Aircraft Design –

Assignment [52, Assignment 2: Preliminary sizing and propulsion**]**

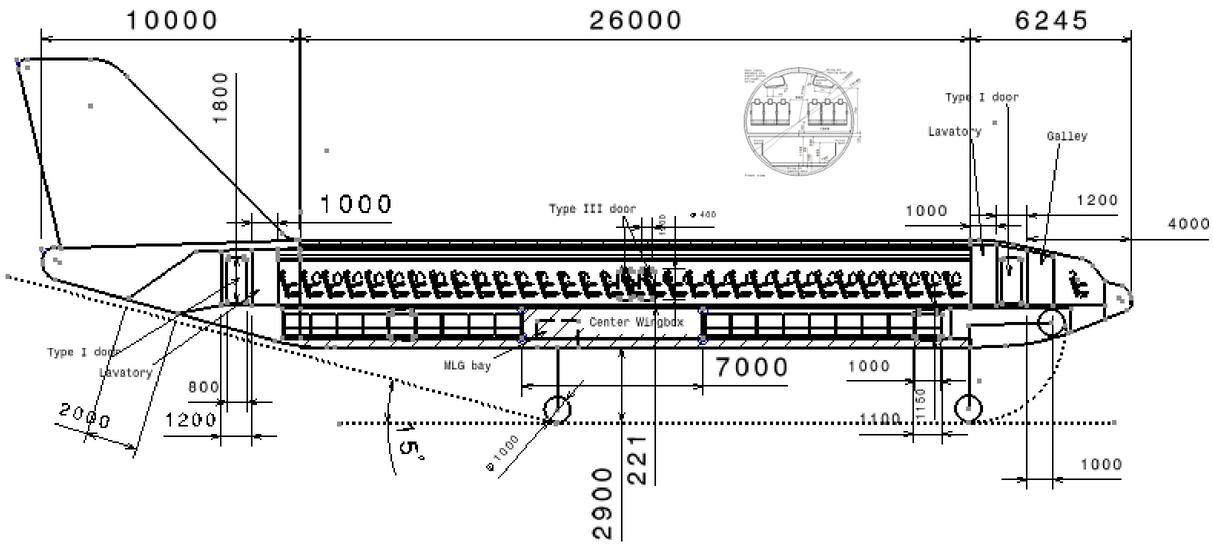
Student Names and Study Numbers:

Instance [first delivery]
Hours spent on assignment: [46]

Aircraft type: Passenger Transport Airliner
Aircraft number: 52

Requirement type	Value	Unit
Payload	190	passengers
Range	5500	km
Cruise altitude	11800	m
Cruise speed	0.82	Mach
Take-off distance	2100	m
Landing distance (at maximum weight)	1650	m

Table 1.1: Matrix Requirements



Side-view Scale =

Front view Scale =

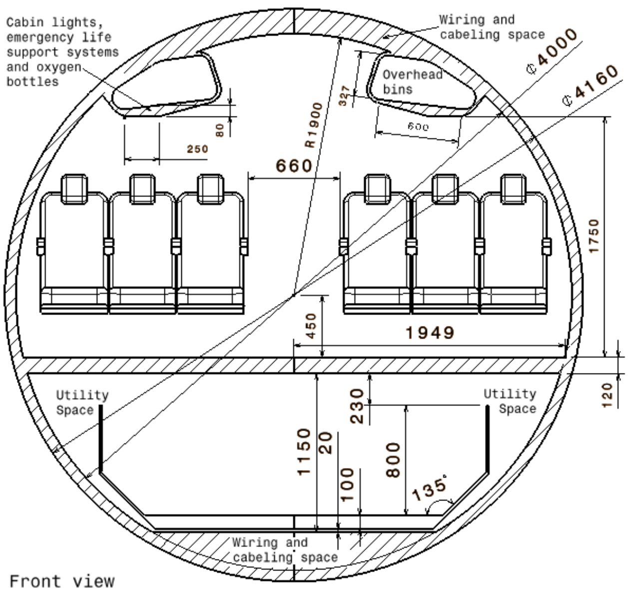


Table of Contents

1. Introduction

This report consists of 2 main parts, one dealing with preliminary sizing of an airliner, using reference aircraft as well as the given requirements, and one dealing with propulsion. The first chapter (chapter 2) will go through the calculations that are needed to estimate the take-off weight. Next, chapter 3 will show how the T/W-W/S graph is made. After that, chapter 4 deals with the generation of the drag polar. Chapter 5 will discuss the optimal takeoff weight, as function of the payload-range diagram.

The second part of this report proposes a propulsion system for our airliner in order to present an order to engine manufacturers. First, in chapter 5, the required thrust for the aircraft at relevant flight conditions will be calculated. Secondly, chapter 6 comprises a selection of the engine type, dimensions and weight. Then in chapter 7, the engine that is closest to the requirements will be selected and scaled to fit the requirements even better. Finally, in chapter 8, some propositions will be made to optimize the thermodynamic cycle of the engine and the specific fuel consumption. Here, also some checks will be performed to validate the previously obtained design-values.

2. Determining the Takeoff weight.

To determine the takeoff weight, the different parts the takeoff weight is built from should be found. The takeoff weight consists of the operational empty weight, the fuel weight and the payload weight. In the following parts, these components of the weight will be calculated. At the end of this chapter, the takeoff weight will be determined. In symbols this forms equation 2.1.

$$W_{TO} = W_{PL} + W_{F} + W_{OE} \qquad \text{Eq. 2.1}$$

2.1 Payload

As already determined in the first assignment, the payload is found by counting the weight per passenger and flight attendant and multiplying this by the number of people on board. The weight per passenger consists of the body weight of this passenger, added baggage and carry-on weight. The average weight of an adult of 80 kg can be assumed. (Reference 1) And after consulting airline policies concerning baggage per person and carry-on, an average of 20 kg of baggage and another 5kg for carry-on (3 kg for a laptop and rest for magazines/documents/bag weight), the total weight per passenger is thus set at 105 kg. Multiplying this by the sum of the number of passengers (190 required) and flight attendants (4), we get 105 * 194 = 20370 kg of payload. A summarization of this can be found in table 2.1. this number is set and should not change.

Baggage per person	20 kg
Carry-on	5 kg
Average Person Weight	80 kg
Total Passengers	190
Total Attendants	4
TOTAL Payload + Attendants	20370kg

Table 2.1: Total payload mass.

2.2 Fuel

The fuel weight of an aircraft exists of the actual used fuel and the reserve fuel (see Eq 2.2). The used fuel is a fraction of the takeoff weight, where standard reference data determines that fraction. For this calculation, the reserve fuel will be set at 0, since the fuel needed for loiter and diversion cruise maneuvers will be calculated instead, and will already be taken into account. The fraction of fuel that makes up the takeoff weight of the aircraft consists of 3 main parts: the normal flight fraction, the cruise fraction and the loiter fraction.

$$W_F = W_{Fres} + W_{Fused} \qquad \text{Eq. 2.2}$$

$$W_{Fused} = M_{used} * W_{TO} = (1 - M_{ff}) * W_{TO} \qquad \text{Eq. 2.3}$$

2.2.1 Normal Flight fraction

Normal flight components can be explained as the default elements that make up each flight. These are e.g the engine startup fraction, the taxi fraction, or climb fraction. All fractions are listed in table 2.2. From the lecture notes page 32(see references) the values for these fractions are given. These fractions are multiplied to get the total 'normal flight' fraction. Note that (according to assignment 1) the climb and descend phase are multiplied twice, since the second (diversion) cruise has to be taken into account.

Engine start and warm-up	0.99
Taxi	0.99
Take off	0.995
Climb	0.98
Descend	0.99
Landing taxi shutdown	0.992
Normal flight, with additional climb an descend	0.9106

Table 2.2: The normal flight fuel fraction.

2.2.1 Cruise fraction

When calculating the fuel fraction used in cruise, equation 2.4 should be used.

$$R = \left(\frac{V}{g.c_j}\right)_{cruise} \cdot \left(\frac{L}{D}\right)_{cruise} \ln\left(\frac{W_1}{W_2}\right) \qquad \text{Eq. 2.4}$$

This can be rewritten as:

$$\frac{W_2}{W_1} = e^{\frac{R}{\left(\frac{V}{g.c_j}\right)_{cruise} \cdot \left(\frac{L}{D}\right)_{cruise}}} = e^{\frac{R * g.c_{j;cruise}}{V_{cruise} \cdot \left(\frac{L}{D}\right)_{cruise}}} \qquad \text{Eq. 2.5}$$

Where $\frac{W_5}{W_4}$ is the mass fraction of fuel used in cruise, R is the cruise range, g is the gravitational constant (9.80665 m/s^2), V_{cruise} is the cruise-speed and L/D is the lift-drag ratio in cruise. From the requirements, R is given as 5500 km, and V_{cruise} = Mach 0.82. However, for the range, the divergence cruise should also be taken into account. The authors decided that a diversion cruise of 500 km at the same altitude as the normal cruise should be acceptable. To find the speed in SI units, the speed of sound has to be calculated at Cruise altitude. This is done using formula 2.5. The temperature at cruise level is the temperature at the given cruise altitude of 11800 m.

$$c = \sqrt{\gamma R T_{cruise}} \qquad \text{Eq. 2.5}$$

The temperature at this altitude can be found using the rule of thumb that per 1000m, the standard sea level temperature decreases by 6.5 degrees, until 11km of altitude, where it stays constant all the way through the stratosphere, until about 70km of altitude. Calculating the temperature at 11.8 km, starting from a standard sea level temperature of 15 degrees gives:

298.15 - 6.5 * 11 = 226.65K or -46.5 degrees Celsius

The speed of sound and the cruising speed then become:

$$c = \sqrt{\gamma R T_{cruise}} = \sqrt{1.4 \cdot 287 \cdot 226.65} = 301.774 m/s$$

$$V_{cruise} = M \cdot c = 0.82 \cdot 301.774 = 247.46 m/s$$

Now, For the constant Cj, which is the specific fuel consumption in cruise for the aircraft, the typical values lie between 0.5 lb/lb/hr and 0.9 lb/lb/hr. Since our aircraft is supposedly very advanced, a value of 0.55 lb/lb/hr will be used. This is equal to 0.000014927 kg/Ns in SI units. And finally, the Lift-Drag ratio L/D usually ranges between 13 and 15 for modern airliners in cruise. Our aircraft will fly with an L/D of 14.5. These values can be filled in in eq. 2.5:

$$\frac{W_2}{W_1} = e^{\frac{R * g . c_{j;cruise}}{V_{cruise} \cdot \left(\frac{L}{D}\right)_{cruise}}} = e^{\frac{6000000 \cdot 9.80665 \cdot 0.00001493}{247.46 \cdot 14.5}} = 0.7829$$

The fuel fraction used during the two cruises is 0.7829.

2.2.1 Loiter fraction

Similarly, the fraction during the loiter maneuver can be found. Eq. 2.6 is used. Here the g is still the gravitational parameter, c_j the specific fuel consumption during loiter, L/D the lift to drag ratio during loiter and E the Endurance (the time the loiter is performed).

$$E = \left(\frac{1}{g . c_j}\right)_{loiter} . \left(\frac{L}{D}\right)_{loiter} \ln\left(\frac{W_1}{W_2}\right) \qquad \text{Eq. 2.6}$$

For the endurance, the authors decided that an endurance of 45 minutes should sufficient to fit all regulations. The L/D during loiter is usually between 14 and 18. Here the authors opted for 16.5, giving a high lift, but on the other side no extensive research has to be done in the flaps of the wings. The Specific fuel consumption is normally between 0.4 lb/lb/hr and 0.6 lb/lb/hr. Here a value of 0.45 lb/lb/hr was chosen, which gives 0.000012213 kg/Ns. Reforming eq. 2.6 and filling in the values in SI units gives:

$$\frac{W_1}{W_2} = e^{\frac{E \cdot g \cdot c_{j;loiter}}{\left(\frac{L}{D}\right)_{loiter}}} = e^{\frac{45 * 60 \cdot 9.80665 \cdot 0.000012213}{16.5}} = 0.9806$$

The fuel fraction used during the loiter maneuver is 0.9806.

2.2.1 Total Fuel

The total fuel fraction can be calculated by simply multiplying all the above calculated fractions.

$$M_{ff} = loiterfraction \cdot cruisefraction \cdot normalflightfraction$$
$$M_{ff} = 0.9806 \cdot 0.7829 \cdot 0.9106 = 0.69907$$
$$W_f = (1 - M_{ff}) \cdot W_{To} = 0.30092 \cdot W_{To} \qquad \text{Eq. 2.7}$$

2.3 Operational Empty weight

Finally, the operational empty weight can be calculated in function of the takeoff weight. The operational empty weight consists of the weight of the crew, the empty weight and the weight of the trapped fuel and oil. This is summarized in Eq 2.8.

$$W_{OE} = W_E + W_{Tfo} + W_{Crew} \qquad \text{Eq. 2.8}$$

The weight of the trapped fuel and oil is a fraction of the fuel weight, which lies between 0.001 and 0.005. The authors decided to go for 0.002. The Crew weight can be calculated in the exact manner of the payload weight, giving 105 * 2 = 210 kg for two pilots. And finally the empty weight is also a fraction of the takeoff weight. But to determine what fraction, a closer look should be taken at the reference aircraft. Plotting the reference aircraft in a 2d plane with on the x-axis the takeoff weight and on the y-axis the empty weight, gives figure 2.1. then the trend line can be found from the data points. Note that the point furthest to the right is the A380. And of course, the takeoff and empty weight of the world's biggest airliner is huge. This is why it 'sticks out' on the graph.

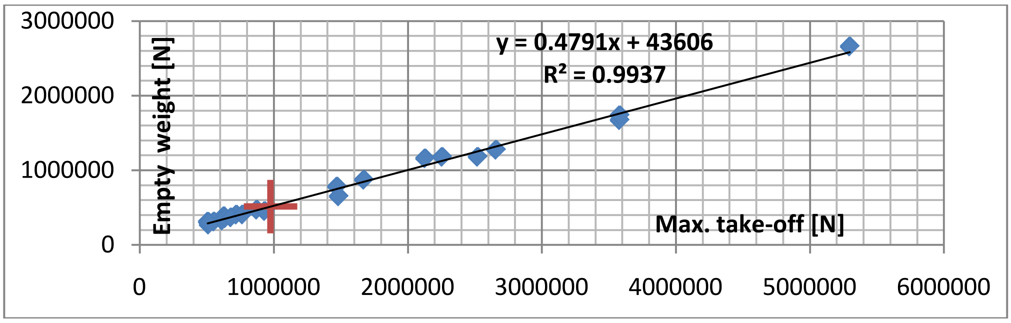


Figure 2.1: Maximum takeoff weight versus empty weight of reference aircraft.

From this graph, the formula of the empty weight in function of the takeoff weight can be found. Equation 2.9 shows this formula in SI units.

$$W_E = 0.4791 \cdot W_{To} + 43606 / g \qquad \text{Eq. 2.9}$$

2.4 Total Weight Estimation

So, from previous equations, we get:

$$W_{TO} = 20370 + 0.330893 \cdot W_{To} + 0.002 \cdot W_{To} + 0.4791 \cdot W_{To} + 43606 + 210$$

$$W_{TO} = \frac{20370 + 43606 / 9.80665 + 210}{1\text{-}0.300921 - 0.4791 - 0.002} = \frac{25027}{0.21798} = 114857.2\text{kg} \qquad \text{Eq. 2.10}$$

This gives us a takeoff weight of 115 metric tons. When compared to the reference aircraft, this is a lot (it even sticks out 35 tons above its competitor, the 737-800). The problem is in the fact that this weight is for full load, over full cruise range! It would be a lot better for the design weight to be set lower, and that the aircraft has better fuel consumption. This has been done in figure 2.2: the payload- range diagram.

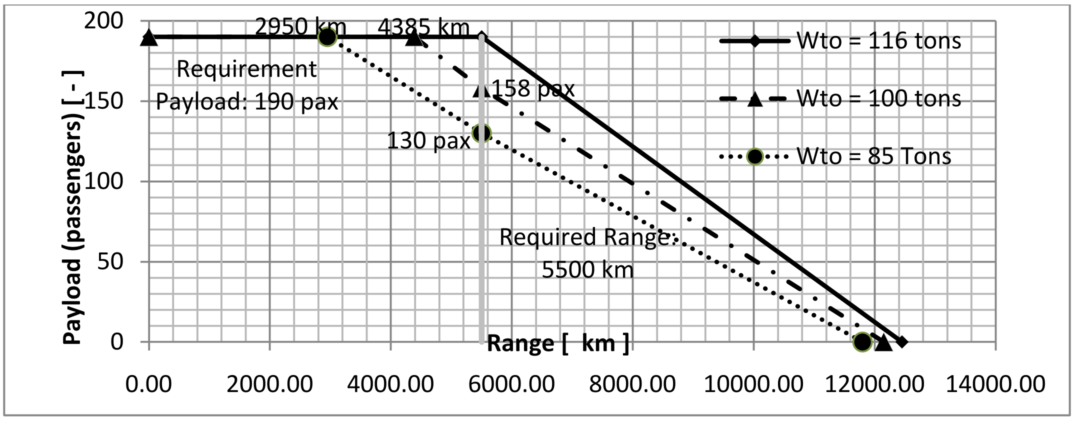


Figure 2.2: The simplified Payload-Range Diagram.

As can be seen from this graph, it would be better to set the take-off weight to 100 tons, to get a better weight, and then not carry the full payload over the full range. Setting it to this W_{to} would allow the transport of the full payload over 4385 km, or a 3-class configuration of the aircraft (in the first report, this was set at 158 seats) for the full range. So the authors will continue with a 3-seat variant, weighing 100.000kg at takeoff and carrying 158 passengers, to improve the performance of this jet. For this takeoff weight, the empty weight is 52.400 kg, and when checked in the graph in figure 2.1m this is on the trend line. Note that the part of the graph right of the maximum range depends on the size of the fuel tanks, which have not been designed yet (of course, a range of 12000km is absurd for this aircraft, since it would mean that almost the whole fuselage is filled with fuel, so the line should go down steeply after reaching the design range)…

3. Determining the T/W-W/S Diagram.

3.1 Stall Sizing

Given is the following equation from lift:

$$\frac{W}{S}=\frac{1}{2}.\rho.V_S^2.C_{L_{max}} \qquad \text{Eq. 3.1}$$

Where Cl_{max} and V_s are unknown. From references, we can estimate these values as following:

Configuration	Cl_{max} [-]	V_{stall} [m/s]
Clean	1.7	77
Take off	2.0	73
Landing	2.4	60

Table 3.1: Clmax and Vstall for different configurations.

Using eq 3.1:
W/S clean = 6173 N/m^2
W/S land = 5292 N/m^2
These values are added on the T/W – W/S graph in appendix A

3.2 Take-off Sizing

All important elements of the take-off are combined in the so-called take-off parameter.

$$TOP_{jet}=\left(\frac{W}{S}\right)_{TO}.\left(\frac{W}{T}\right)_{TO}.\frac{1}{C_{L_{max}}}.\frac{1}{\sigma} \qquad \text{Eq. 3.2}$$

W/S and W/T are the variables, sigma is the relative density with relation to sea level (rho/rho_0) and Cl_{max} is a variable which, for commercial airliners lies between 1.6 and 2.2. The TOP- value can be found from a graph with the TOP of the reference aircraft in relation with their take-off distance. This graph is shown in figure 3.1

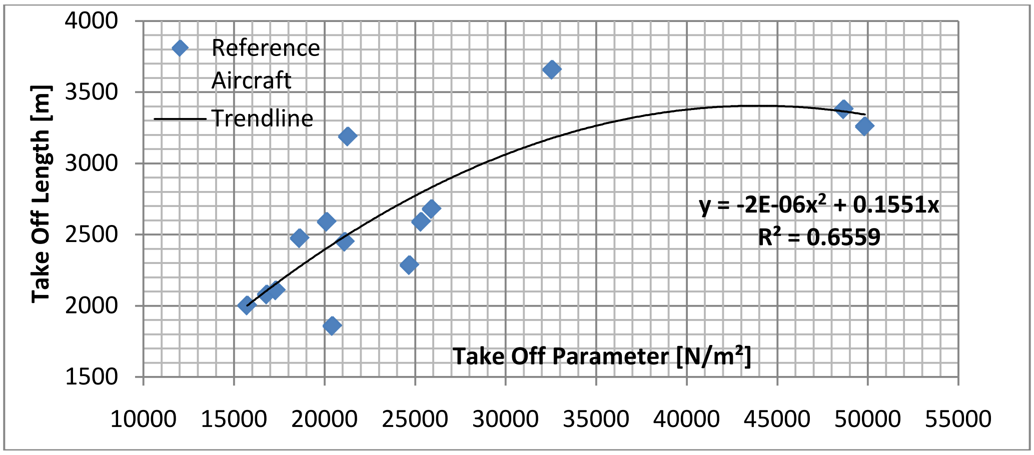


Figure 3.1: The Take off distance as function of the TOP.

Using the given take off length, the take off parameter can be found by simply substituting it in the equation given by the trend line. Using a value of 2100m returns a TOP of 17541 or 59859. For future reference only the first TOP will be used, namely 17541. Now every parameter of equation 3.2 is known, and it can be plotted in the T/W - W/S graph in appendix A.

3.3 Landing Sizing

According to statistics,

$$s_L = 0.5847 * V_{s_{land}}^2 \qquad \text{Eq. 3.3}$$

Using statistical data, W_l/W_{to} = f = 0.75 then;

$$\left(\frac{W}{S}\right)_{TO} = \frac{C_{L_{max}} \cdot \rho \cdot \frac{s_{land}}{0.5847}}{2.f} \qquad \text{Eq. 3.4}$$

By varying $Cl_{max,Land}$, the vertical Lines of (W/S) to can be plotted in the graph in Appendix A.

3.4 Drag Polar

To determine and comply with the climb requirements, the drag polar has to be found. This can be done using eq. 3.5.

$$C_D = C_{D_0} + \frac{C_L^2}{\pi A e} \qquad \text{Eq. 3.5}$$

Where $C_{D_0} = \frac{f}{S}$, in which f is the parasite drag and S is the wing area. To find f, use is made of eq 3.6 and eq 3.7.

$$f = 10^{(a+b \cdot {}^{10}\log S_{wet})} \qquad \text{Eq. 3.6}$$

$$S_{wet} = 10^{(c+d \cdot {}^{10}\log W_{to})} \qquad \text{Eq. 3.7}$$

Here, W_{to} is the takeoff weight in lbs, S_{wet} is the wetted area of the wing (in ft^2), and a, b, c and d are determined from statistical data (a and b are dependant of the friction coefficient which was chosen to be 0.03, this gives a = -2,5229 and b = 1, while statistics show that for commercial jets, c = 0.0199 and d = 0.7531). Using these values gives a wetted area of 1023m2 and an f of 3.07m2. This combined gives a C_{D_0} of 0.01283. This is low, since this value should be between 0.014 and 0.02. After a brief examination of the reference aircraft, it can be seen that these aircraft are exactly the size or larger than our aircraft in terms of takeoff weight. Taking only reference aircraft in the close range of our aircraft gives a C_{D_0} of 0.019. Finally, we set e, the Oswald efficiency factor, at 0.8. Eq 3.5 now has 3 unknowns: A, Cl and Cd. which can be plotted out as a graph. In this graph, the curves for all values have also been plotted in case of a takeoff, approach or landing (= approach with gear out) or (this increases the Cd0 with 0.015, 0.07 and 0.09 respectively).

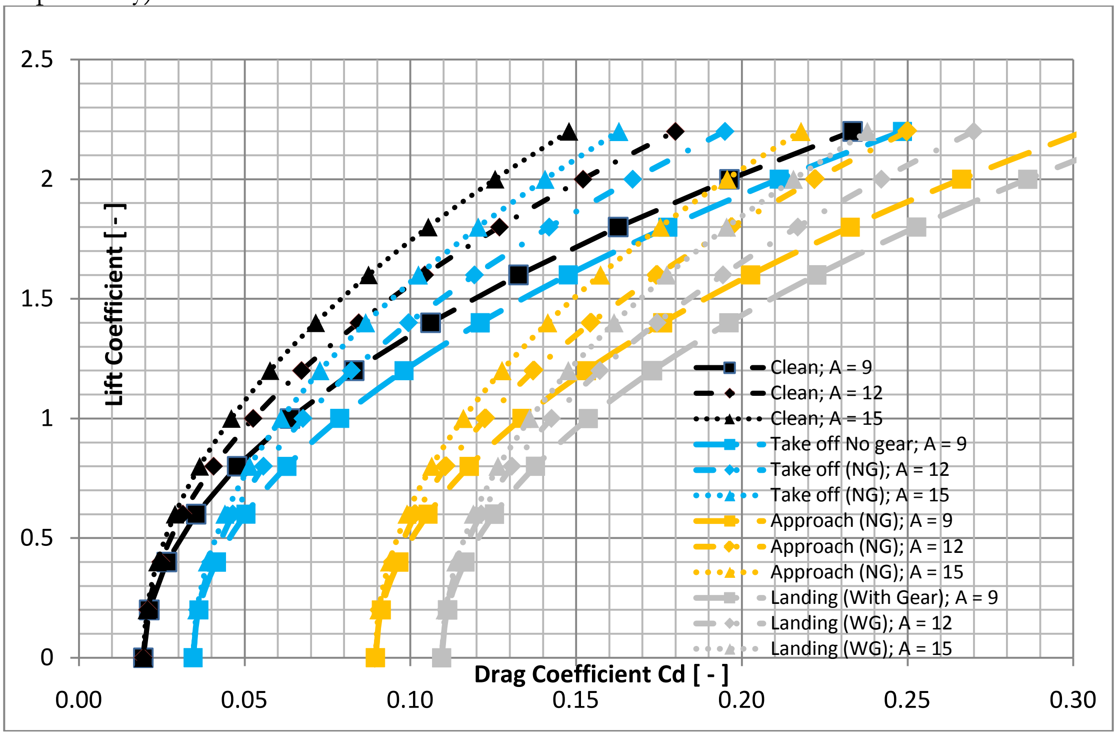


Figure 3.2: The drag polar (Cl – Cd curve).

3.5 Climb performance requirements

In order to determine the climb performance requirements, reference is made to certification regulations. For our aircraft, the climb rate was set at 18m/s, for maximum continuous power and flaps in take-off position. The T/W relation in function of W/S is then:

$$c = \left(\frac{T}{W} - \frac{C_D}{C_L}\right) \cdot \sqrt{\frac{W}{S}\frac{2}{\rho}\frac{1}{C_L}} \qquad \text{Eq. 3.8}$$

And when maximized for climb, this gives:

$$C_L = \sqrt{3C_{D_0}\pi A e}$$

$$C_D = 4C_{D_0}$$

With Cd0 = 0.019.

This can be plotted in the T/W – W/S graph, and these results in the curved lines that run from the top-left to the right side.

3.6 Climb gradient requirements

To find the optimal climb gradient requirements, the maximum lift to drag ratio has to be found. The T/W at which this counts can be done with eq. 3.9:

$$\frac{T}{W}=\frac{c}{V}+\frac{C_D}{C_L} \qquad \textit{Eq. 3.9}$$

This is not related to W/S, so the graph is a horizontal line. To find the T/W, the maximum C_d/C_l is necessary. This is so if

$$C_D = 2\cdot C_{D0}$$

$$C_L = \sqrt{A\cdot e\cdot \pi\cdot C_{D0}}$$

The value of C_{d0} is known, e is the Oswald efficiency factor, which was set at 0.8 and the aspect ratio A is variable. The result can be seen in chapter 4.

4. T/W – W/S Diagram

Combining all calculations done in chapter 3, gives the T/W – W/S diagram. From this diagram the, to this stage, final set of important characteristics in preliminary sizing can be found. This diagram is depicted in appendix A.

A value of T/W of 0.26 was chosen, along with a wing loading of 6150 N/m^2.

The resulting wing area is then 157.94 m^2, and the required take-off thrust is set at 126.76kN per engine.

A summary of the most important values is given in table 4.1.

Table 4.1: Most important Weight parameters.

Parameters	Value	Unit
$c_{j\ cruise}$	0,55	lb/lbf/hr
$c_{j\ loiter}$	0,45	lb/lbf/hr
$c_{j\ climb}$	0,6	lb/lbf/hr
W_{to}	99400	kg
W_E	52409	kg
W_w	29918	kg
W_{Pl}	17010	Kg (158 pax)
S_{wing}	157.94	m^2
A	9	-
e	0,8	-
climb gradient c/V	18	m/s
Rho_{cruise} (at 11800 m)	0,3018	kg/m³
climb gradient engine failure (c/V)	0,024	-
cruise velocity	247,5	m/s
$N_{engines}$	2	-
C_{d0}	0,0194339	-

5. Propulsion design

In this chapter, the required thrust for takeoff, climb and cruise flight will be calculated using the design parameters determined in Part 2a. Those parameters are listed in table 2. Also taking into account the airworthiness requirements (CS/JAR/FAR), for this transport aircraft more than one engines are required. For this aircraft two engines are used.

Table 5.1: Our aircraft parameters from the previous part

Parameters	Value	Unit
$c_{j\ cruise}$	0,55	lb/lbf/hr
$c_{j\ loiter}$	0,45	lb/lbf/hr
$c_{j\ climb}$	0,6	lb/lbf/hr
W_{to}	99400	kg
W_E	52409	kg
Fuel Weight	29918	kg
A	9	-
e	0,8	-
climb gradient c/V (15 - 18 %)	0,18	-
Rho_{cruise} (at 11800 m)	0,3018	kg/m³
climb gradient engine failure (c/V)	0,024	-
cruise velocity	247,5	m/s
$N_{engines}$	2	-
C_{d0}	0,0194339	-

	climb	cruise	climb engine failure
C_l	0,9919	0	0,57269
C_d	0,058	0,029	0,029

5.1 $T_{req,TO}$ all engines operating

From previous chapters, the thrust to weight-ratio T/W was set to be 0.26 and the takeoff weight is 975083 N. The required thrust per engine then becomes:

$$\left(\tfrac{T}{W}\right)_{TO} = 0.26 \rightarrow T_{TO} = \frac{0.263 \cdot 975083N}{2} = 126.76\text{kN} .$$

5.2 $T_{req,cruise}$ all engines operating (AEO)

In climb flight, the following equation from chapter 3 can be used again, but this time in combination with the following relations for C_l and C_d. Now, from table 5.1, c/V is estimated to be around 18%, the values of C_{D_0}, A, e and the takeoff weight are listed and give a required climb thrust of 116.26 kN per engine.

$$\frac{T}{W} = \frac{c}{V} + \frac{C_D}{C_L} \qquad \text{Eq. 3.9}$$

$$C_L = \sqrt{3 \cdot C_{D_0} \cdot \pi \cdot A \cdot e}$$

$$C_D = 4 \cdot C_{D_0}$$

In the requirements matrix (Table 1.1), the cruise speed and cruise altitude are given. At this altitude the air density is 0.310 kg/m^3. Assuming that in cruise flight Drag is equal to Thrust, equation 4.1 is valid using the values listed in table 5.1 for S and a C_d, Thrust cruise = 21.81 kN per engine.

$$T = D = C_D \cdot \frac{1}{2} \cdot \rho \cdot V^2 \cdot S \qquad \text{Eq. 5.1}$$

5.3 $T_{req,climb}$ one engine inoperative (OEI)

According to " *CS/JAR/FAR 25.121 Climb: one-engine inoperative*", the (minimum)climb gradient engine failure (c/V) is set to be 2.4%. One engine should thus produce enough thrust to keep on taking on climbing and flying when the other engine has failed. Using eq 3.9, with $C_L = \sqrt{C_{D_0} \cdot \pi \cdot A \cdot e}$ and $C_D = 2 \cdot C_{D_0}$ for normal, regular flight conditions gives climb thrust = 72.78 kN.

In table 5.2, the required thrust for relevant flight conditions is listed.

Table 5.2: The required thrust for relevant flight conditions per engine in kN

T_{TO} per engine	126.76
T_{climb} (max c_l/c_d required)	116.26
T_{cruise}	21.81
$T_{climb\ engine\ failure}$	72.78

6. Engine Selection and modification

In this section, a selection of existing engines that fits most the thrust requirements, defined above, is made and the engine that is closest to the requirements is selected using trade off methods.

6.1 Good choices

From the given requirements in tables 5.1 and 5.2, high bypass turbofan engines were selected using the "Jane's aero-engine directory" and tabulated in table 6.1. The choice of turbofan engines is because these engines are as good as turboprops in terms of fuel efficiency and the fan is enclosed by the inlet and composed of many blades; it can operate efficiently at highest speeds far exceeding those of a simple propeller. This allows it to cruise with Mach 0.82.

When analyzing the current engine-use and market with respect to our required thrust (126 – 138 kN), the CFM International CFM56 manufacturer is clearly dominating the list and that's because their engines have a high efficiency (0,32 – 0,4) and a high dispatch reliability of 99.96.

6.1: Existing engines fitting most the thrust requirements

Manufacturer Model	CFM 56-5B1/P	CFM 56-5B2/P	Kuznetsov NK-86	IAE V2500-A1
Certification/ IES	Feb 94	May 93	July 1974,	May 1989
T-O thrust kN (ISA +15°C)	133.50	137.90	127.5	133.4
Cruise thrust kN	25,976	25,976	31.6	25.6
Mass flow kg/s	427.7	433.6		357.9
Flat-rate T°F(°C)	86 (30)	86 (30)		
BPR	5.5	5.5		
Maximum climb kN	28.556	28.556		
OPR	34.4	34.4	28.8	
Maximum cruise lb	5,840	5,840		
Redline EGT °C	950	950		
Fan rpm	5,200	5,200		
Core rpm	15,183	15,183		
LP Comp. stages	3	3		3
HP Comp. stages	6	6		10
Pressure ratio	32	32.9	12.93	31.6
Bypass ratio	5.5	5.6	2.4	5.4
SFC T-O, S/L:	16.98 mg/Ns (0.600 lb/h/lb)	16.98 mg/Ns (0.600 lb/h/lb)	14.74 mg/Ns (0.52 lb/h/lb)	
SFC cruise	0.324-0.354 lb/h/lb	0.324-0.354 lb/h/lb	20.97 mg/Ns (0.74 lb/h/lb)	16.26 mg/Ns (0.575 lb/h/lb)
Dimensions in m				
Length (flange to flange):	2,601	2,601	3,638	2,960
with reverser			5,278	
Fan diameter: m	1,735	1,735	1,455	1,613
Inlet diameter (all)			1,600	
Weight, Dry [kg]	2119		2750	2242
Applications	A321	A321	Ilyushin Il-86	A320

6.2 The CFM56 -5B1/P

This engine is the best-selling airliner engine of all time, not only because it is the only one with specific numbers about reduced emissions, but also because it has a high thrust, enough for the required take off. It is also limited in size and has a low weight (2119 kg) making it perfect for this design. However the specific fuel consumption at take off is higher 0.6 lb/h/lb compared to the Kuznetsov 0.52 lb/h/lb, but the CFM has a low weight compared to the Kuznetsov which has also a low max TO thrust (127.5 kN vs. 133kN of the CFM56-5B1/P).

6.3 The CFM56 -5B2/P

This engine is the updated version of the CFM56 -5B1/P. It produces more max Thrust 137 kN compared to 133 kN of the CFM56 -5B1/P and reduces NO_x emissions by more than 45 per cent compared with other engines of series -5B. Although this engine will make a nice green choice for our aircraft, it will still need to be scaled from 137 kN to our required thrust of 126,7 kN which will make it costly in the detailed design portion.

6.4 Kuznetsov NK-86

This engine should be the best fitting engine for our aircraft in term of the takeoff thrust, 127 kN compared to 126 kN required for our aircraft. But unfortunately it is much heavier, less efficient and has large dimensions compared to the other engines.

6.5 IAE V2500-A1

This engine is also less efficient (0.575 lb/h/lb versus 0.37 lb/h/lb) and it is heavier and has slightly less TO thrust (133.4 versus 133.5 kN compared to the CFM56 -5B1/P).

6.6 Selection of the optimal engine

With respect to the requirements, The CFM56-5B1/P is chosen as the design engine. This because it not only fits the requirements for our aircraft, but it also complies with current emission laws (reduced emissions). It also has a low weight and a lower specific fuel consumption, especially during cruise compared to other suitable engines. The latter is the most important parameter for our airliner. because most of the flight time is spent in the cruise flight and most of the fuel is consumed there. This gives a low SFC_{cruise}, meaning a low fuel-weight to be taken aboard for the flight..

7. Engine Scaling of the CFM56 -5B1/P

In table 7.1, the parameters values of the CFM56 -5B1/P and the parameters values of our aircraft are put together. Clearly the take off thrust of the CFM56 -5B1/P is higher than the required thrust by our aircraft (133.5 vs. 126,76 kN).

Table 7.1: Specifications of The CFM56 -5B1/P and of the design aircraft

Specifications	CFM56 -5B1/P	Our aircraft
Performance Ratings	5B1/P	
Certification	Feb 94	2009 - …
T-O thrust kN to ISA + 15°C	133.50	126.76
Cruise thrust kN	25.976	20.70
Climb thrust kN		116.26
Climb thrust engine failure kN		72.78
Max sfc	0.600	
Cj Cruise lb/h/lb	0.324 - 0.6 lb/h/lb	0.55
Cj Loiter lb/h/lb	0.324-0.354 lb/h/lb	0.45
Cj Climb lb/h/lb		0.6
Length (flange to flange) m	2.601	
Fan diameter: m	1.735	
Weight, Dry	2.119 kg (4,672 lb)	

Using Turbo-Jet Engine Scaling as depicted in the course lectures, this engine can be scaled according to empirical formulas. For scaling the engine, the following equations are given:

$$D = D_{ref} \cdot \left(\frac{T}{T_{ref}} \right)^{\frac{1}{2}} \qquad \text{Eq. 7.1}$$

$$L = L_{ref} \cdot \left(\frac{T}{T_{ref}} \right)^{\frac{2a-1}{2}} \qquad \text{Eq. 7.2}$$

$$W = W_{ref} \cdot \left(\frac{T}{T_{ref}} \right)^{a} \qquad \text{Eq. 7.3}$$

Selecting a = 1 and noting that subscript ref refers to the CFM56-5B1/P parameters, the scaled values are obtained. The newly scaled values are shown in table 7.2.

Table 7.2: The Engine after rescaling

		CFM56-5B1/P	New values
T_{ref}	T-O thrust to ISA + 15°C [kN]	133.5	126.76
L_{ref}	Length (flange to flange) [m]	2.601	2.530
D_{ref}	Fan diameter: [m]	1.735	1.69
W_{ref}	Weight, Dry [kg]	2119	2012.01

8. Required checks and engine improvements

In this section, a sanity check will be made to see if the amount of fuel calculated in the previous chapters is sufficient for the specified range with the selected engine. Also on basis of the design parameters, i.e. the fuel weight and engine weight and the payload will be checked. If needed some parameters will be changed in order to reach the performance requirements. Browsing back through previous chapters, we summarize the design parameters in table 8.1.

Table 8.1: The design parameters

cj cruise	0,55	lb/h/lb		
cj loiter	0,45	lb/h/lb		
Cj climb	0,6	lb/h/lb		
WTO	99400	kg	975083	N
WE	52409	kg	513954	N
Fuel Weight	29918	kg		N
Payload	17010	kg		
Range	4385 – 5500	km		
V cruise	0.82	M		
L/D cruise	14.5			
fuel fraction cruise	0.7829			

8.1 Fuel weight check

To calculate the fuel weight, one can consider the fuel necessary for different phases of the flight. Using eq. 8.1 the fuel necessary for the first cruise (cruise1) and the loiter phases can be calculated.

$W_F = SFC * E * T_{cruise}$ *Eq. 8.1*

The thrust in eq. 8.1 is obtained from figure 8.1, which shows the plot of thrust to max thrust ratio vs. The corresponding Mach number.

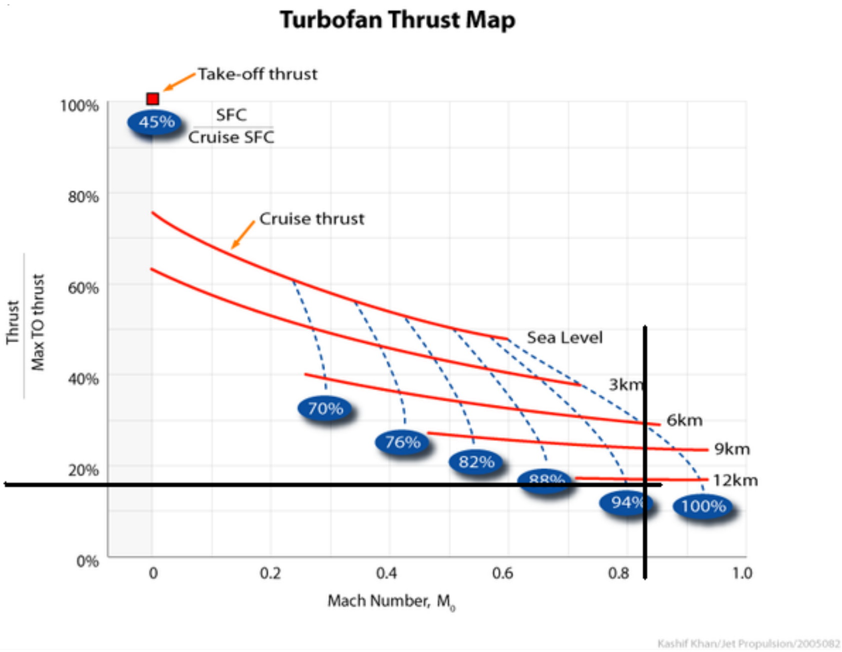


Figure 8.1: Turbofan Thrust Map(ref lecture notes)

Indicating the thrust to maximum take of thrust ratio, with the corresponding cruise Mach number of 0.82, a value of 18% can be obtained which is quite correct compared to table 8.1.

Table 8.2: T_{cruise} / T_{max}

Fuel weight check	T_{cruise} [kN]	T_{max} [kN]	T_{cruise} / T_{max} -
Our engine	21,8	126	0,17
CFM56-5B1/P kN	126	133,7	0,19
From Turbofan Thrust Map			0,18

Therefore the thrust at cruise altitude of 11800 m becomes 22,81 kN for one engine and 45,63 kN for 2 engines. The weight fractions obtained before will be used to determine the fuel necessary for the other phases. Keeping the take-off weight constant, 99431 kg, the necessary fuel weight for the engine start and warm up, taxi, take-off and the first climb is found to be,

Wf 1= (1-(0.99*0.99*0.995*0.98) WTO = 4405,23 kg

The weight at the end of the first climb then becomes 95026 kg. The necessary fuel weight Wf2 for the first cruise phase will be calculated using equation 8.1, with E (endurance) for cruise time plus loiter time:

$$(W_F)_2 = (SFC)_{cruise} * E * T_{cruise} = (16.98*10^{-6}*24918.96)*50704.32 = 16660.36 \text{ kg}$$

with $E_{cruise1+loiter} = \dfrac{5500*10^3}{247} + 2700 = 24918.96s$, and $(SFC)_{cruise} = 15.56*10^{-6} mg/Ns$

The weight at the end of the first cruise W2 becomes 75359 kg. Then using weight fraction from before, the necessary fuel weight for first descent, second climb, Wf3, is found to be:

$(W_F)_3 = (1-(0.99*0.98))*(W2) = 2245.70kg$

The weight at the end of the first descent and the second climb, W3, is 73114 kg. The necessary fuel weight for the second cruise phase, Wf4, will again be calculated using equation (6),

$(W_F)_4 = SFC*E*T = (15.56*10^{-6}*2019.90*50704.32) = 1594.13$ kg

with $E_{cruise2} = \dfrac{500*10^3}{247} = 2018.90s$, $SFC = 15.56*10^{-6} mg/Ns$

The weight at the end of the second cruise and the loiter phase, W4, is 71519 kg. Finally, the necessary fuel weight for the second descent, landing, taxi and shut down, Wf5, is found to be,

$(W_F)_5 = (1-(0.99*0.98))*(W4) = 1281.62kg$

The total fuel weight therefore becomes:

$W_F = (W_F)_1 + (W_F)_2 + (W_F)_3 + (W_F)_4 + (W_F)_5 = 29193.06kg$

The value calculated for the fuel weight in part 2a is 29918 kg which is slightly larger than the just calculated value by 735 kg or 2.4 %. This can be seen as part of the reserve fuel amount to be taken. Also by lowering the SFC from .55 to .4 lb/h/h, the total fuel weight becomes lower, namely 23655.62 kg. which means that the aircraft can perform even better (larger range or take more payload) since the design engine has high efficiency (sfc between 0.35 – 0.6). But here the design sfc was chosen to be 0.55 according to part 2a which should be lowered after general discussion with the engine manufacturer and for the final design.

8.2 Engine weight check

The weight of the selected engine was one of the reasons that the engine CFM56 -5B1/P was selected. Comparing the percentage of the engine weight to the empty weight of this aircraft with other similar aircrafts in the reference, (see table 8.3), it is obvious that the selected engine, the CFM56 -5B1/P, is lighter. Its weight consists only of 8.09 % of the empty weight. For all other aircraft, this value is above 10 % of their aircraft empty weight. Also looking at the NK 86 and the V2500-A1 which were considered during the selection of the engine, and taking their engine weight percentage to our empty weight (52409 kg), just in case those engines were chosen, one can clear see, in column 7 in table 8.3, that the CFM56 -5B1/P is still the lightest one.

Table 8.3: W_{Engine}/W_E in %

Airliner	$N_{Engines}$	Engine	W_E [kg]	W_{Engine} [kg]	W_{Engine}/W_E %	Our engines
our aircraft	2	CFM56 -5B1/P	52409	2119	8.09	8.09
A320-200	2	CFM56-5A3	41310	2265,68	10.96	8.65
A320 231	2	V2500-A1	42400	2242	10.58	8.56
B737-800	2	CFM56-7B24	41480	2365,9	11.41	9.03
Il-86	4	4x NK 86	115000	2750	9.57	10.49

8.3 Payload check

The total weight of the engine is stated as 2x 2119 kg, which is about 8.1% of the empty weight. The fuel weight is previously calculated and has a value of 27791 kg. The value calculated previously is larger with a difference of 735 kg. The take-off weight will change, but since the change is small, it is assumed to be constant. This airliner is thus able to transport more than the normal payload of the aircraft which can be for example, more payload (pax or cargo) or extra fuel for a larger range.

8.4 Improvements on Specific Fuel Consumption

In this part, a few ways of improving the specific fuel consumption (SFC) are proposed. This will help the engine manufacturer when improving the engine thermodynamic cycle will be considered.

8.4.1 The specific fuel consumption (sfc):

By definition, the specific fuel consumption of an engine describe the fuel efficiency of the engine with respect to its mechanical output. This SFC has a large influence on the amount of fuel required for a certain amount of work. That is why having a SFC as low as possible is beneficial, meaning that the engine needs a low amount of fuel to produce a specific amount of thrust for a certain period. This will mean that less fuel will be required for a certain range which will reduces costs, the fuel weight and emissions, which is preferred.

8.4.2 Proposed Improvements:

The figure below represents a plot of the thrust T vs. entropy S which represents the thermodynamic cycle of a turbojet engine.

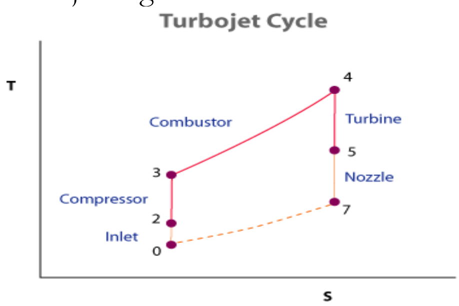


Figure 2: Thermodynamic cycle of a turbojet engine(ref. lecturenotes)

From figure 2 it can clearly be seen that the total amount of work produced is the line from point 3 to 4. To increase the amount of work produced by a certain amount of fuel, the entropy of point 4 should be increase, or the temperatures for point 3 and 4 should be increased. A few ways to do this are:

Increasing the compression ratio. This can either be done by the compressor or already at the inlet. This requires a more advanced compressor which will also increase the total weight of the engine.

Reduce losses between the turbine and compressor. The turbine operates the compressor. The amount of energy the turbine requires can be found in the figure as line 2-3 and the amount of energy it delivers at the compressor is line 4-5. If this delivery would be ideal, both lines would be of the same length making point 3 reach a higher value, and thus the total amount of work produced by the engine increases.

Higher combustion temperature. This is usually limited by the chemical behavior of the propellant.

So in order to find a better fuel consumption, these optimizations would be advised to the engine manufacturer.

Sources and references

Websites
Info on average human weight.
http://www.cdc.gov/nchs/data/ad/ad347.pdf

Information on aviation regulations
http://www.easa.eu.int/home/

General engine data
http://www.howellinst.com/HowellWeb/pdf/EngineRefByManufact.pdf

General information on aircraft design & Information on existing aircraft.
Also used to find engine data.
http://www.janes.com

Data on Large Turbofan Engines
http://adg.stanford.edu/aa241/propulsion/largefan.html

Lectures notes
http://blackboard.tudelft.nl
Course AE3-021 Aircraft Design (2009-2010)

Files:
Lecture notes preliminary sizing of aircraft 2009.ppt
AE3-021 Propulsion Part-1.pdf
AE3-021 Propulsion Part-2.pdf
AE3-021 Propulsion Part-3.pdf

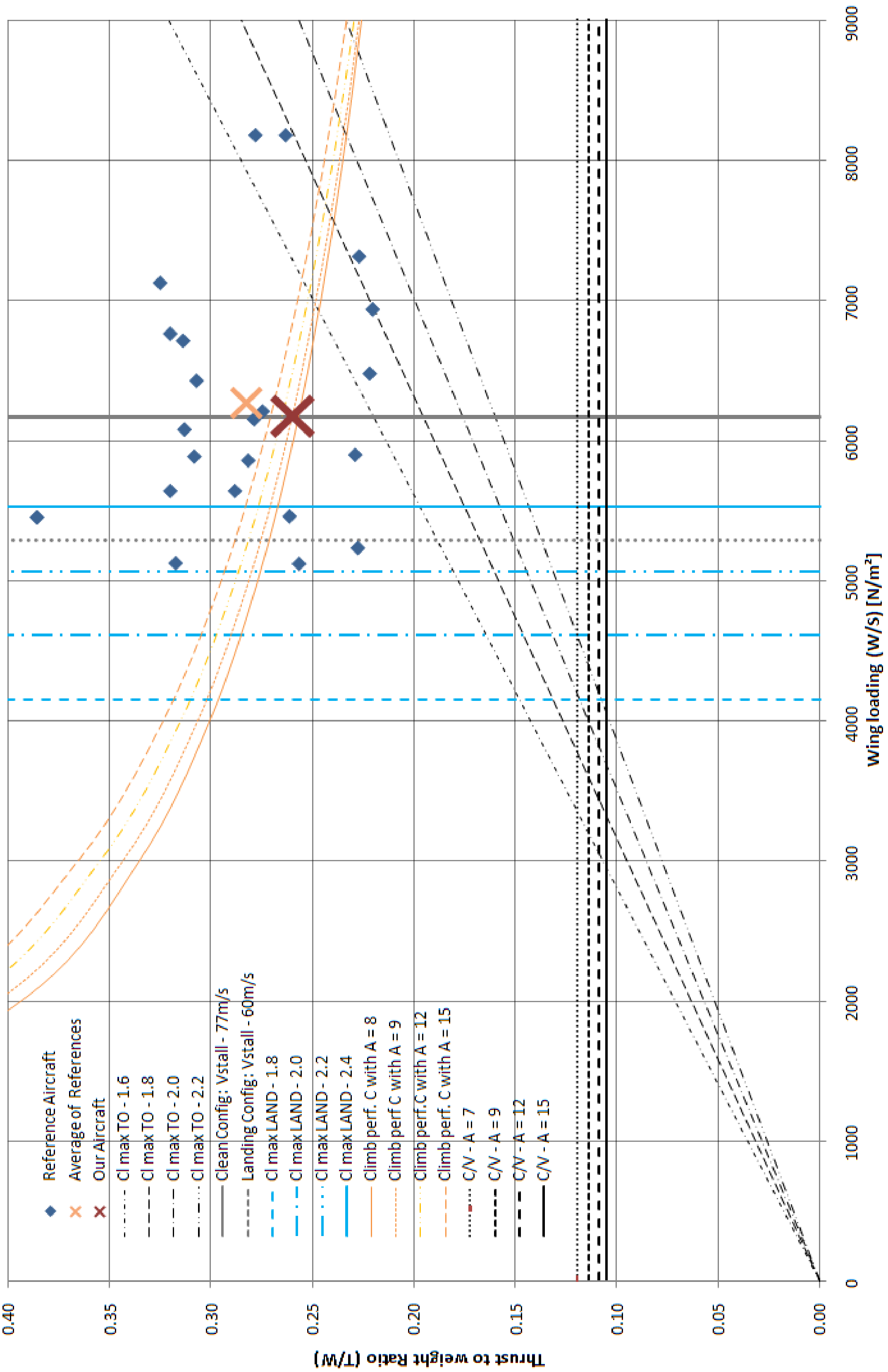
Books
Elements of Airplane Performance
G.J.J. Ruijgrok. Print 3

Introduction to flight,
Anderson, 5th edition

Aircraft Design: Synthesis and Analysis
Desktop aeronautics, Version 0.99

Appendices

Appendix A: T/W – W/S graph.



Appendix B: Table of reference aircraft and their characteristics

		Length [m]	engines	Cockpit crew	nr passenger (1-class)	Range (at MTOW) [km]	range (km)
airbus	A320-200	44,51	2	2	180		5600
Airbus	A321-200				220		5600
Boeing	737-800	39,5			189		5670
Boeing	707-320B	46,61	4	3	202	10650	6920
Boeing	727-200	46,7	3	3	189		4450
Airbus	A350-800	60,7			270		15400
Douglas	DC-8-63CF	57,1	4	3	180		3445
Ilyushin	Il-62	53,12	4	5	186		9,2
Shanghai	Y-10	42,93	4	5	178		8300
Tupolev	Tu-114 Rossiya	54,1	4	5	220		6200
Tupolev	Tu-154	48	3		180	2500	
Boeing	757-200	47,32	2	2	234	7,222	
Boeing	747-400	70,7	2	4	450	12491	
McDonnell Douglas DC-9	DC-9-50	49,7			135		3030
Boeing	737-400	36,5	2	2	168	4204	
Yakovlev	Yak-42D	36,38	3		120		41000
Lockheed	L-1011-1	54,2	3		253	7420	
Airbus	A300B4	54,08	2	3	266	6670	
Airbus Beluga	A300-600ST	56,15		2	47000t	2779	
McDonnell Douglas	MD-81	45,1			172	2910	

		Cruise altitude [m]	Service ceiling [m]	Cruise speed [Mach]	max cruise speed [Mach]	Take off distance [m]	Landing distance MLW [m]	Empty weight [kg]
airbus	A320-200	12000	12000	0,82		2180		48200
Airbus	A321-200	1200		0,82		2180		
Boeing	737-800	12500		0.785	2525			41413
Boeing	707-320B			972 km/h		3280	1813	66406
Boeing	727-200			.81		1768	1585	

Airbus	A350-800	13100		0.85				
Douglas	DC-8-63CF							6636
Ilyushin	Il-62		12000					67500
Shanghai	Y-10		12000					58120
Tupolev	Tu-114 Rossiya		12000	770 km/h				93000
Tupolev	Tu-154		12100					50700
Boeing	757-200		12800	0.80				
McDonnell Douglas	DC-9-50			898 km/h				
Boeing	737-400		11277,73	0.74	0.82			33200
Yakovlev	Yak-42D		8800		810km/h			
Lockheed	L-1011-1		11000	0.86	0.86		101867	101867
Airbus	A300B4			0.78	0.82			90060
Airbus Beluga	A300-600ST							86000
McDonnell Douglas	MD-81			.76				

		MTOW [kg]	Maximum speed	Wingspan [m]	Wing Area $[m^2]$	Wing aspect ratio	Wing Sweepback [degrees]	Height [m]
airbus	A320-200	93500		34,1	122,6		25	11,76
Airbus	A321-200							
Boeing	737-800	79010	Mach 0.82	35,7			25,02	
Boeing	707-320B	151320		44,42				
Boeing	727-200		.90 Mach	32,9				
Airbus	A350-800			64,8	443		31,9	17,1
Douglas	DC-8-63CF	161000	959 km/h	45,24	271,9			13,11
Ilyushin	Il-62	165000	900 km/h	43,2	279,5			12,35
Shanghai	Y-10	110227	639 km/h	42,24	244,5			13,42
Tupolev	Tu-114 Rossiya	175000	870 km/h	51,1	311,1			15,44
Tupolev	Tu-154	99000	950 km/h	37,55	201,5			11,4
Boeing	757-200	115680		38,05	181,25	7,8	25	
Boeing	747-400	401300						
McDonnell Douglas	DC-9-50	54900		28,47				
Boeing	737-400	68050		28,9			25	11,1
Yakovlev	Yak-42D			34,88	150			9,83
Lockheed	L-1011-1	200000		47,3	321,1			16,9
Airbus	A300B4	165000		44,85	260			16,62
Airbus Beluga	A300-600ST	155000		44,84	122,4			17,24
McDonnell Douglas	MD-81			32,8				9,05

		Maximum landing weight [kg]	Maximum zero-fuel weight [kg]	Takeoff run at MTOW [m]	Fuselage Width [m]	Fuselage Height	Cabin width [m]	Cabin Height [m]	Cabin length [m]	Tail Height
airbus	A320-200				3,95		3,7			
Airbus	A321-200									
Boeing	737-800			2525	3,8	4.0 m	3,5	2,2		
Boeing	707-320B				3,76					
Boeing	727-200									
Airbus	A350-800				5,96	6.09 m	5,59			
Douglas	DC-8-63CF				3,73					
Boeing	757-200			2911			3,54		36,09	
Boeing	747-400									
McDonnell Douglas DC-9	DC-9-50									8.38 m
Boeing	737-400	56200	53100	2540	3,76	4.11 m	3,54	2,2		
Lockheed	L-1011-1	167000					5,7			
Airbus	A300B4						5,28			

Appendix C: Current Aircraft Parameters Table

Symbol	Parameters	Value	Unit
	Cabin characteristics		
	Cabin length	33	m
	Cabin length	25.34	m
	Maximum diameter	4.1	m
	Maximum cabin height	2.53	m
	Maximum width	4,1	m
	Aisle width	0.66	m
	Aisle height	2.35	m
	Wall thickness	0.08	m
	Chair width	0.48	m
	Chair Pitch	0.83	m
	Cockpit Characteristics		
	Overnose angle	15	Degrees
	Overside angle	35	Degrees
	Grazing angle	32	Degrees
	Upward angle	20	Degrees
	Divergence angle	19	Degrees
	Flight deck length	2.5	m
	Fuselage Characteristics		
L	Total length	42.5	m
	Fineness	10.73	-
	Nose fineness	1.92	-
	After body fineness	2.56	-
	Tail length	10,66	m
	Aircraft Wing geometry		
b	Wing span	37.7	m
S	Wing area	157	m^2
A	Aspect ratio	9	-
Λ	Wing sweep angle	-	
	Weights and loadings		
	Fuel weight	29 918	kg
	Payload weight	17 010	kg
W_E	Empty weight	52 408	kg
W_{MTO}	Maximum take-off weight	100 000	kg
W/S	(maximum) Wing loading	6174	N/m^2
T/W	Thrust to weight ratio	0.26	-
	Flight parameters		

h_{cruise}	Cruise altitude	11800	m
V_{cruise}	Cruise speed	0.82	Mach
$C_{Lcruise}$	Cruise lift coefficient	1.7	-
C_{Lmax}	Maximum lift coefficient (take-off)	2.4	-
s_{TO}	Take-off distance	2100	m
s_{L}	Landing distance	1650	m
	Range	5500	km