Aircraft Design –

Assignment [52, Assignment 4: Wing Design]

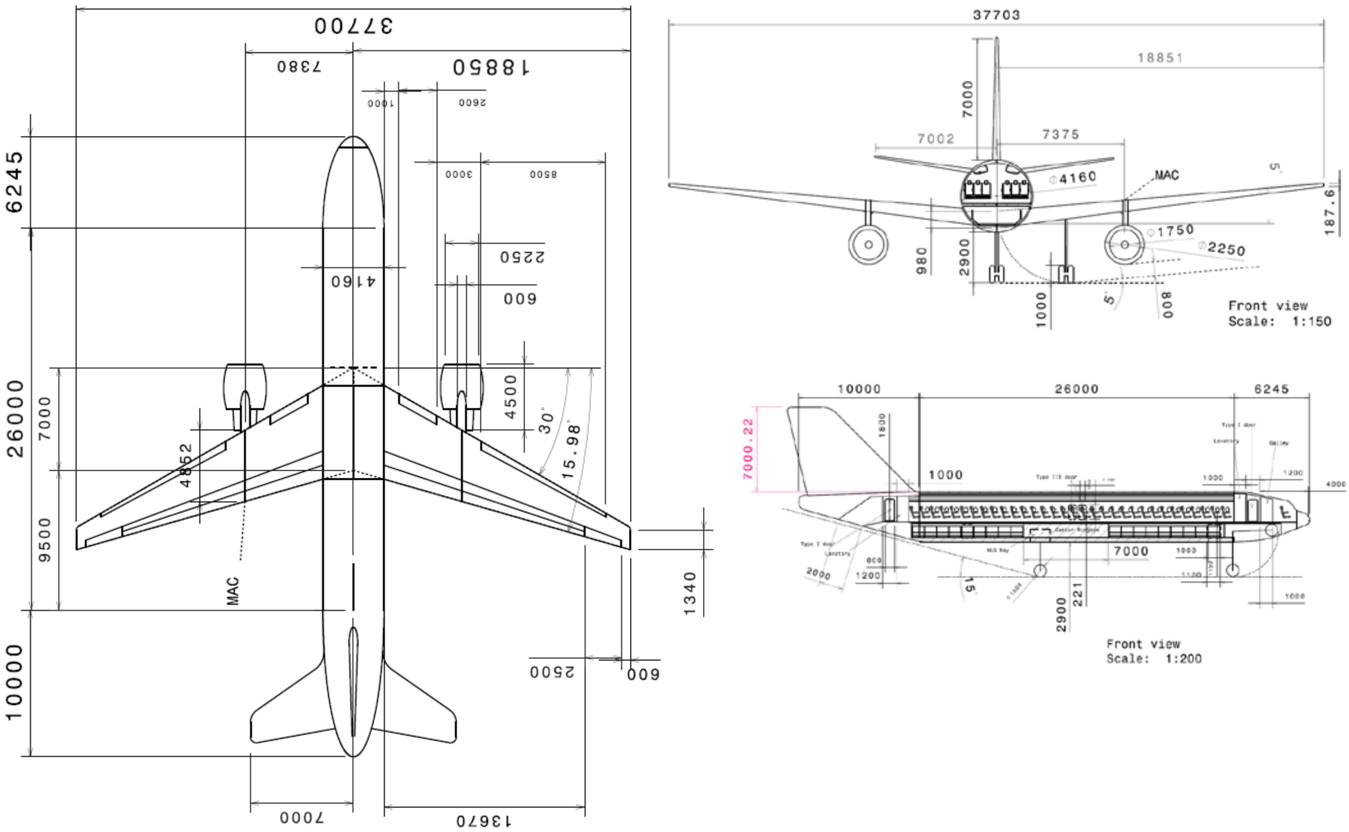
Student Names and Study Numbers:

Hours spent on assignment:

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



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1. Introduction

In report, the wing of the aircraft with specifications given in table 1.1 will be designed. For this, several steps are necessary. The first chapter (chapter 2) will go through the calculations that are needed to estimate the wing size and properties. Next, in chapter 3, the designed wing will be checked for take-off and landing conditions and the required flaps will be chosen and sized. Finally, chapter 4 will check the fuel storage capacity of the wing structure, and this will be compared to the required mission fuel.

2. Clean wing design

To determine the wing, first the clean wing is designed without any concerns for fuel, gear or flaps. First the wing will be regarded as infinite to define the correct airfoil. Then that airfoil will be transferred to a 3D wing design.

2.1 The infinite wing

First, the total required lift is calculated. This is done by multiplying the weight of the aircraft by 1.1, in order to account for gusts and maneuvering loads. The total required lift is then:

$$1.1\ W_{TO} = 1.1\ L_{tot} = 1.1*975083N = 1072591N \qquad \textit{Eq. 2.1}$$

So the whole wing needs to generate $1.07*10^6$N of lift in nominal cruise conditions. Looking back to assignment 2, a wing loading of 6174N/m^2 was found at take-off. Assuming this remains the same up until the beginning of cruise gives us the start cruise wing loading. From assignment 2 we can also calculate the wing loading at the end of cruise (using the fact that the aircraft uses 78% of its fuel during cruise.) this gives a wing loading of 4719N/m^2. The calculations for this are discussed in assignment 2, and are therefore not repeated. From equation 1.1, we can say, with the definition of lift that the lift coefficient is equal to:

$$L = C_{L\,des}\frac{\rho_{cruise}V_{cruise}^2}{2}\cdot S \quad \text{or} \quad C_{L\,des} = \frac{\rho_{cruise}V_{cruise}^2}{2}\cdot\frac{S}{W} \qquad \textit{Eq. 2.2}$$

The V_{Cruies} and the ϱ_{cruise} are determined from cruising conditions, and are set for the cruise condition. The W/S is the average weight during cruise, and the total CLdes should be multiplied by 1.1, for the same reasons the lift was multiplied in equation 2.1. This gives equation 2.3.

$$C_{L\,des} = 1.1\frac{2}{\rho V_{cruise}^2}\left\{\frac{1}{2}\left[\left(\frac{W}{S}\right)_{start\_cruise} + \left(\frac{W}{S}\right)_{end\_cruise}\right]\right\} \qquad \textit{Eq. 2.3}$$

Filling in with the known values, an average design lift coefficient for the whole wing of 0.284 is found. Using the reference aircraft, an average quarter chord sweep angle of the wing is found at 27.66 degrees. Evaluating for the start and end conditions separately, gives a $C_{L,crStart}$ = 0.317, and a $C_{L,crEnd}$ = 0.251. For our aircraft, a leading edge sweep angle of 30 degrees is used for now. Table 2.1 summarizes the important parameters to determine the design lift coefficient.

Parameter	Value	Unit
W_{start_cruise}	931886	N
W_{end_cruise}	739019	N
S	158	m^2
ϱ @ sea level	1.225	kg/m^3
ϱ @ average altitude 5900 m	0,667	kg/m^3
V_∞ (=V_{cruise})	247,5	m/s

Table 1.1: Parameters for determining the design lift coefficient

Another property of the wing is the thickness to chord ratio. Using a mach number of 0.82, and using fig. 2.1, a thickness to chord ratio of 0.14 is found.

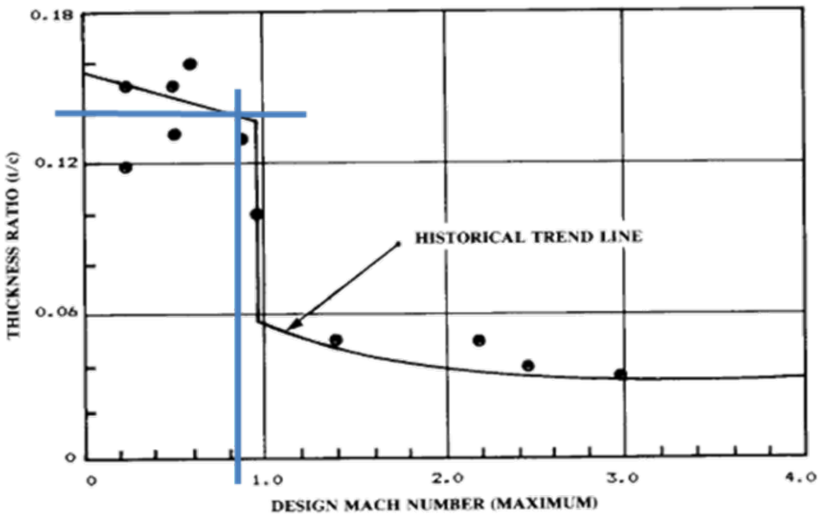


Figure 2.1: Design Mach number versus thickness ratio of historical aircraft

Using previously found data, and equation 2.4, the lift coefficient of the airfoil of the wing can be found to be 0.379.

$$C_{ldes} = \frac{C_{Ldes}}{\cos^2(\lambda)} = \frac{0.284}{\cos^2(30^\circ)} = 0.379 \qquad \text{Eq. 2.4}$$

So, when determining the airfoil, a design lift coefficient should be close to 0.4. The important design parameters are summarized in table 2.1.

C_{ldes}	0.4
Sweep angle λ (LE)	30^0
T/C	0.14

Table 2.2: Important wing design parameters

2.2 NACA airfoil Selection

Using DesignFOIL and JAVAFOIL, the airfoil of the wing was selected. A supercritical airfoil is almost imminent, since the speed of the aircraft is mach 0.82, and into the transonic region. So a NACA 6X- Series is chosen. Since the C_{ldes} is 0.4, the first digit of the foil is 4. Then, ideally, using a t/c of 0.14, the name of the airfoil is NACA 6X-414.

First the speed acting on the airfoil is calculated. Since the swept wing is at a speed of mach 0.82, the unswept airfoil travels at a mach-number of 0.82*COS(30) = 0.710. To aid in the choosing process, the Reynolds number is calculated using formula 2.5. All conditions are taken at cruise level, and at low speed and low altitude level. The chord that is used is the Mean aerodynamic chord. This is calculated in section 2.3.

$$Re = \frac{\rho\ V\ C}{\mu} \qquad \text{Eq. 2.5}$$

This, for cruise conditions, yields a Reynolds number of 26.8 million.

$$Re = \frac{\rho_{cruise} V_{cruise} C_{mac}}{\mu_{cruise}} = \frac{0.3212 \cdot 247.5 \cdot 4.818}{0,00001425} = 26.88 \cdot 10^{6}$$

For low speed and low altitude this gives 12.6 million.

$$Re = \frac{\rho_{low\_speed} V_{low\_speed} X_{mac}}{\mu_{low\_speed}} = \frac{0.6676 \cdot 66.76 \cdot 4.818}{1.61 \cdot 10^{-5}} = 12.63 \cdot 10^{6}$$

After some optimization was performed on the cruise performance, the airfoil NACA 64-414 was chosen. The 4- digit indicates that the maximum thickness is at 40 percent of the chord. This was chosen to optimize structural stiffness and effectiveness. The 4, as mentioned, stands for the design lift coefficient, being 0.379.

Chord location of the minimum pressure	40%
Design lift coefficient	0.4-
Maximum thickness	14%
Mean line parameter	1-

Table 2.3: Important wing design parameters for the NACA 64-414:

The flow-field of this airfoil at approach speed (mach 0.2) is shown in figure 2.2. The drag polar and lift curve at low speed are shown in figure 2.3. In figure 2.4, a zoom in is done of the lift curve, to indicate some typical values. Those typical values, as derived from the graphs are summarized in table 2.2.

Symbol	Info	Value	Unit
Alpha0L	Zero lift angle of attack	-3	deg
		=-0,03142	rad
dCl/dalpha	Lift curve slope	0,10	1/deg
		=5,7295	1/rad
cd0	minimum drag coefficient	0,008	-
AlphaStall	Stall angle of attack	12.5	deg
		=0,2182	rad
Clmax	Maximum lift coefficient	1,65	-
alhpa0	zero angle of attack lift coefficient	0,43	-
T/c	Thickness to chord ratio	0,14	-
Mcr	Critical Mach number at 0^{o} Angle of attack	0,705	-

Table 2.4: Important parameters of the NACA 64-414 Airfoil.

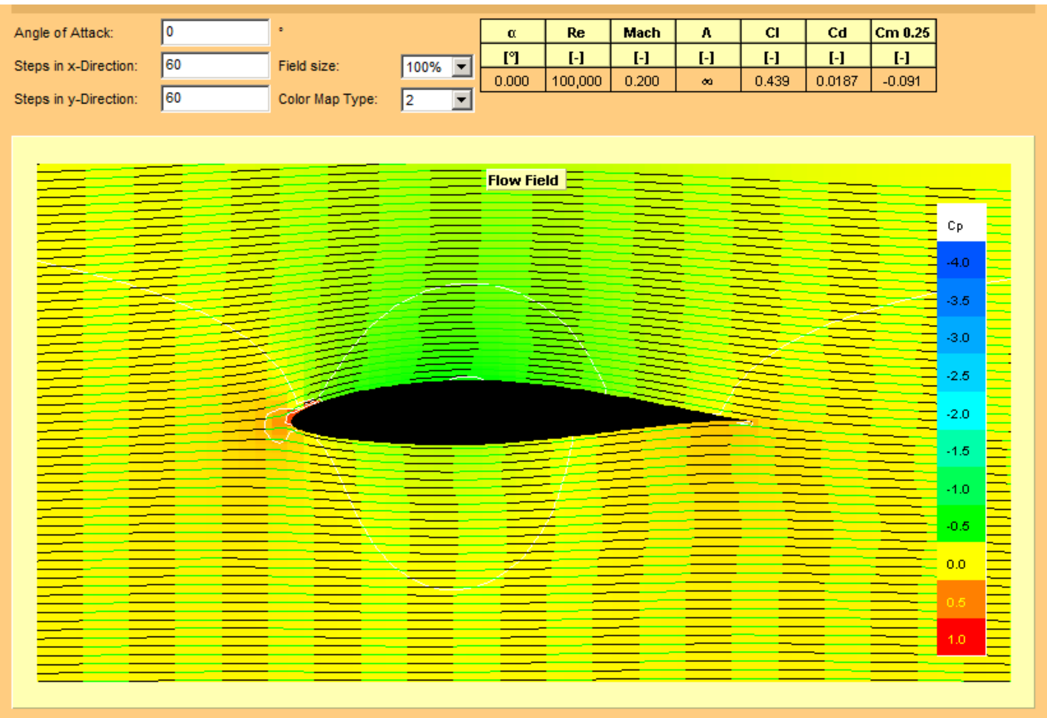


Figure 2.2: The flow-field around the NACA 64-414 airfoil in JavaFoil.

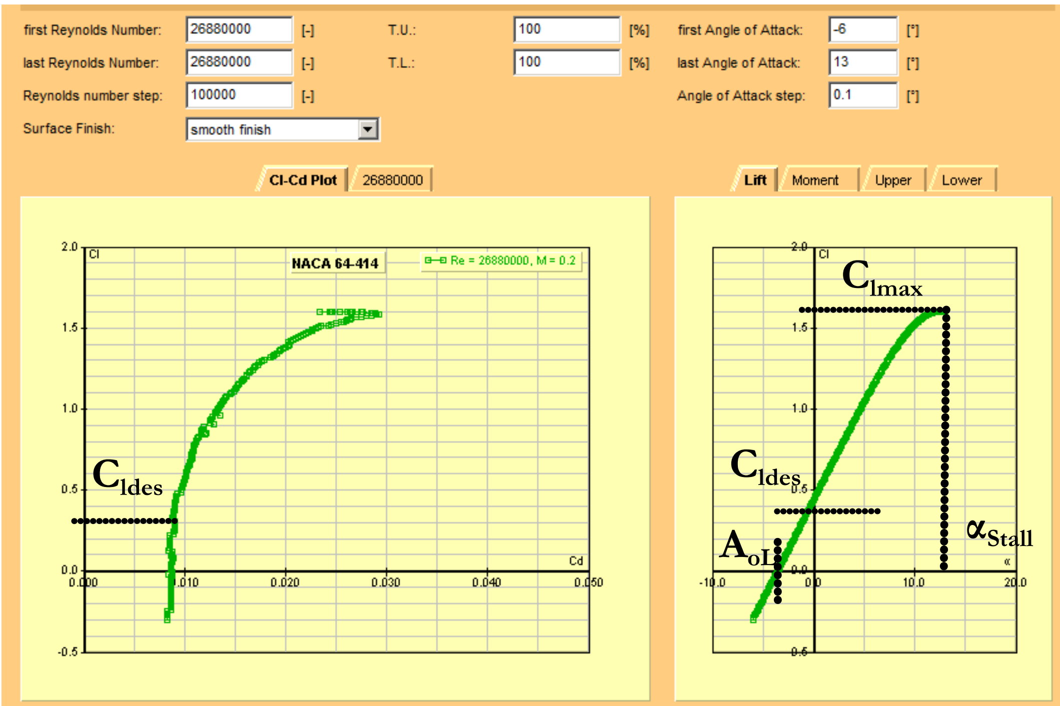


Figure 2.3: The NACA 64-414 airfoil polars at high speed in JavaFoil.

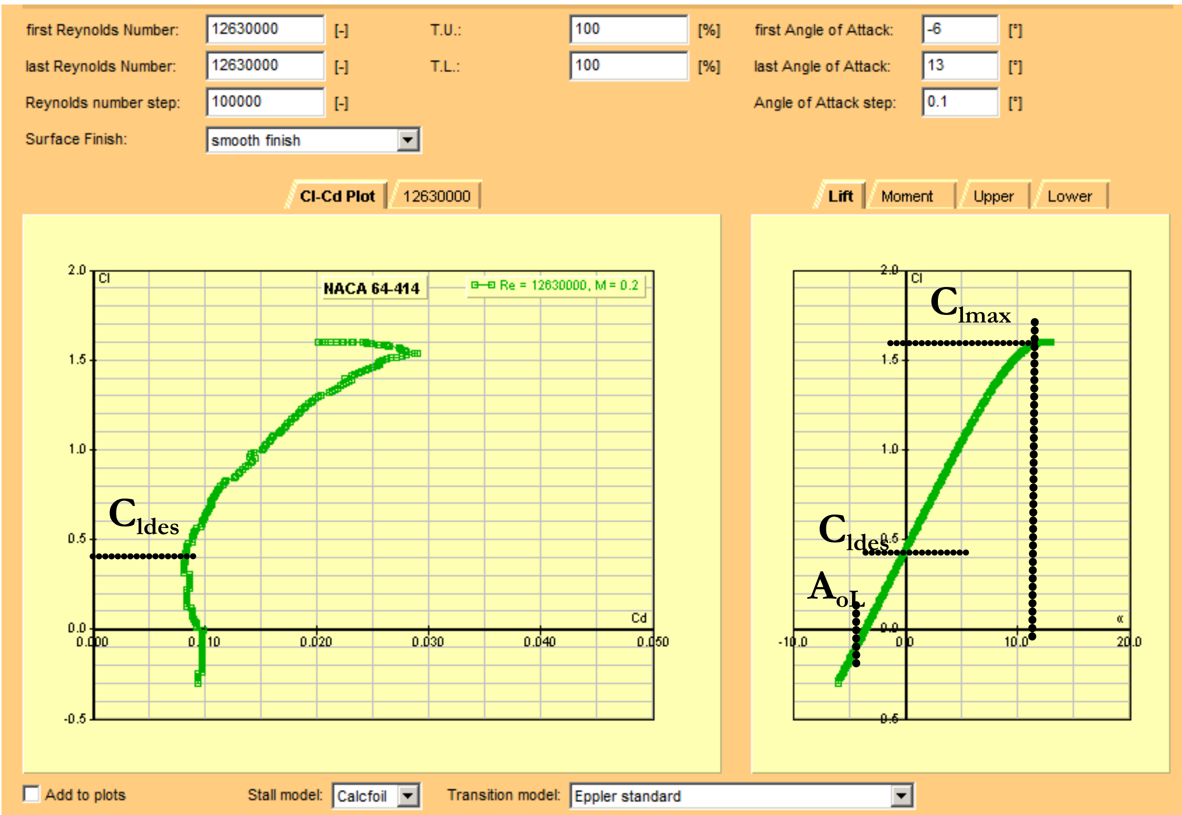


Figure 2.4: The NACA 64-414 airfoil polars at low speed in JavaFoil.

2.3 The finite wing and mean aerodynamic chord

Now the transition to the 3d wing is done. From assignment 2, a wing surface of 158m² was found, together with an aspect ratio of 9. Using formula 2.5, a wingspan of 37.70m is found.

$$A = \frac{b^2}{S} \quad \text{or} \quad \sqrt{AS} = b = \sqrt{158 \cdot 9} = 37.7\text{m} \qquad \text{Eq. 2.6}$$

Or the length of one wing is 18.85m. In the drawings from the designing of the aircraft, 7m of space was left to put the wing. The root chord of the wing will then also be chosen as 7m. Using equation 2.6, the taper ratio of the wing can be determined. From this, the tip chord c_T can be determined.

$$A = \frac{2b}{c_r(1+\lambda)} \quad \text{or} \quad \lambda = \frac{2b}{c_r A} - 1 = 0.197 \qquad \text{Eq. 2.7}$$

$$\lambda = \frac{c_t}{c_r} \quad \text{or} \quad c_t = c_r \cdot \lambda = 1.38\text{m} \qquad \text{Eq. 2.8}$$

This gives a tip chord of 1.378m. The mean span of one wing can be calculated using Eq. 2.8. the Mean aerodynamic Chord is then found using equation 2.9.

$$\overline{Y} = \frac{b}{6}\frac{1+2\lambda}{1+\lambda} = 7.3\text{m} \qquad \text{Eq. 2.9}$$

$$M.A.C = \bar{C} = \frac{2C_r}{3}\frac{(1+\lambda+\lambda^2)}{(1+\lambda)} = 4.8m \qquad \textit{Eq. 2.10}$$

This implies a quarter mean aerodynamic chord of 1.204m. This is the location where the lift force-resultant acts on the wing. This is also the ideal location, in terms of bending loads and wing bending relief, to hang the engine pod. In the drawings in the appendix, the new location of the engines can be seen. They are suspended exactly below the Mean aerodynamic chord, at a distance of 7.318m from the centre of the fuselage.

2.4 Validity and sanity checks

The following subsections will perform some validity and sanity checks, to make sure none of the values are out of bounds in comparison with other aircraft.

2.4.1 Pitch-up Tendency

The pitch-up tendency can be checked using figure 2.2. The quarter chord sweep angle can be calculated from equation 2.X, where x/c is 0.25, the root chord is 7m, the taper ratio is 0.197 and the span is 37.7m. The leading edge sweep angle is 30 degrees. This gives a quarter chord sweep angle of 26.7 degrees.

$$\Lambda_{x/c} = a\tan\left[\tan\Lambda_{LE} - \frac{x}{c}\frac{2C_r}{b}(1-\lambda)\right] = 26.7^{o} \qquad \textit{Eq. 2.11}$$

From this figure it is clear that the aircraft has a high pitch-up tendency. However, this is without the contribution of the tail. It is assumed that the pitch-up tendency will be lowered due to the contribution of the tail, when the tail is designed. Therefore, this graph is not really applicable to the situation, but it does give an idea about what the wing alone will do when flown.

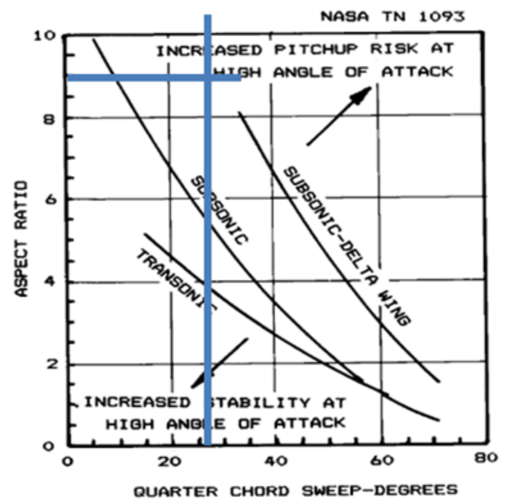


Figure 2.5: Historic data of aircraft in relation to pitch-up tendency.

2.4.2 Sweep angle

Figure 2.3 shows the leading edge sweep angle versus the Mach number. If the sweep angle that was chosen earlier at 30 degrees is checked versus the maximum Mach number, being 0.82, the graph shows a point on the historical trend line. So the values chosen are valid.

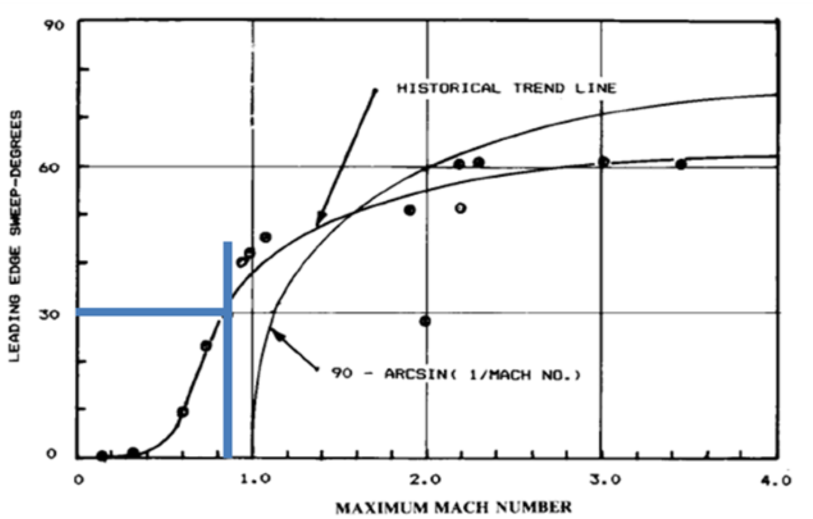


Figure 2.6: Historic data of aircraft which shows Mach number in relation to sweep angle.

2.4.3 Aspect Ratio

To check whether the wing is viable for construction, the aspect ratio is checked using the wing cantilever ratio. By definition this calculated as follows.

$$R_c = \frac{b/2}{t_r \cos \Lambda_{.50c}} \quad \text{Eq. 2.12}$$

The span b is known, the thickness at the root t_r can be calculated using the root chord and the thickness to chord ratio. This is found to be 0.98m. The sweep angle at a half chord length can be calculated using equation 2.13, using an x/c of 0.5.

$$\Lambda_{x/c} = a\tan\left[\tan \Lambda_{LE} - \frac{x}{c}\frac{2C_r}{b}(1-\lambda)\right] \quad \text{Eq. 2.13}$$

This gives a Cantilever Ratio R_c of 20.9. To be a valid number, this should be between 18 and 22. As calculated, this value fits perfectly within this interval.

2.4.4 Taper Ratio

Another test is the taper ratio test. Checking our taper ratio of 0.197 and a quarter chord sweep angle of 26.7 degrees shows right on the trend line of the graph.

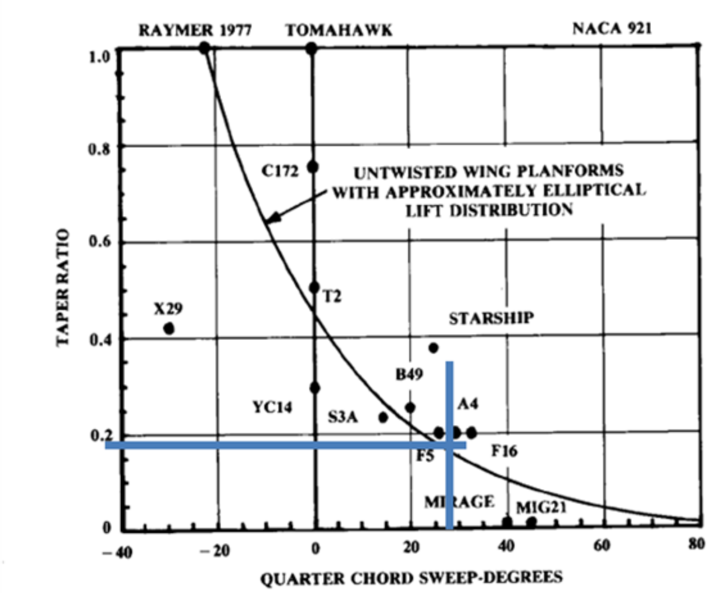


Figure 2.7: Historic data of aircraft which shows sweep in relation with taper ratio.

Note that the taper ratio should, preferably be larger than 0.2. A value of 0.197 is still within limits and requires no special approach.

2.4.5 Drag divergence

To check if the drag is not large, and below the drag divergence, the mach of the drag divergence is calculated. This can be done by using eq 2.14. to fill in the formula, the previously calculated value of the quarter chord sweep angle can be used, together with the critical mach number found from the airfoil choice.

$$M_{DD} = M_{cr}\left[1.02 + 0.08(1-\cos\Lambda_{0.25c})\right] = 0.736 \qquad \text{Eq. 2.14}$$

This results in a drag divergence mach of 0.736, which is 2.9 percent larger than the critical Mach number. This is enough clearance to ensure a low-drag flight.

2.4.6 Some other wing parameters

The undetermined parameters of the wing are the incidence angle (the angle between the chord line and the horizontal), the dihedral angle (the angle between the quarter chord points of the wing and the horizontal) and the twist angle (the difference between the incidence angle at the tip and the incidence angle at the root).

For simplicity, the twist angle will be set at 0 degrees, since its exact value requires several aerodynamical computations before it can be optimally determined, and this requires a wind tunnel or aerodynamical finite elements test.

The incidence angle can best be determined from the ideal angle of attack in combination with the airfoil. From our airfoil, since it is designed for a lift coefficient of 0.2, the actual value for lift coefficient at zero angle of attack is 0.378. This is close to the 0.4 value, so no changes will be made, and the incidence angle will be set at 0^0.

The dihedral angle is set at 5 degrees. This value, in combination with the landing gear size and engine nacelle size, makes for enough clearance with the ground. Reference is made to the drawings in appendix 2.

2.5 The clean wing lift curve

There is now almost enough information to determine the lift curve for the aircraft. Some angles will still need to be determined, such as the slope angle, the trim angle and the stall angle. When these angles are determined, the aircrafts wing lift curve can be determined.

2.5.1 Curve slope angle

The curve slope angle will be determined both at cruise and on approach at low speed (60m/s). First the Prandtl-Glauert compressibility factor is calculated using equation 2.15. The M_{eff} used here is the mach that is on the airfoil: the infinite Mach number, multiplied by the cosine of the leading edge sweep angle. For the sweep angle, 30 degrees is used. For the cruise speed, Mach 0.82 and for the landing speed Mach 0.15, as calculated from the approach-stall speed in assignment 2.

$$\beta_{cruise} = \sqrt{1 - M_{eff}^2} = \sqrt{1 - 0.71^2} = 0.70$$
$$\beta_{landing} = \sqrt{1 - M_{eff}^2} = \sqrt{1 - 0.15^2} = 0.99$$

Eq. 2.15

Now, the dCL/dα can be found using equation 2.16, where A is the aspect ratio (A=9), η is the wing efficiency factor (set at 0.95) and $\lambda_{0.5c}$ is the half chord sweep angle, calculated with equation 2.13. ($\lambda_{0.5c}$=0.4 rad).

$$\frac{dC_L}{d\alpha} = C_{L\alpha} = \frac{2\pi A}{2 + \sqrt{4 + (\frac{A\beta}{\eta})^2 (1 + \frac{\tan^2 \Lambda_{0.5C}}{\beta^2})}}$$

Eq. 2.16

The Lift curve slope for cruise then becomes dCL/dα = 5.622/rad = 0.098 /deg.
The Lift curve slope for approach then becomes dCL/dα = 4.561/rad = 0.080/deg, whilst 2pi, the lift curve slope for an infinite wing is 6.283/rad.

2.5.2 Trim angle

To determine the trim angle, the following formula is used:

$$\alpha_{trim} = \frac{C_{Ldes}}{C_{L\alpha}} + \alpha_{0L}$$

Eq. 2.17

$Alpha_{0L}$ is the angle determined in table 2.4, C_{Ldes} is the design lift coefficient of 0.378, determined in chapter 2.1 and C_{Lalpha} is the Curve slope, determined on cruise, from equation 2.16 (C_{LAlpha} = 5.622). The trim angle is then calculated as 0.015 rad, or 0.85 degrees. When checked with JavaFoil, this angle is not critical for the Mach divergences or critical Mach number. This means that no sonic conditions appear at this trim setting, so the wing is designed properly.

2.5.3 Stall angle

To determine the stall angle, first a check needs to be done on the taper ratio. From historical data, the aspect ratio should be larger than the value from the right hand side of equation 2.18.

$$A > \frac{4}{(C_1 + 1)\cos \Lambda_{LE}}$$

Eq. 2.18

The C_1 in the equation can be derived from figure 2.5 to be 0.4, using a taper ratio of 0.197.

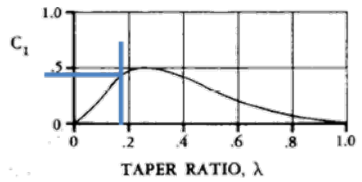


Figure 2.8: Historic data of aircraft which shows taper ratio in relation with the C_1 coefficient.

This gives a minimum aspect ratio of 3.3, and since the aircraft's aspect ratio was set at 9, no problems arise here, and the high aspect ratio method is used. First the sharpness of the leading edge of the wing has to be determined. This can be done by finding the difference between the vertical location of the 0.0015c point, and the 0.06c point of the top surface of the wing. From JavaFoil, this is found to be 3.2019. For low speed flight, the following figures then apply. Those figures can be used to find the C_{Lmax}/C_{lmax} and the $\delta\alpha C_{Lmax}$. As can be found from assignment 2, the approach Mach speed is 0.176, so close to the M=0.2, shown in these graph.

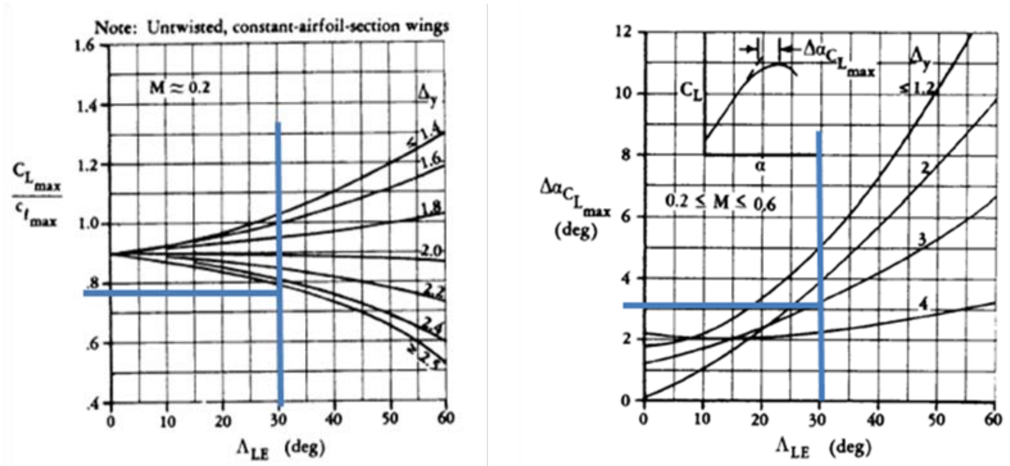


Figure 2.9: Determination of the C_{Lmax}/C_{lmax} and the $\delta\alpha C_{Lmax}$.

From these graphs, the C_{Lmax}/C_{lmax} = 0.78 and the $\delta\alpha C_{Lmax}$= 3 ^{0} = 0.052 rad. From equation 2.19, the C_{Lmax} can now be determined, using a δC_{Lmax} equal to 0, since this is at low speed.

$$C_{L\max} = \left[\frac{C_{L\max}}{C_{l\max}}\right] C_{l\max} + \Delta C_{L\max} = 1.32 \qquad \text{Eq. 2.19}$$

Now, using equation 2.20, the stall angle of attack can be calculated, since all values are known.

$$\alpha_s = \frac{C_{L\max}}{C_{L\alpha}} + \alpha_{0L} + \Delta\alpha_{C_{L\max}} = 0.289 \text{ rad} \quad = 16.58 \deg \qquad \text{Eq. 2.20}$$

$Alpha_{0L}$ is the same for the airfoil and for the wing, and C_{Lalpha} was determined in eq 2.16, for slow speed.

2.5.4 The graph

Summarizing all above values in equation 2.21 gives the CL-Alpha curve in figure 2.10.

$$C_L = C_{L\alpha}\alpha - C_{L\alpha}\alpha_{0L} \qquad \text{Eq. 2.21}$$

Note, the rounding near the stall angle and the maximum lift coefficient is estimated, and not calculated.

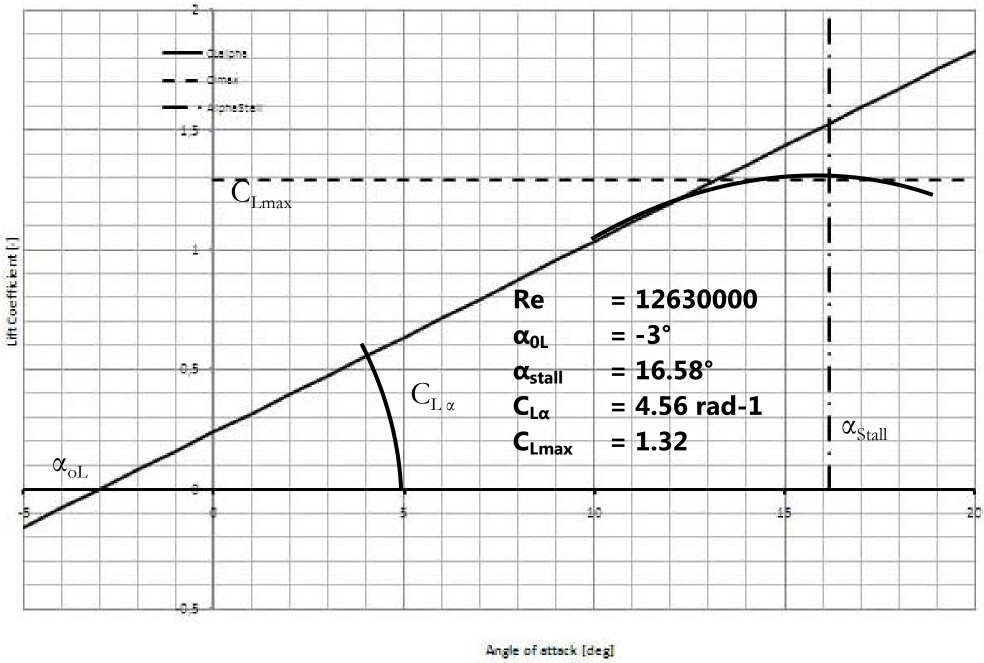


Figure 2.10: Determination of the C_{Lmax}/C_{lmax} and the $\delta\alpha C_{Lmax}$.

3. High Lift Devices

To determine whether the aircraft will perform well on approach and take off, those conditions will first be checked. This will be done in this chapter.

3.1 Landing and take-off conditions

In Chapter 2, the maximum lift coefficient of the clean wing was found to be 1.32, However when calculating the performance, a lift coefficient at landing of 2.4 was assumed. And for take-off, this needed to be 2.0. This means the wing lacks 1.08 in lift coefficient on landing, and 0.67 of lift coefficient on take-off. But another 10% needs to be added to this, to ensure a safe landing, away from the stall characteristics. This means that flaps need to be installed to increase the lift coefficient with 1.19 in take-off and 0.737 in landing conditions.

3.2 Selecting the appropriate high lift devices

For this design aircraft the high lift devices that will be installed on the aircraft are double slotted Fowler flaps at the trailing edge, TE, and slats at the leading edge, LE. Reference is made to table 3.1 and figure 3.1 for the properties of the chosen HL devices. Those devices are considered for their higher effectiveness in regard to the required $\Delta(C_L)_{\max} = 1.19$ to be generated. Countless other transport aircraft use the same setup of flaps and slats, the system is also easy to maintain.

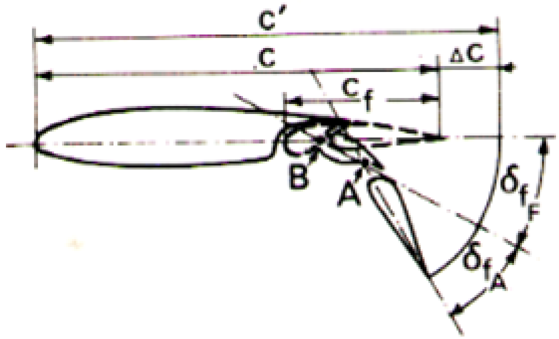


Figure 3.1: Chord length definitions

High lift devices	$\Delta(C_l)_{\max}$ when fully deployed	Typical angles [Deg]	Deflection	Position in chord length x/c	cf/c
		Landing	Take off		
TE devices (flaps)					
Double slotted	1.6 c'/c	50	20	0.7	0.30
LE devices Slats					
Slats	0,4	40	20	0.15	0.15

Table 3.1: High lift devices given/assumed properties

3.2.1 Sizing the trailing edge devices to meet the required maximum lift coefficient

Using the sizing method (shown in lecture notes), the relation between the amount of $\Delta(C_L)$ and a high lift device surface S_{wf}/S (ratio between the wing flapped area and the wing trapezoidal reference area), can be shown by the given formula.

$$\Delta C_{L\max} = 0.9\Delta C_{l\max}\frac{S_{wf}}{S}\cos(\Lambda_{hingeline})$$

or *Eq. 3.1*

$$\frac{\Delta C_{L\max}}{0.9\Delta C_{l\max}\cos(\Lambda_{hingeline})} = \frac{S_{wf}}{S}$$

Where $\Delta(C_l)_{\max}$ is given in table 3.1 for each chosen high lift devices as function of the c'/c value which is related to the assumed value of c_f/c, the ratio between the length of the flap chord and the airfoil chord.

By using the MAC of 4.818m as the reference chord length, the location of the trailing edge high lift devices can be defined. At the trailing edge, the flaps location should be such that the space left between the front and rear spars is between 40 and 50% of the wing chord. A value of 50 % is chosen here. Then leaving a margin of 5% between spars and the flaps positions, the assumed values, listed in table 3.2, can be used to calculate in the corresponding hinge angles using Eq. 3.2.

$$\tan\Lambda_{x/c} = \tan\Lambda_{LE} - \frac{x}{c}\frac{2C_r}{b}(1-\lambda)$$ *Eq. 3.2*

Fowler Flaps	x/c	Angle [deg]	hinge_line	Deflection angle δ [deg]	δc/cf	c'/c
TE Fowler	0.7	20.23		40	0.78	1.189

Table 3.2: high lift devices determined properties

Then using the given deflections angles at landing (see Table 1), the value of δc/cf can be determined using figure 3.2. This value can then be multiplied by Cf/C(=0.3) to find the DeltaC'/C. The C'/C is δC'/C+1 (Eq. 3.3). Those values are listed in table 3.2.

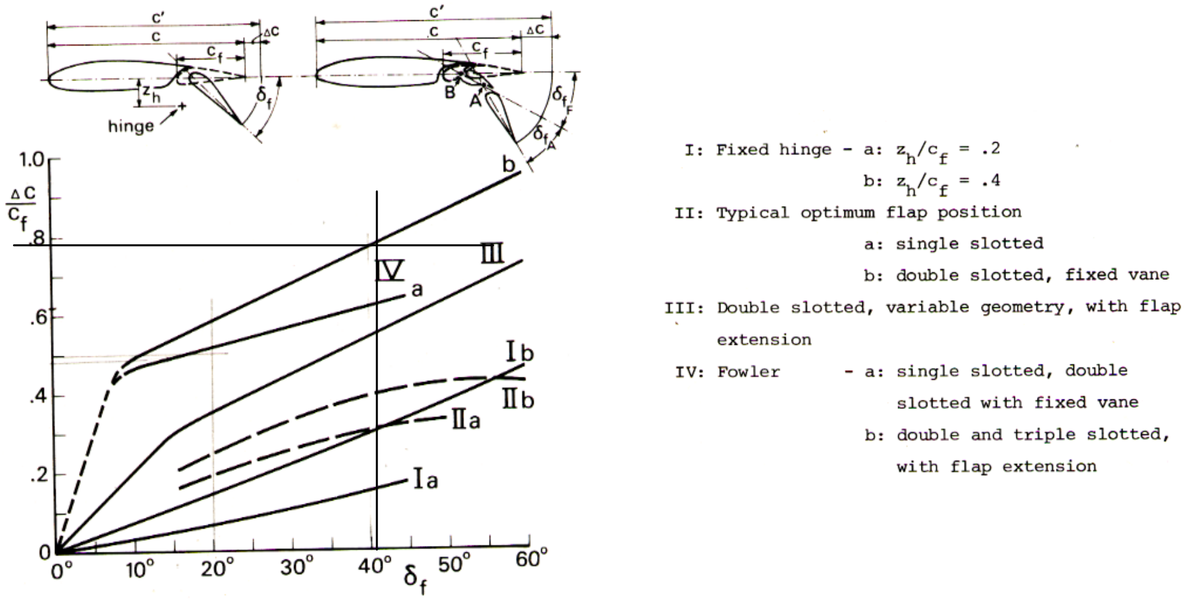


Figure 3.2 Chord extensions for different types of flaps.

$$\frac{c'}{c}=\left(1+\frac{\Delta c}{c}\right)=\left(1+\frac{\Delta c}{c_f}\frac{c_f}{c}\right) \qquad \text{Eq. 3.3}$$

Calculating the lift coefficient change with formula 3.3 and table 3.1 gives that

$$\Delta(C_l)_{\max}=1.6\frac{c'}{c}=1.6\left(1+\frac{\Delta c}{c_f}\frac{c_f}{c}\right)=1.6(1+0.78\cdot 0.3)=1.974 \qquad \text{Eq. 3.4}$$

$$\frac{\Delta C_{L\max}}{0.9\Delta C_{l\max}\cos(\Lambda_{hingeline})}=\frac{2.4-1.32}{0.9\cdot 1.97\cos(20.23)}=\frac{S_{wf}}{S}=0.639$$

Now all values are known to find the Swf/S. This is found to be 0.639. Or 64 percent of the wing needs to be flapped with a trailing edge flap. Calculating this in terms of length and width: the mean aerodynamic chord of the flap needs to be 1.45m and the width of the flaps in one wing is 11.90m. Subtracting the 2.05m of the fuselage section from the half wing span of 18.85m, gives a wing width of 16.8m. Now, removing the 11.90m of flaps leaves a width of 4.90m of room fill the ailerons in. Taking into account that the actual tip of the wing isn't used for ailerons, and that 0.65m is reserved for the wing tip structure, the aileron is sized to be 4.25m wide, and has a chord of 0.25 times the wing chord.

Now, the rear wing spar can be placed. Using the 0.25 chord of the trailing edge flaps, and the extra 10 percent needed to store systems, gives that the rear spar of the wing needs to be placed at 0.6c. The total landing $C_{L\max}$ is then 1.32+1.18 = 2.5. For take-off, assuming a lift coefficient increase of 0.8 times the landing increase of lift coefficient, this value is 1.32+0.8*1.28= 2.264.

3.2.2 Leading edge devices

Leading edge slats are added to the wing to give another extra lift security when landing. The lift coefficient is calculated in the same manner as for the trailing edge flaps. It is chosen that the front spar is at 0.2C, and then, removing 5% room for the flap devices, gives a total x/c of 0.15.

HL devices	**x/c**	**Angle hinge_line [deg]**	**Deflection angle δ [deg]**	**c'/c**
LE Slats	0.15	28.04	40	1.09

Table 3.3: high lift devices determined properties

Using the same equations as for the design of the trailing edge flaps, the leading edge slat contribution is found. One value is different though: the Swf/S is now known. It is assumed that most of the leading edge is covered in slats, as can be made up from the drawings in appendix 2. From the geometry, a slat span of 2 times 11.1m can be reserved from the wing. This results in an Swf/S of 0.67. Using equation 3.1 and evaluating the hinge line angle and using the data in table 3.3, an additional increase in $C_{L\max}$ of 0.4378 can be found, which becomes 0.8*0.4378 = 0.35 on take-off.

3.3 Evaluation of the CL-α curve for the flapped wing

In this section, a generation of the flapped CL-α curve will be approximated.

Due to the introduction of TE HL devices, the CL-α curve has changed. The max landing CL has increased with 1.18 to give a max CL of (= CL clean(1.32) + Delta CL needed(1.18)) 2.5 with slats and flaps. To represent this graphically, we need to generate the flapped CL-α curves for both for the take-off and landing conditions on the same graph where the clean wing CL-α curve, which was found in chapter 2, is plotted. Then the following values of C_{Lmax}, $C_{L\alpha}$, α_{stall} and α_{0L} for the flapped wing curve will be indicated with the contributions of both LE and TE HL devices separately, but on the same graph, to get an overview of the effect of the flaps.

3.3.1 Evaluation of $\alpha_{0L,land}$

First, the angle at the zero lift coefficient has to be determined for the flaps using the following relation:

$$\Delta\alpha_{0L_{flapped}} = (\Delta\alpha_{0L})_{airfoil}\frac{Swf}{S}\cos\Lambda_{hinge\_line} \quad \text{Eq. 3.5}$$

The $(\Delta\alpha_{0L})_{airfoil}$ can be assumed to be -10° at take-off and = - 15 deg at landing (from Lecture notes). With the known values of the total $\frac{Swf}{S}$ = 0.65 at landing and Λ_{hinge_line} = 20.23 deg both at TE , Delta alpha zero lift flapped , $\Delta\alpha_{0L}$, becomes -9.148 deg at landing and -6.09 at take off. Therefore the alpha zero lift flapped, $\alpha_{0Lflapped}$, is (alpha zero lift clean + Delta alpha zero lift flapped) = -3 + -9.148 =-12.148 deg at landing and -9.09 deg at take off.

3.3.2 Determination of the slope of the flapped curve

Since in this design case, the TE HL devices increase the wing surface, can be used to calculate C_{la} flapped, the slope of the linear part of the flapped curve

$$C_{L\alpha\,flapped} = \frac{S'}{S}C_{L\alpha-clean} \quad \text{Eq. 3.6}$$

Where S' is the increased wing surface by the extended flap (S'/S >1), defined as S'/S= 1+Swf/S(c'/c -1). Equation 3.6 then becomes:

$$C_{L\alpha\,flapped} = \left(1+\frac{Swf}{S}\frac{c'}{c}\right)C_{L\alpha-clean} \quad \text{Eq. 3.7}$$

Using the known value of the slope at clean wing of 0.080/deg or 4.561/rad, the flapped slope, C_{la} flapped, results in 0.081/deg or 0.21 /rad which is below 2π and $C_{L\alpha}$ also stays below 2π. Then the linear part of the flapped curve can be plotted, as function of alpha, using Equation 2.20.

$$C_L = C_{L\alpha}\alpha - C_{L\alpha}\alpha_{0L} \quad \text{Eq. 2.20}$$

The stall angle at landing and take-off can then be calculated using the following equation.

$$\alpha_{sflapped} = \frac{C_{L\max flapped}}{C_{L\alpha flapped}} + \alpha_{0Lflapped} + \Delta\alpha_{C_{L\max flapped}} \quad \text{Eq. 3.8}$$

With $C_{L\max flapped}$ is equal to $C_{LmaxFlaps}$ = 2.264

$C_{L\alpha flapped}$ = lift curve slope flapped, C_{la} flapped= 0.08/deg

$\alpha_{0Lflapped}$ = alpha zero lift flapped = -12.148 deg.

Then stall angle flapped, $\alpha_{sflapped}$, = 18.78 deg.

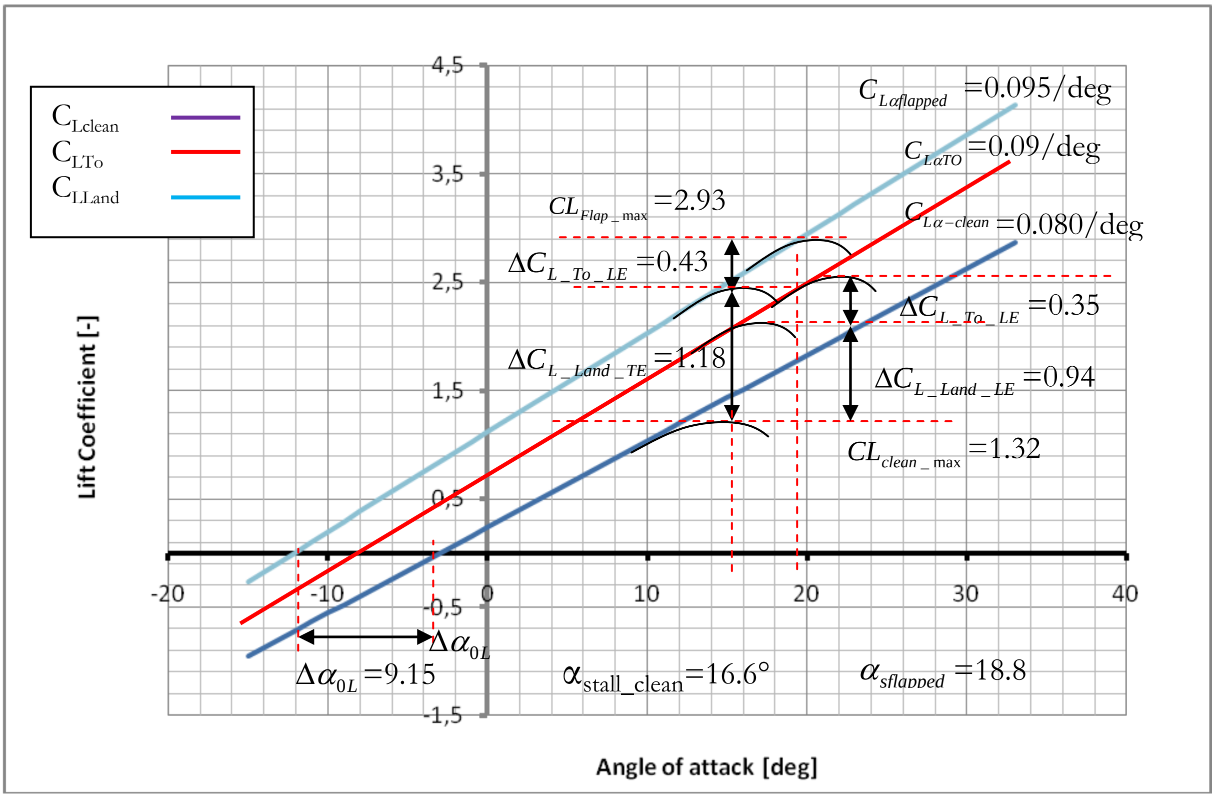
Finally, adding the contribution of the LE devices of delta C_{lmax} LE = 0.4378, see section 3.2.2, to the C_{Lmax} with Flaps of 2.5 gives a Final C_{Lmax} with Flaps that can be reached due to HL devices of 2.93 and then the α_{stall} becomes 18.8 deg. See CL-alpha graph below.

3.3.3 Flapped wing Cl-α Curve

<u>Important parameters:</u>

Condition	**CLEAN**	**Take Off**		**Land**	
		Flaps	**Flaps+slats**	**Flaps**	**Flaps+Slats**
C_{Lmax}	1.32	2.26	2.61	2.5	2.93
α_{stall}	16.58 deg		18.4 deg		18.78 deg
$C_{L\alpha}$	0.08/deg	0.09/deg		0.095/deg	
α_{0L}	-3 deg	-8.2 deg		-12.15 deg	

Note that an extra of 10 percent has been added to the initial required Cl's, to ensure sufficient margin of safety.



4. Fuel storage

Now the wings have been fully designed, it is time to check whether there is enough room to fit all the mission fuel in the wings. To store the fuel, 3 separate tanks will be used: one in the center between the wings, and one in each wing. The volume and capacity of those tanks will be calculated in this chapter, and a comparison will be made with the required fuel to fulfill the mission. If necessary other additional locations to store the fuel will be considered.

4.1 Inboard Center fuel tank

The center fuel-tank will be taken as having a box-shape. The dimensions of this box can be found easily by looking at the drawings in appendix 2. The maximum size of this tank, limited by the height of the cargo-space, the width of the fuselage and the chord of the wing, minus the location needed to store the landing gear.

The maximum height of this tank is 1.1m, the maximum width of the tank is 4m and the maximum length is 3.5m. leaving 3.5m chord for the storage of the gear. The volume of this tank then becomes 15.4m³. The density of Kerosene at sea level and normal pressure is 819kg/m³. The tank can then hold a maximum of 12613 kg. However, some space in the tank is lost, due to cabling, supports or systems. This renders 9% of the space unusable. The total tank capacity can thus only be 11477kg

4.2 Wing fuel tank

For the volume of the wing tank, it is assumed that the fuel can only be stored between the front and rear spar of the wing. The location of those spars was determined in chapter 3. The shape of the tank can then be estimated with the following volumetric shape:

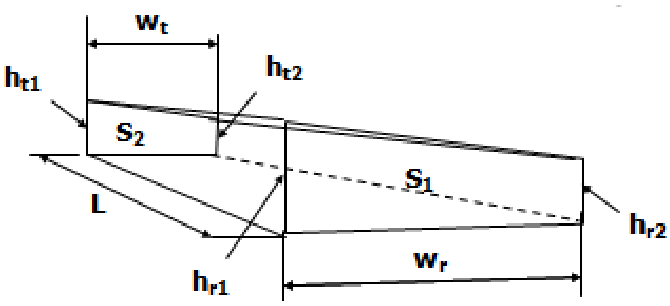


Figure 4.1: Representative volume of the fuel-tank in the wings

The volume of the tank is then determined by equation 4.1, 4.2 and 4.3.

$$V = \frac{L}{3}\left(S_1 + S_2 + \sqrt{S_1 S_2}\right)$$ Eq. 2.19

Where S1 and S2 are determined by:

$$S_1 = \frac{1}{2}\left(h_{r_1} + h_{r_2}\right)w_r$$ Eq. 2.19

$$S_2 = \frac{1}{2}\left(h_{t_1} + h_{t_2}\right)w_t$$ Eq. 2.19

From the geometry of the wing, the following values can be concluded and used:

Wr	2.45m
Hr1	0.848m
Hr2	0.64m
Wt	0.9m
Ht1	0.109m
Ht2	0.082m
L	15m

Table 4.1: Important fuel tank parameters

Filling in these values gives a total wing tank volume of 14.39 m^3. Also accounting for the 9% of capacity losses due to structures and systems, gives a total capacity of 10725kg per wing.

4.3 Total fuel storage capacity

To find the total fuel storage capacity, the capacity of the tanks is simply added to one another. From the first tank we had 11477kg, and from the two wing-tanks the capacity was 2*10725kg. This gives a total fuel storage capacity of 32928 kg. In assignment 2, the mission fuel was estimated at 29918kg, comparing this with the volumetric capacity of the wings it is clear that the tanks can actually be designed smaller, or that an extended range can be flown if the payload is lowered. Since most inaccuracies are accounted for, the tanks will be kept at their current size.

Since the capacity is large enough, the wing design can be concluded.

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http://www.mh-aerotools.de/airfoils/javafoil.htm

General information on aircraft design & Information on existing aircraft.
http://www.janes.com

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

Files:
WingDesign_Part_I_2009-10.ppt
WingDesign_Part_II_2009-10.ppt
WingDesign_Part_III_2009-10.ppt

Books
Aircraft Design: Synthesis and Analysis
Desktop aeronautics, Version 0.99

Appendices

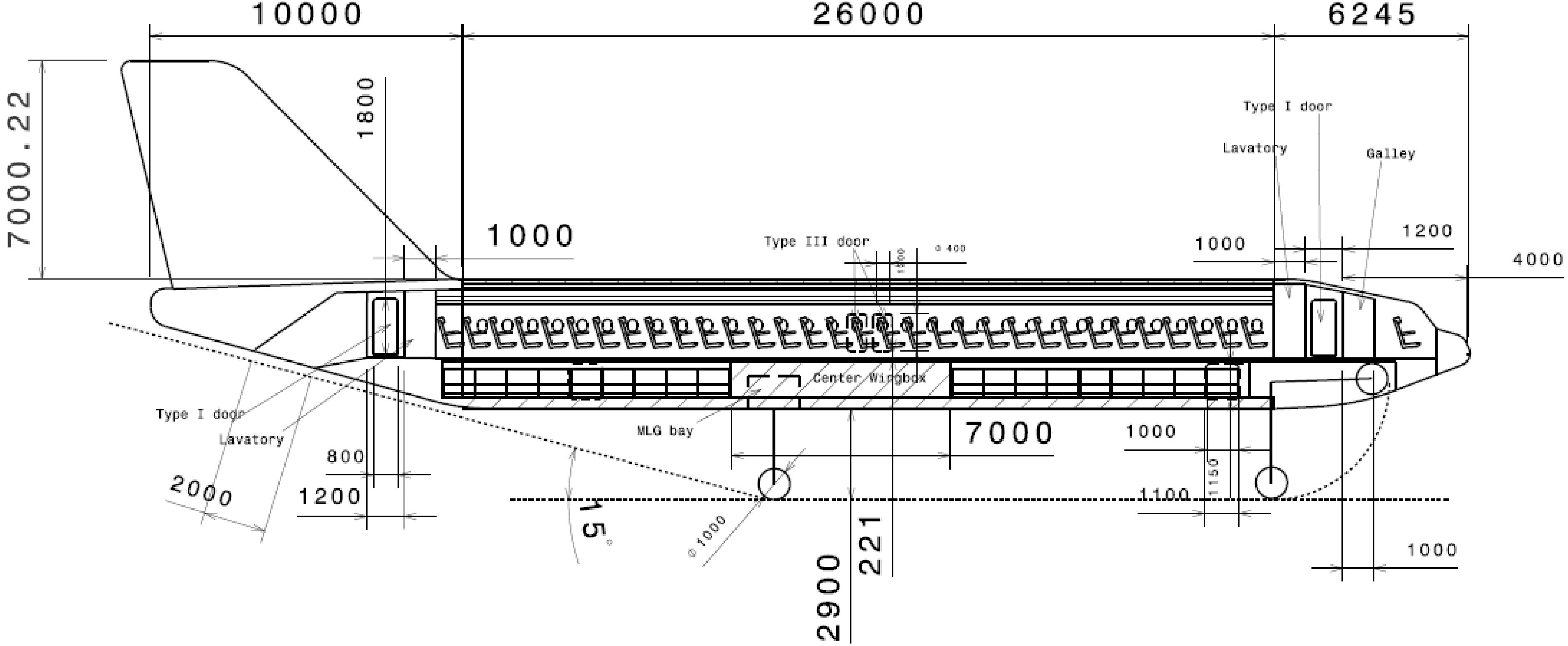
1 Aircraft parameters

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	^{o}
	Overside angle	35	^{o}
	Grazing angle	32	^{o}
	Upward angle	20	^{o}
	Divergence angle	19	^{o}
	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
	Wing airfoil geometry		
	NACA Airfoil series	64-414	-
MAC	Mean aerodynamic Chord	4.8	m
Y	MAC location, engine suspension point	7.3	m
dCl/dα	Lift curve slope	0.10	1/rad
A_{0L}	Zero lift angle of attack	-3	^{o}
C_{d0}	Minimum drag coefficient	0.0082	-
C_{lmax}	Maximum lift coefficient	1.45	-
α0	Zero Angle of attack lift coefficient	0.4	-
T/c	Thickness to chord ratio	0.14	-
M_{cr}	Critical Mach number	0.736	-
C_{ldes}	Design lift coefficient	0.4	-
	Aircraft Wing geometry		

b	Wing span	37.7	m
S	Wing area	158	m^2
A	Aspect ratio	9	-
Λ_{LE}	Leading Edge Wing sweep angle	30	^{O}
$\Lambda_{0.25C}$	Quarter chord wing sweep angle	26.77	^{O}
	Incidence angle	0	^{O}
	Twist angle	0	^{O}
	Dihedral angle	5	^{O}
Re	Cruise Reynolds number	$2.688*10^7$	-
C_r	Root Chord	7	m
	Taper ratio	0.197	-
T_r	Root thickness	0.98	M
R_c	Cantilever Ratio	20.9	-
M_{Dd}	Mach of drag divergence	0.757	-
α_{trim}	Trim angle	-0.249	^{O}
$C_{L\alpha}$	Lift curve slope, low speed	0.08	-
C_{Lmax}	Maximum wing lift coefficient	1.32	m
$M_{fuelmax}$	Total fuel tank capacity	32928	kg
	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

2 Aircraft technical drawings

See the following 3 pages.



Front view
Scale: 1:200

Aircraft Design

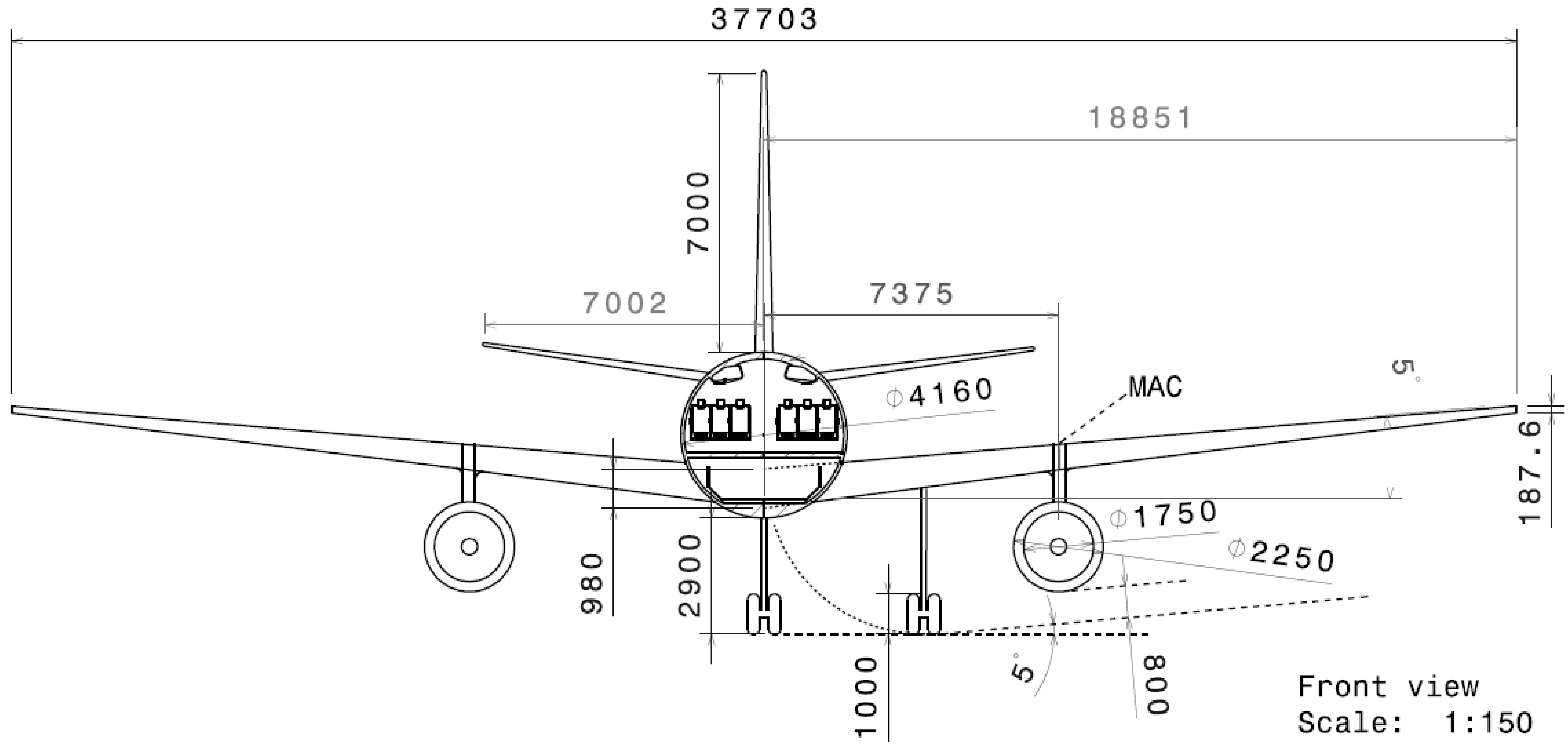
DRAWN BY	DATE
Wout De Backer	20/09/2009
CHECKED BY	DATE
DESIGNED BY	DATE

DRAWING TITLE

Side View

SIZE	DRAWING NUMBER	REV
A4		X

SCALE 1:200	WEIGHT(kg)		SHEET 1/1



Front view
Scale: 1:150

Aircraft Design	
DRAWING TITLE	Front View
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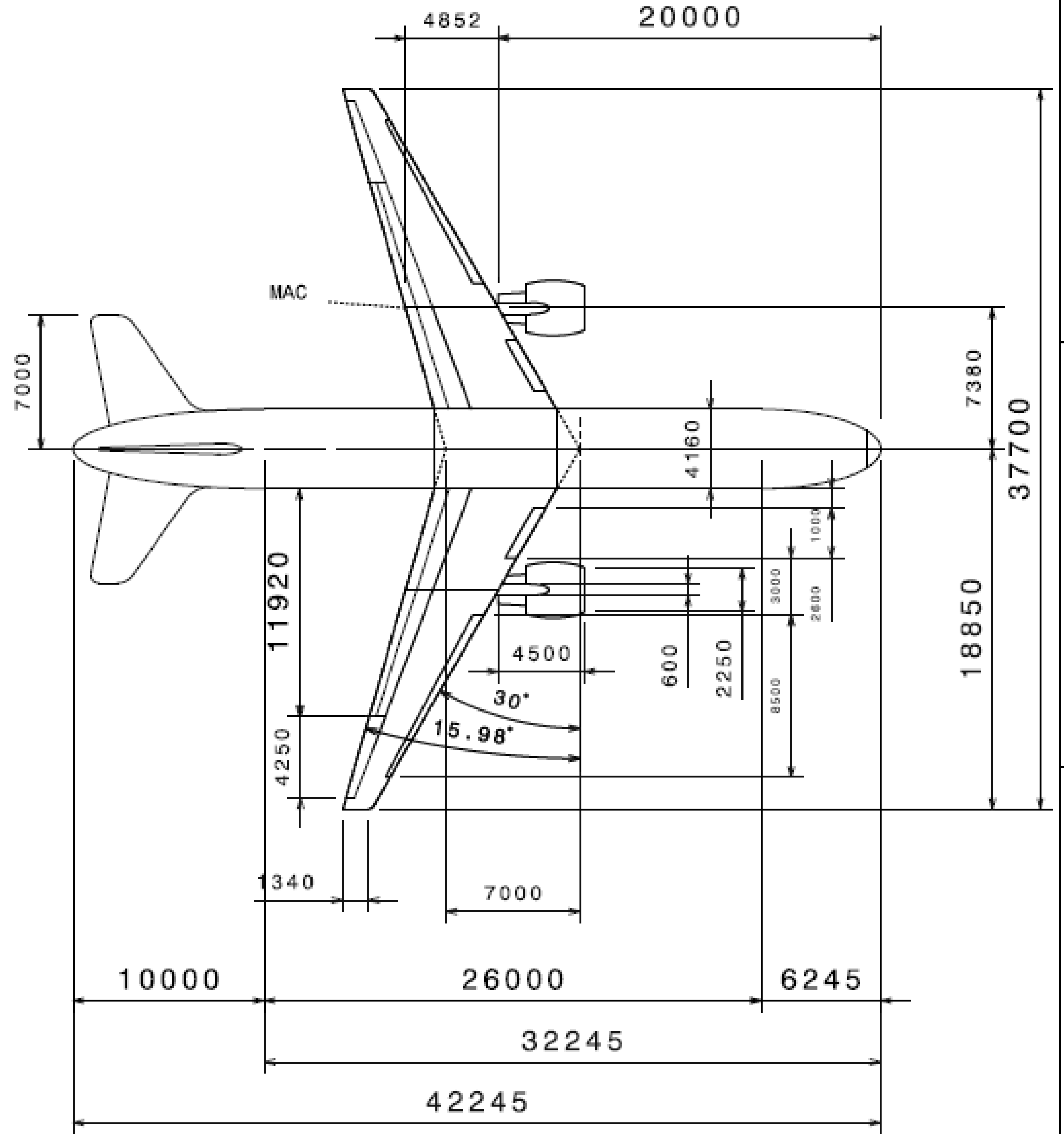
CHECKED BY	DATE
DESIGNED BY	DATE
SIZE: A4	DRAWING NUMBER: Product1
REV	
SCALE 1:1	WEIGHT(kg)
SHEET 1/1	



Top
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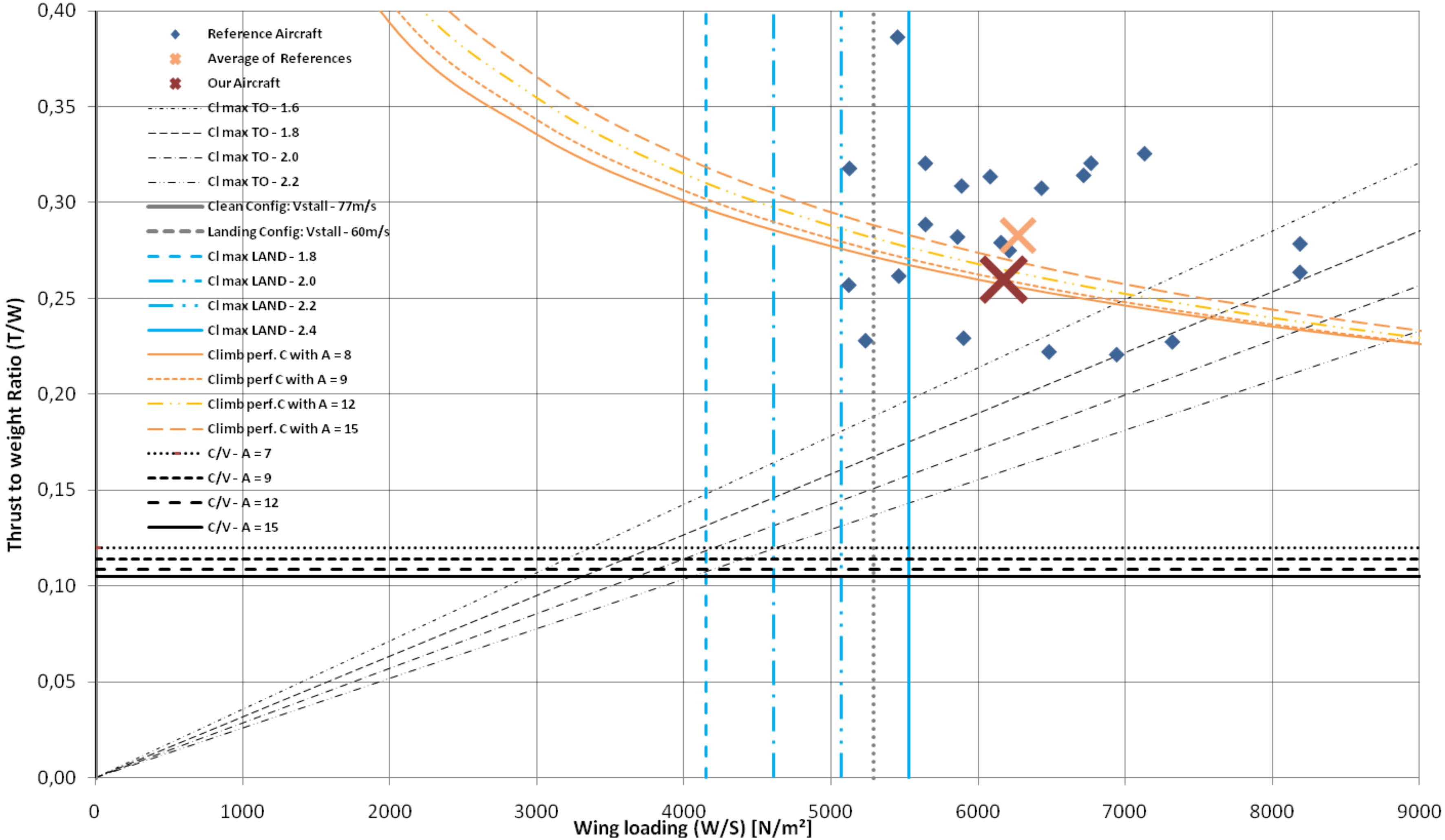
DRAWN BY Wout De Backer	DATE 07/12/2009				
CHECKED BY	DATE	SIZE A4	DRAWING NUMBER		REV
DESIGNED BY	DATE	SCALE	WEIGHT(kg)		SHEET 1/1

3 T/W – W/S Diagram



4 Reference aircraft table

4.1 Airbus aircraft parameters

Manufacturer Type Model	**AIRBUS A300-600R**	**AIRBUS A310-300**	**AIRBUS A319-100**	**AIRBUS A320-200**	**AIRBUS A321-200**	**AIRBUS A330-200**	**AIRBUS A330-300**	**AIRBUS A340-200**	**AIRBUS A340-300**	**AIRBUS A340-500**	**AIRBUS A340-600**	**AIRBUS A380-800**
Initial service date	1974	1983	1995	1988	1993	1998	1994	1993	1994	2002	2002	2004
In service (ordered)												
Africa	28	11	2(1)	27(6)	4	——>	-	——>	8(1)	——>	(2)	-
Middle East/Asia/Pacific	205(5)	69(5)	(4)	162(47)	17(32)	——>	59(47)	——>	54(10)	——>	(6)	-
Europe & CIS	83	87(1)	48(82)	244(105)	84(68)	——>	17(42)	——>	69(15)	——>	(27)	-
North & South America	72(37)	57	57(264)	237(146)	-	——>	4(42)	——>	9(8)	——>	(5)	-
Total aircraft	388(42)	224(6)	107(351)	670(304)	105(102)	——>	80(131)	——>	140(34)	——>	(40)	-
Engine Manufacturer	P&W	P&W	CFMI	CFMI	CFMI	GE	GE	CFMI	CFMI	R-R	R-R	R-R/GE/PW
Model / Type	4158	4152	CFM56-5A4	CFM56-5A3	CFM56-5B3	CF6-80E1A4	CF6-80E1A2	CFM-56-5C2	CFM-56-5C4	Trent 553	Trent 556	Options
No. of engines	2	2	2	2	2	2	2	4	4	4	4	4
Static thrust (kN)	257,0	231,0	99,7	111,2	142,0	310,0	300,0	139,0	151,0	235,8	249,1	294,0
Operational Items:												
Accomodation:												
Max. seats (single class)	375	280	153	179	220	380	440	440	440	440	475	850
Two class seating	266	218	124	150	186	293	335	303	335	350	440	
Three class seating	228	187	-	-	-	253	295	262	295	313	380	555
No. abreast	9	9	6	6	6	9	9	9	9	9	9	10+8
Hold volume (m³)	116,00	79,90	27,00	38,76	51,76	136,00	162,90	136,00	162,90	134,10	187,74	
Volume per passenger	0,31	0,29	0,18	0,22	0,24	0,36	0,37	0,31	0,37	0,30	0,40	
Mass (Weight) (kg):												
Ramp	171400	150900	64400	73900	89400	230900	217900	257900	271900	365900	365900	540900
Max. take-off	170500	150000	64000	73500	89000	230000	217000	257000	271000	365000	365000	540000
Max. landing	140000	123000	61000	64500	73500	177150	179000	181000	190000	236000	254000	381000
Zero-fuel	130000	113000	57000	60500	71500	165142	169000	172000	178000	222000	240000	356000
Max. payload	41100	33300	17390	19190	22780	36400	48400	49400	48150	51635	63000	85000
Max. fuel payload	27100	21500	5360	13500	19060	0	18600	21220	33160	31450	29311	15400
Design payload	25270	20710	11780	14250	17670	24035	28025	24890	28025	29735	36100	52725
Design fuel load	56330	49624	13020	17940	23330	85765	70786	111882	113125	164875	151890	216275
Operational empty	88900	79666	39200	41310	48000	120200	118189	120228	129850	170390	177010	271000
Weight Ratios:												
Ops empty/Max. T/O	0,521	0,531	0,613	0,562	0,539	0,523	0,545	0,468	0,479	0,467	0,485	0,502
Max. payload/Max. T/O	0,241	0,222	0,272	0,261	0,256	0,158	0,223	0,192	0,178	0,141	0,173	0,157
Max. fuel/Max. T/O	0,316	0,322	0,295	0,256	0,210	0,478	0,358	0,430	0,412	0,423	0,423	0,464
Max. landing/Max. T/O	0,821	0,820	0,953	0,878	0,826	0,770	0,825	0,704	0,701	0,647	0,696	0,706
Fuel (litres):												
Standard	68150	61100	23860	23860	23700	139090	98250	140000	141500	195620	195620	317000

Optional	75350	68300			26600				148700	213120		
DIMENSIONS												
Fuselage:												
Length (m)	53,30	45,13	33,84	37,57	44,51	57,77	62,47	58,21	62,47	65,60	69,57	67,46
Height (m)	5,64	5,64	4,14	4,14	4,14	5,64	5,64	5,64	5,64	5,64	5,64	8,50
Width (m)	5,64	5,64	3,95	3,95	3,95	5,64	5,64	5,64	5,64	5,64	5,64	7,02
Finess Ratio	9,45	8,00	8,57	9,51	11,27	10,24	11,08	10,32	11,08	11,63	12,34	9,61
Wing:												
Area (m^2)	260,00	219,00	122,40	122,40	122,40	363,10	363,10	363,10	363,10	437,30	437,30	817,00
Span (m)	44,84	43,89	33,91	33,91	33,91	58,00	58,00	58,00	58,00	61,20	61,20	79,80
MAC (m)	6,44	5,89	4,29	4,29	4,29	7,26	7,26	7,26	7,26	8,35	8,35	12,02
Aspect Ratio	7,73	8,80	9,39	9,39	9,39	9,26	9,26	9,26	9,26	8,56	8,56	7,79
Taper Ratio	0,300	0,283	0,240	0,240	0,240	0,251	0,251	0,251	0,251	0,220	0,220	0,213
Average (t/c) %	10,50	11,80										
1/4 Chord Sweep (°)	28,00	28,00	25,00	25,00	25,00	29,70	29,70	29,70	29,70	31,10	31,10	30,00
High Lift Devices:												
Trailing Edge Flaps Type	F2	F1	F1	F1	F2	S2	S2	S2	S2	S2	S2	S2
Flap Span/Wing Span	0,800	0,840	0,780	0,780	0,780	0,665	0,665	0,665	0,665	0,625	0,625	0,737
Area (m^2)	47,3	36,68	21,1	21,1	21,1							
Leading Edge Flaps Type	slats	slats	slats	slats	slats	slats	slats	slats	slats	slats	slats	slats
Area (m^2)	30,3	28,54	12,64	12,64	12,64							
Vertical Tail:												
Area (m^2)	45,20	45,20	21,50	21,50	21,50	47,65	45,20	45,20	45,20	47,65	47,65	134,20
Height (m)	8,60	8,10	6,26	6,26	6,26	9,44	8,45	8,45	8,45	9,44	9,44	13,66
Aspect Ratio	1,64	1,45	1,82	1,82	1,82	1,87	1,58	1,58	1,58	1,87	1,87	1,39
Taper Ratio	0,365	0,395	0,303	0,303	0,303	0,350	0,350	0,350	0,350	0,350	0,350	0,424
1/4 Chord Sweep (°)	40,00	40,00	34,00	34,00	34,00	45,00	45,00	45,00	45,00	45,00	45,00	
Tail Arm (m)	24,90	20,20	10,67	12,53	15,20	25,20	27,50	25,50	27,50	27,50	27,50	28,99
S_v/S	0,174	0,206	0,176	0,176	0,176	0,131	0,124	0,124	0,124	0,109	0,109	0,164
S_vL_v/S_b	0,097	0,095	0,055	0,065	0,079	0,057	0,059	0,055	0,059	0,049	0,049	0,060
Horizontal Tail:												
Area (m^2)	69,45	64,00	31,00	31,00	31,00	31,00	72,90	72,90	72,90	93,00	93,00	222,57
Span (m)	16,26	16,26	12,45	12,45	12,45	12,45	19,06	19,06	19,06	21,50	21,50	31,29
Aspect Ratio	3,81	4,13	5,00	5,00	5,00	5,00	4,98	4,98	4,98	4,97	4,97	4,40
Taper Ratio	0,420	0,417	0,256	0,256	0,256	0,256	0,360	0,360	0,360	0,360	0,360	0,383
1/4 Chord Sweep (°)	34,00	34,00	29,00	29,00	29,00	29,00	30,00	30,00	30,00	30,00	30,00	
Tail Arm (m)	25,60	22,50	11,67	13,53	16,20	16,20	28,60	26,50	28,60	28,60	28,60	31,29
S_h/S	0,267	0,292	0,253	0,253	0,253	0,253	0,201	0,201	0,201	0,213	0,213	0,272
S_hL_h/S_c	1,062	1,116	0,689	0,799	0,957	0,957	0,791	0,733	0,791	0,729	0,729	0,709
Undercarriage:												
Track (m)	9,60	9,60	7,60	7,60	7,60	7,60	10,70	10,70	10,70	10,70	10,70	13,47
Wheelbase (m)	18,60	15,21	12,60	12,63	16,90	16,90	25,40	23,20	25,40	28,53	32,50	27,60
Turning radius (m)	34,00	31,40	20,60	21,90	29,00	29,00	41,40		40,60			
No. of wheels (nose;main)	2;8	2;8	2;4	2;4	2;4	2;8	2;8	2;10	2;10	2;12	2;12	2;22
Main Wheel diameter (m)	1,245	1,168	1,143	1,143	1,270							
Main Wheel width (m)	0,483	0,406	0,406	0,406	0,455							
Nacelle:												
Length (m)	6,70	6,30	4,44	4,44	4,44	7,00	7,00	4,95	4,95	6,10	6,10	7,30

Max. width (m)	2,70	2,70	2,37	2,37	2,37	3,10	3,10	2,37	2,37	3,05	3,05	3,20
Spanwise location	0,359	0,352	0,338	0,338	0,338	0,312	0,312	0.312/0.672	0.312/0.672	0.296/0.625	0.296/0.625	0.348/0.678
PERFORMANCE												
Loadings:												
Max. power load (kg/kN)	331,71	324,68	320,96	330,49	313,38	370,97	361,67	462,23	448,68	386,98	366,32	459,18
Max. wing load (kg/m^2)	655,77	684,93	522,88	600,49	727,12	633,43	597,63	707,79	746,35	834,67	834,67	660,95
Thrust per engine												
Thrust/Weight Ratio	0,3073	0,3140	0,3176	0,3084	0,3253	0,2748	0,2819	0,2205	0,2272	0,2634	0,2783	0,2220
Take-off (m):												
ISA sea level	2280	2290	1750	2180	2000	2470	2320	2790	3000	3100	3100	
ISA +20ºC SL.	3189	2450	2080	2590	2286	2590	2680	3260	3380	3550	3550	3350
ISA 5000ft		2950	2360	2950	3269	3900	3840	4320	4298	4250	4250	
ISA +20ºC 5000ft		3660	2870	4390								
Landing (m):												
ISA sea level.	1489	1490	1350	1440	1580	1750	1600	1856	1964	2090	2240	
ISA +20ºC SL.	1489	1490	1350	1440	1580	1750	1600	1856	1964	2090	2240	
ISA 5000ft	1701	1686	1530	1645	1795	1970	1920	2094	2227	2390		
ISA +20ºC 5000ft	1701	1686	1530	1645	1795	1970	1920	2094	2227	2390		
Speeds (kt/Mach):												
V_2	153	156	133	143	143	158	144	154	158			
V_{app}	136	138	131	134	138	135	136	134	136	139	144	<150
V_{no}/M_{mo}	335/M0.82	360/M0.84	381/M0.89	350/M0.82	350/M0.82	330/M0.86	330/M0.86	330/M0.86	330/M0.86	330/M0.86	330/M0.86	
V_{ne}/M_{me}	395/M0.88	420/M0.90	350/M0.82	381/M0.89	TBD/M0.89	365/M0.93	365/M0.93	365/M0.93	365/M0.93	365/M0.93	365/M0.93	
$C_{L\max}$ (T/O)	2,44	2,45	2,58	2,56	3,10	2,21	2,51	2,60	2,61			
$C_{L\max}$ (L/D @ MLM)	2,98	3,02	2,97	3,00	3,23	2,74	2,73	2,84	2,89	2,86	2,87	
Max. cruise :												
Speed (kt)	480	484	487	487	487		500	500	500			507
Altitude (ft)	31000	35000	33000	28000	28000		33000	33000	33000			35000
Fuel consumption (kg/h)	5120	4690	3160	3200	3550		5000	7180	7300			
Long range cruise:												
Speed (kt)	456	458	446	448	450	470	465	475	475			
Altitude (ft)	35000	37000	37000	37000	37000	39000	39000	39000	39000			
Fuel consumption (kg/h)	4300	3730	1980	2100	2100		4700	5400	5700			
Range (nm):												
Max. payload	3283	3645	1355	637	1955	4210	3888	6393	6371	7050	5700	6200
Design range	4000	4300	1900	2700	2700	6370	4500	7350	7150	8500	7500	7650
Max. fuel (+ payload)	4698	5076	4158	3672	2602		7046	8834	8089	9000	7800	-
Ferry range										9800	8800	10300
Design Parameters:												
$W/SC_{L\max}$	2160,21	2227,23	1726,69	1962,27	2211,48	2269,21	2150,33	2445,04	2529,97	2865,71	2857,63	
W/SC_{LtoST}	2678,25	2702,77	2071,39	2423,85	2590,29	3146,34	2906,75	4224,13	4242,69	4144,91	3912,54	
Fuel/pax/nm (kg)	0,0529	0,0529	0,0553	0,0443	0,0465	0,0460	0,0470	0,0502	0,0472	0,0554	0,0460	
Seats x Range (seats.nm)	1064000	937400	235600	405000	502200	1866410	1507500	2227050	2395250	2975000	3300000	

4.2 Boeing aircraft parameters

Manufacturer Type Model	**BOEING 707- 320C**	**BOEING 717- 200**	**BOEING 727- 200Adv**	**BOEING 737- 200**	**BOEING 737- 300**	**BOEING 737- 400**	**BOEING 737- 500**	**BOEING 737- 600**	**BOEING 737- 700**	**BOEING 737- 800**
Initial service date	1962	1999	1970	1967	1967	1967	1967	1998	1997	1998
In service (ordered)										
Africa	53	-	58	85	14(1)	7	17	(7)	(2)	2(12)
Middle East/Asia/Pacific	31	-	52	114	194(19)	142(5)	49(2)	-	9(24)	7(45)
Europe & CIS	9	-	94	147	272(12)	216(4)	145(1)	6(49)	21(35)	20(110)
North & South America	25	(60)	799	532	573(11)	97(5)	165(1)	-	36(146)	13(211)
Total aircraft	120	(60)	1003	878	1053(43)	462(14)	376(4)	6(56)	66(207)	42(378)
Engine Manufacturer	P&W	BMW R-R	P&W	P&W	CFMI	CFMI	CFMI	CFMI	CFMI	CFMI
Model / Type	JT3D-7	715	JT8D-15A	JT8D-15A	CFM56-3-B1	CFM56-3B-2	CFM56-3-B1R	CFM56-7B18	CFM56-7B20	CFM56-7B24
No. of engines	4	2	3	2	2	2	2	2	2	2
Static thrust (kN)	84,5	97,9	71,2	71,2	89,0	97,9	82,3	82,0	89,0	107,0
Operational Items:										
Accomodation:										
Max. seats (single class)	219	110	189	130	149	170	130	132	149	189
Two class seating	147	106	136	115	128	146	108	108	128	160
Three class seating	-	-	-	-	-	-	-	-	-	-
No. abreast	6	5	6	6	6	6	6	6	6	6
Hold volume (m³)	50,27	25,00	43,10	24,78	30,20	38,90	23,30	23,30	30,2	47,1
Volume per passenger	0,23	0,23	0,23	0,19	0,20	0,23	0,18	0,18	0,20	0,25
Mass (Weight) (kg):										
Ramp	152405	52110	95238	52615	56700	63050	52620	65310	69610	78460
Max. take-off	151315	51710	95028	52390	56470	62820	52390	65090	69400	78220
Max. landing	112037	46266	72575	46720	51710	54880	49900	54650	58060	65310
Zero-fuel	104330	43545	63318	43091	47630	51250	46490	51480	54650	61680
Max. payload	38100	12220	18597	15445	16030	17740	15530	9800	11610	14690
Max. fuel payload	12852	8921	24366	9118	8705	13366	5280	7831	10996	15921
Design payload	13965	10070	12920	10925	12160	13870	10260	10260	12160	15200
Design fuel load	71126	9965	35944	13819	12441	15580	11170	18390	19655	21540
Operational empty	66224	31675	46164	27646	31869	33370	30960	36440	37585	41480
Weight Ratios:										
Ops empty/Max. T/O	0,438	0,613	0,486	0,528	0,564	0,531	0,591	0,560	0,542	0,530
Max. payload/Max. T/O	0,252	0,236	0,196	0,295	0,284	0,282	0,296	0,151	0,167	0,188
Max. fuel/Max. T/O	0,471	0,212	0,255	0,341	0,281	0,253	0,303	0,316	0,296	0,263
Max. landing/Max. T/O	0,740	0,895	0,764	0,892	0,916	0,874	0,952	0,840	0,837	0,835
Fuel (litres):										
Standard	90299	13892	30622	19532	20105	20105	20105	26024	26024	26024
Optional		16065	40068	22598	23170	23170	23170			
DIMENSIONS										
Fuselage:										
Length (m)	44,35	33,00	41,51	29,54	32,30	35,30	29,90	29,88	32,18	38,08

Height (m)	3,76	3,61	3,76	3,73	3,73	3,73	3,73	3,73	3,73	3,73
Width (m)	3,76	3,61	3,76	3,73	3,73	3,73	3,73	3,73	3,73	3,73
Finess Ratio	7,30	4,30	7,00	7,40	7,40	7,40	7,40	7,40	7,40	7,40
Wing:										
Area (m^2)	283,40	92,97	157,90	91,04	91,04	91,04	91,04	124,60	124,60	124,60
Span (m)	44,42	28,40	32,92	28,35	28,90	28,90	28,90	34,30	34,30	34,30
MAC (m)	7,36	3,88	5,46	3,80	3,73	3,73	3,73	4,17	4,17	4,17
Aspect Ratio	6,96	8,68	6,86	8,83	9,17	9,17	9,17	9,44	9,44	9,44
Taper Ratio	0,259	0,196	0,309	0,266	0,240	0,240	0,240	0,278	0,278	0,278
Average (t/c) %	10,00	11,60	11,00	12,89	12,89	12,89	12,89			
1/4 Chord Sweep (°)	35,00	24,50	32,00	25,00	25,00	25,00	25,00	25,00	25,00	25,00
High Lift Devices:										
Trailing Edge Flaps Type	F1	S2	F3	F3	S3	S3	S3	S2	S2	S2
Flap Span/Wing Span	0,670	0,650	0,740	0,740	0,720	0,720	0,720	0,599	0,599	0,599
Area (m^2)	44,22		36,04							
Leading Edge Flaps Type	flaps	slats	slats/flaps	slats/flaps	slats/flaps	slats/flaps	slats/flaps	slats/flaps	slats/flaps	slats/flaps
Vertical Tail:										
Area (m^2)	30,47	19,50	33,07	19,70	23,13	23,13	23,13	23,13	23,13	23,13
Height (m)	7,20	4,35	4,60	5,85	6,00	6,00	6,00	6,00	6,00	6,00
Aspect Ratio	1,70	0,97	0,64	1,74	1,56	1,56	1,56	1,56	1,56	1,56
Taper Ratio	0,410	0,780	0,780	0,288	0,310	0,310	0,310	0,310	0,310	0,310
1/4 Chord Sweep (°)	30,00	45,00	53,00	35,00	35,00	35,00	35,00	35,00	35,0	35,0
Tail Arm (m)	21,00	12,80	14,20	12,10	13,68	14,90	12,90	13,55	14,7	17,7
S_v/S	0,108	0,210	0,209	0,216	0,254	0,254	0,254	0,186	0,186	0,186
S_vL_v/S_b	0,051	0,095	0,090	0,092	0,120	0,131	0,113	0,073	0,080	0,096
Horizontal Tail:										
Area (m^2)	58,06	24,20	34,93	31,31	31,31	31,31	31,31	32,40	32,40	32,40
Span (m)	13,95	10,80	10,90	12,70	12,70	12,70	12,70	13,40	13,40	13,40
Aspect Ratio	3,35	4,82	3,40	5,15	5,15	5,15	5,15	5,54	5,54	5,54
Taper Ratio	0,400	0,380	0,380	0,260	0,260	0,260	0,260	0,186	0,186	0,186
1/4 Chord Sweep (°)	36,00	30,00	36,00	30,00	30,00	30,00	30,00	30,00	30,00	30,00
Tail Arm (m)	20,50	14,30	20,10	14,78	14,78	16,00	14,00	13,58	14,73	17,68
S_h/S	0,205	0,260	0,221	0,344	0,344	0,344	0,344	0,260	0,260	0,260
S_hL_h/S_c	0,571	0,959	0,814	1,338	1,363	1,475	1,291	0,847	0,919	1,102
Undercarriage:										
Track (m)	6,73	4,88	5,72	5,23	5,25	5,25	5,25	5,70	5,7	5,7
Wheelbase (m)	17,98	17,60	19,28	11,38	12,40	14,30	11,00		12,4	
Turning radius (m)			25,00		19,50				19,5	
No. of wheels (nose;main)	2;8	2;4	2;4	2;4	2;4	2;4	2;4	2;4	2;4	2;4
Main Wheel diameter (m)	1,117		1,245		1,016	1,016	1,016	1,016	1,016	1,016
Main Wheel width (m)	0,406		0,432		0,368	0,368	0,368	0,368	0,368	0,368
Nacelle:										
Length (m)	6,00	6,10	7,00	7,00	4,70	4,70	4,70	4,70	4,70	4,70
Max. width (m)	1,60	1,75	1,50	1,50	2,00	2,00	2,00	2,06	2,06	2,06
Spanwise location	0.44/0.71	-	-	0,350	0,340	0,340	0,340	0,282	0,282	0,282
PERFORMANCE										
Loadings:										
Max. power load (kg/kN)	447,47	264,10	444,89	367,91	317,25	320,84	318,29	396,89	389,89	365,51

Max. wing load (kg/m^2)	533,93	556,20	601,82	575,46	620,28	690,03	575,46	522,39	556,98	627,77
Thrust per engine										
Thrust/Weight Ratio	0,2278	0,3860	0,2291	0,2884	0,3133	0,3203	0,3203	0,2568	0,2615	0,2789
Take-off (m):										
ISA sea level	3054		3033	1829	1939	2222	1832			
ISA +20ºC SL.			3658	1859	2109	2475	2003	1878	2042	2316
ISA 5000ft			3962	2886	2432		2316			
ISA +20ºC 5000ft			4176	3292	2637		2649			
Landing (m):										
ISA sea level.	1905	1445	1494	1350	1396	1582	1362	1268	1356	1600
ISA +20ºC SL.			1494	1350	1396	1582	1362	1268	1356	1600
ISA 5000ft			1661	1615	1576	1695	1533			
ISA +20ºC 5000ft			1661	1615	1576	1695	1533			
Speeds (kt/Mach):										
V_2		150	166	147	148	159	142			
V_{app}	135	130	137	131	133	138	130			
V_{no}/M_{mo}	383/M0.90	438/M0.76	390/M0.90	350/M0.84	340/M0.82	340/M0.82	340/M0.82	392/M0.84	392/M0.84	392/M0.84
V_{ne}/M_{me}	425/M0.95		M0.95							
$C_{L\max}$ (T/O)		2,15	1,90	2,32	2,47	2,38	2,49			
$C_{L\max}$ (L/D @ MLM)	2,22	3,01	2,51	3,06	3,28	3,24	3,32			
Max. cruise :										
Speed (kt)	521		530	488	491	492	492			
Altitude (ft)	25000		25000	25000	26000	26000	26000	41000	41000	41000
Fuel consumption (kg/h)			4536	4005	3890	3307	3574			
Long range cruise:										
Speed (kt)	478	438	467	420	429	430	429	450	452	452
Altitude (ft)		35000	33000	35000	35000	35000	35000	39000	39000	39000
Fuel consumption (kg/h)			4309	2827	2250	2377	2100	1932	2070	2186,84
Range (nm):										
Max. payload	3150		2140	1549	1578	1950	1360			
Design range		1375	2400	1900	2850	2700	1700	3191	3197	2897
Max. fuel (+ payload)	5000			2887	3187	2830	3450	3229	3245	2927
Ferry range										
Design Parameters:										
$W/SC_{L\max}$	2360,53	1811,43	2356,82	1845,48	1852,54	2090,56	1701,59			
W/SC_{LtoST}	3947,87	1788,04	3918,96	2537,71	2196,64	2506,93	2024,27			
Fuel/pax/nm (kg)		0,0684	0,1101	0,0632	0,0341	0,0395	0,0608	0,0534	0,0480	0,0465
Seats x Range (seats.nm)		145750	326400	218500	364800	394200	183600	344628	409216	463520