# Aircraft Design –

**Assignment [52, Assignment 4**: Stability, Control, Weight and Balance**]**

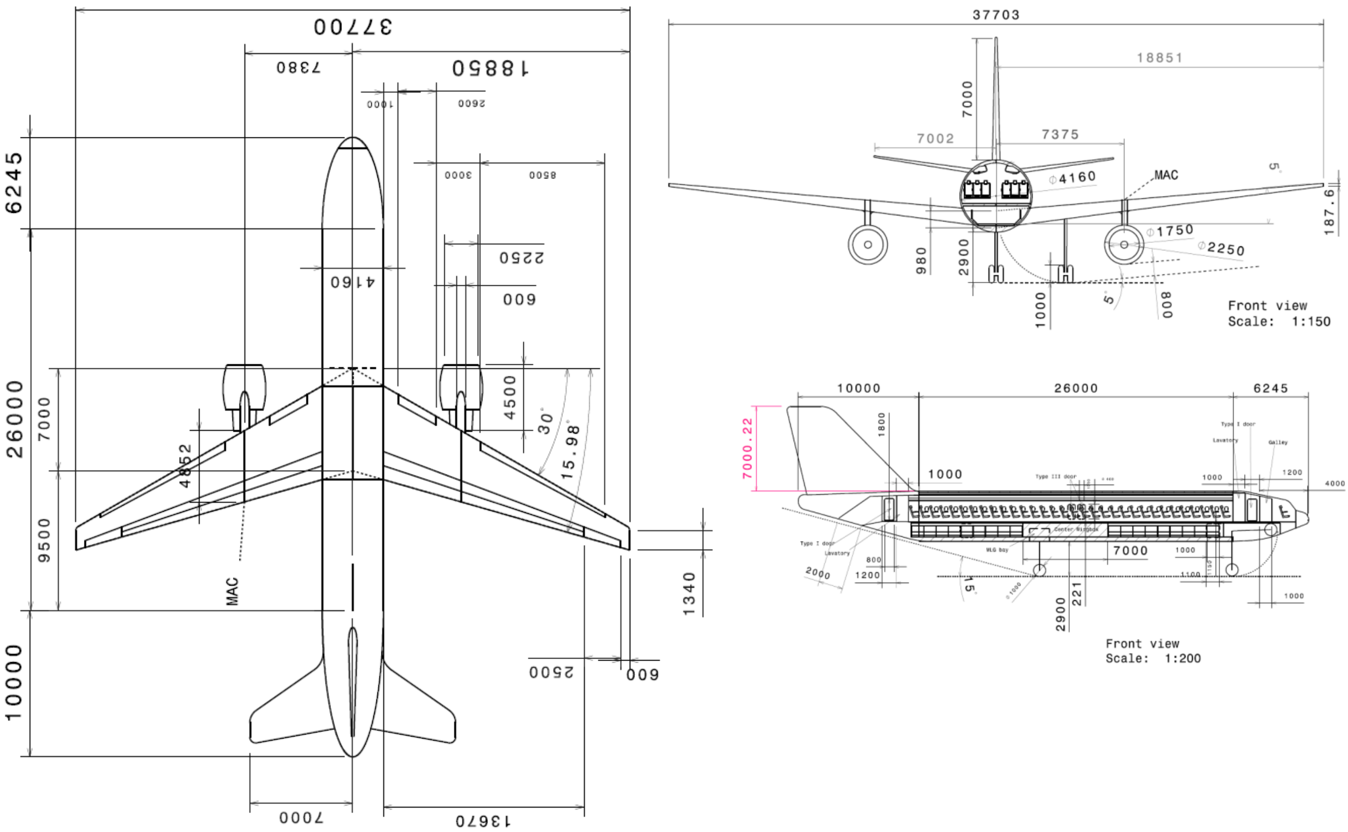
Student Names and Study Numbers:

Instance [first delivery]

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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# Chapter 1: Introduction

In this report, the stability, control, weight and balance of the aircraft with specifications given in table 1.1 will be discussed. For this, several steps are necessary. The first chapter (chapter 2) will go through the calculations that are needed to estimate the tail size and properties using the V-bar method. Next, in chapter 3, the ultimate load factor will be determined for further design. This load factor will be then used in chapter 4 when estimating the aircraft OEW and cog. position using the Torenbeek Class II handbook. Then in chapter 5 a reasonable position of the center of gravity with respect to the fuselage ($X_{LEMAC}$) will be proposed. In chapter 6, the Scissor plot will be made. This will finally be used in chapter 7, where the final tail is checked against its initial v-bar values.

# Chapter 2: Tail Plane Geometry

In this chapter, the V-bar method will be used to size the horizontal and vertical tail-planes. The tail-planes, as can be seen in the drawings, are conventional tail-planes, intersecting with the fuselage in the tail-cone.

## 2.1 Volume coefficients

The V-bar method is used early in the design stages, to determine a preliminary sizing of the tail, which allows a quick startup for the weight and aerodynamic group. The following formulas come from statistical data concerning vertical and horizontal tail planes, and allow sizing of the volume coefficients of the tail-planes. $\bar{V}_h$ is the horizontal tail volume coefficient, $\bar{V}_v$ is the vertical tail coefficient.

$$\bar{V}_h = \frac{S_h l_h}{S \cdot \bar{c}} \qquad \textit{Eq. 2.1}$$

$$\bar{V}_v = \frac{S_v l_v}{S \cdot b} \qquad \textit{Eq. 2.2}$$

In these equations, S is the surface of the wing, $S_h$ and $S_v$ are the surfaces of the horizontal and vertical tail planes respectively, l is the distance between the quarter chord point of the mean aerodynamic centre of the wing and the quarter chord point of the mean aerodynamic centre of the tail plane, c is the mean chord and b is the wing span. These values can be obtained from tables 8.7a, 8.7b and 8.7c in Roskam.

Since our aircraft, from overall characteristics is very narrowly related to the Boeing 737-300, the these values are used for the design of the volume coefficients. The values for the 737 are summarized below.

| | |
|---|---|
| $\bar{V}_h$ | 1.35 |
| $\bar{V}_v$ | 0.1 |

*Table 2.1: Volume coefficients of the Boeing 737*

The actually used values differ slightly from the 737, this is to ensure better results later on in the assignment. The chosen values are shown below.

| | |
|---|---|
| $\bar{V}_h$ | 1.10 |

| | |
|---|---|
| $\bar{V}_v$ | 0.085 |

*Table 2.2: Design volume coefficients of our aircraft*

Now the tail plane geometry can be calculated.

## 2.2 The vertical tail

First an estimation is done to find the surface of the tail plane using reference aircraft. From the closely related 737-800 from the references and the values for the 737-300 from Roskam, the following surfaces are found.

| | |
|---|---|
| $S_v$ Roskam | 22.20 m² |
| $S_v$ 737-800 ref | 23.13 m² |

*Table 2.3: Reference Surfaces*

For our aircraft, since it is slightly larger, a surface of 25 m² is chosen. Using equation 2.2, the length between the two quarter points of the aerodynamic chords can be found.

$$l_v = \frac{S \cdot b \cdot \bar{V}_v}{S_v} = 20.247\text{m}$$

The rest of the geometry can be determined with some more parameters found from lecture notes. The following values of quarter chord sweep angle, aspect ratio, taper ratio and thickness ratio can be found for jet aircraft. The sweep angle is chosen, using the average of the reference aircraft. This is 39 degrees.

| | |
|---|---|
| Taper Ratio | 0.3 - |
| Aspect ratio | 1.9 - |
| Sweep angle (0.25c) | 35-45 degrees (39) |

*Table 2.4: Vertical tail parameters.*

For the thickness to chord ratio, a value of 0.13 is chosen. This is slightly lower than the value for the wing, which was 0.14. this should postpone shock induced separation. The length of the root and tip chord can now be determined from the taper ratio and aspect ratio. This is the same method as for the wing design. First the span is calculated:

$$b_v = \sqrt{A_v S_v} = 6.892\text{m} \qquad \textit{Eq. 2.3}$$

From the reference aircraft, the average height is found to be 7.3m, so this height of the tail is acceptable. The root and tip chord can then be found from equations 2.4 and 2.5.

$$C_{rv} + C_{tv} = \frac{2 \cdot S}{b} \qquad \textit{Eq. 2.4}$$

$$C_{rv} = \frac{C_{rv} + C_{tv}}{1 + \lambda} \qquad \textit{Eq. 2.5}$$

Or, solving for Cr and Ct gives:

$$C_{rv} = \frac{2 \cdot S}{b \cdot (1+\lambda)} = 5.581\text{m}$$
$$C_{tv} = \lambda \cdot C_{rv} = 1.674\text{m}$$

The mean aerodynamic chord can be calculated then with equation 2.6.

$$\bar{c} = \frac{2c_r(1+\lambda+\lambda^2)}{3 \cdot (1+\lambda)} = 3.978\text{m} \quad \textit{Eq. 2.6}$$

## 2.3 The horizontal tail

In the same manner, the horizontal tail can be sized. From the values of the 737-300 in Roskam and the 737-800 we again get that the surface equals:

| | |
|---|---|
| $S_h$ Roskam | 36.23 m² |
| $S_h$ 737-800 ref | 32.40 m² |

Table 2.5: Reference Surfaces

For our aircraft, again, a larger surface of 45 m² is chosen. Using equation 2.1, the length between the two quarter points of the aerodynamic chords can be found.

$$l_h = \frac{S \cdot \bar{c} \cdot \bar{V}_h}{S_h} = 18.601\text{m}$$

Again, using the following values from the lecture notes:

| | |
|---|---|
| Taper Ratio | 0.4 - |
| Aspect ratio | 5 - |
| Sweep angle (0.25c) | 10° more than wing (36.7°) |

Table 2.6: Horizontal tail parameters.

The thickness to chord ratio is 1 percent smaller than that of the wing, and is chosen at 0.13. Using the same equations as in chapter 2.2, the span, tip root, chord root and mean aerodynamic cord can be found. These values are summarized in table 2.7.

| | |
|---|---|
| Span | 15.000m |
| Root Chord | 4.286m |
| Tip chord | 1.714m |
| MAC-htail | 3.184m |

Table 2.7: Horizontal tail parameters.

Now, the rudder and elevator need to be sized. Using reference aircraft, it is found that the flap area is about 25pct of the horizontal and vertical tail surfaces. They will be drawn as such on the technical drawings. The span of these surfaces will be the whole exterior part of the tail surfaces, from the tip till the interception with the fuselage

# Chapter 3: The load factor

In this chapter the generation of the V-n diagram will be treated. With this diagram the maximum load factor, $n_{max}$ , of the aircraft can be determined. This maximum load factor is used in the weight calculations of the wing and other components. Table 3.1, gives the Characteristic values of maximum load factors of transport aircraft.

| Aircraft Type | Load factor |
|---|---|
| Commercial Transport | $1.5 \leq n \leq 3.5$ |

*Table 3.1: Characteristic values of maximum load factors of commercial aircraft*

In order to determine the maximum load factor, two cases must be evaluated first; the load factor during maneuvers and during gust. The largest one, multiplied by a safety factor of 1.5, determines the limit load factor which will be used for class II weight estimation method in chapter 4.

## 3.1 Maneuver loading

### 3.1.1. The maximum maneuver loads

The loading during maneuvers are specified in the Airworthiness certification specifications. For different flight conditions different requirements are set. According to the airworthiness-regulations, this aircraft, a transport aircraft, falls under the CS/FAR-25 regulations.

| CS | Aircraft type | | $n_{max}$ |
|---|---|---|---|
| CS-23 | Normal + Commuter | | $2.1 + 24{,}000 / W_{TO} + 10{,}000$ |
| | Utility | | 4.4 |
| | Aerobatic | | 6.0 |
| CS-25 | Transport | $\leq 4100$ [lbs] | 3.8 |
| | | $4100 < W_{TO} \leq 50{,}000$ [lbs] | $2.1 + 24{,}000 / W_{TO} + 10{,}000$ |
| | | $> 50{,}000$ [lbs] | 2.5 |

*Table 3.2: Maximum positive load factor*

| CS | Aircraft type | $-n_{max}$ |
|---|---|---|
| CS-23 | Normal + Commuter + Utility | $-0.4 \cdot n_{max}$ |
| | Aerobatic | $-0.5 \cdot n_{max}$ |
| CS-25 | Transport | $-1$ |

*Table 3.3: Maximum negative load factor*

Since this aircraft has a MTOW of 219107 Lbs, it can be seen in Table 3.2 and Table 3.3 that the maximum maneuver loads **are 2.5 and -1** as positive and negative loads respectively.

### 3.1.2. Plot manoeuvre V-n Diagram

These load factors can be given as a function of the velocity of the aircraft and a V-n Diagram, Figure 3.1, can be plotted to show the load factor envelope for the maneuvering of the aircraft.

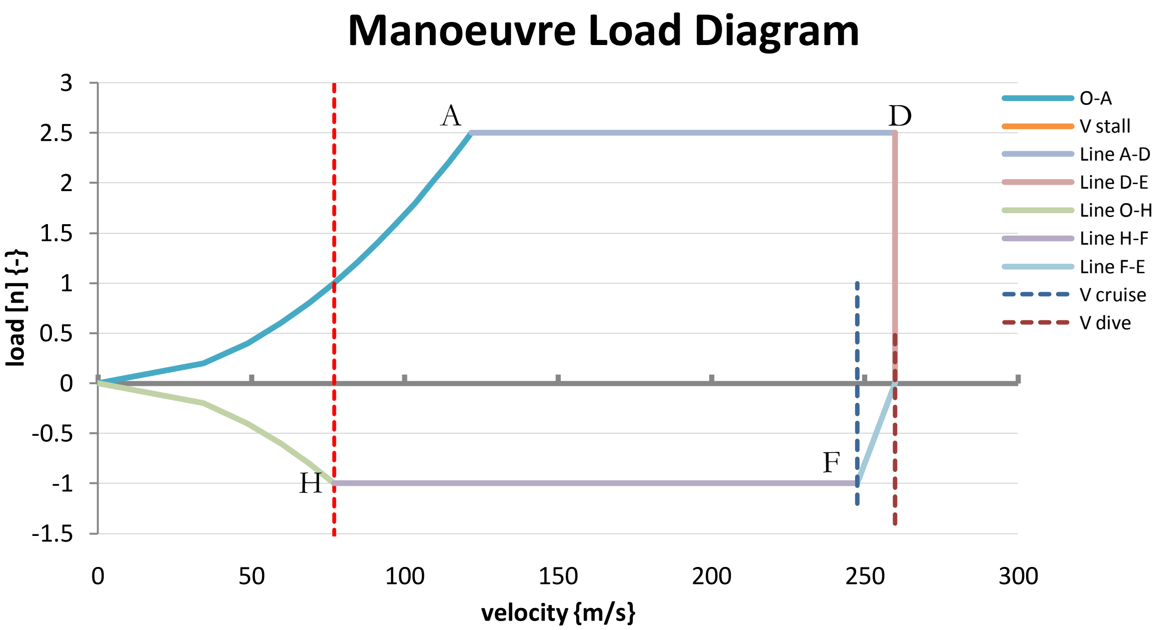



*Figure 3.1: Reference maneuver load diagram.*

**Curve 0-A**

In the V-n Diagram, Figure 3.1, curve 0-A represents the maximum normal component load due to a high angle of attack and thus the maximum positive lift. This curve is defined by equation Eq.3.1:

$$n = \frac{qC_{L_{max}}}{W/S} \qquad \textit{Eq.3.1}$$

Where the wing loading W/S is equal to 6572.3 N/m$^2$, as determined in assignment 2, $C_{Lmax,clean}$ is the maximum lift coefficient at low speed (=1.7) and q is the dynamic pressure taken at sea level with an air density of 1.225 kg/m3 and airspeed v is variable.

The stall speed is visualized in Figure 3.2 there where curve 0-A crosses the n=1 line and it is equal to 77 m/s, a little bit higher than the assumed value of 66 m/s in assignment 2. The limit of curve 0-A is imposed by the value of the maximum limit load factor of line A-D. This value is specified in CS-25 for transport aircraft categories to be 2.5 as specified in Table 3.2.

**Curve A-D**

Then line A-D follows parallel to the x-axis until the dive velocity. This dive velocity is also defined in the certification specifications CS-25 to satisfy the relation : $V_C/M_C \leq 0.8 \cdot V_D/M_D$. But in general $M_D$ is 0.05M higher than $M_c$ ,which gives a dive velocity of $V_D$ = 1.05 * $V_{cruise}$ = 262.6 m/s.

**Curve 0-H**

Line 0-H represents the maximum load produced by a high angle of attack flight in negative direction and is also given by Equation 3.1.

**Curve H-F**
Line H-F represents the largest negative load factor of this aircraft as specified in CS-25 to be -1.

Finally from point F, at the cruise velocity of 247 m/s, a line runs to zero load to the dive velocity.

## 3.2 Gust loading

In this section, the second part of the V-n diagram, given by the load factor during a gust, will be generated. In Figure 3: the gust load diagram of this aircraft is given.

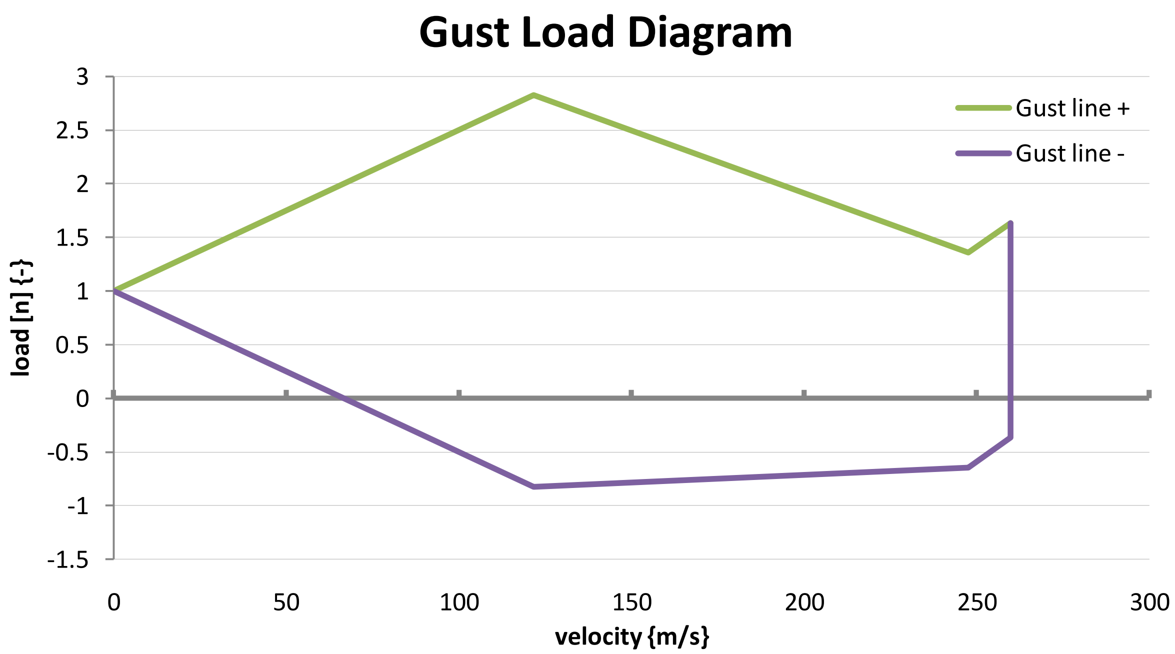



*Figure 3.2: Gust load diagram*

First, the gust at 3 velocities was evaluated: the speed at which highest lift and maximum angle of attack occurs ($V_B$), the cruise speed ($V_C$) and the dive speed ($V_D$). The corresponding gust speeds vary with the condition and with altitude. The maximum load factor due to gust is then given by equation **Fout! Verwijzingsbron niet gevonden.**. The value of 1 represents the mean load factor. It is assumed that the aircraft has this load factor of 1 before the gust occurs.

$$n_{peak} = 1 + \Delta n \qquad \textit{Eq.3.2}$$

The value of $\Delta n$ is the incremental load factor and can be obtained by equation Eq.3.3:

$$\Delta n = \frac{\rho V C_{L_\alpha} u}{2(W/S)} \qquad \textit{Eq.3.3}$$

To be able to calculate this incremental load factor the normal component of the gust velocity u is required. This value can be calculated using statistical gust velocities ($\hat{u}$) and the so-called load alleviation factor K in equation Eq.3.4.

$$u = K\hat{u} \qquad \textit{Eq.3.4}$$

For subsonic aircraft, K changes according to equation Eq.3.5:

$$K = \frac{0.88\mu}{5.3 + \mu} \qquad \textit{Eq.3.5}$$

Where μ is an equivalent mass ratio and is defined in equation Eq.3.6:

$$\mu = \frac{2W/S}{\rho g \bar{c} C_{L_\alpha}} \qquad \textit{Eq.3.6}$$

With $g = 9.81\ \text{m/s}^2$ the gravitational constant, $\bar{c} = 4.818\ \text{m}$ the mean aerodynamic chord and $C_{L_\alpha}$ the slope of the lift curve and varies with different conditions. Then the values for û, based on statistics and change with altitude and flight condition, are presented in Table 3.4 and Figure 3.3.

| Flight condition | Altitude range [f] | û [f/s] |
|---|---|---|
| High angle of attack | 0-20,000 | 66 |
| | 50,000 | 38 |
| Level flight | 0-20,000 | 50 |
| | 50,000 | 25 |
| Dive condition | 0-20,000 | 25 |
| | 50,000 | 12.5 |

*Table 3.4: Gust velocities for different conditions*

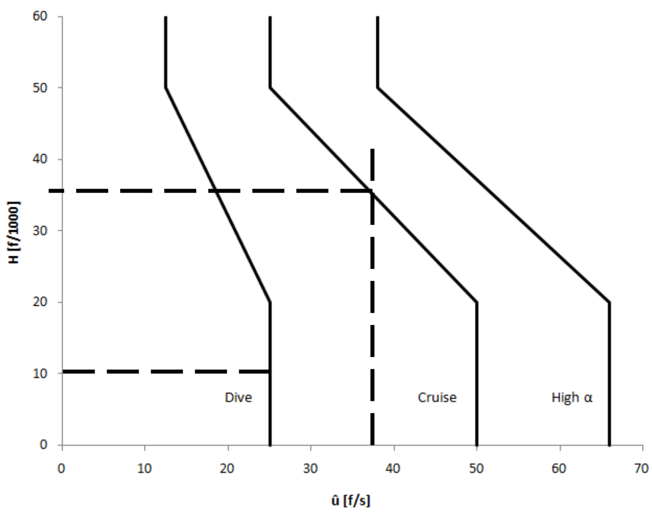



*Figure 3.3: Variation of gust velocity û with altitude and flight condition*

For High angle, the aircraft is assumed to be flying at sea level conditions, giving a statistical gust velocities **û** of 66 ft/s. Then the level flight condition is assumed to be at cruise altitude (11800 m) and **û becomes 38 ft/s**. Finally, during dive, the aircraft is assumed to be flying at a lower altitude than the cruise altitude. For dive condition, an altitude of 3440 m is assumed which gives an air density of 0.8689 kg/m3 and **û = 25 ft/s.** Then using these values and equation 3.2 up to 3.6 the maximum load factor can be calculated for different conditions as summarized in Table 3.5.

| Symbol | Definition | Unit | High angle | Level flight | Dive |
|---|---|---|---|---|---|
| H | Altitude | {m} | 0 | 11800 | 3440 |
| V | Velocity | {m/s} | 66 | 248 | 263 |
| ϱ | Air density | {kg/m3} | 1.225 | 0.321 | 0.590 |
| $C_{La}$ | Lift Slope flapped | {1/rad} | 5.455 | 5.622 | 5.622 |
| u | u equivalent mass ratio | - | 39 | 145 | 79 |
| K | K load alleviation factor | - | 0.775 | 0.849 | 0.825 |
| û | statistical gust velocities ( ) | {ft/s} | 66 | 38 | 25 |
| | | {m/s} | 20 | 11.6 | 7.62 |
| u | normal component of the gust velocity | {m/s} | 15.590 | 9.832 | 6.283 |
| Δn | the incremental load factor | - | 1.827 | 0.356 | 0.443 |

*Table 3.5 Incremental load factor calculations for different conditions*

**Plot Gust V-n Diagram**

The maximum and minimum load factors during gust can be finally derived and are found by a high angle of attack to be (1+ 1.827 =) 2.8 as maximum and -0.82 as minimum. This means that this aircraft has a higher wing loading and a lower $C_{L_\alpha}$ values and is less sensitive to gust. This can be easily explained by the fact that a higher $C_{L_\alpha}$ or higher wing surface S (for a given aircraft weight) yield a higher $C_L$ with a certain angle of attack, and thus a larger increment of lift. By superimposing the maneuver and gust diagrams from section 0 the final V-n diagram can be generated as presented in Figure 3.4:

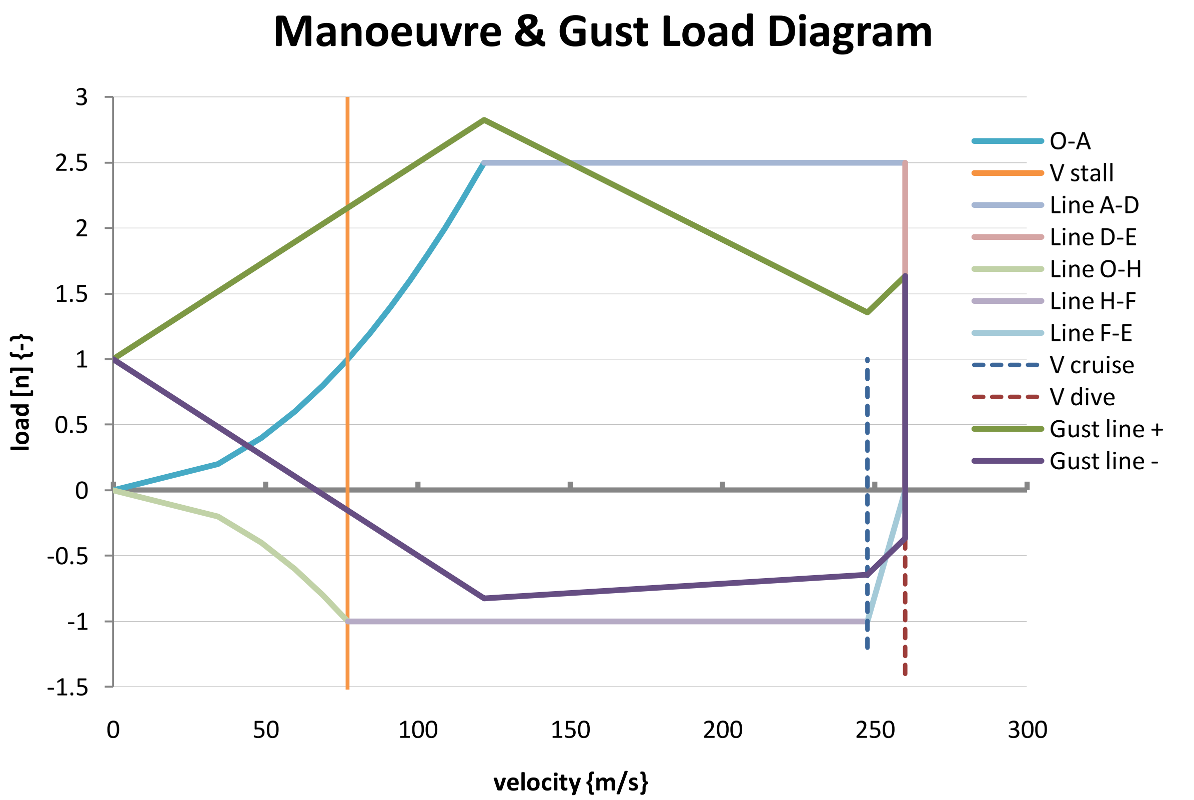



*Figure 3.4: Combined load diagram*

To obtain the load factors for the further design ($n_{design}$) the largest values in Figure 3.4 must be multiplied with the standard safety factor 1.5 which gives the ultimate design load factor. As it can be seen in Figure 3.4 and with eq.3.7, the design load factors obtained are 4.2 and -1.5.

$$n_{design} = 1.5 \cdot n_{max} = 1.5 * 2.8 = 4.2$$
$$n_{design} = 1.5 \cdot n_{max} = 1.5 * -1 = -1.5 \qquad \text{Eq.3.7}$$

# Chapter 4: OEW estimation using a class II method

Weight minimization in aircraft design is of the utmost importance. Now that most components of this aircraft have been scaled and designed since the first assignment, a more proper and close look at the weight of each section is necessary for a higher accuracy of the initial weight prediction. There are many methods to approach this, in this chapter, the operating empty weight and center of gravity position will be estimated, using methods described in a class II handbook. In this case, the Torenbeek method was used.

## The Torenbeek method

The Torenbeek weight prediction data & methods provides 9 weight groups for a conventional transport aircraft, presented in Table 4.1 together with their respective percentages of the take off weight:

| weight groups | % WTO |
|---|---|
| The airframe structure | - |
| The wing weight | 10-14 |
| The Tail group weight | 1.9-2.7 |
| The Fuselage body group: | 9-13 |
| The Alighting Gear group weight | 3.4-4.5 |
| The Surface controls group | 0.8-2.1 |
| The Engine nacelle group | 0.8-2 |
| The Propulsion group | 5-10 |
| The weight of the airframe services group | 8-11 |

*Table 4.1 The Torenbeek weight prediction of 9 weight groups*

Using the positive ultimate load $n_{ult}$ of 4.2 calculated in chapter 3, the weight prediction of each of these components can now be looked into more closely in terms of the takeoff weight and the estimated empty weight in assignment 2 of 99.4 tons and 52409 kg respectively. In the following subsections, this method will be briefly summarized.

## 4.1 The airframe structure

The airframe structure weight can be calculated using eq. 4.1.

$$\frac{W_s}{W_{to}} = k_s \sqrt{n_{ult}} \left( \frac{b_f h_f l_f}{W_{to}} \right)^{0.24} \qquad \text{Eq.4.1}$$

Where

| | | | |
|---|---|---|---|
| $W_s$ | Total airframe structure weight | | |
| $W_{to}$ | = Take off weight | 99.4 | Tons |
| $k_s$ | = structure | 0.447 | - |
| $b_f$ | = fuselage width | 4.16 | m |
| $h_f$ | = fuselage height | 4.16 | m |
| $l_f$ | = fuselage length | 42.5 | m |
| $n_{ult}$ | = ultimate load factor | 4.2 | - |
| **Total airframe structure weight** | | **28056** | **Kg** |

Table 4.2 The input parameters and group weight

## 4.2 The wing weight

The weight of the wing group can be calculated using eq 4.2:

$$\frac{W_W}{W_G} = k_w b_s^{0.75}\left\{1+\sqrt{\frac{b_{ref}}{b_s}}\right\} n_{ult}^{0.55}\left(\frac{b_s/t_r}{W_G/S}\right)^{0.30}\cdot f_{NE} \qquad \text{Eq.4.2}$$

Where

| | | | |
|---|---|---|---|
| $W_G$ | $W_{MZFW}$ (maximum zero-fuel weight) | 69512 | kg |
| $k_w$ | constant | 0.01 | - |
| $b_s$ | Structural Span | 41.01 | m |
| $b_{ref}$ | reference span | 37.70 | m |
| $t_r$ | root chord thickness | 0.98 | m |
| $W_G$ | Spoilers and speedbrakes factor | 1.02 | - |
| $f_{NE}$ | Reduction factor 2-engines mounted on wing | 0.95 | - |
| $S$ | Wingsurface | 158 | $m^2$ |
| **Total** | **Wing weight $W_W$** | **15504** | **kg** |

Table 4.3 The input parameters and group weight

## .3 The Tail group weight

According to Torenbeek, the Tail group weight is consists of the summation of the weights of the vertical and horizontal tails. First, for the **horizontal** tail:

$$\frac{W_h}{S_h} = k_h f\left(\frac{S_h^{0.2} v_D}{\sqrt{\cos(\wedge_h)}}\right) \qquad \text{Eq.4.3}$$

Where

| | | | |
|---|---|---|---|
| $k_h$ | 1 for fixed tail, 1.1 for variable incidence | 1.1 | - |
| $f\left(\frac{S_h^{0.2} v_D}{\sqrt{\cos(\wedge_h)}}\right)$ | | 29.50 | $Kg/m^3$ |
| **Total** | **Horizontal Tail weight** | **1460** | **kg** |

Table 4.4 The input parameters and horizontal tail weight

For the **vertical** Tail**,** eq 4.4 can be used:

$$\frac{W_v}{S_v} = k_v f\left(\frac{S_v^{0.2} v_D}{\sqrt{\cos(\wedge_v)}}\right) \qquad \textit{Eq.4.4}$$

Where

| | | | |
|---|---|---|---|
| $k_h$ | 1 for conventional tail | 1 | - |
| $f\left(\frac{S_v^{0.2} v_D}{\sqrt{\cos(\wedge_v)}}\right)$ | | 29 | Kg/m³ |
| **Total** | **Vertical Tail weight Wv** | **725** | **kg** |

Table 4.5 The input parameters and vertical tail weight

Then the total Tail group weight becomes 2185 kg.

## 4.4 Fuselage body group:

The weight of the fuselage body group is calculated as following:

$$W_f = k_{wf}\sqrt{v_D \frac{l_t}{b_f + h_f}} S_G^{1.2} \qquad \textit{Eq.4.5}$$

Where

| | | | |
|---|---|---|---|
| $k_{wf}$ | Fuselage constant | 0.23 | - |
| $l_t$ | 0.25MAC tail and wing distance | 18.601 | m |
| $\lambda$ | Fuselage fineness | 10.216 | - |
| $S_G$ | Gross Shell area | 485 | m² |
| | Calculated using $S_G = \pi b_f l_f \left(1-\frac{2}{\lambda}\right)^{(2/3)}\left(1+\frac{1}{\lambda^2}\right)$ and $\lambda = \frac{l_f}{b_f}$ | | |
| | Add 8 % for Pressurized cabin | 1.08 | - |
| $v_D$ | dive speed | 263 | kts |
| $b_f$ | Fusalage diameter | 4.1 | m |
| $h_f$ | Fuselage height | 4.1 | m |
| $l_f$ | Fuselage length | 33 | m |
| **Total** | **Fuselage (body) group weight** $W_f$ | **10055** | **kg** |

Table 4.6 The input parameters and group weight

Then eq. 4.5, gives 10055 kg for the Fuselage (body) group weight.

## 4.5 The Alighting Gear group weight

The Alighting Gear group weight consists of the weight of the main gear and of the front gear. This can be calculated using eq 4.6:

$$W_{uc} = k_{uc}\ \ A + B\cdot W_{to}^{3/4} + C\cdot W_{to} + D\cdot W_{to}^{3/2} \qquad \textit{Eq.4.6}$$

Where

| | | | |
|---|---|---|---|
| $k_{uc}$ | for low wing aircraft | 1.00 | - |
| A | For retractable Main gears | 18.1 | - |
| B | Coefficient Main gear | 0.131 | - |
| C | Coefficient Main gear | 0.019 | - |
| D | Coefficient Main gear | 0.0000223 | - |
| | Main gear | 3340 | kg |
| A | For retractable Front gears | 9.1 | - |
| B | Coefficient Front gear | 0.082 | - |
| C | Coefficient Front gear | 0 | - |
| D | Coefficient Front gear | 0.00000297 | - |
| | Front gear | 561 | kg |
| **Total Gear weight** | | **3901** | **kg** |

Table 4.7 The input parameters and group weight

Then the total Alighting Gear group weight becomes 3901 kg.

## 4.6 Surface controls group

Using eq. 4.7, an approximation can be made:

$$W_{sc} = k_{sc} W_{to}^{(2/3)} * L_{dumperSlat} \qquad Eq.4.7$$

With

| | | | |
|---|---|---|---|
| $k_{sc}$ | | 0.49152 | - |
| $L_{dumperSlat}$ | Lift dumper and slats | 1.35 | - |
| **Total** | **Surface control group** $W_{sc}$ | **1424** | **kg** |

Table 4.8 The input parameters and group weight

The Surface control group weight is found to be 1424 kg.

## 4.7 Engine nacelle group

In using the Torenbeek method, the weights of firewalls, splitter plattes and flaps and baffles are neglected since they are not relevant for this aircraft. Then the weight of the engine nacelle group consists of different parts as listed below.

| Weight Definition | Equation | Values [kg] |
|---|---|---|
| Weight Absorbers 5% $W_{eng}$ | $W_{eng\_absorbers} = 0.05 \cdot N_e \cdot W_e$ | 161 |
| Nacelle structrure | $W_{str\_nacelle} = 0.03\sqrt{(v_D \cdot 1.94)} S_{nacelle\_nacelle}$ | 650 |
| Pylons and struts | $W_{Pylons\_struts} = 0.405\sqrt{V_D} S_{wet\_nacelle}^{1.3}$ | 541 |
| Engines cowlings | $W_{eng\_cowling} = 0.9 S_{wet\_nacelle}$ | 35 |
| Gas generator | $W_{Gas\_gen} = 0.14 S_{wet\_nacelle}$ | 48 |
| Nacelle walls | $W_{nacelle\_wall} = 0.9 S_{wet\_nacelle}$ | 72 |
| Noise suppression | $W_{noise\_suppression} = 1.71 S_{wet\_nacelle}$ | 124 |
| Extra material weight | $W_{extra\_material} = 8.53 S_{wet\_nacelle}$ | 768 |

Table 4.9 The group components

Where the parameters are given as:

| Symbol | Definition | Equations | Values | Unit |
|---|---|---|---|---|
| $N_e$ | Number of engines | | 2 | - |
| $N_e$ | Dive velocity | | 260 | m/s |
| $S_{wet\_nacelle}$ | nacelle Wetted area | $S_{wet\_nacelle} = \pi D_{nacelle} l_{nacelle}$ | 35 | m2 |
| $l_{nacelle}$ | Length nacelle | | 4.5 | m |
| $D_{nacelle}$ | Diameter nacelle | | 2.5 | m |

Table 4.10 The input parameters

Then summing up all the weights, the total engine nacelle group weight becomes 2326 kg.

## 4.8 The propulsion group weight

The weight of the propulsion group consist also of different parts, including the weight of the two CFM56-5B1P installed turbofan engines, as listed below. Note again, that the weights of a water injection is neglected since they are not relevant for the selected engines. Then the weights of different components are defined as:

| Weight Definition | Equation | Values [kg] |
|---|---|---|
| Engine installation | $W_{eng\_install} = N_e \cdot W_{eng}$ | 4024 |
| Accessory gear boxes and drives | $W_{acc\_gearbox\_drives} = 0.0343 \cdot N_e \cdot W_{fto}$ | 81 |
| Tailpipes | $W_{tailpipes} = 14.63 \cdot D_t \cdot l_t \cdot \pi$ | 46 |
| Silencers | $W_{silencers} = 0.01 \cdot N_e \cdot T_{to}$ | 258 |
| Oil system and cooler | $W_{oil\_cooler\_system} = 0.01 \cdot N_e \cdot W_{engine}$ | 40 |
| Fuel system | $W_{fs} = 36.3(N_e + N_{ft}) + 4.366 N_{ft}^{0.5} V_{ft}^{0.333}$ | 376 |
| Thrust reversers | $W_{thrust\_reverser} = 0.18 \cdot N_e \cdot W_{engine}$ | 724 |

Table 4.11 The group components

Where the parameters are defined as follow:

| Symbol | Definition | Values | Units |
|---|---|---|---|
| $W_{fto}$ | Fuel flow/engine during TO | 428 | kg/sec |
| $D_t$ | Diameter pipe | 1 | m |
| $l_t$ | Length pipe | 1 | m |
| $T_{to}$ | Thrust at take-off | 126 | kN |
| $N_{ft}$ | Number of integral fuel tanks | 3 | - |
| $V_{ft}$ | Volume fuel tanks | 28797 | L |

Table 4.12 The input parameters

The total propulsion weight is then found to be 5550 kg.

## 4.9 The weight of the airframe services group

The weight of the airframe services group has different contributions are defined as following:

| Weights | Equations | Values [kg] |
|---|---|---|
| APU | $W_{APU} = 11.7 \cdot k_{APU} \cdot (w_{ba} \cdot l_{cabin})^{3/5}$ | 126 |
| Instruments and electronics group | $W_{ieg} = k_{ieg} \cdot W_{DE}^{\ 5/9} \cdot R_D^{\ 1/4}$ | 1274 |
| Triplex hydraulics | $W_{hydr} = 0.015 \cdot W_{DE} + 272$ | 1053 |
| Electrical system | $W_{el} = 16.3 \cdot P_{el} \;\; 1 - 0.033 \cdot \sqrt{P_{el}}$ | 1817 |
| Fournishing equip. | $W_{fournishing} = 0.196 \cdot W_{zf}^{\ 0.91}$ | 4995 |
| Airco and anti-ice | $W_{airco\_anti\_icing} = 14.0 \cdot l_{cabin}^{\ 1.28} - l_{cabin}$ | 1197 |
| Miscellaneous | $W_{miscellaneous} = 0.01 \cdot W_{DE}$ | 260 |
| Crew previsions | $W_{crew\_previsions} = 93 \cdot N_{fc} + 68 \cdot N_{CC}$ | 458 |
| Computers | $W_{Compters} = 0.453 \cdot pax$ | 72 |
| Snacks | $W_{snacks} = 2.27 \cdot pax$ | 359 |
| Main meal | $W_{main\_meal} = 6.35 \cdot pax$ | 1003 |
| Portable water and WC | $W_{portable\_water/chemicals} = 1.36 \cdot pax$ | 215 |
| Safety requirements | $W_{safety\_requirements} = 3.4 \cdot pax$ | 537 |
| Residual fuel | $W_{residual\_fuel} = 0.151 \cdot V_{ft}^{\ 2/3}$ | 145 |

Table 4.13 The group components

Where the parameters are defined as follows:

| Symbol | Definition | Formula | Values | Unit |
|---|---|---|---|---|
| $W_{DE}$ | Aircraft delivery weight | | 52084 | kg |
| $k_{APU}$ | Installation factor (2…… 2,5) | | 2.000 | |
| $w_{ba}$ | The bleed airflow for high density layout | | 0.500 | kg/min/m3 |
| $l_{cabin}$ | Cabin length | | 33 | m |
| $k_{ieg}$ | Constant | | 0.347 | - |
| $R_D$ | The maximum range | | 6000 | km |
| $V_{pc}$ | passenger cabin volume | $V_{pc} = \pi \cdot \frac{D_f^{\ 2}}{4} \cdot l_f$ | 578 | m3 |
| $D_f$ | Diameter passenger cabin | | 4.1 | m |
| $l_f$ | Length passenger cabin | | 26 | m |
| $P_{el}$ | electrical power | $P_{el} = 3.64 \cdot V_{pc}^{\ 0.7}$ | 217 | - |
| $N_{fc}$ | Number flight crew | | 2 | |
| $N_{CC}$ | Number cabin crew members | | 4 | |
| $pax$ | Number passengers | | 158 | |
| $V_{ft}$ | Mission Fuel Weight | | 29918 | kg |

Table 4.14 The input parameters

Then the total airframe services group weight becomes 10809 kg.

In Table 4.2, the weights of the 9 components of Torenbeek weights prediction method are summarized with their percentage contributions to the takeoff weight. Organized from large to small contribution, and with normal percentages to compare the weight contribution.

| **Components** | **Weight [kg]** | **% Wto** | **Normal % Wto** |
|---|---|---|---|
| Wing weight | 15504 | 15.59% | 10-14% |
| Airframe Services Group | 10809 | 10.87% | 11 - 13,6% |
| Fuselage (body) group weight | 10055 | 10.11% | 9 -13% |
| Propulsion | 5550 | 5.58% | 10% |
| Main gear | 3340 | 3.36% | |
| Nacelle engine group | 2326 | 2.34% | 1,3 - 2,1% |
| Surface control group | 1424 | 2.20% | 1,4 - 2,1% |
| Horizontal Tail | 1460 | 1.47% | |
| Vertical Tail | 725 | 0.73% | |
| Front gear | 561 | 0.56% | |

*Table 4.15 Summarization of previous obtained values and normal WTO percentages*

As can be seen from the table, the wing weight is high compared to the normal percentages. The airframe services, fuselage body group,nacelle group and surface control group are within good range of normal values. The propulsion group is, however, low for our aircraft. The gear summed gear weight is 3.92% of the Wto, and this value is good, since it is between the recommended 3.5 and 4.5 percent. The same outcome is achieved when looking to the tail group: the horizontal and vertical tail combined is 2.2% of the Wto, where it should be somewhere between 1.9 and 2.7%. So, except from the high wing weight, and low engine weight, this aircraft is in good ranges, with respect to the key group weights. Now. Summing the values and comparing to the Weight class 1 estimation gives:

| **Total empty weight class II** | **51754** | **Kg** |
|---|---|---|
| Total empty weight class I | 52408 | Kg |
| Difference | 655 | Kg or 1.25%Wto |

*Table 4.16: Comparison of class I and class II weight estimation values*

The total empty weight of the aircraft is only 655 kg lighter than the empty weight obtained with the class I estimation method. Since there is only a 1.25 % difference, one can neglect this difference and continue with the values of the previous weight estimation.

# Chapter 5: Center of Gravity Estimation

In this chapter, the location of the centre of gravity will be discussed. This will be done using all the weight groups that have been calculated in chapter 4. When loading the aircraft with pilots, passengers, luggage and fuel the location of the centre of gravity will change a certain range. The travel of the location of the centre of gravity has to be determined for further designing of the gear and tail. This will be done using a loading diagram.

## 5.1 The empty aircraft

To find the CoG of the aircraft, it is divided in components weights in two categories: the fuselage and wing group. This is shown in table. 5.1 and 5.2 where the weight of the fuselage group consists of the weights of the airframes services, the fuselage landing gear, the fuselage and of the tail. The distance to the front of the aircraft is also indicated, since that is required to find the cg.

$$COG = \frac{\sum W_{element} \cdot COG_{element}}{\sum W_{element}} \qquad \text{Eq.5.1}$$

Using the simplification that the aircraft is symmetric, and that all elements are on the longitudinal axis, together with equation 5.1, gives that the Center of gravity of the Body group is 20.499 meters from the nose.

| **Fuselage Group** | Weight [kg] | $X_{front}$ [m] |
|---|---|---|
| Fuselage (1) | 10055 | 21.25 |
| Airframe services (2) | 10809 | 17.00 |
| Horizontal tail (3) | 1460 | 39.00 |
| Vertical tail (4) | 725 | 37.00 |
| Nose landing gear (5) | 561 | 5.00 |

Table 5.1: The components weights of the fuselage group

The total mass of the body group is 23610kg. All distances above are derived from the technical drawings that were made up in earlier assignments.

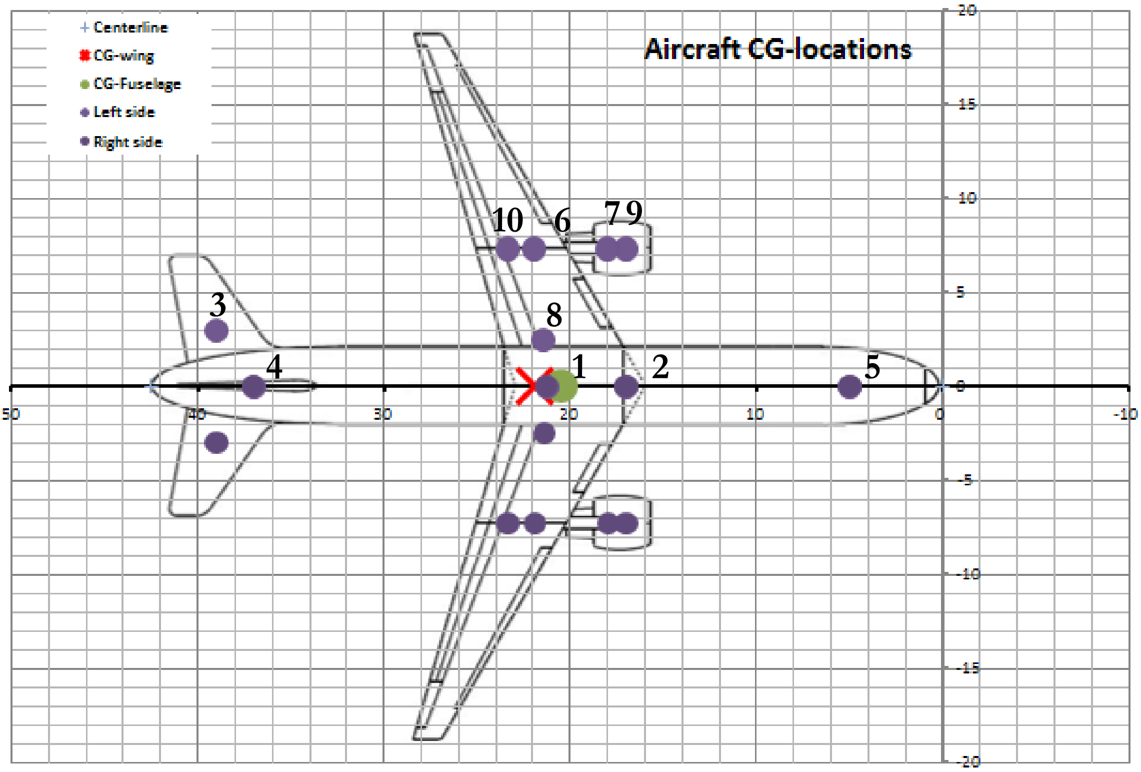



Figure 5.1: Body group contributions to the COG (axes show distance in m)

As a visual aid, the body group is represented in the image above, with the most contributing points on the aircraft.

The second group is the wing group. The elements in this group are mainly the wing itself, the propulsion, the nacelles and main landing-gear. The total mass of the wing group is 28144kg, the individual components are shown below.

| **Wing Group** | Weight [kg] | $X_{front}$ [m] |
|---|---|---|
| Wing (6) | 15504 | 21.91 |
| Propulsion (7) | 5550 | 17.00 |
| Main landing gear (8) | 3340 | 22.5 |
| Nacelle (9) | 2326 | 18.00 |
| Surface controls (10) | 1424 | 23.35 |

*Table 5.2: The components weights of the wing group*

Again using equation 5.1 and figure 5.1, the COG can be determined to be 21.91m from the front of the aircraft.

The COG of the whole aircraft is then, again using eq 5.1, located at 21.266m from the nose of the aircraft. From the drawings, it is found that the $X_{lemac}$ is 19.5m from the front of the nose. The ratio of the leading edge of the MAC over the length of the fuselage $\frac{x_{LeMAC}}{L_{fus}}$ is then 0.46. The COG position can thus be rewritten in terms of $X_{lemac}$: the Empty weight CoG is then at 36.35%* MAC past the $X_{lemac}$, or the CoG is at 36.35%$X_{lemac}$/MAC. This, together with the empty-weight of 51754kg, as earlier found using the Class 2 method, is the starting point of the Loading diagram.

## 5.2 The loading diagram

In this section the change in centre of gravity position of the aircraft due to loading of passengers (and pilots), luggage and fuel will be considered. The pilots will be counted as part of the passengers. The plot will consist of 4 loading contributions (in the following order): loading of the cargo pallets, loading of the window-seat passengers, loading of the middle-seat passengers, loading of the aisle seat passengers and finally fueling the aircraft, using the center tank, and the wing-tanks.

Using the previously determined starting point, and adding per row of the aircraft 160kg for 2 passengers, or 210 kg for a loaded cargo pallet (the mass of one loaded pallet was determined in assignment 1, and the mass in the fuel tanks in assignment 4). For completeness, the aircraft is loaded in the full economy position, seating 190 passengers, and holding 18 cargo pallets. In a similar method as the one described above for each fuselage or wing element, the weight is added, and the new cog position is calculated for the different steps in the loading process.

Since the process is trivial, it will not be explained further. The following image shows the loading of the aircraft, with the 5 contributions described above, in 2 loading conditions (front to back, and back to front). This gives 10 lines. Along which the COG travels. See figure 5.2. Note that the X-scale is set in $X_{lemac}$/MAC, as described above, and that the Y-axis is in kg.

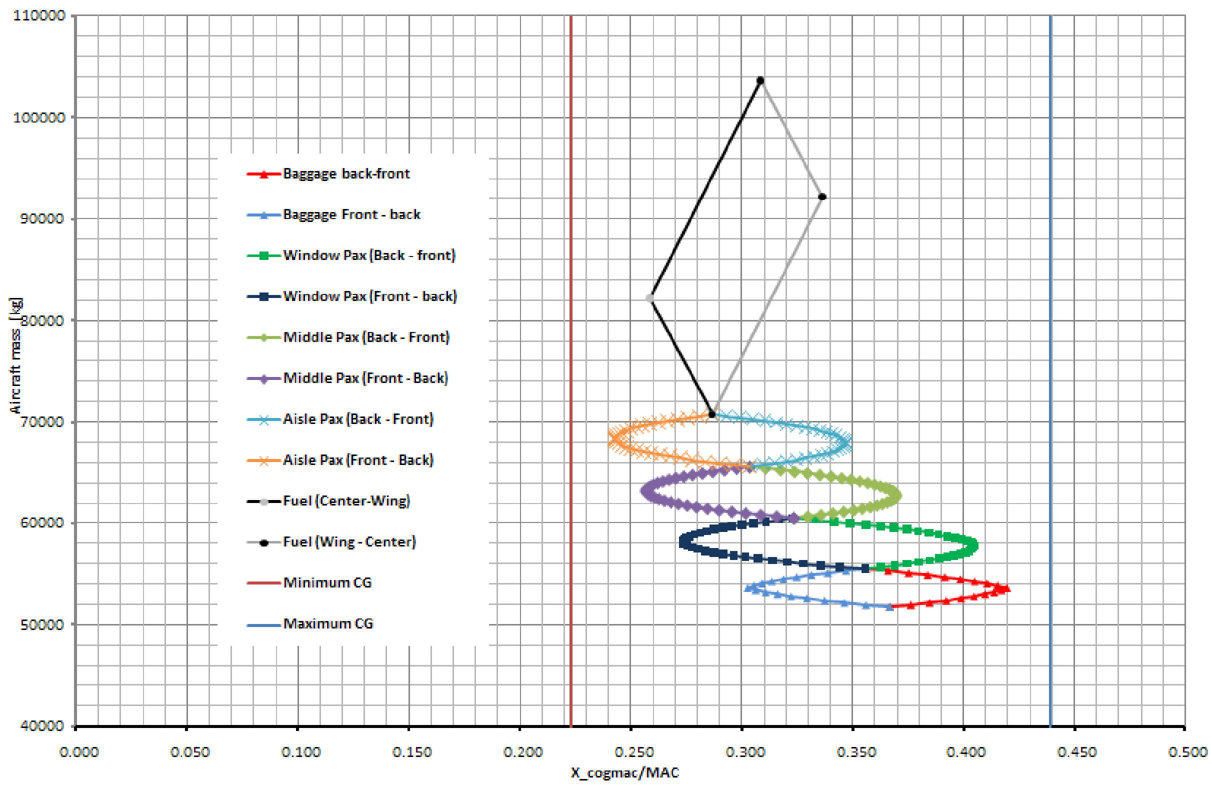



*Figure 5.2: Loading diagram of initial position*

The maximum and the minimum location of the centre of gravity can be derived from the figure above. However, a 2% margin is added to account for the centre of gravity variations caused by the passengers and the attendants moving, landing gear retracting, food and drinks served etc.

This gives us a minimum CG of 0.22 and a maximum CG of 0.44 for this wing position and a COG range of 0.22 MAC. The gear position has been put at 22.5m, or 0.62 $X_{gear}$/MAC. Using the maximum aft cg position of 0.44, this gives a difference of 18.3%MAC between the max aft cg and the gear position. This value is larger than the minimum required of 15%, so the gear is at the correct position.

## 5.3 The loading diagram with 2 new wing positions

To determine the optimal wing position, the wings are shifted 5 meter to the front, and 5 meter to the rear, creating 2 new Loading conditions. Following similar procedure as in chapter 5.2, the 2 new loading conditions are shown below. (the legend is the same as figure 5.2).

Assembling the characteristics of these plots in one table, gives us table 5.3:

| | X_cogLEMAC/MAC | | |
|---|---|---|---|
| Xlemac/lfus | 0.5760 | 0.4590 | 0.3410 |
| Min cog | -0.3888 | 0.2228 | 0.7148 |
| Max cog | -0.0539 | 0.4394 | 0.9761 |
| Xlemac | 24.5 | 19.5 | 14.5 |
| Cog - Range | 0.3350 | 0.2170 | 0.2610 |

*Table 5.3 The centre of gravity shift for different positions*

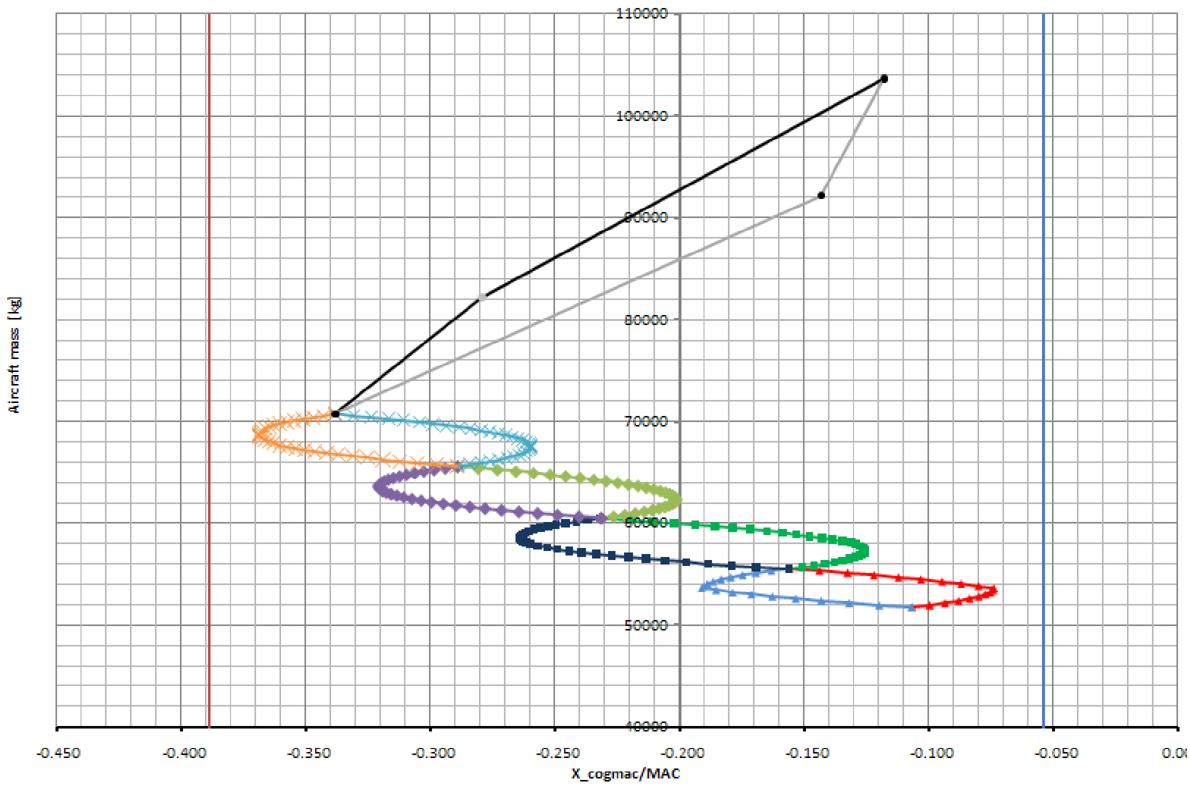



*Figure 5.3: Loading diagram with the wing shifted forward 5m*

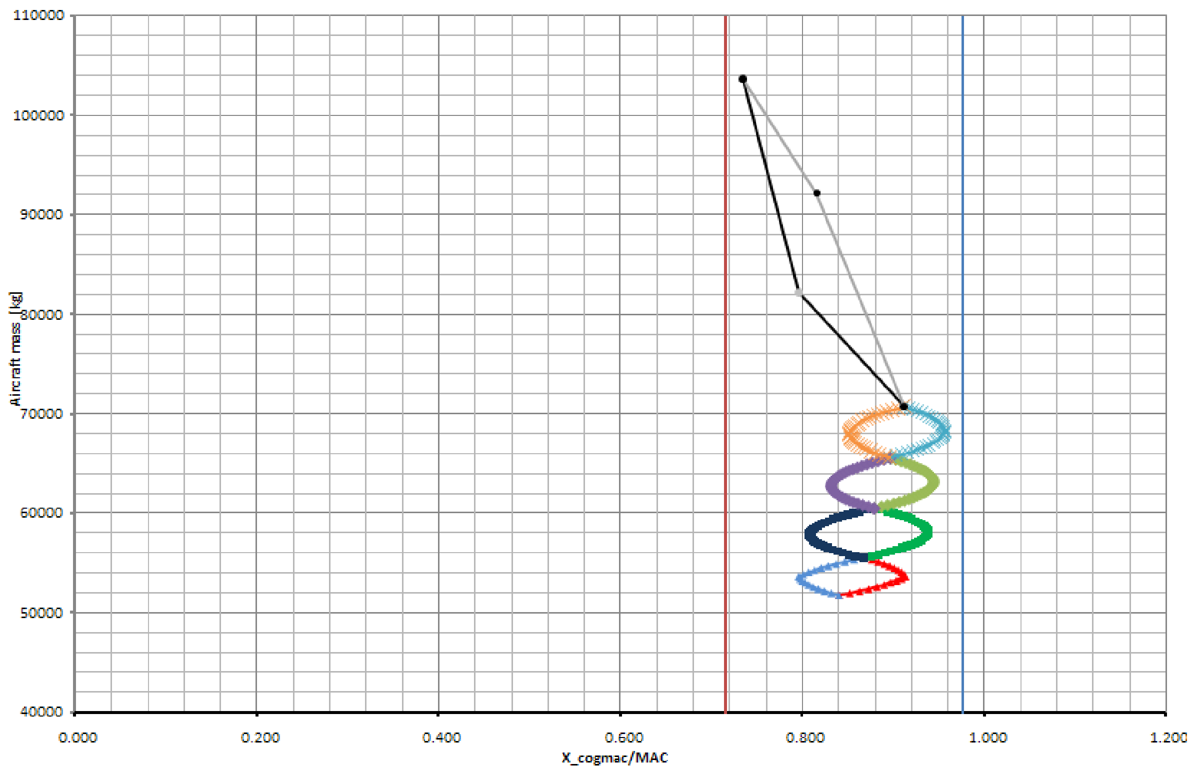



*Figure 5.4: Loading diagram with the wing shifted backwards 5m.*

From table 5.3, figure 5.5 can be made.

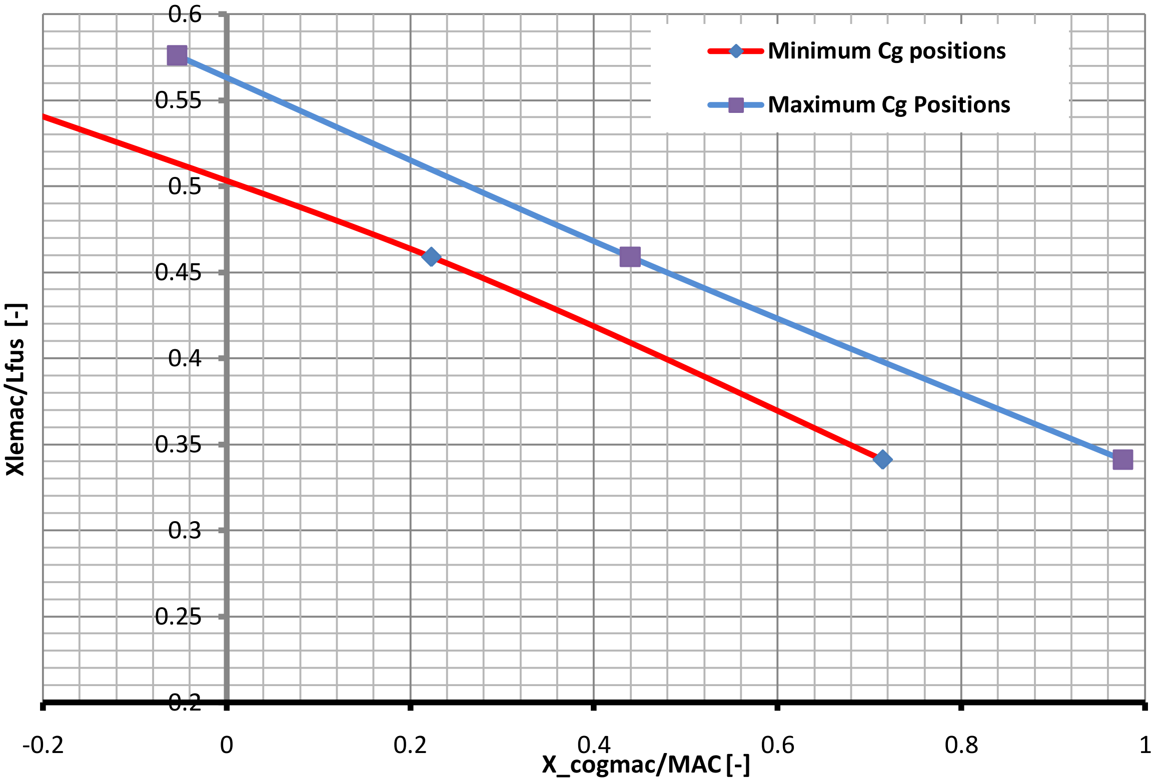



*Figure 5.5: The wing effect of wing shift on centre of gravity travel*

# Chapter 6: The scissor plot

In this chapter the tail section will be evaluated more closely since the centre of gravity is now defined more properly.

## 6.1 The stability Curve

A stick-fixed neutral point is assumed for the rear centre of gravity. This stick-fixed stability is described with Equation 6.1.

$$\bar{x}_{np} = \bar{x}_{ac} + \frac{C_{L\alpha_h}}{C_{L\alpha}}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\frac{S_h l_h}{S\bar{c}}\left(\frac{V_h}{V}\right)^2 \qquad Eq.6.1$$

This equation investigates the relation between the horizontal tail size and the aerodynamic centre position to evaluate the neutral point. Many variables have their own formulas and will be elaborated.

The first variable $x_{ac}$-bar, in Eq. 6.1, represents the aerodynamic centre of the aircraft minus the tail, so far the aerodynamic centre of the wing has been calculated yet the addition of the fuselage is needed. The value $C_{L\alpha h}$ is the lift-rate coefficient of the horizontal tail itself, while $C_{L\alpha}$ is the final lift-rate coefficient of the total aircraft shown in Equation 6.2. The variable $d\varepsilon/d\alpha$ is the downwash effect of the wing on the tail, while $(V_h/V)^2$ is the ratio of the tail to wing speed squared, which in this case is 0.85.

$$C_{L\alpha} = C_{L\alpha_{A-h}} + C_{L\alpha_h} \frac{S_h}{S}\left(1 - \frac{d\varepsilon}{d\alpha}\right)\left(\frac{V_h}{V}\right)^2 \qquad Eq.6.2$$

### 6.1.1. The aircraft Lift coefficient without the tail

To find the final lift coefficient of the total aircraft $C_{L\alpha}$, some new variables are needed, $C_{L\alpha(A\text{-}h)}$ is the lift-rate coefficient of the aircraft without the tail. This can be found using using Equation 6.3.

$$C_{L\alpha_{A-h}} = C_{L\alpha_{wf}} = C_{L\alpha_w}\left(1 + 2.15\frac{b_f}{b}\right)\frac{S_{net}}{S} + \frac{\pi}{2}\frac{b_f^2}{S} \qquad Eq.6.3$$

The lift-rate coefficient of the aircraft without the tail $C_{L\alpha(A\text{-}h)}$ or lift-rate coefficient of the wing fuselage system $C_{L\alpha(wf)}$ needs the variable $b_f$ which is fuselage width which still remains 4.1 m. Also $S_{net}$ is needed which is the equal to the main wing surface area S minus the projection of the central wing part inside the fuselage and comes out to 128 m$^2$. The last variable is $C_{L\alpha(w)}$ which is the main wing lift-rate coefficient at high speed, this can be calculated according to Assignment 4 which comes is 5.622 $C_L$/rad.

| | |
|---|---|
| $C_{L\alpha w}$ | 5.622 1/rad |
| $b_f$ | 4.16 m |
| $b$ | 37.7 m |
| $S_{net}$ | 128.825 m$^2$ |
| $S$ | 157.9 m$^2$ |

*Table 6.1: Variables used to calculate the lift coefficient without tail*

Thus when using all values, the lift-rate coefficient of the wing fuselage system $C_{L\alpha(wf)}$ or $C_{L\alpha(A\text{-}h)}$ is equal to 5.845 $C_L$/rad.

### 6.1.2. The wing downwash gradient effect

Now this value can be used back into Equation Eq. 6.2 once some other variables are calculated, first with the downwash dε/dα which is described in Equation Eq. 6.4.

$$\frac{d\varepsilon}{d\alpha} = \frac{K_{\varepsilon_\Lambda}}{K_{\varepsilon_{\Lambda=0}}}\left(\frac{r}{r^2+m_{tv}^2}\frac{0.4876}{\sqrt{r^2+0.6319+m_{tv}^2}} + \left[1+\left\{\frac{r^2}{r^2+0.7915+5.0734m_{tv}^2}\right\}^{0.3113}\right]\left[1-\sqrt{\frac{m_{tv}^2}{1+m_{tv}^2}}\right]\right)\frac{C_{L_{\alpha w}}}{\pi A} \qquad Eq.6.4$$

The downwash dε/dα is also dependent on variable $K_{\varepsilon(\Lambda)}$ and $K_{\varepsilon(\Lambda=0)}$ which is given by Equation Eq. 6.5 and Eq. 6.6 and r which is given by Equation 2.7.

$$K_{\varepsilon_\Lambda} = \frac{0.1124+0.1265\Lambda+0.1766\Lambda^2}{r^2} + \frac{0.1024}{r} + 2 \qquad Eq.6.5$$

$$K_{\varepsilon_{\Lambda=0}} = \frac{0.1124}{r^2} + \frac{0.1024}{r} + 2 \qquad Eq.6.6$$

$$r = \frac{2L_h}{b} \qquad Eq.6.7$$

The variable r is dependent on the distance between the main wing an horizontal tail aerodynamic centers which is estimated at around 18.601 m, which makes r equal to 0.987. This makes $K_{\varepsilon(\Lambda)}$ equal to 2.337 and $K_{\varepsilon(\Lambda=0)}$ equal to 2.219. Now looking back to Equation Eq. 6.4, all it needs is $m_{tv}$ which is twice vertical distance between horizontal tail and the main wing (3.6m) devided by the wingspan (37.7m), and is found to be 0.191.

| | |
|---|---|
| $K_{\varepsilon_\Lambda}$ | 2.337 |
| $K_{\varepsilon_{\Lambda=0}}$ | 2.219 |
| $r$ | 0.987 |
| $m_{tv}$ | 0.191 |
| $A$ | 9 |

*Table 6.2: Variables used to calculate wing downwash gradient effect*

The same main wing lift-rate coefficient at high speed $C_{L\alpha(w)}$ can be used here. This finally leads to the downwash coefficient $d\varepsilon/d\alpha$ which is then equal to 0.38.

### 6.1.3. The lift-rate coefficient of the horizontal tail

For Equation Eq. 6.2, the lift-rate coefficient of the horizontal tail $C_{L\alpha h}$ also needs to be calculated and is given by Equation 6.8 and 6.9.

$$C_{L\alpha_h} = \frac{2\pi A_h}{2+\sqrt{4+\left(\frac{A_h\beta}{\eta}\right)^2\left(1+\left[\frac{\tan^2\Lambda_{0.5c_h}}{\beta^2}\right]\right)}} \qquad Eq.6.8$$

$$\beta = \sqrt{1-M_h{}^2} \qquad Eq.6.9$$

The Prandtl-Glauert compressibility correction factor β in Equation Eq. 6.9 is equal to 0.795, because the Mh is multiplied by $(Vh/V)^2$ (=0.85). The factor η equal to 0.95 for Equation Eq. 6.9. The only variable left for Equation Eq. 6.8 is $\Lambda_{0.5ch}$ which is the sweep angle at half chord of the horizontal tail which is calculated as 0.523 rad.

This formulates the lift-rate coefficient of the horizontal tail $C_{L\alpha h}$ to 5.845 $C_L$/rad.

Using equation 6.2,the final lift coefficient of the total aircraft $C_{L\alpha}$ turns out to be 6.445 1/rad, using all the previously calculated variables.

### 6.1.4 The aerodynamic centre of the aircraft without the tail

Next, the aerodynamic centre of the aircraft without the tail needs to be determined. This is done using three steps. Firstly, the contribution of the aerodynamic centre of wing and fuselage systems need to be determined, which is different than the position of the wing aerodynamic centre alone. Then the contribution of the nacelles is calculated and finally the contribution of the jet stream is determined.

The aerodynamic centre of the wing minus the tail $x_{ac}$-bar is given by Equation Eq. 6.10.

$$\bar{x}_{ac} = \bar{x}_{ac\ wf} + \bar{x}_{ac\ n} + \bar{x}_{ac\ T} \qquad Eq.6.10$$

This variable is split up into the aerodynamic centre of the wing and fuselage $(x_{ac}\text{-bar})_{wf}$, the nacelle contribution $(x_{ac}\text{-bar})_n$ and the jet stream contribution $(x_{ac}\text{-bar})_T$.

A: Wing & fuselage contribution

In order to find ($x_{ac}$-bar)$_{wf}$, Equation Eq. 6.11 is needed.

$$\left(\frac{x_{ac}}{\bar{c}}\right)_{wf} = \left(\frac{x_{ac}}{\bar{c}}\right)_{w} - \frac{1.8}{C_{L_{\alpha wf}}}\frac{b_f h_f l_{fn}}{S c_{MAC}} + \frac{0.273}{1+\lambda}\frac{b_f c_{av}}{c_{MAC}^2}\frac{b-b_f}{b+2.15 b_f}\tan\Lambda_{0.25c} \qquad \textit{Eq.6.11}$$

This equation needs the aerodynamic centre of the main wing ($x_{ac}$-bar)$_w$ which is calculated using Figure 6.1. Since the sweep of the main wing Λ is 30°, the variable β from Figure 6.1 makes $\Lambda_\beta$ equal to around 40°. The taper ratio of the main wing λ is 0.2 and the 3[d] graph should be used since the aspect ratio of the main wing A times β is closest to 6, (namely 6.34).

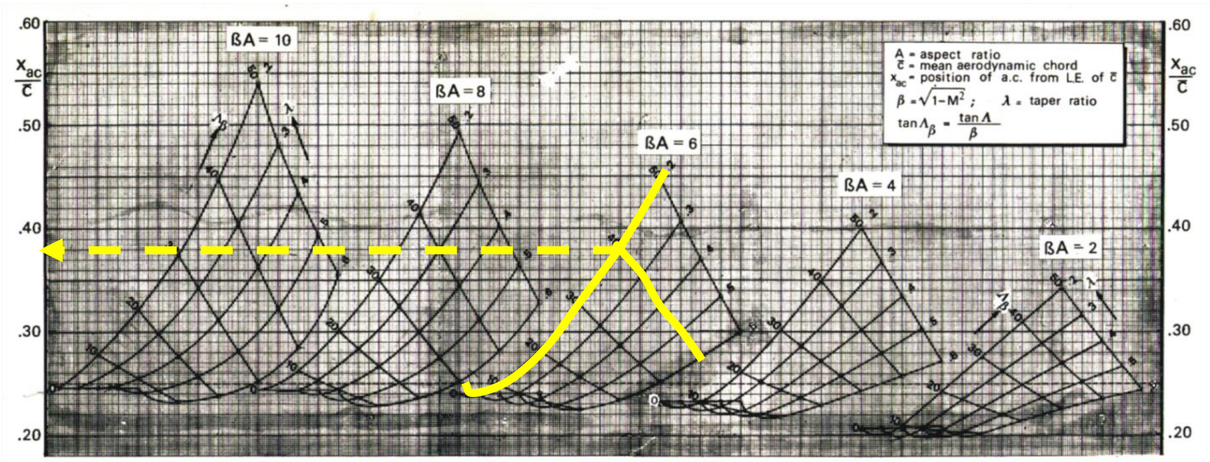



Fig. E-10. Aerodynamic center of lifting surfaces at subsonic speeds (Ref. E-31).

*Figure 6.1: Finding the aerodynamic centre of the main wing*

This means that from the graph the aerodynamic centre of the main wing ($x_{ac}$-bar)$_w$ is around 0.38. The height of the fuselage $h_f$ is also needed and is equal to 4.16 m. The lift-rate coefficient of the fuselage $C_{L\alpha(wf)}$ is the same as previously mentioned. The variable $l_{fn}$ is the distance from the front of the aircraft to the leading edge of the wing that is not attached to the fuselage and is equal to 16.5 m. The main wing sweep angle at quarter chord $\Lambda_{0.25c}$ is equal to 0.47 rad. The geometric average chord $c_{av}$ is equal to 4.18 m. A summary of variables used in this calculation is given in table 6.3.

**Wing Fuselage contribution**

| | |
|---|---|
| $C_{L_{\alpha wf}}$ | 5.845 1/rad |
| $b_f$ | 4.16 m |
| $h_f$ | 4.16 m |
| $l_{fn}$ | 16.1 m |
| $S$ | 157.9 m2 |
| $\lambda$ | 0.20 |
| $\bar{c}$ | 4.818 m |
| $c_g$ | 4.18 m |
| $b$ | 37.7 m |
| $\Lambda_{1/4}$ | 0.47 rad |

*Table 6.3. Variables used to calculate the aerodynamic centre of the w-f contribution*

This makes the fuselage and wing aerodynamic centre ($x_{ac}$-bar)$_{wf}$ equal to $\bar{x}_{ac\ \ wf}$ = 32.635% MAC.

B: Nacelle contribution

The nacelle have a negative contribution as demonstrated with Equation Eq. 6.12.

$$\bar{x}_{ac\ \ n} = \sum \frac{k_n b_n^2 l_n}{S\bar{c}C_{L_{\alpha wf}}} \qquad \text{Eq.6.12}$$

There are 2 engines (thus this formula is multiplied by 2 for its final contribution), the nacelle distribution factor $k_n$ is found to be 4 for wing mounted engines using Figure 6.2. The nacelle diameter of this aircraft is $b_n$ = 2.25 m and the distance from the nacelle front to quarter MAC Ln is 5.70 m (this is for all 2 engines and is found using Ln =4.5+0.25MAC). Used variable are summarized in Table 6.2.2.

$$\Delta_n \frac{x_{ac}}{\bar{c}} = \sum k_n \frac{b_n^2 l_n}{S\bar{c}\left(C_{L_\alpha}\right)_{wf}} \quad (C_{L_\alpha} \text{ in rad}^{-1}) \quad \text{(E-41)}$$

where $k_n \sim -4.0$ for nacelles mounted in front of the LE of the wing or in the fuselage nose

$k_n \sim -2.5$ for jet engine pods mounted to the sides of the rear fuselage.

Figure 6.2: Kn values for wing mounted engines

| | |
|---|---|
| $k_n$ | -4 |
| $b_n$ | 2.25 m |
| $l_n$ | 5.70 m |
| S | 158 m$^2$ |
| $\bar{c}$ | 4.818 m |
| $C_{L_{\alpha wf}}$ | 5.85 1/rad |

Table 6.4: Variables used to calculate the nacelles contribution

This makes ($x_{ac}$-bar)$_n$ equal to -0.0519 %MAC with $C_{L\alpha h}$ is 5.85 $C_L$/rad. (Note the negative sign since it is a negative contribution).

C: Jet stream contribution

The jet stream contribution ($x_{ac}$-bar)$_T$ applies only to jet engines and can be calculated using Eq. 6.13.

$$\left(\frac{X_{ac}}{\bar{c}}\right)_T = -\sum_{1}^{N} \frac{T}{W}\frac{z_T}{\bar{c}} \qquad \text{Eq.6.13}$$

The thrust to weight ratio T/W is calculated previously in assignment 2 which has a value of 0.26. A value of 1.5 m is chosen for $z_T$ and the value for the MAC is equal to 4.818 m. Therefore, the jet stream contribution gives a negative contribution of -0.162.

D: The aerodynamic Center.

Revaluating Equation Eq. 6.10, the aerodynamic centre $x_{ac}$-bar is equal to $\bar{x}_{ac}$ = 0.115 or 11.53% MAC.

Since all the variables of equation 6.1 are known, the neutral point graph can now be computed. Rewriting this equation gives:

$$\frac{S_h}{S} = \frac{1}{\left[\frac{C_{L\alpha_h}}{C_{L\alpha}}\left(1-\frac{d\varepsilon}{d\alpha}\right)\frac{l_h}{\bar{c}}\left(\frac{V_h}{V}\right)^2\right]}\bar{x}_{cg} - \frac{\bar{x}_{ac}}{\frac{C_{L\alpha_h}}{C_{L\alpha}}\left(1-\frac{d\varepsilon}{d\alpha}\right)\frac{l_h}{\bar{c}}\left(\frac{V_h}{V}\right)^2} \quad \textit{Eq.6.14}$$

Of which all parameters are known. Filling in these values gives the Neutral point line. However, for uncertainties, it is best if an additional 5% MAC is subtracted from these values, such that $\bar{x}_{np} - \bar{x}_{cg} = 0.05$. This gives the stability curve, given by following equation:

$$\frac{S_h}{S} = \frac{1}{\left[\frac{C_{L\alpha_h}}{C_{L\alpha}}\left(1-\frac{d\varepsilon}{d\alpha}\right)\frac{l_h}{\bar{c}}\left(\frac{V_h}{V}\right)^2\right]}\bar{x}_{cg} - \frac{\bar{x}_{ac}-0.05}{\frac{C_{L\alpha_h}}{C_{L\alpha}}\left(1-\frac{d\varepsilon}{d\alpha}\right)\frac{l_h}{\bar{c}}\left(\frac{V_h}{V}\right)^2} \quad \textit{Eq.6.15}$$

Now it is possible to evaluate this equation and plot the position of the centre of gravity with reference to the surface ratio $S_h/S$. This is graphed in figure 6.6 further in this chapter. Note that the stick fixed neutral point is the just the rear centre of gravity limit with an added safety margin of 5% MAC.

## 6.2 Forward control limit

After examining the stability of the aircraft, the next thing to do is to determine the controllability of the aircraft. This can be found using equation 6.16.

$$\frac{S_h}{S} = \frac{1}{\frac{C_{L_h}}{C_{L_{A-h}}}\frac{l_h}{\bar{c}}\left(\frac{V_h}{V}\right)^2}\bar{x}_{cg} + \frac{\frac{C_{m_{ac}}}{C_{L_{A-h}}} - \bar{x}_{ac}}{\frac{C_{L_h}}{C_{L_{A-h}}}\frac{l_h}{\bar{c}}\left(\frac{V_h}{V}\right)^2} \quad \textit{Eq.6.16}$$

For this, the zero lift pitching moment coefficient has to be determined.

### 6.3.1. The zero lift pitching moment coefficient

Equation Eq. 6.17, Eq. 6.18, Eq. 6.19 and Eq. 6.20 are the formulas for the zero pitching moment which are needed in order to find the controllability. The controllability is based on the forward centre of gravity limit. The value of $C_{mACw}$ is -0.1 while $\Delta_{nac}C_{mAC}$ is -0.05 for wing-mounted engines.

$$C_{m_{ac}} = C_{m_{ac_w}} + \Delta_f C_{m_{ac}} + \Delta_{fus} C_{m_{ac}} + \Delta_{nac} C_{m_{ac}} \quad \textit{Eq.6.17}$$

$$\Delta_{fus} C_{m_{ac}} = -1.8\left(1-\frac{2.5b_f}{l_f}\right)\frac{\pi b_f h_f l_f}{4S\bar{c}}\frac{C_{L_0}}{C_{L_{\alpha wf}}} \quad \textit{Eq.6.18}$$

$$\Delta_f C_{m_{0.25MAC}} = \mu_2 \left\{ -\left[ \mu_1 \Delta C_{l\max} \frac{c'}{c} \right] - \left[ C_L + \Delta C_{l\max} \left( 1 - \frac{S_{wf}}{S} \right) \right] \frac{1}{8} \frac{c'}{c} \left( \frac{c'}{c} - 1 \right) \right\} + 0.7 \frac{A}{1 + 2/A} \mu_3 \Delta C_{l\max} \tan \Lambda_{0.25MAC} \qquad Eq.6.19$$

$$C_{m0.25c} = C_{mAC} + C_L \left( 0.25 - \frac{x_{ac}}{\bar{c}} \right) \qquad Eq.6.20$$

First the variables $\mu_1$, $\mu_2$ and $\mu_3$ need to be calculated using, Figure 6.2. figure 6.3 and figure 6.4 the required values for these plots were found in assignment 4.

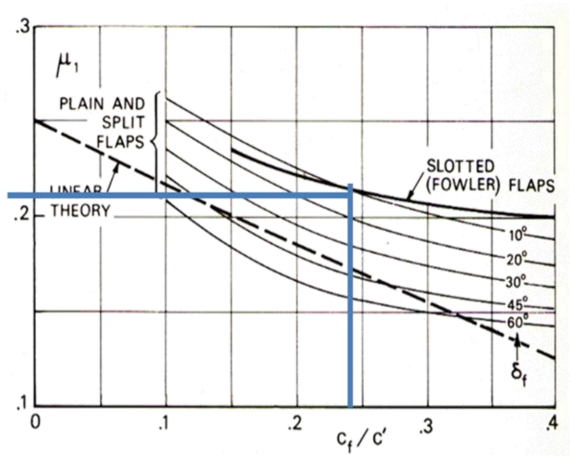



Figure 6.2: calculation of µ1

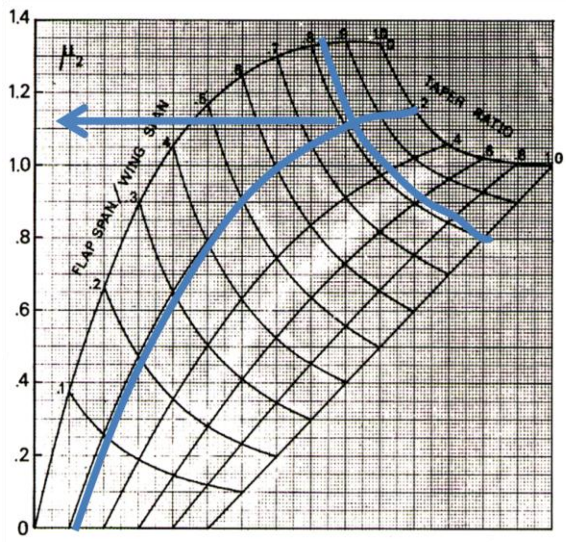



Figure 6.3: calculation of µ2

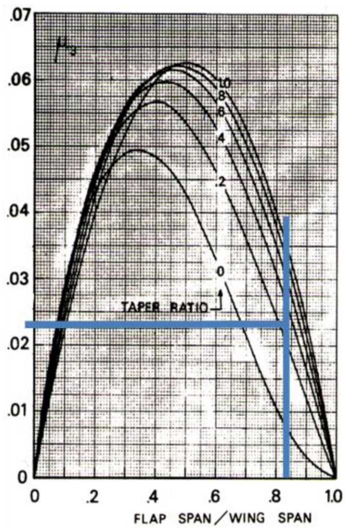



Figure 6.4: calculation of µ3

The values for $\mu_1$, $\mu_2$ and $\mu_3$ are found to be 0.22, 1.1 and 0.022 respectively. The zero lift pitching moment coefficient without tail $C_{mAC}$ from Equation Eq. 6.17 is dependent on four variables. Next $\Delta_f C_{mAC}$ needs to be calculated from $\Delta_f C_{m0.25c}$ using Eq. 6.19 and Eq. 6.20. The value for the lift coefficient at landing $C_L$ is 1.13 while the increase in airfoil $C_{Lmax}$ by flaps at landing is 1.29. The value for $\Delta_f C_{m\ 0.25MAC}$ equals -0.5924. Both equations can be merged and simplified, thus $\Delta_f C_{m\ AC}$ equals -0.748.

Now only $\Delta_{fus}C_{mAC}$ needs to be calculated. The zero angle lift $C_{L0}$ is calculated as 1.29 according to Assignment 4. Applying all previous values, see Table 6.5 , $\Delta_{fus}C_{mAC}$ is equal to -0.28.

| | | | |
|---|---|---|---|
| $C_{m_{ac_w}}$ | -0.1 | $\mu_2$ | 1.1 |
| $\Delta_{nac}C_{m_{ac}}$ | -0.05 | $\mu_3$ | 0.022 |
| $b_f$ | 4.16 m | $\Delta_f C_L$ | 0.963 |
| $l_f$ | 16.1 m | $\frac{c_f}{c'}$ | 0.24 |
| $h_f$ | 4.16 m | $\frac{c'}{c}$ | 1.23 |
| $S$ | 157.8 m$^2$ | $C_L$ | 1.13 |
| $\bar{c}$ | 4.818 m | $\Delta_f c_l$ | 0.465 |
| $C_{L_0}$ | 1.29 | A | 9 |
| $C_{L_{\alpha wf}}$ | 4.775 | $\Lambda_{1/4}$ | 0.467 rad |
| $\mu_1$ | 0.22 | | |

Table 6.5: Variables used to calculate the zero lift pitching moment coefficient

Equation 6.15 can now be evaluated and simplified to find the zero lift pitching moment coefficient $C_{m\,AC}$ equal to $C_{m_{ac}}$ = -1.174.

### 6.3.2. Controllability curve

Since all the values are known the neutral point can be plotted. A figure with the forward controllability limit, rear centre of gravity limit and neutral point will be plotted using eq. 6.16:

$$\frac{S_h}{S}=\frac{\frac{\Delta x_{cg}}{\bar{c}}+\frac{x_{np}-x_{cg}}{\bar{c}}-\frac{C_{m_{ac}}}{C_{L_{\max}}}}{\left\{\frac{C_{L_{h_\alpha}}}{C_{L_{w_\alpha}}}\left(1-\frac{d\varepsilon}{d\alpha}\right)-\frac{C_{L_h}}{C_{L_{max}}}\right\}\left(\frac{V_h}{V}\right)^2\frac{l_h}{\bar{c}}}$$

The two unknowns on the formula are $\frac{S_h}{S}$ and $\frac{\Delta x_{cg}}{\bar{c}}$. The lift of the horizontal tail is $C_{L_h}$ =-0.8. The maximum lift coefficient of the aircraft is 2.21 and the lift rate coefficient of the aircraft without tail at Low speed is $C_{L\alpha_{A-h}}$ = 4.775 1/rad given by Eq.6.3 where the variables are summarized the table below. Since the rest of the variables are known, Figure 6.5 can be plotted.

| | |
|---|---|
| $C_{L\alpha w}$ | 4.561 |
| $\frac{b_f}{b}$ | 0.110 m |
| $\frac{S_{net}}{S}$ | 0.816 m |

## 6.3 The Scissor plot

Using equations 6.14, 6.15, 6.16 and all the calculated values, the scissor plot can be made. This is shown in figure 6.5.

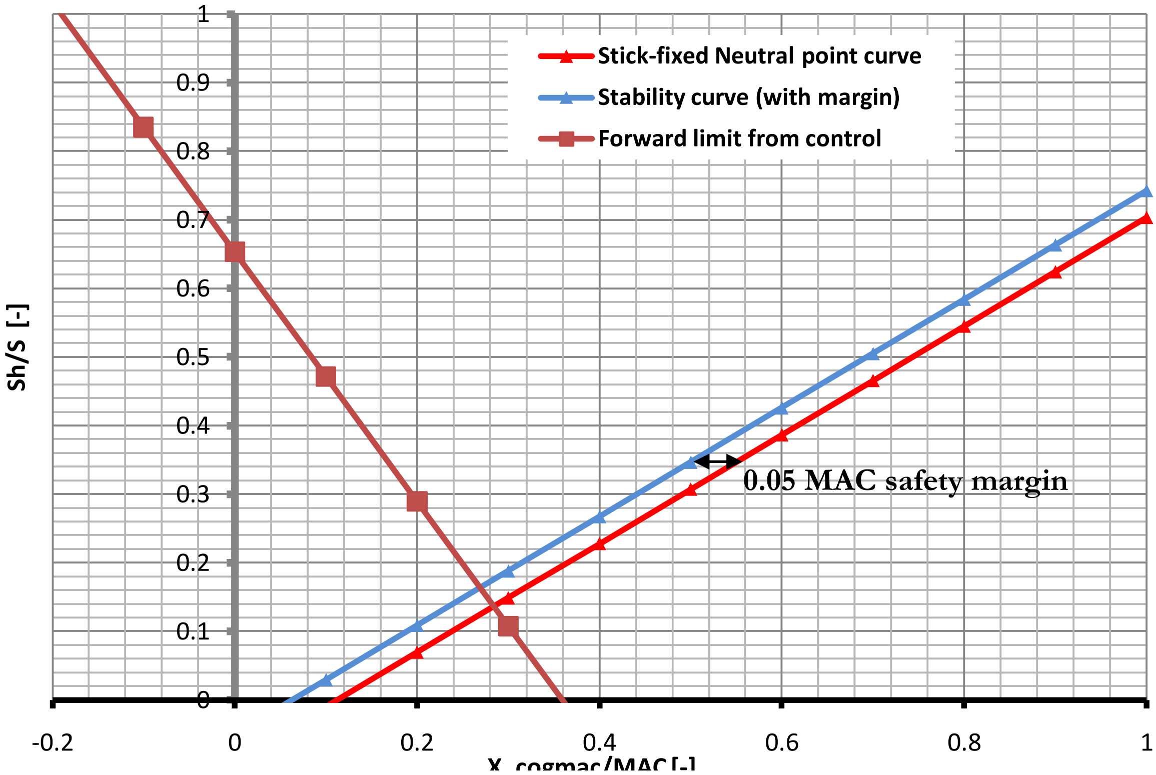


Figure 6.5: The obtained scissor plot.

## Chapter 7: Evaluation and conclusion

In this chapter the tail section will be evaluated more closely since all plots are made. Combining figure 5.5 with figure 6.6 gives figure 7.1. The x-values of the point of intersection represent the optimal forward and aft centre of gravity limit. The y-values of the point of intersection represent the optimal $x_{LEMAC}/L_{fus}$ and $S_h/S$. These values give the optimal position of the wing. See graph 7.1 below.

The optimal value are Sh/S =0.29 for the ratio between the horizontal tail area and the wing area which results in an area of 45.8 $m^2$. The difference between the area calculated by means of the V-method (Sh/h=0.285) and the area found from the figure is 1.79%, which is smaller than the 10% criteria.

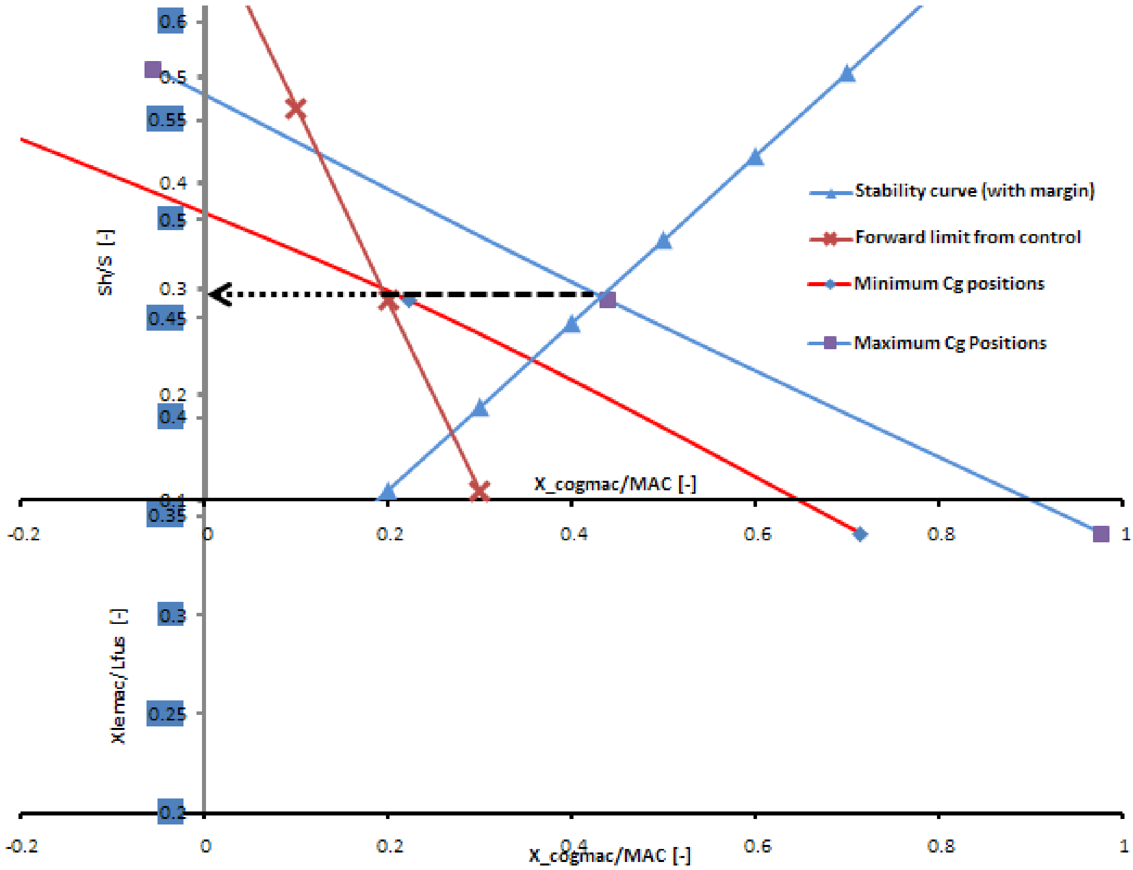



*Figure 7.1: Overlaying the scissor plot with the cg-ranges plot gives the optimum position.*

The optimal value found $X_{leMAC}/L_{fus}$ is indicated at 0.46. This leads to a value of 19.55m for the distance between the leading edge of the MAC and the nose of the aircraft. The value used previously in the calculations is 19.50m, however since the difference is only 1.79% the value used in the calculations is assumed to have sufficient accuracy. To obtain an even better value, the leading edge of the mean aerodynamic chord should be moved to the rear by 5 cm.

## References


**Websites**

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

**Lectures notes**

Files:

- PowerPoint sheets,
- Design of Aircraft: Thomas, Corke, Torenbeek
- Synthesis of Subsonic Airplane Design: E.Torenbeek

**Books**

Aircraft Design: Synthesis and Analysis
Desktop aeronautics, Version 0.99

# Appendices

## 1 Aircraft parameters

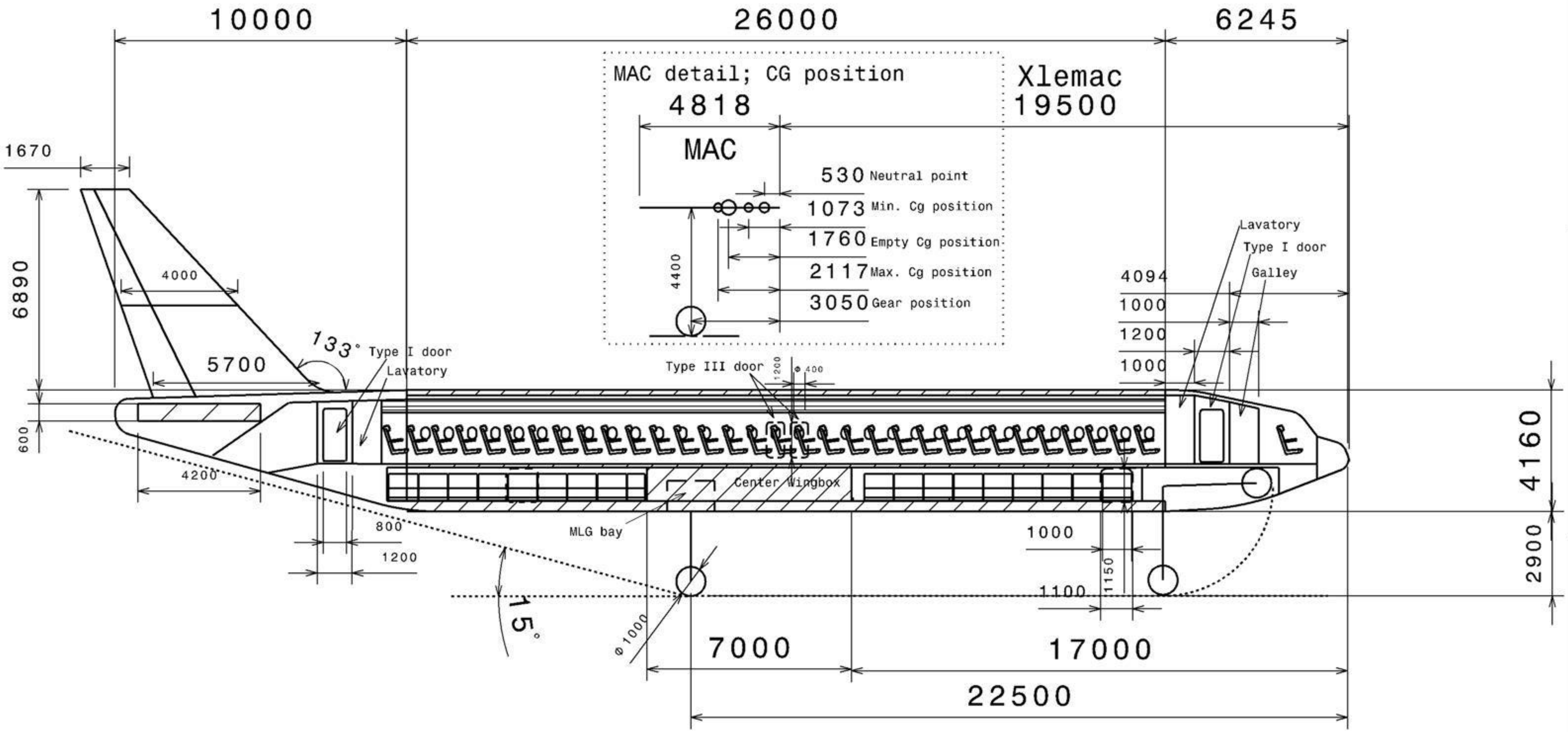
| Symbol | Parameters | Value | Unit |
|---|---|---|---|
| | **Cabin characteristics** | | |
| | Cabin length | 33 | m |
| | Maximum diameter | 4.16 | m |
| | Maximum cabin height | 2.53 | m |
| | Maximum width | 4.1 | m |
| | Aisle width | 0.66 | m |
| | Aisle height | 2.35 | m |
| | Wall thickness | 0.08 | m |
| | Chair width | 0.48 | m |
| | Chair Pitch | 0.83 | m |
| | Design configuration | 3-class | - |
| | Total passengers for design configuration | 158 | - |
| | **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 | 64-214 | - |
| MAC | Mean aerodynamic Chord | 4.818 | m |
| Y | MAC location, engine suspension point | 7.318 | m |
| dCl/dα | Lift curve slope | 0.10 | 1/rad |
| $A_{0L}$ | Zero lift angle of attack | -1.8 | $^{O}$ |
| $C_{d0}$ | Minimum drag coefficient | 0.0062 | - |
| $C_{lmax}$ | Maximum lift coefficient | 1.45 | - |
| $C_{lmax\ clean}$ | Maximum clean lift coefficient | 1.70 | - |
| α0 | Zero Angle of attack lift coefficient | 0.229 | - |
| T/c | Thickness to chord ratio | 0.14 | - |
| $M_{cr}$ | Critical Mach number | 0.736 | - |
| $C_{ldes}$ | Design lift coefficient | 0.203 | - |
| | **Aircraft Wing geometry** | | |

| | | | |
|---|---|---|---|
| b | Wing span | 37.703 | m |
| S | Wing area | 157.945 | $m^2$ |
| A | Aspect ratio | 9 | - |
| $\Lambda_{LE}$ | Leading Edge Wing sweep angle | 30 | $^{O}$ |
| $\Lambda_{0.25C}$ | Quarter chord wing sweep angle | 26.77 | $^{O}$ |
| $\Lambda_{0.5C}$ | Half chord wing sweep angle | 23.183 | $^{O}$ |
| | Incidence angle | 0 | $^{O}$ |
| | Twist angle | 0 | $^{O}$ |
| | Dihedral angle | 5 | $^{O}$ |
| Re | Reynolds number at cruise | 2.688E+07 | - |
| $C_r$ | Root Chord | 7 | m |
| | Taper ratio | 0.197 | - |
| $T_r$ | Root thickness | 210.000 | m |
| $R_c$ | Cantilever Ratio | 20.926 | - |
| $M_{Dd}$ | Mach of drag divergence | 0.757 | - |
| $\alpha_{trim}$ | Trim angle | -0.249 | $^{O}$ |
| $C_{L\alpha}$ | Lift curve slope, low speed | 0.080 | 1/deg |
| $C_{Lmax}$ | Maximum wing lift coefficient | 1.131 | m |
| $M_{fuelmax}$ | Total fuel tank capacity | 32928 | kg |
| | Engine location | on MAC | |
| | Front spar location | 0.20 | c |
| | Rear spar location | 0.65 | c |
| | | | |
| | **Weights and loadings** | | |
| | Cargo weight | 4300 | kg |
| | Fuel weight | 29918 | kg |
| | Payload weight | 17010 | kg |
| $W_E$ | Empty weight | 52409 | kg |
| $W_{MTO}$ | Maximum take-off weight | 99431 | 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 | 0.152 | - |
| $C_{Lmax}$ | Maximum lift coefficient (take-off) | 2.4 | - |
| $s_{TO}$ | Take-off distance | 2100 | m |
| $s_L$ | Landing distance | 1650 | m |
| R | Range | 5500 | km |
| E | Loiter Endurance | 45 | min |
| $V_{stallLand}$ | Landing stall speed | 60 | m/s |
| $V_{stallCr}$ | Cruise stal speed | 77 | m/s |
| | | | |
| | **Vertical tailplane geometry** | | |
| Sv | Area | 25 | m² |
| Lv | Distance of 0.25MAC to 0.25 MAC wing | 20.247 | m |
| | quarter chord sweep angle | 39 | $^{O}$ |
| | Aspect ratio | 1.9 | - |

| | | | |
|---|---|---|---|
| | Taper ratio | 0.3 | - |
| | Thickness to chord ratio | 0.13 | - |
| | Tail height | 6.892 | m |
| | Root chord | 5.581 | m |
| | Tip chord | 1.674 | m |
| | MAC-V-Tail | 3.978 | m |
| | | | |
| | **Horizontal tailplane geometry** | | |
| | quarter chord sweep angle | 36.69 | o |
| Sh | Area | 45 | $m^2$ |
| | Aspect ratio | 5 | - |
| | Taper ratio | 0.4 | - |
| | Thickness to chord ratio | 0.13 | - |
| | Span | 15.000 | m |
| | Root chord | 4.286 | m |
| | Tip chord | 1.714 | m |
| | MAC-H-Tail | 3.184 | m |

## 2 Aircraft technical drawings

See the following 3 pages.

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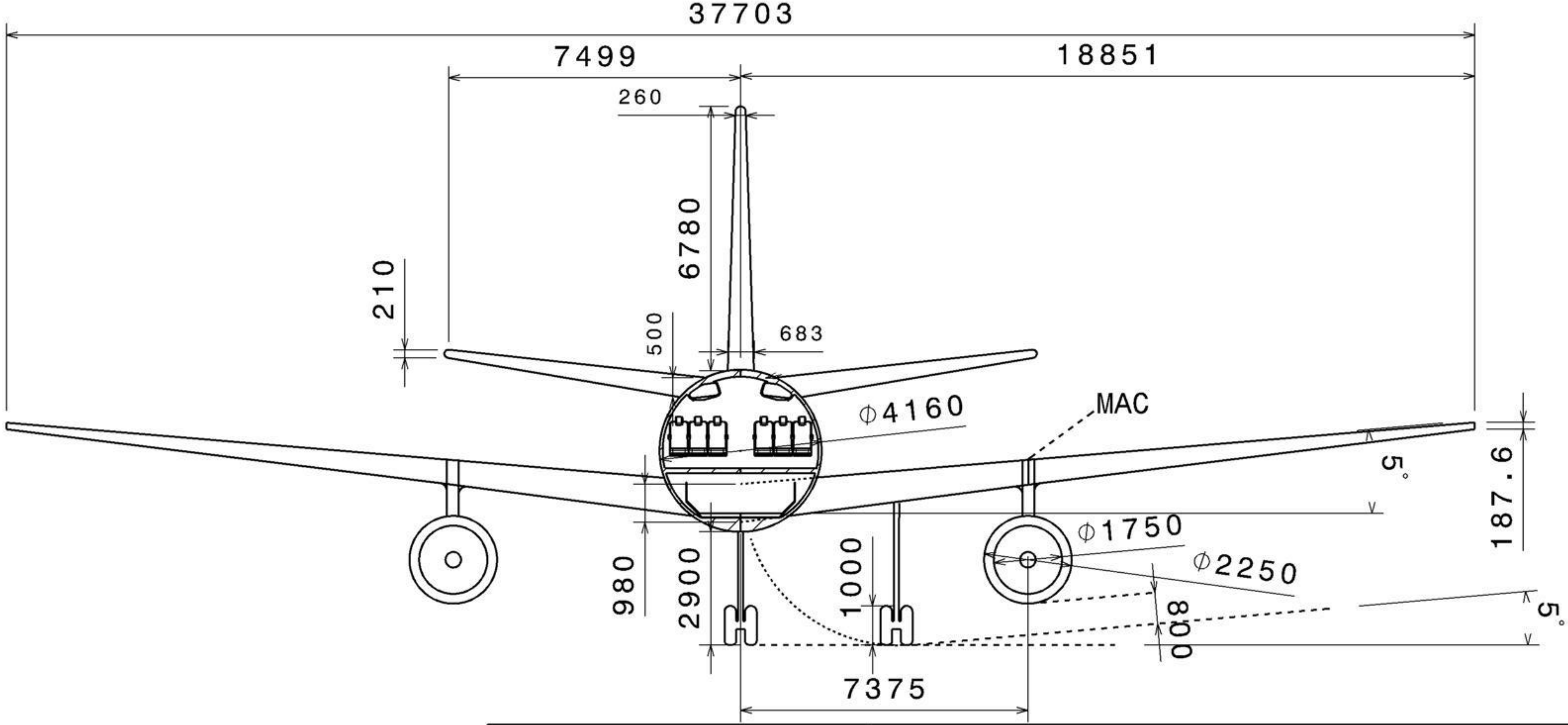
# Aircraft Design

DRAWING TITLE
D.3:Fuselage side-view and CG position

| | | | |
|---|---|---|---|
| DRAWN BY<br>Wout De Backer | DATE<br>05/01/2010 | | |
| CHECKED BY | DATE | SIZE<br>A4 | DRAWING NUMBER | REV<br>X |
| DESIGNED BY | DATE | SCALE 1:200 | WEIGHT(kg) | SHEET 1/1 |

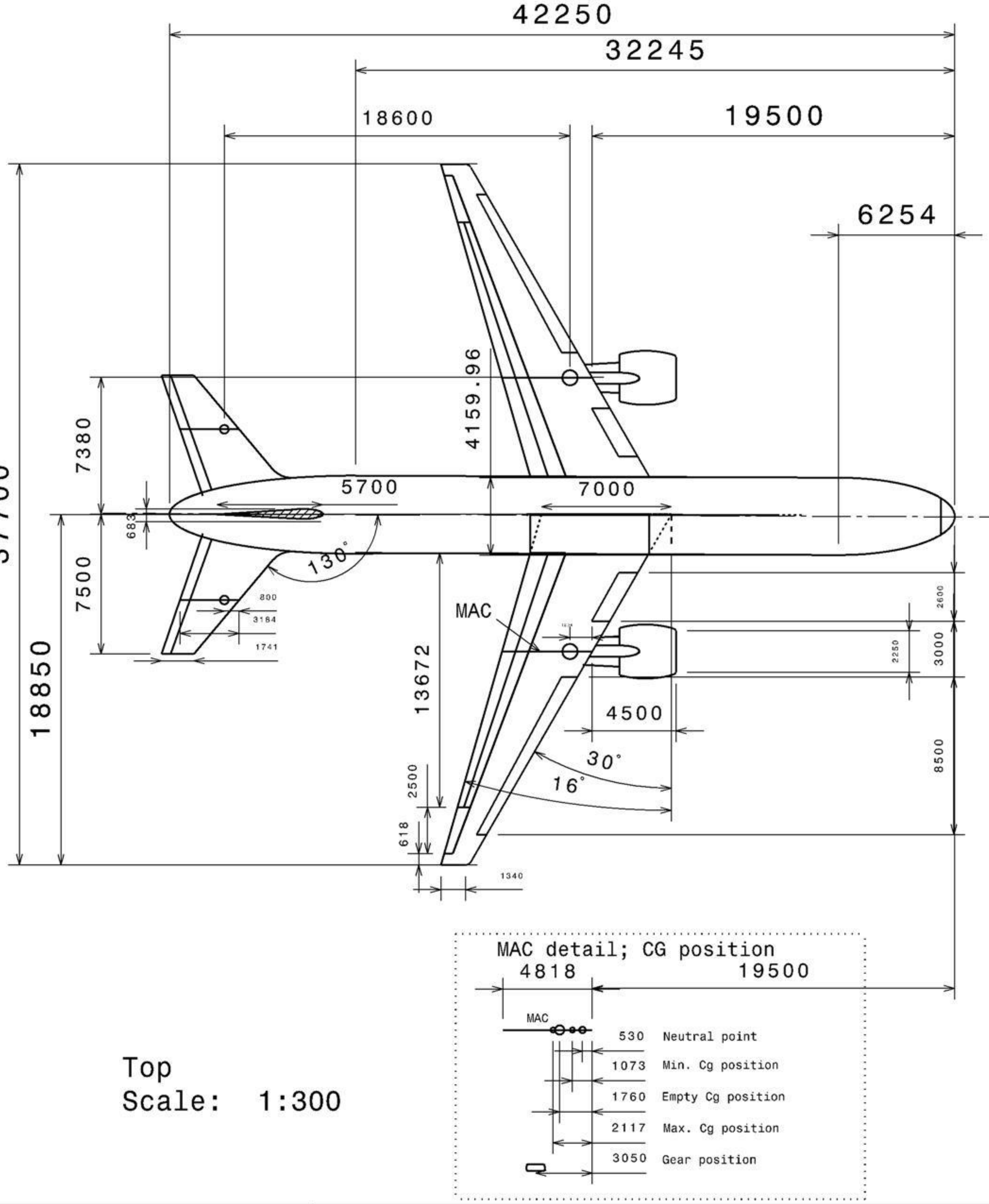



Front view
Scale: 1:150



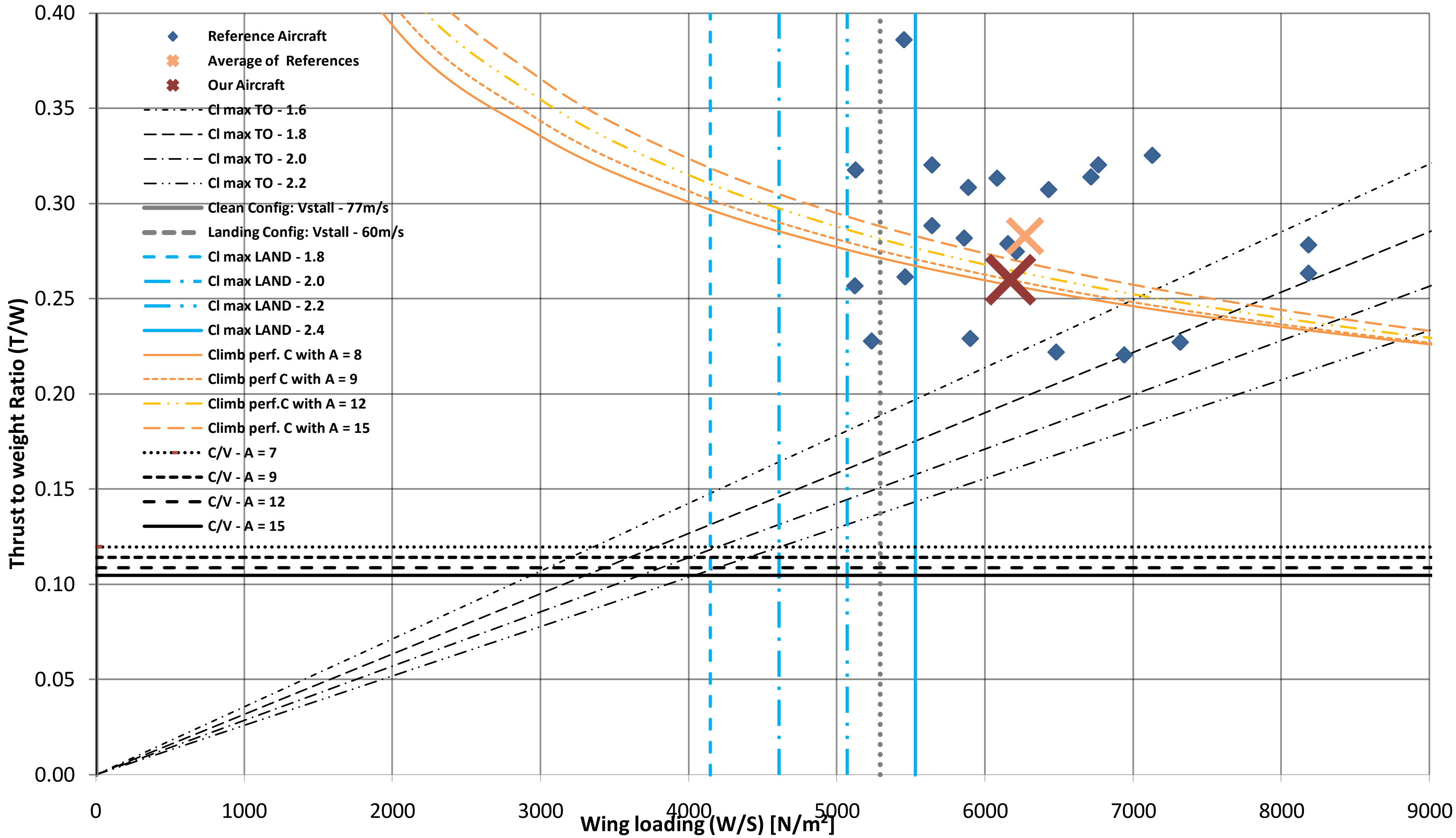
Aircraft Design

| | | | |
|---|---|---|---|
| DRAWN BY: Wout De Backer | DATE: 05/01/2010 | DRAWING TITLE: Front View | |
| CHECKED BY | DATE | SIZE: A4 | DRAWING NUMBER: Product1 | REV |
| DESIGNED BY | DATE | WEIGHT(kg) | SHEET 1/1 |




| | | | | |
|---|---|---|---|---|
| This drawing is our property. It can't be reproduced or communicated without our written agreement. | | Aircraft Design | | |
| DRAWN BY: Wout De Backer | DATE: 06/01/2009 | DRAWING TITLE: Top View | | |
| CHECKED BY | DATE | SIZE: A4 | DRAWING NUMBER | REV |
| DESIGNED BY | DATE | SCALE 1:1 | WEIGHT(kg) | SHEET 1/1 |

# 3 T/W – W/S Diagram



# 4 Reference aircraft table

## 4.1 Airbus aircraft parameters

| Manufacturer<br>Type<br>Model | AIRBUS<br>A300-<br>600R | AIRBUS<br>A310-<br>300 | AIRBUS<br>A319-<br>100 | AIRBUS<br>A320-<br>200 | AIRBUS<br>A321-<br>200 | AIRBUS<br>A330-<br>200 | AIRBUS<br>A330-<br>300 | AIRBUS<br>A340-<br>200 | AIRBUS<br>A340-<br>300 | AIRBUS<br>A340-<br>500 | AIRBUS<br>A340-<br>600 |
|---|---|---|---|---|---|---|---|---|---|---|---|
| Initial service date | 1974 | 1983 | 1995 | 1988 | 1993 | 1998 | 1994 | 1993 | 1994 | 2002 | 2002 |
| **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 |
| Model / Type | 4158 | 4152 | CFM56-5A4 | CFM56-5A3 | CFM56-5B3 | CF6-80E1A4 | CF6-80E1A2 | CFM-56-5C2 | CFM-56-5C4 | Trent 553 | Trent 556 |
| No. of engines | 2 | 2 | 2 | 2 | 2 | 2 | 2 | 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 |
| **Operational Items:** | | | | | | | | | | | |
| ***Accomodation:*** | | | | | | | | | | | |
| Max. seats (single class) | 375 | 280 | 153 | 179 | 220 | 380 | 440 | 440 | 440 | 440 | 475 |
| 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 |
| No. abreast | 9 | 9 | 6 | 6 | 6 | 9 | 9 | 9 | 9 | 9 | 9 |
| Hold volume ($m^3$) | 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 |
| Max. take-off | 170500 | 150000 | 64000 | 73500 | 89000 | 230000 | 217000 | 257000 | 271000 | 365000 | 365000 |
| Max. landing | 140000 | 123000 | 61000 | 64500 | 73500 | 177150 | 179000 | 181000 | 190000 | 236000 | 254000 |
| Zero-fuel | 130000 | 113000 | 57000 | 60500 | 71500 | 165142 | 169000 | 172000 | 178000 | 222000 | 240000 |
| Max. payload | 41100 | 33300 | 17390 | 19190 | 22780 | 36400 | 48400 | 49400 | 48150 | 51635 | 63000 |
| Max. fuel payload | 27100 | 21500 | 5360 | 13500 | 19060 | 0 | 18600 | 21220 | 33160 | 31450 | 29311 |
| Design payload | 25270 | 20710 | 11780 | 14250 | 17670 | 24035 | 28025 | 24890 | 28025 | 29735 | 36100 |
| Design fuel load | 56330 | 49624 | 13020 | 17940 | 23330 | 85765 | 70786 | 111882 | 113125 | 164875 | 151890 |
| Operational empty | 88900 | 79666 | 39200 | 41310 | 48000 | 120200 | 118189 | 120228 | 129850 | 170390 | 177010 |
| ***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 |
| 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 |
| 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 |
| 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 |
| ***Fuel (litres):*** | | | | | | | | | | | |
| Standard | 68150 | 61100 | 23860 | 23860 | 23700 | 139090 | 98250 | 140000 | 141500 | 195620 | 195620 |
| 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 |
| Height (m) | 5,64 | 5,64 | 4,14 | 4,14 | 4,14 | 5,64 | 5,64 | 5,64 | 5,64 | 5,64 | 5,64 |

| | | | | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|
| Width (m) | 5,64 | 5,64 | 3,95 | 3,95 | 3,95 | 5,64 | 5,64 | 5,64 | 5,64 | 5,64 | 5,64 |
| Finess Ratio | 9,45 | 8,00 | 8,57 | 9,51 | 11,27 | 10,24 | 11,08 | 10,32 | 11,08 | 11,63 | 12,34 |
| ***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 |
| Span (m) | 44,84 | 43,89 | 33,91 | 33,91 | 33,91 | 58,00 | 58,00 | 58,00 | 58,00 | 61,20 | 61,20 |
| $MAC$ (m) | 6,44 | 5,89 | 4,29 | 4,29 | 4,29 | 7,26 | 7,26 | 7,26 | 7,26 | 8,35 | 8,35 |
| Aspect Ratio | 7,73 | 8,80 | 9,39 | 9,39 | 9,39 | 9,26 | 9,26 | 9,26 | 9,26 | 8,56 | 8,56 |
| Taper Ratio | 0,300 | 0,283 | 0,240 | 0,240 | 0,240 | 0,251 | 0,251 | 0,251 | 0,251 | 0,220 | 0,220 |
| Average (t/c) % | 10,50 | 11,80 | | | | | | | | | |
| 1/4 Chord Sweep (º) | 28,00 | 28,00 | 25,00 | 25,00 | 25,00 | 29,70 | 29,70 | 29,70 | 29,70 | 31,10 | 31,10 |
| ***High Lift Devices:*** | | | | | | | | | | | |
| Trailing Edge Flaps Type | F2 | F1 | F1 | F1 | F2 | 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 |
| 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 |
| 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 |
| Height (m) | 8,60 | 8,10 | 6,26 | 6,26 | 6,26 | 9,44 | 8,45 | 8,45 | 8,45 | 9,44 | 9,44 |
| Aspect Ratio | 1,64 | 1,45 | 1,82 | 1,82 | 1,82 | 1,87 | 1,58 | 1,58 | 1,58 | 1,87 | 1,87 |
| Taper Ratio | 0,365 | 0,395 | 0,303 | 0,303 | 0,303 | 0,350 | 0,350 | 0,350 | 0,350 | 0,350 | 0,350 |
| 1/4 Chord Sweep (º) | 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 |
| $S_v/S$ | 0,174 | 0,206 | 0,176 | 0,176 | 0,176 | 0,131 | 0,124 | 0,124 | 0,124 | 0,109 | 0,109 |
| $S_vL_v/S_b$ | 0,097 | 0,095 | 0,055 | 0,065 | 0,079 | 0,057 | 0,059 | 0,055 | 0,059 | 0,049 | 0,049 |
| ***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 |
| Span (m) | 16,26 | 16,26 | 12,45 | 12,45 | 12,45 | 12,45 | 19,06 | 19,06 | 19,06 | 21,50 | 21,50 |
| Aspect Ratio | 3,81 | 4,13 | 5,00 | 5,00 | 5,00 | 5,00 | 4,98 | 4,98 | 4,98 | 4,97 | 4,97 |
| Taper Ratio | 0,420 | 0,417 | 0,256 | 0,256 | 0,256 | 0,256 | 0,360 | 0,360 | 0,360 | 0,360 | 0,360 |
| 1/4 Chord Sweep (º) | 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 |
| $S_h/S$ | 0,267 | 0,292 | 0,253 | 0,253 | 0,253 | 0,253 | 0,201 | 0,201 | 0,201 | 0,213 | 0,213 |
| $S_hL_h/S_c$ | 1,062 | 1,116 | 0,689 | 0,799 | 0,957 | 0,957 | 0,791 | 0,733 | 0,791 | 0,729 | 0,729 |
| ***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 |
| Wheelbase (m) | 18,60 | 15,21 | 12,60 | 12,63 | 16,90 | 16,90 | 25,40 | 23,20 | 25,40 | 28,53 | 32,50 |
| 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 |
| 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 |
| 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 |
| Spanwise location | 0,359 | 0,352 | 0,338 | 0,338 | 0,338 | 0,312 | 0,312 | 0.312/0.672 | 0.312/0.672 | 0.296/0.62<br>5 | 0.296/0.62<br>5 |
| **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 |
| Max. wing load (kg/m²) | 655,77 | 684,93 | 522,88 | 600,49 | 727,12 | 633,43 | 597,63 | 707,79 | 746,35 | 834,67 | 834,67 |
| 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 |
| ***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 |
| 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 |
| $V_{no}/M_{mo}$ | 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 |
| $C_{Lmax}$ (T/O) | 2,44 | 2,45 | 2,58 | 2,56 | 3,10 | 2,21 | 2,51 | 2,60 | 2,61 | | |
| $C_{Lmax}$ (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 | | |
| Altitude (ft) | 31000 | 35000 | 33000 | 28000 | 28000 | | 33000 | 33000 | 33000 | | |
| 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 |
| Design range | 4000 | 4300 | 1900 | 2700 | 2700 | 6370 | 4500 | 7350 | 7150 | 8500 | 7500 |
| Max. fuel (+ payload) | 4698 | 5076 | 4158 | 3672 | 2602 | | 7046 | 8834 | 8089 | 9000 | 7800 |
| Ferry range | | | | | | | | | | 9800 | 8800 |
| **Design Parameters:** | | | | | | | | | | | |
| $W/SC_{Lmax}$ | 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 | BOEING | BOEING | BOEING | BOEING | BOEING | BOEING | BOEING | BOEING | BOEING | BOEING |
|---|---|---|---|---|---|---|---|---|---|---|
| **Type** | **707-** | **717-** | **727-** | **737-** | **737-** | **737-** | **737-** | **737-** | **737-** | **737-** |
| **Model** | **320C** | **200** | **200Adv** | **200** | **300** | **400** | **500** | **600** | **700** | **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 (m$^3$) | 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 (m²) | 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 (º) | 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 (m²) | 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 (m²) | 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 (º) | 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 (m²) | 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 (º) | 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/m²) | 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 | | | | | | | |
| $C_{Lmax}$ (T/O) | | 2,15 | 1,90 | 2,32 | 2,47 | 2,38 | 2,49 | | | |
| $C_{Lmax}$ (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:** | | | | | | | | | | |
| $W/SC_{Lmax}$ | 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 |