Climb: On course at best climb speed at intermediate thrust (normal power for props) to best cruise altitude (not to exceed cruise ceiling).

9. Cruise Back: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

10. Descent: Descend to sea level (no fuel used, no distance gained).

11. Reserve: Fuel allowance equal to 20 minutes (30 minutes for props) loiter at sea level at speeds for maximum endurance (maximum range for props) with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 10.
(b) Cycle Time: Items 2 through 11.

MINELAYING

1. For Taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust (10 minutes for propeller engine aircraft at normal power) plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust (normal power for props) to best cruise altitude (not to exceed cruise ceiling).

3. Cruise Out: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

4. Descent: Descend to 200 feet (no fuel used, no distance gained).

5. Penetrate: At maximum continuous power for 300 nautical miles at 200 feet.

6. Attack: At maximum continuous power for 100 nautical miles at 200 feet.

7. Release Mines.

8. Escape: On course at maximum continuous power for 300 nautical miles.

9. Climb: On course at best climb speed at intermediate thrust (normal power for props) from 200 feet to best cruise altitude (not to exceed cruise ceiling).

10. Cruise Back: At speeds and altitudes for best range, using a cruise climb flight path (not to exceed cruise ceiling).

11. Descent: Descend to sea level (no fuel used, no distance gained).

12. Reserve: Fuel allowance equal to 20 minutes (30 minutes for props) loiter at sea level at speeds for maximum endurance (maximum range for props) with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 11.
(b) Cycle Time: Items 2 through 12.

WEAPONS DELIVERY/GUNNERY

1. For Taxi, warmup, takeoff and acceleration to best climb speed: Fuel allowance at sea level static equal to 4.6 minutes at intermediate thrust plus 30 seconds afterburner thrust if afterburner is used for takeoff.

2. Climb: On course at best climb speed at intermediate thrust to 20,000 feet.

3. Cruise Out: 60 nautical miles from initial climb point at speed for best range at 20,000 feet.

4. Descent: Descend to gunnery altitude (10,000 feet) or weapons delivery altitude (sea level), (no fuel used, no distance gained).

5a. Gunnery Option: Time for gunnery shall be allocated as follows: 60% at intermediate thrust (Mach to be designated by NAVAIR), 25% at 80% intermediate thrust fuel flow (Mach to be designated by NAVAIR), and 15% at speed for best range. (no distance gained).

5b. Weapons Delivery Option: Time for weapons delivery shall be allocated as follows: 50% at intermediate thrust (Mach to be designated by NAVAIR), and 50% at 80% intermediate thrust fuel flow (Mach to be designated by NAVAIR). (no distance gained).

6. Climb: On course at best climb speed at intermediate thrust from gunnery (10,000 feet) or weapons delivery (sea level) altitude.

7. Cruise Back: 60 nautical miles from climb point at speed for best range at 20,000 feet.

8. Descent: Descend to sea level (no fuel used, no distance gained).

9. Reserve: Fuel allowance equal to 20 minutes loiter at sea level at speeds for maximum endurance with all engines operating plus 5% of initial total fuel (internal plus external).

(a) Mission Time: Items 2 through 8.
(b) Cycle Time: Items 2 through 9.

B2. Airplane Stability, Control and Flying Qualities

NOTES: 1. FOR SYMBOLS SEE P. 330-341
2. FOR CONTENTS SEE P. 341-343

B2.1 MIL-F-8785C: Military Specification, Flying Qualities of Piloted Airplanes, November 1980.

This specification is approved for use by all Departments and Agencies of the Department of Defense.

1. SCOPE

1.1 Scope. This specification contains the requirements for the flying and handling qualities, in flight and on the ground, of U.S. Military, manned, piloted airplanes except for flight at airspeeds below V_{con} (MIL-F-83300).

It is intended to assure flying qualities that provide adequate mission performance and flight safety regardless of design implementation or flight control system mechanization. The structure of the specification allows its use to guide these aspects in design tradeoffs, analyses and tests.

1.2 Application. The flying qualities of all airplanes proposed or contracted for shall be in accordance with the provisions of this specification. The requirements apply as stated to the combination of airframe and related subsystems. Stability augmentation and control augmentation are specifically to be included when provided in the airplane. The automatic flight control system is also to be considered to the extent stated in MIL-F-9490 or MIL-C-18244, whichever applies. The requirements are written in terms of cockpit flight controls that produce essentially pitching, yawing and rolling moments. This approach is not meant to preclude other modes of control for special purposes. Additional or alternative requirements may be imposed by the procuring activity in order to fit better the intended use or the particular design.

1.3 Classification of airplanes. For the purpose of this specification, an airplane shall be placed in one of the following Classes:

Class I Small, light airplanes such as:
- Light utility
- Primary trainer
- Light observation

Class II Medium weight, low-to-medium maneuverability airplanes such as:
- Heavy utility/search and rescue
- Light or medium transport/cargo/tanker
- Early warning/electronic countermeasures/airborne command, control or communications relay
- Antisubmarine
- Assault transport
- Reconnaissance
- Tactical bomber
- Heavy attack
- Trainer for Class II

Class III Large, heavy, low-to-medium maneuverability airplanes such as:
- Heavy transport/cargo/tanker
- Heavy bomber
- Patrol/early warning/electronic countermeasures/airborne command, control, or communications relay
- Trainer for Class III

Class IV High-maneuverability airplanes such as:
- Fighter/interceptor
- Attack
- Tactical reconnaissance
- Observation
- Trainer for Class IV

The procuring activity will assign an airplane to one of these Classes, and the requirements for that Class shall apply. When no Class is specified in a requirement, the requirement shall apply to all Classes. When operational missions so dictate, an airplane of one Class may be required by the procuring activity to meet selected requirements ordinarily specified for airplanes of another Class.

1.3.1 Land- or carrier-based designation. The letter -L following a Class designation identifies an airplane as land-based. Carrier-based airplanes are similarly identified by -C. When no such differentiation is made in a requirement, the requirement shall apply to both land-based and carrier-based airplanes.

1.4 Flight Phase Categories. The Flight Phases have been combined into three Categories which are referred to in the requirement statements. These Flight Phases shall be considered in the context of total missions so that there will be no gap between successive Phases of any flight and so that transition will be smooth. In certain cases, requirements are directed at specific Flight Phases identified in the requirement. When no Flight Phase or Category is stated in a requirement, that requirement shall apply to all three Categories. Flight Phases descriptive of most military airplane missions are:

Nonterminal Flight Phases:

Category A: Those nonterminal Flight Phases that require rapid maneuvering, precision tracking, or precise flight-path control. Included in this Category are:

a. Air-to-air combat (CO)
b. Ground Attack (GA)
c. Weapon delivery/launch (WD)
d. Aerial recovery (AR)
e. Reconnaissance (RC)
f. In-flight refuelling (receiver) (RR)
g. Terrain following (TF)
h. Antisubmarine search (AS)
i. Close formation flying (FF)

Category B: Those nonterminal Flight Phases that are normally accomplished using gradual maneuvers and without precision tracking, although accurate flight-path control may be required. Included in this Category are:

a. Climb (CL)
b. Cruise (CR)
c. Loiter (LO)
d. In-flight refueling (tanker) (RT)
e. Descent (D)
f. Emergency descent (ED)
g. Emergency deceleration (DE)
h. Aerial delivery (AD)

Terminal Flight Phases:

Category C: Terminal Flight Phases are normally accomplished using gradual maneuvers and usually require accurate flight-path control. Included in this category are:

a. Takeoff (TO)
b. Catapult takeoff (CT)
c. Approach (PA)
d. Wave-off/go-around (WO)
e. Landing (L)

When necessary, recategorization or addition of Flight Phases or delineation of requirements for special purpose situations, e.g. zoom climbs, will be accomplished by the procuring activity.

1.4 Levels of flying qualities. Where possible, the requirements of Section 3 have been stated in terms of three values of the stability or control parameter being specified. Each value is a minimum condition to meet one of three Levels of acceptability related to the ability to complete operational missions for which the airplane is designed. The Levels are:

Level 1: Flying qualities clearly adequate for the mission Flight Phase.

Level 2: Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists.

Level 3: Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.

2. APPLICABLE DOCUMENTS

2.1 Issues of documents. The following documents, of the issue in effect on the date of invitation for bids or request for proposal, form a part of this specification to the extent specified herein:

SPECIFICATIONS

MILITARY

MIL-D-8708	Demonstration Requirements for Airplanes
MIL-A-8861	Airplane Strength and Rigidity Flight Loads
MIL-F-9490	Flight Control Systems- Design, Installation and Test of, Piloted Aircraft, General Specification for
MIL-C-18244	Control and Stabilization Systems, Automatic, Piloted Aircraft, General Specification for
MIL-F-18372	Flight Control Systems, Design, Installation and Test of, Aircraft, General Specification for
MIL-W-25140	Weight and Balance Control Data (for Airplanes and Rotorcraft)
MIL-F-83300	Flying Qualities of Piloted V/STOL Aircraft
MIL-S-83691	Stall/Post-Stall/Spin Flight Test Demonstration Requirements for Airplanes

STANDARDS

MIL-STD-756	Reliability Prediction

(Copies of specifications and standards required by contractors in connection with specific procurement functions should be obtained from the procuring activity or as directed by the contracting officer).

3. REQUIREMENTS

3.1 General Requirements

3.1.1 Operational missions. The procuring activity will specify the operational missions to be considered by the contractor in designing the airplane to meet the flying quality requirements of this specification. These missions will include all associated Flight Phases and tasks, such as takeoff, takeoff abort, landing and missed approach. Operational missions include the entire spectrum of intended usage including aircrew upgrade and training.

3.1.2 Loadings. The contractor shall define the envelopes of center-of-gravity and corresponding weights that will exist for each Flight Phase. These envelopes shall include the most forward and aft center-of-gravity positions as defined in MIL-W-25140. In addition, the contractor shall determine the maximum center-of-gravity excursions attainable through failures in systems or components, such as fuel sequencing, hung stores, etc., for each Flight Phase to be considered in the Failure States of 3.1.6.2. Within these envelopes, plus a growth margin to be specified by the procuring activity, and for the excursions cited above, this specification shall apply.

3.1.3 Moments and products of inertia. The contractor shall define the moments and products of inertia of the airplane associated with all loadings of 3.1.2. The requirements of this specification shall apply for all moments and products of inertia so defined.

3.1.4 External stores. The requirements of this specification shall apply for all combinations of external stores required by the operational missions. The effects of external stores on the weight, moments of inertia, center-of-gravity position, and aerodynamic characteristics of the airplane shall be considered for each mission Flight Phase. When the stores contain expendable loads, the requirements of this specification apply throughout the range of store loadings. The external stores and store combinations to be considered for flying qualities design will be specified by the procuring activity. In establishing external store combinations to be investigated, consideration shall be given to asymmetric as well as to symmetric combinations.

3.1.5 Configurations. The requirements of this specification shall apply for all configurations required or encountered in the applicable Flight Phases of 3.4. A (crew-) selected configuration is defined by the positions and adjustments of the various selectors and controls available to the crew except for pitch, roll, yaw, throttle and trim controls. Examples are: the flap control setting and the yaw damper ON or OFF. The selected configurations to be examined must consist of those required for performance and mission accomplishment. Additional configurations to be investigated may be defined by the procuring activity.

3.1.6 State of the airplane. The State of the airplane is defined by the selected configuration together with the functional status of each of the airplane components or systems, throttle setting, weight, moments of inertia, center-of-gravity position, and external store complement. The trim setting and the positions of the pitch, roll and yaw controls are not included in the definition of Airplane State since they are often specified by the requirements.

3.1.6.1 Airplane Normal States. The contractor shall define and tabulate all pertinent items to describe the Airplane Normal (no component or system failure) State(s) associated with each of the applicable Flight Phases. This tabulation shall be in the format and shall use the nomenclature specified in 6.2. Certain items, such as weight, moments of inertia, center-of-gravity position, wing sweep, or thrust setting may vary continuously over a range of values during a Flight Phase. The contractor shall replace this continuous variation by a limited number of values of the parameter in question which will be treated as specific States, and which include the most critical values and extremes encountered during the Flight Phase in question.

3.1.6.2 Airplane Failure States. The contractor shall define and tabulate all Airplane Failure States, which consist of Airplane Normal States modified by one or more malfunctions in airplane components or systems, for example, a discrepancy between a selected configuration and an actual configuration. Those malfunctions that result in center-of-gravity positions outside the center-of-gravity envelope defined in 3.1.2 shall be included. Each mode of failure shall be considered. Failures occurring in any Flight Phase shall be considered in all subsequent Flight Phases.

3.1.6.2.1 Airplane Special Failure States. Certain components, systems, or combinations thereof may have extremely remote probability of failure during a given flight. These failure probabilities may, in turn, be very difficult to predict with any degree of accuracy. Special Failure States of this type need not be considered in complying with the requirements of Section 3 if justification for considering the Failure States as Special is submitted by the contractor and approved by the procuring activity.

3.1.7 Operational Flight Envelopes. The Operational Flight Envelopes define the boundaries in terms of speed, altitude and load factor within which the airplane must be capable of operating in order to accomplish the missions of 3.1.1. Envelopes for each applicable Flight Phase shall be established with the guidance and approval of the procuring activity. In the absence of specific guidance, the contractor shall use the representative conditions of Table I for the applicable flight phases.

TABLE I OPERATIONAL FLIGHT ENVELOPES

Flight Phase Category: A

Flight Phase	Airspeed		Altitude		Load Factor	
	$V_{o_{min}}$ ($M_{o_{min}}$)	$V_{o_{max}}$ ($M_{o_{max}}$)	$h_{o_{min}}$	$h_{o_{max}}$	$n_{o_{min}}$	$n_{o_{max}}$
Air-to-air Combat (CO)	$1.4V_S$	V_{MAT}	MSL	Combat Ceiling	-1.0	n_L
Ground Attack (GA)	$1.3V_S$	V_{MRT}	MSL	Medium	-1.0	n_L
Weapon Delivery/ Launch (WD)	V_{range}	V_{MAT}	MSL	Combat Ceiling	0.5	*
Aerial Recovery (AR)	$1.2V_S$	V_{MRT}	MSL	Combat	0.5	n_L
Reconnaissance (RC)	$1.3V_S$	V_{MAT}	MSL	Combat Ceiling	*	*
In-Flight Refuel (Receiver) (RR)	$1.2V_S$	V_{MRT}	MSL	Combat Ceiling	0.5	2.0
Terrain Following (TF)	V_{range}	V_{MAT}	MSL	10,000 ft	0	3.5
Antisubmarine Search (AS)	$1.2V_S$	V_{MRT}	MSL	Medium	0	2.0
Close Formation Flying (FF)	$1.4V_S$	V_{MAT}	MSL	Combat Ceiling	-1.0	n_L

Flight Phase Category: B

Flight Phase	Airspeed		Altitude		Load Factor	
	$V_{o_{min}}$ ($M_{o_{min}}$)	$V_{o_{max}}$ ($M_{o_{max}}$)	$h_{o_{min}}$	$h_{o_{max}}$	$n_{o_{min}}$	$n_{o_{max}}$
Climb (CL)	$0.85V_{R/C}$	$1.3V_{R/C}$	MSL	Cruise Ceiling	0.5	2.0
Cruise (CR)	V_{range}	V_{MRT}	MSL	Cruise Ceiling	-1.0	2.0
Loiter (LO)	$0.85V_{end}$	$1.3V_{end}$	MSL	Cruise Ceiling	0.5	2.0
In-Flight Refuel (Tanker) (RT)	$1.4V_S$	V_{MAT}	MSL	Cruise Ceiling	0.5	2.0
Descent (D)	$1.4V_S$	V_{MAT}	MSL	Cruise Ceiling	0.5	2.0
Emergency Descent (ED)	$1.4V_S$	V_{max}	MSL	Cruise Ceiling	0.5	2.0
Emergency Decelera-tion(DE)	$1.4V_S$	V_{max}	MSL	Cruise Ceiling	0.5	2.0
Aerial Delivery (AD)	$1.3V_S$	200 kts	MSL	10,000 ft	0	2.0

TABLE I OPERATIONAL FLIGHT ENVELOPES (Cont'd)

Flight Phase Category: C

Flight Phase	Airspeed		Altitude		Load Factor	
	$V_{o_{min}}$ ($M_{o_{min}}$)	$V_{o_{max}}$ ($M_{o_{max}}$)	$h_{o_{min}}$	$h_{o_{max}}$	$n_{o_{min}}$	$n_{o_{max}}$
Takeoff (TO)	Minimum Normal Takeoff Speed	V_{max}	MSL	10,000 ft	0.5	2.0
Catapult Takeoff (CT)	Min. Catapult End Airspeed	$V_{o_{min}}$ + 10 kts	MSL		0.5	n_L
Approach (FA)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft	0.5	2.0
Wave-off/Go-around (WO)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft	0.5	2.0
Landing (L)	Minimum Normal Approach Speed	V_{max}	MSL	10,000 ft	0.5	2.0

* Appropriate to the operational mission

3.1.8 Service Flight Envelopes. For each Airplane Normal State the contractor shall establish, subject to the approval of the procuring activity, Service Flight Envelopes showing combinations of speed, altitude and normal acceleration derived from airplane limits as distinguished from mission requirements. For each applicable Flight Phase and Airplane Normal State, the boundaries of the Service Flight Envelopes can be coincident with or lie outside the corresponding Operational Flight Envelopes, but in no case shall they fall inside those Operational boundaries. The boundaries of the Service Flight Envelopes shall be based on considerations discussed in 3.1.8.1, 3.1.8.2, 3.1.8.3 and 3.1.8.4.

3.1.8.1 Maximum service speed. The maximum service speed, V_{max} or M_{max}, for each altitude is the lowest of:

a. The maximum permissible speed

b. A speed which is a safe margin below the speed at which intolerable buffet or structural vibration is encountered

c. The maximum airspeed at MAT, for each altitude, for dives (at all angles) from V_{MAT} at all altitudes, from which recovery can be made at 1,000 ft above MSL or higher without penetrating a safe margin from loss of control, other dangerous behavior or intolerable buffet, and without exceeding structural limits.

3.1.8.2 Minimum service speed. The minimum service speed, V_{min} or M_{min}, for each altitude is the highest of:

a. $1.1V_S$

b. V_S + 10 knots equivalent airspeed

c. The speed below which full airplane-nose-up pitch control power and trim are insufficient to maintain steady, straight flight.

d. The lowest speed at which level flight can be maintained with MRT and, for Category C Flight Phases:

e. A speed limited by reduced visibility or an extreme pitch attitude that would result in the tail or aft fuselage contacting the ground.

3.1.8.3 Maximum service altitude. The maximum service altitude, h_{max} for a given speed is the maximum altitude at which a rate of climb of 100 feet per minute can be maintained in unaccelerated flight with MAT.

3.1.8.4 Service load factors. Maximum and minimum service load factors, n(+) (n(-)), shall be established as a function of speed for several significant altitudes. The maximum (minimum) service load factor, when trimmed for 1g flight at a particular speed and altitude, is the lowest (highest) algebraically of:

a. The positive (negative) structural limit load factor

b. The steady load factor corresponding to the minimum allowable value of lift coefficient for stall warning (3.4.1.1.1.1)

c. The steady load factor at which the pitch control is in the full airplane-nose-up (nose-down) position

d. A safe margin below (above) the load factor at which intolerable buffet or structural vibration is encountered.

3.1.9 Permissible Flight Envelopes. The contractor shall define Permissible Flight Envelopes which encompass all regions in which operation of the airplane is both allowable and possible, consistent with 3.1.10.3.3. These Envelopes define boundaries in terms of speed, altitude and load factor.

3.1.10 Application of Levels. Levels of flying qualities as indicated in 3.5 are employed in this specification in realization of the possibility that the airplane may be required to operate under abnormal conditions. Such abnormalities that may occur as a result of either flight outside the Operational Flight Envelope, failure of airplane components, or both, are permitted to comply with a degraded Level of flying qualities as specified in 3.1.10.1 through 3.1.10.3.3 (see also 4.1.1).

3.1.10.1 Requirements for Airplane Normal States. The minimum required flying qualities for Airplane Normal States (3.1.6.1) are as specified in Table II.

TABLE II LEVELS FOR AIRPLANE NORMAL STATES

Within Operational Flight Envelope	Within Service Flight Envelope
Level 1	Level 2

3.1.10.2 Requirements for Airplane Failure States. When Airplane Failure States exist (3.1.6.2), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in 3.1.10.1 is sufficiently small. At intervals established by the procuring activity, the contractor shall determine, based on the most accurate available data, the probability of occurrence of each Airplane Failure State per flight and the effect of that Failure State on the flying qualities within the Operational and Service Flight Envelopes. These determinations shall be based on MIL-STD-756 except that:

a. All airplane components and systems are assumed to be operating for a time period, per flight, equal to the longest operational mission time to be considered by the contractor in designing the airplane, and

b. Each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is the most critical (in the flying qualities sense). From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight, that one or more flying qualities are degraded to Level 2 because of one or more failures. The contractor shall also determine the probability that one or more flying qualities are degraded to Level 3. These probabilities shall be less than the values specified in Table III.

In no case shall a Failure State (except an approved Special Failure State) degrade any flying quality parameter outside the Level 3 limit.

TABLE III LEVELS FOR AIRPLANE FAILURE STATES

Probability of Encountering	Within Operational Flight Envelope	Within Service Flight Envelope
Level 2 after failure	$< 10^{-2}$ per flight	
Level 3 after failure	$< 10^{-4}$ per flight	$< 10^{-3}$ per flight

3.1.10.2.1 Requirements for Specific Failures. The requirements on the effects of specific types of failures, e.g. propulsion or flight control system, shall be met on the basis that the specific type of failure has occurred, regardless of its probability of occurrence.

3.1.10.3 Exceptions

3.1.10.3.1 Ground operation and terminal flight phases. Some requirements pertaining to takeoff, landing and taxiing involve operations outside the Operational, Service and Permissible Flight Envelopes, such as at V_G or on the ground. When requirements are stated at conditions such as these, the Levels shall be applied as if the conditions were in the Operational Flight Envelope.

3.1.10.3.2 When Levels are not specified. Within the Operational and Service Flight Envelopes, all requirements that are not identified with specific Levels shall be met under all conditions of component and system failure except approved Airplane Special Failure States (3.1.6.2.1).

3.1.10.3.3 Flight outside the Service Flight Envelope. From all points in the Permissible Flight Envelopes, it shall be possible readily and safely to return to the Service Flight Envelope without exceptional pilot skill or technique, regardless of component failure or system failures. The requirements on flight at high angle of attack, dive characteristics, dive recovery devices and dangerous flight conditions shall also apply.

3.1.11 Interpretation of subjective requirements. In several instances throughout the specification subjective terms, such as objectionable flight characteristics, realistic time delay, normal pilot technique and excessive loss of altitude or buildup of speed, have been employed to permit latitude where absolute quantitative criteria might be unduly restrictive. Final determination of compliance with requirements so worded will be made by the procuring activity (1.9).

3.1.12 Interpretation of quantitative requirements. The numerical requirements of this specification generally are stated in terms of a linear description of the airplane. Certain factors, for example flight control system nonlinearities and higher-order characteristic or aerodynamic nonlinearities, can cause the aircraft response to differ significantly from that of the linear model. The contractor shall define equivalent classical systems which have responses most closely matching those of the actual aircraft. Then those numerical requirements of section 3 which are stated in terms of linear system parameters (such as frequency, damping ratio and modal phase angles) apply to the parameters of that equivalent system rather than to any particular modes of the actual higher-order system. The procuring activity shall be the judge of the adequacy of the response match between equivalent and actual aircraft.

3.1 Longitudinal flying qualities. For Levels 1 and 2 there shall be no tendency for airspeed to diverge aperiodically when the airplane is disturbed from trim with the cockpit controls fixed and with them free. This requirement will be considered satisfied if the variations of pitch control force and pitch control position with airspeed are smooth and the local gradients stable, with:

a. Trimmer and throttle controls not moved from the trim setting by the crew, and

b. 1g acceleration normal to the flight path, and

c. constant altitude

over a range about the trim speed of +/- 15 percent or +/- 90 knots equivalent airspeed, whichever is less (except where limited by the boundaries of the Service Flight Envelopes). Alternatively, this requirement will be considered satisfied if stability with respect to speed is provided through the flight control system, even though the resulting pitch control force and deflection gradients may be zero. For Level 3 the requirements may be relaxed, subject to approval by the procuring activity of the maximum instability to be allowed for the particular case. In no event shall its time to double amplitude be less than 6 seconds. In the presence of one or more other Level 3 flying qualities, no static longitudinal instability will be permitted unless the flight safety of that combination of characteristics has been demonstrated to the satisfaction of the procuring activity. Stable gradients mean that the pitch controller deflection and force increments required to maintain straight, steady flight at a different speed are in the same sense as those required to initiate the speed change, that is, airplane-nose-down control to fly at a faster speed, airplane-nose-up control to fly at a slower speed. The term gradient does not include that portion of the control force or control position versus airspeed curve within the breakout force range.

3.2.1.1.1 Relaxation in transonic flight. The requirements of 3.2.1.1 may be relaxed in the transonic speed range provided any divergent airplane motions or reversals in slope of pitch control force and position with speed are gradual and not objectionable to the pilot. In no case, however, shall the requirements of 3.2.1.1 be relaxed more than the following:

a. Levels 1 and 2 - For center-stick controllers, no local force gradient shall be more unstable than 3 lbs per 0.01 M nor shall the force change exceed 10 lbs in the unstable direction. The corresponding limits for wheel controllers are 5 lbs per 0.01 M and 15 lbs, respectively.

b. Level 3- For center-stick controllers, no local force gradient shall be more unstable than 6 lbs per 0.01 M nor shall the force ever exceed 20 lbs in the unstable direction. The corresponding limits for wheel controllers are 10 lbs per 0.01 M and 30 lbs respectively.

This relaxation does not apply to Level 1 for any Flight Phase which requires prolonged transonic operation.

3.2.1.1.2 Pitch control force variations during rapid speed changes. When the airplane is accelerated and decelerated rapidly through the operational speed range and through the transonic speed range by the most critical combination of changes in power, actuation of deceleration devices, steep turns and pullups, the magnitude and rate of the associated trim change shall not be so great as to cause difficulty in maintaining the desired load factor by normal pilot techniques.

3.2.1.2 Phugoid stability. The long-period airspeed oscillations which occur when the airplane seeks a stabilized airspeed following a disturbance shall meet the following requirements:

a. Level 1 ----- ζ_p at least 0.04

b. Level 2 ----- ζ_p at least 0

c. Level 3 ----- T_2 at least 55 seconds

These requirements apply with the pitch control free and also with it fixed. They need not be met transonically in cases where 3.2.1.1.1 permits relaxation of the static stability requirement.

3.2.1.3 Flight-path stability. Flight-path stability is defined in terms of flight-path-angle change where the airspeed is changed by the use of pitch control only (throttle setting not changed by the crew). For the landing approach Flight Phase, the curve of flight-path angle versus true airspeed shall have a local slope at $V_{o_{min}}$ which is negative or less positive than:

a. Level 1 ----- 0.06 degrees/knot

b. Level 2 ----- 0.15 degrees/knot

c. Level 3 ----- 0.14 degrees/knot

The thrust setting shall be that required for the normal approach glide path at $V_{o_{min}}$. The slope of the curve of flight-path angle versus airspeed at 5 knots slower than $V_{o_{min}}$ shall not be more than 0.05 degrees/knot more positive than the slope at $V_{o_{min}}$, as illustrated by:

3.2.2 Longitudinal maneuvering characteristics

3.2.2.1 Short-period response. The short-period response of angle-of-attack which occurs at approximately constant speed, and which may be produced by abrupt control inputs, shall meet the requirements of 3.2.2.1.1 and 3.2.2.1.2. These requirements apply, with the cockpit control free and with it fixed, for responses of any magnitude that might be experienced in service use. If oscillations are nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation. In addition to meeting the numerical requirements of 3.2.2.1.1 and 3.2.2.1.2, the contractor shall show that the airplane has suitable response characteristics in atmospheric disturbances (3.7 and 3.8).

3.2.2.1.1 Short-period frequency and acceleration sensitivity. The equivalent short-period undamped natural frequency, $\omega_{n_{sp}}$, shall be within the limits shown on Figures B1, B2 and B3. If suitable means of directly controlling normal force are provided, the lower bounds on $\omega_{n_{sp}}$ and n/α of Figure B3 may be relaxed if approved by the procuring activity.

3.2.2.1.2 Short-period damping. The equivalent short-period damping ratio, ζ_{sp} shall be within the limits of Table IV.

TABLE IV SHORT-PERIOD DAMPING RATIO LIMITS

	Category A and C Flight Phases		Category B Flight Phases	
Level	Minimum	Maximum	Minimum	Maximum
1	0.35	1.30	0.30	2.00
2	0.25	2.00	0.20	2.00
3	0.15*	-	0.15*	-

*May be reduced at altitudes above 20,000 ft if approved by the procuring activity.

3.2.2.1.3 Residual oscillations. Any sustained residual oscillations in calm air shall not interfere with the pilot's ability to perform the tasks required in service use of the airplane. For Levels 1 and 2, oscillations in normal acceleration at the pilot's station greater than +/-0.05g will be considered excessive for any Flight Phase, as will pitch attitude oscillations greater than +/-3 mils for Category A Flight Phases requiring precise control of pitch attitude. These requirements shall apply with the pitch control fixed and with it free.

3.2.2.2 Control feel and stability in maneuvering flight at constant speed. In steady turning flight and in pullups at constant speed, there shall be no tendency for the airplane pitch attitude or angle of attack to diverge aperiodically with controls fixed or with controls free. For the above conditions, the incremental force and control deflection required to maintain a change in normal load factor and pitch rate shall be in the same sense (aft-more positive, forward-more negative) as those required to initiate the change. These requirements apply for all local gradients throughout the range of service load factors defined in 3.1.8.4.

3.2.2.2.1 Control forces in maneuvering flight. At constant speed in steady turning flight, pullups and pushovers, the variation in pitch controller force with steady-state normal acceleration shall have no objectionable non-linearities within the following load factor ranges:

Class	Minimum	Maximum
I, II and III	0.5	$0.5(n_D(+) + 1)$ or 3
IV	0	whichever is less

Figure B1 Short-Period Frequency Requirements-Category A Flight Phases

Figure B2 Short-Period Frequency Requirements-Category B Flight Phases

Figure B3 Short-Period Frequency Requirements-Category C Flight Phases

Outside this range, a departure from linearity resulting in a local gradient which differs from the average gradient for the maneuver by more than 50 percent is considered excessive, except that larger increases in force gradient are permissible at load factors greater than $0.65n_L$. All local force gradients shall be within the limits of Table V. In addition, F_e/n_z should be near the Level 1 upper boundaries of Table V for combinations of high frequency and low damping. The term gradient does not include that portion of the force versus n_z curve within the breakout force.

Since the range of acceptable force gradients for side stick controllers varies with the control deflection gradient and the task to be performed, the contractor shall show that the control force gradients will provide suitable flying qualities.

3.2.2.2.2 Control motions in maneuvering flight. For all types of pitch controllers, the control motions in maneuvering flight shall not be so large or so small as to be objectionable. For Category A Flight Phases, the average gradient of pitch-control force per unit of pitch-control deflection at constant speed shall not be less than 3.0 pounds per inch for wheel and center-stick controllers or 2.0 pounds per degree for side-stick controllers for Levels 1 and 2.

3.2.2.3 Longitudinal pilot-induced oscillations. There shall be no tendency for pilot-induced oscillations, that is, sustained or uncontrollable oscillations resulting from the efforts of the pilot to control the airplane. The pitch attitude response dynamics of the airplane plus control system shall not change abruptly with the motion amplitudes of pitch, pitch rate or normal acceleration unless it can be shown that this will not result in a pilot-induced oscillation. The requirements of 3.2.2.3.1 and 3.2.2.3.2 shall be met for all expected airplane motion amplitudes and frequencies, starting at any service load factor.

3.2.2.3.1 Dynamic control forces in maneuvering flight. The frequency response of normal acceleration at the pilot to pitch control force shall be such that the inverse amplitude is greater than the following for all frequencies greater than 1.0 rad/sec. Units are pounds per g.

Controller Type	Level 1	Level 2	Level 3
One-handed Controllers	$14/(n_L - 1)$	$12/(n_L - 1)$	$8/(n_L - 1)$
Two-handed Controllers	$30/(n_L - 1)$	$23/(n_L - 1)$	$17/(n_L - 1)$

3.2.2.3.2 Control feel. The deflection of the pilot's control must not lead the control force throughout the frequency range of pilot control inputs. In addition, the peak control forces developed during abrupt maneuvers shall not be objectionably light, and the buildup of control force during the maneuver entry shall lead the buildup of normal acceleration.

TABLE V PITCH MANEUVERING FORCE GRADIENT LIMITS

For Center Stick Controllers:

Level	Maximum Gradient, $(F_s/n)_{max}$, lbs/g	Minimum Gradient, $(F_s/n)_{min}$, lbs/g
1	240/(n/s) but not more than 28.0 nor less than 56/(n_L - 1)*	The higher of 21/(n_L - 1) and 3.0
2	360/(n/s) but not more than 42.5 nor less than 85/(n_L - 1)	The higher of 18/(n_L - 1) and 3.0
3	36.0	The higher of 12/(n_L - 1) and 3.0

*For n_L < 3, $(F_s/n)_{max}$ is 28.0 for Level 1, and 42.5 for Level 2.

For Wheel Controllers:

Level	Maximum Gradient, $(F_s/n)_{max}$, lbs/g	Minimum Gradient, $(F_s/n)_{min}$, lbs/g
1	300/(n/s) but not more than 120.0 nor less than 120/(n_L - 1)	The higher of 33/(n_L - 1) and 6.0
2	375/(n/s) but not more than 102.0 nor less than 102/(n_L - 1)	The higher of 30/(n_L - 1) and 6.0
3	240.0	5.0

3.1.3 Longitudinal control

3.1.3.1 Longitudinal control in unaccelerated flight. In erect unaccelerated flight at all service altitudes, the attainment of speed between V_S and V_{max} shall not be limited by the effectiveness of the longitudinal control or controls.

3.1.3.2 Longitudinal control in maneuvering flight. Within the Operational Flight Envelope, it shall be possible to develop, by use of the pitch control alone, the following range of load factors:

Levels 1 and 2 ----- $n_o(-)$ to $n_o(+)$

Level 3 ----- n = 0.5g to the lower of:

a) $n_o(+)$

b) n = 2.0 for $n_o(+) \leq 3g$

= $0.5(n_o(+) + 1)$ for $n_o(+) > 3g$

This maneuvering capability is required at the 1g trim speed and, with trim and throttle settings not changed by the crew, over a range about the trim speed the lesser of +/-15 percent or +/-50 knots equivalent airspeed (except where limited by the boundaries of the Operational Flight Envelope). Within the Service and Permissible Flight Envelopes, the dive-recovery requirements of 3.3.3.3 and 3.3.3.6, respectively, shall be met.

3.2.3.3 Longitudinal control in takeoff. The effectiveness of the pitch control shall not restrict the takeoff performance of the airplane and shall be sufficient to prevent over-rotation to undesirable attitudes during takeoffs. Satisfactory takeoffs shall not be dependent upon use of the trimmer control during takeoff or on complicated control manipulation by the pilot. For nose-wheel airplanes it shall be possible to obtain, at $0.9V_{min}$, the pitch attitude which will result in takeoff at V_{min}. For tail-wheel airplanes, it shall be possible to maintain any pitch attitude up to that for a level thrust-line at $0.5V_S$ for Class I airplanes and at V_S for Class II, III and IV airplanes. These requirements shall be met on hard-surfaced runways. In the event that an airplane has a mission requirement for operation from unprepared fields, these requirements shall be met on such fields.

3.2.3.3.1 Longitudinal control in catapult takeoff. On airplanes designed for catapult takeoff, the effectiveness of the pitch control shall be sufficient to prevent the airplane from pitching up or down to undesirable attitudes in catapult takeoffs at speeds ranging from the minimum safe launching speed to launching speed 30 knots higher than the minimum. Satisfactory catapult takeoffs shall not depend upon complicated control manipulation by the pilot.

3.2.3.3.2 Longitudinal control force and travel in takeoff. With the trim setting optional but fixed, the pitch-control forces required during all types of takeoffs for which the airplane is designed, including short-field takeoffs and assisted takeoffs such as catapult or rocket-augmented, shall be within the following limits:

Nose-wheel and bicycle-gear airplanes

Classes I, IV-C -------- 20 pounds pull to 10 pounds push

Classes II-C, IV-L ----- 30 pounds pull to 10 pounds push

Classes II-L, III ------ 50 pounds pull to 20 pounds push

Tail-wheel airplanes

Classes I, II-C, IV ---- 20 pounds push to 10 pounds pull

Classes II-L, III ------ 15 pounds push to 15 pounds pull

The pitch-control travel during takeoffs shall not exceed 75 percent of the total travel, stop-to-stop. Here the term takeoff includes the ground run, rotation and lift-off, the ensuing acceleration to V_{max}(TO), and the transient caused by assist cessation. Takeoff power shall be maintained until V_{max}(TO) is reached, with the landing gear and high-lift devices retracted in the normal manner at speeds from $V_{o_{min}}$ (TO) to V_{max}(TO).

3.2.3.4 Longitudinal control in landing. The pitch control shall be sufficiently effective in the landing Flight Phase in close proximity to the ground, that in calm air:

a. The geometry-limited touchdown attitude can be maintained in level flight or

b. The lower of V_S(L) or the guaranteed landing speed can be obtained.

This requirement shall be met with the airplane trimmed for the approach Flight Phase at the recommended approach speed. The requirements of 3.2.3.4 and 3.2.3.4.1 define Levels 1 and 2, and the requirements of 3.4.1e define Level 3.

3.2.3.4.1 Longitudinal control forces in landing. The pitch-control forces required to meet the requirements of 3.2.3.4 shall be pull forces and shall not exceed:

Classes I, II-C ----- 35 pounds

Classes II-L -------- 50 pounds

3.2.3.5 Longitudinal control forces in dives - Service Flight Envelope. With the airplane trimmed for level flight at speeds throughout the Service Flight Envelope, the control forces in dives to all attainable airspeeds within the Service Flight Envelope shall not exceed 50 pounds push or 10 pounds pull for center-stick controllers, nor 75 pounds push or 15 pounds pull for wheel controllers. In similar dives, but with trim optional following the dive entry, it shall be possible with normal piloting techniques to maintain the forces within the limits of 10 pounds push or pull for center-stick controllers, and 20 pounds push or pull for wheel controllers. In event that operation of the trim system requires removal of one hand from a wheel control the force limits shall be as for a center-stick. The forces required for recovery from these dives shall be in accordance with the gradients specified in 3.2.3.2.1 although speed may vary during the pullout.

3.2.3.6 Longitudinal control forces in dives - Permissible Flight Envelope. With the airplane trimmed for level flight at V_{MAX} but with trim optional in the dive, it shall be possible to maintain the pitch control force within the limits of 50 pounds push or 35 pounds pull in dives to all attainable speeds within the Permissible Flight Envelope. The force required for recovery from these dives shall not exceed 120 pounds. Trim and deceleration devices, etc., may be used to assist in recovery if no unusual pilot technique is required.

3.2.3.7 Longitudinal control in sideslips. With the airplane trimmed for straight, level flight with zero sideslip, the pitch-control force required to maintain constant speed in steady sideslips with up to 50 pounds of pedal force in either direction shall not exceed the pitch-control force that would result in a 1g change in normal acceleration. In no case, however, shall the pitch-control force exceed:

Center-stick controllers ----- 10 lbs pull to 2 lbs push

Wheel controllers ---------- 15 lbs pull to 10 lbs push

If a variation of pitch-control force with sideslip does exist, it is preferred that increasing pull force accompany increasing sideslip, and that the magnitude and direction of the force change be similar for right and left sideslips. These requirements define Levels 1 and 2. For Level 3 there shall be no uncontrollable pitching motions associated with the sideslips discussed above.

3.3 Lateral-directional flying qualities

3.3.1 Lateral-directional mode characteristics

3.3.1.1 Lateral-directional oscillations (Dutch roll). The frequency, ω_{n_d}, and damping ratio, ζ_d, of the lateral-directional oscillations following a yaw disturbance input shall exceed the minimum values in Table VI. The requirements shall be met in trimmed and in maneuvering flight with the cockpit controls fixed and with them free, in oscillations of any magnitude that might be expected in operational use. If the oscillation is nonlinear with amplitude, the requirements shall apply to each cycle of the oscillation. In calm air residual oscillations may be tolerated only if the amplitude is sufficiently small that the motions are not objectionable and do not impair mission performance. For Category A Flight Phases, angular deviations shall be less than +/- 3 mils.

3.3.1.2 Roll mode. The roll-mode time constant, T_r, shall be no greater than the appropriate value in Table VII.

TABLE VI MINIMUM DUTCH ROLL FREQUENCY AND DAMPING

Level	Flight Phase Category	Class	Min. ζ_d*	Min. $\zeta_d\omega_{n_d}$* rad/sec.	Min. ω_{n_d} rad/sec.
1	A (CO and GA)	IV	0.4	-	1.0
	A	I, IV	0.19	0.35	1.0
		II, III	0.19	0.35	0.4**
	B	All	0.08	0.15	0.4**
	C	I, II-C, IV	0.08	0.15	1.0
		II-L, III	0.08	0.10	0.4**
2	All	All	0.02	0.05	0.4**
3	All	All	0	-	0.4**

* The governing damping requirement is that yielding the larger value of ζ_d, except that a ζ_d of 0.7 is the maximum required for Class III.

** Class III airplanes may be excepted from the minimum ω_{n_d} requirement, subject to approval by the procuring activity, if the requirements of 3.3.2 through 3.3.2.4.1, 3.3.5 and 3.3.5.4 are met.

When $(\omega_{n_d})^2|\phi/\beta|_d$ is greater than 20 $(rad/sec)^2$, the minimum $\zeta_d\omega_{n_d}$ shall be increased above the $\zeta_d\omega_{n_d}$ minimum listed in Table VI by:

Level 1 - $\Delta\zeta_d\omega_{n_d} = 0.014[(\omega_{n_d})^2|\phi/\beta|_d - 20]$

Level 2 - $\Delta\zeta_d\omega_{n_d} = 0.009[(\omega_{n_d})^2|\phi/\beta|_d - 20]$

Level 3 - $\Delta\zeta_d\omega_{n_d} = 0.005[(\omega_{n_d})^2|\phi/\beta|_d - 20]$

with ω_{n_d} in rad/sec.

TABLE VII MAXIMUM ROLL-MODE TIME CONSTANT, τ_r, SECONDS

Flight Phase Category	Class	Level 1	Level 2	Level 3
A	I, IV	1.0	1.4	
	II, III	1.4	3.0	
B	All	1.4	3.0	10.0
C	I, II-C, IV	1.0	1.4	
	II-L, III	1.4	3.0	

3.3.1.3 Spiral stability. The combined effects of spiral stability, flight-control-system characteristics and rolling moment change with speed shall be such that following a disturbance in bank of up to 20 degrees, the time for the bank angle to double shall be greater than the values in Table VIII. This requirement shall be met with the airplane trimmed for wings-level, zero-yaw-rate flight with the cockpit controls free.

TABLE VIII SPIRAL STABILITY - MINIMUM TIME TO DOUBLE

	AMPLITUDE, $T_{1/2}$		
Flight Phase Category	Level 1	Level 2	Level 3
A and C	12 sec.	8 sec.	4 sec.
B	20 sec.	8 sec.	4 sec.

3.3.1.4 Coupled roll-spiral oscillation. For flight phases which involve more than gentle maneuvering, such as CO and GA, the airplane characteristics shall not exhibit a coupled roll-spiral mode in response to the pilot roll control commands. A coupled roll-spiral mode will be permitted for Category B and C flight phases provided the product of frequency and damping ratio exceeds the following requirements:

Level	$\zeta_{cs}\omega_{n_{cs}}$, rad/sec
1	0.3
2	0.3
3	0.15

3.3.2 Lateral-directional dynamic response characteristics. Lateral-directional dynamic response characteristics are stated in terms of response to atmospheric disturbances and in terms of allowable roll rate and bank angle oscillations, sideslip excursions, roll control forces and yaw control forces that occur during specified rolling and turning maneuvers to the right and to the left. The requirements of 3.3.2.2, 3.3.2.3 and 3.3.2.4 apply for roll commands of all magnitudes needed to meet the roll performance requirements of 3.3.4 and 3.3.4.1.

3.3.2.1 Lateral-directional response to atmospheric disturbances. The combined effect of ω_{n_d}, ζ_d, T_r, $|\phi/\beta|_d$, $\angle(p/\beta)$, gust sensitivity, and flight-control-system nonlinearities on response and controllability characteristics in atmospheric disturbances shall be considered (see 3.8.3). In particular, the roll acceleration, rate and displacement responses to side gusts shall be investigated for airplanes with large rolling moment due to sideslip (i.e. large dihedral effect).

3.3.2.2 Roll rate oscillations. Following a yaw-control-free step roll control command, the roll rate at the first minimum following the first peak shall be of the same sign and not less than the following percentage of the roll rate at the first peak:

Level	Flight Phase Category	Percent
1	A and C	60
	B	25
2	A and C	25
	B	0

For all Levels, the change in bank angle shall always be in the direction of the roll control command. The roll command shall be held fixed until the bank angle has changed at least 90 degrees.

3.3.2.2.1 Additional roll rate requirement for small inputs. The value of the parameter p_{osc}/p_{av} following a yaw-control-free step roll command shall be within the limits shown on Figure B4 for Levels 1 and 2. This requirement applies for step roll-control commands up to the magnitude which causes a 60-degree bank angle change in $1.7T_d$ seconds.

3.3.2.3 Bank angle oscillations. The value of the parameter ϕ_{osc}/ϕ_{av} following a yaw-control-free impulse roll control command shall be within the limits as shown on Figure B5 for Levels 1 and 2. The impulse shall be as abrupt as practical within the strength limits of the pilot and the rate limits of the roll control system.

Figure B4 Roll Rate Oscillation Limitations

Figure B5 Bank Angle Oscillation Limittions

3.3.3.4 Sideslip excursions. Following a yaw-control-free step roll control command, the ratio of the sideslip increment, $\Delta\beta$ to the parameter k (4.2.6) shall be less than the values specified herein. The roll command shall be held fixed until the bank angle has changed at least 90 degrees.

Level	Flight Phase Category	Adverse Sideslip (Right roll command causes RIGHT sideslip	Proverse Sideslip (Right roll command causes LEFT sideslip
1	A	6 degrees	2 degrees
	B and C	10 degrees	3 degrees
2	All	15 degrees	4 degrees

3.3.3.4.1 Additional sideslip requirement for small inputs. The amount of sideslip following a yaw-control-free step roll control command shall be within the limits shown on Figure B6 for Levels 1 and 2. This requirement shall apply for step roll control commands up to the magnitude which causes a 60-degree bank angle change within T_d or 2 seconds, whichever is longer.

Figure B6 Sideslip Excursion Limitations

3.3.3.5 Control of sideslip in rolls. In the rolling maneuvers described in 3.3.4, but with coordination allowed for all Classes, directional-control effectiveness shall be adequate to maintain zero sideslip with pedal force not greater than 50 pounds for Class IV airplanes in Flight Phase Category A, Level 1, and 100 pounds for all other combinations of Class, Flight Phase Category and Level.

3.3.3.6 Turn coordination. It shall be possible to maintain steady coordinated turns in either direction, using 60 degrees of bank for Class IV airplanes, 45 degrees of bank for Class I and II airplanes, and 30 degrees of bank for Class III airplanes, with a pedal force not exceeding 40 pounds. It shall be possible to perform steady turns at the same bank angles with yaw-controls-free, with a roll-stick force not exceeding 5 pounds or a roll-wheel force not exceeding 10 pounds. These requirements constitute Levels 1 and 2, with the airplane trimmed for wings-level straight flight.

3.3.3 Pilot-induced oscillations. There shall be no tendency for sustained or uncontrollable lateral-directional oscillations resulting from efforts of the pilot to control the airplane.

3.3.4 Roll control effectiveness. Roll performance in terms of a bank angle change in a given time, ϕ_t, is specified in Table IXa for Class I and Class II airplanes, in 3.3.4.1 for Class IV airplanes, and in 3.3.4.2 for Class III airplanes. For rolls from banked flight, the initial condition shall be coordinated, that is, zero lateral acceleration. The requirements apply to roll commands to the right and to the left, initiated both from steady bank angles and from wings-level flight except as otherwise stated. Inputs shall be abrupt, with the time measured from the initiation of control force application. The pitch control shall be fixed throughout the maneuver. Yaw control pedals shall remain free for Class IV airplanes for Level 1, and for all carrier-based airplanes in Category C Flight Phases for Levels 1 and 2, but otherwise, yaw control pedals may be used to reduce sideslip that retards roll rate (not to produce sideslip which augments roll rate) if such control inputs are simple, easily coordinated with roll control inputs and consistent with piloting techniques for the airplane class and mission. For Flight Phase TO, the time required to bank may be increased proportional to the ratio of the rolling moment of inertia at takeoff to the largest rolling moment of inertia at landing, for weights up to the maximum authorized landing weight.

TABLE IXa ROLL PERFORMANCE FOR CLASS I AND II AIRPLANES

Class	Level	Category A		Category B		Category C	
		Time to Achieve the Following Bank Angle Change (Seconds)					
		60°	45°	60°	45°	30°	25°
I	1	1.8		1.7		1.3	
I	2	1.7		2.5		1.8	
I	3	2.4		3.4		2.6	
II-L	1		1.4		1.9	1.8	
II-L	2		1.9		2.8	3.5	
II-L	3		2.8		3.8	3.4	
II-C	1		1.4		1.9		1.8
II-C	2		1.8		2.6		1.5
II-C	3		2.8		3.6		2.8

TABLE IXb ROLL PERFORMANCE FOR CLASS IV AIRPLANES

Level	Speed Range	Category A			Category B	Category C
		Time to Achieve the Following Bank Angle Change (Seconds)				
		30°	60°	90°	90°	30°
1	VL	1.1			1.8	1.1
	L	1.1			1.7	1.1
	M			1.3	1.7	1.1
	H		1.1		1.7	1.1
2	VL	1.6			2.8	1.3
	L	1.5			2.5	1.3
	M			1.7	2.5	1.3
	H		1.3		2.5	1.3
3	VL	2.4			3.7	2.0
	L	2.0			3.4	2.0
	M			2.6	3.4	2.0
	H		2.6		3.4	2.0

3.3.4.1 Roll performance for Class IV airplanes. Roll performance in terms of ϕ_t for Class IV airplanes is specified in Table IXb. Additional or alternate roll performance requirements are specified in 3.3.4.1.1 and 3.3.4.1.2; these requirements take precedence over those in Table IXb. Roll performance for Class IV airplanes is specified over the following ranges of airspeeds:

Speed Range		Equivalent Airspeed Range
VL	Level 1:	$V_{o_{min}} \leq V < V_{min} + 20$ kts
	Levels 2 and 3:	$V_{min} \leq V < V_{min} + 20$ kts
L	Level 1:	$V_{min} + 20$ kts[1] $\leq V < 1.4V_{min}$
	Levels 2 and 3:	$V_{min} + 20$ kts $\leq V < 1.4V_{min}$
M	Level 1:	$1.4V_{o_{min}} \leq V < 0.7V_{max}$[2]
	Levels 2 and 3:	$1.4V_{min} \leq V < 0.7V_{max}$
H	Level 1:	$0.7V_{max}$[2] $< V \leq V_{o_{max}}$
	Levels 2 and 3:	$0.7V_{max} \leq V \leq V_{max}$

(1) or $V_{o_{min}}$ whichever is greater

(2) or $V_{o_{max}}$ whichever is less

3.3.4.1.1 Roll performance in Flight Phase CO. Roll performance for Class IV airplanes in Flight Phase CO is specified in Table IXc in terms of ϕ_t for 360 degree rolls initiated at 1g, and in Table IXd for rolls initiated at load factors between $0.8n_o(-)$ and $0.8n_o(+)$.

3.3.4.1.2 Roll performance in Flight Phase GA. The roll performance requirements for Class IV airplanes in Flight Phase GA with large complements of external stores may be relaxed from those specified in Table IXb, subject to approval by the procuring activity. For any external loading specified in the contract, however, the roll performance shall not be less than that in Table IXe where the roll performance is specified in terms of ϕ_t for rolls initiated at load factors between $0.8n_o(-)$ and $0.8n_o(+)$.

For any asymmetric loading specified in the contract, roll control power shall be sufficient to hold the wings level at the maximum load factors specified in 3.2.2.2 with adequate control margin (3.4.10).

3.3.4.1.3 Roll response. Stick-controlled Class IV airplanes in Category A Flight Phase shall have a roll response to control force not greater than 15 degrees in 1 second per pound for Level 1, and not greater than 25 degrees in 1 second per pound for Level 2. For Category C

TABLE IXc FLIGHT PHASE CO ROLL PERFORMANCE IN 360° ROLLS

Time to Achieve the Following Bank Angle Change (Seconds)

Level	Speed Range	30°	90°	180°	360°
1	VL	1.0			
	L		1.4	2.3	4.1
	M		1.0	1.6	2.8
	H		1.4	2.3	4.1
2	VL	1.6			
	L	1.3			
	M		1.3	2.0	3.4
	H		1.7	2.8	4.4
3	VL	1.5			
	L	2.0			
	M		1.7	3.0	
	H		2.1		

TABLE IXd FLIGHT PHASE CO ROLL PERFORMANCE

Time to Achieve the Following Bank Angle Change (Seconds)

Level	Speed Range	30°	50°	90°	180°
1	VL	1.0			
	L		1.1		
	M			1.2	1.2
	H		1.0		
2	VL	1.6			
	L	1.3			
	M			1.4	2.8
	H		1.4		
3	VL	2.5			
	L	2.0			
	M			1.7	3.4
	H		1.7		

Flight Phases, the roll sensitivity shall be not greater than 7.5 degrees in 1 second per pound for Level 1, and not greater than 12.5 degrees in 1 second per pound for Level 2. In case of conflict between the requirements of 3.3.4.1.3 and 3.3.4.3, the requirements of 3.3.4.1.3 shall govern. The term sensitivity does not include breakout force.

TABLE IXe FLIGHT PHASE GA ROLL PERFORMANCE

Time to Achieve the Following Bank Angle Change (Seconds)

Level	Speed Range	30°	50°	90°	180°
1	VL	1.5			
	L		1.7		
	M			1.7	3.0
	H		1.5		
2	VL	2.8			
	L	2.2			
	M			3.4	4.3
	H		3.4		
3	VL	4.4			
	L	3.8			
	M			3.4	4.0
	H		3.4		

TABLE IXf CLASS III ROLL PERFORMANCE

Time to Achieve 30° Bank Angle Change (Seconds)

Level	Speed Range	Category A	Category B	Category C
1	L	1.0	1.5	1.5
	M	1.5	1.0	1.5
	H	2.0	2.5	3.5
2	L	3.4	3.9	4.0
	M	3.0	3.5	4.0
	H	3.9	3.0	4.0
3	All	5.0	5.0	6.0

3.3.4.3 Roll performance for Class III airplanes. Roll performance in terms of ϕ_t for Class III airplanes is specified in Table IXf over the following ranges of airspeeds:

Speed Range		Equivalent Airspeed Range
L	Level 1:	$V_{o_{min}} \leq V < 1.6V_{min}$
	Levels 2 and 3:	$V_{min} \leq V < 1.6V_{min}$
M	Level 1:	$1.6V_{min}$ (1) $\leq V < 0.7V_{max}$ (2)
	Levels 2 and 3:	$1.6V_{min} \leq V < 0.7V_{max}$
H	Level 1:	$0.7V_{max}$ (1) $\leq V \leq V_{o_{max}}$
	Levels 2 and 3:	$0.7V_{max} \leq V \leq V_{max}$

(1) or $V_{o_{min}}$ whichever is greater

(2) or $V_{o_{max}}$ whichever is less

3.3.4.3 Roll control forces. The stick or wheel force required to obtain the rolling performance specified in 3.3.4, 3.3.4.1 and 3.3.4.2 shall be neither greater than the maximum in Table X nor less than the breakout force plus:

a. Level 1 ----- one-fourth the values in Table X

b. Level 2 ----- one-eighth the values in Table X

c. Level 3 ----- zero

TABLE X MAXIMUM ROLL CONTROL FORCES

Level	Class	Flight Phase Category	Maximum Stick Force (lbs)	Maximum Wheel Force (lbs)
1	I,II-C, IV	A, B	20	40
		C	30	20
	II-L, III	A, B	25	50
		C	35	25
2	I, II-C, IV	A, B	30	60
		C	20	20
	II-L, III	A, B	30	60
		C	50	30
3	All	All	35	70

3.3.4.4 Linearity of roll response. There shall be no objectionable nonlinearities in the variation of rolling response with roll control deflection or force. Sensitivity or sluggishness in response to small control deflections or force shall be avoided.

3.3.4.5 Wheel control throw. For airplanes with wheel controllers, the wheel throw necessary to meet the roll performance requirements specified in 3.3.4 and 3.3.4.2 shall not exceed 60 degrees in either direction. For completely mechanical systems, the requirement may be relaxed to 90 degrees.

3.4.5 Directional control characteristics. Directional stability and control characteristics shall enable the pilot to balance yawing moments and control yaw and sideslip. Sensitivity to yaw control pedal forces shall be sufficiently high that directional control and force requirements can be met and satisfactory coordination can be achieved without unduly high pedal forces, yet sufficiently low that occasional improperly coordinated control inputs will not seriously degrade the flying qualities.

3.3.5.1 Directional control with speed change. When initially trimmed directionally with symmetric power, the trim change of propeller-driven airplanes with speed shall be such that wings-level straight flight can be maintained over a speed range of +/- 30 percent of the trim speed or +/- 100 knots equivalent airspeed, whichever is less (except where limited by boundaries of the Service Flight Envelope) with yaw-control-pedal forces not greater than 100 pounds for Levels 1 and 2 and not greater than 180 pounds for Level 3, without retrimming. For other airplanes, yaw-control-pedal forces shall not exceed 40 pounds at the specified conditions for Levels 1 and 2 or 180 pounds for Level 3.

3.3.5.1.1 Directional control with asymmetric loading. When initially trimmed directionally with each asymmetric loading specified in the contract at any speed in the Operational Flight Envelope, it shall be possible to maintain a straight flight path throughout the Operational Flight Envelope with yaw-control-pedal forces not greater than 100 pounds for Levels 1 and 2 and not greater than 180 pounds for Level 3, without retrimming.

3.3.5.2 Directional control in wave-off (go-around). For propeller-driven Class IV, and all propeller-driven carrier-based airplanes the response to thrust, configuration and airspeed change shall be such that the pilot can maintain straight flight during wave-off (go-around) initiated at speeds down to V_s (PA) with yaw-control-pedal forces not exceeding 100 lbs when trimmed at $V_{o_{min}}$ (PA).

For other airplanes, yaw-control-pedal forces shall not exceed 40 pounds for the specified conditions. The preceding requirements apply for Levels 1 and 2. For all

airplanes the Level 3 requirement is to maintain straight flight in these conditions with yaw-control-pedal forces not exceeding 100 pounds. For all Levels, bank angles up to 5 degrees are permitted.

3.3.6 Lateral-directional characteristics in steady sideslips. The requirements of 3.3.6.1 through 3.3.6.3.1 and 3.3.7.1 are expressed in terms of characteristics in yaw-control-induced steady, zero-yaw-rate sideslips with the airplane trimmed for wings-level straight flight. Requirements of 3.3.6.1 through 3.3.6.3 apply at sideslip angles up to those produced or limited by:

a. Full yaw-control-pedal deflection, or

b. 150 pounds of yaw-control-pedal force, or

c. Maximum roll control or surface deflection,

except that for single-propeller-driven airplanes during wave-off (go-around), yaw-control-pedal deflection in the direction opposite to that required for wings-level straight flight need not be considered beyond the deflection for a 10-degree change in sideslip from the wings-level straight flight condition.

3.3.6.1 Yawing moments in steady sideslips. For sideslips specified in 3.3.6, right yaw-control-pedal deflection and force shall produce left sideslips and left yaw-control-pedal deflection and force shall produce right sideslips. For Levels 1 and 2 the following requirements shall apply. The variation of sideslip angle with yaw-control-pedal deflection shall be essentially linear for sideslip angles between +15 degrees and -15 degrees. For larger sideslip angles, an increase in yaw-control-pedal deflection shall always be required for an increase in sideslip. The variation of sideslip angle with yaw-control-pedal force shall be essentially linear for sideslip angles between +10 degrees and -10 degrees. Although a lightening of pedal force is acceptable for sideslip angles outside this range, the pedal force shall never reduce to zero.

3.3.6.2 Side forces in steady sideslips. For the sideslips of 3.3.6, an increase in right bank angle shall accompany an increase in right sideslip, and an increase in left bank angle shall accompany an increase in left sideslip.

3.3.6.3 Rolling moments in steady sideslips. For the sideslips of 3.3.6, left roll-control deflection and force shall accompany left sideslips, and right roll-control deflection and force shall accompany right sideslips. For Levels 1 and 2, the variation of roll-control deflection and force with sideslip angle shall be essentially linear.

3.3.6.3.1 Exception for wave-off (go-around). The requirement of 3.3.6.3 may, if necessary, be excepted for wave-off (go-around) if task performance is not impaired and no more than 50 percent of roll control power available to the pilot, and no more than 10 pounds of roll-control force, are required in a direction opposite to that specified in 3.3.6.3.

3.3.6.3.2 Positive effective dihedral limit. For Levels 1 and 2, positive effective dihedral (right roll control for right sideslip and left roll control for left sideslip) shall never be so great that more than 75 percent of roll control power available to the pilot, and no more than 10 pounds of roll-stick force or 20 pounds of roll-wheel force, are required for sideslip angles which might be experienced in service employment.

3.3.7 Lateral-directional control in crosswinds. It shall be possible to take off and land with normal pilot skill and technique in 90-degree crosswinds, from either side, of velocities up to those specified in Table XI. Roll-control force shall be within the limits specified in 3.3.4.2, and yaw-control-pedal forces shall not exceed 100 pounds for Level 1 or 180 pounds for Levels 2 and 3. This requirement can normally be met through compliance with 3.3.7.1 and 3.3.7.2.

TABLE XI CROSSWIND VELOCITY

Level	Class	Crosswind
1 and 2	I	20 knots
	II, III, and IV	30 knots
	Water-based airplanes	20 knots
3	All	One half the values for Levels 1 and 2

3.3.7.1 Final approach in crosswinds. For all airplanes except land-based airplanes equipped with crosswind landing gear, or otherwise constructed to land in a large crabbed attitude, yaw- and roll-control power shall be adequate to develop at least 10 degrees of sideslip (3.3.6) in the power approach with yaw-control-pedal forces not exceeding the values specified in 3.3.7. For Level 1, roll control shall not exceed either 10 pounds of force or 75 percent of control power available to the pilot. For Levels 2 and 3, roll-control force shall not exceed 20 pounds.

3.3.7.2 Takeoff run and landing rollout in crosswinds. Yaw and roll control power, in conjunction with other normal means of control, shall be adequate to maintain a straight flight path on the ground or other landing surface. This requirement applies in calm air and in crosswinds up to the values specified in Table XI with cockpit control forces not exceeding the values specified in 3.3.7.

3.3.7.2.1 Cold- and wet-weather operation. The requirements of 3.3.7.2 apply on wet runways for all airplanes, and on snow-packed and icy runways for airplanes intended to operate under such conditions. If compliance is not demonstrated under these adverse runway conditions, directional control shall be maintained by use of aerodynamic controls alone at all airspeeds above 10 knots for Class IV airplanes and above 30 knots for all others. For very slippery runways, the requirement need not apply for crosswind components at which the force tending to blow the airplane off the runway exceeds the opposing tire-runway frictional force with the tires supporting the entire weight of the airplane.

3.3.7.2.2 Carrier-based airplanes. All carrier-based airplanes shall be capable of maintaining a straight path on the ground without the use of wheel brakes, at airspeeds of 30 knots and above, during takeoffs and landings in a 90-degree crosswind of at least 10 percent V_S(L). Cockpit control forces shall be as specified in 3.3.7.

3.3.7.3 Taxiing wind speed limits. It shall be possible to taxi at any angle to a 35-knot wind for Class I airplanes and to a 45-knot wind for Class II, III and IV airplanes.

3.3.8 Lateral-directional control in dives. Yaw and roll control power shall be adequate to maintain wings level and sideslip zero, without retrimming, throughout the dives and pullouts of 3.2.3.3 and 3.2.3.4. In the Service Flight Envelope, roll control forces shall not exceed 20 pounds for propeller-driven airplanes or 10 pounds for other airplanes. Yaw-control-pedal forces shall not exceed 120 pounds for propeller-driven airplanes or 10 lbs for other airplanes.

3.3.9 Lateral-directional control with asymmetric thrust. Asymmetric loss of thrust may be caused by many factors including engine failure, inlet unstart, propeller failure or propeller-drive failure. Following sudden asymmetric loss of thrust from any factor, the airplane shall be safely controllable in the crosswinds of Table XI from the unfavorable direction. The requirements of 3.3.9.1 through 3.3.9.4 apply for the appropriate Flight Phases when any single failure or malperformance of the propulsive system, including inlet or exhaust, causes loss of thrust on one or more engines or propellers, considering also the effect of the failure or malperformance on all subsystems or driven by the failed propulsive system.

3.3.9.1 Thrust loss during takeoff run. It shall be possible for the pilot to maintain control of an airplane on the takeoff surface following sudden loss of thrust from the most critical factor. Thereafter, it shall be possible to achieve and maintain a straight path on the takeoff surface without a deviation of more than 30 feet from the path originally intended, with yaw-control-pedal forces not exceeding 180 pounds. For the continued takeoff, the requirement shall be met when thrust is lost at speeds from the refusal speed (based on the shortest runway from which the airplane is designed to operate) to the maximum takeoff speed, with takeoff thrust maintained on the operative engine(s), using only controls not dependent upon friction against the takeoff surface or upon release of the pitch, roll, yaw or throttle controls. For the aborted takeoff, the requirement shall be met at all speeds below the maximum takeoff speed, however, additional controls such as nosewheel steering and differential braking may be used. Automatic devices which normally operate in the event of a thrust failure may be used in either case.

3.3.9.2 Thrust loss after takeoff. During takeoff it shall be possible without a change in selected configuration to achieve straight flight following sudden asymmetric loss of thrust from the most critical factor at speeds from V_{min} (TO) to V_{max} (TO), and thereafter to maintain straight flight throughout the climbout. The yaw-control-pedal force required to maintain straight flight with asymmetric thrust shall not exceed 180 lbs. Roll control shall not exceed either the force limits specified in 3.3.4.3 or 75 percent of available control power, with takeoff thrust maintained on the operative engine(s) and trim at normal setting for takeoff with symmetric thrust. Automatic devices which normally operate in the event of a thrust failure may be used, and the airplane may be banked up to 5 degrees away from the inoperative engine.

3.3.9.3 Transient effects. The airplane motions following sudden asymmetric loss of thrust shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay (3.4.3) of at least 1 second shall be incorporated.

3.3.9.4 Asymmetric thrust - yaw controls free. The static directional stability shall be such that at all speeds above $1.4V_{min}$, with asymmetric loss of thrust from the most critical factor while the other engine(s) develop normal rated thrust, the airplane with yaw-control-pedals free may be balanced directionally in steady straight flight. The trim settings shall be those required for wings-level straight flight prior to the engine failure. Roll-control forces shall not exceed the Level 3 upper limits specified in 3.3.4.3 for Levels 1 and 2 and shall not exceed the Level 3 upper limits for Level 3.

3.3.9.3 Two engines inoperative. At the one-engine-out speed for maximum range with any engine initially failed, it shall be possible upon failure of the most critical remaining engine to stop the transient motion and thereafter to maintain straight flight from that speed to the speed for maximum range with both engines failed. In addition, it shall be possible to effect a safe recovery at any service speed above $V_{o_{min}}$ (CL) following sudden simultaneous failure of the two most critical engines.

3.4 Miscellaneous flying qualities. Dangerous conditions may exist where the airplane should not be flown. When approaching these flight conditions, it shall be possible by clearly discernible means for the pilot to recognize the impending dangers and take corrective action. Final determination of the adequacy of all warning of impending dangerous flight conditions will be made by the procuring activity, considering functional effectiveness and reliability.

3.4.1.1 Warning and indication. Warning and indication of approach to a dangerous condition shall be clear and unambiguous. For example, a pilot must be able to distinguish readily among stall warning (which requires pitching down or increasing speed), Mach buffet (which may indicate a need to decrease speed), and normal airplane vibration (which indicates no need for pilot action).

3.4.1.2 Devices for indication, warning, prevention, recovery. It is intended that dangerous flight conditions be eliminated and the requirements of this specification met by appropriate aerodynamic design and mass distribution, rather than through incorporation of a special device or devices. Such devices may be used only if the procuring activity approves the need, the design criteria, possible Special Failure States (3.1.4.2.1) and the devices themselves. As a minimum, these devices shall perform their function whenever needed but shall not limit flight within the Operational Flight Envelope. Neither normal nor inadvertent operation of such devices shall create a hazard to the airplane. For Levels 1 and 2, nuisance operation shall not be possible. Functional failure of the devices shall be indicated to the pilot.

3.4.2 Flight at high angle of attack. The requirements of 3.4.2 through 3.4.2.2.2 concern stall warning, stalls, departure from controlled flight, post-stall gyrations, spins, recoveries and related characteristics. They apply at speeds and angles of attack which in general are outside the Service Flight Envelope. They are intended to assure safety and the absence of mission limitations due to high angle of attack characteristics.

3.4.2.1 Stalls. The stall is defined in terms of airspeed and angle of attack in 6.2.1 and 6.2.2 respectively. It usually is a phenomenon caused by airflow separation induced by high angle of attack, but it may instead be determined by some limit on usable angle of attack. The stall requirements apply for all Airplane Normal States in straight unaccelerated flight and in turns and pullups with attainable normal accelerations up to n_L. Specifically, the Airplane Normal States associated with the configurations, throttle settings and trim settings of 6.2.2 shall be investigated. Also, the requirements apply to Airplane Failure States that affect stall characteristics.

3.4.2.1.1 Stall approach. The stall approach shall be accompanied by an easily perceptible warning consisting of shaking of the cockpit controls, buffeting or shaking of the airplane, or a combination of both. The onset of this warning shall occur within the ranges specified in 3.4.2.1.1.1 and 3.4.2.1.1.2 but not within the Operational Flight Envelope. The increase in buffeting intensity with further increase in angle of attack shall be sufficiently marked to be noted by the pilot. The warning shall continue until the angle of attack is reduced to a value less than that for warning onset. At all angles of attack up to the stall, the cockpit controls shall remain effective in their normal sense, and small control inputs shall not result in departure from controlled flight. Prior to the stall, uncommanded oscillations shall not be objectionable to the pilot.

3.4.2.1.1.1 Warning speed for stalls at 1g normal to the flight path. Warning onset for stalls at 1g normal to the flight path shall occur between the following limits when the stall is approached gradually:

Flight Phase	Minimum Speed for Onset	Maximum Speed for Onset
Approach	Higher of $1.05V_S$ or V_S + 5 knots	Higher of $1.10V_S$ or V_S + 10 knots
All other	Higher of $1.05V_S$ or V_S + 5 knots	Higher of $1.15V_S$ or V_S + 15 knots

3.4.2.1.1.2 Warning range for accelerated stalls. Onset of stall warning shall occur outside the Operational Flight Envelope associated with the Airplane Normal State and within the following range or fraction of lift at stall at that airspeed, in that Airplane State, when the stall is approached gradually:

Flight Phase	Minimum Lift at Onset	Maximum Lift at Onset
Approach	$0.82C_{L_{stall}}$	$0.98C_{L_{stall}}$
All other	$0.75C_{L_{stall}}$	$0.90C_{L_{stall}}$

3.4.2.1.2 Stall characteristics. In the unaccelerated stalls of 3.4.2.1, the airplane shall not exhibit rolling, yawing or downward pitching at the stall which cannot be controlled to stay within 10 degrees for Classes

I, II and III or 30 degrees for Class IV airplanes. It is desired that no pitchup tendencies occur in unaccelerated or accelerated stalls. In unaccelerated stalls, mild nose-up pitch may be acceptable if no pitch control force reversal occurs and if no dangerous, unrecoverable flight conditions result. A mild nose-up tendency may be acceptable in accelerated stalls if the operational effectiveness of the airplane is not compromised and:

a. The airplane has adequate stall warning

b. Pitch control effectiveness is such that it is possible to stop the pitchup promptly and reduce the angle of attack and

c. At no point during the stall, stall approach or recovery does any portion of the airplane exceed structural limit loads.

The requirements apply for all stalls, including stalls entered abruptly.

3.4.2.1.3 Stall prevention and recovery. It shall be possible to prevent the stall by moderate use of the pitch control alone at the onset of the stall warning. It shall be possible to recover from a stall by simple use of the pitch, roll and yaw controls with cockpit control forces not to exceed those of 3.4.4.1, and to regain level flight without excessive loss of altitude or buildup of speed. Throttles shall remain fixed until speed has begun to increase and an angle of attack below the stall has been regained unless compliance would result in exceeding engine operating limitations. In the straight-flight stalls of 3.4.2.1, with the airplane trimmed at an airspeed not greater than $1.4V_S$, pitch control power

shall be sufficient to recover from any attainable angle of attack.

3.4.2.1.3.1 One-engine-out stalls. On multi-engine airplanes, it shall be possible to recover safely from stalls with the critical engine inoperative. This requirement applies with the remaining engines at up to:

Flight Phase	Thrust
TO	Takeoff
CL	Normal climb
PA	Normal approach
WO	Waveoff

3.4.2.2 Post-stall gyrations and spins. The post-stall gyration and spin requirements apply to all modes of motion that can be entered from upsets, decelerations, and extreme maneuvers appropriate to the Class and Flight Phase Category. Entries form inverted flight shall be included for Class I and IV airplanes. Entry angles of attack and sideslip up to maximum control capability and under dynamic flight conditions are to be included, except as limited by structural considerations. For all Classes and Flight Phase Categories, thrust settings up to and including MAT shall be included, with and without one critical engine inoperative at entry. The requirements hold for all Airplane Normal States and for all states of stability and control augmentation systems, except approved Special Failure States. Store release shall not be allowed during loss of control, spin or gyration, recovery, or subsequent dive pullout. Automatic disengagement of augmentation systems, however, is permissible if it is necessary and does not prevent meeting any other requirements. Re-engagement shall be possible in flight following recovery.

3.4.2.2.1 Departure from controlled flight. All Classes of airplanes shall be extremely resistant to departure from controlled flight, post-stall gyrations and spins. The airplane shall exhibit no uncommanded motion which cannot be arrested promptly by simple appropriate application of pilot control. In addition, the procuring activity may designate that certain training airplanes shall be capable of a developed spin and consistent recovery.

3.4.2.2.2 Recovery from post-stall gyrations and spins. For airplanes which, according to MIL-A-8861 must be structurally designed for spinning, the following requirements apply. The proper recovery technique(s) must be readily ascertainable by the pilot, and simple and easy to apply under the motions encountered. Whatever the motions, safe consistent recovery and pullout shall be possible without exceeding the control forces of 3.4.4.1 and without exceeding structural limitations. A single technique shall provide prompt recovery from all post-stall gyrations and incipient spins, without requiring the pilot to determine the direction of motion and without tendency to develop a spin. The same technique used to recover from post-stall gyrations and incipient spins, or at least a compatible one, is also desired for spin recovery. For all modes of spin that can occur, these recoveries shall be attainable within the number of turns, measured from the initiation of recovery action as follows:

Class	Flight Phase Category	Turns for Recovery
I	A, B	1.5
I	PA	1
Others	PA	1
Others	A, B	2

Avoidance of a spin reversal or an adverse mode change shall not depend upon precise pilot control timing or deflection. It is desired that all airplanes be readily recoverable from all attainable attitudes and motions. The post-stall characteristics of those airplanes not required to comply with requirements of this paragraph shall be determined by analysis and model test.

3.4.3 Cross-axis coupling in roll maneuvers. For Class I and IV airplanes in yaw-control-free, pitch-control-fixed, maximum performance rolls through 360 degrees, entered from straight flight or from turns, pushovers, or pullups ranging from 0g to 0.8n_L, the resulting yaw or pitch motions and sideslip or angle of attack changes shall neither exceed structural limits nor cause other dangerous flight conditions such as uncontrollable motions or roll autorotation.

During combat-type maneuvers involving rolls through angles up to 360 degrees and rolls which are checked at a given bank angle, the yawing and pitching shall not be so severe as to impair the tactical effectiveness of the maneuver. These requirements define Level 1 and 2 operation. For Class II and III airplanes, these requirements apply in rolls through 120 degrees and rolls which are checked at a given bank angle.

3.4.4 Control harmony. The pitch- and roll-control force and displacement sensitivities and breakout forces shall be compatible so that intentional inputs to one control axis will not cause inadvertent inputs to the other.

3.4.4.1 Control force coordination. The cockpit control forces required to perform maneuvers which are normal for the airplane should have magnitudes which are related to the pilot's capability to produce such forces in combination. The following control force levels are considered to be limiting values compatible with pilot's capability to apply simultaneous forces:

Control Type	Pitch	Roll	Yaw
Side-stick or Center-stick	50 lbs	25 lbs	
Wheel	75 lbs	40 lbs	
Pedal			175 lbs

3.4.5 Buffet. Within the boundaries of the Operational Flight Envelope, there shall be no objectionable buffet which might detract from the effectiveness of the airplane in executing its intended missions.

3.4.6 Release of stores. The intentional release of any stores shall not result in objectionable flight characteristics for Levels 1 and 2. However, the intentional release of stores shall never result in dangerous or intolerable flight characteristics. This requirement applies for all flight conditions and store loadings at which normal or emergency store release is permissible.

3.4.7 Effects of armament delivery and special equipment. Operation of moveable parts such as bomb bay doors, cargo doors, armament pods, refueling devices and rescue equipment, or firing of weapons, release of bombs, or delivery or pickup of cargo shall not cause buffet, trim changes, or other characteristics which impair the tactical effectiveness of the airplane under any pertinent flight condition. These requirements shall be met for Level 1 and for Level 2.

3.4.8 Transients following failures. The airplane motions following sudden system or component failures shall be such that dangerous conditions can be avoided by pilot corrective action. A realistic time delay between the failure and initiation of pilot corrective action shall be incorporated when determining compliance. This time delay should include an interval between the occurrence of the failure and the occurrence of a cue such as acceleration, rate, displacement, or sound that will definitely indicate to the pilot that a failure has occurred, plus an additional interval which represents the time required for the pilot to diagnose the situation and initiate corrective action.

3.4.9 Failures. No single failure of any component or system shall result in dangerous or intolerable flying qualities. Special Failure States (3.1.4.2.1) are excepted. The crew member concerned shall be provided with immediate and easily interpreted indications whenever failures occur that require or limit any flight crew action or decision.

3.4.10 Control margin. Control authority, rate and hinge moment capability shall be sufficient to assure safety throughout the combined range of all attainable angles of attack (both positive and negative) and sideslip. This requirement applies to the prevention of loss of control and to recovery from any situation for all maneuvering, including pertinent effects of factors such as regions of control-surface-fixed instability, inertial coupling, fuel slosh, the influence of symmetric and asymmetric stores (3.1.4), stall/post-stall/spin characteristics (3.4.2 through 3.4.2.2.3), atmospheric disturbances (3.8) and Airplane Failure States (3.1.10.1 and 3.1.10.2 with maneuvering flight appropriate to the Failure State to be included). Consideration shall be taken of the degrees of effectiveness and certainty of operation of limiters, e.g. control malfunction or mismanagement, and transients from failures in the propulsion, flight control and other relevant systems.

3.4.11 Direct force controls. Use of devices for direct normal-force control and direct side-force control shall not produce objectionable changes in attitude for any amount of control up to the maximum available. This requirement shall be met for Levels 1 and 2.

3.5 Characteristics of the primary flight control system

3.5.1 General characteristics. As used in this specification, the term primary flight control system includes the pitch, roll and yaw controls, stability augmentation systems, and all mechanisms and devices which they operate. The requirements of this section are concerned with those aspects of the primary flight control system which are directly related to the flying qualities. These requirements are in addition to the requirements of the applicable control system design specifications, e.g.: MIL-F-9490 or MIL-C-18244.

3.5.2 Mechanical characteristics. Some of the important mechanical characteristics of control systems (including servo valves and actuators) are: friction and preload, lost motion, flexibility, mass imbalance and inertia, nonlinear gearing, and rate limiting. Requirements for some of these characteristics are contained in 3.5.2.1 through 3.5.2.4. Meeting these separate requirements, however, will not necessarily ensure that the overall system will be adequate, the mechanical characteristics must be compatible with the nonmechanical portions of the control system as well as with the airframe dynamic characteristics.

3.5.2.1 Control centering and breakout forces. Longitudinal, lateral and directional controls should exhibit positive centering in flight at any normal trim setting. Although absolute centering is not required, the combined effects of centering, breakout force, stability and force gradient shall not produce objectionable flight characteristics, such as poor precision-tracking ability, or permit large departures from trim conditions with controls free. Breakout forces, including friction, preload, etc., shall be within the limits of Table XII. The values in Table XII refer to the cockpit control force required to start movement of the control surface in flight for Levels 1 and 2, the upper limits are doubled for Level 3.

Measurement of breakout forces on the ground will ordinarily suffice in lieu of actual flight measurement, provided that the qualitative agreement between ground measurement and flight observation can be established.

3.5.2.2 Cockpit control free play. The free play in each cockpit control, that is, any motion of the cockpit control which does not move the control surface in flight, shall not result in objectionable flight characteristics, particularly for small-amplitude control inputs.

3.5.2.3 Rate of control displacement. The ability of the airplane to perform the operational maneuvers required of it shall not be limited in the atmospheric disturbances specified in 3.7 by control surface deflection rates (3.6.3.1, 3.6.3.2 and 3.6.10). For powered or boosted controls, the effect of engine speed and the duty cycle

TABLE XII ALLOWABLE BREAKOUT FORCES, POUNDS

Control		Classes I, II-C, IV Min.	Max.	Classes II-L, III Min.	Max.
Pitch	Stick	0.5	3.0	0.5	5.0
	Wheel	0.5	4.0	0.5	7.0
Roll	Stick	0.5	2.0	0.5	4.0
	Wheel	0.5	3.0	0.5	6.0
Yaw	Pedal	1.0	3.0	1.0	14.0

of both primary and secondary control together with the pilot control technique shall be included when establishing compliance with this requirement.

3.5.2.4 Adjustable controls. When a cockpit control is adjustable for pilot physical dimensions and comfort, the control forces defined in 6.1 refer to the mean adjustment. A force referred to any other adjustment shall not differ by more than 10 percent from the force referred to the mean adjustment.

3.5.3 Dynamic characteristics. A linear or smoothly varying airplane response to cockpit-control deflection and to control force shall be provided for all amplitudes of control input. The response of the control surfaces in flight shall not lag the cockpit-control force inputs by more than the angles specified in Table XIII, for frequencies equal to or less than the frequencies specified in Table XIII.

TABLE XIII ALLOWABLE CONTROL SURFACE LAGS

Level	Allowable Lag, degrees: Flight Phase Category A and C	Flight Phase Category B
1	15	30
2	30	45
3	60	60

Control	Upper Frequency, rad/sec
Pitch	the larger of $\omega_{n_{sp}}$ and 3.0
Roll and Yaw	the largest of ω_{n_d}, $1/T_r$ and 2.0

In addition, the response of the airplane motion shall not exhibit a time delay longer than the values given in Table XIV for a pilot-initiated step control force input.

Further, the values of the equivalent time delay derived from equivalent system match of the aircraft response to cockpit controls shall not exceed the values given in Table XIV.

TABLE XIV ALLOWABLE AIRPLANE RESPONSE DELAY

Level	Allowable Delay, Seconds
1	0.10
2	0.20
3	0.25

3.5.3.4 Damping. All control system oscillations shall be well damped, unless they are of such an amplitude, frequency and phasing that they do not result in objectionable oscillations of the cockpit controls or the airframe during abrupt maneuvers and during flight in atmospheric disturbances.

3.5.4 Augmentation systems. Operation of stability augmentation and control augmentation systems and devices shall not introduce any objectionable flight or ground handling characteristics.

3.5.5 Failures. The following events shall not cause dangerous or intolerable flying qualities:

a. Complete or partial loss of any function of the augmentation system following a single failure

b. Failure-induced transient motions and trim changes either immediately after failure or upon subsequent transfer to alternate control modes

c. Configuration changes required or recommended following failure.

3.5.5.1 Failure transients. With controls free, the airplane motions due to failures described in 3.5.5 shall not exceed the following limits for at least 2 seconds following the failure, as a function of the Level of flying qualities after the failure transient has subsided:

Levels 1 and 2 (after failure)	+/- 0.5g incremental normal or lateral acceleration at the pilot's station and +/- 10 degrees per second roll rate, except that neither stall angle of attack nor structural limits shall be exceeded. In addition for Category A, vertical or lateral excursions of 5 feet, +/- 2 degrees bank angle
Level 3 (after failure)	No dangerous attitude or structural limit is reached, and no dangerous alteration of the flight path results from which recovery is impossible.

3.5.5.2 Trim changes due to failures. The changes in control forces required to maintain attitude and sideslip for the failures described in 3.5.5 shall not exceed the following limits for at least 5 seconds following the failure:

Pitch----------10 pounds
Roll----------10 pounds
Yaw----------30 pounds

3.5.6 Transfer to alternate control modes. The transient motions and trim changes resulting from the intentional engagement or disengagement of any portion of the primary flight control system by the pilot shall be such that dangerous flying qualities never result.

3.5.6.1 Transfer transients. With controls free, the transients resulting from the situations described in 3.5.6 shall not exceed the following limits for at least 2 seconds following the transfer:

Within the Operational Flight Envelope	+/- 0.1g normal or lateral acceleration at the pilot's station and +/- 3 deg/sec roll
Within the Service Flight Envelope	+/- 0.3g at the pilot's station, +/- 5 deg/sec roll, the lesser of +/- 5 degrees sideslip and the structural limits.

These requirements apply only for Airplane Normal States.

3.5.6.2 Trim changes. The changes in control forces required to maintain attitude and sideslip for the situations described in 3.5.6 shall not exceed the following limits for at least 5 seconds following the transfer:

Pitch----------20 pounds
Roll----------10 pounds
Yaw----------30 pounds

These requirements apply only for Airplane Normal States.

3.6 Characteristics of secondary control systems

3.6.1 Trim system. In straight flight, throughout the Operational Flight Envelope the trimming devices shall be capable of reducing all the cockpit control forces to zero for Levels 1 and 2. For Level 3 the untrimmed steady-state cockpit-control forces shall not exceed 10 pounds

in pitch, 5 pounds in roll and 10 pounds in yaw (pedal). The failures to be considered in applying the Level 1 and 3 requirements shall include trim sticking and runaway in either direction. It is permissible to meet the Level 2 and 3 requirements by providing the pilot with alternate trim mechanisms or override capability. Additional requirements on trim rate and authority are contained in Mil-F-9490 and Mil-F-18372.

3.6.1.1 Trim for asymmetric thrust. For all multi-engine airplanes, it shall be possible to trim the cockpit-control forces to zero in straight flight with up to two engines inoperative following asymmetric loss of thrust from the most critical factors (3.3.8). This requirement defines Level 1 in level-flight cruise at speeds from the maximum-range speed for the engine(s)-out configuration to the speed obtainable with normal rated thrust on the functioning engine(s). Systems completely dependent on the failed engine(s) shall also be considered failed.

3.6.1.2 Rate of trim operation. Trim devices shall operate rapidly enough to enable the pilot to maintain low control forces under changing conditions normally encountered in service, yet not so rapidly as to cause oversensitivity or trim precision difficulties under any conditions. Specifically, it shall be possible to trim the pitch control forces to less than +/- 10 pounds for center-stick airplanes and +/- 20 pounds for wheel-control airplanes throughout:

a. dives and ground attack maneuvers required in normal service operation and

b. level-flight acceleration at maximum augmented thrust from 350 knots or $V_{R/C}$, whichever is less, to V_{max} at any altitude when the airplane is trimmed for level flight prior to initiation of the maneuver.

In the event that operation of the trim system requires removal of one hand from the wheel-control, Level 1 force limits shall be as for a center-stick.

3.6.1.3 Stalling of trim systems. Stalling of a trim system due to aerodynamic loads during maneuvers shall not result in an unsafe condition. Specifically, the longitudinal trim system shall be capable of operating during the dive recoveries of 3.2.5.4 at any attainable permissible n, at any possible position of the trimming device.

3.6.1.4 Trim system irreversibility. All trimming devices shall maintain a given setting indefinitely unless changed by the pilot, or by a special automatic interconnect (such as to the landing flaps), or by the operation of an augmentation device. If an automatic interconnect or augmentation device is used in conjunction with a trim device, provision shall be made to ensure the accurate return of the device to its initial trim position on removal of each interconnect or augmentation command.

3.6.2 Speed and flight-path control devices. The effectiveness and response times of the longitudinal controls shall be sufficient to provide adequate control of flight path and airspeed at any flight condition within the Operational Flight Envelope. This requirement may be met by use of devices such as throttles, thrust reversers, auxiliary drag devices and flaps.

3.6.3 Transients and trim changes. The transients and steady-state trim changes for normal operation of secondary control devices (such as throttle, thrust reversers, flaps, slats, speed brakes, deceleration devices, dive recovery devices, wing sweep and landing gear) shall not impose excessive control forces to maintain the desired heading, altitude, attitude, rate of climb, speed or load factor without use of trimmer control. This requirement applies to all in-flight configuration changes and combinations of changes made under service conditions, including the effects of asymmetric operations such as unequal operation of the landing gear, speed brakes, slats or flaps. In no case shall there be any objectionable buffeting or oscillation caused by such devices. More specific requirements on secondary control devices are contained in 3.6.1.1, 3.6.4 and 3.6.5 and in Mil-F-9490 and Mil-F-18372.

3.6.3.1 Pitch trim changes. The pitch trim changes caused by operation of secondary control devices shall not be so large that a peak pitch control force in excess of 10 lbs for center-stick controllers or 20 lbs for wheel controllers is required when such configuration changes are made in flight under conditions representative of operational procedure. Generally, the conditions listed in Table XV will suffice for determination of compliance with this requirement. (For airplanes with variable-sweep wings, additional requirements will be imposed consistent with operational employment of the vehicle). With the airplane trimmed for each specified initial condition, the peak force required to maintain the specified parameter constant following the specified configuration change shall not exceed the stated value for a time interval of at least 5 seconds following the completion of the pilot action initiating the configuration change. The magnitude and rate of trim change subsequent to this time period shall be such that the forces are easily trimmable by use of the normal trimming devices. These requirements define Level 1. For Levels 2 and 3, the allowable forces are increased by 50 percent.

3.6.4 Auxiliary dive recovery devices. Operation of any auxiliary device intended solely for dive recovery shall always produce a positive increment of normal acceleration, but the total normal load factor shall never exceed $0.8n_L$, controls free.

TABLE XV PITCH TRIM CHANGE CONDITIONS

No.	Flight Phase	Initial Trim Condition: Altitude	Speed	Landing Gear	High Lift Devices and Wing Flaps	Thrust	Configuration Change	Parameter held constant
1	Approach	$h_{o_{min}}$	Normal pattern entry speed	Up	Up	TLF	Gear down	Altitude and airspeed*
2	Same	Same	Same	Up	Up	TLF	Gear Down	Altitude
3	Same	Same	Same	Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude and airspeed*
4	Same	Same	Same	Down	Up	TLF	Extend high-lift devices and wing flaps	Altitude
5	Same	Same	Same	Down	Down	TLF	Idle thrust	Airspeed
6	Same	Same	$V_{o_{min}}$	Down	Down	TLF	Extend approach drag device	Airspeed
7	Same	Same	Same	Down	Down	TLF	Takeoff thrust	Airspeed
8	Same	Same	Same	Down	Down	TLF	Takeoff thrust plus normal cleanup for wave-off (go-around)	Airspeed
9	Takeoff	$h_{o_{min}}$	$V_{o_{min}}$	Down	Takeoff	Takeoff thrust	Gear up	Pitch attitude
10	Same	Same	Minimum flap retract speed	Up	Takeoff	Takeoff thrust	Retract high lift devices and wing flaps	Airspeed
11	Cruise and air-to-air combat	$h_{o_{min}}$ and $h_{o_{max}}$	Speed for level flight	Up	Up	MRT	Idle thrust	Pitch Attitude
12	Same	Same	Same	Up	Up	MRT	Actuate deceleration device	Same
13	Same	Same	Same	Up	Up	MRT	Maximum augmented thrust	Same
14	Same	Same	Speed for best range	Up	Up	TLF	Actuate deceleration device	Same

Notes: 1. Auxiliary drag devices are initially retracted, and all details of configuration not specifically mentioned are normal for the Flight Phase.

2. If power reduction is permitted in meeting the deceleration requirements established for the mission, actuation of the deceleration device in Numbers 12 and 14 shall be accompanied by the allowable power reduction.

* Throttle may be changed during the maneuver

3.7 Atmospheric disturbances

3.7.1 Form of the disturbance models. Where feasible, the Von Karman form shall be used for the continuous turbulence model, so that the flying qualities analyses will be consistent with comparable structural analyses. When no comparable structural analysis is performed or when it is not feasible to use the Von Karman form, use of the Dryden form will be permissible. In general, both the continuous turbulence model and the discrete gust model shall be used. The scales and intensities used in determining the gust magnitudes for the discrete gust model shall be the same as those in the continuous model.

3.7.1.1 Continuous turbulence model (Von Karman form). The Von Karman form of the spectra for the turbulence velocities is:

$$\Phi_{u_g}(\Omega) = \frac{(\sigma_u)^2(2L_u/\pi)}{(1 + (1.339L_u\Omega)^2)^{5/6}}$$

$$\Phi_{v_g}(\Omega) = \frac{(\sigma_v)^2(L_v/\pi)(1 + (8/3)(1.339L_v\Omega)^2)}{(1 + (1.339L_v\Omega)^2)^{11/6}}$$

$$\Phi_{w_g}(\Omega) = \frac{(\sigma_w)^2(L_w/\pi)(1 + (8/3)(1.339L_w\Omega)^2)}{(1 + (1.339L_w\Omega)^2)^{11/6}}$$

3.7.1.2 Continuous turbulence model (Dryden form). The Dryden form of the spectra of turbulence velocities is:

$$\Phi_{u_g}(\Omega) = \frac{(\sigma_u)^2(2L_u/\pi)}{(1 + (L_u\Omega)^2}$$

$$\Phi_{v_g}(\Omega) = \frac{(\sigma_v)^2(L_v/\pi)(1 + 3(L_v\Omega)^2}{(1 + (L_v\Omega)^2)^2}$$

$$\Phi_{w_g}(\Omega) = \frac{(\sigma_w)^2(L_w/\pi)(1 + 3(L_w\Omega)^2}{(1 + (L_w\Omega)^2)^2}$$

3.7.1.3 Discrete gust model. The discrete gust model may be used for any of the three gust-velocity components and, by derivation, any of the three angular components. The discrete gust has the '(1 - cosine)' shape given by:

$v = 0$ for $x < 0$

$v = (v_m/2)(1 - \cos\pi x/d_m)$, for $0 < x < d_m$

$v = v_m$ for $x > d_m$

In graphical form this is:

The discrete gust above may be used singly or in multiples to assess airplane response to, or pilot control of, large disturbances. Step function or linear ramp gusts may also be used.

3.7.2 Medium/high-altitude model. The scales and intensities are based on the assumption that turbulence above 2,000 feet is isotropic. Then:

$$\sigma_u = \sigma_v = \sigma_w$$

and:

$$L_u = L_v = L_w$$

3.7.2.1 Turbulence scale lengths. The scales to be used are:

$L_u = L_v = L_w = 2,500$ feet using the Von Karman form, or

$L_u = L_v = L_w = 1,750$ feet using the Dryden form.

3.7.2.2 Turbulence intensities. Root-mean-square turbulence intensities are shown in Figure B7 as functions of altitude and probability of exceedance. Simplified variations for application to the requirements of this specification are indicated.

3.7.2.3 Gust lengths. Several values of d_m shall be used, each chosen so that the gust is tuned to each of the natural frequencies of the airplane and its flight control system (higher-frequency structural modes may be excepted). The magnitude of v_m shall be determined from Figure B8.

Figure B7 Turbulence Exceedance Probability

Figure B8 Magnitude of Discrete Gusts

3.7.2.4 Gust magnitudes. The Light and Moderate gust magnitudes u_g, v_g and w_g shall be determined from Figure B8 using the values of d_x, d_y and d_z determined according to 3.7.3.3. and the appropriate RMS turbulence intensities from Figure B7. Severe gusts shall be:

a. 66 ft/sec EAS at V_G, gust penetration speed

b. 50 ft/sec EAS at $V_{G_{max}}$

c. 25 ft/sec EAS at V_{max}

d. 50 ft/sec EAS at speeds up to V_{max} (PA) with the landing gear and other devices which are open or extended in their maximum open or maximum extended positions

e. For altitudes above 20,000 feet the gust magnitudes may be reduced linearly from:

(1) 66 ft/sec EAS at 20,000 feet to 38 ft/sec EAS at 50,000 feet for the V_G condition

(2) 50 ft/sec EAS at 20,000 feet to 25 ft/sec EAS at 50,000 feet for the $V_{G_{max}}$ condition

(3) 25 ft/sec EAS at 20,000 feet to 12.5 ft/sec EAS at 50,000 feet for the V_{max} condition

f. For altitudes above 50,000 feet the equivalent gust velocity specified at 50,000 feet shall be multiplied by the factor $(\rho/\rho_{50})^{1/2}$, the square root of the ratio of air density at altitude to standard atmospheric density at 50,000 feet.

3.7.3 Low-altitude disturbance model. This section specifies the model of atmospheric disturbances to be used for all Category C operations. The effects of wind shear, turbulence and gusts may be analysed separately. Some analysis and piloted simulation is required considering a complete environmental representation, demonstrating compliance with the requirements with the cumulative effects of wind shear, turbulence and gusts. A non-Gaussian turbulence representation together with a wind model may also be used to represent the patchy, intermittent nature of actual measured turbulence.

3.7.3.1 Wind speeds. The wind speed at 20 feet above the ground, u_{20}, is shown in Figure B9 as a function of probability of occurrance. The values to be used for the different intensities of atmospheric disturbance are indicated.

Figure B9 Probability of Exceeding Mean Wind Speed at 20 Feet

3.7.3.3 Wind shear. The magnitude of the wind scalar shear is defined by the use of the following expression for the mean wind profile as a function of altitude:

$$u_w = u_{20}[\ln(h/z_o)]/[\ln(20/z_o)]$$

where: z_o = 0.13 feet for Category C Flight Phase

= 1.0 feet for other Flight Phases.

3.7.3.3 Vector shear. Different orientations of the mean wind relative to the runway for Category C, or relative to the aircraft flight path for other Flight Phases, shall be considered. In addition, changes in direction of the mean wind speed over a given height change shall be considered as follows:

Disturbance intensity	Change in mean wind heading, degrees	Height of vector shear, feet
LIGHT	0	----
MODERATE	90	600
SEVERE	90	300

A range of values for the initial wind orientation and the initial altitude for onset of the shear shall be considered. Relative to the runway, magnitudes of $u_{20}\sin\phi_w$ greater than the crosswind values in 3.3.7 or tailwind component at 20 feet greater than 10 knots need not be considered. At any altitude other than 20 feet these limits do not apply.

3.7.3.4 Turbulence. The turbulence models of 3.7.1.1 or 3.7.1.2 shall be used. The appropriate scale lengths are shown in Figure B10 as functions of altitude. The turbulence intensities to be used are:

$\sigma_w = 0.1u_{20}$, and σ_u and σ_v given by Figure B11 as functions of σ_w and altitude.

3.7.3.5 Gusts. Discrete gusts of the form specified in 3.7.2.3 shall be used, with both single and double ramps to be considered. Several values of d_m shall be used, each chosen so that the gust is tuned to each of the natural frequencies of the airplane and its flight control system. The gust magnitudes shall be determined from Figure B8 using the appropriate values from Figures B10 and B11. The two halves of a double gust do not have to be the same length or magnitude.

3.7.4 Carrier landing disturbance model. This section specifies the model of atmospheric disturbances to be used for carrier landing operations. This model shall be used in analysis and piloted simulation to determine aircraft control response and path control accuracy during carrier landing. This model supplements but does not replace the low-altitude model of 3.7.3.

The terminal approach carrier landing disturbance model shall be used during simulation of the last 1/3 mile of the carrier approach. The u velocity component is aligned with the wind over the deck. Total disturbance velocities are computed by adding segments caused by random free-air turbulence, u_1, v_1, w_1, steady ship-wake disturbance, u_2, v_2, periodic ship-motion-induced turbulence, u_3, v_3, and random ship-wake disturbance, u_4, v_4, w_4. The total air disturbance components u_g, v_g and w_g are then computed as:

$$u_g = u_1 + u_2 + u_3 + u_4$$

$$v_g = v_1 + v_4$$

$$w_g = w_1 + w_2 + w_3 + w_4$$

The input to all of the random disturbance filters shall be generated by filtering the wide-band, Gaussian output of zero-mean, unit-variance random-number generators.

3.7.4.1 Free-air turbulence components. The free-air turbulence components which are independent of aircraft relative position are represented by filtering the output of white-noise generators described in 3.7.4 to produce the following spectra:

$$\phi_{u_1}(\Omega) = \frac{100}{(1 + (100\Omega)^2)} \quad \text{per radian/ft}$$

$$\phi_{v_1}(\Omega) = \frac{3300(1 + (400\Omega)^2)}{(1 + (1000\Omega)^2)(1 + (4000/\Omega)^2)} \quad \text{per radian/ft}$$

$$\phi_{w_1}(\Omega) = \frac{71.6}{(1 + (100\Omega)^2)} \quad \text{per radian/ft}$$

Figure B10 Low Altitude Turbulence Intensity Scales

Figure B11 Horizontal Turbulence RMS Intensities

3.7.4.1 Steady component of carrier airwake. The steady components of the carrier airwake consist of a reduction in the steady wind and a predominant upwash aft of the ship which are functions of range. Figure B12 illustrates the steady wind functions $u_2/V_{w/d}$ and $w_2/V_{w/d}$ as functions of position aft of the ship center of pitch.

3.7.4.2 Periodic component of carrier airwake. The periodic component of the airwake varies with ship pitching frequency, pitch magnitude, wind over deck and aircraft range. These components are computed as follows:

$$u_3 = \theta_a V_{w/d}(1.22 + 0.0005X)C$$

$$w_3 = \theta_a V_{w/d}(4.98 + 0.0013X)C$$

with: C =

$$\cos[\omega_p(t + t(V - V_{w/d})/(0.85V_{w/d}) + X/(0.85V_{w/d})) + P]$$

where: ω_p = Ship pitch frequency, rad/sec.

θ_a = Ship pitch amplitude, rad.

P = Random phase, rad.

X = the distance from the ship center of pitch, in ft, as defined in Figure B13.

Notes from the author: 1. The center of pitch location depends on the type of carrier. In preliminary design, the center of pitch may be assumed to be at one half the length of the carrier.

2. Values for ship pitch frequency, pitch amplitude and random phase must be obtained from the procuring activity. These values depend on ship type and on sea state.

The u component is set to zero for X < -2336 feet, and the w component is set to zero for X < -2336 feet.

3.7.4.4 Random component of carrier air wake. The ship-related random velocity components are computed by filtering white noise (3.7.4) as follows:

$$u_4 = \frac{\sigma(X)[2\tau(X)]^{1/2}(\text{Input})}{[\tau(X)j\omega + 1]}$$

$$w_4 = v_4 = \frac{0.835V_{w/d}(5.66)^{1/2}(\text{Input})}{(3.33j\omega + 1)}$$

where:

$\sigma(X)$ = RMS Amplitude in ft/sec., see Figure B13

$\tau(X)$ = Time constant in sec., see Figure B13

Input = (Random number output)$[j\omega/(j\omega + 0.1)]\sin(10\pi t)$

3.7.5 Application of the disturbance model in analyses. The gust and turbulence velocities shall be applied to the airplane equations of motion through the aerodynamic terms only, and the direct effect on the aerodynamic sensors shall be included when such sensors are part of the airplane augmentation system. When using the discrete gust model, all significant aspects of the penetration of the gust by the airplane shall be incorporated in the analyses. Application of the disturbance model depends on the range of frequencies of concern in the analyses of the airframe. When structural modes are significant, the exact distribution of turbulence velocities should be considered. For this purpose, it is acceptable to consider u_g and v_g as being one-dimensional functions only of x, but w_g shall be considered two-dimensional, a function of both x and y, for the evaluation of aerodynamic forces and moments.

When structural modes are not significant, airframe rigid-body responses may be evaluated by considering uniform gust or turbulence immersion along with linear gradients of the disturbance velocities. The uniform immersion is accounted for by u_g, v_g and w_g defined at the airplane center of gravity. The angular velocities due to turbulence are equivalent in effect to airplane angular velocities. Approximations for these angular velocities are defined (precisely at very low frequencies only) as follows:

$$\dot{\alpha}_g = q_g = \partial w_g/\partial x,\quad p_g = \partial w_g/\partial y,\quad r_g = \partial v_g/\partial x$$

The spectra of the angular velocity disturbances due to turbulence are then given by:

$$\Phi_{p_g}(\Omega) = [(\sigma_w)^2/L_w]\frac{(0.8(\pi L_w/4b)^{1/3})}{(1 + (4b\Omega/\pi)^2)}$$

$$\Phi_{q_g}(\Omega) = [(\Omega)^2/(1 + (4b\Omega/\pi)^2)]\Phi_{w_g}(\Omega)$$

$$\Phi_{r_g}(\Omega) = [(\Omega)^2/(1 + (3b\Omega/\pi)^2)]\Phi_{v_g}(\Omega)$$

where: b is the wing span.

The turbulence components u_g, v_g, w_g and p_g shall be considered mutually independent (uncorrelated) in a statistical sense. However, q_g is correlated with w_g, and r_g is correlated with v_g. For the discrete gusts the linear gradient gives angular velocity perturbations of the following form:

$$p_g = p_m \sin(\pi x/d_m) \qquad 0 \leq x \leq d_m$$

For the low-altitude model, the turbulence velocity components u_g, v_g and w_g are to be taken along axes with u_g aligned along the relative mean wind vector and with v_g vertical.

Figure B12 CVA Ship Burble Steady Wind Ratios

Figure B13 u-Component Burble Time Constant and Variance

3.8 Requirements for use of the disturbance models

Explicit consideration of the effects of disturbances on flying qualities, if required by the procuring activity, shall be in accordance with requirements in 3.8.2 through 3.8.3.1. In particular, 3.8.3.1 will replace 3.1.10.1 and 3.8.3.2 will replace 3.1.10.2.

3.8.1 Use of disturbance models. Paragraphs 3.7.1 through 3.7.4.4 specify models of wind shear, continuous random turbulence and discrete gusts that shall be used to assess:

a. The effects of certain environmental conditions on the flying qualities of the airplane.

b. The ability of the pilot to recover from upsets caused by environmental conditions.

c. Flight path control precision during manual and automatic carrier landing.

For the purpose of this specification the atmosphere shall be considered to consist of three regions:

(1) low-altitude (ground level to approximately 2,000 feet AGL),

(2) medium/high-altitude (above approximately 2,000 feet) and,

for carrier landing only,

(3) terminal approach (0 - 300 feet altitude and 1/2 mile to touchdown).

The low-altitude model shall apply to Category C and any other Flight Phase (e.g. ground attack, terrain following) designated by the procuring activity. The medium/high-altitude model is intended to apply to those Flight Phases where proximity to the ground is not a factor, generally Categories A and B. In application it will be permissible to use conditions at an average altitude for the medium/high-altitude model only. The carrier landing disturbance model will apply to carrier-based aircraft only.

3.8.2 Qualitative degrees of suitability. In assessing the qualitative suitability of flying qualities three intensities and disturbances shall be considered. These intensities are Light, Moderate and Severe as defined in 3.7. The requirements for the effects of these disturbances are contained in 3.8.3.1 and 3.8.3.2 for the different Flight Envelopes and Airplane States.

The qualitative degrees of suitability of flying qualities are categorized as follows:

Satisfactory	Flying qualities clearly adequate for the mission Flight Phase
Acceptable	Flying qualities adequate to accomplish the mission Flight Phase, but some increase in pilot workload or degradation in mission effectiveness, or both, exists
Controllable	Flying qualities such that the airplane can be controlled safely, but pilot workload is excessive or mission effectiveness is inadequate, or both. Category A Flight Phases can be terminated safely, and Category B and C Flight Phases can be completed.
Recoverable	Flying qualities such that control can be maintained long enough to fly out of a disturbance. All Flight Phases can be terminated safely and a wave-off/go-around can be accomplished.

3.8.3 Effects of atmospheric disturbances. Levels of flying qualities as indicated in 3.5 are employed in this specification in realization of the possibility that the airplane may be required to operate under abnormal conditions. Such abnormalities may occur also as a result of extreme atmospheric disturbances, or some combination of conditions. For these factors a degradation of flying qualities is permitted as specified in 3.8.3.1 and in 3.8.3.2 (see also 4.1.3).

3.8.3.1 Requirements for Airplane Normal States. In atmospheric disturbances the minimum required flying qualities for Airplane Normal States (3.1.4.1) are as specified in Table XVI.

3.8.3.2 Requirements for Airplane Failure states. When Airplane Failure States exist (3.3.4.2), a degradation in flying qualities is permitted only if the probability of encountering a lower Level than specified in 3.8.3.1 is sufficiently small. At intervals established by the procuring activity, the contractor shall determine, based on the most accurate available data, the probability of occurrence of each Airplane Failure State per flight and the effect of that Failure State on the flying qualities within the Operational and Service Flight Envelopes. These determinations shall be based on Mil-STD-756 except that:

a. All airplane components and systems are assumed to be operating for a time period, per flight, equal to the longest operational mission time to be considered by the contractor in designing the airplane, and

b. Each specific failure is assumed to be present at whichever point in the Flight Envelope being considered is most critical (in the flying qualities sense).

From these Failure State probabilities and effects, the contractor shall determine the overall probability, per flight, that one or more flying qualities are degraded to Level 3 because of one or more failures. The contractor shall also determine the probability that one or more

flying qualities are degraded to Level 3. Table XVII specifies the requirements as functions of the probability of encountering the degradation in flying qualities.

TABLE XVI LEVELS FOR AIRPLANE NORMAL STATES

Atmospheric Disturbances	Within Operational Flight Envelope	Within Service Flight Envelope
LIGHT to CALM	Quantitative requirements Level 1. Qualitative requirements Satisfactory	Quantitative requirements Level 2. Qualitative requirements Acceptable
MODERATE to LIGHT	Quantitative requirements Level 1, Qualitative requirements Acceptable or better	Quantitative requirements Level 2. Qualitative requirements Controllable or better
SEVERE to MODERATE	Qualitative requirements Controllable or better	Qualitative requirements Recoverable or better

TABLE XVII LEVELS FOR AIRPLANE FAILURE STATES

Atmospheric Disturbances	Failure State I*	Failure State II**
LIGHT to CALM	Quantitative requirements Level 2 and qualitative requirements Acceptable or better	Quantitative requirements Level 3 and qualitative requirements Controllable or better
MODERATE to LIGHT	Quantitative requirements Level 2 and qualitative requirements Controllable or better	Quantitative requirements Level 3 and qualitative requirements Recoverable or better
SEVERE to MODERATE	Qualitative requirements Recoverable or better	

* For flight in the Operational Flight Envelope: Probability of encountering degraded levels of flying qualities due to failure(s) < 1 per 100 flights

** For flight in the Operational Flight Envelope: Probability of encountering degraded levels of flying qualities due to failure(s): < 1 per 10,000 flights, and for flight in the Service Flight Envelope. Probability of encountering degraded levels of flying qualities due to failure(s): < 1 per 100 flights

4. QUALITY ASSURANCE

4.1 Compliance demonstration. Compliance with all requirements of section 3 shall be demonstrated through analysis. In addition, compliance with many of the requirements will be demonstrated by simulation, flight test or both. The methods for demonstrating compliance shall be established by agreement between the procuring activity and the contractor. Representative flight conditions, configurations, external store complements, loadings, etcetera, shall be determined for detailed investigations in order to restrict the number of design and test conditions. The selected design points must be sufficient to allow accurate extrapolation to other conditions at which the requirements apply.

Table XVIII specifies the general guidelines but the peculiarities of the specific airplane design may require additional or alternate test conditions. The required failure analyses shall be thorough, excepting only approved Special Failure States (3.1.6.3.1).

4.1.1 Analytical compliance

4.1.1.1 Effects of Failure States. To determine theoretical compliance with the requirements of 3.1.10.3, the following steps must be performed:

a. Identify those Airplane Failure States which have a significant effect on flying qualities (3.1.6.2)

b. Define the longest flight duration to be encountered during operational missions (3.3.1)

c. Determine the probability of encountering various Airplane Failure States per flight, based on the above flight duration (3.1.10.2)

d. Determine the degree of flying qualities degradation associated with each Airplane Failure State in terms of Levels as defined in the specific requirements

e. Determine the most critical Airplane Failure States (assuming the failures are present at whichever point in the Flight Envelope being considered is most critical in a flying qualities sense), and compute the total probability of encountering Level 2 flying qualities in the Operational Flight Envelope due to equipment failures. Likewise, compute the probability of encountering Level 3 flying qualities in the Operational Flight Envelope, etc.

f. Compare the computed values above with the requirements specified in 3.1.10.2 and 3.1.10.1. An example which illustrates an approximate estimate of the probabilities of encounter follows: If the failures are all statistically independent, determine the sum of the probabilities of encountering all Airplane Failure

States which degrade flying qualities to Level 2 in the Operational Flight Envelope. This sum must be less than 1 per 100 flights.

If the requirements are not met, the designer must consider alternate courses such as:

a. Improve the airplane flying qualities associated with the more probable Failure States, or

b. Reduce the probability of encountering the more probable Failure States through equipment redesign, redundancy, etc.

Regardless of the probability of encountering any given Airplane Failure States (with the exception of Special Failure States) the flying qualities shall not degrade below Level 3.

4.1.1.2 Effects of atmospheric disturbances. Paragraph 4.1.1.1 indicates a procedure for satisfying the requirements on the degrading effects of Airplane Failure States, without consideration of atmospheric disturbances. Atmospheric disturbances also may cause a degradation in pilot opinion as specified in 3.8.2. In application, numerical values of control force and deflection, and of steady-state and dynamic response parameters (for example $n_{o_{max}}$, $\partial F_s/\partial n$, θ_t) are to be considered as mean values in the presence of atmospheric disturbances. These are frequently equivalent to the values in calm air. Numerical values of frequency-response parameters and of control authority are effective values for the airplane in each particular intensity of atmospheric disturbance. The qualitative requirements of 3.8.3.1 and 3.8.3.2 should then be assessed for both Airplane Normal States and critical Failure States identified in 4.1.1.1.

4.1.1.3 Computational assumptions. Assumptions a and b of 3.1.10.2 are somewhat conservative, but they simplify the required computations in 3.1.10.1 and provide a set of workable ground rules for theoretical predictions. The reasons for these assumptions are:

a. '...components and systems are ... operating for a time period per flight equal to the longest operational mission time ...'. Since most component failure data are in terms of failures per flight hour, even though continuous operation may not be typical (e.g. yaw damper ON during supersonic flight only), failure probabilities must be predicted on a per flight basis using 'typical' total flight time. The 'longest operational mission time' as 'typical' is a natural result. If acceptance cycles-to-failure reliability data are available (MIL-STD-756), these data may be used for prediction purposes based on maximum cycles per operational mission, subject to procuring activity approval. In any event, compliance with the requirements of 3.1.10.2, as determined in accordance with Section 4, is based on the probability of encounter per flight.

b. '... failure is assumed to be present at whichever point ... is most critical ...'. This assumption is in keeping with the requirements of 3.1.6.2 regarding Flight Phases subsequent to the actual failure in question. In cases that are unrealistic from the operational standpoint, the specific Airplane Failure States might fall in the Airplane Special Failure State classification (3.1.6.2.1).

4.1.2 Simulation. The danger, extent or difficulty of flight testing may dictate simulation rather than flight test to evaluate some conditions and events, such as the influence of Severe disturbances, events close to the ground (except 3.2.3.4 shall be demonstrated in flight), combined Failure States and disturbances, etc. In addition, by agreement with the procuring activity, piloted simulation shall be performed before first flight of a new airplane design in order to demonstrate compliance with qualitative requirements in atmospheric disturbances and in the critical conditions identified in 4.1.1.1. Where simulation is the ultimate method of demonstrating compliance for a requirement, the simulation model shall be validated with flight test data and approved by the procuring activity.

4.1.3 Flight test demonstration. The required flight tests will be defined by operational, technical and safety considerations as decided jointly by the procuring activity, the test agency and the contractor using results from 4.1.1 and 4.1.2. It is not expected that flight test demonstration of the requirements in Moderate and Severe disturbances will be done unless required by the airplane mission. Some flights can be expected to encounter actual disturbances. Then the qualitative requirements would apply if the disturbance intensity could be categorized.

4.2 Airplane States. The parameters defining Airplane States shall be tabulated. Table XIX illustrates an acceptable format.

4.2.1 Weights and moments of inertia. Terms specified in Table XVIII such as 'heaviest weight' and 'greatest moment of inertia' mean the heaviest and greatest consistent with 3.1.2 and 3.1.3. When a critical center-of-gravity position is identified, the airplane weight and associated moments of inertia shall correspond to the most adverse service loading in which that critical center-of-gravity position is obtained.

4.2.2 Center-of-gravity positions. Terms specified in Table XVIII such as 'most forward c.g.' and 'most aft c.g.' mean the most forward or aft consistent with 3.1.2. When a critical weight or inertia is identified, the center-of-gravity position shall correspond to the most adverse service loading in which that critical weight or moment of inertia is obtained.

4.2.3 Thrust settings. Thrust settings shall be as listed in Table XIX.

TABLE XVIII DESIGN AND TEST CONDITION GUIDELINES

Requirement Number	Title	Critical Loading (4.1.1 and 4.1.2)	Load Factor	Altitude (4.1.1)	Speed	Flight Phase
Section 3.2	**Longitudinal Flying Qualities**					
3.2.1.1	Longitudinal static stability	Most aft c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}, transonic	CO, CR, LO, RR, FF, RT, PA, L, WO TO, CT
3.2.1.1.1	Relaxation in transonic flight	Same	Same	Same	Same	Same
3.2.1.1.2	Elevator control force variations during rapid speed changes	----	As reqd	Same	Same	CO, GA, DE
3.2.1.2	Phugoid stability	Most fwd c.g.*	1.0	Same	V_{min} to V_{max}	CR, LO, PA, RT
3.2.1.3	Flight-path stability	----	Same	Same	$V_{o_{min}}$, $V_{o_{min}}$ - 5 kts	PA
3.2.2.1.1	Short period frequency and acceleration sensitivity	Most fwd* and most aft** c.g.	Same	Same	V_{min} to V_{max}	(*), CR, RT, PA, L, CT
3.2.2.1.2	Short period damping	Most fwd c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	(*), CR, RT, PA, CT
3.2.2.1.3	Residual oscillations	----	Same	Same	$V_{o_{min}}$ to $V_{o_{max}}$	(*), PA
3.2.2.2	Control feel and stability in maneuvering flight	Most aft c.g.	n(-) to n(+)	Same	V_{min} to V_{max}	(*), RT, CR, PA, L, CT
3.2.2.2.1	Control forces in maneuvering flight	Most fwd* and most aft** c.g.	n_o(-) to $n_{(+)}$	Same	Same	Same
3.2.2.2.2	Control motions in maneuvering flight	Most fwd c.g.*	Same	Same	Same	Same
3.2.2.3	Longitudinal pilot-induced oscillations	----	Min. to max. permissible	Same	Same	(*), RT, CR, PA, L, CT
3.2.2.3.1	Dynamic control forces in maneuvering flight	Most fwd c.g.*	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	----
3.2.2.3.2	Control feel	Most aft c.g.**	Same	Same	Same	----
3.2.3.1	Longitudinal control in unaccelerated flight	Most fwd c.g.	Same	Same	Same	----

* Combined with heaviest weight ** Combined with lightest weight

TABLE XVIII (CONT'D) DESIGN AND TEST CONDITION GUIDELINES

Requirement Number	Title	Critical Loading (4.2.1 and 4.2.2)	Load Factor	Altitude (4.3.1)	Speed	Flight Phase
3.2.3.1	Longitudinal control in maneuvering flight	Most fwd c.g.*	As reqd	Same	$V_{o_{min}}$ to $V_{o_{max}}$	CO, GA, AR, TF, CR, PA
3.2.3.3	Longitudinal control in takeoff	Most fwd c.g. for nosewheel and most aft c.g. for tailwheel airplanes	1.0	Low	As reqd	TO
3.2.3.3.1	Longitudinal control in catapult takeoff	Most fwd and most aft c.g.	As reqd	Same	Min. safe launch speed to min. + 30 kts	CT
3.2.3.3.2	Longitudinal control force and travel in takeoff	Most fwd and most aft c.g.	As reqd	Low	0 to V_{max} (TO)	TO, CT
3.2.3.4	Longitudinal control in landing	Most fwd c.g.	1.0	Same	V_s(L) or geometric limit	L
3.2.3.4.1	Longitudinal control forces in landing	Same	Same	Same	Same	L
3.2.3.5	Longitudinal control forces in dives (SFE)	Most fwd* and most aft** c.g.	As reqd	2000 ft above MGL to h_{max}	V_{min} to V_{max}	D, ED, CO, CR
3.2.3.6	Longitudinal control forces in dives (PFE)	Same	Same	As reqd	V_{MAT} to max permissible	Same
3.2.3.7	Longitudinal control in sideslips	----	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO, CR, PA, L
Section 3.3	Lateral-Directional Flying Qualities					
3.3.1.1	Lateral-directional oscillations (Dutch roll)	Greatest yawing moment of inertia	1.0 and n_o(+)	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to $V_{o_{max}}$	(*), CR, RT, PA, L
3.3.1.2	Roll mode	Greatest rolling moment of inertia	Same	$h_{o_{max}}$	Same	(*), CR, PA, L
3.3.1.3	Spiral stability	----	1.0	Same	Same	(*), CL, CR, LO, RT, DE, PA, L
3.3.1.4	Coupled roll-spiral oscillations	----	1.0 and n_o(+)	Same	Same	(*), CR, PA, L
3.3.2.1	Lateral-directional response to atmospheric disturbances	----	1.0	Same	Same	----
3.3.2.2	Roll rate oscillations	----	1.0 and n_o(+)	Same	Same	(*), CR, PA, L
3.3.2.2.1	Additional roll rate reqm't for small inputs	----	Same	Same	Same	Same

* Combined with heaviest weight
SFE = Service Flight Envelope

** Combined with lightest weight
PFE = Permissible Flight Envelope

TABLE XVIII (CONT'D) DESIGN AND TEST CONDITION GUIDELINES

Requirement Number	Title	Critical Loading (4.3.1 and 4.3.3)	Load Factor	Altitude (4.3.1)	Speed	Flight Phase
3.3.2.3	Bank angle oscillations	----	1.0 and $n_o(+)$	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to $V_{o_{max}}$	(*), CR, PA, L
3.3.2.4	Sideslip excursions	Greatest yawing and rolling moment of inertia	1.0	Same	Same	Same
3.3.2.4.1	Additional sideslip reqm't for small inputs	Same	1.0	Same	Same	Same
3.3.2.5	Control of sideslip in rolls	Greatest rolling moment of inertia	As reqd	Same	Same	CO, GA, AR, TF, CR, PA, L
3.3.2.6	Turn coordination	----	Same	Same	$V_{o_{min}}$	CO, CR, LO, PA
3.3.3	Pilot-induced oscillations	----	Min. to max. permissible	MSL to h_{max}	V_{min} to V_{max}	----
3.3.4	Roll control effectiveness	Greatest rolling moment of inertia	As reqd (not above $0.8n_L$)	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	CO, GA, AR, TF, CR, PA, L
3.3.4.1	Roll performance for Class IV airplanes	Same	Same	Same	Same	Same
3.3.4.1.1	Roll performance in Flight Phase CO	Same	Same	$h_{o_{min}}$	Same	CO
3.3.4.1.2	Roll performance in Flight Phase GA	Same	Same	Same	Same	GA
3.3.4.1.3	Roll response	Smallest rolling moment of inertia	Same	$h_{o_{min}}$ medium $h_{o_{max}}$	Same	----
3.3.4.2	Roll performance for Class III airplanes	Greatest and smallest rolling moment of inertia	Same	Same	Same	CO, GA, AR, TF, CR, PA, L
3.3.4.3	Roll control force	Greatest rolling moment of inertia	As reqd (not above $0.8n_L$)	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	----
3.3.4.4	Linearity of roll response	Same	Same	Same	Same	CO, GA, AR, TF, CR, PA, L
3.3.4.5	Wheel control throw	Same	Same	Same	Same	Same
3.3.5	Directional control characteristics	----	n(-) to n(+)	Same	Same	(*), CR, PA, L

* Combined with heaviest weight ** Combined with lightest weight

TABLE XVIII (CONT'D) DESIGN AND TEST CONDITION GUIDELINES

Requirement Number	Title	Critical Loading (4.1.1 and 4.2.1)	Load Factor	Altitude (4.3.1)	Speed	Flight Phase
3.3.1.1	Directional control with speed change	----	1.0	Same	Same	CO, GA, CR, D, PA, L
3.3.1.1.1	Directional control with asymmetric loading	----	Same	Same	$V_{o_{min}}$ to $V_{o_{max}}$	----
3.3.1.2	Directional control in wave-off (go-around)	Lightest weight	1.0	Low	V_{min} (PA) or guaranteed landing speed	WO
3.3.6 (3.3.6.1, 3.3.6.2, 3.3.6.3, 3.3.6.3.1, 3.3.6.3.2)	Lateral-directional characteristics in steady sideslips	Same	Same	$h_{o_{min}}$ medium, $h_{o_{max}}$	V_{min} to V_{max}	CO, CR, PA, L
3.3.7	Lateral-directional control in cross winds	----	Same	Low	As reqd	TO, L
3.3.7.1	Final approach in cross winds	----	Same	Same	V_{min} to V_{max}	PA
3.3.7.2 (3.3.7.2.1, 3.3.7.2.2)	Takeoff run and landing rollout in cross wind	----	As reqd	Same	As reqd	TO, L
3.3.7.3	Taxiing wind speed limits	----	As reqd	Same	All taxiing speeds	TAXI
3.3.8	Lateral-directional control in dives	----	As reqd	3,000 ft above MSL to h_{max}	V_{MAT} to V_{max}	D, ED
3.3.9.1	Thrust loss during takeoff run	Lightest weight	1.0	$h_{o_{min}}$	0 to max. takeoff speed	TO
3.3.9.2	Thrust loss after takeoff	Same	Same	Same	Down to V_{mc} (TO)	TO, CT
3.3.9.3	Transient effects	Same	Same	All	V_{min} to V_{max}	CO, GA, TF, CR, CL, TO, CT
3.3.9.4	Asymmetric thrust, rudder pedals free	Same	Same	$h_{o_{min}}$, medium, $h_{o_{max}}$	$1.4V_{min}$	CR
3.3.9.5	Two engines inoperative	----	Same	Same	V_{range} (1 and 2 engines out)	----

* Combined with heaviest weight ** Combined with lightest weight

TABLE XVIII (CONT'D) DESIGN AND TEST CONDITION GUIDELINES

Requirement Number	Title	Critical Loading (4.2.1 and 4.2.2)	Load Factor	Altitude (4.3.1)	Speed	Flight Phase
Section 3.4	**Miscellaneous Flying Qualities**					
3.4.2	Flight at high angle of attack	See MIL-S-83691 or MIL-D-8708, whichever is applicable for flight demonstration. More severe conditions generally will be investigated by analysis and by model testing. ***				
3.4.2.1	Stalls (3.4.2.1.1 through 3.4.2.1.3.1)	See *** above				
3.4.2.2	Post-stall gyrations and spins (3.4.2.2.1, 3.4.2.2.2)	See *** above				
3.4.3	Cross-axis coupling in roll maneuvers	----	ϕ to $0.8n_z$	$h_{o_{min}}$, medium $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	CO, GA, AR, TF
3.4.4 (3.4.4.1)	Control harmony	----	$n_o(-)$ to $n_o(+)$	Same	Same	----
3.4.5	Buffet	----	Same	Same	Same	(*)
3.4.6	Release of stores	----	Same	Same	Same	CO, GA, WD, D
3.4.7	Effects of armament delivery and special equipment	----	Same	Same	Same	(*), RT
3.4.8	Transients following failures	----	All	$h_{o_{min}}$ to $h_{o_{max}}$	All	----
3.4.10	Control margin	----	Same	Same	Same	----
3.4.11	Direct force control	----	1.0 +/- max. DLC authority	Same	$V_{o_{min}}$ to $V_{o_{max}}$	----
Section 3.5	**Characteristics of the Primary Flight Control System**					
3.5.2 (3.5.2.1, 3.5.2.2, 3.5.2.3)	Mechanical characteristics	----	$n_o(-)$ to $n_o(+)$	$h_{o_{min}}$ and $h_{o_{max}}$	V_{min} to V_{max}	----
3.5.3	Dynamic characteristics	Most fwd c.g. and lowest rolling and yawing moments of inertia	1.0			----
3.5.5 (3.5.5.1, 3.5.5.2)	Failures	----	All			----
3.5.6 (3.5.6.1, 3.5.6.2)	Transfer to alternate control modes	----	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$		----

NOTES: 1. ---- indicates that no general guidance can be provided.

2. The phrase 'As reqd' means the flight conditions are specified in the requirement or are determined by the nature of the test maneuver.

3. (*) means all applicable Category A Flight Phases.

TABLE A.11 (CONT'D) DESIGN AND TEST CONDITION GUIDELINES

Requirement Number	Title	Critical Loading (4.2.1 and 4.2.2)	Load Factor	Altitude (4.3.1)	Speed	Flight Phase
Section 3.6	**Characteristics of Secondary Control Systems**					
3.6.1	Trim system	Most fwd and most aft c.g.	1.0	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	----
3.6.1.1	Trim for asymmetric thrust	Same	Same	$h_{o_{min}}$ and max. attainable	V_{range} to V_{MRT} (with 1 and 2 engines out)	CK
3.6.1.2	Rate of trim operation	----	Same	As reqd	As reqd	CO, GA, D, ED
3.6.1.3	Stalling of trim system	Most fwd c.g. combined with heaviest weight	As reqd	Same	Start of dive recovery to V_{max}	D, ED, CO, CK
3.6.1.4	Trim system irreversibility	----	1.0	MSL to h_{max}	V_{min} to V_{max}	----
3.6.2	Speed and flight-path control devices	----	1.0 to n_o(+)	$h_{o_{min}}$ medium, $h_{o_{max}}$	$V_{o_{min}}$ to $V_{o_{max}}$	(*), RT, ED, DE, PA, WO, GA
3.6.3	Transients and trim changes	----	n_o(-) to n_o(+)	$h_{o_{min}}$, medium, $h_{o_{max}}$	V_{min} to V_{max}	----
3.6.3.1	Pitch trim changes	----	As reqd	As reqd	As reqd	CO, CR, FA TO, CT
3.6.4	Auxiliary dive recovery devices	Most aft c.g. combined with lightest weight	Same	MSL to to h_{max}	$V_{o_{min}}$ to V_{max}	D, ED
Section 3.7	**Atmospheric Disturbances**	----	1.0	Same	V_{min} to V_{max}	----
Section 3.8	**Requirements for Use of the Disturbance Models**	----	All	Same	Same	----

NOTES: 1. ---- indicates that no general guidance can be provided.

2. The phrase 'As reqd' means the flight conditions are specified in the requirement or are determined by the nature of the test maneuver.

3. (*) means all applicable Category A Flight Phases.

TABLE XIX DEFINITION OF AIRPLANE NORMAL STATES

Flight Phase	Symbol	Weight	C.G.	External Stores	Thrust	Thrust Vector Angle	High Lift Devices	Wing Sweep
Takeoff	TO							
Climb	CL							
Cruise	CR							
Loiter	LO							
Descent	D							
Emergency Descent	ED							
Emergency Deceleration	DE							
Approach	PA							
Wave-off/ Go-around	WO							
Landing	L							
Air-to-air Combat	CO							
Ground Attack	GA							
Weapon Delivery/ Launch	WD							
Aerial Delivery	AD							
Aerial Recovery	AR							
Reconnaissance	RC							
Refuel Receiver	RR							
Refuel Tanker	RT							
Terrain Following	TF							
Antisubmarine Search	AS							
Close Formation Flying	FF							
Catapult Takeoff	CT							

TABLE XIX (CONT'D) DEFINITION OF AIRPLANE NORMAL STATES

Flight Phase	Symbol	Wing Incidence	Landing Gear	Speed Brakes	Bomb Bay or Cargo Doors	Stability Augmentation	Other
Takeoff	TO						
Climb	CL						
Cruise	CR						
Loiter	LO						
Descent	D						
Emergency Descent	ED						
Emergency Deceleration	DE						
Approach	PA						
Wave-off/ Go-around	WO						
Landing	L						
Air-to-air Combat	CO						
Ground Attack	GA						
Weapon Delivery/ Launch	WD						
Aerial Delivery	AD						
Aerial Recovery	AR						
Reconnaissance	RC						
Refuel Receiver	RR						
Refuel Tanker	RT						
Terrain Following	TF						
Antisubmarine Search	AS						
Close Formation Flying	FF						
Catapult Takeoff	CT						

4.3 Design and test conditions

4.3.1 Altitudes. For terminal Flight Phases, it will normally suffice to examine the selected Airplane States at only one altitude below 10,000 feet (low altitude). For nonterminal Flight Phases, it will normally suffice to examine the selected Airplane States at one altitude below 10,000 feet or at the lowest operational altitude (low altitude), the maximum operational altitude ($h_{o_{max}}$), and one intermediate altitude. When the maximum operational altitude is above 40,000 feet or when stability or control characteristics vary rapidly with altitude, more intermediate altitudes than specified in Table XVIII shall be investigated. When the Service Flight Envelope extends far above or below the Operational Flight Envelope, the service-altitude extremes must be considered.

4.3.2 Special conditions. In addition to the flight conditions previously indicated, the speed-altitude combinations that result in the following shall be investigated, where applicable:

a. Maximum normal acceleration response per degree of controller deflection

b. Maximum normal acceleration response per pound of control force

c. Highest dynamic pressure and highest Mach Number.

4.4 Tests at specialized facilities. Certain tests, by their nature can be conducted only at specialized facilities which are not accessible to either the procuring activity or the contractor except at the option of a third organization. In such cases, when an agreement of test support at the specialized facility is obtained by the procuring activity, an analysis of results obtained in the tests, is a necessary part of the analytical compliance demonstration.

5. PREPARATION FOR DELIVERY

5.1 Section 5 is not applicable to this specification.

6. NOTES

6.1 Intended use. This specification contains the flying qualities requirements for piloted airplanes and forms one of the bases for determination by the procuring activity of airplane acceptability. The specification consists of design requirements in terms of criteria for use in stability and control calculations, analysis of wind tunnel test results, simulator evaluations, flight testing, etc. The requirements should be met as far as possible by providing an inherently good basic airframe. Cost, performance, reliability, maintenance, etc. trade-offs are necessary in determining the proper balance between airframe characteristics and augmented dynamic response characteristics. The contractor should advise the procuring activity of any significant design penalties which may result from meeting any particular requirement.

6.2 Definitions. Terms and symbols used throughout this specification are defined as follows:

6.2.1 General.

S	wing area
s	Laplace operator
$\bar{q}$	dynamic pressure
MSL	mean sea level
T_2	time to double amplitude: $T_2 = -0.693/(\omega_n$ for oscillations; $T_2 = 0.693\tau_s$ or τ for first order divergences
Airplane Normal States	nomenclature and format of Table XIX shall be used in defining the Airplane Normal States (3.1.6.1)
Service ceiling	altitude at a given airspeed at which the rate of climb is 100 ft/min at the stated weight and engine thrust
Combat ceiling	altitude at a given airspeed at which the rate of climb is 500 ft/min at the stated weight and engine thrust
Cruising ceiling	altitude at a given airspeed at which the rate of climb is 300 ft/min at MRT at the stated weight
h_{max}	maximum service altitude (defined in 3.1.8.3)
$h_{o_{max}}$	maximum operational altitude (3.1.7)
$h_{o_{min}}$	minimum operational altitude (3.1.7)
c.g.	airplane center of gravity

6.2.2 Speeds.

Equivalent airspeed, EAS	true airspeed multiplied by $(\sigma)^{1/2}$, where σ is the ratio of free-stream density at the given altitude to standard sealevel air density
Calibrated airspeed	airspeed-indicator reading corrected for position and instrument error but NOT for compressibility
Refusal speed	the maximum speed to which the airplane can accelerate and then stop in the available runway length

M	Mach number
V	airspeed along the flight path (where appropriate, may be replaced by M in this specification)
V_s	stall speed (equivalent airspeed), at 1g normal to the flight path, defined as the highest of:

a) speed for steady straight flight at $C_{L_{max}}$, the first local maximum of the curve of lift coefficient ($L/\bar{q}S$) versus angle of attack which occurs as C_L is increased from zero

b) speed at which uncommanded pitching, rolling or yawing occurs (3.4.1.1.1)

c) speed at which intolerable buffet or structural vibration is encountered

<u>Conditions for determining V_s.</u>

The airplane shall be initially trimmed at approximately $1.2V_s$ with the following settings, after which the trim and and throttle settings shall be held constant:

Flight Phase	Thrust Settings*	Trim Setting
Climb (CL)	Normal climb	For straight flight
Descent (D)	Normal Descent	For straight flight
Emergency descent (ED)	Idle	For straight flight
Emergency deceleration (DE)	Idle	For straight flight
Takeoff (TO)	Takeoff	Recommended takeoff setting
Approach (PA)	Normal approach	For normal approach
Wave-off/ Go-around (WO)	Takeoff	For normal approach
Landing (L)	Idle	For normal approach
All other	TLP at $1.2V_s$	For straight flight

* Either on all engines or on remaining engines with critical engine inoperative whichever yields the higher value of V_s.

In flight test, it is necessary to reduce speed very slowly (typically 0.5 knot per second or less) to minimize dynamic lift effects. The load factor will generally not be exactly 1g when stall occurs, when this is the case, V_s is defined as follows:

$$V_s = V/(n_f)^{1/2}$$

where V and n_f are the measured values at stall, n_f being the load factor normal to the flight path.

$V_s(X)$, $V_{min}(X)$, $V_{max}(X)$	short-hand notation for the speeds V_s, V_{min}, V_{max}, for a given configuration, weight, c.g. position, and external store combination associated with some Flight Phase X. For example, the designation $V_{max}(TO)$ is used in 3.3.3.3.2 to emphasize that the speed intended (for the weight, center of gravity, and external store combination under consideration) is V_{max} for the configuration associated with the takeoff Flight Phase. This is necessary to avoid confusion, since the configuration and the Flight Phase change from takeoff to climb during the maneuver.
V_{con}	speed below which control is lost
V_{trim}	trim speed
V_{end}	speed for maximum endurance
$V_{L/D}$	speed for maximum lift-to-drag ratio
$V_{R/C}$	speed for maximum rate of climb
V_{range}	speed for maximum range with zero wind
V_{NRT}	high speed, level flight, normal rated thrust
V_{MRT}	high speed, level flight, military rated thrust
V_{MAT}	high speed, level flight, maximum augmented thrust
V_{max}	maximum service speed (see 3.1.8.1)
V_{min}	minimum service speed (see 3.1.8.2)

$V_{o_{max}}$	maximum operational speed (see 3.1.7)
$V_{o_{min}}$	minimum operational speed (see 3.1.7)
V_g	gust penetration speed

6.3.3 Thrust and power.

Thrust and power	for propeller-driven airplanes the word 'thrust' shall be replaced by the word 'power' throughout this specification
TLF	thrust for level flight
NRT	normal rated thrust, which is the maximum thrust at which the engine can operate continuously
MRT	military rated thrust, which is the maximum thrust at which the engine can be operated for a specified period
MAT	maximum augmented thrust: the maximum thrust, augmented by all means available for the Flight Phase
Takeoff thrust	maximum thrust available for takeoff

6.3.4 Control parameters.

Pitch, roll, yaw controls	the stick or wheel and pedals manipulated by the pilot to produce pitching, rolling and yawing moments respectively, the cockpit controls
Pitch control force, F_S	component of applied force, exerted by the pilot on the cockpit control, in or parallel to the plane of symmetry, acting at the center of the stick grip or wheel in a direction perpendicular to a line between the center of the stick grip and the stick or control column pivot
Roll control force	for a stick control, the component of control force exerted by the pilot in a plane perpendicular to the plane of symmetry, acting at the center of the stick grip in a direction perpendicular to a line between the center of the stick grip and the stick pivot. For a wheel control, the total moment applied by the pilot applied about the wheel axis in the plane of the wheel, divided by the average radius from the wheel pivot to the pilot's grip
Yaw-control pedal force	difference of push-force components of forces exerted by the pilot on the yaw-control pedals, lying in planes parallel to the plane of symmetry, measured perpendicular to the pedals at the normal point of application of the pilot's instep on the respective yaw-control pedals
Direct normal force control	a device producing direct normal force for the primary purpose of controlling the flight path of the airplane. Direct normal force control is the descriptive title given to the concept of directly modulating the normal force on the airplane by changing its lifting capabilities at a constant angle of attack and constant airspeed or by controlling the normal force component of such items as jet exhausts, propellers, and fans
Control power	effectiveness of control surfaces in applying forces or moments to an airplane. For example, 30 percent of available roll control power is 30 percent of the maximum rolling moment that is available to the pilot with allowable roll control force

6.3.5 Longitudinal parameters.

ζ_{sp}	damping ratio of the short-period oscillation
$\omega_{n_{sp}}$	undamped natural frequency of the short-period oscillation
ζ_p	damping ratio of the phugoid oscillation
ω_{n_p}	undamped natural frequency of the phugoid oscillation
n	normal acceleration or normal load factor, measured at the c.g.
n_L	symmetrical flight limit load factor for a given Airplane Normal State, based on structural considerations
n_{max}, n_{min}	maximum and minimum service load factors
n(+), n(-)	for a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Service Flight Envelope
$n_{o_{max}}$, $n_{o_{min}}$	maximum and minimum operational load factors

$n_o(+)$, $n_o(-)$ for a given altitude, the upper and lower boundaries of n in the V-n diagrams depicting the Operational Flight Envelope

α angle of attack, the angle in the plane of symmetry between the fuselage reference line and the line tangent to the flight path at the airplane c.g.

α_S the stall angle of attack at constant speed for the configuration, weight, c.g. position and external store combination associated with a given Airplane Normal State, defined as the lowest of the following:

a) Angle of attack for the highest steady load factor, normal to the flight path, that can be attained at a given speed or Mach number

b) Angle of attack, for a given speed or Mach number, at which uncommanded pitching, rolling or yawing occurs (3.4.3.1.2)

c) Angle of attack, for a given speed or Mach number, at which intolerable buffeting is encountered.

$C_{L_{stall}}$ lift coefficient at α_S defined above

n/α the steady-state normal acceleration change per unit change in angle of attack for an incremental pitch control deflection at constant speed (airspeed and Mach number)

F_S/n gradient of steady-state pitch control force versus n at constant speed (see 3.1.1.2.1)

γ climb angle, positive for climbing flight:

$\gamma = \sin^{-1}$(vertical speed/true airspeed)

θ pitch attitude, the angle between the fuselage reference line (body x-axis) and the horizontal

L aerodynamic lift plus thrust component normal to the flight path

6.2.6 <u>Lateral-directional parameters.</u>

δ_{AS} displacement of the roll control stick or wheel along its path

T_r first-order roll mode time constant, positive for stable mode

T_s first-order spiral mode time constant, positive for stable mode

λ_r $(-1/T_r)$

λ_s $(-1/T_s)$

ω_ϕ undamped natural frequency of numerator quadratic of ϕ/δ_{AS} transfer function

ζ_ϕ damping ratio of numerator quadratic of ϕ/δ_{AS} transfer function

ω_{n_d} undamped natural frequency of the Dutch roll oscillation

ζ_d damping ratio of the Dutch roll oscillation

T_d damped period of the Dutch roll:

$T_d = 2\pi/(\omega_{n_d}(1 - \zeta_d^2)^{1/2})$

$\omega_{n_{rs}}$ undamped natural frequency of a coupled roll-spiral oscillation

ζ_{rs} — damping ratio of a coupled roll-spiral oscillation

ϕ — bank angle measured in the y-z plane, between the y-axis and the horizontal (6.3.1)

ϕ_t — bank angle change in time t, in response to control deflection of the form given in 3.3.4

p — roll rate about the x-axis (6.3.1)

p_{osc}/p_{av} — a measure of the ratio of the oscillatory component of roll rate to the average component of roll rate following a yaw-control-free step roll control command:

For $\zeta_d < 0.2$: $p_{osc}/p_{av} = (p_1 + p_3 - 2p_2)/(p_1 + p_3 + 2p_2)$

For $\zeta_d > 0.2$: $p_{osc}/p_{av} = (p_1 - p_2)/(p_1 + p_2)$

where p_1, p_2, and p_3 are roll rates at the first, second and third peaks respectively. (Figures B14 and B15)

ϕ_{osc}/ϕ_{av} — a measure of the ratio of the oscillatory component of bank angle to the average component of bank angle following a pedals-free impulse aileron command:

For $\zeta_d < 0.2$: $\phi_{osc}/\phi_{av} = (\phi_1 + \phi_3 - 2\phi_2)/(\phi_1 + \phi_3 + 2\phi_2)$

For $\zeta_d > 0.2$: $\phi_{osc}/\phi_{av} = (\phi_1 - \phi_2)/(\phi_1 + \phi_2)$

where: ϕ_1, ϕ_2 and ϕ_3 are bank angles at the first, second and third peaks respectively. (Figures B14 and B15)

β — sideslip angle at the center of gravity, angle between undisturbed flow and plane of symmetry. Positive, or right sideslip corresponds to incident flow approaching from the right side of the plane of symmetry.

$\Delta\beta$ — maximum change in sideslip occurring within 3 seconds or one half-period of the Dutch roll, whichever is greater, for a step roll-control command. (Figures B14 and B 15)

k — ratio of 'command roll performance' to 'applicable roll performance requirement' of 3.3.4 or 3.3.4.1, with:

$k = (\phi_t)_{command}/(\phi_t)_{requirement}$

where:

a) 'Applicable roll performance requirement', $(\phi_t)_{requirement}$, is determined from 3.3.4 and 3.3.4.1 for the Class, Flight Phase Category and Level under consideration

b) 'Commanded roll performance', $(\phi_t)_{command}$, is the bank angle attained in the stated time for a given step roll command with yaw control pedals employed as specified in 3.3.4 and 3.3.4.1

t_{n_β} — time for the Dutch roll oscillation in the sideslip response to reach the nth local maximum for a right step or pulse roll-control command, or the nth local minimum for a left command. In the event a step control input cannot be accomplished, the control shall be moved as abruptly as practical and, for purposes of this definition, time shall be measured from the instant the cockpit control deflection passes through half the amplitude of the commanded value. For pulse inputs, time shall be measured from a point halfway through the duration of the pulse.

ψ_β — phase angle expressed as a lag for a cosine representation of the Dutch roll oscillation in sideslip, where:

$\psi_\beta = -(360/T_d)t_{n_\beta} + (n - 1)360$ deg.

with n as in t_n above.

$\angle(p/\beta)$ — phase angle between roll rate and sideslip in the free Dutch roll oscillation. This phase angle is positive when p leads β by an angle between 0 and 180 degrees.

$|\phi/\beta|_d$ — at any instant, the ratio of amplitudes of the bank-angle and sideslip-angle envelopes in the Dutch roll mode

Examples showing measurement of roll-sideslip coupling parameters are shown on Figure B14 for right rolls and Figure B15 for left rolls. Since several oscillations of the Dutch roll are required to measure these parameters, and since for proper identification large roll rates and bank angle changes must generally be avoided, step roll control inputs should be small. It should be noted that since Ψ_β is the phase angle for the Dutch roll component of sideslip, care must be taken to select a peak far enough downstream that the position of the peak is not influenced by the roll mode. In practice, peaks occurring one or two roll mode time constants after the aileron input will be relatively undistorted. Care must also be taken when there is ramping of the sideslip trace, since ramping will displace the position of a peak of the trace from the corresponding peak of the Dutch roll component. In practice, the peaks of the Dutch roll component of sideslip are located by first drawing a line through the ramping portion of the sideslip trace and then noting the times at which the vertical distance between the line and the sideslip trace is the greatest. (See the following sketch for Case (a) of Figures B14 and B15; P. 839.)

Since the first local maximum of the Dutch roll component of the sideslip response occurs at t = 2.95 seconds,

$$\Psi_\beta = -(360/T_d)t_{n_\beta} + (n - 1)360 =$$

$$= -(360/3.5)(2.95) = -303 \text{ deg.}$$

Level 1 flying qualities of a Class IV airplane in the approach are under examination, so the roll performance requirement from Table IX upon which the parameter 'k' in the sideslip excursion requirement (Figure B6) is based, is ϕ_t = 30 degrees in 1 second with rudder pedals free (as in the rolls of 3.3.2.4). From the definitions, 'k' for this condition is:

$$k = (\phi_1)_{command}/(\phi_1)_{requirement}$$

Therefore from Figures B14 and B15:

Case (a), k = 9.1/30 = 0.30

Case (b), k = 8.1/30 = 0.27

Case (c), k = 4.3/30 = 0.13

Case (d), k = 4.0/30 = 0.10

6.3.7 <u>Atmospheric disturbance parameters</u>.

j	$(-1)^{1/2}$
Ω	spatial (reduced) frequency (radians per foot)
ω	temporal frequency (radians per sec), where: $\omega = \Omega V$
t	time (seconds)
u_g	disturbance velocity along the x-axis, positive forward (ft/sec)
v_g	disturbance velocity along the y-axis, positive to the pilot's right (ft/sec)
w_g	disturbance velocity along the z-axis, positive down (ft/sec)
	Note: Random u_g, v_g and w_g have Gaussian (normal) distributions
$V_{w/d}$	magnitude of wind over the aircraft carrier deck (ft/sec)
σ, RMS	root-mean-square disturbance intensity, where: $\sigma^2 = \int_0^\infty \Phi(\Omega)d\Omega = \int_0^\infty \Phi(\omega)d\omega$
σ_u	root-mean-square intensity of u_g
σ_v	root-mean-square intensity of v_g
σ_w	root-mean-square intensity of w_g
L_u	scale for u_g (feet)
L_v	scale for v_g (feet)
L_w	scale for w_g (feet)
$\Phi_{u_g}(\Omega)$	spectrum for u_g, where: $\Phi_{u_g}(\Omega) = V\Phi_{u_g}(\omega)$
$\Phi_{v_g}(\Omega)$	spectrum for v_g, where: $\Phi_{v_g}(\Omega) = V\Phi_{v_g}(\omega)$
$\Phi_{w_g}(\Omega)$	spectrum for w_g, where: $\Phi_{w_g}(\Omega) = V\Phi_{w_g}(\omega)$
v_m	generalized discrete gust intensity, positive along the positive axes, m = x, y, z (ft/sec)
d_m	generalized discrete gust length (always positive), m = x, y, z (feet)
w_{30}	wind speed at 30 feet above the ground

$$p/\delta_A\Big|_{STEP} = \frac{K(s^2+2\zeta_\phi\omega_{n\phi}s+\omega_{n\phi}^2)}{(s-\lambda_R)(s-\lambda_s)(s^2+2\zeta_d\omega_{nd}s+\omega_{nd}^2)}$$

ROLLRATE TO AILERON TRANSFER FUNCTION

Figure B14 Roll-Sideslip Coupling Parameters: Right Rolls

Figure B14 (Cont'd) Roll-Sideslip Coupling Parameters: Right Rolls

Figure B15 Roll-Sideslip Coupling Parameters: Left Rolls

Figure B15 (Cont'd) Roll-Sideslip Coupling Parameters: Left Rolls

SKETCH CORRESPONDING TO COMMENT ON FIGS B14 AND B15 ON P. 335, LEFT

x distance from airplane to ship center of pitch, negative aft of ship (feet)

V_w mean wind direction relative to the runway (3.7.3.3)

6.2.8 Terms used in high angle of attack requirements.

Post-stall: The flight regime involving angles of attack greater than nominal stall angles of attack. The airplane characteristics in the post-stall regime may consist of three more or less distinct consecutive types of airplane motion following departure from controlled flight: post-stall gyration, incipient spin, and developed spin.

Post-stall gyration (PSG): Uncontrolled motions about one or more airplane axes following departure from controlled flight. While this type of airplane motion involves angles of attack higher than the stall angle, lower angles may be encountered intermittently in the course of the motion.

Spin: That part of post-stall airplane motion which is characterized by a sustained yaw rotation. The spin may be erect or inverted, flat (high angle of attack) or steep (low, but still stalled angle of attack) and the rotary motions may have oscillations in pitch, roll and yaw superimposed on them. The incipient spin is the initial, transient phase of the motion during which it is not possible to identify the spin mode, usually followed by the developed spin, the phase during which it is possible to identify the spin mode.

6.3 Interpretation of F_s/n limits of Table V. Because the limits on F_s/n are a function of both n_L and n/α, Table V is rather complex. To illustrate its use, the limits are presented on Figure B16 for an airplane having a center-stick controller and $n_L = 7.0$.

6.4 Gain scheduling. Changes of mechanical gearings and stability augmentation gains in the primary flight control system are sometimes accomplished by scheduling the changes as a function of the settings of secondary control devices, such as flaps or wing sweep. This practice is generally acceptable, but gearings and gains normally should not be scheduled as a function of trim control settings since pilots do not always keep airplanes in trim.

6.5 Engine considerations. Secondary effects of engine operation may have an important bearing on flying qualities and should not be overlooked in design. These considerations are:

a) the influence of engine gyroscopic moments on airframe dynamic motions

Figure B16 Example of Pitch Maneuvering Force Gradient Limits: Center-stick Controller, $n_L = 7.0$

b) the effects of engine operation (including flameout and intentional shutdown) on characteristics of flight at high angle of attack (3.4.2)

c) the reduction at low rpm of engine-derived power for operating the flight control system.

6.6 Effects of aeroelasticity, control equipment and structural dynamics. Since aeroelasticity, control equipment and structural dynamics may exert an important influence on the airplane flying qualities, such effects should not be overlooked in calculations or analyses directed toward investigation of compliance with the requirements of this specification.

6.7 Application of levels. Part of the intent of 3.1.10 is to ensure that the probability of encountering significantly degraded flying qualities because of component

or subsystem failure is small. For example, the probability of encountering very degraded flying qualities (Level 3) must be less than specified values per flight.

4.7.1 Level definitions. To determine the degradation in flying qualities parameters for a given Airplane Failure State the following definitions are provided:

a. Level 1 is better than or equal to the Level 1 boundary, or number, specified in Section 3

b. Level 2 is worse than Level 1, but no worse than the Level 2 boundary, or number

c. Level 3 is worse than Level 2, but no worse than the Level 3 boundary, or number.

When a given boundary, or number, is identified simultaneously as Level 1 and Level 2, this means that the flying qualities outside the boundary conditions shown, or worse than the number given, are at best Level 3 flying qualities. Also, since Level 1 and Level 2 requirements are the same, flying qualities must be within this common boundary, or number, in both the Operational and Service Flight Envelopes for Airplane Normal States (3.1.10.1). Airplane Failure States that do not degrade flying qualities beyond this common boundary are not considered in meeting the requirements of 3.1.10.3. Airplane Failure States that represent degradations to Level 3 must, however, be included in the computation of the probability of encountering Level 3 degradations in both the Operational and Service Flight Envelopes. Again, degradation beyond the Level 3 boundary is not permitted regardless of component failures.

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COURTESY: SAAB

APPENDIX C: THE AIRWORTHINESS CODE AND THE RELATIONSHIP BETWEEN FAILURE STATES, LEVELS OF PERFORMANCE AND LEVELS OF FLYING QUALITIES

The purpose of this appendix is to provide airplane designers with a link between the airworthiness regulations, the failure of flight crucial systems and the required level of performance and flying qualities.

<u>LEMMA:</u> It is NOT possible to design an airplane so that the probability of catastrophic failure is zero.

For the airplane designer, the consequence of this Lemma is that:

1. The airplane and all its systems must be designed so that catastrophic failures do not occur above some agreed upon (low) level of probability.

2. The consequences of each flight crucial failure must be understood and the probability of its occurrence must be predicted.

The civil airworthiness code of Figure C1 reflects this design philosophy in a qualitative and quantitative manner. The airworthiness code of Figure C1 reflects the intent of the civil regulations of Appendix A.

The military airworthiness code of Figure C2 reflects this design philosophy in a qualitative and quantitative manner. The airworthiness code of Figure C2 reflects the intent of the military regulations as presented in Appendix B.

There is <u>no formal</u> linkage between the civil and the military airworthiness requirements. However, Figure C2 when compared to Figure C1 infers such a linkage. This inference was made by the author!

The airworthiness codes of Figures C1 and C2 depict the 'level of airplane capability' left, following some failure. The following definitions of capability levels are suggested by the author. They are based on those stated in Appendix B, Section B3:

Level 1: Characteristics clearly adequate for the conduct of the flight.

Figure C1 The Civil Airworthiness Code

Figure C2 The Military Airworthiness Code

Level 2: Characteristics adequate to conduct the flight, but some increase in pilot workload or some degradation in mission effectiveness, or both, exists.

Level 3: Characteristics such that the flight can be continued and a safe landing conducted, but pilot workload is excessive or mission effectiveness is inadequate, or both.

Note: for flight characteristics worse than Level 3 a catastrophic crash is likely to occur.

The word 'characteristics' as used here implies: performance capabilities and/or flying qualities. Therefore, from a preliminary design viewpoint, two failure aspects are important:

C1: Failures Affecting Performance
C2: Failures Affecting Stability and Control (i.e. flying qualities to the military and flight characteristics to the civilians)

C1: FAILURES AFFECTING PERFORMANCE

The civil and military performance regulations of Appendices A and B cover three failure situations:

Failure Situation:	Level Definition
1. All engines operating (AEO):	Level 1
2. One engine inoperative (OEI):	Level 2
3. Two or more engines inoperative:	Level 3

The Level association given above is suggested by the author: it is NOT found in any of the regulations.

C2 STABILITY AND CONTROL

The civil regulations for stability and control behavior (flight characteristics) as stated in Appendix A apply to the required characteristics with ALL systems functioning normally.

The military regulations for stability and control (flying qualities) as stated in Appendix B allow for three Levels of flying qualities. The definition of these Levels is given on page 286. Each Level of flying qualities is associated with some failure in the flight control system AND with a probability of occurrence: see

Table II on page 290. Appendix B provides numerical values for all pertinent flying qualities at each flying quality Level and for each phase of flight.

The civil regulations do not formally recognize a link between failures and degradation (Level) in flying qualities. However, industry design practice is to use the military flying quality Levels in design, but with failure probabilities which are typically a factor 100 times better than those allowed by the military.

As seen in Figures C1 and C2, the relative level of failure probability in civil airplanes is a factor 100 lower than that for military airplanes.

With regard to flight control system design, it is suggested to consider the following two extremes:

TYPE 1: Airplanes with reversible control systems and inherent stability.

In these airplanes the assumption should be made that all control system components are purely mechanical so that their probability of failure is equal to the probability of failure in the primary airplane structure. Such airplanes should be designed to Level 1 flying qualities when intended for the military: see Appendix B.

For civil airplanes the regulations of Appendix A should be used where possible. In cases where the civil regulations fail to provide numerical design guidelines, the pertinent military regulation of Appendix B should be used.

TYPE 2: Airplanes with irreversible control systems and inherent instability.

With all systems operating, civil and military airplanes should be designed according to the Type 1 philosophy stated before.

With one or more failures, <u>civil and military</u> airplanes should be designed to the appropriate handling quality levels as stated in Appendix B. The failure probability levels associated with civil airplanes should be at least a factor 100 lower than that associated with military airplanes.

For airplanes which fit in between Types 1 and 2, the design philosophy for Type 2 airplanes should be used.

COURTESY : SAAB

APPENDIX D: INERTIA TRANSFORMATIONS

The purpose of this appendix is to provide equations for the transformation of airplane moments and product of inertia from one body fixed axis system to another. The two axis systems are shown in Figure D1. Note that both are centered on the airplane center of gravity.

The transformation equations are as follows:

$$\begin{Bmatrix} I_{xx_s} \\ I_{zz_s} \\ I_{xz_s} \end{Bmatrix} = \begin{bmatrix} \cos^2\alpha_1 & \sin^2\alpha_1 & -\sin 2\alpha_1 \\ \sin^2\alpha_1 & \cos^2\alpha_1 & \sin 2\alpha_1 \\ 0.5\sin 2\alpha_1 & -0.5\sin 2\alpha_1 & \cos 2\alpha_1 \end{bmatrix} x \begin{Bmatrix} I_{xx_B} \\ I_{zz_B} \\ I_{xz_B} \end{Bmatrix}$$

The angle α_1 represents the steady state angle of attack of the airplane. The subscript 'B' refers to an arbitrarily selected body-fixed axis system. The subscript 's' refers to the so-called stability axis system (its x-axis is aligned with the steady state velocity vector, U_1 of the airplane) which is ALSO a body-fixed axis system!

Figure D1 Airplane Axis Systems

AIRPLANE DESIGN

PART VIII: AIRPLANE COST ESTIMATION: DESIGN, DEVELOPMENT, MANUFACTURING AND OPERATING

by

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1. GENERAL SYMBOLS

Symbol	Definition	Dimension
a	speed of sound	ft/sec or kts
A	Aspect ratio (wing)	-----
A (alternate)	Regression coefficient in Equation (8.1)	-----
AEP	Airplane Estimated Price	USD
AFP	Airframe price	USD
AH	Number of flight hours for a crew member, per year	hrs/yr
AMP	Airplane Market Price	USD
ANW	Airplane net worth	USD
ASP	Avionics systems price per airplane	USD
ATF	Airplane type factor	-----
b	wing span	ft
B	Regression coefficient in Equation (8.1)	-----
c_f	skin friction coefficient	-----
c_j	Engine specific fuel consumption (jet)	1/hr
c_p	Engine specific fuel consumption (piston or turboprop)	1/hphr
C	See Eqn. (8.3)	-----
C_{aed}	Airframe engineering and design cost	USD
C_{amb}	Applied maintenance burden	USD/nm
C_{apc}	Airplane production cost	USD
C_{apctrl}	Cost of airplane control	USD/nm
C_{aps}	Cost of airplane service	USD/nm

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COURTESY: SAAB

COURTESY: FOKKER

C_{aplf}	Airplane landing fee per landing	USD/landing
C_{apnf}	Navigation fee charged per flight	USD/flight
C_{auc}	Acquisition unit cost	USD/airplane
$C_{avionics}$	Avionics systems cost	USD
C_{cat}	Cost for cabin attendants	USD/nm
C_{crew}	Crew cost	USD/nm
C_{crewpr}	Crew cost for airplane program	USD
C_{dap}	Cost of airframe depreciation	USD/nm
C_{dapsp}	Cost of airplane spare parts depreciation	USD/nm
C_{dav}	Cost of avionics systems depreciation	USD/nm
C_{deng}	Cost of engine depreciation	USD/nm
C_{dengsp}	Cost of engine spare parts depreciation	USD/nm
C_{dgef}	Cost of depreciation of ground facilities and equipment	USD/nm
C_{dprp}	Cost of propeller depreciation	USD/nm
C_{dst}	Development support and test cost	USD
C_{e}	Engine unit cost	USD
$C_{(e + a)}$	Cost of engine and avionics	USD
C_{fin}	Financing cost	USD
C_{frt}	Cost of handling freight	USD/nm
C_{fta}	Cost for flight test airplanes	USD
C_{fto}	Flight test operations cost	USD
C_{ins}	Cost of insurance	USD/nm
C_{int}	Cost of interiors	USD
$C_{lab/ap}$	Labor cost of airframe and systems	USD/nm
$C_{lab/eng}$	Labor cost of engine maintenance	USD/nm

C_{lf}	Direct operating cost due to landing fees	USD/nm
C_{lcuc}	Life cycle unit cost	USD/airplane
C_{man}	Manufacturing labor cost	USD
C_{mat}	Materials cost	USD
$C_{mat/ap}$	Cost of maintenance materials for airplane and systems	USD/nm
$C_{mat/apblhr}$	Cost of airframe and systems maintenance materials	USD/nm
$C_{mat/eng}$	Cost of maintenance materials for engines	USD/nm
$C_{mat/engblhr}$	Cost of engine maintenance materials per block hour	USD/hr
C_{mgef}	Cost of maintaining ground equipment and facilities	USD/nm
C_{mls}	Cost for meal service	USD/nm
$C_{mpersdir}$	Progam cost for direct maintenance personnel	USD
C_{muc}	Manufacturing unit cost	USD/airplane
C_{nf}	Direct operating cost due to navigation fees	USD/nm
$C_{n_{\delta_R}}$	Yawing moment derivative due to rudder deflection	1/rad
$C_{ops_{dir}}$	Direct operating cost for entire airplane program	USD
$C_{ops/hr}$	Operating cost per hour	USD/hr
$C_{ops_{ind}}$	Indirect operating cost for entire airplane program	USD
C_p	Propeller unit cost	USD
$C_{pax/gen}$	General passenger cost	USD/nm
$C_{pax/ins}$	Cost of passenger insurance	USD/nm
C_{pol}	Fuel and oil cost	USD/nm
C_{pro}	Profit (cost)	USD
C_{puc}	Program unit cost	USD/airplane
C_{qc}	Quality control cost	USD
C_{rt}	Direct operating cost due to registry taxes	USD/nm

C_{tool}	Tooling cost	USD
C_{ACQ}	Total cost of acquisition	USD
C_{CONMAT}	Program cost of consumable materials	USD
C_D	Drag coefficient	-----
C_{D_o}	Zero-lift drag coefficient	----
C_{DEPOT}	Program cost associated with depots	USD
C_{DISP}	Total cost of the disposal phase	USD
CEF	Cost Escalation Factor	-----
CEF-ratio	See below Eqn.(2.5) for a definition	-----
C_L	Lift coefficient	-----
$C_{L_{max}}$	Maximum lift coefficient	-----
C_{MAN}	Total cost of the manufacturing phase	USD
C_{MISC}	Program miscellaneous cost	USD
C_{OPS}	Total cost of the operations phase	USD
$C_{PERSDIR}$	Program cost of direct personnel	USD
$C_{PERSIND}$	Program cost of indirect personnel	USD
C_{POL}	Program cost of fuel, oil and lubricants	USD
C_{PRO}	Manufacturer's profit in the manufacturing phase	USD
C_{PROT}	Prototype cost	USD
C_{RDTE}	Total cost of the RDTE phases	USD
C_{SPARES}	Program cost of spares	USD
D	See Eqn.(8.4)	lbs
DOC	Direct Operating Cost	USD/nm
DOC_{depr}	Direct operating cost of depreciation	USD/nm
DOC_{fin}	Direct operating cost of financing	USD/nm
DOC_{flt}	Direct operating cost of flying	USD/nm

DOC_{lnr}	Direct operating cost of landing fees, navigation fees and registry taxes	USD/nm
DOC_{maint}	Direct operating cost of maintenance	USD/nm
DP_{ap}	Airplane depreciation period	yrs
DP_{apsp}	Airplane spare parts depreciation period	yrs
DP_{av}	Avionics systems depreciation period	-----
DP_{engsp}	Engine spare parts depreciation period	yrs
DP_{prp}	Propeller depreciation period	yrs
e	Oswald's efficiency factor	----
E	Endurance	hrs
$EP = C_e$	Engine price	USD
ESPPF	Engine spare parts price factor	-----
$f_{amb/lab}$	overhead distribution factor for labor cost	-----
$f_{amb/mat}$	overhead distribution factor for material cost	-----
f_{depot}	depot cost fraction of C_{OPS}	-----
$f_{ins_{hull}}$	annual hull insurance rate	USD/USD/yr
f_{ioc}	indirect operating cost fraction of DOC	-----
f_{lf}	(landing fee) factor which depends on airplane size	-----
f_{misc}	miscellaneous cost fraction of C_{OPS}	-----
$f_{persdir}$	direct personnel cost fraction of C_{OPS}	-----
$f_{persind}$	indirect personnel cost fraction of C_{OPS}	-----
f_{pol}	fuel, oil and lubricant cost fraction of C_{OPS}	-----
f_{rt}	(registry tax) factor which depends on airplane size	-----

f_{spares}	indirect cost of spares as a fraction of C_{OPS}	-----
F_{apsp}	Airplane spare parts factor	---
F_{cad}	Judgement factor of relative company experience with CAD	
F_{dap}	Airframe depreciation factor	--
F_{dapsp}	Airplane spare parts depreciation factor	-----
F_{dav}	Avionics systems depreciation factor	-----
F_{deng}	Engine depreciation factor	----
F_{dengsp}	Engine spare parts depreciation factor	-----
F_{diff}	Judgement factor of relative program difficulty	-----
F_{dprp}	Propeller depreciation factor	-----
F_{engsp}	Engine spare parts factor	-----
F_{fin}	Finance cost fraction	-----
F_{ftoh}	Overhead factor associated with flight test activities	---
F_{int}	Interior cost factor which depends on number of pax	USD/pax
F_{mat}	Judgement factor to account for differing materials cost	-----
F_{obs}	Judgement factor to account for observables characteristics	
F_{pro}	Profit fraction	-----
F_{tsf}	Cost adjustment factor for extra test and simulation facilities	-----
FD	Fuel density	lbs/gallon
F_{OL}	Factor which accounts for cost of oil and lubricants	-----
FP	Fuel price	USD/gallon
H_{em}	Attained number of hours between engine overhaul	hrs
IOC	Indirect Operating Cost	USD/nm
IOC_{ascf}	Indirect operating cost for airplane and traffic servicing, control and freight	USD/nm

V_C	Design cruising speed	kts
V_s	Stall speed	kts
W	Weight	lbs
W_{ampr}	Aeronautical Manufacturers Planning Report Weight	lbs
W_{crew}	Crew weight	lbs
W_{eng}	Weight per engine	lbs
W_{feq}	Fixed equipment weight	lbs
$W_{ol_{bl}}$	Block oil used	lbs
$W_{pwrplt} = W_{pwr}$	Powerplant weight	lbs
W_{struct}	Airplane structural weight	----
W_A	Airframe weight	lbs
W_E	Airplane empty weight	lbs
$W_{F_{bl}}$	Block fuel used	lbs
$W_{F_{used}}$	Mission fuel used	lbs
W_{PL}	Payload weight	lbs
W/S	Wing loading	lbs/ft^2
W_{TO}	Takeoff Weight	lbs
W_{US}	Useful load	lbs
y_T	Engine-out moment arm	ft

2. Greek Symbols

δ_R	rudder deflection	rad
η_p	propeller efficiency	-----
ρ	air density	$slugs/ft^3$

3. Subscripts

aed	airframe engineering and design
api	airconditioning and pressurization

IOC_{gaa}	Indirect operating cost for general administrative expenses	USD/nm
IOC_{pax}	Indirect operating cost for passenger services	USD/nm
IOC_{pse}	Indirect operating cost for promotion, sales and entertainment	USD/nm
IOC_{sta}	Indirect operating cost for maintaining ground equipment and facilities	USD/nm
K_j	Vacation (etc) pay factor	-----
$K_{H_{em}}$	Attained period between overhaul factor	-----
LCC	Life Cycle Cost	USD (United States Dollars)
L/D	Lift-to-drag ratio	-----
L_R	Annual loss rate in number of airplanes lost per 10^5 hrs	1/hrs
M_1	Steady state Mach number	-----
M_{ff}	Overall mission fuel fraction	-----
M_{res}	Reserve fuel fraction	-----
M_{tfo}	Trapped fuel and oil fraction	-----
MHR_{aed}	Engineering manhours	hrs
$MHR_{aed_{program}}$	Engineering manhours for entire program	hrs
MHR_{flthr}	Total maintenance manhours per flight hour	hrs/hr
MHR_{man}	Manufacturing manhours	hrs
$MHR_{map_{bl}}$	Airframe and systems maintenance manhours per block hour	hrs/hr
$MHR_{map_{flt}}$	Airframe and systems maintenance manhours per flight hour	hrs/hr
$MHR_{man_{program}}$	Manufacturing manhours for entire program	hrs

$MHR_{map_{bl}}$	Maintenance manhours for airframe and systems	hrs/blhr
$MHR_{meng_{bl}}$	Maintenance manhours for engines per block hour	hrs/hr
MHR_{tool}	Tooling manhours	hrs
$MHR_{tool_{program}}$	Tooling manhours for entire program	hrs
n	learning curve exponent	-----
n_{c_j}	number of crew members of type j	-----
N_{acq}	Number of airplanes acquired	--
N_{crew}	Number in crew	-----
N_e	Number of engines per airplane	-----
N_{loss}	Number of airplanes lost through accidents over the airplane service life	-----
N_m	Number of airplanes built to production standards	-----
N_{market}	Estimated number of airplanes in a given market	-----
$N_{mission}$	Number of missions flown per year	-----
N_p	Number of propellers per airplane	-----
N_{pax}	Number of passengers	-----
$N_{program}$	Total number of airplanes built in a program	-----
N_{prot}	Number of prototypes	-----
N_{rdte}	Number of airplanes built in rdte phases	-----
N_r	Number of airplanes built per month	-----
N_{res}	Number of airplanes kept in reserve	-----
N_{serv}	Number of airplanes in active service	-----
N_{st}	Number of static test airframes built	-----
N_{yr}	Number of years an airplane is operated	-----

OD	Oil density	lbs/gallon
OHR_{crew}	Overhead rate factor associated with crew pay	-----
OLP	Oil price	USD/gallon
P	P percent learning curve	-----
Pay_{crew}	Annual pay of crew	USD/yr
P_{pot}	Profit potential	USD
$PP = C_p$	Propeller price	USD
$\bar{q}$	dynamic pressure	lbs/ft^2
R	Range	nm
R_{bl}	Block distance	nm
$R_{bl_{ann}}$	Annual block miles flown	nm/yr
R_{cl}	Climb distance	nm
R_{conmat}	Average cost of consumable materials per maintenance manhour	USD/hr
R_{cr}	Crew ratio	-----
R_{de}	Descent distance	nm
R_e	Engineering manhour rate	USD/hr
$R_{l_{ap}}$	Airplane maintenance labor rate	USD/hr
$R_{l_{eng}}$	Engine maintenance labor rate	USD/hr
R_m	Manufacturing manhour rate	USD/hr
R_{man}	Maneuvering distance	nm
$R_{m_{ml}}$	Military maintenance labor rate (incl. overhead)	USD/hr
R_t	Tooling manhour rate	USD/hr
REV	Revenue per nm	USD/nm
ROI	Return on investment	-----
S	Wing (reference) area	ft^2
S_{wet}	Wetted area	ft^2
SAL_j	Annual salary of crew member of type j	USD/yr

t_{bl}	block time	hrs
t_{cl}	climb time	hrs
t_{cr}	cruise time	hrs
t_{de}	descent time	hrs
t_{flt}	average flight time	hrs
t_{gm}	ground maneuver time	hrs
t_{pft}	Number of flight test hours flown by manufacturer	hrs
t_{man}	maneuvering time	hrs
t_{mis}	average mission time	hrs
tx_{inv}	investment tax rate	-----
tx_{rev}	income (revenue) tax rate	-----
T	Thrust	lbs
TEF	Travel expense factor	-----
TOC	Total operating cost	USD/nm
T_{TO}	Total takeoff thrust	lbs
T/W	Thrust to weight ratio	-----
$U_{ann_{bl}}$	Annual utilization in blockhours	hrs
$U_{ann_{flt}}$	Annual utilization in flight hours	hrs
V	Speed	keas
V_{bl}	Block speed	kts
V_{cr}	Cruise speed	kts
V_{de}	Descent speed (forward)	kts
V_{flt}	Average flight speed	kts
V_{man}	Speed while maneuvering	kts
V_{max}	Maximum design speed	kts
V_{mc}	Minimum control speed	kts

apu	auxiliary power unit
arm	armament and fire control
avionics	avionics
AMPR, ampr	Aeronautical Manufacturers Planning Report (Also called DCPR)
cl	climb
clean	clean airplane: gear and flaps up (retracted)
cr	cruise
C	Design Cruise
CAD	Computer Aided Drafting
dst	development support and testing
DCPR	Defense Contractors Planning Report (Also called AMPR)
DISP	Disposal
e	engine
els	electrical system
ess	engine starting system
E	Empty
fin	finance
fta	flight test airplanes
fto	flight test operations
F	Fuel
H	Maximum Level
iae	instruments, avionics and electronics
j	number which indicates type of crew member, j=1, 2, 3
L	Landing
m	manufacturing phase
mat	materials
max	maximum design
OPS	Operations
pro	profit
program	for entire program
PL	Payload
qc	quality control

r	rdte phase
rdte, RDTE	Research, Development, Test and Evaluation
tfo	trapped fuel and oil
tool	tooling
tsf	test and simulation facilities
TO	Takeoff

4. Acronyms and Abbreviations

AAA	Advanced Aircraft Analysis
ANG	Air National Guard
Ch	Chapter
CL	Climb
CR	Cruise
DESC	Descent
ECM	Electronic Counter Measures
FBO	Fixed Base Operator
IFF	Identification Friend or Foe
keas	knots equivalent airspeed
LCC	Life Cycle Cost
LTR	Loiter
Pt	Part
SLS	Sea level standard
TO	Takeoff
USAF	United States Air Force
USMC	United States Marine Corps
USN	United States Navy

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CLEARANCE
DIHEDRAL
DIAMETER
CLEARANCE
Lateral Control Spoilers
Ground Spoilers
COURTESY OF:
DE HAVILLAND CANADA
★ NOTE: DIMENSIONS ARE APPROXIMATE AND MAY VARY DEPENDING ON AIRCRAFT CONFIGURATION AND LOADING CONDITIONS.
HEIGHT ★
BILL HEIGHT
STATIC GROUND LINE
LENGTH

1. INTRODUCTION

The purpose of this series of textbooks on Airplane Design is to familiarize aerospace engineering students with the design methodology and design decision making involved in the process of designing airplanes. The series of books is organized as follows:

Part I:	PRELIMINARY SIZING OF AIRPLANES
Part II:	PRELIMINARY CONFIGURATION DESIGN AND INTEGRATION OF THE PROPULSION SYSTEM
PART III:	LAYOUT DESIGN OF COCKPIT, FUSELAGE, WING AND EMPENNAGE: CUTAWAYS AND INBOARD PROFILES
PART IV:	LAYOUT DESIGN OF LANDING GEAR AND SYSTEMS
PART V:	COMPONENT WEIGHT ESTIMATION
PART VI:	PRELIMINARY CALCULATION OF AERODYNAMIC, THRUST AND POWER CHARACTERISTICS
PART VII:	DETERMINATION OF STABILITY, CONTROL AND PERFORMANCE CHARACTERISTICS: FAR AND MILITARY REQUIREMENTS
PART VIII:	AIRPLANE COST ESTIMATION: DESIGN, DEVELOPMENT, MANUFACTURING AND OPERATING

In this part (Part VIII) frequent reference is made to other parts in this Airplane Design series: Refs 1-7.

The purpose of Part VII is to present preliminary cost estimating methods for newly designed airplanes. The methods are presented in such a manner that they can be applied to commercial and to military airplanes of all types.

Chapter 2 presents a discussion of cost definitions and concepts. The idea of Life Cycle Cost is defined and its relation to the design decision making process is outlined.

The major cost sources of LCC are treated in some detail in Chapters 3 through 6 in the form of systematic cost estimating procedures and examples for:

* Research, development, technology and evaluation cost in Chapter 3

* Manufacturing and Acquisition cost in Chapter 4

* Operating cost for commercial airplanes in Chapter 5

* Operating cost for military airplanes in Chapter 6

An example of LCC determination for a military airplane is given in Chapter 7.

It is important that students of airplane design understand the role of optimization. Chapter 8 contains some fundamental discussions on this subject. The topic of 'design-to-cost' is also introduced in this chapter. Finally, a list of 88 design guidelines for 'low cost' is included in Chapter 8.

The decision on whether or not to proceed with full scale development of a new design involves a variety of factors. The purpose of Chapter 9 is to give an overview of key factors in airplane program decision making. This includes a list of 19 lessons learned on past programs.

Actual price data for airplanes and airplane components are hard enough to find for practicing engineers. For students this is often a very time consuming problem. To help with the problem of getting 'on track' with price data several appendices are included with specific information on prices:

* Appendix A includes airplane price data

* Appendix B includes engine and propeller price data

* Appendix C includes avionics systems price data

2. COST: DEFINITIONS AND CONCEPTS

The purpose of this chapter is to introduce aerospace engineering students to several important definitions and concepts, all dealing with airplane cost. The material is organized as follows:

2.1 Cost, Price and Profit
2.2 Airplane Program, Life Cycle and Life Cycle Cost
2.3 Examples of Aeronautical Enterprises
2.4 Some Observations on Profit and Cost

2.1 COST, PRICE AND PROFIT

First, it is necessary to define the meaning of the terms: COST, PRICE and PROFIT.

Definition 1: The COST of an airplane is the total amount of expenditure of resources, usually measured in dollars, needed to manufacture that airplane.

Definition 2: The PRICE of an airplane is the amount of dollars paid for the airplane by customers.

Definition 3: Profit = Price - Cost (2.1)

How cost, price and profit are viewed depends on which position is occupied in the economic process. Five examples of different viewpoints are:

Example 1: Position of the airplane manufacturer.

Assume that the cost to produce a certain airplane is USD 100,000. If the selling price is USD 110,000 a profit is made of USD 10,000. In the presence of competition it will be of prime interest to the manufacturer to offer his airplane at the lowest possible price (and still make a profit) while assuring that his airplane will be profitable or affordable to operate by his customer. This means that design, development and manufacturing costs must be kept as low as possible.

Example 2: Position of the commercial airplane operator.

To the commercial airplane operator, the price paid for an airplane is a cost (acquisition cost). Another cost is that incurred to operate the airplane.

The operator hopes to sell enough revenue ton miles (RTM) or revenue passenger miles (RPM) to make the operation of the airplane profitable. To make this possible, the acquisition and the operating cost must be as low as possible.

Example 3: Position of the personal or corporate airplane operator.

To this type of operator the airplane is a tool used to enhance certain aspects of business or to satisfy certain personal needs. Affordability is of prime concern to this type of operator. Affordability usually translates into requirements for low acquisition cost, low operating cost and high resale value.

Government tax policies and tax laws tend to have a significant impact on airplane affordability for this type of operator.

Example 4: Position of the government/military airplane operator.

The government/military airplane operator normally has a limited budget available with which to carry out certain missions. The price paid for an airplane and the cost incurred while operating the airplane all are costs to this operator. What this operator hopes to achieve is to carry out the required missions within the available budget. To make this possible the acquisition cost and the operating cost need to be as low as possible.

Military operators are normally directly involved in the design and development process with the airplane manufacturer. The cost associated with the design and development process can be significant particularly if the production run is small.

Example 5: Position of the consumer/taxpayer.

The consumer/taxpayer is the person who ends up paying for all activities required to design, manufacture, operate and dispose of airplanes. The consumer/taxpayer is ultimately interested in lowering the total cost associated with all aeronautical activities.

At this point it should be observed that all costs mentioned so far are those which can be measured directly

in dollars. Such costs are referred to also as <u>economic costs</u>. There are other types of cost as well: <u>social</u>, <u>psychological</u> and <u>environmental</u>, to name a few. Examples of causes of these costs are:

<u>Social Cost:</u> If a company decides to relocate its operations to another part of the country (or the world) then social costs are incurred because of loss of employment, relocation of families, etc. The burden of carrying the economic costs associated with social costs are normally carried by the taxpayer.

<u>Psychological Cost:</u> Airplanes tend to create a significant amount of noise during takeoff and landing. This can cause distress in people and can be thought of as a psychological cost. The psychological distress can become so severe as to lead to anti-noise legislation. Examples are found in FAR 36 which places the burden of the noise problem on the designer and thus, ultimately on the consumer.

<u>Environmental Cost:</u> Pollution caused by certain airplane operations represents an environmental cost. Examples are: atmospheric pollution due to exhaust gasses, fuel dumping in emergencies and chemical discharges resulting from manufacturing and maintenance processes.

Although these types of cost are important, they are difficult to quantify and will not be considered in this text. Many of these costs are ultimately borne by taxpayers although some of them are borne by the consumer (fare paying passenger) through increased operating cost as in the case of costs incurred by meeting FAR 36.

There is much confusion about the relationship between cost and price. If an airplane manufacturer buys certain equipment for installation in an airplane which is being manufactured, he pays a certain price for that equipment. That price paid then becomes part of his cost to produce that airplane!

What a manufacturer charges in the marketplace for a certain airplane becomes the price charged. That price may or may not have a rational relationship to the actual cost. <u>Example:</u> true or not, it was said that Cessna actually 'lost' money on each Cessna 152 it produced. The reason, true or not, was that it was felt that Cessna 152 customers eventually would 'step-up' to a more expensive airplane model on which a profit was actually made. This is the so-called 'loss-leader' policy which is popular in supermarkets as well!

2.2 AIRPLANE PROGRAM, LIFE CYCLE AND LIFE CYCLE COST

The evolution of an airplane from design to manufacturing, operation and finally, disposal is referred to as an airplane program. A typical airplane program can be divided into the following six phases:

Phase 1: Planning and Conceptual Design.

During this phase the activities depicted in Figure 2.1 take place. Planning here consists primarily of mission requirements research. This eventually leads to a mission specification. Conceptual design here consists of the design activities associated with Preliminary Design Sequence I as outlined in Parts I and II of this text.

Some very preliminary cost studies are also conducted during this phase.

Phase 2: Preliminary Design and System Integration.

During this phase the activities depicted in Figure 2.2 take place. The accompanying design activities are those associated with Preliminary Design Sequence II as outlined in Parts I and II of this text. Design trade studies are conducted to find that combination of technology and cost which might result in a viable airplane program.

Phase 3: Detail Design and Development.

During this phase the airplane and system integration design is finalized for certification flight testing and for production.

Phase 4: Manufacturing and Acquisition.

During this phase the airplane is manufactured and delivered to (acquired by) the customer.

Phase 5: Operation and Support.

During this phase the airplane is being acquired by the user and is being operated with the accompanying support activities.

Note: Phases 4 and 5 normally overlap.

Figure 2.1 Phase 1: Planning and Conceptual Design

Figure 2.2 Phase 2: Preliminary Design and System Integration

Phase 6: Disposal.

This phase marks the end of the operational life of the airplane. Disposal activities may include destruction of the airplane and disposal of the remaining materials. In case of military airplanes it may also include storage (at Davis-Monthan AFB for example). Disposal becomes necessary when an airplane has reached the limit of its technological or economical life.

Note: Phases 5 and 6 normally overlap.

Figure 2.3 shows these six phases with an indication of their relative impact on costs associated with a typical airplane program. Note the following definitions:

Definition 4: The time elapsed during the six phases of an airplane program (Fig.2.3) is called the Airplane Life Cycle.

Definition 5: The total cost of an airplane program incurred during the airplane life cycle is called the Life Cycle Cost.

Note: the vertical scale in Figure 2.3 indicates the cost associated with each program phase as a percentage of life cycle cost: LCC.

For preliminary cost estimating purposes the life cycle cost of an airplane program is broken down into four cost sources:

1. Research, development, test and evaluation cost: C_{RDTE}.

   This cost source accounts for all costs incurred in phases 1, 2 and 3 in Figure 2.3. A method for estimating C_{RDTE} is presented in Chapter 3.

2. Acquistion cost: C_{ACQ}.

   This cost source includes the manufacturing cost: C_{MAN} and the manufacturer's profit: C_{PRO}. Thus:

   $$C_{ACQ} = C_{MAN} + C_{PRO} \qquad (2.2)$$

   These costs are incurred during Phase 4 in Figure 2.3. A method for estimating C_{ACQ} is presented in Chapter 4.

BASED ON AN IDEA IN REF.8

Figure 2.3 Impact of Airplane Program Phases on Life Cycle Cost

3. Operating cost: C_{OPS}.

   This cost source represents the cost incurred while operating the airplane: see Phase 5 in Figure 2.3. The airplane manufacturer and his suppliers (vendors) usually incur certain support costs during this phase. A method for estimating C_{OPS} is given in Chapter 5 for commercial airplanes and in Chapter 6 for military airplanes.

4. Disposal cost: C_{DISP}.

   This is the cost incurred in disposing of the airplane: see Phase 6 in Figure 2.3. Chapter 7 presents an approach to the estimation of C_{DISP}.

The life cycle cost of an airplane program (LCC) can now be expressed as:

$$LCC = C_{RDTE} + C_{ACQ} + C_{OPS} + C_{DISP} \tag{2.3}$$

Chapter 7 presents an example of LCC estimation for a military airplane program. An example for a commercial airplane program is given in Section 5.4.

Figure 2.4 shows example cost histories for two airplane programs. Note that the operating cost source is much larger than the acquisition cost source. The latter in turn is much larger than the research, development, test and evaluation cost source. The cost histories of Figure 2.4 are schematically represented in Figure 2.5. It is seen that:

$$C_{OPS} \gg C_{ACQ} \gg C_{RDTE} \tag{2.4}$$

What emerges from Figures 2.3 - 2.5 is the following important conclusion:

<u>THE CONCEPTUAL AND PRELIMINARY DESIGN PHASES ARE RESPONSIBLE FOR LOCKING IN MOST OF THE LIFE CYCLE COST OF AN AIRPLANE!!</u>

The implication for airplane design engineers and for airplane program managers is that:

<u>SIGNIFICANT LEVERAGE AFFECTING LIFE CYCLE COST EXISTS ONLY IN THE CONCEPTUAL AND PRELIMINARY DESIGN PHASES!!</u>

Figure 2.4a Life Cycle Cost History of the B-52 Program

Figure 2.4b Life Cycle Cost History of the F-111 Program

Figure 2.5 Schematic Representation of Life Cycle Cost History of Typical Airplane Programs

Figure 2.6 The Iceberg Effect in Airplane Program Management

Another way of illustrating these facts is found in Figure 2.6. The airplane program manager is like the captain of a ship navigating close to an iceberg. The program manager has many pressures upon him to lower RDTE costs (the short term outlook). Yielding to these pressures may in fact increase LCC! Performing more RDTE work, when directed at designing to minimize LCC, can result in significant savings to the customer and therefore to society as a whole. It takes courage and wisdom to purposely increase RDTE costs so that LCC will be lower.

During the preliminary airplane design process as described in Parts I through VII many design decisions are made which have a significant effect on airplane life cycle cost. Airplane designers must be aware of these effects if their designs are to be 'cost-effective'.

Step 36 in the preliminary design process (See Part II, page 23) requires the designer to perform a preliminary cost analysis for the new airplane. This in turn means that designers must have available a systematic method for estimating LCC. Chapters 3 through 7 present methods for estimating the major LCC sources of Eqn.2.3.

In addition to having a method for estimating LCC, the designer must have an understanding of how either total LCC or any of its cost sources can be minimized. Several important principles in the areas of DESIGN OPTIMIZATION, DESIGN-TO-COST and DESIGN-TO-PRICE are discussed in Chapter 8.

Finally, it is necessary to have an understanding of various factors (in addition to cost) which may affect an airplane program decision. That subject is taken up in Chapter 9.

Because it is hard for aeronautical engineering students to 'find' hard data on prices, the author has included price data for airplanes, engines, propellers and avionics systems in the form of several appendices. It is hoped that these data (even though they will become obsolete rather rapidly) will be useful to the reader.

2.3 EXAMPLES OF AERONAUTICAL ENTERPRISES

Aeronautical engineering students should be aware of the various types of aeronautical enterprises which exist. After all, these enterprises are most likely to be their future employers.

From an ownership point of view there are three types of aeronautical enterprises:

1. Privately owned.

2. Government owned (country, state, local).

3. Part private and part government owned.

In the following, some selected examples of each are given. There is no implication of importance, relative or absolute, to be attached to these examples. They are merely what came to the author's mind while developing this text. An extensive list of aeronautical enterprises can be found in Reference 9. When consulting Ref.9 the reader will come to the conclusion that aeronautics is very much an international affair! Aeronautics is expected to become even more international in the future.

1. Examples of privately owned aeronautical enterprises are:

Manufacturers of airplanes:

In the USA:

Boeing, McDonnell-Douglas, Lockheed, General Dynamics (including Cessna), Grumman, Raytheon-Beech, Piper and Learjet Corp.

Foreign:

Robin, Grob, Reims-Aviation, Pilatus, Mudry.

Operators of airplanes:

In the USA:

American Airlines, United Airlines, Federal Express, US Air, TWA , Northwest and Southwest Airlines.

Foreign:

British Airways, Martin Air, Cross Air and Tyrolean.

2. Examples of government owned aeronautical enterprises are:

Manufacturers of airplanes:

In the USA: none. But: the U.S. government owns several plants which it leases to private companies.

Foreign:

Fabrica Militar de Aviones, Embraer, CASA, IPTN, Antonov, Ilyushin, PZL, Airbus GmbH and Aeritalia.

Operators of airplanes:

In the USA:

USAF, USNavy, U.S.M.C., US Coast Guard and FAA.

Foreign:

Aeroflot, SAS, Garuda Indonesian and Air France.

3. Examples of part private and part government owned aeronautical enterprises are:

Manufacturers of airplanes:

In the USA: none.

Foreign:

SIAI-Marchetti, Piaggio, Partenavia, Dassault-Breguet, MBB and Fokker.

Operators of airplanes:

In the USA: none.

Foreign:

KLM (Royal Dutch Airlines), Lufthansa, and UTA.

There are many other aeronautical enterprises which supply parts, systems and engines. To save space, these were not included as examples. Nevertheless, these supplier enterprises are important. For an airplane manufacturing industry to be viable, a network of parts and system suppliers (the aeronautics infra-structure) must exist! Airplanes cannot be built without:

* Materials such as: aluminum, steel and titanium alloys, composites, etc.
* Wheels, brakes and tires
* Engines and APU's
* Avionics systems and instruments
* Pumps, lines, sensors, cables and fasteners
* Interiors, including insulating materials

Reference 9 provides an annual listing of airplane operators, manufacturers and suppliers with details about their corporate structure. This reference deals with foreign as well as domestic enterprises.

Last (and not least!) a healthy aeronautics industry requires very long term research capabilities. In most countries such long term research facilities are owned by the government. NASA in the USA is an example.

2.4 SOME OBSERVATIONS ON PROFIT AND COST

In general, the objective of a privately owned aeronautical enterprise is to make a profit. The profit made before taxes is sometimes referred to as the operating margin. Examples of operating margins for several aeronautical enterprises is presented in Table 2.1. Since the tax situation of a company can vary from year to year and from country to country, the operating margin of a company is not the same as its profit. In day-to-day use of these terms they are frequently interchanged.

In the case of military, government owned enterprises, profit is not an objective. Instead, getting the 'best' defense or offense per dollar paid ('biggest bang per buck') is usually the objective.

In the case of civilian, government owned enterprises, profit may not be an immediate objective. Significant participation in some market segment to generate jobs and general prosperity is an important objective of many government owned enterprises.

In the case of part private and part government owned aeronautical enterprises the objective is usually to make a profit albeit not necessarily each year.

Another objective which can play a role in privately owned enterprises is: to 'have fun at building airplanes and not lose money'.

It is expected that the manner in which an aeronautical enterprise is managed will depend on its objective.

Table 2.1 Operating Margins in Percentage of Sales for the Period 1985 - 1988

Aerospace Manufacturers	1985	1986	1987	1988
Boeing Commercial	4.7	4.2	3.6	5.1
Boeing Military	8.0	7.5	1.5	-2.6
Chrysler (Gulfstream)	N/A	9.0	12.9	11.7
General Dynamics Convair	13.3	12.5	10.1	9.3
General Dynamics Cessna	N/A	-82.5	1.7	5.5
General Electric Engines	14.3	14.5	13.9	15.4
G.M. (Hughes)	N/A	7.8	8.6	8.8
Grumman Aerospace	4.4	3.3	1.6	5.3
Honeywell/Sperry	3.4	6.1	6.3	-7.7
Kaman	7.9	7.8	8.2	8.1
Learjet	N/A	N/A	0.0	0.0
Litton Adv. Electronics	11.3	5.7	8.0	7.8
Lockheed Aeronautical	10.6	10.4	8.1	6.3
Loral (Goodyear)	N/A	20.0	18.7	N/A
LTV Aircraft Products	12.3	8.8	9.4	3.4
McDonnell Douglas				
Combat Aircraft	8.7	6.7	5.7	6.6
Transport Aircraft	3.7	3.0	2.7	2.6
Moog (Hydr. Actuators)	15.2	13.8	11.4	-1.0
Northrop Aircraft	9.0	1.6	4.5	1.9
Northrop Electronics	9.0	8.0	6.2	3.6
Raytheon Beech	-2.3	1.4	-1.3	3.0
Rockwell North American	9.3	10.3	13.8	12.4
Teledyne (aviation, electronics)	8.6	6.9	10.2	9.4
United Tech (P and W)	7.6	6.1	12.0	10.8
U.T. (Sikorsky, Hamilton)	9.8	-8.8	3.0	-0.4

Data from: Flight International, June 10, 1989

Airlines	1985	1986	1987	1988
American Airlines	8.7	6.8	6.4	9.1
USAir	9.5	9.2	10.6	7.6
United Airlines	-4.5*	1.3	3.0	7.4
Delta Airlines	7.8	0.8	7.6	7.2
Northwest Airlines	2.9	4.6	3.8	3.6

*Strike

Data from: Aviation Week and Space Techn., June 19, 1989

NOTE: Operating margin is the profit before taxes.

When profit is the primary objective, the temptation is for management decisions to focus on the short term: 'maximizing profit this quarter'. This can lead to the iceberg disaster depicted in Figure 2.6.

When profit is not the immediate objective, management decision making tends to focus on the long term.

The time required for development and certification of an airplane does affect management decision making. Because of the long time it takes to develop and certify airplanes, management decision making in the airplane industry should focus on the long term.

Whatever management objectives are, the cost of developing, certifying, producing and operating an airplane must be known with some certainty before a decision to 'launch' an airplane program is made.

Because airplane programs take many years to evolve through the phases shown in Figure 2.3, inflation (= cost escalation) plays an important role in estimating program cost. So does interest on investment: the cost of money.

Estimates for cost magnitudes are usually given in 'then-year' dollars. It is customary to scale cost data from one 'then-year' to another with a cost escalation factor: CEF. Figure 2.7 shows how this CEF has varied with calendar time since 1965. In Figure 2.7 the CEF has been arbitrarily normalized to a value of 1.0 for 1970. Cost from one year to another may be 'scaled' as follows:

$$Cost_{19XX} = Cost_{19YY}(CEF_{19XX}/CEF_{19YY}) \qquad (2.5)$$

where: the ratio (CEF_{19XX}/CEF_{19YY}) is the CEF-ratio.

The reader is referred to References 8 and 10 for a detailed discussion and development of the ideas of airplane program cost. Chapters 3 through 7 present rapid methods for estimating airplane program costs. These methods are aimed strictly at Phases 1 and 2 as depicted in Figure 2.3. They should be used for preliminary cost estimating purposes only.

Figure 2.7 Variation of Cost Escalation Factor with Time

3. METHOD FOR ESTIMATING RESEARCH, DEVELOPMENT, TEST AND EVALUATION COST AND A METHOD FOR ESTIMATING PROTOTYPING COST

The purpose of this chapter is to present:

1) A method for estimating research, development, test and evaluation cost (C_{RDTE}) for airplanes.

2) A method for estimating prototyping cost (C_{PROT}) for airplanes.

The RDTE cost, C_{RDTE}, is the first cost source seen in Eqn.(2.3). RDTE cost is accumulated during Phases 1-3 in Figure 2.3. Phases 1-3 involve those activities which take a new airplane all the way from the planning and conceptual design stage to certification. This applies not only to military, but also to commercial airplanes. Phases 1-3 normally include the design, construction, ground and flight testing of a number of static and flight test airplanes. Figure 2.3 shows how the RDTE activities of Phases 1-3 fit into an airplane program.

The method for estimating RDTE cost is based on that of Ref.11. It should be used for preliminary cost estimation purposes only. The method can be applied to military as well as to commercial airplane programs.

RDTE costs are normally broken down into seven cost categories:

3.1 Airframe Engineering and Design Cost: C_{aed_r}

3.2 Development Support and Testing Cost: C_{dst_r}

3.3 Flight Test Airplanes Cost: C_{fta_r}

3.4 Flight Test Operations Cost: C_{fto_r}

3.5 Test and Simulation Facilities Cost: C_{tsf_r}

3.6 RDTE Profit: C_{pro_r}

3.7 Cost to finance the RDTE phases: C_{fin_r}

The total RDTE cost for a new airplane program may be estimated from:

$$C_{RDTE} = C_{aed_r} + C_{dst_r} + C_{fta_r} + C_{fto_r} + C_{tsf_r} + C_{pro_r} + C_{fin_r} \quad (3.1)$$

Expressions for estimating these seven categories of RDTE cost are presented in Sections 3.1 through 3.7. An example application is given in Section 3.8.

Not all airplane programs are aimed at eventual production. Some are started for reasons of developing and/or demonstrating some aspect of advanced technology: the X-29 and X-31 programs are typical examples. A method for estimating the cost of such 'experimental' or 'prototyping' programs, together with an application is given in Section 3.9.

3.1 AIRFRAME ENGINEERING AND DESIGN COST: C_{aed_r}

Airframe engineering and design activities typically break down as follows:

1. Planning, conceptual design and associated cost studies (Phase 1 activities of Figures 2.1 and 2.3).

2. Preliminary design and system integration studies, including any associated cost studies (Phase 2 activities of Figures 2.2 and 2.3).

3. Engineering for windtunnel models, mock-ups and engine tests.

4. Design of windtunnel models and mock-ups.

   Note: Figure 3.1 records the number of windtunnel test hours needed in the past for several airplane programs.

5. Design and construction of dedicated test facilities, conduct of developmental tests and static tests including systems tests.

   Note: Items 3-5 are normally part of Phases 1 and 2 of Figures 2.1 through 2.3.

6. Detail design and development (This is represented by Phase 3 in Figure 2.3).

Figure 3.1 Windtunnel Hours Required in Typical Airplane Programs

Figure 3.2 Data and Trend Line of AMPR Weight as Related to Takeoff Weight

7. Release and maintenance of drawings and specifications. It is of interest to mention that the first Boeing 747 required 75,000 drawings.

   Note: During the last ten years most companies have switched from 'hand-drafting' to CAD (Computer Aided Drafting). The effect of this on airframe engineering and design cost, C_{aed_r} is to reduce this cost. By how much, varies from one manufacturer to another.

8. Liaison with manufacturing and with vendors.

9. Incorporation and analysis of design changes.

10. Development of specifications for materials, processes and for items purchased from vendors.

11. Analysis of reliability.

12. Analysis of maintainability and accessibility.

    Note: items 6-12 are normally part of Phase 3 of Figure 2.3.

The total engineering manhours required to complete Phases 1-3 of Figure 2.3 may be estimated from:

$$MHR_{aed_r} = \tag{3.2}$$

$$0.0396(W_{ampr})^{0.791}(V_{max})^{1.526}(N_{rdte})^{0.183}(F_{diff})(F_{cad})$$

The airframe engineering and design cost associated with Phases 1-3 of Figure 2.3 can be computed from:

$$C_{aed_r} = (MHR_{aed_r})(R_{e_r}) \tag{3.3}$$

The various terms in Equations (3.2) and (3.3) are defined as follows:

W_{ampr} is the so-called AMPR weight of the airplane. AMPR stands for: Aeronautical Manufacturers Planning Report. This weight is also referred to as the DCPR weight, where DCPR stands for: Defense Contractors Planning Report. The AMPR weight is defined by:

$$W_{ampr} = W_E - (\text{SUM } W_i)_{i=1}^{i=11} \qquad (3.4)$$

with:

W_E = airplane empty weight. A preliminary method for estimating airplane empty weight is given in Part I. A more detailed method for estimating empty weight and airplane component weights is given in Part V.

W_1 = weight of wheels, brakes, tires and tubes. Note: this is NOT the total landing gear weight, W_g of Part V.

W_2 = W_e, weight of engines (See Part V)

W_3 = W_{ess}, weight of starter (See Part V)

W_4 = weight of cooling fluid (Cooling fluid weight should be based on recommendations from the engine manufacturer)

W_5 = weight of rubber or nylon fuel cells (See Part V)

W_6 = W_{els}, weight of batteries, electrical power supply and electrical power conversion equipment (See Pt V)

W_7 = W_{iae}, weight of instruments, avionics and electronic equipment (See Pt V)

W_8 = W_{arm}, weight of armament and fire-control system (See Part V)

W_9 = W_{api}, weight of airconditioning units and fluid (See Part V)

W_{10} = W_{apu}, weight of APU (See Part V)

W_{11} = W_{tfo}, weight of trapped fuel and oil (See Parts I and V)

Many of the weight items 1-11 are often purchased from vendors. These items are NOT normally manufactured by the airframer.

IMPORTANT NOTES:

1. Lacking detailed knowledge of airplane component weights, an alternate (but less accurate) way of finding W_{ampr} is to use

Figure 3.2 which translates into:

$$W_{ampr} = \text{invlog}(0.1936 + 0.8645(\log W_{TO})) \quad (3.5)$$

where: W_{TO} is the takeoff weight. Methods for determining airplane takeoff weight are given in Parts I and V.

2. The cost of 'engineering' weight items W_i into the airframe is included in Eqn(3.1) despite the fact that W_{ampr} excludes these items!

V_{max} = V_C for civil airplanes (keas)

= V_H for military airplanes (keas)

For a definition of these speeds, see Section 4.2 in Part V of this text.

N_{rdte} is the number of airplanes produced for the RDTE phase. This number includes flight test airplanes as well as airframes used for static testing.

For military programs use: N_{RDTE} = 6 - 20

For commercial programs use: N_{RDTE} = 2 - 8

F_{diff} is a judgement factor which accounts for the difficulty (i.e. complexity) of a new airplane program. It is suggested to use:

F_{diff} = 1.0 for programs involving fairly conventional airplanes. Airplanes of the 1970 era fall in this class of relative difficulty.

F_{diff} = 1.5 for programs involving moderately aggressive use of advanced technology.

F_{diff} = 2.0 for programs involving very aggressive use of advanced technology. Examples would be: the X-29 and a laminar flow transport.

NOTE: The reader should not try to 'lowball' this difficulty factor. As a general rule, airplane development tends to be more difficult than anticipated.

F_{cad} is a judgement factor which accounts for the effect of computer aided design capability on airframe engineering and design cost. The following values are suggested:

F_{cad} = 1.2 for manufacturers which are in a CAD learning mode.

F_{cad} = 1.0 for manufacturers which are using 'manual' drafting techniques.

F_{cad} = 0.8 for manufacturers which are experienced in the use of CAD.

R_{e_r} is the engineering dollar rate per hour charged for the airframe engineering activity. It is preferred to use currently hourly rates. However, such information is often not available. Lacking data on current rates, the rates shown in Figure 3.3 may be used. Engineering manhour rates vary with factors such as: local cost-of--living, number of engineers in the job market, union contracts etc.

NOTES: 1. The engineering dollar rate per hour includes the following cost items:

*direct engineering labor
*overhead
*general and administrative costs
*miscellaneous direct charges such as: overtime premiums, travel, per diem, miscellaneous taxes, etc.

2. For airplane programs with a significant security requirement (such as 'black' programs) the hourly rates should be assumed to be 25 percent higher! Security measures add a lot of extra cost to a program.

For future estimates it is suggested to use:

$$(R_{e_r})_{\text{then year}} = \tag{3.6}$$

$$(R_{e_r})_{1989}\{(CEF_{\text{then year}})/(CEF_{1989})\}$$

where: $(R_{e_r})_{1989}$ is the engineering manhour rate for 1989 as found from Figure 3.3 or from actual data.

$\{(CEF_{\text{then year}})/(CEF_{1989})\}$ is the CEF-ratio as found from Fig.2.7.

<u>NOTE:</u> Equation (3.6), suitably modified, can also be used to adjust the manufacturing and tooling manhour rates of Section 3.3.

Figure 3.3 Variation of Engineering Manhour Rates with Calendar Time

3.2 DEVELOPMENT SUPPORT AND TESTING COST: C_{dst_r}

Typical activities which are responsible for this cost category are:

1. Windtunnel testing
2. Systems testing (for example the iron bird mentioned in Part IV)
3. Structural testing (strength, fatigue and materials testing)
4. Propulsion testing
5. Simulation for development support testing

The total cost of the above named activities is called C_{dst_r} and may be estimated from:

$$C_{dst_r} = 0.008325(W_{ampr})^{0.873}(V_{max})^{1.890}(N_{rdte})^{0.346}(CEF)(F_{diff}) \quad (3.7)$$

where: CEF is the cost escalation factor of Figure 2.7. Note that the value of CEF is 1.0 for 1970.

All other quantities are defined in Section 3.1.

3.3 FLIGHT TEST AIRPLANES COST: C_{fta_r}

This cost category is normally broken down into the following cost components:

1. Cost of engine and avionics as acquired from vendors: $C_{(e + a)_r}$
2. Manufacturing labor cost: C_{man_r}
3. Manufacturing material cost: C_{mat_r}
4. Tooling cost: C_{tool_r}
5. Quality control cost: C_{qc_r}

To estimate C_{fta_r} it is suggested to use:

$$C_{fta_r} = C_{(e + a)_r} + C_{man_r} + C_{mat_r} + C_{tool_r} + C_{qc_r} \quad (3.8)$$

where: $C_{(e + a)_r}$ is the cost of engines and avionics for the airplanes and airframes required for the RDTE phases. It can be found from:

$$C_{(e + a)_r} = \tag{3.9}$$

$$(C_{e_r} N_e + C_{p_r} N_p + C_{avionics_r})(N_{rdte} - N_{st})$$

where: C_{e_r} is the cost per engine. This may be estimated with the help of Appendix B.

N_e is the number of engines per airplane.

C_{p_r} is the cost per propeller. This may be found with the help of Appendix B.

N_p is the number of propellers per airplane. Note that N_p is not necessarily the same as N_e!

$C_{avionics_r}$ is the cost of avionics equipment per airplane. This equipment normally consists of: radar(s), navigation equipment, automatic flight control systems and computers, maintenance and fault detection equipment.

For military airplanes the following may also be included: fire control equipment, missile guidance equipment and other items required for the mission of that particular airplane.

After preparing a listing of all avionics systems required for the mission of the airplane, an individual cost estimate of each such system may be obtained with the help of Appendix C. This appendix also contains a 'short-cut' method for estimating avionics cost.

N_{rdte} is defined in Section 3.1.

N_{st} is the number of static test airplanes. These are NOT normally equipped with engines, propellers or avionics systems.

C_{man_r} is the manufacturing cost of the flight test airplanes. It may be estimated from:

$$C_{man_r} = (MHR_{man_r})(R_{m_r}) \tag{3.10}$$

where: (MHR_{man_r}) is the number of manufacturing manhours required in Phases 1-3 of Figure 2.3. This may be found from:

$$MHR_{man_r} = \tag{3.11}$$

$$28.984(W_{ampr})^{0.740}(V_{max})^{0.543}(N_{rdte})^{0.524}(F_{diff})$$

R_{m_r} is the manufacturing labor rate in dollars per manhour: see Figure 3.4. Also: read the note after Eqn.(3.6).

Other factors are defined in Section 3.1.

C_{mat_r} is the cost of materials to manufacture the flight test airplanes. It is found from:

$$C_{mat_r} = 37.632(F_{mat})(W_{ampr})^{0.689} *$$

$$*(V_{max})^{0.624}(N_{rdte})^{0.792}(CEF) \tag{3.12}$$

where: F_{mat} is a correction factor which depends on the type of materials used in the construction of the airplane:

F_{mat} = 1.0 for airframes made primarily of conventional aluminum alloys

= 1.5 for stainless steel airframes

= 2.0 - 2.5 for airplanes where the primary structure is made with 'conventional' composite materials, Li/Al alloys and/or ARAL

= 3.0 for carbon composite airframes

In most cases, the reader will have to 'estimate' his own 'correction' factor F_{mat}.

All other quantities in Eqn (3.12) are defined in Section 3.1.

Figure 3.4 Variation of Manufacturing Manhour Rates with Calendar Time

Figure 3.5 Variation of Tooling Manhour Rates with Calendar Time

C_{tool_r} is the tooling cost associated with the manufacturing of flight test airplanes. It can be estimated from:

$$C_{tool_r} = (MHR_{tool_r})(R_{t_r}) \tag{3.13}$$

where: (MHR_{tool_r}) represents the tooling manhours needed in Phases 1-3 of Fig.2.3. This may be found from:

$$MHR_{tool_r} = 4.0127(W_{ampr})^{0.764}(V_{max})^{0.899} *$$

$$*(N_{rdte})^{0.178}(N_{r_r})^{0.066}(F_{diff}) \tag{3.14}$$

R_{t_r} is the tooling labor rate in dollars per manhour which can be found from Figure 3.5. Also: read the comment after Eqn.(3.6).

N_{r_r} is the RDTE production rate in units per month. For RDTE a typical rate is 0.33 units per month.

Other factors are defined in Section 3.1.

C_{qc_r} is the quality control cost associated with manufacturing of the flight test airplanes. It may be estimated from:

$$C_{qc_r} = 0.13(C_{man_r}) \tag{3.15}$$

where: (C_{man_r}) is found from Eqn.(3.10)

<u>NOTE:</u> The manufacturing and tooling manhour rates depend on the same factors listed for the engineering manhour rates on page 27. In addition, if the manufacturing and tooling processes involve toxic materials, special facilities may be required. This would tend to increase the overhead component of the hourly rates. The designer must somehow account for this in any cost estimates.

3.4 FLIGHT TEST OPERATIONS COST: C_{fto_r}

The following activities are normally responsible for this cost category:

1. Flight testing of flight test airplanes. As an example, the Boeing 747 required 1,400 flight test hours before reaching its first 'type' certification.

2. Simulation activities associated with flight testing. This type of activity becomes very significant in the development of inherently unstable airplanes.

3. If the 'observables' of an airplane are important a significant amount of flight test time will have to be dedicated to the determination of the various 'signatures' of the airplane such as:

   * radar cross section

   * infrared signature

   * acoustic signature

   * optical signature

The total cost for these activities may be estimated from:

$$C_{fto_r} = 0.001244(W_{ampr})^{1.160}(V_{max})^{1.371} *$$

$$*(N_{rdte} - N_{st})^{1.281}(CEF)(F_{diff})(F_{obs}) \qquad (3.16)$$

where: F_{obs} is a factor which depends on the importance of having 'low' observables. Based on the author's judgement it is suggested to use:

For commercial airplanes and for military airplanes without a 'stealth' requirement: $F_{obs} = 1.0$

For 'stealthy' airplanes: $F_{obs} = 3.0$

All other terms are as previously defined.

3.5 TEST AND SIMULATION FACILITIES COST: C_{tsf_r}

In many airplane programs it will be found necessary to build new, dedicated test facilities. For example, in the case of a fly-by-wire, inherently unstable airplane a simulation facility will be required which couples the flight control software to the actual flight control hardware and which includes a sophisticated moving base cab with realistic displays. In addition, extensive software validation tests may be required.

There are not sufficient data in the literature to predict this type of cost. It is the responsibility of the designer to understand the facilities available to him during the development of the airplane. If, because of the special nature of the airplane new facilities must be constructed, an estimate of these costs must be made with the help of facilities people.

For preliminary cost estimating purposes it is suggested to use:

$$C_{tsf_r} = (F_{tsf})(C_{RDTE}) \tag{3.17}$$

where: C_{RDTE} is found from Eqn.(3.1)

F_{tsf} is a cost adjustment factor which depends on judgement. The author suggests to use:

F_{tsf} = 0.00 if no extra facilities are required

= 0.20 if extensive test and simulation facilities are required. The X-29, ATF and B-2 programs are examples of this.

3.6 RDTE PROFIT: C_{pro_r}

In most cases a privately held enterprise will want to make a profit on RDTE activities. The total cost of RDTE activities can be a very substantial amount and it can also take many years. The profit cost category may be estimated from:

$$C_{pro_r} = (F_{pro_r})(C_{RDTE}) \tag{3.18}$$

where: F_{pro_r} = 0.10 for a suggested profit of 10 percent.

C_{RDTE} is found from Equation (3.1).

NOTE: The actual profits vary significantly with politics (military), with market conditions (civilian) and with management strategy. Table 2.1 presents examples of profits made by various enterprises during the 1985-1988 time period. The reader should note that these profit figures are for the corporate entities listed. The profit made on a specific RDTE program may be more or it may be less. Many military programs have a fixed profit clause built in.

3.7 COST TO FINANCE THE RDTE PHASES: C_{fin_r}

In many instances a manufacturer will borrow money to finance the RDTE phases depicted in Figure 2.3. Borrowing money in turn costs money. However, even if internally generated cash is used for RDTE financing, there are costs associated with this: the money used could have earned interest!

Methods for estimating these financing costs are judged to be beyond the scope of this text. To estimate financing costs the calendar time element associated with an airplane program becomes of critical importance. A very general discussion of the calendar time factor is included in Chapter 9.

For a fundamental discussion of the cost of money the reader is referred to Reference 8, Section 2.4.4.

Lacking better information on the cost of financing the RDTE phase it is suggested to use:

$$C_{fin_r} = (F_{fin_r})(C_{RDTE}) \tag{3.19}$$

where: F_{fin_r} = 0.1 to 0.2, depending on the interest rates which are available.

C_{RDTE} is found from Equation (3.1).

3.8 EXAMPLE APPLICATION OF ESTIMATING RDTE COST

In this section an example application of the method for estimating RDTE costs is given. The example applies to the Ourania jet transport of Parts I, II and V of this text. Assume that it is required to estimate RDTE costs for the Ourania program for the year 1992.

The material is organized as follows:

3.8.1 RDTE Cost Input Data
3.8.2 RDTE Cost Calculations

3.8.1 RDTE Cost Input Data

The following input data are required to estimate RDTE costs with the method of Sections 3.1 through 3.7.

W_{TO} = 123,683 lbs from p.11, Part V. This assumes that the airframe will be made mostly from Li/Al alloys.

The AMPR weight, W_{ampr} is determined from Eqn.(3.5):

$$W_{ampr} = \text{invlog}\{0.1936 + 0.8645(\log 123{,}683)\} =$$

$$= 39{,}437 \text{ lbs}$$

$V_{max} = V_C$ = 295 kts from p.43, Part V. Note that this speed is in KEAS, at sea level!

N_{rdte} = 5, consisting of 3 flying prototypes and 2 airframes for structural testing. Because of the decision to use Li/Al alloys, it is prudent to perform a detailed structural evaluation! One airframe will be used for static testing (which includes watertank pressurization testing of the fuselage), the other for dynamic testing to determine the fatigue life. Therefore:

N_{st} = 2

CEF = 3.10, extrapolated from Figure 2.7 for 1992.

F_{diff} = 1.8 from page 26/27. This is based on the judgement that the combined use of Li/Al alloys and relaxed static stability (See Part II, p.272) represents a fairly aggressive use of new technology.

F_{cad} = 0.90 from page 27. This assumes moderate experience in the use of computer aided design methods.

N_e = 2 (number of engines per airplane: Pt I, p.55)

C_{e_r} = the cost per engine. This depends on the takeoff thrust of the engine, according to Appendix B. From Part I, p.184: T_{TO} = 47,625 lbs., therefore, the design takeoff thrust <u>per engine</u> is: 47,625/2 = 23,813 lbs.

The engine cost to the manufacturer is the price, EP, charged by the engine manufacturer. From Eqn(B9):

$$EP_{1989} = \text{invlog}\{2.3044 + 0.8858(\log 23,813)\} =$$

$$= \text{USD } 1,518,385$$

This price must be modified with the 1992 to 1989 CEF-ratio according to Eqn (B1). The CEF-ratio is found from Figure 2.7. The result is:

$$C_{e_r} = (3.10/3.02)1,518,385 = \text{USD } 1,558,607$$

N_p = 0 since the Ourania has no propellers.

$C_{avionics_r}$ = the cost of avionics per RDTE airplane.

If a detailed list of required avionics equipment were available, a reliable estimate of this cost contribution could be made. As a shortcut, it will be assumed that the installed avionics cost for the RDTE airplanes will be roughly 10 percent of the airplane market price. AMP_{1992}. The latter is found from Eqns (A14) and (A1):

$$AMP_{1992} = \{(3.10)/(3.02)\}[\text{invlog}\{3.3191 +$$

$$+ 0.8043(\log 123,683)\}] = \text{USD } 26,680,852$$

Therefore, ten percent of this amounts to roughly:

$C_{avionics_r}$ = USD 2,670,000 per RDTE airplane

R_{e_r} = (3.10/3.02)62.00 = 63.64 USD/mhr for the engineering manhour rate in 1992 as estimated with the help of Eqn.(3.6) and Figure 3.3.

CEF is the cost escalation factor obtained from Figure 2.7. For 1992:

$CEF_{1992} = 3.10$

$R_{m_r} = (3.10/3.02)34.44 = 35.35$ USD/mhr for the manufacturing manhour rate in 1992 as estimated with the help of Eqn.(3.6) and Figure 3.4.

$F_{mat} = 2.3$ from page 31. This is because of the decision to use Li/Al alloys.

$R_{t_r} = (3.10/3.02)43.06 = 44.20$ USD/mhr for the tooling manhour rate in 1992 as estimated with the help of Eqn.(3.6) and Figure 3.5.

$N_{r_r} = 0.33$ units per month (assumed).

$F_{obs} = 1.0$, according to page 34.

F_{tsf} = an adjustment factor which depends on the extra test and simulation facilities required for a new airplane program. Because of the relaxed stability decision made on the Ourania program, page 35 suggests:

$F_{tsf} = 0.2$

$F_{pro_r} = 0.10$, assumed from p.35.

$F_{fin_r} = 0.10$, assumed from p.36.

This completes the list of input requirements for estimating RDTE cost. The RDTE cost calculations are presented in sub-section 3.8.2.

3.8.2 RDTE Cost Calculations

With the input data from sub-section 3.8.1 the following RDTE cost component estimates can be made:

From Eqns (3.3) and (3.2):

$$C_{aed_r} = 0.0396(39,437)^{0.791}(295)^{1.526}(5)^{0.183} * (1.8)(0.9)(63.64) = \text{USD } 139,046,557$$

From Eqn (3.7):

$$C_{dst_r} = 0.008325(39,437)^{0.873}(295)^{1.890}(5)^{0.346} * (3.1)(1.8) = \text{USD } 38,819,483$$

From Eqn (3.9):

$$C_{(e+a)_r} = \{(1,558,607)(2) + (2,670,000)\}*(3) = \text{USD } 17,361,642$$

From Eqns (3.10) and (3.11):

$$C_{man_r} = 28.984(39,437)^{0.740}(295)^{0.543}(5)^{0.524} * (1.8)(35.35) = \text{USD } 236,680,646$$

From Eqn (3.12):

$$C_{mat_r} = 37.632(2.3)(39,437)^{0.689}(295)^{0.624}(5)^{0.792} * (3.1) = \text{USD } 48,975,376$$

From Eqns (3.13) and (3.14):

$$C_{tool_r} = 4.0127(39,437)^{0.764}(295)^{0.899}(5)^{0.178} * (0.33)^{0.066}(1.8)(44.20) = \text{USD } 213,015,201$$

From Eqn (3.15):

$$C_{qc_r} = 0.13(236,680,646) = \text{USD } 30,768,484$$

From Eqn (3.8):

$$C_{fta_r} = 17,361,642 + 236,680,646 + 48,975,376 + 213,015,201 + 30,768,484 = \text{USD } 546,801,349$$

From Eqn (3.16):

$$C_{fto_r} = 0.001244(39,437)^{1.160}(295)^{1.371}(3)^{1.281} * (3.1)(1.8) = \text{USD } 14,791,812$$

From Eqn (3.17):

$$C_{tsf_r} = 0.20(C_{RDTE})$$

From Eqn(3.18):

$$C_{pro_r} = 0.10(C_{RDTE})$$

From Eqn.(3.19):

$$C_{fin_r} = 0.10(C_{RDTE})$$

From Eqn (3.1):

$$C_{RDTE} = 139,046,557 + 38,819,483 + 546,801,349 +$$
$$+ 14,791,812 + 0.20(C_{RDTE}) + 0.10(C_{RDTE}) + 0.10(C_{RDTE})$$

from which it follows that:

C_{RDTE} = USD 739,459,201/(0.6) = USD 1,232,432,002 or rounding off to next highest million:

C_{RDTE} = <u>USD 1,232,000,000</u>

This represents the estimated 1992 cost of the entire RDTE part of the Ourania jet transport program.

Figure 3.6 shows the contribution of this RDTE cost to the unit cost of the Ourania transport. Note, that this type of RDTE program makes financial sense only if more than, say 250 transports can be expected to be sold.

<u>Figure 3.6 Effect of Number of Airplanes Produced on the RDTE Cost Contribution per Airplane</u>

3.9 METHOD FOR ESTIMATING PROTOTYPE PROGRAM COST WITH AN EXAMPLE APPLICATION

The purpose of this section is to present a method for estimating the cost of developing, manufacturing and flight testing prototypes. Prototypes are usually built rather quickly and without provisions required for a follow-on production program.

Nevertheless, many prototype programs do lead to a follow-on production program. Examples are the F-104, T-38, B-47, B-52, C-130 and KC-135 programs.

In many cases a prototype program is conducted mainly for the purpose of evaluating certain technologies: the NASA X-planes, such as the X-29 and the X-31 are good examples of this.

The method presented here should be applied ONLY to prototype programs where 1-4 airplanes are constructed. If more than 4 prototypes are to be constructed the RDTE method of Chapter 3 should be used instead. The method is based on Reference 23.

The prototype program cost may be estimated from:

$$C_{PROT} = (1115.4)(10^3)(W_{ampr})^{0.35}(N_{prot})^{0.99} *$$

$$*(CEF_{then\ year})/(CEF_{1973}) \qquad (3.20)$$

where: W_{ampr} is found from Eqn.(3.4) or from Eqn.(3.5).

N_{prot} is the number of prototypes to be built. This number should not exceed 4. When it does, the RDTE method of Chapter 3 should be used.

$CEF_{then\ year}$ is the Cost Escalation Factor (CEF) of Fig. 2.7 for a given 'then year'.

CEF_{1973} is the CEF for 1973 as obtained from Figure 2.7: $CEF_{1973} = 1.14$.

To illustrate the use of the method it will now be applied to the Grumman X-29 prototype program. A three-view of the X-29 is shown in Figure 3.7. The X-29 program was funded by DARPA (Defense Advanced Research Projects Agency). Grumman was awarded the X-29 contract in

December of 1981 for a total of USD 80,000,000. An anonymous source has indicated that Grumman invested a total of USD 20,000,000 of its own money on top of the DARPA funding. The total cost of the X-29 program (until delivery to NASA) was thus estimated at USD 100,000,000.

The following data are found in the 1985-1986 issue of Jane's All the World Aircraft:

W_{TO} = 17,800 lbs

N_{prot} = 2

Using Eqn.(3.5) it is found that:

W_{ampr} = 7,380 lbs

The 1982 CEF value is found from Figure 2.7 as:

CEF_{1982} = 2.5

With Eqn.(3.20): C_{PROT} =

$(1115.4)(10^3)(7,380)^{0.35}(2)^{0.99}(2.5)/(1.14)$ =

= USD 109,724,907

This is in good agreement with the actual cost which was quoted before.

Figure 3.7 Three-view of the Grumman X-29

SIAI - MARCHETTI S.211 AVIONIC SYSTEM

KEY

1. FLIGHT DIRECTOR COMPUTER
2. ADF
3. VERTICAL GYRO
4. DIRECTIONAL GYRO
5. RATE GYRO
6. AVIONIC CONTROL UNIT
7. TACAN
8. DME
9. V/UHF
10. HF
11. UHF ANTENNA
12. V/UHF ANTENNA
13. VOR/ILS ANTENNA
14. ALTITUDE SENSOR
15. R/ALTIMETER
16. IFF
17. R/ALTIMETER ANTENNA
18. VOR/ILS
19. ADF LOOP ANTENNA
20. MARKER BEACON ANTENNA

4. METHOD FOR ESTIMATING MANUFACTURING AND ACQUISITION COST

The purpose of this chapter is to present a method for estimating manufacturing cost: C_{MAN} and acquisition cost: C_{ACQ} for airplane programs. These costs are incurred during Phase 4 of an airplane program as shown in Figure 2.3. The acquisition cost represents the second cost source of the life cycle cost defined in Eqn.(2.3).

The following definitions are important:

<u>Definition 1:</u> The number of airplanes produced during an airplane program is $N_{program}$.

<u>Definition 2:</u> The number of airplanes produced to production standard during an airplane program is N_m.

<u>Definition 3:</u> The number of airplanes produced during the RDTE phases of an airplane program is: N_{rdte}.

Consequently: $N_{program} = N_m + N_{rdte}$ (4.1)

Table 4.1 shows examples of the total number of airplanes built in several airplane programs.

<u>Definition 4:</u> The total manufacturing cost associated with an airplane program is: C_{MAN}.

<u>Definition 5:</u> The total acquisition cost associated with an airplane program is: C_{ACQ}.

The difference between acquisition cost and manufacturing cost is the profit made by the manufacturer: C_{PRO}, so that:

$$C_{ACQ} = C_{MAN} + C_{PRO} \quad (4.2)$$

Again, the costs mentioned sofar represent program costs. The price paid by the user of an airplane (which is his acquisition cost) depends on a number of factors:

Table 4.1 Examples of Airplane Program Production Runs

Note: No. Produced equals $N_{program}$ of Eqn.(4.1).

Fighters

Type	No. Produced
Gen.Dyn. F-111	563
Gen.Dyn. F-16	>3,000
Gloster Meteor	3,545
Gloster Javelin	435
Grumman F9F2-5	1,325
Grumman F9F6-8	1,985
Grumman F11F	201
Grumman F14	> 900
Lockheed F-94	387
Lockheed F-80	1,732
Lockheed T-33	5,691
Lockheed F-104	2,578
McDonnell F-4	>5,000
McDD F-15	>2,000
McDD F-18	>1,500
SAAB JA37	329
SAAB J35A	604

Commercial Transports

Type	No. Produced
Douglas DC-7	338
Douglas DC-8	556
McDD DC-9, MD-80+	>1,000
Boeing 707	870
Boeing 727	1,834
Boeing 737	>2,600
Boeing 747	> 750

General Aviation

Type	No. Produced
Beech Duke	471
Beech Duchess	437
Beech Model 18	8,000
Cessna 140	5,429
Cessna 150	23,839
Cessna 310	5,447
Cessna 421	> 1,900
Cessna 500/501	315

Mil. Trainers/Liaison

Type	No. Produced
Cessna L-19	3,399
Beech T-34B	423
Beech T-34C	> 300
Cessna T-37	1,838
Fokker S-14	20
NAA T2C	521
Northrop T-38	1,189

Mil. Transports/Bombers

Type	No. Produced
Northrop C-125	23
Fairchild C-132	300
Lockheed C-130	>2,000
Lockheed C-141	285
Boeing B-47	2,004
Boeing B-52	742
Boeing KC-135	760
Douglas C-133	50

Regional Transports

Type	No. Produced
Beech M-99	234
Beech 1900	> 100
Fairchild Merlin/ Metroliner	> 800
Embraer-110	> 500
Embraer-120	> 150
DHC-6	> 832
DHC-7	> 115
DHC-8	> 137
Dornier 228	> 133
Fokker F-27	786
Fokker F-28	241
Fokker F-100	> 300
BAC-146	> 100
Pilatus B/N-2B	>1,077
Shorts 330	> 165
Shorts Skyvan	150

1. the total number of airplanes built by the manufacturer: $N_{program}$

2. the number of airplanes acquired: N_{acq}. In commercial programs this number tends to be only a fraction of $N_{program}$. For example, an airplane program may see 1,000 transports built. One airline customer may buy (acquire) 60 airplanes.

   In military programs (discounting foreign sales) N_{acq} is usually the same as N_m.

3. the manufacturers profit which can be negotiated: for large fleet buys a manufacturer often offers a lower profit to enhance his market share.

4. the cost of the RDTE program: C_{RDTE} (See Ch.3)

An estimate of the unit price per airplane can be obtained from:

$$AEP = \{(C_{MAN} + C_{PRO} + C_{RDTE})/N_m\} \qquad (4.3)$$

This assumes that the RDTE airplanes are not sold during the program. The effect of RDTE cost on the unit cost of an airplane was shown in Figure 3.6.

Another assumption made in Eqn.(4.3) is that no spare parts are bought by the user. In most cases commercial and military customers will want to buy a certain number of spare parts. The cost of spare parts is discussed in Chapters 5 and 6.

Having stated these fundamentals, the remainder of this chapter deals with the development of a method for estimating manufacturing cost: C_{MAN} and manufacturer's profit: C_{PRO}.

The method is a modified version of the methods of References 11 and 14. It should be used for preliminary cost estimation purposes only. The method can be applied to military as well as to commercial airplane programs.

The total airplane program manufacturing cost can be broken down into the following cost categories:

4.1 Airframe Engineering and Design Cost: C_{aed_m}

4.2 Airplane Production Cost: C_{apc_m}

4.3 Production Flight Test Operations Cost: C_{fto_m}

4.4 Cost of financing the manufacturing program: C_{fin_m}

The total manufacturing cost may be estimated from:

$$C_{MAN} = C_{aed_m} + C_{apc_m} + C_{fto_m} + C_{fin_m} \qquad (4.4)$$

Expressions for estimating the first three cost categories of the manufacturing cost are presented in Sections 4.1 through 4.3. The cost of financing the manufacturing part of the program is briefly discussed in Section 4.4. In Section 4.5 an equation is presented for estimating the manufacturer's profit. An example application is developed in Section 4.6. A discussion of the effect of several design and production parameters on the estimated price of an airplane is given in Section 4.7.

4.1 AIRFRAME ENGINEERING AND DESIGN COST: C_{aed_m}

Airframe engineering and design typically consists of the following activities:

1. Engineering design work necessitated by problems uncovered during RDTE phases 1-3 (Figure 2.3).
2. Design studies and integration studies associated with special customer requests.
3. Sustaining engineering associated with:
   * errors made in the manufacturing process
   * changes made in manufacturing of components
4. Release and maintenance of drawings and specifications.
   NOTE: during the last ten years most companies have switched from 'hand-drafting' to CAD (Computer Aided Design). The effect of this on C_{aed_m} is to reduce these costs.

   By how much, varies from one manufacturer to another.
5. Liaison engineering with manufacturing and with vendors.
6. Incorporation and analysis of design changes.
7. Development of specifications for materials and processes.
8. Analysis of reliability.
9. Analysis of maintainability and accessibility.

The total (= program) cost of the above named activities is called: C_{aed_m} and may be estimated from:

$$C_{aed_m} = (MHR_{aed_{program}})(R_{e_m}) - (MHR_{aed_r})(R_{e_r}) \qquad (4.5a)$$

or, with Eqn.(3.3):

$$C_{aed_m} = (MHR_{aed_{program}})(R_{e_m}) - C_{aed_r} \qquad (4.5b)$$

where: $MHR_{aed_{program}}$ is the total amount of engineering manhours required for the entire airplane program, that is for <u>all</u> $N_{program}$ airplanes. These manhours may be estimated from:

$$MHR_{aed_{program}} = 0.0396(W_{ampr})^{0.791}(V_{max})^{1.526} *$$

$$*(N_{program})^{0.183}(F_{diff})(F_{cad}) \qquad (4.6)$$

C_{aed_r} is computed with Eqn.(3.3).

R_{e_m} is the engineering manhour rate in dollars per hour for the entire airplane program. It may be assumed equal to R_{e_r} which in turn is obtained from Figure 3.3.

$N_{program}$ is the total number of airplanes produced: See Eqn.(4.1).

Other symbols are defined in Section 3.1.

4.2 AIRPLANE PROGRAM PRODUCTION COST: C_{apc_m}

The airplane (program) production cost category, C_{apc_m}, normally consists of the following cost components:

1. Cost of engine and avionics as acquired from vendors: $C_{(e + a)_m}$
2. Cost of interior: C_{int_m}
3. Manufacturing labor cost: C_{man_m}
4. Manufacturing material cost: C_{mat_m}
5. Tooling cost: C_{tool_m}
6. Quality control cost: C_{qc_m}

The following equation is suggested for estimating airplane program production cost, C_{apc_m}:

$$C_{apc_m} = C_{(e + a)_m} + C_{int_m} + C_{man_m} + C_{mat_m} + C_{tool_m} + C_{qc_m} \quad (4.7)$$

where: $C_{(e + a)_m}$ is the cost of engines and avionics equipment as paid to vendors. This can be estimated from:

$$C_{(e + a)_m} = (C_{e_m} N_e + C_{p_m} N_p + C_{avionics_m})(N_m) \quad (4.8)$$

where: C_{e_m} is the cost per engine during the manufacturing phase. This cost can differ from the cost per engine incurred in the RDTE phases: C_{e_r}.

Lacking better information, the method of Appendix B may be used to estimate the cost of production engines.

N_e is the number of engines per airplane.

C_{p_m} is the cost per propeller during the manufacturing phase. This cost can differ from the cost per propeller incurred in the RDTE phases: C_{p_r}.

Lacking better information, the method of Appendix B may be used to estimate the cost of production propellers.

N_p is the number of propellers per airplane. Note that N_p may not be the same as N_e!

$C_{avionics_m}$ is the cost of avionics equipment per airplane. This cost may differ from the avionics cost associated with the RDTE airplanes. Lacking better ininformation, the method of Appendix C may be used.

<u>Note:</u> in <u>civilian</u> airplanes the total cost of avionics typically is 5 to 15 percent of the airplane cost without avionics.

in <u>military</u> airplanes the total cost of avionics, depending on the mission of the airplane, may range from 10 to 100 percent of the airplane cost without avionics.

C_{int_m} is the cost of the airplane interior.

<u>Note:</u> The cost estimating methods of References 11 and 14 do not account for interior cost as a separate cost component. In military airplanes, for which these methods were primarily derived, this is acceptable because their interior cost is included in the statistical equations used for estimating cost. This includes items like passenger seats, ejection seats, insulating materials, etc.

The cost of interiors in commercial airplanes can be substantial and must be accounted for as a separate item. In commercial airplanes the cost of the interior includes items such as:

seats, partitions, cabin sidewall, ceiling and floor materials, galleys, lavatories, closets, storage bins, entertainment systems, lighting systems, insulating materials, etc.

The reader will have observed that interior cost was NOT accounted for in Eqn.(3.8) for the RDTE part of the airplane program. The reason is that flight test airplanes are normally NOT equipped with anything approaching a production interior.

To account for the interior cost component it is suggested to use:

$$C_{int_m} = (F_{int})(N_{pax})(N_m)*$$

$$*(CEF_{then\ year})/(CEF_{1990}) \qquad (4.9)$$

where: F_{int} is the interior cost factor expressed in USD/pax. The following values are suggested for calendar year 1990:

F_{int} = 0 USD/pax for military airplanes

= 500 USD/pax for light general avition airplanes

= 1,000 USD/pax for regional transport airplanes

= 2,000 USD/pax for jet transports

= 3,000 USD/pax for business jets

N_{pax} is the number of passengers.

CEF values are obtained from Figure 2.7.

C_{man_m} is the labor cost incurred in manufacturing N_m airplanes to production standard. It may be estimated from:

$$C_{man_m} = \qquad (4.10a)$$

$$(MHR_{man_{program}})R_{m_m} - (MHR_{man_r})R_{m_r}$$

or, with Eqn.(3.10):

$$C_{man_m} = (MHR_{man\ program})R_{m_m} - C_{man_r} \qquad (4.10b)$$

where: $MHR_{man_{program}}$ represents the total number of manhours required for the manufacturing of $N_{program}$ airplanes. It can be found from:

$$MHR_{man_{program}} = 28.984(W_{ampr})^{0.740}(V_{max})^{0.543} * (N_{program})^{0.524}(F_{diff}) \quad (4.11)$$

R_{m_m} is the manufacturing labor rate per hour for the entire program. This may be assumed to be equal to R_{m_r} which in turn is determined from Figure 3.4.

Other symbols are as defined before.

C_{mat_m} is the materials cost incurred while manufacturing N_m airplanes to production standard. It may be found from:

$$C_{mat_m} = C_{mat_{program}} - C_{mat_r} \quad (4.12)$$

where: $C_{mat_{program}}$ is the total materials cost associated with building $N_{program}$ airplanes. It follows from:

$$C_{mat_{program}} = 37.632F_{mat}(W_{ampr})^{0.689}(V_{max})^{0.624} * (N_{program})^{0.792}(CEF) \quad (4.13)$$

C_{mat_r} is found from Eqn.(3.12).

All other symbols are as defined before.

C_{tool_m} is the tooling cost to produce N_m airplanes. It may be estimated from:

$$C_{tool_m} = \tag{4.14a}$$

$$(MHR_{tool_{program}})R_{t_m} - (MHR_{tool_r})R_{t_r}$$

or, with Eqn.(3.13):

$$C_{tool_m} = (MHR_{tool_{program}})R_{t_m} - C_{tool_r} \tag{4.14b}$$

where: $MHR_{tool_{program}}$ is the total number of tooling manhours required to build $N_{program}$ airplanes.

It can be estimated from:

$$MHR_{tool_{program}} = 4.0127(W_{ampr})^{0.764}(V_{max})^{0.899} *$$

$$*(N_{program})^{0.178}(N_{r_m})^{0.066}(F_{diff}) \tag{4.15}$$

R_{t_m} is the tooling labor rate in dollars per manhour as found from Figure 3.5. Note: R_{t_m} may differ from R_{t_r}. It is suggested to ignore this difference in preliminary cost estimations.

N_{r_m} is the airplane manufacturing rate in units per month.

Other symbols are as defined before.

C_{qc_m} is the quality control cost associated with building N_m airplanes. It may be found from:

$$C_{qc_m} = 0.13(C_{man_m}) \tag{4.16}$$

where: C_{man_m} is found from Eqn.(4.10)

4.3 PRODUCTION FLIGHT TEST OPERATIONS COST: C_{fto_m}

This cost category is normally neglected in preliminary cost estimating calculations. Depending on the airplane program it may not be negligible. The following equation is suggested for estimating C_{fto_m}:

$$C_{fto_m} = N_m(C_{ops/hr})(t_{pft})(F_{ftoh}) \tag{4.17}$$

where: N_m is the number of airplane built to production standard.

$C_{ops/hr}$ is the airplane operating cost per hour as computed with the method of Chapter 5 (commercial airplanes) or Chapter 6 (military airplanes).

t_{pft} is the number of flight test hours flown by the manufacturer before airplane delivery to the customer. It is suggested to use:

t_{pft} = 2 hrs for general aviation airplanes
10 hrs for jet transports
20 hrs for military airplanes

F_{ftoh} is the overhead factor associated with the production flight test activities. Lacking actual overhead data it is suggested to use:

$F_{ftoh} = 4.0$

4.4 COST TO FINANCE THE MANUFACTURING PHASE: C_{fin_m}

Usually a manufacturer will borrow money to finance the manufacturing phase (Phase 4 in Figure 2.3) of an airplane program. Borrowing money in turn costs money. However, even if internally generated cash would be used for production financing, there are costs associated with this: the money used could have been earning interest!

Methods for estimating these financing costs are judged to be beyond the scope of this text. To estimate financing costs, the calendar time element associated with an airplane program becomes critically important. A discussion of the effect of calendar time on interest is also beyond the scope of this text.

For a fundamental discussion of the cost of money the reader should consult Reference 8.

For preliminary cost estimating purposes it is suggested to use:

$$C_{fin_m} = (F_{fin_m})(C_{MAN}) \tag{4.18}$$

where: F_{fin_m} = 0.1 to 0.2, depending on the interest rates which are available.

C_{MAN} is found from Equation (4.4).

4.5 PROFIT: C_{PRO}

As indicated in Chapter 2, privately held aeronautical enterprises will want to make a profit on airplane manufacturing activities. This profit, which is part of the cost of airplane acquisition (Eqn.4.2), may be estimated from:

$$C_{PRO} = (F_{pro_m})(C_{MAN}) \tag{4.19}$$

where: F_{pro_m} = 0.10 for a suggested profit of 10 percent.

C_{MAN} is found from Equation (4.4).

NOTE: Profits can vary significantly with politics (military) and with market conditions (civilian). Table 2.1 presents examples of profits made by various manufacturers during the 1985-1988 time period. The reader must keep in mind that these profit figures are for the corporate entities listed. The profit made on any given manufacturing program may be more or less.

4.6 EXAMPLE APPLICATION OF ESTIMATING MANUFACTURING AND ACQUISITION COST

The purpose of this section is to present an example application of the method for estimating the manufacturing and acquisition costs of an airplane program. In addition, an estimate for the unit price of the airplane will be determined. The example applies to the Ourania jet transport of Parts I, II and V of this text.

Since cost and price are tied to calendar time, it will be assumed that the cost and price estimates are required for the year 1992.

The material is organized as follows:

4.6.1 Manufacturing and Acquisition Cost Input Data

4.6.2 Manufacturing and Acquisition Cost Calculations

4.6.3 Determination of Estimated Unit Price

4.6.1 Manufacturing and Acquisition Cost Input Data

The following input data are required to estimate the manufacturing and acquisition cost for the Ourania program with the method of Sections 4.1 through 4.5.

W_{TO} = 123,683 lbs from p.11, Part V. This assumes that the airframe will be made mostly from Li/Al alloys.

The AMPR weight, W_{ampr}, is determined from Eqn.(3.5):

$$W_{ampr} = \text{invlog}\{0.1936 + 0.8645(\log 123{,}683)\} =$$

$$= 39{,}437 \text{ lbs}$$

$V_{max} = V_C$ = 295 kts from p.43, Part V. Note that this speed is in KEAS, at sealevel!

N_{rdte} = 5 (see p.37)

The number of airplanes produced to production standard, N_m, must be assumed at this point. The assumption is made that a marketing analysis has shown that 500 airplanes can be sold, so that:

$N_m = 500$

With Eqn.(4.1):

$N_{program} = 5 + 500 = 505$

$F_{diff} = 1.8$, according to p.37

$F_{cad} = 0.90$, according to p.38

$R_{e_m} = 63.64$ USD/mhr (Set equal to R_{e_r} of p.38)

C_{e_m} = the cost per engine during the production program. Lacking better information at this point, this cost will be assumed equal to the engine price paid in the RDTE program: C_{e_r} of p.38:

C_{e_m} = USD 1,558,607

$N_e = 2$ (number of engines per airplane: Pt I, p.55)

$N_p = 0$ since the Ourania has no propellers

$C_{avionics_m}$ = the cost of avionics per airplane in the production phase of the program. At this point it will be assumed that the avionics cost for production airplanes is the same as that for the RDTE airplanes: $C_{avionics_r}$. For the reasons stated on p.38:

$C_{avionics_m}$ = USD 2,670,000 per airplane

F_{int} = 2,000 USD/pax from p.52

$N_{pax} = 150$ from p.55, Part I.

$R_{m_m} = 35.35$ USD/mhr (Set equal to R_{m_r} of p.39)

$F_{mat} = 2.3$ from p.31

$CEF_{1992} = 3.10$ from p.20

$CEF_{1990} = 3.05$ from p.20

$R_{t_m} = 44.20$ USD/mhr (Set equal to R_{t_r} of p.39)

N_{r_m} = the unit production rate per month. It is assumed that the Ourania delivery rate will be 5 per month:

N_{r_m} = 5 units per month

$C_{ops/hr}$ = the total operating cost per block hour for the Ourania. This cost includes the direct and indirect cost of the manufacturer to operate this airplane during production flight testing. Operating cost can be computed with the method of Chapter 5. At this point it will be assumed that this cost for 1992 is USD 2,500 per block hour:

$C_{ops/hr}$ = USD 2,500 per block hour

t_{pft} = the number of flight test hours flown by the manufacturer before delivery to the customer. From Section 4.3:

t_{pft} = 10 hours

The flight test overhead factor (p.55) is set at:

F_{ftoh} = 4.0

The following data from the RDTE cost calculations are also needed:

C_{aed_r} = USD 139,046,557, from p.39

C_{man_r} = USD 236,680,646, from p.40

C_{mat_r} = USD 48,975,376, from p.40

C_{tool_r} = USD 213,015,201, from p.40

F_{fin_m} = 0.10 (assumed from p.56)

F_{pro_m} = 0.10 (assumed from p.56)

C_{RDTE} = USD 1,232,000,000, from p.41

4.6.2 Manufacturing and Acquisition Cost Calculations

With the input from sub-section 4.6.1 the following manufacturing cost categories can now be determined:

From Eqns (4.5) and (4.6):

$$C_{aed_m} = 0.0396(39,437)^{0.791}(295)^{1.526}(505)^{0.183} *$$

$$*(1.8)(0.90)(63.64) - 139,046,557 =$$

$$= 323,557,187 - 139,046,557 = \text{USD } 184,510,630$$

From Eqn.(4.8):

$$C_{(e + a)_m} = [(1,558,607)(2) + 2,670,000](500) =$$

$$= \text{USD } 2,893,607,000$$

From Eqn.(4.9):

$$C_{int_m} = (2,000)(150)(500)(3.10/3.05) =$$

$$= \text{USD } 152,459,016$$

From Eqns (4.10) and (4.11):

$$C_{man_m} = 28.984(39,437)^{0.740}(295)^{0.543}(505)^{0.524} *$$

$$*(35.35)(1.8) - 236,680,646 =$$

$$= 2,657,217,752 - 236,680,646 =$$

$$= \text{USD } 2,420,537,106$$

From Eqns (4.12) and (4.13):

$$C_{mat_m} = 37.632(2.3)(39,437)^{0.689}(295)^{0.624} *$$

$$*(505)^{0.792}(3.1) - 48,975,376 =$$

$$= 1,894,088,669 - 48,975,376 =$$

$$= \text{USD } 1,845,113,293$$

From Eqns (4.14) and (4.15):

$$C_{tool_m} = 4.0127(39,437)^{0.764}(295)^{0.899}(505)^{0.178} *$$

$$*(5)^{0.066}(1.8)(44.20) - 213,015,201 =$$

$$= 579,548,654 - 213,015,201 = \text{USD } 366,533,453$$

From Eqn.(4.16):

$$C_{qc_m} = 0.13(2,420,537,106) = \text{USD } 314,669,824$$

With Eqn.(4.7):

$$C_{apc_m} = 2,893,607,000 + 152,459,016 +$$

$$+ 2,420,537,106 + 1,845,113,293 +$$

$$+ 366,533,453 + 314,669,824 =$$

$$= \text{USD } 7,992,919,692$$

From Eqn.(4.17):

$$C_{fto_m} = 500(2,500)(10)(4) = \text{USD } 50,000,000$$

The financing cost, C_{fin_m}, will be determined from Eqn.(4.18) with $F_{fin_m} = 0.10$ so that:

$$C_{MAN} = (184,510,630 + 7,992,919,692 +$$

$$+ 50,000,000)/(0.9) = \text{USD } 9,141,589,247$$

From Eqns (4.2) and (4.19):

$C_{ACQ} = (1.1)(9,141,589,247) = \text{USD } 10,055,748,000$, or rounding off to the next highest million:

$$C_{ACQ} = \text{USD } 10,056,000,000$$

4.6.3 Determination of Estimated Unit Price

The price per airplane may now be estimated from Eqn.(4.3) as:

$$AEP_{1992} = (10,056,000,000 + 1,232,000,000)/500 =$$

$$= \text{USD } 22,576,000$$

This price should be checked for realism by comparison to the statistical market price data in Appendix A. Eqn.(A14) with Eqn.(A1) yields:

$$AMP_{1992} = \{(3.10)/(3.02)\}[\text{invlog}\{3.3191 +$$

$$+ 0.8043(\log 123.683)\}] = \text{USD } 26,680,852$$

The estimated price, AEP, is in fair agreement with the market price, AMP. A discussion of various factors which affect the airplane estimated price is presented in section 4.7.

4.7 EFFECT OF SEVERAL DESIGN AND PRODUCTION PARAMETERS ON AIRPLANE ESTIMATED PRICE: AEP

When inspecting the cost estimating equations in Sections 4.1 through 4.5 it is clear that the following parameters play a role in determining the program manufacturing cost and thereby in determining airplane price:

1. Airplane takeoff weight, W_{TO}, because of its effect on W_{ampr} which appears in Eqns.(4.6), (4.11), (4.13) and (4.15).
2. Airplane design cruise speed, V_{max}, which appears in the same equations as W_{ampr}.
3. Total number of airplanes built, $N_{program}$ which appears in the same equations as W_{ampr}.
4. Airplane production rate, N_{r_m}, which appears in Eqn.(4.15) for the tooling cost.
5. Airplane RDTE cost and the number of airplanes (N_m) over which this cost is to be recovered.

The purpose of this section is to show the results of some trade studies conducted to illustrate the effect of these design and production parameters on airplane estimated price. The author gratefully acknowledges the the work of Mr. Larry Bellmard who conducted these trade studies for the Ourania program of Sections 3.8 and 4.6.

Figure 4.1 shows the effect of the following parameters on the airplane estimated price, AEP:

* Total number of airplanes built, $N_{program} = 5 + N_m$

* Takeoff weight, W_{TO}

Observe that for the first 1,000 airplanes the airplane unit price is strongly affected by the number of airplanes built. After the 1,000 mark is reached there is still some price reduction to be seen but the effect is much less dramatic. There are two reasons for the rapid initial unit price reduction:

1) the so-called 'learning curve' effect

2) the RDTE cost contribution per airplane decreases hyperbolically with the total number of airplanes produced. This was already observed in Figure 3.6.

Figure 4.2 illustrates the 'learning curve' effect for three airplane programs: B-727, B-737 and McDD F-15. Note that the number of manhours required per airplane per pound decreases linearly when plotted on a log-log scale. This learning curve effect can be mathematically expressed as follows:

$$(\text{MHRS per unit}) = (\text{MHRS})_1/(N_{program})^n \qquad (4.20)$$

where: $MHRS_1$ is the number of manhours required to produce unit (= airplane) number 1.

$N_{program}$ is the total number of airplanes built.

n is the learning curve exponent. The meaning of n is best illustrated with a graphical example. This is done in Figure 4.3. Note that for a so-called 80 percent learning curve the exponent value is: n = 0.322. For a P percent learning curve the exponent n may be found from:

$$P = 100/(2^n) \qquad (4.21)$$

The learning curves shown in Figure 4.2 represent a 73 percent learning curve. Learning curves of 73 to 80 percent are fairly typical for the airframe manufacturing industry. The 'learning curve' effect noted here is already 'built' into the statistical cost equations of Sections 4.1 through 4.5.

The effect of airplane takeoff weight, W_{TO}, on airplane price is also shown in Figure 4.1. It is seen that the effect of weight on price is important when the number of airplanes produced is small.

Figure 4.4 illustrates the effect of airplane speed, V_{max}, on airplane price. It may be seen that increased speed requirements would increase airplane price rather significantly, when not many airplanes are produced.

Finally, Figure 4.5 shows that airplane production rate has a minor effect on airplane price except for the case where very small numbers are produced.

Figure 4.1 Effect of Number of Airplanes Produced and of Takeoff Weight on Airplane Estimated Price

Figure 4.2 Learning Curves for Three Airplane Programs

Figure 4.3a Reduction of Manhours with Units Produced

Figure 4.3b Percentage Reduction of Manhours with Units Produced

Figure 4.4 Effect of Number of Airplanes Produced and of Cruise Speed on Airplane Estimated Price

Figure 4.5 Effect of Number of Airplanes Produced and of Production Rate on Airplane Estimated Price

5. METHOD FOR ESTIMATING OPERATING COST OF COMMERCIAL AIRPLANES

The purpose of this chapter is to present a method for estimating the operating cost of commercial airplanes. A commercial airplane is defined as any non-military airplane. The method applies to airliners (all types), corporate airplanes and personal airplanes. A method for estimating the operating cost of military airplanes is presented in Chapter 6.

IMPORTANT ASSUMPTION:

Commercial operators acquire airplanes over many calendar years. As a result, commercial airplane production may vary considerably from year to year. The effect of all this on operating cost is difficult to model. It will therefore be ignored. Instead, the assumption is made that whatever numbers of airplanes are acquired by various operators, are acquired in one particular year and at one particular price.

The methodology presented in this chapter is based on methods and data presented in References 15 - 21. The material in this chapter is organized as follows:

5.1 Method for Estimating the Program Operating Cost of Commercial Airplanes: C_{OPS}

5.2 Method for Estimating the Direct Operating Cost of Commercial Airplanes: DOC

5.3 Method for Estimating the Indirect Operating Cost of Commercial Airplanes: IOC

5.4 Example Application

5.5 Some Important Observations about Commercial Airplane Operating Cost

5.1 METHOD FOR ESTIMATING THE PROGRAM OPERATING COST OF COMMERCIAL AIRPLANES: C_{OPS}

The total (or program) operating cost of commercial airplanes, C_{OPS}, is shown in relationship to other life cycle cost (LCC) sources in Eqn.(2.3) and in Figure 2.3.

The operating cost source is divided into the following two cost categories:

$$C_{OPS} = \sum_{i=1}^{i=n} (C_{ops_{dir}})_i (N_{acq})_i + \sum_{i=1}^{i=n} (C_{ops_{ind}})_i (N_{acq})_i \quad (5.1)$$

where: C_{OPS} is the program operating cost source of Eqn.(2.3). It is expressed in USD.

$(C_{ops_{dir}})_i$ is the program direct operating cost for the ith airplane customer, expressed in USD/airplane.

$(N_{acq})_i$ is the number of airplanes acquired by the ith customer. Table 5.1 shows examples of fleet sizes acquired by various customers.

<u>Note:</u> because commercial loss rates due to accidents are relatively small, the effect of this on the number of active airplanes will be ignored.

$(C_{ops_{ind}})_i$ is the program indirect operating cost for the ith airplane customer, expressed in USD/airplane.

The airplane program direct and indirect operating costs (per operator and per airplane) as defined by Equation (5.1) may be estimated from:

$$(C_{ops_{dir}})_i = (DOC)_i (R_{bl_{ann}})_i (N_{yr})_i \quad (5.2)$$

and:

$$(C_{ops_{ind}})_i = (IOC)_i (R_{bl_{ann}})_i (N_{yr})_i \quad (5.3)$$

where: $(DOC)_i$ is the direct operating cost per nautical mile of the airplane as flown by the ith customer and expressed in USD/nm (per airplane). A method for estimating DOC is presented in Section 5.2.

Table 5.1 Typical Fleet Sizes in 1989

Airline	Boeing					McDonnell Douglas			Lockheed	Airbus
	727	737	747	757	767	DC-9	DC-10	MD80-82	L-1011	A300-310
Domestic										
American	164	24	2		45		63	153		13
Continental	96	97	8			48	15	65		12
Delta	130	74		50	30	36		40	40	
Eastern	116			25		81	2		20	20
Fed.Express	118		21				24			
Northwest	209		40	33		139	20	8		
Pan Am	70		34							31
TWA	70		19		11	48		33	35	
United	145	134	31		19		56			
US Air	44	201			6	74		31		
Foreign										
KLM		16	21				5			10
SAS					5	60	7	31		
British Airways		49	40	35			8		17	
Air France	24	16	35							22
Lufthansa	24	58	25				11			19
Alitalia			12			43		42		16

$(R_{bl_{ann}})_i$ is the total annual block miles flown by the ith customer, expressed in nautical miles (nm) per airplane. This is related to block speed and to annual utilization as shown in Eqn.(5.4).

$(N_{yr})_i$ is the number of years during which the airplane is operated by the ith customer. It must be recognized that airplanes are sold and resold during their operational lives. For the purpose of program cost estimating it is suggested to use:

For transport airplanes: N_{yr} = 20 years

For corporate airplanes: N_{yr} = 20 years

For personal owner airplanes: N_{yr} = 20 - 40 years

$(IOC)_i$ is the indirect operating cost per nautical mile of the airplane as flown by the ith customer and expressed in USD/nm per airplane. A method for estimating IOC is presented in Section 5.3.

The reason for using the nautical mile (as opposed to the statute mile) as the operating cost basis is an arbitrary one. The reader will find that both mile types are used in the reference literature.

The reason for expressing the program cost, C_{OPS}, as a sum over i customers is the fact that direct and indirect operating costs vary considerably from one customer to another. However, there are no reliable methods for predicting what the cost variations among various customers will be. For the reader who wishes to actually calculate C_{OPS} for any given airplane program it is suggested to use 'reasonable averages' for $(C_{ops_{dir}})_i$ and for $(C_{ops_{ind}})_i$ while setting $(N_{acq})_i$ equal to N_m, the total number of airplanes built to production standard. For the remainder of this chapter the subscript i can therefore be omitted without ambiguity.

The total annual block miles, $R_{bl_{ann}}$, (now without subscript i) in Eqns (5.2) and (5.3) can be expressed as:

$$(R_{bl_{ann}}) = (V_{bl})(U_{ann_{bl}}) \tag{5.4}$$

where: V_{bl} is the block speed in nm/hr. The concept of block speed is illustrated in Figure 5.1. V_{bl} may be calculated with Eqn.(5.5).

$U_{ann_{bl}}$ is the annual utilization in block hours. This depends on airplane type and on routes flown: see the discussion on page 76.

For zero wind, the block speed, V_{bl}, is found from:

$$V_{bl} = (R_{bl})/t_{bl} \tag{5.5}$$

where: R_{bl} is the block distance measured in nm. It is defined in Figure 5.1. The block distance depends on the routes flown. It is suggested to use the 'great circle' distance between city pairs for R_{bl}. Table 5.2 gives examples of block distances between typical city pairs.

t_{bl} is the block time in hours. Figure 5.1 shows how t_{bl} is defined. It may be found from:

$$t_{bl} = t_{gm} + t_{cl} + t_{cr} + t_{de} \tag{5.6}$$

where: t_{gm} is the time spent in ground maneuvers, such as pulling away from the gate, taxiing to the active runway, takeoff run, landing ground run and taxiing to the gate at the point of final destination. It is expressed in hours. The ground maneuver time may be estimated from:

$$t_{gm} = 0.51(10)^{-6}(W_{TO}) + 0.125 \tag{5.7}$$

where: W_{TO} is the takeoff weight of the airplane as found with the methods of Part I, Chapter 2 or, more accurately with those of Part V, Chapters 2 and 4.

<u>Note:</u> Eqn.(5.7) includes an allowance of one minute for the takeoff run.

Figure 5.1 Mission Profile Definitions

Table 5.2 Example Block Distances, R_{bl}, Between City Pairs

City-Pairs Under 500 nm		R_{bl} (nm)			R_{bl} (nm)
Detroit	- Chicago	220	Wash. DC*	- Columbus	301
Buffalo	- Wash. DC*	270	Chicago	- Pittsburg	380
Chicago	- Cleveland	274	Reno	- Portland	424
Rochester	- Wash. DC*	283	Omaha	- Denver	430
Chicago	- Des Moines	288	Atlanta	- Pittsburg	487

City Pairs From 500 - 1,000 nm		R_{bl} (nm)			R_{bl} (nm)
Oakland	- Portland	501	Boise	- Denver	572
S.L. City	- S.Francisco	551	Seattle	- S.Francisco	669
Chicago	- Wash. DC**	555	N.Y.***	- Chicago	719
Huntsville	- Wash. DC*	558	Cleveland	- Ft.Lauderd.	972
Las Vegas	- Denver	561	Rochester	- Tampa	960

City Pairs From 1,000 - 1,700 nm		R_{bl} (nm)			R_{bl} (nm)
Denver	- Cleveland	1,066	Chicago	- Seattle	1,522
S.L. City	- Chicago	1,104	Los Ang.	- Chicago	1,538
Chicago	- Boise	1,264	Birmingh.	- Los Ang.	1,589
Memphis	- Los Ang.	1,430	Huntsville	- Los Ang.	1,610
Chicago	- Reno	1,469	S.Francis.	- Chicago	1,644

City Pairs From 1,700 - 2,300 nm		R_{bl} (nm)			R_{bl} (nm)
Los Ang.	- Detroit	1,749	N.Y.***	- Los Ang.	2,190
S.Francis.	- Detroit	1,846	N.Y.***	- S.Francis.	2,284
Los Ang.	- Wash. DC**	2,016			

* National Airport ** Dulles *** JFK

Chicago as listed here is O'Hare.

Data based on United Airlines Information, Ref.20.

t_{cl} is the time required to climb and to accelerate to the cruise speed, expressed in hours. A rapid method for predicting t_{cl} is presented in Part VII, pages 127-128.

t_{cr} is the time spent in cruise, expressed in hrs. It is suggested to use for:

<u>Domestic operations:</u>

$$t_{cr} = (1.06R_{bl} - R_{cl} - R_{de} + R_{man})/V_{cr} \quad (5.8a)$$

<u>International operations:</u>

$$t_{cr} = (1.01R_{bl} - R_{cl} - R_{de} + R_{man})/V_{cr} \quad (5.8b)$$

where: the factors 1.06 and 1.01 account for the fact that great circle distances cannot usually be flown for a variety of reasons.

R_{cl} is the distance to climb and to accelerate to cruise speed, in nm. It can be estimated from:

$$R_{cl} = (V_{cl})(t_{cl}) \quad (5.9)$$

where: V_{cl} is the projected horizontal speed of the airplane during the climb, in kts. For shallow climbs this may be assumed to be the same as the true airspeed in the climb. This speed is a by-product of the climb rate calculations for finding t_{cl}.

The reader is referred to pages 124-128 in Part VII.

R_{de} is the distance covered during the descent, in nm. It can be estimated from:

$$R_{de} = (V_{de})(t_{de}) \quad (5.10)$$

where: V_{de} is the projected horizontal speed of the airplane during the descent, in kts.

<u>Note:</u> For shallow descents this is roughly the same as the true airspeed during the

descent. This speed is a by-product of the descent rate calculations of pages 159-161 in Part VII.

t_{de} as used in Eqns (5.6) and (5.10), is the time taken to descend. It may be computed with the method of p.159-161 in Part VII.

R_{man} is the distance covered while maneuvering due to ATC (air traffic control) constraints (in nm), with:

$$R_{man} = (V_{man})(t_{man}) \qquad (5.11)$$

where: V_{man} is the speed while maneuvering as required by ATC constraints. It is suggested to use:

below 10,000 ft: $V_{man} = 250$ kts

above 10,000 ft: $V_{man} = V_{cr}$ (kts)

t_{man} is the time spent in ATC maneuvering. It can be found from:

$$t_{man} = 0.25(10)^{-6}W_{TO} + 0.0625 \qquad (5.12)$$

V_{cr} is the airplane cruise speed in kts. Different operators cruise airplanes at different speeds. For preliminary costing it is recommended to use:

$$V_{cr} = V_C \qquad (5.13)$$

where: V_C is the design cruise speed of an airplane in <u>nautical miles</u> per hour (kts).

<u>NOTE:</u> V_C is normally a 'given' in the mission specification of an airplane. Examples are found on pages 50 and 55 in Part I. The reader should ascertain that V_C of Eqn.(5.13) is consistent with the V_C defined in his V-n diagram calculations of Ch.4, Part V. However, note that V_C in the V-n diagram is expressed in KEAS and not in kts.

The annual utilization in block hours, $U_{ann_{bl}}$, as defined in Equation (5.4) depends on the type of airplane and on the routes flown. The weekly magazine Aviation Week and Space Technology publishes utilization data on a quarterly basis for passenger transports. The monthly magazine Business and Commercial Aviation publishes data on the utilization of other commercial airplanes. The reader will find representative utilization data for several types of airplanes in Table 5.3.

Lacking more detailed information it suggested to determine $U_{ann_{bl}}$ as follows:

<u>For passenger transports:</u>

$$U_{ann_{bl}} = (10^3)[3.4546(t_{bl}) + 2.994 + - \{12.289(t_{bl})^2 - 5.6626(t_{bl}) + 8.964\}^{1/2}] \quad (5.14a)$$

<u>For cargo transports:</u>

$$U_{ann_{bl}} = (10^3)[6.053(t_{bl}) + 5.70 + - \{37.771(t_{bl})^2 + 13.494(t_{bl}) + 32.490\}^{1/2}] \quad (5.14b)$$

where: t_{bl} is determined from Equation (5.6).

Eqns (5.14) are shown graphically in Figure 5.2.

<u>For All Other Types of Airplanes:</u> see Table 5.3.

Observe that annual airplane utilization so far has been expressed in terms of block hours. It is sometimes desirable (for a variety of reasons) to express the annual utilization in terms of flight hours: $U_{ann_{flt}}$. This can be done as follows:

$$U_{ann_{flt}} = (U_{ann_{bl}})(V_{bl})/(V_{flt}) \quad (5.15)$$

where: V_{flt} is the average flight speed at which the airplane is operated, in kts. It is noted that the average flight speed is always greater than the block speed. In turn, the average

Table 5.3 Utilization of Commercial Airplanes

Operator	1988 Airplane Utilization of Narrow-Body Jets in Block Hours per Day				
	727-200	737	757	MD-80	DC-9
American	7.8	9.2	NA	8.8	NA
United	9.2	7.9	NA	NA	NA
Delta	8.9	7.9	8.4	8.0	7.3
US Air/Piedmont	8.2	7.6	NA	NA	7.9
TWA	6.6	NA	NA	8.7	6.0

	1988 Airplane Utilization of Wide-Body Jets in Block Hours per Day				
	747	767	A300	DC-10	L-1011
American	10.2	11.1	NA	9.0	NA
Pan American	11.0	NA	8.4	NA	NA
United	12.0	10.8	NA	10.0	10.2
TWA	11.0	10.8	NA	NA	10.0
Continental	8.0	NA	7.0	10.0	NA

The following data are representative <u>annual-utilization</u> numbers <u>in block hours</u> for various types of airplanes:

Type of Operation:	Long Haul Block Time: $t_{bl} > 5$	Medium Haul Block Time: $2 < t_{bl} < 5$	Short Haul Block Time: $0.5 < t_{bl} < 2$
Airplane Type			
Jet Transports	3,600-4,400	2,100-3,300	1,000-3,000
Regional Transports	NA	2,000-3,000	1,000-2,500
Corporate Transp.	500-1,500	400-1,200	300-1,000
Personal Transp.	NA	200-800	200-800
Agricultural	NA	NA	500-1,000
Trainers	NA	NA	1,000-2,500

NA = Not Applicable

<u>Data Sources:</u> Various issues of 'Aviation Week and Space Technology' and 'Business and Commercial Aviation'.

Figure 5.2 Effect of Block Time on Annual Utilization

flight speed is always less than the design cruise speed. This leads to the inequality:

$$V_{cr} > V_{fl} > V_{bl} \tag{5.16}$$

The average flight speed may be found from:

$$V_{flt} = V_{cr}(t_{cr}/t_{flt}) \tag{5.17}$$

where: t_{cr} is found from Equations (5.8)

t_{flt} is the average flight time which can be computed from:

$$t_{flt} = t_{cl} + t_{cr} + t_{de} \tag{5.18}$$

where: t_{cl} and t_{de} are defined on pages 74 and 75, respectively.

Note: it is observed that t_{man} of Eqn.(5.11) is already included in the definition of t_{cr} via the term R_{man} in Eqn.(5.8)!

Remember, that C_{OPS} in Equation (5.1) represents one of four <u>program</u> cost sources of Life Cycle Cost (LCC) as expressed by Equation (2.3). This cost source as such is normally NOT of any particular interest to a commercial airplane operator. However, the following cost categories are VERY IMPORTANT to typical airplane operators:

1a) Direct operating cost per nautical mile: (DOC)

and:

1b) Direct operating cost per block hour: $\{(DOC)(V_{bl})\}$

2a) Indirect operating cost per nautical mile: (IOC)

and:

2b) Indirect operating cost per block hour: $\{(IOC)(V_{bl})\}$

Sections 5.2 and 5.3 present methods for estimating DOC and IOC respectively. An example application is provided in Section 5.4. Finally, a number of important observations regarding DOC and IOC are made in Section 5.5.

5.2 METHOD FOR ESTIMATING THE DIRECT OPERATING COST OF COMMERCIAL AIRPLANES: DOC

The purpose of this section is to present a method for estimating the direct operating cost (DOC) in USD/nm incurred while operating commercial airplanes.

The method which is presented here is an adaptation of the so-called ATA-method of References 15 and 16 with various inputs from References 17-21. The ATA-method, as such, applies only to passenger and cargo transport airplanes. With the adaptation given here the method can be applied to any type of commercial airplane.

Table 5.4 presents an overview of cost components and cost elements which contribute to direct operating cost per nautical mile: DOC, which is broken down as:

$$DOC = DOC_{flt} + DOC_{maint} + DOC_{depr} + \\ + DOC_{lnr} + DOC_{fin} \qquad (5.19)$$

where: DOC_{flt} is the direct operating cost of flying in USD/nm: see Sub-section 5.2.1

DOC_{maint} is the direct operating cost of maintenance in USD/nm: see Sub-section 5.2.2

DOC_{depr} is the direct operating cost of depreciation in USD/nm: see Sub-section 5.2.3

DOC_{lnr} is the direct operating cost of landing fees, navigation fees and registry taxes in USD/nm: see Sub-section 5.2.4

DOC_{fin} is the direct operating cost of financing in USD/nm: see Sub-section 5.2.5

NOTE: The right hand side of Table 5.4 indicates which DOC elements apply to various airplane types.

5.2.1 Direct Operating Cost of Flying: DOC_{flt}

The direct operating cost of flying (in USD/nm) is broken down into the following flying cost elements:

$$DOC_{flt} = C_{crew} + C_{pol} + C_{ins} \qquad (5.20)$$

Table 5.4 Overview of Direct Operating Cost (DOC)

DIRECT OPERATING COST ~ DOC ~ USD/nm			AIRLINE	CORPORATE	PERSONAL
DOC COMPONENTS	DOC ELEMENTS				
FLYING ~ DOC_{flt} (5.20)	CREW ~ C_{crew}	(5.21)	✓	✓	NO
	FUEL+ ~ C_{pol}	(5.25)	✓	✓	✓
	INSURANCE ~ C_{ins}	(5.31)	✓	✓	✓
MAINTENANCE ~ DOC_{maint} (5.33)	AIRFRAME LABOR ~ $C_{lab/ap}$	(5.34)	✓	✓	✓
	ENGINE LABOR ~ $C_{lab/eng}$	(5.36)	✓	✓	✓
	AIRFRAME MATERIALS ~ $C_{mat/ap}$	(5.37)	✓	✓	✓
	ENGINE MATERIALS ~ $C_{mat/eng}$	(5.38)	✓	✓	✓
	APPLIED MAINTENANCE BURDEN ~ C_{amb}	(5.39)	✓	✓	NO
DEPRECIATION ~ DOC_{depr} (5.40)	AIRFRAME ~ C_{dap}	(5.41)	✓	✓	✓
	ENGINE(S) ~ C_{deng}	(5.42)	✓	✓	✓
	PROP(S) ~ C_{dprp}	(5.43)	✓	✓	✓
	AVIONICS ~ C_{dav}	(5.44)	✓	✓	✓
	AIRFRAME SPARE PARTS ~ C_{dapsp}	(5.45)	✓	✓	NO
	ENGINE SPARE PARTS ~ C_{dengsp}	(5.46)	✓	✓	NO
LANDING & NAVIG. FEES, REGISTRY TAXES ~ DOC_{lnr} (5.47)	LANDING FEES ~ C_{lf}	(5.48)	✓	✓	NO
	NAVIGATION FEES ~ C_{nf}	(5.52)	✓	✓	NO
	REGISTRY TAXES ~ C_{rt}	(5.53)	✓	✓	✓
FINANCE ~ DOC_{fin} (5.55)			✓	✓	✓

where: C_{crew} is the crew cost in USD/nm

C_{pol} is the fuel and oil (pol means: petroleum, oil and lubricants) cost in USD/nm

C_{ins} is the airframe insurance cost in USD/nm

Methods for estimating these cost elements are presented in sub-sub-sections 5.2.1.1 through 5.2.1.3.

5.2.1.1 Crew cost per nautical mile: C_{crew}

The crew cost per nautical mile, C_{crew}, can be found from:

$$C_{crew} = \sum_{j=1}^{j=4} [(n_{c_j})\{(1 + K_j)/V_{bl}\}(SAL_j/AH_j) + (TEF_j/V_{bl})] \quad (5.21)$$

where: For transport airplanes:

n_{c_j} is the number of crew members of each type, j.

j indicates the type of crew member. The following distinctions are normally made:

n_{c_1} stands for captain.

n_{c_2} stands for co-pilot.

n_{c_3} stands for flight engineer.

Note: the cost of the cabin crew (flight attendants) is accounted for in the indirect operating cost, see Section 5.3.

The number of crew members which must be carried depends on government regulations (FARs), company rules and union rules. Lacking precise information the following numbers are suggested:

n_{c_1} = 1 for scheduled block times < 10 hours
= 2 for scheduled block times > 10 hours

n_{c_2} = as for n_{c_1}

n_{c_3} = 0 for airplanes with a 2-man cockpit

= as for n_{c_1} for airplanes with a 3-man cockpit

<u>For personal owner type airplanes:</u>

It is common <u>NOT</u> to count the cost of the pilot, so that:

n_{c_j} = 0 and j=1 only.

<u>For agricultural airplanes:</u>

n_{c_j} = 1 and j=1 only.

K_j is a factor which accounts for such items as: vacation pay, cost of training, crew premium, crew insurance, and payroll tax. This factor varies from one operator to another. Lacking more detailed data on operator work rules and benefits, it is suggested to use: $K_j = 0.26$.

V_{bl} is the airplane block speed in nautical miles per hour. The block speed may be found from:

$$V_{bl} = R_{bl}/t_{bl} \tag{5.22}$$

where: R_{bl} is the block distance in nautical miles. It is defined in Figure 5.1.

t_{bl} is the block time in hours: see Fig. 5.1 and Equation (5.6).

SAL_j is the annual salary paid to a crew member of type j. Crew member salaries depend on the following factors:

*equipment flown: salaries tend to increase with airplane weight and with airplane speed
*seniority
*union and company rules

References 15 and 16 contain formulas which relate crew salaries to airplane weight and speed. These formulas applied reasonably well in the sixties because the airline industry was heavily regulated by the CAB:

Civil Aeronautics Board. The author believes that these salary relations can no longer be used. Table 5.5 contains recent data on crew salary ranges. The reader must decide which salaries apply to his airplane.

To estimate crew salaries for future years it is suggested to use:

$$SAL_{then\ year} = \tag{5.23}$$

$$= SAL_{1989}\{(CEF_{then\ year})/(CEF_{1989})\}$$

where: the CEF values are found from Fig.2.7.

AH_j is the number of flight hours per year for a crew member of type j. Lacking actual data, the following numbers are suggested as applicable to all flight crew members:

<u>For domestic operations:</u>

AH_j = 800 hrs for jets
= 900 hrs for props

<u>For international operations:</u>

AH_j = 750 hrs for jets
= 850 hrs for props

TEF_j is the travel expense factor associated with each type of crew member. Since flight crews normally stay in the same hotel it is not necessary to vary the TEF from one crew member to another. It is suggested to use for calendar year 1990:

<u>Domestic routes:</u> TEF = USD 7.0/bl.hr

<u>International routes:</u> TEF = USD 11.0/bl.hr

For future years use:

$$TEF_{then\ year} = \tag{5.24}$$

$$= TEF_{1990}(CEF_{then\ year})/(CEF_{1990})$$

where: the CEF values are found from Fig.2.7.

Table 5.5 Salaries and Wages Paid by Operators in 1989

Cockpit Crew: Salary Range in USD/year

Airplane Type:	Captain	First Officer	Flight Engineer
Jet Transport BAC 111 - B747	35,000- 144,000	24,000- 67,000	20,000- 62,000
Corp. Jet Transp. Learjet M23 - Gulfstr.III	30,000- 72,000	22,000- 52,000	NA
Corp. Turboprops MU-2 - King Air 300	25,000- 52,000	20,000- 32,000	NA
Regional Turboprops DHC-6 - F-27	20,000- 52,000	11,000- 21,000	NA
Corporate Recips Single - Twin	20,000- 47,000	19,000- 26,000	NA

Cabin Crew: Salary Range in USD/year

	Purser	Cabin Attendant
All airplane types	25,000-47,000	19,000-32,000

Maintenance Personnel: Hourly Wage Range, USD/hr

	Fixed Base Operators (FBO)	Airlines
Avionics specialist	6.60 - 15.60	8.00 - 20.60
Dispatcher	5.00 - 10.40	7.00 - 15.50
Inspector	5.00 - 16.60	10.00 - 22.60
Line Personnel	3.35 - 12.80	5.00 - 10.30
Mechanic with A/P	5.00 - 18.40	10.00 - 22.60
Mechanic without A/P	5.00 - 11.30	8.00 - 15.50
Stock Clerk	3.35 - 7.20	5.00 - 10.30

NOTES: 1. For years other than 1989 use:

$$(\text{SAL or Wage})_{\text{then year}} = (\text{SAL or Wage})_{1989}\{(\text{CEF}_{\text{then year}})/(\text{CEF}_{1989})\}$$

where: the CEF data are obtained from Fig.2.7.

2. Salary and wage data do NOT include overhead!
3. Sources: Professional Pilot, Discussions with industry people and CEF data, Fig.2.7.
4. NA = Not Applicable

5.2.1.2 Fuel and oil cost per nautical mile: C_{pol}

The fuel and oil cost per nautical mile, C_{pol}, can be estimated from:

$$C_{pol} = (W_{F_{bl}}/R_{bl})(FP/FD) + $$

$$+ (W_{ol_{bl}}/R_{bl})(OLP/OD) \qquad (5.25)$$

where: $W_{F_{bl}}$ is the block fuel used in lbs. This amount of fuel is the same as the mission fuel used, $W_{F_{used}}$:

$$W_{F_{bl}} = W_{F_{used}} \qquad (5.26)$$

A preliminary method for computing $W_{F_{used}}$ is presented as Eqn.(2.14) of page 16, Part I. More accurate methods to compute $W_{F_{used}}$ are given in Sections 5.4 and 5.5 of Part VII.

NOTE: Reference 20 shows that airline experience indicates that the mission fuel consumption of an airplane tends to increase with calendar time for the following reasons:

* engine deterioration
* airframe surface deterioration (seals, finish)

If the reader wishes to account for this, 0.5 percent increase in fuel consumption per year of service seems to be the industry average.

R_{bl} is the block distance in nautical miles as defined in Figure 5.1.

FP is the price of fuel in USD/gallon

Figure 5.3 shows how fuel prices have varied with calendar time. There is no accurate way to predict how fuel prices will vary in the future. The reader is encouraged to call a local fixed base operator (FBO) to determine actual fuel prices.

NOTE: in this text 'gallon' means 'US gallon'!

Figure 5.3 Variation of Fuel Prices with Calendar Year

FD is the fuel density in lbs/gallon. The following values are suggested:

<u>Aviation Gasoline:</u>

grades 100/130: FD = 6.0 lbs/gallon
grades 108/135: FD = 5.9 lbs/gallon
grades 115/145: FD = 5.8 lbs/gallon

<u>Aviation Petroleum:</u>

Kerosene:	FD = 6.7 lbs/gallon
JP-1:	FD = 6.65 lbs/gallon
JP-3:	FD = 6.45 lbs/gallon
JP-4:	FD = 6.55 lbs/gallon
JP-5:	FD = 6.82 lbs/gallon
Jet A	FD = 6.74 lbs/gallon

$W_{ol_{bl}}$ is the weight of oil and lubricants used, in lbs. It depends strongly on the type of powerplants and systems in need of lubricants. Manufacturers of engines and systems normally specify the hourly consumption of oil and lubricants. Lacking such specific information, it is suggested to use:

<u>for reciprocating engines:</u>

$$W_{ol_{bl}} = (W_{F_{used}})/70 \qquad (5.27)$$

<u>for turbine engines:</u>

$$W_{ol_{bl}} = (0.70)(N_e)(t_{bl}) \qquad (5.28)$$

where: N_e is the number of engines

t_{bl} is the block time in hours as found from:

$$t_{bl} = R_{bl}/V_{bl} \qquad (5.29)$$

Note that Eqn.(5.29) is a modification of Eqn.(5.5).

Eqns (5.27) and (5.28) may be assumed to account for consumption of oil and lubricants together.

OD is the oil density in lbs/gallon. The following values are suggested:

Oil (MIL-6082B, grade 1100): OD = 7.40 lbs/gallon
Synthetic oil (MIL-L-7808-1): OD = 7.74 lbs/gallon

OLP is the price of oil and lubricants in USD/gallon.

Figure 5.3 also shows the price of oil. Just like is the case for fuel prices, there is no accurate way to predict oil prices in the future.

<u>An alternate method</u> to account for the direct operating cost of oil and lubricants is to assume that it is 5 percent of the direct operating cost of fuel. This leads to:

$$C_{pol} = 1.05(W_{F_{bl}}/R_{bl})(FP/FD) \qquad (5.30)$$

5.2.1.3 Cost of airframe insurance per nautical mile: C_{ins}

Airplane operators carry airframe insurance for the following reasons:

1. ground and flight risk of experiencing airframe damage or total loss
2. passenger liability in case of injury or death
3. third party liability in case of injury or death
4. cargo damage risk

Most airplane operators will want to carry airframe damage insurance (to cover item 1) to an amount equal to the replacement value of the airplane. Although some operators will prefer to 'carry' a certain amount of the insurance themselves, there is a cost associated with that as well: a reserve will have to be created and the interest cost on the reserve can be thought of as part of the cost of insurance.

<u>NOTE WELL:</u> In references 15-20 the convention has been adopted to consider item 1 above as part of the DOC while items 2-4 are considered as part of the IOC (Section 5.3). This convention is followed here also!

Insurance rates for hull damage or loss depends on the so-called hull loss rate. Figure 5.4 gives historic insight into the hull losses of jet transports. Note that the annual hull loss rate on the average is about 2.0 per 1,000,000 departures.

For details on insurance rates the reader should consult an aviation underwriter. Lacking such precise information it is suggested to use:

$$C_{ins} = (f_{ins_{hull}})(AMP)/[(U_{ann_{bl}})(V_{bl})] \qquad (5.31)$$

where: $f_{ins_{hull}}$ is the annual hull insurance rate in USD per USD airplane price, per airplane and per year. Typical values for this hull insurance rate range from:

$f_{ins_{hull}}$ = 0.005 to 0.030 USD/USD/airplane/year

AMP is the airplane market price in USD. AMP may be estimated rapidly with the method given in Appendix A. A more detailed method for computing AMP is presented in Sub-section 4.6.3.

$U_{ann_{bl}}$ is the annual block hour utilization as determined from the discussion on p.76.

V_{bl} is the block speed in nm/hr (kts) as defined by Eqn.(5.5).

An alternate method to account for hull insurance is to set:

$$C_{ins} = 0.02(DOC) \qquad (5.32)$$

where: DOC is found from Eqn.(5.19).

SOURCE: AVIATION WEEK AND SPACE TECHNOLOGY JULY 24, 1989

Figure 5.4 Worldwide Accident and Hull Loss Rates

5.2.2 Direct Operating Cost of Maintenance: DOC_{maint}

The direct operating cost of maintenance is broken down as follows:

$$DOC_{maint} = C_{lab/ap} + C_{lab/eng} + C_{mat/ap} + C_{mat/eng} + C_{amb} \quad (5.33)$$

where: $C_{lab/ap}$ is the labor cost of airframe and systems (other than the engines) maintenance in USD/nm. See: Eqn.(5.34).

$C_{lab/eng}$ is the labor cost of engine(s) maintenance in USD/nm. See: Eqn.(5.36).

$C_{mat/ap}$ is the cost of maintenance materials for the airframe and systems (other than the engines) in USD/nm. See: Eqn.(5.37).

$C_{mat/eng}$ is the cost of maintenance materials for the engines in USD/nm. See Eqn.(5.38).

C_{amb} is the applied maintenance burden in USD/nm. See Eqn.(5.39).

The maintenance labor cost per nautical mile of airframe and systems, $C_{lab/ap}$, can be estimated from:

$$C_{lab/ap} = 1.03(MHR_{map_{bl}})(R_{l_{ap}})/(V_{bl}) \quad (5.34)$$

where: 1.03 is called a non-revenue factor. It accounts for extra maintenance costs incurred due to flight delays.

$MHR_{map_{bl}}$ is the number of <u>airframe and systems</u> maintenance man hours needed per block hour. This is related to the number of maintenance man hours per flight hour, $MHR_{map_{flt}}$ as follows:

$$MHR_{map_{bl}} = (MHR_{map_{flt}})(t_{flt}/t_{bl}) \quad (5.35)$$

where: t_{bl} is found from Eqn.(5.6)

t_{flt} is found from Eqn.(5.18)

It is desirable to use actual man hour data where possible. Lacking such information the data of Table 5.6 (given per flight hour) and/or Fig. 5.5 (given per block hour) may be used as a guide.

NOTE WELL: Aviation Week and Space Technology publishes data on airframe and systems maintenance on a regular basis. The reader should check the latest issues for updated information. Ref.20 contains detailed maintenance data for United Airlines airplanes for the year 1975.

$R_{l_{ap}}$ is the airplane maintenance labor rate per manhour in USD/hr. It is desirable to use actual labor rate data. Lacking such information, Table 5.5 may be used: use the 'mechanic with A/P' data.

V_{bl} is the block speed as found from Eqn.(5.5).

The maintenance labor cost per nautical mile of engines, $C_{lab/eng}$ can be estimated from:

$$C_{lab/eng} = 1.03(1.3)(N_e)(MHR_{meng_{bl}})(R_{l_{eng}})/(V_{bl}) \qquad (5.36)$$

where: the factor 1.3 accounts for cycle-dependent labor as opposed to cycle-independent labor.

N_e is the number of engines per airplane.

$MHR_{meng_{bl}}$ is the number of engine maintenance hours needed per block hour per engine. Whenever possible, it is desirable to use actual manhour data. Lacking such information the data of Table 5.6 and/or Figures 5.6 and 5.7 may be used.

$R_{l_{eng}}$ is the engine maintenance labor rate per man hour in USD/hr. It may be found from Table 5.5: use the 'mechanic with A/P' data.

The cost of maintenance materials for airframe and systems (other than the engines) per nautical mile may be estimated from:

$$C_{mat/ap} = 1.03(C_{mat/apblhr})/V_{bl} \qquad (5.37)$$

where: $C_{mat/apblhr}$ is the airframe and systems maintenance materials cost per airplane block

Table 5.6 Maintenance Manhour Data for Commercial Airplanes

Airplane Type	Annual Utilization in flight hours, $U_{ann_{flt}}$	Maintenance Manhours* per flight hour, MHR_{flthr}	Data from year
Cessna 150	250	0.3	1974
Cessna Skywagon	250	0.5	1974
Beech Kingair 90	350	1.0	1974
Cessna Citation I	400	3.0	1974
McDD DC-9-30	2,900	6.4	1973
B-707-300	3,196	8.4	1973
B-727-100	2,670	7.9	1973
B-727-200	2,800	6.5	1973
B-737-200	2,200	6.6	1973
B-747-100	3,525	14.5	1973
L-1011	1,870	14.1	1974
McDD DC-10-10	2,450	10.9	1973

Data Source: Nicolai, Ref.14

* sum of airframe (+ systems) and engine maintenance hrs:

$$MHR_{flthr} = MHR_{map_{flt}} + MHR_{meng_{flt}}$$

For additional data, see Ref.20 (1976). The distribution between airframe (+ systems) and engine maintenance manhours varies <u>considerably</u> with airplane and engine type, and in particular with engine 'maturity'.

Engine maintenance manhours have been reported anywhere from 10 to 50 percent of the total maintenance manhours per flight hour. For modern jet transports with highly mature engines, the trend is for the engine maintenance manhours to constitute about 10 percent of the total maintenance manhours per flight hour.

Lacking the specific information given in Ref.20 it is suggested to split the total maintenance manhours per flight hour evenly between the airframe (+ systems) and the engine(s).

Figure 5.5 Effect of Airframe Weight on Airframe and Systems Maintenance Man Hours

Figure 5.6 Effect of Takeoff Thrust and Takeoff Shaft Horsepower on Turbine Engine Maintenance Manhours

Figure 5.7 Effect of Engine Weight on Reciprocating Engine Maintenance Man Hours

hour in USD/hr. It may be found from Figure 5.8.

The cost of maintenance materials for the engine(s) per nautical mile may be estimated from:

$$C_{mat/eng} = 1.03(1.3)(N_e)(C_{mat/engblhr})/V_{bl} \qquad (5.38)$$

where: the factor 1.3 accounts for cycle-dependent as opposed to cycle-independent material.

$C_{mat/engblhr}$ is the engine maintenance materials cost per engine per airplane block hour in USD/hr. It may be found from Figures 5.9-5.11. Note that one factor which must be decided on is H_{em},

the attained number of hours between engine overhauls. That number depends on the 'maturity' of the engine in question. Lacking data based on actual operating experience, use:

For turbine engines: H_{em} = 3,000 - 5,000 hrs

For recipr. engines: H_{em} = 1,000 - 2,000 hrs

The cost of the applied maintenance burden per nautical mile, C_{amb}, may be estimated from:

$$C_{amb} = 1.03[(f_{amb/lab})[(MHR_{map_{bl}})(R_{1_{ap}}) + (N_e)(MHR_{eng_{bl}})(R_{1_{eng}})] + (f_{amb/mat})[C_{mat/apblhr} + (N_e)C_{mat/engblhr}]]/(V_{bl}) \qquad (5.39)$$

where: $f_{amb/lab}$ and $f_{amb/mat}$ are overhead distribution factors for labor and for material cost, respectively. These overhead factors are intended to cover such costs as: building, lighting, heating as well as administrative costs associated with airplane maintenance. Lacking actual overhead data it is suggested to use (see top of p.102):

Figure 5.8 Effect of Airframe Price on Airframe and Systems Maintenance Materials Cost

Figure 5.9 Effect of Engine Price on Turbine Engine Maintenance Materials Cost

Figure 5.10 Effect of Engine Price on Reciprocating Engine Maintenance Materials Cost

	Personal	Corporate	Airlines
$f_{amb/lab}$ =	0.80-0.90	0.90-1.00	1.00-1.40
$f_{amb/mat}$ =	0.20-0.30	0.30-0.40	0.40-0.70

The other terms in Eqn.(5.39) were defined before.

Figure 5.11 Factor Depending on Attained Engine Period Between Engine Overhauls

5.2.3 Direct Operating Cost of Depreciation: DOC_{depr}

The direct operating cost of depreciation is broken down as follows:

$$DOC_{depr} = C_{dap} + C_{deng} + C_{dprp} + C_{dav} + C_{dapsp} + C_{dengsp} \quad (5.40)$$

where: C_{dap} is the cost of airplane depreciation without engines and without propellers, avionics systems and spares (= airframe), in USD/nm. See Eqn.(5.41).

C_{deng} is the cost of engine depreciation (as mounted on the airplane but without propellers), in USD/nm. See Eqn.(5.42).

C_{dprp} is the cost of depreciation of propellers, in USD/nm. See Eqn.(5.43).

C_{dav} is the cost of depreciation of avionics systems in USD/nm. See Eqn.(5.44).

C_{dapsp} is the cost of depreciation of airplane spare parts in USD/nm. See Eqn.(5.45).

C_{dengsp} is the cost of depreciation of engine spare parts in USD/nm. See Eqn.(5.46).

The cost of airframe depreciation per nautical mile, C_{dap}, can be estimated from:

$$C_{dap} = \quad (5.41)$$

$$= \frac{(F_{dap})\{(AEP) - (N_e)(EP) - (N_p)(PP) - (ASP)\}}{(DP_{ap})(U_{ann_{bl}})(V_{bl})}$$

where: F_{dap} is an airframe depreciation factor. This factor depends on the perceived resale value of the airplane: see Table 5.7, p.107.

AEP is the airplane estimated price in USD. It can be estimated with the methods of Ch.4 or Appendix A.

N_e is the number of engines per airplane.

EP is the engine price (per engine) in USD. It can be found from Appendix B.

N_p is the number of propellers per airplane.

PP is the price per propeller in USD. It can be determined from Appendix B.

ASP is the avionics systems price per airplane in USD. This can be found from Appendix C.

DP_{ap} is the airplane depreciation period. This depends on the operator's business strategy. Table 5.7 may be used as a guide.

$U_{ann_{bl}}$ is the annual utilization in block hours. It may be found from page 76.

V_{bl} is the block speed in nm/hr as found from Equation (5.5).

The cost of engine depreciation per nautical mile, C_{deng}, may be determined from:

$$C_{deng} = \frac{(F_{deng})(N_e)(EP)}{(DP_{eng})(U_{ann_{bl}})(V_{bl})} \qquad (5.42)$$

where: F_{deng} is the engine depreciation factor. This factor depends on the perceived resale value of each engine: see Table 5.7, p.107.

DP_{eng} is the engine depreciation period. This depends on the operator's business strategy. The data of Table 5.7 may serve as a guide.

The cost of depreciation of propellers, C_{dprp}, in USD/nm can be found from:

$$C_{dprp} = \frac{(F_{dprp})(N_p)(PP)}{(DP_{prp})(U_{ann_{bl}})(V_{bl})} \qquad (5.43)$$

where: F_{dprp} is the propeller depreciation factor. This factor depends on the perceived resale value of each propeller: see Table 5.7, p.107.

DP_{prp} is the propeller depreciation period. This depends on the operator's business strategy. The data of Table 5.7 may serve as a guide.

The cost of depreciation of avionics systems, C_{dav}, in USD/nm may be estimated from:

$$C_{dav} = \frac{(F_{dav})(ASP)}{(DP_{av})(U_{ann_{bl}})(V_{bl})} \qquad (5.44)$$

where: F_{dav} is the avionics systems depreciation factor. This factor depends on the perceived resale value of the avionics systems. Experience has indicated little or no resale value as shown in Table 5.7, p.107.

DP_{av} is the avionics systems depreciation period. This depends on the operator's business strategy AND on federal regulations (in terms of requiring new types of equipment: TCAS is an example). The data of Table 5.7, p.107 may serve as a guide.

The cost of depreciation of airplane spare parts, C_{dapsp}, in USD/nm may be estimated from:

$$C_{dapsp} = \frac{(F_{dapsp})(F_{apsp})\{(AEP) - (N_e)(EP)\}}{(DP_{apsp})(U_{ann_{bl}})(V_{bl})} \qquad (5.45)$$

where: F_{dapsp} is the airplane spare parts depreciation factor. This factor depends on the perceived resale value of the airplane spare parts. The data of Table 5.7, p.107 may serve as a guide.

F_{apsp} is the airplane spare parts factor. It is equal to the ratio of airplane spare parts cost to the cost of the airplane minus engines. This factor depends on repair and maintenance operating experience with the airplane. Lacking actual data it is suggested to use: $F_{apsp} = 0.10$.

DP_{apsp} is the airplane spare parts depreciation period. This depends on the operator's business strategy. The data in Table 5.7 may serve as a guide.

Note: the airplane spare parts factor may be asmed to account also for spare parts needs for propellers and avionics systems.

The cost of engine spare parts depreciation, C_{dengsp}, in USD/nm can be estimated from:

$$C_{dengsp} = \frac{(F_{dengsp})(F_{engsp})(N_e)(EP)(ESPPF)}{(DP_{engsp})(U_{ann_{bl}})(V_{bl})} \qquad (5.46)$$

where: F_{dengsp} is the engine spare parts depreciation factor. It depends on the perceived resale value of the engine spare parts. The data of Table 5.7 should serve as a guide.

F_{engsp} is the engine spare parts factor. It is equal to the ratio of the cost of engine spare parts to the cost of one engine. The engine manufacturer usually recommends a ratio for each given engine depending on experience in the field. For preliminary cost estimating purposes it is suggested to use: $F_{engsp} = 0.50$.

ESPPF is the engine spare parts price factor. It depends on the engine manufacturer's policy in pricing spare parts. If all engine components could be purchased at the same price as a fully assembled engine, the value of ESPPF would be 1.0. It is suggested to use: ESPPF = 1.50.

DP_{engsp} is the depreciation period for engine spare parts. It depends on the operator's business strategy. The data of Table 5.7 may be used as a guide.

Table 5.7 Suggested Depreciation Periods and Depreciation Factors for Airframe and Equipment

Item	Suggested Depreciation Period	Residual Value in Percent	Depreciation Factor*
Airframe	$DP_{ap} = 10$	15	$F_{dap} = 0.85$
Engines	$DP_{eng} = 7$	15	$F_{deng} = 0.85$
Propellers	$DP_{prp} = 7$	15	$F_{dprp} = 0.85$
Avionics	$DP_{av} = 5$	0	$F_{dav} = 1.00$
Airplane Spares	$DP_{apsp} = 10$	15	$F_{dapsp} = 0.85$
Engine Spares	$DP_{engsp} = 7$	15	$F_{dengsp} = 0.85$

* Depreciation factor =

= {1 - (Residual Value)/(Original Price)}

5.2.4 Direct Operating Cost of Landing Fees, Navigation Fees and Registry Taxes: DOC_{lnr}

There is no uniform agreement whether the operating costs caused by landing fees, navigation fees and registry taxes are to be considered direct or indirect operating costs. In this text the decision has been made to include them as direct operating costs.

The direct operating cost of landing fees, navigation fees and various taxes can be broken down in the following manner:

$$DOC_{lnr} = C_{lf} + C_{nf} + C_{rt} \tag{5.47}$$

where: C_{lf} is the direct operating cost due to landing fees in USD/nm. This can be found from:

$$C_{lf} = (C_{aplf})/\{(V_{bl})(t_{bl})\} \tag{5.48}$$

where: C_{aplf} is the airplane landing fee per landing. This is a function of

airplane takeoff weight (it is assumed, that after landing, it will take off again!) AND of local airport authority policy. Lacking actual landing fee data it is suggested to use:

$$C_{aplf} = 0.002W_{TO} \text{ USD/lbs} \quad (5.49)$$

V_{bl} is the block speed in nm/hr as found from Eqn.(5.5).

t_{bl} is the block time as defined by Equation (5.6).

An alternate method is to consider C_{lf} to be a simple fraction of the DOC. Based on data from Ref.20 it is suggested to use:

$$C_{lf} = (f_{lf})(DOC) \quad (5.50)$$

where: f_{lf} is a factor which depends on airplane size. This factor depends on local airport authorities which set these landing fees. Lacking actual data it is suggested to use:

$$f_{lf} = 0.036 + 4(10^{-8})W_{TO} \quad (5.51)$$

where: W_{TO} is the airplane takeoff weight in lbs.

C_{nf} is the cost of navigation fees in USD/nm. It may be estimated from:

$$C_{nf} = (C_{apnf})/\{(V_{bl})(t_{bl})\} \quad (5.52)$$

where: C_{apnf} is the navigation fee charged per airplane per flight, expressed in USD/flight. This fee varies from route to route and from country to country.

Lacking actual data it is suggested to use:

For operations in the USA:

C_{apnf} = 0 USD/flight

For operations outside the USA:

C_{apnf} = 10.00 USD/flight

C_{rt} is the direct cost of registry taxes expressed in USD/nm. Based on data from Ref.20 it is suggested to use:

$$C_{rt} = (f_{rt})(DOC) \tag{5.53}$$

where: f_{rt} is a factor which depends on airplane size. This factor depends on state and country governments which set these registry taxes. Lacking actual data it is suggested to use:

$$f_{rt} = 0.001 + (10^{-8})W_{TO} \tag{5.54}$$

where: W_{TO} is the airplane takeoff weight in lbs.

5.2.5 Direct Operating Cost of Financing: DOC_{fin}

The direct operating cost of financing the airplane in USD/nm, DOC_{fin}, depends on how an operator is financing his fleet of airplanes. Operators can elect to borrow money to pay for airplane(s) and spare parts as well as to finance their operation. They frequently elect to lease part or all of their flying equipment.

Even when an operator decides to use his own money to finance the airplanes and their operation, there is a cost associated with that: interest.

Methods for estimating financing cost are judged to be beyond the scope of this text. However, the reader may wish to account for financing costs by the following simple 'rule-of-thumb':

$$DOC_{fin} = 0.07(DOC) \tag{5.55}$$

This 'rule-of-thumb' is based on the observation that financing costs typically run to 7 percent of the total DOC as found from Eqn.(5.19).

5.3 METHOD FOR ESTIMATING THE INDIRECT OPERATING COST OF COMMERCIAL AIRPLANES: IOC

The purpose of this section is to provide a method for estimating the indirect operating cost (IOC) in USD/nm, incurred while operating commercial airplanes.

Table 5.8 presents an overview of cost components and cost elements which contribute to the indirect operating cost per nautical mile.

NOTE: The right hand side of Table 5.8 indicates which IOC elements apply to various airplane types.

The indirect cost associated with airplane operations vary significantly from one operator to another. Also, as will be seen, the airplane designer has very little influence over this cost category. The method proposed here to estimate the indirect operating cost assumes that the IOC can be expressed as a simple fraction of DOC:

$$IOC = f_{ioc}(DOC) \tag{5.56}$$

where: f_{ioc} is the fraction of DOC by which IOC is estimated. When reviewing published data on the relationship between IOC and DOC, it is found that f_{ioc} is strongly dependent on block distance, R_{bl}, as well as on the type of airplane. Figure 5.12 provides some data for this fraction.

Lacking actual IOC data it is suggested to use:

For transport operations: f_{ioc} as in Fig.5.12

For personal owner and for corporate airplanes: $f_{ioc} = 0$.

As will be seen, airplane designers have very little design leverage over the magnitude of the indirect operating costs of an airplane operator. Despite this fact, the author believes that it is desirable for aeronautical engineering students to develop at least a basic understanding of those factors which determine IOC. For that reason a discussion of IOC breakdown is presented next. For a discussion of IOC trends, see Section 5.5.

Table 5.8 Overview of Indirect Operating Cost (IOC)

INDIRECT OPERATING COST ~ IOC ~ USD/nm		AIRLINE	CORPORATE	PERSONAL
IOC COMPONENTS	IOC ELEMENTS			
PASSENGERS ~ IOC_{pax} (5.58)	MEALS ~ C_{mls}	✓	✓	NO
	INSURANCE ~ $C_{pax/ins}$	✓	✓	✓
	CABIN ATTENDANTS ~ C_{cat}	✓	✓	NO
	PASSENGER HANDLING, BAGGAGE HANDLING, SALES & RESERVATIONS, SECURITY, MISC ~ $C_{pax/gen}$	✓	NO	NO
STATION ~ IOC_{sta} (5.59)	MAINTENANCE OF GROUND EQUIPM'T & FACILITIES ~ C_{mgef}	✓	✓	NO
	DEPRECIATION OF GROUND EQUIPM'T & FACILITIES ~ C_{dgef}	✓	✓	NO
AIRPLANE SERVICE, CONTROL AND FREIGHT ~ IOC_{ascf} (5.60)	AIRPLANE SERVICE ~ C_{aps}	✓	✓	✓
	AIRPLANE CONTROL ~ C_{apc}	✓	✓	NO
	FREIGHT HANDLING ~ C_{frt}	✓	NO	NO
PROMOTION, SALES AND ENTERTAINMENT ~ IOC_{pse} (SUB-SECT. 5.3.4)	COMMISSIONS TO TRAVEL AGENCIES,	✓	NO	NO
	PUBLICITY AND ADVERTISING,	✓	NO	NO
	ENTERTAINMENT	✓	NO	NO
GENERAL AND ADMINISTRATIVE ~ IOC_{gaa} (SUB-SECT. 5.3.5)	ADMINISTRATIVE, ACCOUNTING AND CORPORATE STAFF + FACILITIES	✓	✓	NO

Figure 5.12 Effect of Block Distance on the Ratio of IOC to DOC

The indirect operating cost per nautical mile for a commercial airplane is broken down into the following IOC cost components:

$$IOC = IOC_{pax} + IOC_{sta} + IOC_{ascf} + \\ + IOC_{pse} + IOC_{gaa} \qquad (5.57)$$

where: IOC_{pax} is the indirect operating cost for passenger services, expressed in USD/nm: see Sub-section 5.3.1.

IOC_{sta} is the indirect cost for maintaining and depreciating ground equipment and ground facilities, expressed in USD/nm: see Sub-section 5.3.2.

IOC_{ascf} is the indirect operating cost for airplane and traffic servicing, control and for freight, expressed in USD/nm: see Sub-section 5.3.3.

IOC_{pse} is the indirect operating cost for promotion, sales and entertainment, expressed in USD/nm: see Sub-section 5.3.4.

IOC_{gaa} is the indirect operating cost for general administrative expenses, expressed in USD/nm: see Sub-section 5.3.5.

Sub-sections 5.3.1 through 5.3.5 contain breakdowns for these indirect operating cost components.

5.3.1 Indirect Operating Cost for Passenger Service: IOC_{pax}

The indirect operating cost for passenger services may be broken down as follows:

$$IOC_{pax} = C_{mls} + C_{pax/ins} + C_{cat} + C_{pax/gen} \qquad (5.58)$$

where: C_{mls} is the cost for meal service. This depends on the routes flown and on the level of service which the operator wants to provide for his passengers.

The airplane designer has minor leverage over this cost item. The design of the airplane

galleys constitutes this leverage. No quantitative method for estimating this leverage is available.

$C_{pax/ins}$ is the cost for passenger insurance. There exists an international agreement between airline operators on the amount of passenger insurance that must be carried: the so-called Warsaw Convention. The cost of this insurance to an airline varies with the underwriter of that insurance and with the accident history of the airline.

Other than designing the airplane to be safer than previous airplanes, the airplane designer has no design leverage over this cost element.

C_{cat} is the cost for cabin attendants. The number of cabin attendants which must be carried aboard an airplane is regulated by the FAA. Part III, p.59 shows how many cabin attendants are required. By using salary data from Table 5.5 it is possible to determine the actual cost of flight attendants.

The designer has no leverage over this cost element.

$C_{pax/gen}$ is the cost associated with the following items:

* Passenger handling
* Passenger baggage handling
* Sales and reservations
* Security
* Miscellaneous passenger costs

No reliable quantitative method is available for estimating this cost element. The designer of the airplane does have some design leverage over the following items:

* design of baggage space and the ease of access to baggage handling personnel or equipment

* boarding and unboarding of passengers is affected significantly by the interior cabin layout.

5.3.2 Indirect Operating Costs for Station Operation: IOC_{sta}

The indirect operating cost for station operation can be broken down as follows:

$$IOC_{sta} = C_{mgef} + C_{dgef} \tag{5.59}$$

where: C_{mgef} is the cost of maintaining ground equipment and facilities

C_{dgef} is the cost of depreciation of ground equipment and facilities

The designer has some leverage over these cost elements: requirements for ground facilities and for ground equipment are dependent on the type of systems which need to be accessed for servicing while the airplane is at the gate. No quantative method is available to estimate this cost element.

5.3.3 Indirect Operating Cost for Airplane Service, Airplane Control and Freight: IOC_{ascf}

The indirect operating cost for aircraft service, control and freight is broken down into the following cost elements:

$$IOC_{ascf} = C_{aps} + C_{apctrl} + C_{frt} \tag{5.60}$$

where: C_{aps} is the cost of airplane service. It depends on how well the airplane was designed from a gate servicing viewpoint: see Part III, pages 82-83.

If all required service vehicles cannot have simultaneous access to the airplane then the cost of servicing will increase.

C_{apctrl} is the cost of airplane control. It depends on the amount of equipment and personnel needed to push (or maneuver) the airplane away from the gate or help it approach the gate.

C_{frt} is the cost associated with handling freight. This freight is NOT passenger baggage. Many airlines will try to carry extra freight when possible to bring in additional revenue.

Depending on how this freight must be handled (people, containers, handling equipment) this cost component can be high or low. The airplane designer has definite design leverage over this cost element. Sub-section 3.3.5 in Part III provides details on the layout of cargo/baggage holds.

5.3.4 Indirect Operating Cost for Promotion, Sales and Entertainment: IOC_{pse}

The following activities contribute to this indirect cost component:

* Commissions to travel agencies
* Publicity and advertising campaigns
* Entertainment

The airplane designer has no leverage over this item unless there is something about the appearance or public appeal of the airplane that causes this cost to increase.

No quantitative method for estimating this cost component is avalable.

5.3.5 Indirect Operating Cost for General and Administrative Cost: IOC_{gaa}

The following activities contribute to this indirect cost component:

* requirements for administrative and accounting personnel as well as for their facilities
* requirements for corporate staffers and their facilities

The mentality of upper level management has a major effect on the magnitude of this cost item: the degree of fanciness in the corporate offices and the various perks associated with company officers all influence this cost item. No quantitative method is available for estimating this cost component. The airplane designer certainly has no design leverage over this cost component.

5.4 EXAMPLE APPLICATION

The purpose of this section is to present an example application of the methods for estimating program operating cost, C_{OPS}, direct operating cost, DOC, and indirect operating cost, IOC, of Sections 5.1, 5.2 and 5.3 respectively. The example has been constructed around the Ourania jet transport, to be consistent with the examples given in Sections 3.8 and 4.6. The data will be computed for calendar year 1990.

The material is organized as follows:

5.4.1 Ourania Mission Data

5.4.2 Calculation of Direct Operating Cost: DOC

5.4.3 Calculation of Indirect Operating Cost: IOC

5.4.4 Calculation of Program Operating Cost: C_{OPS}

5.4.5 Discussion of Results

5.4.1 Ourania Mission Data

Before operating costs can be determined, some baseline data about the airplane mission are needed. These baseline data are essentially those dealing with block distance, block speed and block times. Definitions and formulas for these factors were given in Section 5.1.

The block distance for the Ourania will be assumed to be equal to the 1,500 nm range requirement specified in Table 2.18 of Part I: R_{bl} = 1,500 nm.

Block time, t_{bl}, is determined from Eqn.(5.6). This in turn requires the ground maneuvering time, t_{gm}, the time-to-climb, t_{cl}, the time to cruise, t_{cr}, and the time to descend, t_{de}. These times will be found next.

The ground maneuvering time, t_{gm}, is obtained from Eqn.(5.7). This in turn requires the takeoff weight, W_{TO}, which is: W_{TO} = 123,683 from p.11, Part V.

Therefore: $t_{gm} = 0.51(0.123683) + 0.125 = 0.19$ hrs

The time-to-climb, t_{cl}, is found from p.56 of Part I as: $t_{cl} = 14$ min. $= 0.23$ hrs

The time-to-cruise, t_{cr}, is found from Eqn.(5.8a). This in turn requires the block distance, R_{bl}, the distance-to-climb, R_{cl}, the distance-to-descend, R_{de}, the maneuvering distance, R_{man}, and the cruise speed, V_{cr}. These distances will be found next.

The distance-to-climb is found from p.56, Part I as: $R_{cl} = 64$ nm.

The distance-to-descend is found from Eqn.(5.10). The average speed during the descent will be assumed to be 250 kts. For the Ourania it will also be assumed that the average descent rate from 35,000 ft is 2,500 ft/min so that:

$t_{de} = 35{,}000/2{,}500 = 14$ min $= 0.23$ hrs. Therefore:

$R_{de} = 250(0.23) = 58$ nm

The maneuvering distance, R_{man}, is determined from Eqn.(5.11). The maneuvering speed is set equal to the cruise speed: $V_{man} = V_{cr} = 473$ kts (p.56, Part I).

The maneuvering time, t_{man}, follows from Eqn.(5.12):

$t_{man} = 0.25(0.123683) + 0.0625 = 0.093$ hrs

Therefore:

$R_{man} = 473(0.093) = 44$ nm

The cruise time now follows from Eqn.(5.8a):

$t_{cr} = \{1.06(1{,}500) - 64 - 58 + 44\}/473 =$

$= 1{,}512/473 = 3.20$ hrs

The block time now follows from Eqn.(5.6):

$t_{bl} = 0.19 + 0.23 + 3.20 + 0.23 = 3.85$ hrs

The block speed can now be found from Eqn.(5.5) as:

$$V_{bl} = 1,500/3.85 = 390 \text{ kts}$$

The annual utilization in block hours may be computed from Eqn.(5.14a) as:

$$U_{ann_{bl}} = (10)^3[3.4546(3.85) + 2.994 +$$

$$- \{12.289(3.85)^2 - 5.6626(3.85) + 8.964\}^{1/2}] = 3.282 \text{ hrs}$$

<u>NOTE:</u> this annual utilization amounts to an average daily utilization of about 9 hours. For a well run airline, on 1,500 nm stage lengths, this is a reasonable number. Table 5.3 also substantiates this.

5.4.2 Calculation of Direct Operating Cost: DOC

The Ourania direct operating cost in USD/nm is computed with Eqn.(5.19). The five cost components of DOC are detailed in sub-sections 5.2.1 through 5.2.5. The calculations for these cost components are presented in sub-sub-sections 5.4.2.1 through 5.4.2.5. The DOC is computed in sub-sub-section 5.4.2.6.

5.4.2.1 Direct operating cost of flying: DOC_{flt}

This cost is found with Eqn.(5.20).

<u>Crew Cost:</u> C_{crew}

This cost may be estimated with Eqn.(5.21).

The mission specification for the Ourania jet transport of Table 2.18, Part I, p.55 specifies a crew of two. Therefore, in Eqn.(5.21):

$$n_{c_1} = 1 \text{ and also: } n_{c_2} = 1$$

From page 83: $K_j = 0.26$

The crew salaries for the captain (n_{c_1}) and for the first officer (n_{c_2}) are determined from Table 5.5. The annual flight hours for each are found from p.84:

SAL_1 = 85,000 USD/yr $\qquad$ AH_1 = 800 hrs

SAL_2 = 50,000 USD/yr $\qquad$ AH_2 = 800 hrs

These data are assumed to be applicable for 1990.

The travel expense factor, TEF, according to p.84 is:

$$TEF_1 = TEF_2 = 7 \text{ USD/blhr}$$

The crew cost per nautical mile can now be estimated from Eqn.(5.21) as:

$$C_{crew} = \{(1.26/390)(85,000/800) + 7/390\} +$$

$$+ \{(1.26/390)(50,000/800) + 7/390\} =$$

$$= 0.361 + 0.220 = 0.581 \text{ USD/nm}$$

<u>Fuel and oil cost:</u> C_{pol}

This cost is determined with Eqn.(5.30)

The block fuel used, $W_{F_{bl}}$, is the same as the mission fuel used, $W_{F_{used}}$, which in turn is given by Eqn.(2.14) of Part I. Reference is made to the example calculation for fuel used in sub-section 2.6.2 of Part I. Note that mission phases 6 and 8 as depicted in the mission profile of Table 2.18 (Part I) should <u>not</u> be counted as normal operational fuel used. Therefore, the M_{ff} calculation of p.58 of Part I must be modified to:

$$M_{ff} = (0.992)(1)(0.990)(1)(0.909)(0.980)(0.995)*$$

Mission Phase: 9 8 7 6 5 4 3

$$*(0.990)(0.990) = 0.853$$

Mission Phase: 2 1

The mission phase numbers refer to Table 2.18 in Part I. This yields for the operational mission fuel used, $W_{F_{used}}$, with Eqn.(2.14) of Part I:

$$W_{F_{bl}} = (1 - 0.853)(123,683) = 18,181 \text{ lbs}$$

The 1990 price of Jet A fuel will be assumed to be 0.65 USD/gallon (Figure 5.3), while the density (p.88) is

6.74 lbs/gallon. Therefore, with Eqn.(5.30):

$$C_{pol} = (1.05)(18,181/1,500)(0.65/6.74) =$$

$$= 1.227 \text{ USD/nm}$$

<u>Insurance cost:</u> C_{ins}

This cost is found from Eqn.(5.32):

$$C_{ins} = 0.02(DOC)$$

The direct operating cost of flying may now be computed from Eqn.(5.20) as:

$$DOC_{flt} = 0.581 + 1.227 + 0.02(DOC) =$$

$$= 1.808 + 0.02(DOC) \text{ USD/nm}$$

<u>5.4.2.2 Direct operating cost of maintenance:</u> DOC_{maint}

This cost is found from Eqn.(5.33).

<u>Maintenance labor cost for airframe and systems:</u> $C_{lab/ap}$

This cost may be calculated from Eqn.(5.34). The number of maintenance manhours per blockhour for airframe and systems, $MHR_{map_{bl}}$, may be found from Figure 5.5. This is seen to depend on the airframe weight, W_A, defined as:

$$W_A = W_E - W_{pwrplt} = 65,133 - 9,891 = 55,242 \text{ lbs},$$

where the weight data were found on p.11 of Part V.

Figure 5.5 now yields:

$$MHR_{map_{bl}} = 3.0 + 0.067(55,242) = 6.7 \text{ mhrs/blhr}$$

The airplane maintenance labor rate per manhour is assumed from Table 5.5 to be: $R_{l_{ap}} = 16$ USD/mhr.

Therefore, with Eqn.(5.34):

$$C_{lab/ap} = 1.03(6.7)(16)/390 = 0.283 \text{ USD/nm}$$

<u>Maintenance labor cost for engines:</u> $C_{lab/eng}$

This cost is found from Eqn.(5.36). The number of engine maintenance hours per engine, per block hour, $MHR_{meng_{bl}}$, may be determined from Figure 5.6. It is seen

to depend on the takeoff thrust per engine, (T_{TO}/N_e), and on the attained period between engine overhauls, H_{em}.

The takeoff thrust per engine is found from page 184 in Part I as:

$$(T_{TO}/N_e) = 47,625/2 = 23,813 \text{ lbs}$$

The attained overhaul period for the engine, H_{em}, is assumed to be 3,000 hrs (see p.98). For a new engine, this is not an unreasonably high number. Therefore, from the equation in Figure 5.6:

$$MHR_{meng_{bl}} = [\{0.718 + 0.0317(23,813)\}* *(1,100/3,000) + 0.10] = 0.64 \text{ mhrs/blhr}$$

The hourly labor rate for engine maintenance will be assumed to be the same as that for the airframe:

$R_{l_{eng}} = 16$ USD/hr so that with Eqn.(5.36):

$$C_{lab/eng} = 1.03(1.3)(2)(0.64)(16)/390 = 0.070 \text{ USD/nm}$$

<u>NOTE:</u> it is of interest to 'check' whether the total manhours for maintenance are reasonable. This total is: 6.7 + 2(0.64) = 7.98 mhrs/blhr. This can be translated into mhrs/flthr with Eqn.(5.35). The value for t_{flt} is found from Eqn.(5.18) as:

$$t_{flt} = 0.23 + 3.20 + 0.23 = 3.66 \text{ hrs}$$

The block time, t_{bl}, was found to be 3.85 hours, so that the maintenance manhours per flight hour follow from Eqn.(5.35) as:

$$\text{MHR per flt hour} = (\text{MHR per bl hour})(t_{bl})/(t_{flt}) = = 7.98(3.85/3.66) = 8.4 \text{ mhrs/flthr.}$$

This number appears reasonable when compared with the data in Table 5.6.

Maintenance materials cost for airplane and systems: $C_{mat/ap}$

This cost can be found from Eqn.(5.37). The cost of airplane maintenance materials per blockhour, $C_{mat/apblhr}$, is determined from Figure 5.8. It depends on the airframe price, AFP:

$$AFP = AEP - N_e(EP) =$$

$$= 22,576,000 - 2(1,558,607) = USD\ 19,458,786$$

where: the airplane and engine prices are found from pages 62 and 58 (Section 4.6), respectively.

With Figure 5.8:

$$C_{mat/apblhr} = 30.0 + 0.79(194.58786) =$$

$$= 183.72\ USD/blhr$$

where: the 1990 to 1989 CEP ratio has been assumed to be equal to 1.0.

Therefore, with Eqn.(5.37):

$$C_{mat/ap} = 1.03(183.72)/390 = 0.485\ USD/nm$$

Maintenance materials cost for engines: $C_{mat/eng}$

This cost is computed from Eqn.(5.38). The cost of engine maintenance materials per blockhour, $C_{mat/engblhr}$, may be found from Figures 5.9 and 5.11. It was already assumed that the attained overhaul period for the engines is H_{em} = 3,000 hrs (See p.122). Thus, from Figure 5.11:

$K_{H_{em}} = 1.40$.

Then, with the equation in Figure 5.9:

$$C_{mat/engblhr} = \{0.0543(1,558.607)(1.5) +$$

$$- 0.47\}/1.40 = 90.34\ USD/blhr$$

Thus, with Eqn.(5.38):

$$C_{mat/eng} = 1.03(1.3)(2)(90.34)/390 = 0.620\ USD/nm$$

Applied maintenance burden cost: C_{amb}

This cost may be found from Eqn.(5.39). The overhead distribution factors are selected from the top of page 102 as:

$f_{amb/lab} = 1.30$ $\qquad$ $f_{amb/mat} = 0.60$

Therefore from Eqn.(5.39):

$$C_{amb} = 1.03[1.30[(6.7)(16) + 2(0.64)(16)] + $$
$$+ 0.60(183.72 + 2(90.34))]/390 = 1.016 \text{ USD/nm}$$

Total direct maintenance cost: DOC_{maint}

The total direct cost for maintenance now follows with Eqn.(5.33) as:

$$DOC_{maint} = 0.283 + 0.070 + 0.485 + 0.620 + 1.016 =$$
$$= 2.474 \text{ USD/nm}$$

5.4.2.3 Direct operating cost of depreciation: DOC_{depr}

This cost is found with Eqn.(5.40).

Airplane Depreciation Cost: C_{dap}

This cost may be computed with Eqn.(5.41).

The following terms are found in Table 5.7:

Airframe depreciation factor: $F_{dap} = 0.85$

Airplane depreciation period: $DP_{ap} = 10$ yrs

The following price terms were found in Section 4.6:

Airplane estimated price: AEP = USD 22,576,000
Engine price: EP = USD 1,558,607
Avionics systems price: ASP = USD 2,670,000

Number of propellers: $N_p = 0$. Now, with Eqn.(5.41):

$$C_{dap} = \frac{0.85(22,576,000 - 2(1,558,607) - 2,670,000)}{10(3,282)(390)} =$$

$$= 1.115 \text{ USD/nm}$$

Engine Depreciation Cost: C_{deng}

This cost is determined with Eqn.(5.42).

The following terms are found in Table 5.7:

Engine price depreciation factor: $F_{deng} = 0.85$

Engine depreciation period: $DP_{eng} = 7$ yrs

With Eqn.(5.42):

$C_{deng} = \{0.85(2)(1,558,607)\}/\{7(3,282)(390)\} =$

$= 0.296$ USD/nm

Propeller Depreciation Cost: C_{dprp}

This cost is zero: the Ourania has no propellers.

Avionics Systems Depreciation Cost: C_{dav}

This cost may be found with Eqn.(5.44).

The following terms are found in Table 5.7:

Avionics systems depreciation factor: $F_{dav} = 1.00$

Avionics systems depreciation period: $DP_{av} = 5$ yrs

With Eqn.(5.44):

$C_{dav} = \{1.0(2,670,000)\}/\{5(3,282)(390)\} =$

$= 0.417$ USD/nm

Airplane Spare Parts Depreciation Cost: C_{dapsp}

This cost may be found with Eqn.(5.45).

The following terms are found from Table 5.7:

Airplane spare parts depreciation factor:

$F_{dapsp} = 0.85$

Airplane spare parts depreciation period:

$DP_{apsp} = 10$ yrs

The airplane spare parts factor, F_{apsp}, is assumed to have the value 0.10: see p.105.

With Eqn.(5.45):

$$C_{dapsp} = \{0.85(0.1)(19,458,786)\}/\{10(3,282)(390)\} =$$

$$= 0.129 \text{ USD/nm}$$

<u>Engine Spare Parts Depreciation Cost:</u> C_{dengsp}

This cost may be estimated from Eqn.(5.46).

The following terms are found in Table 5.7:

Engine spare parts depreciation factor:

$$F_{dengsp} = 0.85$$

Engine spare parts depreciation period:

$$DP_{engsp} = 7 \text{ yrs}$$

The engine spare parts factor, F_{engsp}, will be set equal to 0.50 as suggested on p.106. The engine spare parts price factor, ESPPF will be set equal to 1.50 as suggested on page 106. Therefore, with Eqn.(5.46):

$$C_{dengsp} = \{0.85(0.50)(2)(1,558,607)*$$

$$*(1.5)\}/\{7(3,282)(390)\} = 0.222 \text{ USD/nm}$$

<u>Total Direct Operating Cost of Depreciation:</u> DOC_{depr}

From Eqn.(5.40):

$$DOC_{depr} = 1.115 + 0.296 + 0 + 0.417 + 0.129 +$$

$$+ 0.222 = 2.181 \text{ USD/nm}$$

5.4.2.4 Direct operating cost of landing fees, navigation fees and registry taxes: DOC_{lnr}

This cost may be determined with Eqn.(5.47).

Cost of Landing Fees: C_{lf}

This cost is estimated with Eqn.(5.48).

The landing fee per landing, C_{aplf}, is determined from Eqn.(5.49) as:

$$C_{aplf} = 0.002(123,683) = \text{USD } 247.37$$

Therefore, with Eqn.(5.48):

$$C_{lf} = 247.37/[(390)(3.85)] = 0.165 \text{ USD/nm}$$

Cost of Navigation Fees: C_{nf}

According to Eqn.(5.52) and p.109, for domestic operations:

$$C_{nf} = 0$$

Cost of Registry Taxes: C_{rt}

This cost is estimated with Eqn.(5.53).

The factor f_{rt} is determined from Eqn.(5.54) as:

$$f_{rt} = 0.001 + 0.00123683 = 0.00224$$

Therefore, with Eqn.(5.53):

$$C_{rt} = 0.00224(DOC)$$

Total Direct Operating Cost of Landing Fees, Navigation and Registry Taxes: DOC_{lnr}

With Eqn.(5.47):

$$DOC_{lnr} = 0.165 + 0 + 0.00224(DOC) =$$

$$= 0.165 + 0.00224(DOC) \text{ USD/nm}$$

5.4.2.5 Direct operating cost of financing: DOC_{fin}

Lacking better information, Eqn.(5.55) will be used:

$$DOC_{fin} = 0.07(DOC)$$

5.4.2.6 Total direct operating cost: DOC

The DOC cost components of sub-sub-sections 5.4.2.1 through 5.4.2.5 can be combined with Eqn.(5.19) to yield:

$$DOC = 1.808 + 0.02(DOC) + 2.474 + 2.180 + 0.165 +$$

$$+ 0.00224(DOC) + 0.07(DOC) =$$

$$= 6.627 + 0.09224(DOC) \text{ USD/nm}$$

or: $DOC = 7.300$ USD/nm

5.4.3 Calculation of Indirect Operating Cost: IOC

The Ourania indirect operating cost may be estimated with Eqn.(5.56) where the factor f_{ioc} is extrapolated from Figure 5.12 as:

$$f_{ioc} = 0.50$$

Therefore: $IOC = 0.50(DOC) = 0.5(7.300) =$

$$= 3.650 \text{ USD/nm}$$

5.4.4 Calculation of Program Operating Cost: C_{OPS}

The Ourania program operating cost may be estimated from Eqns (5.1) - (5.3).

It will be assumed that a total of 500 airplanes will be acquired and that all operators have the same route system:

$$N_{acq} = 500$$

The total annual block miles per airplane, $R_{bl_{ann}}$, may be found from Eqn.(5.4):

$$R_{bl_{ann}} = (390)(3,282) = 1,279,980 \text{ nm}$$

The number of years during which the airplane is being operated will be assumed to be equal to the airframe depreciation period, which is 10 years:

$$N_{yr} = 10 \text{ yrs}$$

Therefore, with Eqns (5.1) - (5.3):

$$C_{OPS} = (7.300)(1,279,980)(10)(500) +$$

$$+ (3.650)(1,279,980)(10)(500) =$$

$$= \text{USD } 70,078,905,000$$

5.4.5 Discussion of Results

The significance of the numerical results which were obtained in this section are discussed as follows:

5.4.5.1 Discussion of DOC results
5.4.5.2 Discussion of Program Cost results
5.4.5.3 Discussion of DOC trade studies

5.4.5.1 Discussion of DOC results

To determine the realism of the DOC predictions for the Ourania transport, several of its DOC components will be compared with DOC data of existing airplanes.

Table 5.9 provides the information. Aviation Week and Space Technology publishes DOC data regularly. The data in Table 5.9 were published in the August 28, 1989 issue, however: these data are for the year 1988.

Note that the data in Table 5.9 are given in USD per block hour. The Ourania DOC predictions of sub-section 5.4.2 are in USD per nautical mile. To convert it was necessary to multiply by the block speed: 390 kts.

The Ourania data are for the year 1990. The CEF-ratio from 1990 to 1988 is 1.003 according to Figure 2.7. This is so close to 1.0 that the Ourania data may be compared with the 1988 data for comparable airplanes.

The Ourania estimates in Table 5.9 appear to be very reasonable when compared to the data for comparable airplanes. However, the following question should be asked:

> 'Can a new design, such as the Ourania, be viable if its total DOC is not significantly better than that of existing airplanes?'

To address this question, the following point should be made: the DOC total for the Ourania is inflated by the selection of a 10 year depreciation period. Figure 5.13 presents the results of a trade study which shows the effect of the depreciation period on DOC. The depreciation period used by the airlines in conjunction with the data in Table 5.9 is not known. The author believes that this

Table 5.9 Direct Operating Cost Data in USD/blhr for Several Jet Transports

Source of airline data: Aviation Week and Space Technology, August 28, 1989*

Type No. of Airlines No. of Airplanes	Relative Cost	Crew	POL	Total Flight DOC	Airplane Maint.	Engine Maint.	Total Maint.	Total DOC
Boeing 727-200	High	568	834	1,633	266	195	539	2,306
11 airlines	Low	217	674	1,091	108	49	341	1,711
809 airplanes	Average	442	711	1,286	182	118	445	1,891
Boeing 737	High	449	549	1,707	269	178	484	2,250
13 airlines	Low	165	399	749	43	14	159	1,159
661 airplanes	Average	282	446	1,104	152	84	318	1,540
Boeing 757	High	547	574	1,546	152	241	415	2,136
4 airlines	Low	187	509	1,075	80	11	190	1,681
98 airplanes	Average	410	549	1,301	106	157	334	1,933
McDD MD-80	High	561	782	1,525	321	859	1,411	3,377
8 airlines	Low	196	455	1,028	44	32	124	1,380
305 airplanes	Average	405	530	1,245	127	203	416	1,884
McDD DC-9	High	485	781	1,953	311	337	773	2,749
8 airlines	Low	181	272	777	52	53	225	1,268
445 airplanes	Average	366	478	1,044	181	143	423	1,569
Ourania**	Estimated in							
500 airplanes	5.4.2***	227	479	766****	300	269	965	2,847

* 1988 Data ** 1990 Estimates

*** The data in 5.4.2 are in USD/nm. They must be multiplied by the block speed of 390 kts to obtain the data for this table.

**** Includes 0.02DOC for insurance

Figure 5.13 Effect of Depreciation Period on the Direct Operating Cost of the Ourania Transport

period is much longer than 10 years! If a 20 year period of depreciation is used, the Ourania data look better!

One final comment on the data in Table 5.9: the wide gap between the 'high' and the 'low' data is caused by a number of factors:

* differences in route structure
* differences in pay scales
* differences in depreciation strategies
* differences in age of airplanes

5.4.5.2 Discussion of program cost results

It is now possible to compare the three program cost sources of Eqn.(2.3) for the Ourania program with each other. The fourth cost source, C_{DISP}, was not considered because the expected life of the airplane is considerably longer than the 10 year depreciation period used in finding the program operating cost data.

The three program cost sources are as follows:

1.) Research, development, test and evaluation program cost, from Section 3.8:

C_{RDTE} = USD 1,232,000,000

2.) The acquisition program cost, from Section 4.6:

C_{ACQ} = USD 10,056,000,000

3.) The operating program cost, from Section 5.4:

C_{OPS} = USD 70,078,905,000

What is clear from all this is that the trends of Figure 2.5 and therefore inequality (2.4) are indeed correct:

$$C_{OPS} \gg C_{ACQ} \gg C_{RDTE}$$

There appears to be roughly an order of magnitude (i.e. a factor of 10) difference between the three major program cost sources. This should emphasize further the points made by Figure 2.3: early design decision making has a large impact on the ultimate cost of an airplane program!

5.4.5.3 Discussion of DOC trade studies

The purpose of this sub-sub-section is to present and discuss the results of several trade studies which show how DOC is affected by the following parameters:

1) Block distance, R_{bl}

2) Crew salary, SAL

3) Fuel Price, FP

4) Specific fuel consumption, c_j

5) Airframe and systems maintenance manhours per block hour, $MHR_{map_{bl}}$

6) Take-off weight, W_{TO}

The results of these trade studies are presented in Figures 5.14 through 5.19.

Figure 5.14 shows the strong effect of block distance on DOC: decreasing the block distance below 1,000 nm has a very inflationary effect on DOC.

The effect of crew salary on DOC is seen to be very weak in Figure 5.15: doubling the crew salary has only a marginal effect on DOC.

Figure 5.16 indicates the very strong effect of fuel price on DOC. Figure 5.17 shows that specific fuel consumption also has a strong effect on DOC. If fuel prices rise too much, there will be strong pressure on airplane designers to seek ways to:

1) reduce fuel consumption: lower drag!

and

2) decrease specific fuel consumption: more advanced engine technology!

It is seen from Figure 5.18 that the number of airplane maintenance manhours per block hour is a strong factor in determining airplane DOC. Anything that the designer can do to lower the need for maintenance will be benificial. However, read the caution notes below!

Finally, Figure 5.19 shows that an increase in take-off weight (as caused by an increase in empty weight) also drives up the DOC of an airplane. Thus, designing for low empty weight helps to keep the DOC down.

NOTES OF CAUTION:

I. If a reduction in takeoff weight is achieved by use of more expensive materials or manufacturing techniques,

and:

If a reduction in drag or engine specifics is achieved by higher cost in manufacturing of airframe and/or engine,

then it will be necessary to evaluate the net benefit on DOC. The data presented here assume that ALL ELSE is kept constant. That is hardly ever the case!

II. Several cost contributions in this chapter were expressed as a fraction of DOC. One example is the IOC of sub-section 5.4.3. There is no reason why IOC should change if parameters such as takeoff weight, fuel price, engine specific fuel consumption, etc. are changed. The effect of these parameters on the total operating cost (TOC) should therefore be evaluated with the help of a baseline IOC. This baseline IOC must be kept constant while parameters which do not affect IOC are being varied.

Figure 5.14 Effect of Block Distance on the Direct Operating Cost of the Ourania Transport

Figure 5.15 Effect of Crew Salary on the Direct Operating Cost of the Ourania Transport

Figure 5.16 Effect of Fuel Price on the Direct Operating Cost of the Ourania Transport

Figure 5.17 Effect of Specific Fuel Consumption on the Direct Operating Cost of the Ourania Transport

Figure 5.18 Effect of Maintenance Manhours per Block Hour on the Direct Operating Cost of the Ourania Transport

Figure 5.19 Effect of Takeoff Weight on the Direct Operating Cost of the Ourania Transport

5.5 SOME IMPORTANT OBSERVATIONS ABOUT COMMERCIAL AIRPLANE OPERATING COST

The purpose of this section is to discuss a number of important aspects of airplane operating costs: direct (DOC), indirect (IOC) and total (TOC). The total operating cost, TOC, is related to the direct (DOC) and indirect (IOC) operating costs by:

$$TOC = DOC + IOC \tag{5.61}$$

Table 5.10 shows some historical trends of the operating cost components as defined by Equations (5.19) and (5.57). Note that the financing cost is not included because this is not normally disclosed by airlines.

Note that the DOC over the last decade has gradually decreased from 56 percent to 53 percent of TOC, while the IOC has increased from 44 percent to 47 percent of TOC. The reason for these changes is not known. Based on the data, it seems reasonable to assume for the year 1985:

$$DOC = 0.53(TOC) \tag{5.62}$$

$$IOC = 0.47(TOC) \tag{5.63}$$

Note also from Table 5.10 that the fuel and oil cost fraction of TOC has increased from 19 percent to 23 percent, despite a decrease in fuel prices (Fig.5.3) from 1981 to 1985! Also, the flight crew cost as a fraction of TOC has decreased from 9 percent to 7 percent. This may be because during this period:

1) inflation was brought under reasonable control
2) average crew salaries have decreased.

It is useful to keep the orders of magnitude of the various operating cost contributions in mind. The 1985 data suggest the following tabulation of TOC components:

$$DOC_{flt} = 0.33(TOC) \tag{5.64}$$

where:

$$C_{crew} = 0.07(TOC) \tag{5.64a}$$

$$C_{pol} = 0.23(TOC) \tag{5.64b}$$

$$C_{ins++} = 0.03(TOC) \tag{5.64c}$$

$$DOC_{maint} = 0.10(TOC) \tag{5.65}$$

Table 5.10 Trends in Airline Total (= DOC + IOC) Operating Cost Components

Cost Item	1976	1981	1982	1983	1984	1985
Direct Operating Cost (DOC)						
Flight crew, C_{crew}	0.086	0.072	0.072	0.072	0.069	0.068
Fuel, oil and lub., C_{pol}	0.193	0.292	0.272	0.245	0.233	0.232
Hull insurance, rentals and training, C_{ins++}	0.039	0.027	0.025	0.027	0.031	0.034
DOC_{flt}	0.318	0.391	0.370	0.344	0.333	0.333
DOC_{maint}	0.125	0.103	0.100	0.100	0.101	0.100
DOC_{depr}	0.080	0.064	0.068	0.072	0.072	0.071
DOC_{lnr}	0.037	0.035	0.033	0.033	0.030	0.030
DOC Total	0.560	0.593	0.571	0.549	0.536	0.534
Indirect Operating Cost (IOC)						
IOC_{pax}	0.094	0.086	0.091	0.092	0.092	0.092
IOC_{sta}	0.133	0.113	0.122	0.126	0.130	0.130
IOC_{pse}	0.149	0.147	0.155	0.164	0.166	0.165
IOC_{gaa}	0.064	0.061	0.061	0.069	0.076	0.079
IOC Total	0.440	0.407	0.429	0.451	0.464	0.466
TOC = DOC + IOC	1.000	1.000	1.000	1.000	1.000	1.000

Notes: 1) IOC_{frt} is included in IOC_{sta} 2) Source: Interavia, Feb.1987

$$DOC_{depr} = 0.07(TOC) \quad (5.66)$$

$$DOC_{lnr} = 0.03(TOC) \quad (5.67)$$

$$IOC_{pax} = 0.09(TOC) \quad (5.68)$$

$$IOC_{sta} = 0.13(TOC) \quad (5.69)$$

$$IOC_{pse} = 0.17(TOC) \quad (5.70)$$

$$IOC_{gaa} = 0.08(TOC) \quad (5.71)$$

It is of interest to consider the effect of block distance, R_{bl}, and of airplane type on the various contributions to TOC.

Figures 5.20 and 5.21 provide this information for four types of airplane for the four major DOC components. It is noted from Fig.5.20 that block distance has essentially no effect on the fractions of DOC_{flt} and DOC_{depr} of total DOC. However, airplane type has a significant effect on their relative contributions.

It is noted from Figure 5.21 that block distance and type of airplane both have a significant influence on the magnitudes of DOC_{maint} and DOC_{lnr} relative to total DOC.

Figures 5.22 - 5.24 provide similar information for the five major IOC components. It is noted from Fig.5.22 that neither the type of airplane nor block distance have much effect on the magnitudes of IOC_{pax} and IOC_{gaa} relative to total IOC.

However, as is seen in Figures 5.23 and 5.24, both airplane type and block distance have a considerable effect on IOC_{sta}, IOC_{pse} and IOC_{ascf} as a fraction of total IOC.

The author believes that even though the data in Figures 5.20 - 5.24 are from 1975, they can be expected to apply to 1990 (and perhaps beyond) because of the fact that (as Table 5.10 indicates) calendar time had only a minor effect on the distribution of TOC components.

Figure 5.20 Effect of Block Distance and Airplane Type on the Direct Operating Cost for Flight and Depreciation as a Fraction of Total DOC

Figure 5.21 Effect of Block Distance and Airplane Type on the Direct Operating Cost for Maintenance and Landing, Navigation and Registration as a Fraction of Total DOC

Figure 5.22 Effect of Block Distance and Airplane Type on the Indirect Operating Cost for Passengers and General Administrative Expenses as a Fraction of Total IOC

Figure 5.23 Effect of Block Distance and Airplane Type on the Indirect Operating Cost for Station Operation as a Fraction of Total IOC

Figure 5.24 Effect of Block Distance and Airplane Type on the Indirect Operating Cost for Promotion, Sales and Entertainment and for Airplane Service, Control and Freight as a Fraction of Total IOC

6. METHOD FOR ESTIMATING OPERATING COST OF MILITARY AIRPLANES

The purpose of this chapter is to present a method for estimating the operating cost of military airplanes in peace time. A method for estimating the operating cost of commercial airplanes is presented in Chapter 5.

A detailed method for estimating operating cost for military airplanes is provided in Ref.10. Despite the level of detail given in Ref.10, it is not sufficient to give the preliminary designer specific insight into what can be done (from a design viewpoint) to lower operating cost of military airplanes. Therefore, the author has decided to combine the methods, ideas and data given in Refs 10 and 14 into a simpler method. This method should be used only for preliminary cost estimating purposes.

IMPORTANT ASSUMPTION:

Military airplanes are acquired in a manner depending considerably on budgetary and political constraints. It is not possible to translate such constraints into any rational format. Therefore, no attempt will be made to represent program operating cost as a summation over a number of calendar years. Instead, the assumption will be made that the number of airplanes to be acquired, N_{acq}, have indeed been acquired by the military service.

Furthermore, the assumption will be made that deliveries have been made to ONE military service only so that cost differences between services need not be accounted for.

The program operating cost of military airplanes can be broken down as follows:

$$C_{OPS} = C_{POL} + C_{PERSDIR} + C_{PERSIND} + C_{CONMAT} + C_{SPARES} + C_{DEPOT} + C_{MISC} \qquad (6.1)$$

where: C_{POL} is the program fuel, oil and lubricants cost: see Section 6.1

$C_{PERSDIR}$ is the program cost of direct personnel (that is aircrew and maintenance): see Section 6.2

$C_{PERSIND}$ is the program cost of indirect personnel: see Section 6.3

C_{CONMAT} is the program cost of consumable materials used in conjunction with maintenance: see Section 6.4

C_{SPARES} is the program cost of spares: see Section 6.5

C_{DEPOT} is the program cost associated with depots: see Section 6.6

C_{MISC} is the program miscellaneous cost: see Section 6.7

The total (or program) operating cost of military airplanes, C_{OPS}, is shown in relationship to other life cycle cost (LCC) sources in Eqn.(2.3) and in Figure 2.3.

A summary for the determination of the program operating cost for a military airplane, C_{OPS}, is presented in Section 6.8. An example application of the method is given in Section 6.9. Some important observations on the subject of military airplane operating cost are included in Section 6.10.

6.1 PROGRAM COST OF FUEL, OIL AND LUBRICANTS: C_{POL}

The program cost of fuel, oil and lubricants depends on the following:

* type of airplane
* mission of the airplane
* annual utilization
* number of airplanes in active service

The program cost for fuel, oil and lubricants can be estimated from:

$$C_{POL} = [(F_{OL})(W_{F_{used}})[(FP)/(FD)]* \\ *(N_{mission})(N_{serv})](N_{yr}) \qquad (6.2)$$

where: F_{OL} is a factor which accounts for the cost of oil and lubricants. According to Ref.14 it is acceptable to use: $F_{OL} = 1.005$

$W_{F_{used}}$ is the mission fuel used in lbs. per mission. This can be found from Eqn.(2.14) in Part I or from the more detailed method of Section 5.4, Part VII.

FP is the price of fuel in USD per gallon. This is obtained from Figure 5.3. In most instances the price of JP-4 can be used.

FD is the fuel density in lbs per gallon. This may be obtained from p.88. In most instances the density of JP-4 can be used.

$N_{mission}$ is the number of missions flown per year. This can be estimated from:

$$N_{mission} = (U_{ann_{flt}})/(t_{mis}) \qquad (6.3)$$

where: $U_{ann_{flt}}$ is the annual utilization in flight hours. Table 6.1 provides typical data for military airplane utilization in peace time.

t_{mis} is the average mission time in hours. Figure 6.1 shows how this is defined.

<u>NOTE:</u> in the case of multi-mission airplanes, a scenario of typical 'mission mixes' per year must be predicted.

N_{serv} is the number of airplanes of the type in actual service. This number is averaged as:

$$N_{serv} = N_{acq} - N_{res} - 0.5(N_{loss}) \qquad (6.4)$$

where: N_{acq} is the number of airplanes which are acquired

N_{res} is the number of airplanes held in active reserve

N_{loss} is the number of airplanes lost through accidents over the service life, N_{yr}

How to determine these three airplane quantities is discussed next.

TABLE 6.1 MILITARY AIRPLANE UTILIZATION IN FLIGHT HOURS PER YEAR AND MILITARY CREW TO AIRPLANE RATIOS

Airplane Type	Flight Hours per airplane per year, $U_{ann_{flt}}$	Crew Ratio per airplane, R_{cr}	Number in crew N_{crew}
Trainers	600 - 1,000	-	2
Fighters	250 - 400	1.1	1 - 2
Bombers	300 - 600	1.5	2 - 5
Tankers	300 - 600	1.5	4 - 6
Transports	< 1,200	1.5	4 - 8
	1,200 - 2,400	2.5	4 - 8
	2,400 - 3,600	3.5	4 - 8

NOTE: average annual utilization for military airplanes is 350 hours

NOTE: FOR EXAMPLE MISSION PROFILES, SEE: P.61 IN PART I AND PAGES 244-248, AND 280-284 IN PART VII.

Figure 6.1 Definition of Mission Time, t_{mis}

The number of airplanes acquired, N_{acq}, is assumed to be equal to the number of airplanes built to production standards, N_m. This in turn is related to the total number of airplanes built, $N_{program}$ by:

$$N_{program} = N_m + N_{rdte} \tag{6.5}$$

where: N_{rdte} is the number of airplanes built during the research, development, technology and evaluation phase. For military programs this number is in the following range:

$$N_{rdte} \approx 6 \text{ to } 20 \tag{6.6}$$

<u>NOTE:</u> Table 4.1 gives examples of airplane program production runs for several military programs.

The number of airplanes held in reserve, N_{res}, depends on factors such as: politics and estimated loss rates in peace time. For preliminary cost estimating purposes it is suggested to use:

$$N_{res} = 0.10(N_{acq}) \tag{6.7}$$

The total number of airplanes lost due to accidents in peace time, N_{loss}, is estimated from:

$$N_{loss} = (L_R)(N_{serv})(U_{ann_{flt}})(N_{yr}) \tag{6.8}$$

where: L_R is the annual loss rate expressed in airplanes lost per 10^5 flying hours. Table 6.2 shows typical loss rates for USAF and for USN/USMC.

For the reader interested in loss rate data in times of military conflict, reference is made to Section 13.3 in Part IV. Methods to predict combat loss rates are given in Ref.22.

N_{yr} is the number of years the airplane is in active service. In peace time this number may be assumed to range from 20 to 40 years. Table 6.3 provides 1987 data for the average age of several U.S. military airplanes.

All other quantities were previously defined.

Table 6.2 Military Airplane Accident Summaries for 1984 and 1986

Type	<<<<<<<<<<<<<<<(1984)>>>>>>>>>>>>>>> Number of Accidents	Total Flight Hours (Fleet)	Accident Rate, L_R	<<<<<<<<<<<<<<<(1986)>>>>>>>>>>>>>>> Number of Accidents	Total Flight Hours (Fleet)	Accident Rate, L_R
USAF:						
T-37	1	333,000	0.3	1	333,300	0.3
T-38	3	375,000	0.8	4	363,600	1.1
F-4	11	275,000	4.0	12	245,000	4.9
RF-4	1	77,000	1.3	5	78,000	6.4
F-5	2	29,000	6.9	6	26,500	22.6
F-15	3	176,000	1.7	7	194,400	3.6
F-16	10	200,000	5.0	11	250,000	4.4
F-111	3	79,000	3.8			
A-7	6	86,000	7.0	1	83,300	1.2
A-10	6	222,000	2.7	3	214,300	1.4
C-130	3	375,000	0.8	2	400,000	0.5
C-135				1	250,000	0.4
C-141	1	333,000	0.3	1	333,300	0.3
B-52	2	105,000	1.9			
Totals:	52	2,665,000	2.0	54	2,771,700	1.9
USN and USMC:						
T-34C	1	179,597	0.6	2	180,900	1.1
T2-J	2	84,734	2.4	2	14,300	14.0
TA-4	9	102,739	8.8	6	118,300	5.1
F-4	4	86,331	4.6	5	54,400	9.2
F-14	4	110,727	3.6	10	120,600	8.3
F-18	1	47,256	2.1	2	107,500	1.9
A-4	4	49,472	8.1	3	59,200	5.1
A-6	7	106,637	6.6	6	104,900	5.7
EA-6	3	26,778	11.2	3	29,800	10.0
A-7	13	146,547	11.3	9	126,400	7.1
AV-8	1	18,297	5.5	3	25,400	13.1
Totals:	49	960,000	5.1	51	941,700	5.4

Data from: Aviation Week and Space Technology, March 11, 1985 and June 8, 1987
Note: Accident Rate = Loss Rate = L_R in airplanes per 100,000 flying hours

TABLE 6.3 SOME AIR FORCE AIRPLANES AND THEIR AGE (1987)

Airplane Type	USAF No. in Service	Ave. Age (yrs)	ANG No. in Service	Ave. Age (yrs)	AF Reserve No. in Service	Ave. Age (yrs)
T-33	54	29.4	21	31.7		
T-37	609	25.3				
T-38	810	21.5				
F-4	596	17.1	602	20.5	119	21.0
F-5	96	11.7				
F-15	732	6.9	66	11.4		
F-16	943	3.7	132	5.5	49	5.0
F-100	10	29.4				
F-106	5	28.1	33	27.9		
F-111	334	16.4				
A-7	31	15.9	342	14.1		
A-10	451	6.9	106	8.0	97	9.0
A-37	22	13.8	60	14.7		
C-5	76	12.1	5	16.9	15	17.0
C-9	23	16.5				
C-10	56	2.9				
C-12	75	6.2	6	1.9		
C-20	13	1.1				
C-21	79	2.7	3	0.1		
C-22	1	3.6	4	2.7		
C-23	18	2.4				
C-130	354	19.3	210	19.8	168	23.4
C-135	608	26.2	104	28.6	24	29.0
C-140	5	24.7				
C-141	255	21.0	8	21.3	8	22.0
E-3	33	7.9				
B-1	60	0.8				
B-52	263	27.0				
FB-111	62	17.1				

Data from: Air Force Magazine, May 1987

NOTE: This table may be used to estimate N_{yr} for a new military airplane program.

6.2 PROGRAM COST OF DIRECT PERSONNEL: $C_{PERSDIR}$

The program cost component of direct personnel can be split into two elements:

$$C_{PERSDIR} = C_{crewpr} + C_{mpersdir} \tag{6.9}$$

where: C_{crewpr} is the program cost for aircrews: see sub-section 6.2.1

$C_{mpersdir}$ is the program cost for direct maintenance personnel: see sub-section 6.2.2

6.2.1 Program Cost of Aircrews: C_{crewpr}

The program cost for aircrews may be estimated from:

$$C_{crewpr} = \tag{6.10}$$

$$= (N_{serv})(N_{crew})(R_{cr})(Pay_{crew})(OHR_{crew})(N_{yr})$$

where: N_{serv} is found from Eqn.(6.4)

N_{crew} is the number of crew members required for the airplane: see Table 6.1.

R_{cr} is the crew ratio per airplane. This may be determined with the help of Table 6.1.

Pay_{crew} is the annual pay in USD per year. The annual pay depends on:

* pay grade
* years of service

Table 6.4 gives military pay data applicable to the calendar years 1989 and 1990. To determine Pay_{crew}, several assumptions must be made about

crew composition in terms of grade and years in service. For preliminary cost estimating purposes it is suggested to assume:

<u>For trainers:</u> average crew member is a 1st Lt. with 5 years of military service.

<u>For fighters:</u> average crew member has the rank of captain with 8 years of military service.

Table 6.4 1989 Military Pay Data for Aircrews*

Years in Service	Basic Pay in USD/year — Rank: Col/Capt	LtCol/Cmdr	Maj/LtCmdr	Capt/Lt	1stLt/LtJG	2ndLt/Ens
4	38,099	32,677	28,488	26,964	24,109	19,418
6	38,099	32,677	29,027	28,253	24,617	19,418
8	38,099	32,677	30,308	29,268	24,617	19,418
10	38,099	33,667	32,375	30,857	24,617	19,418
12	38,099	35,474	34,196	32,375	24,617	19,418
14	39,391	37,854	35,759	33,174	24,617	19,418
16	45,619	40,687	37,325	33,174	24,617	19,418
18	47,952	43,020	38,358	33,174	24,617	19,418
20	48,996	44,323	38,358	33,174	24,617	19,418
22	51,836	45,871	38,358	33,174	24,617	19,418

Aviation Career Incentive Pay (ACIP), excl.' Hazardous Duty Pay' in USD/month

Years of Aviation Officer Service	<or =2	>2	>3	>4	>6	>18	>20	>22
Monthly Rate	125	156	188	206	400	370	340	310

Aviation Continuation Pay (Re-up Bonus) in USD/year

Years of Active Federal Commissioned Service	6	7	8	9	10	11	12
1989 Bonus	9,000	9,000	8,250	8,250	7,125	6,000	4,875
Subsequent annual bonus	12,000	12,000	11,000	11,000	9,500	8,000	6,500

* Source: Professional Pilot Magazine, April, 1989

For bombers: average crew member has the rank of major with 12 years of military service.

For transports: average crew member has the rank of 1st Lt. with 5 years of military service. Note: transports often carry non-commissioned personnel!

IMPORTANT NOTE: the ranks given are for 'average' crew members in the USAF and the USMC only. For the USN use the 'equivalent' ranks of Table 6.4.

An example military pay calculation will now be given. For this example it is assumed that the crew pay needs to be found for a USAF fighter:

Page 152 suggests that for preliminary costing purposes it may be assumed that for a fighter the average crew member has the rank of captain with 8 years of military service. In that case, it is seen from Table 6.4 that:

$$Pay_{crew} = (29,268 + 12*400 + 12,000) =$$

$$= 46,068 \text{ USD/year}$$

The 12,000 pay factor assumes 1990 and beyond aviation continuation pay.

NOTE: Military pay scales are subject to congressional action. For years beyond 1990, the reader should check whether these pay scales are valid!

OHR_{crew} is an overhead rate factor, associated with the actual crew pay. Lacking actual military overhead data, it is suggested to use: $OHR_{crew} = 3.0$. This overhead rate factor is similar in magnitude to industry overhead rates.

N_{yr} is the number of years the airplane is in active service. This quantity may be estimated with the help of Table 6.3.

6.2.2 Program Cost of Direct Maintenance Personnel: $C_{mpersdir}$

The program cost of direct maintenance may be estimated from:

$$C_{mpersdir} = (N_{serv})(N_{yr})(U_{ann_{flt}})*$$

$$*(MHR_{flthr})(R_{m_{ml}}) \qquad (6.11)$$

where: N_{serv} is found from Eqn.(6.4)

N_{yr} is found from Table 6.3.

$U_{ann_{flt}}$ is found from Table 6.1

MHR_{flthr} is the number of maintenance manhours required per flight hour. This depends on the following factors:

* number of years in service
* equipment density (this affects accessibility)
* maturity of the technologies used in the airplane systems: propulsion, avionics, flight controls, brakes, wheels and tires.

Table 6.5 provides some historical data on the effect of 'years in service' on MHR_{flthr} for a number of military airplanes. Figure 6.2 shows some data based on actual USNavy experience for A4 type airplanes.

Figure 6.3 illustrates how equipment density affects the maintenance hours per flight hour.

The cost analyst should not attemp to arbitrarily 'lowball' a number for MHR_{flthr}. Instead, a list should be prepared of all factors which affect maintenance and which compare the new design with a comparable existing design for which MHR_{flt} is known. Only then can an educated estimate for MHR_{flthr} be made.

Nicolai (in Ref.14) implies that the number of maintenance manhours per flight hour is related

Table 6.5 Maintenance Man-Hours per Flight Hour for Military Airplanes

Type	1973	1974	1975	1976	1977	1978	1979	1980	1981	1982	1983
T-37	7.8										
T-38	10.0		8.8	11.6	8.7	9.0	12.2	9.0	10.3	6.8	10.9
F-100			19.8	17.1	17.1	16.1					
F-4D			55.6	41.9	54.2	56.2	47.7	44.8	28.4	26.9	42.0
F-4E	33.0										
RF-4C			39.6	31.9	33.8	34.2	32.8	31.1	22.9	18.2	36.1
F-5	17.0										
FB-111A			57.2	47.7	36.4	58.2	47.6	45.8	60.8	49.0	56.0
F-111D	40.0		19.8	17.1	17.1	16.1					
F-15		20.0	29.1	27.8	26.0	30.6	31.9	28.6	36.9	30.5	44.1
F-15C								13.6	24.3	22.3	32.8
F-16A	25.0						25.4	22.6	11.2	14.6	21.8
F-16B								18.4	13.9	15.3	30.6
A-7D	25.0										
A-10		13.0				7.2	11.8	12.0	11.2	8.9	13.9
C-130	20.0			25.1	37.1	22.4	19.6	19.8	21.6	19.7	24.7
C-141	21.0		17.5	18.6	18.2	17.3	15.0	15.0	20.7	19.3	25.2
C-5A	40.0		107.9	89.4	48.2	41.9	53.4	47.3	52.5	37.0	64.9
C-9	12.0										
KC-135	26.6		32.8	31.6	34.0	26.6	21.2	23.2	20.2	17.5	24.2
B-52D	37.0										
B-52G	49.0		46.5	46.9	44.9	45.5	42.1	40.6	39.6	34.0	50.7
B-52H			45.5	48.9	47.6	49.3	39.5	33.9	34.6	30.2	42.9

The following ranges are suggested for preliminary cost estimating purposes:

For Trainers: 6 - 10 For Fighters: 15 - 35
For Bombers: 25 - 50 For Transports: 25 - 60

Data from: Refs 14, 24 and Air Force Magazine, September 1983.

to the total area (in square feet) of all access panels in an airplane!

$R_{m_{ml}}$ is the military maintenance labor rate in USD/hr. This depends on many factors, including the ratio of civilian versus military personnel used in maintenance functions. For 1989, this labor rate, <u>including all overhead</u> contributions may be assumed to be:

$$(R_{m_{ml}})_{1989} = 45 \text{ USD/hr}$$

For future years it is suggested to use:

$$(R_{m_{ml}})_{\text{then year}} = (R_{m_{ml}})_{1989} * $$

$$*(CEF_{\text{then year}})/(CEF_{1989}) \qquad (6.12)$$

where: the CEF-values are found from Figure 2.7.

<u>Figure 6.2 Effect of Flying Hours per Month on Maintenance Manhours per Flight Hour</u>

Figure 6.3 Effect of Equipment Density on Maintenance Manhours per Flight Hour

6.3 PROGRAM COST OF INDIRECT PERSONNEL: $C_{PERSIND}$

This cost component covers the cost of all squadron level personnel which are not directly involved in flight operations or airplane maintenance. This cost component may be estimated from:

$$C_{PERSIND} = (f_{persind})(C_{OPS}) \qquad (6.13)$$

where: C_{OPS} is given by Eqn.(6.1)

$f_{persind}$ may be estimated with the help of Table 6.6

6.4 PROGRAM COST OF CONSUMABLE MATERIALS: C_{CONMAT}

This cost component covers the cost of consumable materials used in conjunction with military airplane maintenance functions. This cost component may be found as follows:

$$C_{CONMAT} = (N_{serv})(N_{yr})(U_{ann_{flt}})*$$

$$*(MHR_{flthr})(R_{conmat}) \qquad (6.14)$$

where: R_{conmat} is the average cost in USD/hr (USD per maint. hour) for consumable materials. Lacking specific information it is suggested to use:

$$R_{conmat} = (6.50)(CEF_{then\ year})/(CEF_{1989}) \qquad (6.15)$$

where: the factor 6.50 is the 1989 cost of consumable materials in USD per maintenance hour.

All other terms were previously defined.

6.5 PROGRAM COST OF SPARES: C_{SPARES}

The program cost of spares may be estimated from:

$$C_{SPARES} = (f_{spares})(C_{OPS}) \qquad (6.16)$$

Table 6.6 Multipliers for Program Operating Cost of Military Airplanes

Operating Cost Component	Operating Cost Fraction (=Multiplier)	Fraction of Total Program Operating Cost For Each Airplane Type				
		B52G	C-141	F-111	F-4	A-7D
C_{POL}	f_{pol}	0.14	0.23	0.05	0.10	0.04
$C_{PERSDIR}$	$f_{persdir}$	0.34	0.32	0.28	0.38	0.38
$C_{PERSIND}$	$f_{persind}$	0.16	0.13	0.14	0.20	0.20
C_{SPARES}	f_{spares}	0.13	0.16	0.27	0.12	0.16
C_{DEPOT}	f_{depot}	0.20	0.15	0.22	0.13	0.16
C_{MISC}	f_{misc}	0.03	0.01	0.04	0.07	0.06
Totals:		1.00	1.00	1.00	1.00	1.00

Source: Ref.14, Figure 24.1

<u>NOTES:</u> 1) The fractional contribution of consumable materials cost, C_{CONMAT}, is included in the miscellaneous cost item in this table.

2) The following assumption may be made: $C_{CONMAT} = 0.25(C_{MISC})$

where: C_{OPS} is given by Eqn.(6.1)

f_{spares} may be estimated with the help of Table 6.6

6.6 PROGRAM COST OF DEPOT: C_{DEPOT}

Many overhaul and maintenance actions on military airplanes are not carried out at the squadron or wing level. Instead, these are carried out at so-called depots. The program depot cost associated with the operation of a given airplane type may be estimated from:

$$C_{DEPOT} = (f_{depot})(C_{OPS}) \tag{6.17}$$

where: C_{OPS} is given by Eqn.(6.1)

f_{depot} may be estimated with the help of Table 6.6

6.7 PROGRAM COST OF MISCELLANEOUS ITEMS: C_{MISC}

The following miscellaneous cost elements contribute to the operating cost of military airplanes:

* requirements for technical data to support maintenance functions
* requirements for training, training data and training equipment
* requirements for support equipment

Reference 10 provides detailed cost estimating relations for these items. For the purpose of preliminary cost estimating it is suggested to use:

$$C_{MISC} = (f_{misc})(C_{OPS}) \tag{6.18}$$

where: C_{OPS} is given by Eqn.(6.1)

f_{misc} may be estimated with the help of Table 6.6

Using the assumption made at the bottom of Table 6.6, it is possible to write C_{MISC} alternatively as:

$$C_{MISC} = 4C_{CONMAT} \tag{6.19}$$

6.8 SUMMARY FOR THE DETERMINATION OF THE PROGRAM OPERATING COST OF MILITARY AIRPLANES : C_{OPS}

The program operating cost of military airplanes, with the help of equations (6.1), (6.2), (6.9) and (6.13) through (6.18) may now be found from:

$$C_{OPS} = \frac{C_{POL} + C_{PERSDIR} + C_{CONMAT}}{(1 - f_{persind} - f_{spares} - f_{depot} - f_{misc})} \qquad (6.20)$$

By using Equation (6.19) it is also possible to cast Equation (6.20) in the following form:

$$C_{OPS} = \frac{C_{POL} + C_{PERSDIR} + C_{CONMAT} + C_{MISC}}{(1 - f_{persind} - f_{spares} - f_{depot})} \qquad (6.21)$$

or:

$$C_{OPS} = \frac{C_{POL} + C_{PERSDIR} + {}^{5}C_{CONMAT}}{(1 - f_{persind} - f_{spares} - f_{depot})} \qquad (6.22)$$

In comparing the program operating costs from one airplane to another it is often desirable to do this on the basis of their so-called program operating cost per hour: $C_{OPS/HR}$.

This program operating cost per flight hour may be obtained from:

$$C_{OPS/HR} = (C_{OPS})/[(N_{serv})(N_{yr})(U_{ann_{flt}})] \qquad (6.23)$$

where: C_{OPS} follows from Equations (6.20), (6.21) or (6.22).

All other terms were previously defined.

6.9 EXAMPLE APPLICATION

The purpose of this section is to present an example application of the method for determining the program operating cost of military airplanes. The material is organized in the following manner:

6.9.1 Program Operating Cost Components

6.9.2 Program Operating Cost Total

6.9.3 Program Operating Cost per Flight Hour

6.9.1 Program Operating Cost Components

The example will be constructed around the ground attack fighter airplane of Parts I and II (called Eris). The mission specification for this fighter is found in Table 2.19 of Part I.

The assumption will be made that all costs must be computed for the year 1990.

Program Fuel, Oil and Lubricants Cost: C_{POL}

These costs are computed with Eqn.(6.2).

The input requirements for Eqn.(6.2) are:

F_{OL} = 1.005, according to page 146.

$W_{F_{used}}$ = 18.500lbs, according to p.67 of Part I.

The assumption will be made that JP-4 is the fuel of choice for this ground attack fighter airplane.

FP = 0.75 USD/gallon (assumed from Fig.5.3, p.87).

FD = 6.55 lbs/gallon, from p.88.

$N_{mission}$ is computed from Eqn.(6.3) as:

$N_{mission}$ = 350/2.67 = 131 missions per year

This assumes an annual utilization of $U_{ann_{flt}}$ = 350 flight hours (see Table 6.1) and a mission duration of t_{mis} = 2.67 hours. The latter is pieced together with the help of Table 2.19 in Part I.

The following data are assumed for the number of airplanes involved in the Eris development. production and acquisition program:

$N_{program} = 750$ $N_{rdte} = 10$ $N_m = 740$

With the assumption made on page 145:

$N_{acq} = 740$ units

Next, the number of airplanes in active service will be determined from Eqn.(6.4).

It will be assumed that 10 percent of the Eris fleet will be held in active reserve: $N_{res} = 74$ units

The loss rate per 100,000 flight hours will be assumed to be: $L_R = 2$, which is an optimistic projection, based on Table 6.2.

The number of years during which the Eris will be in active service is assumed to be: $N_{yr} = 25$ years.

With Eqns (6.8) and (6.4):

$$N_{serv} = 740 - 74 - [\{(0.5)(2)(350)(25)\}/100,000]N_{serv} =$$

$$= 740 - 74 - 0.0875N_{serv}, \text{ so that:}$$

$$N_{serv} = 666/1.0875 = 612 \text{ units}$$

Now with Eqn.(6.2):

C_{POL} = USD 4,266,978,750

<u>Program Cost of Direct Personnel:</u> $C_{PERSDIR}$

This cost may be determined with Eqn.(6.9).

The program cost for aircrews is found with the help of Equation (6.10). In this equation the following input data remain to be determined:

Since the Eris requires only one pilot:

$N_{crew} = 1.0$

The crew ratio, $R_{cr} = 1.1$ from Table 6.1.

The crew pay is assumed to be that of the example of page 154, so that:

Pay_{crew} = 46,068 USD/year

The crew overhead factor, OHR_{crew} is assumed to be:

OHR_{crew} = 3.0 (p.154).

With these data, the program cost for aircrews can be found from Eqn.(6.10) to be:

$$C_{crewpr} = (612)(1)(1.1)(46,068)(3)(25) =$$

$$= \text{USD } 2,325,973,320$$

The program cost for direct maintenance personnel is found with Eqn.(6.11). In this equation the following input data remain to be determined:

For maintenance manhours per flight hour, Table 6.5 has been used to project the following number:

MHR_{flthr} = 12

The maintenance labor rate per manhour for 1990 (including overhead) will be determined from Eqn.(6.12) as:

$$R_{m_{ml}} = 45(3.04)/(3.02) = \text{USD } 45.30$$

The program cost for direct maintenance personnel can now be computed:

$$C_{persdir} = (612)(25)(350)(12)(45.30) =$$

$$= \text{USD } 2,910,978,000$$

The program cost of direct personnel, aircrews and maintenance personnel is found with Eqn.(6.9) as:

$$C_{PERSDIR} = 2,325,973,320 + 2,910,978,000 =$$

$$= \text{USD } 5,236,951,320$$

<u>Program Cost of Indirect Personnel:</u> $C_{PERSIND}$

This cost component may be determined from Eqn.(6.13) and Table 6.6 as:

$C_{PERSIND} = (0.20)(C_{OPS})$, where $f_{persind} = 0.20$

Program Cost of Consumable Materials: C_{CONMAT}

This cost is determined with Eqn.(6.14). The following input data remain to be determined:

With Eqn.(6.15):

$R_{conmat} = 6.50(3.04)/(3.02) =$

$= 6.54$ USD per maint.hour

With Eqn.(6.14):

$C_{CONMAT} = (612)(25)(350)(12)(6.54) =$

$=$ USD 420,260,400

Program Cost of Spares: C_{SPARES}

This cost is found with Eqn.(6.16) and Table 6.6 as:

$C_{SPARES} = (0.14)(C_{OPS})$, where $f_{spares} = 0.14$

Program Cost of Depot: C_{DEPOT}

This cost is found from Eqn.(6.17) and Table 6.6 as:

$C_{DEPOT} = (0.15)(C_{OPS})$, where $f_{depot} = 0.15$

Program Cost of Miscellaneous Costs: C_{MISC}

This cost component is found from Eqn.(6.19) as:

$C_{MISC} = 4C_{CONMAT} = 4(420,260,400) =$

$=$ USD 1,681,041,600

6.9.2 Program Operating Cost Total

The program operating cost total may now be computed with Equation (6.21) as follows:

$$C_{OPS} = \frac{(4,266,978,750 + 5,236,951,320 + 420,260,400 + 1,681,041,600)}{(1 - 0.20 - 0.14 - 0.15)} =$$

$$= 11,605,232,070/0.51 = \text{USD } 22,755,000,000,$$

which has been rounded off to the nearest million.

6.9.3 Program Operating Cost per Flight Hour

The program operating cost per flight hour may be determined with Equation (6.23) as:

$$C_{ops/hr} = 22,755,000,000/[(612)(25)(350)] =$$

$$= 4,249 \text{ USD/hour}$$

The reader will have observed the fact that neither cost of depreciation nor cost of financing is included in the military program operating cost per flight hour.

Since military airplanes 'belong' to the tax payers, they are not depreciated in any commercial sense of the word. Also, the cost of financing is not included, because military operating costs are deferred out of annual budgets alotted by the congress.

6.10 SOME OBSERVATIONS ON OPERATING COST OF MILITARY AIRPLANES

The purpose of this section is to provide the reader with additional insight and with additional references relative to the operating cost of military airplanes. The material is organized as follows:

6.10.1 Military Service Organization, Typical Airplane Numbers in Service and Reasons for High Operating Cost

6.10.2 Discussion of Operating Cost Distribution

6.10.3 Examples of System Maintainability and Reliability

6.10.1 Military Service Organization, Typical Airplane Numbers in Service and Reasons for High Operating Cost

Most military airplane operations are conducted at the squadron or wing level. The number of airplanes per squadron, the number of squadrons per wing and the number of wings per Air Force or per Navy Group depends on many tactical, strategic and political considerations which are beyond the scope of this text. For the interested reader, Air Force Magazine and the United States Naval Institute Proceedings (both are monthly magazines) publish annual details about USAF, USN and USMC service organizations. Although the US Army does operate several fixed wing airplane types, this service concentrates on rotary wing airplanes for combat roles. Since this text deals only with fixed wing airplanes, USArmy airplanes have not been considered.

Next, some examples are given of how military airplanes are distributed numerically at the squadron level. The examples given apply to the USAF. Table 6.7 shows how many airplanes are typically assigned to a squadron. Table 6.8 indicates the number of squadrons dedicated to specific operational tasks. Table 6.9 shows the number of active airplanes and the annual flying hours produced by each for recent years.

Because military forces have to be able to operate relatively independently of large fixed bases during war times, a certain amount of duplication in support and infrastructure is inevitable. This in turn implies that the operation of military airplanes cannot be conducted

TABLE 6.7 NUMBER OF AIRPLANES PER SQUADRON

Airplane Type	Number per Squadron	Airplane Type	Number per Squadron
A-10	18 or 24	B-1	16
F-4	12 or 24	B-52	13 - 19
RF-4	18	C-5	15 or 16
F-5	11, 18 or 20	C-9	3 or 11
F-15	15, 18 or 24	C-130	16
F-16	18 or 24	AC-130	10
F-106	15	KC-10	19
F-111	12, 18 or 24	KC-135	13 - 25
FB-111	8 or 11	C-141	13 - 17
		E-3	2, 4 or 16

Source: Air Force Magazine, May 1988

TABLE 6.8 USAF FLYING SQUADRONS BY MISSION TYPE

Active Forces	FY'85	FY'87	FY'89*
Strategic Bomber	22	22	25
Air Refueling	35	36	35
Strategic Comm. and Ctrl	6	6	6
Intelligence	3	3	3
Strategic Reconnaissance	1	1	1
Strategic Interceptor	4	3	2
Fighter	78	81	79
Tactical Reconnaissance	8	7	5
Tactical Electr. Warfare	3	4	4
Special Operations Forces	5	5	5
Tact. Air Comm. Ctrl Syst.	3	3	3
Tactical Air Ctrl Syst.	7	7	7
Weather	2	2	1
Rescue	8	9	7
Tactical Airlift	14	13	12
Strategic Airlift	17	17	18
Special Mission	1	1	1
Aeromedical Airlift	3	3	3
Total Active Forces	220	223	217
Reserve Forces:			
ANG Selected Reserve	91	91	91
Air Force Reserve	56	57	58
Total Squadrons	367	371	366

Source: Air Force Magazine, May 1988 * Estimated

TABLE 6.9 NUMBER OF ACTIVE AIRPLANES AND FLYING HOURS

Airplane Type	Number of Active Airplanes in: FY'85	FY'87	FY'89*
Strategic Bomber	330	393	420
Tanker	559	576	566
Fighter/Interceptor/Attack	3,057	3,033	2,998
Recon./Electronic Warfare	418	432	438
Cargo/Transport	859	848	852
Search and Rescue (fixed wing only)	37	35	31
Helicopter (incl. rescue)	234	191	171
Trainer	1,613	1,595	1,515
Utility/Observation/Other	180	110	112
Total USAF	7,287	7,213	7,103
Air National Guard Total	1,688	1,732	1,736
Air Force Reserve Total	468	502	514
Total Active USAF, ANG and USAF Reserve	9,443	9,447	9,353
	Flying Hours in Thousands:		
USAF	2,914	2,837	2,770
ANG	423	435	447
AF Reserve	140	153	156
Total Fl. Hrs in Thousands	3,477	3,425	3,373
Average Flying Hours per year, per airplane:	368	363	361

Source: Air Force Magazine, May 1988 * Estimated

COURTESY: SAAB

at levels of efficiency comparable to commercial airplane operations. This is one reason why the operating cost per hour for airplanes such as the Eris ground attack fighter is predicted to be rather high: USD 4,249 per flight hour as determined on page 167.

The challenge to the airplane designer is to design military airplanes and their systems layout in such a way that their mission can be accomplished at an affordable cost to the taxpayers.

The following subsections present observations on several aspects of operating cost distribution as well as on system reliability and maintainability.

6.10.2 Discussion of Operating Cost Distribution

Figure 6.4 shows the contribution to total operating cost of each cost component considered in Equation (6.1) for the Eris fighter. The cost data were calculated in Section 6.9. The percentage contribution of the seven operating cost components can be summarized as follows:

1)	C_{POL} (Fuel, oil and lubricants):	19
2)	$C_{PERSDIR}$ (Direct personnel): aircrew 10, maintenance 13	23
3)	C_{SPARES} (Spares):	14
4)	C_{CONMAT} (Consumable materials):	2
5)	$C_{PERSIND}$ (Indirect personnel):	20
6)	C_{DEPOT} (Depot):	15
7)	C_{MISC} (Miscellaneous):	7
	C_{OPS} (Total operating cost percentage)	100

The following operating cost items are directly affected by the airplane designer and the design decisions he makes, mostly those in the early phases of preliminary design:

* Fuel, oil and lubricants (see Parts I and VII, specifically for what governs W_F and W_{fto})

Figure 6.4 Distribution of Program Operating Cost Components for the Eris Fighter Program

Table 6.10 CLASSICAL EXAMPLES OF POOR DESIGN PRACTICES

EXAMPLE 1: In the F-4 fighter, the radio was a high failure rate item. It was installed underneath the rear ejection seat. To remove and to replace this radio this ejection seat had to be removed first. This made radio removal and replacement very expensive.

EXAMPLE 2: In the AV-8B V/STOL fighter, it is required to remove the wing before the engine can be removed. This adds significantly to the cost and time of engine removal.

EXAMPLE 3: In the B-1B bomber, a 'quick access' port was installed to allow for rapid checking and adding of engine oil levels. However, the oil gauge itself was hidden behind the very panel which included the 'quick access' panel. Maintenance personnel had to remove that entire panel before they could access the oil gage. That defeated the entire purpose of the 'quick access' port.

* Aircrew and maintenance (see Parts II, III and IV, for details on configuration, cockpit, structure and system layout design practices)

From the tabulation on page 171 it is clear that the designer has direct leverage over 58 percent of the operating cost of a typical ground attack fighter. Two examples illustrate this:

EXAMPLE 1: Maintenance Manhours

In Section 6.9 it was shown that the program operating cost for the Eris fighter is USD 22,755,000,000 over a projected 25 year life. The maintenance manhours per flight hour were assumed to be 12.

Through careless design it is conceivable that this number could increase to 24 maintenance manhours per flight hour. Table 6.5 shows that this can indeed occur. In such a case, the program operating cost increases by:

USD 1.2x2x2,910,978,000 (see p.165) = USD 6,986,347,200.

This means an increase of 31 percent of the total program operating cost!

If, on the other hand, by careful design the maintenance manhours per flight hour could be decreased from 12 to 6, a program operating cost savings of:

USD 1.2x0.5x2,910,978,000 (see p.165) = USD 1,746,586,800

would be the result. That is a savings of 8 percent!

EXAMPLE 2: Mission Fuel Used

It was shown on p.164 that the Eris program operating cost for fuel, oil and lubricants is approximately: USD 4,300,000,000. The designer should be able to reduce this cost by paying careful attention to details of:

* engine inlet design
* nozzle and boattail design
* aerodynamic design
* c.g. excursion diagram

A ten percent reduction in this cost item should be within reach: this shaves USD 430,000,000 off the program operating cost!

A thoughtless designer can needlessly increase operating cost. Table 6.10 gives three classic examples of poor design decisions from a maintenance viewpoint.

Chapter 8 contains a tabulation of specific design guidelines aimed at reducing cost.

NOTE OF CAUTION:

On p.171 all seven of the operating cost contributions are expressed as a fraction of the total operating cost. Whenever fundamental airplane design parameters are changed, there is no intrinsic reason why the cost items 4-7 should change as the fractions would dictate.

Therefore, whenever a cost trade study is conducted, a baseline should be established for the cost items 4-7. These baseline costs should then be kept constant while the effect of certain design parameters which affect cost items 1-3 is being investigated.

6.10.3 Observations on System Maintainability and Reliability

Chapter 8 and References 25-28 are recommended for further reading about ways to improve airplane maintainability and reliability.

Many military missions require the use of very complex avionics systems. Such systems are required to allow the airplane to function as a weapon system rather than merely as a gunnery or bombing platform.

Also, the nature of military missions, particularly in terms of speed, altitude and weather capability often dictates the use of very advanced technology. This is inherently expensive, risky and to some extent can cause frequent breakdowns.

As a result of poor system and component reliability as well as poor accessibility of those components which require frequent attention, the operational readiness of military airplanes has tended to be poor: see Table 6.11! Table 6.12 indicates how various avionics components affect the time between maintenance. Recollecting that, on the average, military airplanes fly about 350 hours per year, the need for maintenance occasions per year is very high, as shown in Table 6.13. Airplane designers should be able to straighten out their act! Again, Chapter 8 contains some tabulated guidelines which can be helpful in reducing cost.

Table 6.11 EXAMPLES OF FIGHTER READINESS

Airplane Type	Airplane Complexity	Not Mission Capable in percentage	Mean Flight Hours between Failure	Maintenance Events per Sortie	Maintenance Manhours per Sortie
USAF					
A-10	Low	32.6	1.2	1.6	18.4
A-7D	Medium	38.6	0.9	1.9	23.8
F-4E	Medium	34.1	0.4	3.6	38.0
F-15	High	44.3	0.5	2.8	33.6
F-111F	High	36.9	0.3	9.2	74.7
F-111D	High	65.6	0.2	10.2	98.4
USN					
A-4M	Low	27.7	0.7	2.4	28.5
AV-8A	Low	39.7	0.4	3.0	43.5
A-7E	Medium	36.7	0.4	3.7	53.0
F-4J	Medium	34.2	0.3	5.9	82.7
A-6E	High	39.3	0.3	4.8	71.3
F-14A	High	47.1	0.3	6.0	97.8

Source: Aviation Week and Space Technology, October 6, 1980

COURTESY: NORTHROP

ACCESSIBILITY FEATURES

TABLE 6.12 AVIONICS SYSTEM OPERATIONAL TIME BETWEEN MAINTENANCE, IN FLIGHT HOURS

Avionics System	F-111	F-4	F-16	A-7	F-15
Instruments	15	17	60	45	32
Autopilot	13	43	NA	NA	61
UHF	42	22	57	64	24
IFF	35	NA	44	89	15
Navigation	66	4	172	59	18
Fire Control	71	3	9	26	6
ECM	7	7	25	20	22

NA = not available
Source: Aerospace America, August 1985

TABLE 6.13 AVIONICS SYSTEM DOWN OCCURRENCES PER YEAR BASED ON 350 FLIGHT HOURS PER YEAR

Avionics System	F-111	F-4	F-16	A-7	F-15
Instruments	23	21	6	8	11
Autopilot	27	8	NA	NA	6
UHF	8	16	6	5	15
IFF	10	NA	8	8	23
Navigation	5	88	2	6	19
Fire Control	5	117	39	13	58
ECM	50	50	14	18	16

NA = not available Source: Table 6.12

7. EXAMPLE OF LIFE CYCLE COST CALCULATION FOR A MILITARY AIRPLANE

The purpose of this chapter is to present an example of the calculation of life cycle cost (LCC) of a military airplane. The example has been constructed for the Eris ground attack fighter of Parts I and II. An equation for life cycle cost (LCC) was given in Ch.2. as Eqn.(2.3):

$$LCC = C_{RDTE} + C_{ACQ} + C_{OPS} + C_{DISP} \tag{7.1}$$

where: C_{RDTE} is the total cost of research, development, test and evaluation

C_{ACQ} is the program acquisition cost

C_{OPS} is the program operating cost

C_{DISP} is the program disposal cost

More detailed definitions of these cost sources are provided in Chapter 2, pages 9 - 11. Methods for determining the first three sources of LCC are developed in Chapters 3 - 6. Applications of these methods to the Eris fighter program are given in Sections 7.1 - 7.3. An approach to the determination of disposal cost (C_{DISP}) is given in Section 7.4. The life cycle cost of the Eris fighter is computed in Section 7.5. Definitions for airplane unit cost are given and illustrated in Section 7.6. The material is organized as follows:

7.1 Research, Development, Test and Evaluation Cost of the Eris Ground Attack Fighter

7.2 Manufacturing and Acquisition Cost of the Eris Ground Attack Fighter

7.3 Operating Cost of the Eris Ground Attack Fighter

7.4 Disposal Cost of the Eris Ground Attack Fighter

7.5 Life Cycle Cost of the Eris Ground Attack Fighter

7.6 Unit Cost of the Eris Ground Attack Fighter

7.1 RESEARCH, DEVELOPMENT, TEST AND EVALUATION COST OF THE ERIS GROUND ATTACK FIGHTER

In this section the research, development, test and evaluation cost, C_{RDTE}, will be estimated for the Eris ground attack fighter. The mission specification for this fighter is stated in Table 2.19 of Part I. The assumption will be made that the cost estimates are needed for calendar year 1990. The method used is that of Ch.3. To use the method the following input data are needed:

W_{TO} = 64,905 lbs from p.14, Part V.

W_E = 33,500 lbs from p.14, Part V.

V_{max} = 450 kts from p.45, Part V.

N_{rdte} = 10 assumed from p.26.

N_{st} = 2. However, it will be assumed that even the two static test airplanes will be fully equipped with engines and avionics.

F_{diff} = 2.0 assumed from p.26 as well as the design decision to use 'de-facto' stability, which was made in sub-section 11.4.3 of Part II.

F_{cad} = 0.8 assumed from p.27.

From Eqn.(3.5):

$$W_{ampr} = \text{invlog}\{0.1936 + 0.8645(\log 64,905)\} =$$

$$= 22,585 \text{ lbs}$$

<u>Airframe Engineering and Design Cost:</u> C_{aed_r}

From Equation(3.2):

$$MHR_{aed_r} = 0.0396(22,585)^{0.791}(450)^{1.526}(10)^{0.183} *$$

$$*(2.0)(0.8) = 3,002,542 \text{ hrs}$$

R_{e_r} = 62.00 USD/hr from Figure 3.3, which assumes that the Eris program does not require a high level of security.

From Eqn.(3.3):

$$C_{aed_r} = 3,002,542(62.00) = \text{USD } 186,157,604$$

<u>Development Support and Test Cost:</u> C_{dst_r}

CEF for 1990 is 3.03 as determined from Figure 2.7.

From Eqn.(3.7):

$$C_{dst_r} = 0.008325(22,585)^{0.873}(450)^{1.890}(10)^{0.346}*$$

$$*(3.03)(2.0) = \text{USD } 73,166,699$$

<u>Flight Test Airplanes Cost:</u> C_{fta_r}

This follows from Eqn.(3.8). First, all five cost components in this equation will be determined.

<u>Cost of engine and avionics:</u> $C_{(e + a)_r}$

The required takeoff thrust per engine, T_{TO_e}, follows from p.191, Part I:

$$T_{TO_e} = 29,670/2 = 14,835 \text{ lbs}$$

From Eqn.(B9) in Appendix B, the 1990 engine price may be estimated as follows:

$$EP_{1990} = [\text{invlog}\{2.3044 + 0.8858\log(14,835)\}]*$$

$$*(3.03/3.02) = \text{USD } 1,001,757$$

The avionics cost, $C_{avionics}$, will be assumed to be 25 percent of the Eris unit flyaway cost, AMP_{1990}. The latter is estimated from Eqn.(A14) in Appendix A:

$$AMP_{1990} = (3.03/3.02)*$$

$$*[\text{invlog}\{2.3341 + 1.0586(\log 64,905)\}] =$$

$$= \text{USD } 26,903,782$$

Thus, $C_{avionics} = 26,903,782/4 = \text{USD } 6,725,946$

From Eqn.(3.9):

$$C_{(e + a)_r} = [2(1,001,757) + 6,725,946](10) =$$

$$= \text{USD } 87,294,600$$

where it has been assumed that all 10 prototypes, including the 2 used for static testing, will be fully equipped with engines and avionics.

<u>Manufacturing cost of flight test airplanes:</u> C_{man_r}

From Eqn.(3.11):

$$MHR_{man_r} = 28.984(22,585)^{0.740}(450)^{0.543}(10)^{0.524})*$$

$$*(2) = \text{USD } 8,906,551 \text{ hrs}$$

$R_{m_r} = 35$ USD/hr from Figure 3.4

From Eqn.(3.10):

$$C_{man_r} = 35(8,906,551) = \text{USD } 311,729,285$$

<u>Materials cost for the flight test airplanes:</u> C_{mat_r}

From Eqn.(3.12):

$$C_{mat_r} = 37.632(1.0)(22,585)^{0.689}(450)^{0.624}*$$

$$*(10)^{0.792}(3.03) = \text{USD } 31,943,800$$

In this application it is assumed that the Eris is manufactured with primarily conventional materials (so that $F_{mat} = 1.0$) and that the 1990 CEF value is 3.03.

<u>Tooling cost for the flight test airplanes:</u> C_{tool_r}

From Eqn.(3.14):

$$MHR_{tool_r} = 4.0127(22,585)^{0.764}(450)^{0.899} *$$

$$*(10)^{0.178}(0.33)^{0.066}(2.0) =$$

$$= 5,784,225 \text{ hrs}$$

where the factor 0.33 represents the assumed manufacturing rate per month for the prototype phase, see p.33.

The tooling hourly rate is assumed to be:

R_{t_r} = 44 USD/hr from Figure 3.5.

From Eqn.(3.13):

$$C_{tool_r} = 44(5,784,225) = \text{USD } 254,505,900$$

<u>Quality control cost for the flight test airplanes:</u> C_{qc_r}

From Eqn.(3.15):

$$C_{qc_r} = 0.13(311,729,285) = \text{USD } 40,524,807$$

Finally, from Eqn.(3.8):

$$C_{fta_r} = 87,294,600 + 311,729,285 + 31,943,800 +$$

$$+ 254,505,900 + 40,524,807 = \text{USD } 725,998,392$$

<u>Flight Test Operations Cost:</u> C_{fto_r}

From Eqn.(3.16):

$$C_{fto_r} = 0.001244(22,585)^{1.160}(450)^{1.371}(8)^{1.281} *$$

$$*(3.03)(2) = \text{USD } 52,739,614$$

<u>Test and Simulation Facilities Cost:</u> C_{tsf_r}

In sub-section 11.4.3 of Part II the decision was made to use 'de-facto' stability in the Eris design. Therefore, from Eqn.(3.17):

$C_{tsf_r} = 0.20(C_{RDTE})$ as suggested on p.35.

<u>Profit over Flight Test Airplanes:</u> C_{pro_r}

From Eqn.(3.18):

$$C_{pro_r} = 0.1(C_{RDTE})$$

<u>Cost to Finance the Flight Test Airplanes:</u> C_{fin_r}

It will be assumed that the manufacturer will have to arrange for the financing of the Eris fighter program. Consistent with Section 3.7, the financing rate will be assumed to be 0.10 (p.36) so that, with Eqn.(3.19):

$$C_{fin_r} = 0.1(C_{RDTE})$$

<u>Total Research, Development, Test and Evaluation Cost:</u> C_{RDTE}

From Eqn.(3.1):

$$C_{RDTE} = (186,157,604 + 73,166,699 + 725,998,392 + 52,739,614)/(1 - 0.2 - 0.1 - 0.1) =$$

$$= 1,038,062,309/0.6 = \text{USD } 1,730,103,848$$

Rounding this off:

$$C_{RDTE} = 1,730,000,000$$

COURTESY: SAAB

7.2 MANUFACTURING AND ACQUISITION COST OF THE ERIS GROUND ATTACK FIGHTER

In this sub-section the manufacturing and acquisition costs, C_{MAN} and C_{ACQ}, will be estimated for the Eris ground attack fighter. The mission specification for this fighter is stated in Table 2.19 of Part I. The assumption will be made that the cost estimates are needed for calendar year 1990. The method used is that of Ch.4. To use the method the input data for Section 7.1 as well as the following input data are needed:

$N_{program}$ = 750 airplanes will be assumed.

N_{rdte} = 10 from page 178.

This yields with Eqn.(4.1):

N_m = 740

The Eris will be assumed to be manufactured with rather conventional materials, thus: F_{mat} = 1.0 as was also assumed on page 180.

Airframe Engineering and Design Cost of Production Airplanes: C_{aed_m}

This will be determined from Eqn.(4.5b). First, the engineering manhours for the Eris program are found from Eqn.(4.6):

$$MHR_{aed_{program}} = 0.0396(22,585)^{0.791}(450)^{1.526} * (750)^{0.183}(2.0)(0.8) =$$

$$= 6,616,455 \text{ hrs}$$

The engineering hourly rate will be assumed to be 62.00 USD/hr, as on p.178. Next, with Eqn.(4.5b) and with the C_{aed_r} of p.179:

$$C_{aed_m} = (6,616,455)(62.00) - 186,157,604 =$$

$$= 410,220,210 - 186,157,604 = \text{USD } 224,062,606$$

Airplane Program Production Cost: C_{apc_m}

This cost will be computed with Eqn.(4.7). First, all six cost components in this equation will be found.

Cost of engine and avionics of production airplanes: $C_{(e + a)_m}$

This follows from Eqn.(4.8). It will be assumed that the engine and avionics cost for the production version of the Eris fighter is the same as that for the RDTE version. With the cost data from pages 179 and 180:

$$C_{(e + a)_m} = \{(2)(1,001,757 + 6,725,946)\}(740) =$$

$$= (8,729,460)(740) = \text{USD } 6,459,800,400$$

Cost of the interiors of production airplanes: C_{int_m}

This cost is determined from Eqn.(4.9) with $F_{int} = 0$ for military airplanes:

$$C_{int_m} = 0$$

Manufacturing cost of production airplanes: C_{man_m}

Equation (4.10b) will be used to estimate this cost item. First, the program manufacturing manhours are computed from Eqn.(4.11):

$$MHR_{man_{program}} = 28.984(22,585)^{0.740}(450)^{0.543} *$$

$$*(750)^{0.524}(2) = 85,554,269 \text{ hrs}$$

The manufacturing labor rate will be assumed to be the same as that used in the RDTE phase on p.180:

R_{m_m} = 35 USD/hr. Thus, with Eqn.(4.10b) and with the data of p.180:

$$C_{man_m} = (85,554,269)(35.00) - 311,729,285 =$$

$$= 2,994,399,415 - 311,729,285 = \text{USD } 2,682,670,130$$

Materials cost for production airplanes: C_{mat_m}

The materials cost of the production program is determined with Equation (4.12). First, the materials cost of the entire program is found from Eqn.(4.13):

$$C_{mat_{program}} = 37.632(1.0)(22.585)^{0.689}(450)^{0.624} *$$

$$*(750)^{0.792}(3.03) = \text{USD } 975,966,932$$

where $F_{mat} = 1.0$ and $CEF_{1990} = 3.03$ were obtained from page 180.

Next, the cost of materials for the production program follows from Eqn.(4.12) as:

$$C_{mat_m} = 975,966,932 - 31,943,800 = \text{USD } 944,023,132$$

Tooling cost for production airplanes: C_{tool_m}

This cost item will be calculated with Eqn.(4.14b). First, the program tooling manhours are determined with Eqn.(4.15):

$$MHR_{tool_{program}} = 4.0127(22.585)^{0.764}(450)^{0.899} *$$

$$*(750)^{0.178}(10)^{0.066}(2) =$$

$$= 15,623,711 \text{ hrs}$$

In this calculation it was assumed that the Eris will be manufactured at a rate of $N_{r_m} = 10$ airplanes per month. For a production program of 740 units, this represents a 6 year program. That is quite reasonable for this type of airplane.

The tooling hourly rate will be assumed to be equal to that used in the RDTE phase on p.181:

$$R_{t_m} = 44.00 \text{ USD/hr}$$

Now, with Eqn.(4.14b) and the data from p.181:

$$C_{tool_m} = (15,623,711)(44.00) - 254,505,900 =$$

$$= 687,443,284 - 254,505,900 = \text{USD } 432,937,384$$

<u>Quality control cost for production airplanes:</u> C_{qc_m}

This cost will be found with Eqn.(4.16):

$$C_{qc_m} = 0.13(2,682,670,130) = \text{USD } 348,747,117$$

It is now possible to determine the entire airplane production cost from Eqn.(4.7) as follows:

$$C_{apc_m} = 6,459,800,400 + 0 + 2,682,670,130 +$$

$$+ 944,023,132 + 432,937,384 + 348,747,117 =$$

$$= \text{USD } 10,868,178,163$$

<u>Cost of Flight Test Operations for Production Airplanes:</u> C_{fto_m}

This cost is estimated with Eqn.(4.17). On p.167 it was found that for the Eris: $C_{ops/hr} = 4,249$ USD/hr.

However, this operating cost includes the effects of military overhead, depot, spares and other factors. It is necessary to eliminate these factors from the p.167 operating cost before Eqn.(4.17) can be applied. Instead of doing this, it is assumed here that the operating cost per hour to the manufacturer is roughly 2,000 USD/hr.

Production airplane flight test hours per airplane are assumed to be:

$t_{pft} = 20$ hrs/airplane as stated on p.55.

The manufacturers flight test overhead factor is assumed to be: $F_{ftoh} = 4.0$ (on p.55). So, with Eqn.(4.17):

$$C_{fto_m} = (740)(2,000)(20)(4.0) = \text{USD } 118,400,000$$

<u>Manufacturing Program Financing Cost:</u> C_{fin_m}

The assumption will be made that the financing cost

for the manufacturing phase of the Eris program will be carried by the manufacturer and/or his lenders. A rate of 10 percent will be assumed. Thus, with Eqn.(4.18):

$$C_{fin_m} = 0.10(C_{MAN})$$

Program Manufacturing Cost: C_{MAN}

This cost can now be found from Eqn.(4.4) and the data determined in this section:

$$C_{MAN} = (224,062,606 + 10,868,178,163 +$$

$$+ 118,400,000)/(1 - 0.1) = 11,210,640,769/0.9 =$$

$$= \text{USD } 12,456,000,000 \text{ after some rounding off.}$$

Profit over the manufacturing phase: C_{pro_m}

According to the suggestion made on p.56, a 10 percent profit for the manufacturing phase is reasonable. Therefore, with Eqn.(4.19):

$$C_{PRO} = (0.10)(C_{MAN}) = \text{USD } 1,245,600,000$$

Program Acquisition Cost: C_{ACQ}

The total Eris program acquisition cost is estimated from Eqn.(4.2) as:

$$C_{ACQ} = 12,456,000,000 + 1,245,600,000 =$$

$$= \text{USD } 13,701,600,000$$

or, after rounding off:

$$C_{ACQ} = \text{USD } 13,702,000,000$$

7.3 OPERATING COST OF THE ERIS GROUND ATTACK FIGHTER

The program operating cost of the Eris fighter was determined with the method of Chapter 6 in Section 6.9. The result from p.167 is:

$$C_{OPS} = \text{USD } 22,755,000,000$$

7.4 DISPOSAL COST OF THE ERIS GROUND ATTACK FIGHTER

There comes a point in time when any airplane no longer has commercial or military value. That point is normally reached when:

1. The airplane has reached the end of its safe structural life and structural repairs are judged to be not economical.

2. The airplane has reached the end of its economic life: it can no longer compete effectively in the face of more modern airplanes.

3. The airplane ceases to have military value. This can occur if either its mission is no longer a viable one, or improvements in technology have rendered it obsolete.

4. It has been damaged beyond repair (such as in certain crashes or by damage caused by weather).

Whenever an airplane ceases to have commercial or military value it needs to be disposed of. The cost associated with such disposal is referred to as the disposal cost. The cost of disposing of airplanes depends on the types of materials used in construction and/or in operation. Disposal usually consists of:

1. Temporary storage

2. Draining of liquids and disposal thereof

3. Disassembly of engines and certain systems (such as computers and instruments)

4. Cutting up of the airframe and disposal of the resulting materials

There are costs associated with each of these actions. However, it is quite possible that there is a salvage value associated with actions 2-3. Examples are: resale value of engines, instruments, scrap metals (for re-cycling) and other materials. Any disposal costs are therefore partially offset by these resale values.

In some airplanes, materials and liquids are used which pose a problem for the environment if they are not disposed of with care. Examples are:

* Beryllium alloys
* Most composites (because they are not inherently degradable)
* Many oils and liquids (such as hydraulic fluids)

The author believes that it should be the responsibility of designers to include in their design decision making some serious thinking (and actions) about disposal problems. It is not ethical to assume that future generations will be able to 'take care' of problems created by design decisions of today.

To responsibly account for future disposal problems will require that designers enlist the help of chemical and environmental engineers in the design decision making process. Only that way can the cost of disposal be estimated with credibility.

One of the reasons that the earth environment is in serious trouble today is the fact that disposal problems are not included in design thinking and design decision making at most levels of industrial activity today.

Examples of potential problems which were caused by the lack of disposal considerations in design thinking are given in Table 7.1.

The author is not aware of any reliable methods for the estimation of airplane disposal costs. Reference 8 mentions disposal cost but provides no guidelines for its estimation.

In view of this sad state of affairs, it is suggested to use the following 'fudge' factor to account for the disposal of airplanes:

$$C_{DISP} = 0.01(LCC) \qquad (7.2)$$

Whether or not this accounts fairly for the actual balance between resale values and disposal costs is very much an open question.

7.5 LIFE CYCLE COST OF THE ERIS GROUND ATTACK FIGHTER

The life cycle cost of the Eris fighter program will now be determined with Equation (7.1). The cost sources in this equation have the following magnitudes for the Eris fighter program:

TABLE 7.1 EXAMPLES OF POTENTIAL ENVIRONMENTAL PROBLEMS

Item	Problem
Nuclear waste	Insufficiently considered in the original decision to move ahead with fission power generation.
Lithium batteries	Used in thousands of personal computers. What will happen to this hazardous waste when personal computers are discarded?
Americium	This hazardous material is used in many smoke detectors. What will happen to this material when smoke detectors are discarded?
Non-biodegradables	These materials are used in nearly all 'fast food' containers today. Their disposal is creating massive waste piles in the environment.
Composites in airplanes and cars	The use of various types of composites is rapidly increasing. What will happen to these materials when disposal time comes around?

Source: This table is based on ideas by Mr. Ben Williams of the Air Force Institute of Technology.

C_{RDTE} = USD 1,730,000,000 (from p.182)

C_{ACQ} = USD 13,702,000,000 (from p.187)

C_{OPS} = USD 22,755,000,000 (from p.187)

C_{DISP} = USD 386,000,000 (with Eqn.(7.2))

With Equation (7.1) this yields:

$LCC_{Eris\ program}$ = USD 38,573,000,000

Observe, that for the Eris program:

$C_{OPS} \gg C_{ACQ} \gg C_{RDTE}$

This was also noted on p.11 in Chapter 2. It was further substantiated by the Ourania jet transport cost calculations which are summarized on p.132 in Ch.5.

The reason why the operating cost of the Eris program is 'only' twice that of the acquisition cost is that Eris is a relatively small airplane. The Eris has a crew of only one and consumes much less fuel per mission than bomber or transport airplanes. Also, the annual utilization of the Eris is much less than that of typical civil transports.

The relative magnitudes of the first three cost sources in the LCC equation do suggest one more time:

THE PRELIMINARY DESIGN PHASE (PART OF RDTE) LOCKS IN MOST OF THE LIFE CYCLE COST OF AN AIRPLANE!

This lesson should not be lost on designers nor on design managers.

7.6 UNIT COST OF THE ERIS GROUND ATTACK FIGHTER

The problem with determining the unit cost of military airplanes is largely one of definition. The author has decided to use the following definitions:

1) The manufacturing unit cost is defined as:

$$C_{muc} = (C_{MAN})/(N_m) \tag{7.3}$$

2) The acquisition unit cost is defined as:

$$C_{auc} = (C_{ACQ})/(N_m) \tag{7.4}$$

3) The program unit cost is defined as:

$$C_{puc} = (C_{ACQ} + C_{RDTE})/(N_m) \tag{7.5}$$

4) The life cycle unit cost is defined as:

$$C_{lcuc} = (LCC)/(N_m) \tag{7.6}$$

The reader will observe that the third definition corresponds to what for commercial airplanes was called the airplane estimated price, AEP, in Equation (4.3).

In the popular press the concept 'unit flyaway cost' is often used. However, the definition of this 'unit flyaway cost' is usually omitted, leaving the reader to guess how it is defined. The author believes that the definitions 1), 2) and 3) are useful in comparing unit

costs of airplanes with similar missions. Airplane designers should keep in mind the fact that only the fourth definition reflects the actual unit cost to the taxpayer!

It is instructive to use the Eris fighter as an example of what these unit costs actually are.

With Eqn.(7.3) and p.187, the manufacturing unit cost is estimated as:

$$C_{muc} = 12,456,000,000/740 = \text{USD } 16,832,432$$

or, rounded off:

$$C_{muc} = \text{USD } 17,000,000$$

With Eqn.(7.4) and p.187, the acquisition unit cost is estimated as:

$$C_{auc} = 13,702,000,000/740 = \text{USD } 18,516,216$$

or, rounded off:

$$C_{auc} = \text{USD } 19,000,999$$

With Eqn.(7.5), p.187 and p.182, the program unit cost is estimated as:

$$C_{puc} = (13,702,000,000 + 1,730,000,000)/740 =$$

$$= \text{USD } 20,854,054$$

or, rounded off:

$$C_{puc} = \text{USD } 21,000,000$$

With Eqn.(7.6) and p.191, the life cycle unit cost is estimated as:

$$C_{lcuc} = (38,573,000,000)/740 = \text{USD } 52,125,676$$

or, rounded off:

$$C_{lcuc} = 52,000,000$$

Note that the airplane market price estimate for 1990 as obtained from Equation (A14) in Appendix A (see p.179) represents an over-estimate! Since the avionics price estimate of p.180 is based on a percentage of AMP the actual Eris cost estimates should be revised downward accordingly. This was not done in this text.

8. AIRPLANE DESIGN OPTIMIZATION AND DESIGN-TO-COST

A major question which every airplane designer asks himself at one time or another is: 'How can airplanes be designed so that they are the best possible?'

The problem with the use of the word 'best' is that it is not immediately clear what is meant by it. Therefore, it is 'better' to ask the following questions:

A) How can airplanes be designed so that their performance is maximized?

B) How can airplanes be designed so that their empty weight is minimized?

C) How can airplanes be designed so that their cost is minimized?

D) How can airplanes be designed to a given cost?

In mathematics, the words MAXIMUM and MINIMUM are both related to the concept: OPTIMUM. Maxima or minima arise by attempts to optimize certain functions of independent variables. Such functions are given the generic name: cost functions. The mathematical theory of optimization of cost functions is well developed and documented: References 29 and 30 are typical examples.

In airplane design, examples of cost functions are:

* Range
* Endurance
* Fuel consumption
* Empty weight
* Operating cost
* Return on investment

Before these cost functions can be optimized it will be necessary to express them functionally in terms of a finite set of design variables. After reviewing the material in Parts I through VII it will be agreed that the number of airplane design variables is very large indeed. Also, many of these variables are not truly independent, AND their interrelationship is not always definable.

That is why the airplane design process in Parts I and II was referred to as non-unique and iterative. It is useful to review these terms before returning to the idea of design optimization.

The airplane design process is <u>non-unique</u> because:

1. It is possible that more than one configuration approach results in an acceptable airplane. This is clearly illustrated by Figure 1.1 in Part II!

2. It is possible that more than one approach to the interior layout design and systems design leads to an acceptable airplane. Parts II, III and IV contain many examples of this.

The airplane design process is <u>iterative</u> because:

1. Most design decisions made early in the airplane design process turn out to have major (and often not foreseen) impact on design decisions which are made later. If that impact is unfavorable, many earlier decisions have to be reconsidered: i.e. the design has to be iterated*.

2. Except for a relatively small number of design variables (to be discussed later), it has not yet been found possible to relate design variables to each other in a mathematically unique manner. Besides, many airplane design variables are not even independent in the mathematical sense.

To arrive at an acceptable airplane design solution, many design decisions are therefore made on the basis of <u>experience or intuition</u>. This in turn makes it necessary to iterate through previous design decisions. The need for this was amply illustrated in Parts I, II and V. The iterative nature of the design decision-making process is 'formalized' through the step-by-step design procedures called: Design Sequences I and II as outlined in Part II.

It should be clear from Parts I - VII and also from Chapters 3 - 7 in this volume that it is nearly always desirable to keep the takeoff weight of an airplane (for a given mission) as low as possible. It is therefore essential that the designer understand the sensitivity of his design to those mission parameters and those design variables which have been found by experience to have a major impact on the takeoff weight of a new airplane. The problem of takeoff weight sensitivity was dealt with in some detail in Part I, Chapter 2. Applications and interpretations of the takeoff weight sensitivity of a new airplane are presented in Section 8.1.

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* The process of arriving at a satisfactory weight and balance situation (Part II, Chapter 10) is one example!

Note the use of the word 'acceptable' (not optimum) in previous paragraphs. To find a true optimum design solution, the following steps must be completed:

1) One or more cost functions must be defined. These cost functions must be consistent with the mission objectives of the airplane AND they must not conflict with each other.

2) Mathematical relationships between each cost function and certain design variables must be formulated.

3) A rule must be formulated which makes clear how the cost function(s) is (are) to be evaluated: maximum or minimum AND whether or not design constraints are to be accounted for.

4) Design constraints and their relationship to certain design variables must be identified.

5) A theory (mathematical procedure) by which the optimum will be found must be agreed upon.

It depends on the specifics of a particular design problem how these steps are to be taken. Sections 8.2 through 8.6 contain suggestions and guidelines to help airplane design students find their way through the maze called: design optimization. Several examples of design optimization studies are discussed in Section 8.7.

Decisions to go ahead with an airplane program are rarely made without economic considerations. Minimizing economic cost is therefore a 'driver' in many airplane design and airplane program decisions.

More and more, cost itself is being viewed as a constraint. In the design and development of military airplanes, budget limitations are playing an increasingly important role. Such budget limitations must be viewed by the designer as a constraint.

Similarly, in the design and development of commercial airplanes, the acquisition cost (cost of ownership) as well as the direct operating cost are being viewed as design constraints.

This raises the question of how to design airplanes within certain cost constraints: design-to-cost. An introduction to this subject is given in Section 8.8.

Assume that the 'best' of all available design optimization tools have been employed in the design of a new airplane. There still are many aspects of the design which were untouched by the process of optimization, if only because the number of variables accounted for in any formal design optimization procedure is relatively small.

Examples of design variables left untouched by most design optimization processes are found in areas such as:

* interior layout design
* systems design and integration
* materials selection
* application of new, maturing technologies

It is in these areas where the experience and the intuition of the designer are the only method to keep cost down. Therefore, it is useful for the designer to have available a list of 'detail design guidelines' which have been found helpful in lowering airplane cost. Such guidelines are given in Section 8.9.

The material in this chapter is therefore organized as follows:

8.1 Takeoff Weight Sensitivity
8.2 Example Cost Functions in Airplane Design
8.3 Relation Between Cost Functions and Airplane Design Variables
8.4 Design Constraints in Airplane Design
8.5 Rules for Evaluating Cost Functions
8.6 Methods for Design Optimization
8.7 Examples of Design Optimization Studies
8.8 An Introduction to the Design-to-Cost Problem
8.9 Design Guidelines for Lowering Cost

8.1 TAKEOFF WEIGHT SENSITIVITY

It was shown in Chapters 3 - 7 that airplane takeoff weight plays an almost dominant role in determining cost. For that reason it is essential that airplane designers understand what 'drives' the takeoff weight. The purpose of this section is to review and discuss the sensitivity of airplane takeoff weight to mission performance parameters and to airframe design parameters.

A mathematical derivation of the sensitivity of airplane takeoff weight to such parameters can be found in Chapter 2 of Part I. In that chapter, the takeoff weight sensitivity to certain design parameters is expressed as the partial derivative: $\partial W_{TO}/\partial y$, where: y is any one of the following parameters:

* Empty weight, W_E
* Payload weight, W_{PL}

and, for any given mission phase i*:

* Range, R_i
* Endurance, E_i
* Lift-to-drag ratio, $(L/D)_i$
* Specific fuel consumption, c_{j_i} or c_{P_i}
* Propeller efficiency, η_{P_i}

The validity of these takeoff weight sensitivities depends on the validity of a statistical relationship between airplane takeoff weight and airplane empty weight which can be written as:

$$\log W_{TO} = A + B \log W_E \qquad (8.1)$$

where: W_{TO} is the takeoff weight in lbs

W_E is the empty weight in lbs

A and B are regression coefficients determined from weight data on existing airplanes

It was shown in Chapter 2 of Part I that this type of relationship exists for airplanes with similar mission orientation. The scatter of actual takeoff weight and empty weight values relative to Eqn.(8.1) was shown to be

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*For a definition of what is meant by mission phases, see Chapter 2 in Part I.

small. Numerical values for the regression coefficients A and B are given in Table 2.15 of Part I for twelve types of airplanes with similar mission orientation.

Whenever new, lighter structural materials are used in a new design, the validity of Eqn.(8.1) is in doubt. A method to account for the use of advanced materials is given in Appendix B of Part I.

Anytime a statistical method is employed it might be expected that the method becomes highly suspect whenever new, untried configurations are proposed. A discussion of circumstances which can 'drive' a designer toward new and untried configurations is presented in Reference 31.

The significance of takeoff weight sensitivities to an airplane designer is best illustrated with an example. For this example an airplane designed to fill the market void identified as Void I in Figure 9.4 (next chapter!) will be used. A preliminary mission specification for such an airplane is given in Table 8.1. For lack of a better name the airplane will be referred to as the SLRT: Small Long Range Transport.

The takeoff weight sensitivities for the SLRT are determined with the help of a user-friendly, preliminary design computer program called AAA: Advanced Aircraft Analysis program. The main features of this program are described in Reference 32. The takeoff weight sensitivities of the SLRT are determined in three steps:

<u>Step 1:</u> Calculation of the regression coefficients A and B in Eqn.(8.1).

Figure 8.1 shows a 'screendump' of the AAA program after entering takeoff weight and empty weight data for the 23 jet transports identified in Table 8.2. Note the following values for A and B:

$$A = 0.15 \qquad\qquad B = 1.02$$

Figure 8.2 shows a logarithmic plot of the weight data from Table 8.2. Note the small amount of scatter, thus supporting the validity of using Equation 8.1.

The mission specification of Table 8.1 is a very aggressive one: there is no such airplane in existence in 1990! It can be expected that to design a workable SLRT, use of more advanced structural materials (compared to those used in the 23 airplanes of Table 8.2) will be required. For that reason, Equation (B2) of Appendix B

Table 8.1 Mission Specification for the Small, Long Range Transport (SLRT)

Passengers: 125 (175 lbs each plus 40 lbs baggage)
Crew: 2 + 4 = 6 (175 lbs each plus 40 lbs baggage)

Range:	5,000 nm still air range
Average cruise altitude:	45,000 ft
Design cruise speed:	450 kts
Fuel reserves:	10 percent of mission fuel used

Table 8.2 Transport Jet Data Used in Obtaining the Regression Coefficients for Eqn.(8.1)

No.	Type	Takeoff Weight (lbs)	Empty Weight (lbs)
1.	Canadair 600	41,100	22,650
2.	601	43,100	23,940
3.	Airbus A300-600	363,765	188,291
4.	600R	375,885	187,251
5.	Airbus A310-200	305,560	166,742
6.	300	330,695	167,477
7.	Airbus A320-100	145,505	83,062
8.	-200	158,730	84,586
9.	Fokker F-100-A	95,000	51,720
10.	Yakovlev 42	124,560	76,180
11.	Tupolev 204	206,130	124,780
12.	British Aerospace 146-100	84,000	48,150
13.	-200	93,000	49,505
14.	Boeing 737-200	124,500	59,738
15.		115,500	59,500
16.	737-400	138,500	71,578
17.		150,000	72,520
18.	757-LR	250,000	124,380
19.	767	300,000	176,470
20.		400,000	195,770
21.	Gulfstream GIV	71,700	40,312
22.	McDonnell Douglas MD-81-A	140,000	77,291
23.	-C	160,000	79,333

Source: Jane's All the Worlds Aircraft, 1989

DATA BASE FILE : ROSKAM

ADVANCED AIRCRAFT ANALYSIS (AAA)

[redacted]	GEOMETRY	DRAG POLAR	WEIGHT & BALANCE	PERF. ANALYSIS	DYNAMICS	COST ANALYSIS	H E L P
PERF. SIZING	HIGH LIFT	STAB. & CONTROL	INST. THRUST	S&C DERIVATIVES	CONTROL	DATA BASE	Q U I T

WEIGHT SIZING

FUEL-FRACTION	TAKE-OFF WEIGHT	[redacted]	SENSITIVITY COEF	AIRPLANE TYPE	R E T U R N

WEIGHT TABLES

NEW TABLE	LOAD TABLE	[redacted]	CURRENT TABLE	PLOT WEIGHTS	R E T U R N

Take-off Weight and Empty Weight

C A L C U L A T E	R E T U R N
ADD ROW	DELETE UNDEFINED
LOAD TABLE	SAVE TABLE

	Wto (lb)	We (lb)		Wto (lb)	We (lb)
1	41100.0	22650.0	13	93000.0	45505.0
2	43100.0	23940.0	14	124500.0	59730.0
3	363765.0	188291.0	15	115500.0	59500.0
4	375805.0	187251.0	16	138500.0	71570.0
5	305560.0	166742.0	17	150000.0	72520.0
6	330695.0	167477.0	18	250000.0	124300.0
7	145505.0	83062.0	19	300000.0	176470.0
8	150730.0	84506.0	20	400000.0	195770.0
9	85000.0	51720.0	21	71700.0	40312.0
10	124560.0	76100.0	22	140000.0	77291.0
11	206130.0	124700.0	23	150000.0	79333.0
12	84000.0	40150.0			

Regression Coefficients A and B

A	B
0.15	1.02

Figure 8.1 AAA Screendump of Takeoff Versus Empty Weight Regression Analysis

Figure 8.2 AAA Screendump of Logarithmic Plot of Takeoff Weight Versus Empty Weight

in Part I is used to find an 'improved' value for the regression coefficient A. The assumption is made that by judicious use of composites and Li-Al alloys a 5 percent reduction in empty weight is achievable: $\eta = 0.95$, thus:

$$A_{new} = 0.15 - 1.02\log(0.95) = 0.173$$

The assumption has been made here that the regression coefficient B is NOT affected by the choice of new structural materials.

<u>Step 2:</u> Calculation of takeoff weight for the SLRT.

The takeoff weight of the SLRT can be estimated from Equation (2.24) in Part I, which is:

$$\log W_{TO} = A + B\log(CW_{TO} - D) \tag{8.2}$$

where: $C = \{1 - (1 + M_{res})(1 - M_{ff}) - M_{tfo}\}$ (8.3)

with: M_{res} the reserve mission fuel fraction, specified in Table 8.1: $M_{res} = 0.10$.

M_{ff} the overall mission (used) fuel fraction, see Eqn.(2.13), p.16, Part I.

M_{tfo} = trapped fuel and oil fraction, see p.69, Part I. Use: $M_{tfo} = 0.005$.

$$D = W_{PL} + W_{crew} \tag{8.4}$$

where: W_{PL} is the payload weight in lbs

W_{crew} is the crew weight in lbs

Next, the AAA program of Ref.32 is used to compute the overall mission fuel fraction, M_{ff}. The critical mission phase for M_{ff} is the one corresponding to the extreme range requirement of Table 8.1. Figure 8.3 shows the input data needed to find the cruise fuel fraction: $(W_{i+1}/W_i)_{cruise}$. Note the aggressive use of:

aerodynamics technology: $(L/D)_{average} = 21.0$ and of

engine technology: $c_j = 0.5$ lbs/lbs/hr

Figure 8.4 shows the result of the takeoff weight calculation: $W_{TO} = 152.548$ lbs.

DATA BASE FILE : ROSKAM

ADVANCED AIRCRAFT ANALYSIS (AAA)

[illegible]	GEOMETRY	DRAG POLAR	WEIGHT & BALANCE	PERF. ANALYSIS	DYNAMICS	COST ANALYSIS	HELP
PERF. SIZING	HIGH LIFT	STAB. & CONTROL	INST. THRUST	S&C DERIVATIVES	CONTROL	DATA BASE	QUIT

WEIGHT SIZING

[illegible]	TAKE-OFF WEIGHT	REGRESSION COEFF	SENSITIVITY COEF	AIRPLANE TYPE	RETURN

MISSION PROFILE	Wi+1/Wi
Warmup	0.9950
Taxi	0.9950
Take-off	0.9950
Climb	0.9891
[illegible]	0.7344
Descent	0.9900
Land/Taxi	0.9900
SEGMENT COMMANDS	

SEGMENT INPUT AND FUEL-FRACTION CALCULATION

NEW SEGMENT	[illegible]	DELETE SEGMENT	INSERT SEGMENT	MOVE SEGMENT	FUEL-FRACTION	HELP	RETURN

Fuel Used in Cruise Segment for a Jet Driven Aircraft

CALCULATE	THEORY	SAVE OUTPUT	RETURN
Range = 5000.0 n.mi	V = 450.00 kts	Cj = 0.500 #/#-hr	L/D = 21.00

Output Parameter

Wi+1/Wi = 0.7675

Figure 8.3 AAA Screendump of Fuel Fraction Calculation for a Cruise Mission Phase

DATA BASE FILE : ROSKAM

ADVANCED AIRCRAFT ANALYSIS (AAA)

[illegible]	GEOMETRY	DRAG POLAR	WEIGHT & BALANCE	PERF. ANALYSIS	DYNAMICS	COST ANALYSIS	HELP
PERF. SIZING	HIGH LIFT	STAB. & CONTROL	INST. THRUST	S&C DERIVATIVES	CONTROL	DATA BASE	QUIT

WEIGHT SIZING

FUEL-FRACTION	[illegible]	REGRESSION COEFF	SENSITIVITY COEF	AIRPLANE TYPE	RETURN

MISSION PROFILE	Wi+1/Wi
Warmup	0.9950
Taxi	0.9950
Take-off	0.9950
Climb	0.9891
Cruise	0.7676
Descent	0.9900
Land/Taxi	0.9900

Take-off Weight

CALCULATE	THEORY	SAVE OUTPUT	RETURN

A	= 0.1738	Wp expend =	0.0 lb	FFunusabl =	0.500 %	WF correc =	0.0 lb
B	= 1.0236	WTOest =	100000.0 lb	Fuel Resr =	10.000 %		
Mff uc	= 0.7329	Wp + Wcrm =	20165.0 lb	Delta WE =	0.5000 %		

Output Parameters

WE	= 70807.1 lb	Wfmission =	44813.0 lb	Mff	= 0.7329
WFuelused =	40739.1 lb	Wto	= 152547.9 lb		

Figure 8.4 AAA Screendump of Overall Mission Fuel Fraction and Takeoff Weight Calculation

Step 3: Calculation of takeoff weight sensitivities.

Formulas for $\partial W_{TO}/\partial y$ were derived in Section 2.7 of Part I. Because of the proposed aggressive use of the following technologies:

* materials: A * aerodynamics: L/D * engine: c_j

it is particularly important that the sensitivity of takeoff weight to these three parameters be understood.

Figure 8.5 shows a AAA screendump of the relevant sensitivity calculations. An interpretation of these results is now given:

1) Sensitivity of takeoff weight to empty weight: W_E

   From Figure 8.5:

   $\partial W_{TO}/\partial W_E$ = 1.98 (2.0 rounded off)

   This signifies that for each extra pound of empty weight, the takeoff weight grows by 2 lbs. Such growth in weight might occur as a result of not being able to achieve the assumed 5 percent empty weight reduction and/or by careless structural design. Management should exercise extra careful weight control during the detail design phase!

2) Sensitivity of takeoff weight to cruise specific fuel consumption: c_j

   From Figure 8.5:

   $\partial W_{TO}/\partial c_j$ = 331,591 (332,000 lbs/lbs/hr/lbs, rounded off)

   This signifies that if the engine s.f.c is not 0.5, but say 0.55, the takeoff weight would increase by ΔW_{TO} = 0.05x332,000 = 16,600 lbs!

   Management has to watch out that the engine manufacturer can indeed achieve the promised levels of cruise specific fuel consumption!

3) Sensitivity of takeoff weight to cruise lift-to-drag ratio: L/D

   From Figure 8.5:

DATA BASE FILE : ROSKAM

ADVANCED AIRCRAFT ANALYSIS (AAA)

	GEOMETRY	DRAG POLAR	WEIGHT & BALANCE	PERF. ANALYSIS	DYNAMICS	COST ANALYSIS	HELP
PERF. SIZING	HIGH LIFT	STAB. & CONTROL	INST. THRUST	S&C DERIVATIVES	CONTROL	DATA BASE	QUIT

WEIGHT SIZING

FUEL-FRACTION	TAKE-OFF WEIGHT	REGRESSION COEFF		AIRPLANE TYPE	RETURN

SENSITIVITY STUDY AND GROWTH FACTORS : INPUT

CALCULATE	THEORY	NEXT PAGE	RETURN

B =	1.0230	Wff uc =	0.7329	Wp expend =	0.0 lb	Wp + Wcrew =	20165.0 lb
Ffunusabl =	0.500 %	Fuel Resr =	10.000 %	WF correc =	0.0 lb	Wto =	152547.9 lb
WE =	70807.1 lb						

SENSITIVITY : OUTPUT

dWto/dWpl =	5.09	dWto/dWe =	1.98

MISSION PROFILE	dWto/dWpExp	dWto/dCj lb/hr	dWto/dR lb/nm	dWto/dLD lb	dWto/dE lb/hr
Warmup					
Taxi					
Take-off					
Climb		12534.1		-459.5	22879.2
Cruise		331598.9	33.1	-7895.0	
Descent					
Land/Taxi					

Figure 8.5 AAA Screendump of Calculation of Takeoff Weight Sensitivity to Mission and Technology Parameters

$\partial W_{TO}/\partial(L/D) = -7895.0$ (-7,900 lbs per unit L/D, rounded off)

This signifies that if the average cruise L/D is not 21.0, but say 19.0, the takeoff weight would increase by $\Delta W_{TO} = 2x7,900 = 15,800$ lbs!

Management has to exercise careful control over the aerodynamics department to ensure that the projected high cruise L/D can indeed be achieved!

4) Sensitivity of takeoff weight to a change in the required minimum fuel reserves of 10 percent as stipulated in Table 8.1.

Suppose that the airworthiness authorities would stipulate an increase in minimum fuel reserves from 10 percent to 15 percent. This increase can be interpreted as a change in the requirement for cruise range from 5,000 nm to 5,250 nm while keeping the old ten percent reserve number.

<u>From Figure 8.5:</u>

$\partial W_{TO}/\partial R = 33.1$ (33.0 lbs/nm rounded off)

This signifies that an increase of 250 nm range due to an increase in fuel reserve minimums would force an increase in takeoff weight of:

$\Delta W_{TO} = 250x33.0 = 8,250$ lbs

Before launching a project such as the SLRT it is essential that the designer assures himself that critical fuel reserve rules are accepted by all parties concerned!

It should be clear from this discussion that takeoff weight is critically dependent on several design parameters. The prelimininary designer should determine the sensitivity of takeoff weight to these parameters early in the design process. In this manner, management can be guided to take the appropriate action with regard to the establishment of controls on critical design parameters.

Having thusly established the sensitivity of a new design to critical design parameters the designer should initiate studies of airplane design optimization. The first step on the road to optimization studies is the identification of cost functions. Section 8.2 presents a discussion of potential airplane design cost functions.

8.2 EXAMPLE COST FUNCTIONS IN AIRPLANE DESIGN

Airplane designers are confronted with a wide range of cost functions which are potential candidates for optimization. Cost functions in airplane design can be generically classified as follows:

1) Performance Cost Functions
2) Weight Cost Functions
3) Economic Cost Functions

Sub-sections 8.2.1 through 8.2.3 provide several examples of each. A major problem in the conduct of airplane design optimization studies is the very selection of cost functions. Cost functions, to be useful, must be consistent with the mission objective of an airplane, AND they must not conflict with each other. A brief discussion of this problem is given in sub-section 8.2.4.

8.2.1 Examples of Performance Cost Functions

The reader should review the example mission specifications of Tables 2.17 through 2.19 in Part I. It is seen that in these mission specifications, several performance characteristics appear repeatedly:

1. Cruise speed, V_{cr} 2. Time-to-climb to altitude, t_{cl}

3. Cruise range, R_{cr} 4. Loiter time (or endurance), E

As long as everything else is the same, performance cost functions 1, 3 and 4 beg to be maximized while cost function 2 begs to be minimized, particularly in the case of fighter interceptors.

Before these cost functions can be optimized it is necessary to relate them to a set of design parameters. How to relate typical cost functions to airplane design parameters is discussed in Section 8.3.

8.2.2 Examples of Weight Cost Functions

The reader is referred again to the mission specifications of Tables 2.17 through 2.19 in Part I. Payload is seen to be a prominent requirement in all. Furthermore, from the performance sizing procedures outlined in Part I, it is obvious that takeoff weight, empty weight and fuel weight all play an important role. Therefore, the following weights suggest themselves as potential candidates for weight cost functions:

1. Payload weight, W_{PL} 2. Takeoff weight, W_{TO}

3. Empty weight, W_E 4. Fuel weight, W_F

It is clear that weight cost functions 2 and 3 beg to be minimized. In any given airplane, the sum of payload weight and fuel weight is referred to as the useful load:

$$W_{US} = W_{PL} + W_F \tag{8.5}$$

It is the useful load (sum of the cost functions 1 and 4) which beg to be maximized.

In Part V, page 26, Eqn.(4.2), the empty weight, W_E, was expressed as:

$$W_E = W_{struct} + W_{pwr} + W_{feq} \tag{8.6}$$

where: W_{struct} is the airplane structural weight

W_{pwr} is the powerplant weight

W_{feq} is the fixed equipment weight

Each of these three empty weight components can be viewed as a weight cost function: their minimization is certainly desirable.

Before any optimization can take place, a mathematical relation between the weight cost functions and a set of design variables will have to be developed. That is discussed in Section 8.3.

8.2.3 Examples of Economic Cost Functions

For airplane operators the economic aspects of airplanes is of prime importance. This is certainly so for commercial operators. However it is true to a large extent even for military operators. Therefore, cost must be made as low as possible. From discussions in Chapters 2 through 7 it follows that candidates for economic cost functions could be:

1. Manufacturing cost, C_{MAN} 2. RDTE cost, C_{RDTE}

3. Direct operating cost, DOC 4. Life cycle cost, LCC

In addition, the following economic cost functions suggest themselves:

5. Return on investment, ROI 6. Profit potential, P_{pot}

Before any of these cost functions can be used in an optimization study it is necessary to relate them to airplane design variables. Examples of how some of these economic cost functions are related to design variables are discussed in Section 8.3.

8.2.4 Consistency of, and Conflict Between Cost Functions

For any airplane design cost function to be useful, it must satisfy two criteria:

1. It must be consistent with the mission objectives of the airplane.

2. It must not conflict with another cost function that has already been selected.

An example of a cost function which is not consistent with a mission objective would be:

Minimizing wave drag in an airplane that spends only a small amount of time in supersonic flight conditions.

Examples of cost functions which conflict with each other are:

A) Optimize lift-to-drag ratio, L/D with wing aspect ratio, A, as a variable, while also minimizing wing structural weight with A as a variable.

B) Minimizing structural weight with properties of structural materials as a variable, while also minimizing manufacturing cost with only materials weight as a variable. It may be, that the cost of advanced materials 'outweighs' their weight advantage!

8.3 RELATION BETWEEN COST FUNCTIONS AND AIRPLANE DESIGN VARIABLES

In this section, some of the example cost functions of Section 8.2 will be expressed in terms of appropriate airplane design variables. Finding such expressions for other example cost functions is left as an exercise for the reader. In addition, a discussion is presented on how some of these design variables themselves depend on or relate to each other. The material is organized as follows:

8.3.1 Performance Cost Functions
8.3.2 Weight Cost Functions
8.3.3 Economic Cost Functions

8.3.1 Performance Cost Functions

The following two performance cost functions of subsection 8.2.1 will be discussed:

* Cruise speed, V_{cr}
* Cruise Range, R_{cr}

8.3.1.1 Cruise speed as a cost function

It will be assumed that the classical parabolic drag polar format is applicable:

$$C_D = C_{D_o} + (C_L)^2/(\pi Ae) \tag{8.7}$$

Below the absolute ceiling, each airplane can fly level at two speeds: a low speed and a high speed. The high speed is referred to as the cruise speed. All this applies at cruise thrust settings only! It may be seen from Equation (3.60) in Part I that airplane cruise speed, V_{cr}, can be written as follows:

$$V_{cr} = \tag{8.8}$$

$$= \left[(W/S)_{cr}\frac{(T/W)_{cr} + \{\{(T/W)_{cr}\}^2 - (4C_{D_o})/(\pi Ae)\}^{1/2}}{(\rho C_{D_o})}\right]^{1/2}$$

where: $(W/S)_{cr}$ is the wing loading for the cruise flight condition being considered

$(T/W)_{cr}$ is the thrust-to-weight ratio for the cruise flight condition being considered

C_{D_o} is the zero lift drag coefficient

A is the wing aspect ratio

e is Oswald's efficiency factor

If these five parameters are viewed as independent design variables (which, as will be shown, they are <u>NOT</u>), and cruise speed is seen as a cost function which should be maximized, four conclusions can be drawn. V_{cr}:

1) increases with increasing wing loading,
2) increases with increasing thrust-to-weight ratio,
3) increases with decreasing zero lift drag coefficient.
4) increases with increasing product Ae

In other words, a maximum cruise speed value does not exist. There are several flaws in this reasoning:

<u>Flaw 1:</u> Wing loading cannot be increased independently.

NOTES: 1. The cruise wing loading differs from the takeoff wing loading because fuel has been used between takeoff and cruise.

2. The takeoff wing loading is constrained by a number of performance constraints as was shown in Chapter 3 of Part I.

<u>Therefore:</u> The cruise wing loading, $(W/S)_{cr}$, cannot be varied independently.

<u>Flaw 2:</u> Thrust-to-weight ratio cannot be increased independently.

NOTES: 1. The cruise thrust-to-weight ratio differs from takeoff thrust-to-weight ratio because of differences in thrust (throttle) setting and also because of altitude effects on engine output.

2. The takeoff thrust-to-weight ratio is constrained on its low side by performance considerations as shown in Chapter 3 of Part I.

3. Selecting a high thrust-to-weight ratio to attain a high cruise speed is probably self defeating: engine weight and cost will increase, as will engine fuel specifics.

<u>Thus:</u> the cruise thrust-to-weight ratio, $(T/W)_{cr}$, cannot be varied independently.

Flaw 3: The zero lift drag coefficient cannot be independently decreased. See Chapter 4 in Part VI for methods to compute C_{D_o} for airplanes.

NOTES: 1. At low subsonic speeds, C_{D_o} depends on:

a) airplane wetted area, S_{wet}. Wetted area in turn depends on the overall layout of the configuration and on the wing area: see Chapter 12 in Part II.

The reader is reminded of the fact that wetted area tends to correlate in a logarithmic manner with airplane takeoff weight for airplanes with similar mission orientation. See Part I, Figures 3.22.

b) skin friction coefficient, c_f. This in turn depends on Reynolds Number, Mach Number and on the smoothness of the exterior surfaces.

c) airplane interference drag and separation drag. These factors depend on the detailed layout of the external surfaces.

2. At high subsonic speeds, C_{D_o} depends on all previous factors AND on the dragrise behavior of an airplane. The dragrise depends on such detail design variables as: thickness ratio, sweep angle and aspect ratio of wing and empennage as well as on the overall progression of crosssectional areas: area ruling!

3. At supersonic speeds, C_{D_o} depends on all previous factors AND on lift induced wave drag.

Therefore: the zero-lift drag coefficient, C_{D_o}, can not be varied independently.

Flaw 4: The product Ae cannot be increased independently.

NOTES: 1. Increasing wing aspect ratio has the following consequences:

a) wing weight increases (This is seen from the wing weight equations in Ch.5 of Part V).

b) aeroelastic effects may become unacceptable. Examples of aeroelastic effects are:

* aileron reversal
* flutter
* aeroelastically induced shift of the wing aerodynamic center

Aeroelastic effects depend on:

* structural arrangement
* properties of structural materials
* aerodynamic load distribution
* mass distribution
* dynamic pressure of the flight condition

Reference 33 is an excellent introduction to the theory of aeroelasticity.

c) wing span may become too large because of operational considerations.

2. Oswald's efficiency factor, e, depends on the following variables:

* wing aspect ratio
* wing sweep angle
* airfoil leading edge radius
* wing lift curve slope (the latter also is a function of aeroelastic effects)

Chapter 4 of Part VI contains a method for estimating 'e'.

<u>Therefore:</u> the product Ae cannot be varied independently.

The cruise speed of the airplane therefore depends not on four variables (which are not independent), but on many more. Most of these additional design variables or design factors are not easily related to each other, certainly not during the early design phases.

8.3.1.2 Cruise range as a cost function

It is shown in Chapter 5 of Part VII that the cruise range depends on how the cruising flight is carried out: cruise at constant altitude, cruise at constant Mach number or some combination of these. Eqns (5.33) through (5.36) in Part VII provide simplified analytical formulas for cruise range under either scenario. Only the case of constant speed cruise of a jet airplane will be taken up here. From Equation (5.36) in Part VII:

$$R_{cr} = (U_1/c_j)_{cr}(L/D)_{cr}\ln(W_{initial}/W_{end}) \qquad (8.9)$$

where: U_1 is the true airspeed in kts

c_j is the engine specific fuel consumption in (lbs/hr)/lbs

L/D is the lift-to-drag ratio

NOTE: all of these quantities are for the cruise flight condition

$W_{initial}$ is the airplane weight at the start of cruise in lbs

W_{end} is the airplane weight at the end of cruise in lbs

Observe that:

$$U_1 = M_1 a \qquad (8.10)$$

where: M_1 is the steady state (cruise) Mach number

a is the speed of sound at the cruise altitude in kts

Substituting Eqn.(8.10) into Eqn.(8.9) yields:

$$R_{cr} = (a/c_j)_{cr}(M(L/D))_{cr}\ln(W_{initial}/W_{end}) \qquad (8.11)$$

It is clear from this equation that for:

* a given cruise altitude (which defines a)
* a given engine cruise thrust setting (which, with the given altitude and known engine inlet conditions, defines c_j)

* a given weight ratio, $W_{initial}/W_{end}$

the cruise range is maximized for maximum {M(L/D)}.

Figure 8.6 illustrates how L/D of a typical transport jet varies with Mach number. Note that {M(L/D)} becomes maximum at a Mach number different from that at which (L/D) becomes a maximum. All of this must be tempered by the fact that the airplane also must be flying at a lift coefficient which is consistent with the L/D at $\{M(L/D)\}_{max}$. From Table 3.1 in Part VI:

$$C_{L_{cr}} = W/\bar{q}S = f\{C_{D_o}(\pi Ae)\}^{1/2} \qquad (8.12)$$

where: f is a factor, such that $0<f<1.0$, which accounts for the fact that $C_{L_{cr}}$ is not the same as the value C_L takes on at maximum L/D.

For a discussion of the design variables which are determinant for C_{D_o}, A and e, the reader should read sub-sub-section 8.3.1.1. The reader should also appreciate the fact that the weight ratio, $W_{initial}/W_{end}$, in Eqn.(8.11) is itself not an independent variable. Values for this ratio, typical for transport jets, are shown in Table 8.3. The actually available weight ratio depends on the overall configuration of the design, in particular on the general structural arrangement and on materials used in the structure.

TABLE 8.3 TYPICAL CRUISE WEIGHT RATIOS, $W_{initial}/W_{end}$ FOR JET TRANSPORTS

Transport Type	W_{TO}(lbs)	W_F(max, lbs)	$W_{initial}/W_{end}$
Boeing 747-400	800,000	387,000	1.70
Boeing 767-200ER	300,000	113,000	1.47
Boeing 757	250,000	75,000	1.34
McD.D. MD81A	140,000	39,000	1.31
McD.D. DC-10-30	572,000	246,000	1.58
Airbus A320-100	150,000	28,000	1.19
Airbus A320-200	162,000	42,000	1.28

Note: data assume that 15 percent of the fuel is used for reserves and for conditions other than cruise.

Figure 8.6 Effect of Mach Number on L/D and on M(L/D)

8.3.2 Weight Cost Functions

The following two weight cost functions of sub-section 8.2.2 will be discussed:

* Empty weight
* Useful load

8.3.2.1 Empty weight as a cost function

From Equation (8.6):

$$W_E = W_{struct} + W_{pwr} + W_{feq} \tag{8.13}$$

Minimizing any one of these empty weight contributions is clearly desirable. A problem with this statement is that in many instances attempts to reduce weight can be costly (in terms of dollars!). Thus, a balance must be struck between weight decrease and cost increase.

The Class II weight estimation formulas of Chapters 5, 6 and 7 of Part V can be used to relate airplane empty weight to fundamental design variables. The Class II weight equations are derived from a data base which consists primarily of airplane structures made from aluminum alloy types of materials. When more advanced materials are planned, downward adjustments have to be made.

There are no simple methods to account for the effect of advanced materials and/or manufacturing processes on dollar cost. To do justice to the effect of dollar cost, much more detailed knowledge about the design of the airplane must be available than is normally the case in advanced design. An interesting approach to the problem of dollar-cost-weight trades in airplane design is found in Reference 34.

The reader should observe that many of the component weight equations in Chapter 5 of Part V require the limit load factor as an input. The limit load factor itself is not an independent variable! The V-n diagrams of Ch.4 of Part V show the design load factor to be a function of:

* the takeoff weight
* the flight envelope
* the gust response behavior

It is clear from this discussion that the relationship between empty weight and design variables is a very complicated one.

8.3.2.2 Useful load as a cost function

The useful load of an airplane is defined in Eqn.(8.5) as:

$$W_{US} = W_{PL} + W_F \tag{8.14}$$

where: W_{PL} is the payload weight

W_F is the fuel weight

Payload weight and payload type are normally defined in the mission specification of an airplane. It turns out that payload weight and payload type both have a significant effect on airplane empty weight. Examples which illustrate this effect are:

1) In a passenger jet transport, the number of passengers carried, the class of service offered and the requirements for on-board meals and other services have a major impact on the weight of cabin interior amenities.

2) In a cargo jet transport, the type of cargo has a major effect on the structural design and layout of the cargo hold:

* if containers are to be carried, roller-rails and tie-down provisions are required
* if concentrated loads are to be carried (any type of roadable or off-highway type vehicle), major floor reinforcements are required
* if front, rear and/or side loading of the payload is/are required, major structural provisions for these features will be required
* if weapons have to be carried and launched, major provisions in the structure to take the various weapons loads will be required

The reader is referred to Parts III and IV for more detailed discussions of these effects.

The point to be made here is that payload itself can be a complex function of design variables involved in the empty weight of an airplane, mostly in the structural and fixed equipment part of this weight.

8.3.3 Economic Cost Functions

The following three economic cost functions of subsection 8.2.3 will be discussed:

* Manufacturing cost (C_{MAN}
* Direct operating cost (DOC)
* Return on investment (ROI)

8.3.3.1 Manufacturing cost as a cost function

A method for determining airplane manufacturing cost was presented in Chapter 4. The method, as reflected by Equations (4.4), (4.5), (4.7), (4.17) and (4.18), shows that airplane manufacturing cost depends on the following design parameters:

1) Takeoff weight, W_{TO} (through the AMPR weight of Equation (3.4) or (3.5))
2) Maximum design speed, V_{max}, as defined on p.26
3) Number of airplanes produced to manufacturing standard, N_m
4) Labor rates for engineering, manufacturing and tooling, R_{e_m}, R_{m_m} and R_{t_m}
5) Number of passengers, N_{pax} (only for commercial airplanes
6) Manufacturing rate, N_{r_m}
7) Various judgement factors, F_{diff}, F_{cad}, F_{int}, F_{mat}, F_{ftoh} and F_{fin_m}
8) Cost escalation factor, CEF
9) Operating cost per hour, $C_{ops/hr}$, and number of production flight test hours, t_{pft}
10) Cost of engines and avionics

Design variables 1) and 2) are the most important ones. It turns out that the engine cost is related to the takeoff weight.

8.3.3.2 Direct operating cost (DOC) as a cost function

A method for determining airplane direct operating cost is presented in Chapters 5 and 6. The DOC of an airplane depends (according to Equations (5.19) and (6.1) and their components) on the following design variables:

1) Takeoff weight, W_{TO}, because many of the DOC component costs are expressed in terms of the takeoff weight

2) Cockpit crew requirements: this is determined by the mission of the airplane

3) Fuel used: this depends on lift-to-drag ratios and on engine efficiencies

4) Accessibility and ease of maintenance

For a discussion of the relative significance of DOC component contributions, see Chapters 5 and 6.

It turns out that takeoff weight, aerodynamic and fuel efficiency are normally the 'driving' factors behind the DOC of commercial and military airplanes.

8.3.3.3 Return-on-investment (ROI) as a cost function

The following definition of return-on-investment (ROI) will be used:

$$ROI = \tag{8.15}$$

$$= \left\{\frac{(REV - DOC - IOC)V_{bl}}{(AEP)(1 - tx_{inv})}\right\}(1 - tx_{rev})U_{ann_{bl}}(100)$$

where: REV is the revenue in USD per nm

DOC is the direct operating cost in USD per nm

IOC is the indirect operating cost in USD per nm

V_{bl} is the block speed in nm/hr

AEP is the airplane estimated price in USD (See page 62)

tx_{inv} is the investment tax credit rate extended by the government to the operator. For preliminary design purposes it is suggested to use: $tx_{inv} = 0.10$ (ten percent).

tx_{rev} is the 'income' or 'revenue' tax rate paid to the government. For preliminary design studies, use: $tx_{rev} = 0.20$.

<u>Note:</u> tax rates are determined by the congress and therefore are a variable over which the designer has little control!

$U_{ann_{bl}}$ is the annual airplane utilization in hours

Note that ROI as defined in Eqn.(8.15) comes out as a percentage.

It is obviously desirable to maximize ROI. However, there are a number of problems:

* The ticket price charged per nautical mile, REV, must be competitive or no passengers will be willing to pay the price

* The ticket price will influence market demand to some extent: the demand for passenger miles is elastic

* DOC and IOC should be kept as low as possible. Methods for computing DOC and IOC are given in Chapters 5 and 6 of this volume. Note the very large number of airplane design and operating variables which enter the picture!

* Annual utilization should be kept as high as possible. There are several limitations to this:

  a) if annual utilization goes above some number, maintenance and repairs become a problem

  b) the route structure of the airline may not make it feasible to have high annual utilization

8.4 DESIGN CONSTRAINTS IN AIRPLANE DESIGN

Aeronautics is one of the most heavily regulated activities of mankind. The reason for this is: SAFETY. To ensure minimum standards of safety in airplane design and in airplane operation, governments adopt and enforce minimum standards of airworthiness and operating safety.

These standards have been referred to as the 'regulations' in this series of books. Many aspects of these regulations were covered in Parts I through VII. Copies of those regulations which directly impact airplane performance, stability and control (and thereby the configuration design) are included as appendices to Part VII.

From a design optimization viewpoint, these regulations should be interpreted as: design constraints. Design constraints limit the numerical ranges of design parameters which are available to the designer.

In addition to regulatory design constraints, there are many other examples of design constraints. Airplane design constraints are generically classified as follows:

1. Design constraints related to airworthiness
2. Design constraints related to operational safety
3. Design constraints related to operational factors other than safety
4. Design constraints related to economics
5. Design constraints related to the infra-structure
6. Design constraints related to typical military requirements
7. Design constraints related to the environment

There is no implied priority in the listing of these design constraints. They can all play important roles in arriving at key design decisions. Examples of airplane design constraints and the role played in design decision making are provided in sub-sections 8.4.1 through 8.4.7.

8.4.1 Design Constraints Related to Airworthiness

Civilian and military airworthiness is ensured by specifying minimum standards of:

1. Airplane performance with and without all engines operating

2. Airplane handling qualities with and without all engines operating

3. Structural integrity in the presence of extreme atmospheric and/or other operational loads

4. Reliability and/or redundancy of flight crucial systems

Parts I through VII contain many examples of the impact of airworthiness regulations on airplane design. In the following, one example of each of these airworthiness constraints will be given.

8.4.1.1 Example of a performance constraint

It was shown in Chapter 3 of Part I that takeoff and landing distances are regulated by certain definitions of fieldlength: ground distance, air distance and obstacle height. Also, factors such as air density, temperature and humidity must be acounted for. If an airplane is to safely operate from a field with a given minimum fieldlength, then the result of these fieldlength constraints is to limit the available choices of wing loading and of thrust-to-weight ratio. This was shown in Chapter 3 of Part I in the form of a plot of takeoff thrust-to-weight ratio, $(T/W)_{TO}$, versus takeoff wing loading, $(W/S)_{TO}$.

Figure 8.7 is an example of this. The range of design parameter values which are denied by the fieldlength constraints is clearly indicated. A complicating factor is that the size of the 'denied' ranges is a function of the flap capability of the airplane, as expressed through the maximum lift coefficients: $C_{L_{max_{TO}}}$ and $C_{L_{max_L}}$.

Table 3.1 in Part I gives ranges of numerical magnitudes for these maximum lift coefficients for mechanical flaps. There are various forms of augmentation (blowing is one example) which can be used to attain significantly higher values of maximum lift coefficient. In any case, there are limits to attainable maximum lift coefficients. These limits can have a number of causes: aerodynamic, flap weight, flap system complexity and flap system cost are only a few of these.

It is possible to formulate these maximum lift coef-

Figure 8.7 Example of Fieldlength Constraints on Wing Loading and Thrust-to-Weight Ratio

Figure 8.8 Effect of Cruise Mach Number and Block Distance on Block Time

ficient limits in a suitable mathematical format. This is most easily done for the case of mechanical flaps. The lift prediction methods of Part VI are examples of how this can be done. Considerations involving calendar time and money restrictions make it difficult to arrive at similar formulations for items such as: flap weight, flap system complexity and flap system dollar cost.

Here is an example of a case where design intuition and design experience will enter the picture. An experienced designer will not hesitate to make a flap design decision based on intuition. However, this makes the design process no longer optimizable in the rational sense of the word.

8.4.1.2 Example of a handling quality constraint

Minimum requirements for safe handling qualities are discussed briefly in Part II and in Part VII. Analytical methods for determining handling quality parameters are presented in References 35 and 36. It can be shown that each requirement for handling qualities translates into some form of design constraint. One example will be discussed: the minimum control speed problem.

In multi-engine airplanes, with one engine inoperative, it is required that the pilot be able to maintain control down to the so-called minimum control speed, V_{mc}.

The airworthiness requirements (see Appendices A and B in Part VII) stipulate the following relationship between the minimum control speed, V_{mc}, and the stall speed, V_s:

$$V_{mc} < 1.2V_s \tag{8.16}$$

The stall speed itself is related to the wing loading and the maximum lift coefficient (clean, flaps up and flaps down!):

$$V_s = \{2(W/S)/(\rho C_{L_{max}})\}^{1/2} \tag{8.17}$$

The minimum control speed (in a takeoff flight condition) for a twin engine airplane can be approximated as follows:

$$V_{mc} = [\{(2T_{TO_e})(y_T)\}/\{(C_{n_{\delta_R}}\delta_{R_{max}})(Sb\rho)\}]^{1/2} \tag{8.18}$$

where: T_{TO_e} is the take-off thrust of the most critical engine in lbs

y_T is the lateral moment arm of the most critical engine in ft

$C_{n_{\delta_R}}$ is the rudder control power derivative in rad^{-1}

$\delta_{r_{max}}$ is the maximum allowable rudder deflection in rad

S is the wing area in ft^2

b is the wing span in ft

ρ is the atmospheric density in $slugs/ft^3$

Each design parameter in Equation (8.18) itself may depend on a number of other design parameters. For example, the control power derivative, $C_{n_{\delta_R}}$, is a function of:

* vertical tail size
* vertical tail geometry (aspect ratio, sweep angle, taper ratio and airfoil)
* vertical tail location on the configuration
* rudder size, shape and location

The maximum allowable rudder deflection, $\delta_{R_{max}}$, is a function of:

* Vertical tail stall behavior and Reynolds Number
* Rudder geometry
* Vertical tail location on the configuration

When all these design parameters have been decided upon, the V_{MC} constraint defines a minimum allowable vertical tail size. This in turn has an impact on overall airplane drag, weight and center of gravity location. The latter will also impact the stability and control of the airplane!

8.4.1.3 Examples of structural integrity constraints

Examples of structural design constraints are all those factors which impact on the airloads and other loads which act on an airplane as it operates in the air or on the ground. In the preliminary design phase the structural integrity requirements impact the design decision making process through factors such as:

* specified limit load factor

  This has an impact on structural weight and thus on overall weight, center of gravity locations and on cost.

* specified design touchdown velocity

  This has an impact on the design of the landing gear which in turn impacts airplane takeoff weight and the volumetric requirement of the gear upon retraction. The latter can also have an economic impact if the retracted landing gear volume limits payload or fuel carrying capability.

8.4.1.4 Examples of system reliability and redundancy constraints

Whenever the failure of an airplane system is considered to have a major impact on safety, it is called a flight crucial system. The hydraulic system is a flight crucial system if it is the only source of control in flight. That is the case in many transports today.

Whenever a system is defined as flight crucial it must satisfy certain standards of reliability and redundancy. For more information on the subject of safety and redundancy, see Chapters 6 and 13 in Part IV and also Appendix C in Part VII.

Flight crucial systems are 'allowed' to fail in a catastrophic manner at the rate of once per 10^9 flight hours (or sometimes flights). One task of the designer is to estimate the reliability of each such system, in terms of catastrophic failure rate per X flight hours. Once this estimate is available it is easy to compute how many independent systems are required to satisfy the 10^9 requirement. It is not difficult to see that the outcome of these system design considerations can have an impact on airplane weight, airplane layout (signal and power

path redundancy!!), airplane complexity, airplane maintenance requirements and airplane cost.

Casting these constraints in mathematical formats suitable for use in optimization is a formidable task.

8.4.2 Design Constraints Related to Operational Safety

Examples of this type of design constraint are:

* Airplane ceiling

  Each airplane has an altitude-weight combination beyond which it is unsafe to operate. The reason for this limit is the proximity of stall speed, buffet limit and maximum allowable Mach number. There are also cases where airplane handling qualities are compromised beyond certain combinations of airplane weight and altitude.

* Airplane operating speed and/or Mach number

  At each altitude, airplanes are limited in minimum and maximum allowable speed for a variety of reasons having to do with engine or airframe limitations or with handling quality limitations.

* Fuel reserves

  When airplanes are flown on a given route, there exist regulations (FAR 135) which specify that flight to alternate airports must be possible in case of weather limitations at the airport of destination. This leads to requirements for fuel reserves. These fuel reserves increase the weight of the airplane. They also require that enough fuel volume is available on board to store it.

* Emergency and evacuation constraints

  These constraints require certain airplanes to be equipped with:

  a) emergency exits of prescribed size and location
  b) evacuation slides
  c) life rafts and floatation equipment
  d) fire safety equipment

  All these requirements translate into weight and volume needs which should be viewed as constraints on the design decision-making process.

These constraints must be translated into functional relations between various airplane design variables. This cannot (in general) be done until detailed data on the aerodynamic and the structural design are available.

8.4.3 Design Constraints Related to Operational Factors Other than Safety

Examples of this type of design constraint are:

* Servicing requirements: commercial

  A jet transport, when arriving at the gate, needs to be serviced by a wide variety of servicing vehicles. Critical here is the so-called 'gate turn-around time' which must be kept to a minimum. Airplanes parked at the gate do not earn money! A discussion of this problem is given in Part III, Chapter 3 (page 82).

  This design constraint implies that there must be sufficient access to various systems of the airplane to make simultaneous servicing possible. This in turn puts constraints on the location of equipment in the airplane and on the size and location of access doors and access covers. These considerations tend to have a significant impact on structural design (strength and fatigue life) and therefore on weight and cost.

* Servicing requirements: military

  A military airplane returning from a combat mission needs te be turned around and readied for its next mission in as short a time as possible.

  This design constraint is similar to that of the jet transport at the gate. However, in the case of military airplanes these functions may have to be carried out by personnel wearing polar and/or chemical warfare clothing. This makes the design of access doors and covers very critical: they tend to be large!

* Landing gear constraints

  The reader is asked to review the landing gear design and disposition criteria of Ch.9 in Part II and in Ch.2 of Part IV. Most of these gear design criteria should be viewed as design constraints.

No attempt has been made here to derive functional relations between these design constraints and the appropriate airplane design variables.

8.4.4 Design Constraints Related to Economics

Examples of design constraints related to economics are:

* Revenue
* Program Cost
* Acquisition cost
* Travel time

A brief discussion of each follows.

* Revenue as a constraint

In certain markets the customer may not be willing to pay more than a certain amount of USD per nm. It is essential that airplane designers understand such limits and incorporate them into their design trade studies to determine the economic viability of a proposed design. Such limits on 'extractable' revenue should be viewed as design constraints.

* Acquisition cost as a constraint

The acquisition cost of an airplane translates into the cost of ownership. There are limitations on the allowable cost of ownership for commercial as well as for military airplanes. Again, such limitations should be viewed as design constraints.

* Program cost as a constraint

In military programs in particular the total allowable program cost should be seen as a design constraint.

These three cost constraints all bear a relation to the 'design-to-cost' problem which is discussed briefly in Section 8.8.

* Travel time as a constraint

Figure 8.8 shows how trip time relates to block-speed for various triplengths (block distances). It is seen that for transpacific distances there is not much payoff, in terms of time gained, for flying beyond Mach 4.0. This design constraint is caused by the 'smallness' of the earth.

8.4.5 Design Constraints Related to the Infra-structure

Examples of design constraints which are related to the infra-structure are:

* Gate space at airports of intended operation

  This may limit the designer to a given wing span. One example of this is the Boeing 747-400. Winglets were added to this airplane largely to avoid greater wing span. In future designs, folding wing tips may have to be considered as well!

* Hangar size

  Airplanes may have to fit into existing hangars. Construction of new hangars is expensive. Some operators may not be willing to spend money to build a new hangar just because the airframer decided to build an airplane which does not 'fit'. This constraint can therefore dictate the outside dimensions of the airplane regardless of any aerodynamic or other considerations!

  Such size constraints are common in the design of carrier based airplanes for the navy: deck spotting, elevator spotting and hangar deck spotting considerations all impose size constraints on the airplane designer.

* Width of existing taxiways

  Construction of airport taxiways involves major expenses. If an airplane manufacturer designs an airplane with a (landing gear) configuration which is not compatible with existing taxiways he may find no buyers for that airplane!

8.4.6 Design Constraints Related to Typical Military Requirements

Examples of such design constraints are:

* Gun integration constraints

  Guns impose a number of serious constraints on the design of an airplane. For a discussion of these, see Chapter 3, Part IV.

* Missile integration constraints

  This may dictate internal and/or external space requirements. It may also dictate a range of design parameters related to aerodynamics, weights and flight controls. For a discussion of weapon integration, see Chapter 3, Part IV.

* Stealth requirements

  Requirements for stealth may dictate the external shape of the airplane and/or the structural arrangement, quite apart from any conventional aerodynamic and/or structural design considerations.

  Prominent examples of such airplanes are the B-2 and the F-117A.

8.4.7 Design Constraints Related to the Environment

Examples of this type of design constraint are:

* Noise

  This design constraint may dictate the size, location and type of powerplant installation. In turn, this has a significant impact on weight, complexity and fuel consumption.

* Fuel dumping

  If a requirement would be formulated against fuel dumping, airplane landing gears would have to be designed for considerably greater weight at touchdown. This would impact airplane weight and landing gear retraction volume requirements.

* Atmospheric pollution

  Requirements to minimize 'visible' atmospheric pollution have already resulted in a redesign of burner systems in jet engines.

  Depending on the outcome of on-going research of potential ozone destruction by high flying supersonic transports, their burner systems may have to be tailored specifically to the prevention of releasing certain combustion products. The effect of these considerations on engine performance is still subject to question.

8.5 RULES FOR EVALUATING COST FUNCTIONS

It will be assumed that at this point the following steps have been taken:

1. One or more cost functions have been selected for optimization. These cost functions must be consistent with the required mission performance of the airplane.

2. The cost functions selected under 1. have been expressed functionally in terms of a range of airplane design variables.

3. Airplane design constraints have been cast in a format which relates them clearly to airplane design variables.

Next, a decision has to be made which of the cost functions are to be maximized or minimized while at the same time observing specific constraints.

NOTE WELL:

A) It is just as important to keep in mind which potential cost functions and which potential design constraints have NOT been considered. Doing this avoids the pitfall of believing that the results of optimization procedures (constrained or not) are somehow absolute.

B) Probably the most difficult aspect of the design optimization process is to decide which cost functions and which constraints should be used.

8.6 METHODS FOR DESIGN OPTIMIZATION

The purpose of this section is to acqaint the reader with reference material in the area of airplane design optimization. A detailed treatment of methods for design optimization is considered beyond the scope of this text.

There exists an impressive amount of literature on the subject of airplane design optimization. References 37 and 38 are recommended as excellent introductions to this subject. Both references contain many citations of other literature pertinent to design optimization.

Optimization methods can be broadly classified as:

1. Optimization within a given discipline

2. Multidisciplinary optimization

In each case, the optimization methods may or may not consider effects of design constraints: constrained or unconstrained optimization.

Most of the early design optimization methods dealt with optimization problems related to a particular design discipline, usually without accounting for constraints.

Examples are methods for optimizing airplane performance and/or aerodynamic parameters. References 39, 40 and 41 fit this category. Typical of the design conclusions drawn from these methods is Kuchemann's conclusion in Ref.39, that the best lift-to-drag ratio which is achievable in supersonic cruise is:

$$(L/D)_{max} = 4(M + 3)/M \qquad (8.19)$$

A problem with optimization within any discipline is that it can result in unrealistic design solutions. For example: induced drag is reduced by allowing the wing aspect ratio to assume very large values. This runs into problems with wing structural weight. Therefore, it is necessary to account for the disciplines of structural and aerodynamic analysis together whenever one impacts the other.

A problem with design optimization without proper regard of constraints is that it also can lead to unrealistic design solutions. Consider the following example:

A supersonic fighter has a requirement for a long subsonic mission leg, followed by a short supersonic dash requirement, followed again by a long subsonic leg to home base.

It is possible, by the application of area-ruling methods, to minimize the supersonic wave drag of this airplane. However, it may be that a result of minimizing wave drag is an increase in wetted area over that needed for efficient subsonic flight. If the fuel used in the short supersonic dash is significantly less than that used during the long subsonic legs, it may be that wave drag minimization makes no sense.

In this case, wave drag should have been minimized subject to the constraint that total mission fuel consumption (subsonic + supersonic) is kept to a minimum.

The subject of airplane design optimization in the presence of design constraints is extensively discussed in References 37 and 38.

Recently, research in airplane design optimization has moved in the direction of multidisciplinary design optimization subject to a multitude of constraints. To become familiar with recent trends in design optimization methods References 42 through 46 should be consulted.

The next section presents some examples of the end product of typical design optimization studies.

8.7 EXAMPLES OF DESIGN OPTIMIZATION STUDIES

The purpose of this section is to present a brief discussion of results of potential design optimization studies. For more details on how to perform such studies references 42 - 46 should be consulted.

Example 1: The economic value of blockspeed

For a given airplane, operated over a given stage length (block distance) and at a given altitude, the question can be asked: at what speed is the operator's profit margin the highest possible?

In this case, a solution can be found by plotting the various operating cost components as a function of blockspeed. Figure 8.9 shows an unscaled example of such a plot. Note the behavior of the fuel cost contribution:

* at low block speeds, fuel usage increases because of the increase in induced drag

* at high block speeds, fuel usage increases because zero-lift drag and compressibility drag increases

Figure 8.9 shows various 'optima' which correspond to different block speeds:

* block speed for minimum fuel cost
* block speed for minimum fuel used per nm per hr
* block speed for minimum DOC
* block speed for maximum profit margin

It is of interest to observe that the block speed for the lowest DOC is lower than the blockspeed for the greatest profit margin!

Figure 8.9 The Economic Consequence of Blockspeed

Figure 8.10 Effect of Fieldlength Constraints on Direct Operating Cost (DOC)

Note that in Figure 8.9 various factors were kept constant. The validity of the study results are subject to question if these factors are allowed to vary!

Example 2: Effect of fieldlength constraints on DOC

The reader is reminded of the various fieldlength constraints placed on commercial aircraft. This topic was extensively covered in Chapter 3 of Part I.

For a given airplane, operated over a given range, the takeoff weight allowed is limited by fieldlength considerations. At any given takeoff weight, the useful load (sum of payload weight and fuel weight) is fixed: for long ranges payload must be off-loaded!

Figure 8.10 shows an unscaled example of the effect of fieldlength on DOC, with range as a second parameter. The 'design field length' is defined as that takeoff distance required with maximum passengers and maximum takeoff weight under critical atmospheric conditions and one engine failing.

Point A reflects an airplane which has been 'optimized' for minimum DOC in cruise: it does not meet the fieldlength constraint. Point B represents an airplane which meets the fieldlength constraint exactly, but at slightly greater DOC. Point C reflects an airplane which meets the fieldlength requirement at any range, but at higher DOC yet.

The lesson from this study is that designing for extreme fieldlength requirements may result in serious compromises in DOC. It may be better to design to Point B AND to offer versions of the airplane with slightly different wings and/or high lift devices to operators with extreme fieldlength requirements.

The airplane design process as described in Parts I and II is a sequential process. If design optimization studies are conducted at each step along the way, it is argued in Reference 47 that the result is a sub-optimal design solution. An example of this is given next.

Example 3: Minimizing wing structural weight in a sequential manner

This example is based on Reference 46. Suppose wing structural weight needs to be minimized subject to some performance measure, PM. This performance measure, PM,

could be the payload for a given range. Constraints C_1 and C_2 (C_1 could be a fieldlength constraint and C_2 could be an engine-out, rate-of-climb constraint) are also imposed on the problem.

Figure 8.11 shows a possible result where the wing aspect ratio is used as an independent variable. The point P_1 is the constrained optimum for this case study. Suppose that, as a result of this type of study, the wing aspect ratio is frozen at a value A_1. Also suppose that at some later point in the design process a flutter study has been conducted which results in an added constraint, reflected by C_3 in Figure 8.12. As a result, the wing structural weight has to be increased to a value corresponding to point P_2. However, if the flutter study had been conducted earlier, point P_3 might have been selected as the desirable design point!

It is worthwhile to end with an observation first made in Reference 47. The reader is asked to refer to Figure 8.13. Observe that as the time into the design process increases, the knowledge about the design also increases. However, this increasing knowledge about the design implies that more and more design variables are 'frozen'. That in turn limits the potential design freedom. Therefore, the ability to act on the results of design optimization studies gradually disappears as time into the design process increases. Reference 47 refers to this state of affairs as the paradox of sequential design decision making. This 'peril' can be avoided by careful selection of design cost functions and judicious application of design optimization studies.

Figure 8.11 Optimization of Wing Weight with Aspect Ratio Subject to Two Constraints

Figure 8.12 Optimization of Wing Weight with Aspect Ratio Subject to an Additional (Later Discovered) Constraint

Figure 8.13 Illustration of the Paradox of Sequential Design

8.8 INTRODUCTION TO THE DESIGN-TO-COST PROBLEM

Budgets for airplane programs are always limited. Also, airplane operators cannot afford airplanes if they are too expensive. Interesting questions therefore are the following:

1. Assume that an airplane has been designed to some mission specification. How much is that airplane worth to the operator?

2. Given a certain net worth and given the need for a certain number of airplanes, can a manufacturer design, develop and manufacture such an airplane and make a profit?

The first question is referred to as the so-called 'net worth' problem. Sub-sections 8.8.1 and 8.8.2 give an approach to estimating airplane net worth.

The second question is referred to as the so-called 'design-to-cost' or the 'design-to-price' problem. An approach to this problem is given in sub-section 8.8.3.

8.8.1 The Commercial Airplane Net Worth Problem

Definition: The net worth of an airplane is the price an operator is willing to pay for an airplane because the potential for profit (Return-on-investment, or ROI) is demonstrable.

The net worth of an airplane can be determined via an operational market analysis. Such a market analysis should result in the following data:

* definition of routes, frequencies, passenger demand and/or cargo demand

* definition of sustainable revenue factors such as USD/paxnm and/or USD/tonnm

A preliminary estimate of the sustainable revenues can be obtained from the literature. Figure 8.14 is just one example. From such data it is possible to estimate the annual revenue which can be obtained per airplane. Assuming a desired return-on-investment percentage, the allowable airplane price (net worth) can be recovered from Equation (8.15) by solving it for AEP, the airplane estimated price.

Figure 8.14 Example of Airline Revenues

A problem with this procedure is that the direct operating cost, DOC, itself is a function of AEP, as shown in Chapter 5, Eqns (5.19), (5.40) and (5.41). Solving for AEP is therefore a complicated matter.

The IOC (indirect operating cost in Eqn.(8.15)) can be assumed to be some constant fraction of DOC as suggested in Section 5.3.

8.8.2 The Military Airplane Net Worth Problem

Estimating the net worth of a military airplane is a tricky proposition. For combat airplanes one approach is to estimate the worth of potential targets which have to be destroyed by the airplane in question. At least in theory, it would not make economic sense to develop an airplane which costs considerably more than the targets it is supposed to destroy. This line of thinking thus sets an upper bound on the allowable airplane cost. Reference 48 can be helpful in this regard.

For military training airplanes the problem is a lot simpler. It is not too difficult to determine the cost of training pilots. Knowing the number of pilots which have to be trained an estimate can be made of the allowable cost of the airplane: its net worth. Reference 49 provides more detailed insight into this problem.

8.8.3 The Design-to-Cost Problem

It will be assumed that as a result of the solution to the net worth problem in sub-sections 8.1.1 or 8.8.2, the following information is available:

- Airplane mission specification

* Airplane net worth, ANW (usually assumed to be the same as airplane estimated price, AEP)

* Number of airplanes required in a market, N_{market}

Assuming that the number of airplanes to be built to production standard, N_m, is equal to:

$$N_m = F_m(N_{market}) \tag{8.20}$$

where: F_m is a factor (> 1.0) which reflects the level of optimism about additional markets for that airplane.

The estimated revenue from this airplane program to the manufacturer is:

$$R_{man} = (N_m)(ANW) \tag{8.21}$$

If the airplane manufacturer is to make a profit on this airplane program, the amount R_{Man} must equal the sum of airplane acquisition cost, C_{ACQ} (Eqn.(4.2)) and airplane RDTE cost, C_{RDTE}:

$$R_{man} = C_{ACQ} + C_{RDTE} \tag{8.22}$$

With Eqns (4.2) and (4.19) this yields:

$$R_{man} = C_{MAN}(1 + F_{pro}) + C_{RDTE} \tag{8.23}$$

Thus, equating (8.21) and (8.23):

$$(N_m)(ANW) = C_{RDTE} + C_{MAN}(1 + F_{pro}) \tag{8.24}$$

By assuming that C_{RDTE} is a constant fraction of C_{MAN} it is now possible to determine the 'allowable' cost of development and manufacturing. Figure 8.15 shows a typical example of the breakdown of various costs in a commercial jet transport program. Similar data can be extracted from the literature or can be estimated with the methods of Chapters 3 - 7.

Knowing the 'allowable' cost of airplane development and manufacturing, the next problem is to decide if it is feasible to design an airplane which can be profitably built within these cost constraints. This problem can be solved iteratively by using the methods of Chapter 4.

Figure 8.15 Breakdown of Costs in a Jet Transport Program

8.9 DESIGN GUIDELINES FOR LOWERING COST

Even after applying the most thorough of preliminary design methods, including design optimization procedures, many aspects of the detail design of the airplane remain to be decided. Particularly at the system design and at the system design integration level, those detail design decisions can have a significant impact on airplane cost, especially the manufacturing and operating costs.

The purpose of this section is to present a series of design guidelines which, when followed, tend to reduce airplane cost. These guidelines should be carefully considered in any new design.

The 88 design guidelines as presented in Tables 8.4 are the direct result of problems encountered with airplanes designed without the benefit of these guidelines. The author acknowledges the fact that the majority of the 88 guidelines come from research done by Mr. B. Williams, Director of the Center for Reliability, Maintainability and Quality, Air Force Institute of Technology, WPAFB.

SIAI-MARCHETTI S.211 FUEL SYSTEM

KEY

1. FUSELAGE TANK
2. HOPPER
3. DROP TANK
4. WING TANK

Table 8.4 DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
SYSTEMS		
1. Minimize complexity and number of parts associated with all items which require frequent servicing and/or access.	x	x
2. Keep ALL plumbing away from heat sources.	x	x
3. Make most components accessible to people while they stand on the ground. Access step-ups, ladders and platforms are sources of trouble and accidents.	x	x
4. Design so that simultaneous refueling and rearming is possible with the engines running.		x
5. Design so that all required servicing vehicles can access the airplane simultaneously while parked at the gate.	x	
6. Design ALL flight crucial systems with redundant drive and with redundant signal path systems.	x	x
7. Design the installation of identical components for complete interchangeability: no installation should be unique.	x	x
8. Design wire bundles so they will 'fit' only one way, avoiding the possibility of miswiring.	x	x
9. If reversed installation of a component can cause malfunctioning, design the installation for a one way fit.	x	x
10. Design all access panels with CAPTIVE fasteners and with retaining devices.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
SYSTEMS (Cont'd)		
11. Design access to units which require servicing during turn-arounds so that personnel wearing winter clothing AND gloves can service them.	x	x
12. Design all systems which contain liquids (oil, hydraulics, fuel, glycol and water) so that these liquids can be easily poured in when necessary.	x	x
13. Design oil and hydraulic systems so that they can be completely replenished by ONE person in 5 minutes.	x	x
14. Put servicing instructions on placards in easy to read letters.	x	x
15. Size fluid reservoirs for one full day of operation.	x	x
16. Design avionics systems for fit in integrated, easily removable racks.	x	x
17. Locate avionics racks so that exposure to vibration is minimized.	x	x
18. Locate airplane rate and acceleration sensors so they cannot read false signals due to gun induced vibrations.		x
19. Standardize all micro-integrated circuits.	x	x
20. Design APU systems to provide for adequate cooling on the ground in a hot day environment.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
SOFTWARE		
21. Establish rigid control over software design AND over software modifications.	x	x
22. Design digital systems so that in-flight computing resources use only HALF of the total system capability (i.e. processing capacity, execution time and memory).	x	x
23. Design software to be compatible with ground systems and with test stations.	x	x
24. Design data management systems to be modular: one update input to update all systems.	x	x
AIRFRAME		
25. Design access doors and panels which require frequent opening with quick opening latches.	x	x
26. Design access doors and panels so that they are easy to close AND so that unlocked doors and panels are easily spotted.	x	x
27. Design door hinges so that if they break, they cannot tear into other primary structure(s).	x	x
28. Design the airframe so that NO structural components have to be removed in order to remove the engine.	x	x
29. Structural areas with a high probability of failure, fatigue or corrosion SHALL be visible and inspectable.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
AIRFRAME (Cont'd)		
30. Design airframe jack points so they do not interfere with gear operation, flap operation and other check-out procedures with the airplane on the jacks.	x	x
31. Provide adequate access to integral fuel tanks so that resealing is made possible.	x	x
32. Design for access to all crew station equipment without having to remove a seat.	x	x
33. Ejection seats must be removable without having to disassemble the canopy.		x
34. Install drag chutes in a pre-container to facilitate re-installation.		x
35. All seat components must be inspectable with the seat installed in the airplane.		x
36. Design the nosegear so that the airplane can be towed without having to disconnect steering power and so that the airplane can be towed within the design steering range.	x	x
37. Design landing gear door actuation system as simple as possible.	x	x
38. Wheel and brake assemblies should be separately removable.	x	x
39. Design landing gear so that jacking is NOT needed for scheduled maintenance and inspection events.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
FLIGHT CONTROLS		
40. Design control surfaces so that they can be replaced without re-rigging the flight control system.	x	x
41. Provide and design for rigging pins with the cockpit controls in the neutral position.	x	x
42. ALL flight control system components must be accessible for all types of maintenance.	x	x
43. Where possible, LIMIT the autopilot authority.	x	x
ENGINES AND INSTALLATION		
44. All engine driven accessories and their electronic controls should be airframe mounted.	x	x
45. Design engine mountings so that each engine can be removed AND replaced within 1.5 hours.	x	x
46. Some engines require access to the top of the engine. Provide easy access in such cases.	x	x
47. Any engine mounted accessories shall be replaceable with the engine installed.	x	x
48. Engine removal shall be possible without jacking or otherwise supporting the airframe.	x	x
49. Design the engine-to-airframe interfaces so they are all located in one area.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
ENGINES AND INSTALLATION (Cont'd)		
50. All engine performance sensors must be replaceable within 10 minutes.	x	x
51. Design engine throttle controls so that rigging is not required.	x	x
52. The engine oil system must be designed for operation at prolonged negative 'g's.		x
AUXILIARY POWER UNIT		
53. Ground checkout of the APU must be possible with one access panel.	x	x
54. Design airframe hardpoints to enable quick APU removal and replacement.	x	x
55. It must be possible to operate the APU via an external panel when the airplane is on the ground.	x	x
ELECTRONIC COUNTERMEASURES SYSTEM		
56. Failure of the automatic temperature control system shall NOT preclude manual operation of the system.		x
57. If ECS cooling ducts are located in inaccessible areas DON'T use duct material which deteriorates with age.		x
58. ECS lubrication systems shall be designed so that operation in extreme cold is possible.		x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
ELECTRICAL POWER SYSTEM		
59. Design for over current protection from any single failure.	x	x
60. Design wire bundles for easy removal for modifications.	x	x
61. Design to prevent wire chafing through holes and hinges in the case of long wire bundles.	x	x
62. Keep wire bundles away from heat sources.	x	x
63. Locate the power distribution panel in a non-hazardous area with the engine(s) running on the ground.	x	x
64. All flight station panels must be connected via pendant cables to allow for removal without disturbing ANY adjacent equipment.	x	x
65. Incorporate service loops in the case of sliding chassis or hinged doors.	x	x
66. All grounding receptacles must be in accessible areas AND must be replaceable.	x	x
67. Design the power generation and main battery system so that the main battery is fully charged after one mission.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
LIGHTING		
68. Incorporate lights in the wheelwells for maintenance.	x	x
69. Isolate lights from vibration.	x	x
70. All external lights must be rapidly replaceable.	x	x
71. Design strobe light location to avoid disoreintation of flight crew.	x	x
72. Minimize the use of retractable lights and/or use highly reliable retraction/extension mechanisms.	x	x
HYDRAULIC SYSTEMS		
73. Design hydraulic line and hose runs to prevent the inadvertent cross-connection of systems.	x	x
74. Identify the required fluid type CLEARLY at all fluid filler points.	x	x
75. Design all filters, shut-off valves, pressure switches and sub-components for separate removal.	x	x
76. Incorporate isolation valves in each sub-system to allow for maintenance, bleeding, etc.	x	x
77. Incorporate quick disconnect fittings for ground servicing and for connecting to a maintenance cart.	x	x
78. Incorporate a central hydraulic system control panel for maintenance personnel.	x	x

Table 8.4 (CONT'D) DESIGN GUIDELINES FOR LOW COST

Guideline	Applicability Commercial	Applicability Military
FUEL SYSTEM		
79. Fuel quantity transmitters should be mounted to accept high 'g' maneuvers.		x
80. Fuel quantity indicators should indicate remaining fuel in a proportional manner.	x	x
81. Design ALL fuel system components so they can be removed WITHOUT draining the fuel tanks.	x	x
82. Any motor operated shut-off valve shall have the motor removable while the valve is in the fuel system.	x	x
83. Design fuel system with common valves and common pumps.	x	x
84. Incorporate a single point refueling and control panel.	x	x
FAULT DETECTION SYSTEM		
85. All individual fault detection systems should report to a master system.	x	x
86. Failure status should be reported to pilot for manual selection of a backup system.	x	x
87. All in-flight failures must be data linked to the ground for maintenance preparations.	x	x
88. The crew station must have a 'mission capability' panel.	x	x

9. FACTORS IN AIRPLANE PROGRAM DECISION MAKING

The purpose of this chapter is to discuss several factors which must be considered before making an airplane 'program-go-ahead' decision. Much of the material included in this chapter is based on ideas which are developed in the series: 'Case Studies in Aircraft Design', published by the American Institute of Aeronautics and Astronautics (AIAA) as References 50 through 53. An excellent overview of the airplane program decision making process is presented by J.E.Steiner in Ref.54. Finally, the reader is urged to familiarize himself with Ref.25 which provides insight into military airplane programs.

The following factors can play a role in a decision to 'launch' or 'not launch' an airplane development and/or an airplane production program:

9.1 Factors Involving Commercial and/or Financial Feasibility
9.2 Factors Involving Technological Feasibility
9.3 Factors Involving Manufacturing Facilities
9.4 Factors Involving Political and/or Environmental Feasibility
9.5 Lessons Learned from Past Airplane Programs

9.1 FACTORS INVOLVING COMMERCIAL AND/OR FINANCIAL FEASIBILITY

Before embarking on a new airplane program it is necessary to investigate the following commercial feasibility factors:

9.1.1 Market Potential
9.1.2 Development Cost and Time
9.1.3 Availability of a Skilled Work Force
9.1.4 Required Production Investment
9.1.5 Potential for Return on Investment (ROI)

These factors are discussed next.

9.1.1 Market Potential

Market research is an essential component of the decision making process leading to an airplane program go-ahead. The following elements deserve attention:

9.1.1.1 Understand the customer and his organization
9.1.1.2 Identify and compare with the competition
9.1.1.3 Identify voids in the market
9.1.1.4 Forecast the size of the market
9.1.1.5 Forecast the net worth of the airplane
9.1.1.6 Decide on ability to produce at a cost below the airplane net worth
9.1.1.7 Identify and contact customer decision makers

9.1.1.1 Understand the customer and his organization

First, it is necessary to recognize the fundamental differences between commercial and military customers. Figure 9.1 shows the overall relationship between the manufacturer, the customer and the regulator for:

* Commercial Airplane Programs
* Military Airplane Programs

Second, it is essential that those involved in the airplane design decision making process understand the individual customer, his organization and his needs. In the early phases of the design it is of particular importance to know and understand those persons working for potential customers who are in charge of 'technical decision makers'. It should be noted that the technical decision makers are often not the financial decision makers and vice versa. Frequent contact between the customer and the manufacturer is essential, particularly during the early phases of the design.

9.1.1.2 Identify and compare with the competition

Identification of the competition is an essential ingredient in any competitive enterprise. It was also made part of the configuration design development procedures on page 11 of Part II.

The following reference material has been found very useful in identification of competitive designs:

* Jane's All the World Aircraft (British, Annually)
* Business and Commercial Aviation (U.S., Monthly)
* Aviation Week and Space Technology (U.S., Weekly)
* Interavia (Swiss, Monthly)
* Flight International (British, Weekly)
* Sales brochures from competitors

There are many ways to study comparative strengths and weaknesses of competing designs. Table 9.1 and Figures 9.2 and 9.3 show typical presentations from which

Figure 9.1a Relation Between Manufacturer, Customer and Regulators: Commercial

Figure 9.1b Relation Between Manufacturer, Customer and Regulators: Military

TABLE 9.1 COMPARISON OF SIMILAR AIRPLANE DESIGNS

Types:	SAAB 340	Boeing/DH Dash 8	Embraer Brasilia
Specifications			
Max. seating capacity	35	36	30
Power plants (2 each)	CT7-5A2	PW120	PW115
Shp per engine	1,735	1,800	1,600
Max seating capacity	5+35	3+40	3+30
Dimensions			
Length (ft)	64.9	73.0	65.6
Span (ft)	70.3	85.0	64.9
Height (ft)	22.5	24.6	20.8
Wing area (ft^2)	450	585	424.4
Wing aspect ratio	11.0	12.4	9.9
Weights			
Max. takeoff wht (lbs)	27,275	33,000	25,353
Std. empty wht (lbs)	17,415	21,590	15,586
Max. useful load (lbs)	9,860	11,410	9,767
Max. usable fuel (lbs)	5,690	10,160	5,862
(gallons)	843	1,451	875
Max landing wht (lbs)	26,500	32,400	24,802
Max. wing loading (psf)	60.6	56.4	59.8
Performance			
Takeoff fieldlength (ft) (SLS, ISA)	3,960	3,370	4,560
Max. climb rate (ft/min)			
AEO	1,850	1,660	2,250
OEI	500	450	630
Best climb rate speed, AEO (kts)	139	117	NA
Stall speed (kts)	82	72	86
Serv. ceiling AEO (ft)	25,000	25,000	31,000
Serv. ceiling OEI (ft)	16,200	13,400	18,500
Normal cruise speed (kts)	277	265	285
At altitude of (ft)	26,000	15,000	25,000
High speed cruise (kts)	281	265	300
At altitude of (ft)	15,000	15,000	20,000
Fuel flow for:			
Normal cruise (lbs/hr)	948	1,244	832
High sp. cruise (lbs/hr)	1,098	1,244	1,080
Turbulent air penetration speed (kts)	190	180	182
Max IFR range, ISA (nm)*	1,345	1,270	1,240
Range, 4,000 lbs payload	581	1,180	615

*Includes 100 nm diversion plus 45 minute hold

Figure 9.2 Comparison of Similar Airplane Designs

Figure 9.3 Comparison of Similar Airplane Designs

the nature of the competition can be discerned.

When comparing a new design to existing competitive airplanes, ALWAYS assume that the competititon will NOT sit still! Instead, try to imagine various scenarios for the competition to respond. Possible responses from the competition might be:

* Launching of a derivative design

  This could be very cost effective and may be hard to beat with a new design!

* Launching of a competitive new design

  Depending on the cost structure of the competition and on the technology level employed, this may or may not be a threat.

9.1.1.3 Identify voids in the market

Figure 9.3 shows a seat-range diagram for existing jet transports. Figure 9.4 shows that same information but with the airplane identifications left off. It is seen that there are several voids in the diagram. The following questions should be asked:

1.) Are there technical or economic reasons for the voids?

2.) If so, can these reasons be removed by clever and new design approaches?

Take for example the void labeled I in Figure 9.4. Void I would suggest an airplane capable of flying 125-175 passengers over a still air range of 4,000-5,000 nm. The question is: what might be the market for such an airplane? The answer could be: very long but very thin international routes.

* Long means: large distances between city pairs.

* Thin means: passenger demand is relatively small and does not justify the deployment of large wide body airliners.

Table 9.2 shows a list of such city pairs. What must be determined next is how many people might want to travel these routes and what prices might they be willing to pay for their tickets. Market research studies can be conducted to find the answers.

Figure 9.4 Identification of Market Voids

TABLE 9.2 CITY PAIRS WHICH FIT VOID I IN FIGURE 9.4

City Pairs		Great Circle Distance
Kansas City, MO, USA	- Amsterdam, Holland	4,700 nm
Sevastopol, USSR	- Trenton, N.J., USA	5,000 nm
Helena, Montana, USA	- Glasgow, UK	4,200 nm
Baton Rouge, LA, USA	- Rio de Janeiro, Brazil	4,900 nm
Norfolk, VA, USA	- Lima, Peru	3,500 nm
Hamburg, W.Germany	- Dar es Salaam, Tangan.	4,500 nm
Vienna, Austria	- Recife, Brazil	4,900 nm
Osaka, Japan	- Surabaja, Indonesia	3,500 nm
Seattle, WA, USA	- Amsterdam, Holland	4,800 nm
Montreal, Canada	- Abidjan, Ivory Coast	4,800 nm

Figure 9.5 Development Times for Four Transport Programs

9.1.1.4 Forecast the size of the market

To make a credible estimate of the size of the market for the new airplane, the information developed in 9.1.1.3 needs to be translated into flight frequencies required to serve example city pairs. Some growth rate is normally accounted for. Typical growth scenarios will postulate that 5 to 10 percent growth occurs annually.

This way the number of airplanes required to serve these new markets can be estimated. With that done, the market share must be estimated. A cautious approach is to assume that a new design might 'catch' 1/3 of the estimated total market.

9.1.1.5 Forecast the net worth of the airplane

The definition of net worth of an airplane is given in Chapter 8 (p.241): it is the price the customer may be willing to pay if there is reasonable assurance of an acceptable return-on-investment (ROI).

A method for determining net worth was also outlined in Chapter 8.

9.1.1.6 Decide on ability to produce at a cost below the airplane net worth

It makes no sense to initiate serious marketing of an airplane until it has been determined that it is possible to manufacture it at a cost well below the airplane net worth. The methods of Chapter 4 should be used to obtain an estimate of the manufacturing cost.

9.1.1.7 Identify and contact customer decision makers

With most of the marketing homework completed, the final step is to try to 'convince' the financial decision makers of one or more customers to commit to ordering the airplane. It should be noted that the financial decision makers are usually not the technical decision makers and vice versa. This tends to be so for commercial as well as for military airplanes.

9.1.2 Development Cost and Time

New airplane development programs can take a very long time. Figure 9.5 illustrates this for several jet transport programs. Also, as shown by the RDTE cost estimation examples of Chapters 3 and 7, the cost of a new airplane development program is extremely high. Management must be prepared to 'sink' a lot of money into the project for many years before any signs of financial returns occur. Figure 9.6 is an illustration of this for the Ourania transport program used in earlier examples.

9.1.3 Availability of a Skilled Work Force

It should not be taken for granted that a skilled work force will always be available to carry out a new airplane development and production program. Historically, aircraft manufacturing employment has gone through many peaks and valleys. Figure 9.7 is an example.

Following major layoffs of production workers, many of these workers will move elsewhere or change job types. Most of these workers may then be 'lost' from that area.

Training of new workers is very expensive but may have to be considered as part of the start-up costs of a new airplane program.

9.1.4 Required Production Investment

Examples of methods for predicting manufacturing costs were given in Chapters 4 and 7. The amount of money involved is usually beyond the capability of an airframer to raise without borrowing money from one or more financial institutions.

The following numbers are recalled:

For the Ourania (150 passenger, medium range jet transport) the estimated program manufacturing cost is:

USD 9,142,000,000 (rounded off from p.61.)

The 'up-front' investment in tooling and engineering cost associated with this manufacturing program is:

USD 552,000,000 (rounded off from pages 60 and 61.)

In addition to these investment needs, there are those for materials, various facilities, engines and avionics. It is true that some of these costs can be

Figure 9.6 Example Cash Sink in a Transport Program

Figure 9.7 Typical Employment Fluctuation in an Airplane Manufacturing Operation

spread out over the duration of the manufacturing program. However, a considerable part has to be invested 'up-front'. The interest over that 'up-front' investment must also be accounted for. If only the tooling and engineering are 'counted' as part of this 'up-front' investment and if the annual interest rate is, say 10 percent, this represents an annual interest payment of USD 55,000,000!!

The need for 'up-front' manufacturing investment money may be so great that risk-sharing arrangements have to be sought with other companies in the USA or abroad.

9.1.5 Potential for Return on Investment (ROI)

At some point well before first deliveries take place, it is customary in the 'airplane business' that the customer begins to make 'progress' payments toward his airplane(s). As a result, some money will start flowing back into the manufacturer's coffers. The difference between program expenditures and program receipts is called the program cash-flow. Figure 9.8 illustrates the cash-flow associated with a typical commercial transport program. Note that the cash-flow does not become positive until 10 years after initiation of the program! However, receipts start coming in after year 3 or 4.

The ultimate profitibility of the program (Return on Investment, or ROI) is determined by the behavior of the cash-flow with time. As it turns out, many airplane programs are viable over the long run ONLY if the manufacturer is willing to adjust his design to new market conditions. This normally results in many 'derivatives' of the original design. However, the design, development, testing and certification of these derivatives is not without cost. Figure 9.9 illustrates the effect of such derivative programs on total program cash-flow.

NOTE TO DESIGNERS: It is important to design the original airplane in such a way that derivatives are in fact possible. For example, if an airplane cannot be grown by lengthening the fuselage because of a landing gear or takeoff rotation limitation it should not have been designed that way in the first place!

Figure 9.8 Example of Cash Flow in a Commercial Jet Transport Program

Figure 9.9 Effect of Derivatives on the Cash Flow of a Commercial Jet Transport Program

9.2 FACTORS INVOLVING TECHNOLOGICAL FEASIBILITY

It makes little sense to proceed with new airplane development and/or manufacturing plans without first determining the technological feasibility of the program.

Technological feasibility (or lack thereof) has many facets. To name a few:

* Configuration selection
* Structures
* Flight controls
* Materials
* Aerodynamics
* Propulsion
* Systems
* Manufacturing

The preliminary design methods outlined in Parts I through VII are aimed at establishing the technological feasibility of a new design. Fundamental flaws in the design are supposed to be 'smoked out' as a result of the two preliminary design sequences: P.D. Sequence I and P.D. Sequence II described in Part II.

However, if major advances (innovations) over past airplane designs are to be achieved, it is often required of the preliminary designer to take a risk in one or more of the technological facets mentioned before. Taking a risk implies the extrapolation of currently available and VALIDATED technologies. Examples are:

* Configuration selection: proposing a new configuration which is radically different from existing design practice

* Aerodynamics: assuming that the lift-to-drag ratio in cruise is significantly above that of existing design practice

* Structures: assuming that a new material with much greater strength-to-weight ratio (compared to current design practice) will work out

* Propulsion: assuming that a new engine with much lower s.f.c. (specific fuel consumption) will materialize

* Flight Controls: assuming that a new approach to flight control system design will work and will be certifiable

* Systems: assuming that a new approach to the design of one or more airplane systems will work out

The emphasis in all this is on the word VALIDATED.

NOTE 1: There is nothing wrong with assuming that major advances in technology can be made AND to integrate these advances in a new design during the preliminary design phases.

NOTE 2: There is much wrong with NOT doing the research, development and test work needed to VALIDATE any of these 'extrapolated' technologies.

NOTE 3: It is foolhardy to decide to proceed with the integration of UNVALIDATED technology into the detail design development phase of an airplane.

There are many historical examples of airplane programs which were launched without adequate VALIDATION of one or more aspects of technology used in the design. The consequences of launching an airplane program without technology validation can range from:

* mere embarrassment to a fatal crash, or from:

* minor cost overruns to bankruptcy.

Any unvalidated technology when used in a new airplane design can therefore become a 'profit suppressor' or even a 'show stopper'. It is essential that designers acquaint themselves with all technology factors which may become either a profit suppressor or a show stopper. The purpose of this section therefore is to provide airplane designers with a discussion of a number of technology factors which in past airplane programs have proven to be a profit suppressor or a show stopper. The material is organized as follows:

9.2.1	Drag prediction	9.2.2	Loads prediction
9.2.3	Laminar flow	9.2.4	Range prediction
9.2.5	Maximum speed prediction	9.2.6	VTOL capability
9.2.7	New materials	9.2.8	New engines
9.2.9	Manufacturing processes	9.2.10	De-facto stability
9.2.11	Summary		

This list of possible show stoppers is far from complete. However, it serves to illustrate problems which manufacturers have stumbled into because of excessive optimism or because of insufficient testing.

Examples of each are given without naming the manufacturer or the program involved.

9.2.1 Drag Prediction

Drag prediction is still a 'troublesome' aspect of aeronautical technology, despite all the advances made in computational aerodynamics as well as in wind-tunnel testing techniques. Drag was under-predicted on several recent military airplane programs. The results were:

a) on a military transport program performance was compromised and costly modifications were made

b) on a tactical fighter program performance was compromised to the point where even after costly modifications the effectiveness of the airplane was significantly reduced

c) a military jet trainer program was cancelled well into the flight test program

9.2.2 Loads Prediction

Airframe airload prediction can be a major headache in the transonic speed regime. On a recent military jet transport program the inboard wing airloads were under-predicted. This led to reduced wing fatigue life and costly modifications to the airframe.

9.2.3 Laminar Flow

Laminar flow has the potential to reduce friction drag very considerably. However, the stability of laminar boundary layers is tenuous. Therefore, those factors which are known to affect laminar boundary layer stability must be completely understood. For example, it is well known that high leading edge sweep angles tend to lead to rapid laminar boundary layer transition to turbulent flow which increases friction drag by a considerable amount. Therefore, it might not make sense to use high leading edge sweep if laminar flow is essential to attain the predicted operating speed of a new airplane.

This has resulted in significant downward revision of the cruise speed of a business airplane.

9.2.4 Range Prediction

If an airplane must have very long range capability, the Breguet range equations of page 15, Pt I show that three factors are essential:

1) High average lift-to-drag ratio: $(L/D)_{cr}$

2) Low weight ratio: W_{end}/W_{begin}

3) Low engine fuel specifics: c_p or c_j

The technical program decision makers had better assure themselves that none of these factors are very sensitive to optimism in their prediction. If they are, a directed research program aimed at obtaining these optimistics values may have to be financed.

An early jet transport program ran into this type of a problem resulting in not meeting transatlantic range predictions.

9.2.5 Maximum Speed Prediction

If an airplane must attain very high speed at sea-level, a number of factors bear watching:

1) Drag, including compressibility drag

2) Aeroelastic deformations which affect drag and controllability

3) Flutter, particularly in not yet encountered modal combinations

It takes a major engineering effort up-front to determine whether or not all technical risks in these three areas have been lowered sufficiently to justify launching the program. A number of jet fighter programs were launched without sufficient validation in these areas. The result was either early cancellation or a significant reduction in operational capability.

9.2.6 VTOL Capability

The following factors must be solidly researched and tested before launching a VTOL program:

* Suckdown effects of the configuration

* Adequacy of the reaction controls in hover

* Efficiency of ejectors used

* Reingestion of high temperature gasses

* Foreign object damage

Several VTOL programs of the past failed because one or more of these factors were not adequately researched. In all cases major cost overruns caused the program to be cancelled. In one case a test pilot also lost his life.

9.2.7 New Materials

On several occasions new materials have been introduced without adequate testing. Examples are:

1) Introduction of 7075-ST aluminum in areas of the wing structure where corrosion/crack induced fatigue led to very low structural life.

In a commercial jet transport program this forced the manufacturer into a very costly re-skinning program.

2) Introduction of composites which were found to be subject to (difficult-to-detect) delaminations resulting in compromises to the integrity of the structure.

This caused the manufacturer to use significantly thicker skins than planned. As a result the airplane 'gained' a lot of unplanned weight which reduced the useful load.

3) Introduction of composites without proper manufacturing control over weight. This led to unpredictable weight variability from one production airplane to the next.

This caused the termination of the program and bankruptcy of the manufacturer.

9.2.8 New Engines

History is full of examples of airplanes which did not succeed because the engines around which they were designed never reached either acceptable performance or acceptable reliability.

There are also many programs where engine problems caused major cost overruns, delays in deliveries and corresponding financial headaches for the manufacturer.

9.2.9 Manufacturing Processes

Consider the following case histories:

1) As part of the manufacturing process, a given airplane component may have to be quelched in water. The water used comes from the city water supply. During one season, the city puts some chlorine compounds in their water. The chlorine alters the structural characteristics of that airplane component.

   It took a lot of 'detective' work by engineers to identify this problem.

2) A manufacturer introduces metal-to-metal bonding in small radius skin structures, such as a horizontal tail. Say that in the analysis of the flutter behavior of the tail, all bonding was assumed to be complete. Suppose that as a consequence of local disbonding, the tail develops flutter which causes inflight airframe failure. Now the probability of a minor flaw in manufacturing has major consequences!

9.2.10 De-facto Stability

A new airplane requires a certain inherent static instability to meet certain mission demands. The manufacturer decides to employ a digital flight control system. However, the manufacturer engages a sub-contractor to assist in the development of the flight control laws. Somehow, inadequate provisions are made to simulate the control laws before first flight. The first prototype crashes, causing major program delays and cost overruns.

9.2.11 Summary

There are several lessons in all these examples:

1. PLAY THE 'WHAT-IF' GAME AT ALL LEVELS OF DESIGN DECISION MAKING.

2. LEARN FROM MISTAKES MADE ON PAST PROGRAMS. DO SO BY READING THE TECHNICAL LITERATURE.

3. THERE IS NO SUBSTITUTE FOR THOROUGH ENGINEERING RESEARCH AND TESTING: IN OTHER WORDS, VALIDATE THE TECHNOLOGY

9.3 FACTORS INVOLVING MANUFACTURING FACILITIES

The following examples are offered as illustrations of the importance of considering requirements for manufecturing facilities in early design decision making.

Example 1

Sometimes, to assure technological feasibility, new manufacturing methods and processes must be worked out. There is no guarantee that these will work. Therefore, a certain amount of manufacturing research may have to be performed to ensure that the product will perform as advertised.

Examples of programs where a very intensive program of manufacturing research was carried out are the F-27 and the F-28 transport programs and the SR-71 reconnaissance airplane program.

Example 2

It is desirable from a structural weight viewpoint to have as few joints in any structure as possible. In wing design this may lead to a requirement for very long wing skin panels. If it is not possible to manufacture these panels on existing milling machines new facilities may have to be acquired. The cost of this must be taken into account. An example of this is the 747 program.

Example 3

If laminar flow must be maintained over a large part of the wing chord it is essential that very tight surface tolerances be maintained. This has been found relatively easy to accomplish when using composite skins because of their inherent high degree of stiffness.

In metallic structures, maintaining large laminar flow runs is critically dependent on the number of joints and fasteners used. Two methods are used to minimize the number of joints and fasteners:

A) Manufacturing the wing torque box out of two sections which are joined on the neutral axis. Figure 9.10 shows an example of that method. This method requires the availability of sophisticated machining equipment.

B) Manufacturing external surfaces with 'outside-in' instead of the more conventional 'inside-out' method.

Figure 9.10 Manufacturing a Wing Torquebox in Two Pieces

Figure 9.11 Inside-out (Old Style) and Outside-in Method (New Style) of Airframe Manufacturing

Figure 9.11 shows an example of this method. It is also possible to manufacture wings in that manner. This method requires a new approach to tooling design.

All this emphasizes the need for preliminary design organizations to discuss their designs with manufacturing experts at the earliest possible time. In this manner it is possible to prevent poor design decisions to go unchallenged before it is too late. In addition, this can help to identify the need for new manufacturing equipment and new manufacturing facilities early in the design program. This way, the cost consequences of design decisions which affect manufacturing receive early scrutiny.

9.4 FACTORS INVOLVING POLITICAL AND/OR ENVIRONMENTAL FEASIBILITY

There are situations where political and/or environmental pressures cause airplane programs to be terminated or altered. The purpose of this section is to acquaint the reader with some of these pressures. Three examples will be discussed.

Example 1: Nuclear Power

In the fifties, major studies were conducted to determine the technological and financial feasibility of using nuclear power for the propulsion of large intercontinental bombers and transports. Flight tests of at least one small nuclear reactor were in fact carried out in a B-36 airframe. This program was eventually terminated for a number of reasons. Questions were raised relative to the technological and financial feasibility (i.e. the projected high cost).

But there were other questions such as: what happens in the case of a crash with perhaps an explosion and/or a fire? How much nuclear contamination of the environment will occur? How many people might get killed due to the radiation effects? What is the probability of such a crash occurring?

These questions all beg for answers because they affect political and environmental issues.

It is the lack of acceptable answers to these questions which, probably more than any technical or financial issue, led to the termination of this and other nuclear aircraft propulsion projects.

But there is another issue here: **ETHICS**.

Should engineers push for starting a flight hardware project if there are serious questions about safety and about the environment?

The author believes that engineers SHOULD NOT DO SO! Engineers have an ethical commitment to society not to engage in activities which lead to major safety problems and/or problems with the environment.

Example 2: Supersonic Transports

Consider the following question: Do large fleets of supersonic transports have the potential of causing significant damage to the earth's ozone layer?

Major study programs of the chemical interactions between engine exhausts and the upper atmosphere are now being conducted by NASA to determine the answer to this question.

Until the answer to this question is available and widely accepted by the scientific and political community there SHOULD NOT be a commercial supersonic transport program.

The reader will remember the cancellation of the US-SST program in the late sixties. That program was cancelled mostly because of environmental pressures. These pressures prevailed because the aeronautical community failed to come up with scientifically sound arguments to defuse them.

The reader may ask: what about military supersonic flights? The answer has to do with relative levels of pollution. Consider the following argument.

Assume that there are, worlwide, 5,000 military airplanes able to sustain supersonic cruising flight. These airplanes fly on the average 350 hours per year. However, of those 350 hours, they probably spend less than 10 percent of their flying time in supersonic, high altitude flight conditions. This represents 175,000 hours of flight in the ozone layer.

Now consider a worldwide fleet of 1,000 commercial supersonic transports. This fleet could consist of a mixture of business jets and airliners. Say that these airplanes fly an average of 3,000 hours per year. About 70 percent of this time will be spent in a supersonic

flight condition. The average takeoff weight of this fleet of airplanes will probably be about 500,000 lbs, compared to at most 50,000 lbs for their military counterparts. The reason why takeoff weight is important has to do with the fact that the cruise thrust (and thus the engine massflow) is proportional to takeoff weight. This represents a factor 10 difference.

This commercial fleet of SST's will therefore produce the military equivalent of (1,000)(0.7)(3,000)(10) = 21,000,000 hours of flight in the ozone layer. This represents a factor of 120 difference. That is more than TWO ORDERS OF MAGNITUDE!! That is why commercial supersonic flying might be a problem whereas military supersonic flying is thought to be insignificant in terms of ozone destruction.

Example 3: Noise

During the late sixties and early seventies serious questions were raised about airplane noise, particularly the following types of noise:

1) Airport and community noise caused by takeoff operations

2) Airport and community noise caused by approach and landing operations

3) Community noise and damage caused by sonic booms from supersonic overflights

The noise levels in many instances had become intolerable to many people. Political action followed. This in turn resulted eventually in legislation under which:

1) Takeoff noise is regulated in FAR 36

2) Approach and landing noise is regulated in FAR 36

3) Supersonic overflight (other than in certain restricted, military areas) is prohibited by law

As a result, airplane propulsion systems must all be designed so that they meet the minimum noise standards of FAR 36. Many older transport jets were (and are still) being phased out because their modification to the latest FAR 36 standards is not economical.

9.5 LESSONS LEARNED FROM PAST AIRPLANE PROGRAMS

The purpose of this section is to present a listing of lessons learned (the hard way) on past airplane programs. The 19 lessons were taken from Reference 54 with minor revisions.

Lesson 1: Engines

Engines usually take at least a year longer for development than airplanes. In planning an airplane program this must be accounted for.

It is advisable to design the airframe around at least two engines from different engine manufacturers. If not, one knowingly takes a large degree of program risk. Figure 9.12 shows the engine options available on the 747 in 1982.

Lesson 2: Being there First

Being the first manufacturer to go ahead with a new, advanced airplane does not assure success. History shows that the eventual prize may go to the manufacturer who is second, provided that one is fast on his feet and is technically and financially sound.

Lesson 3: Losing a Competition

A competitive loss can motivate the loser more than the winner. This possibility should be carefully considered by management!

Lesson 4: Validated Technology

Generally speaking, the highest VALIDATED state of technology has produced the longest term, and thus, the most financially rewarding production program. Note the emphasis on the word VALIDATED.

Management should always weigh the risks and unanticipated costs associated with advanced technology. It should be ready to take program actions to minimize these.

Lesson 5: Hidden Conservatisms

Engineering should not be allowed to make conservative design assumptions AND keep these hidden from program management.

Figure 9.12 Engine Options Available in the 747 Program

Figure 9.13 Early Delivery History of the 727 Program

A consequence of not heeding this lesson may be that the design is invalidated and/or that growth is not felt to be possible. This could stop an actually promising design from being launched!

Lesson 6: Non-recurring Investment

The RDTE cost associated with airplane development programs is also referred to as a 'non-recurring' cost. If a company wants to be in the commercial airplane market it should be prepared for investments of roughly twice the original RDTE investment cost. These costs may cover a period of 3-10 years.

Lesson 7: Staying power

Commercial airplane programs require that the manufacturers have 'staying power' to carry them over temporary market depressions or unexpected actions taken by the government.

Figure 9.13 shows the early delivery history of the Boeing 727, clearly indicating the need for staying power.

Lesson 8: Red-White-and-Blue Team Approach

To arrive at the 'best possible' design solution for a given mission it has been found to be cost effective to deploy two or three design teams. Each team should be given access to the same budget and talent pools as well as other required resources. This way, two to three different design approaches to a given mission can be compared and the 'best' one selected for full scale development.

It is vital that all three teams know and understand the rules by which what is 'best' will be decided.

Lesson 9: Up-Front Investment

To get a truly good airplane requires a large 'up-front' investment. This investment is to cover cost for new technology development, new machinery which may be required, training and equipment.

In commercial developments, management is free to decide on such investments AND to implement such decisions.

In U.S. government acquisition procedures there is no provision to adequately accommodate this.

Lesson 10: Producing Paper Airplanes

Government paper competitions which start airplane production programs without prototypes MAY result in selection of the bravest and least informed winner, with uncorrectable consequences.

Lesson 11: Readying Technology

Readying new technology developments for application to a production program can be a long and very expensive process. This process will frequently warrant funding stimulation. The payoff, when the need and technology readiness match, can be enormous.

Lesson 12: Sizing a New Airplane

Past decisions that have resulted in airplanes which were improperly sized indicate that designers should moderate their thinking to find a middle ground between 'highs' and 'lows' in projected market growth, competitor proliferation and airport constraints.

Lesson 13: Need for Maturation

A good airplane program requires at least two years of concentrated 'pre go-ahead' study and planning.

The penalty for omitting this phase is greater exposure to large changes late in the development program.

Lesson 14: Understanding Real Customer Needs

There is no substitute for understanding the real needs of a customer. This applies to commercial as well as to military programs. The job of the airplane designer is to design the airplane which the customer wants WHILE still meeting or providing for product capabilities the customer will want five years after all the basic design decisions have been made.

Lesson 15: Definition of Program Success

Management should clearly have in mind what is meant by ultimate program success. Designing two (or even more) airplanes 'head on' for the same market may

result in financial failure of one manufacturer unless the market is very large or segmented.

Lesson 16: Catering a Design to One Customer

Most good airplanes are designed to meet a broad spectrum of market requirements. Therefore, design compromises are ESSENTIAL. In the commercial market case, designing to the detailed requirements of one customer can result in an unsatisfactory program. Starting a program with one customer, while designing to a broader market need, can be satisfactory.

Lesson 17: Product Support

Product support after delivery is just as important as is successful design. Product support represents a growing financial burden with time in a successful airplane program. Without it, a manufacturer will soon find himself losing customers.

Lesson 18: World Competition

The airplane business is an international business.

In the race to acquire the most rewarding aeronautical technology, the prize will go to the nation and/or to the manufacturer who runs the fastest. The nation and/or the manufacturer who stops running will lose, regardless of lead or government protection.

Lesson 19: Design Emphasis

In future designs, the most constructive emphasis is to build a superior product at the lowest manufacturing cost.

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*ONLY REQUIRED WHEN APU IS NOT BEING USED

COURTESY: BOEING

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ACOUSTIC TREATMENT

APPENDIX A: AIRPLANE PRICE DATA

The purpose of this appendix is to provide airplane price data as well as a rapid method for 'ball-parking' airplane prices for future designs. Most of the information presented was obtained from the following sources:

1. Business and Commercial Aviation: this monthly magazine publishes an annual Planning and Purchasing Handbook, usually as part of the April or May issue.

2. Aviation Week and Space Technology: this weekly magazine publishes an annual Aerospace Forecast and Inventory issue, usually as part of the March issue.

3. Flight International: this British weekly magazine publishes regular overviews of airplane price data.

4. AOPA Pilot: this monthly magazine publishes a general aviation aircraft directory usually as part of its March issue.

5. Professional Pilot Magazine: this monthly magazine publishes regular listings of airplane prices.

6. Interavia: this Swiss monthly magazine contains regular overviews of airplane price data.

Definition: The amount of dollars charged to an airplane operator to acquire that airplane is called the airplane market price: AMP.

Airplane prices vary with the following factors:

* Takeoff weight
* Performance
* Technology features
* Manufacturing cost
* Calendar time (if only because of cost escalation)
* Market conditions (demand and competition)
* Manufacturer sales strategies
* Government rules and regulations (certification standards, politics)
* Legal issues (product liability)

Except for the effect of cost escalation, no easy

rules for predicting the effect of these factors on airplane price can be given. Since all airplane price data are 'dated' by calendar year, the following equation is suggested to allow for cost escalation:

$$AMP_{19XX} = AMP_{\text{then year}}(CEF_{19XX}/CEF_{\text{then year}}) \qquad (A1)$$

where: AMP_{19XX} is the desired airplane price in 19XX

$AMP_{\text{then year}}$ is the given airplane price in 'then year' dollars

CEF is the cost escalation factor obtained from Figure 2.7. The ratio $(CEF_{19XX}/CEF_{\text{then year}})$ is called the CEF-ratio.

Sources 1-6 provide price data for many different calendar years. Therefore, Eqn.(A1) was used to 'correct' all data to 1989 dollars.

Airplane prices can be correlated with many factors. The most common factors used are the takeoff weight and the number of seats. In this appendix airplane prices are correlated with takeoff weight only. A brief discussion of other price parameters is given in Section A11. The airplane price data are organized as follows:

A1 SAILPLANE PRICE DATA

A2 ULTRALIGHT AIRPLANE PRICE DATA

A3 AGRICULTURAL AIRPLANE PRICE DATA

A4 SINGLE ENGINE PISTON AIRPLANE PRICE DATA

A5 MULTIENGINE PISTON AIRPLANE PRICE DATA

A6 MULTIENGINE TURBOPROP AIRPLANE PRICE DATA

A7 BUSINESS JET AIRPLANE PRICE DATA

A8 TURBOPROP COMMUTER AIRPLANE PRICE DATA

A9 COMMERCIAL TRANSPORT PRICE DATA

A10 MILITARY AIRPLANE PRICE DATA

A11 A GENERAL LOOK AT AIRPLANE PRICE DATA

A1 SAILPLANE PRICE DATA

Sailplane prices are plotted as a function of their takeoff weight in Figure A1. These data are based on information from the March, 1985 issue of AOPA Pilot Magazine. The data were adjusted to 1989 dollars with Equation (A1) and a CEP ratio of 1.082 obtained from Fig.2.7.

It is seen that there is a large spread in the data. The trend line is seen to 'fit' the data (+/- 25 percent) for unpowered as well as powered sailplanes. Based on this trend line the following equation is suggested for 'ballparking' sailplane prices:

$$AMP_{1989} = \text{invlog}\{1.1592 + 1.0705(\log W_{TO})\} \qquad \text{(A2)}$$

where: AMP_{1989} is the predicted market price for sailplanes in 1989 dollars.

This equation applies to the following range of takeoff weights: 600 lbs < W_{TO} < 2,000 lbs and should NOT be used outside that range.

A2 ULTRALIGHT AIRPLANE PRICE DATA

The March, 1985 issue of AOPA Pilot Magazine contains a lengthy listing of ultralight prices. Because these aircraft are usually delivered in the form of a kit (raw materials or pre-cut) most prices apply to the unassembled vehicle. There did not seem to be a reasonable correlation between prices and takeoff weights. Based on the aforementioned data, the following ranges are suggested for 'ballparking' ultralight airplane prices in 1989 dollars:

Pre-cut kits: $AMP_{1989} = 5(W_{TO})$ to $10(W_{TO})$ (A3)

Raw material kits: $AMP_{1989} = 3(W_{TO})$ to $10(W_{TO})$ (A4)

According to the data, ultralight takeoff weights range from 400 to 800 lbs. Note: ultralights are not certified to FAR standards.

A3. AGRICULTURAL AIRPLANE PRICE DATA

Prices for agricultural airplanes are plotted as a function of takeoff weight in Figure A2. These data are based on information from the March, 1984 issue of AOPA Pilot Magazine and from the October, 1984 issue of Agricultural Aviation Magazine. The data were adjusted to 1989 dollars using Equation (A1). The CEF ratio of 1.119 was obtained from Figure 2.1.

Some of the price data were judged to be unrealistically low by the author. These 'suspect' data are identified in Figure A2. After disregarding these data, a regression analysis was performed. This resulted in the following formula for 'ballparking' prices of agricultural airplanes:

$$AMP_{1989} = \text{invlog}\{-0.6681 + 1.5799\log(W_{TO})\} \qquad \text{(A5)}$$

where: AMP_{1989} is the predicted market price for agricultural airplanes in 1989 dollars.

This equation applies to the following range of takeoff weights: 3,000 lbs < W_{TO} < 15,000 lbs and should NOT be used outside that range.

It is interesting to observe that engine type does not seem to have a significant effect on agricultural airplane price trends.

A4. SINGLE ENGINE PISTON AIRPLANE PRICE DATA

The price of single engine piston airplanes depends on the following factors:

* engine aspiration: normal or turbocharged
* cabin pressurization (most are not!)
* takeoff weight and performance
* product liability insurance

Figure A3 shows the calendar time history of a single engine piston training airplane compared to the CEF. The price of this airplane compares reasonably with CEF trends until about 1982 when product liability became a major factor in airplane pricing. In 1985 Cessna decided to discontinue the production of this type: the price had become unreasonably high!

Price data for single engine piston airplanes were taken from the March, 1989 Planning and Purchasing Handbook issue of Business and Commercial Aviation Magazine. The data are plotted in Figure A4. Two trend lines suggest themselves: one for airplanes with normally aspirated engines and the other for airplanes with turbocharged engines. Note, that the pressurized Piper Malibu follows the latter trend line. Also note, that the only (in 1989) certified single engine turboprop airplane (the Cessna Caravan I) follows the normally aspirated trend line.

Note: the Lake-4 amphibian and the Beech BE-A36 were excluded from the data base because of the 'specialty nature of these airplanes. The Piper Super Cub was added to the data as follows: W_{TO} = 1,750 lbs at USD 45,000.

A regression analysis was performed to determine the trend line equations which may be used for 'ballparking' single engine piston and single engine turboprop prices.

For single engine airplanes with normally aspirated piston engine or with a turboprop engine:

$$AMP_{1989} = \text{invlog}\{-1.2435 + 1.8459(\log W_{TO})\} \qquad (A6)$$

For single engine airplanes with turbocharged piston engines (with and without cabin pressurization):

$$AMP_{1989} = \text{invlog}\{-1.1174 + 1.8459(\log W_{TO})\} \qquad (A7)$$

where: AMP_{1989} is the predicted airplane market price for single engine airplanes in 1989 dollars.

These equations apply inside the following range of takeoff weights: $1,500 < W_{TO} < 10,000$ and should NOT be used outside that range.

A5. MULTIENGINE PISTON AIRPLANE PRICE DATA

Airplane price data for 1989 and 1980 were obtained from the Planning and Purchasing Handbook (March issue of Business and Commercial Aviation Magazine). The data for 1980 were adjusted to 1989 with a CEF-ratio of 2.045 obtained from Figure 2.7. The resulting price data are plotted in Figure A5. Observe that the effect on price of turbocharging (over normally aspirated engines) and of cabin pressurization (over unpressurized) is negligible.

The author decided to eliminate the relatively very

highly priced Partenavia airplanes from the data before determining the trend line (See Figure A5) with a regression analysis. As a result, the following equation can be used for 'ballparking' prices for multiengine piston airplanes:

$$AMP_{1989} = \text{invlog}\{-0.8526 + 1.7413(\log W_{TO})\} \qquad \text{(A8)}$$

where: AMP_{1989} is the predicted market price for multiengine piston airplanes in 1989 dollars.

This equation applies to the following range of takeoff weights: 3,000 lbs < W_{TO} < 8,000 lbs and should NOT be used outside that range.

A6. MULTIENGINE TURBOPROP BUSINESS AIRPLANE PRICE DATA

This airplane category consists of airplanes which are configured for business and NOT for airline use. The 1989 Planning and Purchasing Handbook of Business and Commercial Aviation Magazine (March issue) was used to plot the price data of Figure A6. A regression analysis was performed on these data. From this, the following equation was derived, which may be used for 'ballparking' airplane prices:

$$AMP_{1989} = \text{invlog}\{1.9153 + 1.1115(\log W_{TO})\} \qquad \text{(A9)}$$

where: AMP_{1989} is the predicted airplane market price for multiengine turboprop business airplanes in 1989 dollars.

This equation applies to the following range of takeoff weights: 8,000 lbs < W_{TO} < 50,000 lbs and should NOT be used outside this range.

A7 BUSINESS JET AIRPLANE PRICE DATA

This airplane category consists of business airplanes, NOT airline configured airplanes. Price data were taken from the March 1989 issue of the Planning and Purchasing Handbook of Business and Commercial Aviation Magazine. Figure A7 shows a plot of these data. As seen, prices of the Gulfstream GIV and several Boeing airplane adaptations for business usage do not fit a linear trend. After eliminating these airplanes from the data a regression analysis was performed. The result is the following

equation which may be used to 'ballpark' airplane prices for this category:

$$AMP_{1989} = invlog\{0.6570 + 1.4133log(W_{TO})\} \qquad (A10)$$

where: AMP_{1989} is the predicted airplane market price for business jets in 1989 dollars.

This equation applies to the following range of takeoff weights: 10,000 lbs < W_{TO} < 60,000 lbs and should NOT be used outside this range.

A8 TURBOPROP COMMUTER AIRPLANE PRICE DATA

Price data for turboprop commuter airplanes were obtained from the March 1989 issue of the Planning and Purchasing Handbook of Business and Commercial Aviation Magazine. The data are plotted in Figure A8. A regression analysis was performed on the data. This resulted in the following equation, which can be used to 'ballpark' airplane prices for this category:

$$AMP_{1989} = invlog\{1.1846 + 1.2625(logW_{TO})\} \qquad (A11)$$

where: AMP_{1989} is the predicted airplane market price for turboprop commuters in 1989 dollars.

These equations apply to the following range of takeoff weights: 6,000 lbs < W_{TO} 50,000 lbs and should NOT be used outside these ranges.

A9 COMMERCIAL JET TRANSPORT PRICE DATA

Commercial jet transport price data were taken from the October 10, 1987 issue of Flight International (British weekly aviation magazine). A CEF ratio of 1.0237 was applied to adjust the data to 1989 dollars. The adjusted data are plotted in Figure A9. The trend line was determined with a regression analysis which yielded the following equation:

$$AMP_{1989} = invlog\{3.3191 + 0.8043(logW_{TO})\} \qquad (A12)$$

where: AMP_{1989} is the predicted airplane market price for jet transports in 1989 dollars.

This equation can be used to obtain a 'ballpark' es-

timate of jet transport prices. The equation applies to the following range of takeoff weights:

60,000 lbs < W_{TO} < 1,000,000 lbs and should NOT be used outside this range.

A10 MILITARY AIRPLANE PRICE DATA

Prices for military airplanes vary with the following factors:

* takeoff weight and performance
* types of technology included
* mission orientation (weapons capability)
* procurement politics

Price data for military airplanes are published on occasion in sources such as magazines number 2, 3 and 6 mentioned on page 297. The author has collected price data (in the form of unit fly-away prices without spares) from these sources and applied the appropriate CEF-ratio to adjust these data to 1989 dollars. Figure A10 represents a plot of these data. It is noted that military airplane prices vary 'all over the lot' when plotted in this manner. After eliminating the price data for what the author considers 'old technology' fighters, a regression analysis was performed and a trend line established. Except for the T-45 (which appears to be extremely expensive) the trend line follows the data rather well. Thus, the following equation is suggested for 'ballparking' prices for military airplanes:

$$AMP_{1989} = \text{invlog}\{2.3341 + 1.0586(\log W_{TO})\} \qquad \text{(A13)}$$

where: AMP_{1989} is the predicted airplane market price for military airplanes in 1989 dollars.

This equation applies within the following range of takeoff weights: 2,500 lbs < W_{TO} < 1,000,000 lbs and should NOT be used outside that range.

A11 A GENERAL LOOK AT AIRPLANE PRICE DATA

Figure A11 presents a plot of all price-versus-takeoff-weight equations discussed in Sections A1 - A10. The suggested ranges of takeoff weight values are indicated with large dots. Note the two <u>overall trend</u> lines drawn in Figure A11. As seen, business jets and military air-

planes differ from these trend lines significantly. It seems that, roughly speaking, the business jets and the military airplanes are 2 to 3.5 times as expensive as all other types of airplane.

An interesting question is: How have airplane prices varied with calendar years in 'then yaer' dollars? The answer may be found in Figures 9, 10 and 15 of Ref.25. The author has taken the trend lines from Figures 9, 10 and 15 of Ref.25 and derived the following equations:

<u>For tactical airplanes (Fig.9, Ref.25):</u>

$$(AMP_{\text{then year}}) = \qquad \text{(A14)}$$

$$= \text{invlog}\{-104.8212 + 0.056604(19XX)\}$$

<u>For bombers (Fig.15, Ref.25):</u>

$$(AMP_{\text{then year}} = \qquad \text{(A15)}$$

$$= \text{invlog}\{-138.4443 + 0.074074(19XX)\}$$

<u>For commercial transports (Fig.10, Ref.25):</u>

$$(AMP_{\text{then year}}) = \qquad \text{(A16)}$$

$$= \text{invlog}\{-99.2972 + 0.054054(19XX)\}$$

where: $(AMP_{\text{then year}})$ is the airplane market price in 'then year' dollars.

19XX is a calendar year anywhere between 1910 and 1990.

Figure A12 presents plots for each equation. Plots of the U.S. annual defense budget and the gross national product are superimposed. Note the following:

I. All airplane prices tend to increase with calendar time. The logarithmic slope of the price trends is larger than a similar slope of economic trends. If that trend is not reversed, airplanes will become unaffordable!

II. The logarithmic slope of bomber price trends is 50 percent greater than that of commercial transports!

In comparing airplane prices with one another, the following parameters are often used:

* Dollars price per pound of takeoff weight
* Dollar price per seat (civilian airplanes only)

Table A1 tabulates the prices per lbs of takeoff weight for all types of airplanes considered in this appendix. Note that, except for the commercial jet transports, the price per lbs <u>increases</u> for all types of airplanes <u>with increasing takeoff weight</u>. The slope constant in Eqns (A2) through (A11) and Eqn.(A13) are all larger than 1.0, but the slope constant for commercial transports (see Eqn.(A12)) is less than 1.0! An explanation for this may be that commercial transports represent a relatively 'mature' technology.

Note again that business jets and military aircraft at the high end of their weight ranges cost about the same in terms of dollars per lbs of takeoff weight!

Table A2 shows the cost per seat for civilian airplanes. It is of interest to observe that the price per seat for transport type airplanes falls in a relatively narrow range: 160,000 to 360,000 dollars/seat. Also note the very high price per seat for the upper end of the business jets!

Figure A1 Effect of Takeoff Weight on the Price of Sailplanes

Figure A2 Effect of Takeoff Weight on the Price of Agricultural Airplanes

Figure A3 Variation of the Price of a Light Trainer Airplane with Calendar Year

Figure A4 Effect of Takeoff Weight on the Price of Single Engine Piston Airplanes

Figure A5 Effect of Takeoff Weight on the Price of Multiengine Piston Airplanes

Figure A6 Effect of Takeoff Weight on the Price of Multiengine Turboprop Business Airplanes

Figure A7 Effect of Takeoff Weight on the Price of Business Jets

Figure A8 Effect of Takeoff Weight on the Price of Turboprop Commuter Airplanes

Figure A9 Effect of Takeoff Weight on the Price of Commercial Jet Transports

• EPSILON

Figure A10 Effect of Takeoff Weight on the Price of Military Airplanes

Figure A11 Composite Plot of Price Versus Takeoff Weight of all Airplane Types

Figure A12 Effect of Calendar Year on the Price of Three Airplane Types

Table A1 Ranges of Airplane Price per Pound for 1989

Airplane Type	Range of Takeoff Weights in Lbs Range of Prices in Dollars/Lbs		
Sailplanes	600	to	2,000
Eqn.(A2)	23	to	25
Ultralight Airplane Kits	400	to	800
Eqns (A3 and A4)	3	to	10
Agricultural Airplanes	3,000	to	15,000
Eqn.(A5)	22	to	57
Single Engine Piston Airplanes			
Normally Aspirated	1,500	to	10,000
Eqn.(A6)	28	to	138
Turbocharged	1,500	to	10,000
Eqn.(A7)	37		185
Multieng. Piston Airpl.	3,000	to	8,000
Eqn.(A8)	53	to	110
Multieng. TBP Airplanes	8,000	to	50,000
(Business). Eqn.(A9)	224	to	275
Business Jets	10,000	to	60,000
Eqn.(A10)	204	to	428
Turboprop Commuter Airplanes	6,000	to	50,000
Eqn.(A11)	150		262
Commercial Transp. Jets	60,000	to	1,000,000
Eqn.(12)	242		140
Military Airplanes	2,500	to	1,000,000
Eqn.(A13)	341		485

Table A2 Ranges of Airplane Price per Seat for 1989

Airplane Type	Range of Price per Seat in Dollars/Seat		
Sailplanes	16,000	to	80,000
Single Engine Piston Airplanes	20,000	to	50,000
Multiengine Piston Airplanes	33,000	to	70,000
Multiengine Turboprop Airplanes (Business)	115,000	to	450,000
Business Jets	320,000	to	1,600,000
Turboprop Commuter Airplanes	160,000	to	360,000
Commercial Transport Jets	170,000	to	250,000

APPENDIX B ENGINE AND PROPELLER PRICE DATA

The purpose of this appendix is to provide engine and propeller price data as well as a rapid method for 'ball-parking' such prices for future designs. The sources listed in Appendix A for airplane price data (which, by the way INCLUDE the cost of the engine to the airframe manufacturer) publish engine and propeller price data only sparingly. The data presented in this appendix were primarily taken from References 55 through 59.

Propulsion system prices are affected by:

* Uninstalled engine thrust or horsepower
* Turbine and compressor technology
* Calendar time (if only because of cost escalation)
* Market conditions (demand and competition)
* Manufacturer sales strategies
* Government rules and regulations (Certification requirements, specifications and politics)

Except for the effect of cost escalation, no easy rules for predicting the effect of these factors on propulsion system prices can be given. Since all engine and propeller price data are 'dated' by calendar year, the following equation is suggested to allow for cost escalation:

$$PSP_{19XX} = PSP_{\text{then year}}(CEF_{19XX}/CEF_{\text{then year}}) \qquad \text{(B1)}$$

where: PSP_{19XX} is the desired propulsion system price in 19XX

$PSP_{\text{then year}}$ is the given propulsion system price in 'then year' dollars

CEF is the cost escalation factor obtained from Fig.2.7. The ratio $(CEF_{19XX}/CEF_{\text{then year}})$ is called the CEF-ratio

Equation (B1) may be applied to adjust prices for engines (EP) and/or propellers (PP) from one year to another. Because the price data in References 55 - 59 are in various 'then year' dollars, Equation (B1) was used frequently to adjust price data to 1989 dollars.

<u>IMPORTANT NOTE:</u> In chapters 4 - 7 the symbols used for engine and/or propeller cost to the airframe manufac-

were C_e and C_p respectively. In this appendix, the symbols used for engine and/or propeller price are EP and PP respectively. For practical purposes it is acceptable to assume:

$$EP = C_e \qquad \text{AND:} \qquad PP = C_p \qquad \text{(B2)}$$

The engine and propeller price data in this appendix are organized as follows:

B1 PISTON ENGINE PRICE DATA

B2 TURBOPROP ENGINE PRICE DATA

B3 PROPELLER PRICE DATA

B4 JET ENGINE PRICE DATA

B1 PISTON ENGINE PRICE DATA

Piston engine price data were taken from Ref.55 and adjusted to 1989 dollars with Eqn.(B1). The prices do NOT include the price of the propeller. Price data for propellers are given in Section B3.

Piston engine price data are plotted versus engine shaft horsepower at takeoff (SLS), SHP_{TO_e} in Figure B1. The solid line represents the trend line obtained from a regression analysis on all data points. The trend line does not represent the data very well. To improve this, the horsepower range was split in two ranges:

80 - 200 shp AND: 200 - 500 shp.

Separate correlation lines were determined for each range. From this, the following formulas were derived for 'ballparking' piston engine prices:

For the shaft horsepower range of 80 - 200 shp:

$$EP_{1989} = \text{invlog}\{2.9923 + 0.4536(\log SHP_{TO_e})\} \qquad \text{(B3a)}$$

For the shaft horsepower range of 200 - 500 shp:

$$EP_{1989} = \text{invlog}\{-0.7770 + 2.0917(\log SHP_{TO_e})\} \qquad \text{(B3b)}$$

where: SHP_{TO_e} is the takeoff shaft horsepower per engine at sealevel standard (SLS) conditions.

Do NOT use these equations outside the ranges of shaft horsepower indicated.

B2 TURBOPROP ENGINE PRICE DATA

Price data for turbopropeller engines (NOT including the price of the propeller) were obtained from Refs 55 - 58. These 1975 data were adjusted to 1989 dollars with the help of Eqn.(B1). The data are plotted in Figure B2. A trend line was determined with a regression analysis. The trend line is seen to represent the data very well. This leads to the following equation for 'ballparking' turbopropeller engine prices:

$$EP_{1989} = \text{invlog}\{2.5262 + 0.9465(\log SHP_{TO_e})\} \qquad \text{(B4)}$$

This equation applies to a shaft horsepower range of 400 to 5,000 shp. It should NOT be used outside that range. For larger engines. Ref.56 suggests the following equations:

$$EP_{1989} = (1.418)(2,160,000)\{0.4(SF) + 0.600\} \qquad \text{(B5)}$$

for SF > 1.0 and:

$$EP_{1989} = (1.418)(2,160,000)\{0.533(SF) + 0.467\} \qquad \text{(B6)}$$

for SF < 1.0

where: the factor 1.418 is the 1989 to 1980 CEF-ratio

2,160,000 is the 1980 base price in US dollars for a turboprop engine with 20,424 shp

SF is an engine shp scale factor determined from:

$$SF = SHP_{TO_e}/20,424 \qquad \text{(B7)}$$

NOTE: the price data in this Section INCLUDE the engine gearbox, but NOT the propeller.

B3 PROPELLER PRICE DATA

Price data for propellers were obtained from Refer-

ences 56 through 59 and adjusted to 1989 dollars. The data are plotted in Figure B3. It is seen that there are not enough data points to establish the effect of number of blades on propeller cost. A trend line was drawn by performing a regression analysis on all METAL propeller data. This line has the following equation which may be used to 'ballpark' METAL propeller prices:

$$PP_{1989} = \text{invlog}\{0.6119 + 1.1432(\log SHP_{TO_e})\} \qquad (B8)$$

For COMPOSITE propellers the following equation may be used:

$$PP_{1989} = \text{invlog}\{0.7746 + 1.1432(\log SHP_{TO_e})\} \qquad (B9)$$

These equations should NOT be used outside a range of 100 - 20,000 shp.

B4 JET ENGINE PRICE DATA

Jet engine price data were taken from Ref.55 and plotted versus maximum sealevel static thrust, T_{TO_e} in Figure B4. The data do NOT indicate a systematically higher price for turbofans than for turbojets. That is why only one regression analysis was performed, resulting in the solid line in Figure B4. The data straddle the trend line within a +/- 20 percent band. Using Eqn.(B1) to correct the jet engine price data from 1975 to 1989, the line labelled 'suggested for 1989' was obtained. The equation which represents this line is:

$$EP_{1989} = \text{invlog}\{2.3044 + 0.8858(\log T_{TO_e})\} \qquad (B10)$$

This equation may be used to 'ballpark' new engine prices. It should NOT be used outside the takeoff thrust range of 1,000 to 50,000 lbs per engine.

Figure B1 Effect of Shaft Horse Power on the Price of Piston Engines

Figure B2 Effect of Shaft Horse Power on the Price of Turboprop Engines

Figure B3 Effect of Shaft Horse Power on the Price of Propellers

Figure B4 Effect of Takeoff Thrust on the Price of Jet Engines

APPENDIX C AVIONICS SYSTEMS PRICE DATA

The purpose of this appendix is to provide price data for airplane avionics systems.

The material is organized as follows:

C1 COMMERCIAL AVIONICS SYSTEMS PRICE DATA

C2 MILITARY AVIONICS SYSTEMS PRICE DATA

C1 COMMERCIAL AVIONICS SYSTEMS PRICE DATA

The author is grateful to Business and Commercial Aviation Magazine for their permission to copy the Avionics Section of the 1989 Planning and Purchasing Handbook. The material presented in this Section is an exact copy of the following pages of the May 1989 issue of Business and Commercial Aviation: pages 151, 152, 154, 156, 158, 160, 161, 162, 164, 166, 168, 170, 172, 174, 176, 178 and 182 through 202. These pages appear here as pages 330 through 366.

The reader should realize that any price data will vary with time and with marketing strategies of manufacturers. Whenever possible, actual and recent price data should be used. Lacking such information, the price of any avionics system listed in this appendix for 'then years' other than 1989 may be estimated from:

$$(\text{Avionics System Price})_{\text{then year}} = \qquad \text{(C1)}$$

$$= (\text{Avionics System Price})_{1989}(\text{CEF}_{\text{then year}}/\text{CEF}_{1989})$$

where: $(\text{Avionics System Price})_{1989}$ may be found from the data in pp. 330 - 366.

CEF is the cost-escalation-factor of Figure 2.7 in Chapter 2.

<u>IMPORTANT NOTE:</u> The symbol used for total avionics cost in Chapters 4 - 7 is: $C_{avionics}$. This cost is intended to be the SUM of the individual avionics system prices obtained from this section.

INTRODUCTION 1989 AVIONICS

Canadair's Challenger 601-3A cockpit utilizes quiet/dark design principles.

Business aircraft operators will remember the last decade of the 20th century as being the period during which digital avionics became dominant on business aircraft. Looking at the array of digital equipment in today's business aircraft, it's hard to remember that little over a decade ago, digital comm/nav/pulse radios were just being introduced. Electronic flight instrument systems (EFIS) were rarely seen in business aircraft prior to the mid-1980s. The proliferation of navigation management systems (NMS) is an even more recent phenomenon.

Integration is the single most prominent development in digital avionics today. The big three avionics houses—Bendix/King, Honeywell and Rockwell/Collins—are placing great emphasis on integrated avionics packages. For example, comprehensive Bendix/King or Honeywell installations may include EFIS, NMS, digital air data computers, comm/nav/pulse radios and weather avoidance radar—all manufactured by one company. Rockwell/Collins packages feature virtually the same broad line-up of equipment except that the firm relies on outside manufacturers for NMS such as Universal Navigation Corporation.

In late 1988, Collins announced its Pro Line 4 fully integrated avionics system that upped the ante for all competitors (see *EFIS* elsewhere in this B/CA *Planning and Purchasing Handbook*). This brand-new design features EFIS, an engine instrument and crew alerting system (EICAS) and cockpit control display units (CDUs)—all branching out from a central computer that constitutes the heart of the system. Left- and right-side navigation computer modules, radios, sensors and data concentrators are all contained in just one remote-mount computer, greatly reducing the volume, weight and number of components in the avionics bay.

Pro Line 4 accelerates the acceptance of Arinc 429 as the digital interface of choice in business aircraft. Other digital and analog interface formats will persist for many years, owing to preferences of individual avionics manufacturers, but Arinc 429 (or possibly GAMA 429) is becoming about as commonplace as the RS 232 serial digital interface used by desktop computers.

Multi-sensor NMS have all but squeezed out single-sensor navigation

The Learjet 55's flight deck boasts an impressive array of standard avionics including five-tube Collins EFIS, Bendix/King HF, Universal UNS-1 navigation management system with VLF/Omega sensor, SELCAL and a Global Wulfsberg Flitefone VI. Digital avionic designs are becoming increasingly prevalent.

systems in the more sophisticated business aircraft. Multi-sensor NMS units are able to take advantage of reliable and accurate short-range DME and VOR signals when in line-of-sight reception range of ground stations. Some systems, such as the Foster LNS 616B, are capable of using UHF TACAN bearing and distance information, thereby escaping the radial scalloping signal distortion associated with some VOR signals.

NMS provide exceptional flexibility in the choice of long-range nav sensors. Some aircraft operators may need Omega/VLF receivers for extended over-water operations. Others may opt for Loran-C sensors, a popular choice for continental U.S. operations. For frequent international operations, it's hard to beat the current generation of laser inertial reference sensors that have achieved drift rates well under the one-nautical-mile-per-hour variation design criteria.

While the launch of Navstar global positioning satellites has resumed (there were eight of a planned 21 satellites in orbit at press time), the cost of putting operational and spare satellites into orbit could be a hot topic on Capitol Hill with today's emphasis on federal budget deficit reduction. If funding constraints delay the launch of additional GPS vehicles, multi-sensor NMS designers may be forced to look at inter-operable multi-sensor technology.

An inter-operable system is quite different from an NMS that simply uses inputs from several independent long-range navigation sensors. Such a system actually combines inputs from many sources to compute the aircraft's position. For example, GPS might be used for an extremely accurate time check. One line of position might be computed from signals received from Omega or VLF stations, while a second line of position might be obtained from a pair of Loran-C transmitters. And eventually all Loran-C stations may be synchronized, permitting a Loran-C receiver to plot lines of position from any two stations in any GRI chain—without the need to receive the master station.

As for ease-of-use features, virtually all long-range and nav management systems now come equipped with comprehensive databases. Operators can expect to find airports and navaids stored in almost all of these systems, with the better units having intersections and even arrival and departure procedures.

Most manufacturers now are making database updating easy. Rather than pulling a unit out of the avionics bay or changing internal components, such as programmable read-only memory chips, many systems may be equipped with optional diskette dataloaders. One innovative approach to updating is the use of database cartridges so compact that they easily fit into a shirt pocket.

Panel-mounted Loran-C receivers will be found in the flight decks of many business aircraft for which nav management systems represent too large an investment; however, the number of active manufacturers of such systems may stabilize at about a half-dozen strong firms.

Manufacturers' top of the line systems such as ARNAV Systems' R-50, Foster Airdata Systems' F4 Phoenix, Northstar's M1 and II-Morrow's 618 SuperBandit, have airport, navaid and special-use airspace databases. Most of these units are certified for VFR operations only, except for the IFR-certified Foster F4 Phoenix. Bendix/King will enter the panel-mount Loran-C market later this year when the firm introduces its cross-chain, IFR-certifiable unit. This innovative product should prove highly capable for a panel-mount Loran-C receiver because of its reception range, its impressive database and its wide selection of features. But don't anticipate existing Loran-C manufacturers to stand still in the face of this new competitor.

By the year's end, we also anticipate that at least one manufacturer will announce a panel-mount NMS, complete with a short-range VOR/DME navigation capability as well as an internal Loran-C receiver. It is not unreasonable to conclude that a GPS sensor will find a home in compact, panel-mount nav systems in the future, right alongside the internal Loran-C sensor.

We also can look forward to the expanded use of EFIS in business aircraft because of the variety and sophistication of information that can be displayed on a single tube. In addition, reversion modes reduce the odds of being grounded by a single tube or component failure.

Growing in importance is the increasing time and cost associated with repairing aging electro-mechanical instruments. The reliability of EFIS units has increased substantially in the last several years, reducing the overall cost of ownership to a level that is becoming competitive with

Dassault's Falcon 900 employs an all-DC electrical system and digital avionics throughout its spacious cockpit. Honeywell furnishes the aircraft's radar, dual laser IRSes, EFIS and dual flight management systems. Collins Pro Line II Navcomms is standard on the aircraft.

conventional "clocks and dials." In recent years, we've witnessed a great deal of research and development activity, accompanied by subtle but effective publicity, regarding flat panel displays. However, we will need to wait beyond 1989 before seeing flat panel displays offered for business aviation.

The trend in EFIS is toward parts commonality. The number and variety of displays, instruments, dials and annunciators is declining in instrument panels. This decline is accompanied by a reduction in shop spares. Ultimately, fleet operators that have equipped all their various aircraft with the same brand of EFIS/EICAS may find their parts shelves bare of all flight and engine instruments, plus annunciators, save one common replacement display tube and maybe one spare generic symbol generator.

TCAS, the traffic alert and collision avoidance system that has been mandated for the air carriers and air taxi operators of larger aircraft, continues to be a popular topic discussed by business aircraft operators. Many industry officials believe that such operators would quickly purchase TCAS units if they were certified and in dealer inventory.

Legitimate concern exists on the part of some observers that active TCAS installed on host aircraft may overload the reply capacity of the conventional ATCRBS transponders on intruder aircraft. Such a limitation may heighten the need for development of passive collision avoidance systems, such as Avion Systems' Sentinel II (B/CA, March 1989, page 49).

Nevertheless, there is little doubt that any future derivative collision avoidance system will rely on conventional transponders as the key cooperative avionics unit needed to identify intruder aircraft.

The arrival of active TCAS is putting pressure on Mode S transponder development. Mode S transponders are used by TCAS systems in two approaching host aircraft to coordinate traffic avoidance maneuvers.

Activity associated with microwave landing systems (MLS) continues to be decidedly flat as a result of the stalemate between the FAA and its MLS ground station supplier, Hazeltine. Delivery delays, cost overruns and organizational problems have plagued the MLS program for more than three years and many business aircraft operators simply have lost interest in buying MLS avionics.

But, it should be noted that the Europeans are going ahead with their

Foster AirData Systems' F4 Phoenix Loran offers impressive performance, operating features, all-lighting readability and relatively low-cost.

MLS installation plans. On the Continent, MLS deployment is being mandated by the success of commercial FM broadcasters that are legally climbing up the frequency band into the lower end of frequencies formerly allocated to ILS localizers. Also, the rock-solid, stable signal characteristics of MLS represent a genuine improvement compared to ILS at airports subject to terrain interference or ground-plane variations caused by snow drifts or tidal changes.

In the United States, MLS deployment is likely to occur first at military and federal government airports plus a few private installations. Because of these developments, business aircraft owners that operate internationally will find a greater need for MLS avionics in the immediate future than operators that fly within the confines of North America.

The next 12 months will be exciting ones in the avionics market. While it's not likely that a third crew seat will be added to business aircraft, the intelligent functions, decision-making assistance and help in controlling the aircraft provided by today's advanced avionics may just make two pilots think that there's an extra crewmember somewhere in the cockpit.

AVIONICS GLOSSARY

ADF—Automatic direction finder, a navigation receiver providing bearing to a nondirectional beacon.
ADI—Attitude deviation indicator.
AHRS—Attitude-heading reference system, a sensor deriving aircraft attitude from a strapdown gyro or other position sensor, accelerometers, and heading from a flux valve; replaces the conventional gyro set.
Air Data—Values computed from pitot, static and temperature measurements, usually by means of a digital computer; Arinc 575 defines outputs.
AM—Amplitude modulation.
AOA—Angle of attack.
Arinc—Acronym for Aeronautical Radio, Incorporated, a nonprofit corporation owned by member airlines to define form, fit and function of avionics equipment and to provide radio communication services.
Arinc 419—A compendium of digital information transfer characteristics as applied in Arinc 500-series equipment specifications.
Arinc 429—A standard for broadcast digital information transfer systems for general applications.
Arinc 561—Inertial navigation system specifications.
Arinc 571—Inertial sensor attitude-heading reference system specifications.
Arinc 575—Digital air-data system specifications.
Arinc 600-series—Equipment specifications.
Arinc 700-series—All digital equipment specifications for new-generation transport category aircraft.
ASCB—Avionics Standard Communication Bus, a digital electronic bus developed by Sperry that provides bidirectional "broadcasting" of data between aircraft systems by means of a "ring network"; similar to, but slower than the MIL-S 1553 network.
ATI and ATR—Arinc form factors (see "Arinc Form Factor" chart).
BCD—Binary coded decimal.
Beam Width—The included angle of a weather radar interrogation signal as defined by the antenna characteristic.
BRG—Bearing.
CDI—Course deviation indicator.
CDP—Counter-drum-pointer, a type of altimeter display that uses a combination of a rotating drum with numerals and a pointer.
CDU—Control-display unit.
Coherent Detection—A technique used in certain ADF receivers to improve useful range by providing better definition of weak signals.
Composite Video—Analog VOR receiver output before processing.
Contact Digitizer—A mechanical device that converts analog information to digital codes by means of electrical contacts.
CRT—Cathode ray tube, similar to a "TV" picture tube.
CW—Continuous wave, a radio carrier broadcast that does not have modulation.
DA—Drift angle.
DG—Directional gyro.
DH—Decision height.
Digitizer—Any electronic device capable of converting information to a digital format; usually refers to encoding devices that convert sensed altitude into a transponder code.
DR—Dead reckoning.
DTK—Desired track.
Duplex—Separate channels for transmission and receiving.
E- (as a prefix)—Used with "ADI" or "HSI," to designate an electronic display using a CRT.
EAROM—Electrically alterable read-only memory, a type of digital memory device.
ECDI—Electronic course deviation indicator.
EFIS—Electronic flight instrumentation system.
EEPROM—Electrically erasable programmable read-only-memory.
Gas discharge—A type of luminescent digital display, usually seven-segment and driven by a high-voltage source; characterized by low heat generation.
FM—Frequency modulation.
GCR—Ground clutter rejection.

The Beechjet 400 cockpit will be the first to be equipped with Collins Pro Line 4 EFIS, an advanced integrated display system that could grow into an all-glass panel.

GMT—Greenwich Mean Time, used in navigation systems.
GS—Groundspeed.
Hot Wire—Any method for providing some continuous electrical power to a memory device from the airplane's primary battery.
HSI—Horizontal situation indicator.
Hz or Hertz—Cycles per second, used to describe radio frequencies; usually with the prefix *k* for kilo (thousands) or *M* for mega (millions).
ILS—Instrument landing system, a precision approach system using a localizer, glideslope, an outer marker beacon, middle marker beacon and approach lights.
INS/IRS—Inertial navigation system/inertial reference system.
ITU—International Telephone Union, refers to certain HF channels.
Laser IRS—An inertial reference system deriving angular rate information by measuring doppler shift between two contra-rotating light beams.
LCD or LC—Liquid crystal display.
LED—Light-emitting diode.
LOC—Localizer, the course guidance component of an ILS.
Looks per minute—Scanning or sweep rate of a weather radar antenna.
Loran-C—Hyperbolic grid navigation system that is based upon measured time differences (TD) from pulse transmissions.
Mid-Continent Gap—A lapse in Loran-C signal coverage roughly spanning the United States and Canada, east of the Rocky Mountains and west of the Mississippi River.
MFD—Multifunction display, the "third or fifth tube" in an EFIS, replaces the weather radar and displays radar data, navigation maps, checklists, and other information.
MS—Military specifications, which define the size of a standard three-inch instrument panel cutout for such mechanical displays as altimeters and airspeed indicators.
N/A—Not applicable.
NDB—Nondirectional beacon.
NMS—Navigation management system.
Nonvolatile—Describes a digital memory that retains information through system shutdown.
Omega—A very-low-frequency navigation system.
Optical Digitizer—Any device using a photosensor to read a light beam through slots or cutouts and thereby to convert analog information to digital codes.
Peak power output—The maximum transmitter power output measured over a very short time period; usually used to rate pulse transmissions.
PEP—Peak envelope power, a standard electronic rating of any AC source, including a radio transmitter.
Pulse—A transmission of very brief duration used to carry information by using time measurement or as a series of pulses representing code.
RA—Radio altitude.
RAM—Random access memory.
RMI—Radio magnetic indicator.
RMS—Root mean square.
RNAV—Area navigation.
ROM—Read-only memory.
RS-232—A serial digital format used in data transmission.
RT—Receiver-transmitter combined in a single line-replaceable unit.
SAR—Search and rescue, usually pertaining to weather radars and navigation systems.
SI—Slip indicator or inclinometer, usually a ball within a tube containing fluid.
Simplex—Single frequency for both transmitting and receiving in communications; generally implies a push-to-talk function and verbal procedures.
Slaving Rate—The rate, in degrees per minute, a compass system can adjust for gyro precession; a "fast-slave" mode is used to align a compass system when first turned on.
SSB—Single sideband, a highly efficient form of radio transmission wherein information is carried on a sideband instead of the center carrier in order to gain range performance; usually used in HF communication transceivers.
STC—(1) Sensitivity-time compensation, in weather radar a technique and circuit for calibrating a display; (2) supplemental type certificate.
Synchro—Any device capable of converting mechanical position into an (analog) electronic signal.
Synthesizer—Frequency synthesizer, an electronic circuit capable of generating multiple frequencies from a single crystal oscillator.
TAS—True airspeed.
TD—Time difference, in Loran-C navigation measurements.
3P—Three-pointer altimeter display.
TKE—Track error.
TRK—Track, or path over the ground.
TSO—Technical standard order, a performance specification and production compliance criteria applied to avionics and defined by FARs and the Radio Technical Commission for Aeronautics.
TTG, TTS or TTW—Time to go, time to station, time to waypoint.
Turbulence Mode—Flight control system mode in which a "softer" response to gust upsets is programmed.
2 X 5—Two-out-of-five, an electronic radio tuning code.
USB—Upper sideband.
VAC—Volts alternating current.
VDC—Volts direct current.
VLF—Very low frequency, refers to Navy VLF transmissions used in long-range navigation systems in combination with Omega.
VNAV—Vertical navigation, or the capability to compute pitch-axis maneuvers and altitude-related points in space.
VOR—Very high frequency omni range.
X-band—Frequency range in which most general aviation weather radars operate.
XTK—Cross track; cross track error.

ARINC FORM FACTOR

The standard air transport radio (ATR) case is shown with all optional projections included. Volumes are given, except for the forward-facing projections called "doghouses."

Not shown on this chart is the short, low-profile (SLP) case used in the Collins Pro Line. All Pro Line remote units are about 3.25 inches high and 14 inches long and vary in width.

Panel-mounted avionics are built to the "mark width," which is about 6.25 inches. Widths will vary beyond the 6.25 inches because of flanges that extend beyond the radio mounting tray.

In the avionics tables, a case size is shown as 2 ATI, 3 ATI, up through 4 ATI. Arinc 600 conversion factors: As a rule of thumb, divide Minimum Configuration Units (MCU) by eight for ATR size:

1 MCU = 1/8 ATR*
2 MCU = 1/4 ATR
3 MCU = 3/8 ATR
4 MCU = 1/2 ATR
5 MCU = 5/8 ATR*
6 MCU = 3/4 ATR
7 MCU = 7/8 ATR*
8 MCU = 1 ATR

AIR TRANSPORT INDICATOR (ATI) CASE SIZES

Case Size Nominal	Case Size Actual	Air Transport Indicator	Length	Max Weight (lb)
2	2.175"	3 ATI 6.2	Expressed to nearest 10th of an inch.	3
3	3.175"			5
4	3.975"			9
5	4.975"			14
6	5.975"			20

STANDARD ATR CASE DIMENSIONS

ATR Size	Approx. Vol.		W		L_1		L_2 (Max)		H (Max)	
	In³	Liter	±.03 In	±.76 mm	±.04 In	±1.0 mm	In	mm	In	mm
Dwarf	95	1.56	2.25	57.15	12.52	318.0	12.62	320.5	3.38	85.8
¼ Short	215	3.52	2.25	57.15	12.52	318.0	12.62	320.5	7.62	193.5
¼ Long	335	5.49	2.25	57.15	19.52	495.8	19.62	498.3	7.62	193.5
⅜ Short	340	5.57	3.56	90.41	12.52	318.0	12.62	320.5	7.62	193.5
⅜ Long	530	8.69	3.56	90.41	19.52	495.8	19.62	498.3	7.62	193.5
½ Short	470	7.70	4.88	123.95	12.52	318.0	12.62	320.5	7.62	193.5
½ Long	725	11.88	4.88	123.95	19.52	495.8	19.62	498.3	7.62	193.5
¾ Short	720	11.80	7.50	190.50	12.52	318.0	12.62	320.5	7.62	193.5
¾ Long	1,120	18.36	7.50	190.50	19.52	495.8	19.62	498.3	7.62	193.5
1 Short	975	15.98	10.12	257.05	12.52	318.0	12.62	320.5	7.62	193.5
1 Long	1,510	24.75	10.12	257.05	19.52	495.8	19.62	498.3	7.62	193.5
1½	2,295	37.62	15.38	390.65	19.52	318.0	19.62	498.3	7.62	193.5

B/CA Art Dept

AUTOMATIC DIRECTION FINDERS

MANUFACTURER	MODEL TSO	MEMORY CAPACITY FREQ. (kHz) DISPLAY	POWER INPUT MEMORY TYPE	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KR 86 none	N/A 200-1750 mechanical	14 or 28 VDC none	2/6.3 6.25 x 2.6 x 9.05	$2,760	Panel-mounted; automatic band switching; self-contained indicator; no RMI capability; KA 42B combined loop-sense antenna available at same price
	KR 87 C41c	2 200-1799 gas discharge	11 to 33 VDC EAROM nonvolatile	3/6.7 6.25 x 1.3 x 11.23	$3,855	Panel-mounted; times flight and approaches; coherent detection; includes slaved KI 227 indicator; KI 226 dual pointer indicator available, KI 229/KNI 582 RMIs available as options
	KDF 806 C41c	2/9* 190-1799 gas discharge	28 VDC EAROM non-volatile	3/7.4 5.0 x 2.0 x 10	$5,935	Digital control head displays both active/standby frequency; coherent detection; flag circuitry; strappable for 2182 kHz, Gables control optional; combined loop-sense antenna included; remote flip-flop optional; *sine-frequency pilot-programmable "A" control head optional; night vision goggle compatible "A" control head optional; two-year warranty
	DFS-43 C41c	2 190-1660* (0.5 kHz) LCD or dichroic	28 VDC non-volatile	3/8.4 4.0 x 4.0 x 12.62	$12,926	ADF left-right steering to HSI; *2182 kHz intl. distress freq std; combined antenna; 0.5 kHz spacing, analog and Arinc 429 interfaces; two-year warranty
Collins Avionics Group Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	ADF-650A C41c	N/A 200-1799 mechanical	14 VDC* none	3/5.0 6.25 x 1.75 x 10.8	$4,425	Panel-mounted; includes ANT-650A combined loop-sense antenna and IND-650A indicator *28 to 14 VDC adapter — $120
	ADF-462 C41d	6 190-1799 gas discharge	28 VDC nonvolatile	3/6.0 ¼ ATR-SL	$10,105	Built-in diagnostics; compatible only with CSDB or Arinc 429 controls, digital signal processing; 2182 kHz emergency frequency; includes CTL-52 CSDB control and ANT-462A antenna; ANT-462B dual antenna optional
	ADF-60A C41c	2/6 190-1750 (½-kHz steps) gas discharge	28 VDC nonvolatile	3/8.7 ¾ ATR-SL	$10,630	Remote-mounted; coherent detection; combined loop-sense antenna; dual antenna optional; three-wire or sine/cosine RMI output available; includes CTL-60 electronic control and ANT-60A; optional CTL-62 with CAD-62 adapter provides serial digital tuning and six stored frequencies
	DF-206 C41b	N/A 190-1750 (0.5-kHz steps) none	28 VDC none	6/15.1 ¼ ATR-S	$20,816	Remote-mounted; azimuth card driven by compass system; automatic pointer stow in receive position; coherent detection; includes flush-fixed loop and sense antennas, antenna cable, control head, antenna coupler; Arinc 570

AUTOMATIC DIRECTION FINDERS

MANUFACTURER	MODEL TSO	MEMORY CAPACITY FREQ. (kHz) DISPLAY	POWER INPUT MEMORY TYPE	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	ADF 841 C41c	2 200-1799 gas discharge	11-32 VDC hot-wire nonvolatile	3/5.7 6.25 x 1.5 x 11.0	$3,022	Digital ADF with dual-frequency, flip-flop capability; elapsed- and flight-timers; includes combined loop-sense antenna and indicator
Terra Corp. 3250 Pan American Freeway NE Albuquerque, NM 87107 (505) 884-2321	TDF 100 none	N/A 200-1799 mechanical	11 to 33 VDC N/A	3/6.8 3.2 x 1.6 x 11.5	$1,895	

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE DISPLAY	NUMBER WAYPOINTS DATABASE	OUTPUTS	INPUTS	POWER INPUT MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Advanced Navigation, Inc. 61 Thomas Johnson Dr. Frederick, MD 21701 (301) 695-4840	AN-7000 Sensor	Loran-C sensor only	N/A N/A	Arinc 419/ 429 pos., GS, track, circular error probability computation	N/A	18 to 32 VDC —	2/14.6	$6,640	TSO C60a; software identical to AN-7000; a Loran-C sensor for use with KMS 690, Tracor 7880 and other navigation management computer systems
	AN-7000	Loran-C incandescent flatnet	200 nonvolatile internal battery	Lat/long and all nav data, with four analog steering signals standard; CDI / HSI; autopilot data; ARINC 429, 561, 568; EFIS, RMI, RS-232; GMT clock	HDG, TAS optional for automatic wind calculation	16 to 32 VDC (60 watts) —	3/19.6	$11,446	TSO C60a full continental US airspace IFR-approved including mid-continent area, limited non-precision approach approval AC 90-45A; AC 20-121
ARNAV Systems 16100 SW 72nd Ave. P.O. Box 23839 Portland, OR 97223 (503) 684-1600	R-25	Loran-C LED	1150 nonvolatile RAM	Waypoint alert; CDI; HSI; autopilot; RS-232	FC-10 Fuel computer; manual input of TAS & HDG for wind calculations; altitude for VNAV gph, fuel remaining for fuel range	10 to 35 VDC —	2/4.0	$2,495	Plain language alphanumeric display; antenna included

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUT / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
ARNAV Systems 16100 SW 72nd Ave. P.O. Box 23939 Portland, OR 97223 (503) 684-1600	R-50 Database	Loran-C / LED	150 / nonvolatile RAM	Waypoint alert warnings; PTK & APR annunciator; CDI; HSI; autopilot; RS-232	FC-10 Fuel Computer; manual input of TAS & HDG for wind calculations; altitude for VNAV advisories	10 to 35 VDC / —	2/4.0	$3,995	Plain language alphanumeric display; 20 route legs; auto extended range; CDI nav bearing; nearest 6 airports; nearest 6 VORs; TCA Alert
	R-40	Loran-C / LED	200 / nonvolatile RAM	Waypoint alert warnings; PTK & APR annunciator; CDI; HSI; autopilot; RS-232	None	10 to 35 VDC / —	1/4.5	$4,495	Plain language alphanumeric display; 20 route legs; Omni (pseudo VOR) extended range; CDI scaling; TSO C50a
	R-40 Database	Loran-C / LED	150 / nonvolatile RAM	Waypoint alert warnings; PTK & APR; annunciator; CDI; HSI; autopilot RS-232	Manual input of TAS & HDG for wind calculations altitude for VNAV	10 to 35 VDC / —	1/4.5	$4,995	IFR in compliance with AC 20-121; 10,000+ waypoint database with emergency nearest airport search; min. safe altitude; auto extended range; 20 route legs; Omni (pseudo VOR); optional aerial survey configuration (VFR certification); available with grid/nav aerial survey software — no database: $4,745
	R-60 Database	Loran-C / LED	200 / nonvolatile RAM	Waypoint alert warnings; PTK & APR; annunciator; CDI; HSI; autopilot; RS-232	Manual input of TAS & HDG for wind calculations altitude for VNAV	10 to 35 VDC / —	2/7.7	$9,495	Designed specifically for pedestal installations; TAS & winds aloft; VNAV; extended range; 20 route legs; Omni (pseudo VOR); available with grid/nav aerial survey software — VFR configuration without database
AZURE Technology San Jose Jet Center 1250 Aviation Ave. San Jose, CA 95110 (408) 947-2070	Long Ranger F/P	Loran-C / fixed panel or portable	99 / none	RS-232	None	12 VDC / —	2/4.2	$1,095	automatic chainswitching and station management; four flight plans up to 10 waypoints each. Antenna and mounting options avail.
Canadian Marconi 2442 Trenton Ave. Montreal, Quebec, Canada H3P 1Y9 (514) 341-7630	CMA-734 OSS (sensor) Mk II	Omega/VLF / —	891 / non-volatile RAM	Position, status, stations used, GS	synchro heading, air data (optional)	28 VDC / —	2/15.0	$23,833	Arinc 429; optional LCD CDU completes system for back-up long-range nav capability; TSO C94; three-year warranty
	CMA-764 OSS (sensor)	Omega/VLF / —	891 / non-volatile RAM	position, status, station used, GS	Arinc 429, heading air data	28 VDC / —	2/8.7	$23,962	Optional LCD CDU completes system for back-up long-range nav capability; TSO C94; three-year warranty

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUT / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Canadian Marconi 2442 Trenton Ave. Montreal, Quebec, Canada H3P 1Y9 (514) 341-7830	CMA-734 "Arrow" Mk II	Omega/VLF / liquid crystal	891 / non-volatile RAM	Pos, steering, nav data, HSI, autopilot	synchro, heading, TAS	28 VDC / —	3/17.2	$27,250	Lightweight system, tracks three Omega and one VLF frequency; TSO C94; three-year warranty
	CMA-734 Alpha Mk II	Omega/VLF / incan-descent filament	4,000 / non-volatile RAM	Position, steering, nav data, HSI, autopilot	synchro, heading, TAS	28 VDC / —	3/20.0	$37,507	CMA-734 with alphanumeric CDU; three-year warranty
	CMA-771 Alpha Mk III	Omega/VLF / incan-descent filament	4,000 / non-volatile RAM	Pos, steering, nav data, HSI, autopilot, digital	synchro, heading, TAS	115 VAC, 400 Hz / —	3/33.0	$54,207	Standard CMA-771 but with alphanumeric CDU; optional HSI synchro outputs; Arinc 599 analog, Arinc 429 digital interfaces; compatible with Global-Wahlsberg NDB-2; three-year warranty
Foster Airdata Systems, Inc. 7020 Hentley Rd. Columbus, OH 43229 (614) 888-9502	LRN 500	Loran-C / LCD	234 / non-volatile RAM	L&R-3, 1K ohm loads; T/F-2, 200 ohm loads; Flag-3, low-level 1K ohm loads; parallel track and WPT alert announ-ciation	none	10 to 32 VDC / —	1/3.1	$1,695	TSO C60a; IFR — AC 20-121; includes panel unit, 5,400 waypoint pilot-changeable database cartridge, antenna and preamp; VNAV advisories, nearby airport locater; "Demo Mode" allows removal from aircraft; carrying case $50
	F-4 Phoenix	Loran-C / LED dot matrix	234 / nonvolatile RAM	see LRN 500	none	10 to 32 VDC / —	1/3.1	$2,595	See LRN 500
	LNS 616B	Loran C/ auto Tacan / LED	400 / non-volatile RAM	HSI, EFIS, RS-232, Arinc 561, RMI, RS-232 Tacan channeling	VOR composite, Arinc 568, Foster serial	18 to 32 VDC / —	3/12.75	$22,600	TSO C60a; provides channeling to Foster DME 670 for automatic navigation; includes NY-153 antenna; optional DME-670 $11,900
Global Systems 2144 Michelson Dr. Irvine, CA 92715 (714) 851-0119	GNS-500A Series 4A	VLF/ Omega / CRT	127 / NDB-2 interface	Arinc 561 and 571 to EFIS; radar; INS; including Pos; TK; DTK; DA; GS; TTW; flight plan waypoints wind; dist. mag. var.	Heading; TAS; Arinc 571; Pos from INS; EFIS; radar; Arinc 561; joystick from EFIS; AFIS performance data	28 VDC at 7.5 amp / —	3/39.0	$64,025	GPS compatible; stores 127 waypoints defined by five letters, nine flight plans up to 20 waypoints each; digital datalink with AFIS; sunlight readable, full-alphanumeric CRT CDU; interfaces with FD, EFIS and autopilots; auto GMT and date; active flight plan capacity 50 waypoints; GNS 500A Series 1 - 3 may be converted to Series 4 configuration

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUT / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Global Systems 2144 Michelson Dr. Irvine, Ca. 92715 (714) 851-0119	GNS-500A Series 4M special mission navigation system	VLF/ Omega / CRT	127 / MDB-2 interface	Arinc 561 and 571 to EFIS; radar; INS; including Pos; TK; DTK; DA; GS; TTW; flight plan waypoints wind; dist. mag. var.; flight data recorder	Heading; TAS; Arinc 571; Pos from INS; EFIS; radar; Arinc 561; joystick from EFIS; AFIS performance data; forward looking infra-red sensor	28 VDC at 7.5 amp / —	3/39.0	$66,525	same capabilities as Series 4A + quick-initialization for scramble launch; special search / surveillance flight patterns; law enforcement, military configuration
Honeywell Inc. P. O. Box 29000 Phoenix, AZ 85038 (602) 863-8000	LASEREF	Laser IRS / —	none / none	Arinc 429, Arinc 419, Arinc 561 incl: attitude, heading (mag & true), GS, wind, track, present position	TAS Alt. V. Spd.	115 VAC, 400 Hz or 28 VDC backup / —	2/48	$146,742	Price includes installation kit; primarily analog interfaces to flight director (EFIS or EM), autopilot, radar, NMS:GNS-1000, LINS-1, KINS-660; fast align; no pressurization required
	LASEREF II	Laser IRS / —	none / none	ASCB, Arinc 429, Arinc 419 incl: attitude, heading (mag & true); GS, wind, track, present, position	TAS Alt V. Spd.	115 VAC, 400 Hz or 28 VDC back-up / —	2/48.0	$149,563	Primarily digital interfaces to Honeywell FMS, EFIS, autopilot; fast align; no pressurization required
Intermar Box 1261 Sydney, Nova Scotia B1P 6J9 Canada (902) 564-2043	LC1210 (sensor)	Loran-C / none	— / —	Position	—	12/26 VDC, 20W / —	2/10.0	$8,000	Loran-C sensor for navigation computer systems; continuous dual-chain operation; Output/ Control via Arinc 429 (dual input, triple output)
	LC1200	Loran-C / vacuum fluorescent	200 / nonvolatile RAM and EPROM	Pos; track error; TTW; GS; HSI; wind; digital interface	Heading; TAS, VOR-DME	12/26 VDC, 40W / —	3/15.0	$12,000	Continuous dual chain operation, Arinc format CDU; 1,200 programmed waypoint database; multiroute storage, optional VOR/DME back-up mode; five character alphanumeric waypoint identifiers position output for air-ground datalink; D/R mode usable outside Loran signal areas; optional SAR program

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUTS / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Litton Aero Products 6101 Condor Road Moorpark, CA 93021 (805) 378-2000	LTN-311	VLF/Omega / 8-line LED	99 / none	Digital pos; track; GS; TTG, HSI; Wind; XTD; TAE; DA; TH	Heading; TAS; ADC valid	115 VAC, 400 Hz / —	3/41.0	$44,190	Direct ARINC replacement for LTN-211; integrated GPS optional; ACARS route entry with OAT option; AC/DC noninterruptible P.S.; XTK discrete; Digital Air Data (575/429); std ARINC 599; BITE/STATUS in English language; pre-heated oscillator for rapid start; 1 year warranty
	LTN-211	Omega / —	9 / none	Pos; track error; TTW; GS; HSI; wind	Heading; TAS	115 VAC, 400 Hz / —	3/34.0	$44,580	HSI synchro and digital outputs standard; Arinc 599; optional VLF receiver available; 1 year warranty
	LTN-710	GPS / LED matrix	— / —	Digital ARINC 429; pos; track; vert track; velocity; acceleration Time (> 100 as); course steering	Altimeter / precise clock	115 VAC, 60-400 Hz / —	3/20.0	$50,000	Designed for aerial survey; present position display only; operates over entire range of signal levels; fixed freq. signals; single correlator; two IF freqs.; phase sampler; time accuracy to 900 nanoseconds; operational worldwide; 24 hr. all-weather coverage; pos. accuracy 40 SEP 3-D; 1 year warranty
	LTN-72	INS / —	9 / none	Pos; track error; TTW; GS; HSI; wind	TAS	115 VAC, 400 Hz / 28 VDC back-up / —	3/62.0	$172,265	Expandable memory; may be transverse-mounted; standby battery optional; Arinc 561; 1 year warranty
	LTN-92	Laser INS / 5-line LED matrix	99 / none	Arinc 561 digital & analog / Arinc 704 digital	VOR/DME; Tacan; Arinc 429; autopilot; steering; EFIS, PMS; Radar; pos; trk error; TTW; GS; HSI; wind	115 VAC / 400 Hz / 28 VDC back-up / —	3/63.6	$174,360	Pin for pin replacement for gyrobaled-gyro inertial nav systems — Arinc 561 in 1 ATR configuration; accommodates Omega/VLF acceleration advisory or GPS; interfaces with bulk storage units; solid-state design; 3 year warranty
	LTN-72R	INS/ RNAV / —	9 / none / none	Pos; track error; TTW; GS; HSI; wind	TAS; VOR-DME	115 VAC, 400 Hz / 28 VDC back-up / —	3/62.0	$175,090	Combined INS and vortac-based RNAV; compatible with Global-Wulfsberg NDB-2; meets requirements for enroute, terminal and approach; may be transverse-mounted; standby battery optional; Arinc 561; 1 year warranty

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUTS / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Litton Aero Products 6101 Condor Road Meorpark, Ca. 93021 (805) 376-2009	LTN-90 -100	Laser IRS / sensor	N/A / N/A	Arinc 429: Pos; attitude; heading; angular rates; acceler-ations; drift angle; VS; GS; flight path angle, wind track	Arinc 429 air data	115 VAC, 400 Hz / 28 VDC back-up / —	2/43.0	$190,030	Optional control display unit; primary inertial sensor for navigation management computer systems; complies with Arinc 704 spec for IRS; growth capability to ADIRS; 10 MCU size; for digital applications; 3 year warranty
	LTN-72RL	INS/ RNAV / —	99 / none	Pos; track error; TTW; GS; HSI; wind	TAS; VOR-DME	115 VAC, 400 Hz / 28 VDC back-up / —	3/62.0	$192,070	Combined INS and VORTAC-based RNAV; compatible with Global-Wulfsberg NDB-2; meets requirements for enroute, terminal and approach; transverse-mounted; standby battery optional; bulk storage with five-line alphanumeric LED display; Arinc 561; 1 year warranty
	LTN-91	Laser IRS / sensor	N/A / N/A	Arinc 429: Pos; attitude; heading; angular rates; acceler-ations; drift angle; VS; GS; flight path angle, wind track	Arinc 429 air data	115 VAC, 400 Hz / 28 VDC back-up / —	2/43.0	$196,030	Optional control display unit; primary inertial sensor for navigation management computer systems; complies with Arinc 704 spec for IRS; growth capability to ADIRS; 10 MCU size; additional analog outputs; 3 year warranty
Northstar Avionics 30 Sudbury Rd. Acton, MA 01720 (617) 897-6600	M1	Loran-C / LED	250 / non-volatile RAM	analog	rotary knob	10 to 30 VDC / EPROM	2/4.5	$3,995	Great ergonomics; VFR-only; no TSO; 20,000+ waypoint database; update with 8 EPROM chips — $75, dealer replaceable; first update included in purchase price; all 6,500 US public airports, 1,150 VORs, 1,600 NDBs and 4,800 low-altitude intersections; 20 waypoint flight plan; 3 year warranty
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	LRN 840	Loran-C / LCD	200 / non-volatile RAM	External CDI / autopilot	RS-232	11 to 32 VDC / RAM with five yr. battery	2/4.0	$1,050	Automatic selection of GRI; secondaries MAG VAR; database available with airports, VORs, other fixes; database update cartridge — $75, plugs in back of unit
STS Avionic Products, Inc. 11600 Lilburn Park Road St. Louis, MO 63146 (314) 567-0304	120MC	Loran-C / LCD	120 / non-volatile RAM	analog, CDI	encoded altitude, heading, air data	10 to 32 VDC / —	2/5.0	$1,995	Cross-chain, master-independent operation; concentric knob controls; 15,000 waypoint database; database updates with unit installed; includes antenna, coupler and mounting hardware; new super-twist LCD; 3 year warranty

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUTS / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Terra Corp. 3520 Pan American Freeway NE Albuquerque, NM 87107 (505) 884-2321	TLC 120 none	Loran-C gas discharge	120 none	CDI autopilot	rotary knob	10 to 40 VDC —	2/2.5	$1,595	2.0-in high by 6.25-in wide by 11.0-in long; panel-mounted; includes whip antenna with CT6 preamp
Tracor, Inc. Avionics Products 6500 Tracor Lane MD 27-23 Austin, TX 78725 (512) 929-2233 Exclusive Distributor: DAC International Inc. 6702 McNeal Dr. Austin, TX 78729 (512) 331-5323	TA-7580	Omega/ VLF color CRT	2,480 30,000 waypoints optional	Pos; track error; TTW; GS; HSI; winds; CDI; autopilot roll steering; radar; cabin display	heading; TAS	21.5 VDC to 36 VDC —	3/17.0	$29,583*	*base price; 2,480 pilot-defined waypoints, 170 flight plans with up to 96 waypoints each, alphanumeric identifiers; optional outputs: analog — HSI, autopilot, synchro and roll steering; digital — Arinc 429, 561; EFIS compatible; Jeppesen world-wide database optional — $11,275; TA-2000 3½" disk database loader — $3,360; TA-7900 large CDU $6,797, exchange; includes E-field antenna and installation kit; optional DME/DME nav, GPS
	TA-7900	Omega/VLF color CRT	2,480 30,000 waypoints optional	Pos; track error TTW; GS; HSI; winds; CDI; autopilot roll steering; radar; cabin display	heading; TAS	21.5 VDC to 36 VDC or 115 AC, 400 Hz —	3/23.0	$44,973*	*base price; Arinc 599 spec'n Omega/VLF; price includes ADF coupler; 2,480 pilot-defined waypoints; 170 flight plans with up to 96 waypoints each, alphanumeric identifiers; standard outputs: analog — HSI, autopilot, synchro and roll steering; digital — Arinc 419, 429, 561; EFIS compatible; 60,000 waypoint Jeppesen database optional — $11,275; TA-2000 3½" disk database loader — $3,360
II Morrow, Inc. P.O. Box 13549 Salem, OR 97309 (503) 581-8101	604	Loran-C LED	200 —	CDI, HSI, autopilot, RS-232	concentric knobs, push buttons	6.5-46 VDC RAM with five-year battery	1/3.68	$1,795	VFR only; TSO C60a; panel-mounted; 6.25W x 2.0H; 350 KTAS A-16 antenna with preamp included; manual magnetic variation input; world-wide coverage — 14 station chains; manual secondary selection; upgradable to 604TCA configuration; includes installation kit; 26-month warranty

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUTS / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Morrow, Inc. P.O. Box 13549 Salem, OR 97309 (503) 581-8101	612B	Loran-C LEC	100 EPROM	CDI, HSI, autopilot RS-232	concentric knobs, push buttons	6.5-48 VDC RAM with five-year battery	1/3.75	$2,695	VFR only; 6.25in. wide by 2.0-in high; panel-mounted; includes A-16 antenna with preamp; TSO C60a; standard database for 48 United States and Alaska; optional database includes 620 Canadian airports; dealer-installed EPROM database updates — $75; alphanumic sort; offset waypoint mode; non-volatile two-leg flight plan; couples to Shadin fuel management system, Glatzer ELT; VNAV option; Model 612C also available in 3 ATI package — $2,995; includes installation kit; 26-month warranty
	604TCAsp	Loran-C LED	100 cartridge	CDI, HSI, autopilot, external annunciators, RS-232	concentric knobs, push buttons	6.5-48 VDC RAM with five-year battery	1/3.75	$2,995	VFR only; 6.25-in wide by 2.0-in high; includes A-16 antenna with preamp; all 19 Loran-C chains; database cartridge user-installed in rear panel — includes 7,000 airports, VORs, VORTACs and NDBs in US, Canada and Alaska; update cartridges available every 56 days, each — $75, annual subscription — $200; couples to Shadin fuel management system, Glatzer ELT; 26-month warranty
	618	Loran-C LED	100 EPROM	CDI, HSI autopilot, external annunciators RS-232	concentric knobs, push buttons	6.5-48 VDC RAM with five-year battery	2/3.75	$3,295	VFR only; 6.25W x 2.0H; panel-mounted; A-16 antenna with preamp included; 20,000+ WP database for lower-48 US; updates — $75; auto secondary search; emergency airport search; couples to Shadin fuel management system, Glatzer ELT; comprehensive special use airspace intrusion feature; 26-month warranty
	612	Loran-C LED	100 EPROM	CDI, HSI autopilot, external annunciators RS-232	concentric knobs, push buttons	6.5-48 VDC RAM with five-year battery	2/3.75	$3,295	TSO C60a; 6.25W x 2.0H; panel-mounted; A-16 antenna with preamp included; IFR — AC 20-121; 2,600 WP database for lower-48 US; updates — $75; auto secondary search; emergency airport search; couples to Shadin fuel management system, Glatzer ELT; NPA software available; 26-month warranty

LONG-RANGE NAVIGATION SYSTEMS

MANUFACTURER	MODEL	SYSTEM TYPE / DISPLAY	NUMBER WAYPOINTS / DATABASE	OUTPUTS	INPUTS	POWER INPUTS / MEMORY	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Morrow, Inc. P.O. Box 13549 Salem, Or. 97309 (503) 581-5101	614R	Loran-C LED	100 EPROM	CDI, HSI, autopilot RS-232	N/A	6.5-48 VDC RAM with five-year battery	2/4.75	$6,995	IFR — AC 20-121; IFR demo approach certification; remote, standard Dzus rail mount for CDU; includes 521 KTAS-rated, A-17 antenna with preamp; TSO C60a; 0 — 50,000' MSL; VNAV standard; all other specifications same as 612IFR

VHF NAVCOMS

MANUFACTURER	MODEL / TSO	DISPLAY / MEMORY / CHANNELS STORABLE	VOR OUTPUT	POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb) / SIZE	PRICE (uninstalled)	REMARKS
Aire-Sciences, Inc. 216 Passaic Ave. Fairfield, NJ 07006 (201) 228-1880	RT-563A none	mechanical none none	CDI/HSI	6	14 VDC*	1/5.5 6.5 x 3.25 x 10.9	$2,175	Built-in converter-indicator; output for remote converter-indicator; built-in intercom; *28 to 14 VDC adapter, $145
	RT-563A none	mechanical none none	CDI/HSI	5	14 VDC*	1/7.8 6.5 x 3.25 x 10.9	$2,995	Automatic omni; built-in converter-indicator; output for remote converter-indicator optional *28 to 14 VDC adapter, $145
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KX 155 C34c C36c C37b C38b C40a	gas discharge EAROM non-volatile 2	CDI/RMI	10	14 or 28 VDC	2/7.0 6.25 x 2.5 x 10.25	$3,240	Auto squelch; includes KI 209 VOR/ LOC/ GS converter-indicator; price includes built-in 40-channel GS receiver; optional audio amplifier; 760 channel;
	KX 165 C34c C36c C37b C38b C40a	gas discharge EAROM non-volatile 2	CDI/RMI HSI	10	14 or 28 VDC	2/7.0 6.25 x 2.0 x 10.25	$7,225	Auto squelch; includes KI 208 VOR/LOC/ GS indicator; price includes built-in VOR converter and GS receiver; price less GS with indicator $6,360; digital radial readout
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	Escort IIA none	gas discharge none none	none	6	14 VDC,	1/3.0 3.25 x 3.25 x 10.75	$1,149	Auto omni, digital OBS, ECDI, and digital bearing; 28 VDC configuration — $1,229

Footnote: All prices and weights, unless noted in Remarks, include a nav indicator.

VHF NAVCOMS

MANUFACTURER	MODEL TSO	DISPLAY MEMORY CHANNELS STORABLE	VOR OUTPUT	POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	MK-12D C34c C37b C38a C40a	gas discharge hot-wire non-volatile 2/2	CDI/HSI	8	14 or 28 VDC	2/4.4 6.25 x 2.5 x 11.0	$3,568	Includes GS receiver and separate ID-825 VOR/LOC/GS indicator; $2,965 without GS and with ID-824 VOR/LOC indicator; includes 2x5 DME channeling; 8-watt transmitter; 720 channels; Cessna ARC form factor, direct-replacement design — $1,796
	MK-12E C34c C37b C38b C40a	gas discharge hot-wire non-volatile 2/2	CDI/HSI	8	14 or 28 VDC	2/5.4 6.25 x 2.5 x 11.0	$3,955	Includes GS receiver and separate ID-825 VOR/LOC/GS indicator; $3,343 without GS and with ID-824 VOR/LOC indicator; includes 2x5 DME channeling; 8-watt transmitter; 760 channels
	NCS-812 C34c C37b C38b C40a	gas discharge hot-wire non-volatile 2/2	CDI/HSI	8	14 or 28 VDC	2/5.4 6.25 x 2.5 x 11.0	$5,510	Includes separate 25-watt IDME-895 DME/VOR/LOC/GS indicator and GS receiver; 8-watt VHF comm transmitter; 760 channels
Terra Corp. 3520 Pan American Freeway NE Albuquerque, NM 87107 (505) 884-2321	TXN 920 none	mechanical/ ECDI none none	CDI	6	14 VDC*	2/2.5 5.25 x 1.5 x 10.0	$1,795	Pushbutton tuning; manual or auto squelch; with GS — $2,195; 2x5 DME channeling; *28 to 14 VDC adapter — $85
	TXN 960 none	mechanical/ ECDI none none	built-in	6	14 VDC*	1/3.7 6.25 x 3.3 x 10.0	$1,795	Pushbutton tuning; auto squelch, manual override; with GS — $2,195; 2x5 DME channeling; *28 to 14 VDC adapter — $85

Footnote: All prices and weights, unless noted in Remarks, include a nav indicator.

VHF PANEL-MOUNTED NAV RECEIVERS

MANUFACTURER	MODEL TSO	DISPLAY MEMORY CHANNELS STORABLE	INCLUDES GS MB	POWER INPUT VOR OUTPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Aire-Sciences, Inc. 216 Passaic Ave. Fairfield, NJ 07006 (201) 228-1880	R-662 C38c C40a	mechanical none none	no no	14 or 28 VDC DC to CDI	2/4.9 3.25 x 2.62 x 11.4	$945	Interfaces with ECDI-541 converter-indicator — $895; glideslope and 2x5 DME channeling optional
	R-552 none	mechanical none none	no no	14 or 28 VDC DC to CDI	2/5.0 3.2 x 2.62 x 11.44	$1,745	Includes converter-indicator; glideslope and 2x5 DME channeling optional; includes ECDI-554A display

VHF PANEL-MOUNTED NAV RECEIVERS

MANUFACTURER	MODEL TSO	DISPLAY MEMORY CHANNELS STORABLE	INCLUDES GS MB	POWER INPUT VOR OUTPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Aire-Sciences, Inc. 216 Passaic Ave. Fairfield, N.J. 07006 (201) 228-1880	R-554 none	mechanical none none	yes no	14 or 28 VDC DC to CDI	2/5.4 3.25 x 2.62 x 11.44	$2,620	Includes indicator; 2x5 DME channeling
	R-564 C34c C36b C40a	mechanical none none	yes no	14 or 28 VDC DC to CDI	2/5.4 3.25 x 2.62 x 11.44	$2,665	Interface with OBS indicator; 2x5 DME channeling
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KN 53 C34c C36c C40a	gas discharge EAROM non-volatile 2	no no	11 to 33 VDC CDI*	2/4.7 6.25 x 1.3 x 10.5	$4,250	Displays both active and preset frequency; price includes *KI 203 indicator; $5,100 with GS and *KI 204 indicator
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	VIR-351 C36c C40a	incandescent filament none none	no no	14 VDC* OBS resolver DRZ'd at 300° 0.05 Vrms to RMAV	2/4.2 3.12 x 2.6 x 12.45	$4,925	Displays either frequency or to/from bearing; electronic frequency readout; includes IND-350A indicator; includes glideslope and DME channeling; Arinc 547 *28 to 14 VDC adapter — $120
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	NAV-824 C36c C40a	gas discharge hot-wire non-volatile 2	no no	11 to 32 VDC Arinc: .5 Vrms at 0°	1/3.3 6.25 x 1.5 x 11.0	$2,132	RMI function; ID 824 indicator included in price; 2x5 DME channeling
	NAV-825 C34c C36c C40a	gas discharge hot-wire non-volatile 2	yes no	11 to 32 VDC N/A	1/3.3 6.25 x 1.5 x 11.0	$2,525	RMI function; ID 825 indicator included in price; 2x5 DME channeling
Terra Corp. 3520 Pan American Freeway, NE Albuquerque, NM 87107 (505) 884-2321	TN 200 none	mechanical none none	optional no	14 VDC* N/A	1/1.5 3.2 x 1.8 x 9.0	$695	2x5 DME channeling; with GS option $995; *28 to 14 VDC adapter — $85

VHF REMOTE-MOUNTED NAV RECEIVERS

MANUFACTURER	MODEL TSO	DISPLAY MEMORY CHANNELS STORABLE	INCLUDES GS MB	POWER INPUT VOR OUTPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-6400	VNS-41 C40b C36c C34c C35d	LCD or dichroic non-volatile 2	yes yes	28 VDC analog, Arinc 429	2/5.5 4.0 x 4.0 x 12.62	$12,695	Digital RMI readout, analog or Arinc 429 interface; includes LCD control head; two-year warranty
	KNR 634 C40a C36c C34c C35d	gas discharge EAROM non-volatile 2*	yes yes	28 VDC CDI, RMI, HSI	2/5.6 2.0 x 5.0 x 10.0	$13,310	Includes GS and MB receivers; self-contained converter; price includes KFS 584 with simultaneous digital display of active and standby frequency; Gables control option; King Serial Data tuning; also available without GS or MB; requires 26 VAC for sine/cosine output; night vision compatible "A" control head available; *optional control head with 9 programmable frequencies
	KNR 634A C40a C36c C34c C35d	gas discharge EAROM non-volatile 2*	yes yes	28 VDC Arinc 429 CDI, RMI, HSI	2/6.5 3.0 x 5.0 x 10.0	$17,465	synchro-interface RMI optional; price includes GS and MB receivers, control head; night vision compatible "A" control head available; *optional control head with 9 programmable channels; two-year warranty
	KTU 709 C68a	252 gas discharge EAROM non-volatile none	no no	28 VDC CDI, RMI	2/6.5 3.0 x 5.0 x 10.0	$18,425	Tacan: includes DME; bearing information derived from DME signal; price includes KDI 572 DME indicator; King serial data tuning; control head optional; night vision compatible "A" control head available; two-year warranty
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	VIR-32 C34c C35d C36d C40a	gas discharge non-volatile 6	yes yes	28 VDC, 26 VAC, 400 Hz sine/cosine serial digital	2/5.7 ¾ ATR-SL	$13,390	RMI compatible; replaces VIR-30A directly; includes CTL-32 serial digital control; digital signal processing; built-in diagnostics; rotor modulation filter optional (strap)
	VIR-432	gas discharge	yes	28 VDC, 26 VAC, 400 Hz	2/5.7	$15,055	Built-in diagnostics; compatible only with CSDB or Arinc 429 controls; includes CTL-32 CSDB control; digital signal processing; auto instrumentation for use with RMI; rotor modulation filter optional (strap)
	51RV-4 C34c C36c C40a	mech. synchro none none	yes no	28 VDC, 26 VAC, 400 Hz electronic synchro driver	1/8.9 ½ ATR-S	$17,184	Self-contained converter; RMI capability; RNAV compatibility; options for glideslope receiver and monitoring of any or all functions; digital circuitry, output and tuning; coherent detector in receiver; Arinc 547; 2x5 DME channeling

VHF REMOTE-MOUNTED NAV RECEIVERS

MANUFACTURER	MODEL TSO	DISPLAY MEMORY CHANNELS STORABLE	INCLUDES GS MB	POWER INPUT VOR OUTPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
	VIR-31A C34c C35d C36c C40a	gas discharge non-volatile 2	yes yes	28 VDC; 26 VAC, 400 Hz three-wire synchro	2/8.0 ¼ ATR-SL	$18,425	RMI compatibility; includes CTL-30 electronic control; 2x5 DME channeling; three-wire synchro VOR output; rotor modulation filtering optional
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4065	VIR-30A C34c C35d C36c C40a	gas discharge nonvolatile 2	yes yes	28 VDC; 26 VAC, 400 Hz sine/cosine	2/6.8 ¼ ATR-SL	$18,840	RMI compatibility; superflag output; includes CTL-30 electronic control; 2x5 DME channeling; four-wire sine-cosine VOR output
Honeywell Inc. P. O. Box 29000 Phoenix, AZ 85038 (602) 863-8000	RNZ-850 C34d C35d C36d C40b C41d C66b	Color CRT non-volatile RAM 1 to 12	yes yes	28 VDC, 26 VAC 400 Hz serial digital, analog	2/—	—	Nav subsystem of SRZ-850 Primus II integrated, digital radio system; full dual system price — $128,427; includes RM-850 color CRT radio management unit, digital audio system, NV-850 VHF nav receiver, DF-850 ADF with AT-860 combined loop sense antenna, DM-850 precision format compatible DME (+/– 100-ft) with station ID display and 6 channel scanning; built-in diagnostics, maintenance log; 1 to 12 stored channels depending on function; self-contained converter; digital signal processing; RTCA DO-160B environmental, DO-178A software

VHF PANEL-MOUNTED TRANSCEIVERS

MANUFACTURER	MODEL TSO	DISPLAY	MEMORY CHANNELS STORABLE	NOMINAL POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Aire-Sciences, Inc. 216 Passaic Ave. Fairfield, NJ 07006 (201) 228-1880	RT-551A none	mechanical	none none	6	14 VDC*	1/2.7 3.25 x 2.62 x 10.88	$2,050	Manual squelch; portable version available *28 to 14 VDC adapter, $145
	RT-661A C37b C38b	mechanical	N/A N/A	6	14 or 28 VDC*	1/2.7 3.25 x 2.82 x 11.44	$2,250	Manual squelch; portable version available *Adapter, $145

VHF PANEL-MOUNTED TRANSCEIVERS

MANUFACTURER	MODEL T30	DISPLAY	MEMORY CHANNELS STORABLE	NOMINAL POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (unit-installed)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KY 96A C37c C38c	LCD	EEPROM 9	7	28 VDC	1/2.9 6.25 x 1.3 x 10.5	$1,140	Auto leveling; auto squelch; stuck mike protection; remote mike keying and channel call-up; 760 channel
	KY 97A C37c C38c	LCD	EEPROM 9	7	14 VDC	1/2.9 6.25 x 1.3 x 10.5	$1,140	Auto leveling; auto squelch; stuck mike protection; remote mike keying and channel call-up; 760 channel
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KY 92 C37b C38b	mechanical	none none	7	14 VDC*	1/3.2 6.25 x 1.5 x 10.5	$2,160	Auto squelch; self-test *28 to 14 VDC adapter, $300; 760 channel
	KY 196A C37b C38b	gas discharge	EAROM non-volatile 9	16	28 VDC	1/3.2 6.25 x 1.3 x 10.5	$3,060	Auto squelch; displays active and standby frequency; remote flip-flop; optional remote call-up; 760 channel; optional extended frequency range available
	KY 197A C37b C38b	gas discharge	EAROM non-volatile 9	7	14 VDC	1/3.2 6.25 x 1.3 x 10.5	$3,060	Same as KY 196A, but 14 VDC power input
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	VHF-253 C37b Class II; C38b	LCD	EAROM non-volatile 6	10	10 to 32 VDC	1/2.7 6.3 x 1.3 x 12.1	$3,910	Remote frequency management; visible active and standby frequencies plus four stored presets; liquid-crystal displays; optional broadband version (VHF-253S) $4,440
General Aviation Electronics, Inc. 8284 E. 250th Hampton, MN 55031 (507) 263-4002	Alpha 720 none	incandescent filament	none none	5	14 VDC	1/4.0 5.5 x 2.5 x 10.0	$1,000	Vehicle- or ground-station versions available; 25kHz spacing; 10-watt carrier option — $200
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	COM-810/ 811 C38b Class C	gas discharge	volatile (.2mA holding) 2	8	14 or 28 VDC	1/3.6 6.25 x 1.5 x 11.0	$1,295	Auto squelch; audio leveling; speaker, mike; 10-watt amp included
Terra Corp. 3520 Pan Am Freeway NE Albuquerque, NM 87107 (505) 884-2321	TX 720 none	mechanical	none none	5	14 VDC*	1/1.2 3.2 x 1.5 x 9.0	$945	Auto squelch; lighted; intercom; portable, vehicle or ground station version available *28 to 14 VDC adapter — $85

VHF REMOTE-MOUNTED TRANSCEIVERS

MANUFACTURER	MODEL TSO	DISPLAY	MEMORY CHANNELS STORABLE	POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KTR 908 C37b C38b	gas discharge	EAROM 2*	20	28 VDC	2/4.3 1.8 x 5.0 x 10.0	$8,310	Includes KFS 598 control with digital display of active and standby frequencies; Gables control option; remote flip-flop *optional control head with 9 programmable frequencies and remote frequency call-up function; 760 channel; 118.0 - 151.975 MHz option; night vision compatible "A" control head available; 2-year warranty
	VCS-40 C37c C38c	LCD or dichroic	non-volatile 2	20	28 VDC	2/5.5 4.0 x 4.0 x 12.62	$10,775	Arinc 429 or analog interface; continuous transmit capable; SELCAL and ACARS compatible; 760 channel; 118.0 - 151.975 MHz option; 2-year warranty
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	VHF-21A C37c C38c	gas discharge	non-volatile 8	20	28 VDC	2/5.8 ¾ ATR-SL	$8,900	Directly replaces VHF-20A; built-in diagnostics; 2x5 serial tuning; includes CTL-22 serial tuning control; VHF-21B, 118.0 - 151.975 MHz — $11,635
	VHF-22A C37c C38c	gas discharge	non-volatile 8	20	28 VDC	2/5.8 ¾ ATR-SL	$8,900	Built-in diagnostics; 2x5 serial digital tuning — includes CTL-22 serial tuning control; VHF-22B, 118.0-151.975MHz — $11,835
	618M-3 C37b C38b	none	none none	25	28 VDC	1/10.1 ½ ATR-S	$8,748	Arinc 566A
	VHF-422A C37c C38c	gas discharge	non-volatile 8	20	28 VDC	2/5.8 ¾ ATR-SL	$9,895	Built-in diagnostics; compatible only with CSDB or Arinc 429 controls; includes CTL-22 CSDB control; VHF-422B, 118-151.975 MHz — $12,795
	VHF-20A C37b C38b	gas discharge	non-volatile 2	20	28 VDC	2/6.8 ¾ ATR-SL	$10,705	CTL-20 electronic control included; VHF-20B, 118 - 151.975 MHz — $13,635
Global-Wulfsberg Systems 2144 Michelson Dr. Irvine, CA 92715 (714) 851-0119	WT-200B C37b C38b	mechanical	none none	20	28 VDC	1/5.6 2.9 x 5.2 x 13.28	$7,962	Arinc and VHF 20A conversion mount available; 1,360 channels at 25 KHz spacing from 118.000 to 151.975 MHz; narrow or wide IF bandwidth models available; all solid state

Footnote: Arinc form factor is shown in the Units/Weight/Size columns. (See Form Factor explanation at the front of the Avionics section.)

VHF REMOTE-MOUNTED TRANSCEIVERS

MANUFACTURER	MODEL TSO	DISPLAY	MEMORY STORABLE CHANNELS	POWER OUTPUT (watts)	VOLTS INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (installed)	REMARKS
Honeywell Inc. P.O. Box 29000 Phoenix, AZ 85038 (602) 863-8800	RCZ-850 C37c C38c C112	color CRT	non-volatile RAM 2 to 12	20, 500*	28 VDC	2/— 14.0 x 6.6 x 3.36	—	Comm subsystem of SRZ-850 Primus II integrated, digital radio system; full dual system price — $128,427; includes RM-850 color CRT radio management unit, 20-channel AV-850 or -851 digital audio system, TR-850 VHF transceiver with 118.0 - 151.975MHz range, XS-850 500-Watt Mode S transponder; built-in diagnostics, maintenance log; 2 to 12 stored channels depending on function; digital signal processing; RTCA DO-160B environmental, DO-178A software

Footnote: Arinc form factor is shown in the Units/Weight/Size columns. (See Form Factor explanation at the front of the Avionics section.)

HF TRANSCEIVERS

MANUFACTURER	MODEL TSO	CHANNELS FREQ CONTROL	FREQ. RANGE (MHz)	POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb)	PRICE COUPLER	REMARKS
Aircom Electronics, Ltd 9501 Ryan Ave. Dorval, Quebec H9P 1A2 Canada (514) 636-3679	SSB 10/20 none	10 crystal	2-29.999 SSB/AM	100 PEP	28 VDC	1/7.3	$10,080* $1,344*	*$12,000 Canadian, $1,600 Canadian; remote-, panel- and direct- (side) mount configurations available; 14 - 28 VDC remote coupler, fully compatible with other HF radios;
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KHF 950 C31c C32c	280,000 synthesizer	2-29.9999 SSB/AM	150 PEP (SSB) 35 (AM)	28 VDC	3/20.2	$16,860 —	Automatic antenna coupler included; 99 operator programmable, 178 ITU radiotelephone channels with standard, KCU 951 control head; optional, space saving KFS 594 control head provides 19 operator programmable, 178 ITU radiotelephone channels; optional dual HF ghost antenna; optional SELCAL compatibility; optional speech processing; explosion proof; two-year warranty
	KHF 990 C31c C32c	280,000 synthesizer	2-29.9999 SSB/AM	150 PEP (SSB) 35 (AM)	28 VDC	3/22.5	$19,270 —	Designed for helicopter use; includes digital antenna coupler/ probe antenna in single unit; 19 preset, 178 ITU radiotelephone channels with standard KFS 594 control head; explosion proof, salt-spray proof; two-year warranty

HF TRANSCEIVERS

MANUFACTURER	MODEL TSO	CHANNELS FREQ. CONTROL	FREQ. RANGE (MHz)	POWER OUTPUT (watts)	POWER INPUT	UNITS/ WEIGHT (lb)	PRICE COUPLER	REMARKS
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	HF-230 C31c C32c	280,000 synthesizer	2-29.9999 SSB/AM AM	100 PEP (SSB)	28 VDC	4/24.4	$16,640 —	Half-duplex; compatible with maritime radiotelephone network; additional channels: 40 operator programmable, 176 ITU radiotelephone preprogrammed channels; optional SELCAL; PAC-230 antenna coupler for helicopters, $17,710 system price; weight 23.2 lbs.
	HF-9000 C31d C32d	280,000 synthesizer	2-29.9999 SSB/AM AM data	175 PEP,	28 VDC	3/27.5	$28,200 —	Fiber-optic interface; 99 operator programmable, 176 ITU radiotelephone programmed channels; selectable power output — 10, 50, 175-Watt PEP; rapid-tune antenna coupler (40 msec computer tuning); BITE
	626T-3A C31c C32c	280,000 synthesizer	2-29.9999 SSB/ AM/CW	400 PEP (SSB) 125 (AM) 25 (CW)	28 VDC; 115 VAC, 400 Hz	3/26.0	$30,336 —	Panel-mounted control; remote-mounted ½ ATR-S transceiver and antenna coupler; power amplifier coupler; shock mounts included; frequencies automatically tuned in 0.1-kHz increments; Arinc 533, 404; Varies with installation
Sunair Electronics 3101 S.W. 3rd Ave. Fort Lauderdale, FL 33315 (305) 525-1505	ASB-500 C31c C32c	32,000 synthesizer	2-17.999 SSB/AM	100 PEP (SSB)	28 VDC	2/36.6	$9,495* —	*Factory dealer price; panel-mounted LED control unit; remote-mounted transceiver; SELCAL compatibility; includes automatic, full HF range ACU-150D microprocessor controlled antenna coupler
	ASB-850 C31c C32c	264,000 synthesizer	2-29.9999 SSB/AM	100 PEP (SSB)	28 VDC	3/30.0	$11,850 —	Integrated package including SCU-90 control-display unit, RE-800 remote-mounted HF transceiver and ACU-810 microprocessor-controlled antenna coupler; SELCAL compatibility

DISTANCE MEASURING EQUIPMENT

MANUFACTURER	MODEL TSO	CHANNELS OUTPUT	DISPLAY	QUANTITIES DISPLAYED	PEAK POWER OUTPUT (watts)	VOLTS INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KN 64 none	200 King serial data	gas discharge	dist.: 0-399 nm gnd. spd.: 0-999 kt TTS: 0-99 min.	35	11 to 33 VDC	1/2.4 6.25 x 1.3 x 12.2	$2,085	Distance accuracy: ±0.1 nm nominal to 99 nm, ±1.0 nm 100 to 200 nm; panel-mounted; includes antenna and installation kit; accepts remote nav channeling

Footnote: Arinc form factor is shown in the Units/Weight/Size columns. (See Form Factor explanation at the front of the Avionics section.)

DISTANCE MEASURING EQUIPMENT

MANUFACTURER	MODEL TSO	CHANNELS OUTPUT	DISPLAY	QUANTITIES DISPLAYED	PEAK POWER OUTPUT (watts)	VOLTS INPUT	UNITS/WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KN 62A C66a	200 King serial data	gas discharge	dist.: 0-389 nm gnd. spd.: 0-999 kt TTS: 0-99 min	100	11 to 33 VDC	1/2.4 6.25 x 1.3 x 12.2	$4,515	Distance accuracy: ±0.1 nm nominal to 99 nm, ±1.0 nm 100 to 200 nm; panel-mounted; includes antenna and installation kit; accepts remote channeling
	KN 63 C66a	200 King serial data	gas discharge	dist.: 0-389 nm gnd. spd.: 0-999 kt TTS: 0-99 min	100	11 to 30 VDC	2/3.6 6.5 x 1.9 x 11.55	$6,850	Distance accuracy: ±0.1 nm nominal to 99 nm, ±1.0 nm 100 to 200 nm; includes KDI 572 indicator; optional slaved indicator
	KDM 706 C66a	252 King serial data	gas discharge	dist.: 0-389 nm gnd. spd.: 0-999 kt TTS: 0-99 min	250	28 VDC	2/5.5 2.5 x 5.25 x 11.8	$11,295	Distance accuracy: ±0.1 nm to 99 nm, ± 100 to 200 nm; includes KDI 572 indicator; optional slaved indicators; remotely mounted; 2-year warranty
	KDM 706A C66a	252 Arinc 429, 568	gas discharge	dist.: 0-389 nm gnd. spd.: 0-999 kt TTS: 0-99 min	250	28 VDC	2/6.3 5.0 x 3.0 x 10.0	$13,385	KDM 706 with Arinc 429, 568 interface; KDI 572 included; optional slaved indicators; remotely-mounted; night vision compatible "A" control head available; 2-year warranty
	DMS-44 C66	252 2 x Arinc 568 2 x 40mv/nm Arinc 429 40-bit serial	LCD or dichroic	dist.: 0-300 nm gnd. spd.: 0-999 kt TTS: 0-19:59 min	325	28 VDC	2/6.6 5.0 x 4.0 x 12.62	$15,690	Triple scanning DME, displays Nav 1 and Nav 2 distance simultaneously; Arinc 429 or analog interface; displays RNAV distance; remotely-mounted; 2-year warranty
	KTU 709 C66a	252 King serial	gas discharge	dist.: 0-389 nm gnd. spd.: 0-999 kt TTS: 0-99 min	250	11 to 33 VDC	6.6 5.0 x 3.0 x 10.0	$17,260	Tacan (DME with Tacan-based bearing independent of VOR); optional KFS-579A control head — $4,180; KDI 572 indicator included; night vision compatible "A" control head available; 2-year warranty
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	DME-450 C66a	200 40 mv/nm	incandescent filament	dist.: 0-199.9 nm gnd. spd.: 0-399 kt TTS: 0-120 min	100	14 or 28 VDC	3/6.1 3.5 x 5.0 x 11.0	$7,605	Distance accuracy: ±0.1 nm or 1%; includes ANT-451 antenna and IND-450 indicator
	DME-451 C66a	200 40 mv/nm	incandescent filament	dist.: 0-199.9 nm gnd. spd.: 0-399 kt TTS: 0-120 min	100	14 or 28 VDC	3/6.4 3.5 x 5.0 x 11.0	$9,215	Distance accuracy: ±0.1 nm or 1%; built-in timer based on GMT clock; includes ANT-451 antenna and IND-451 indicator

Footnote: Arinc form factor is shown in the Units/Size/Weight column. (See Form Factor explanation in the front of the Avionics section.)

DISTANCE MEASURING EQUIPMENT

MANUFACTURER	MODEL TSO	CHANNELS OUTPUT	DISPLAY	QUANTITIES DISPLAYED	PEAK POWER OUTPUT (watts)	VOLTS INPUT	UNITS/WEIGHT (lb) SIZE	PRICE (installed)	REMARKS
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	DME-42 C66b	252 40 mv/nm Arinc 568B serial digital	gas discharge	dist.: 0-300 nm gnd. spd.: 0-999 kt TTS: 0-120 min ID: 2,3,4 letters	300	28 VDC	2/6.2 ½ ATR-SL	$12,935	Includes hex-digit-code diagnostics; directly replaces DME-40; tracks three channels simultaneously when used with CTL-32 and IND-42A (CTL-32 controls nav receiver); serial digital, 2 x 5 tuning; digital signal processing; echo monitor; includes IND-42A indicator
	DME-442 C66b	252 40 mv/nm Arinc 568B serial digital	gas discharge	dist.: 0-300 nm gnd. spd.: 0-999 kt TTS: 0-120 min ID: 2,3,4 letters	300	28 VDC	2/6.2 ½ ATR-SL	$14,510	Tracks three channels simultaneously when linked to CTL-32, IND-42A; decodes / displays station ident; compatible interfaces: Collins Serial Data Bus (CSDB) or ARINC 429, only; digital signal processing; echo monitor; built-in diagnostics; includes IND-42A
	DME-40 C66a	252 40 mv/nm Arinc 568 pulse pair	gas discharge	dist.: 0-250 nm gnd. spd.: 0-999 kt TTS: 0-99 min	300	28 VDC	2/8.1 ½ ATR-SL	$15,950	Distance accuracy: ±0.1 nm or 0.01%; includes IND-40 indicator
	860E-5 C66a	252 Arinc 568B	N/A	dist.: 0-390 nm	700	115 VAC, 400 Hz	1/16.0 ½ ATR-SL	$26,016	Distance accuracy: ±0.1 nm nominal; indicator not included
	860E-4 C66a	252 Arinc 521D	N/A	dist.: 0-390 nm	700	115 VAC, 400 Hz	1/20.0 ½ ATR	$30,080	Distance accuracy: ±0.2 nm; channeling and lock-on less than 2.5 seconds; indicator not included
Foster Airdata Systems, Inc. 7020 Huntley Rd. Columbus, OH 43229 (614) 888-9502	DME 670 C66b	252 RS-422, Arinc 568, DME pulse pair, 40 mv/nm, 400 HzAC	LED dot matrix	dist.: 0-400 nm gnd. spd.: 0-720 kt TTS: 0-99 min. bearing: (Tacan) 0-359° ID: 3, 4 letter Morse	250	28 VDC	2/6.8 10.25 x 3.25 x 5.0 1/2-3ATI x 6.5	$11,900	Tracks up to four stations; displays distance and ident or Tacan radial from two stations; distance, ident, Tacan radial and speed from single station; also available with ARINC 429 output for interface with long-range navigation systems
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	DME 890 none	200 Narco digital	gas discharge	dist.: 0-160 nm gnd. spd.: 44-600 kt TTS: 0-99 min	25	11 to 32 VDC	1/3.3 6.25 x 1.5 x 11.0	$1,949	Microprocessor controlled; meets all minimum operating characteristics; Narco RNAV 860, NS-801 compatible
	IDME 891 none	200 Narco digital	LED	dist: 0-160 nm gnd. spd.: 44-600 kt	25	11 to 32 VDC	1/26 three-inch cutout	$2,639	Requires MK 120; package with GS, $3,650; includes VOR/LOC indicator and marker lights; compatible with Mark 12D, NAV 825, RNAV 860, NS-801

Footnote: Arinc form factor is shown in the Units/Weight/Size column. (See Form Factor explanation in the front of the Avionics section.)

TRANSPONDERS

MANUFACTURER	MODEL TSO	INTERRO-GATION MODES	POWER OUTPUT (watts)	VOLTS INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Aire-Sciences, Inc. 216 Passaic Ave. Fairfield, NJ 07006 (201) 228-1880	RT-787 C74c Class 1A	A,C	500	14 VDC*	2/3.4 8.5 x 1.8 x 9.9	$1,495	Manual reply-light dimmer *28 to 14 VDC adapter, $95
	RT-887 C74c Class 1A	A,C	500	14 VDC*	2/4.0 3.25 x 3.25 x 10.12	$1,795	Manual reply-light dimmer; electronic digit display; mounts into 3.25-inch instrument cutout *28 to 14 VDC adapter, $95
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KT 76A C74c Class 1A	A,C	250	14 or 28 VDC*	1/3.1 6.25 x 1.6 x 10.0	$1,225	Automatic reply-light dimmer; system test; remote ident capability *Adapter, $30
	KT 79 C74c Class 1A	A,C	250	11 to 33 VDC	1/3.4 6.25 x 1.3 x 10.5	$3,315	Auto 1200 select; displays encoded altitude; all solid-state, no cavity tube; remote ident capability
	KXP 756 C74c Class 1A	A,C	250	28 VDC	2/4.7 5.0 x 1.75 x 10.0	$6,975	Includes KFS 578 digital control head with auto 1200 select; test displays encoded altitude; solid-state; no cavity tube; Gables control optional; remote ident capability; night vision compatible "A" control head available; 2-year warranty
	TRS-42 C74c	A,B,C	325	28 VDC	2/5.8 4.0 x 4.0 x 12.62	$10,125	Analog or Arinc 429; controls two RT's; auto 1200 select; digital Mode C altitude readout; 2-year warranty
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	TDR-850 C74c Class 1A	A,C	250	14 VDC*	2/2.0 5.3 x 1.6 x 8.2	$2,010	Manual reply-light dimmer; optional remote ident *28 to 14 VDC adapter, $125
	TDR-90 C74b C74c	A,C	500	28 VDC	2/4.9 ½ ATR-SL	$8,045	Remote-mounted; manual reply-light dimmer; includes CTL-90 electronic control; optional CTL-92 with CAD-92 adapter provides serial digital tuning, two stored codes
	621A-5A C74a	A,B,C (D opt.)	700	115 VAC, 400 Hz	3/15.0 ¾ ATR-S	$17,808	Remote-mounted; must add control head with reply-light; Arinc 572
	TPR-720 C74c C112	A,C,S	250-631	115 VAC, 400 Hz	1/12.0 ½ ATR	$19,348	Mode S transponder; 4 MCU package (approx. ½ ATR); LCD CDU — 5.0 lbs

ENCODING ALTIMETERS/DIGITIZERS

MANUFACTURER	MODEL TSO	INTERRO-GATION MODES	POWER OUTPUT (watts)	VOLTS INPUT	UNITS/ WEIGHT (lb) SIZE	PRICE (uninstalled)	REMARKS
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	AT-150 C74C Class 1A	A, C	250	14 or 28 VDC*	1/2.3 6.25 x 1.75 x 11.3	$1,249	Manual reply-light dimmer; remote ident available *Adapter, $27
Terra Corp. 3520 Pan American Freeway, NE Albuquerque, NM 87107 (505) 884-2321	TRT 250 C74C Class 2a	A,C	200	10 to 30 VDC	1/1.7 3.2 x 1.6 x 10.0	$1,195	All solid-state; no cavity tube; remote ident capability; push-button code selection; 3" ATI configuration available

ENCODING ALTIMETERS/DIGITIZERS

MANUFACTURER	MODEL TYPE	RANGE (x 100 ft)	DISPLAY CASE	LENGTH WEIGHT	DIGITIZER DESIGN	POWER INPUT	PRICE (uninstalled)	REMARKS
ACK Technologies 424 W. Julian St. San Jose, CA 95110 (408) 287-8021	A-30 altitude digitizer	−10+307	— —	6.0 .45	solid state	10 to 32 VDC	$325	TSO C88a; remotely mounted
Aero Mechanism 20327 Nordhoff St. Chatsworth, CA 91311-5161 (818) 709-2851	61410-35LW —	−10+350	CDP 3" ATI	6.26 .75	optical	14 or 28 VDC	$4,743	Millibar baro setting optional; mating connector included; internal lighting: 5, 14, 28 VDC available; TSO C10b, C88
Aerosonic Corp. P.O. Box 4627 Clearwater, FL 33515 (813) 461-3000	102200 -01812 —	−10+200	3P 3 ATI	5.3 2.0	optical	14 or 28 VDC	$1,391	Millibar baro setting, dual scale, internal lighting and altitude alerter optional; TSO C10b, C88
	102200 -01818 —	−10+350	3P 3 ATI	5.1 2.0	optical	14 or 28 VDC	$1,470	Millibar baro setting, dual scale, internal lighting and altitude alerter optional; TSO C10b, C88
	102200 -01831 —	−10+350	CDP 3 ATI	5.4 2.4	optical	14 or 28 VDC	$2,531	Millibar baro setting, dual scale, internal lighting and altitude alerter optional; TSO C10b, C88
	101400- 01235 —	−10+500	CDP 3 ATI	5.2 7.7	optical	14 or 28 VDC	$2,575	Millibar baro setting, dual scale, internal lighting and altitude alerter optional; other configurations available; TSO C10b, C88

ENCODING ALTIMETERS/DIGITIZERS

MANUFACTURER	MODEL TYPE	RANGE (x 100 ft)	DISPLAY CASE	LENGTH WEIGHT	DIGITIZER DESIGN	POWER INPUT	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KE 127 altitude digitizer	−10+200	— —	3.25 1.00	optical	14 or 28 VDC	$1,075	Digitizer only; remotely-mounted;
	KEA 129 encoding altimeter	−10+2000	3P MS	5.57 1.90	optical	14 or 28 VDC	$2,585	Millibar / inches baro setting; kit and rack included; internally lighted
	KEA 130 encoding altimeter	−10+350	3P MS	5.57 1.90	optical	14 or 28 VDC	$2,850	Millibar / inches baro setting; internally lighted; kit and rack included; KEA 130A with altitude pre-select / alerter compatibility $3,035
	KEA 346 servoed encoding altimeter	−10+500	CDP 3 ATI	8.94 2.90	optical	28 VDC	$8,500	Internal lighting; altitude-alert outputs; DH annunciation; dual baro set in millibars or inches
Kollsman Avionics Division 220 Daniel Webster Hwy. Merrimack, NH 03054 (603) 889-2500	B4515 Alti-coder	−10+200	CDP MS	4.75 2.50	optical	28 VDC + 115 VAC	N/A	Price dependent on specific aircraft; internal white or red lighting; three-digit display; 35,000-ft model available
	B4450 Alti-coder II	−10+350	3P MS	5.30 2.50	optical	14 or 28 VDC	N/A	Price dependent on specific aircraft; optional internal lighting and failure warning flag
	24829 encoding altimeter	−10+200	CDP 3 ATI	6.9 2.9	optical	14 or 28 VDC	$14,400*	* base price; red, white or blue lighting; 35,000-ft / 50,000-ft models available
	28007 encoding altimeter	−10+500	CDP 3 ATI	8.3 3.5	contact	26 VAC 400 Hz	$15,504*	* base price;,above options, features; includes altitude alert, altitude rate outputs
	28702 encoding altimeter	−10+350	CDP 3 ATI	7.0 3.5	contact	28 VDC	$16,044*	* base price; above options, features
	29007 encoding altimeter	−10+50	CDP 3 ATI	8.0 3.8	contact	26 VAC 400 Hz	$17,536*	* base price; above options, features; includes radio altimeter (RA) display; links to GPWS; requires separate RA receiver/transmitter
	29702	−10+350	CDP 3 ATI	8.0 3.6	contact	28 VDC	$18,512*	* base price; optional altitude alert light; extended baro range; above options, features
Narco Avionics Inc. P.O. Box 1807 Laguna Beach, CA 92652-1807 (714) 497-5077	AR-850 altitude digitizer	−10+250 −10+307 altitude digitizers	— —	6.15 0.7	solid state	11 to 32 VDC	$260 $300	−10;250 model TSO C88a −10;307 model not TSOed

ENCODING ALTIMETERS/DIGITIZERS

MANUFACTURER	MODEL TYPE	RANGE (x 100 ft)	DISPLAY CASE	LENGTH WEIGHT	DIGITIZER DESIGN	POWER INPUT	PRICE (uninstalled)	REMARKS
Pointer, Inc. 1027 N. Stadem Dr. Tempe, AZ 85281 (602) 968-1674	L115 altitude digitizer	−10+200	— —	7.3 1.1	solid state	14 or 28 VDC	$575	tentative price; remotely mounted; installation kit included
Terra Corp 3520 Pan American Freeway, N.E. Albuquerque, NM 87107 (505) 884-2321	AT 3000 blind encoder	−10+300	— —	6.2 0.5	solid state	14 or 28 VDC	$495	Digitizer only; remote-mounted
Trans-Cal Industries, Inc. 16141 Cohasset Street Van Nuys, CA 91406 (818) 787-1221	D120P2-T D125P2-T D130P2-T D135P2-T D150P2-T altitude digitizers	−10+200 −10+250 −10+300 −10+350 −10+500	— —	3.75 1.0	optical	14 or 28 VDC	$755 $850 $850 $925 $1,700	replaceable electronic module for ease of repair; uses generic aneroid designed for sensitive baro altimeters
United Instruments 3625 Comotara Ave. Wichita, KS 67226 (316) 685-9203	5035 encoding altimeter	−10+200 −10+350	3P MS	5.50 1.60	optical	14 or 28 VDC	$1,617 $1,760	Optional white or blue-white lighting

HORIZONTAL SITUATION INDICATORS/COMPASS SYSTEMS

MANUFACTURER	MODEL TSO	DISPLAY CASE GYRO	SLAVING RATE (norm. fast)	VOLTS INPUT AUTOPILOT OUTPUT	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Aeronetics 455 Kehoe Blvd. Carol Stream, IL 60188 (312) 668-3040	5000 C6c, C34c C36c, C40b	3 ATI remote	3 degrees/min 150 degrees/min	28 VDC Arinc analog	3/8.7	$7,370	Arinc format; automatic emergency mode function (flux valve input); autopilot outputs; with bearing pointer — $8,460; 3 wire synchro boot strap output to RMI; two-year warranty
	421 Series C6c, C34c C36c, C40b	4 ATI remote	3 degrees/min 300 degrees/min (nominal)	28 VDC Arinc analog	3/7.5	$7,600	Arinc format; with bearing needle — $8,480; with bearing needle and digital distance and course readout — $10,080; individual components available; Arinc 568 or King Serial Digital; 40 mv/nm optional; two-year warranty
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KCS 305 C6c	4 ATI remote	3 degrees/min 240 degrees/min	115 VAC 400 Hz Hdg. 393 mv AC/degree Course: 393 mv AC/degree	4/10.4	$5,755*	*base price; includes autopilot outputs; many options available; includes boot strap heading in HSI; night vision goggle compatible instruments available; 2-year warranty

HORIZONTAL SITUATION INDICATORS/COMPASS SYSTEMS

MANUFACTURER	MODEL	DISPLAY CASE TYPE GYRO	SLAVING RATE (contr. fast)	VOLTS INPUT AUTOPILOT OUTPUT	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KCS 55A C5c	3 ATI remote	3 degrees/min 180 degrees/ min	14 or 28 VDC Hdg.:±550 mv DC/ deg.: max. ±12.5v Course: ±220 mv DC/deg.: max ±12.5v	4/9.4	$7,790	Includes autopilot outputs: RMI output $815 additional; night vision goggle compatible instruments available
Century Flight Systems Mineral Wells, TX 76067 (817) 325-2517	NSD-360A-15 C5c C9c C52a	3 ATI panel mounted	non-slaved	14 or 28 VDC potentiometer	1/4.5	$3,805	Includes autopilot outputs; requires 5-in vacuum
	NSD-360A(S) -26 C6c C9c C52a	3 ATI panel mounted	2 degrees to 3 degrees/min N/A	14 or 28 VDC potentiometer	4/5.1	$5,525	Includes autopilot outputs; requires 5-in vacuum; NSD-360A-35(S) provides RMI output, requires 26 VAC, 400 Hz power — $6,135
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4095	MCS-65 C6c	— remote	2 degrees/min 7 degrees/sec	26 VDC three-wire synchro	3/6.7	$9,055	Includes autopilot outputs; digital interface with EHSI-74
	PN-101 C6c	3 ATI remote,	3 degrees/min 300 degrees/ min	14 or 28 VDC three-wire synchro	4/12.7	$24,520	Includes autopilot outputs, slaving accessory and 331A-36 course indicator

Footnote: Avionic form factor is shown in the Units/Weight/Size column. (See Form Factor explanation in the front of the Avionics section.)

RADIO ALTIMETERS

MANUFACTURER	MODEL TSO	ALTITUDE RANGE PITCH/ROLL LIMITS	ACCURACY	UNITS SIZE/WEIGHT	DISPLAY	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KRA 10A FM/CW none	-20-2,000 ft ±20 degrees/20 degrees	40-500 ft ±5 ft or 5%; 500-2,000 ft ±7%	rcvr./xmtr. 3.1 x 3.5 x 8 in; incl. 3 ATI; 1 ant. 3.8 lb	pointer and dial type; linear scale expanded from 0-500 ft	$4,535*	*28 VDC system; continuous DH set; "skewed" antenna optional for mounting on angled aircraft surface; 14 VDC configuration — $4,807
	KRA 405 FM/CW C87	-20-2,000 ft ±20 degrees/ ±25 degrees	-20-500 ft ±3 ft or 3%; 500-2,000 ft ±5%	rcvr./xmtr. 11.5 x 5 x 3 in; incl. 3 ATI; 2 ant.; 10.7 lb	pointer and dial type; linear scale expanded from 0-500 ft	$11,490	Continuous DH set; tracking to 2,500 ft; expanded helicopter version available; night vision goggle compatible version available; 2-year warranty
Collins Avionics Div. Rockwell International 400 Collins Rd. NE Cedar Rapids, IA 52498 (319) 395-4085	ALT-50A CW C87	-0+2,000 ft ±40 degrees/ ±50 degrees	±2 ft or 2%;	rcvr./xmtr. ¼ ATR-SL; indicator, .83 x 3.38 in two antennas; 6.7 lb	digital	$11,935	DH light and four trips settable over entire range; includes DRI-55 indicator and two ANT-52 antennas
	ALT-55B CW C87	-0+2,500 ft ±40 degrees/ ±50 degrees	±2 ft or 2%;	rcvr./xmtr. ¼ ATR-SL; indicator, .83 x 3.38 in two antennas; 6.7 lb	digital	$14,010	DH light and four trips settable over entire range; includes DRI-55 indicator and two ANT-52 antennas
	AL101 FM/CW C87	-20+2,500 ft ±40 degrees/ ±50 degrees	-20+500 ft ±2 ft or 2%; 500-2,500 ft ±5%	rcvr./xmtr. ½ ATR-S; incl. 3 ATI; 2 ant. 7-in dia. x 2.75 in; 22.5 lb	pointer and dial type; linear -20+500 ft; logarithmic 500-2,500 ft	$39,236	ARINC 552, 552A specification design, includes vertical tape instrument
Honeywell Inc. P.O. Box 29000 Phoenix, AZ 85038 (602) 863-6009	AA-300 pulse C87	0-2,500 ft ±45 degrees/ ±45 degrees	0-100 ft ±5ft; 100-500 ft ±5%; 500-2,500 ft ±7%	rcvr./xmtr. 10.7 x 4 x 7.5 in; incl. 3.3-in dia. x 4 in; 2 ant. 8.25 lb	pointer and dial type; linear scale expanded from 0-500 ft or 0-200 ft	$14,307	Altitude rate optional; continuous DH set; tracking to 3,000 ft; includes RT-300 receiver/transmitter, RA-315 indicator and AT-300 antenna
Terra Corp. 3520 Pan American Freeway NE Albuquerque, NM 87107 (505) 884-2321	TRA 3000 FM/CW none	40-2,500 ft ±20 degrees/ ±30 degrees	40-100 ft ±5 ft; 100-500 ft ±5%; 500-2,500 ft ±7%	rcvr./xmtr. 1.0 x 5.0 x 7.6 in; indicator 3.5 x 3.25 x 4.0 in. antenna built into R/T unit; 2.5 lb	—	see remarks	*Prices: with TRI 20 LED indicator — $2,395; with TRI 30 analog indicator — $2,995; with TRI 40 LED indicator, with 100' AGL gear warning — $3,295; continuous DH set; aux analog output and STWT
	TRA 3500 FM/CW none	0-2,500 ft ± 20 degrees/ ± 30 degrees	0-10 ft ± 1 ft; 10-100 ft ± 5 ft; 100-2,500 ft ± 5-7%	rcvr./xmtr. 3.0 x 3.0 x 6.9 in; indicator: 3.5 x 3.25 x 4.0 in; —	pointer and dial type; linear 40-2,500 ft; (enlarged linear 40-500 ft)	$5,495	Price includes TRI 40 LED indicator, including 100 ft gear warning; TSO C87 pending; anti-lower circuit design

Footnote: Arinc form factor is shown in the Units/Size/Weight column. (See Form Factor explanation in the front of the Avionics section.)

THUNDERSTORM DETECTION SYSTEMS

MANUFACTURER	MODEL	RANGE (nm)	DISPLAY SIZE VIEWING ANGLE	DISPLAY TYPE	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Honeywell Inc. P.O. Box 29000 Phoenix, AZ 85038 (602)863-8000	LSZ-850	Same as radar display to 100 nm	Interfaces on radar display 120 or 360 degrees	Quantitative electrical discharge activity displayed on CRT	3 11.0	$19,703	Frequency of lightning discharge causes one of three unique symbols to be displayed at the relative location of such activity; integrated radar & lightning system sensor controls available
3M Stormscope Systems 6530 Singletree Dr. Columbus, OH 43229 (614) 885-3310	WX-8	100	3-in dia. 135 degrees	liquid-crystal display	1 + ant. 4.0	$3,975	Panel-mounted unit/display; three-color liquid-crystal display range resolution; storm intensity enhancing; 10 – 30 VDC
	WX-1000 Stormscope Series II	25 50 100 200	3-in ATI 120 or 360 degrees	vertical electrical discharge activity on CRT	2 + ant. 10.9	$8,895	Displays only vertical electrical discharge activity associated with thunderstorms; up to six checklists; clock/calendar mode with elapsed and flight time; built-in test and error code display; future expansion capability; 10.5 - 32 VDC; 28 Watts; TSO C110
	WX-1000+ Stormscope Series II	25 50 100 200	3 in. ATI 120 or 360 degrees	vertical electrical discharge activity on CRT	2 + ant. 10.9	$9,995	Heading stabilized — 5 to 50 VAC, 400 Hz; displays only vertical electrical discharge activity associated with thunderstorms; up to six checklists; clock/calendar mode with elapsed and flight time; built-in test and error code display; future expansion capability; 10.5 - 32 VDC; 28 Watts; TSO C110

MLS RECEIVERS

MANUFACTURER	MODEL TSO	CHANNELS CONTROL/ DISPLAY	SIZE/WEIGHT	POWER	PRICE (uninstalled)	REMARKS
Bendix/King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	MLS-21 C104	200 ICAO standard rotary knob LCD	receiver: 11.75 x 5 x 3.9 in. control: 2.5 x 3.125 x 2.5 in. (2) antenna: flush-mount system weight: 8.7 lbs	28 VDC/ .75 amps	$20,162	Arinc 429 compatible; ICAO magnetic course reference; price includes installation tray, kit and two antennas; PDME compatible & EFIS compatible; night vision goggle compatible control head available; two-year warranty
Canadian Marconi 2442 Trenton Ave. Montreal, Quebec, Canada H3P 1Y9 (514) 341-7630	CMA-2000 NA	200 ICAO standard rotary knob LCD	receiver: 3 MCU control: 3.75 x 5.75 x 2.375 in. (3) antenna system weight: 11.5 lbs	115 VAC 400 Hz, 75 VA max	$31,040	In full compliance with Arinc 727 and ICAO specifications; analog and Arinc 429 interfaces; manual and fully automatic self-test; auto/manual angle modes; three-year warranty
Honeywell Inc. P.O. Box 29000 Phoenix, AZ 85038 (602) 863-6000	MLZ-850 C104	200 ICAO standard rotary knob LED	receiver: 13.9 x 3.3 x 3.9 in. control: 2.375 x 2.625 x 5.5 in. (2) antenna: 1.2 x 2.5 in. flush aperture system weight: 6.6 lbs	28 VDC	$23,033	Price includes tray and installation kits; analog output to ADI, HSI, FCS; CSDB digital interface; 2x5 DME channeling; updatable to future ICAO magnetic course reference; Primus II MLS sensor configuration, 5.5 lbs — $19,520

NAVIGATION MANAGEMENT SYSTEMS

MANUFACTURER	MODEL TSO	TOTAL WAYPOINTS FLIGHT PLANS / WAYPOINTS DATA ENTRY	PRIMARY SENSOR(S) ADDITIONAL SENSORS	INPUTS OUTPUTS	POWER INPUT COOLING	ATR SIZE / WEIGHT / UNITS	PRICE	REMARKS
Bendix / King Avionics 400 N. Rogers Rd. Olathe, KS 66062 (913) 782-0400	KNS 660 C40b C90a	3,000 400 / 25 keyboard (non-volatile)	DME/DME VOR/DME Omega/VLF TACAN GPS Loran C IRS	analog digital analog digital	28 VDC 26 VAC / 400 Hz forced air required	2/20.2 —	$28,570*	*47,995 with optional Omega/VLF sensor; world-wide database: VORs, TACANs, airports, intersections, airport data, SIDs, STARs, with updating every 28 days with Bendix / King supplied diskette; monochrome CRT; VNAV advisory; optional full-alpha keyboard; coupled VNAV with KFC 400; compatible with Bendix / King GPS sensor; direct interface with Bendix EFIS; radar mapping available through Bendix / King weather radar with optional GC 360A radar graphics unit; optional rubidium frequency standard; night vision compatible CDU available; special packages may be available; two-year warranty

NAVIGATION MANAGEMENT SYSTEMS

MANUFACTURER	MODEL TSO	TOTAL WAYPOINTS FLIGHT PLANS / WAYPOINTS DATA ENTRY	PRIMARY SENSOR(S) ADDITIONAL SENSORS	INPUTS OUTPUTS	POWER INPUT COOLING	ATR SIZE / WEIGHT / UNITS	PRICE	REMARKS
Canadian Marconi 2442 Trenton Ave. Montreal, Quebec, Canada H3P 1Y9 (514) 341-7630	CMA-900 pending	35,000 400 / 99 keyboard disk data loader	DME / DME Omega/VLF IRS MLS GPS Loran-C DADC	analog Arinc 429 analog Arinc 429 Arinc 561	28 VDC / 60 Watts or 115 VAC / 80 VA no external cooling required	Color CDU: 5.75 x 6.8 x 7.8; 6 lbs. NMU: 2 MCU; 10 lbs.	$50,000	price varies with number and type of interfaces and optional long-range sensors; internal Omega/VLF included, may be replacedwith optional, internal GPS; external Omega/VLF sensor available; 10 Arinc 429 input ports; fuel computer optional; frequency management optional; 3.5" diskette data loader optional; ColorCDU capable of controlling other sub-systems
Global-Wulfsberg Systems 2144 Michelson Dr. Irvine, CA 92715 (714) 851-0119	GNS X C115	256 49 / 30" keyboard	DME/DME VOR/DME Loran C VLF/Omega IRS	Arinc 429 Arinc 571 CSDB analog Arinc 429 Arinc 561 Arinc 571 CSDB analog	28 VDC / 1.8 Amps 26 VAC / 400 Hz** no external cooling required	Computer: ¼ATR-S / 8.4 lbs CDU: 4.5 x 5.75 x 6.5 / 5.5 lbs	$41,652	*up to 50 route waypoints on active flight plans; **AC power for analog interfaces only; price includes installation kit, Loran C sensor board, 50,000 waypoint, worldwide internal database, fuel flow interface, full alpha keyboard CDU; no VNAV; frequency management system standard; AFIS compatible, optional Data Management Unit — $26,700, first year subscription included; 3.5" diskette Data Transfer Unit — $2,600; with AFIS option, flight plans and messages can be interchanged via datalink; optional VLF/Omega RPU — $22,500
	GNS 1000 C115 C109 C94a	256 49 / 30" keyboard	DME/DME VLF/Omega VOR/DME IRS	Arinc 429 Arinc 571 CSDB analog Arinc 429 Arinc 561 Arinc 571 CSDB analog	28 VDC / 5.2 Amps 26 VAC / 400 Hz** no external cooling required	Computer: ½ATR-S / 15.0 lbs Receiver / Processor: ½ATR-S / 16.0 lbs NDB-2: ½ATR-S / 9.1 lbs CDU: 4.5 x 5.75 x 6.5 / 5.5 lbs	$88,143	*up to 50 route waypoints on active flight plans; ** AC power for analog interfaces only; price includes installation kit, H-field antenna; dual systems — $159,177; monochrome CRT; optional full alpha keyboard — $1,220; no VNAV; frequency management system standard; AFIS compatible, optional Data Management Unit — $26,700, includes first year subscription; 3.5" diskette Data Transfer Unit — $2,600; with AFIS option, flight plans and messages can be interchanged via datalink; compatible with almost all fuel flow sensors or indicators; 25,000 waypoint NDB-2 database includes North American / European or world-wide data; remove database from aircraft every 28 days to update; VLF/Omega Receiver Processor Unit includes in price, but may be deleted for credit

NAVIGATION MANAGEMENT SYSTEMS

MANUFACTURER	MODEL TSO	TOTAL WAYPOINTS FLIGHT PLANS / WAYPOINTS DATA ENTRY	PRIMARY SENSOR(S) ADDITIONAL SENSORS	INPUTS OUTPUTS	POWER INPUT COOLING	ATR SIZE / WEIGHT / UNITS	PRICE	REMARKS
Honeywell Inc. P.O. Box 29000 Phoenix, AZ 85038 (602) 863-8000	LASERNAV II —	255 20 / 20 keyboard	Laser IRS VOR DME VLF/Omega	Arinc 429 Arinc 419 Arinc 429 Arinc 561 Arinc 571 CSDB	28 VDC / 5 VDC or 115 VAC / 400 Hz no external cooling required	INU: modified 10 MCU / 46.5 lbs CDU: Arinc 704-4 / 7.0 lbs	$153,214	certificated as a primary sole-source, long-range navigation system; supplies attitude and heading reference to flight instruments; ring-laser-gyros used for inertial reference; monochrome display; NDB-2 database, updated with Honeywell-supplied tape; accommodates 2 x VORs, 2 x DMEs, 2 x VLF/Omega sensors; compliance with RTCA DO-160A
	FMZ-800 C115	1,400 100 / 50 3.5" disk data loader	DME/DME VOR Omega/VLF IRS AHRS (MLS) (GPS) air data	2 x ASCB 3 x Arinc 429 6 x CSDB ASCB Arinc 429 CSDB	28 VDC / 2.7 Amps no external cooling required	Color CDU: 5.4 x 7.5 x 10.0; 9.4 lbs Computer: ½ATR-S; 14.3 lbs	$92,326	system "learns" operating characteristics of aircraft from previous flights; 200 pilot-defined waypoints in addition to those listed; non-volatile EEPROM memory; flight plans computed by Lockheed JetPlan and down loaded via 3.5" diskette; mono display also available; 3-D navigation with full VNAV; linked J-route / V-airway mode; radial intercept mode; full-autothrottle growth mode; ACARS / datalink growth mode; Loran-C capability — TBA; no TACAN; US/IFR-RNAV, NAT MNPS, RNAV Approach, sole-source over-water nav approvals; full Jeppesen database, updated with 3.5" diskette data loader; standard equipment on Canadair 601-3A, Gulfstream IV and Falcon 900 aircraft; 30,000 waypoint database standard; optional 1.25-Mbyte KZ-910 international nav database including 4,000+ paved airports, airways, SIDs/ STARs; 28-day updates take less than 8 minutes; 1-year warranty
Universal Navigation Corporation 3260 East Lerdo Rd. Tucson, AZ 85706 (602) 741-2300	UNS-1 NMS C115 C66b C94a C60a	3,000 200 / 96 keyboard alpha or line-item selection (3.5" disk data loader)	DME/DME VOR IRS Omega/VLF Loran-C GPS TACAN DADC	analog Arinc 429 Arinc 575 CSDB analog Arinc 429 Arinc 561 Arinc 571 Arinc 575 ASCB	28 VDC / 1.8 Amps 26 VAC / 400 Hz no external cooling required	CDU: 5.75 x 4.5 x 6.75, 4.5 lbs MCU: 2 MCU, 7.0 lbs 5.75W x	$30,000	price varies with up to three optional long-range sensors; 40,000 waypoint worldwide database; auto-scanning DME / DME / DME; diskette data loader — $3,600; advisory VNAV; direct-to key; two engine fuel management — DC-analog inputs; optional Super CDU to upgrade to UNS-1A configuration — mono CRT, adds 3.0 lbs and $7,000; color CRT, adds 3.0 lbs and $12,000; direct-to key; 3-D approach mode; direct-to mode; optional frequency management system; multi-chain Loran-C sensor — $12,000; UNS-RRS DME/VOR/ TACAN sensor — $16,000; optional UNS-764 VLF/Omega sensor — $22,000; three-year warranty includes sensors

NAVIGATION MANAGEMENT SYSTEMS

MANUFACTURER	MODEL TSO	TOTAL WAYPOINTS FLIGHT PLANS / WAYPOINTS DATA ENTRY	PRIMARY SENSOR(S) ADDITIONAL SENSORS	INPUTS OUTPUTS	POWER INPUT COOLING	ATR SIZE / WEIGHT / UNITS	PRICE	REMARKS
Universal Navigation Corporation 3260 East Lerdo Rd. Tucson, Az. 85706 (602) 741-2300	UNS-1A Compact FMS C115 C66a C94a C60a	3,000 200 / 98 keyboard alpha or line item select (3.5" disk data loader)	DME / DME VOR IRS Omega/VLF Loran-C TACAN GPS DADC	analog Arinc 429 Arinc 575 CSDB analog Arinc 429 Arinc 561 Arinc 571 Arinc 575 ASCB	28 VDC / 2.1 Amps 26 VAC / 400 Hz no external cooling required	SuperCDU: 5.75 x 6.4 x 7.88 7.6 lbs NCU: 2 MCU, 7.8 lbs	$55,000	price varies with CDU and up to five long-range sensors; 40,000 waypoint worldwide database; auto-scanning DME/DME/DME and VOR/DME; monochrome super CDU, deduct $5,000; frequency management unit — $10,000; dual system data cross-talk mode; off-line flight planning; diskette data loader — $4,500; plus all features and options of UNS-1 NMS above; three-year warranty includes sensors
	UNS-1A FMS C115 C66b C94a C60a	3,000 200 / 98 keyboard alpha or line item select (3.5" disk data loader)	DME / DME VOR IRS Omega/VLF Loran-C TACAN GPS DADC	analog Arinc 429 Arinc 575 CSDB analog Arinc 429 Arinc 561 Arinc 571 Arinc 575 ASCB	28 VDC / 2.3 Amps 26 VAC / 400 Hz no external cooling required	SuperCDU: 5.75 x 6.4 x 7.88 7.6 lbs NCU: 2 MCU, 11.0	$72,500	price varies with CDU and up to five optional long-range sensors; includes color Super CDU; optional SID/STAR/Approach database — $10,000; fuel management with up to four inputs; all features and options of UNS 1A Compact FMS listed above; three-year warranty includes sensors

AIRBORNE TELEPHONE SYSTEMS

MANUFACTURER	MODEL COLORS	MODE OF OPERATION	POWER OUTPUT (watts) MAX ALT	POWER INPUT	UNITS/ WEIGHT (lb)	PRICE (uninstalled)	REMARKS
Global-Wulfsberg Systems 2144 Michelson Dr. Irvine, CA 92715 (714) 851-0119	Filstone VI 7	full duplex, HF optional	10 51,000 ft	28 VDC	2/9.5	$7,664	Remote handsets and controls available; direct-dial capable
Terra Corp. 3520 Pan American Freeway, N.E. Albuquerque, NM 87107 (505) 884-2321	Jetone II 4	full duplex	8 40,000 ft	28 VDC	1/4.25	$3,000	Self-contained; panel-mounted unit; includes handset
	TD-3000 N/A	full duplex, AGRAS	10 55,000 ft	28 VDC	2/6.8	$7,500	Direct dial compatible; automatic channel selection, operator alert; additional handsets, $535; digital channel readout; includes cockpit control unit

C2 MILITARY AVIONICS SYSTEMS PRICE DATA

The prices of military avionics systems depend strongly on the following factors:

* Requested performance
* Operational environment (particularly high 'g' levels and extreme temperatures)
* Frequency of modifications requested by various military services
* Number of systems procured

In some cases the military will purchase 'off-the-shelf' commercial avionics equipment. In that case the price data of Section C1 might be applicable.

In most cases military avionics systems are developed for specific military purposes. Since prices of such equipment are not published regularly in the open literature, another approach is recommended. That approach is to use fractional cost data based on published information.

Figures C1 - C3 are examples of such fractional data. It appears from these data that military avionics costs typically range from:

$$C_{avionics/military} = 0.05(AEP) \qquad (C1)$$

to:

$$C_{avionics/military} = 0.40(AEP) \qquad (C2)$$

where: AEP is the airplane estimated price as estimated from Equation (4.3)

The reader has to apply 'judgment' to determine the magnitude of the fraction (0.05 to 0.40) which applies to a new design.

Figure C1 Avionics as a Percentage of Total Airplane Cost

Figure C2 Avionics and Other Equipment as a Percentage of Flyaway Cost

Figure C3 Historical Distribution of Avionics and Other Equipment as a Percentage of Flyaway Cost

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page 51, Eqn. (2.9)	units for c_P= (lbs/hr)/hp
line 6 from bottom	replace 1,000 nm with 1,000 sm
page 52, line 3	1000 is standard miles
page 61, Table 2.19	Take-off and Landing: groundrun of less than 2,400 ft.
page 64, line 3 and 5	W is the take-off weight: W_{TO}
page 69, Eqn. (2.23)	$D = \left(W_{PL} + W_{CREW}\right) + W_{p_{\exp}}$ Where $W_{p_{\exp}}$ is the weight of the expended payload. (i.e., Missles, bombs, etc.)
page 95, line 14 from bottom	remove “range of”
page 98, line 8 from bottom	replace $C_{L_{TO_{\max}}}$ by $C_{L_{\max_{TO}}}$
page 106, line 2	replace four factors: with five factors:
page 115, Eqn. (3.18)	$V_A = 1.1 V_{s_{PA}}$
page 132, 4th line	delete “an”
page 150, Eqn. (3.32)	$RC_h = RC_0\left(1 - h / h_{abs}\right)$
page 152, Eqn. (3.38)	$\sin\gamma = \frac{T}{W}\left[P_{dl} - \sqrt{P_{dl}^2 - P_{dl} + \left(1 + \left(\frac{L}{D}\right)^2\right)^{-1}\left(\frac{T}{W}\right)^{-2}}\right]$

page 186, Section 3.7.4.2

line 3	...groundrun as < 2,400 ft.
line 7	S_L = 1.9 x 2,400 = = 4,500 ft.
From Figure 3.16	S_L = 4,500/0.6 = 7,500 ft.
From Figure 3.17 this yields:	V_A^2 = 25,000 kts^2. $V_A = \left\{21{,}200(1.3/1.2)^2\right\}^{1/2} = 158\,kts$ should be $V_A = \{25{,}000\}^{1/2} = 158\,kts$

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page 102, Step 3.1, 2nd line ...falls into one of the <u>eight</u> catagories...

page 158, Step 6.10 *Eqn. (6.1) should be Eqn. (6.2)*

page 170, Eqn (7.8) and (7.11-18) C_l should be c_l

pages 176 to 185 The example problems of Section 7.2 are incorrect. The K_Λ factor was multiplied instead of divided to yield the $C_{l_{max}}$ values for Step 7.4 of 7.2.1, 7.2.2, and 7.2.3. Each example will be addressed below.

Section 7.2.1 Twin Engine Propeller Driven Airplane

The results of step 7.4 should be:

	Take-off flaps		Landing flaps	
$\frac{S_{wf}}{S}$	0.3	0.6	0.3	0.6
$\Delta C_{l_{max}}$	0.58	0.29	2.32	1.16

Z_{fh} should be $\frac{Z_{fh}}{c}$.

For Step 7.5, the referenced equations and figures are wrong:

Eqn. (7.15) should be Eqn. (7.16).
Eqn. (7.14) should be Eqn. (7.15).
Eqn. (7.13) should be Eqn. (7.14).
Figure 7.7 should be Figure 7.8.
Figure 7.3b should be Figure 7.4.
Eqn. (7.10) should be Eqn. (7.11).

Section 7.2.2 Jet Transport

The results of step 7.4 should be:

	Take-off flaps		Landing flaps	
$\frac{S_{wf}}{S}$	0.6	0.8	0.6	0.8
$\Delta C_{l_{max}}$	3.00	2.24	3.84	2.88

For Step 7.5, the referenced equations and figures are wrong:
Eqn. (7.10) should be Eqn. (7.11)
Figure 7.3b should be Figure 7.4

Section 7.2.3 Fighter

The results of Step 7.4 should be:

	Take-off flaps		
$\frac{S_{wf}}{S}$	0.4	0.8	1.0
$\Delta C_{l_{max}}$	4.00	2.000	1.60

These corrections will affect the results of each sample problem. It is left to the reader to complete the sample problems using the correct results of Step 7.4. The summary and referenced drawings of Step 7.6 may change due to these corrections.

pages 178, 181, and 184

Under Step 7.4, K_{Δ} should be K_{Λ}

page 267
Eqn (11.13)

$N_D = 0.75 N_{t_{crit}}$ should be $N_D = 0.25 N_{t_{crit}}$

Eqn (11.14)

$N_D = 0.25 N_{t_{crit}}$ should be $N_D = 0.10 N_{t_{crit}}$

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page 15, Table 2.3	Switch values for β_1 and β_2
page 194, 8th line from top	replace “stability” with “Stability”.

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page 54, Table 2.18	For fighters and trainers, the details are in Figure 2.26, not Figure 2.25.
page 380, 7th line from top	replace “is” with “are”

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page 34, Eqn (4.13)	$n_{\lim pos} \geq 2.1 + 24{,}000 / (GW + 10{,}000)$
page 43, 12th line from top	V_A should be *217 kts.*
page 45, 8th line	Change *Selene* to *Eris*
page 61, under Step 6	n_{ult} = 7.33 should read n_{lim} = *7.33*
page 71	
Eqn (5.12)	*57.5* should be *174.04*
Eqn (5.13)	*15.6* should be *639.95* Replace exponent *1.249* with *0.1249*
page 72	
6th line	replace 'ib' with 'in'
13th line	replace $1/4_V$ by $\Lambda_{1/4_V}$
page 72 - 74	All comments about the canard weight should reference 5.2.1.1.
page 74	
Eqn (5.19)	Replace $1/2_H$ by $\Lambda_{1/2_H}$
Eqn (5.20)	$W_v = K_v S_v \left[3.81 \left(S_v^{0.2} V_D \right) \Big/ \left(1000 \sqrt{\cos \Lambda_{c/2_v}} \right) - 0.287 \right]$ Replace $1/2_V$ by $\Lambda_{1/2_V}$
page 75, Eqn (5.23)	$W_f = 0.04682 (W_{TO})^{0.692} (P_{\max})^{0.374} (l_{f-n})^{0.590}$ where P_{max} is the maximum fuselage perimeter expressed in feet.
page 76	
Eqn (5.24)	N_{pax} is the number of passengers including the pilot.
Eqn (5.26)	Replace 2 x 10.43 by 10.43
page 77, Eqn (5.27)	l_h = distance from wing root c/4 to horizontal tail root c/4 in ft
page 78, Eqn (5.29)	$W_n = K_n W_{TO}$ should be $W_n = K_n P_{TO}$

page 79 *2nd line*	Eliminate and replace by "P_{TO} = Take-off power in HP"
Eqn (5.32)	$W_n = 0.045(P_{TO})^{5/4}(N_e)^{-1/4}$
page 81	
Eqn (5.38)	$W_g = 0.013W_{TO} + 0.362(W_L)^{0.417}(n_{ult.l})^{0.950}(l_{s_m})^{0.183} + 6.2 + $ $+ 0.0013W_{TO} + 0.007157(W_L)^{0.749}(n_{ult.l})(l_{s_n})^{0.788}$
7th line from bottom	Replace $N_{ult.l}$ by $n_{ult.l}$
4th line from bottom	Replace 62.61 by 62.21
page 85 *Section 6.1.2, 9th line*	Should read: *Equations (6.4) and (6.6) may also be used...*
page 87, Eqn (6.8)	$W_{ai} + W_p = 1.03(N_e)^{0.3}(P_{TO})^{0.7}$
page 90, Eqn (6.14)	Method assumes that the number of engines equals the number of propellers
page 92, Eqn (6.23)	$W_{fs} = 1.6\left[\frac{W_f}{K_{fsp}}\right]^{0.727}$
	See the associated insert for the comparison of Torenbeek and GD methods.
page 93,	Add an increment of 5 to all equation numbers throughout Chapter 6 (i.e., Eqn. (6.22) becomes Eqn. (6.27)).
page 95, Eqn (6.34b)	$W_{apsi} = 0.4K_b(N_e)^{0.2}(P_{TO})^{0.8}$
page 98,	Add an increment of 1 to all equation numbers throughout Chapter 7.
page 99, Eqn (7.4)	$W_{fc} = 0.33(W_{TO})^{2/3}$
page 108,	Eqn (7.44) Remove the (before N_{pax} in the cabin windows weight component.
page 109, Eqn (7.46)	$K_{st} = 0$ for no ejection seat
page 111, *3rd line of Section 7.12*	Should read: *Part IV, Chapter 3.*

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page xx $\bar{x}_{ac_h} = x_{ac_h} / \bar{c}$

page 27, Eqn (4.9)

$$C_{L_w} = C_L - C_{L_c} \frac{S_c}{S} - C_{L_h} \frac{S_h}{S}$$

page 46, Eqn (4.33)

$$C_{D_{L_{fus}}} = 2\alpha^2 \frac{S_{b_{fus}}}{S} + \eta c_{d_c} |\alpha|^3 \frac{S_{plf_{fus}}}{S}$$

page 47, Figure 4.20 $M_c = M \sin|\alpha|$

page 49

Eqn (4.39)

$$C_{D_{L_{fus}}} = \alpha^2 \left(S_{b_{fus}} \right) / S$$

Eqn. (4.41)

$$C_{D_{o_{fus}}} = \left(C_{f_{fus}} \left(\frac{S_{wet_{fus}}}{S_{fus}} \right) + C_{D_{N_2}} + C_{D_A} + C_{D_{A(NC)}} + C_{D_{b_{fus}}} \right) \frac{S_{fus}}{S}$$

page 52, Eqn (4.43)

$$C_{D_{L_{fus}}} = F \left\{ 2\alpha^2 \frac{S_{b_{fus}}}{S} + c_{d_c} \frac{S_{plf_{fus}}}{S} |\alpha|^3 \right\}$$

page 70, 9th line Should read "... of 4.2.2.1 but with the appropriate..."

page 73

Eqn. (4.60) Add: $\varepsilon_n > 0$ *for upwash and* $\varepsilon_n < 0$ *for downwash*

Last line Should read: *Chapter 8.*

page 77, Eqn. (4.63) $\Delta c_{l_2} = +0.056(i_n)$ with i_n expressed in degrees

page 86

3rd line Should read: ... *from Eqn. (4.6)*

Eqn. (4.74) '*Chapter 9*' should be '*Chapter 8*'

page 105, Eqn (4.84)

$$\Delta C_{D_{trim_{prof}}} = \left(\Delta C_{D_p} \right)_{\Lambda_{c/4_h}=0} \cos \Lambda_{c/4_h} \left(\frac{S_{ef}}{S_h} \right) \left(\frac{S_h}{S} \right) + \left(\Delta C_{D_p} \right)_{\Lambda_{c/4_c}=0} \cos \Lambda_{c/4_c} \left(\frac{S_{cf}}{S_c} \right) \left(\frac{S_c}{S} \right)$$

page 142 — The reference to Chapter 6 in Part IV should be Chapter 7

page 171, Eqn. (6.25) — Replace $\dot{m}_{gas}$ with $\dot{m}_a$

Where $\dot{m}_a$ follows from Eqn. (6.19)

page 177, 22nd line — Section on supersonic jet inlets should be Section 6.2.3.4

page 181, Eqn. (6.44)

$$F_{inl} = 1 + 1.75\left\{\left(\frac{\mu_{inl}-1}{\mu_{inl}}\right)\left(\frac{1}{\frac{A_m}{A_c}-1}\right)\right\}$$

page 229, Eqn. (8.7)

$$\Delta c_l = \eta_1\left(c_{l_{\delta f_1}}\right)\left(\delta_{f_1}\right)\left\{\left(c+c_1\right)/c\right\} + \eta_2\left(c_{l_{\delta_{f_2}}}\right)\left(\delta_{f_2}\right)\left(c'/c\right)$$

page 233, Eqn. (8.10)

$$\Phi_{TE_{UPPER}} = arctan\{10\frac{y_{90}-y_{100}}{c}\}$$

page 245, Eqn. (8.20)

$$\alpha_w = \alpha + i_w$$

page 268, Eqn. (8.32) — Should be

$$C_{L_o} = C_{L_{o_{wf}}} + C_{L_{\alpha_h}}\eta_h\left(S_h/S\right)\left(-\varepsilon_{o_h} - \alpha_{o_{L_h}}\right) + C_{L_{\alpha_c}}\eta_c\left(S_c/S\right)\left(\varepsilon_{o_c} - \alpha_{o_{L_c}}\right)$$

where: $\alpha_{o_{L_h}}$ and $\alpha_{o_{L_c}}$ can be found using the method of Section 8.1.3.1.

page 269, Eqn (8.37) — For jet airplanes, the horizontal tail dynamic pressure should be calculated from:

$$\eta_h = 1 - \frac{2.42\sqrt{C_{D_{o_w}}}\cos^2\left(\frac{\pi\, z_{h_{wake}}}{2\,\Delta z_{wake}}\right)}{\frac{x_{h_{wake}}}{\bar{c}} + 0.30}$$

where:

$C_{D_{o_w}}$ is the wing zero-lift drag coefficient as found from 4.2.1.1.

$$z_{h_{wake}} = a\sin\left(\gamma_h - \alpha - i_w + \varepsilon_h\right)$$

$$x_{h_{wake}} = a\cos\left(\gamma_h - \alpha - i_w + \varepsilon_h\right)$$

$$\Delta z_{wake} = 0.68\bar{c}\sqrt{C_{D_{o_w}}\left(\frac{x_{h_{wake}}}{\bar{c}} + 0.15\right)}$$

page 269, Eqn (8.37) (Cont.)

with:

a and γ_h shown in Figure 8.63;
$\bar{c}$ as the wing mean geometric chord;
α as the airplane angle of attack;
i_w as the wing incidence angle; and

$$\varepsilon_{cl} = \frac{1.62 C_{L_w}}{\pi A}$$

page 270, Figure 8.63 Should be

page 275, Eqn. (8.50) Parenthesis after ε_{0h} should be moved to after $d\varepsilon/d\alpha$.

page 279, Fig. (8.70) Units for $\dfrac{\left(\dfrac{\Delta\varepsilon_f A b_f}{b}\right)}{\Delta C_{L_w}}$ should be *degrees*

page 280, Eqn. (8.54)

$$\Delta C_{L_{max}} = K_{cw}\Delta C_{L_{max_w}} - \left(C_{L_{\alpha_w}}\right)_\delta \Delta\alpha_{w/c} + (S_c/S)\Delta C_{L_{max_c}} + (S_h/S)C_{L_{\alpha_h}}\left(-\Delta\varepsilon_f\right)$$

page 305, Eqn (8.73) The bar over the 2 should be over the c.

page 311, Eqn (8.74) The first term on the r.h.s. should read: $\left(\bar{x}_{ref} - 0.25\right)\Delta C_{L_w}$

where: ΔC_{L_w} is the wing lift increment due to flaps.

page 320, Eqn (8.78) Replace 'i_w' with '$-i_w$'.

page 323, 1st line	Should read *airplane zero angle of attack. . .*
page 333, Eqn. (8.97)	$\varepsilon = \varepsilon_{o_h} + \alpha\left(\frac{d\varepsilon_h}{d\alpha}\right)$
page 333, Eqn. (8.97)	$\varepsilon = \varepsilon_{o_c} + \alpha\left(\frac{d\varepsilon_c}{d\alpha}\right)$
page 340, Eqn. (8.107)	$K_{T_i} = \frac{550 SHP_{AV_i}\sqrt{\rho}}{\left(2W/S\right)^{3/2} D_{P_i}{}^2}$
page 342, Eqn. (8.108)	$\left(dC_m/dC_L\right)_N = \sum_{i=1}^{n}\left[\frac{\left(dC_N/d\alpha\right)_{P_i}\left(d\bar{\varepsilon}_{P_i}/d\alpha\right)\left(l_{P_i}\right)(0.79)\left(D_{P_i}\right)^2}{S\bar{c}C_{L_{\alpha_w}}}\right]$
page 374, Eqn. (10.8)	$C_{T_{X_1}} = C_{D_1}$
page 375, Last Line	$\partial C_D/\partial C_m$ should be $\partial C_D/\partial M$.
page 377 *Eqn. (10.12)*	$C_{m_u} = -C_{L_1}\left(\partial\bar{X}_{ac_a}/\partial M\right)M$
Eqn. (10.13)	$C_{T_{x_u}} = (1/\bar{q}S)\left(\partial P_{reqd}/\partial u\right) - 3C_{T_{x_1}}$
Eqn. (10.15)	$C_{T_{X_u}} = (M_1/\bar{q}S)(\partial T_{reqd}/\partial M) - 3C_{T_{X_1}}$
page 382, *Eqn (10.24)*	Add a ‘)’ to the end of Equation 10.24
2nd line below Eqn (10.24)	Equation ‘(10.23)’ should be equation ‘(10.22)’.
page 390, Fig. (10.16)	z_h should be ‘the vertical distance between the horizontal tail aerodynamic center to the fuselage center line’.
page 397, *6th line from bottom*	Replace C_{y_β} by $C_{y_{\beta_v}}$
page 398, *Eqn (10.44)*	$C_{n_{T_\beta}} = -\underset{i=1}{\overset{i=n}{Sum}}\left[\left\{\left(dC_N/d\alpha\right)_{p_i}(0.79)\left(D_{P_i}\right)^2\left(l_{p_i}\right)\right\}\Big/Sb\right]$
page 401~415, Fig.30~33	$z_v = z_p\cos\alpha_f - l_p\cos\alpha_f$

page 417, Eqn (10.50)

$$C_{y_p} = 2C_{y_{\beta_v}}\left(\frac{z_v \cos\alpha - l_v \sin\alpha - z_v}{b}\right) + 3\sin\Gamma\left(1 - \frac{4z}{b}\sin\Gamma\right)\left(C_{l_p}\right)_{\substack{\Gamma=0\\ C_L=0}}$$

where: z - is the vertical distance between the cg and the wing root quarter-chord point.

$$\left(C_{l_p}\right)_{\substack{\Gamma=0\\ C_L=0}} = \frac{k}{\beta}\left(\frac{\beta C_{l_p}}{k}\right)_{C_L=0}$$

page 418, Figure 10.35 Replace $\frac{\beta C_{l_p}}{k}$ with $\left(\frac{\beta C_{l_p}}{k}\right)_{C_L=0}$

page 419, Eqn. (10.55)

$$\frac{\left(C_{l_p}\right)_{\Gamma}}{\left(C_{l_p}\right)_{\Gamma=0}} = 1 - \frac{4z_w}{b}\sin\Gamma + 12\left(\frac{z_w}{b}\sin\Gamma\right)^2$$

page 421, Eqn.(10.62)

$$C_{n_{p_w}} = \left\{\left(C_{n_p}/C_L\right)_{\substack{C_L=0\\ M}}\right\}C_{L_w} + \left(C_{n_p}/\varepsilon_t\right)\varepsilon_t + \left[\Delta C_{n_p}\Big/\left(\alpha_{\delta_f}\delta_f\right)\right]\alpha_{\delta_f}\delta_f$$

page 422, Eqn. (10.66)

$$\alpha_{\delta_f} = \Delta c_l / \left(c_{l_\alpha}\delta_f\right)$$

page 424, Eqn. (10.71)

$$\left(C_{L_{q_w}}\right)_{M=0} = \left(0.5 + 2\frac{x_w}{\bar{c}}\right)C_{L_{\alpha_w}}$$

page 435, Eqn. (10.89)

$$C_{D_{i_h}} = \frac{2C_L}{\pi Ae}C_{L_{\alpha_h}}\eta_h\frac{S_h}{S}$$

page 439,

Eqn. (10.97)

$$C_{D_{i_c}} = \frac{2C_L}{\pi Ae}C_{L_{\alpha_c}}\eta_c\frac{S_c}{S}$$

Eqn. (10.100)

$$C_{m_{i_c}} = C_{L_{\alpha_c}}\eta_c\bar{V}_c$$

page 440, Eqn (10.102) Replace $c_{l_{\alpha_h}}$ with $c_{l_{\alpha_c}}$

page 446, Eqn (10.110)

$$c_{l_\delta} = \frac{c_{l_\delta}}{\left(c_{l_\delta}\right)_{theory}}\left(c_{l_\delta}\right)_{theory}k'$$

where k' is found from Figure 8.13

page 447,

Eqns. (10.111) & (10.113) It is assumed that: $\left(C_{l_\delta}\right)_{left} = \left(C_{l_\delta}\right)_{right}$

page 461, Eqn. (10.123) Should read:

$$C_{y_{\delta_r}} = K_\delta C_{L_{\alpha_v}} \frac{S_v}{S}\left\{\frac{c_{l_\delta}}{\left(c_{l_\delta}\right)_{theory}}\right\}\left(c_{l_\delta}\right)_{theory}\left(\frac{k'}{c_{l_{\alpha_v}}}\right)\left\{\frac{\left(\alpha_\delta\right)_{C_L}}{\left(\alpha_\delta\right)_{c_l}}\right\}$$

Eqn (10.123) is correct for a single vertical tail <u>only</u>. For a twin vertical tail:

$$C_{y_{\delta_r}} = 2\left(\frac{C_{y_{\beta_{v_{(wfh)}}}}}{C_{y_{\beta_{v_{eff}}}}}\right)K_b C_{L_{\alpha_v}} \frac{S_v}{S}\left\{\frac{c_{l_\delta}}{\left(c_{l_\delta}\right)_{theory}}\right\}\left(c_{l_\delta}\right)_{theory}\left(\frac{k'}{c_{l_{\alpha_v}}}\right)\left\{\frac{\left(\alpha_\delta\right)_{C_L}}{\left(\alpha_\delta\right)_{c_l}}\right\}$$

where: $\left(\frac{C_{y_{\beta_{v_{(wfh)}}}}}{C_{y_{\beta_{v_{eff}}}}}\right)$ is found from Figure 10.17.

All other parameters are the same.

page 467, Eqn. (10.129) $\left(c_{h_\alpha}\right)_{theory}$ is found from Figure 10.63b. The parameter $\frac{c_{l_\alpha}}{\left(c_{l_\alpha}\right)_{theory}}$ in Figure 10.63b is itself found from Figure 10.64a with the assumption that $\tan\frac{\Phi'_{TE}}{2} = \frac{t}{c}$.

page 470,
1st line (10.126) should be (10.128).

Eqn. (10.130) $\frac{c_{l_\alpha}}{\left(c_{l_\alpha}\right)_{theory}}$ is obtained from Figure 10.64a with the assumption that $\tan\frac{\Phi'_{TE}}{2} = \frac{t}{c}$.

page 484, Eqn (10.145) Replace 'α_δ' with '$-\alpha_\delta$'.

page 521, (Pratt and Whitney handbook errors)
Pressure (psia) for 200,000 ft should be 0.002655 psia
Pressure (psia) for 200,131 ft should be 0.002641 psia
Pressure Ratio, δ, for 154,199 ft should be 0.001095

DESIGN, ANALYSIS AND RESEARCH CORPORATION

ERRATA: AIRPLANE DESIGN PART VII

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page 11,
last sentence of 3rd para. — Typical numerical values for gearing ratios are given in Table 4.1 of Part IV, not Part VI.

page 39, Figure 2.5 — z_T should be negative as shown

page 44, Eqn. (2.39) — Should read:

$$S_h = \frac{-z_T T + z_D D + W\left(x_{mg} - x_{cg} + \mu_g z_{mg}\right) - L_{wf}\left(x_{mg} - x_{ac_{wf}} + \mu_g z_{mg}\right) - C_{m_{ac_{wf_g}}} \bar{q}_{rot} S\bar{c} + I_{yy_{mg}} \ddot{\Theta}}{\bar{q}_{rot}\left(x_{mg} - x_{ac_h} + \mu_g z_{mg}\right) C_{L_{h_{\max}}}}$$

page 49, Eqn. (2.54) — The 4th term on the left hand side should read:
'+ $Y_{ng}\left(x_{ng} + x_{cg}\right)$ - ...'

page 50, Eqn. (2.57) — The 4th term on the left hand side should read:
$0.025\left(\Psi_A + \Psi_{steer}\right) P_{ng}\left(x_{ng} + x_{cg}\right)$

page 98, 1st sentence — Should read, "Civil: FAR 23.201, 23.203, 23.205, ..."

page 152, Eqn. (5.59) — $T\sin\left(\alpha + \phi_t\right) + C_L \bar{q} S - W\cos\phi - (W/g) U_1 Q_1 \sin\phi = 0$

page 159,
Last sentence of 1st para. — Should read, "See Section 5.3."

page 161, Eqn. (5.77) — $RD = \left\{(W/S)(2/\rho)\left(C_D^2 / C_L^3\right)(\cos\gamma)^3\right\}^{1/2}$

page 164, AS-5263 — thrust required for level flight at $1.15V_s$, not $1.15V_{s_{PA}}$.