Some aspects of

Aircraft Design

mainly from the Loads and Structures point of view

by

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Aircraft Design

For Load Calculations (certainly for Certification of a Large Aircraft), we need to have some knowledge of

- Aircraft Weights
- Aerodynamic Data
- Design Speeds
- Stiffness Data
- Systems Data (Fuel, EFCS, Hydraulics, Airconditioning, Galleys Toilets etc)
- Geometry
- Mass Data
- V-n Information
- Engine Trust/Power

Initially, it will be a TOP down approach evolving into a BOTTOM up situation

Some Initial Remarks….

- Remember that aircraft design is a “open end” problem: multiple solutions do exist!

- This is by no means a complete design exercise. It only covers a first preliminary design iteration!

- All the dimensions, weights and performance metrics associated to the individual aircraft configuration are meant to be representative only!

- To cover various configurations and sizes, the following aircraft have been used, they may not all be represented in this note.
  They are A320 (150 seats), ATR72-600 (72 seats),
  BAe Jetstream 41 (30 seats), Britten-Norman Islander (10 seats)
  and Rans S6 Coyote (2 seats)

Aircraft Design

So how or where do we start

CS-25

This regulation applies to aircraft with a TOW >12500lb(5700kg)

For the Design Project, the specification might include

Payload 150 pax @ 100kg
Range 3000 nm
Design Cruising Speed
M_{CR}= 0.75 to 0.85
TOFL/LFL ??
Approach Speed 130 to 140 kts
(Defines V_S)
$M_{NO} = M_{CR} + 0.04$

CS-VLA

This regulation has some limits
AUW = 750 kg
VS(LDG)not to exceed 45 kts
Not more than 2 seats
Max. Weight of one occupant = 86 kg

We might add a flight duration limit of say 3 hours – this will give a fuel load

Note: Latest issue of CS-25 asks for rational analysis (e.g. time histories), at this stage of the project, suggest using earlier issues of CS-25 or CS-23 Amdt 4

Aircraft Design

The starting point is to make some "informed" guesses on the important design parameters

MTOW
OWE
Fuel (Mission)
Wing Area (and may be Aspect Ratio and Sweep)
Max $C_{L_{TO}}$ and $C_{L_{LDG}}$

TOW = OWE + Payload + Fuel (Mission + Reserves)
LW = OWE + Payload + Fuel (Reserves)
$C_{L_{MAX}}$ is a function of Flap type, Flap Angle, c_f/c, b_f/b and Leading Edge devices
V_{APP} = 1.3 times $V_{S_{LDG}}$

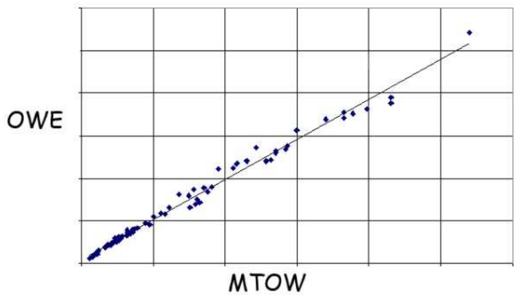
Aircraft Design

Initial aircraft weights can be derived as follows
for our example of 150 pax for 3000 nm

TOW = ZFW + FUEL = OWE + Payload + Fuel

OWE

OWE is a function of TOW
from Jenkinson et al



OWE = 0.47MTOW + 4800kg
(approx)

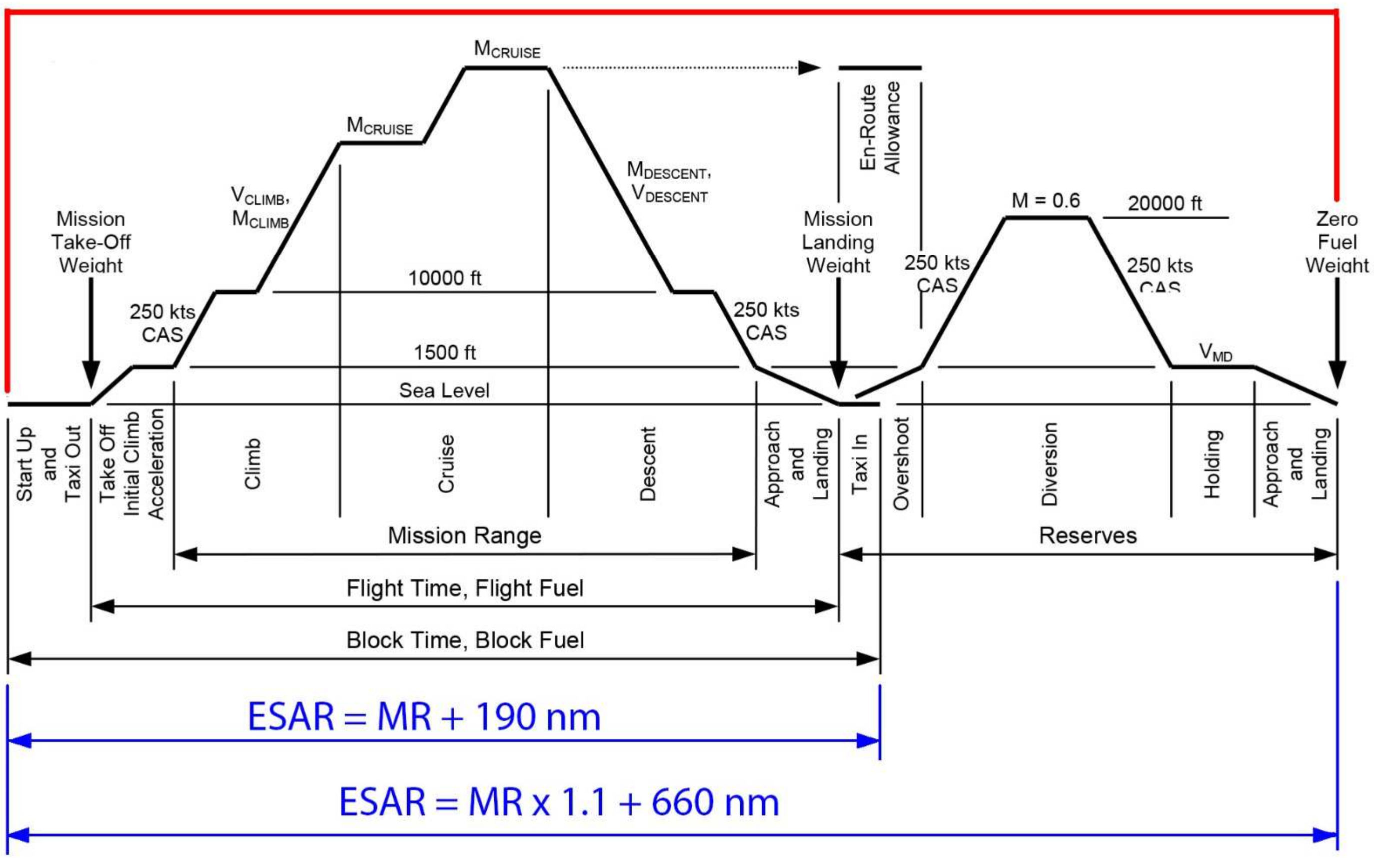
Pax

150 x 100 = 15000kg
made up of
77kg pax (CS25.785)
5kg Hand Baggage
(in Cabin)
18kg Baggage in Hold

Fuel

Mission + Reserve
from Breguet Range Eqn
which involves
Range
L/D
Speed (cruise)
Engine sfc

Typical International Flight Profile



ESAR -- Equivalent Still Air Range
MR -- Mission Range

L/D is approx 17 to 20

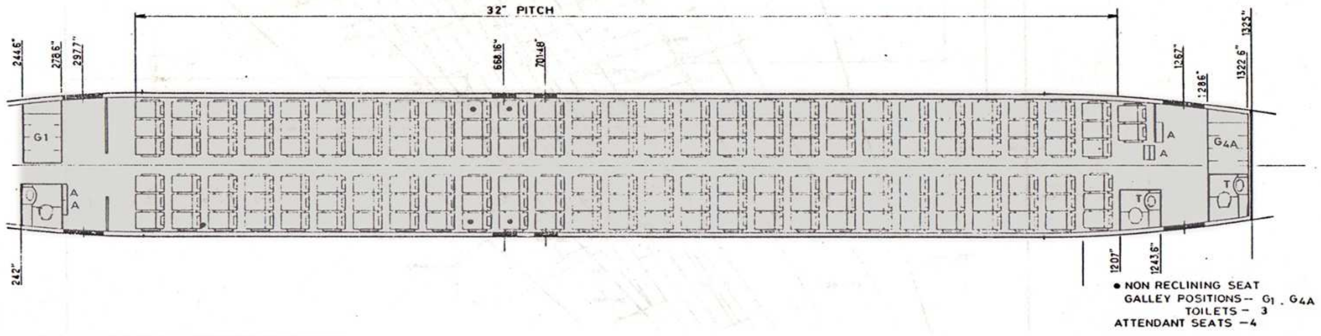
sfc is approx 0.45 to 0.59 depending on engine technology (includes BPR)

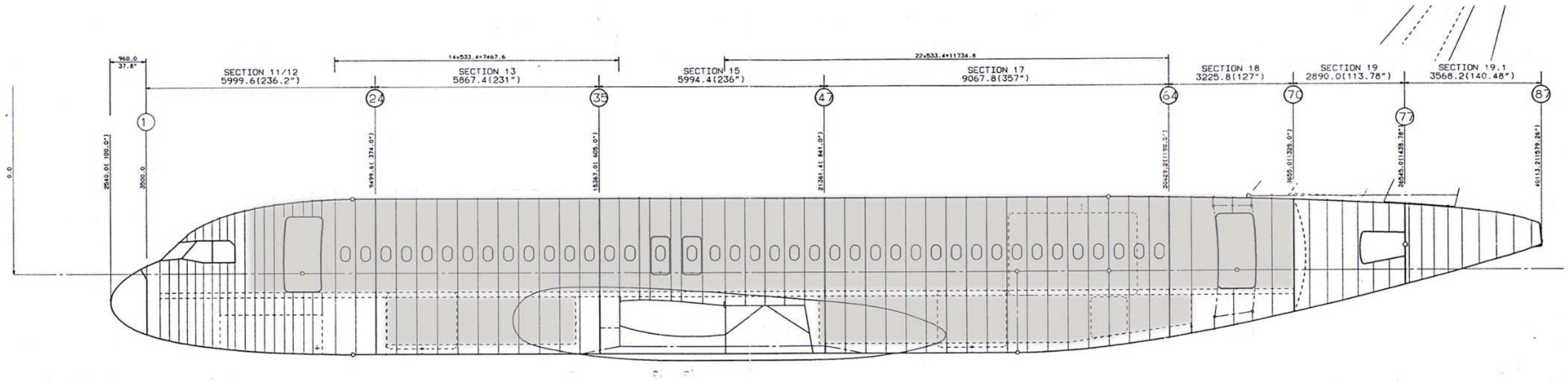
For a Conventional Aircraft layout or configuration,
the Fuselage is usually sized by the payload.

For a passenger Aircraft, the number of pax is the main driver which in turn will influence Cabin Width (Narrow Body or Wide Body or even an A380 style).

Need to provide Toilets, Galleys etc
Number of Emergency Exits (size and access to them)
Cabin Crew Seats (along with Flight Crew, these are usually counted as part of the OWE from a Mass point of view)

Typical Fuselage Layout



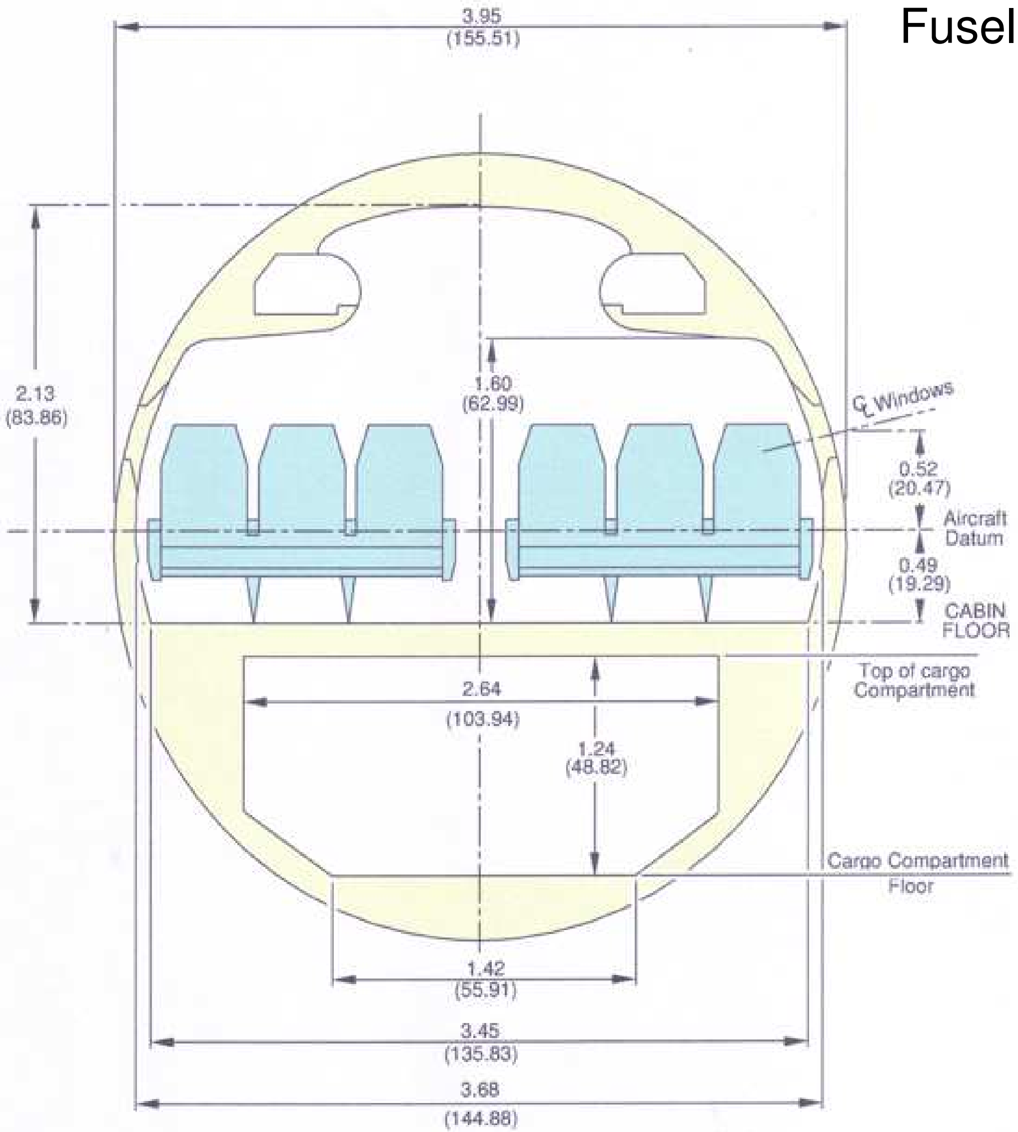


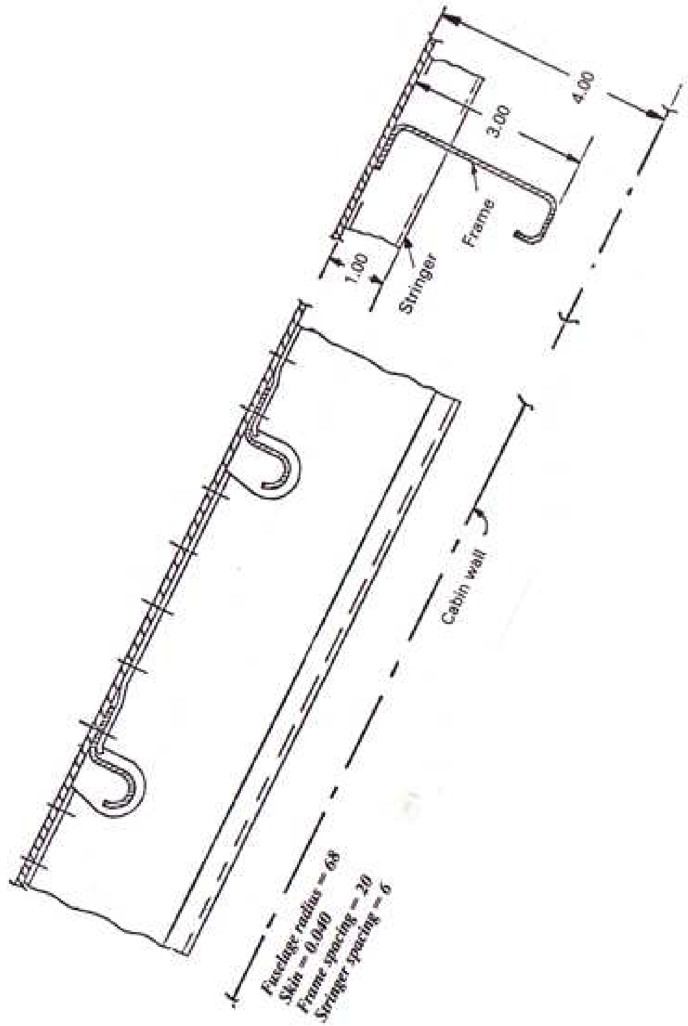
Fuselage Length is approximately 2.5 times Fuselage Width plus Cabin Length

Cabin Length is Flight Deck Door to Pressure Bulkhead

A320

Fuselage Cross-Section and Frame Depth





A320

NOTE: DIMENSIONS m (in)

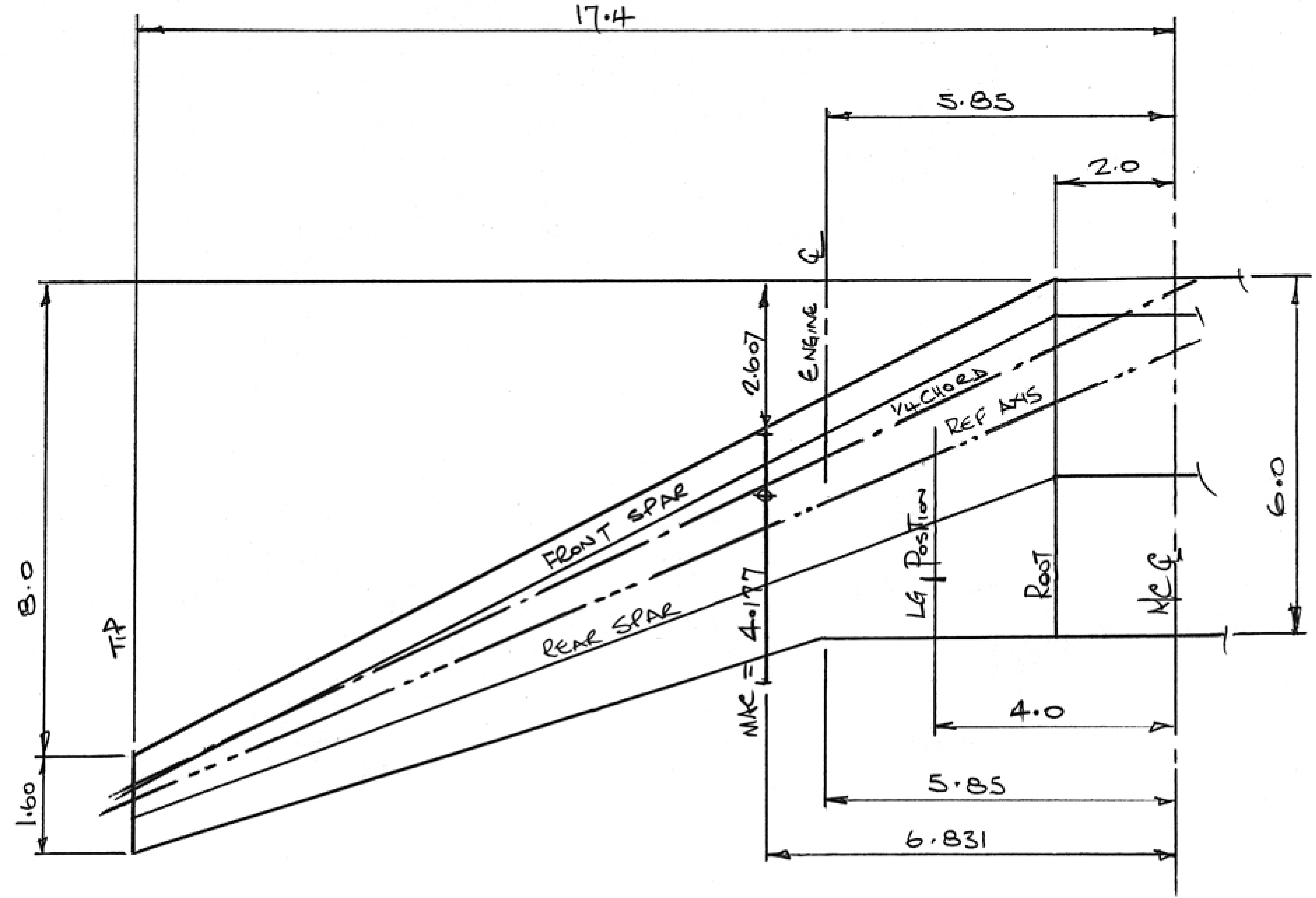
Still staying with a Conventional Aircraft,
the Wing is usually sized by the Performance needs like
Cruise, Take-off and Landing and Fuel Capacity

As a guide, based on the Fuel required (Mission plus Reserves),
Wing Area correlates as

$S = ((C \times \text{Fuel})/16)^{2/3}$ fuel in kg or

$S = ((C \times \text{Fuel})/20)^{2/3}$ fuel in litres

Where S is Wing Area (m^2)
C = 1.0 for Wing and Centre Tanks
C = 1.5 for Wing only (Dry Centre Tank)
and a Fuel Density of 800kg/m^3 is implied



Dimensions in metres

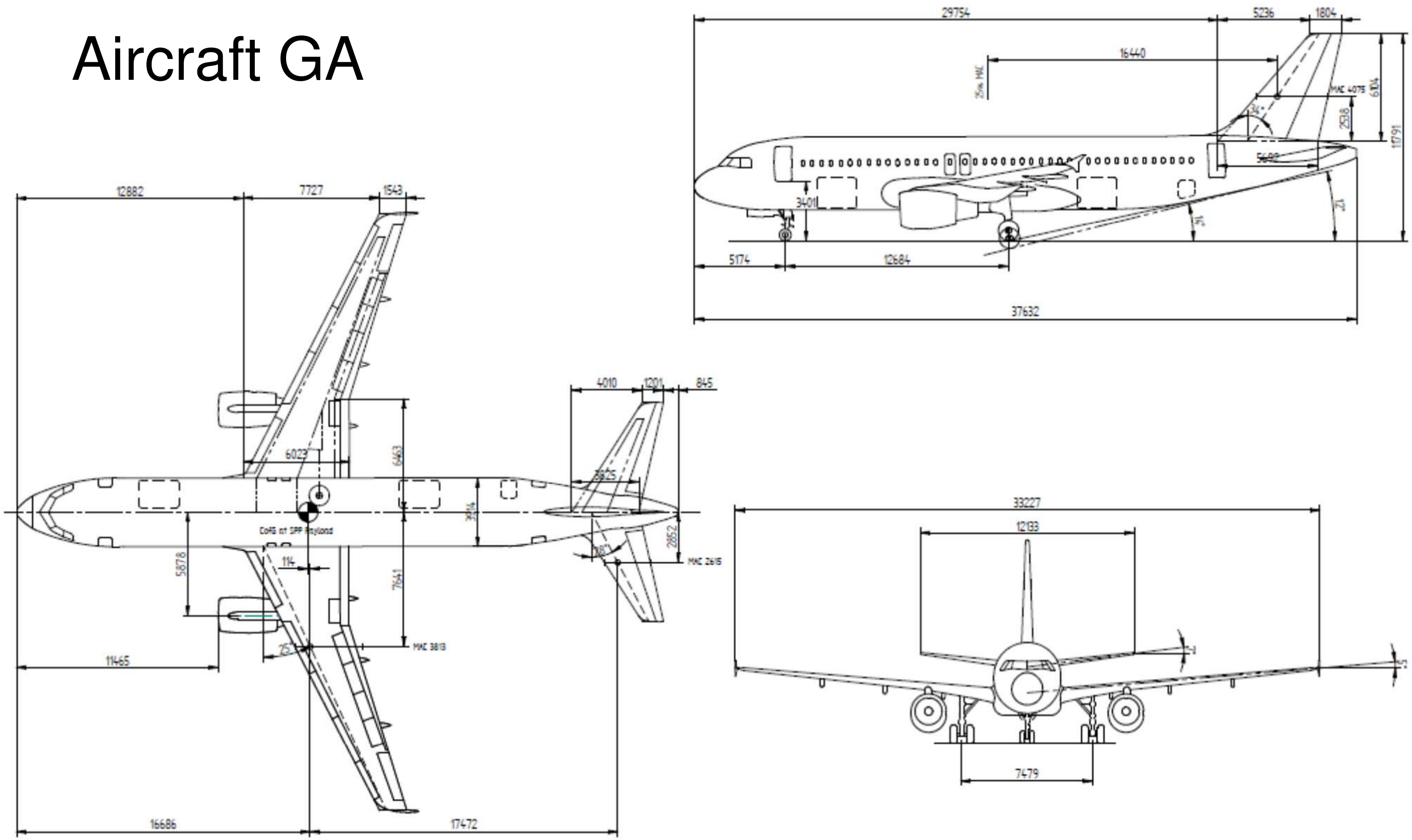
Basic Wing Geometry

Spar Positions

Position		F/S	R/S
Root	A320	0.101	0.560
Mid Span		0.143	0.634
Tip		0.348	0.609
Root	A310	0.147	0.526
Mid Span		0.201	0.657
Tip		0.323	0.605
Root	A330	0.117	0.614
Mid Span		0.154	0.654
Tip		0.263	0.665
Root	A340-600	0.100	0.659
Mid Span		0.152	0.696
Tip		0.211	0.737
Root	A380	0.190	0.704
Mid Span		0.122	0.671
Tip		0.176	0.559
Root	A400M	0.150	0.625
Mid Span		0.150	0.625
Tip		0.150	0.625

Note; Values are fractions of local chord
These values have been derived by scaling small scale drawings
They are some indication of what can occur

Aircraft GA



	Wing	HTP	VTP
Ref. Area (m²)	120	32	20
Aspect Ratio	9.2	4.6	1.9
¼ Chord Sweep (Deg °)	25	28	34
Taper Ratio	0.26	0.31	0.32
T/C Ratio (%)	15 / 12 / 10	10	11

A320-200 3-View taken from A320 "Airplane Characteristics" Manual, page 38. Issued 30th September 1985, revision September 1st 2010. Available at:
http://www.airbus.com/en/services/customer-services/maintenance-engineering/tech-data/
ONLY TO BE USED AS A TEACHING EXAMPLE. DETAILS SUCH AS SPAR LINES AND DIMENSIONS ARE NOT ACCURATE.

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AIRBUS
Future Projects Office
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SHEET 1 OF 1 | SCALE - NOT TO SCALE
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Typical Supercritical Wing Section

Wing sections on the latest commercial airliners use what is known as supercritical sections. These are usually developed by the wing designer to meet the needs of the particular aircraft specification.

The wing design is a 3 dimensional exercise ending with a wing section that can change all along the span from root to tip. Equally the thickness will vary but in a smooth manner, however t/c will probable have kinks where there are kinks in the planform.

The wing section given has been manually digitised from a very small drawing.

The process was to define a basic symmetric profile and a mean camber line following the general principles of the theory as given in 'Theory of Wing Sections' by Ira H. Abbott and Albert E. Von Doenhoff in Chapter 6 (in particular paragraphs 6.4 and 6.6)

The constants in the equations where derived using the trend line in Microsoft Excel.

The basic thickness and the mean line are defined by the following equation

$$y = t\,(a_o\sqrt{x} + a_1 x + a_2 x^2 \;\; a_3 x^3 + a_4 x^4)$$

where

- y is the vertical ordinate
- x is the distance along the chord line (maximum value = 1.0 for a non-dimensional profile)
- t is the thickness (0.1 is equivalent to a t/c = 10%)
- $a_{o,}\, a_{1,}\, a_{2,}\, a_{3,}\, a_4$ are constants derived from the trend line fit

Note: for the mean line $t = 1$

Typical Supercritical Wing Section

Values for the ‘a’ constants are

	basic	mean line
a_o	1.529035	0.0
a_1	-0.788940	0.004714
a_2	-1.449138	-0.013111
a_3	1.263147	0.119557
a_4	-0.554104	-0.111160

Leading Edge Radius is given by $(a_0^2 \times t^2)/2$

Upper and lower profiles are then derived by

$$y_u = y_m + y_b$$

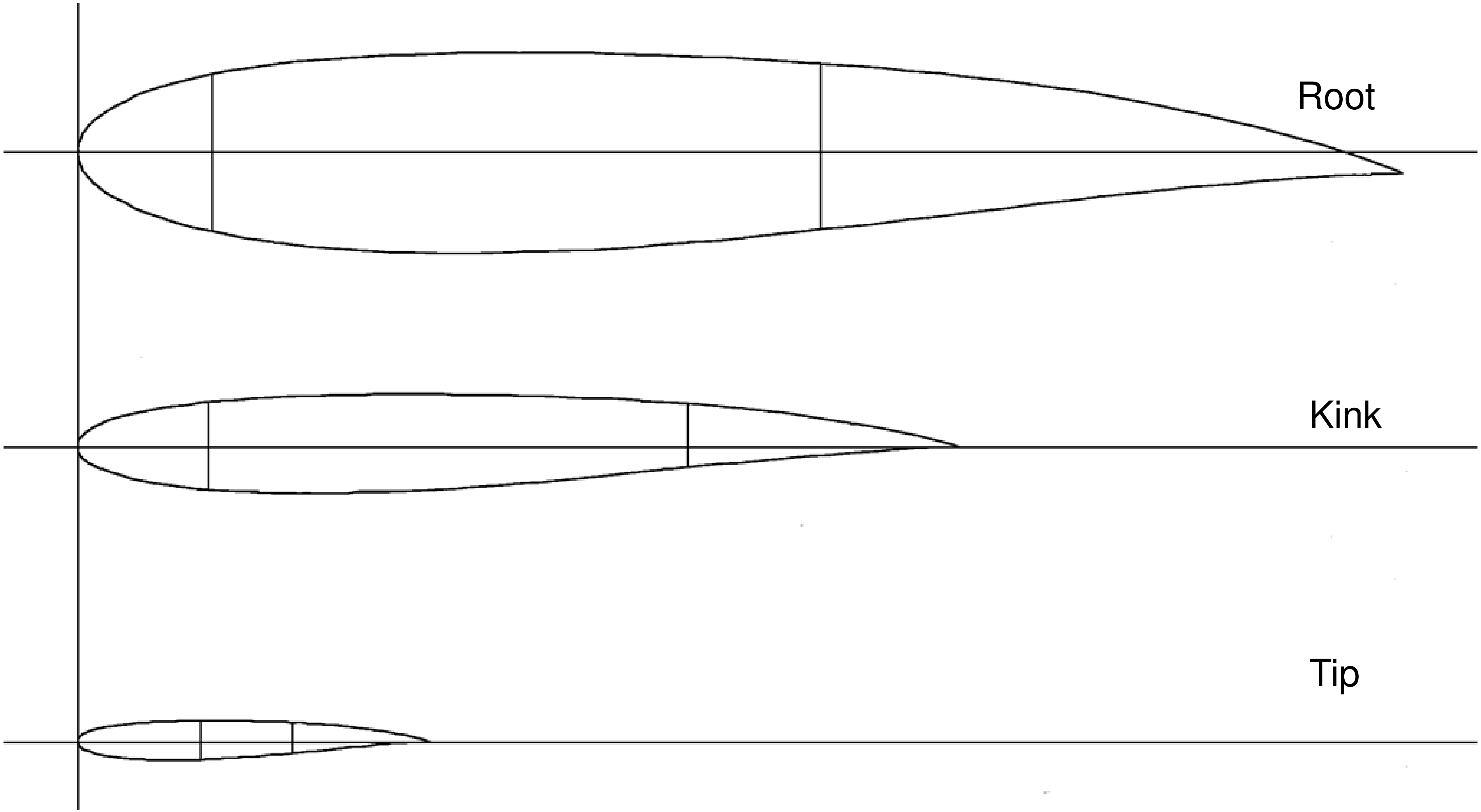
$$y_l = y_m - y_b$$

The method given in Abbott & Doenhoff for combining the base section and the mean line involves the local slope of the mean line (see equation 6.1), this was ignored in the derivation of the profiles.

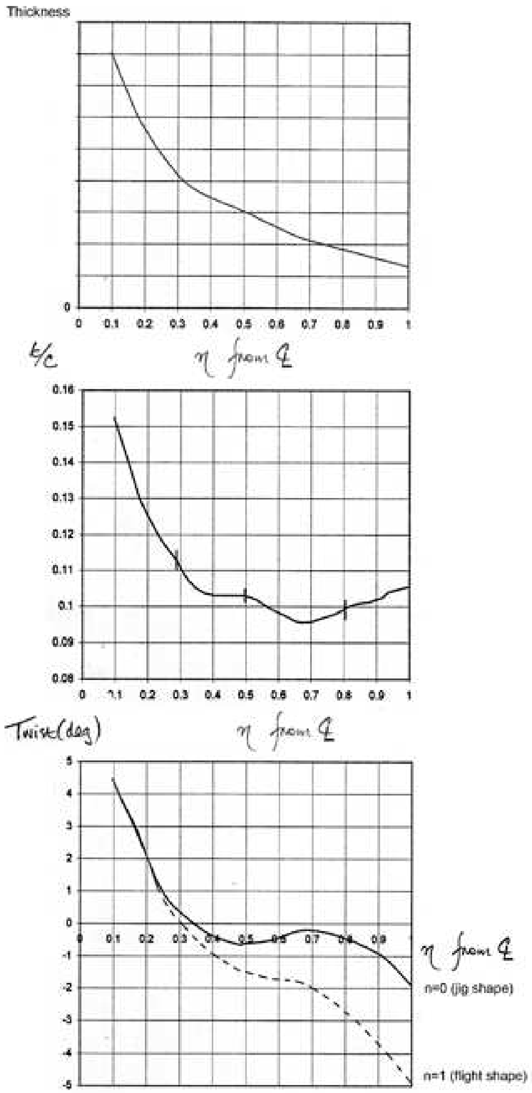
Note that $\cos 5^o$ is 0.9962
$\sin 5^o$ is 0.0872
$\tan 5^o$ is 0.0875

Wing Sections

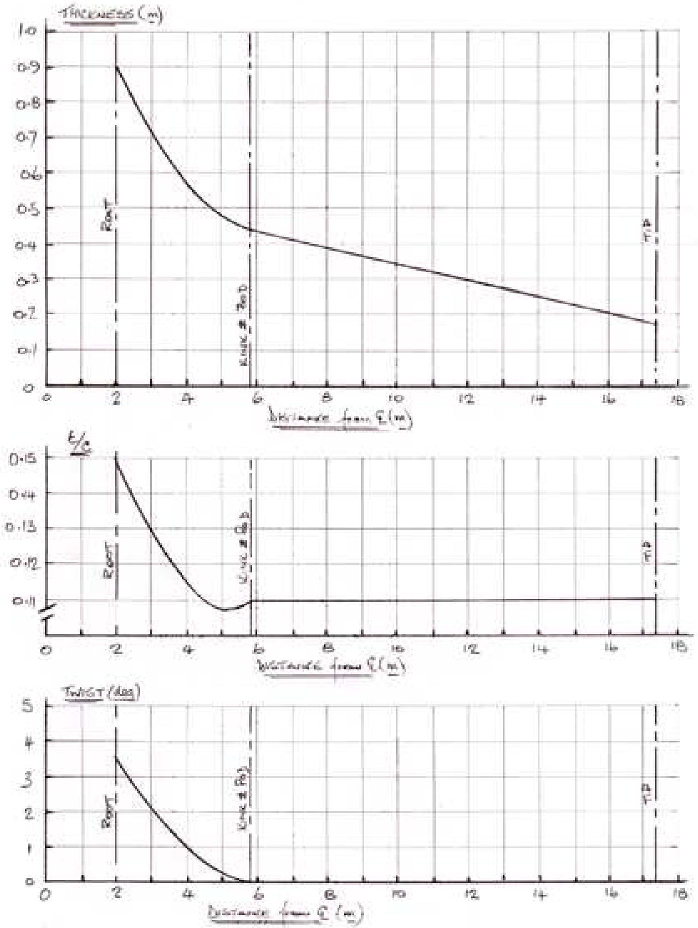
Shows Front and Rear Spar positions



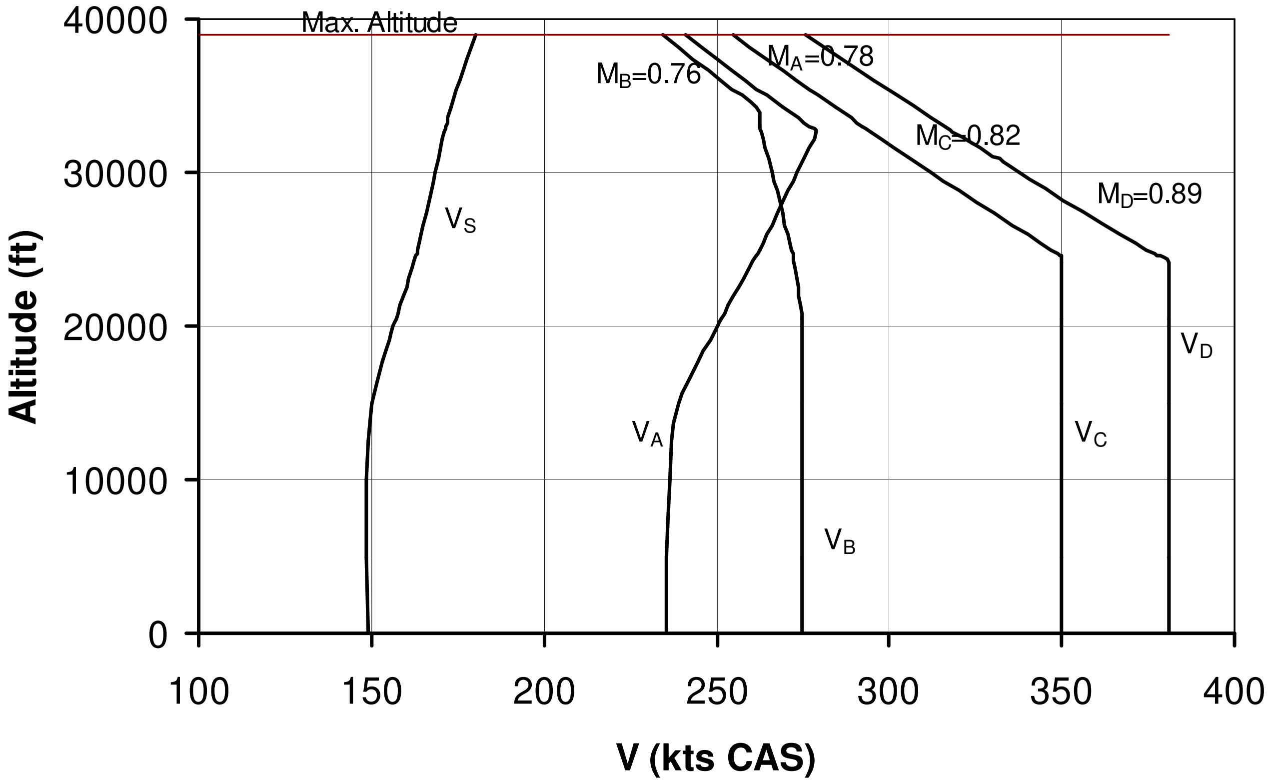
Modern Airliner



Wing Design Exercise

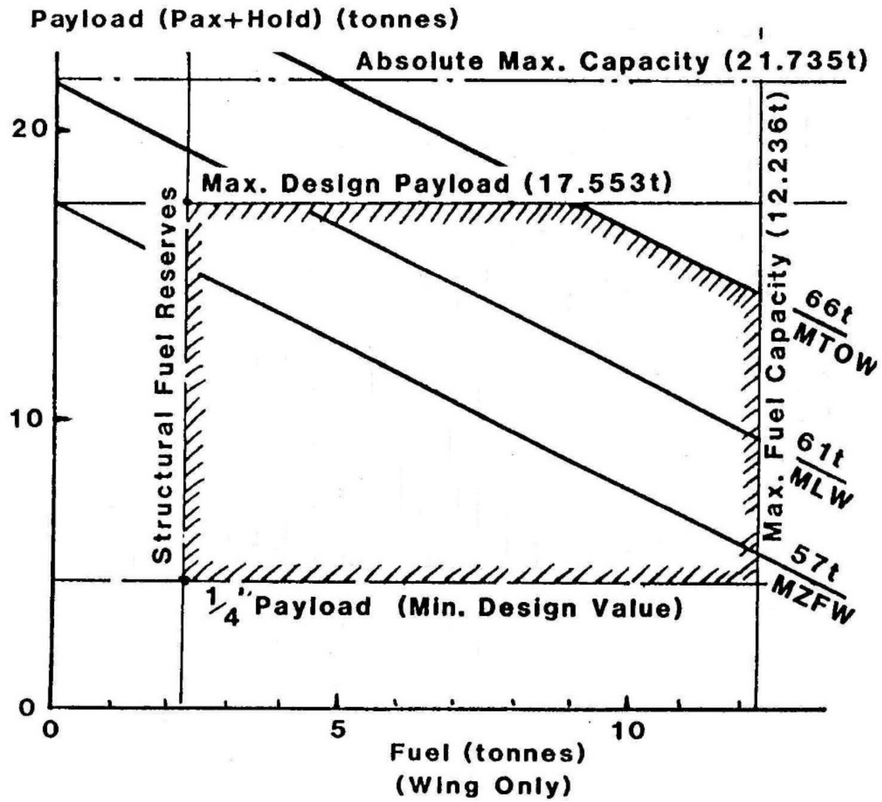


Design Speeds



A320

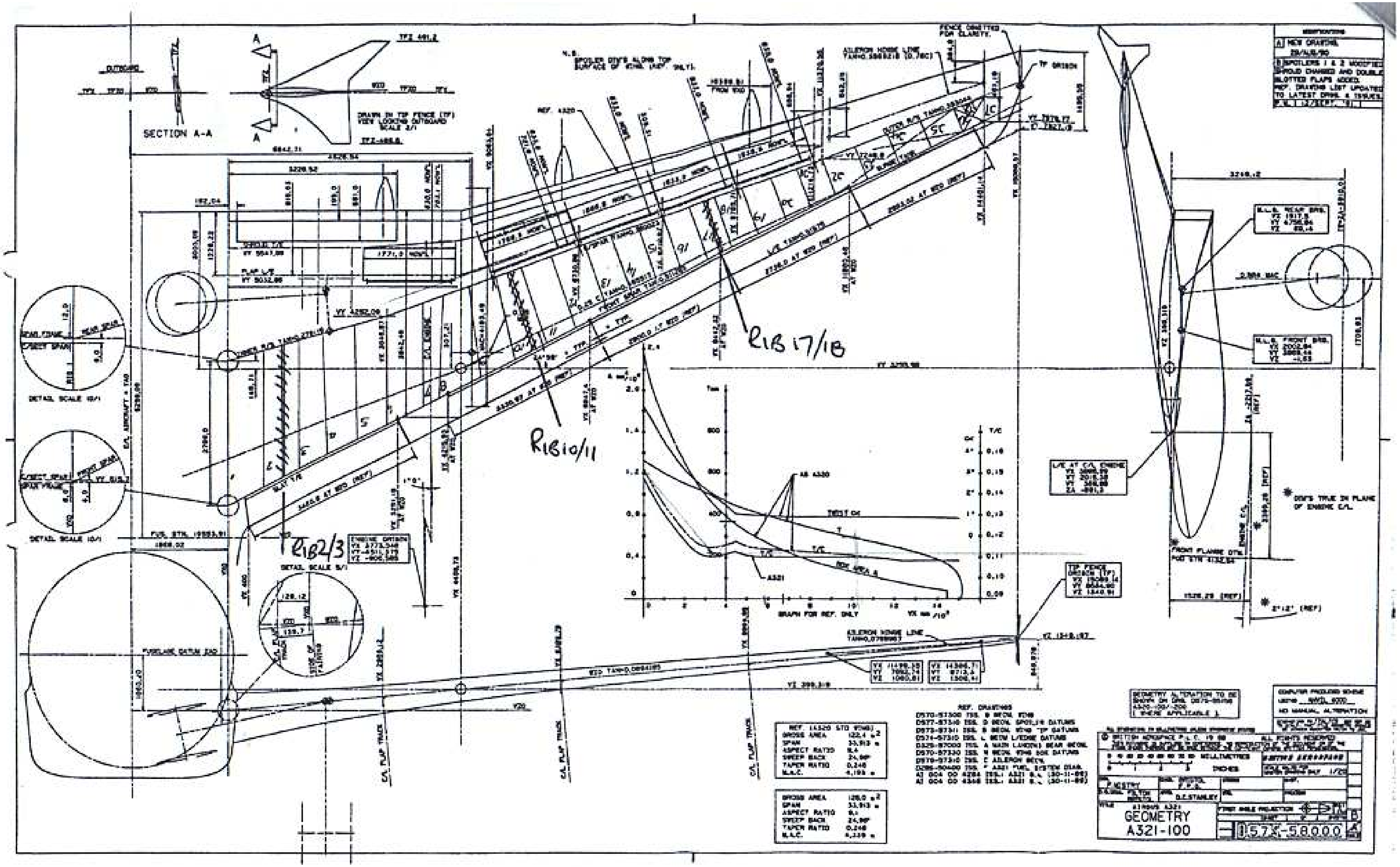
PAYLOAD - FUEL LOADING



NOTES

- Passengers (75kg) and Baggage (15kg)
  165 X (75 + 15) = 14850 kg
  Max Hold Capacity = 9435 kg
- Structural Fuel Reserves - Fuel required to climb to 10000 ft and then cruise at speed appropriate to that for maximum range for 45 mins
- Payload loaded from front or back to give CG range
- Payload for Fatigue Mission is between 65% and 80% of maximum Design Payload

A320


SECTION A-A
RIB 17/18
RIB10/11
RIB2/3
GEOMETRY
A321-100
D57X-58000

MISSION COMPARISON

	MAXIMUM DESIGN WEIGHTS	DESIGN MISSION	FATIGUE MISSION
OWE (kg)	39447	39447	39447
Payload (kg)	17553	15580	11410
ZFW (kg)	57000	55027	50857
Fuel (reserve) (kg)	4000	3109	2543
Landing Weight (kg)	61000	58136	53400
Fuel (block) (kg)	5000	7864	2600
TOW (kg)	66000	66000	56000
Fuel (max) (kg)	12200		
Range (nm)		1500	370
Cruise Altitude (ft)		35000	31000
Cruise Mach No		0.76	0.78
Flight Time (mins)		219	60
Block Time (mins)		233	74

A320

Mass Configurations

	AUW	Fuel	Payload	
Mission 1	66000	10973	15580	Single class 164 pax @ 95kg
Mission 2	66000	12303	14250	2 class 150 pax @ 95kg (Wing Tanks Full)
Take-off 1	66000	9000	17553	Max Payload
Take-off 2	66000	12303	14250	Max Fuel
Landing 1	61000	4000	17553	Max Payload
Landing 2	61000	7130	14423	High Fuel load
Flight 1	59250	2250	17553	Max P/L + Str Res
Flight 2	46085	2250	4388	quarter P/L + Str Res
Flight 3	64130	7143	17553	Max P/L less fuel burn to altitude
Flight 4	57000	0	17553	Zero Fuel, Max P/L
Flight 5	52002	7130	5425	Not sure of the reasoning for this case
Flight 6	43835	0	4388	Zero Fuel, quarter P/L
Flight 7	40447	1000	0	Min Fuel for ferry flight
Flight 8	64130	10433	14250	Max Fuel less fuel burn to altitude

Masses in kg

OWE= 39447kg

A320

Each case can be loaded from front or back giving rise to forward CG or Aft CG

OWE Weight Breakdown

MTOW = 66000kg

		Fuselage	Wing C/S	Wing	Pylon	Engine	Fin	Tailplane		TOTAL
Structure	Wing		1020	7569						8589
	Fuselage	8140	4							8144
	Tailplane							606		606
	Fin						429			429
	MLG	1153		790						1943
	NLG	324								324
	Pylon				896					896
	Total	9617	1024	8359	896	0	429	606		20931
Power Unit	Engine					6818				6818
	Bleed Air	72		54	94					220
	Controls	13		3	5					21
	Fuel System	16	18	190	20					244
	Total	101	18	247	119	6818	0	0		7303
Systems	APU	213								213
	Hyd Gen	366		59	62					487
	Hyd Dist	304		107	12		12	12		447
	Air Cond	660								660
	De-Ice			39						39
	Fire Pre	31		5	70					106
	Flt Contol	340		328			49	32		749
	Instruments	108								108
	Auto FS	130					13			143
	Nav	458					7			465
	Com	161		2						163
	Elec Gen	341		18	12					371
	Elec Dist	986	3	100	56			4		1149
	Total	4098	3	658	212	0	81	48		5100
Furnishing	Furn	2418								2418
	Oxygen	144								144
	Light	207		19						226
	Water	88								88
	Total	2857	0	19	0	0	0	0		2876
MWE		16673	1045	9283	1227	6818	510	654		36210
Operators	Items	3087		73		77				3237
OWE		19760	1045	9356	1227	6895	510	654		39447

A320

Typical Wing Weight Breakdown

The following data may be used to establish the potential weight savings that might be obtained using modern materials in the construction of a medium sized civil transport wing.

1 Datum wing.

The mass of a typical civil transport wing around the size of say an A310 is in the range 80~90 kg/m^2
The materials used in a typical wing of that era are shown on Fig. 57 of the Airworthiness and Loads Lecture.

2 Overall mass breakdown

The wing mass can be typically split as follows :

Total upper skins (inc. centre section)	17.5	%
Total lower skins (inc. centre section)	22.0	%
Total spars (inc. centre section)	12.0	%
Total ribs (inc. centre section)	8.0	%
Fixed LE	6.0	%
Slats	6.0	%
Fixed TE	3.5	%
Flap surfaces & fairings	8.0	%
Flap tracks etc.	12.5	%
U/C support structure	2.5	%
Engine mounting reinforcement	2.0	%
Total	100.0	%

3 Typical unit masses

Outside the main wing torque box, typical structural masses for aircraft in this category are :
Fixed LE 20~30 kg/m^2
Moveable LE 30~38 kg/m^2
Fixed TE (composite) 10 kg/m^2
Flap system (total surfaces supports & fairings) 55~65 kg/m^2 flap area
Spoilers (composite) 12 kg/m^2
Ailerons (composite) 24 kg/m^2

Typical Fuselage Structural Weight Breakdown

This data is based on civil passenger aircraft with the Engines mounted on the wing and the Main Landing Gear pintle point on the wing (typically A320, A310, B737 etc)

Skin & Stringers	33	%
Frames	8	%
Access Doors & Apertures	10	%
Cabin Windows	3.5	%
Nose & Tail Fairings	1.5	%
NLG Installation	3	%
MLG Installation	3	%
Cabin Floor Structure And Panels	16	%
Freight/baggage holds Including Floor)	6	%
Tail Unit Attacment Structure	3.5	%
Windscreen (Frame and Glazings)	2	%
Pressure Bulkheads	2.5	%
Fuselage/Wing Attachment	5	%
Paint/Sealant	1.5	%
APU Attachment	0.3	%
Miscellaneous/Special Features	1.2	%
TOTAL	100	%

Suggested Loads Cases that you should be aware of

Loads Cases Aircraft Design Project

The list of cases that follows might not give the worst case for the structure but should be within about 5%. We are looking for understanding of the principles involved and getting close to the worst case.

Don't forget to apply the appropriate Ultimate Factor (use 1.5 unless specified)

Wing Design Cases (Flight)

Consider n= 2.5g, 1.0g and -1.0g trimmed Manoeuvres with full wing fuel at MTOW.
This is not probably the worst case but will give a good indication and save a lot of calculations for varying fuel and payload states.
Assume Gust cases will be equal to or less than Manoeuvres due to the use of Load Alleviation devices.

Fuselage Design Cases (Flight)

Use Mission Payload uniformly distributed (will give a middling CG). Consider n= 2.5g, 1.0g and -1.0g trimmed Manoeuvres. Worst Case likely to be a Gust with Max Payload at a low fuel load or low payload at forward or aft CG with low fuel.

Suggested Loads Cases that you should be aware of

Wing Design Cases (Ground)

Depends on aircraft configuration and time available whether they are considered for the Design Project

Fuselage Design Cases (Ground)

The landing cases can give rise to some large vertical g's in dynamic conditions. One way of assessing the loads is as follows. Landing Case is 1.0g flight condition prior to impact plus the increment due to impact (Use next fig for g along the fuselage – it already includes the 1.0g for trimmed flight).

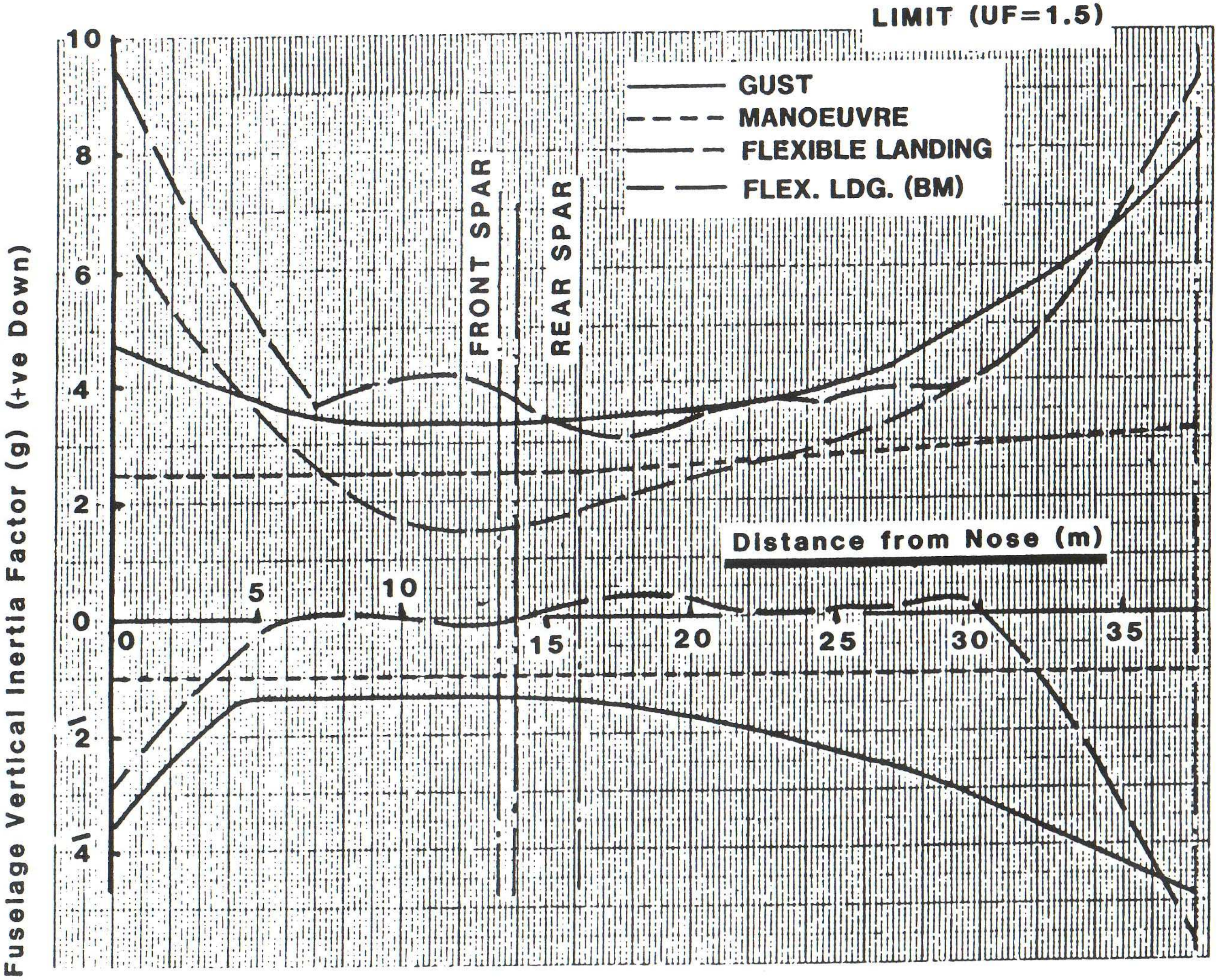
Tailplane

The loads on the tailplane result from the Trimmed state (2.5g, 1.0g and -1.0g), Pitching Manoeuvres and Vertical Gusts (down gust will be critical).

Fin

Critical cases are Lateral Gust and Rudder Manoeuvre. While Engine Failure might size the fin and rudder in terms of area, it is not normally a critical strength case.

Fuselage Vertical Inertia Factor



A320

Suggested Loads Cases that you should be aware of

Landing Gear Design Cases

Critical cases are those that give the maximum fore/aft, side and vertical loads (plus moment cases for torque links and steering.

Don't forget to derive and assess the attachment loads of the landing gear to the wing or fuselage.

For the MLG these are

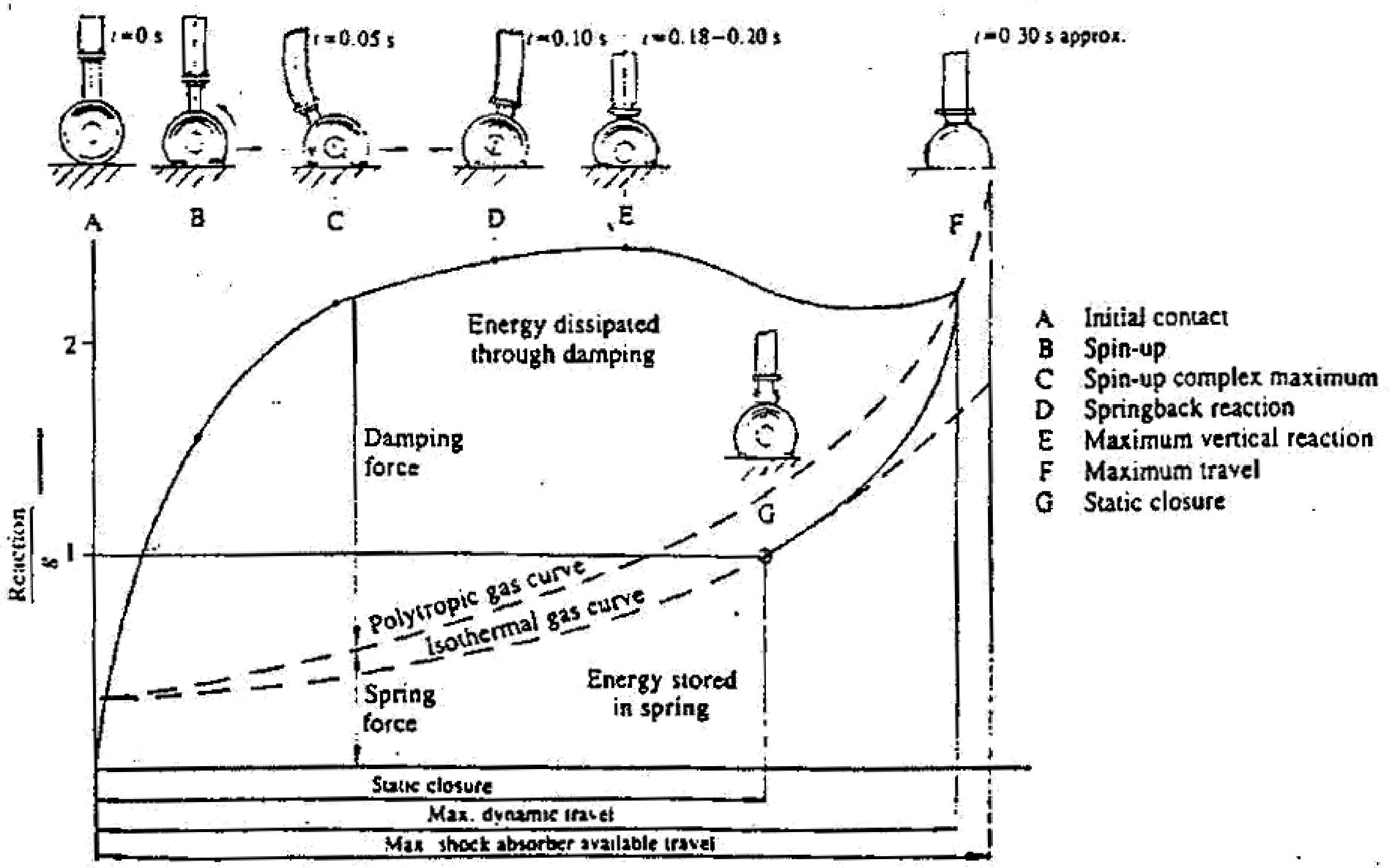
- Take-off
- Braked Roll (max. drag load)
- Zmax
- Spring Back (max. forward load)
- Turn Case (max.side load)
- Pivot Case (max. moment for torque links)

Note that a 60/40 distribution is used between tyre/wheels (see CS23.511 Amdt 4), CS25.511 has a more rational analysis

For the NLG these are

- Braked Roll (No drag Load)
- Towing (consider push and pull)
- Nose Wheel Yaw (only consider para(a)of CS 25.499, that is 1g vertical and 0.8*Z laterally)
- Zmax

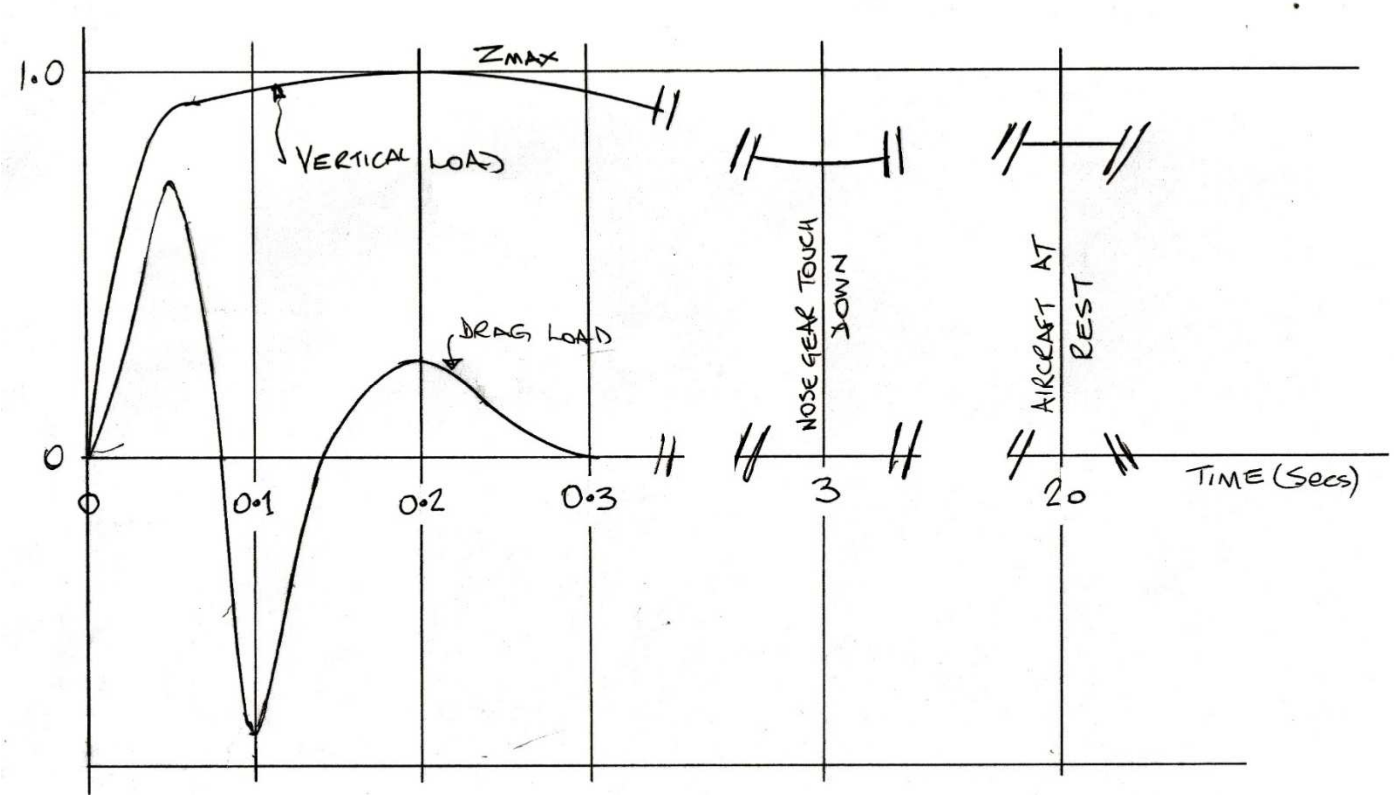
Landing Gear Load Sequence



Significant events during the landing cycle

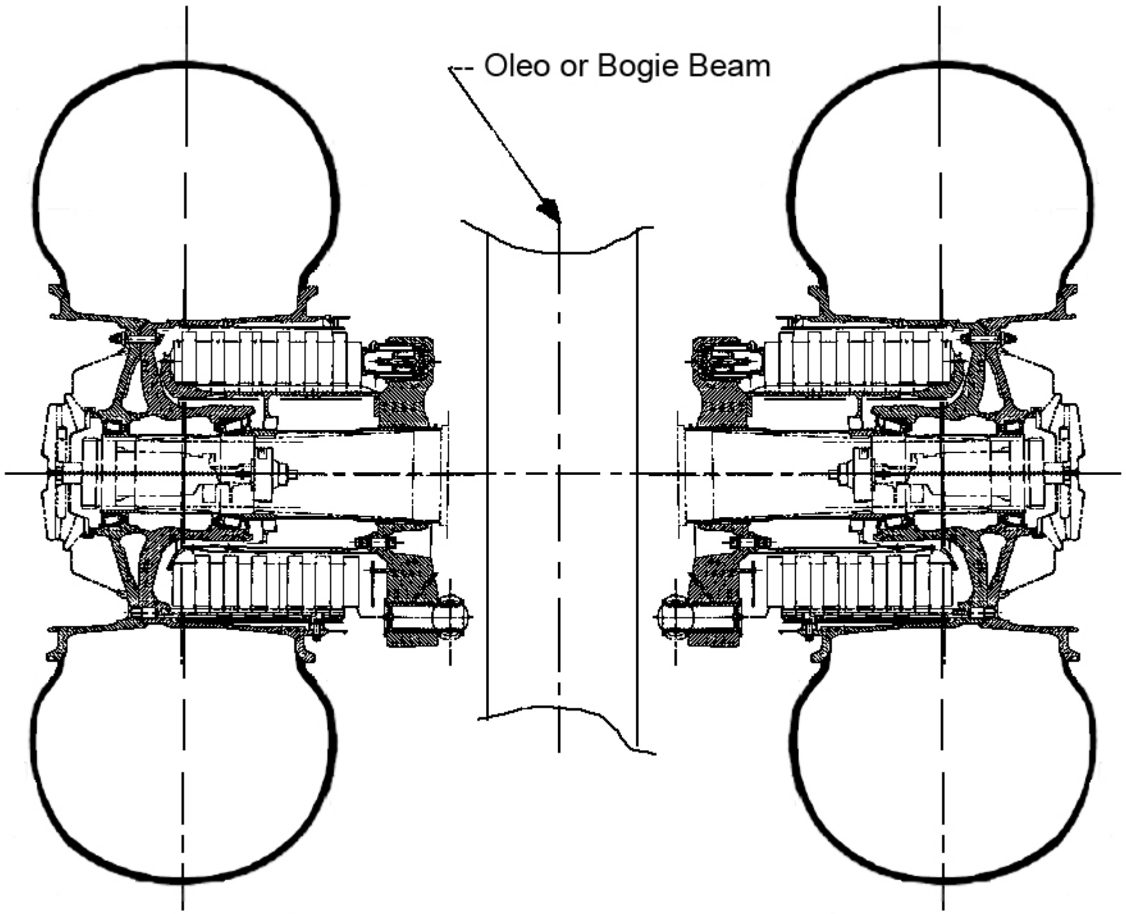
Landing Gear Time History

The Reaction Factor (λ) is the ratio of the Peak Vertical Load (Zmax) during the landing impact to the 1g Static Load at Maximum Landing Weight. For a light aircraft it can be as high as 3.0, for something like an A320 or larger, industry tends to use something like 1.25



Wheel, Tyre and Brakes

Rim Dia = 20 inches
Energy/Wheel = 70 MJ
This is for a RTO at MTOW
Brakes on speed is typically
85 to 90 m/s



Lateral Spacing of Wheels is determined
by the ACN requirements and the need
to accommodate the brakes

Suggested Loads Cases that you should be aware of

General

There are many other design cases that need to be considered for full certification (see figs 21, 22, and 23 of Airworthiness and Loads handout).

You should be aware that some of these cases can only be acted on when the design has progressed further.

At this stage you should certainly consider cabin pressurisation (the following values cover short range to long range aircraft). Values in mb

Max Nominal Operating Differential Pressure	556 ± 7	to	605 ± 7
Relief Valve Setting (actuating)	590 ± 7	to	635 ± 7
Max Relief valve overpressure (max flow)	621 ± 7	to	665 ± 7
Max Negative Differential Pressure	-70		
Stressing Values			
– Fatigue	556	to	605
- Static (relief valve setting)	597	to	642

Engine and Pod Loads (TurboFan)

The g and thrust values that are given in this section are applied to the combined mass of the Engine and Pod, at the combined CG.

Wing Mounted

Condition	Ult Factor	Fx	Fx	Fy	Fz
		Thrust	Mass	Mass	Mass
Flight	1.5	MCP or Zero		±3g	1g
	1.5	MCP or Zero			±3g
Landing	1.5	Idle or Zero			+3g
	1.5	Idle or Zero			-1g
Max Thrust	1.0	2 x Max TO			1.5g or Zero
	1.0	2 x Reverse			1.5g or Zero
Blade Loss	1.0				

Note: Should be aware of Wheels up Landing but not enough data available to derive

Rear Fuselage Mounted

Condition	Ult Factor	Fx	Fx	Fy	Fz
		Thrust	Mass	Mass	Mass
Flight	1.5	MCP or Zero		±3g	1g
	1.5	MCP or Zero			+5g
	1.5	MCP or Zero			-3g
Landing	1.5	Idle or Zero			+4g
	1.5	Idle or Zero			-2g
Max Thrust	1.0	2 x Max TO			1.5g or Zero
	1.0	2 x Reverse			1.5g or Zero
Blade Loss	1.0				
Ground	1.0		12g (30° Cone)		

Note: Take Reverse Thrust as 40% of Forward Thrust.
Blade Loss gives an out of balance rotating system (probably ignore at this stage)

Engine and Pod Loads (TurboProp)

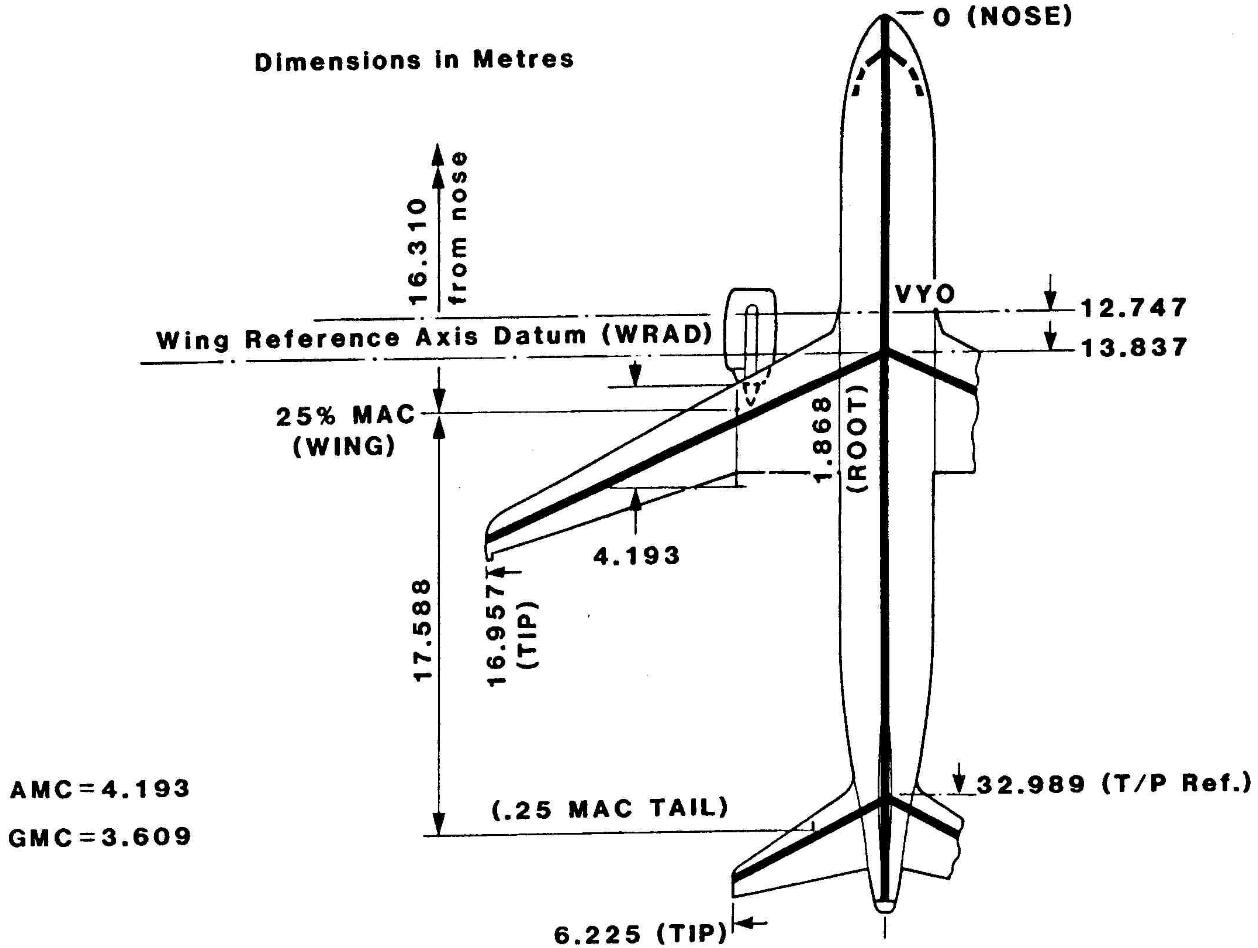
The g and thrust values that are given in this section are applied to the combined mass of the Engine, Gearbox, Propeller and Nacelle, at the combined CG.

Wing Mounted

Condition	Ult Factor	Fx	Fx	Fy	Fz	Mx
		Thrust	Mass	Mass	Mass	Torque
Flight	1.5	MCP or Zero		±3g	1g	
	1.5	MCP or Zero			±3g	
Landing	1.5	Idle or Zero			+3g	
	1.5	Idle or Zero			-1g	
Max Thrust	1.0	2 x Max TO			1.5g or Zero	
	1.0	2 x Reverse			1.5g or Zero	
Torque	1.5				1g	2 x Mean Torque
"Failsafe"	1.0	0.5 x Max TO		±2g	2g	Mean Torque
"Fatigue"	1.0				1g	Mean Torque

Note: Take Reverse Thrust as 40% of Forward Thrust.
The gearbox for the propeller is often supported by struts or rds from the gas generator. The failure of these rods has to be considered (take the failure of 1 or 2 rods depending on design).
Fatigue life to be taken as 20000 flights.

Aircraft Reference Axes



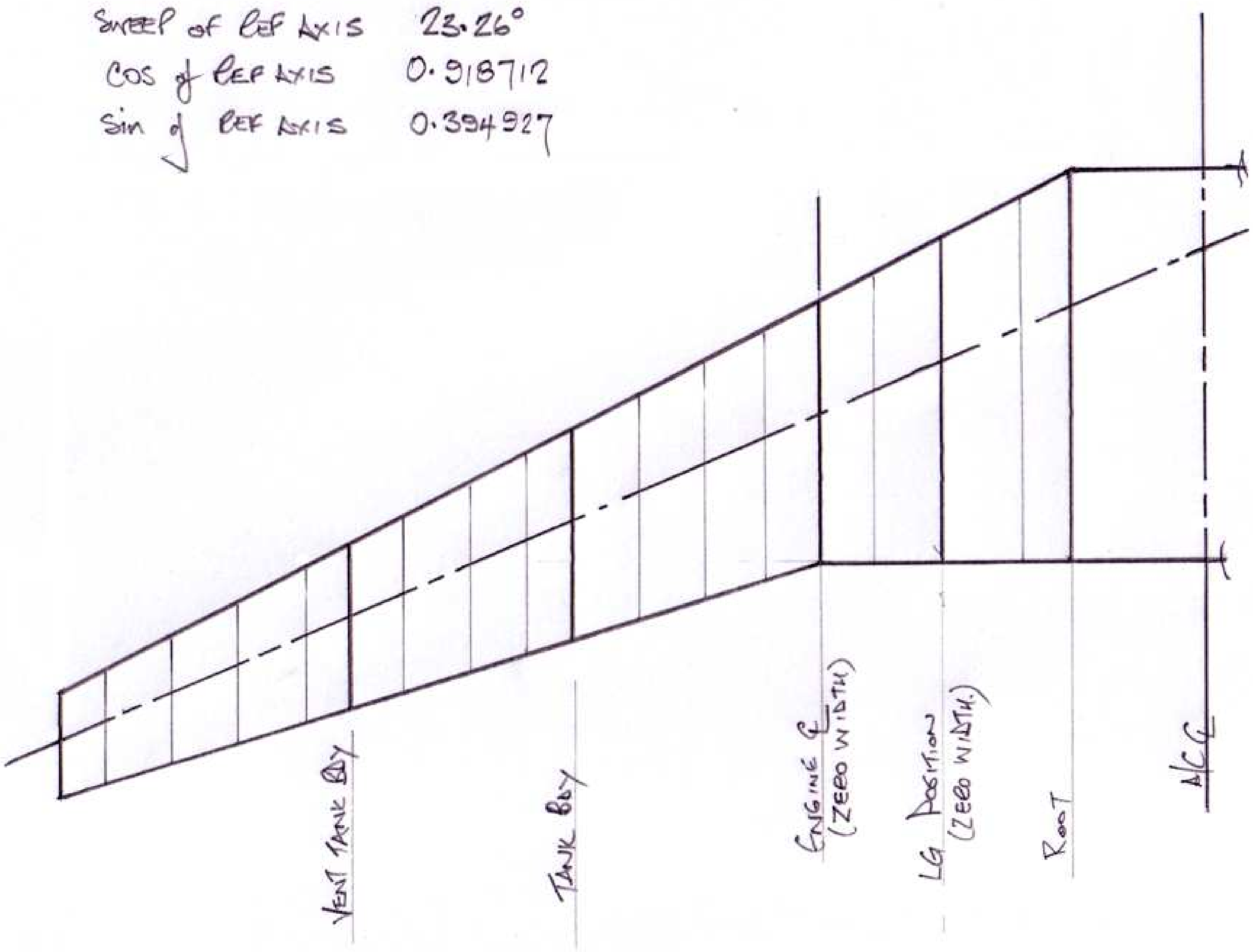
A320

Aircraft Reference Axes for Loads

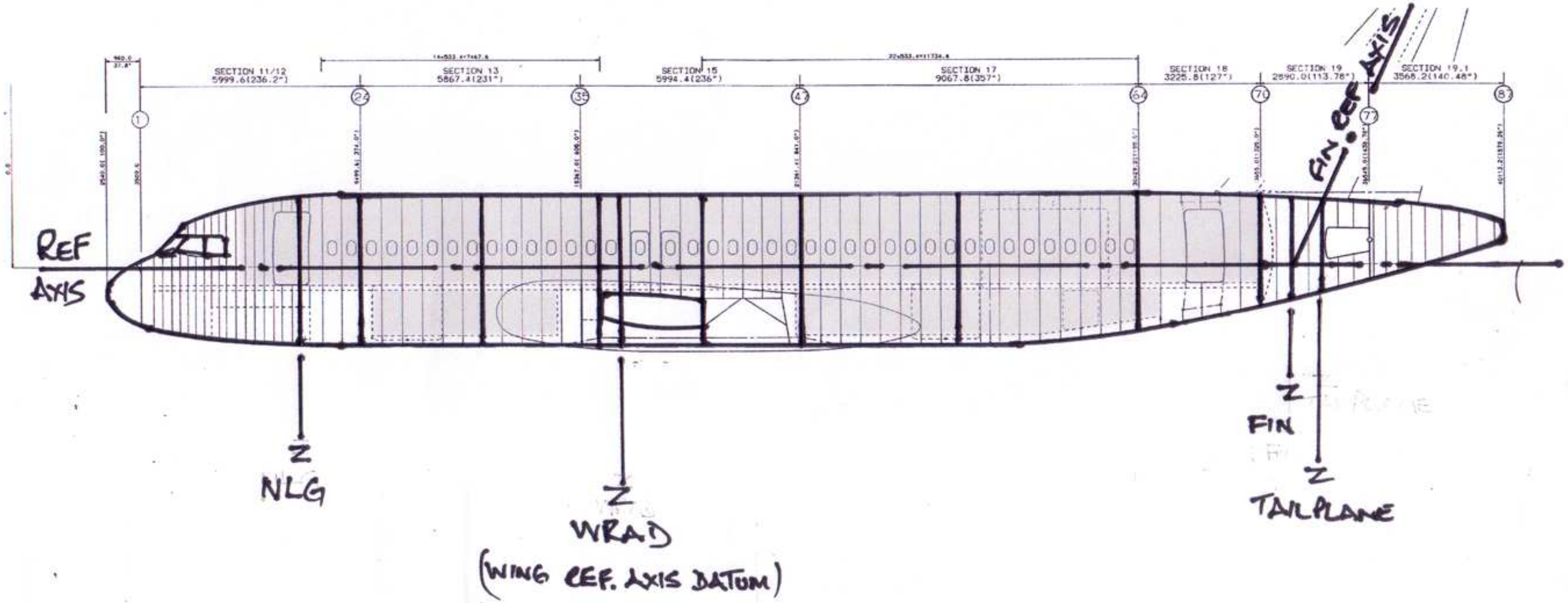
Fuselage	Straight line through Horizontal Fuselage Datum
Wing	Straight line in the Wing Plane (assume ‘Flat’ Wing) through 40% Chord at Wing Root and 40% Chord at Wing Tip (extrapolate to Aircraft Centre Line)
Tailplane	Straight line in the Tailplane Plane through 40% Chord at Root and Tip (extrapolate to Aircraft Centre Line)
Fin	Straight line in the Fin Plane through 40% Chord at Root and Tip (extrapolate to Horizontal Fuselage Datum)

These are suggestions, depending on the layout of the aircraft, other values may be appropriate

Wing Strip Layout



Fuselage Strip Layout



Payload and Fuel information

Payload

The specification says that the pax weight is 100kg including a check-in baggage of approx 40lb (18kg).
We can assume that 100-18 = 82kg/pax is in the cabin either on seats or in the overhead lockers while the 18kg/pax is in the hold.
The 82kg/pax can be uniformly distributed over the cabin length for this exercise (that will include any galley toilet areas).
The 18kg/pax will be uniformly distributed over the hold length (which will be less than the cabin length)

NOTE: For simplicity, it would be reasonable to distribute the 100kg/pax over the whole length of the cabin with effectively nothing in the holds.

Fuel

The fuel volume can be calculated using the GROSS volume of the tanks and then allowing for airspace and structure. Usually this is 10%. Hence NETT fuel volume is 90% of gross.
Experience suggests that fuel volume of an outer wing tank will be 85% of gross volume. Inner wing tank 90% and the centre tank within the fuselage is 92%
Don’t forget to apply the appropriate fuel density to derive the actual mass.
Fuel density can vary between 743 to 851 kg/m3. The nominal or typical value is 785 kg/m3 and this value should be used.


A320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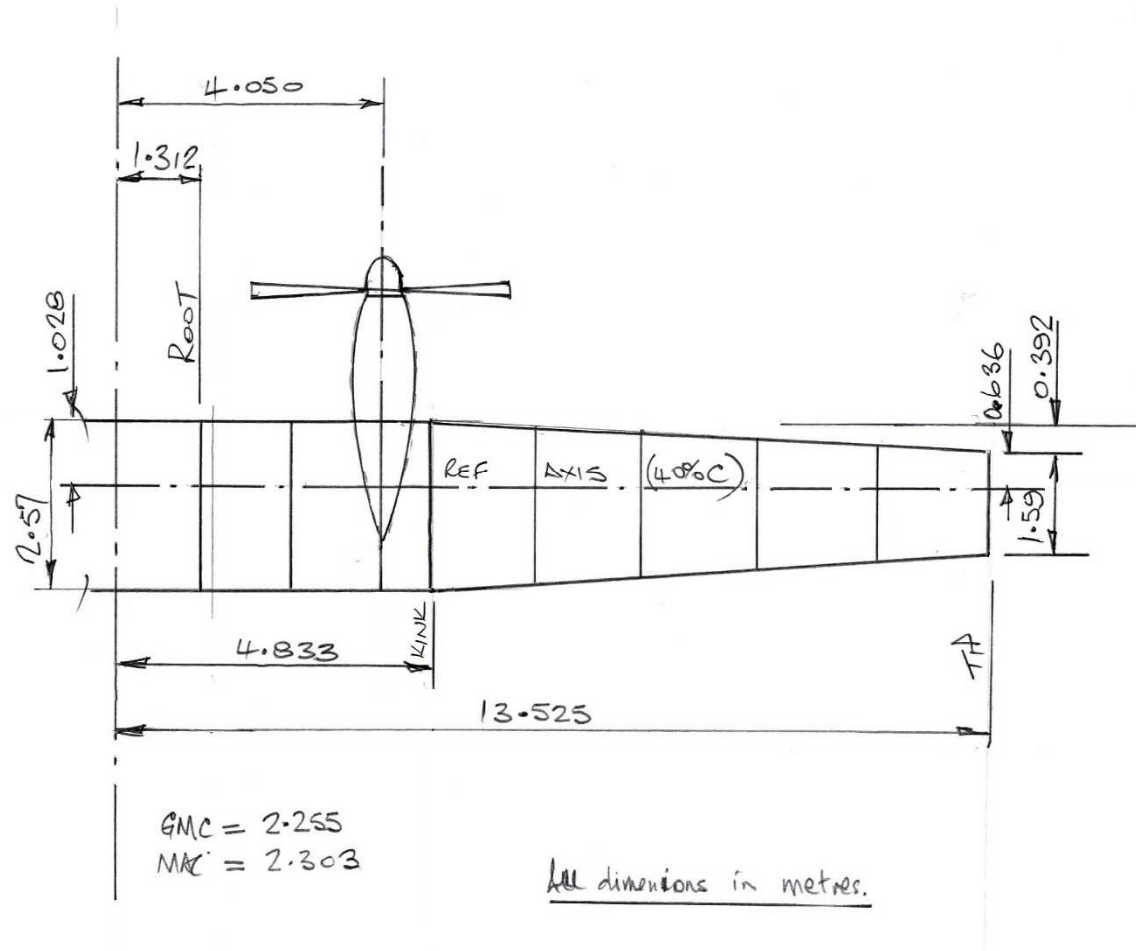
Wing Geometry



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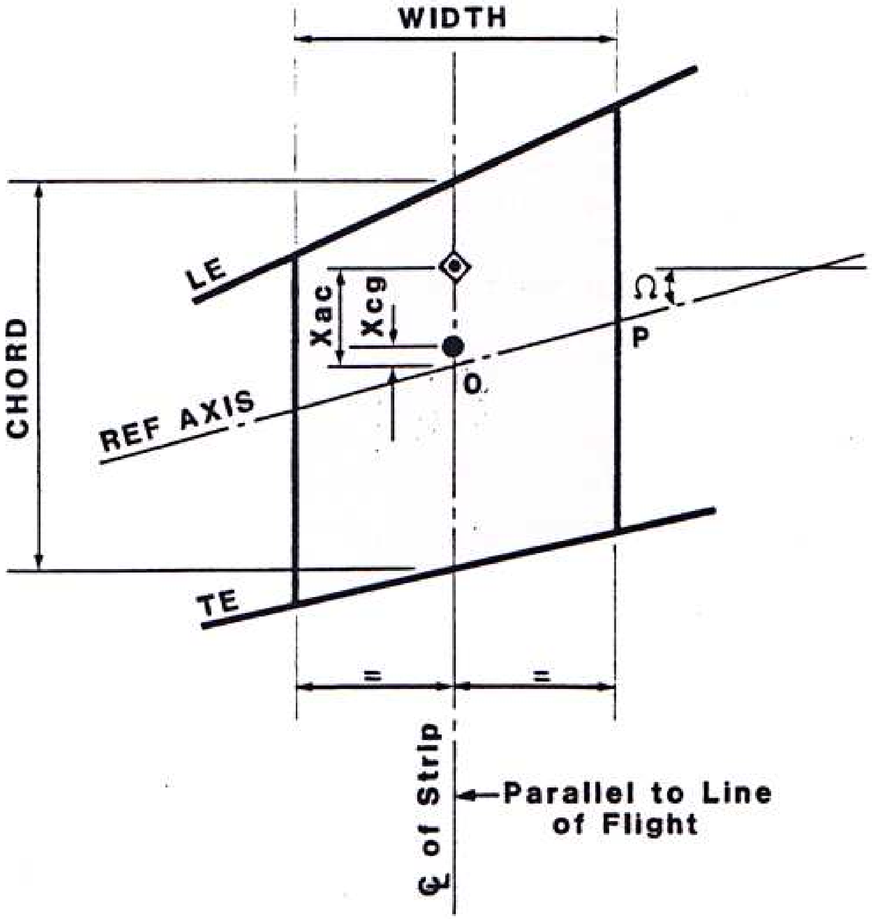
Strip Definition and Forces on Strip



TOTAL LOAD ON STRIP

$$F_S = -W_S \times n_S + (L_{S0}/qS)\, qS + (L_S/L_{WFP})\, L_{WFP}$$

MOMENT (PITCHING) ABOUT O (INTERSECTION OF STRIP ℄ AND REF. AXIS)

$$M_{PS} = -W_S \times n_S \times x_{cg} + (L_{S0}/qS)\, qS \times x_{ac} + (L_S/L_{WFP})\, L_{WFP} \times x_{ac} + C_{M_{0_L}} \times \text{width} \times \text{chord}^2 \times q$$

FORCES AND MOMENTS AT P (INNER BOUNDARY OF STRIP)

$$S_A = F_S$$
$$M_R = F_S \times \text{width}/2$$
$$M_P = M_{PS} - F_S \cdot \tfrac{w}{2} \cdot \tan\Omega$$

IN LINE OF FLIGHT OR A/C AXES

TRANSFORM TO REFERENCE AXES

$$S_{REF} = S_A$$
$$M_{REF} = M_R/\cos\Lambda - M_{PS}\sin\Lambda$$
$$T_{REF} = M_{PS}\cos\Lambda$$

Strip Definition and Forces on Strip

Assuming a flat wing (no dihedral) and tan, sin & cos refer to the sweep of the reference axis.

Taking the definitions of FS and MPS as given for the force and moment at the centre of the strip.

Values at the INNER boundary in line of flight will be

$S_A=F_S$
$M_R=F_S*(w/2)$
$M_P=M_{PS}- F_S *(w/2)*tan$

Convert to reference axis, the values at the INNER boundary will be

$S_{REF}=S_A$
$M_{REF}=M_R*cos-M_P*sin$
$T_{REF}=M_R*sin+M_P*cos$

Substituting for M_R and M_P

$M_{REF}= F_S*(w/2)*cos- M_{PS}*sin+F_S *(w/2)*tan*sin$
$T_{REF}= F_S*(w/2)*sin+ M_{PS}*cos-F_S *(w/2)*tan*cos$

Re-arranging the equations

$M_{REF}= F_S*(w/2)*(cos+sin*sin/cos)- M_{PS}*sin$
$(cos+sin*sin/cos)=1/cos$

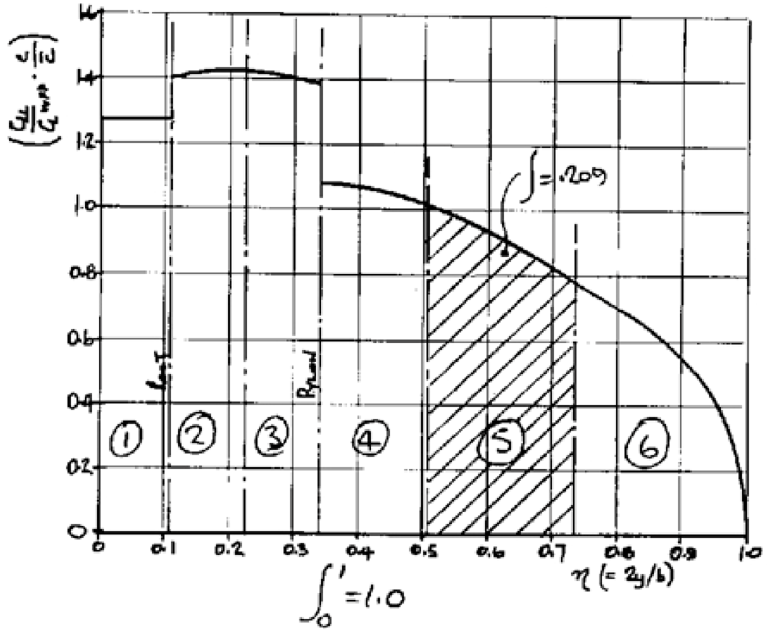
$T_{REF}= F_S*(w/2)*(sin-sin*cos/cos)+M_{PS}*cos$
$(sin-sin*cos/cos)=0$

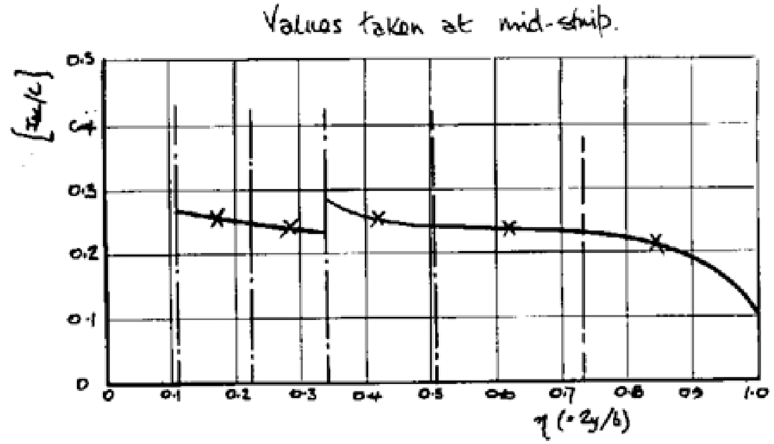
Which gives the final equations

$S_{REF}=S_A$
$M_{REF}= M_R/cos- M_{PS}*sin$
$T_{REF}= M_{PS}*cos$

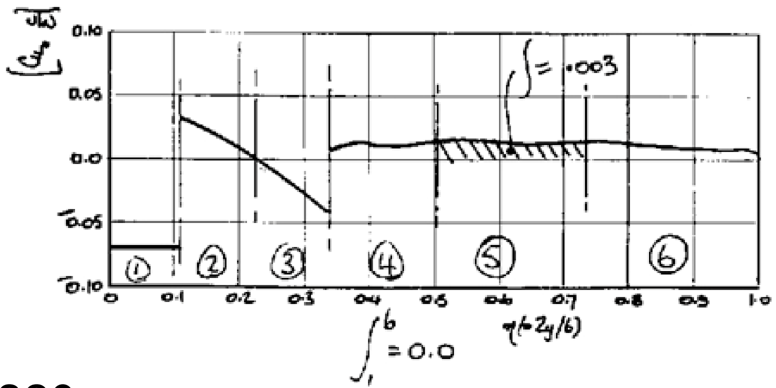
These values at the INNER boundary of the strip have to be added to the values from the OUTER boundary (or from the previous strip)

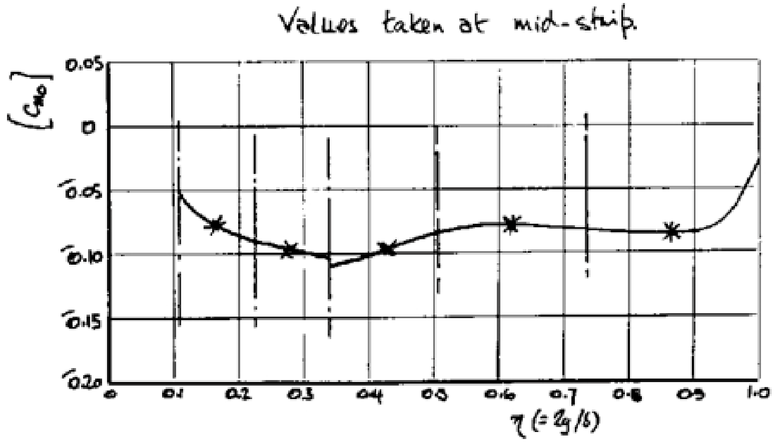
Aerodynamic Data - Wing





WING AERODYNAMIC DATA (UB90)





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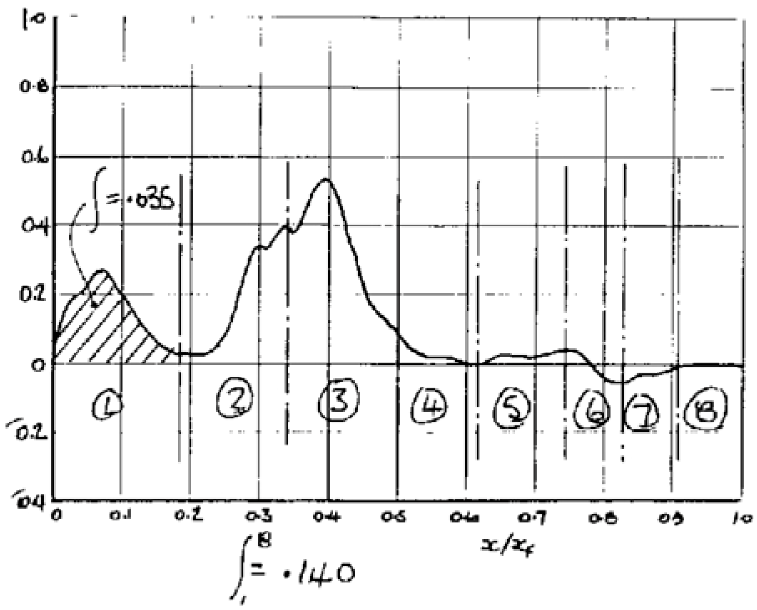
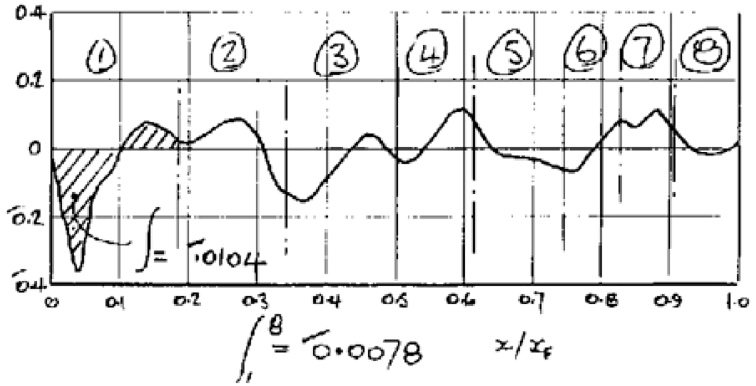
Aerodynamic Data - Fuselage

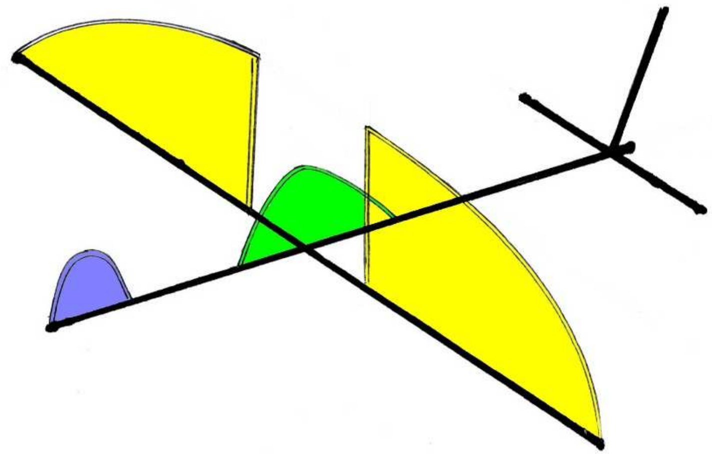


FUSELAGE AERODYNAMIC DATA (UB90)



A320

Aero Lift Distribution



Wing + Fuselage Lift = 100 ($L_W + L_F = 100$)

Typically $L_W = 86$ & $L_F = 14$

Lift on Fuselage is in 2 parts (Wing Carry over and Nose contribution)

$L_{F(NOSE)} = 4$ and acts over approx 1.5 fuse diameters.

$L_{F(CARRY\ OVER)} = 10$ and acts over 1.5 Wing Root chords centred on 1/4 Chord (Root)

Schrenk's Method

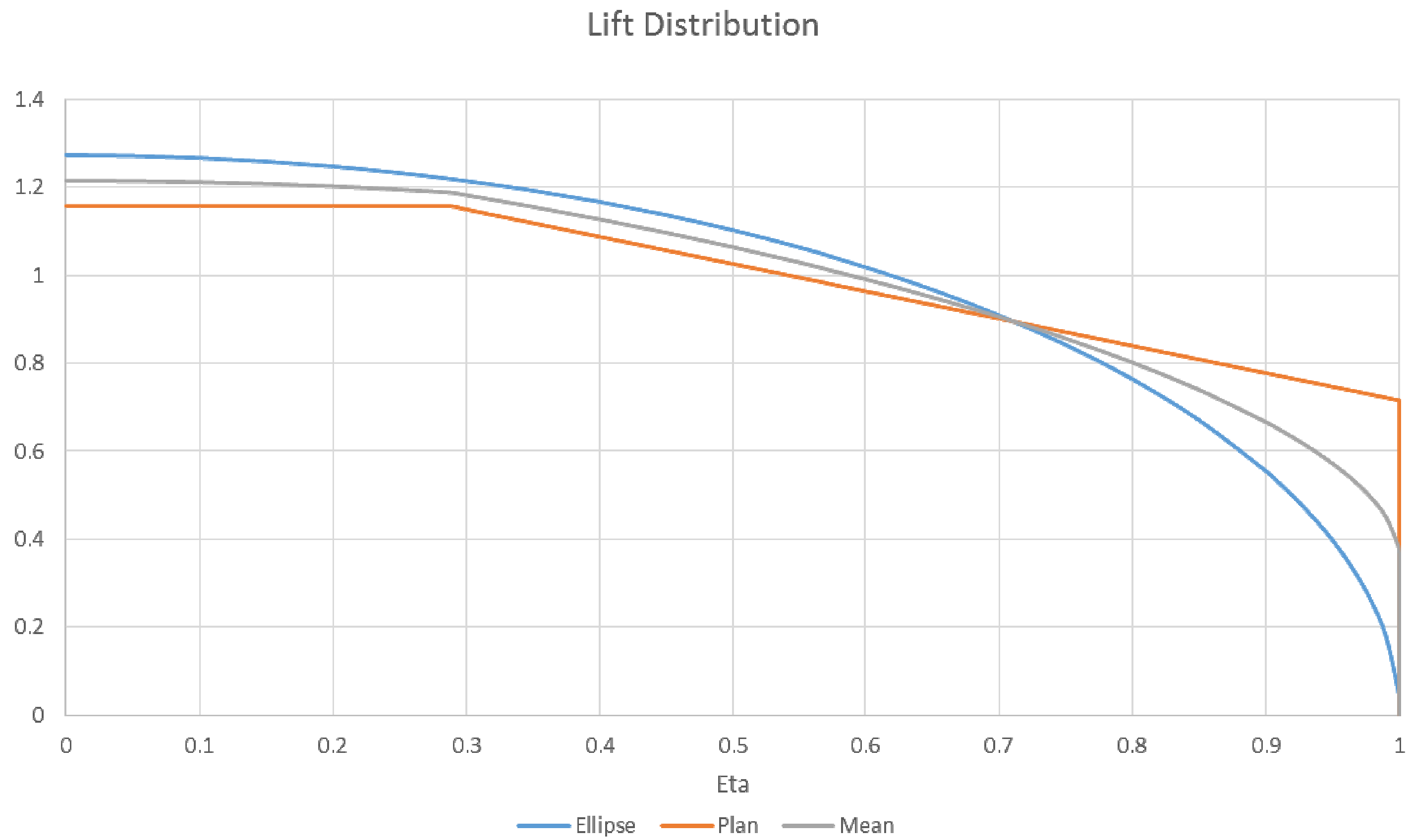
- O. Schrenk, (1940), A Simple Approximation Method for Obtaining the Spanwise Lift Distribution, NACA TM 948. Available online at the URL:

  http://naca.central.cranfield.ac.uk/reports/1940/naca-tm-948.pdf

- The method is valid for wings whose lift distribution is close to be elliptical

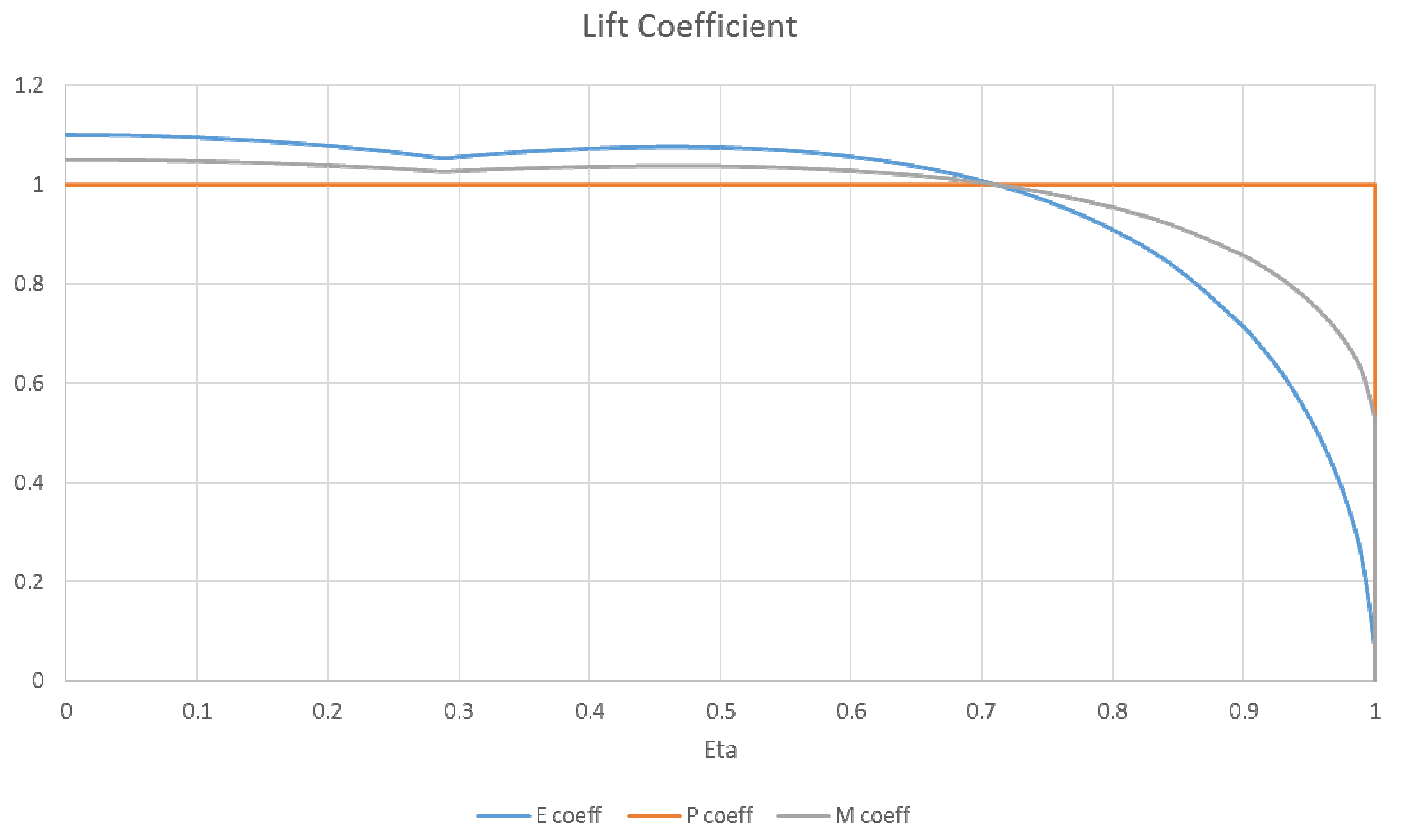
- The method is based on the following steps:
  1. **Consider ellipse whose semi-axis is equal to the wing-span and whose area is equivalent to the wing area**
  2. **Consider the chord distribution associated with a quarter of the ellipse**
  3. **Take the average between the quarter-ellipse chord distribution and the actual one**
  4. **The aforementioned average represents the local product of the actual cord times the sectional lift coefficient**
  5. **Divide the average by the local chord in order to get the local lift coefficient for a global lift coefficient of 1**

Schrenk's Method



ATR 72

Schrenk's Method



ATR 72

Schrenk Method – Lift Distribution

- Let us consider a condition at V_C and at an the altitude depending on the aircraft in question (ATR72 probably at sea level while the A320 will be at a higher altitude)

- The lift distribution is normally derived non-dimensionally and is then scaled to the appropriate total lift which will depend on the load factor n.

- Assume that the lift generated by the horizontal tail is negative and equal to the lift on the fuselage. This means that the lift on the exposed wing (outbd of the root) is equal to the aircraft weight times the load factor n. – *this is an overestimation suitable only for early design iterations*

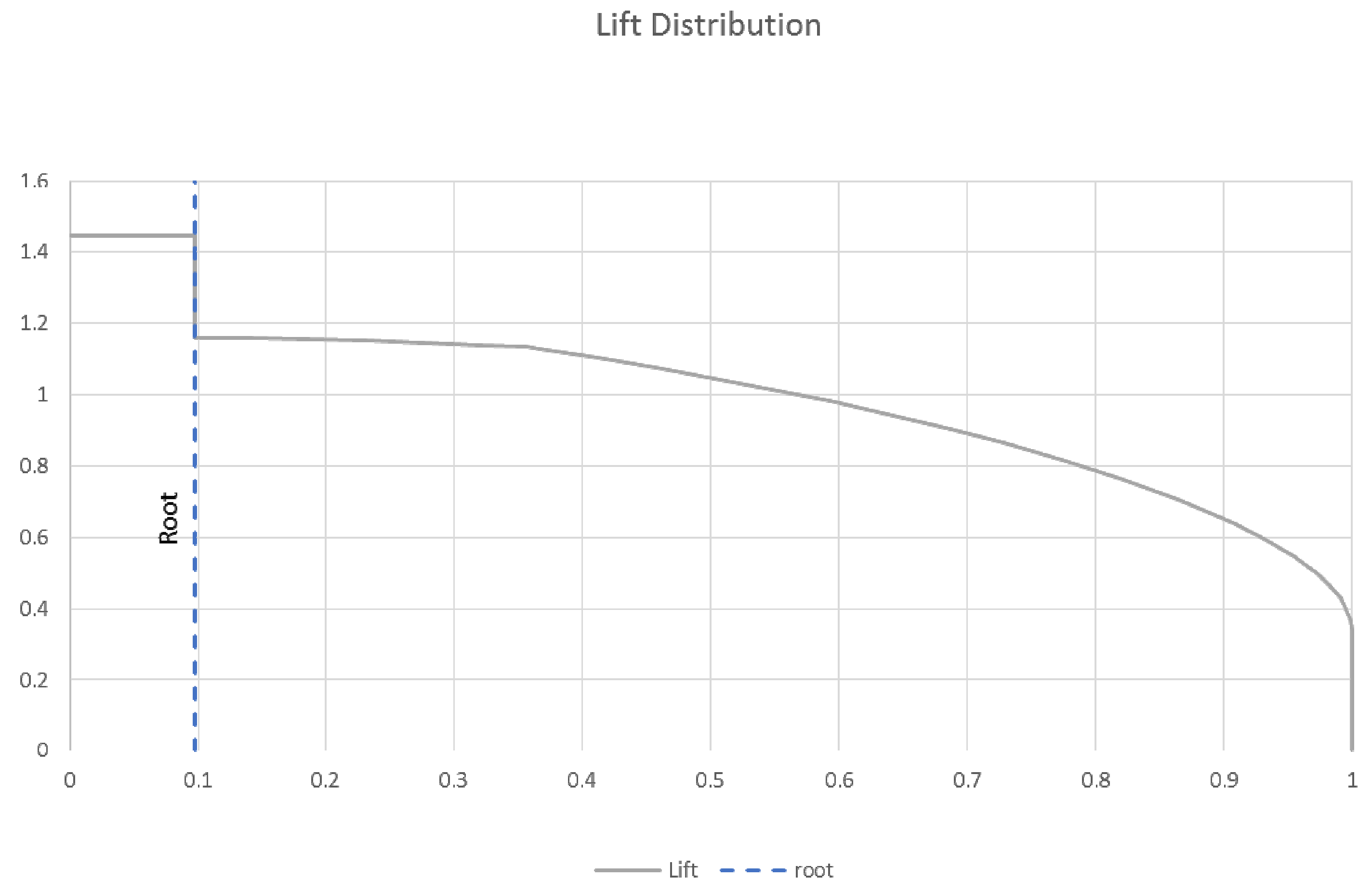
- Thus $L = nW = (L_W + L_F) + L_T$

  If $L_W + L_F = 100$, typically, the breakdown between wing and fuselage is

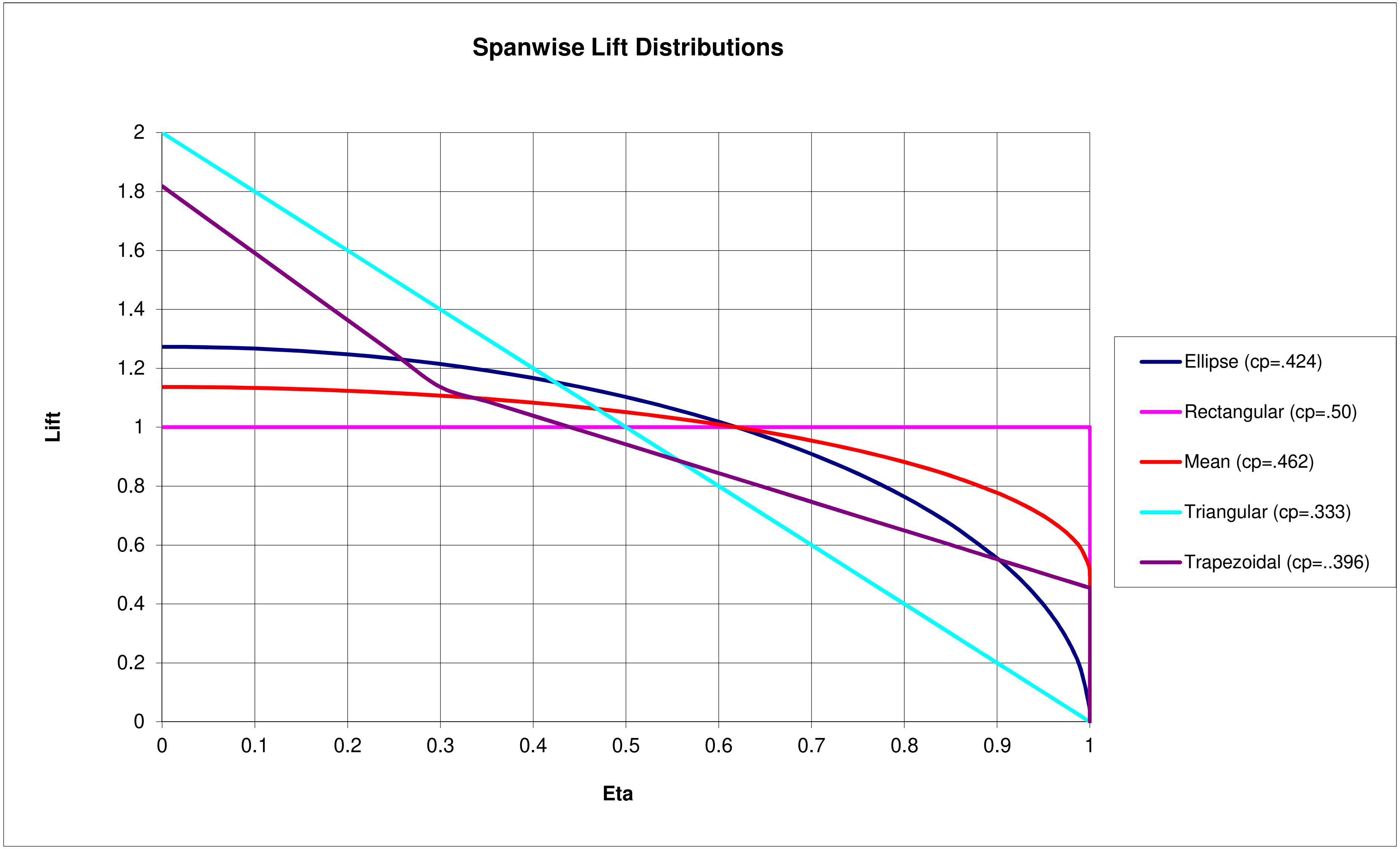
  $$L_W = 86 \text{ and } L_F = 14$$

  Thus $(L_W + L_F)$ needs to be scaled to give $L_W = L$

Schrenk's Method – Lift Distribution



ATR 72


Spanwise Lift Distributions
Lift
2
1.8
1.6
1.4
1.2
1
0.8
0.6
0.4
0.2
0
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1
Eta
Ellipse (cp=.424)
Rectangular (cp=.50)
Mean (cp=.462)
Triangular (cp=.333)
Trapezoidal (cp=..396)

Wing Weight Estimation

- A typical figure for civil transport aircraft:

  W_W = 10 to 15% of W_T

- A semi-empirical formula given in Torenbeek (p280) is

$$W_W = 0.0017 W_T \left(\frac{b}{\cos\Lambda_{1/2}}\right)^{\frac{3}{4}} \left[1+\sqrt{\frac{6.25}{b}\cos\Lambda_{1/2}}\right] \left(n_{ult}\right)^{\frac{11}{20}} \left(\frac{b S_W}{t_r W_T \cos\Lambda_{1/2}}\right)^{\frac{3}{10}}$$

- Dimensions in feet (areas in ft^2); weight in pounds;
- The following is advised but should be used with care

  For 2 wing mounted engines reduce weight by 5% (4 engines; -10%)

  If the landing gear is mounted on the fuselage, reduce weight by 5%.

Other formulae are given in Shevell, Standford University, Raymer and Roskam

Structural Weight

- We have an estimation of the total weight of the wing (structure and systems = 2468+394 = 2862kg). But how is it distributed? It can depend on the size and type of aircraft.

- Let us assume that it is constant in the center wing box (i.e. between the intersections with the fuselage) and triangular for the outer wing. But why?

- Note that the joints are a key component, both in terms of structural function and contribution to the overall weight of the wing assembly

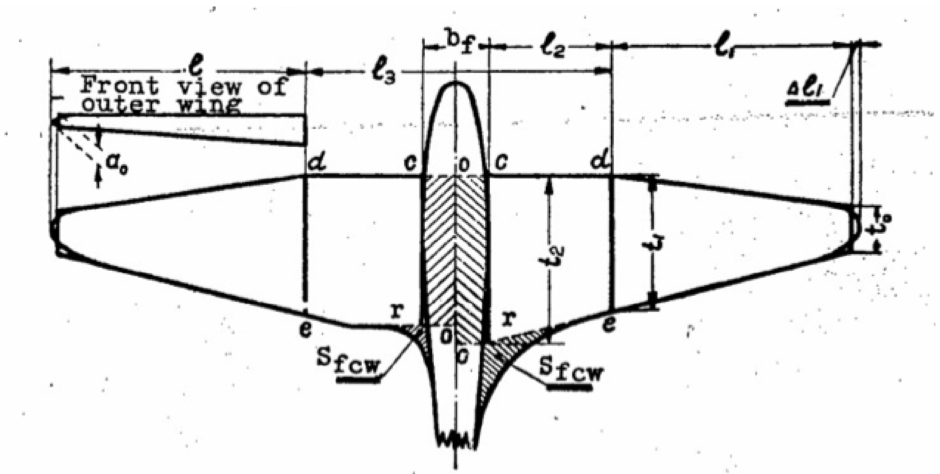
ATR 72

Structural Weight

- A general “geometrical” method for estimating the structural weight distribution in a wing:

  Savelyev, V.V., (1946), Distribution of Structural Weight of Wing Along the Span, NACA TM-1086

- According to the reference above a uniform weight distribution is suitable for the center wing and a triangular distribution is adequate for the outer wing



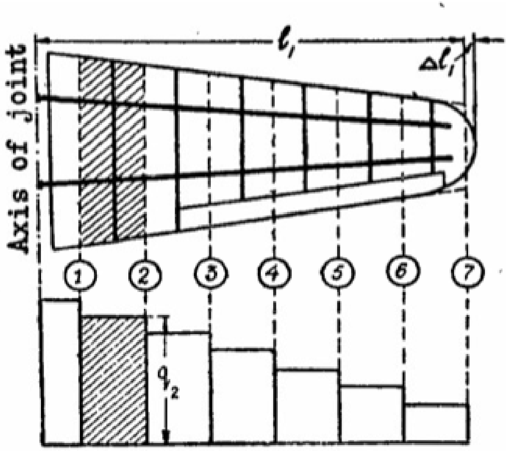


Figure 2

Wing Spanwise Mass Distribution

This can depend on the size of the aircraft and the layout of the wing and we are looking mainly what happens outboard of the root.

Light aircraft tend to be uniformly distributed e.g. kg/m is constant

The next is a triangular distribution which seems to work best for almost straight wings (little or no sweep).

The final distribution is effectively a function of thickness v chord ratio. It can be linear (2 slopes) or (t/c) times c to the power of 3/2

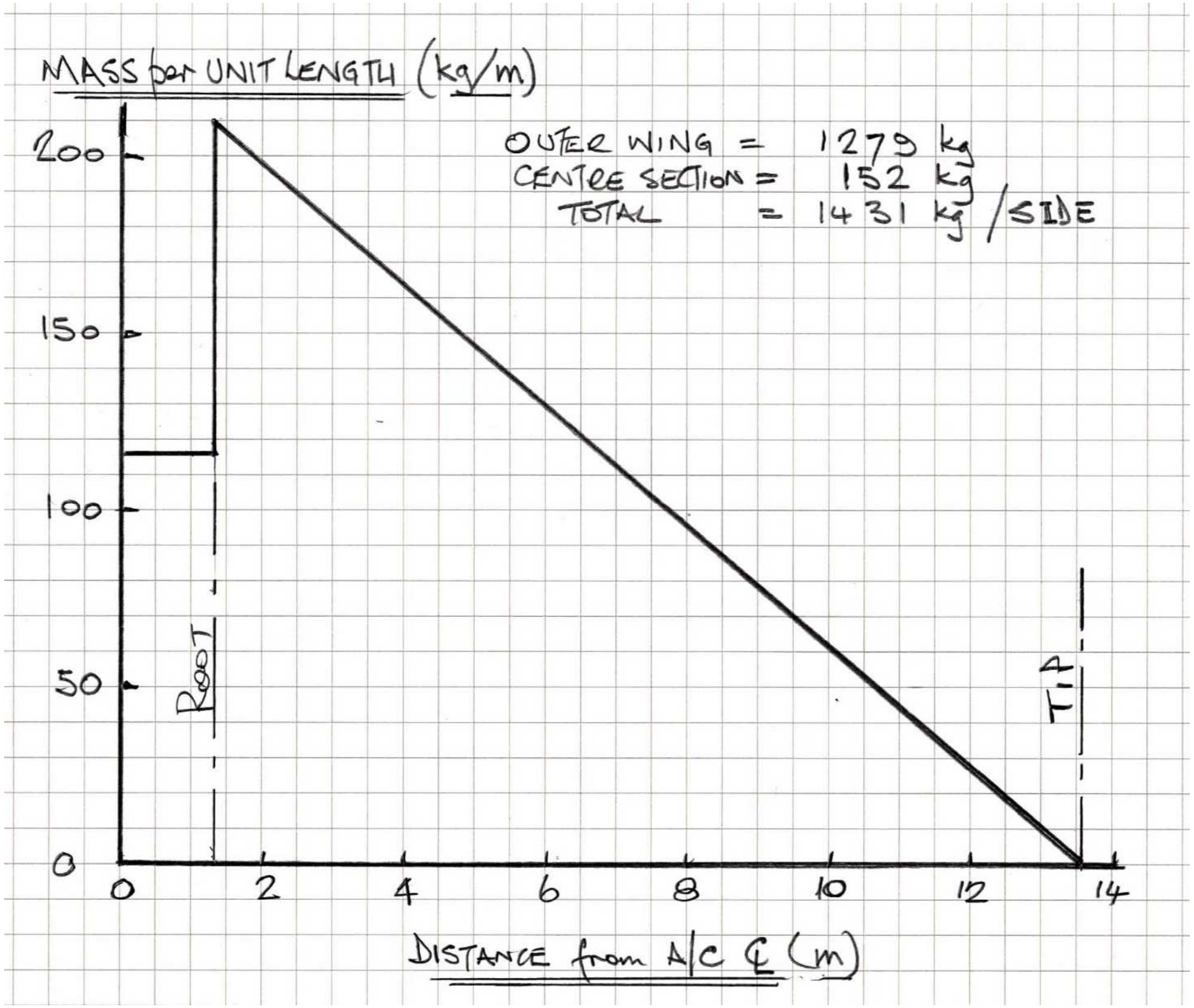
$$W = K \int_{Root}^{Tip} \left(\frac{t}{c}\right) c^n \ d_y$$

W = 9356 /2 kg per side (from sheet 24)

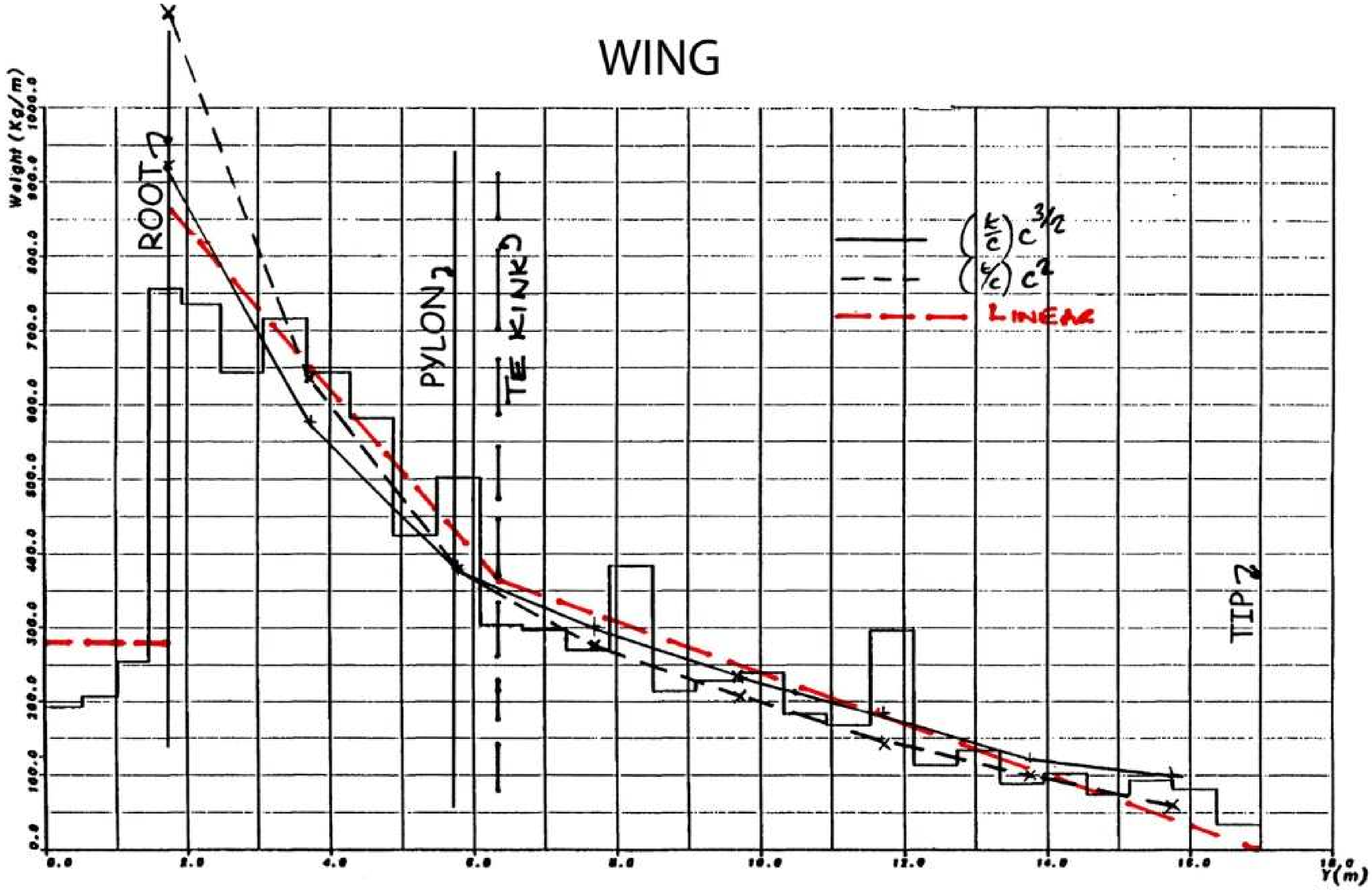
n = 3/2 ----- simple geometry suggests n=2

$$w(y) = K \left[\left(\frac{t}{c}\right) c^n\right](y)$$

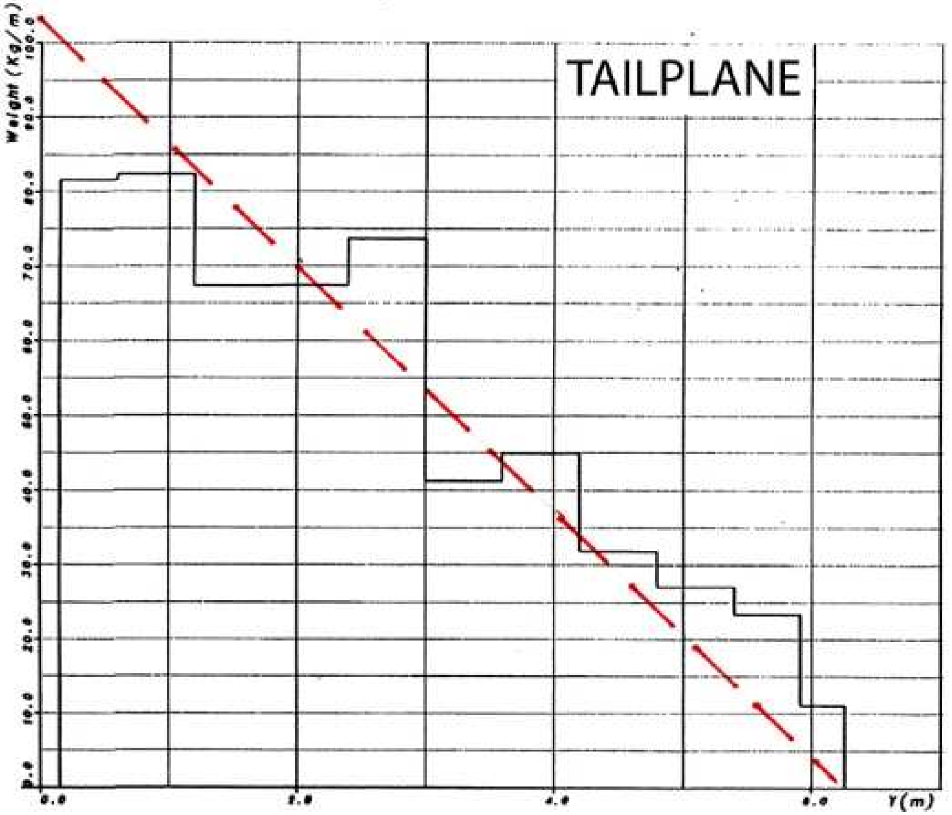
Wing Mass Distribution

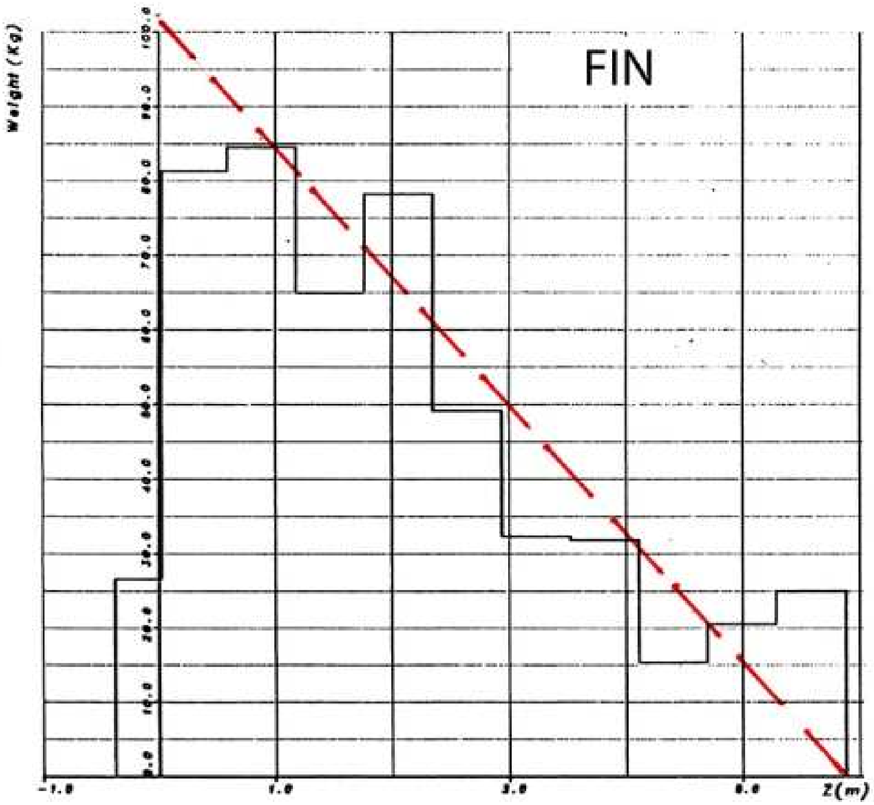


ATR 72



A320





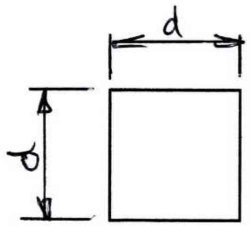
A320

Fuel Weight

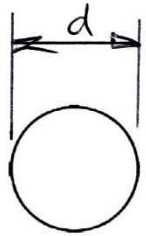
- The maximum stipulated fuel capacity is 5000 kg, corresponding to 6.37 m^3 (Nett Volume for Fuel Density of 785 kg/m^3). This the aircraft value.
- The fuel is stored in the outer wing, from the root rib to 60% of the semi-span.
- Gross fuel volume is that corresponding to the outer geometry of the wing box. This is the outer skin line and front and rear spar datums.
- Nett fuel volume is 90% of Gross volume. This takes into account the structure and airspace required in the tank.

ATR 72

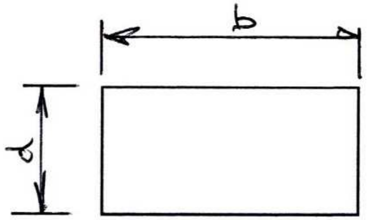
Area Info



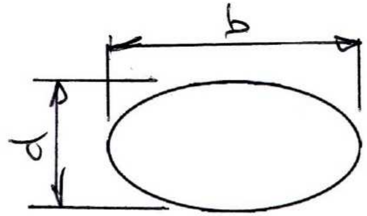
AREA $= d^2$



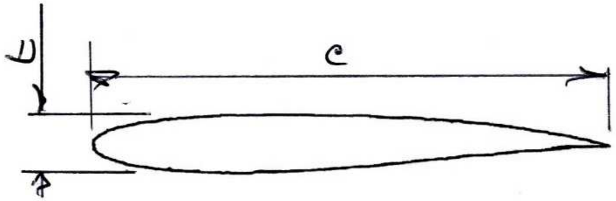
AREA $= \frac{\pi}{4} d^2 = 0.7584\, d^2$



AREA $= bd$

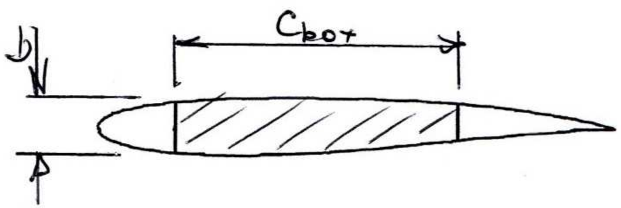


AREA $= \frac{\pi}{4} bd = 0.7854\, bd$



NACA SECTION AREA $= 0.684\, tc$

SUPER CRITICAL AREA $= 0.693\, tc$



SPARS at 20 & 60 AREA $= 0.938\, tC_{box}$

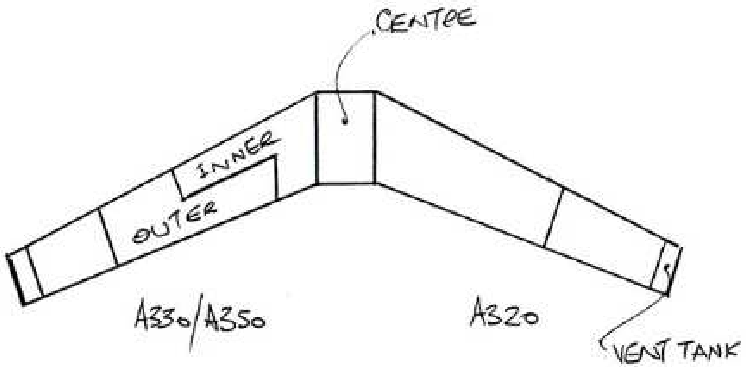
SPAR at 10 & 60 AREA $= 0.927\, tC_{box}$

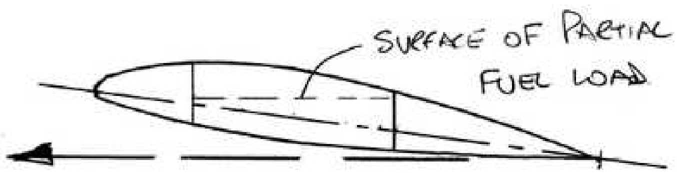
SPAR at 15 & 65 AREA $= 0.918\, tC_{box}$.

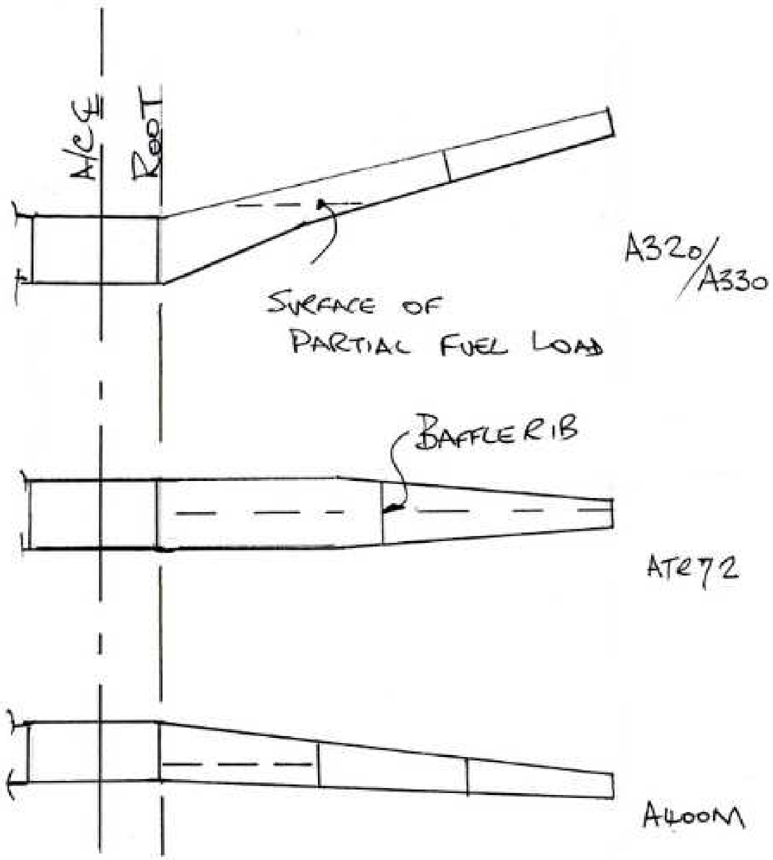
SPAR at 15 & 60 AREA $= 0.937\, tC_{box}$

Some Information about Fuel Tanks

For Design Projects, it is suggested that full Fuel Tanks are assumed.
Calculating Volumes and CG positions could be very involved

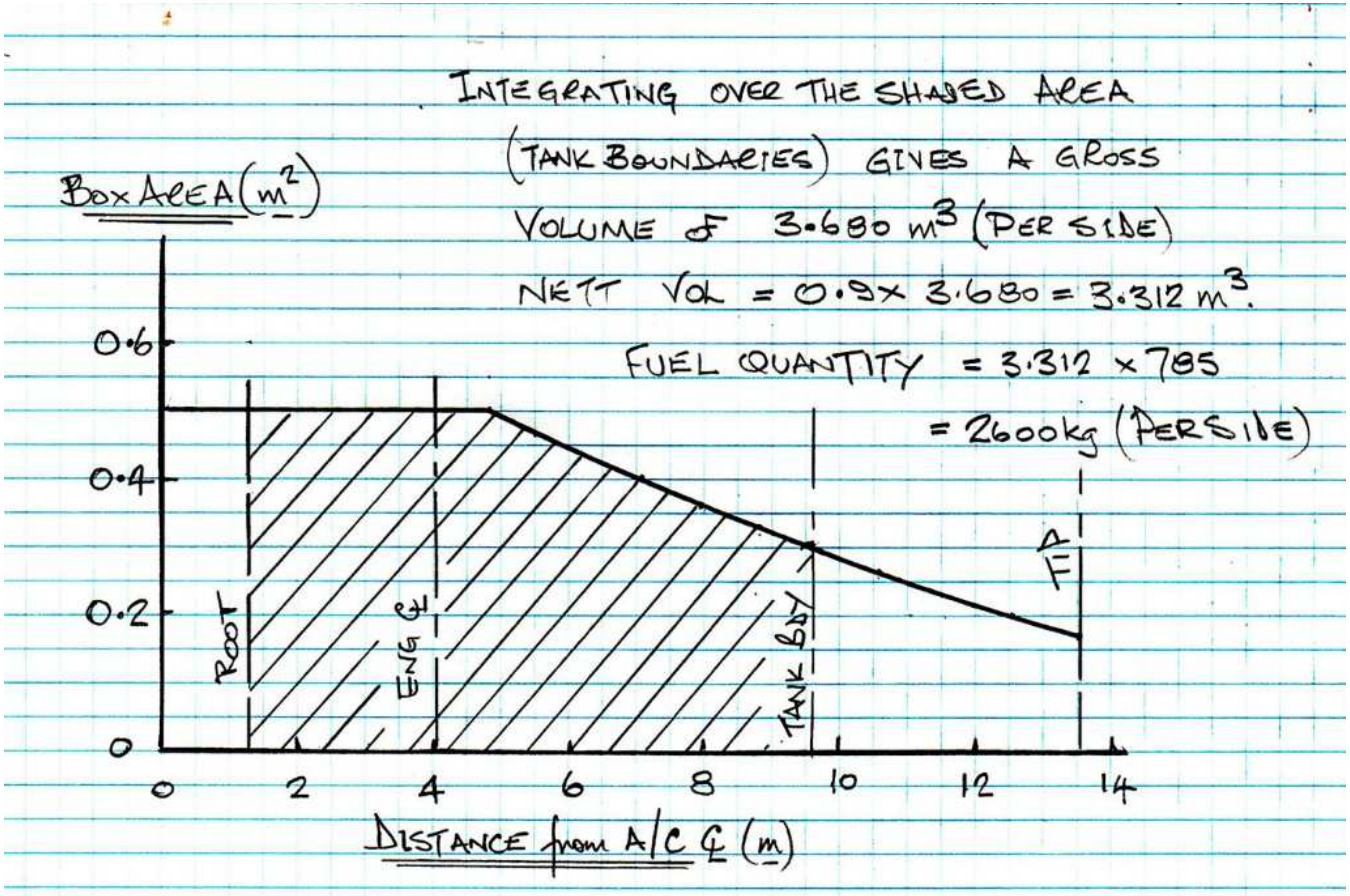






Box Area & Fuel Mass

- The fuel mass is distributed using wing box area



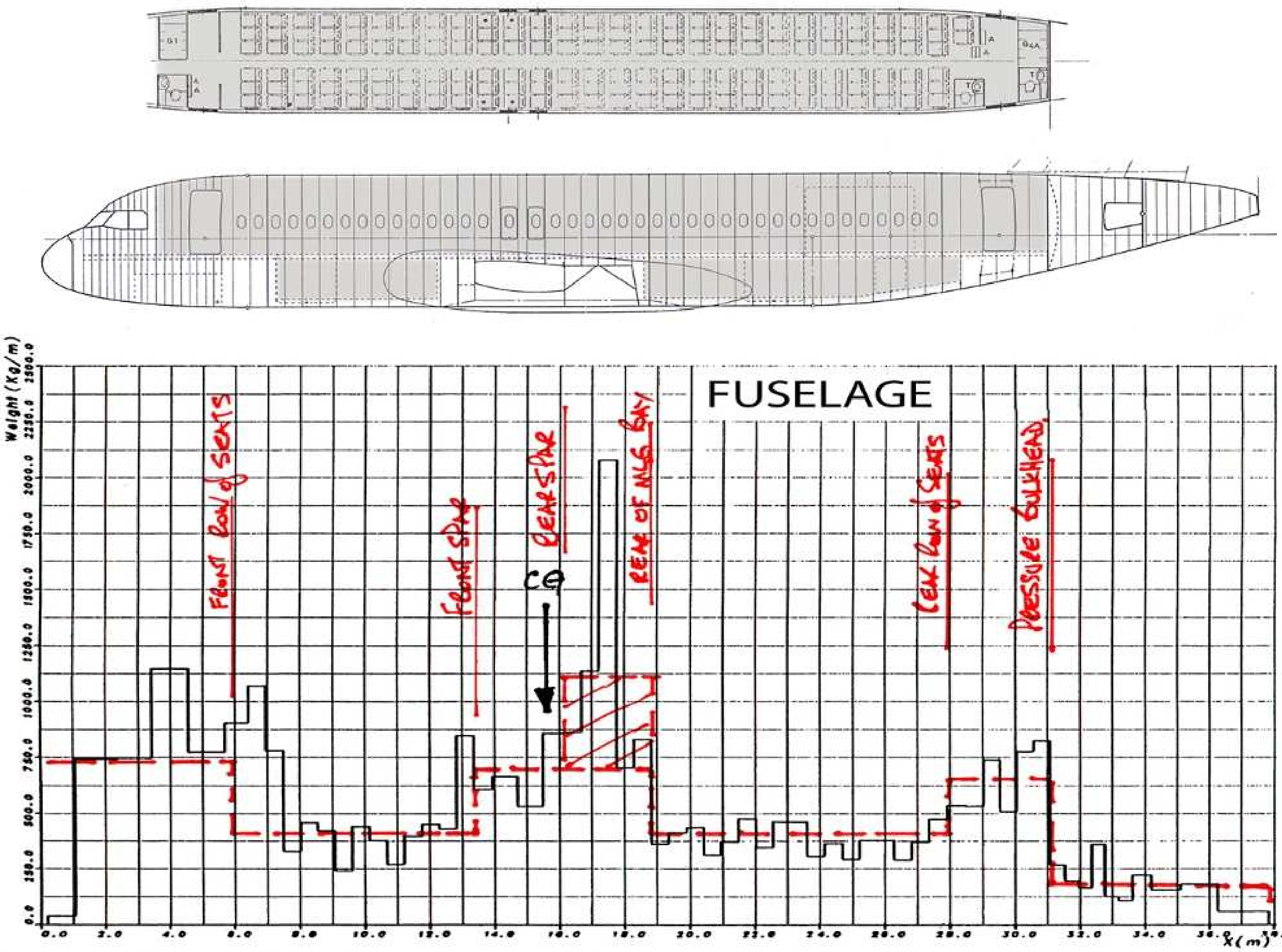
Chordwise CG position at OWE

Wing	Aircraft CL	0.45 c
	Root	0.45 c
	Inner Engine	0.43 c
	Outer Engine	0.43 c
	Tip	0.43 c
Tailplane	CL	0.45 c
	Tip	0.45 c
Fin	CL	0.45 c
	Tip	0.45 c

Fuel in the Wing is assumed to have a chordwise CG midway between the spars

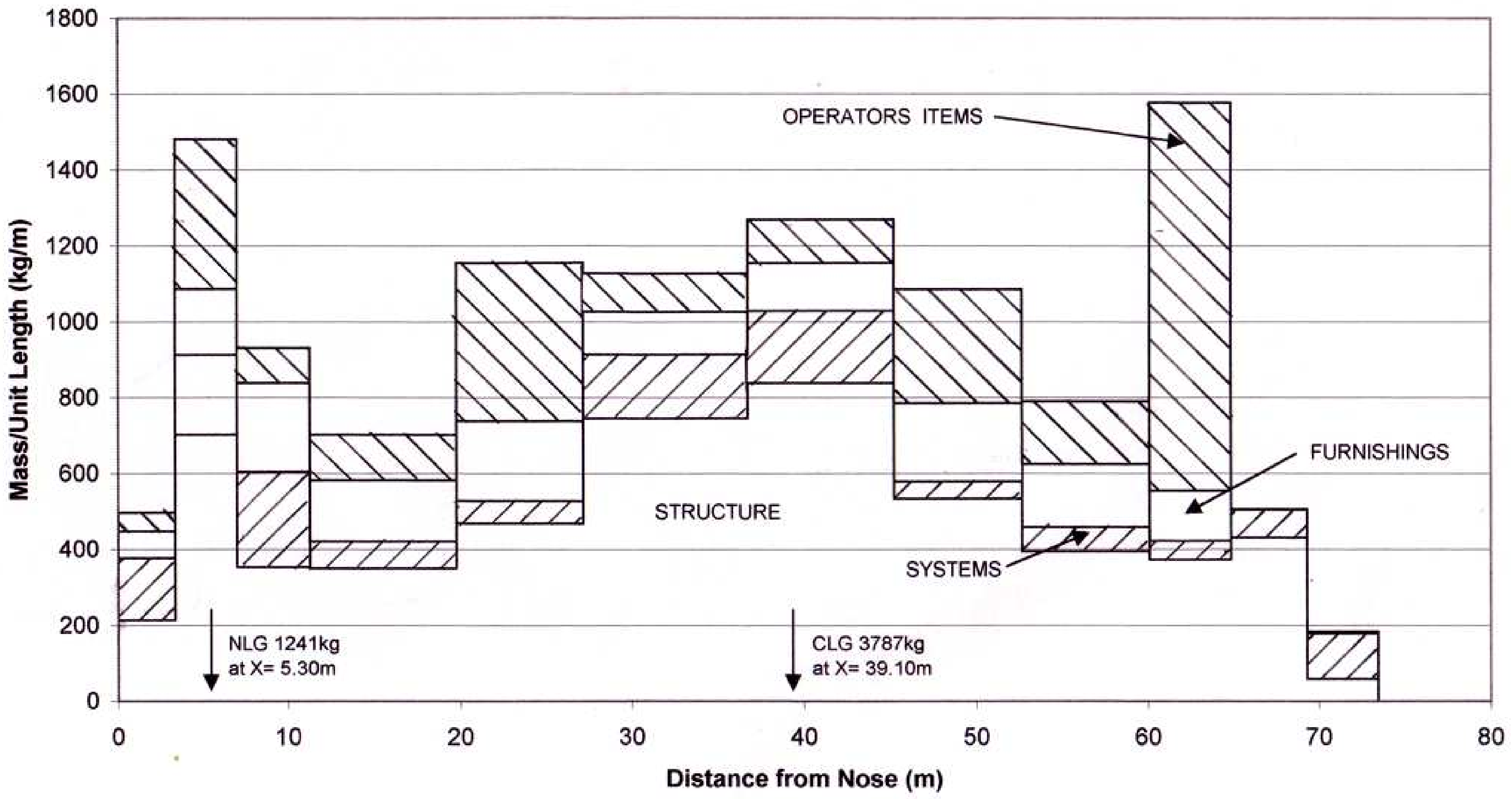
These are suggestions, depending on the layout of the aircraft,
other values may be appropriate.
For example, small light aircraft, 0.40 c might be more appropriate.

Fuselage Mass Distribution



A320

Fuselage Mass Distribution



Engine Weight

- Pratt and Whitney PW127 W
- 2750 SHP rating
- Bare weight: 522 kg (propeller NOT included)
- Mounted at 4.05m from fuselage centerline

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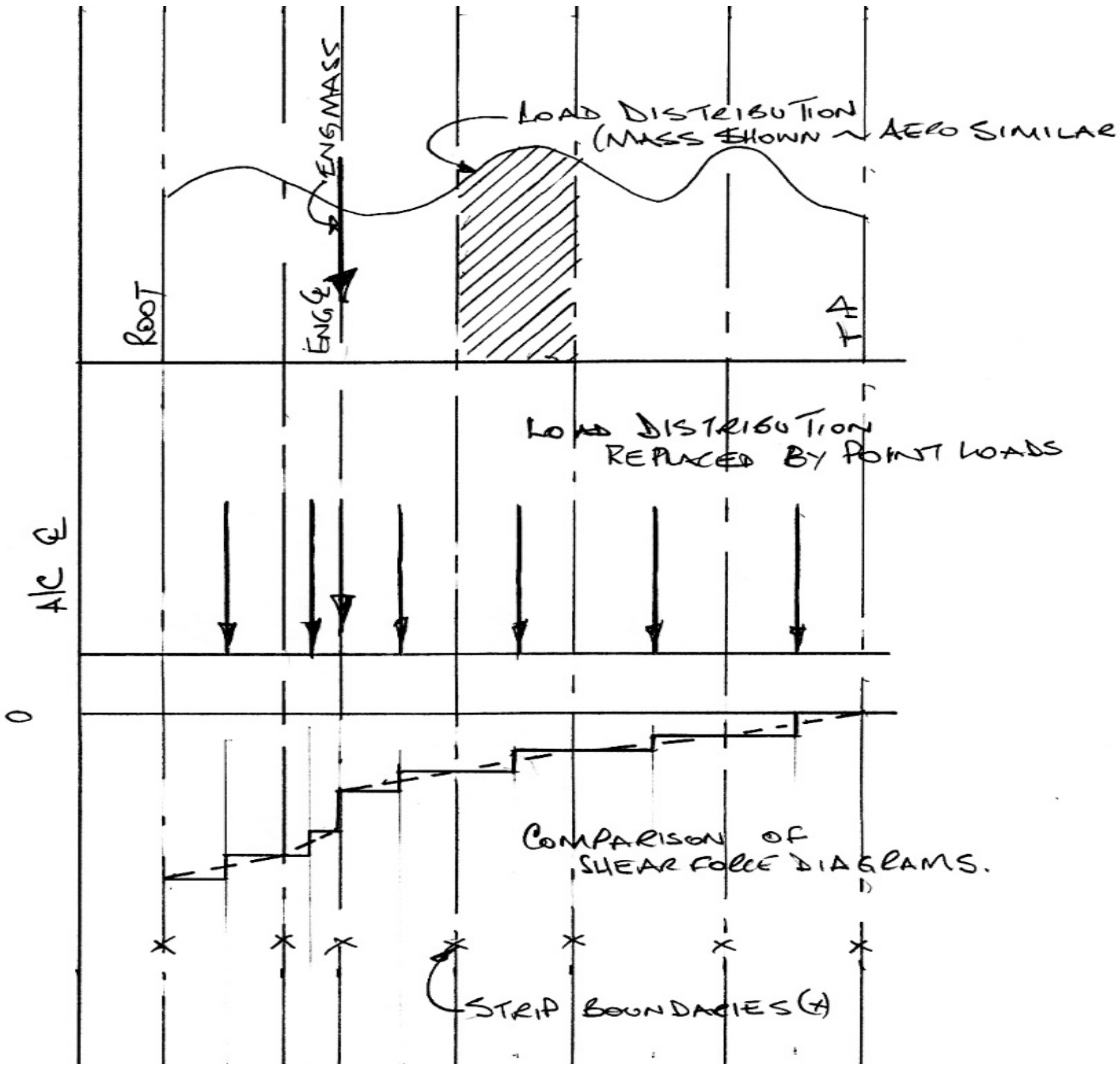
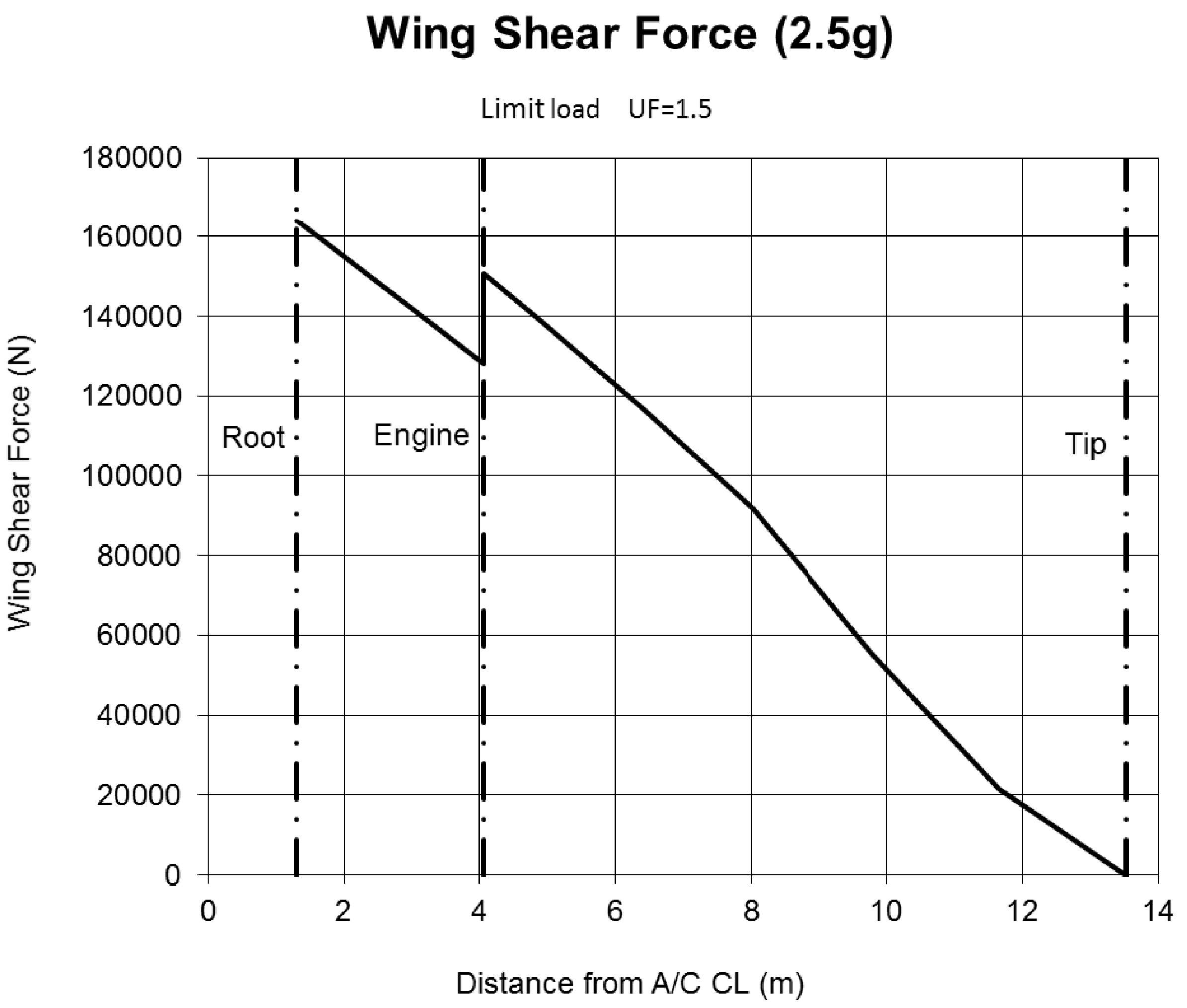
Engine + Nacelle Mass

	Mass(kg)	CG(m fwd of LE)
Engine	1044	0.9
Nacelle	228	0.0
Propeller	400	2.0
Fuel System	12	0.0
Systems	140	0.0
Operator Items	30	0.0
Total	1854	0.938

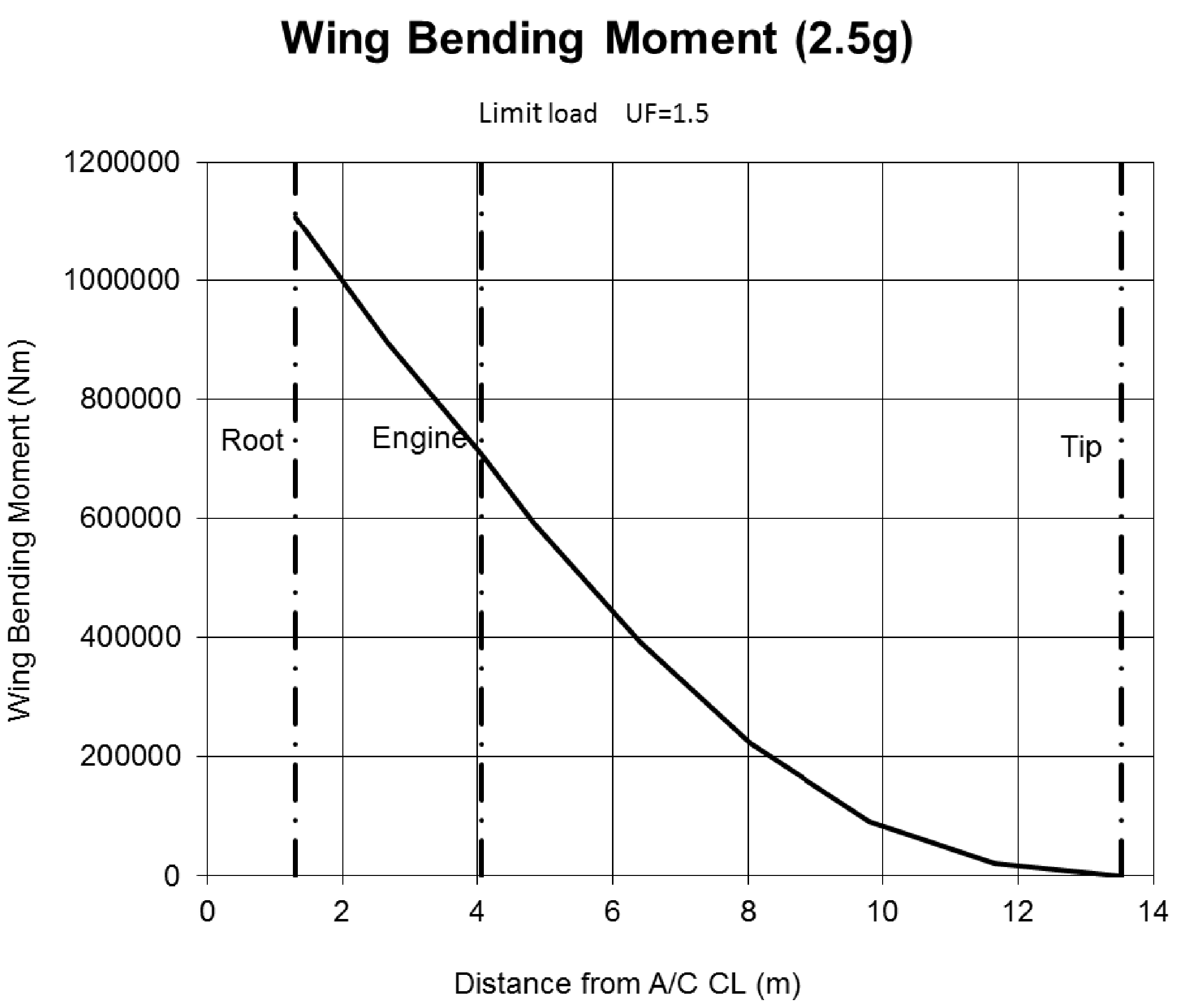
Use 927kg at 2.0m fwd of Ref Axis (per side)

Distributed Load to Point Load

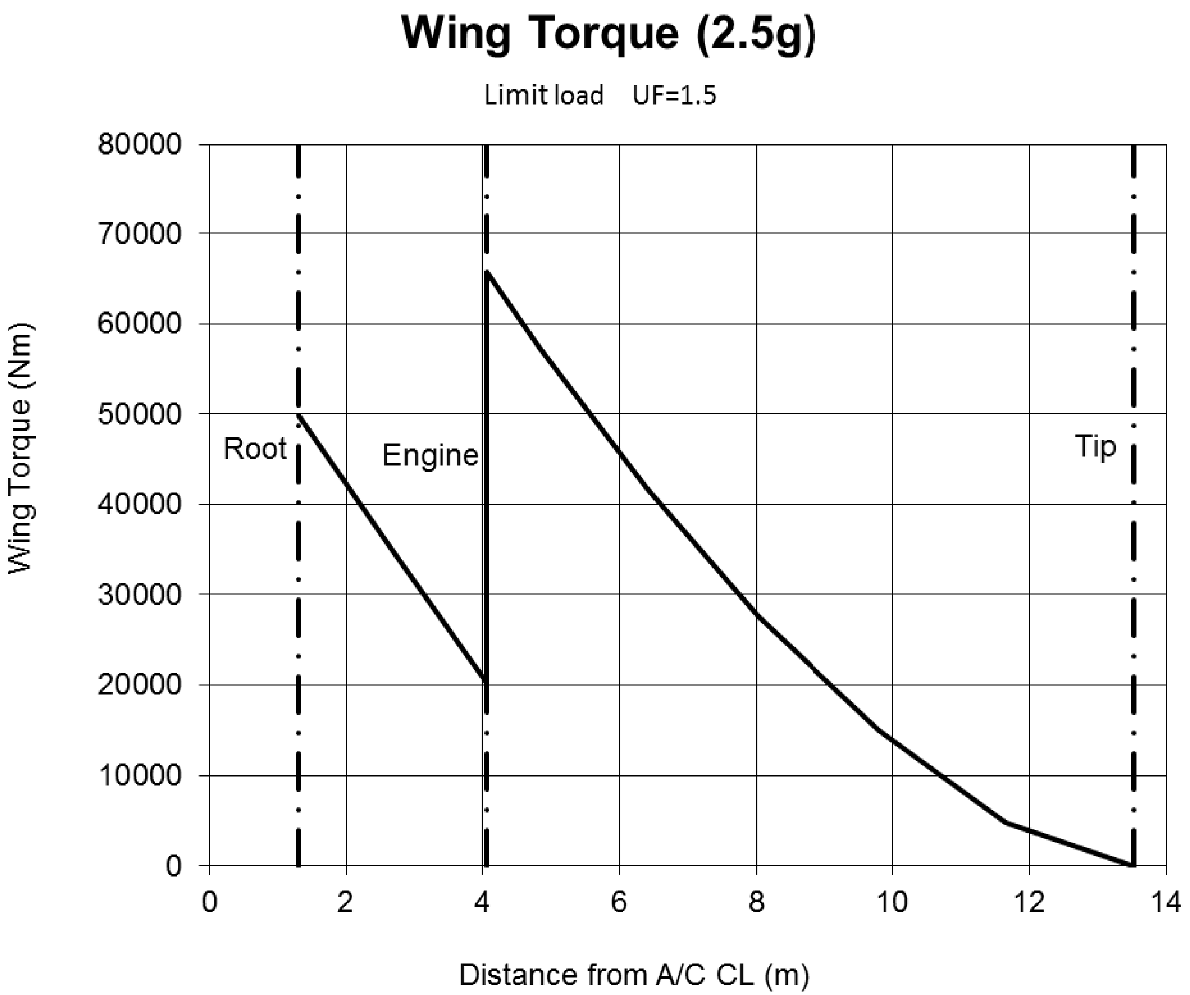



Wing Shear Force (2.5g)
Limit load UF=1.5
180000
160000
140000
120000
100000
80000
60000
40000
20000
0
Wing Shear Force (N)
Root
Engine
Tip
0
2
4
6
8
10
12
14
Distance from A/C CL (m)

ATR 72


Wing Bending Moment (2.5g)
Limit load UF=1.5
1200000
1000000
800000
600000
400000
200000
0
Wing Bending Moment (Nm)
Root
Engine
Tip
0
2
4
6
8
10
12
14
Distance from A/C CL (m)

ATR 72


Wing Torque (2.5g)
Limit load UF=1.5
Wing Torque (Nm)
80000
70000
60000
50000
40000
30000
20000
10000
0
Root
Engine
Tip
0
2
4
6
8
10
12
14
Distance from A/C CL (m)

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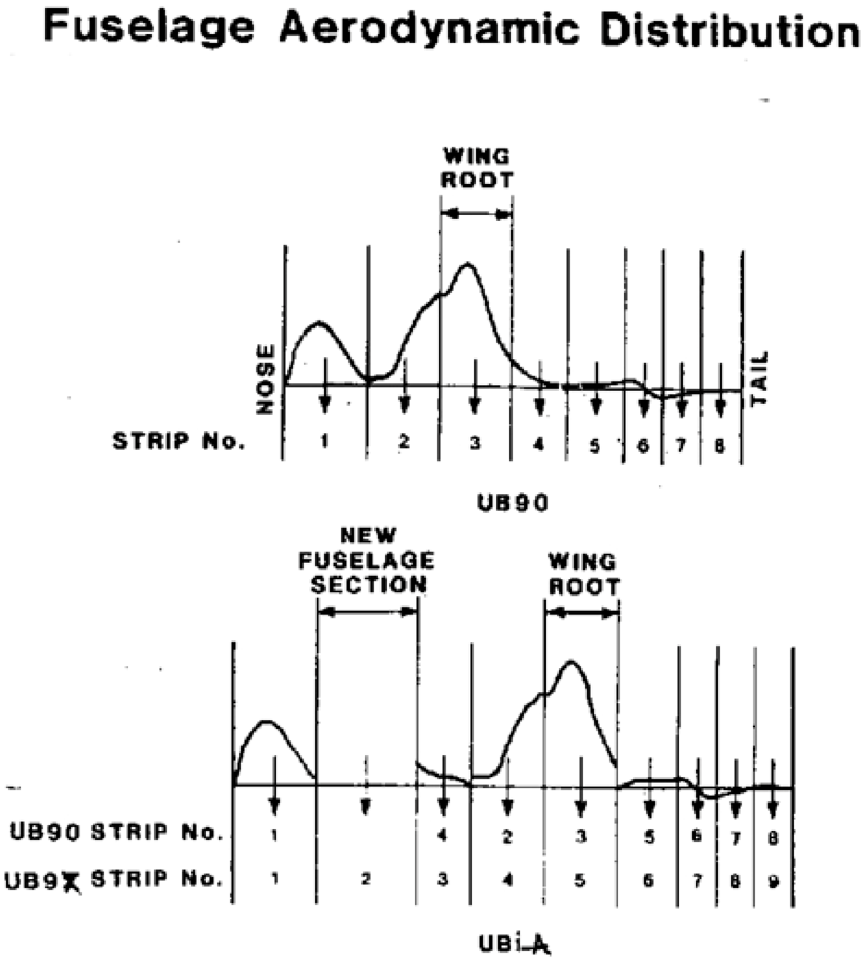
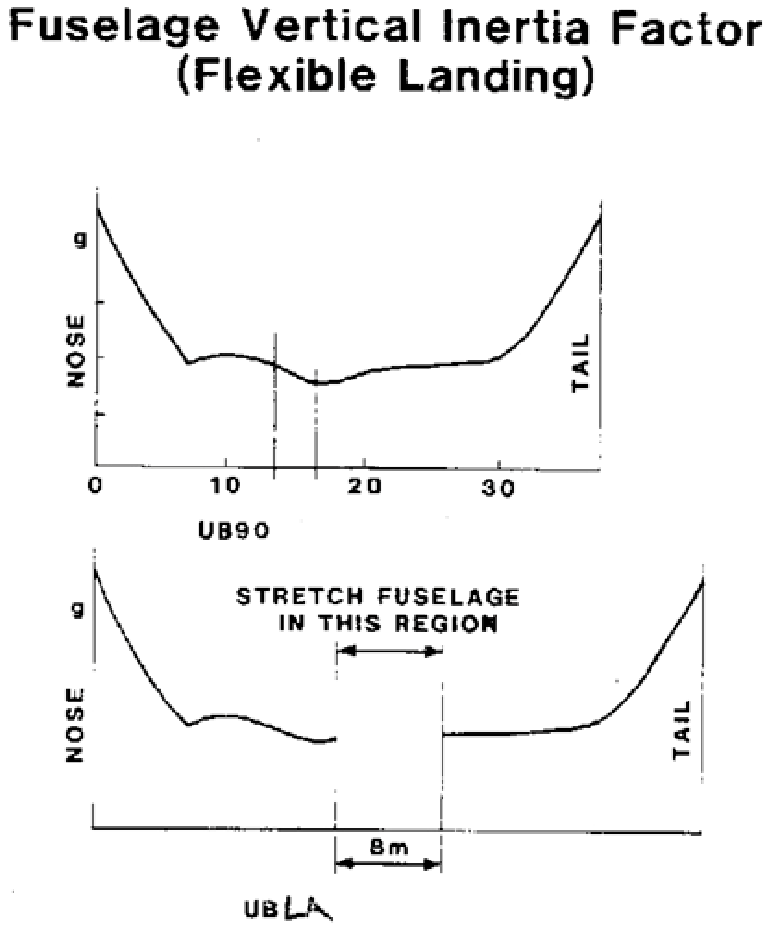
Suggestions when Changing Fuselage Length





Wing Geom

		given	calc	calc	given			
				wing geom				
	strip	bdy	width	CL	chord	wc^2	Ref Axis	LE
		(m)	(m)	(m)	(m)	(m^3)	(m)	(m)
							aft Datum	aft Datum
cl		0						0
	1		2	1	6.000	72.0000	1.670	0.0000
Root		2						
	2		2	3	5.481	60.0722	2.530	0.5195
		4						
LG	3		0	4	4.961	0.0000	2.960	1.0390
		4						
	4		1.85	4.925	4.481	37.1389	3.357	1.5195
		5.85						
Pod	5		0	5.85	4.000	0.0000	3.755	2.0000
		5.85						
	6		3.75	7.725	3.610	48.8809	4.561	2.9740
Tank Bdy		9.6						
	7		3.4	11.3	2.868	27.9573	6.098	4.8312
Vent T Bdy		13						
	8		4.4	15.2	2.057	18.6201	7.774	6.8571
Tip		17.4						

Wing Mass

		g=	9.80665	m/s^2		Wing Area	127.18		Case Defn	Weight=	74000	74000	64000	
		n=	2.5							n=	2.5	2.5	1	
										dyn press=	18500	18500	2700	
												for flight	for ldg	
	OWE	OWE	Fuel	Fuel	Total	Total								
strip	mass	CG(RA)	mass	CG(RA)	mass	CG(RA)	Fs	SF	mean SF	dMR	MR	MPs	MP	
	(kg)	(m)	(kg)	(m)	(kg)	(m)	(N)	(N)	(N)	(Nm)	(Nm)	(Nm)	(Nm)	
		fwd RA		fwd RA		fwd RA								
								-396189			-2434728		-250896	
1	510	-1.0299		-0.3099	510	-1.0299	-12503		-389937	-779875		12877		
								-383686			-1654854		-263773	
2	1368	-0.4274	2632	0.1091	4000	-0.0744	-98060		-334655	-669311		7295		
								-285625			-985543		-271068	
3	0	-0.2601	0	0.0982	0	0.0000	0		-285625	0		0		
								-285625			-985543		-271068	
4	779	-0.1103	1474	0.0881	2253	0.0195	-55237		-258007	-477313		-1078		
								-230388			-508230		-269990	
5	4000	2.7	0	0.0780	4000	2.7000	-98067		-181355	0		-264780		
								-132322			-508230		-5210	
6	1166	0.0345	1905	0.0575	3071	0.0488	-75299		-94672	-355021		-3675		
								-57023			-153209		-1536	
7	748	0.0336	989	0.0185	1737	0.0250	-42594		-35726	-121467		-1066		
								-14428			-31742		-470	
8	589	0.0326	0	-0.0240	589	0.0326	-14428		-7214	-31742		-470		
								0			0		0	
	9160													
	18320		7000.014		16160.01									

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Wing Aero

S(kN)	M(kNm)	T(kNm)					
437	3860	-153	Lwfp=	1886799	qS=	2352830	Values are per side
Values at ROOT							

strip	Ls/Lwfp	xac/c	xac(RA) (m)	Lso/qS	Cmol	Ls	Lso	Fs	SF	mean SF	dMR	MR	Mol	Fs*xac	MPs	MP
									820783			6782788.4				97598
1		0	0		0	0	0	0		820783	1641565.6		0	0	0	
									820783			5141222.7				97598
2	0.1656	0.2842	0.4528	0.00227	-0.0927	156211	2673	158884		741341	1482681.8		-103021	71947	-31074	
									661899			3658540.9				128672
3	0.0000	0.0000	0.0000	0.00000	0.0000	0	0	0		661899	0		0	0	0	
									661899			3658540.9				128672
4	0.1524	0.2632	0.6587	-0.00186	-0.1153	143777	-2194	141584		591107	1093548.3		-79247	93262	14016	
									520315			2564992.6				114656
5	0.0000	0.0000	0.0000	0.00000	0.0000	0	0	0		520315	0		0	0	0	
									520315			2564992.6				114656
6	0.2300	0.2759	0.5908	0.00253	-0.1136	216994	2982	219976		410328	1538728.7		-102728	129956	27228	
									300340			1026263.9				87428
7	0.1749	0.2537	0.5391	0.00261	-0.0768	165048	3065	168113		216284	735364.04		-39717	90630	50913	
									132227			290899.85				36515
8	0.1374	0.2064	0.4925	0.00225	-0.0830	129577	2650	132227		66114	290899.85		-28607	65122	36515	
									0			0				0

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Wing Results

										cos=	0.91871248				
										sin=	0.39492705				
strip	SF (mass)	SF (aero)	SF (total)	MR (mass)	MR (aero)	MR (total)	MP (mass)	MP (aero)	MP (total)	Sref (N)	Mref (Nm)	Tref (Nm)	Defn	strip	bdy (m)
	-396189	820783	424594	-2434728	6782788	4348060	-250896	97598	-153298	424594	4793317	-140837	cl		0
1														1	
	-383686	820783	437097	-1654854	5141223	3486369	-263773	97598	-166175	437097	3860469	-152667	Root		2
2														2	
	-285625	661899	376274	-985543	3658541	2672998	-271068	128672	-142397	376274	2965741	-130821			4
3													LG	3	
	-285625	661899	376274	-985543	3658541	2672998	-271068	128672	-142397	376274	2965741	-130821			4
4														4	
	-230388	520315	289927	-508230	2564993	2056762	-269990	114656	-155334	289927	2300090	-142707			5.85
5													Pod	5	
	-132322	520315	387994	-508230	2564993	2056762	-5210	114656	109446	387994	2195521	100549			5.85
6														6	
	-57023	300340	243317	-153209	1026264	873055	-1536	87428	85892	243317	916381	78910	Tank Bdy		9.6
7														7	
	-14428	132227	117799	-31742	290900	259158	-470	36515	36045	117799	267853	33115	Vent T Bdy		13
8														8	
	0	0	0	0	0	0	0	0	0	0	0	0	Tip		17.4

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Fuselage Geom

	strip	bdy	width	CL	chord
		given	calc	calc	given
		strip	strip	strip	Fuse
	strip	bdy	width	CL	width
		(m)	(m)	(m)	(m)
Nose		0			
	1		5.08	2.54	4.058
		5.08			
NLG	2		0	5.08	4.058
		5.08			
	3		1.88	6.02	4.058
F24		6.96			
	4		3.2	8.56	4.058
F30		10.16			
	5		3.2	11.76	4.058
FS		13.36			
	6		0.48	13.6	4.058
		13.84			
WRAD	7		0	13.84	4.058
		13.84			
	8		2.28	14.98	4.058
RS		16.12			
	9		2.7	17.47	4.058
F47		18.82			
	10		4.26	20.95	4.058
F55		23.08			
	11		4.82	25.49	4.058
F64		27.9			
	12		3.22	29.51	4.058
P Bulkhead		31.12			
	13		0.88	31.56	4.058
		32			
Fin	14		0	32	4.058
		32			
	16		1	32.5	4.058
		33			
Tailplane	17		0	33	4.058
		33			
	18		4.58	35.29	4.058
Tail		37.58			

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Fuselage Mass

		9.80665	m/s^2		127.18		Case Defn	Weight=	74000
		2.5						n=	2.5
								dyn press=	18500
strip	OWE mass (kg)	Pax mass (kg)	Hold mass (kg)	Total mass (kg)	Fs (N)	SF (N)	mean SF (N)	dM (Nm)	M (Nm)
						0			0
1	3744	0	0	3744	-91788		-45894	-233141	
						-91788			-233141
2	0	0	0	0			-91788	0	
						-91788			-233141
3	1681	0	0	1681	-41220		-112398	-211308	
						-133008			-444449
4	1426	1880	558	3864	-94734		-180375	-577199	
						-227741			-1021647
5	1480	1880	558	3917	-96042		-275762	-882439	
						-323783			-1904087
6	309	282	0	590	-14476		-424569	-203793	
						-338260			-2107880
7									
						-525356			-4453507
8	1521	1339	0	2860	-70127		-490292	-1117866	
						-455229			-3335641
9	2993	1586	0	4579	-112263		-399097	-1077562	
						-342965			-2258080
10	1693	2502	743	4938	-121069		-282431	-1203155	
						-221896			-1054924
11	1767	2831	841	5439	-133341		-155226	-748190	
						-88556			-306734
12	2099	0	0	2099	-51451		-62830	-202314	
						-37105			-104420
13	196	0	0	196	-4805		-34703	-30538	
						-32300			-73882
14	0	0	0	0	-5001		-29800	-8287	
						-27299			-65595
16	262	0	0	262	-6428		-24085	-24085	
						-20871			-41510
17	0	0	0	0	-6414		-17664	-8402	
						-14458			-33108
18	590	0	0	590	-14458		-7229	-33108	
						0			0
	19760	12300	2700	34760					
					Fin Mass	510	kg at	33.657	m from nose
					Fs=	-5001	N		
					dM=	-8287	Nm		
					Tailplane M	654	kg at	34.31	m from nose
					Fs=	-6414	N		
					dM=	-8402	Nm		
					Note	Tailplane Mass is aircraft total (NOT per side)			

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Fuselage Aero

				Lwfp=	1886799	qS=	2352830	Values are per side	
strip	Ls/Lwfp	Lso/qS	Ls	Lso	Fs	SF	mean SF	dM	M
						0			0
1	0.030	-0.023	56839	-54970	1869		935	4747	
						1869			4747
2	0.000	0.000	0	0	0		1869	0	
						1869			4747
3	0.003	0.004	6124	9408	15532		9635	18114	
						17401			22861
4	0.005	0.005	10097	12840	22936		28869	92382	
						40337			115244
5	0.036	-0.003	67944	-6202	61742		71208	227867	
						102079			343111
6	0.008	-0.003	15124	-6153	8971		106565	51151	
						111050			394262
7	0.000	0.000							
						61614			-733591
8	0.037	-0.004	69035	-9879	59156		32037	73043	
						2459			-806634
9	0.014	0.001	27080	3153	30233		-12658	-34176	
						-27774			-772458
10	0.004	0.008	7201	18449	25649		-40599	-172950	
						-53423			-599509
11	0.005	-0.003	9606	-6111	3496		-55171	-265924	
						-56919			-333585
12	-0.001	0.000	-1523	1018	-505		-56666	-182465	
						-56414			-151120
13	-0.001	0.002	-1290	4682	3393		-58110	-51137	
						-59806			-99984
14	0.000	0.000	0	0	0		-59806	0	
						-59806			-99984
16	-0.001	0.004	-1545	9333	7788		-63700	-63700	
						-67594			-36284
17	0.000	0.000	0	0	-72569		-31309	-47678	
						4976			11394
18	-0.001	0.003	-1104	6080	4976		2488	11394	
						0			0
					Tailplane A	72569	N at	33.657	m from nose
					Fs=	-72569	N		
					dM=	-47678	Nm		

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Fuselage Results

strip	SF (mass)	SF (aero)	SF (total)	M (mass)	M (aero)	M (total)		S (N)	M (Nm)		Defn	strip	bdy (m)
	0	0	0	0	0	0		0	0		Nose		0
1												1	
	-91788	1869	-89919	-233141	4747	-228394		-89919	-228394				5.08
2											NLG	2	
	-91788	1869	-89919	-233141	4747	-228394		-89919	-228394				5.08
3												3	
	-133008	17401	-115606	-444449	22861	-421587		-115606	-421587		F24		6.96
4												4	
	-227741	40337	-187404	-1021647	115244	-906404		-187404	-906404		F30		10.16
5												5	
	-323783	102079	-221704	-1904087	343111	-1560976		-221704	-1560976		FS		13.36
6												6	
	-338260	111050	-227210	-2107880	394262	-1713618		-227210	-1713618				13.84
7											WRAD	7	
	-525356	61614	-463741	-4453507	-733591	-5187098		-463741	-5187098				13.84
8												8	
	-455229	2459	-452770	-3335641	-806634	-4142275		-452770	-4142275		RS		16.12
9												9	
	-342965	-27774	-370739	-2258080	-772458	-3030538		-370739	-3030538		F47		18.82
10												10	
	-221896	-53423	-275320	-1054924	-599509	-1654433		-275320	-1654433		F55		23.08
11												11	
	-88556	-56919	-145474	-306734	-333585	-640319		-145474	-640319		F64		27.9
12												12	
	-37105	-56414	-93519	-104420	-151120	-255541		-93519	-255541		P Bulkhead		31.12
13												13	
	-32300	-59806	-92106	-73882	-99984	-173866		-92106	-173866				32
14											Fin	14	
	-27299	-59806	-87105	-65595	-99984	-165578		-87105	-165578				32
16												16	
	-20871	-67594	-88465	-41510	-36284	-77793		-88465	-77793				33
17											Tailplane	17	
	-14458	4976	-9482	-33108	11394	-21714		-9482	-21714				33
18												18	
	0	0	0	0	0	0		0	0		Tail		37.58

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Short Range vs Long Range OR Small vs Large

The following values are for a mainly metal structure

	Short Range OR Small	Long Range OR Large
Wing Stringer Pitch Upper cover Lower cover	165 mm 165mm	205mm 180mm
Rib Pitch	650mm	820mm
Design Service Goal	60000 Flight Cycles 100000 Flight Hours	19000 Flight Cycles 140000 Flight Hours
Mission	Short Range	17% Short Range 59% Medium Range 24% Long Range
Fuselage Frame Pitch	21inches(533mm)	533 or 600mm
Fuselage Stringer Pitch	180mm	180mm

Summary of Objectives for Design Project

- This is NOT to demonstrate that you can carry out a skin/stringer compression panel analysis or similar detailed design exercise, but will assess you on understanding of the structural components, their positioning, their functions and their classification.
- You will need to demonstrate attention to:
  - **Loads**

  Know what load cases drive which types of structure, and what kind of loads
    - Load cases (via n-v diagram and prescribed cases, also via weight-cg and mission profile)
    - Derive SMT loading and running loads for wings, SF, BM, torque diagrams for wing and fuselage for all identified cases
    - Ground loads, identify cases
    - Load alleviation – can you make structure lighter by active or passive load alleviation
    - Spanwise CP position – ib = good for structures bad for aero, where do you put it?
    - Internal pressure loads – what are the cases? (NB Airbus may be able to provide an envelope acceleration plot)
    - Brazier loads, use "classical" methods to derive the loads
    - Thermal loads if dis-sililar materials are used anywhere, if differential thermal effects take place can we design the structure to avoid picking up the loads?

Summary of Objectives for Design Project

- **Geometry –**
- **wing**
  - Structure positioning, front spar, rear spar,
  - Key rib positions,
  - rib pitches and orientation,
  - stringer pitches and orientation,
  - Gear, flap track and engines
  - Systems, vents, pumps etc
  - Manholes
- **fuselage**
  - Cross section geometry
  - Frame geometry and stringer positions
  - Floor beams
  - Doors – cargo and pax
  - Main structural item positons,
    - Gear
    - Engines (if applicable)
    - Empenage
    - APU
    - Wingbox

Summary of Objectives for Design Project

- **Integration**

  Understand the influence of each of these with respect to the wingbox
  - LE devices (slats etc, if applicable)
  - MLG (if applicable)
  - Engine (if applicable)
  - Flaps
  - failsafe and crashworthiness issues
- **Wing fuel tank layouts**

  Basic fuel systems integration, engine feed, venting, refueling pipe and pump locations, tank boundaries, lightning strike requirements, dry bay requirements
- **Principal structural design drivers**
  - Damage tolerance
  - Fatigue
  - Stability
  - Static Strength
  - Max and min thickness drivers
- **Material selection**

  Weight savings for composite structures should be considered carefully, a clean wing will enable much larger weight savings than one with many local load inputs.
  - Composite
    - Manufacturing processes, build procedures, tolerances, section join-ups
    - Laminate design (stacking rules, ply orientations etc)
  - Metallic
    - Manufacturing processes
    - Alloy choices
- **Weights**
  - Bottom up calc– this will be difficult at this stage of the design but spot checks would be useful

Summary of Objectives for Design Project

– **Integration**

Understand the influence of each of these with respect to the wingbox

- LE devices (slats etc, if applicable)
- MLG (if applicable) – what support structure will this need?
- Engine (if applicable)
- Flaps
- failsafe and crashworthiness issues

– **Wing fuel tank layouts**

Basic fuel systems integration,

- engine feed routing,
- venting,
- refuel/defuel piping,
- pump locations,
- tank boundaries (considering unconstrained rotor burst failure),
- lightning strike zone requirements,
- dry bay requirements

Summary of Objectives for Design Project

- In your section you should ALWAYS include:
  - a 3-view GA
  - Wing planform showing
    - high lift surfaces and types (eg slats, krugers, droop nose, single or double slotted flaps
    - control surfaces ailerons and spoilers
    - structural datums for spars, ribs etc
    - tank geometry,
    - fuel systems basic architecture,
    - gear position, ie footprint and support positions
    - a/c CG position,
    - Wing Aerodynamic Centre position
  - Fuselage layout, showing;
    - Nose gear position
    - Pressure bulkheads
    - Wing centre box
    - Gear positions (if applicable)
    - Engine positions (if applicable)
  - Diagrams of engine attachment, gear attachment showing load cases applied
  - Show how the structures can assist in the low cost carrier design goal, investigate the compromises along the design path.