Model of the Boeing 747-400

with CF6-80C2B1F engines

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Contents

Introduction

This document concerns a model of the Boeing 747-400, a wide-body passenger aircraft with 4 General Electric CF6-80C2B1F turbofan engines. The following chapters will give models of the operational, aerodynamic and engine characteristics of this aircraft. All models are obtained from the *Performance Engineers Manual* of the Boeing 747-400 (reference 1). The specific figures and tables which are used to derive the models are given in each section. All models are accurate within a 5% error, unless stated otherwise.

Chapter 1

Principal Characteristics

The Boeing 747-400 was the world's first wide-body jet airliner. The programme was launched on the 25^{th} of July 1966 and was first delivered on December 30^{th} of the same year to Pan Am. The series 400 with extended range and capacity first flew on April 29^{th} 1988. Several versions of the series 400 are available, such as the basic passenger version, the full-freight version, the passeneger/freight combi and the extended range version. There is also a choice in engines, between the Pratt & Whitney PW4056, the General Electric CF6-80C2B1F or the Rolls-Royce RB211-524G. This model only applies to the Boeing 747-400 passenger version with General Electric CF6-80C2B1F engines.The following principal characteristics obtained from references 2, 3, 4, 5, 6 and 7 are applicable to this model:

Geometry:

Wing Surface Area	=	541.16 m^2
Wing Span	=	64.44 m
Aspect Ratio	=	7.7
Length	=	70.67 m
Wing Sweep	=	$27°$
Dihedral	=	$7°$

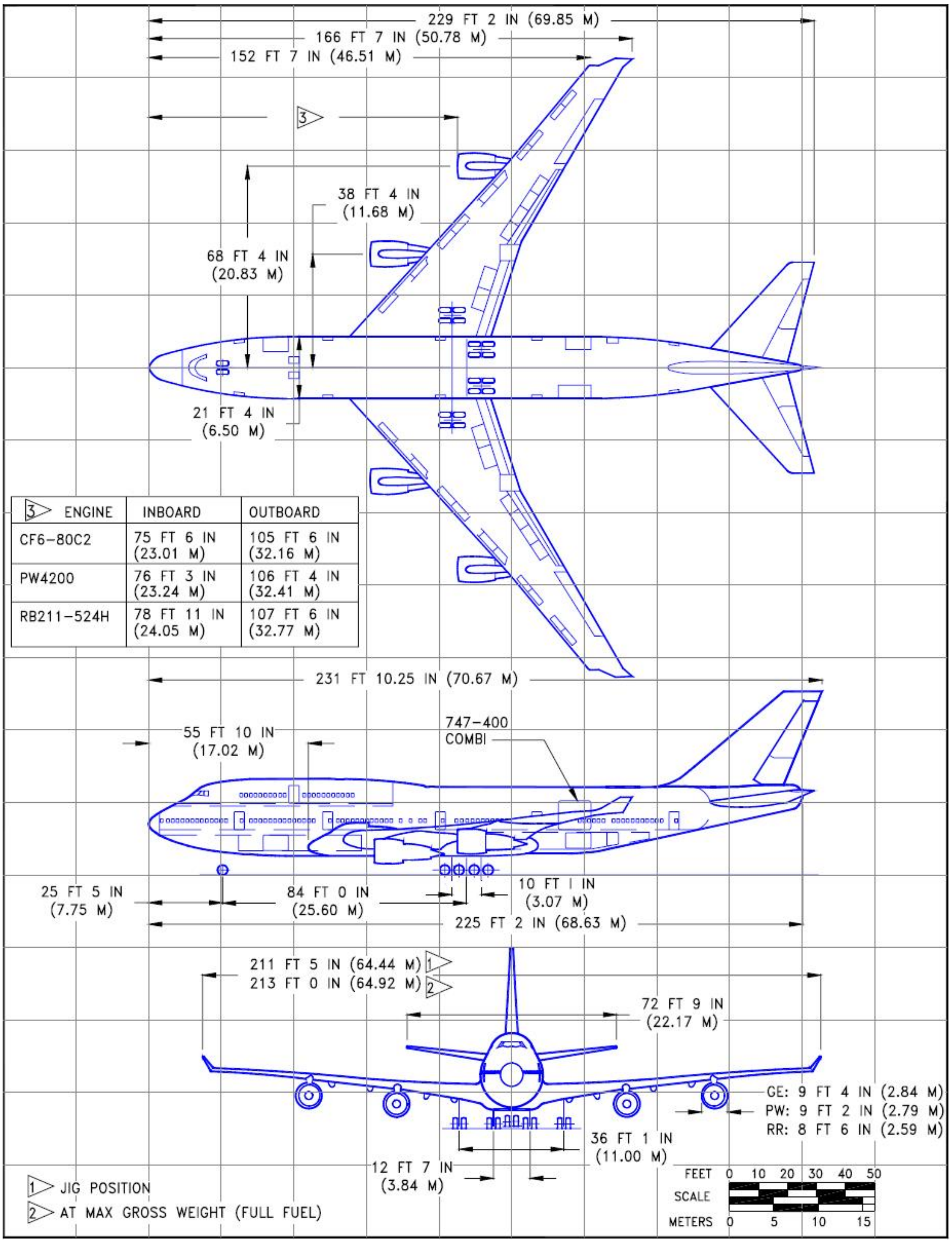

Figure 1.1: Three-View of the Boeing 747-400 (reference 7)

Weight:

MTW	=	364235 kg
MTOW	=	362874 kg
MLW	=	260362 kg
MZFW	=	242672 kg
OEW	=	178756 kg
Maximum Payload	=	63917 kg
Maximum Seating Capacity	=	42 (up), 24 (first), 32 (business), 302 (economy)
Maximum Cargo Volume	=	157 m^3 Containers, 24 m^3 Bulk Cargo
Usable Fuel	=	163396 kg

Engines:

Rated Output	=	254.26 kN (57900 lb)
Length	=	4.267 m
Diameter	=	2.692 m
Dry Weight	=	4441 kg
Fan Diameter	=	2.362 m
T-O Mass Flow	=	802 kg/s
Bypass Ratio	=	5.1
Number of Compressor Stages	=	4LP, 14HP
Core Air Flow	=	154 kg/s
Compressor Pressure Ratio	=	29.9
Number of Turbine Stage	=	2HP, 5LP
Specific Fuel Consumption	=	8.95 mg/Ns (0.316 lb/h/lb st)

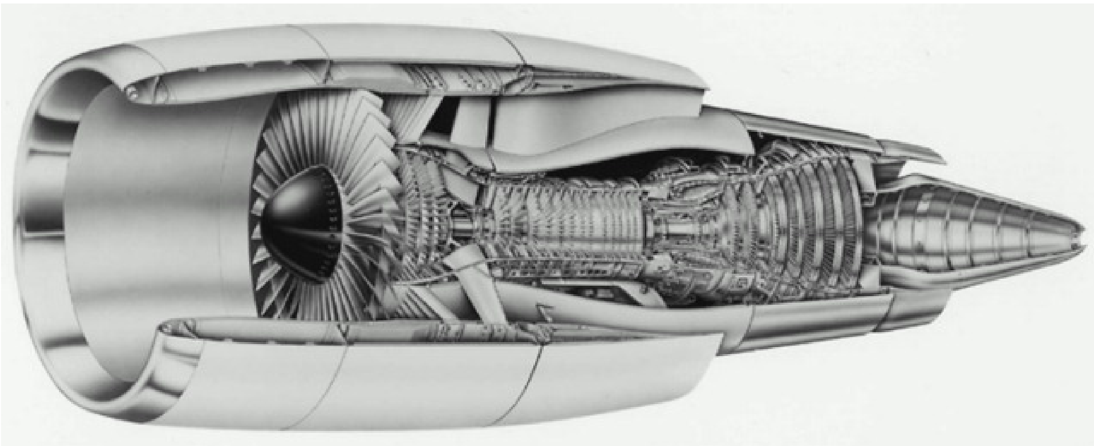

Figure 1.2: Cutaway of the General Electric CF6-80C2B1F (reference 4)

Performance:

M_{cruise}	=	0.85
Design Range	=	6185 nm (11454 km)
Cruise Altitude	=	35000 ft
Service Ceiling	=	45000 ft
Takeoff Field Length	=	2820 m
Landing Field Length	=	1905 m
Approach Speed	=	146 kts (270 km/h)

Figure 1.3: A Boeing 747-400 touching down (reference 2)

Chapter 2

Operational Characteristics

The flap speed schedules in this chapter are taken from reference 1. A reference speed is used to determine the flap extension and retraction speeds. V_{REF} is equal to 1.2 times the stall speed and depends on the aircraft weight and the flap setting. The following table obtained from reference 8 gives an overview of some values of V_{REF} in kts IAS.

Weight [ton]	360	350	340	330	320	310	300	290	280	270	260
V_{REF} 30	150	148	146	144	143	141	139	137	136	134	132
V_{REF} 25	155	153	151	149	148	146	144	142	141	139	138

2.1 Flap Extension Speed Schedule

The following flap extension initiation speed schedule is assumed (in kts IAS):

	flap setting for landing/approach					
	up	1°	5°	10°	20°	25°
flaps 1° at	$V_{REF}30+80$					
flaps 5° at		$V_{REF}30+60$				
flaps 10° at			$V_{REF}30+40$			
flaps 20° at				$V_{REF}30+20$		
flaps 25° at					$V_{REF}30$	
flaps 30° at						$V_{REF}25$

2.2 Flap Retraction Speed Schedule

The following flap retraction initiation speed schedule can be assumed (in kts IAS):

	flap setting for takeoff					
	30°	25°	20°	10°	5°	1°
flaps 25° at	$V_{REF}30$					
flaps 20° at		$V_{REF}25$				
flaps 10° at			$V_{REF}30 + 10$			
flaps 5° at				$V_{REF}30 + 20$		
flaps 1° at					$V_{REF}30 + 40$	
flaps up at						$V_{REF}30 + 60$

2.3 Maximum Operating Limit Speed

The maximum operating limit speed V_{MO} of the Boeing 747-400 is equal to 365 kts or M = 0.90.

2.4 Maximum Flaps Operating Speeds

The following maximum flaps operating speeds can be assumed.

slats, flaps up	280 kts
slats, flaps 5°	260 kts
slats, flaps 10°	240 kts
slats, flaps 20°	230 kts
slats, flaps 25°	205 kts
slats, flaps 30°	180 kts

2.5 Landing Gear Limit Speeds

The following landing gear limit speeds can be assumed.

maximum landing gear operating speed	$V_{LO} =$	280 kts
maximum speed with gear down and locked	$V_{LE} =$	260 kts

Chapter 3

Aerodynamic Characteristics

The following models of the aerodynamic characteristics of the Boeing 747-400 are derived from data in reference 1.

3.1 Lift Curves

The following lift curves were derived from figure 2.3 and are valid for the indicated range of α_{wing}.

flaps up/gear up ($4 \leq \alpha_{wing} \leq 10.5$):
$C_L = 0.025 + 0.075\alpha_{wing}$

flaps 1/gear up ($7 \leq \alpha_{wing} \leq 12.5$):
$C_L = -0.2 + 0.1\alpha_{wing}$

flaps 5/gear up ($7 \leq \alpha_{wing} \leq 14$):
$C_L = -0.065 + 0.095\alpha_{wing}$

flaps 10/gear up ($6 \leq \alpha_{wing} \leq 15.5$):
$C_L = 0.035 + 0.0975\alpha_{wing}$

flaps 20/gear up ($4.5 \leq \alpha_{wing} \leq 15.5$):
$C_L = 0.25 + 0.1\alpha_{wing}$

flaps 25/gear down ($4 \leq \alpha_{wing} \leq 13.5$):
$C_L = 0.405 + 0.10125\alpha_{wing}$

flaps 30/gear down ($4.5 \leq \alpha_{wing} \leq 13$):
$C_L = 0.64 + 0.10167\alpha_{wing}$

The angles α_{wing} and α_{body} are both in degrees and are related by the following function.

$$\alpha_{wing} = \alpha_{body} + 2^\circ$$

Unfortunately no data was available on the values of $C_{L_{max}}$.

3.2 Low Speed Drag Polar

Figures 2.6 and 2.7 are used to determine a quadratic polynomial for the low speed drag polar of the Boeing 747-400 at several flap settings. These polynomials are valid up to the indicated C_L value.
For the gear up (climb-out/landing phase), the following polynomial expressions have been determined:

flaps up ($C_L \leq 0.9$):
$C_D = 0.0233 - 0.0454C_L + 0.1037C_L^2$

flaps 1 ($C_L \leq 1.2$):
$C_D = 0.0299 - 0.0307C_L + 0.0718C_L^2$

flaps 5 ($C_L \leq 1.6$):
$C_D = 0.0344 - 0.0129C_L + 0.0571C_L^2$

flaps 10 ($C_L \leq 1.6$):
$C_D = 0.0463 - 0.0424C_L + 0.0726C_L^2$

flaps 20 ($C_L \leq 1.7$):
$C_D = 0.0387 + 0.0085C_L + 0.0402C_L^2$

For the gear down (landing phase), the following polynomial expressions have been determined:

flaps 25 ($C_L \leq 1.85$):
$C_D = 0.1003 - 0.0497C_L + 0.0687C_L^2$

flaps 30 ($C_L \leq 2.05$):
$C_D = 0.1017 - 0.0096C_L + 0.0522C_L^2$

3.3 High Speed Drag Polar

Figure 2.8 is used to determine the 1^{st} order polynomials of C_L^2 for the high speed drag. The values of the coefficients are given in a table-lookup representation for several Mach numbers. For intermediate values of the Mach number the polynomial coefficients can be found by local quadratic interpolation.
For the high speed drag of the Boeing 747-400, a parabolic drag polar is assumed which is parameterized as follows:

$$C_D(M, C_L) = C_{D_0}(M) + K(M)C_L^2$$

The following table of drag coefficients is available:

M	0.3	0.5	0.6	0.7	0.8	0.85
$C_{D_0}(M)$	0.012564	0.012469	0.012423	0.012443	0.012367	0.012173
$K(M)$	0.053463	0.05373	0.05474	0.058231	0.064035	0.070273

3.4 Landing Gear Drag Increment

The drag increment due to the extension of the landing gear can be calculated with the following equations for different flap settings derived from figure 2.14 . The ranges of lift coefficients for which the equations are valid are also given.

flaps up ($0.3 \leq C_L \leq 0.9$):
$C_{D_{gear}} = 0.0240 + 0.0152C_L - 0.0192C_L^2$

flaps 5 and 10 ($0.6 \leq C_L \leq 1.4$):
$C_{D_{gear}} = 0.0355 - 0.0230C_L + 0.00877C_L^2$

flaps 20 ($0.65 \leq C_L \leq 1.7$):
$C_{D_{gear}} = 0.0336 - 0.0197C_L + 0.00676C_L^2$

flaps 25 ($0.85 \leq C_L \leq 1.8$):
$C_{D_{gear}} = 0.0303 - 0.0175C_L + 0.00473C_L^2$

flaps 30 ($1.0 \leq C_L \leq 2.0$):
$C_{D_{gear}} = 0.0347 - 0.0292C_L + 0.00947C_L^2$

3.5 Initial Buffet Boundary

The following table obtained from figure 2.15 and table 7.35 can be used to calculate the initial buffet boundary at a certain Mach number. For a Mach number which is not in the table the appropriate value of $C_{L_{IB}}$ can be found by linear interpolation between the two closest data points.

Mach number	0.20	0.50	0.70	0.80	0.82	0.84	0.86	0.88	0.90	0.92
$C_{L_{IB}}$	.945	.783	.754	.739	.728	.711	.685	.646	.588	.505

Chapter 4

Engine Characteristics

Various models for the thrust of the Boeing 747-400 are calculated from the data in reference 1. The thrust can be set anywhere between the idle thrust and the maximum thrust, where the maximum thrust depends on the flight segment (takeoff, climb, or cruise).

$$T_{idle} \leq T \leq T_{max}$$

4.1 Installed Takeoff Corrected Net Thrust

The installed takeoff corrected net thrust per engine in pounds is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 3.3 through 3.12:

$$\begin{aligned} T_{max}(M,h) = \delta(h)[&(56283 + 1.3231h - 4.8825e^{-5}h^2) + \\ &+ (-55343 - 0.41746h + 1.3332e^{-5}h^2)M + \\ &+ (37825 + 1.1609h - 3.1028e^{-5}h^2)M^2] \end{aligned}$$

Where the quantity $\delta(h)$ is the atmospheric pressure ratio given by:

$$\delta(h) = \frac{p(h)}{p_0}$$

Where $p(h)$ is the ambient pressure. The equation is valid for an altitude range of 0 ft up to 17000 ft, a Mach number range of 0.0 to 0.6 and a temperature range of ISA −75°F to ISA +31°F, and only if the air-conditioning inside the aircraft is turned off.

In case the air-conditioning is turned on, the equation becomes:

$$\begin{aligned}T_{max}(M,h) = \delta(h)[&(55524 + 1.2927h - 4.8103e^{-5}h^2) + \\ &+ (-55509 - 0.44601h + 1.4014e^{-5}h^2)M + \\ &+ (37531 + 1.1096h - 3.1025e^{5}h^2)M^2]\end{aligned}$$

4.2 Maximum Climb/Continuous Corrected Thrust

The maximum net climb or continuous thrust per engine in pounds is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.17 through 7.19.

$$\begin{aligned}T_{max}(M,h) = \delta(h)[&(52150 + 0.72668h - 1.5837e^{-5}h^2) + \\ &+ (-53118 + 1.2828h - 8.4802e^{-6}h^2)M + \\ &+ (26330 - 0.89757h + 1.585e^{-5}h^2)M^2]\end{aligned}$$

This equation is valid for an altitude range of 0 ft up to 45000 ft, a Mach number range of 0.1 to 0.92 and a temperature range of $-40°C$ to $+10°C$. It should be noted that 3 A/C packs are turned on.

4.3 Maximum Cruise Corrected Thrust

The maximum net cruise thrust per engine in pounds is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.23 through 7.25.

$$\begin{aligned}T_{max}(M,h) = \delta(h)[&(50056 + 0.98223h - 2.1996e^{-5}h^2) + \\ &+ (-50872 + 0.67884h + 1.0478e^{-5}h^2)M + \\ &(23025 - 0.50108h + 2.3638e^{-6}h^2)M^2]\end{aligned}$$

This equation is valid for an altitude range of 0 ft up to 45000 ft, a Mach number range of 0.1 to 0.92 and a temperature range of $-40°C$ to $+10°C$. It should be noted that 3 A/C packs are turned on.

4.4 Minimum Idle Thrust

The minimum net idle thrust per engine in pounds is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.32 and 733.

$$\begin{aligned} T_{idle}(M,h) = \delta(h)[&(2723.3 - 0.28954h + 1.4242e^{-5}h^2) + \\ &+ (1620.5 + 0.61135h - 2.7882e^{-5}h^2)M + \\ &+ (-20287 - 0.023299h + 1.0421e^{-5}h^2)M^2] \end{aligned}$$

This equation is valid for an altitude range of 0 ft up to 45000 ft with ISA temperature and a Mach number range of 0.2 to 0.9. It should be noted that 3 A/C packs are turned on. Overall the accuracy of this model is acceptable but for some combinations of altitude and Mach number the error can be larger than 5%.

4.5 Reverse Thrust

Table 4 gives a method for calculating the installed reverse thrust in pounds. First of all the uninstalled reverse thrust in pounds must be calculated with the following equation which is valid for an airport pressure altitude range of 0 ft to 14000 ft and a speed range of 60 kts TAS up to 250 kts TAS.

$$\begin{aligned} T_{reverse_{uninstalled}}(V,h) = &(7823.1 - 0.21697h - 1.5904e^{-6}h^2) + \\ &+ (75.077 - 0.0015655h + 1.0705e^{-8}h^2)V + \\ &+ (0.086572 - 2.1089e^{-6}h + 1.9756e^{-13}h^2)V^2 \end{aligned}$$

During the spinup phase both an effectiveness factor and a spinup factor must be incorporated to obtain the installed reverse thrust.

$$T_{reverse_{installed}} = T_{reverse_{uninstalled}} * K_1 * K_2$$

When the reversers are fully deployed only the effectiveness factor must be incorporated.

$$T_{reverse_{installed}} = T_{reverse_{uninstalled}} * K_1$$

Both factors can be obtained from the following tables.

Flap position	Effectiveness Factor - K_1
10	0.774
20	0.750
25	0.728
30	0.671

Time from reverser deployment (sec)	Spinup factor - K_2
0.0	0.390
0.5	0.399
1.0	0.426
1.5	0.476
2.0	0.578
2.5	0.728
3.0	0.850
3.5	0.936
4.0	0.983
4.5	0.997
5.0	1.000

4.6 Corrected Fuel Flow

The minimum idle fuel flow per engine in pounds per hour is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.11 through 7.16.

$$\begin{aligned} W_F(T,M,h) = \delta(h)\theta(h)^{0.62}[&(826.15 + 2140.5M - 382.94M^2) + \\ &+ (0.2332 + 0.14859M - 0.095481M^2)T_n + \\ &+ (1.461e^{-6} + 9.01e^{-6}M - 1.0795e^{-5}M^2)T_n^2] \end{aligned}$$

Where the normalized thrust T_n is given by the following equation with the thrust in pounds:

$$T_n = \frac{T}{\delta(h)}$$

And the quantity $\theta(h)$ is the atmospheric temperature ratio given by:

$$\theta(h) = \frac{T_a(h)}{T_{a_0}}$$

Where T_a is the ambient temperature. The equation is valid for a thrust range of -2343 lb up to 62125 lb with ISA temperature, a Mach number range of 0.1 to 0.7 and 3 A/C packs on.
The equation is obtained with fuel flow data for an altitude of 5000 ft but can be used for an altitude range of 0 ft up to 10000 ft, however with a larger error.
This model for the corrected fuel flow can also be used to calculate the idle fuel flow by inputting the idle thrust. However the model for the idle fuel flow in the next paragraph is more accurate, but it is not perfectly continuous with the corrected fuel flow model.

4.7 Minimum Idle Fuel Flow

The minimum idle fuel flow per engine in pounds per hour is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from table 7.34.

$$\begin{aligned} W_F(M,h) = {} & (1464.3 - 0.027679h + 1.5314e^{-7}h^2) + \\ & + (86.964 - 0.014647h + 1.681e^{-7}h^2)M + \\ & + (-404.59 + 0.046619h - 7.332e^{-7}h^2)M^2 \end{aligned}$$

This equation is valid for an altitude range of 0 ft up to 45000 ft with ISA temperature, a Mach number range of 0.2 to 0.9 and 3 A/C packs turned on.

4.8 Ground Operation Fuel Flow

Fuel consumption during ground operation may be approximated by the following fuel flow rates.

APU average operation under normal load	660 lb/h (300 kg/h)
taxi operation	100 lb/min (45 kg/min)

Chapter 5

Engine Out Characteristics

This chapter gives some models for the engine out characteristics of the Boeing 747-400 obtained from reference 1.

5.1 Windmill Drag

From table 7.69 the following equation is derived to calculate the windmill drag $\Delta(D/\delta)$ in pounds as a function of Mach number in case of an inoperative engine.

$$\Delta(D/\delta) = e^{11.75M} + 3500M - 1$$

This equation is valid for a Mach number range of 0.0 to 0.9.

5.2 Locked Rotor Drag

From table 7.70 the drag of a locked rotor in case of an inoperative engine can be approximated with the following quadratic polynomial:

$$\Delta(D/\delta) = 3502.3 - 17716M + 33007M^2$$

This equation expresses the drag increment in pounds as a function of the Mach number and is valid for a Mach number range of 0.3 up to 0.92.

5.3 Maximum Climb/Continuous Corrected Thrust (3 Engines)

The maximum net climb or continuous thrust per engine in pounds for 3 engines operative is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.41 through 7.43.

$$\begin{aligned} T_{max}(M,h) = \delta(h)[&(49520 + 1.1142h - 3.2788e^{-5}h^2) + \\ &+ (-54640 + 0.62728h + 3.4738e^{-5}h^2)M + \\ &+ (29836 - 0.798h - 9.0638e^{-6}h^2)M^2] \end{aligned}$$

This equation is valid for an altitude range of 0 ft up to 45000 ft, a Mach number range of 0.0 to 0.92, a temperature range of −40°C to +10°C and 3 A/C packs turned on.

5.4 Maximum Cruise Corrected Thrust (3 Engines)

The maximum net cruise thrust per engine in pounds for 3 engines operative is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.47 through 7.51.

$$\begin{aligned} T_{max}(M,h) = \delta(h)[&(47261 + 1.1499h - 1.925e^{-5}h^2) + \\ &+ (-38148 - 0.15531h + 6.7361e^{-6}h^2)M + \\ &+ (9401.1 + 0.37408h + 9.0143e^{-7}h^2)M^2] \end{aligned}$$

This equation is valid for an altitude range of 0 ft up to 45000 ft, a Mach number range of 0.1 to 0.92, a temperature range of −40°C to +10°C and 3 A/C packs turned on.

5.5 Maximum Continuous Corrected Thrust (2 Engines)

The maximum continuous thrust per engine in pounds for 2 engines operative is a function of the Mach number and the altitude in feet and can be expressed by the following function, obtained from tables 7.58 through 7.62.

$$
\begin{aligned}
T_{max}(M,h) = \delta(h)[&(47811 + 1.0675h - 1.7611e^{-5}h^2) + \\
&+ (-39348 + 0.079579h - 1.4169e^{-6}h^2)M + \\
&+ (10359 + 0.23677h + 7.9601e^{-6}h^2)M^2]
\end{aligned}
$$

This equation is valid for an altitude range of 0 ft up to 45000 ft, a Mach number range of 0.2 to 0.92, a temperature range of −40°C to +10°C and 1 A/C pack turned on.

5.6 Fuel Flow Factors

The following tables of fuel flow factors from table 7.3 can be multiplied with the corrected fuel flow to account for differences in flight segment, configuration and bleed.

Configuration	Segment	Fuel Flow Factor
all engine	cruise/holding	0.9925
all engine	climb	0.9995
one engine inoperative	cruise/holding	0.9975
two engine inoperative	driftdown/climbing cruise	0.9925

Cargo A/C Pack Mode # in normal / # in high	Fuel Flow Factor
3 / 0	0.9925
2 / 1	0.9950
1 / 2	0.9975
0 / 3	0.9995

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