Royal Institute of Technology

Bachelor Thesis

Aeronautics

Transport Aircraft Conceptual Design

Albin Karlsson & Anton Lomaeus
Supervisor: Christer Fuglesang & Fredrik Edelbrink
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Abstract

A conceptual design for a transport aircraft has been created, tailored for humanitarian missions along the equator with its home base in the European Union while optimizing for fuel efficiency and speed. An initial estimate of the empty weight was made using historical data and Breguet equations, based on a required payload of 60 tonnes and range of 5 500 nautical miles. A constraint diagram consisting of requirements for stall speed, takeoff distance, climb rate and landing distance was used to determine wing loading and thrust to weight ratio, resulting in a main wing area of 387 m$^2$ and thrust to weight ratio of 0.224, for which two Rolls Royce Trent 1000-H engines were selected. A high aspect ratio wing was designed with blended winglets to optimize against lift induced drag. Wing placement and tail volume were decided by iterative calculations, resulting in a centre of lift located aft of the centre of gravity during all stages of the mission. The resulting aircraft model has a high wing with a span of 62 m, length of 49 m with a takeoff gross weight of 221 tonnes, of which 83 tonnes are fuel.
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1 Background

The task was to design an aircraft capable of delivering one week of necessities to 5 000 people from Europe to the equator and back without landing. The mission includes dropping the necessities with parachutes. The aircraft was optimized toward fuel efficiency and speed, as the nature of the mission requires the aircraft to be able to deliver quickly while being fuel efficient.

2 Specification

2.1 Payload

The required necessities are food, water purification tablets, medicine and tents. “Meal ready to eat” (MRE) is made for delivering by plane and each MRE contains 1 200 kcal and weighs about 500 g (MRE Info, 2017). To meet the daily calorie requirement (World Health Organization, 2017) adults need two MREs per day, which results in 35 tonnes food per delivery. Contaminated water is obtainable but needs to be disinfected, which is solved with water purification tablets, with an estimated weight of one tonne. Medicine is estimated to weigh one tonne. Other things that are not consumables like tents and equipments together with pallets and parachutes needed for air drop are estimated to weigh up to 23 tonnes. Together the total required payload weight was set to 60 tonnes.

2.2 Flight requirements

To reduce the onboard fuel needed a shorter flight distance is desired, Malta international airport’s southerly position makes it a good choice and was therefore chosen as the takeoff airport. Nairobi international airport is located close to the equator and will be the point the range will be calculated to. Malta to Nairobi and back is a distance of approximately 5 100 nm (Great Circle Mapper, 2017), with an additional 8% added for the possibility of longer missions, putting the range demand at 5 500 nm.

To meet the goals of fuel efficiency and high cruise speed, high-bypass turbofan engines were the best alternative, because of their fuel efficiency and speed during long range flights (Gudmundsson 2014). Turbofan engines are most efficient at high altitude (Gudmundsson 2014), therefore to optimize for fuel efficiency the cruise altitude was set to 10 km. Whilst higher cruise speed is desired, the fuel consumption increases because of the increased drag. At speeds above 0.80 M transonic airflow starts to occur which increases the drag significantly (NASA, 2017), therefore the cruise speed was set to 0.80 M.

For better precision during drop-off the aircraft needs to be able to fly at low speeds at low altitudes, the drop-off altitude is therefore set to 500 m above the ground. The parachutes have a maximum deployment speed of 150 kn (Coleman, 2017).

According to MIL-C5011A military aircraft must be able to climb at least 500 feet per minute, and with one engine inoperative during takeoff the requirement is 150 feet per minute (Raymer 2012, 988 Table F.2).
Malta airport’s runway length is roughly 11,500 ft (Airportguide, 2017), which was set as the maximum takeoff distance. Landing distance was set to 8,200 ft whilst Federal Aviation Regulations (FAR) was followed in order to comply with most major airports, in case change of landing airport is needed.

Table 1: Specification summary

<table>
<thead>
<tr>
<th>Specification</th>
<th>Value</th>
</tr>
</thead>
<tbody>
<tr>
<td>Payload</td>
<td>60 tonnes</td>
</tr>
<tr>
<td>Range</td>
<td>5,500 nm</td>
</tr>
<tr>
<td>Engine type</td>
<td>High-bypass turbofan</td>
</tr>
<tr>
<td>Cruise altitude</td>
<td>10 km</td>
</tr>
<tr>
<td>Cruise speed</td>
<td>0.80 M</td>
</tr>
<tr>
<td>Drop-off speed</td>
<td>150 kn</td>
</tr>
<tr>
<td>Climb rate</td>
<td>500 ft/min</td>
</tr>
<tr>
<td>Climb rate OEI</td>
<td>150 ft/min</td>
</tr>
<tr>
<td>Takeoff distance</td>
<td>11,500 ft</td>
</tr>
<tr>
<td>Landing distance</td>
<td>8,200 ft</td>
</tr>
</tbody>
</table>

2.3 Mission profile

The mission profile starts with warm up, taxi and takeoff at Malta international airport. Continuing with a climb to the cruise altitude and cruise to Nairobi where it descents to the drop-off altitude. After reaching drop-off altitude the aircraft will have the ability to loiter for 30 minutes, including dropping the cargo. After the 30 minutes it starts to climb to cruise altitude and return to Malta including cruise, descent and landing.

Figure 1: Sketch of the mission profile
3 Initial weight estimation

3.1 Total takeoff weight

An estimation of the takeoff gross weight, $W_0$, including payload, crew, fuel and empty weight. $W_0$ was calculated with historical data and Breguet equations, using assumptions of constant fuel consumption, lift to drag ratio, altitude and speed during cruise and loiter (Raymer 2012, 28-52).

$$W_0 = \frac{W_{\text{crew}} + W_{\text{payload}}}{1 - \frac{W_{\text{fuel}}}{W_0} - \frac{W_{\text{empty}}}{W_0}}$$

(1)

Where a roughly estimated weight of a 3 or 4 man crew with luggage, $W_{\text{crew}}$, of 500 kg.

3.2 Empty weight

The empty weight and fuel weight is dependent on the takeoff gross weight. After estimating the $W_0$ an approximate empty weight fraction could be calculated.

$$\frac{W_{\text{empty}}}{W_0} = A \cdot W_0 \cdot K_{VS} \cdot K_{\text{composite}}$$

(2)

Where $A$ was set to 0.88, $C$ was set to -0.07 as used for military cargo, $K_{VS}$ was set to 1 for fixed sweep (Raymer 2012, 31) and $K_{\text{composite}}$ was set to 0.95 for using light weight composite material (Raymer 2012, 48).

3.3 Fuel weight

Depending on the mission profile the fuel weight varies as a percentage of the takeoff gross weight.

$$\frac{W_{\text{fuel}}}{W_0} = 1.06 \cdot (1 - \frac{W_{11}}{W_0})$$

(3)

To account for trapped fuel and reserves an additional 6% is added to the total fuel weight (Raymer 2012, 42). $W_{11}$ is the total weight of the aircraft after landing. $W_{11}$ is calculated by multiplying the estimated percentage of the weights that have been consumed during the mission profile’s stages shown in Figure 1. Most of the stages have historical data that can be used, except for cruise and loiter that are dependent on the range and endurance and needs to be calculated.

$$W_5 = \exp\left(-\frac{R \cdot C_{\text{cruise}}}{V \cdot L/D_{\text{cruise}}}\right) \cdot W_4$$

(4)

$$W_7 = \exp\left(-\frac{E \cdot C_{\text{loiter}}}{L/D_{\text{max}}}\right) \cdot W_6$$

(5)
To calculate the fuel fraction for cruise and loiter the Breguet equations assumes that the distance traveled during descent and climb is part of the cruise range (Raymer 2012, 33-35). Therefore the range, $R$, was split in half to 2 750 nm for each direction. Specific fuel consumption for high-bypass turbofan during cruise, $C_{\text{cruise}}$, was set to 0.5 and loiter, $C_{\text{loiter}}$, was set to 0.4 (Raymer 2012, 36 Table 3.3). Cruise velocity, $V$, was set to 470 kn approximately 0.80 M at 10 km altitude (Aerospaceweb, 2017). Lift to drag ratio, $L/D_{\text{max}}$, was set to 20 similar to the DC-8 (Raymer 2012, 39 fig 3.5), where $L/D_{\text{cruise}}$ is 86.6% of $L/D_{\text{max}}$ for jet engines (Raymer 2012, 41). The loiter time, $E$, was set to 1800 s, that equals 30 minutes.

Table 2: Historical fuel fraction for military cargo (Roskam, 1985)

<table>
<thead>
<tr>
<th>Warmup and taxi</th>
<th>W1 = 0.99</th>
</tr>
</thead>
<tbody>
<tr>
<td>Takeoff</td>
<td>W2 = 0.99 · W1</td>
</tr>
<tr>
<td>Initial climb</td>
<td>W3 = 0.995 · W2</td>
</tr>
<tr>
<td>Climb</td>
<td>W4 = 0.98 · W3</td>
</tr>
<tr>
<td>Descent</td>
<td>W6 = 0.99 · W5</td>
</tr>
<tr>
<td>Climb</td>
<td>W8 = 0.98 · W7</td>
</tr>
<tr>
<td>Descent</td>
<td>W10 = 0.98 · W9</td>
</tr>
<tr>
<td>Landing and taxi</td>
<td>W11 = 0.992 · W10</td>
</tr>
</tbody>
</table>

Note that this equation does not handle payload drop, but calculates the needed fuel as if the aircraft performed the whole mission without dropping the cargo.

**Weight results**

The iterative method resulted in a preliminary takeoff gross weight of 221 tonnes where 35% of the weight or 78 tonnes was empty weight and 37% or 83 tonnes was fuel. The weights were calculated in MATLAB, see attached code in Appendix A.
4 Wing loading & thrust to weight ratio

With the takeoff gross weight, \( W_0 \), estimated, it is necessary to calculate wing loading, \( W/S \), and thrust to weight ratio, \( T/W \), while also taking into account certain performance parameters for the aircraft. Requirements for stall speed, takeoff distance, landing distance and rate of climb are evaluated in respect to their impact on wing loading and thrust to weight ratio.

\[
\frac{W}{S} = \frac{1}{2} \cdot \rho \cdot (V_{\text{stall}})^2 \cdot CL_{\text{max}}
\]  

(7)

Equation 7 calculates the required wing loading dependent on air density, stall speed and the aircraft’s maximum lift coefficient. The stall speed, \( V_{\text{stall}} \), was set to 120 kn. The density of the air, \( \rho \), was set to 1.225 kg/m\(^3\) to simulate standard sea level conditions. The maximum lift coefficient, \( CL_{\text{max}} \), was set to 2.4, based on historical values for commercial transport aircraft with flaps and slats (Raymer 2012, 127).

\[
\frac{W}{S} = \frac{\text{TOP} \cdot \sigma \cdot CL_{\text{TO}} \cdot T}{W}
\]  

(8)

Equation 8 calculates required wing loading to achieve chosen takeoff distance depending on the takeoff parameter, density ratio, coefficient of lift at takeoff and thrust to weight ratio. The takeoff parameter, \( \text{TOP} \), was chosen as 260, due to the aircraft being a twin engine jet and a FAR balanced takeoff distance of 11500 ft (Raymer 2012, 130. Fig5.4). The density ratio, \( \sigma \), is the ratio between takeoff air density and standard sea level air pressure. It was assumed to be 1. The coefficient of lift at takeoff, \( CL_{TO} \), was estimated to be \( CL_{\text{max}} \) divided by 1.21, which gave a takeoff lift coefficient of 1.9835 (Raymer 2012, 131).

\[
\frac{W}{S} = \frac{(S_{\text{landing}} - S_a) \cdot (\sigma \cdot CL_{\text{max}})}{80}
\]  

(9)

Equation 9 calculates required wing loading with regard to desired landing distance, obstacle clearing distance, density ratio and maximum coefficient of lift. Landing distance, \( S_{\text{landing}} \), was set to 8200 ft(2500 m) according to specifications. Obstacle clearing distance, \( S_a \), was set to 1000 ft(305 m) to be able to clear a 50 ft(15.24 m) obstacle while on approach speed on a 3-degree glide slope, according to FAR 23 (Raymer 2012, 132-133). The density ratio and maximum coefficient of lift were the same as used in equation 7 and 8.

\[
G = \frac{V_{\text{climb}}}{V_{\text{TO}}}
\]  

(10)

\[
\frac{W}{S} = \frac{[T/W - G] - \sqrt{[T/W - G]^2 - (4C_{D0}/\pi A e)}}{2/q \pi A e}
\]  

(11)

Equation 11 calculates required wing loading with regard to desired rate of climb and thrust to weight ratio. The rate of climb was evaluated at takeoff conditions, where MIL-C5011A dictates a takeoff speed, \( V_{\text{TO}} \), at least ten percent greater than stall speed and a
climb speed, $V_{climb}$, of 500 ft/min with all engines operating (Raymer 2012, 988 Appendix F). The aspect ratio of the aircraft, $A$, was set to 10 to optimize against lift induced drag (Raymer 2012, 76-78). The zero lift drag coefficient, $CD_0$, was estimated to be 0.015 and Oswald’s span efficiency factor, $e$, to be 0.8 for no flap settings (Raymer 2012, 135). For takeoff flap settings, the zero lift drag coefficient was increased by about 0.02 and Oswald’s span efficiency decreased by about 10% (Raymer 2012, 142). The dynamic pressure, $q$, was evaluated at takeoff speed and sea level conditions. Eventual engine failure was also taken into account, by halving $T/W$ and lowering the climb speed to 150 ft/min with one engine inoperative, abiding by MIL-C5011A (Raymer 2012, 988).

All units were converted to imperial units and equations (7-11) were plotted in a constraint diagram. All calculations were scaled to takeoff conditions. The code performing these calculations, Constraint diagram, can be found in appendix A.

Based on the constraint diagram, Figure 2, a $W/S$ of 115 lb/ft$^2$ and $T/W$ of at least 0.224 were chosen to comply with all requirements. It should be noted that the landing distance requirement was trivial compared to the rest. The minimum thrust required at takeoff is 486 kN, for which two Rolls Royce Trent 1000-H engines were selected, delivering a continuous thrust of 524 kN. The dry weight of the engine, not including nacelle, is 6120 kg (EASA 2017).
5 Wing parameters

5.1 Wing sweep, taper ratio & twist

The wing sweep is used to reduce the adverse effects of transonic flow which is increased with higher Mach-numbers. A leading edge sweep of 27.5° was chosen to combine efficient cruise at 0.80 Mach and low stall speed (Raymer 2012, 80 Fig 4.20).

Taper ratio, \( \lambda \), affects the distribution of lift along the wing span. For swept wings the taper ratio historically varies between 0.2 and 0.3 (Raymer 2012, 83), where the taper ratio was set to 0.3.

The aileron control surface was placed on the outer part of the wing and with a twist set to \(-3^\circ\) it retains the roll capability when the inner part of the wing starts to stall (Raymer 2012, 86).

5.2 Wing dimensions

The wing loading was calculated to 115 lb/ft\(^2\) in Section 4, and a weight, \( W_0 \), of 221 tonnes resulted in a main wing area, \( S_w \), of 387 m\(^2\).

\[
b = \sqrt{A \cdot S} \quad (12)
\]

Equation 12 in turn yields a main wing span, \( b_w \), of roughly 62 m.

\[
C_{\text{root}} = \frac{2S}{b(1 + \lambda)} \quad (13)
\]

\[
C_{\text{tip}} = \lambda \cdot C_{\text{root}} \quad (14)
\]

With the taper ratio, span and wing area known the wing chords were calculated using equations 13 and 14 resulting in main wing root chord of 9.6 m and wingtip chord of 2.9 m (Raymer 2012, 75).

\[
\bar{c} = \frac{2}{3} \cdot C_{\text{root}} \cdot \frac{1 + \lambda + \lambda^2}{1 + \lambda} \quad (15)
\]

The wing’s mean aerodynamic chord, \( \bar{c} \), was calculated using equation 15 and was needed to determine the location of the wing’s centre of lift and centre of gravity (Raymer 2012, 75).

For the main wing, the airfoil NACA 23012, was selected. It is a cambered airfoil that provides good aerodynamic properties (Raymer 2012, 967). For the vertical and horizontal tail, a symmetrical airfoil, NACA 63015, was selected. The airfoils were chosen arbitrarily, as the study of optimal airfoil was deemed out of the scope of this project.
5.3 Wing vertical location & dihedral

By utilizing a high wing location, it enables the fuselage to be closer to the ground, allowing the aircraft to be loaded and unloaded without special ground-handling gear. This could be beneficial in disaster struck areas, where the aircraft can be loaded directly from trucks due to the high wing providing plenty of ground clearance for the engines allowing the cargo bay floor to be closer to the ground (Raymer 2012, 89-90).

High wing position and sweep together add up to an effective dihedral. Excessive dihedral causes "Dutch roll", a repeated side-to-side motion involving yaw and roll. To counteract this excessive effect a negative dihedral, anhedral, was used. Preliminary an anhedral angle of 3° was selected. It was merely an estimation and requires further testing and validation in order to get satisfactory dihedral effects (Raymer 2012, 88-89).

5.4 Winglet & flaps

High lift devices on the main wing can substantially improve the maximum coefficient of lift, $CL_{max}$, of the wing. By utilizing triple slotted flaps and leading edge slats, the conservative estimate of 2.4 for the maximum coefficient of lift can be increased, resulting in decreased stall speed.

As can be seen in Figure 4, the aircraft utilizes a blended winglet. A standard winglet, as invented by Richard Whitcomb at NASA, is believed to be able to increase the lift-to-drag ratio by up to 20% (Raymer 2012, 95). Even conservative increases to the lift-to-drag ratio can have tremendous effect on fuel consumption, as the initial weight estimation does not take into account the effect of winglets.
6 Aircraft dimensions and wing placement

6.1 Fuselage length

With the takeoff gross weight and wing area known, it was necessary to estimate the length of the fuselage in order to determine wing placement, centre of gravity, tail volumes and centre of lift.

\[ L = a W_0^C \] (16)

Equation 16 calculates the fuselage length based on different aircraft types and how their takeoff gross weight, \( W_0 \), correlates to fuselage length. As the aircraft is based on a military cargo/bomber type, \( a \) was chosen as 0.104 and \( C \) as 0.50 with a takeoff gross weight of 221 tonnes (Raymer 2012, 156-157). The fuselage length, \( L \), was calculated and rounded to 49 m.

6.2 Tail volume & dimensions

With the fuselage length known it was possible to estimate the size of the horizontal and vertical tail surfaces, which in turn enables the finding of the aircraft’s centre of gravity and centre of lift. An approach based on historical data was used in estimating tail size.

\[ C_{VT} = \frac{L_{VT} S_{VT}}{b_w S_w} \] (17)

\[ C_{HT} = \frac{L_{HT} S_{HT}}{C_w S_w} \] (18)

The vertical tail volume coefficient, \( C_{VT} \), and the horizontal tail volume coefficient, \( C_{HT} \), were given the values of 0.076 and 0.950 respectively due to the aircraft being a military cargo/bomber type and the tail being a T-tail, based on historical values (Raymer 2012, 160).

The tail moment arm, the distance from the main wing’s centre of lift to the tail surface’s centre of lift, was initially estimated to be 50% of the fuselage length, for both the horizontal and vertical tail surfaces (Raymer 2012, 160). Equation 17 and 18 were rewritten to solve for surface areas, \( S_{VT} \) and \( S_{HT} \).

The taper ratio for the vertical tail, \( \lambda_{VT} \) was set to 0.8 and the aspect ratio, \( A_{VT} \), to 1 based on historical trends for a T-tail (Raymer 2012, 113 Table 4.3). The taper ratio, \( \lambda_{HT} \), and aspect ratio, \( A_{HT} \), for the horizontal tail were set to 0.3 and 5 respectively. The sweep of the horizontal tail was set to 32.5°, 5° greater than the main wing sweep, as this tends to make the tail stall later than the main wing (Raymer 2012, 112). The vertical tail sweep was set to 30°. With the surface areas of the tail known, equations 13 and 14 were used to calculate tail chord lengths.
6.3 Weights

In order to estimate the weight of individual aircraft components a statistical approach was used, based on historical values for transport & bomber aircraft.

Table 3: Approximate empty weight build up

<table>
<thead>
<tr>
<th>Transport &amp; Bomber</th>
<th>kg/m²</th>
<th>Multiplier</th>
<th>Approximate location</th>
</tr>
</thead>
<tbody>
<tr>
<td>Wing</td>
<td>49</td>
<td>$S_{\text{exposed\ planform}}$</td>
<td>40% MAC</td>
</tr>
<tr>
<td>Horizontal tail</td>
<td>27</td>
<td>$S_{\text{exposed\ planform}}$</td>
<td>40% MAC</td>
</tr>
<tr>
<td>Vertical tail</td>
<td>27</td>
<td>$S_{\text{exposed\ planform}}$</td>
<td>40% MAC</td>
</tr>
<tr>
<td>Fuselage</td>
<td>24</td>
<td>$S_{\text{wetted\ area}}$</td>
<td>50% length</td>
</tr>
<tr>
<td>Landing gear*</td>
<td>0.043</td>
<td>W₀</td>
<td>centroid</td>
</tr>
<tr>
<td>Installed engine</td>
<td>1.3</td>
<td>Engine weight</td>
<td>centroid</td>
</tr>
</tbody>
</table>

* 25% to nose gear and 75% to main gear (Raymer 2012, 584)

The wetted area of the fuselage, $S_{\text{wetted\ area}}$ was approximated as the surface area of a 49 m long cylinder, 6 m in diameter. Table 3 was used to estimate weights. Should the weights of the wing, tail, fuselage, landing gear and installed engine be lower than that of the previously calculated empty weight, the difference should be added to the fuselage weight, to account for avionics, hydraulics and various other aircraft systems.

The total fuel weight was divided between the forward fuel tank, rear fuel tank and fuel tanks in the wing. It was estimated that the wings carry half of the fuel, added to the total wing weight, with the remaining fuel divided equally between the forward and rear fuel tanks.

6.4 Wing placement

What follows is a calculation of the location of the centre of gravity and the centre of lift, best followed in the code named Wing placement found in appendix A. Briefly explained, a first initial estimate for the wing placement was entered and the code answers with centre of gravity, centre of lift and actual tail moment arms. The tail moment arms were then updated to the correct values, and the calculation was repeated with new correct tail surface areas. This was iterated until satisfactory results converge, with centre of lift
behind the centre of gravity during all stages of the planned mission. The centre of lift was calculated by finding the equilibrium point of the lifting forces of the wing, horizontal tail and the fuselage. The lift was assumed to be proportional to wing area. The fuselage was estimated to provide lift proportional to 25% of its wetted fuselage. It was assumed that the wing’s lifting force acted at 25% of its mean aerodynamic chord.

It is assumed that the fuel tanks are cross fed and that fuel is drawn evenly from each tank. The centre of gravity was calculated at five segments, illustrating how the centre of gravity moves during a mission.

- Section 0: Pre Take-off
- Section 1a: Post Take-off, Climb, Cruise, Descent, Loiter
- Section 1b: Post Take-off, Climb, Cruise, Descent, Loiter and Dropping payload
- Section 2a: Climb, cruise, descent, land with payload
- Section 2b: Climb, cruise, descent, land without payload

Table 4: Centre of Gravity during mission

<table>
<thead>
<tr>
<th>Section</th>
<th>Centre of Gravity (x-coord.)</th>
<th>Centre of Lift (x-coord.)</th>
<th>Aircraft mass [tonnes]</th>
</tr>
</thead>
<tbody>
<tr>
<td>0</td>
<td>22.6</td>
<td>26.0</td>
<td>221</td>
</tr>
<tr>
<td>1a</td>
<td>23.0</td>
<td>26.0</td>
<td>180</td>
</tr>
<tr>
<td>1b</td>
<td>24.5</td>
<td>26.0</td>
<td>120</td>
</tr>
<tr>
<td>2a</td>
<td>23.3</td>
<td>26.0</td>
<td>140</td>
</tr>
<tr>
<td>2b</td>
<td>25.7</td>
<td>26.0</td>
<td>83</td>
</tr>
</tbody>
</table>

Satisfactory results were achieved with a wing placement corresponding to a distance between the nose and the start of leading edge of 20 m, illustrated in Table 4.

7 Suggestions for future work

Due to the limitations of available time and manpower, the project has been substantially limited in scope. The airfoils were chosen arbitrarily from a selection of airfoils in Aircraft Design (Raymer 2012), requiring further study. Historical data has been used for most variables, in particular for drag and lift coefficients. Using Computational Fluid Dynamics (CFD) on the 3D-model will yield more accurate values for drag and lift characteristics of the aircraft. The next logical step would be to repeat the iteration with updated values from the CFD results, engine parameters and account for air supply drop during weight estimation. Finally construct a carpet plot, optimizing the aircraft for certain performance parameters such as range and payload capability.
8 References


Appendix A

Weight estimation

```matlab
clc, close all, clear all

%Estimated weight
WtoGuess=input('Viktuppskattning i Ton: ')\*1000%[kg], Total weight ...
(221.1tonnes)
Wpl=60000; %[kg], Payload weight
Wc=500; %[kg], Crew weight

%Constants
R=2750\*1852; %[m], Range/distance one way
P=1; %Efficiency constant
C=0.5/3600\*P; %Specific fuel consumption during Cruise
L=0.4/3600\*P; %Specific fuel consumption during Loiter
V=470\*0.5144; %[m/s], cruise speed
LD=20; %Maximum Lift to drag ratio
LCruise=LD\*0.886;%Lift to drag ratio during cruise
E=0.5\*3600; %[s], Endurance (30min)
WeFrac=0.88\*WtoGuess^(-0.07)*0.95%Empty weight fraction

%Procent of total mass left after the different sections
W0=0.99*WtoGuess; %Start and warmup
W1=0.99*W0; %Taxi
W2=0.995*W1; %Takeoff
W3=0.98*W2; %Climb
W4=exp(-R*P/(V*LCruise))*W3;%Cruise
W5=0.99*W4; %Descent
W6=exp(-E*L/LD)*W5; %Loiter
W7=0.98*W6; %Climb
W8=exp(-R*P/(V*LCruise))*W7;%Cruise
W9=0.99*W8; %Descent
W10=0.992*W9; %Landing and taxi

WendFrac=W10/WtoGuess%Total weight fraction after landing
Wfrac=W1.06*(1-WendFrac)%Fuel weight fraction
Wto=(Wpl+Wc)/(1-Wfrac-WeFrac); %[kg], Total takeoff weight
Wfuel=Wto-Wfrac; %[kg], Weight fuel
Wempty=Wto-Wfuel-Wpl-Wc;%[kg], Empty weight

%Result display
disp('Fraktioner:')
disp([WendFrac, WeFrac, WfFrac])
disp('Totalvikt i ton: ')
disp([WtoGuess/1000, Wto/1000, Wfuel/1000, Wempty/1000])
disp(x)
```
clear all, close all, clc

%Converting constants
mToFeet=3.2808399; %Length from SI to imperial, m -> ft
lbft2ToNm2=47.88; %Pressure from imperial to SI, lb/ft^2 -> N/m^2
kgm2tolbft2=0.20482; %Pressure from SI to imperial, kg/m^2 -> lb/ft^2
kg3toslugfeet3=0.0019403; %Density from SI to imperial, kg/m^3 -> slug/ft^3

%Constants

m=9.8*mToFeet; %Imperial gravity
p=1.225*kgm3ToSlugfeet3; %[slug/ft^3] Density at sea level, (0.974)
sigma=1; %Density ratio, (0.794 value for hot day @5000ft)
CLmax=2.4; %CLmax, commercial air transport with flaps and ...
slats, s.127
A=10; %Aspect ratio, wing span squared over wing+canard ...
area s.78
e=0.8; %Oswald's span efficiency factor s.135
CD0=0.015; %Zero lift drag coefficient s.135
CD0climb=CD0+0.02; % Cd,0 during climb
eclimb=e*0.95; %Adjusted for takeoff flap settings, s.142
Landingscale=0.6663; %Vktcoefficient at landing
Vvert=500/60*mToFeet; %[ft/s] Vertical speed 500m/min
Vstall=120*0.514444*mToFeet; %[ft/s] Stall Speed, 120kn to ft/s
hc=10000*mToFeet; %[ft] Cruise altitude
vc=470*0.514444*mToFeet; %[ft/s] Cruise speed 470kn
p=0.41351*kgm3ToSlugfeet3; %[slug/ft^3] Density at cruise altitude s.958
hloiter=500*mToFeet; %[ft] Loiter altitude
vloiter=150*0.514444*mToFeet; %[ft/s] Loiter speed 150kt
ploiter=1.16730*kgm3ToSlugfeet3; %[slug/ft^3] Air density at 500m s.958

%Calculated constants

v=1.1*Vstall; %[ft/s] Horizontal speed, 110% of stall speed = ...
take off speed
G=Vvert/v; %Climb gradient
g=0.5*pc*(vc)ˆ2; %[Pa] Dynamic pressure during cruise
qclimb=0.5*p*vˆ2;
gloiter=0.5*ploiter*(vloiter)ˆ2;

%T/W can never be less than climb gradient + other stuff eq 5.31:
H=G+2*sqrt(CD0climb/(pi*A*eclimb));
TW=[0]; %Creating vector for Thrust to weight ratio
for i=1:100 TW(i,1)=i/100+H; end

%%Stall Speed
WSstallp=0.5*p*(Vstall)ˆ2*CLmax; %Wing area needed for set stall speed.
for i=1:100 WSstall(i,1)=WSstallp; end

%%Takeoff Distance
CLTO=CLmax/1.21; %Coefficient for lift at takeoff
TOP=260; %Twin engine, FAR Takeoff, max 11.5kft take off ...
distance s.130
WSTO=TOP*sigma*CLTO*TW;

%%Landing Distance
Sa=305*mToFeet; %[ft] Obstacle Clearing Distance 305m, FAR
Slanding=2500*mToFeet; %[ft] Desired landing distance 2500m, FAR
WSlandingp=(Slanding-Sa)*(sigma*CLmax)/80; %
WSlandingp = WSlandingp / (Landingscale); % OBS tveksam åp denna!!
for i = 1:100
    WSlanding(i,1) = WSlandingp;
end

%% Maximum Jet Range
WSjetp = q * sqrt((pi * A * e * CD0) / 3); %
Cruisescale = 0.9557; % Viktkoefficient innan cruise
WSjetp = WSjetp / Cruisescale; % OBS tveksam åp denna!!
for i = 1:100
    WSjet(i,1) = WSjetp;
end

%% Climb
x = (4 * CD0climb) / (pi * A * eclimb); %
y = (2 / (qclimb * pi * A * eclimb)); %
WSclimb = (TW - (G - sqrt((TW - G).^2 - x))) / y; %
G2 = 150 / 60 / v; %
WSclimbhalf = (0.5 * TW - (G2 - sqrt((0.5 * TW - G2).^2 - x))) / y; % One engine gone

%% Wing Loading for Loiter Endurance || Used to optimise for loiter!
WSloiterp = qloiter * sqrt(3 * pi * A * e * CD0); %
Loiterscale = 0.8101; %
WSloiterp = WSloiterp / Loiterscale; %
for i = 1:100
    WSloiter(i,1) = WSloiterp;
end

%% PLOTS
figure(1)
plot(WSTO, TW, 'r', 'LineWidth', 2)
hold on
plot(WSclimb, TW, 'b', 'LineWidth', 2)
% plot(WSjet, TW, 'c', 'LineWidth', 2)
plot(WSlanding, TW, 'g', 'LineWidth', 2)
% plot(WSloiter, TW, 'm', 'LineWidth', 2)
plot(WSstall, TW, 'k', 'LineWidth', 2)
plot(WSclimbhalf, TW, 'y', 'LineWidth', 2)
xlim([113 119])
ylim([0.218 0.232])
grid ON
ylabel('T/W')
xlabel('W/S [lb/ft^2]')
legend('Takeoff', 'Climb', 'Landing Distance', 'Stall Speed', 'Climb OEI')
Wing placement

clear all, close all, clc

%%
% Calculating wing placement to ensure stable flight in regards to center of mass. Cargo hold starts at x=5 and ends x=35
L=49; % [m] effective length of fuselage
Wtot=221100; % [kg] TAKE OFF GROSS WEIGHT
Wempty=78107.4; % [kg] Empty weight
WFuelStart=82500; % [kg] Total fuel weight
bw=62; % [m] Wingspan

%% Tail Design
% Tailwing calculations
Sw=387; % [m^2] Main wing area
bw=62; % [m] Main wingspan
Cw=6.8430; % [m] Main wing mean chord

% Historical values for military cargo/bomber with T-tail
CHT=1;
CVT=0.08;
CHT=0.95*CHT;
CVT=0.95*CVT;

% Moment arms, engines on the wing 50-55% of fuselage length
LVT=0.5*Lf;
LHT=0.55*Lf;

LVT=18.9433; % Actual values measured in CAD with wing start at x=24.5. Used as preliminary guess
LHT=22.7700;

%% Tail wing areas
SVT=CVT*bw*Sw/LVT;
SHT=CHT*Cw*Sw/LHT*1.2;

% Vertical Tail
Av=1; % Aspect ratio
tv=0.8; % Taper
bVT=sqrt(Av*SVT); % Span
CrootVT=SVT/(bVT*(1+tv)); % Root chord
Ctipv=CrootVT*tv; % Tip chord

% Horizontal Tail
Ah=5; % Aspect ratio
th=0.3; % Taper
bHT=sqrt(Ah*SHT); % Span
CrootHT=SHT/(bHT*(1+th)); % Root chord
Ctiph=CrootHT*th; % Tip chord

%% Empty Weight
% Location, mass and distances from cockpit, centre of mass
% Horizontal tail
MACHT=2/3*CrootHT*(1+0.3+thˆ2)/(1+th);
YHT=bHT/6*(1+2*th)/(1+th); % Assume lift proportional to chord
WHT=27*SHT; % [kg] according to Raymer p584

% Vertical Tail
MACVT=2/3*CrootVT*(1+tv+tvˆ2)/(1+tv);
VVT=bVT/3*(1+2*tv)/(1+tv); % Assume lift proportional to chord
WVT=27*SVT; % [kg] Vertical tail weight
DHT=44.65-(CrootVT-4.8)+bVT*tan(30)+0.4+MACVT; %Cockpit to HT COM

DVT=44.65-(CrootVT-4.8)+YVT*tan(30)+0.4+MACVT;

%Fuselage

Sfuselage=6*pi*49; %Fuselage wetted area approximated as a 49m long 6m ... diameter cylinder

LF=0.5*L; %[m] location 50% of fuselage length

WF=24*Sfuselage; %[kg] Fuselage weight

WF=W+0.17*Wtot; %[kg] Fuselage weight and "All-else empty" weight ...

%Wing

Swing=387; %[m^2]

CrootW=9.6; %[m] Wing root chord

MACW=2/3*CrootW*(1+0.3+0.3^2)/(1+0.3);

Ywing=bw/6*(1+2*0.3)/(1+0.3); %[Y-cord of MACW from aircraft centerline

Wwing=49*Swing; %[kg] weight of wing (both)

%Engine

Trent=6120*2; %[kg] %Weight of two Trent 1000 engines

Weng=1.3*Trent; %[kg] Weight of engines installed in nacelles

%Landing Gear

WFLG=0.043*0.15*Wtot; %[kg] weight of nose gear

LFLG=6; %[m] location of nose gear

WRLG=0.043*0.85*Wtot; %[kg] weight of rear gear

% Wet weight

%Payload

WP=60000; %[kg] 60 tonne payload

LP=5+30*0.5; %[m] Payload evenly distributed in cargo hold

%Forward fuel tank

WF1=0.25*WFuelStart;

LF1=5+30*0.25; %[Fuel tank COM placed in first half of cargo hold

%Rear fuel tank

WF2=0.25*WFuelStart;

LF2=5+30*0.75; %[Fuel tank COM placed in second half of cargo hold

%Wing integrated fuel tank

WF3=0.5*WFuelStart;

%People

Wpeople=500; %[kg]

Lpeople=5*2/3; %[m] assume evenly distributed in last two thirds of cockpit

% Calculating CoG/CoM

Totalweight0=WHT+WVT+WF1+WF2+WF3+Wpeople+WFLG+WRLG+Weng+Wwing+WF+WP;

Diff=Wtot-Totalweight0;

WF=WF+Diff; %Add/remove the remaining difference between Totalweight ... and TOGW to Fuselage weight

Totalweight0=WHT+WVT+WF1+WF2+WF3+Wpeople+WFLG+WRLG+Weng+Wwing+WF+WP;

Winglocation=input('Enter wing location: ')

%Wing placement dependent distances
Lwing=Winglocation+0.4*MACW;
Leng=Winglocation+10*tand(27.5); % Location of engine COM, placed 10m out on the wing with a sweep angle of 27.5°
LRLG=27; % [m] location of rear gear, 2m behind wing COM
L3=Lwing; % Assume same COM as wing structural weight
CoG=(WHT*DHT+WVT*DVT+WF1*LF1+WF2*LF2+WF3*LF3+WPeople*Lpeople+WFLG*LFLG+WRLG*LRLG+Weng*Leng+Wwing*Lwing+WF*LF+WP*LP)/Totalweight0;

%% Flight Envelope
% Based on mission profile. Assume cross-fed fuel tanks, even consumption
% Mission profile (from code WeightEstimation)

W0=0.99; % Engine startup
W1=0.99*W0; % Taxi
W2=0.995*W1; % Take off
W3=0.98*W2; % Climb
W4=0.8102; % First cruise
W5=0.99*W4; % Descent into target zone
W6=0.7942; % Loiter in target zone
W7=0.98*W6; % Ascent from target zone
W8=0.6598; % Cruise home
W9=0.99*W8; % Descent into home base
W10=0.992*W9; % Landing

CoG0=CoG;

1. Post take-off, cruise, descent, loiter

WF1x=W6*TOGW-TOGW+Fuelweight;
WFuelLeft1=WFuelStart-Wtot*(1-W6);
WF1=WFuelLeft1*0.25; % WF1 (W6-W10)/(1-W10); % WFuelLeft1*0.25
WF2=WFuelLeft1*0.25; % WF2 (W6-W10)/(1-W10); % WFuelLeft1*0.25
WF3=WFuelLeft1*0.5; % WF3 (W6-W10)/(1-W10); % WFuelLeft1*0.5
Totalweight1=WHT+WVT+WF1+WF2+WF3+WPeople+WFLG+WRLG+Weng+Wwing+WF+WP;
CoG1=(WHT*DHT+WVT*DVT+WF1*LF1+WF2*LF2+WF3*LF3+WPeople*Lpeople+WFLG*LFLG+WRLG*LRLG+Weng*Leng+Wwing*Lwing+WF*LF+WP*LP)/Totalweight1;

2. Dropping payload
WP=0;
Totalweight2=WHT+WVT+WF1+WF2+WF3+WPeople+WFLG+WRLG+Weng+Wwing+WF+WP;
CoG2=(WHT*DHT+WVT*DVT+WF1*LF1+WF2*LF2+WF3*LF3+WPeople*Lpeople+WFLG*LFLG+WRLG*LRLG+Weng*Leng+Wwing*Lwing+WF*LF+WP*LP)/Totalweight2;

3a. Climb, cruise, descent, landing
WFuelLeft2=WFuelStart-Wtot*(1-W10);
WF12=WFuelLeft2*0.25; % WF1 (W10-W10)/(1-W10); % WFuelLeft2*0.25
WF22=WFuelLeft2*0.25; % WF2 (W10-W10)/(1-W10); % WFuelLeft2*0.25
WF32=WFuelLeft2*0.5; % WF3 (W10-W10)/(1-W10); % WFuelLeft2*0.5
Totalweight3=WHT+WVT+WF12+WF22+WF32+WPeople+WFLG+WRLG+Weng+Wwing+WF+WP;
CoG3=(WHT*DHT+WVT*DVT+WF12*LF1+WF22*LF2+WF32*LF3+WPeople*Lpeople+WFLG*LFLG+WRLG*LRLG+Weng*Leng+Wwing*Lwing+WF*LF+WP*LP)/Totalweight3;

3b. Climb cruise descent landing with payload,
WP3b=60000;
WFuelLeft2=WFuelStart-Wtot*(1-W10);
WF12=WFuelLeft2*0.25; % WF1 (W10-W10)/(1-W10); % WFuelLeft2*0.25
WF22=WFuelLeft2*0.25; % WF2 (W10-W10)/(1-W10); % WFuelLeft2*0.25
WF32=WFuelLeft2*0.5; % WF3 (W10-W10)/(1-W10); % WFuelLeft2*0.5
Totalweight3b=WHT+WVT+WF12+WF22+WF32+WPeople+WFLG+WRLG+Weng+Wwing+WF+WP3b;
CoG3b=(WHT*DHT+WVT*DVT+WF12*LF1+WF22*LF2+WF32*LF3+WPeople*Lpeople+WFLG*LFLG+WRLG*LRLG+Weng*Leng+Wwing*Lwing+WF*LF+WP3b*LP)/Totalweight3b;

% CoG Limits
WingCL=winglocation+Ywing*tand(27.5)+0.25*MACW; %Assume CoL at 25% of MAC

FuselageCL=0.25*Sfuselage*LF; %Assume 25% of fuselage wetted area provides lift, acting at half of fuselage length

COL=(Swing*WingCL+SHT*(44.65-(CrootVT-4.8)+bVT*tand(30)+0.25*MACHT-WingCL)+FuselageCL)/(Swing+SHT+0.25*Sfuselage);

T=table(categorical({'Section 0'; 'Section 1'; 'Section 2'; 'Section 3a'; 'Section 3b'},[CoG0;CoG1;CoG2;CoG3;CoG3b],[COL;COL;COL;COL;COL]),'VariableNames',{'Section' 'COG' 'COL'});

%% Moment arms

MomentHT=44.65-WingCL+bVT*tand(30)+0.25*MACHT;

MomentVT=44.65-WingCL+YVT*tand(30)+0.25*MACVT;

%% Compare Assumed Moment arms with actual moment arms

if (MomentHT-LHT)<0
disp ('HTail wing moment arm is OK')
else
disp ('Redesign Horizontal Tail in regards to MomentHT')
end

if (MomentVT-LVT)<0
disp ('Vertical tail moment arm is OK')
else
disp ('Redesign Vertical Tail in regards to MomentVT')
end

U=table(categorical({'Moment [m]'; 'Root chord [m]'; 'Tip chord [m]'; 'Tip Chord [m]'; 'Span [m]'; 'Sweep [degrees]'},[MomentHT;CrootHT;Ctiph;bHT;(32.5)];[MomentVT;CrootVT;Ctipv;bVT;(30)]),'VariableNames',{'Type' 'HorizontalTail' 'VerticalTail'});

disp(U)
Appendix B

Sketch