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The design process of an environmentally neutral aircraft
Group 8

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January 14, 2021

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Chapter 1

Introduction

1.1 Goal & requirements

A climate neutral regional airliner concept is to be designed for the TITAN group, who are working for Airbus. An aircraft concept must be worked out and presented to TITAN, who will asses this concept. The concept itself should be promoted by reasoning on its performance. Apart from aerodynamic and structural performance, finances are to be taken into account as well. The goal of this project report is to enlighten on the choices and decisions made during the conceptualization of a climate neutral aircraft. The key aspects of the design are elaborated on as well as being evaluated and reviewed. Basic criteria are setup from given requirements. The found criteria corresponding to the requirements are shown in table A.1. The requirements given in table A.1 are evaluated and assessed. The criteria are first checked on overlap and relevance. They are also quantified. The unimportant or unquantifiable requirements are disregarded in table A.1. Criteria that overlap sufficiently are merged together as one and quantified as one. Also criteria that are not mentioned, but are of importance, are added to the list.

Also, the criteria are assessed upon possible improvement. They are scored on whether the target group would be interested in an improvement in this subject in order to get a better performing aircraft. The requirements that are to be improved to get more worth are graded from 1 to 5 where 5 is the criteria which can be improved on the best to get a more attractive airplane regarding production, cost and performance.

Relevance and Weighing

From table A.2, it can be clearly seen that operating costs and building costs are the most relevant parameters, it is clear that for companies excellent performance is desired. After taking into account the costs and safety aspects, certifications should be considered in order to be permissible to fly. Also performance factors like range and cruise speed are checked and weighed. Table A.2 represents the results of this analysis.

Evaluation

From the weighing and the improvement score, it can be found which requirements have priority when it comes to fulfilling them and/or improving them. When both table A.1 and table A.2 are combined, the following is noticed: cost is weighed heavily and wins almost every consideration. The cost is prioritized over many of the criteria. This holds for build cost, as well as operation time and cost. There can also be seen that range is prioritized over most criteria. From this table, there can be found which requirements to prioritize.

Chapter 2

Concept

Within the concept it is chosen to maximize efficiency. This means that the fuel cost should be very low. Since the fuel cost is a substantial part of the operating cost, this aircraft is built to have a low operation cost. The initial build cost of the aircraft will be high. This means that the break even point, i.e the point where the aircraft starts earning money, is rather late. Therefore, apart from the low operating costs, the design life should also be long to ensure that the aircraft can serve long after it reaches its break-even point to make an income.

The decision to conceptualize an aircraft with maximum efficiency is in accordance with the business plan. The final chosen concept is given in table A.4 and a rough sketch of the craft is presented in figure A.1.

Design aspect	Concept	Explanation
Engine type	Turboprop in pusher configuration	Large thrust is required at low velocity. Besides this, a pusher is more economical than a puller since it sucks in disturbed air.
Engine location	Close to the CG and on fuselage	So the bending stresses are as low as possible on the fuselage and less reinforcements are required, lowering the weight of the craft.
Fuel type	Hydrogen (liquid stored)	Hydrogen is a clean alternative to kerosene and has a higher energy density than batteries.
Wing design	Boxwing	To lower the induced drag on the craft (higher Oswald efficiency factor). Fuel consumption is reduced by approximately 9% [1].
Tail design	Horizontal stabilizer on aft wing. Vertical stabilizer between fuselage and aft wing.	No extra horizontal stabilizer is necessary, the control surfaces are on the aft wing. The vertical stabilizer also functions as a pylon between the fuselage and the aft wing.
Fuselage	Standard	In order to keep the costs down of the full aircraft.
Undercarriage	Tricycle	So the front wheel can steer during take-off and the distance between the propellers and the runway is increased.

Table 2.1: Choices made for the concept for maximum efficiency

Preliminary Weight Estimate

It is possible to use equation 2.1 to get a rough estimation of the gross weight of the aircraft, where the weight of the crew consist of 8 people of an estimated $90kg$ and the payload weight is 8800kg.

$$W_0 = W_{crew} + W_{payload} + W_{fuel} + W_{empty} \tag{2.1}$$

Regarding to the weight of the fuel, it can be assumed that the liquid hydrogen needed is 0.33 times the kerosene for the same amount of power produced. Looking at the ATR 72-500, it requires 2855 kg of kerosene to travel about 1000 km [2]. Besides this, by using a box-wing design approximately 9% less fuel is consumed compared in this case to the ATR 72-500. Resulting in a fuel weight of $W_{fuel} = 91\%(\frac{2855}{3}) = 866\ kg$

On the other hand, this weight reduction is counteracted by a considerable weight increase in the wing configuration, which according to [1] is around twice as heavy as conventional wing configurations. The wing weight contribution to the total empty weight of the ATR 72-500 is about 23.5%: $W_{BWextra} = 23.5\%(14150) = 3325\ kg$.

The weight of the hydrogen storage tanks should also be added. It is estimated from literature that a pressurized tank will weigh about 920 kg [3]. Hence, the empty weight of the aircraft is as shown in equation 2.2. Adding the previously mentioned values into the gross weight equation results in equation 2.3. According to this solution, the ratio W_{empty}/W_0 of the Box wing concept is 0.87.

$$W_{empty} = 14,150 + 3325 + 920 = 18395\ kg \tag{2.2}$$

$$W_0 = 720 + 8800 + 866 + 18395 = 28781\ kg \tag{2.3}$$

Chapter 3

Performance

In this section the design of a box wing aircraft is worked out fully. The design choices and modelling are to be explained to reason why the optimal result was achieved.

3.1 Structural analysis

The aircraft can be structurally analyzed by choosing a material and finding the relevant forces and moments.

Fuselage Design
To proceed with the stress analysis of the aircraft, we need to know the dimensions of the aircraft, including the fuselage. According to the requirements the airplane has to be able to transport 80 passengers. This means that if rows of 4 seats are used, 20 rows are necessary. The required diameter is given by equation 3.1, in which bench width (2 seats) is $1.03m$ and aisle width is $0.457m$. The distance between outer and inner diameter is $0.2m$ which means that the outer diameter should be 2.77 m.

$$d_{F,I} = (2 \cdot \text{Bench width} + \text{Aisle width}) + 2 \cdot 0.025 = 2.57m \tag{3.1}$$

For the seat pitch, a value of $K_{cabin} = 0.95$ is used (including galley and toilets). Which means that a total cabin length of $0.95 \cdot 20 = 19m$ is necessary. The total fuselage length is calculated in equation 3.2.

$$l_F = l_C + l_t + l_n = 31.32m \tag{3.2}$$

With length of cabin $l_C = 19m$, length of tail $l_T = 8.82m$ and nose plus cockpit length $l_n = 3.5m$ according to similar size aircraft [4]. The final design, taking the volume for the hydrogen tank from section 3.3.4, can be seen in figure A.2.

Stresses by cabin pressurization are given by equations 3.3.

$$\sigma_c = \frac{\Delta p R}{t} \qquad \sigma_l = \frac{\Delta p R}{2t} \tag{3.3}$$

Here, circumferential and longitudinal stresses on the fuselage are considered with Δp being the pressure difference, R the radius of the cylinder, and t the thickness of the fuselage skin, which values are listed in table 3.1. The results are $\sigma_c = 34.1MPa$ and $\sigma_l = 17.05MPa$.

Stres by bending torsion is calculated in equation 3.4 as a torsion of a hollow cylinder caused by side wind onto the vertical wing of the aircraft, which a torque applied to one end of the fuselage.

$$\tau = \frac{T \cdot R}{J} = 561.21Pa \tag{3.4}$$

p_o	30.8 kPa
p	80.06 kPa
R	1.385 m
t	0.002 m

Table 3.1: Values needed for σ_c and σ_l calculation

in which $T = 10459.9Nm$ is the torque caused by the wind assumed to hit an area $S_{vt} = 14.07m^2$ with a speed of $V_{wind} = 17.89m/s$ at sea level. While $R = 1.385m$ is the radius of the cylinder and $J = 23.95m^4$ is the polar moment of inertia of a hollow cylinder.

Stresses by wing bending can be calculated with equation 3.5 and 3.6.

$$\sigma_{max}(x) = \frac{-M(x) \cdot h}{I(x)} \quad (3.5) \qquad M(x) = 0.4x(L/4 - W_w) = 325714.24Nm \quad (3.6)$$

With $M(x)$ being the bending moment on the wing, h the half wing thickness, and $I(x)$ the second moment of area. For the bending moment, one of the front wings is taken isolated and the lift distribution for a tapered wing is considered elliptical, which means that the resultant force of the lift is located at 0.4 of the wingspan. The same holds for the weight of the wing. The equation for the bending moments is given in equation 3.6.

Using the wingspan $b = 26.457m$ and length of one wing $x = 13.23m$, the lift produced by one wing during take-off is $L/4 = 77500$, and the weight of one of the wings is $W_w = 15944.6N$. Once the bending moment is obtained, with the assumption of using a NACA 4412 airfoil and the mean geometric chord of 4 meters, the maximum thickness of the airfoil is given as 12% of $4m$ which is $h = 0.24m$. The second moment of area was calculated for an I-beam and is $I(x) = 0.002m^4$

Using above values, equation 3.5 gives $\sigma_{max} = -39MPa$. Here the assumption is made that each wing of the box wing configuration produces the same lift and therefore the total lift is divided by 4. It should be noted that this might not be the case as the front wings, being closer to the ground, produce most of the lift. The weight of the wings is taken according to the preliminary weight estimate previously made. As the iterative process will continue some changes might be needed.

Stresses by landing gear into fuselage can be calculated for two cases. The first case is the assumption that the pilot lands at an approach angle of 3°and that the landing gear has a stroke of $D_s = 0.5m$ to stop a mass of $m_{landing} = 27850kg$. With the landing speed of $V_{landing} = 55.56\frac{m}{s}$, a vertical speed of $V_{landing,v} = 2.9\frac{m}{s}$ is obtained. Consequently, the average sinking speed is around $V_s = 1.45m/s$ and the time that this process takes is $t_s = 0.345s$. This makes the deceleration equal to $a_s = 8.41m/s^2$. The slam force is then equal to:

$$F = m_{landing} \cdot a_s = 234.2kN \tag{3.7}$$

Which, applied to an assumed area of $0.1521m^2$, would turn into $\sigma_{LG} = 1.54MPa$.
The second case is the worst case scenario, where the moment of deceleration is from landing speed to $0\ \frac{m}{s}$. Although the requirements indicate that the runway length available is 2400m, it would be ideal to land and stop at half runway length. The energy to dissipate this brake is calculated with equation 3.8. The braking force for a 1200m runway length is then $F_{braking} = 35.8kN$

$$E = \frac{M_{landing}}{2}(V_{landing}^2 - V_{stop}^2) = 43 \cdot 10^6 kWs \tag{3.8}$$

Material selection

Figure A.3 shows the different materials that are used commonly in aircraft design. It shows that approximately 50 percent of the total weight of an airplane depends on composites, 20 percent aluminium, 15 percent titanium, 10 percent steel and 5 percent other.

The box wing is a really complicated wing configuration, therefore a strong structure is needed inside the wing. The most important properties to consider when choosing a material are the yield/tensile strength, the deform-ability, the stiffness, the fatigue limit and the fracture toughness. Above this section the maximum bending stress and bending moment were calculated. Taking a safety factor into account, those numbers can be compared to the yield strength of the needed material, using ANSYS Granta software.

In the end the material configuration will look similar to the configuration shown in figure A.3.

Skin material

For the skin of the wing a composite will be used (GLARE). Compared to aluminum, composites can be more lightweight and can cope better with fatigue and corrosion. Furthermore, in some parts a honeycomb structure is used to strengthen the wing and soft-spots in the frame.

Using Ansys Granta it is possible to compare various composites. 2 materials are considered: PEEK + Carbon fibre reinforcement (Polyetheretherketone) and Cyanate ester with carbon fiber reinforcement. Both composites are strong and have a low density and are already used in existing aircraft. Because PEEK + Carbon fibre reinforcement is the less expensive option, this will be the material we used for the skin.

Structural components

Structural components such as spars, ribs and the main construction of the fuselage will carry most of the loads. To be able to choose the right material for these structural component, we need to consider the bending stresses in the aircraft. The maximum stress in the wing is calculated to be -39 MPa and the stress in the fuselage has a circumferential stress of about 34.8MPa and a longitudinal stress of about 17.4MPa. When drawing a vertical line at the minimum stress the material needs to handle, it is possible to get a selection of materials that can be used for the structural component (see figure A.5), the strongest aluminum composite is aluminum 7068 T6511, wrought. This will be used for the structural components.

Force analysis

To understand the forces on an aircraft during straight level flight, a free body diagram is made. These can be found in the appendix in figures A.6 and A.7. Using these free body diagrams it is possible to set up the equations of equilibrium. Since we are talking about straight level flight, it means that the forces in y direction equals zero, which means that $L = W$. And when looking at the moments it is possible to come up with an expression for the moment about the center of gravity, given in equation 3.9.

$$M_{cog} = (L_1 + L_2) \cdot d_{cg-ac} + (M_1 + M_2) \tag{3.9}$$

$$d_{cg-ac} = (L_1 \cdot d_1 + L_2 \cdot d_2)/(L_1 + L_2) \tag{3.10}$$

From the aerodynamics calculations made in section 3.3.1 the total lift is calculated as 295000N. We assume the value of $d_1 + d_2$ is 11m, which is $\frac{1}{3}$ of the total length of the airplane. From this we calculate the distance between the total aerodynamic center and the center of gravity is to be 5.5m.

To calculate the moment around the center of gravity we first need to calculate the total moment due to the two aerodynamic moments in the wings. The total aerodynamic moment is given by equation 3.11(assuming the two wings have identical lift).

$$M_{tot} = [C_{m,0} \cdot (1/2) \cdot \rho_\infty \cdot V_\infty^2 \cdot c \cdot s]_{M1} + [C_{m,0} \cdot (1/2) \cdot \rho_\infty \cdot V_\infty^2 \cdot c \cdot s]_{M2} \tag{3.11}$$

In 3.1 the used values can be found, as taken from the aerodynamics performance analysis (section 3.3.1). The total aerodynamic moment is calculated to be $-7.46 \cdot 10^5 Nm$.

Finally the moment around the center of gravity can be calculated according to equation 3.9. The moment around center of gravity is calculated to be 876500Nm.

$C_{m,0}$	-0.10
ρ_∞	1.225 kg/m^3
V_∞	153 m/s
c	4 m
s	65 m^2

Table 3.2: Values needed for M_{tot} calculation

3.2 Weight estimate

To find more accurate results and optimize the structure, a second weight estimate has to be done. The initial weight estimate holds for a basic aluminium alloy conventional aircraft of a box wing configuration.
Previously it was found that the fuselage can be produced from composites. These materials have a lower density, which results in a weight that decreases.

The part of the fuselage of the aircraft that can be made from composites for a boxwing is responsible for 50% of the weight, if aluminium were to be used. This is, according to the previous weight estimate, 14155 kg. The carbon fibre that is used has a density that is approximately 40% lower. This means that the 14155 kg decreases to 9823.5 kg.

As calculated in section 3.3.2, 339 kg hydrogen is required in a tank of about 350 kg. This makes the weight of the fuel and tank 689 kg.

The new weight estimate results then in an aircraft of 22291 kg. Since the composite is a new material, additional safety factors should be applied, which increase the weight of the aircraft. The final estimation will be set at a mass of 25500 kg using a safety factor of 1.3, which results in a weight of 250.000 Newtons.

3.3 Performance analysis

During performance analysis, the critical aspects of aircraft performance are checked. These consist of a structural analysis, as well as an analysis on aerodynamics and propulsion. After an in depth analysis, these aspects are optimized for most efficient performance.

3.3.1 Aerodynamics

Cruise in 2D

Firstly cruise is analyzed. During cruise it is assumed that $L = W$. This characterises steady flight. This assumption will be used throughout the cruise analysis. An additional set of assumptions must be made first as mentioned next.

- Cruise velocity is equal to 550 km/h at 9000m (comparable to that of an ATR 72-500)
- The mean geometric chord is 4 meters
- A NACA 4412 airfoil was used

To maximize efficiency, cruise must take place at a point where the lift to drag ratio is at a maximum. For the previously mentioned parameters, this lift to drag ratio maximizes to a value of 170 at an angle of attack α of 4°. This can be seen in figure A.9. The C_l value corresponding to this angle of attack is 1.0. The weight was estimated in section 3.2 to be approximately 300.000 Newtons (initial weight estimate, in optimization, the 2nd weight estimate was used), which, according to the assumption for level flight, should be equal to the lift generated.

Using the cruise velocity given before and the mass density of air at 9000m ($0.4671 kg/m^3$), the critical surface area will be $55m^2$. This is the lowest possible surface area to have level flight at the given velocity.

Landing and Take-off in 2D

Let us assume that a velocity of 200 km/h is required for take off, which is a velocity often seen in comparable aircraft. Let us assume a wing area of 70 m^2 is used. Even though this might be less ideal for cruising (increase in 2D drag), a compromise must be made to allow take off and landing. The needed lift-coefficient to take off and land is around 2.2. According to figure A.9, the wing without flap would not be able to deliver this lift coefficient, since the aircraft would stall before delivering this amount of lift. Using flaps this value can more easily be reached.

With an XFLR5 analysis, in figure A.9, it is shown that if trailing edge flaps on a NACA4412 are deployed at 20° the maximum lift coefficient reaches 2.3 at an α of 10.6°, which would suffice.

Non lifting geometries

Up until this point, the non lifting geometries like the fuselage were aerodynamically neglected. Let us now analyze these. Using the assumption that the fuselage is a flat plate, and the estimation of the pressure drag being equal to the friction drag, the drag of the fuselage can be found.

During landing, an extra non lifting geometry plays a role. The landing gear is extended during landing and take off, creating extra drag forces that need to be accounted for. To estimate the drag of the landing gear, it is assumed that they resemble a circle cylinder. Using estimated dimensions, the drag can be approximated. Vertical stabilizers are also causing drag, but not providing any lift. This calculation and analysis is done in XFLR5, where a simple symetric airfoil is considered, with a typical surface area of 25% of the main wing.

Table 3.3: Drag of non lifting geometries

Components	Drag Cruise	Drag Take off
Fuselage	4742 N	1718 N
Vertical stabilizer	547 N	182N
Landing gear	0 N	1653 N
Total	5289 N	4419N

Now that all non lifting geometries are analyzed, the total drag during cruise of non lifting geometries is 5289 N and for take off and landing 4419 N. The drag contribution of each component can be found in 3.3.

Cruise 3D

The friction and pressure drag coefficient can be calculated using XFLR5 and at cruise are approximately 0.0055, but the lift induced drag is calculated using formula 3.12 when $\phi = 1$. It is assumed that a perfect elliptical airfoil is used, which in reality is not the case. Here e is the Oswald efficiency factor, where it is assumed that this is equal to 0.9. At the assumed velocity of cruise and with a first estimation of the Aspect Ratio of 10, the lift induced drag coefficient is approximately $C_{d,i} = 0.021$. This would result in a total drag coefficient of $C_d = 0.026$. This results in a total drag of $10150N$. This is the total drag and for the wing of the aircraft. Adding the additional drag of the non lifting geometries to this gives **the total drag of the aircraft**: 15439 N. This decreases the L/D ratio to an approximate of 19.1. Note that this airfoil is not optimized for lift induced drag yet (see section 3.4.1).

Landing and Take-off in 3D

The lift can be calculated, where the surface area is equal to $70m^2$ and the lift coefficient is 2.2, corresponding to the the required value to take off and land using the applied flaps. The 3D and 2D drag are calculated using the same method used for cruise, but for a lower velocity.

In the 3D drag term, an extra parameter comes into play. An additional term ϕ is introduced, which changes the amount of drag by changing the drag coefficient. This effect is shown in equation 3.12.

$$C_{d,i,land} = \frac{\phi Cl^2}{\pi e AR} \qquad \phi = \frac{(16h/b)^2}{1 + (16h/b)^2} \tag{3.12}$$

Here, ϕ takes the ground effect into account, which is essentially the obstruction of vortex formation of the wing by the ground. This means that less induced drag is expected.

However, this aircraft consists of 2 wings. When we assume both wings are situated at the top of the fuselage (worst case) $h = d_{fuselage} + h_{landing-gear} = 2.77 + 1 = 3.77\ m$, the h/b ratio would be 0.14. This results in a ϕ of 0.83. And this, in turn, results in a total drag during take off of 23358 N.

Conclusions

The aircraft is initially optimized for a 2D situation in steady flight. Later in section 3.4.1, a possible optimization for 3D drag will be discussed. This 2D optimization is done by decreasing the surface area of the wing, thus increasing the aspect ratio. This will decrease the pressure drag, as well as the friction drag. An optimized surface area, using a safety factor, of 70 m^2 is used to fly at a C_l of 0.771. This corresponds to the highest L/D ratio, which equals 19.1, where the non-lifting geometries are also included. The **total drag** of the aircraft is 15439 N. The **total lift** of the aircraft is $2.9 \cdot 10^5$ N.

Take off and landing performance are also analyzed. With flaps applied at an angle of 20 °, a lift coefficient of 2.2 can be reached to provide $2.9 \cdot 10^5$ N. A max lift of $3.1 \cdot 10^5$ can be reached by increasing C_l to 2.3 A **total drag** of $2.34 \cdot 10^4$ N is present on the aircraft during take off and landing.

3.3.2 Propulsion

Cruise

In this section the power required for cruising will be discussed. The maximum cruise altitude will be used for this and is set at 9000m according to the requirements. A cruising speed of at least 153 m/s is desired. It will be checked if the choice of wing dimensions and engine type are able to produce the amount of power required to fly at the desired speed. Equation 3.13 will be used for the power required (P_R) calculation.

$$P_R = \frac{1}{2}\rho V_\infty^3 S \left[C_{D,0} + \frac{(W/\frac{1}{2}\rho V_\infty^2 S)^2}{\pi e AR} \right] \tag{3.13}$$

The values to be filled in these equations are taken from section 3.3.1. The engine to be used is the PW 150A with an estimate for efficiency for a typical turboprop engine. The values are shown in table 3.4

Table 3.4: Values needed for P_R and P_A calculation [5]

S	70 m^2
AR	10
e	0.95
W	$2.82 \times 10^5 N$
$C_{D,0}$	0.019
ρ_0	1.225 kg/m^3
P_{eng}	3282 kW
η	0.85

The power available (P_A) from the two turboprop engines can be calculated as follows: $P_A = 2 \cdot \eta \cdot P_{eng} = 5579 kW$.

At 9000m, the density, $\rho = 0.467\ kg/m^3$. Ratios need to be taken to determine the P_A and P_R graphs at the cruising altitude. This can be seen in equations 3.14, with which the graph in figure A.10 can be plotted.

$$V_{\infty,9000m} = \left(\frac{\rho_0}{\rho}\right)^{\frac{1}{2}} V_\infty \qquad P_{R,9000m} = \left(\frac{\rho_0}{\rho}\right)^{\frac{1}{2}} P_R \qquad P_{A,9000m} = \frac{\rho}{\rho_0} P_A \tag{3.14}$$

From figure A.10 it can be seen that the engines chosen provide sufficient power to cruise at the required altitude. The minimum cruise velocity requirement of 153 m/s is also met with the engines being able to operate at velocities up to 160 m/s. For cruising at this altitude, about 2 MW of power is required.

Fuel

The range and endurance of the airplane depend on the amount of hydrogen stored in the plane and the efficiency of the engine. Our business plan (section 4) states that a range of 2000 km is required. With a cruise velocity of 153 m/s, this gives a required endurance of at least 3 hours and 40 minutes. As mentioned before, the power required for cruise is 2MW. This means that for 3 hours and 40 minutes of flying at cruise speed, about 26400 MJ of energy is required. To account for climbing and descending to 9000m, an additional 30 minutes times the average power between 0m and 9000 m is added, this is equal to $30 \cdot 60 \cdot 2.7MJ = 4860MJ$. This means a total of 31260 MJ is required. The energy density of iquid hydrogen is 120 MJ/kg, which means the airplane needs to carry at least 260.5 kg of hydrogen. To account for reserve fuel and fuel for unforeseen events, a safety factor of 1.3 is used. This means 339 kg hydrogen has to be stored. Since the density of liquid hydrogen is 71 $\frac{kg}{m^3}$, this corresponds to 4.7 m^3 liquid hydrogen. How this is stored can be read in section 3.3.4.

Take-off

The liftoff distance is the running length along the ground required by an airplane, starting from zero velocity, to gain flight speed and lift from the ground. In order to calculate the power required for landing and take-off, the drag D and lift L at $0.7 \cdot V_{LO}$, with V_{LO} equal to 3.15 should be used in equation 3.16.

$$V_{LO} = 1.2V_{stall} = 1.2\sqrt{\frac{2W}{\rho_\infty S C_{L,max}}} = 55.56m/s \tag{3.15}$$

The drag and lift at takeoff are equal to what was calculated in chapter 3.3.1. The value of S_{LO} can be chosen as the average length of runway an aircraft of similar size needs, which is 1500 m [6].

$$T = \frac{1.44W^2}{g\rho_\infty S C_{L,max} S_{LO}} + D - \mu_r(W - L) = 73.01kW \tag{3.16}$$

From there the power required during take-off can be calculated using $P_R = T \cdot V_{LO} = 4.06MW$.

3.3.3 Stability

Cruise

During cruise, the requirement in equation 3.17 must be met in order to ensure longitudinal stability.

$$\frac{dC_{M,cg}}{d\alpha} < 0 \tag{3.17}$$

This ensures that, without active control surfaces, the airplane will return to a given pitch at which $C_{M,cg} = 0$. To ensure this stability requirement the envelope of the center of gravity is taken into account. Where the most forward limit is referred to as the controllability limit and the aft point is named the neutral point. If the CG passes the neutral point the aircraft becomes unstable without any active control action to counter this behaviour. Also, it is required that at $C_l = 0$, $C_{M,cg} > 0$. This ensures that the equilibrium condition of the aircraft produces positive lift. For this analysis drag and thrust are both neglected.

The to be analyzed configuration is shown in figure A.11. Only the lift forces, moments from the airfoils and the mass of the plane are considered in this analysis. It is also assumed that the aerodynamic center is at 25% of the chord of the airfoil.
Taking the balance of moments results in equation 3.18. Converting this to their coefficients results in 3.19

$$M_{cg} = L_1(h - h_0)\bar{c}_1 - L_2 l + M_1 + M_2 = 0 \tag{3.18}$$

$$C_{M,cg} = C_L(h - h_0)\frac{\bar{c}_1}{\bar{c}} - C_{L_2}\frac{l' S_2}{\bar{c}S} + C_{M,1}\frac{\bar{c}_1 S_1}{\bar{c}S} + C_{M,2}\frac{\bar{c}_2 S_2}{\bar{c}S} = 0 \tag{3.19}$$

If we rewrite the following terms $s_i = \frac{S_i}{S_1+S_2}$ and $\bar{c_i'} = \frac{\bar{c}_i}{\bar{c}_1 s_1 + \bar{c}_2 s_2}$ for $i = 1; 2$, we end up with equation 3.20

$$C_{M,cg} = C_L(h - h_0)\bar{c_1'} - C_{L_2}\frac{l'}{\bar{c}}s_2 + C_{M,1}\bar{c_1'}s_1 + C_{M,2}\bar{c_2'}s_2 = 0 \tag{3.20}$$

applying the stability conditions results in equations 3.21 and 3.22.

$$-C_{L_2}\frac{l'}{\bar{c}}s_2 + C_{M,1}\bar{c_1'}s_1 + C_{M,2}\bar{c_2'}s_2 > 0 \quad (3.21)$$

$$h < h_0 + \frac{dC_{L_2}}{dC_L}\frac{l's_2}{\bar{c}'\bar{c_1}} \quad (3.22)$$

One can see from equation 3.22 that to have a large CG envelope, $\frac{dC_{L_2}}{dC_L}$ should be as large as possible. From studies, described in [7], it was shown that this value is mostly determined by the sweep-angles of the front and aft wing. Increasing the sweep-angle of the front wing with 35° increases this value with 10%.
The aft wing is swept forward and studies show that decreasing the forward sweep from −25° to 0° increases $\frac{C_{L_2}}{dC_L}$ with 6%.
So in order to have the largest envelope, the aft wing should have little sweep, while the front wing has as much sweep as possible.
One can also see that increasing l' would increase the envelope, this means that a boxwing with a wide airfoil spacing is more stable.

3.3.4 Other design aspects

Apart from the optimized and analyzed components and aspects, additional design choices must be made. Aspects like landing gear, storage of fuel and cargo as well as access to the aircraft will be discussed here.
Landing gear must withstand the loads which it is subjected to mentioned in section 3.1. Regarding placement of the landing gear; this must be done in a stable manner. Conventionally, the wheels are placed on the tip of the fuselage and under the wings. Since the chosen box wing design is unconventional, placing the wheels under the wing is not possible, since the center of gravity is then behind all wheels, which will cause it to tip. The turboprops are mounted between both wings. These are connected to the fuselage using relatively large pylons (to avoid the prop hitting the fuselage). This is a strong structural component. The landing gear itself will be mounted to the fuselage with a wide landing gear base.

When it comes to aircraft passenger access and storage of cargo; this is very conventional. The fuselage is one that is being used already in aviation for a long time (only with the addition of hydrogen tanks in the aft part). This means that in terms off passengers and cargo, all existing methods and means can still be used on this aircraft.

The liquid hydrogen can be stored in storage tanks. To avoid boil-off of hydrogen, a spherical or cylindrical shape should be used for the tanks to optimize the surface area to volume ratio. Currently, most ground based liquid hydrogen tanks have a perlite with vacuum insulation, resulting in boil-off rates of 1-5% per day. However, using a storage tank based on liquid helium technology (multi layer insulation and active shielding), the boil-off can be reduced to 0.01-0.05% per day [8]. These tanks do not exist yet for hydrogen, but we guess that they will be there when the plane will be used. We will assume that the volume of this tank is 10% of the volume of the hydrogen it stores. 4.7 m^3 would require a 5.2 m^3 tank, which can be achieved with a cylinder with a radius of 1m and a length of 1.66m. Taking in mind that weight of a 1.5 m^3 hydrogen tank is about 100 kg [3], we can roughly estimate that a hydrogen tank for 5.2 m^3 is about 350 kg. The tank can be stored in the tail as shown in figure A.2.

3.4 Optimization and iteration

The design is reviewed and checked on performance. Requirements that are not met, or can be improved on drastically, are reviewed and the design will be altered to meet or improve on this requirement. When changing a design aspect, the compromises made are discussed. Optimizations and/or iterations will be made on all big design aspects, to ensure the optimal design for the chosen aircraft configuration is reached.

3.4.1 Aerodynamics

Lift induced drag can be optimized for. This is done by calculating the total drag and finding the surface area corresponding to the lowest drag. For this, the following assumptions were made.

- NACA 4412 was used with an aspect ratio of 10
- Cruise speed is 153 m/s at an altitude of 9000m
- Weight is 250 000 Newtons

	General
Surface	$110m^2$
Wing	NACA 4412
Alpha	-1 °
Cl	0.4157
Lift	250.000 N = W
Total drag	6615 + 5289 = 11904 N

Table 3.5: Properties at cruise with given Surface area of wing

In figure A.12 can be seen that for all drags included, it is optimal to fly at an area of $110m^2$. Let us now apply this surface area to the aircraft designed. This changes the lift and drag properties of the aircraft. The results can be found in table 3.5.

In table ?? can be seen that the drag is decreased by a significant amount. This is due to the decrease of lift induced drag, which was originally caused by the high lift coefficient. Using a bigger area will decrease this. This progression will increase the L/D ratio, which will ultimately make the aircraft more efficient in cruise. The new L/D ratio is equal to 21 and is reached at a velocity of 153 m/s. This is be shown in figure A.13.

Figure A.13 shows that the designed aircraft flies at maximum efficiency at 153 m/s for the parameters given in ??. These values can be easily reached and applied during cruise, giving this aircraft maximum efficiency.

Increasing the wing area could also beneficially effect the lift properties during take off and landing. Having a bigger surface area could give the option of creating more lift during this phase. The maximum lift coefficient the airfoil (without flaps) can provide is 1.8. Before, using an area of $70m^2$, it was found that a lift coefficient of 2.2 was necessary to take off. Applying the new surface area of $110m^2$, a lift coefficient of 1.45 would suffice. This means that take off could take place without flaps. Note that flaps can still be applied to optimize for take off even further, by being able to decrease take off speed or decrease runway length.

Using, a lift coefficient of 1.45 gives a lift of 301 500 N during take off. Using a max lift coefficient, without flaps, of 1.8 would provide 366 900 N during take off, which would lead to shorter runway length. Drag during take off (using 1.45) will be 36 700 N.

3.4.2 Propulsion

In this section, the results of the optimization from the aerodynamics section will be used to see how it affects the propulsion of the aircraft. The values that will be used are shown in table 3.6.

Table 3.6: Values for propulsion calculations

S	110 m^2
AR	10
e	0.95
W	$250{,}000N$
$C_{D,0}$	0.019
ρ_0	1.225 kg/m^3
P_{eng}	3282 kW
η	0.85

After applying the same calculations as in section 3.3.2, the power required for cruising and takeoff were found. For cruising at 9000m, a similar amount of around 2 MW is needed. However the max velocity attainable by the engines under these new parameters is 148 m/s which is a little less than the required 153 m/s. This also does mean that the L/D ratio is slightly less ($\approx$ 20.8). The power required during takeoff increases to 4.92 MW which is still within the power available from the engines. Overall, the requirement of cruising at a velocity of 153 m/s is not met. However, this is a compromise for the aircraft to cruise as close as possible to it's most efficient L/D ratio. Getting larger engines that would be more expensive and heavier also does not make sense for the small velocity deficit.

3.4.3 Structural analysis

For most of the parts, stresses keep similar to the previously mentioned values. However, some of them change after the final weight estimation and new aerodynamic iteration values. Here the final values for the stresses will be presented using these values.

For the local buckling analysis, the boundary condition was assumed as a simply supported thin plate for which equation 3.23 holds,

Stress	**Value**	**Parameter**	**Wings**	**Fuselage**
Cabin Pressurization	$\sigma_c = 34.1MPa$ and $\sigma_l = 17.05MPa$	**Distance between stringers**	b = 0.2m	b = 0.29m
Wing Bending	$\sigma_{max,w} = 64.2MPa$	**Rib-to-Rib distance**	a = 0.6m	a = 1m
Fuselage Bending	$\sigma_{max,f} = 886.18kPa$	**Skin thickness**	t = 2mm	t = 2mm
Fuselage Torsion	$\tau = 704Pa$	**Young's modulus**	E = 200GPa	E = 200GPa
Landing Gear	$\sigma_{lg} = 1.32MPa$	**Poisson's ratio**	$\nu = 0.3$	$\nu = 0.3$

Table 3.7: All stresses and dimensions for the structural analysis.

When filling the equation with the values listed on the table above, the critical buckling stresses for wing and fuselage are found to be equal to $\sigma_{CR,w} = 86MPa$ and $\sigma_{CR,f} = 35.15MPa$ respectively. It should be mentioned that the material had to be changed as more strength was needed, although it is a carbon fiber reinforced material (carbon laminate) still. TORAYCA T800S can be used for the skin, which is the kind of material used by Boeing in the 787 fuselage and wings.

For the global buckling analysis, a simply supported beam was assumed for which its critical buckling load is given by equation 3.24.

$$\sigma_{cr} = \frac{k\pi^2 E}{12(1-v^2)} \cdot (t/b)^2 \tag{3.23}$$

$$\sigma_{cr} = \frac{\pi^2 E I_{xx}}{L_e^2} / A_{tot} \tag{3.24}$$

Parameter	Wings	Fuselage
Thickness stiffener	$t_2 = 1mm$	$t_2 = 1mm$
Stiffener type Z-beam dimensions	$l_1 = l_3 = 30mm$ and $l_2 = 40mm$	Same as wing
I_{xx} $stiffenerandskin$	$I_{xx} = 51152mm^4$	$I_{xx} = 51152mm^4$
Young's modulus	E = 73.1GPa	E = 73.1GPa

When filling the equation with the values listed in table 3.4.3, the critical global buckling stresses for wing and fuselage are found to be equal to $\sigma_{CR,w} = 465.6MPa$ and $\sigma_{CR,f} = 167.74MPa$ respectively. The same material mentioned in 3.1 was used for the structural components as ribs and stringers.

Chapter 4

Business plan

4.1 Market

Region

The aviation market is developing, and will nearly double to 7.8 billion air passengers in 2036 [9]. Figure A.14 shows the sources of the passengers over the next two decades. It can be seen that all major domestic markets will grow, and that China will surpass the US in amount of passengers. In numbers, China is expected to have 921 million new passengers for a total of 1.5 billion. US is expected to have 401 million new passengers for a total of 1.1 billion. Even though China will grow to a larger size, both options are worth looking into. To make a good decision, we need to look at which market has the most opportunities for the introduction of a new, environmentally friendly aircraft.

The big competitors for a regional airliner are high speed trains. High speed trains are quicker than airplanes if you want to travel short distances, they are very environmentally friendly and can take you directly to city centers. The train infrastructure in the US can be seen in figure A.15 . Unlike planes, long-distance trains usually depart only 1 time per day, meaning less flexibility for travelers. In the figure can be seen that, even though east and west are connected, direct connections between cities are lacking. The high-speed rail in China is the worlds longest high speed railway network and most extensively used (figure A.16).

Considering the popularity and the connections of the train system in China, it is chosen to focus on the market of the US. The current flight connections in the US can be seen in figure A.17. It can be seen that the central part of the US has a lot less connectivity than the rest of the US. Hence, there is a gap in the market to offer domestic flights to and within this central region. We can also improve the connectivity from major airports to smaller airports in the center of US by offering those flights as well. From this decision, the range can be determined, which will be about 2000 km.

Customer group

Our target group is the middle-class, environmentally conscious traveler. The prices of the aircraft can exceed those of a budget-airline, but should not be too expensive. Because our plane will only have space for 80 passengers, it is chosen to not have a business class.

Marketing strategy

To airliners: The big point that will make our plane stand out is the fact that it is environmentally friendly. It will use sustainable energy sources and as technology develops, those will only get cheaper the coming years [10]. It will also attract a new group of travelers who don't like to travel by plane because of so called 'flight shame'. The fact that the plane will travel to small airports, will cause a low turn around time, which makes it possible to have more flights per day. This will create more revenue.
To passengers: Passengers can fly in a new and exciting designed aircraft, which is still very safe to use. They won't have to feel guilty about leaving a big carbon footprint since there will be 0 emission. The aircraft is especially attractive to people who want to travel from and/or to smaller airports.

4.2 Financial

Production

The chosen boxwing configuration focusess on operation efficiency. It consumes low amounts of fuel and is compatible with smaller airports. This means that a compromise is made where production cost is sacrificed for efficiency and compatibility. Still, there was attempted to keep the production cost as low as possible. A cost estimation can be made by using data[11] of existing aircraft. For a Boeing 777-200, a cost estimation for different components per unit mass was found. A boeing 777-200 is a conventional airliner, widely used in aviation. The size of this aircraft is large compared to the designed aircraft in this research, however the cost estimations are per unit mass. For a rough estimation the following estimations in figure A.18 was found to be valid for a regional airliner as well;

Note that this estimation holds for a conventional aircraft. In this case, the designed aircraft will be a hydrogen powered aircraft with a boxwing design configuration. This increases the cost for the payload (fuel storage) as well as the turboprop engine. Additionally, the fuselage is created from composite, which will add cost as well [12] (See figure A.19).

Every components is analyzed and a weight was found for each of these. Using the data in figure A.18, the cost of each component can be found. Note that initally all components must be designed, which will be 40% of all costs. Once an aircraft is designed, it only needs to be produced. This production cost will determine the price of the aircraft if it were to be bought by an airline. Note that the engine has a total cost being equal to production cost. This is since the engine is simply bought, and does not have to be engineered and/or developed.

This means that the total aircraft will have a production and development cost of **54.9 millions dollars**. The production cost itself will be 40% less, according to figure A.18, which results in **34 million dollars**. This will be the price of the aircraft if it is to be bought by an airline.

Table 4.1: Total cost of aircraft components in million dollars

Components	Total cost	Production cost
Wing	33	19.8
Fuselage	13.8	8.3
Engine	2.6	2.6
Payload	5	3
Landing gear	0.5	0.3
Full aircraft	54.9	34

Let us now compare this to similar aircraft. An ATR 72-500, which is a turboprop aircraft of similar size, has a unit cost of 26 million dollars [13]. This is 20% cheaper than the newly designed aircraft. The IPTN CN-235 is also a turboprop engine aircraft that is meant for military purposes. This has a unit cost of 34 million dollar, according to the manufacturer. From these comparisons there can be found that the bowxwing aircraft has a relatively high build cost for its size and specifications. However, it is still feasible and fitting within the financial decisions by large aircraft companies. This aircraft benefits mostly from its efficiency, where it can compensate from the initially high price per unit.

Operating

Most parts of the operating costs will be similar to that of other airplanes the same size. However, the price for the fuel cost will be different. According to the International Energy Agency, hydrogen production through renewable energy costs 3.0 to 7.5 kg/USD [14]. We can predict that as research into renewable energy sources continues, production costs will decrease over the years. However, transmission and distribution costs should also be accounted for. Therefore we assume a price of 5.0 kg/USD in 2035. A full tank would then cost 1695 dollars. A full tank for a similar size and range aircraft, the ATR 72-600, is 5000 kg, which is equal to 6172 L. The price for jet fuel in December 2019 is 1.89 USD per gallon [15], which is 0.5 USD per liter. A full tank for an ATR 72-600 would cost 3063.5 USD. Also, turnaround time will not differ too much compared to conventional aircraft (which is around 30 minutes, approximately the same as for an ATR 72-500).

Conclusion

Initially, the price to pay for a single aircraft is high. It has a relatively high cost, comparing it to similar aircraft of the same size. Though, it is not unfeasible and is still within the range of interest for most airlines. Its operation cost on the other hand is low, comparing it to other conventional aircraft. Due to its efficiency it uses a low amount of fuel, which dominates operation cost. Due to this low operation cost (and therefore on the long term low ticket prices), it can rival with train and car transport in the designated region. Also, turnaround time will not differ too much compared to conventional aircraft.

Chapter 5

Conclusions

5.1 Final Design

The designed aircraft is shown in figure 5.1. A boxwing type aircraft with turboprop engines in a pusher configuration is designed. The essential dimensions are shown in the figure.

The aircraft structure is designed using carbon fibres and aluminium alloys. The fuselage and wings are manufactured from carbon fibres (TORAYCA T800S). This is able to provide strength, while still being lighter than using aluminium alloys. The structural components are made of aluminium (AL 7068 T6511), due to its high load properties and cheaper manufacturing.

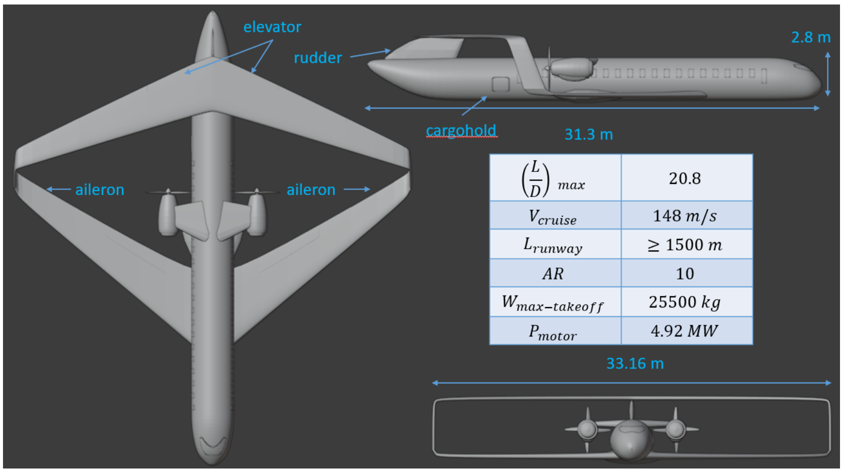

Figure 5.1: The final design with useful specifications

The aerodynamic properties and parameters can be found in figure 5.1. The boxwing exceeds great values in this aspect. Its L/D ratio of 20.8, exceeds efficiency of conventional aircraft by a lot, which makes it perform very well in operation. Apart from that, using 2 turboprop engines, it requires a short runway length of 1500 m, where the engines provide 4.92 MW of power.

Financially, this performance is beneficial in operation cost. Assuming the aircraft will be bought in sufficiently large amounts, the aircraft will have a unit cost of 36 million dollars. This is a high initial investment, but can be quickly earned back, due to its low operation cost, because of its efficiency.

5.2 Suitability for Target

During stress calculations it was assumed the cabin is pressurized to comfortable pressures. Also, seat-pitch is taken into account, which means that people have enough leg-room. This means that cabin comfort, mentioned in the requirements, is taken into account. Also, turnaround time, as discussed, will not differ too much compared to conventional aircraft. Which means that the airplane makes most use of its service life by flying as frequent as possible (increasing revenue).Thirdly, the minimum requirements are met regarding minimum payload, cruise velocity, runway length, ascent/ descent angle, costs and range. The appeal of this aircraft is that it not only runs completely climate neutral but also performs at a higher aerodynamic efficiency than other typical aircraft. Due to this higher aerodynamic efficiency, it also more fuel efficient which keeps the operating costs lower than typical aircraft. This would be a good selling point for passengers who are looking for more environmentally greener methods of travel while still being able to experience comfort and the technological advances in aviation. For Airbus, who are looking for an efficient aircraft with low operating costs, this concept aircraft fits the mould perfectly.

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Appendix A

Appendix

Criteria from Requirements	Quantify	Improvement score
A. Minimum payload	8800 kg	4
B. Minimum Cruise Velocity	550 km/h	4
C. Minimum Cruise Altitude	9000 m	3
D. Runway length	2400 m	2
E. Maximal ascent and descent angle	7.5 degree	2
F. Minimal cabin pressure	0.8 ATM	N/A
G. Energy Storage	t.b.d	2
H. Minimal operation time	36 hours	3
I. Operating cost	$ 2000 /hour [16]	5
J. Turn around time	30 min	3
K. Design life	20000 flight hours	3
Extra's		
L. Build cost	500 Million [17]	5
M. Temperature in Cabin	20 C	N/A
N. Minimum Range	1100 km	4

Table A.1: Design requirements

Criteria	A	B	C	D	E	F	G	H	I	J	K	L	M	N
A	-	A	A	D	A/E	F	A	A	I	J	A	L	A	A/N
B		-	B	D	B	F	B	B	B	J	K	L	B	B
C			-	C	C	F	C	C	I	C	C	L	C	N
D				-	D	D/F	D	H	I	J	K	L	D	N
E					-	F	E	H	I	E	K	E	M	N
F						-	F	F	F	F	F	F	F	F
G							-	H	I	J	K	L	M	N
H								-	I	H	H	L	H	N
I									-	I/J	I	L	I	I
J										-	K	L	J	N
K											-	L	K	N
L												-	L	L
M													-	N
N														-

Table A.2: Pairwise Comparison

Functions	Solutions	Composed of	Specific solutions				
Provide power	Have engines	Engine	Hydrogen piston	Electric prop			
		APU	None				
	Have fuel	Fuel type	Hydrogen	Electricity	Nuclear		
		Fuel storage	Battery	Fuel cell	Cryogenic hydrogen		
Provide lift/maneuverability	Wings	Wing Chord	Constant	Tapered	Constant with tapered	Eliptical	
		Wing sweep angle	None	Backwards	Forwards		
		Wing angle	Dihedral	Anhedral	Gull wing		
		Flaps	Plain	Split	Slotted	Leading edge	
		Aelerons	Single Acting	Wing tip	Frise	Diifertential	
		Airbrake	None	Yes			
	Tail	Tail Configuration	Conventional	Cruciform	T-Tail	Carnard	VXY Tail
		Rudder	Twin tail	Twin Boom	Wing mounted		
		Horizontal stabilizers					
Provide cargo capabilities	Fuselage	Cabin					
		Cargo space					
Provide landing and take off capabilities	Undercarriage	Landing gear	Multi bogey	Tricycle	Taildragger		

Table A.3: Functional analysis

Design aspect	Concept	Explanation
Engine type	Turboprop in pusher configuration	Large thrust is required at low velocity. Besides this, a pusher is more economical than a puller since it sucks in disturbed air.
Engine location	Close to the CG and on fuselage	So the bending stresses are as low as possible on the fuselage and less reinforcements are required, lowering the weight of the craft.
Fuel type	Hydrogen (liquid stored)	Hydrogen is a clean alternative to kerosene and has a higher energy density than batteries.
Wing design	Boxwing	To lower the induced drag on the craft (higher Oswald efficiency factor). Fuel consumption is reduced by approximately 9% [1].
Tail design	Horizontal stabilizer on aft wing. Vertical stabilizer between fuselage and aft wing.	No extra horizontal stabilizer is necessary, the control surfaces are on the aft wing. The vertical stabilizer also functions as a pylon between the fuselage and the aft wing.
Fuselage	Standard	In order to keep the costs down of the full aircraft.
Undercarriage	Tricycle	So the front wheel can steer during take-off and the distance between the propellers and the runway is increased.

Table A.4: Choices made for the concept for maximum efficiency

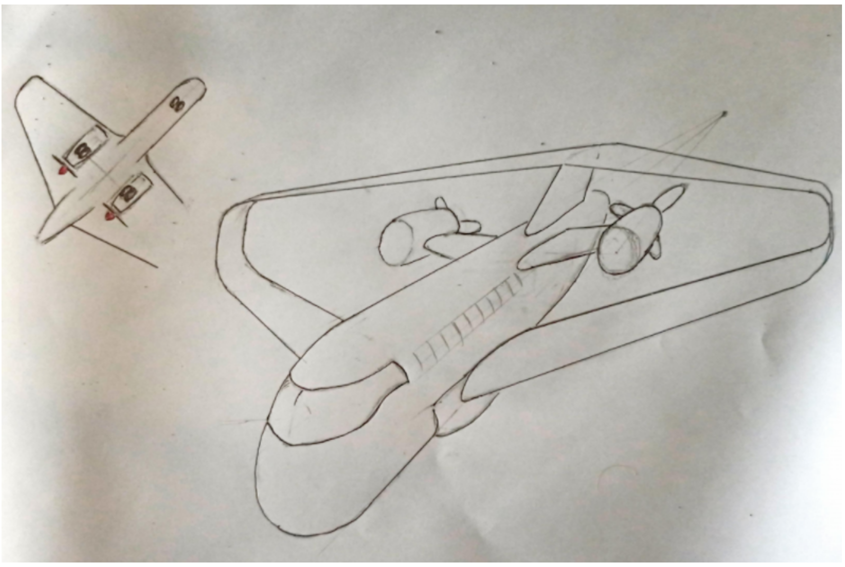

Figure A.1: A sketch of the maximum efficiency aircraft

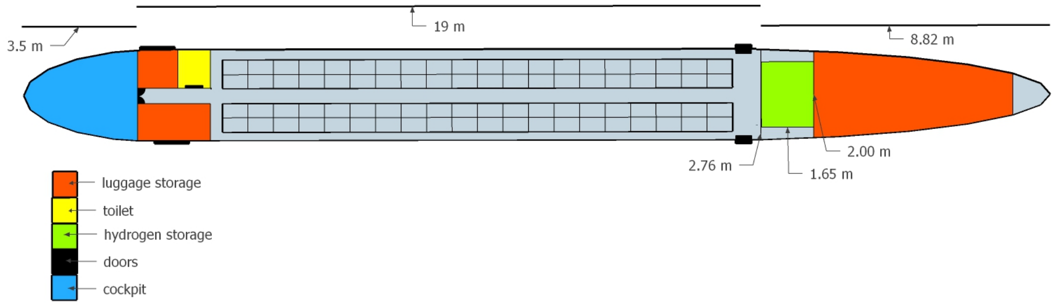

Figure A.2: Plan view of the designed fuselage, on scale

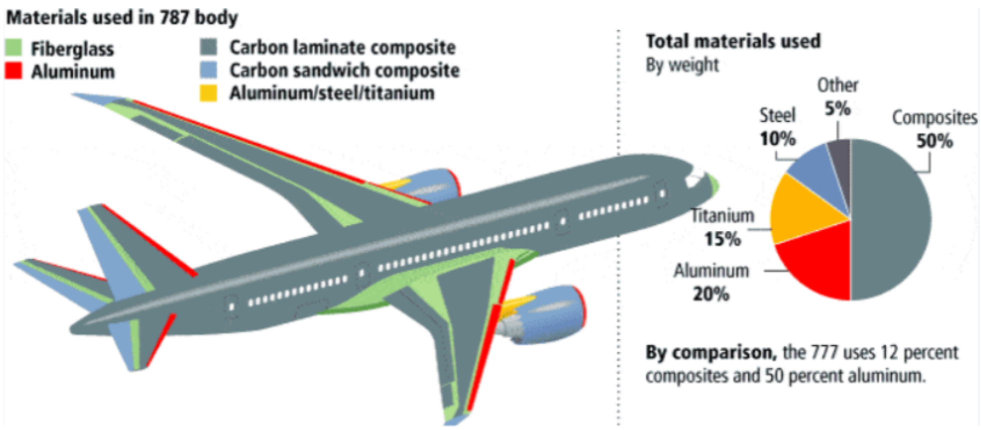

Figure A.3: materials used in airplane

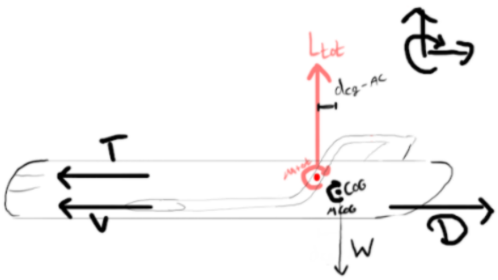

Figure A.6: Schematic of forces on an aircraft

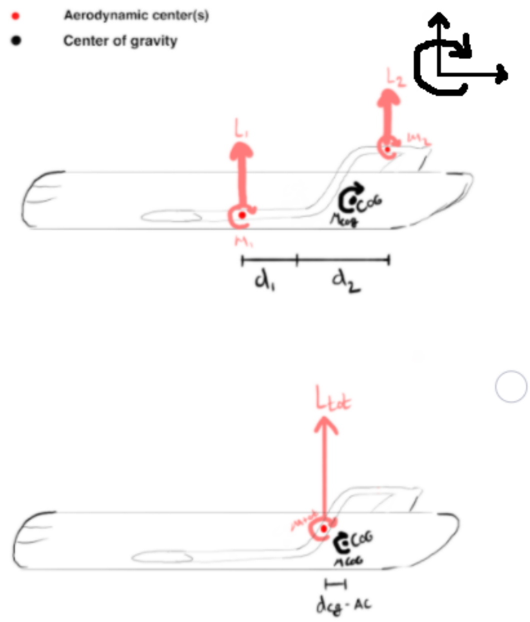

Figure A.7: Schematic of forces on an aircraft in depth

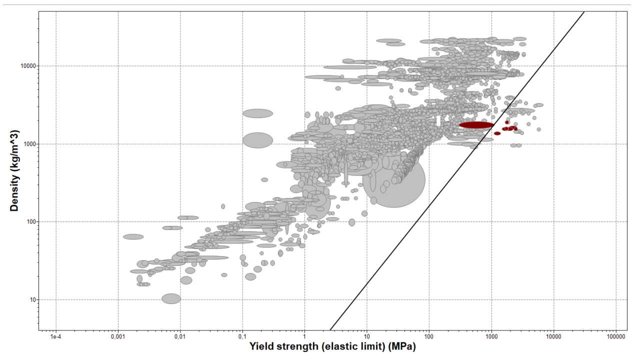

Figure A.4: Ansys granta results for skin

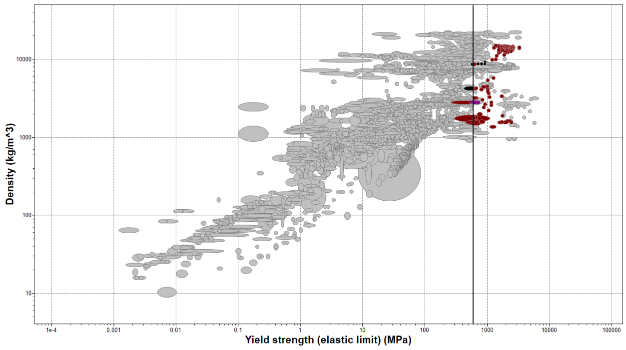

Figure A.5: Ansys granta results for structural components

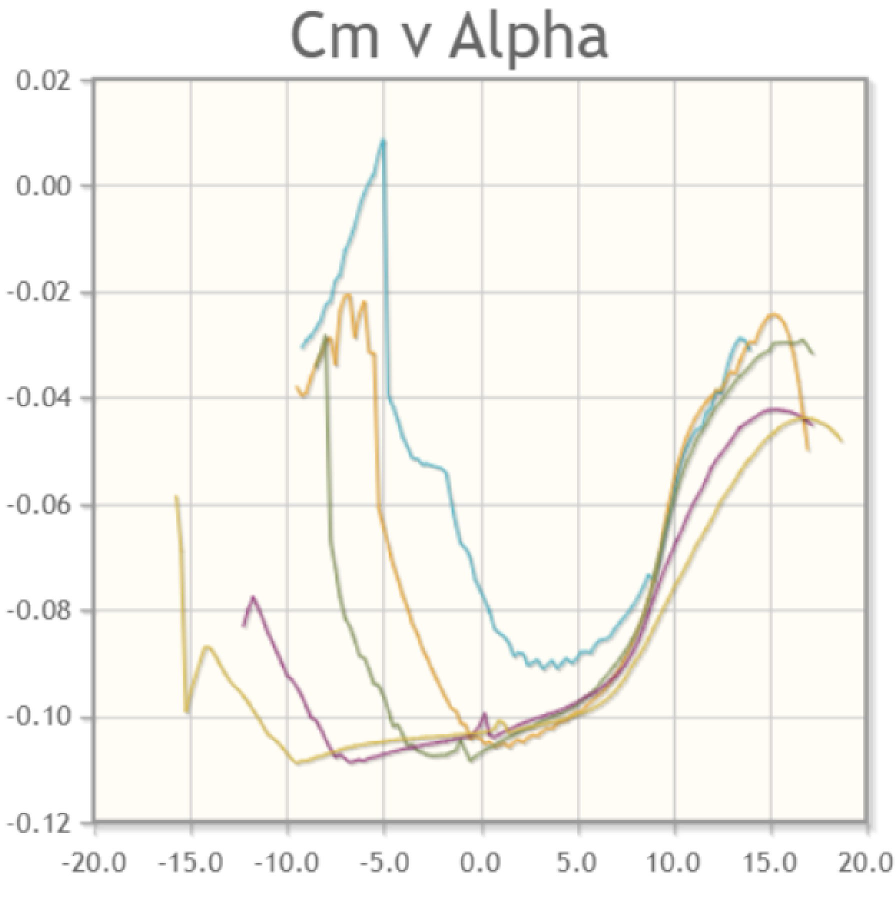

Figure A.8: moment coefficient vs the angle of attack

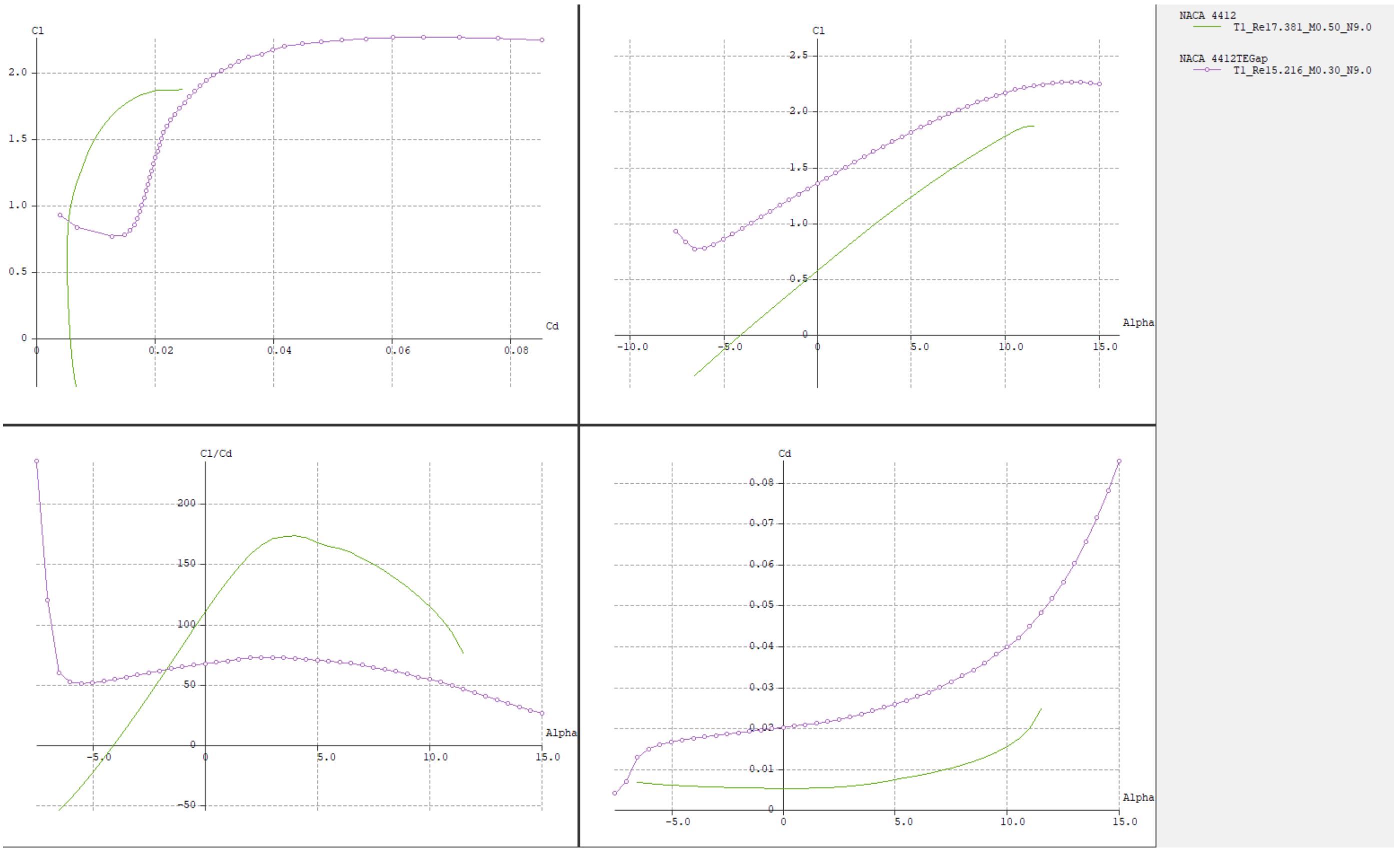

Figure A.9: The properties of the NACA4412 airfoil, with (purple) and without flaps (green) deployed to 20°

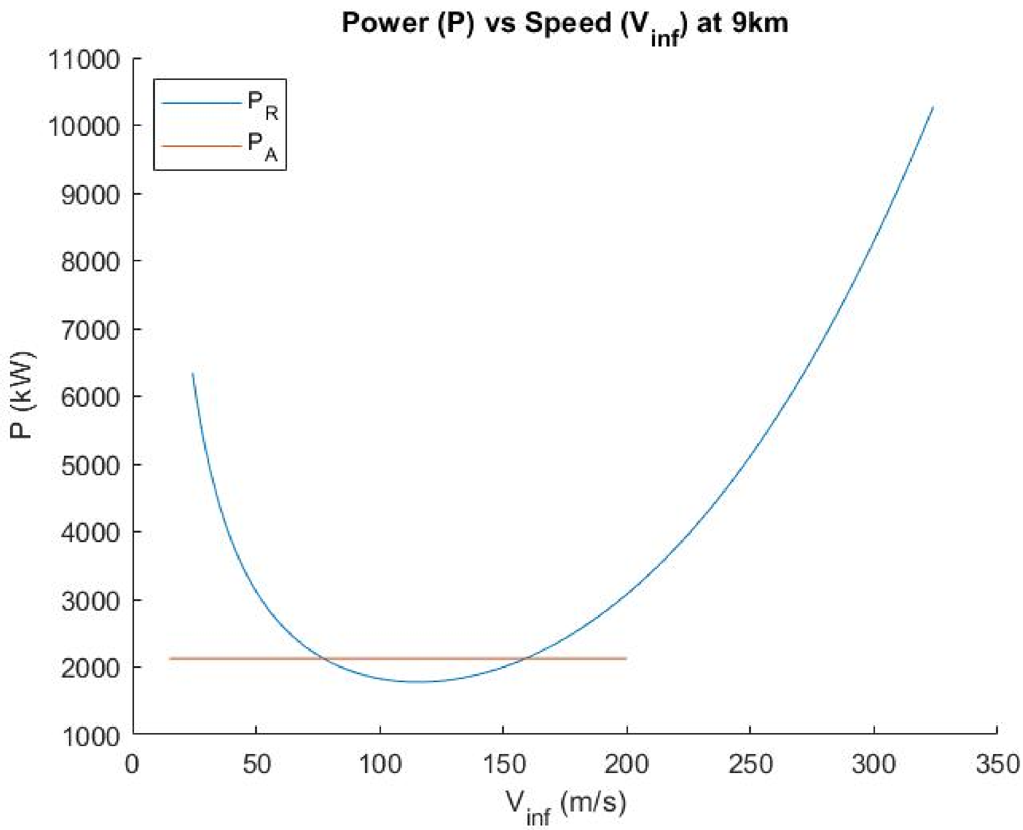

Figure A.10: Power graphs for cruising at 9000m

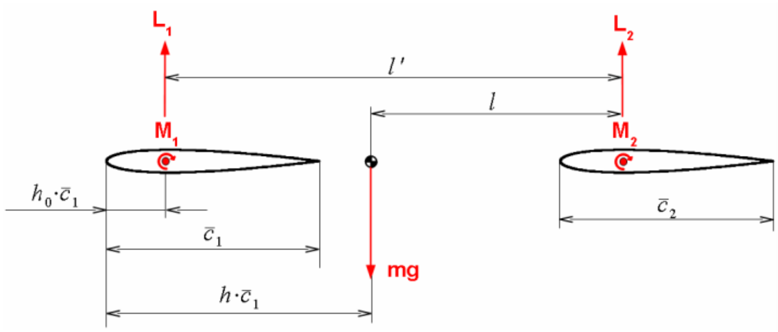

Figure A.11: The force balance of a dubble wing configuration [7], by *D. Schiktanz and D. Scholz, 2011, Aircraft Design and Systems Group, Hamburg University of Applied Sciences, Hamburg, Germany*

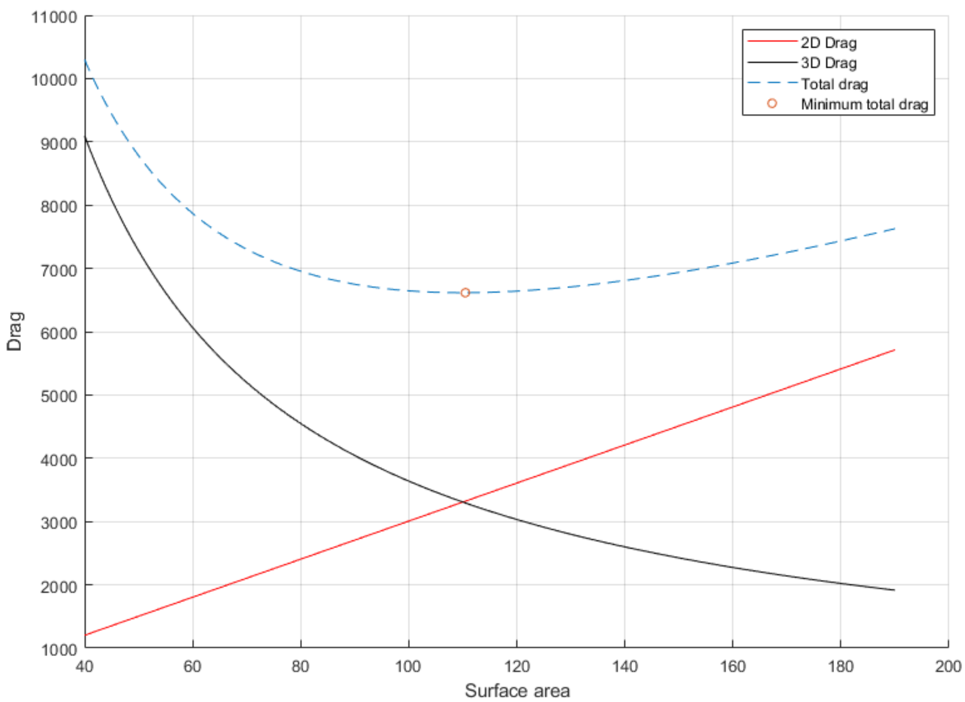

Figure A.12: 3D Drag optimization

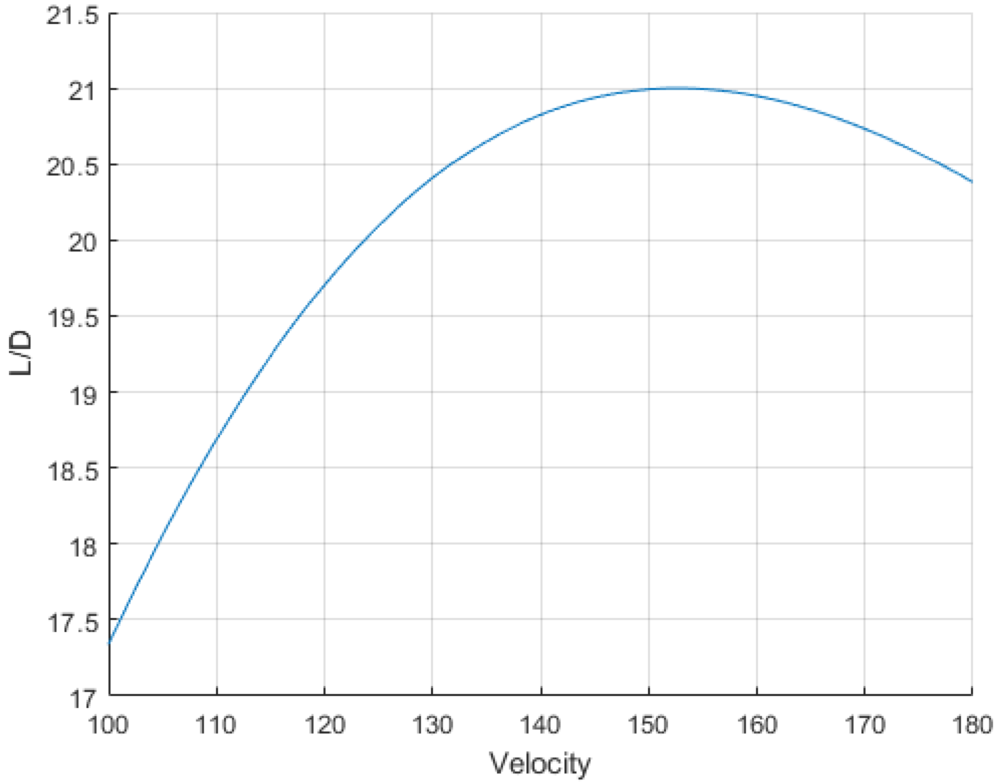

Figure A.13: L/D ratio for a set of velocities

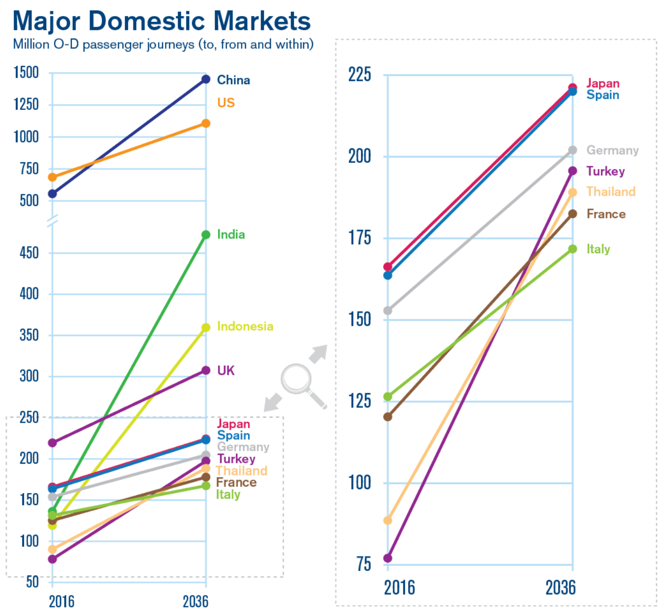

Figure A.14: Prediction of developing aviation market, by IATA [9]

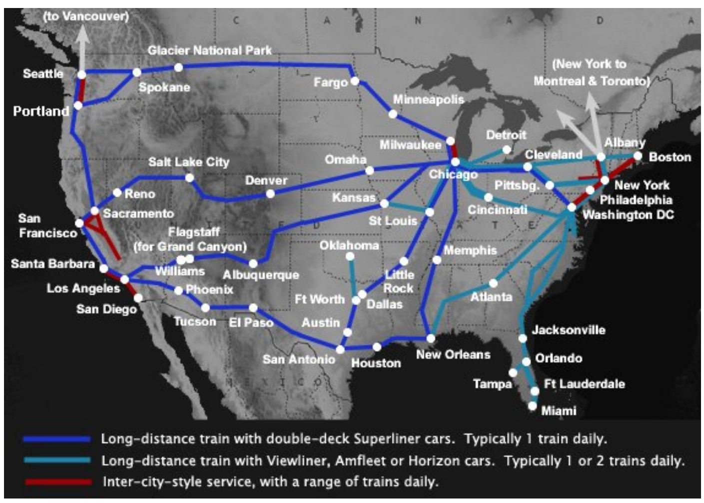

Figure A.15: Train connections in the US (simplified) [18]

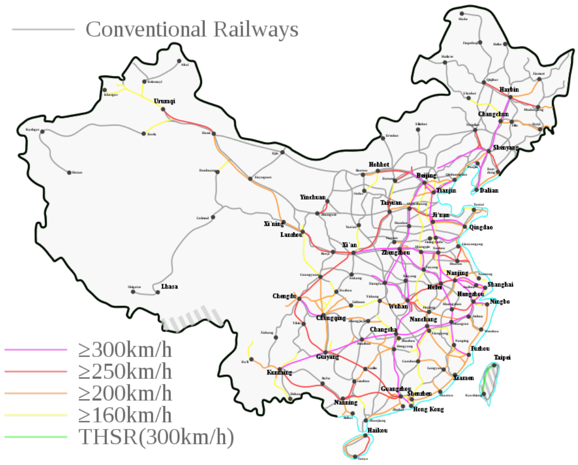

Figure A.16: Railway network China [19]

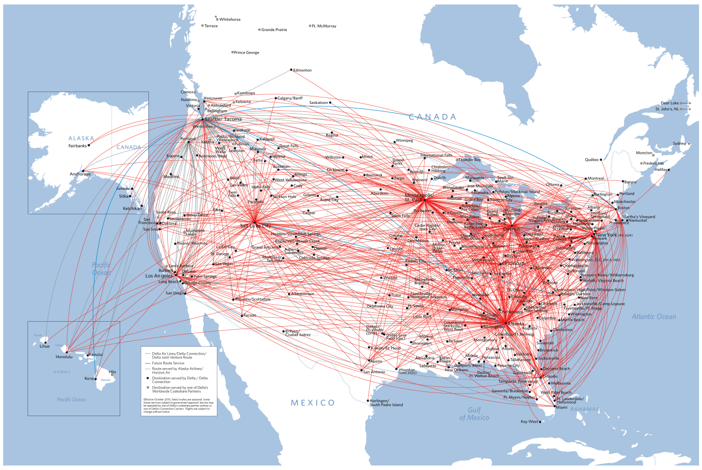

Figure A.17: Flight connections from Delta airlines in the US [20]

	Labor	Materials	Other	Total
Wing	$609	$204	$88	$900
Empennage	$1,614	$484	$233	$2,331
Fuselage	$679	$190	$98	$967
Landing Gear	$107	$98	$16	$221
Installed Engines	$248	$91	$36	$374
Systems	$315	$91	$46	$452
Payloads	$405	$100	$59	$564
Final Assembly	$58	$4	$3	$65

Figure A.18: Cost per pound for a set of aircraft components

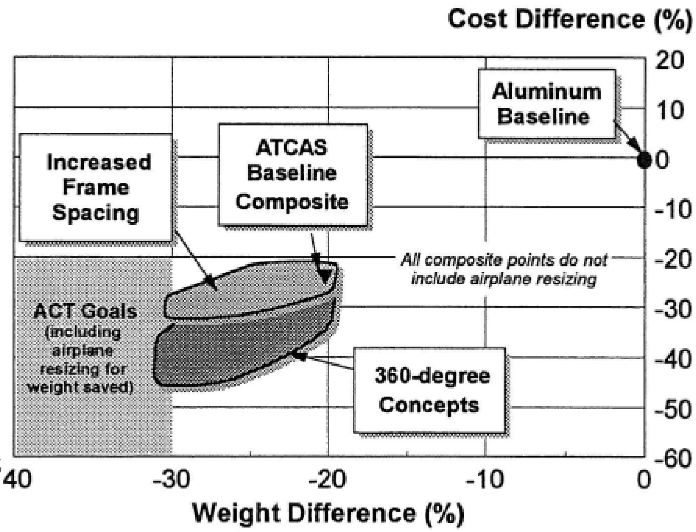

Figure A.19: Cost estimation composites

Appendix B

Cooperation

B.1 Division of Tasks

B.1.1 Organizational roles

Project leader
Felix

Minutes
Sameer

B.1.2 Report

Everyone that started of contributed to the project. Here you find a short description of who did what in which chapter.
During the project we made sure that every chapter was discussed with everyone. So the right information was passed on with each chapter.

Introduction Everyone

Requirements
Final requirements: Luis, Felix

Concepts
First concepts generation: Marly, Sameer, Danny, Joep, Luis
Final concept: Danny, Joep

Performance
Structural analysis: Danny, Luis
Aerodynamics: Felix, Joep
Propulsion: Sameer, Marly
Stability: Joep
Other design aspects & optimization: Felix, Marly

Business plan
Preliminary analysis: Sameer, Joep, Marly, Danny
Financial analysis: Felix, Marly

Conclusion
Render (visualization): Joep
Final design: Felix
Suitability for target: Joep, Sameer