VTOL Aırcraft

ME 429 Mechanical and Thermal Design



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# Executive Summary

Since readers look into the executive summary before they read the entire report. An executive summary should summarize the key points of the report. It should restate the purpose of the project, highlight the major progress in the project execution, and describe any results, conclusions, or recommendations from the report.

An executive summary is usually 500-1000 words in length, it is written as one page, it may include numerical information about the procedure and the results, it should not include any information that is not reported in the report, abbreviations should not be used unless they are spelled out in the summary, citations or references are not given in the summary.

# Introduction

This section of the report usually problem statement which states why that particular subject is chosen. It establishes the importance of the subject by reviewing relevant literature, including academic papers, patents, books, web sites, etc. Relevant references are discussed and a theoretical background is provided based on the literature review. Significance of the project should be clearly stated. Already existing products and designs should be benchmarked and the drawbacks should be stated. This section should be kept brief and to the point in 8-10 pages.

# Design Process

This section should be 10-20 pages long and should include the following subsections:

## Design Criteria and Product Design Specifications

The reasons why the design criteria are chosen and the relevance of the criteria to the product in particular should be explained. All assumptions should be stated. Product design specification should be brief and clear. Use the template provided. Binary Dominance Matrix should be stated here.

## Overview of Possible Solutions

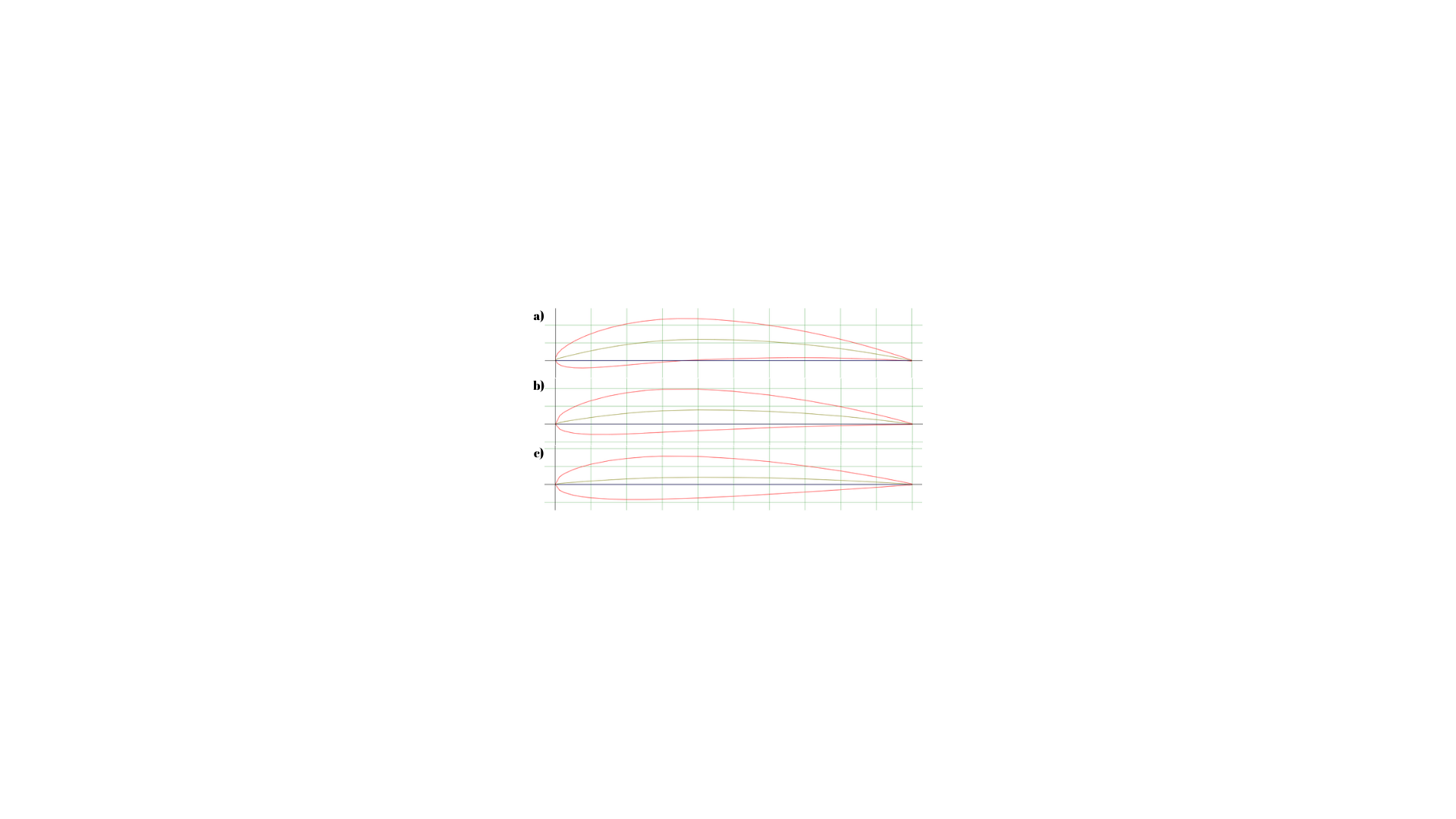
Possible Solutions should be proposed with clear sketches and explained clearly. Decision Matrix should be provided here.

## Detailed Design and Analysis

VTOL which is an efficient combination of two types of flying vehicle, flies as a fixed wing on majority of its flight time. It was decided to design a fixed wing vehicle that can operate stably, and then construct the tri-copter frame on the fixed wing aircraft. The reason behind the decision is that fixed wing aircrafts have more inflexible design parameters such as wing length, location of the CoG (Centre of Gravity), AoA (Angle of Attack) of wing etc. On the other hand, it is relatively easy to adapt rotary-wing aircrafts to different frame designs, and they provide a wider range for the specification of design parameters.

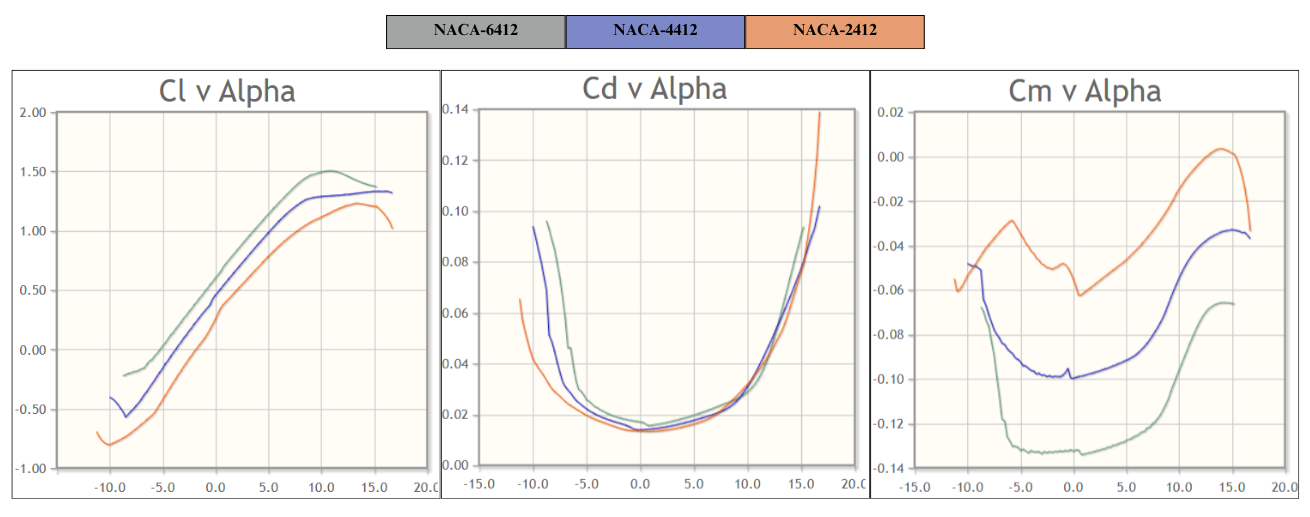
The most important part of the fixed wing vehicles is their wings which produce lift to keep the aircraft on-air and a significant portion the drag. For this reason, the wing design is considered as the most prioritised stage in the design process of the aircraft. The aspect was taken into account in the progress of the project and it was aimed to select the airfoil that will meet to the design criteria of the vehicle. Therefore, airfoils commonly used in commercial RC (Radio-Controlled) aircrafts were surveyed, and it was found three profiles that stand out with their differing advantages in application. These are NACA-6412, NACA-4412 and NACA-2412; these airfoils are shown in the *Figure 2.1*.

National Advisor Committee for Aeronautics (NACA) is an organisation founded in the USA in 1915 and conducting aeronautics research. The aerodynamic surface shapes developed and tested by the organisation, whose name has changed to NASA, are called NACA airfoils. [1] Each digit in NACA 4-digit airfoils refers specific characteristic of the airfoil. First digit refers maximum camber as percentage of the chord length, second digit refers to location of maximum camber with respect to leading edge and last two digits refer maximum thickness of the wing profile as percentage of the chord length. [2]



**Figure 2.1** Commonly Used Airfoils: (a) NACA-6412; (b) NACA-4412; (c) NACA-2412.

The airfoiltools.com website was used to make comparative investigation of the selected airfoils. This web tool plots α (Angle of Attack) dependent variations of (Lift Coefficient), (Drag Coefficient), (Moment Coefficient) values of the airfoil for a certain Re (Reynolds number). As seen in the *Figure 2.2*, polar diagrams are generated for Re=100.000. NACA-6412 has a high value. NACA-2412 has a lower value at 0-5 degrees, which is the AoA of operation. NACA-4412 has intermediate values in all graphs, but it also can be an optimal option for different manufacturing techniques such as balsa spar-rib construction thanks to its semi-linear bottom line.



**Figure 2.2** Polar Diagrams of Airfoils (Re=100.000).

The next stage is to design a wing that can generate enough lift to compensate for the expected take-off weight at lengths and speeds within the design limitation. The wing must produce at least 15N lift force to balance minimum take-off weight at a maximum speed of 13m/s and its length must be in the range of 1300-1500mm. XFLR5, a numerical aircraft analysis and fixed wing design software, was used for this purpose.

XFLR5 is a software developed specifically for model aircraft, unmanned aerial vehicles (UAV) and small-scale fixed-wing vehicles. The program is capable of both 2D aerodynamic and 3D numerical analysis. In wing design, XFLR5 is useful with its three-dimensional analysis capability. Using the panel method, aerodynamic properties of the wing such as lift, induced drag and moment can be computed. 3D Panel Method is ideal for evaluating the performance of wings with different geometries. Furthermore, the user can determine the aerodynamic centres of the wing and stabilisers and examine the effect of in-flight moments on stability.

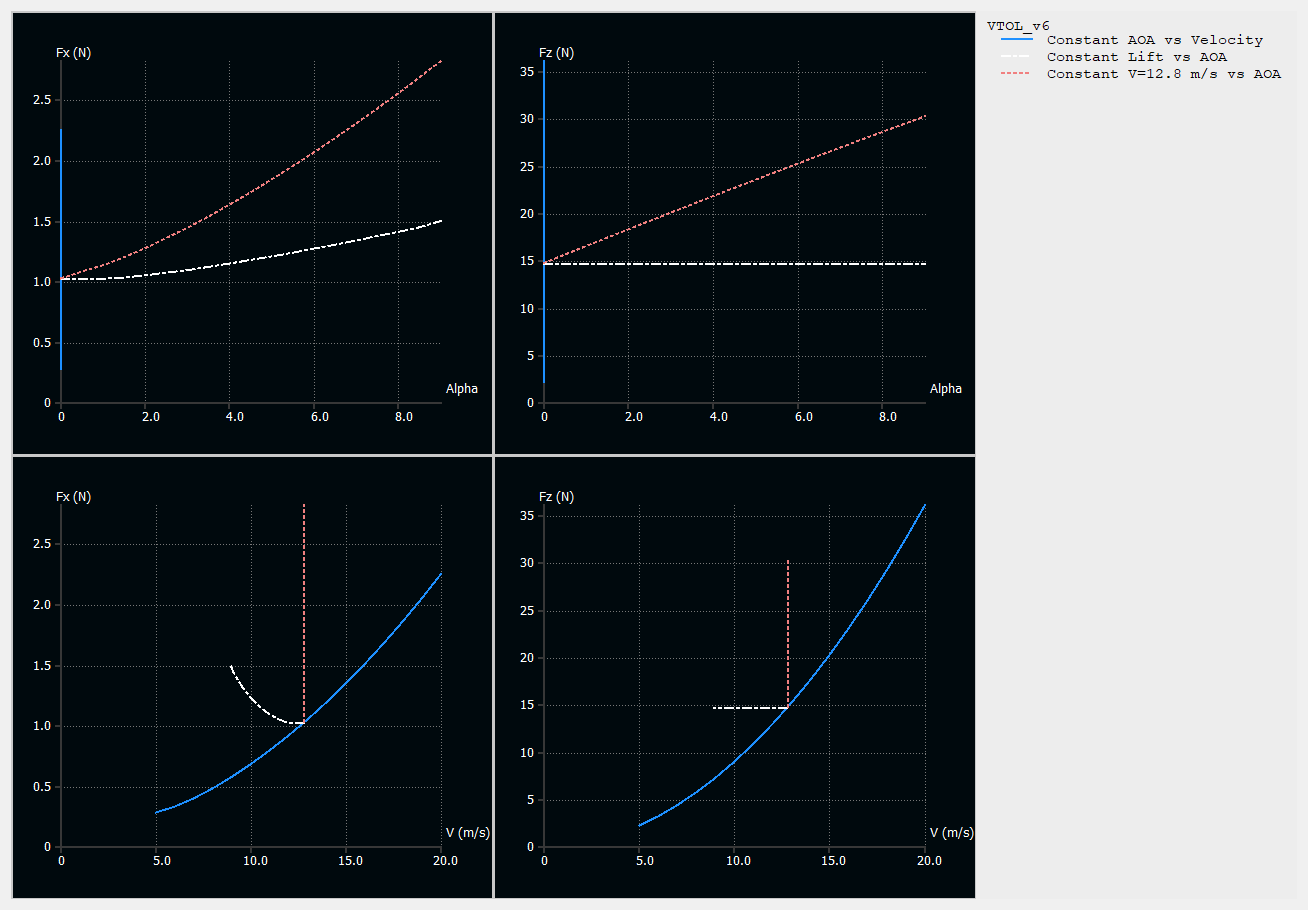
The process was carried out through iterations and the optimal wing geometry was tried to be obtained. The primary focus was on the lift force, while the total drag force and pitching moment were also considered. 3D Panel Analysis method was used and constant lift analysis condition was applied. 1.5 kg of point mass is located at quarter chord length from the leading edge. The algorithm calculates the minimum speed required for the wing to generate sufficient lift at different AoA values, and the software stores the operation points. Total of 6 iterations were performed. Some significant geometric parameters of the wing designs and their required minimum speed obtained from the analysis are shown in *Table 2.1.*

**Table 2.1.** Geometric Parameters and Analysis Results of Design Iterations.

|  |  |  |  |  |  |  |
| --- | --- | --- | --- | --- | --- | --- |
| Parameters | 1st  Iteration | 2nd  Iteration | 3rd  Iteration | 4th  Iteration | 5th  Iteration | 6th  Iteration |
| Wing Span *[m]* | 1.300 | 1.300 | 1.500 | 1.500 | 1.500 | 1.640 |
| Wing Area *[m2]* | 0.191 | 0.210 | 0.251 | 0.251 | 0.252 | 0.266 |
| Airfoil (NACA) | N-4412 | N-4412 | N-4412 | N-6412 | N-6412 | N-6412 |
| Root Chord *[m]* | 0.230 | 0.230 | 0.230 | 0.230 | 0.250 | 0.250 |
| M.A.C. *[m]* | 0.191 | 0.191 | 0.191 | 0.191 | 0.191 | 0.186 |
| Wing Load *[kg/m2]* | 7.843 | 7.143 | 5.970 | 5.970 | 5.980 | 5.726 |
| Tip Twist *[˚]* | 0 | 0 | 0 | 0 | -3.0 | -3.0 |
| Aspect Ratio | 8.837 | 8.048 | 8.995 | 8.955 | 8.970 | 10.10 |
| Tilt Angle *[˚]* | 2.0 | 2.0 | 2.0 | 2.0 | 3.0 | 3.0 |
| Cruse Speed *[m/s]* | 18.75 | 17.37 | 15.23 | 13.22 | 13.16 | 12.77 |
| Stall Speed *[m/s]* | 12.2 | 11.28 | 9.854 | 9.249 | 9.202 | 8.923 |

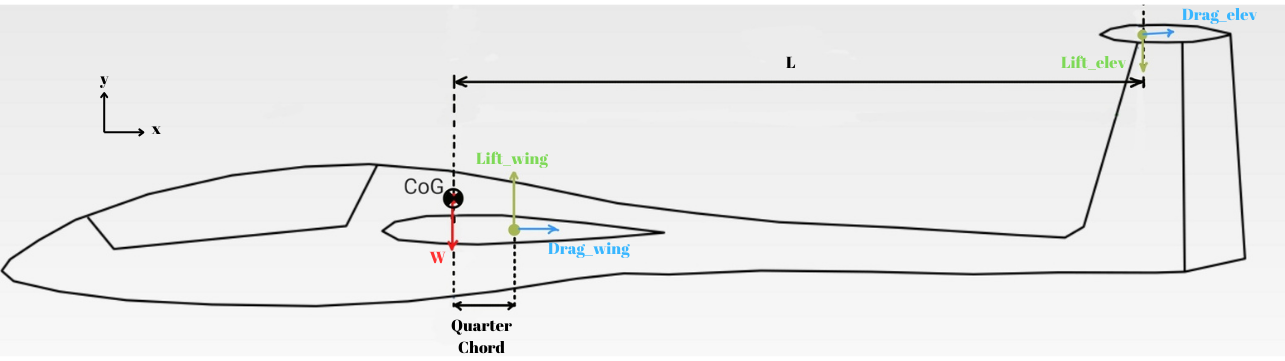
In the first iteration, the rough dimensions of the concept vehicle were taken as the initial values. Since the minimum velocity to produce sufficient lift was outside the design limits, it is decided to increase the wing area in the second and third iterations. At the fourth iteration, it was realised that a higher airfoil was needed. Therefore the profile changed to NACA-6412 which offers almost 20% higher at 2˚ AoA (See *Figure 2.2*). At the fifth iteration, the tilt angle of the wing is increased to 3˚ with negative tip twist angle of 3˚ and the root chord was increased to compensate for the resulting lift. This type of negative tip twist is called wing washout and provides many benefits. Especially at high angles of attack, flow separation starting from the wing tips is observed and it causes a dangerous situation, loss of aileron control [3]. The washout is useful technique to prevent it. Additionally, it reduces the rotational moment generated by the wing. Final iteration is completed with the addition of winglets. They are beneficial for reduces the wing tip vortices and cause little improvement of the lift force. Also, in all design sweep angle is applied to keep maximum thickness position of the airfoil as straight line along the wing. It is the line where we will position the spar tube that will provide the strength of the wing, and a straight maximum thickness line gives the flexibility in determination of the tube diameter.

Final geometry of the wing is given in *Appendix A*. For a more comprehensive performance evaluation, 3 different analyses are performed for final iteration. Those are Constant AoA vs. Velocity which computes generated forces, Constant Lift Force (15N) vs. AoA which computes required air speed and Constant Velocity (12.8 m/s) vs. AoA which computes generated forces. Drag and lift forces versus velocity and α is plotted and polar are shown in *Figure 2.3*.



**Figure 2.3** Results of Force Analysis of the Final Wing Design.

After the wing design process, the aircraft needed a tail. The tail is used to compensate the moments of the aircraft and responsible for generating pitching and yawing movements by affecting resulting moments of the aircraft. Another important concept for fixed-wing aircrafts is Tail Volume coefficient. It is a non-dimensional scale of tail effectiveness. It can be easily derived from span-wise Moment Equilibrium of the aircraft [4]:



**Figure 2.4** Span-wise Free-Body Diagram of the Aircraft.

During calculations of the horizontal stabiliser is neglected, and it is assumed that y-axes of the CoG is such that the moments produced by drag forces of the wing and the horizontal stabiliser cancel each other. Also, x-axes position of the CoG is located on quarter-chord length behind from the leading edge. FBD of the aircraft is shown on the *Figure 2.4*.

Eq. .

Lift forces calculated by *Eq. 2.1* where *q* is dynamic pressure and *A* is area of a member. The total moment about the CoG is:

Eq. .

The moment caused by the lift at quarter-chord of the wing is also calculated by this:

Eq. .

By implementing *Eq. 2.1* and *Eq. 2.3* to total moment equation (*Eq. 2.2*) it gives:

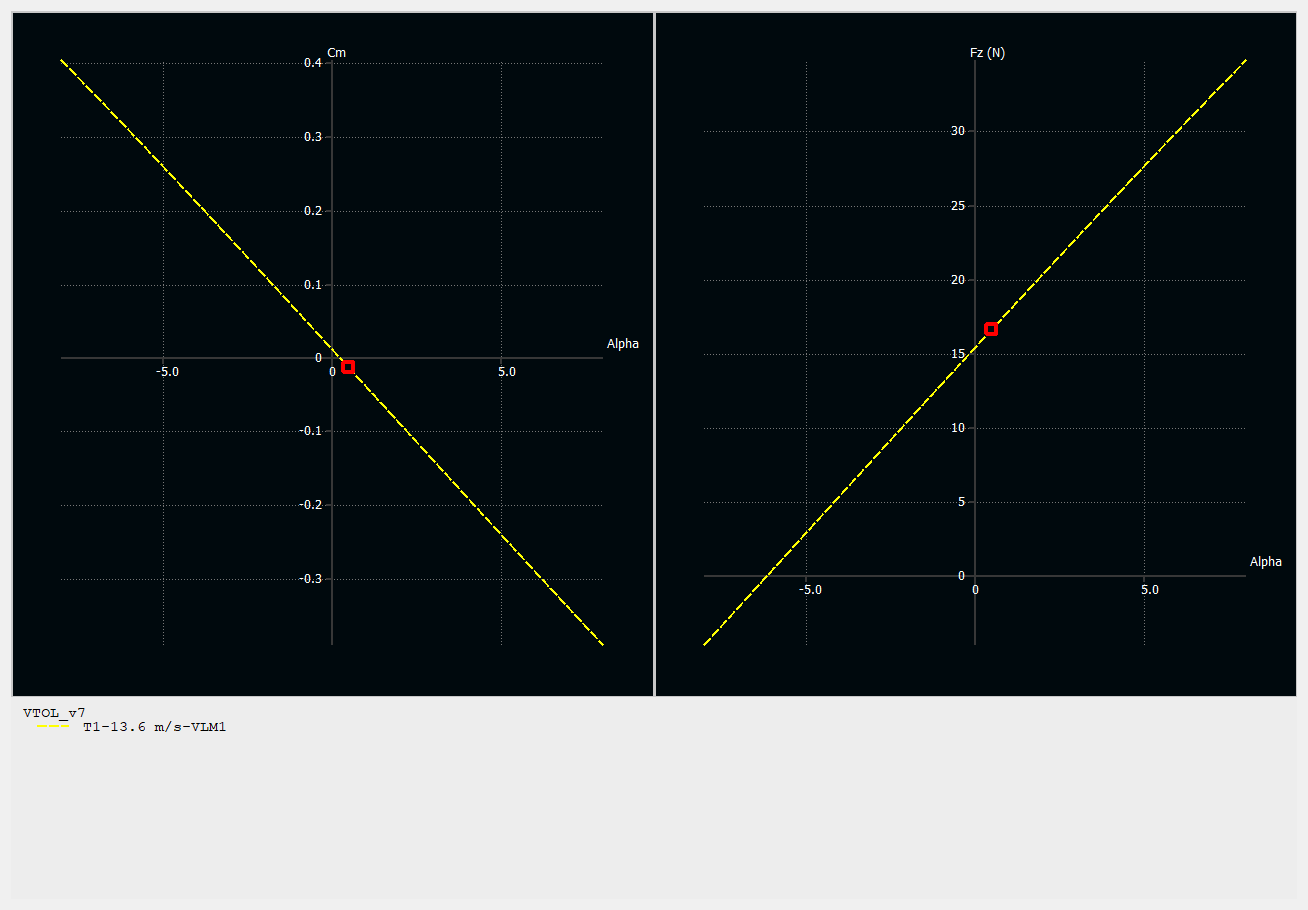
Eq. .

The Tail volume constant is calculated by the *Eq. 2.4* where *M.A.C.* is mean aerodynamic chord length and *L* is lever length of the horizontal stabiliser. For sailplanes, TV for horizontal stabiliser is between 0.5 and 0.7, and during design process of the tail, it was worked in the interval [4]. A horizontal stabiliser was designed with aspect ratio of 3.5, and it was converted a V-Type tail design without changing its projected area. Some geometric properties of the V-Tail stabiliser can be seen in *Table 2.2*. Technical drawings and detailed dimensions of the tail is provided in *Appendix B*.

**Table 2.2** Geometric Properties of the V-Tail Stabiliser.

|  |  |
| --- | --- |
| P. Stabiliser Span *[m]* | 0.500 |
| P. Stabiliser Area *[m2]* | 0.06 |
| Airfoil (NACA) | N-0012 |
| Root Chord *[m]* | 0.180 |
| Tip Chord *[m]* | 0.100 |
| M.A.C. *[m]* | 0.152 |
| Tilt Angle *[˚]* | -3.0 |
| Tail Volume | 0.61 |
| Dihedral Angle *[˚]* | 35.0 |
| Lever Arm, L *[m]* | 0.650 |

Another important concept for the tail design is consideration of the static stability. XFLR5 is able to compute the total moment coefficient of the vehicle. There are two graphs in XFLR5 that should be examined to verify static stability. vs. α graph must have negative slope, it means that the increasing α creates negative pitching moment that will bring the vehicle back to its equilibrium. At equilibrium α, the wings could generate sufficient lift to keep the vehicle on air. As seen in *Figure 2.5*, the aircraft can fulfil both condition at *13.6* *m/s* with generated 16*N* lift . Although the velocity is a bit out of the design margin, it is still acceptable.



**Figure 2.5** Constant Velocity (13.6 m/s) vs. AoA Analysis Results of the Aircraft with V-Tail.

After that stage, the 3D design and dynamic stability analysis were carried out simultaneously. The main reason behind this was the fact that the moment of inertia of the vehicle was required for dynamic stability analysis. SolidWorks had been used as the 3D design software. The vehicle was designed as an outer shell and an internal structure to support the shell. Since additive manufacturing was preferred for production, designing complex internal structure will not cause any major production problem. It was aimed to overcome the disadvantage of low-strength property of the FDM printing technique by utilising pre-produced composite materials in sections requiring strength.

The dimensions of the spar tube to be selected to reduce the deflection and increase the overall strength of the wing were determined by numerical analysis. In order to simplify calculation, internal structure of the wing was not taken into account, and only the wing shell and composite tube were included in the analysis. The second moment of area of the wing profile was computed by the SolidWorks. I Shell for a profile with 250*mm* of chord length and 0.8*mm* of thickness equals to 55082 *mm4*. It was observed that I Shell is proportional to the cube of the chord length with a multiplication constant of 3.53ₓ10-6. The distance between neutral axis of tubular section and neutral axis of the total section is observed as proportional to the chord length with a multiplication constant of 1.22ₓ10-2. Similarities are checked for 5 different section of wing profile and multiplication constants are given in *Table 2.3.*

**Table 2.3** The Geometric Properties and Multiplication Constants of Wing Section with Different Chord Lengths.

|  |  |  |  |  |
| --- | --- | --- | --- | --- |
| Chord Length *[m]* | I Shell *[m4]* | D trans *[m]* | CI *[m]* | CD,T |
| 0.250 | 55082ₓ10-12 | 3.06ₓ10-3 | 3.52ₓ10-6 | 1.22ₓ10-2 |
| 0.220 | 37568ₓ10-12 | 2.68ₓ10-3 | 3.52ₓ10-6 | 1.22ₓ10-2 |
| 0.200 | 28242ₓ10-12 | 2.44ₓ10-3 | 3.53ₓ10-6 | 1.22ₓ10-2 |
| 0.190 | 24213ₓ10-12 | 2.30ₓ10-3 | 3.53ₓ10-6 | 1.21ₓ10-2 |
| 0.180 | 20645ₓ10-12 | 2.18ₓ10-3 | 3.54ₓ10-6 | 1.21ₓ10-2 |

XFLR5 is able to compute bending moment caused by the lift and drag, the results that causes maximum bending moment were exported to MATLAB. Since positions of the motors haven’t yet been decided, the vertical point force generated by the propellers in vertical flight was assumed to be applied at the farthest point of the wing from the centre. The total Flexural Rigidity for a composite beam is derived below.

[5]

Eq. 2.5

Where and are second moment of area about neutral axis of composite beams, and is the equivalent second moment of area for entire beam made of a material with elastic modulus of . The Flexural Rigidity, D is calculated as;

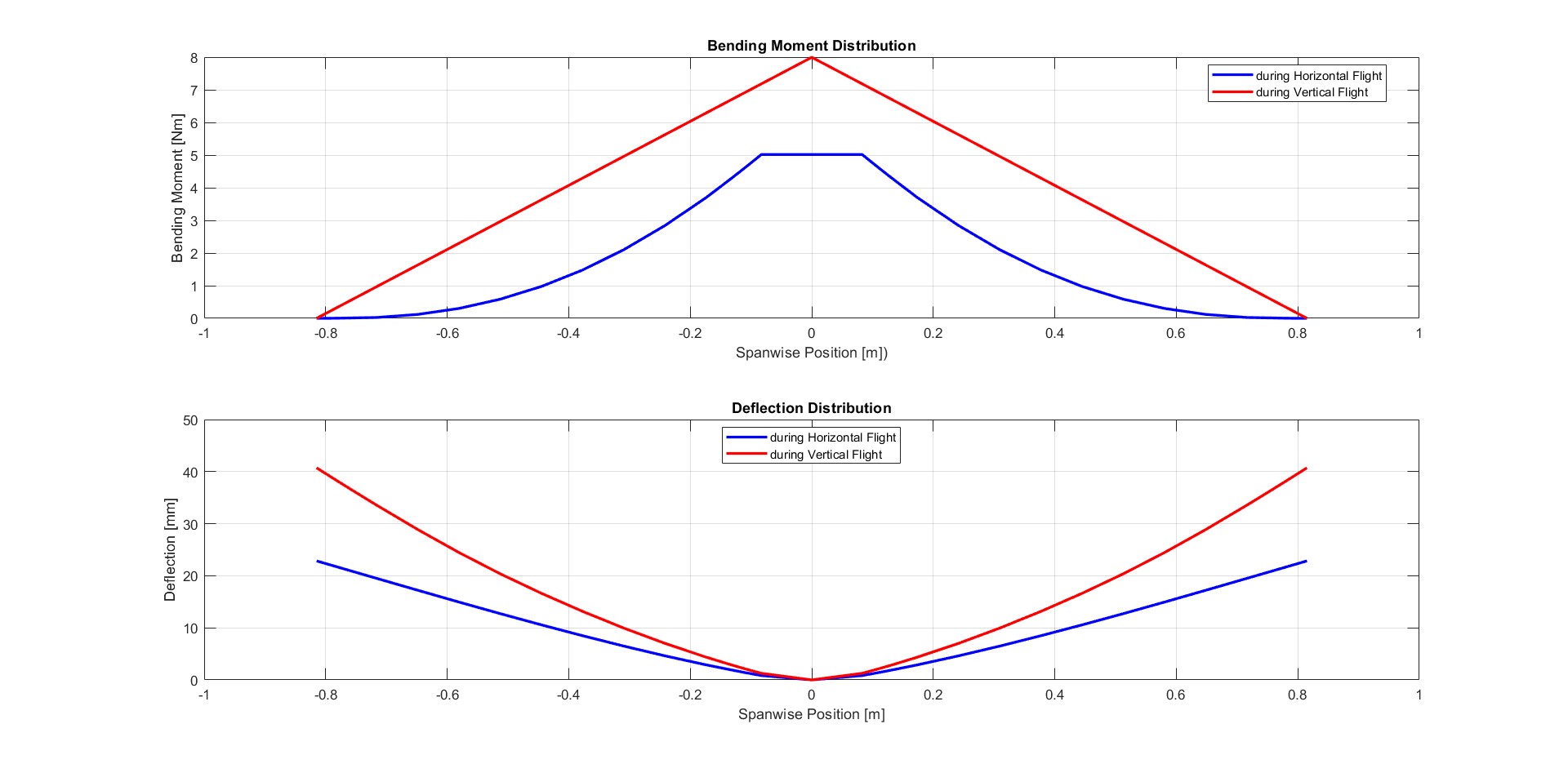
Eq. 2.6

And so on, the deflection of the wing is computed by using Eq. 2.7 and *cumtrapz* function of the MATLAB by applying boundary conditions where the deflection and its slope are equal to zero at the centre. MATLAB Code to compute the deflection is given in *Appendix C.*

[6]

Eq. 2.7

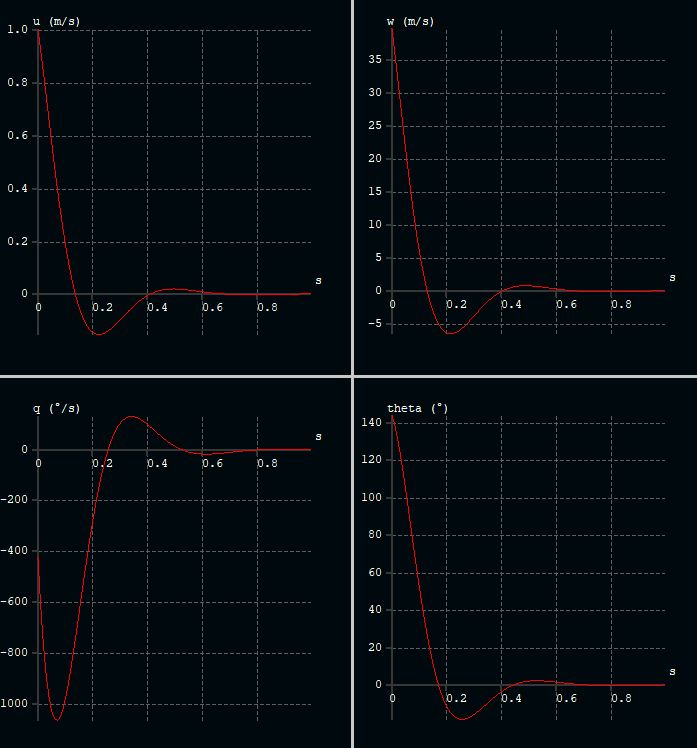
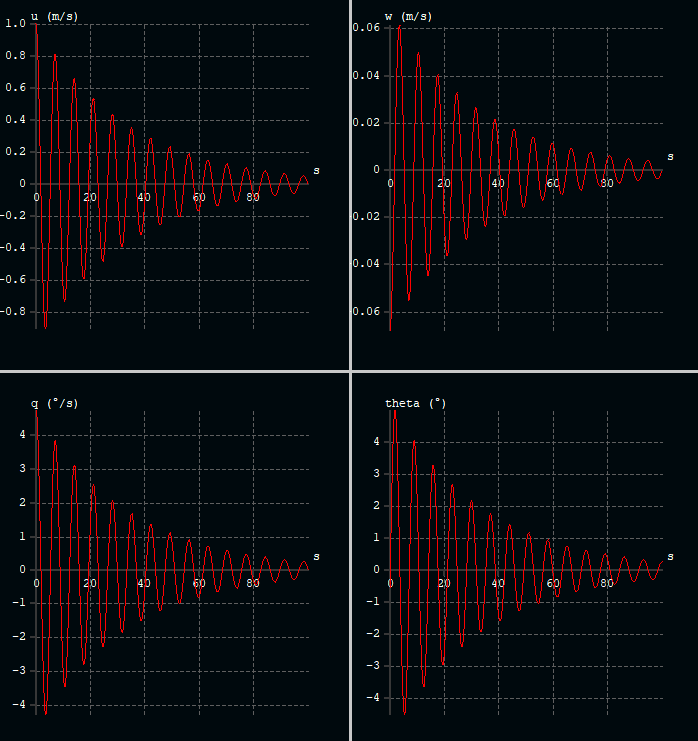
The maximum deflection is expected not to exceed 10% of the wing length, which is 80*mm*. Accordingly, a 500*mm* long CFRP tube with an inner diameter of 10*mm* and an outer diameter of 12*mm* was used as the spar tube. Since the CFRP tube is sold in 1 metre pieces, not exceeding 0.5 metres for each wing will reduce the cost. As seen in the *Figure 2.6*, the maximum deflection which located at wing tips is not exceed 25*mm* in horizontal flight. Although a higher deflection is emerged during vertical flight, the results are still within an acceptable level.



**Figure 2.6** The Numerical Deflection Analysis Results.

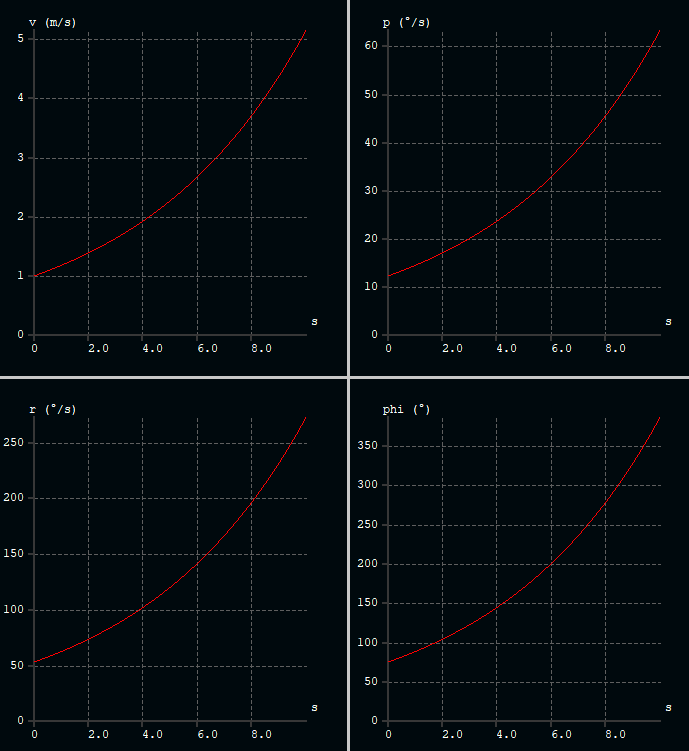
The rough design was finalised taking into account the mentioned concepts. Moment of Inertias of the plane were exported from SolidWorks, and the battery, motors and other significant masses added to XFLR5 as point masses. Thus, the inputs required for the stability analysis were roughly fed into the programme. During the analysis, the damping ratios and frequencies of the natural aerodynamic modes are computed as eigenvectors and eigenvalues by XFLR5. The real parts of these eigenvalues are related to the damping coefficient and their imaginary parts correspond the frequencies. The resulting eight modes can be divided into, four longitudinal and four lateral modes, some of which are symmetric.

The longitudinal modes are two symmetric Phugoid Modes and two symmetric Short-Period Modes. The phugoid is a long period oscillation of change in altitude, that is caused by the exchange of kinetic and potential energy and it is usually lightly damped. For our plane the damping ratio, ζ of this mode is computed to be 0.033 and the damped natural frequency is 0.896 *Hz*. Its duration could be several minutes for a stable aircraft and the settling time is nearly 1.5*min* for the aircraft [7].Modal response is shown in right 4-graphs of *Figure 2.7*. The other longitudinal mode is the Short-Period mode. This mode is related to pitch rate and vertical displacement. It is usually high frequency and damped well. For the plane, the damping ratio, ζ of this mode is computed to be 0.553 and the damped natural frequency is 11.57 *Hz*. Its settling time is expected to be less than a second for a stable aircraft, it takes 0.6*s* stabilise for the aircraft [7]. Modal response is shown in left 4-graphs of *Figure 2.7*.



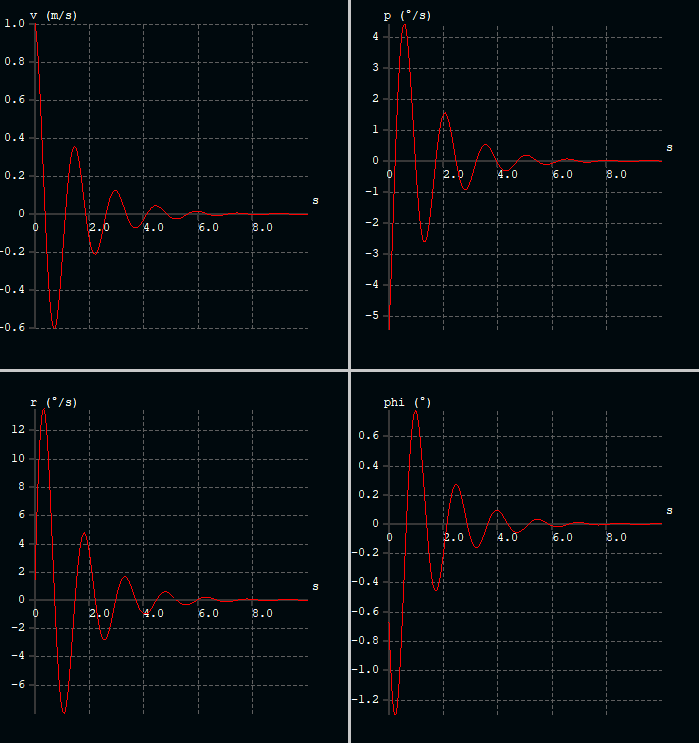
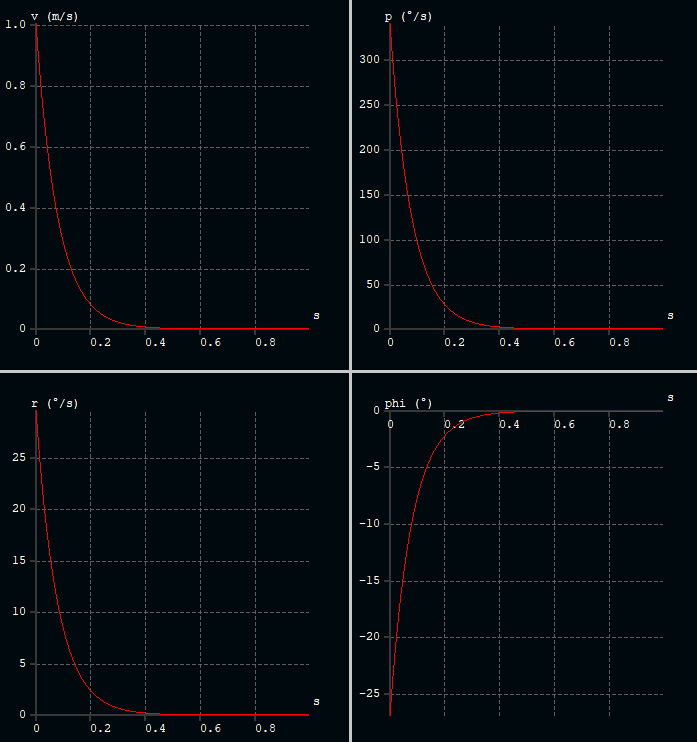
**Figure 2.7** Longitudinal Modal Responses of the Vehicle: (Right-4) Phugoid Mode; (Left-4) Short-Period Mode.

The lateral modes are Spiral Mode, one Roll-Damping Mode and two symmetric Dutch-Roll Modes. The Spiral Mode is primarily a change in heading and it is a non-oscillatory and slow mode and it is usually unstable. This mode is also unstable for the plane as can be seen from the modal response in *Figure 2.8*. Although it is unstable in Spiral mode, it can be easily corrected by the pilot since it is very slow. For the plane, the doubling time is computed to be 4.23*s* and the task of stabilizing it was assigned to the flight controller. Flight controllers with flight control software such as PX4 or Ardupilot, can easily handle such stabilization operations. It also improves the controllability of the vehicle, offers easier flight modes for the pilot and can operate the vehicle in autonomous flight modes [8].



**Figure 2.8** Spiral Mode Modal Responses of the Vehicle.

Another lateral mode is Roll-Damping which is related to a change in roll. This mode is non-oscillatory and usually fast. For our plane the halving time is computed to be 0.055*s*. The modal response of Roll-Damping mode is shown in right 4-graphs of the *Figure 2.9*. Lastly there is the Dutch-Roll mode which is a combination of roll and yaw change with a 90˚ phase difference. Dutch-Roll mode is usually lightly damped. For the plane, the damping ratio, ζ of Dutch-Roll mode is computed to be 0.165 and the damped natural frequency is 4.22 Hz. Modal response is shown in left 4-graphs of the *Figure 2.9*



**Figure 2.9** Lateral Modal Responses of the Vehicle: (Right-4) Roll-Damping; (Left-4) Dutch-Roll Mode.

One of the important parameters for a reliable vertical flight is the thrust-to-weight ratio. It is calculated as the maximum thrust obtained from the propellers multiplied by the number of rotors divided by the weight of the vehicle. While thrust-to-weight ratio is accepted in the range of 2-4 for quadcopters, it has been found that the range of 1.5-3 is recommended for VTOL vehicles that perform vertical flight for a short period of their flight. Considering a 3-rotor VTOL with a maximum take-off weight of 2 kg, the motors are expected to generate 1 to 2 kg-force static thrust. After the market research, Emax RS2205S-2300Kv model BLDC motors were selected. They are able to produce 1281 gr-force thrust with 3 blade 5045 propellers and they consume maximum of 33A current with nominal voltage of 16V. With this configuration, the thrust-to weight ratio is calculated as minimum of 1.92. Another market research was also conducted out for servo actuators to be used in the tilt mechanism. It was decided that Emax ES09MD Dual–Bearing servo actuators are suitable for the system. They have 2.6 kg-force.cm stall torque with angular speed of 6.55 rad/s at operating voltage of 6V. Thanks to dual-bearing design of output shaft, they can withstand not only twisting but also bending moments.

As the plane has three rotors, two at the front and one at the back; their positions must be designated. In the designation of rotors, there are several important criteria. In the vertical thrust, it is desired that the rotors should produce the same thrust. That requires proper placement of rotors with respect to centre of gravity. As *equation 2.8* shows, the position of back rotor is the double distance of front rotors with centre of gravity in z axis. As the propellers produces vertical thrust, the thrust should not interfere with either the wing or the fuselage. The diameter of 3 blade 5045 propellers is *127mm*. With that size and the dimension of plane in account, the positions of front and back rotors are located *160mm* and *320mm* away from the centre of gravity. The other important criterion for the designation is the distances between rotors. To achieve optimized balance and stability, the rotors are placed equidistant from each other. That allows symmetrical thrust distribution and more stable control. With that the designation of rotors are finalized as shown in *Figure 2.10*.

Eq. 2.8

A drawing of a couple of cones

Description automatically generated with medium confidence

**Figure 2.10** The Designation of Rotor Positions.

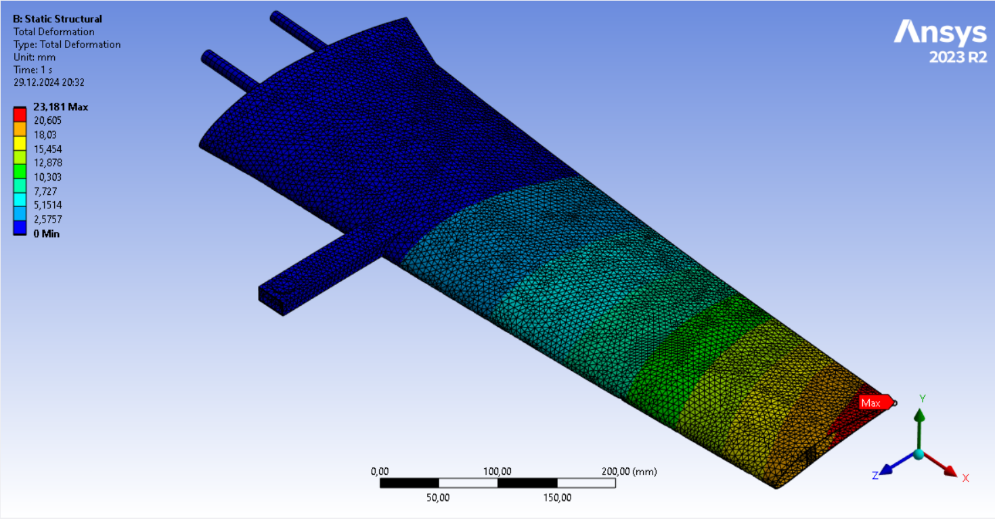
After the positions and necessary mechanism are determined, tilt mechanism must be designed accordingly. The positions of the BLDC motors are sufficiently enough from the wings. That brings additions in the wings. According to these considerations, it is decided that the servo actuators can be directly connected to the BLDC motors. That simplifies the tilting mechanism design and allows more stable structure. The servo motors are located in the inside of the support addition. Therefore, the design of these support units is based on the size of servo motors, while maintaining structural integrity and strength. The connection between servo motors and BLDC motors are designed in a way that the thrust produced by the propellers does not create any moment in the rotating axis of servo motors. That lowers the force applied on the servo motors. The finalized design of tilting mechanism is shown in *Figure 2.11.*

*A close-up of a propeller

Description automatically generated*

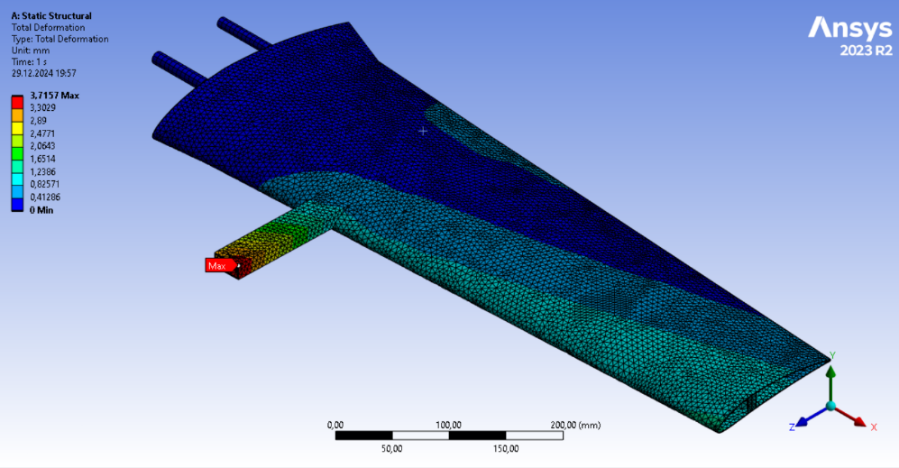
**Figure 2.11** The Tilting Mechanism.

The next stage of the design is the structural analysis of the wing members. In the process, the finite element analysis software Ansys was used. The loads acting on the wing during both horizontal and vertical flights were applied and wing mounting joints were established, and the results was examined for both loadings. Since the materials produced with the 3D printer exhibit orthotropic behaviour, it wouldn’t be a reasonable approach to assign isotropic materials from the material library. Therefore, academic article research was carried out, and a paper that has an experimental approach to mechanical properties of 3D printed PLA samples was found. The mechanical properties of the material were taken from this article and entered into the software. Mechanical properties such as shear yield stress, Poisson’s Ratio which are not mentioned in the article were assumed to be the same as raw PLA. Young’s Modulus of FDM printed PLA material is *2251.4 MPa* in axis parallel to layers and *1951.5 MPa* in axis perpendicular to layer lines for *0.2mm* layer heights. [9] From the analysis; displacement, equivalent Von-Misses stress and safety factor according to the yield strength of the assigned material were obtained.



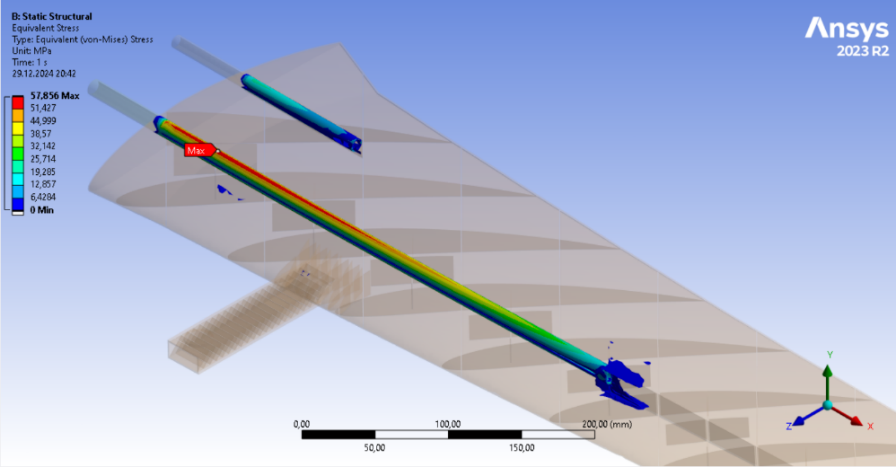
**Figure 2.12** Wing Total Deformation Results of FEA for Horizontal Flight.

As can be seen in the *Figure 2.12*, the maximum deflection during horizontal flight occurs at the wing tips, as expected. Also the maximum deflection was in consistent with the results of the numerical analysis. The maximum deformation found to be *25mm* in the numerical analysis was observed as *23mm* in the FEA results. In addition, the maximum elastic deformation in vertical flight was observed as *3.7mm* at the end of the tilt mechanism where the motors are attached. It can be seen in *Figure 2.13.* The deflection was kept so low thanks to the dense interior infill applied to the extension arm and wing contact surface.



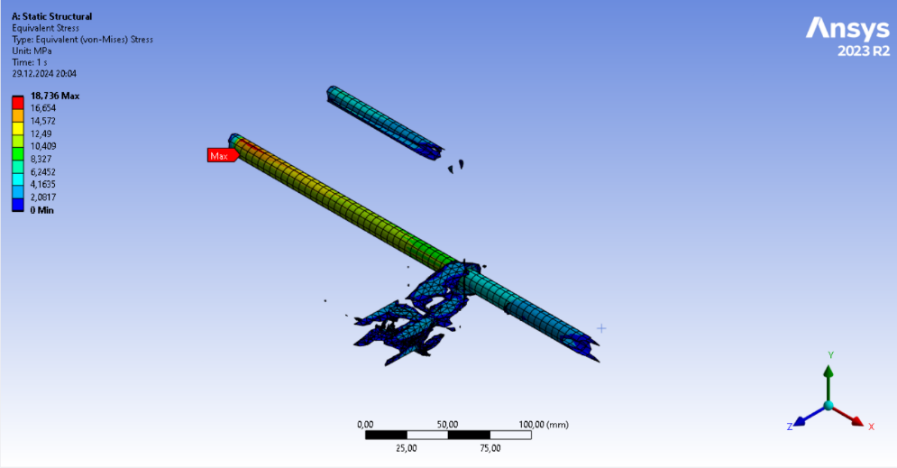
**Figure 2.13** Wing Total Deformation Results of FEA for Vertical Flight.

The Equivalent stress on the wing during horizontal flight was found to be concentrated on the carbon spar pipe. Therefore, iso-clipping is performed for stresses below *5MPa* for better visualisation of stress concentration points which are marked in both *Figure 2.14* and *Figure 2.15*. The maximum equivalent stress is found to be *57.8MPa* for horizontal flight. The stress development area on the spar pipe which is coloured red-orange lays between *50-58MPa* range. Both can be seen in *Figure 2.14.*



**Figure 2.14** Equivalent Stress on the Wing Results of FEA for Horizontal Flight.

For vertical flight loads were applied at the tip of the arm, and maximum equivalent stresses were found lower than the horizontal flight results (See *Figure 2.15*). It is as expected because load is 12N which is maximum static thrust can be generated by rotors and also moment arm is shorter than the horizontal flight load. The stress development starting from the contact point of the arm and wing to tip of spar pipe is shown in *Figure 2.15.*



**Figure 2.15** Equivalent Stress on the Wing Results of FEA for Vertical Flight.

Even if the applied load of 19N at wing tip which develops higher bending moments than generated by the lift at all points of the wing, resulting stresses way lower than the materials yield stresses. Safety factor with respect to yield stress of the materials was also provided by Ansys. Under horizontal flight loads; minimum safety factor is 1.96 for structures made of PLA, and minimum safety factor is calculated as 3.94 for carbon structures which are spar pipe and back mounting pipe. Under vertical flight loads; minimum safety factor is 1.72 for structures made of PLA, and minimum safety factor is calculated as 11.4 for carbon structures which are spar pipe and back mounting pipe.

At the end of the design, it was intended to evaluate the performance of the aircraft. For that purpose, the power consumption of the aircraft in horizontal and vertical flight had to be calculated with reasonable assumption and formulas. The height at which the transition to horizontal flight from vertical flight is called the transition height and the required times to smooth transition is called the transition time. In Ardupilot, the flight control software, the default value of transition height is *30m*, and the default value of transition time is *7s*. The vertical speed up value for autonomous modes is set to be *1m/s* as default. With these parameters, it takes *30s* to reach the transition height. It is assumed that the power consumption during transition is approximately equals to the power consumption of the vertical flight.

Generated drag during vertical ascending is assumed to be 20% of the aircraft weight which is *19.62N.* The force required to be generated by the rotors to maintain static equilibrium is

Eq. 2.9

Dynamic thrust generated by a propeller with diameter of *d = 5inch*, pitch of *p = 4.5inch* and rotational speed ω in RPM can be calculated with Eq. 2.10. Simplified form of the equation is given, and it was derived by Gabriel Staples.

[10]

Eq. 2.10

Where is propeller axial velocity in *m/s.* During vertical flight, propeller axial velocity equals to *1m/s.* After solving Eq. 2.9 and Eq. 2.10 together, it was concluded that required angular velocity of propeller is approximately *23600 RPM*. When the manufacturer data sheets are analysed, it can be seen that the motor rotating at this angular speed consumes approximately *355W* electrical power. Total energy consumption is calculated as *1065W* during vertical flight.

The aircraft is experienced a drag force of *2.5N* at cruse speed. It is sum of parasitic and induced drag forces generated by the wing and the elevator. XFLR5 is capable of compute these forces. Drag force experienced by the fuselage could not be calculate. It is assumed that fuselage drag equals to what rest of the aircraft is experienced. So that, total drag is equals to *5N.* The force required to be generated by the rotors to maintain static equilibrium is *2.5N* for each tilting rotor.

By using equation Eq. 2.9 where equals to cruise speed which is *13.6m/s,* it was concluded that required angular velocity of propeller is approximately *17200 RPM*. The motor rotating at this angular speed consumes approximately *145W* electrical power. Total energy consumption is calculated as *290W* during vertical flight.

The battery planned to be used is a 4S Li-Po battery with a nominal voltage of 14.8V and a capacity of 4500mAh. It can storage *71.1Wh* energy, and 80% of it can be used without damaging the battery. The aircraft consumes *21.9Wh* energy during take-off, landing and transition. With the remaining *35Wh* of energy, the aircraft can perform *7.2min* of horizontal flight. With a total flight time of 8.5 minutes, the vehicle has a range of *5.9km*.

## Project Management

Work Packages (tasks and subtasks), roles of team members, resources, Ghent Chart including this term (ME 429) and next term (ME492) with clearly stated subtasks, milestones, etc.

# Discussion

This section may be 2-4 pages. In this section, statements given in Design Process are discussed and interpreted. Future work should also be stated. What further research can be done in the field you have chosen? Highlight any failures, problems or constraints that have affected progress, and describe the measures taken to respond to them. Describe key lessons learned, that are important to your project or that may be of use to others doing future work related to the project. They may relate to any of the following: successes, strategies adopted, challenges you are facing, surprise results, management processes, or technical understanding. Explain the importance of the topic in your future professional life and society in general. Provide some self-reflection about the design and report writing process. How do you evaluate the contributions of this design process to your academic development? Do you intend to work in the future in the field in general and the topic you have chosen in particular?

-Cannot performed cfd

-more reliable structure for arm connection to develop stresses gently

-lw pla option

# Conclusion

This section is a restatement of the information given in the report overall. No new topics are introduced or discussed. Conclusions/implications are drawn. This section may be 1-2 pages.

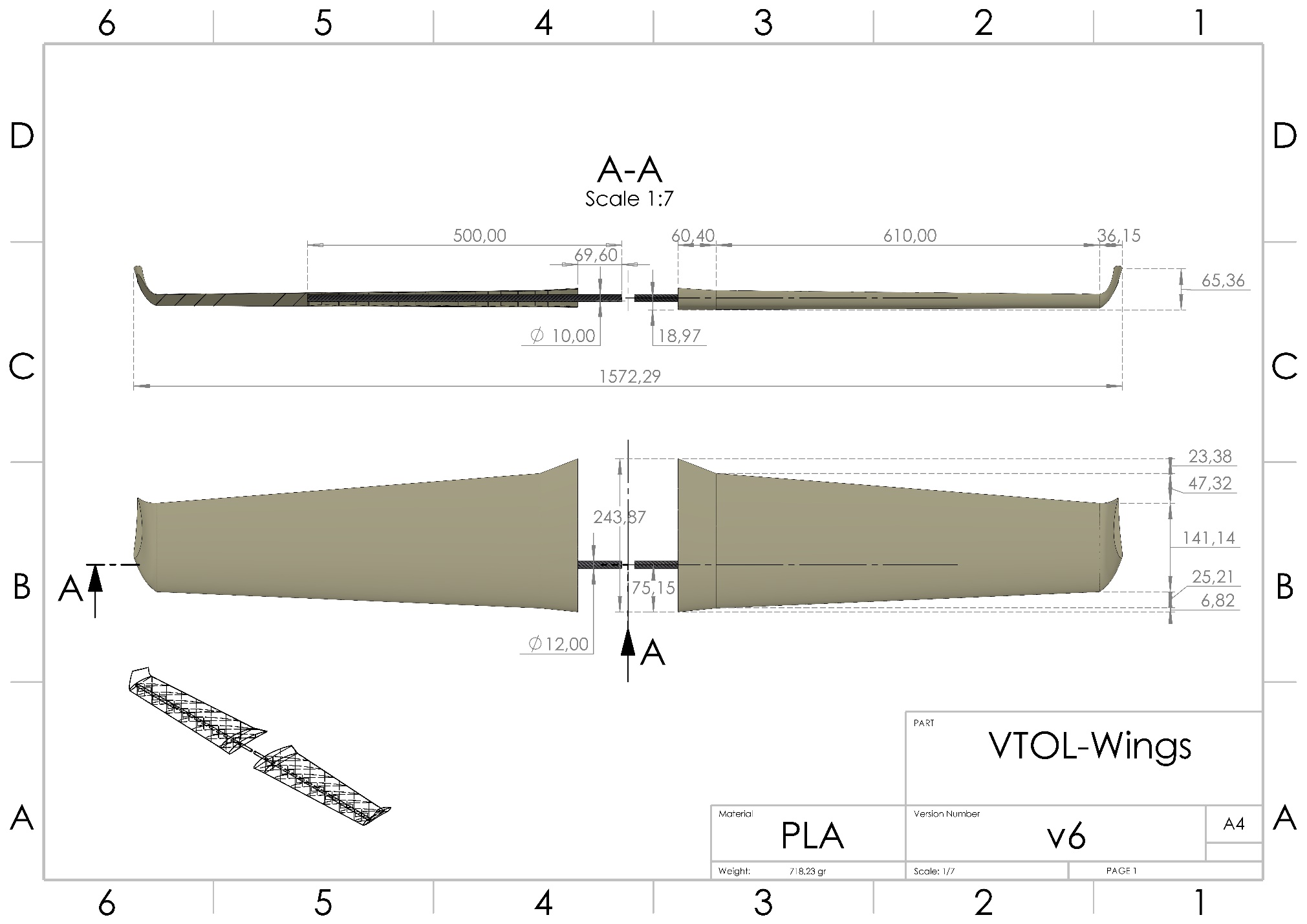
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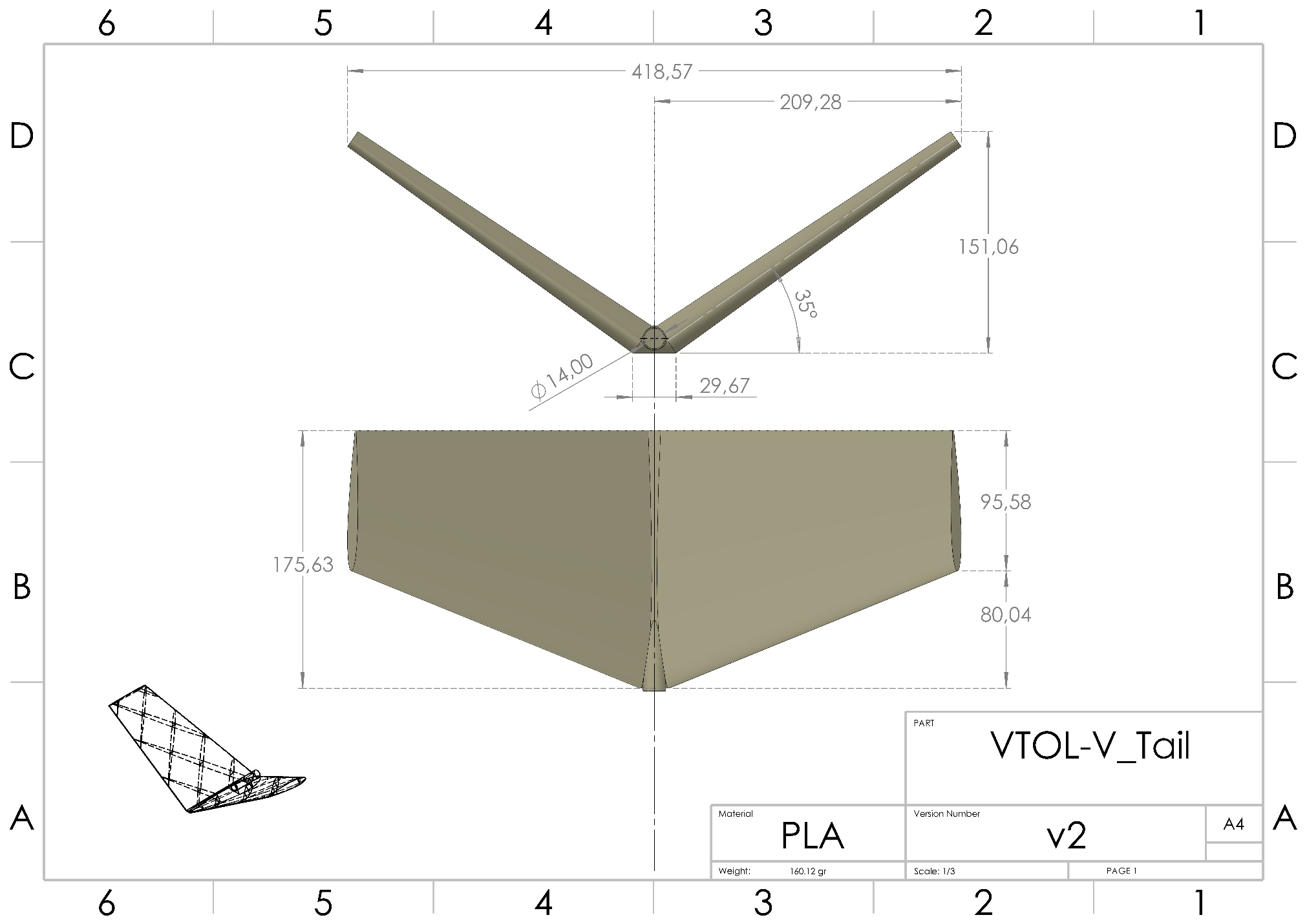
# Appendices

All source codes, technical drawings with dimensions, material data sheets etc. should be givens in appendices as “Appendix A: Python source code for pattern recognition” Appendix B: “Material Data Sheet for foaming Agent” etc.

**Appendix A:** The Technical Drawings of the Wings



**Appendix B:** The Technical Drawing of the V-Tail



**Appendix C:** MATLAB Code for Numerical Deflection

% Imported data from xflr5

wing\_y = [-0.815 -0.805 -0.7975 -0.7925 -0.7875 -0.7825 -0.7775 -0.7725 -0.7675 0.7625 -0.7588 -0.7562 -0.7538 -0.7512 -0.7161 -0.6483 -0.5806 -0.5128 -0.445 -0.3772 -0.3094 -0.2417 -0.1739 -0.137 -0.131 -0.125 -0.119 -0.113 -0.107 -0.101 -0.095 -0.089 -0.083 0 0.083 0.089 0.095 0.101 0.107 0.113 0.119 0.125 0.131 0.137 0.1739 0.2417 0.3094 0.3772 0.445 0.5128 0.5806 0.6483 0.7161 0.7512 0.7538 0.7562 0.7588 0.7625 0.7675 0.7725 0.7775 0.7825 0.7875 0.7925 0.7975 0.805 0.815];

M\_b\_hrz = [0 0.0004 0.0013 0.002 0.003 0.0041 0.0055 0.0071 0.0088 0.0106 0.012 0.0128 0.0137 0.0146 0.0258 0.1201 0.3031 0.5857 0.976 1.48 2.1015 2.8414 3.6968 4.2199 4.3059 4.3927 4.4801 4.5682 4.6569 4.7462 4.836 4.9263 5.0169 5.0169 5.0169 4.9263 4.836 4.7462 4.6569 4.5682 4.4801 4.3927 4.3059 4.2199 3.6968 2.8415 2.1015 1.48 0.976 0.5857 0.3031 0.1201 0.0258 0.0146 0.0137 0.0128 0.012 0.0106 0.0088 0.0071 0.0055 0.0041 0.003 0.002 0.0013 0.0004 0];

chord = [0.08 0.09 0.0975 0.1025 0.1075 0.1125 0.1175 0.1225 0.1275 0.1325 0.1362 0.1387 0.1412 0.1437 0.1492 0.1575 0.1658 0.1742 0.1825 0.1908 0.1992 0.2075 0.2158 0.2215 0.2245 0.2275 0.2305 0.2335 0.2365 0.2395 0.2425 0.2455 0.2485 0.2485 0.2485 0.2455 0.2425 0.2395 0.2365 0.2335 0.2305 0.2275 0.2245 0.2215 0.2158 0.2075 0.1992 0.1908 0.1825 0.1742 0.1658 0.1575 0.1492 0.1437 0.1412 0.1387 0.1362 0.1325 0.1275 0.1225 0.1175 0.1125 0.1075 0.1025 0.0975 0.09 0.08];

% Vertical Flight Bending Moment

M\_b\_vrt = 9.81\*2\*(0.815-abs(wing\_y));

d\_o = 0.012; % Outer diameter of the tube [m]

d\_i = 0.010; % Inner diameter of the tube [m]

A\_tube = pi \* (d\_o^2 - d\_i^2) / 4; % Tube Area [m^2]

I\_tube = pi \* (d\_o^4 - d\_i^4) / 64; % Tube moment of inertia [m^4]

E\_tube = 39e9; % Elastic modulus of CFRP [Pa]

d\_st = (1.22e-2) \* chord; % Distance btw General Neutral Axis and Tube Neutral Axis [m]

I\_tube\_trasformed = I\_tube + A\_tube\*d\_st.^2; % Transformed I of shell

L\_tube = 0.5; % Length of the tube in half of the wing [m]

I\_shell = (55082.565563e-12/0.25^3) \* chord.^3; % Shell moment of inertia [m^4]

E\_shell = 1.951e9; % Elastic modulus of PLA [Pa]

% ---------Flexural Ridigity Computation---------

EI = zeros(0,length(wing\_y));

for i = 1 : length(wing\_y)

if abs(wing\_y(i)) < 0.084

EI(i) = E\_tube\*I\_tube;

elseif abs(wing\_y(i)) < L\_tube

EI(i) = E\_shell\*I\_shell(i) + E\_tube\*I\_tube\_trasformed(i);

else

EI(i) = E\_shell .\* I\_shell(i);

end

end

% ---------Horizontal Flight Deflection---------

% First integration: Calculate slope

slope = cumtrapz(wing\_y, M\_b\_hrz ./ EI);

% Adjust slope to enforce boundary condition: slope(0) = 0

slope = slope - slope(find(wing\_y == 0, 1));

% Second integration: Calculate deflection

deflection = cumtrapz(wing\_y, slope);

% Adjust deflection to enforce boundary condition: deflection(0) = 0

deflection\_hrz = (deflection - deflection(find(wing\_y == 0, 1))).\*1000;

% ---------Vertical Flight Deflection---------

% First integration: Calculate slope

slope = cumtrapz(wing\_y, M\_b\_vrt ./ EI);

% Adjust slope to enforce boundary condition: slope(0) = 0

slope = slope - slope(find(wing\_y == 0, 1));

% Second integration: Calculate deflection

deflection = cumtrapz(wing\_y, slope);

% Adjust deflection to enforce boundary condition: deflection(0) = 0

deflection\_vrt = (deflection - deflection(find(wing\_y == 0, 1))).\*1000;

% Plot bending moment

figure;

subplot(2, 1, 1);

plot(wing\_y, M\_b\_hrz, 'b-', 'LineWidth', 2);

hold on;

plot(wing\_y, M\_b\_vrt, 'r-', 'LineWidth', 2)

grid on;

xlabel('Spanwise Position [m])');

ylabel('Bending Moment [Nm]');

title('Bending Moment Distribution');

legend("during Horizontal Flight”, “during Vertical Flight”, Location="best");

% Plot deflection

subplot(2, 1, 2);

plot(wing\_y, deflection\_hrz, 'b-', 'LineWidth', 2);

hold on;

plot(wing\_y, deflection\_vrt, 'r-', 'LineWidth', 2);

grid on;

xlabel('Spanwise Position [m]');

ylabel('Deflection [mm]');

title('Deflection Distribution');

legend("during Horizontal Flight”, “during Vertical Flight”, Location="best");

# *Appendix D: The Technical Drawing of the VTOL Assembly.*

