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Commercial Aircraft Wing Structure

Design of a Carbon Fiber Composite Structure

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Abstract

This project explores the classical wing structure of an commercial aircraft for an all carbon fiber reinforced polymer unmanned aerial vehicle(UAV). It is part of a collaborative work consisting of several groups researching different parts of the aircraft. The objective of this report is to present the design of the inner wing structure for a greener, more efficient scaled 2:1 version of the Skywalker X8. In order to make the aircraft as efficient as possible, the structure needs to be lightweight. The loads were first approximated using XFLR5 and a first design made. The design was then tested using finite element analysis (FEA) in the programme Ansys Static Structural. The material that was tested was carbon fiber/epoxy prepreg. The final design of the wing weighs 3.815 kg, and consists of one spar and a skin thickness of 1 mm. The weight of the whole aircraft, including the propulsion system and a sharklet at both wingtips researched by other groups, is 20.262 kg. The lift-to-drag ratio was also calculated, and the most efficient angle of attack was concluded to be around 2-3°.

Sammanfattning

Detta projekt utforskar den klassiska vingstrukturen av ett kommersiellt flygplan för en obemannad luftfarkost gjord helt i kolfiberarmerad polymer. Det är en del av ett samarbete som består av flera projektgrupper som forskar på olika delar av flygplanet. Målet med projektet är att designa den inre vingstrukturen för en miljövänligare, mer effektiv uppskalad 2:1 version av drönaren Skywalker X8. För att göra flygplanet så effektiv som möjligt så behöver den vara lättviktig. Lasterna var först uppskattade via XFLR5 och en första design gjordes. Designen testades sedan med finita elementmetoden (FEM) i programmet Ansys Static Structural. Materialet som testades var kolfiber/epoxi prepreg. Den slutgiltiga vingdesignen väger 3.815 kg, och består av en bom och en tjocklek på 1 mm av vingskalet. Totala vikten av flygplanet, inklusive framdrivningssystemet samt virveldämpare på båda vingspetsarna som är framtagna av andra grupper, är 20.262 kg. Glidtalet beräknades även, och är som mest effektiv runt 2-3°.

Acknowledgements

We would like to thank our supervisor Raffaello Mariani, for his guidance and support when we where struggling. Stefan Hallström for his help and suggestions, Magnus Burman for meeting with us and answering our questions to the best of his ability. Lastly, a big thank you to the other groups of the Aeronautical Bachelor for a great group effort taking on this project together.

Contents

Chapter 1

Introduction

1.1 Background

The need for more sustainable flight is becoming increasingly important in the aviation industry. Aviation today stands for more than 2% of the global emission of carbon dioxide, and the air traffic is only expected to increase[1]. It is therefore interesting to look for ways to make airplanes more efficient, to boost the performance, and therefore to reduce the fuel consumption during flight. One important aspect for drastically reducing fuel consumption is to make the airplane as light as possible, whilst still being able to hold its shape and support its cargo. This is the aspect that relates the most to the wing structure and is what this report is going to focus on going forward.

In recent years, a lot of research has been focusing on the use of composites as an alternative to the standard aluminium used in commercial aircraft structures. Especially carbon fiber has been gaining quite a lot of attention, with many companies building light weight aircraft using different amounts of carbon fiber. For example, the new commercial airplane Airbus A350 XWB is made out of roughly 70% so called advanced materials (53% carbon fiber), and the aircraft is estimated to have a 25% advantage in fuel burn and CO_2 emissions[2].

The main advantage of using composites in aircraft structures is the low weight of the materials, coupled with a high weight-to-strength ratio. The properties of the materials are customizable and the material is also easy to mold into complex shapes. This means that a manufacturer can form the full airframe as one part and only have the strength in the directions needed, minimizing the weight even further. The disadvantages however, are the difficulties associated with manufacturing these materials and the cost of production.[3]

Another way to minimize the weight of the airplane, is reducing the need for material in the aircraft structure. This is usually done by cutting away unnecessary materials in frames or adding stringers to the skin of the aircraft, rather than increasing the thickness of the skin. Other types of structures are honeycomb structures, or other types of sandwich structures that greatly reduce the weight at the same time as offering high strength.

1.2 Problem definition

This research project is done as one of four projects developing different aspects of the same plane. The main goal of all of these projects is to create a twice as big version of the Skywalker X8, which is a small unmanned aerial vehicle with a wingspan of about 2 m and a max speed of 19 m/s. This should be done whilst coming up with ways to make it as efficient as possible.

The initial problem statement entailed a 3D CAD model of the original Skywalker (Figure 1.1), with a provided airfoil called MH49. Scaling the model creates an aircraft with a wingspan of 4 m. The goal maximum height is 2000 m over sea level and the max speed is 100 km/h (or about 28 m/s).

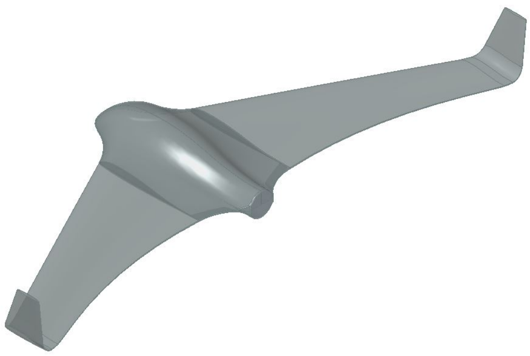

Figure 1.1: The original Skywalker X8 in Solid Edge.

1.3 Purpose

The main purpose of this project is to create the wing structure for the scaled up version of the Skywalker X8, whilst meeting the specifications of the problem definition. This will be done with the goal of minimizing the weight whilst still being able hold its shape and withstand the stresses from the aerodynamic loads of the wing. The main interest for this project is the use of composite materials, mainly carbon fiber, to build a structure consisting of spars and ribs. This will give a greater understanding of some of the realistic problems that today's aeronautics engineers are facing.

1.4 Limitations

Because of the model already given, the shape of the aircraft and the airfoil was fixed and could not be changed. Moreover, the aircraft is assumed to be subsonic with a top cruising speed of 100 km/h, and the properties of the air is assumed to be the same as at sea level. The scope was limited to working with composite materials, and that later became only carbon fiber epoxy reinforced polymer. The worst case condition is considered to be when the aircraft is at max speed at a 15°angle of attack.

Chapter 2

Theory

2.1 Aerodynamic loads

An aircraft in flight is subjected to both the air loads and the gravitational pull of the weight of the aircraft. The distributed pressure on the surface of the aircraft create the aerodynamic loads. These loads change in magnitude and position depending on different flight conditions such as level flight, maneuvers and gusts of wind. The lift magnitude and distribution over the wingspan is dependant on the shape of the airfoil and the angle of attack of the wings. Therefore, it is very important that the airfoil keeps it shape along the wing during flight, since even a small twist of the wing can affect the angle of attack and thereby the lift. At the same time, the lift bends the wing upward, creating both a bending moment, and compression and tension in the skin of the wing. All of this means that the wing needs to be able to withstand direct loads as well as shearing, bending and torsion in all parts of the wing structure.[4]

There are other types of loads and stresses associated with the thrust of the engine and the landing of the plane, but in this project the main focus will be on the aerodynamic loads. These can be simplified into a simple bending case with the lift evenly distributed, see Figure 2.1. In this case, the wing is the beam, the lift is the load and the fuselage is the fixed end. This is an oversimplification, because in reality the load is not even. Nonetheless, this approach can be used as a rough guide for this design that will later also be tested in a finite element analysis (FEA).

The worst case loads are expected to be when the wings are at 15°angle of attack, which is assumed to be the angle of flow separation[5]. The purpose of this project is to make sure that the wing can remain undamaged under this critical load. Bending and buckling loads contribute to the structurally most important stresses concerning the wing. However, only the bending loads will be considered for determining the design initially.

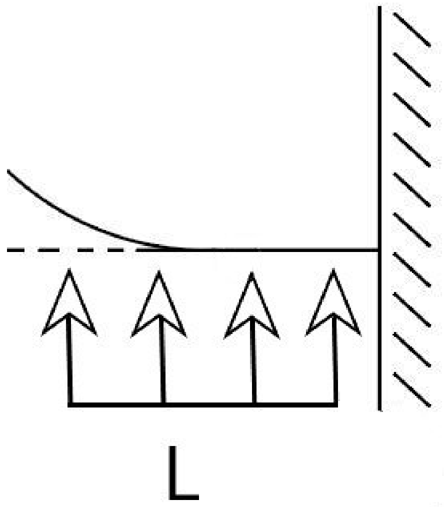

Figure 2.1: Evenly distributed load in a bending case, where the wing is the beam, the lift is the load, and the fuselage is the fixed end.

2.2 The structural components of the wing

The basic function of the structural components is to withstand the aerodynamic and other loads on the wing. At the same time, it is also important to keep the weight of the aircraft down. For most aircraft, this usually results in a structural layout consisting of a thin walled skin combined with stiffening elements both along and transverse the wing. An example of this is shown in the Figure 2.2. Depending on the size and the purpose of the aircraft, the amount and complexity of this structure varies, but the functions of these structural components stay the same. The spar, which is a beam along the span of the wing, is there to take up the majority of the loads on the wing and resist bending. Longitudinal stringers help increase the buckling stress of the skin by dividing the skin into smaller panels. The ribs, frames transverse the wing, also increase the buckling stress of the skin, as well as distribute the stresses in the structure, and upholds the shape of the airfoil[4]. In other words, the structure needs to be rigid enough not to break from the loads, but having a little flex in the wing is also beneficial. Especially for commercial airplane wings, that will usually flex between 0-7°during a normal flight and around 10°in heavy turbulence. By allowing some flexibility in a wing, airplanes are able to suppress gusts and different load changes in an efficient way making the flight smoother[6].

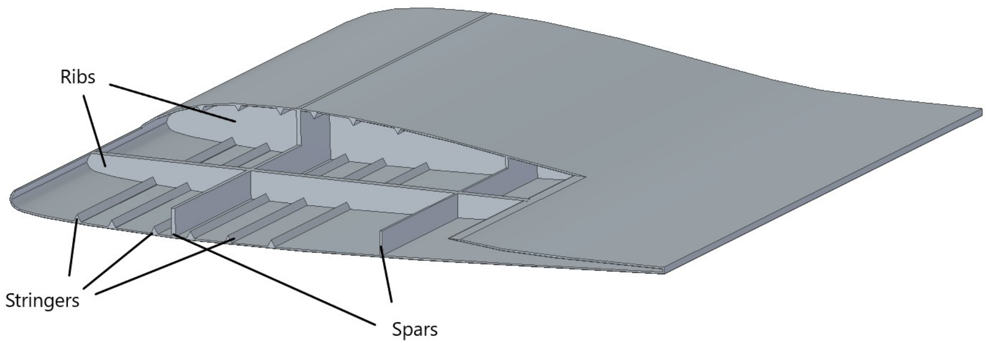

Figure 2.2: Visual representation of spars, ribs and stringers in an aircraft.

2.3 Materials

The most common material used in commercial aviation is aluminium. However, the use of composite materials is becoming more and more usual as substitutes for metals in the aviation industry. These advanced materials have many advantages, the most important being the high strength and its tailorability. This means that the material can be designed by selecting filler and matrix depending on the function and need of the part. Carbon fibre reinforced polymer(CFRP) is a composite material that is known for its impressive strength-to-weight ratio, and is therefore considered to be the main material for this project. In recent years, carbon fibers have been more and more incorporated into commercial aircraft structures, like the previously mentioned Airbus A350-XWB. Another leading contributor to furthering the use of CFRP within aviation is the Swedish company Blackwing, who've created smaller, ultra light aircrafts solely made out of carbon fibre reinforced polymer. One of their first planes made in 2018, also named Blackwing, has a total weight of 300 kg and a top speed around 311 km/h at a max altitude of 3000 m [7].

However, CFRP can be complicated to work with. The largest disadvantages of working with carbon fibre are the high cost of production, and the need to understand the orientation of the fibers in order to extract maximal use of its strength. The optimal strength is achieved when the carbon fibers are oriented in the same direction, but that means the material only has that strength in that specific direction. The strength in the other directions are then decided by the matrix, that is usually a lot less strong. In order to get similar strength in both x,y-directions, the fibers can be layered at a 90°angle with respect to each other, to reduce buckling of the layers, and also 45°, to reduce the effect of torsional loads depending on the need for strength [8].

The material properties for composite materials are difficult to determine, because it depends on the properties and volume of each material involved, as well as how they interact. There are different techniques to characterizing a composite material, such as calculating the volume fraction between fibre and matrix, or even testing the mechanical properties of the actual material [9]. Because of how challenging these calculations can be, this project instead uses a single source for the material properties, the Ansys Composite Library, to get an approximation of how the material could work in the design. Ansys is an engineering simulation program that can be used to simulate a finite element analysis, among other things. The benefit of using their library is that the programme contains the properties of the material in all three directions, making the results of the FEA more reliable.

From the composite library, the main material was chosen to be a carbon/epoxy prepreg. The most relevant properties are presented in Table 2.1. Worth noting is that the yield strength of the material is decided by the strength of the matrix rather than the fiber. Prepreg is a sort of premade fabric, made of carbon fibers in a epoxy matrix. Unlike a wet layup, where the liquid epoxy resin and the carbon fibres are set together when the product is shaped, prepreg is a dry fabric that can be shaped around the mould in layers and then cured under pressure. This makes it less messy than a wet layup, and leads to fewer mistakes that can affect the strength of the material. However this also makes it a lot more expensive.[10]

Youngs modulus* [GPa]	61.34
Density [kg/m^{-}3]	1420
Tensile strength* [GPa]	0.805
Tensile/yield of epoxy resin [MPa]	54.6

Table 2.1: Material properties for Epoxy carbon woven (230GPa) prepreg, taken from Ansys Composite Library. * In x,y-direction

Chapter 3

Method

3.1 Calculation of lift and drag forces

The lift coefficient C_L and the drag coefficient C_D, are both dimensionless and are calculated in order to determine the lift and drag forces for the wing. They are also studied to examine the lift-to-drag ratio, which is a ratio that helps determine the most efficient angle of attack for the airfoil. C_L for every angle α from 0 to 15 °is determined with

$$C_L = a(\alpha - \alpha_{L=0}) \tag{3.1}$$

Where α is the airflow's angle of attack, $\alpha_{L=0}$ is the angle of the lift force when it's zero, and is set to 1.5 for this airfoil. The a here is the approximated slope of the infinite wing calculated as

$$a = \frac{a_0}{1 + \frac{57.3a_0}{\pi eAR}} \tag{3.2}$$

Where AR is the aspect ratio, e is the span efficiency factor and a_0 is a factor obtained from the lifting line theory [11].

If the profile drag coefficient is assumed to be $c_d = 0.003$, then the drag coefficient C_D can now be solved with

$$C_D = c_d + \frac{C_L^2}{\pi eAR} \tag{3.3}$$

Both lift and drag forces for every angle can now be calculated with the the stream velocity V, air density ρ, the surface area of the wing s and each respective coefficient as

$$F_i = \frac{\rho V^2}{2} sC_i \tag{3.4}$$

3.2 Approximating the loads with XFLR5

Despite previously calculating the lift and drag forces, the complete distribution of load over the wing needs to be known in order to calculate the stresses. To approximate this, the programme XFLR5 is used. XFLR5 is a tool to analyze airfoils and wings operating at low reynolds numbers. By using the coordinates to the airfoil MH49 in XFLR5, the wing could be approximated and then analyzed at different angles of attack. The speed used is the maximum speed of 28m/s and the biggest loads are assumed to be at an angle of attack of 15°. Using XFLR5, the lift coefficient for different sections along the wing can be produced, and from that the aerodynamic loads can be calculated. If the y-axis is along the span of the wing, the distributed lift is

$$L(y) = \frac{1}{2}\rho V^2 C_L(y) chord(y) \tag{3.5}$$

Where ρ is the density of the air, V is the speed of the aircraft, C_L is the lift coefficient, and chord is the chord length at that section.

Since the lift is countered by the weight of aircraft, the net vertical force is calculated as

$$Q(y) = L(y) - nW(y) \tag{3.6}$$

Where Q is the load and W is the weight assumed to be linearly distributed along the wing. n is the load factor, which is the ratio of weight of the aircraft to its lift and can differ during flight. From the load, the shear and the bending moment, along the length of the wing b, can be calculated using Equation 3.7 and 3.8 below, assuming that both the shear and the bending moment is zero at the edge of the wing.

$$S(y) = \int_0^b Q(y)dy \tag{3.7}$$

$$M(y) = \int_0^b S(y)dy \tag{3.8}$$

3.3 Defining the geometry

The wing needs to be able to withstand the bending stresses. Calculating these stresses can become very difficult because of the complicated structure of the wing box. However, when the wing is subjected to bending stresses, for example when the wing is bent upward because of lift, the wing can be seen as a cantilever beam with the fuselage as the fixed support. In order to simplify even further, the spar can be assumed to be the only carrier of the shear stresses. This means that the maximum stress can be calculated using simple beam theory

$$|\sigma(y)| = \frac{|M(y)|}{I(y)} z_{max} \tag{3.9}$$

Where *M(y)* is the maximum bending moment, *I(x)* the cross-sectional moment of inertia of the spar, and z_{max} is the maximum height of the cross-section.

Clearly, the maximum stress is dependent on the geometry of the cross-section of the beam. Using the moment of inertia for a rectangle and the yield strength σ_s of the material, the minimum width can be factored out of Equation 3.9 and be calculated as

$$b = 12\frac{|M(y)|z_{max}}{h^3\sigma_s} \tag{3.10}$$

If it is then assumed that the weight is evenly distributed, see Figure 2.1 in section 2.1, then a simple elementary bending case can be assumed. From this, the deflection can be calculated as

$$d = \frac{3L_{tot}l^3}{24EI} \tag{3.11}$$

Where l is the length of the wing, E is the youngs modulus of the material, and I is the moment of inertia of the beams cross section. L_{tot} used here is the max load.

To test the design, the engineering programme Ansys Static Structural was used. In the finite element analysis(FEA), the worst case maximum pressure was assumed to be evenly distributed over the wing and the same as the stagnation pressure. This is a very extreme case, which should give a good margin of safety. The stagnation pressure is calculated according to

$$p = \frac{1}{2}\rho V^{2\prime} \tag{3.12}$$

With the density of the air as $\rho = 1.225$ kg/m^3, and the velocity $V = 28$ m/s.

Chapter 4

Results

4.1 Aerodynamic Loads

The calculated lift-to-drag ratio is seen in Figure 4.1. The resulting coefficients C_L and C_D, both calculated and from XFLR5, is presented in Figures 4.2-4.3 and 4.4-4.5 respectively. The distributed loads and bending moment are presented in Appendix A1.

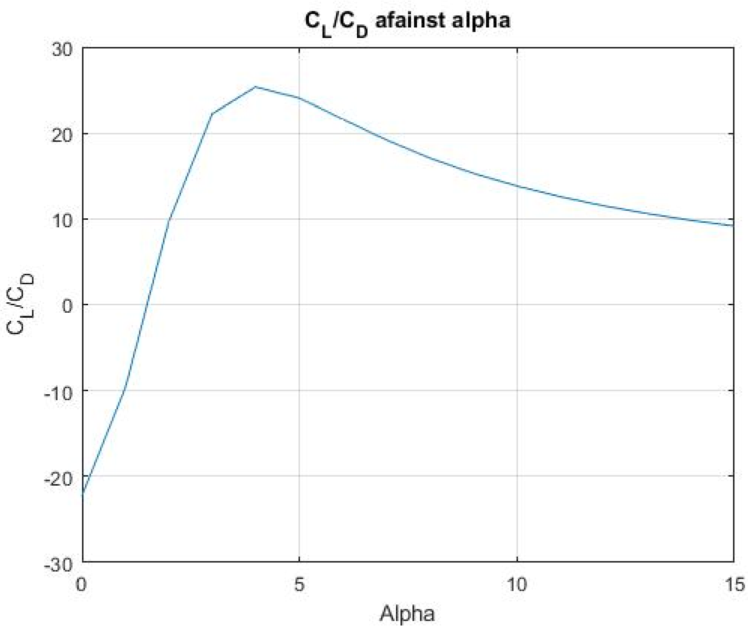

Figure 4.1: The lift-to-drag-ratio.

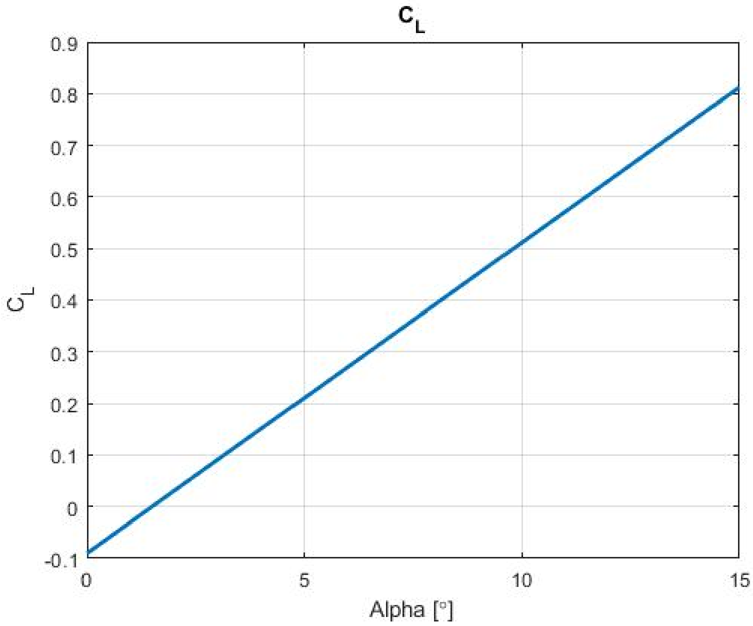

Figure 4.2: Calculated lift coefficient plotted against angle α.

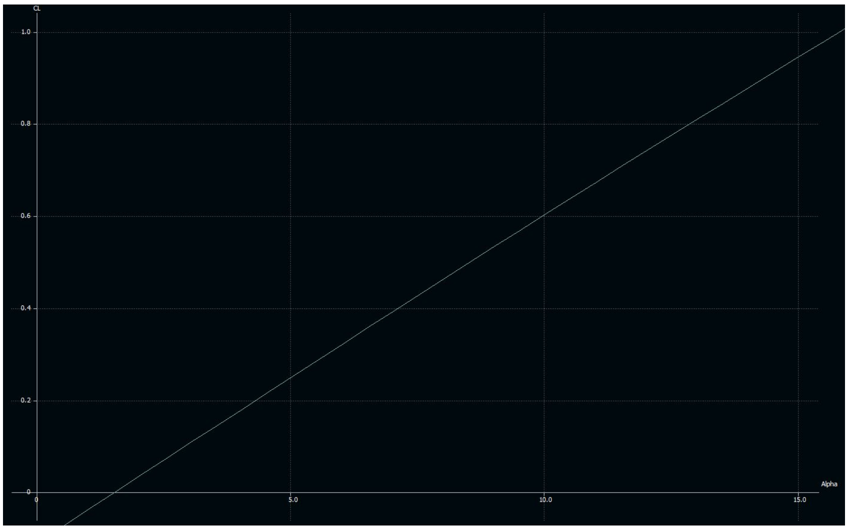

Figure 4.3: Lift coefficient from XFLR5 plotted against angle α.

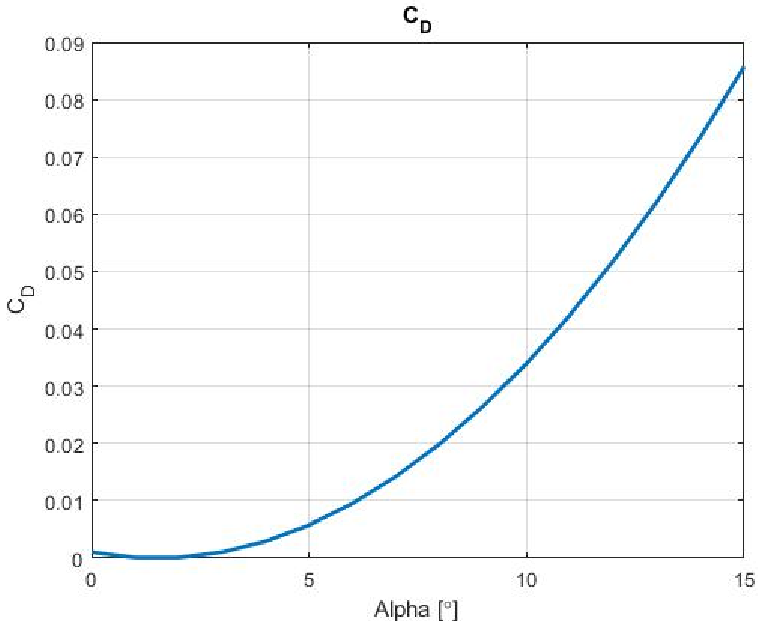

Figure 4.4: Calculated drag coefficient plotted against angle α.

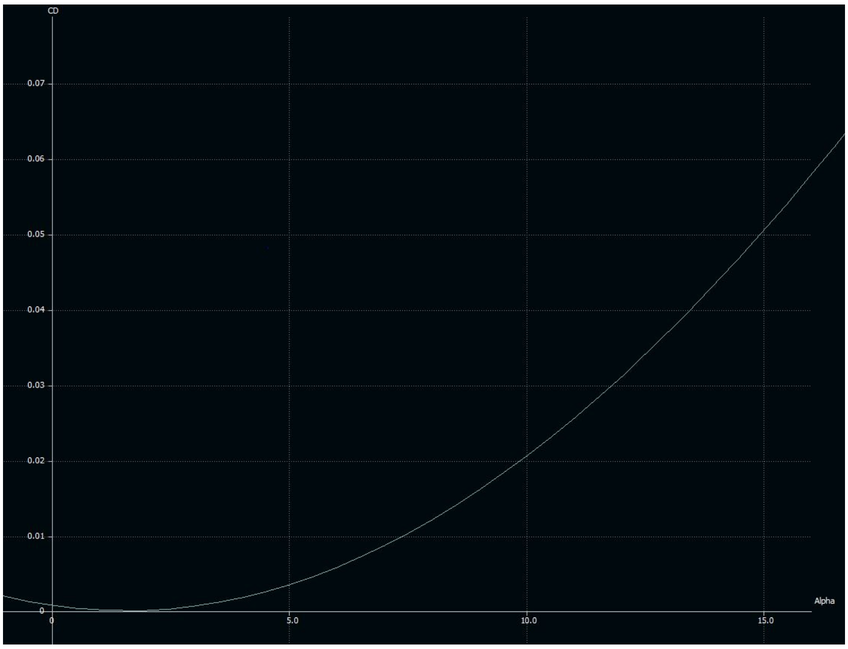

Figure 4.5: Drag coefficient from XFLR5 plotted against angle α.

4.2 Stresses and Finite Element Analysis

The results from a few selected cases of skin thicknesses is presented in Table 4.1. The calculated deflection, maximum stress and stagnation pressure are presented in Table 4.2.

	2mm	1.5mm	1mm
Weight [kg]	6.102	4.959	3.815
Max stress [MPa]	16.729	21.334	30.834
Max deformation [mm]	9.1283	11.572	16.425

Table 4.1: Properties of different skin thicknesses, at 15°.

Calculated deflection [mm]	Calculated max stress	Stagnation pressure [Pa]
15.7	53.0124	480

Table 4.2: Calculated deflection, max stress and max pressure of the wing.

4.3 The Final Design

The resulting wing design made in Solid Edge is shown in Figure 4.6. The wing has one spar set at 25% of the chord length, whilst the trailing edge is solid to about 10% of the chord. Stringers and ribs were not included in this design, which is later discussed in section 5.2.

The complete plane with the sharklet and fuselage is presented in Figure 4.7, and the final values and dimensions are presented in Table 4.3. The weight of the full aircraft is calculated with the weight of the solid sharklet[12] and the propulsion system[13], both made in other projects.

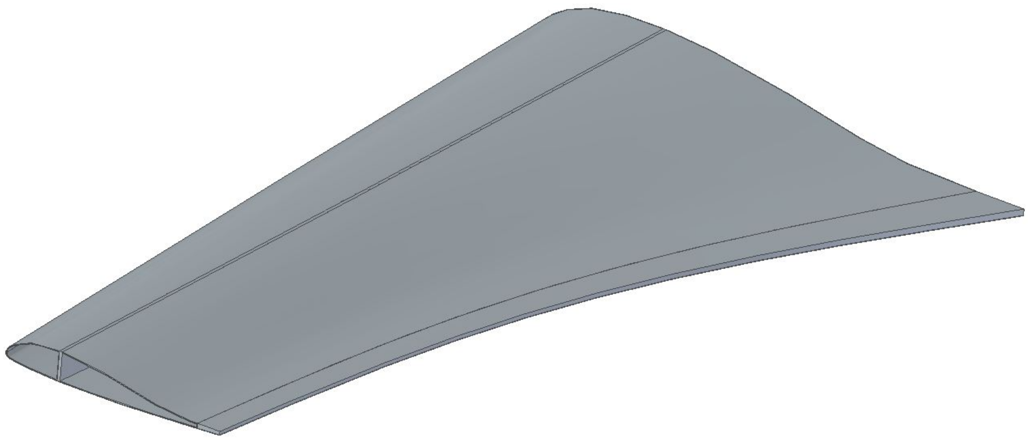

Figure 4.6: The wing design made in Solid Edge.

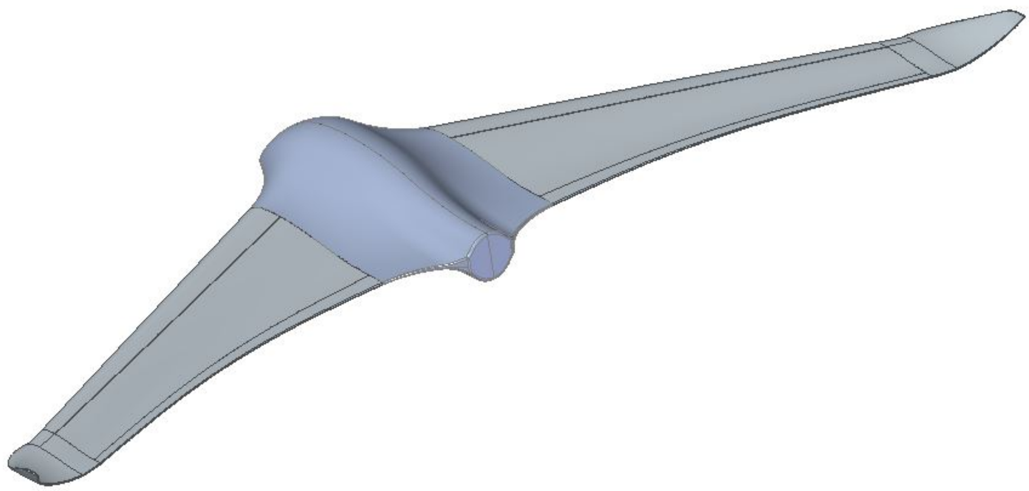

Figure 4.7: The whole plane with sharklet and fuselage, made in Solid Edge.

Spar thickness[mm]	Skin thickness[mm]	Weight of wing[kg]	Weight of plane[kg]
4.3	1	3.815	20.262

Table 4.3: Dimensions and properties of the design.

Chapter 5

Discussion

5.1 The Approach

The main objective for this project was to design a light wing whilst still being able to withstand forces for specific flight conditions. This problem was approached by first doing a literature review, where the classical ways of calculating loads and stresses on wings were learnt, as well as the common materials used in aviation. The main material chosen for this project became carbon fiber reinforced polymer, mostly for its exceptional strength and its increasing usage in commercial aircraft. It was also considered interesting, since the use of composites as alternative and more efficient materials is becoming increasingly important as more is researched about these materials and their complex structure. It was decided early on to focus only on the classic ”spars-and-ribs” approach to the wing structure, and excluded more complex structures like honeycomb structures or sandwich structures. Having a rough idea of what was of interest, the loads were calculated in order to achieve the minimum thickness of the spar. This resulted in a CAD design that was tested in Ansys Static Structural, and then changed until the final design was settled.

5.2 The final design

The final design consisted of a wing with one spar, made completely out of carbon/epoxy prepreg. The dimensions for this are presented in Table 4.3. Considering the max stress of the design is below the yield strength of the epoxy, the design is considered to be able to withstand the worst case pressure of 480 Pa.

Several designs with different spar thicknesses and with/without ribs were tried in Ansys. Ribs didn't seem necessary considering the strength of the material, and the thickness of spar didn't seem to make that much of a difference. Instead, the biggest difference in weight and stress came from trying different skin thicknesses. As can be seen in Table 4.1, both the deformation and the maximum stress increase with thinner skin. However, none of these stresses are larger than the yield strength of the epoxy resin at 54.6 MPa. This means that plastic deformation does not occur and the deformation should be purely elastic. The deformation also doesn't change significantly between the different thicknesses, although any deformation could potentially affect the lift of the airfoil. Considering that an angle of attack of 90°can't generate lift at all, this deformation shouldn't be a problem in this case. Instead it should be enough that the wing can stand these forces without permanently deforming.

The final weight of the aircraft is 20.262 kg, which include; the weight of the wings (7.63 kg for both), the estimated weight of the fuselage (2.548 kg), the weight of the solid sharklets (6.074 kg for both) and the weight of the propulsion system (4.01 kg) positioned in the fuselage. All except the weight of the propulsion system are taken from Solid Edge, using the density of the material. However, there is room for improvement of the weight. For example, the weight of the sharklets could be significantly reduced if they were hollowed out, which they probably would be during manufacturing. This would have to be tested though, to see that the sharklets can withstand the pressures hollowed out. The solid trailing edge could possibly be thinner, or an alternative design of two spars and a thinner trailing edge could be even lighter.

5.3 Loads and stresses

In order to confirm the results of the simulation done in XFLR5, the calculated coefficients C_L and C_D can be compared to the ones approximated with the programme. Comparing the graphs in Figures 4.2 - 4.5, there can be seen very little difference. Especially the lift coefficients have almost the same slope, however there is a slight difference when it comes to the drag coefficient. The drag calculated is much larger than the drag from XFRL5, but this could be because the model used in XFLR5 doesn't take into account the drag of the profile. It could also be because the calculations are based on simplifications and standard values that are not specific to the wing, whilst XFLR5 takes the whole wing into account. Considering how similar the graphs are besides this, and the fact that the drag is partly neglected in this project, it seems that the distributed load from XFLR5 should be an acceptable approximation to use to calculate the shear force and bending moment over the wing. The calculated drag and lift coefficients were also used to calculate the lift-and-drag-ratio for this wing. The most efficient angle of attack can be found where this ratio has its steepest slope, and that is the area for the best angle for level flight. As can be seen in Figure 4.1, the most efficient angle of attack for this wing should be around 2-3°.

The deflection and maximum stress (Table 4.2) were calculated to make sure that the spar would be able to handle the stresses at 15°angle of attack, and then the design was tested using the much more extreme load of the stagnation pressure(Table 4.1. These values do look very similar, but because of the difference in load it is difficult to compare. The calculated values have a smaller load, but are also heavily simplified. The deflection and maximum stress are calculated only for the spar, which means they do not take into account the whole 3D design of the wing, like the skin of the wing or the solid trailing edge. These take up some of the loads, which would lower the maximum stress. This however gives a margin of safety, by making sure that the spar itself can uphold the loads at 15°angle of attack, the wing should be able to handle other unexpected situations that could arise during flight. Nonetheless, the FEA is needed to test that these assumptions are justified and to test the design. Since the FEA also takes into account the material properties in all directions and the whole design of the wing, it shouldn't be a problem that the FEA shows lower stress than the calculated ones, even though it has a much larger load. The conclusion then is that the design seem to be able to handle the loads in both extreme cases, and that gives the design some extra security.

5.4 Material and manufacturing

As mentioned, it was decided early on in the project to work with CFRP. However, the material properties for composites can be hard to find because of the difference depending on manufacturing. The chosen material is therefore found in the Ansys Composite Library, since the programme already had the material properties in all directions which would make the FEA more reliable. Prepreg, the pre-woven carbon/epoxy fabric, was something of interest from the start. Since it is premade by machine, there is less room for errors and the material properties are much more controlled than for a wet layup. The downside is that it is a lot more expensive, which is something we haven't taken into account for this project.

The prepreg is a pre-made fabric of woven carbon fibers, preimpregnated with epoxy resin. This fabric can be molded around most shapes in layers, and then cured. The whole plane would probably have to be manufactured in parts, especially the sharklet. The wing would need to be cut in half, presumably through the airfoil of the wing. Depending on the thickness of the prepreg that is used, one to three layers of prepreg would then be clad around a premade mold of the aircraft, to form the desired 1 mm thickness of the skin. The biggest problem would be the spar, that would either be built from the bottom up by layers of prepreg that is 4.3 mm thick or using layers on its side, fastened to the skin. Building the spar from the bottom up could take some time to cure since it would be significantly thicker than the rest of the aircraft, however it is probably better considering that the material properties would be more consistent.

5.5 Considerations and further research

During the course of this project, a number of assumptions and simplifications were made. The use of lifting line theory, for example, is not completely justified because the theory does not account for swept wings. With wind tunnel testing, the accuracy of this approximation could have been examined more. Instead, these results were used in order to get a rough approximation for the calculations.

This project has also only considered one material. It is possible that there is a better material or material combination out there that is more lightweight and still able to uphold the pressures. It is also important to note that, since the properties of composite materials are not always consistent, these results can only be seen as an estimate and would need to be looked at for the specific material used during manufacturing. This project also didn't take into consideration the buckling of the wing, which has the potential to greatly affect the way the structure upholds the stresses.

Further research therefore might be to test the pressures on the wing in a test tunnel, or to examine the buckling stress to see if the wing buckles from these pressures. Other lightweight materials such as balsa wood, plastics or other composites could also be further researched, possibly in combination with sandwich structures.

5.6 Conclusion

In conclusion, the final design is a wing made completely of carbon/epoxy prepreg and weighs 3.815 kg. The structure consists of only one spar, and is able to stand the stresses of the worst case scenario of 480 MPa in an FEA. The weight of the whole aircraft is 20.262 kg, which includes: the fuselage, the propulsion system and two wings with a sharklet each.

Bibliography

[1] European Commission, 2020. https://ec.europa.eu/clima/policies/transport/aviation, [Accessed: 2020-04-08].

[2] Airbus Media Relations, 2020. https://www.airbus.com/aircraft/passenger-aircraft/a350xwb-family.html [Accessed: 2020-04-08].

[3] L. Sowntharya S. D. Sharma and Kamal K. Kar. *Composite Materials(Chapter 1); Polymer-Based Composite Structures: Processing and Applications*. Springer-Verlag Berlin Heidelberg, Heidelberger Platz 3, 14197 Berlin, Germany, 2017.

[4] A. Saddington. *Aircraft strcutures for Engineering Students*. Butterworth–Heinemann, The Boulevard, Langford Lane, Kidlington, Oxford, OX5 1 GB, UK, 2000.

[5] SKYbrary, 2020. `https://www.skybrary.aero/index.php/Stall` [Accessed: 2020-05-30].

[6] Capt. Les Jetson, 2020. https://captainjetson.com/featured/can-the-airplane-wings-break-off-in-strong-turbulence/, [Accessed: 2020-05-07].

[7] Ann-Katrin Öhman, 2018. https://www.kth.se/aktuellt/nyheter/blackwing-planet-som-bara-vager-300-kilo-1.898321, [Accessed: 2020-05-07].

[8] Rock West Composites, 2018. https://www.rockwestcomposites.com/blog/how-fiber-direction-influences-tube-and-plate-properties/, [Accessed: 2020-05-07].

[9] L. Sowntharya Deepak Pandey Kamal K. Kar Jayesh Cherusseri, Sumit Pramanik and S. D. Sharma. *Composite Materials(Chapter 2); Polymer-Based Composite Materials: Characterizations*. Springer-Verlag Berlin Heidelberg, Heidelberger Platz 3, 14197 Berlin, Germany, 2017.

[10] Flight-Mechanic, 2020. https://www.flight-mechanic.com/composite-repairs-types-of-layups/ [Accessed: 2020-05-30].

[11] John J. Bertin Russell M. Cummings. *Aerodynamics for Engineers (6th Edition)*. Pearson, 2013.

[12] Hanlin GongZhang and Eric Axtelius. *Aircraft Winglet Design*. 2020.

[13] Oscar Andersson and Dennis Bogatov Wilkman. *Propulsion system for a small unmanned aerial vehicle*. 2020.

Appendix A

First Appendix

A.1 Distributed loads over the wing

The distributed loads over the wing, calculated with Equations 3.6, 3.7 and 3.8.

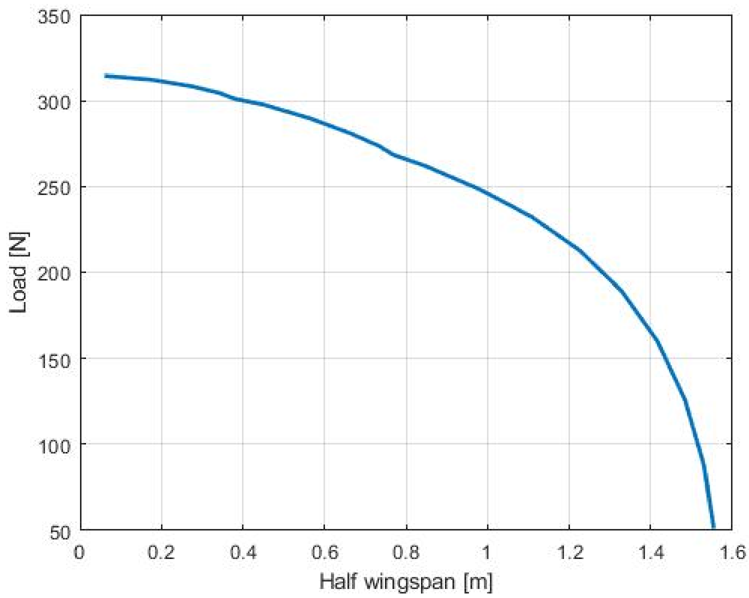

Figure A.1: The distributed load over the wing.

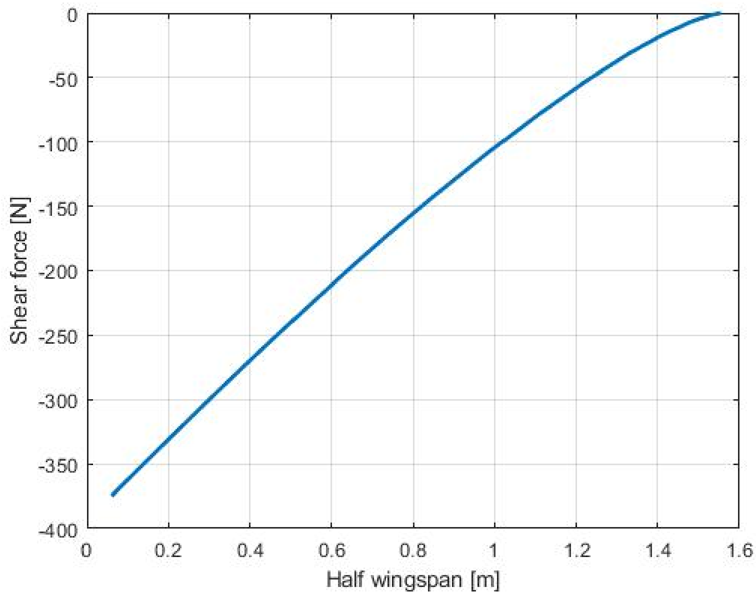

Figure A.2: The distributed shear force over the wing

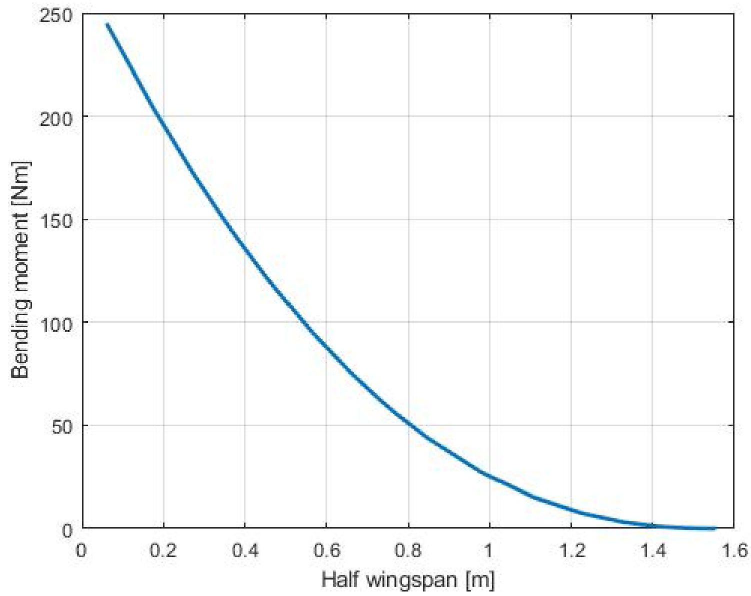

Figure A.3: The distributed bending moment over the wing.