Faculty of Engineering Technology

Minor Aircraft Engineering

Design Report

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Final Design

Table 1: Better weight estimation

Part	Weight [kg]
Landing gear	1117
Engines	1580
Wings	4773.4
Payload & luggage	8800
Seats	640
Crew	400
Empennage	3000
Fuel	1068
Fueltank	2537
Fuselage	2500
Electronics etc	800
Cockpit	800
Total weight	28042

Table 2: Spec's final design

S	99.8 m^2
AR	9.0
Main wing	NACA 64(3)618
Tail wing	NACA 64(2)015
C_{D0}	0.022
$C_{D0runway}$	0.026
α_{cruise}	1.5936°
COG nose	16.66 m
AC nose	17.15 m

Table 3: Engine spec's

Length	261.6 cm
Diameter	79.2 cm
Mass	790 kg
Power	3096-4549 kW
SFC	$7.63 \cdot 10^{-7}$ kg/W·s

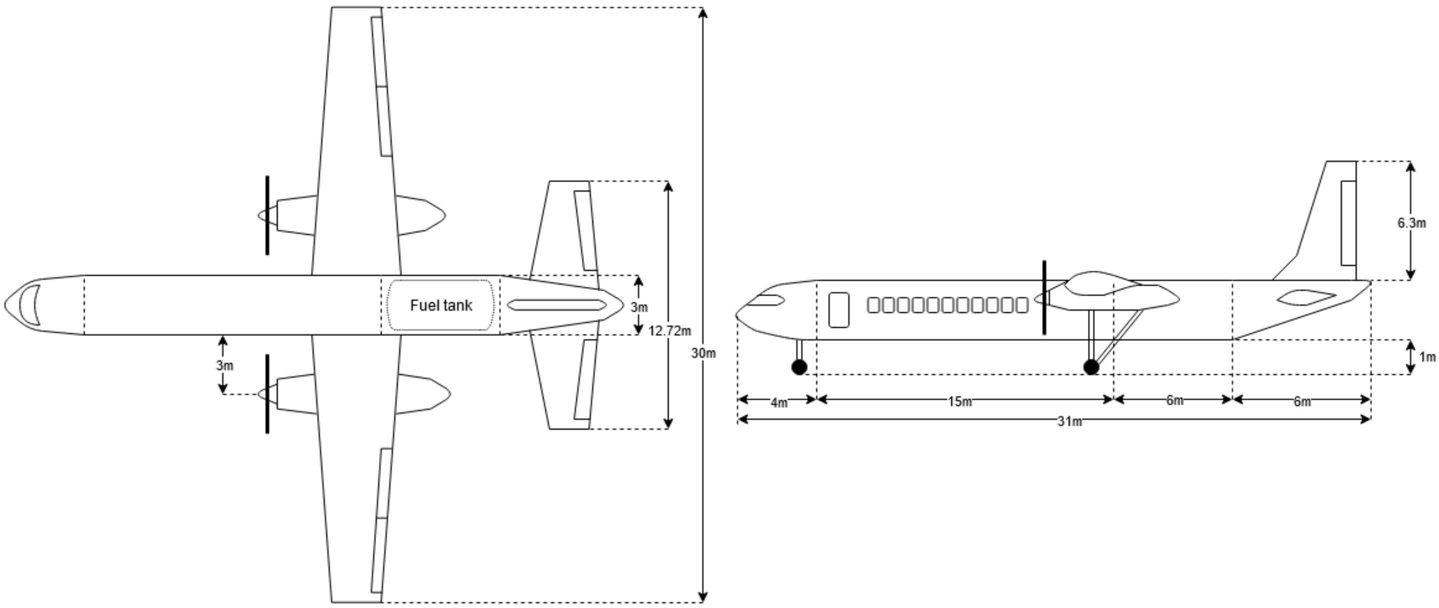


Figure 1: Sketch of top and side view of the airplane with dimensions.

1 Design of the Aircraft

In this section the weight, propulsion (type), wing, cabin and empennage for the aircraft is designed.

1.1 Weight Estimation

The weight of an aircraft can be described by the sum of weights of the different components present on the aircraft. These are prescribed initially by four weight components: The crew, the payload, the fuel and the empty weight of the aircraft itself in Equation 1, which can be rewritten to Equation 2. There are expected to be 10 crew members present in the aircraft and there are a maximum of 80 passengers. Resulting in a weight of the crew of 900kg and payload of 8800kg, calculated with Equation 3 and 4. The Breguet range equation for propeller driven aircraft, Equation 5, can be used to determine the weight of the fuel.

$$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty} = W_{crew} + W_{payload} + \frac{W_{fuel}}{W_0} \cdot W_0 + \frac{W_{empty}}{W_0} \cdot W_0 \tag{1}$$

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - \frac{W_{fuel}}{W_0} - \frac{W_{empty}}{W_0}} \tag{2}$$

$$W_{crew} = 10 \cdot 90 = 900kg = 1,984lbs \tag{3}$$

$$W_{payload} = 80 \cdot (90 + 20) = 8,800kg = 19,400lbs \tag{4}$$

$$R = \frac{\eta_{prop} v}{c} \frac{L}{D} ln \left(\frac{W_0}{W_{nofuel}} \right) \tag{5}$$

$$\frac{W_{nofuel}}{W_0} = e^{-\frac{Rc}{\eta_{prop} v} \frac{1}{\frac{L}{D}}} \tag{6}$$

$$\frac{W_{nofuel}}{W_0} = \frac{W_0 - W_{fuel}}{W_0} = 1 - \frac{W_{fuel}}{W_0} \tag{7}$$

Setting Equation 6 equal to Equation 7, the fraction of the weight of the fuel compared to the total weight of the aircraft becomes Equation 8

$$\frac{W_{fuel}}{W_0} = 1 - e^{-\frac{Rc}{\eta_{prop} v} \frac{1}{\frac{L}{D}}} \tag{8}$$

The fraction between the empty weight and the total weight of any aircraft is almost always equal to approximately 0.62, which will be used to get a first estimate as well. The range R is known from the requirements, the propeller efficiency η_{prop} is estimated to be 85% during cruise, the specific fuel consumption c during cruise is for propeller driven aircraft approximately 0.6 $\frac{lb}{lb \cdot h}$ [1] and the lift to drag ratio is estimated to be approximately 15. A correction for the fuel is used of 15% which explains the multiplication in front of Equation 9. Filling out the numbers in Equation 9, gives the ratio between the weight of the fuel and the total weight, which is namely 0.0868. With all values filled in, Equation 10 gives a first weight estimation of 33082 kg. With these calculations, also the fuel weight can be approximated with Equation 11 and is 2871 kg.

$$\frac{W_{fuel}}{W_0} = 1.15 \left(1 - e^{-\frac{621.504 \cdot 0.6}{372.82 \cdot 0.85} \cdot \frac{1}{15}} \right) = 0.0868 \tag{9}$$

So the total first estimate of the weight is equal to:

$$W_0 = \frac{W_{crew} + W_{payload}}{1 - \frac{W_{fuel}}{W_0} - \frac{W_{empty}}{W_0}} = \frac{1,984 + 19,400}{1 - 0.0868 - 0.62} = 72,933lbs = 33,082kg \tag{10}$$

With these calculations, also the fuel weight can be approximated:

$$W_{fuel} = \frac{W_{fuel}}{W_0} W_0 = 0.0868 \cdot 33,082 = 2,871kg \tag{11}$$

1.2 Hydrogen

The calculations performed in section 1.1 gave a good impression of the expected total weight and fuel weight of the aircraft. These calculations were based on historical and statistical data regarding airplane designs, as well as the assumption that engines run on aviation gasoline. For the GHG-neutral propulsion of the airplane, a fuel alternate to aviation gas needs to be selected. Suitable solutions are batteries used for powering motors or hydrogen used as a combustion fuel for engines. For this conceptual design hydrogen was chosen, because it has 200 times the specific energy of lithium-ion batteries and therefore is the lower weight solution [2]. Substituting hydrogen for aviation gas results in lower mass required due to the significant difference in specific energy. In order to come up with a good estimation for the amount of hydrogen needed for a standard business flight, the assumption is made that the aircraft engines retain similar efficiencies whilst being fueled by aviation gasoline or hydrogen. As a result, the total energy stored in the fuel needs to be the same for both aviation gasoline and hydrogen. The specific energy for aviation gasoline and hydrogen equal $44.65\frac{MJ}{kg}$ and $120\frac{MJ}{kg}$ [3][4]. Now the required weight for hydrogen can be calculated with Equation 12 and is 1068 kg.

$$W_{Hydrogen} = W_{fuel} \cdot \frac{\epsilon_{Gasoline}}{\epsilon_{Hydrogen}} = 2871 \cdot \frac{44.65}{120} = 1,068kg \tag{12}$$

1.2.1 Storage Tank

In order to properly store hydrogen, some general dimensions for the tank have to be chosen and derived. The storage tank is placed behind the cabin, in the back of the fuselage. A cylindrical tank with hemisphere capes is used where during refuelling, gaseous hydrogen is pumped in. The hydrogen is stored at $300K$ and a pressure of $850Bar$. The volume can be calculated using the ideal gas law which is shown in Equation 13. Equation 14 gives the compressibility factor for hydrogen [5].

$$V = W_{Hydrogen}\frac{ZRT}{P} \tag{13}$$

$$Z = 0.99704 + 6.4149 \cdot 10^{-9} \cdot P \tag{14}$$

For the previously stated values, these equations result in a volume of $24.98m^3$.
Since the tank needs to be placed in the fuselage, the radius is limited to $1.5m$. In order to ensure enough space, a radius of $1.25m$ is chosen for the tank. Equation 15 gives a required length of $5.93m$.

$$L = \frac{V}{\pi r^2} + \frac{2}{3}r = \frac{24.98}{4 \cdot \pi \cdot 1.25^2} + \frac{2}{3} \cdot 1.25 = 5.9288m \tag{15}$$

A material needs to be chosen which is capable of handling high pressures. Carbon reinforced plastics are a type of material with low densities and high yield strength. They are also often mentioned in tank design and therefore a suitable material choice for this operation. A yield strength of $1900MPa$ and density of $1520\frac{kg}{m^3}$ is assumed. Equations 16 and 17 are used to estimate the mass for the tank. A safety factor (FoS) of 1.25 is chosen. The thickness of the tank wall is estimated to be $35mm$ which result in a total tank weight of $2537.2kg$.

$$M = \rho(\frac{4}{3}\pi[(r+t)^3-r^3]+\pi(L-2r)[(r+t)^2-r^2]) \tag{16}$$

$$t = FoS\frac{Pr}{2\sigma_y} \tag{17}$$

1.3 Wing Design

In this section the main wing parameters are calculated, namely the wing loading, required power and dimensions based on the first weight estimation.
Firstly for the maximum wing loading based on the stall speed at sea level, a minimum value of 42.8 $\frac{m}{s}$ was defined for the stall speed. The maximum lift coefficient is 2.9 for a NACA 64(3)-618 with the plain flaps 35° extended and slats applied. The maximum wing loading thereby is $3.25 \cdot 10^3$ $\frac{N}{m^2}$, calculated with Equation 18.

$$\frac{W}{S} = 0.5 \cdot \rho_{sea} \cdot V_{stall}^2 \cdot C_{l,max} = 0.5 \cdot 1.225 \cdot 42.8^2 \cdot 2.9 = 3.25 \cdot 10^3 \ N/m^2 \tag{18}$$

Secondly the wing area is calculated with the use of Equation 19, resulting in a wing area of of 99.8 m^2. With a span of 30 meter, the aspect ratio is calculated with Equation 20, which is 9.0.

$$S = \frac{W_0}{\frac{W}{S}} = \frac{33082 \cdot 9.81}{3.25 \cdot 10^3} = 99.8 \ m^2 \tag{19}$$

$$AR = \frac{b^2}{S} = \frac{30^2}{99.8} = 9.0 \tag{20}$$

With a taper ratio of 0.5 this leads to a chord at the root of 4.4 m and at the tip of 2.2 m, calculated with Equation 21.

$$C_{root} = \frac{2 \cdot S}{(\lambda + 1) \cdot b} = \frac{2 \cdot 99.8}{(0.5 + 1) \cdot 30} = 4.4 \ m \tag{21}$$

With Equations 22 and 23 the mean aerodynamic chord is 3.4 meter and it's location from the fuselage is 6.7 meter.

$$C_{Aerodynamic} = \frac{2 \cdot C_r}{3} \cdot \frac{1 + \lambda + \lambda^2}{1 + \lambda} = \frac{2 \cdot 4.4}{3} \cdot \frac{1 + 0.5 + 0.5^2}{1 + 0.5} = 3.4 \ m \tag{22}$$

$$Span_{Locationmeanchord} = \frac{b}{6} \cdot \frac{1 + 2 \cdot \lambda}{1 + \lambda} = \frac{30}{6} \cdot \frac{1 + 2 \cdot 0.5}{1 + 0.5} = 6.7 \ m \tag{23}$$

Thirdly the power required is determined by the minimum cruise velocity at an altitude of 9000 m. The required power for the engine is calculated with Equation 27, where the lift and drag coefficient are defined in Equation 24 and 26. The zero lift drag is approximated with Equation 25 to be 0.0275, where C_{fe} is approximated 0.0055 and $\frac{S_{wet}}{S_{ref}}$ to be 5 [6]. This leads a lift coefficient of 0.6 and drag of 0.04. For the engine power a propeller efficiency, η, of 0.85 was taken into account, which lead to a motor power of 4.2 $\cdot 10^3$ kW. The RR AE2100P has been chosen as it can provide enough power for the job, specification can be found in Table 3. For the propeller attached to the engine a diameter of 3.5 meter is assumed.

$$C_l = \frac{2 \cdot 33082 \cdot 9.81}{0.46 \cdot 153^2 \cdot 99.8} = 0.6 \tag{24}$$

$$C_{d,0} = C_{fe} \cdot \frac{S_{wet}}{S_{ref}} = 0.0055 \cdot 5 = 0.0275 \tag{25}$$

$$C_d = C_{d,0} + \frac{Cl^2}{\pi \cdot e \cdot AR} = 0.0275 + \frac{0.6^2}{\pi \cdot 0.8 \cdot 9.0} = 0.04 \tag{26}$$

$$P_{required} = T_{required} \cdot V = \frac{W}{\frac{C_l}{C_d}} \cdot V = \frac{33082 \cdot 9.81}{\frac{0.6}{0.04}} \cdot \frac{550}{3.6} = 3.6 \cdot 10^3 \ kW \tag{27}$$

$$P_{motor} = \frac{P_{required}}{\eta} = 4.2 \cdot 10^3 \ kW \tag{28}$$

1.4 Cabin Design

The cabin is the section of the aircraft which is designed to locate and bring comfort to the passengers. It should bring all the comfort and facilities that travellers could need during the flight. It must accommodate: seats for passenger and flight attendants, lavatories, galleys and wardrobes inside of the cabin. It should have a capacity of 80 passengers as was stated in the requirements. In order to calculate the number of seats per row Equation 29 can be used.

$$n_{sa} = 0.45 * \sqrt{n_{pax}} \tag{29}$$

Where n_{pax} is the total number of passengers , and it results in approximately 4 seats per row. For the design of the airplane in this report the seat size selected is 18 inches, which is typically used by Airbus in all their small planes [7].
The other dimensions for the seats were designed, taking the recommendations of Raymer into account [6]. The dimensions for the seats and cabin are shown in table 4 and figure 2. The majority of the airplanes have small lavatories to take advantage of the available space. According to Raymer an area of $1.03m^2$ is recommended [6]. There is a formula to calculate the space of the galleys [6], using Equation 30 and taking the value for factor k_{galley} from [8] as 8. The area for the galleys is calculated to be 1.14 m^2.

$$S_{galley} = k_{galley} \cdot \frac{n_{pax}}{1000} + 0.5 \tag{30}$$

Finally the configuration for the wardrobes. There is an equation to define the space needed as a function of the numbers of passengers [6]. Following Equation 31 it results in $2.31m^2$.

$$S_{wardrobe} = 0.03 \left(1 - 3\frac{n_{aisle}}{n_{pax}}\right) * n_{pax} \tag{31}$$

Table 4: Description of dimensions

	Description	Dimensions [cm]
A	Seat width * 2	91.44
B	Height of the seat	140
C	Aisle Width	71
D	Seat width minus arm rest	43
E	Distance from the floor to the seat	45.72
F	Cargo height	97
G	Cabin passenger height	203
H	Diameter	300

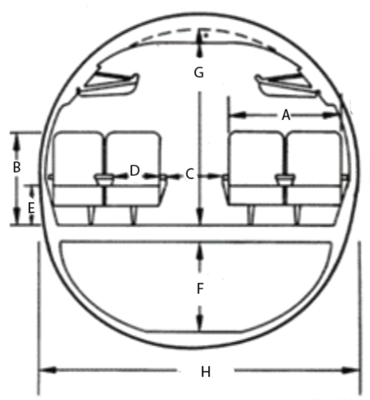


Figure 2: Cabin dimensions

1.5 Empennage Design

For the empennage, a conventional tail configuration is used, which is positioned at the back of the airplane. The tail volume coefficients for both the horizontal and vertical tail are chosen by comparing existing aircraft that, for the purpose of this aircraft design, make use of turboprop engines and are suited for passenger transport. The coefficients are 1.05 and 0.119 for the horizontal and vertical tail respectively. Using these coefficients, as well as some general dimensions of the main wing, the optimum tail moment arm (l_T) in order to minimize the aircraft drag and weight can be calculated using Equation 32. Note that K_c is a correction factor, which is selected to be 1.2 [9].

$$l_T = K_c\sqrt{\frac{4\overline{C}S\overline{V}_H}{\pi D_f}} = 1.2\sqrt{\frac{4\cdot 3.3\cdot 99.8\cdot 1.05}{\pi\cdot 3}} = 14.53 \tag{32}$$

Next, both the horizontal and vertical surface area can be calculated using Equations 33 and 34.

$$S_h = \frac{\overline{C}S\overline{V}_H}{l_t} = \frac{3.3\cdot 99.8\cdot 1.05}{14.53} = 23.79m^2 \tag{33}$$

$$S_v = \frac{bS\overline{V}_v}{l_t} = \frac{30\cdot 99.8\cdot 0.119}{14.53} = 24.51m^2 \tag{34}$$

Some general characteristics for the empennage are chosen. This is done by looking at existing literature. According to Tulapurkara, some value ranges are commonly used for subsonic airplanes [9]. The values are shown at the right side for the horizontal and vertical tail respectively.

1. AR= 4 and 1.5
2. λ= 0° and 20°
3. Λ= 0.5

1.5.1 Horizontal Tail

The optimal arm, which is the distance from the center of gravity to the aerodynamic center of the tail wing, is calculated using Equation 32. In order to position the horizontal stabilizer correctly, the position of the aerodynamic center needs to be determined. This can be done by taking the quarter chord point of the mean aerodynamic chord projected on the root chord. The MAC for a trapezoid shape can be calculated using Equation 35.

$$MAC = \frac{2}{3}C_r\left[\frac{1+\lambda+\lambda^2}{1+\lambda}\right] = \frac{2}{3}\cdot 3.2 \tag{35}$$

Figure 3 shows the general dimensions of the horizontal tail wing, as well as the position of the aerodynamic center including the distance from the leading edge.
Next, the required lift coefficient of the tail needs to be calculated using Equation 36.

$$C_{L_h} = \frac{C_{m_{owf}} + C_L(h-h_0)}{\overline{V}_H} \tag{36}$$

Where C_L is the aircraft cruise lift coefficient, $(h - h_0)$ is the distance from the aerodynamic center to the center of gravity, and C_{mowf} is the wing-fuselage aerodynamic pitching moment coefficient given by Equation 37. Here, AR is the aspect ratio of the main wing, $C_{m_{af}}$ is the moment coefficient of the main wing, λ is the sweep angle of the main wing, and α_t is the twist angle of the main wing.

$$C_{m_{owf}} = C_{m_{af}}\frac{AR\cos\lambda^2}{AR+2\cos\lambda} + 0.01\alpha_t \tag{37}$$

Substituting Equation 37 into Equation 36 and filling in the values gives -0.3577 for C_{L_h}.

$$C_{L_h} = \frac{-0.1388\frac{9.0\cdot\cos 0^2}{9.0+2\cos 0} + 1\cdot 10^{-8} + 0.524(-0.5)}{1.05} = -0.3577 \tag{38}$$

The angle of attack of the tail needed to realize this lift coefficient can be found using an Xfoil approximation for the NACA 64(2)015. This angle is approximated to be $-2.63°$.
When accounting for downwash, both the downwash angle ϵ_0 at zero attack and the downwash slope $d\epsilon/d\alpha$ need to be calculated using formulas 39 and 40. For these equations $C_{L_{aw}}$ is the lift slope curve.

$$\epsilon_0 = \frac{2C_L}{\pi \cdot AR} = \frac{2 \cdot 0.524}{\pi \cdot 9.0} = 0.0371 \quad (39)$$

$$\frac{d\epsilon}{d\alpha} = \frac{2C_{Law}}{\pi \cdot AR} = \frac{2 \cdot 6.76}{\pi \cdot 9.0} = 0.4782 \quad (40)$$

With these values, the required tail setting angle can be calculated using Equation 41.

$$i_h = \alpha_h - \alpha_f + \epsilon_0 + \frac{d\epsilon}{d\alpha}\frac{i_w}{57.3} = -2.63 - 2.1 + 0.0371 + 0.4782 \cdot \frac{1.5936}{57.3} = -4.68° \quad (41)$$

For α_h the approximation for the angle of attack of the horizontal tail was used. α_f is the angle of attack of the aircraft during cruise and i_w is the setting angle of the main wing. This a tail setting angle of $-4.68°$.
Finally, the design for the horizontal stabilizer can be verified by checking for the static longitudinal stability. The stability derivative is determined in Equation 42.

$$C_{m_a} = C_{Law}(h - h_0) - C_{Lah}\eta_h \frac{S_h}{S}\left[\frac{l_T}{\overline{C}} - h\right]\left[1 - \frac{d\epsilon}{d\alpha}\right] = -6.8019 \quad (42)$$

In this equation, $C_{L_{ah}}$ is the lift curve slope of the stabilizer airfoil at cruise and $\overline{C}$ is the mean aerodynamic chord of the main wing. Overall, the stability equation holds for this situation since the outcome is negative.

1.5.2 Vertical Tail

Both the MAC and position of the aerodynamic center are found using methods similar to section 1.5. Figure 4 shows the general dimensions of the vertical tail including the position of the aerodynamic centre on the tail surface, as well as the distance from the leading edge.

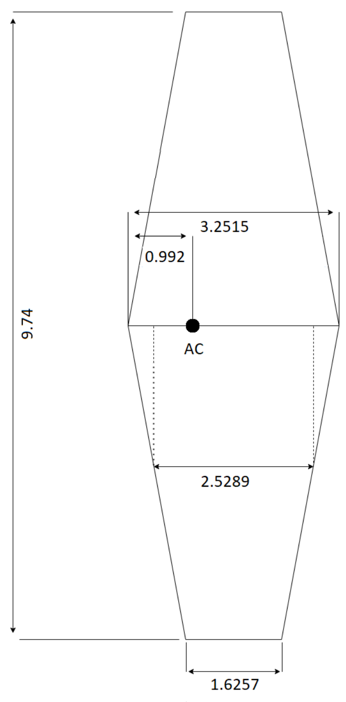


Figure 3: Horizontal tail

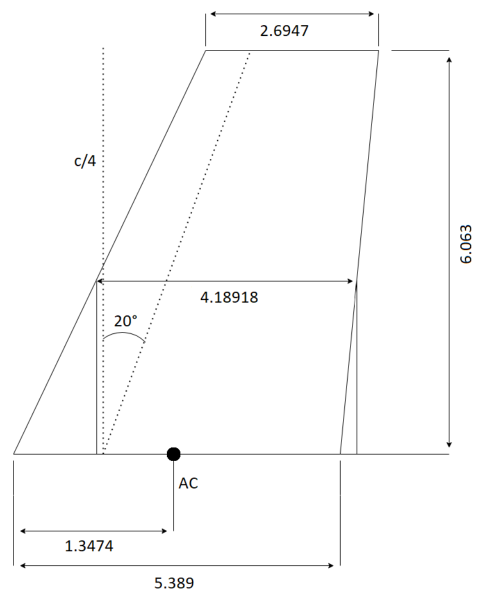


Figure 4: Vertical tail

1.6 Center of Gravity

To determine the position of the centre of gravity, the masses of individual components were estimated and the position of there centre of gravity relative to the nose of the plane. An overview can be found in table 5. With these masses and distances the position of the total centre of gravity is calculated by knowing that the total mass times the distance of the total centre of gravity, must be equal to the individual moments added. So the position of the centre of gravity is calculated by dividing the moment's by the total weight.
When the plane is flying it gets lighter as it looses fuel, which moves the centre of gravity forward and the plane stays stable. As it will never be behind the aerodynamic center.

Table 5: Components taken into account

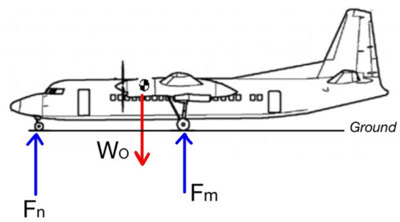
Component	Mass (Kg)	Distance (m)
Wing	4800	18
Engine's	1580	18
fuel	1068	22
Fuel tank	2537	22
Tail	3000	30
Fuselage	3300	15.5
Payload + Seats	9440	11.5
Crew	400	9
Nose wheel	267	2
Cockpit	800	1
Landing gear	850	18
Total	28042	16.66

1.7 Landing Gear

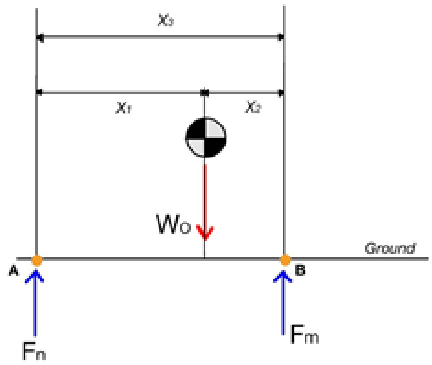
The landing gear is a fundamental part of the design of an airplane as it supports almost all the weight of the aircraft during take-off and landing. It is necessary to know the loads that will be exerted on the landing gear as a starting point. In Figure 5 the loads present on the airplane are shown. Where Fn is the force that the ground exerts on the nose wheel and Fm the force exerted on the two main wheels, this will be Fm/2 for each main wheel. W_O is the total weight which is applied on the center of gravity. To determine the unknown forces, Fn & Fm, equilibrium of moments will be used at point A, which gives Equation 43 as result,obtaining an Fm of 35429 lbf and Fn of 1621 lbf.

$$F_m = \frac{W_O * x_1}{x_2} \qquad \& \qquad F_n = \frac{W_O * x_2}{x_3} \tag{43}$$

With the information obtained above, the diameter of the wheels can be found [6]. For the main landing gear 4 wheels will be used on each side as it has to withstand heavier loads than the nose gear, for which 2 wheels will be used. The formula used to calculate the dimensions of the wheel is: $Diameter\ or\ Width = A * F^B$, where the values for A and B can be found on Table 7 and F is the load. The dimensions for the wheels are shown in Table 6.



(a) Loads on the Airplane



(b) Free body diagram of the forces exerted on the airplane

Figure 5: Loads and Free Body Diagram of the Airplane

Table 6: Wheels dimensions

	Diameter [in]	**Width [in]**
Main	33.5	11
Nose	18.5	7

Table 7: Constant for wheels[6]

	A	**B**
Wheel diameter (in)	1.51	0.349
Wheel width (in)	0.715	0.312

Shock Strut
The selected design is an oleo shock strut which will work as a damper to reduce the impact at landing. The dimensions were found in relation with the load exerted on the plane [10]. Using the loads, the area of the strut can be found with the following formula: $A_h = \frac{F_m}{p}$, where p is the static load which is 1500 psi found using the weight distribution approximation [10]. The outside diameter is chosen to be 30 % bigger than the inner. The results are shown in Table 8.

Table 8: Struts diameters for main and nose landing gears

	Inner diameter [in]	**Outer diameter**[in]
Main gear	5	6.5
Nose gear	1.5	2

1.7.1 Weight Estimation

The weight of the landing gear typically is 3 to 5% of the total gross weight of an airplane. The empirical relations to find the landing gear weight was developed for airliner-type aircraft in which a relation between the total gross weight and the weight of the main and nose landing gear was used [10]. Equation 44 and Equation 45 correspond to the main and nose landing gear weights in lbf respectively [10]. Of course the total weight is the summation of the main and nose landing gear weight. Using the equations given and with the correct conversion the results in Table 9 were obtained.

$$W_{MG} = 40 + 0.16W_g^{\frac{3}{4}} + 0.019W_g + 1.5 * 10^{-5}W_g^{\frac{3}{2}} \tag{44}$$

$$W_{NG} = 20 + 0.10W_g^{\frac{3}{4}} + 2 * 10^{-6}W_g^{\frac{3}{2}} \tag{45}$$

Table 9: Landing gear weights

	Main landing gear	**Nose landing gear**	**Total landing gear**
Mass [kg]	919.39	197.30	1116.69

1.7.2 Material Selection

The material selected for the landing gear should be able to withstand heavy landing and take-off conditions. The material therefore must have a high static strength, good fracture toughness, and high fatigue strength. As a final selection, aluminium 7049 T73511 was chosen, which is commonly used for manufacturing landing gears. This material has a yield stress equal to 442 MPa, which can withstand the stress of 34 MPa, which was calculated using the area of the struts of the main landing gear. A safety factor of 2 is used, to guarantee security during hard landings [11].

2 Aerodynamics

In this section first the lift and lift curve are explained of the chosen airfoil. Also the zero lift drag approximations during cruise and takeoff will be determined, after which the drag coefficient is determined for both situations. The lift and drag is required in order to determine the aerodynamic efficiency.

Lift and lift curve

The lift of the aircraft is created by the wings by using an airfoil. The airfoil chosen for the design is the NACA64(3)618, since it had good lift characteristics, good stall behaviour and is reasonably thick (0.18c) to make sure there is enough room for the internal structure to be placed. The slope of the airfoils lift curve is determined using XFoil and is estimated to be approximately $a_0 = 0.1192$ per degree (see figure 6). This is in a 2D case and will be different in reality, which is approximated to change according to the following relation:

$$a = \frac{a_0}{1 + \frac{a_0}{\pi A R e}} = 0.1186 \; per \; degree \tag{46}$$

The zero lift angle of attack for the NACA64(3)618 is $\alpha_{L=0} = -4.1519°$. This leads to the function for the linear region of the lift coefficient with respect to the angle of attack being:

$$C_L = a\left(\alpha - \alpha_{L=0}\right) = 0.1186\left(\alpha + 4.1519\right) \tag{47}$$

The lift coefficient during cruise can be determined using the approximation that the lift equals the weight. In that matter the lift coefficient during cruise should be equal to $C_L = 0.5240$. Using equation 47, this will result in an effective angle of attack $\alpha_{eff} = 0.2663°$. The geometric angle of attack is larger due to the 3D effects, the induced angle of attack should be added to the effective angle of attack. The induced angle of attack can be estimated using the following relation:

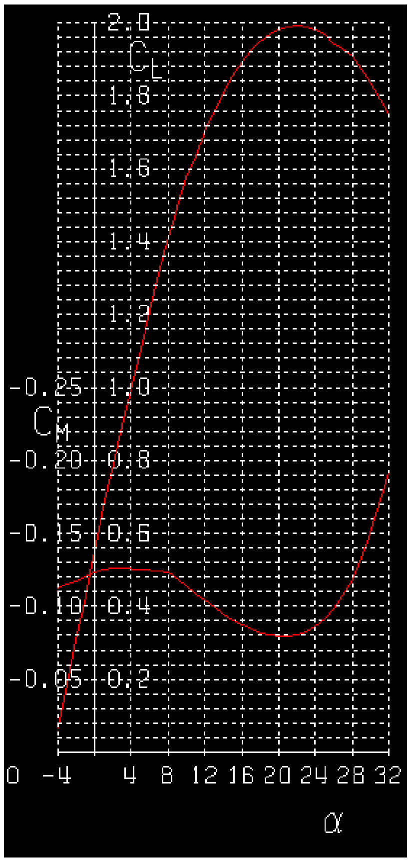


Figure 6: Polar plot NACA64(3)618

$$\alpha_i = \frac{C_L}{\pi A R e} = \frac{0.5240}{\pi \cdot 9.0 \cdot 0.8} = 0.023165 \; rad = 1.327° \tag{48}$$

The geometric angle of attack will be $\alpha_{geo} = \alpha_{eff} + \alpha_i = 1.5936°$.

Zero lift drag coefficient during cruise

During cruise all drag components can be determined. The fuselage and vertical tail are first of all estimated using a flat plate approximation causing friction drag to occur. For the fuselage drag the coefficient of friction is multiplied by a factor 2 to approximate also the pressure drag of the fuselage. For a flat plate, the coefficient of friction for fully turbulent flow can be approximated using:

$$C_f = \frac{0.074}{{Re_L}^{0.2}} \tag{49}$$

Therefore with the use of the Reynolds number of the vertical stabilizer and the fuselage, the friction coefficient can be approximated. The drag coefficient is equal to 0.0035 and 0.0026 for the fuselage and vertical stabilizer respectively. The total drag components can be determined with the use of:

$$Drag = C_d \cdot \frac{1}{2}\rho v^2 \cdot S \tag{50}$$

The areas of all the components must be determined to get to the drag forces. The fuselage is assumed to be a circle cylinder of 3 meter in diameter and 22 meter length, a halve sphere of radius 3 meter and a cone of height 6 meter and radius 1.5 meter. The wings and stabilizers areas are determined with their known geometries.

The drag of lifting parts is a different case due to the effect of lift induced drag. The total drag coefficient of a lifting surface can be estimated using:

$$C_d = C_{d,0} + \frac{{C_L}^2}{\pi ARe} \tag{51}$$

In this relation the $C_{d,0}$ is the drag coefficient when there is no lift generated and is airfoil dependent. For the determination of the zero lift drag coefficient during cruise, it is assumed the lifting surfaces do not create any lift. This means the lift induced drag of the main wing and the horizontal stabilizer are zero. The resulting force is determined filling in the zero lift drag coefficient in equation 50. Now all the forces can be determined in order to get the zero lift drag coefficient of the whole airplane using:

$$C_{d,0} \approx \frac{\sum F_{drag}}{\frac{1}{2}\rho v^2 S} = \frac{D_{wing} + D_{horstab} + D_{vertstab} + D_{fuselage}}{\frac{1}{2}\rho v^2 S} = 0.0209 \tag{52}$$

The Oswald efficiency factor can be estimated from the $C_{d,0}$ according to Scholz and Nita [12]. With $C_{d,0} = 0.02$ and $AR = 9$, this would lead to an Oswald efficiency factor of approximately $e = 0.8$, which seems to be a reasonable value to use.

Zero lift drag coefficient during landing/take-off
In the previous section a zero lift drag coefficient of 0.0209 is calculated. This is not valid for taking off and landing as the landing gear is deployed, which lead to an increase of the drag. The landing gear is assumed to be a cylinder with outer radius of the main gear being 0.16 meter and for the nose gear 0.05 meter, based on Table 8. The length of the main landing gear is 4 meter and 1 meter for the nose wheel. This leads to a Reynolds number for the main landing gear of $4.0 \cdot 10^5$ and nose landing gear of $1.2 \cdot 10^5$. This leads to a drag coefficient of 1.0 for the nose landing gear and 0.5 for the main landing gear, which is assumed to be the zero lift drag [13]. With Equation 53 the drag forces are calculated for the main and nose landing gear and the drag force of the aircraft with the zero lift drag coefficient of 0.0209 and wing area as reference area instead of length and diameter.

$$D = 0.5 \cdot \rho_{sea} \cdot C_{d,gear} \cdot V_{landing}^2 \cdot L_{gear} \cdot D_{gear} \tag{53}$$

The drag of the main landing gear is 257 N and nose gear is 39 N. The drag of the zero lift drag coefficient during cruise is $1.64 \cdot 10^3$ N.

$$C_{d,0,takeoff} = \frac{D_{landinggear} + D_{cruise}}{0.5 \cdot \rho_{sea} \cdot V_{landing}^2 \cdot S} \tag{54}$$

These values added together in Equation 54 leads to a zero lift drag coefficient of 0.0247 when the landing gear is deployed.

Total drag of the aircraft The total drag of the whole aircraft can now be approximated using the zero lift drag coefficient and the lift induced drag approximation, which will be mainly created by the main wing.

$$C_{d,cruise} = 0.0209 + \frac{{C_L}^2}{\pi ARe} \quad \& \quad C_{d,takeoff} = 0.0247 + \frac{{C_L}^2}{\pi ARe} \tag{55}$$

3 Performance

In this section the design verification is done with the use of formulas to calculate: stall speed, range, endurance and landing & take off distance and cargo space.

Stall velocity

The maximum stall velocity was defined as 42.8 $\frac{m}{s}$. To determine if this requirement was met, it was calculated with with the total weight of 28042 kg, wing area of 99.8 m^2 and Cl_{max} of 2.9. The result is, calculated with Equation 56, a stall speed of 39.4 m/s was obtained thus the requirement is met.

$$V_{stall} = \sqrt{\frac{2 \cdot W0}{\rho_{sea} \cdot Cl_{max} \cdot S}} = \sqrt{\frac{2 \cdot 28042 \cdot 9.81}{1.225 \cdot 2.9 \cdot 99.8}} = 39.4 m/s \tag{56}$$

Range & Endurance

With the use of Equations 57 and 58, the range and endurance were calculated. For this the SFC, which is based on kerosene mass, was divided by three as hydrogen has the same energy, but is three times lighter and to take head wind into account during cruise it was multiplied by 1.1 to assume 10% more fuel consumed. The lift over drag terms were determined for cruising at 550 km/h at 9 km with an air density of 0.47 kg/m^3, which result in a $\frac{C_l}{C_d}$ of 15.7 and $\frac{C_l^{1.5}}{C_d}$ of 11.2 based on Figure 8. The propeller efficiency is assumed to be 0.85, the wing area is 99.8 m^2. W_0 is the total weight and is $2.75 \cdot 10^5$ N and W_1 is the total weight minus the fuel weight and is $2.65 \cdot 10^5$ N. This results in a range of 1854 km and endurance of 3.4 hour, which is enough to meet the requirement of a range of 1000 km with head wind taken into account.

$$R = \frac{\eta}{SFC} \cdot \frac{C_l}{C_d} \cdot \ln \frac{W_0}{W_1} = \frac{0.85}{2.8 \cdot 10^{-7}} \cdot 15.7 \cdot \ln \frac{2.75 \cdot 10^5}{2.75 \cdot 10^5 - 2.65 \cdot 10^5} = 1854\ km \tag{57}$$

$$E = \frac{0.85}{2.8 \cdot 10^{-7}} \cdot 11.2 \cdot \sqrt{2 \cdot 0.47 \cdot 99.8} \cdot \left(\sqrt{\frac{1}{2.65 \cdot 10^5}} - \sqrt{\frac{1}{2.75 \cdot 10^5}}\right) = 3.4\ hour \tag{58}$$

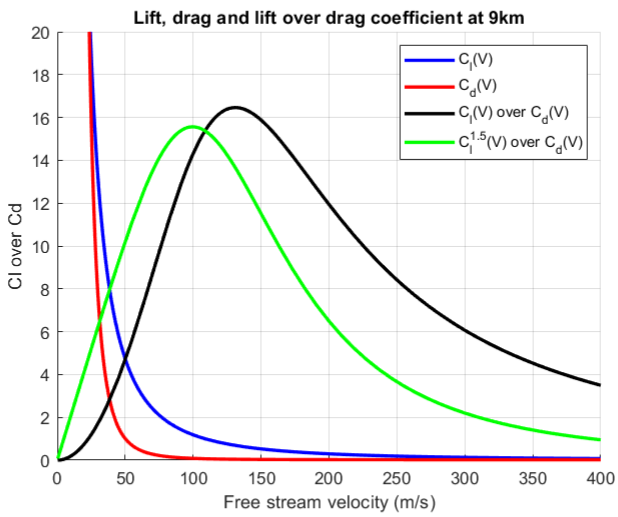


Figure 7: lift/drag components vs velocity

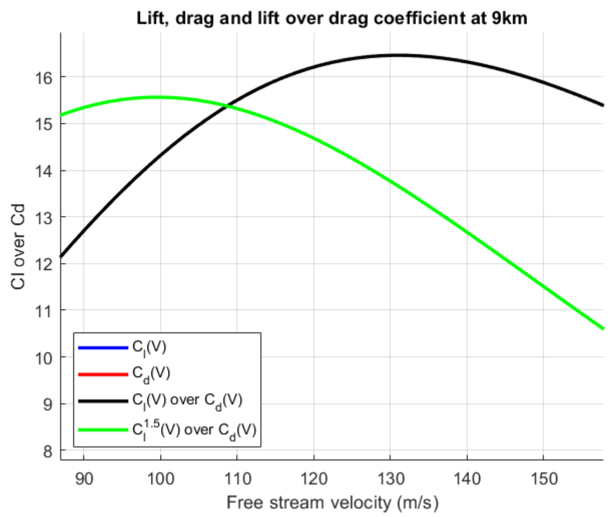


Figure 8: Zoomed in of Figure 7

Take-off

For take-off the ground effect is taken into account, the span is 30 meters and the height of the wing above the ground is defined as 4 meters, 1 meter clearance between the fuselage and the ground plus 3 meters of the fuselage diameter. This results in a ground effect of 0.82, calculated with Equation 59

$$\phi = \frac{(16\frac{h}{b})^2}{1 + (16\frac{h}{b})^2} = \frac{(16\frac{4}{30})^2}{1 + (16\frac{4}{30})^2} = 0.82 \tag{59}$$

The average required velocity for take-off is defined as 0.7 times 1.2 the stall velocity, which is according Equation 60, 33.1 m/s.

$$V_{liftoff} = 1.2 \cdot 0.7 \cdot V_{stall} = 33.1\ m/s \tag{60}$$

With this velocity the lift and drag are calculated, while riding on the runway. For this the coefficient of lift, $C_{l,\alpha=0}$, is assumed to be 0.524 as the angle of attack is assumed to be zero while it rides on the runway, based on Figure 6. The Oswald efficiency factor is assumed to be 0.8. Zero lift drag coefficient is 0.0247 as landing gear is out and comes from section section 2.

$$L = 0.5 \cdot \rho_{sea} \cdot V_{liftoff}^2 \cdot S \cdot Cl_{\alpha 0} = 0.5 \cdot 1.225 \cdot 33.1^2 \cdot 99.8 \cdot 0.524 = 3.5 \cdot 10^4\ N \tag{61}$$

$$D = 0.5 \cdot \rho_{sea} \cdot V_{liftoff}^2 \cdot S \cdot (Cd_0 + \frac{\phi \cdot Cl_{\alpha 0}^2}{\pi \cdot e \cdot AR}) = 0.5 \cdot 1.225 \cdot 33.1^2 \cdot 99.8 \cdot (0.0247 + \frac{0.82 \cdot 0.524^2}{\pi \cdot 0.8 \cdot 9.0}) = 2.3 \cdot 10^3\ N \tag{62}$$

With all data gathered and filled in into Equation 63 the lift off distance is found to be 587 meters. In this equation the thrust is $5.2 \cdot 10^4$ N and the friction coefficient of the paved runway is assumed to be 0.02.

$$S_{liftoff} = \frac{1.44 \cdot (2.75 \cdot 10^5)^2}{1.225 \cdot 9.81 \cdot 99.8 \cdot 2.9 \cdot (5.2 \cdot 10^4 - 2.3 \cdot 10^3 - 0.02 \cdot (2.75 \cdot 10^5 - 3.5 \cdot 10^4)} = 702\ m \tag{63}$$

This results in a required runway length of 703 meter, which is below 1500 meter, thus the requirement is met.

Landing

With the use of Equation 64 the landing distance is calculated. In this equation the touch down velocity is approximated by Equation 65, which is depending on the descent angle. The drag and lift are evaluated at 0.7 times the touch down velocity, as this is the average velocity during ground roll. the friction coefficient, μ, is assumed to be 0.4 for a dry runway and 0.2 for a wet runway.

$$S_{TD} = \frac{0.5 \cdot \frac{W0}{g} \cdot V_{TD}^2}{(D + \mu \cdot (W0 - L))_{0.7V_{TD}}} \tag{64}$$

$$V_{TD} = \frac{1.3}{cos(\theta)} \cdot V_{stall} \tag{65}$$

The drag is calculated for two cases, when the spoilers are applied and when forgotten. The zero lift drag used is 0.0247, because the landing gear is deployed. In the case when the spoilers are applied the induced drag is neglected as lift is destroyed. For the induced drag and the lift a lift coefficient for a zero angle of attack of 0.524 is used. The results are displayed in Table 10 for an approach angle of 0, 4 and 7.5°. It can be concluded that in all cases the requirement of a maximum runway length of 1500 is met and that the runway length requirement for an increased descent angle of 4 to 7.5° is met as it doesn't increase with 5%, but with 1.6% at maximum.

Table 10: Runway length for landing

Runway length [m]	0°	4°	7.5°
Dry & spoiler	328	330	334
Dry & no spoiler	381	383	389
Wet & spoiler	645	648	656
Wet & no spoiler	740	744	754

Luggage Space

In Figure 9 the interior with the luggage space is displayed, where BD is the floor where the cargo is placed below. From Table 2 the distance AB is defined as 0.97 meter and the distance CD is 1.5 meter, as this is the radius of the fuselage. As a result distance BC is 0.53 meter. With this information it is possible to calculate the angle α.

$$\alpha = \arccos(\frac{BC}{CD}) = \arccos(\frac{0.53}{1.5}) = 69.3^{\circ} \tag{66}$$

With this angle the length BD can be calculated.

$$BD = CD \cdot sin(\alpha) = 1.5 \cdot sin(69.3) = 1.40m \tag{67}$$

With this information the total area can be calculated by calculating the total area of 2 times α divided by the two triangles.

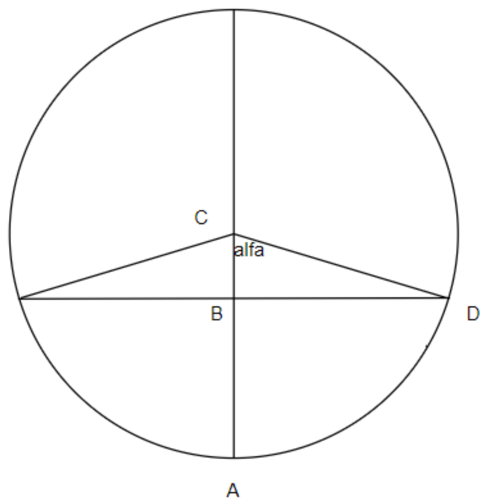


Figure 9: Fuselage interior

$$Area = \pi \cdot R^2 \cdot \frac{2 \cdot \alpha}{360} - 0.5 \cdot 2 \cdot BC \cdot BD = \pi \cdot 1.5^2 \cdot \frac{2 \cdot 69.3}{360} - 0.53 \cdot 1.40 = 1.98m^2 \tag{68}$$

For the total luggage volume the area is multiplied by the length of the cabin where passengers are seated. This length is 15 meter and thus there is a cargo volume of 29.67 m^3. The average luggage is assumed to be 161 kg/m^3 and the amount of luggage is 20 kg for 80 passengers [14]. The required volume for the luggage is 1600 kg divided by 161 kg/m^3, which is 9.94 m^3, thus this requirement is met.

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