Preliminary Design of High Altitude and High Endurance UAV: **SAURON**

Spring 2014

Team: NSFW

Team Members: Nisherag Gandhi, Thomas Gempp, Douglas Rohrbaugh, Gregory Snyder, Stephen Stanek, Victor Thomas
# Contents

INTRODUCTION ...................................................................................................................... 1

DESIGN CHALLENGES ........................................................................................................... 2

DESIGN REQUIREMENTS ........................................................................................................ 3

FUSELAGE DESIGN ................................................................................................................. 9

WING AND TAIL DESIGN ....................................................................................................... 10

- Wing – Airfoil and Drag Analysis ....................................................................................... 35
- Tail – Airfoil and Drag Analysis ......................................................................................... 36

DRAG AND POWER REQUIRED IN TRIMMED FLIGHT ...................................................... 39

- Sea level ............................................................................................................................... 39
- 45,000 ft. Altitude ............................................................................................................... 40
- 61,000 ft. Altitude ............................................................................................................... 41
- 79,000 ft. Altitude ............................................................................................................... 42

FLIGHT ENVELOPE .................................................................................................................. 45

- Absolute and Operational Ceiling .................................................................................... 45
- Climb rate ............................................................................................................................ 46
- Time to climb ....................................................................................................................... 47

MANEUVERS ............................................................................................................................ 48

OPERATIONAL RANGE ............................................................................................................ 53

PROPULSION SYSTEM ............................................................................................................. 54

LANDING GEAR ...................................................................................................................... 55

TAKEOFF AND LANDING ........................................................................................................ 56

- Takeoff ................................................................................................................................. 56
- Landing ................................................................................................................................. 57

FINAL THOUGHTS .................................................................................................................... 58
List of Figures

Figure 1. Sauron fuselage, side profile ................................................................. 9
Figure 2. Reynolds Number along the length of the fuselage ................................. 10
Figure 3. Sauron wing, horizontal and vertical stabilizers in AVL ............................. 13
Figure 4. Four-View of the Sauron .................................................................. 9
Figure 5. SM701 Drag Bucket Analysis ................................................................ 36
Figure 6. NACA 63-015A Drag Bucket Analysis ................................................... 37
Figure 7. Joukovsky 0015 Drag Bucket Analysis .................................................... 38
Figure 8. Drag build up and power required to cruise at sea level ............................ 40
Figure 9. Drag build up and power required to cruise at 45,000 ft ......................... 41
Figure 10. Drag build up and power required to cruise at 61,000 ft ......................... 42
Figure 11. Drag build up and power required to cruise at 79,000 ft ......................... 43
Figure 12. h vs. v graph to determine absolute ceiling .......................................... 46
Figure 13. Rate of climb graph ......................................................................... 47
Figure 14. Time to climb graph ........................................................................ 48
Figure 15. Plots used in determining the load factor .............................................. 49
Figure 16. Load Factor vs. Velocity at Sea Level for Instantaneous Maneuvers ......... 50
Figure 17. Steady state bank radius plots at operational altitudes ............................ 51
Figure 18. Steady state turn rate plots at operational altitudes ............................... 52
Figure 19. Map of operational limitations .............................................................. 53
Figure 20. JM1S Electric Joby Motor .................................................................. 54
Figure 21. Sample variable pitch propulsion system from Northwest UAV ............. 54
Figure 22. Sample Aero Telemetry landing gear .................................................... 55
Figure 23. Anticipated landing gear configuration ............................................... 56
Figure 24. Takeoff configuration schematic .......................................................... 57
Figure 25. Landing configuration schematic ......................................................... 58

List of Tables

Table 1. US Border Lengths by Geographical Area ................................................. 1
Table 2. Wing and Tail specifications .................................................................. 13
Table 3. SM701 coefficients at operational floor and ceiling ................................. 35
Table 4. Airfoil Analysis of NACA 63-015A and Joukovsky 0015 ......................... 37
Table 5. Elevator deflection required to trim the aircraft at different lift coefficients .. 32
Table 6. Summary of cruise conditions are varying altitudes ............................... 45
Table 7. Summary of maneuvers at operating altitudes ......................................... 50
Table 8. Summary of takeoff parameters at different altitudes .............................. 56
Table 9. Summary of landing parameters under different conditions at different altitudes ........ 57
INTRODUCTION

Every day, the United States and many other States face attacks and illegal activity along their borders; therefore there is a much needed demand for long term surveillance over a wide area. Every day, thousands of ships travel in and out of US ports requiring additional security and tracking of all before entering US waters. In 2012, approximately 1,292,080,082 metric tons of cargo was moved in and out of US ports. There has been a decrease in the metric tonnage of waterborne cargo in recent years due an increase in air transport. However, monitoring this maritime traffic, outside of radar tracking, should not be overlooked.

Although ports serve as the main gateway for imports and exports to the US and a serious security concern, there needs to be concern for all US border locations. As Table 1 shows, the US has approximately 16,803 miles of border, both coastal and inland, to protect at all times. This is a daunting task for Customs and Border Protection (CBP) to monitor such a wide area.

Table 1. US Border Lengths by Geographical Area

<table>
<thead>
<tr>
<th>Location</th>
<th>Length (mi)</th>
</tr>
</thead>
<tbody>
<tr>
<td>US/Canada Border</td>
<td>2,069</td>
</tr>
<tr>
<td>US/Mexico Border</td>
<td>1,631</td>
</tr>
<tr>
<td>Atlantic Coast</td>
<td>2,069</td>
</tr>
<tr>
<td>Gulf Coast</td>
<td>1,631</td>
</tr>
<tr>
<td>Pacific Coast</td>
<td>7,623</td>
</tr>
<tr>
<td>Arctic Coast (Alaska)</td>
<td>1,060</td>
</tr>
<tr>
<td><strong>Total</strong></td>
<td><strong>16,083</strong></td>
</tr>
</tbody>
</table>

The number of legal border crossings at the US/Canada and US/Mexico borders has been on a steady decrease in recent years; however, the need to additional surveillance along these borders has not changed. In February, 2013, 4,800 illegal immigrants were arrested in the Rio Grande Valley of South Texas. A month later in March, 7,500 more were arrested in the same location; this number is an increase of 2,800 from January. Additionally, over the decade, as the number of border agents has increased, the number of apprehensions of illegal immigrants at borders has decreased. This could possibly be due to the lack of wide area surveillance that is needed to cover 16,803 miles of border. Another problem that faces CBP, is the distance which
international waters start. US jurisdiction reaches out 200 nautical-miles, which requires surveillance over an even wider area then along physical borders.

With the increasing number of cuts in government spending affecting the Department of Homeland Security (DHS), there is certainly a demand for improvements which will prove to be the most effective without the extreme costs included. Currently, CBP has implemented the MQ-9 Predator B unmanned aircraft system (UAS) for homeland security operations. While the Predator is known for its maneuverability and speed, the cost of running and maintaining such a system is not feasible for the future if these budget cuts are to continue. Additionally, the Predator needs to be operated from the ground using Ground Control Stations (GCS), which requires two trained pilots to maintain flight operations at all times while the predator is airborne. The solution to this problem would be the implementation of a high-altitude/long-endurance (HALE) UAV. This document will outline the mission requirements, constraints and design challenges facing this team and how those topic areas will be implemented in continuing the design of a UAV for border protections and surveillance.

**DESIGN CHALLENGES**

The reason to fly at high altitudes is to stay outside of controlled air space for uninterrupted flight. As altitude is increases, density drops significantly. This poses a problem as density drops, velocity must increase to produce enough lift to maintain flight. Furthermore, because many variables depend upon velocity a few are affected negatively. Such variables include lift, drag, and the power required to produce enough thrust.

Endurance depends upon many factors. A greater endurance means the plane can stay up longer. Endurance can be affected by many factors, some which can be controlled and others that cannot, such as weather. The weather is addressed with the aircraft flying at high altitudes where weather can easily be predicted or avoided. Factors that can be controlled consist of wing span and power from the batteries. As wingspan increases, the aircraft’s weight will increase and more power will be demanded from the battery. To sustain battery power, solar cells are used to harvest solar energy. During the night, when the sun’s energy is not available, the aircraft will go into a sinusoidal descent, to 45,000 ft., which will minimize power usage and then ascend to 61,000 ft. during the day. Also, today’s battery technology consists of a charge density of 350 W-hr/kg. In order to carry less battery weight, a higher lift to drag ratio and a lower cruising velocity are needed.

One of the challenges of this design is to meet is to be able to carry out a variety of missions, ranging from surveillance to scientific. This means that the design has to be able to accommodate a wide range of equipment, with varying weights at forty to sixty-five thousand feet. This means
that the aircraft has to be able to accommodate a reasonable payload at the intended altitude for one to several days reliably.

**DESIGN REQUIREMENTS**

The mission requirements that drove this design include making an aircraft that could carry several pieces of surveillance equipment at an altitude of at least 45,000 ft. for several days at a time using electrical power. To meet these requirements we will use a solar powered aircraft with a battery that is capable of maintaining the necessary thrust, for steady level flight, for around twenty-six hours without recharging. This craft will use propellers to generate thrust as it will work best with a battery powered aircraft. A payload weight of at least 200 pounds was decided upon to conservatively accommodate several pieces of surveillance and/or scientific equipment, dependent upon the drone’s particular mission, as well as an additional battery to power them.

To meet the aerodynamic requirements of designing a high-altitude aircraft with long endurance, historical data was used from other long endurance aircraft to find the range our design should have in terms of Aspect Ratio, Lift/Drag ratio (L/D), cruise speed, empty weight ratio, and propeller efficiency that would best fulfill our mission requirements. Looking at other high altitude manned and unmanned aircraft, the goal was to achieve high Aspect Ratio which is very similar to sailplanes and would be reasonably achievable to meet the mission goals. Since this aircraft would operate in a similar manner as sailplanes a range of 60 to 90 knots would be the range in which this design would be assumed to travel at. For our initial design specs, we chose values that were similar to sailplanes that had similar mission parameters as our aircraft. As with a typical sailplane, an initial airfoil with a trim coefficient of lift of 0.8 was chosen. An initial empty weight ratio estimation of 0.57 was chosen as most other high altitude prototypes had an empty ratio between 0.5 and 0.6. A propeller efficiency of 0.88 was chosen as other prototype HALE UAVs used the same propeller efficiency. Through several design iterations some of these specifications have significantly changed. Sauron currently has a trimmed lift coefficient of 0.66 at cruise altitude. We have also changed our empty weight ratio estimation to 0.5, close to the low end of the initial range we were looking at. No changes were made to the initial proposed propeller efficiency. We also changed other design requirements through these iterations. Our UAV now has a designed payload of 250 lbs. and operates at higher altitudes than initially proposed to better navigate the aircraft outside commercial and military air traffic.
DESIGN CONSTRAINTS

One of the major constraints directing the initial concept design of this aircraft was finding a battery with the necessary charge density to have a reasonable takeoff weight. The battery would then be used to find a comparable wing area to other aircraft that operate under similar conditions, which would allow us to calculate the anticipated wing loading and generate an aspect ratio for the design. Research into battery charge densities showed that a lithium sulfur battery, with a charge density of 350 W-hr/kg, will be used. With that, from historic data, stall velocity of 37 ft/s at sea level was determined. Working with these assumptions, the design would have a battery ratio of 0.21.

![Figure 1. Aspect Ratio, L/D, and Takeoff Weight Relationship](image)

Since we are designing solar power UAV, we need a large wing area to generate and store enough power throughout the day. The generated power will come from our solar cells and will be used for twenty-four hours of flight, on solar energy. We started with choosing different aspect ratios since we needed a high aspect ratio. However, aspect ratio will affect the lift to drag
ratio, which at the end, partially affects takeoff weight. So we plotted aspect ratio vs. lift to drag ratio and then lift to drag vs. takeoff weight to figure what our aspect ratio has to be. Even though, high aspect ratio gives a high lift to drag ratio, it decreases the takeoff weight. If the takeoff weight is too low, it will bring up structural issues and affect stability at altitude due to wind. An aspect ratio of 29.6 was chosen, and yielded an L/D of approximately 41.2. This decided our takeoff weight of roughly 860 lbs.

When designing the aircraft, it is very important to research other UAV aircrafts that already exist. By doing this, a general idea of properties such as wing loading and thrust to weight ratio could be determined. When looking at the other UAV aircraft, a general range of these properties is constructed, giving a sense on whether or not the specifications of the new aircraft are reasonable. Also, we wanted to see where we stand with our other competitors. Therefore, a plot comparing different UAV, with similar missions, is generated.

![Constraint Diagram](image)

**Figure 2: Constraint Diagram**

Figure 2 shows where our UAV stands versus other competitors. We picked our ground roll to be 300 ft. because we wanted to be able to take-off and land in any condition within minimum distance. The value for $C_{L_{max}}$ was chosen to be 1.4 because, at that high altitude we wouldn’t generate enough lift or drag because of density and the speed we are cruising at. Therefore, picking a really high $C_L$ value would not be any useful, since we could never obtained that. As
you can see from the constraint diagram we have wind loading of around 2 and thrust to weight ratio of around 0.1. Low thrust to weight ratio would help us to achieve low required thrust for steady level flight and that would ultimately lower power required for cruise condition.

**DESIGN CHANGES**

Figures below shows the four different views of our aircraft: top, isometric, front and side, and how the design changed over the duration of this project. There were couple changes that were made in this current design that made it more efficient than previous designs. The wing span was increased to 128.6 ft and the tapered positions were changed. The horizontal stabilizer span was increased to 18 ft to have better elevator deflections. Having the bigger area to put solar cells on helped to generate more power and reach higher altitudes. The height of the vertical stabilizer was increased to 7 ft. Also, the total length of the aircraft was decreased to 35 ft. This helped to reduce the parasite drag.

![Figure 3: Initial Sketch](image-url)
Figure 4: Intermediate Design
FUSELAGE DESIGN

To make the VSP model of Sauron, a previous model of another UAV named Euphoria was modified. This fuselage has a smooth curve and thinner body which helps to reduce drag. Also, the slight upward curve helps to put the horizontal stabilizer above the wing’s vortices. This way there isn’t much of a disturbance from the wing that will affect stabilizer’s performance. The total length of the fuselage now is 35 feet. Reducing a fuselage a little helped to reduce parasite drag. Also, the fuselage mainly has elliptical cross-sectional area. Once we had the fuselage design finalized, the cross sectional areas were calculated at different length to calculate the varying Reynolds’s number. This analysis was critical to gather the skin friction coefficient for each cross section. The goal was to ultimately find the fuselage drag coefficient and the drag force generated from fuselage at cruise altitude.

Figure 2 represents the Reynolds number along the fuselage length at different altitudes. The Reynolds number is in linear relationship with the fuselage location. Also, since Reynolds number is function of length, the Reynolds number increases as the fuselage location increases.
Also, and most importantly, the Reynolds number decreases drastically as the altitude increases. At sea level the Reynolds number at the end of fuselage is about 10 million. Whereas, at 79,000 feet it drops down to only 2.2 million.

**Figure 7. Reynolds Number along the length of the fuselage**

**WING AND TAIL DESIGN**

AVL was used to create wing and tail design. The main constraint for our wing design was the area needed to put enough solar cells to generate required power for cruise conditions and sinusoidal pattern. Therefore, the wing was designed in such a way that it would yield the area of around 550 ft². The airfoil chosen for the wing is SM701. This wing design has the span of about 128 ft and reference chord or 4 ft. The tip chord was chosen to be 2 ft; this gave us the needed aspect ratio of 30 and enough chord length for the Reynolds number to be high enough for the SM701 airfoil. It is double evenly tapered wing. The wing’s shape was based off of Titan Aerospace’s Solora 50 which has similar mission requirements as ours. There is a dihedral angle about 45 ft out from root of the wing. This helps to reduce the effect of natural dihedral angle generated while in cruise at high altitude. Also, the apex of the wing is 11 ft behind the nose of the aircraft.

The tail was also utilized in the same manner as the wing. A different airfoil, Joukovsky was chosen for the horizontal and vertical stabilizer. The horizontal stabilizer is single leading edge tapered and has span of 18 ft. The root chord is 3 ft and the tip chord is 2 ft. This tail configuration is placed 32 ft from the nose of the aircraft. This makes the distance between wing and tail about 20 ft. This distance and the size of the horizontal stabilizer help up to trim the aircraft at lower elevator deflection. The vertical stabilizer is 7 ft tall and has root chord of 3 and tip chord of 2 ft. Detailed analyze for the vertical stabilizer will be performed when the stability derivatives will be calculated and that would decide the size of the vertical stabilizer.

Following figures shows the wing design changes throughout the duration of the project.
Figure 8: Initial wing design with just single leading edge taper

Figure 9: Second wing design with double evenly tapered wing with small winglets
Figure 10: Third wing design change with implementation of horizontal and vertical stabilizers.
Table 2 summarizes the important specs for the wing and tail. It also provides the other characteristics of the aircraft such as different weights, power generated and also the location of neutral point and center of gravity. The span efficiency for this configuration is 1.0. This design yields the lift coefficient of 0.66 in cruise condition and the maximum lift coefficient is 1.4 for the chosen airfoil. We have a static margin of about 20 percent. Even when we use 10 percent static margin, it didn’t bring down the drag and power require by any significant amount. Therefore, we are keeping the static margin to be about 20 percent.

<table>
<thead>
<tr>
<th></th>
<th>Wing</th>
<th>Tail</th>
</tr>
</thead>
<tbody>
<tr>
<td>Airfoil</td>
<td>SM701</td>
<td>Jouk0015</td>
</tr>
<tr>
<td>Span (ft.)</td>
<td>128.6</td>
<td>18.0</td>
</tr>
<tr>
<td>Reference Chord (ft.)</td>
<td>4.0</td>
<td>2.5</td>
</tr>
<tr>
<td>Area (ft.²)</td>
<td>557.5</td>
<td>45.0</td>
</tr>
<tr>
<td>Cruise Cₗ</td>
<td>0.66</td>
<td>0.09</td>
</tr>
<tr>
<td>Power Generated (kW)</td>
<td>16.93</td>
<td></td>
</tr>
<tr>
<td>Aspect Ratio</td>
<td></td>
<td>29.6</td>
</tr>
<tr>
<td>Empty Weight (lbs.)</td>
<td>430.0</td>
<td></td>
</tr>
<tr>
<td>Payload (lbs.)</td>
<td>250.0</td>
<td></td>
</tr>
<tr>
<td>Battery Weight (lbs.)</td>
<td>180.0</td>
<td></td>
</tr>
<tr>
<td>Total Takeoff Weight (lbs.)</td>
<td>860.0</td>
<td></td>
</tr>
<tr>
<td>Span Efficiency</td>
<td></td>
<td>1.01</td>
</tr>
<tr>
<td>Max Cₗ</td>
<td></td>
<td>1.4</td>
</tr>
<tr>
<td>Neutral Point Location (ft.)</td>
<td>13.4</td>
<td></td>
</tr>
<tr>
<td>C.G. Location (ft.)</td>
<td></td>
<td>13.2</td>
</tr>
</tbody>
</table>

This wing and tail configuration yield the best range of elevator deflection needed to trim the aircraft, which is discussed later on in this report. Also, there is more than enough combined area to operate on solar power in cruise condition. This design is feasible because Solora 50 has already found a way to make this work. Solora 50 has wing span of around 164 ft, tail span of 80 ft, total length of 50 ft and have larger dihedral than us. Despite their larger dimensions, the total weight of their aircraft is only 350 lbs and that’s with 70 lbs of payload. While, our aircraft is slightly smaller and we have total weight of around 860 lbs. This gives us higher probability to manufacture stronger structure. Therefore, this is the best suitable design for our mission requirements.
SPAR DESIGN AND WEIGHT BREAK-DOWN

We selected D-Tube spar for our wing design because it met the weight limitation and structural criteria better than any other spar designs. We are using Carbon fiber Prepreg for our spar because of its low density and high strength compared to Aluminum and other metals. This would give us the needed light weight and high strength structure. The proposed material is HexPly M91 which is latest aerospace primary structure epoxy matrix with excellent toughness including very high residual compression strength after impact (CAI). It has outstanding properties when combined with HexTow IM10 continuous carbon fiber 12k which has 95% carbon content, enhanced tensile properties and modulus of 310 GPa. Using these two materials provides with the theoretical calculated laminate density of approximately 1.4 g/cc, modulus of approximately 200 GPa, and theoretical calculated cured laminate thickness of around 0.0072 inches. One of the advantages of using this material is that it has been specially designed for automated processing and is particularly suited for both automated tape laying (ATL) and advanced fiber placement (AFP) methods. Also, it has good tack life and out-of-shelf life, which provides flexibility on the shop floor and has low exothermic behavior that allows simple cures of thick structures up to 2.76 inches. Figure below shows the compression of M91/IM10 material with several other high performance materials. The figure shows the overall performance of M91/IM10 and proves that it has well-rounded properties.

![Figure 12: Outstanding all-rounded Prepreg performance with HexPly M91](image)

Table below shows the overall material properties for selected material. Other epoxy matrix properties such as gel time, cure cycle viscosity profiles, and Prepreg curing conditions are mentioned in appendix.

| HexTow IM10 Carbon Fiber |  |  |
# of Filaments | 12000
---|---
Filament Diameter (microns) | 4.4
Tensile Strength (MPa) | 6964
Tensile Modulus (GPa) | 310
Strain (%) | 2.0
Density (g/cm³) | 1.79

**Epoxy-Fiber (Prepreg) Combination (M91/IM10)**

<table>
<thead>
<tr>
<th>Theoretical Values</th>
</tr>
</thead>
<tbody>
<tr>
<td>Cured Ply Thickness (in)</td>
</tr>
<tr>
<td>Fiber Volume (%)</td>
</tr>
<tr>
<td>Laminate Density (g/cm³)</td>
</tr>
<tr>
<td>Laminate Modulus (GPa)</td>
</tr>
<tr>
<td>Tensile Strength (MPa)</td>
</tr>
</tbody>
</table>

**WING SPAR DESIGN**

To minimize the wing deflection, some of the weight was distributed along the wing span. All of the battery weight was distributed along the wing span starting from 30 ft out from the root of the wing. Also, fifty pounds of payload weight is also distributed along the wing. Then the solar cell weight was distributed along the entire wing as well. The wing, fuselage, horizontal stabilizer and vertical stabilizer skins are made up of 4 plies of M91/IM10, which gives it the total skin thickness of about 0.03 inches. Then wing structure weight was calculated using the carbon fiber density mentioned above and the total volume of the wing skin. This weight was then also distributed along the wing span. Motors were placed 10 ft and 25 ft from the root of the wing. That is why there is a big spike at those locations.

The spar was designed in such a way that the spar height varies as a function of chord length. Therefore, the spar didn’t result in heavy weight structure or overly designed structure. The spar thickness is 0.15 inches and the width is 2.15 inches as shown in the figure above. Which means
that we will require to lay up about 20 plies to get this thickness. It is lots of plies to layout; however, since this process can be automated and is also designed for parts up to 2.76 inch thick, we should have any problem. This combination of height, thickness and width gave us the spar with total weight of 70.38 lbs. At the end of the iteration, spar weight was then distributed to get the complete weight distribution along the wing span which is shown in the figure below.

**Figure below** shows the lift distribution along the wing span with and without the weight distribution. As noticed, the wing has an elliptical lift distribution before distributing the weights along the wing. However, after distributing the motor, wing, spar, battery, payload, and solar cell weight the lift distribution decreases along the wing span. It is noticeable that at the location where the motors are placed, there is a huge deficiency of lift.
After distributing all the weights and finding the lift distribution along the wing, the total loading was calculated by subtracting weight distribution from lift distribution at each section of the wing. This resulting load was then used to calculating total bending moment on the wing. Same theoretical approach was used to find bending moment as the cantilever beam, which has one free end and one constrained end. With our spar design we only have about 2750 ft*lbs of bending moment at the root of the wing. The bending moment along the wing span is showed in the figure below. As moving away from the root of the wing to the tip of the wing bending moment decreases exponentially. As predicted the bending moment at the tip of the wing is zero since it’s a free end and is able to move freely.
Once the bending moment and moment of inertia were calculated, the next step was to figure out the maximum stress generated due to all the moments and loads. Since the stress is greatest at the farthest point from the neutral axis, the location was calculated at different section of the wing. And then the stress was calculated using bending moment, distance from neutral axis and moment of inertia. Figure below shows the stress generated in the spar along the wing. Once again, the stress is highest at the root of the wing. During normal condition, 1g load, the stress at the root is about 50,000 Psi. This is far lower than what this composite structure is capable of withstanding. Even with 4g gust, the stress at the root will be small and wouldn’t cause any damage to the structure. And the structure integrity will remain 100%. Also, stress decreases exponentially as span increases. Once again, same as cantilever beam, the stress at the tip of the wing is zero.
Finally, the main factor that governed the spar design was wing deflection. The goal was not to exceed 6 ft of deflection at 4g gust. So the spar was designed around this constraint. Figure below shows the wing deflection at 1g, 2g, 3g, and 4g gusts. In any of those conditions, the maximum wing deflection achieved is 6 ft which is at 4g gust. Under normal conditions, at 1g gust, the maximum wing deflection is only about 1.5 ft. Since we met all of our criteria in terms of small wing deflection, small moments and low stress; the spar design was successful and is best suitable for this aircraft.
HORIZONTAL STABILIZER SPAR DESIGN

As mentioned earlier, the horizontal stabilizer skin is made up of 4 plies of M91/IM10 carbon Prepreg. This gives it the total thickness of about 0.03 inches. The constraint for the horizontal stabilizer spar design was the allowable deflection. Since the horizontal stabilizer is not as long as the wing and since composite structures are not very ductile; the maximum allowable deflection at 4g gust was determined to be about third of an inch. This is probably very conservative approximation. First, the total horizontal weight was calculated using the surface area and thickness of the horizontal stabilizer and then multiplying it by the density of the material. Then, this weight was distributed along the span. Also, we are using D-Tube for our spar design for horizontal stabilizer. Also, the spar thickness was determined to be 0.035 inches which is 5 plies of M91/IM10 and the width of 1.0 inch. Once again, height of the beam was varied with the chord length. Figure below shows the spar design and related dimensions for horizontal stabilizer.
This combination of height, thickness and width gave us the spar with total weight of 1.87 lbs for horizontal stabilizer. Also, the calculated horizontal stabilizer skin weight is 10.24 lbs. Since, the loading on the horizontal stabilizer is fairly low, it results in much smaller spar design which results in really small spar weight. At the end of the iteration, spar weight was then distributed to get the complete weight distribution along the horizontal stabilizer span which is shown in the figure below.
Figure below shows the lift distribution along the horizontal stabilizer span with and without the weight distribution at two different $C_L$ values: 0.1, and 1.4. We decide to study these three different conditions because depending on the lift coefficient horizontal stabilizer the lift loading on the span changes. Most important thing to notice is that the horizontal stabilizer has an elliptical lift distribution along the span in all three conditions. As shown in the figure on the left, for the lowest value of $C_L$ of 0.1, horizontal stabilizer is producing downward lift and it’s because it has positive elevator deflection. Lastly, for the maximum value of $C_L$ of 1.4, the elevator has shifted from positive to negative deflection, thus generating positive lift on the horizontal stabilizer, which is show in the far most right figure. Since the loading is high for the $C_L$ of 0.1, that would be the driving factor for the spar design of horizontal stabilizer.

After distributing all the weights and finding the lift distribution along the horizontal stabilizer, the total loading was calculated by subtracting weight distribution from lift distribution at each section of the horizontal stabilizer. This resulting load was then used to calculating total bending moment on the horizontal stabilizer. Same theoretical approach was used to find bending moment as the cantilever beam, which has one free end and one constrained end. The moment along the horizontal stabilizer span was plotted for both conditions. Since horizontal stabilizer has negative lift for $C_L = 0.1$, it causes the negative banding moment, which means that the top surface of the spar is in tension. For $C_{L,max} = 1.4$, there is positive lift on the horizontal stabilizer it causes positive bending moment; this means that the top surface of the spar is in compression. With our spar design banding moment at the root varies from -90 ft*lbs to 25 ft*lbs. The bending moment along the wing span is shown in the figures below. As moving away from the root of the wing to the tip of the wing bending moment decreases exponentially. As predicted the bending moment at the tip of the wing is zero since it’s a free end and is able to move freely.
Once the bending moment and moment of inertia were calculated, the next step was to figure out the maximum stress generated due to all the moments and loads. Since the stress is greatest at the farthest point from the neutral axis, the location was calculated at different section of the horizontal stabilizer. And then the stress was calculated using bending moment, distance from neutral axis and moment of inertia. Once again, this was done for two different $C_L$ conditions: $C_L = 0.1$, and $C_{L,\text{max}} = 1.4$. Since horizontal stabilizer has negative bending moment for $C_L = 0.1$, it causes the negative value of stress. For $C_{L,\text{max}} = 1.4$, there is positive bending moment on the horizontal stabilizer therefore it causes positive value of stress. Figures below show the stress generated in the spar along the horizontal stabilizer. Once again, the stress is highest at the root of the wing. During normal condition (1g load) the stress at the root is about -5500 Psi, and 1700 Psi for $C_L = 0.1$, and 1.4 respectively. This is far lower than what this composite structure is capable of withstanding. Even with 4g gust, the stress at the root will be small and wouldn’t cause any damage to the structure. And the structure integrity will remain 100%. Also, stress decreases exponentially as span increases. Once again, same as cantilever beam, the stress at the tip of the wing is zero.
Finally, the main factor that governed the spar design was wing deflection. The goal was not to exceed 0.30 inches of deflection at 4g gust as mentions earlier. So the spar was designed around this constraint. Figures below shows the wing deflection at 1g, 2g, 3g, and 4g gusts for $C_L$ of 0.1, and 1.4. In any of those conditions, the maximum horizontal stabilizer deflection achieved is 0.26 inches which is at 4g gust for $C_L$ of 0.1. Under normal conditions, at 1g gust, the maximum wing deflection is only about -0.06 inches at $C_L$ of 0.1. Since we met all of our criteria in terms of small horizontal stabilizer deflection, small moments and low stress; the spar design for horizontal stabilizer was successful and is best suitable for this aircraft.

VERTICAL STABILIZER SPAR DESIGN

Since the vertical stabilizer doesn’t produce any lift for cruise condition because no rudder deflection is needed, the spar was design for worst case scenario. The spar was designed such that vertical stabilizer can easily withstand 10° of side-wash, which requires about 19° of rudder deflection. For this condition, the vertical stabilizer would be generating maximum lift to maintain the yaw moment to be zero by deflection the rudder. As mentioned earlier, the vertical stabilizer skin is made up of 4 plies of M91/IM10 carbon Prepreg. This gives it the total thickness of about 0.03 inches. First, the total vertical weight was calculated using the surface area and thickness of the horizontal stabilizer and then multiplying it by the density of the material. Then, this weight was distributed along the span. Also, we are using D-Tube for our spar design for horizontal stabilizer. The spar dimensions were determined to be the same as the horizontal stabilizer spar. The same spar design gave us the smallest deflection with really light spar weight.
This combination of height, thickness and width gave us the spar with total weight of 0.71 lbs. At the end of the iteration. Also, the vertical stabilizer skin weight was calculated to be 3.98 lbs. since the loading on the vertical stabilizer is really low, the spar doesn’t need to withstand much loads resulting in lower weight. Spar weight and the vertical stabilizer skin weight was then distributed to get the complete weight distribution along the wing span which is shown in the figure below. The vertical stabilizer is 7 ft tall which is represented by the y-axis and the x-axis show the weight at each station along the span.

Figure below shows the lift distribution along the vertical stabilizer span with and without the weight distribution. It is noticeable that the vertical stabilizer doesn’t exactly have elliptical loading. This is a result of side-wash and the rudder deflection, which changes the lift loading to counteract the yaw moment.
After distributing all the weights and finding the lift distribution along the vertical stabilizer, the total loading was calculated by subtracting weight distribution from lift distribution at each section of the span. This resulting load was then used to calculating total bending moment on the vertical stabilizer. Same theoretical approach was used to find bending moment as the cantilever beam, which has one free end and one constrained end. With our spar design we only have about 90 ft*lbs of bending moment at the root of the vertical stabilizer. The bending moment along the span is showed in the figure below. As moving away from the root of the vertical stabilizer to the tip of the vertical stabilizer, bending moment decreases exponentially. As predicted the bending moment at the tip of the vertical stabilizer is zero since it’s a free end and is able to move freely.
Once the bending moment and moment of inertia were calculated, the next step was to figure out the maximum stress generated due to all the moments and loads. Since the stress is greatest at the farthest point from the neutral axis, the location was calculated at different section of the wing. And then the stress was calculated using bending moment, distance from neutral axis and moment of inertia. **Figure below** shows the stress generated in the spar along the vertical stabilizer. Once again, the stress is highest at the root of the vertical stabilizer. During normal condition, 1g load, the stress at the root is about 5500 Psi. This is far lower than what this composite structure is capable of withstanding. Even with 4g gust, the stress at the root will be small and wouldn’t cause any damage to the structure. And the structure integrity will remain 100%. Also, stress decreases exponentially as span increases. Once again, same as cantilever beam, the stress at the tip of the wing is zero.
Finally, the main factor that governed the spar design was that the vertical stabilizer should be easily able to withstand 10° of side-wash, which requires about 19° of rudder deflection. Figure below shows the vertical stabilizer deflection at 1g, 2g, 3g, and 4g gusts. In any of those conditions, the maximum vertical stabilizer deflection achieved is 0.011 inches, which is at 4g gust. Under normal conditions, at 1g gust, the maximum vertical stabilizer deflection is only about 0.0025 inches. Once again, small deflection is the result of having really small loading on the vertical stabilizer. Since we met all of our criteria in terms of small deflection, small moments and low stress; the spar design was successful and is best suitable for this aircraft.
The weights of different components were then calculated using selected material. As mentioned earlier the thickness for wing, fuselage, horizontal and vertical stabilizer skins was chosen to be 4 plies of M91/IM10 carbon Prepreg. Which turns out to be about 0.03 inches after cure. Then using the density of the material the weights were calculated. The empty weight ratio was selected to be 0.5 from design requirements which gave us the empty weight of 430 lbs. After the analysis, the total empty weight of our aircraft come out to be 404.44 lbs. This means that we managed to stay under the allowable empty weight. Also, the battery weight is 180 lbs. and the payload of 250 lbs. This gives us the total weight of 834.44. This is great since we are within our maximum takeoff weight which is 860 lbs.

<table>
<thead>
<tr>
<th>Component</th>
<th>Weight (lbs)</th>
</tr>
</thead>
<tbody>
<tr>
<td>Empty Weight</td>
<td></td>
</tr>
<tr>
<td>Wing</td>
<td>126.89</td>
</tr>
<tr>
<td>Fuselage</td>
<td>32.77</td>
</tr>
<tr>
<td>Horizontal Stabilizer</td>
<td>10.24</td>
</tr>
<tr>
<td>Vertical Stabilizer</td>
<td>3.98</td>
</tr>
<tr>
<td>Solar Cell</td>
<td>87.53</td>
</tr>
<tr>
<td>Wing Spar</td>
<td>70.38</td>
</tr>
<tr>
<td>Item</td>
<td>Weight</td>
</tr>
<tr>
<td>---------------------------</td>
<td>--------</td>
</tr>
<tr>
<td>Horizontal Stabilizer Spar</td>
<td>1.87</td>
</tr>
<tr>
<td>Vertical Stabilizer Spar</td>
<td>0.71</td>
</tr>
<tr>
<td>4 Motors</td>
<td>16.00</td>
</tr>
<tr>
<td>Fuselage Formers</td>
<td>15.00</td>
</tr>
<tr>
<td>Gear System</td>
<td>40.00</td>
</tr>
<tr>
<td>Total Empty Weight</td>
<td>404.44</td>
</tr>
<tr>
<td>Battery</td>
<td>180.00</td>
</tr>
<tr>
<td>Payload</td>
<td>250.00</td>
</tr>
<tr>
<td>Total</td>
<td>834.44</td>
</tr>
</tbody>
</table>

**INBOARD SCHEMATICS**

The remaining payload of 200 lbs. is located in the 2x2 cross section of the UAV and is 10 feet in length total volume of approximately 40 ft\(^3\). The 10 feet of length is comprised of a 5 ft. section where the wing structure will be located. Also in the 5 foot section the front camera will be located underneath the fuselage as is seen in figure below. The remaining 5 feet will contain all of the onboard electronics such as radar equipment, avionics, batteries, etc. A second camera will also be placed in the remaining 5 feet area. This camera is used for a more localized view while the front camera has a broader view for surveillance. The figure below shows an example of how the payload would fit and a schematic of the payload fuselage area with dimensions. The payload volume is unusually large but it is designed to handle extra components depending on customer requests and as long as the overall weight stays under 200 lbs.
CONTROL SURFACE SIZING

Table 5 shows the elevator deflection required to trim the aircraft for given lift coefficient. The goal was to have minimum deflection at cruise $C_L$ that way we reduce the drag generated by having higher horizontal tail deflection. For our design, we have the horizontal stabilizer deflection in the range from -2.5 to 1.6 degrees for lowest to highest $C_L$ range. If the maximum elevator deflection is assumed to be 20 degrees, this gives us higher range of elevator deflection that can be used to perform maneuvers.
Table 3. Elevator deflection required to trim the aircraft at different lift coefficients

<table>
<thead>
<tr>
<th>Lift Coefficient, $C_L$</th>
<th>Aileron Deflection (Degrees)</th>
<th>Elevator Deflection (Degrees)</th>
<th>Rudder Deflection (Degrees)</th>
</tr>
</thead>
<tbody>
<tr>
<td>0.1</td>
<td>0.00</td>
<td>1.55</td>
<td>0.00</td>
</tr>
<tr>
<td>0.4</td>
<td>0.00</td>
<td>0.90</td>
<td>0.00</td>
</tr>
<tr>
<td>0.66</td>
<td>0.00</td>
<td>0.25</td>
<td>0.00</td>
</tr>
<tr>
<td>1.0</td>
<td>0.00</td>
<td>-0.74</td>
<td>0.00</td>
</tr>
<tr>
<td>1.4</td>
<td>0.00</td>
<td>-2.14</td>
<td>0.00</td>
</tr>
</tbody>
</table>

Aileron Sizing

Elevator Sizing

Rudder Sizing
Table below shows the stability derivatives obtained from the AVL for our $C_{L,cruise}$ of 0.66. The configuration of all the combined control surfaces gives us the best performance with minimum deflections needed in trimmed flight. Also, our aircraft is spirally stable.

<table>
<thead>
<tr>
<th></th>
<th>Alpha, $\alpha$</th>
<th>Beta, $\beta$</th>
</tr>
</thead>
<tbody>
<tr>
<td>$z'$ force $C_L$</td>
<td>$C_{L\alpha}$</td>
<td>$C_{L\beta}$</td>
</tr>
<tr>
<td>$y$ force $C_Y$</td>
<td>$C_{Y\alpha}$</td>
<td>$C_{Y\beta}$</td>
</tr>
<tr>
<td>$x'$ mom. $C_t'$</td>
<td>$C_{t\alpha}$</td>
<td>$C_{t\beta}$</td>
</tr>
<tr>
<td>$y$ mom. $C_m$</td>
<td>$C_{m\alpha}$</td>
<td>$C_{m\beta}$</td>
</tr>
<tr>
<td>$z'$ mom. $C_n'$</td>
<td>$C_{n\alpha}$</td>
<td>$C_{n\beta}$</td>
</tr>
</tbody>
</table>
Figures below show the lift distribution along the span of the wing and horizontal stabilizer for cruise condition and max lift coefficient value in trimmed flight. As mentioned in the table above, we need about 0.25° elevator deflection for trimmed flight during cruise condition. The positive elevator deflection means downward lift from horizontal stabilizer. Figure # (first) shows very well presentation of this situation. At our highest lift coefficient of 1.4, we need -2.14° of elevator deflection to trim the flight for zero pitching moment. Negative elevator deflection means upward lift. This is shown in the figure # (second).
Wing – Airfoil and Drag Analysis
Given a of 0.66 several airfoils were compared with the goal to minimize the drag coefficient. XFOIL was utilized to generate polar files which were then used to create graphs illustrating each airfoil’s drag bucket, from a Reynolds number of 200,000 to 5 million, shown in Figure 5 on next page is the SM701. The Somers/Maughmer’s airfoil was chosen because of its relatively low drag coefficients, compared to the other tested airfoils. Table 3 illustrates the and of the airfoil. Given our cruise coefficient of lift, the wing’s airfoil falls into the drag bucket at our cruising Reynolds numbers. Originally a of 0.83 was chosen and then modified to improve the overall performance of the aircraft. The decrease in the coefficient is beneficial for the wing’s SM701 such that we are in the drag bucket at higher Reynolds Numbers. Going forward, more airfoils will be examined and emphasis will be put on increasing Oswald’s efficiency across the wing.

Table 4. SM701 coefficients at operational floor and ceiling

<table>
<thead>
<tr>
<th></th>
<th>SM701 45,000 ft</th>
<th>SM701 61,000 ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>C_l cruise</td>
<td>0.66</td>
<td>0.66</td>
</tr>
<tr>
<td>C_d cruise</td>
<td>0.0082</td>
<td>0.0098</td>
</tr>
<tr>
<td>C_l max</td>
<td>1.4</td>
<td>1.4</td>
</tr>
<tr>
<td>C_d max</td>
<td>0.0195</td>
<td>0.0242</td>
</tr>
</tbody>
</table>
Tail – Airfoil and Drag Analysis

Figure 6 shows the drag bucket for the NACA 63-015A airfoil. Originally, the NACA 63-015A airfoil was chosen and later replaced by the Joukovsky 0015 airfoil. The original NACA airfoil was chosen primarily based upon the thickness near 50% of the chord, with emphasis on choosing a laminar symmetric airfoil. Upon the selection, our chosen flight conditions were ignored and the drag bucket for the NACA 63-015A illustrates that neither the lift nor drag coefficients were examined prior to the selection.
Figure 14. NACA 63-015A Drag Bucket Analysis

Table 4 on the next page illustrates these coefficients and serves as a comparison between the NACA and Joukovsky airfoils prior to improving our previous design by decreasing to 0.09.

Table 5. Airfoil Analysis of NACA 63-015A and Joukovsky 0015

<table>
<thead>
<tr>
<th></th>
<th>NACA 63-015A</th>
<th>Joukovsky 0015</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>45,000 ft</td>
<td>61000 ft</td>
</tr>
<tr>
<td>$C_l$ cruise</td>
<td>0.15</td>
<td>0.15</td>
</tr>
<tr>
<td>$C_d$ cruise</td>
<td>0.0073</td>
<td>0.0086</td>
</tr>
<tr>
<td>$C_l$ max</td>
<td>1.1</td>
<td>1.11</td>
</tr>
<tr>
<td>$C_d$ max</td>
<td>0.032</td>
<td>0.04</td>
</tr>
</tbody>
</table>
After examining the flight conditions, both airfoils performed similar at the cruise value. The yieldd roughly a 8.75% increase and a decrease of roughly 105% for the value corresponding to the NACA’s max coefficient of lift. Due to the Joukovsky 0015 airfoil, shown below, performing better at higher coefficients of lift and having a larger max coefficient of lift, the Joukovsky 0015 airfoil was chosen. This airfoil also behaves better at lower Reynolds number than NACA 63-015A. Analyzing the drag bucket below, the airfoil performs the best around a lift coefficient between 0.1 and 0.2. The Joukovsky performs better for a wider range of Reynolds numbers. Due to the aforementioned, with emphasis on a lower drag coefficient with larger lift coefficients, the Joukovsky 0015 airfoil will be used.

As mentioned above, to further optimize the aircraft, the tail’s cruise lift coefficient was decreased to 0.09. **TABLE 5 BELOW** shows the new values relating to .

**Table 5. Airfoil Analysis of NACA 63-015A and Joukovsky 0015**

<table>
<thead>
<tr>
<th></th>
<th>NACA 63-015A</th>
<th>Joukovsky 0015</th>
</tr>
</thead>
<tbody>
<tr>
<td>45,000 ft</td>
<td>61000 ft</td>
<td>45,000 ft</td>
</tr>
</tbody>
</table>
### DRAG AND POWER REQUIRED IN TRIMMED FLIGHT

**Sea level**

Figure 8 shows the drag build up and the power required to cruise at sea level. The blue vertical line mark the stall speed and the pink vertical line on the right marks the maximum speed that can be achieved before we run out of power at sea level. The stall speed at sea level is 37 ft/s and the maximum speed limit is 113 ft/s. we are cruising at 44.4 ft/s which is 1.2 times the stall speed as per standards and that seems to be the speed required for minimum drag at sea level.

However, it's not the speed required for minimum power, which is about 46 ft/s. Therefore, our total drag at sea level in trimmed flight is 18.4 lbs. We will require about 1.05 kW of power to cruise at sea level for our cruising speed. Also, the Oswald efficiency at this altitude is 0.76 and the drag coefficient $C_{D_0}$ is 0.0087. This produced maximum lift to drag ratio of 46.7.
Figure 16. Drag build up and power required to cruise at sea level

**45,000 ft. Altitude**

Figure 9 shows the drag build up and the power required to cruise at 45000 feet. The blue vertical line marks the stall speed and the pink vertical line on the right marks the maximum speed that was obtained from power required calculations at this altitude. The stall speed at this altitude is 83.9 ft/s and the maximum speed limit is 191.5 ft/s. We are cruising at 100.7 ft/s which is 1.2 times the stall speed. However, the speed required for minimum drag is about 91.7 ft/s and the speed required for minimum power is about 80 ft/s at this altitude. Therefore, our total drag at this altitude in trimmed flight is 20.3 lbs. We will require about 2.7 kW of power to cruise at this altitude. Also, the Oswald efficiency at this altitude is 0.73 and the drag coefficient $C_{D_0}$ is 0.010. This produced maximum lift to drag ratio of 42.4 at this altitude.
Figure 17. Drag build up and power required to cruise at 45,000 ft

61,000 ft Altitude

Figure 10 shows the drag build up and the power required to cruise at 61000 feet. The blue vertical line mark the stall speed and the pink vertical line on the right marks the maximum speed that can be achieved before we run out of power at this altitude. The stall speed at this altitude is 122.3 ft/s and the maximum speed limit is 245.3 ft/s. We are cruising at about 146.8 ft/s which is 1.2 times the stall speed as per standards, but it is not the speed required for minimum drag at this altitude. The speed for minimum drag is about 125 ft/s and speed for minimum power is about 119 ft/s. Therefore, our total drag at sea level in trimmed flight is 22.5 lbs. We will require about 4.3 kW of power to cruise at this altitude. Also, the Oswald efficiency
at this altitude is 0.69 and the drag coefficient $C_{D_o}$ is 0.0105. This produced maximum lift to drag ratio of 48.2 at this altitude.

![Drag Force vs. Velocity Graph]

**Figure 18. Drag build up and power required to cruise at 61,000 ft**

**79,000 ft Altitude**

Figure 11 shows the drag build up and the power required to cruise at 79,000 feet. Since it’s out maximum operational ceiling, we can also cruise at this altitude if mission required us to do so. The blue vertical line mark the stall speed and the pink vertical line on the right marks the maximum speed obtained from power required plot at 79,000 feet. The stall speed at this altitude is 188.7 ft/s and the maximum speed limit is 294 ft/s. we are cruising at about 226.5 ft/s which is 1.2 times the stall speed as per standards. However, the speed required for minimum drag is approximately 190 ft/s. Also, the speed required for minimum power flight is around 184 ft/s. Therefore, our total drag at this altitude in trimmed flight is 26.9 lbs. we will require about 8.1
kW of power to cruise at this altitude. Also, the Oswald efficiency at this altitude is 0.63 and the drag coefficient $C_D$ is 0.0125. This produced maximum lift to drag ratio of 31.9 at this altitude.

Figure 19. Drag build up and power required to cruise at 79,000 ft

Figure below provides well representation of total drag at different altitudes as well as power required and maximum speed that can be achieved at each altitude. It compares the total drag produced in trimmed flight at four different altitudes and how it varies with speed and altitude. The black line indicates the cruising speed at each of those altitudes which then points to the total drag produced for that condition. The power plots shows the maximum available power which is shown by red line as well as the power required for cruise for each altitudes. The point where power required crosses the maximum power available line marks the maximum speed that can be achieved at that altitude. Once again, the black lines shows marks the cruise speeds which then points to the power required for cruise at that particular altitude.
Table 6 provides an overview of the cruise conditions which have been calculated using the drag analyses discussed in this section.
Table 6. Summary of cruise conditions are varying altitudes

<table>
<thead>
<tr>
<th></th>
<th>Sea Level</th>
<th>45,000 feet</th>
<th>61,000 feet</th>
<th>79,000 feet</th>
</tr>
</thead>
<tbody>
<tr>
<td>Stall Speed (ft/s)</td>
<td>37.0</td>
<td>83.9</td>
<td>122.3</td>
<td>188.7</td>
</tr>
<tr>
<td>Cruise Speed (ft/s)</td>
<td>44.4</td>
<td>100.7</td>
<td>146.8</td>
<td>226.5</td>
</tr>
<tr>
<td>Max Speed (ft/s)</td>
<td>113.0</td>
<td>191.5</td>
<td>245.3</td>
<td>294.0</td>
</tr>
<tr>
<td>Total Drag (lbs)</td>
<td>18.4</td>
<td>20.3</td>
<td>22.5</td>
<td>26.9</td>
</tr>
<tr>
<td>Power Required (kW)</td>
<td>1.05</td>
<td>2.7</td>
<td>4.3</td>
<td>8.1</td>
</tr>
<tr>
<td>Reynolds’ Number</td>
<td>1,129,663.40</td>
<td>626,856.80</td>
<td>429,692.6</td>
<td>274,504.6</td>
</tr>
<tr>
<td>$C_{D_0}$</td>
<td>0.0087</td>
<td>0.01</td>
<td>0.0105</td>
<td>0.0125</td>
</tr>
<tr>
<td>Oswald’s Efficiency</td>
<td>0.76</td>
<td>0.73</td>
<td>0.69</td>
<td>0.63</td>
</tr>
<tr>
<td>Max L/D</td>
<td>46.7</td>
<td>42.4</td>
<td>38.2</td>
<td>31.9</td>
</tr>
</tbody>
</table>

FLIGHT ENVELOPE

Absolute and Operational Ceiling

To determine the absolute ceiling and the maximum operational ceiling altitude versus velocity plot was generated. Two different velocities were taken into account. One was stall speed and another was the maximum speed that can be achieved using available power at that altitude. In Figure 12, stall speed is marked by blue line and the power dependent maximum speed is marked
by green line. The stall speed increases exponentially as the altitude increases. As does the maximum speed, which starts to gradually decrease as altitude increases. The intersection of both line gives the absolute ceiling for the aircraft. In our case, the absolute ceiling is 94,000 feet. It is the altitude where power required is equal to power available and we cannot go above this ceiling.

![Graph of h vs. v](image)

**Figure 20. h vs. v graph to determine absolute ceiling**

**Climb rate**

Once the absolute ceiling was found, the climb rate versus altitude was plotted to figure out the maximum operational ceiling for our aircraft. At sea level climb rate is about 633 ft/min. As seen from Figure 13, there is a linear relationship between altitude and climb rate. As altitude increase, climb rate decreases. The maximum operational ceiling is the altitude at which the climb rate is 100 ft/min. In our case at 79,000 feet the climb rate is 100 ft/min thus that is our maximum operation ceiling. It is well above our initial operational ceiling which is 61,000 ft. Therefore, if the mission requires, we can go well above our intended ceiling.
Once we found out our absolute and maximum operational ceiling, the next important step was to figure out the time it will take the aircraft to reach to intended altitude. Therefore, altitude versus time to climb plot was generated. Figure 14 shows the exponential relationship between altitude and time to climb. As the altitude increases, the time to climb also increase. From the plot, to reach to 61,000 ft altitude, our aircraft takes about 2.6 hours and to reach to maximum operational ceiling it takes approximately 4.6 hours. Since we are doing sinusoidal pattern it is important for us to climb back to 61,000 feet or 79,000 feet as fast as possible from 45,000 feet. In our case, to climb from 45,000 feet to 61,000 feet it will take the aircraft only about 50 minutes. And to climb from 45,000 feet to 79,000 feet it will take about 2.8 hours.
MANEUVERS

An important factor in designing an aircraft are the limiting load factors. There exists Federal Aviation Regulations (FARS, 14 CFR) that designate what the positive and negative limiting load factors are. According to the regulation, the positive limit maneuvering load factor may not be less than 4.31 and the negative must be greater than -1.72 for our aircraft, using the regulation for normal category aircrafts. Figure 15 shows the plots that are used to find our load factor. The plots on the left correspond to steady maneuvers and on the right instantaneous maneuvers. The associated values can be seen Table 7. Though our max load factor is 3.32, which is less than the designated 4.31, part c of the FAR dictates that “Maneuvering load factors lower than those specified in this section may be used if the airplane has design features that make it impossible to exceed these values in flight.” Going forward, features will be looked in to either preventing our aircraft from exceeding these values. However, this doesn’t affect our aircraft since we are unmanned aircraft and these regulations apply to manned flight. At 45 and 61 thousand feet our flight envelope sees noticeable change as the shape is comparable to other manned aircraft. At 45,000 ft. altitude, our positive maximum limit load is reached at 145 ft/s and our negative maximum limit load is achieved at 92 ft/s. At 61,000 feet the velocity range of our aircraft’s maximum limit load in both directions constricted again. At this altitude our aircraft reaches its maximum positive and negative limit loads at 207 ft/s and 131 ft/s respectively.
Figure 23. Plots used in determining the load factor
Table 7. Summary of Maneuvers at operating altitudes

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Sea Level</th>
<th>45,000 ft</th>
<th>61,000 ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>Max Load Factor</td>
<td>3.32</td>
<td>3.21</td>
<td>3.16</td>
</tr>
<tr>
<td>Bank Angle (deg)</td>
<td>72.4</td>
<td>71.8</td>
<td>71.5</td>
</tr>
<tr>
<td>Turn Radius (ft)</td>
<td>26.5</td>
<td>137.7</td>
<td>300.5</td>
</tr>
<tr>
<td>Turn Rate (deg/sec)</td>
<td>112.3</td>
<td>48.3</td>
<td>32.5</td>
</tr>
<tr>
<td>Pull Up Radius (ft)</td>
<td>41.2</td>
<td>215.2</td>
<td>460.6</td>
</tr>
<tr>
<td>Pull Up Rate (deg/sec)</td>
<td>77.2</td>
<td>32.9</td>
<td>22.3</td>
</tr>
<tr>
<td>Pull Down Radius (ft)</td>
<td>22.13</td>
<td>112.9</td>
<td>239.3</td>
</tr>
<tr>
<td>Pull Down Rate (deg/sec)</td>
<td>143.66</td>
<td>62.75</td>
<td>22.9</td>
</tr>
</tbody>
</table>

At sea level our flight envelope (Figure 16) follows the same format as manned aircrafts. For this analysis we will be assuming our UAV fits into the utility aircraft category stipulated by FAR regulations with a positive limit load of 4.4 and a negative limit load of -1.76. Our aircraft reaches these points at 64 and 40 ft/s respectively. At sea level we performed a gust analysis with ± 25, ± 50 and ± 65 ft/s gusts. As seen in the figure below, our aircraft can comfortably operate under ± 25 and 50 ft/s gusts. At ± 65 ft/s the load exceeds our aircrafts capabilities due to its low wing loading. This is not a major concern for us as these regulations are designed for manned aircraft. This will, at worst, limit the number of runways our aircraft can use among other restrictions. We can try to make our aircraft operate successfully under the missed condition if we preform structural analysis on the wing to determine a best fix should we need it.

![Figure 24. Load Factor vs. Velocity at Sea Level for Instantaneous Maneuvers](image)

**Figure 24. Load Factor vs. Velocity at Sea Level for Instantaneous Maneuvers**
When analyzing the relationship between our aircraft’s velocity and bank radius for a steady state maneuver, the plots show an exponential relationship between the two as expected. These relationships can be seen in Figure 17 where we plotted the relationship between turn radius and velocity at the maximum achievable limit load of around 3.2 for all operating altitudes. At sea level our aircraft reaches the maximum turn radius of about 500 feet at 100 ft/s. At 45 and 61 thousand feet the relationship is maintained thought the turn radii increase dramatically to 2,500 and 3,300 feet respectively. These values however do not concern us as our mission plan does not depend on how sharply our aircraft can turn. At usual operating conditions our aircraft will not exceed a turn radius of 350 feet at cruise velocity.

Figure 25. Steady state bank radius plots at operational altitudes
Continuing our analysis on a steady state turn we explored the relationship between our aircraft’s turn rate and velocity. In Figure 18 you can see the expected relationship between our aircrafts steady state turn rates and velocities with a bank angle determined by the aircraft’s maximum limit load. Also, the reason for the turn rate for our aircraft being so high is that we have really low wing loading and also high turn radius. The maximum turn rates our aircraft will experience at sea level, 45,000 and 61,000 feet are 112, 48 and 32 degrees/second respectively. Like with our turn radius, the turn rate of our aircraft is not affected by our mission constraints.

![Steady State Statevel Turn Rate](image1)

![Steady State 45k Turn Rate](image2)

![Steady State 61k Turn Rate](image3)

*Figure 26. Steady state turn rate plots at operational altitudes*
OPERATIONAL RANGE

One requirement of the UAV being designed is that it must be high endurance, meaning it must be in flight for five to seven days. To enable Sauron to do this, solar cells must be efficient in providing enough power for the aircraft to maintain enough power to fly through the night with no sunlight. Due to this restriction, the UAV can only go so far North or South before there is not enough solar power being generated.

Range was calculated using the following equation:

\[ \frac{P}{\eta_{sc} S_{sc}} \times \sin(90 - \theta) \]

In this equation, \( \eta_{sc} \) is the solar panel efficiency, \( S_{sc} \) is the wing area, and \( \theta \) is 90 minus angle of latitude. \( P_{sol} \) is a numerical value that changes based on elevation and angle of latitude. Calculations showed that when flying at an elevation of 79,000 feet, the maximum altitude of Sauron’s flight envelope, the aircraft had a range of about 77° North and South latitude. At 79,000 feet, the aircraft requires 8.4 kW to fly and within these latitudes, the aircraft receives enough solar power to maintain flight throughout the night. At 61,000 feet, the altitude of Sauron’s daily flight elevation, 4.6 kW of power is required. Calculations show that within about 76° North and South latitude, the aircraft gets enough solar power to fly throughout the night. At 45,000 feet, the UAV has a range of about 68° North and South latitude. In this range, enough solar power is generated to meet the necessary 3 kW needed for flight.

Although sunlight is not as strong of a power source at lower elevations, the power needed to operate the aircraft is much lower at a lower elevation. Because higher elevations give a less direct angle of sunlight to the solar panels, power generated from the sunlight is not as high as it is at lower elevations. This causes the range to decrease at higher elevations. The figure below shows latitude range lines of the UAV at each elevation.

Figure 27. Map of operational limitations
PROPULSION SYSTEM

The UAV is being powered by solar energy so an electrical motor was needed to be found. The proposed motor is JM1S from Joby Motors. It has a max RPM of 9000 and a continuous torque at 13 Nm. The continuous shaft required power is 8.2 kW, which allows us to operate about 4 of these motors on our UAV with the amount of power being created by the solar cells.

![JM1S Electric Joby Motor](image1.jpg)

**Figure 28. JM1S Electric Joby Motor**

We are technically generating about 16.9 kW of power from the solar cells which allows us to operate 2 motors at full capacity at a given time. We are planning on operating each of them at lower required power. Having four motors gives us an ability to turn off one of the motor on the opposite side of the wing if there is a motor failure and still be able to generate enough thrust to fly without worrying about stability.

The propeller for this motor is a variable pitch customized propeller. This propeller allows us to take off and land in shorter distances as well as make maneuvers at steeper angles. These propellers are custom made, therefore we can have these propeller sized to our requirements.

![Sample variable pitch propulsion system from Northwest UAV](image2.png)

**Figure 29. Sample variable pitch propulsion system from Northwest UAV**
LANDING GEAR

The landing gear chosen is from Aero Telemetry. They provide custom built landing gear for UAV’s that are created from steel alloy and capable of supporting 3000 lbs. Since Our UAV is well under that weight load this company’s model will be more than enough for our design.

![Sample Aero Telemetry landing gear](image)

Figure 30. Sample Aero Telemetry landing gear

The proposed landing gear will be in a reverse tricycle setup. The front two wheels are located one foot behind the apex of the wing which is located at 11 feet behind the nose. Therefore, the front two wheels are located 12 feet behind the nose. Our center of gravity is located at 13.2 feet behind the nose; therefore, there would not be any critical nose-down moment at landing. Two landing gears under the wing are 25 feet apart from the center of the fuse on each side. This means that we will require runway that is at least 50 ft wide. However, all of our targeted airports are wider than that with smallest of them being 180 ft wide. The third landing gear will just be a wheel integrated in the fuselage therefore eliminating the extra weight and drag. It is 22 feet behind the front two landing gears. This landing gear combination gives us 10°-12° of angle of attack which is required for landing and taking-off without damaging the tail. As of now the landing will be retractable with its hydraulic actuators. The landing gear will fold in to the wings and fuse reducing drag. The landing gear is also light, customizable ones will weight approximately 40 lbs together.
TAKEOFF AND LANDING

Takeoff

It has yet to be finalized whether or not Sauron will takeoff under its own power, or using a field launcher. Engineered Arresting systems (ESCO) have developed field launchers that are capable of launching UAV’s that weigh up to 1,225 pounds and launch velocities up to 80 knots. These launchers are moved via pickup trucks or trailers. UAV’s that currently employ this launch system include the Aquila, Altair, Sky Eye, Killer Bee, Storm, and Fury. This system would allow operators to launch the aircraft from shorter runways in a much shorter time. However, if Sauron were to launch under its own power, the following were calculated: takeoff velocity \(V_{lof}\), ground roll \(D_{roll}\), airborne distance to clear a 35 and 50 ft. object \(d_{ab}\), takeoff angle \(\gamma\), and trust required.

The takeoff distance was calculated using the maximum available thrust and maximum climb angle that can be achieved. The analysis was done for four different airports and for two different height clearances. Using the maximum climb angle and the radius the airborne distance was calculated and then added to the ground roll distance to find the total takeoff distance to clear certain height clearance. Table 8 outlines the takeoff parameters for the Sauron at varying geographical locations; as can been seen from the table, takeoff distance increases as altitude increases, as expected.

Table 8. Summary of takeoff parameters at different altitudes

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Sea Level</th>
<th>Edwards AF Base (California) – 2,302 ft</th>
<th>Buckley AF Base (Colorado) – 5,662 ft</th>
<th>Bagram Airfield (Afghanistan) – 4,895 ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>(V_{lof}) [ft/s]</td>
<td>44</td>
<td>46</td>
<td>48</td>
<td>48</td>
</tr>
</tbody>
</table>
Landing

Similar to the takeoff analysis above, the landing parameters were calculated using the baseline 35 and 50 ft. object clearances at varying altitudes. This time, however, an analysis was completed for Sauron landing under power and under glide in the event the conditions call for a glide landing (i.e., complete motor failure). The following parameters were calculated for landing: approach velocity \( (V_a) \), approach angle \( (\gamma_a) \), radius of landing pattern \( (R) \), flare height \( (h_f) \), flare velocity \( (V_f) \), distance to flare \( (d_a) \), distance from flare to touchdown \( (d_f) \), touchdown velocity \( (V_{TD}) \), ground roll after touchdown \( (S_{roll}) \) and thrust \( (T) \). From Table 9, it is shown that the distances required for landing increase with altitude, similar to the takeoff analysis results. As can be noted at the bottom of the table, the values of thrust are negative, this indicates the requirement of reverse thrust to be able to stop our aircraft. Sauron will likely use this in lieu of a braking system on the landing gear system; further analysis is ongoing at this time.

![Figure 32. Takeoff configuration schematic](image)

Table 9. Summary of landing parameters under different conditions at different altitudes

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Sea Level</th>
<th>Edwards AF Base (California) – 2,302 ft</th>
<th>Buckley AF Base (Colorado) – 5,662 ft</th>
<th>Bagram Airfield (Afghanistan) – 4,895 ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>( V_a [\text{ft/s}] )</td>
<td>44</td>
<td>46</td>
<td>48</td>
<td>48</td>
</tr>
<tr>
<td>( R [\text{ft}] )</td>
<td>322</td>
<td>345</td>
<td>382</td>
<td>373</td>
</tr>
<tr>
<td>( V_{flare} [\text{ft/s}] )</td>
<td>46</td>
<td>47</td>
<td>50</td>
<td>49</td>
</tr>
<tr>
<td>( V_{TD} [\text{ft/s}] )</td>
<td>41</td>
<td>42</td>
<td>44</td>
<td>44</td>
</tr>
<tr>
<td>( T_{avail} [\text{lbs}] )</td>
<td>223</td>
<td>215</td>
<td>205</td>
<td>207</td>
</tr>
</tbody>
</table>

<table>
<thead>
<tr>
<th>Parameter</th>
<th>Sea Level</th>
<th>Edwards AF Base (California) – 2,302 ft</th>
<th>Buckley AF Base (Colorado) – 5,662 ft</th>
<th>Bagram Airfield (Afghanistan) – 4,895 ft</th>
</tr>
</thead>
<tbody>
<tr>
<td>( \gamma_a [\text{deg}] )</td>
<td>13.70</td>
<td>13.18</td>
<td>12.45</td>
<td>12.62</td>
</tr>
<tr>
<td>( h_f [\text{ft}] )</td>
<td>9</td>
<td>9</td>
<td>9</td>
<td>9</td>
</tr>
</tbody>
</table>
Please note that all of the calculations in this section are assuming a non-variable pitch propulsion system is being used. A variable pitch system would allow for shorter takeoff and landing distances, and steeper climb and decent angles, as described in the propulsion system section above.

**FINAL THOUGHTS**

This concludes another phase in the conceptual design of the Sauron HALE. Our team has been working diligently towards completing our mission of designing a HALE UAV using alternative fuel sources to support homeland security efforts with a concentration in long term border security. In this report, we discussed the new and improved wing and tail designs that have taken place; a new drag and power in trimmed flight analysis; established a flight envelope; conducted an in-depth maneuver analysis; analyzed our operational range due to solar power constraints; began discussing the propulsion system we will be using; began solidifying our landing gear configuration and possible manufacturer; and finally, an in-depth takeoff and
landing analysis. The next phase of design changes and requirements will be coming in the near future.
APPENDIX

HexPly M91 Epoxy Matrix Characteristics

Gel Time

Cure Cycle Viscosity Profiles

*Typical Autoclave Cure – Monolithic Part < 15mm (0.6inch) thick*
Prepreg Curing Conditions

1. Apply full vacuum (1 bar).
2. Apply 7 bar gauge autoclave pressure.
3. Reduce vacuum to a safety value of 0.2 bar when the autoclave pressure reaches ~ 1 bar gauge.
4. Set heat-up rate from room temperature to 180°C ± 5°C (356°F ± 9°F) to achieve an actual component heat-up rate between 1.2°C/minute (2.4°F/minute).
5. Hold at 180°C ± 5°C (356°F ± 9°F) for 120 minutes ± 5 minutes.
6. Cool component at an actual cool down rate of 2-3°C/minute (4-9°F/minute).
7. Vent autoclave pressure when the component reaches 60°C (140°F) or below.
Bagging Configuration for curing Mechanical Laminates

Bagging Configuration for curing Components
Codes:

UAV Characteristics, Drag in Trimmed Flight, and Flight Envelopes

%
% Drag Computation
%

close all
clear all
clc

global atmosphere aircraft

%==========================================================================
%%%                   ATMOSPHERIC PARAMETERS
%==========================================================================

%%%      SEA LEVEL
atmosphere.rho=23.77e-4; % slug/ft^3
atmosphere.mu=3.737e-7; % lbs-s/ft^2
atmosphere.g=32.174; % ft/s^2;
Vmax = 113.5;
lim = 100;
lim1 = 120;

%%%      5662 ft
% atmosphere.rho=20.1e-4; % slug/ft^3
% atmosphere.mu=3.623e-7; % lbs-s/ft^2
% atmosphere.g=32.153; % ft/s^2;
% Vmax = 113;
% lim = 100;
% lim1 = 120;

%%%      2302 ft
% atmosphere.rho=22.26e-4; % slug/ft^3
% atmosphere.mu=3.691e-7; % lbs-s/ft^2
% atmosphere.g=32.167; % ft/s^2;
% Vmax = 113;
% lim = 100;
% lim1 = 120;

% 4895 ft
% atmosphere.rho=20.55e-4; % slug/ft^3
% atmosphere.mu=3.692e-7; % lbs-s/ft^2
% atmosphere.g=32.159; % ft/s^2;
% Vmax = 113;
% lim = 100;
% lim1 = 120;

% 5000 LEVEL
% atmosphere.rho=20.48e-4; % slug/ft^3
% atmosphere.mu=3.637e-7; % lbs-s/ft^2
% atmosphere.g=32.159; % ft/s^2;

% 10000 LEVEL
% atmosphere.rho=17.56e-4; % slug/ft^3
% atmosphere.mu=3.534e-7; % lbs-s/ft^2
% atmosphere.g=32.143; % ft/s^2;

% 15000 LEVEL
% atmosphere.rho=14.96e-4; % slug/ft^3
% atmosphere.mu=3.430e-7; % lbs-s/ft^2
% atmosphere.g=32.128; % ft/s^2;

% 20000 LEVEL
% atmosphere.rho=12.67e-4; % slug/ft^3
% atmosphere.mu=3.324e-7; % lbs-s/ft^2
% atmosphere.g=32.112; % ft/s^2;

% 25000 LEVEL
% atmosphere.rho=10.66e-4; % slug/ft^3
% atmosphere.mu=3.217e-7; % lbs-s/ft^2
% atmosphere.g=32.097; % ft/s^2;

% 30000 LEVEL
% atmosphere.rho=8.91e-4; % slug/ft^3
% atmosphere.mu=3.107e-7; % lbs-s/ft^2
% atmosphere.g=32.082; % ft/s^2;

% 35000 LEVEL
% atmosphere.rho=7.38e-4; % slug/ft^3
% atmosphere.mu=2.995e-7; % lbs-s/ft^2
% atmosphere.g=32.066; % ft/s^2;

% 40000 LEVEL
% atmosphere.rho=5.87e-4; % slug/ft^3
% atmosphere.mu=2.969e-7; % lbs-s/ft^2
% atmosphere.g=32.051; % ft/s^2;
45,000 FEET
% atmosphere.rho=4.62e-4; % slug/ft^3
% atmosphere.mu=2.969e-7; % lbs-s/ft^2
% atmosphere.g=32.036; % ft/s^2;
% Vmax = 191.45;
% lim = 110;
% lim1 = 250;

50,000 FEET
% atmosphere.rho=3.64e-4; % slug/ft^3
% atmosphere.mu=2.969e-7; % lbs-s/ft^2
% atmosphere.g=32.020; % ft/s^2;

60,000 FEET
% atmosphere.rho=2.26e-4; % slug/ft^3
% atmosphere.mu=2.969e-7; % lbs-s/ft^2
% atmosphere.g=31.990; % ft/s^2;

61,000 FEET
% atmosphere.rho=0.002173; % slug/ft^3
% atmosphere.mu=2.9705e-7; % lbs-s/ft^2
% atmosphere.g=31.99; % ft/s^2;
% Vmax = 245.3;
% lim = 120;
% lim1 = 350;

70,000 FEET
% atmosphere.rho=1.39e-4; % slug/ft^3
% atmosphere.mu=2.984e-7; % lbs-s/ft^2
% atmosphere.g=31.96; % ft/s^2;

79,000 FEET
% atmosphere.rho=0.913e-4; % slug/ft^3
% atmosphere.mu=3.014e-7; % lbs-s/ft^2
% atmosphere.g=31.932; % ft/s^2;
% Vmax = 294;
% lim = 80;
% lim1 = 350;

80,000 FEET
% atmosphere.rho=0.86e-4; % slug/ft^3
% atmosphere.mu=3.018e-7; % lbs-s/ft^2
% atmosphere.g=31.929; % ft/s^2;

90,000 FEET
% atmosphere.rho=0.56e-4; % slug/ft^3
% atmosphere.mu=3.052e-7; % lbs-s/ft^2
% atmosphere.g=31.897; % ft/s^2;

100,000 FEET
% atmosphere.rho=0.33e-4; % slug/ft^3
% atmosphere.mu=3.087e-7; % lbs-s/ft^2
% atmosphere.g=31.868; % ft/s^2;
%%% AIRCRAFT REFERENCE PARAMETERS
%======================================================================

aircraft.S = 557.5; % wing area
aircraft.Stail = 45; % Tail area
aircraft.b = 128.6; % wing span
aircraft.c = 4.0; % mean chord
aircraft.fpar = 1.26; % parasite drag area (ft^2)
aircraft.CLmax = 1.4; % CLmax

P_eff = 0.88; % Prop efficiency
m_eff = 0.9; % Motor Efficiency
S_eff = 0.3; % Solar Cell Efficiency
eC = 350;  % Battery charge density, watt-hr/kg
We = 0.5;  % Requirement, Empty weight ratio
Wp = 250; % Requirement, Payload weight, lb
t = 24; % Requirement, Endurance, hr
Cdo = 0.011; % Historic data
eo = 0.8; % Historic data
g_roll = 350; % Assumed
VS = 37; % Stall Speed, picked

% Aspect Ratio
aircraft.AR = aircraft.b^2/aircraft.S;

% Stall Speed, ft/s
aircraft.Vs = VS/(sqrt(atmosphere.rho/23.77e-4));

% Cruise Speed, ft/s
aircraft.Vc = 1.2*aircraft.Vs;

% Lift to Drag ratio
LD = sqrt((pi*eo*aircraft.AR)/(4*Cdo));

% Battery weight ratio
Wb = (VS*0.3048*9.807*t)/(eC*P_eff*LD);

% Takeoff Wright
aircraft.W = Wp/(1-We-Wb);

% Wing Loading
WS = aircraft.W/aircraft.S;

% Thrust to Weight Ratio
TW = 1.21*aircraft.W/(atmosphere.rho*atmosphere.g*aircraft.CLmax*g_roll*aircraft.S);

% mass (slugs)
aircraft.m = aircraft.W/atmosphere.g;

% Cruise Cl
CLc = (2*aircraft.W)/(atmosphere.rho*aircraft.S*(aircraft.Vc^2));
% Power Generated, kw
Pgen = (aircraft.S + aircraft.Stail) * S_eff * 0.092903;

% Power Available, kw
Pava = Pgen * m_eff * P_eff;

% Reynolds Number
Re1 = (atmosphere.rho * aircraft.c * aircraft.Vc) / (atmosphere.mu);

figure();
subplot(2,1,1)
AR1 = 0:50;
LD1 = sqrt((pi * eo * AR1) / (4 * Cdo));
plot(AR1, LD1)
xlabel('Aspect Ratio')
ylabel('L/D')
subplot(2,1,2)
LD2 = 25:50;
W1 = Wp ./ (1 - We - (VS * 0.3048 * 9.807 * t) ./ (eC * P_eff * LD2));
plot(LD2, W1)
xlabel('L/D')
ylabel('Weight (lbs)')

%==========================================================================
% %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
% DRAG CALCULATION
% %%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
%==========================================================================

CL = 0.1:0.1:1.2;
NCL = length(CL);

% airfoil polar files (from XFOIL)
wing_airfoil_name = 'SM701.dat';
wing_airfoil_file = {
    'SM701_Re100k_f.pol';
    'SM701_Re200k_f.pol';
    'SM701_Re400k_f.pol';
    'SM701_Re600k_f.pol';
    'SM701_Re800k_f.pol';
    'SM701_Re1000k_f.pol';
    'SM701_Re1500k_f.pol';
    'SM701_Re2000k_f.pol';
    'SM701_Re3000k_f.pol';
    'SM701_Re4000k_f.pol';
    'SM701_Re5000k_f.pol';
};

tail_airfoil_name = 'jouk0015.dat';
tail_airfoil_file = {
    'jouk0015_Re0100k.pol';
    'jouk0015_Re0200k.pol';
    'jouk0015_Re0400k.pol';
    'jouk0015_Re0600k.pol';
    'jouk0015_Re0800k.pol';
    'jouk0015_Re1000k.pol';
    'jouk0015_Re1500k.pol';
};
'jouk0015_Re2000k.pol';
'jouk0015_Re3000k.pol';
'jouk0015_Re4000k.pol';
'jouk0015_Re5000k.pol';

surfacefile={
  'n1_1';
  'n1_2';
  'n1_3';
  'n1_4';
  'n1_5';
  'n1_6';
  'n1_7';
  'n1_8';
  'n1_9';
  'n1_10';
  'n1_11';
  'n1_12';
};

% get airfoil data
wing_alpha_range=-5:0.1:11;
wing_CL_range=0.01:0.1:1.51;
wing_airfoil=ParseXFOILData_alpha(1,wing_airfoil_file,
[1e5 2e5 4e5 6e5 8e5 1e6 1.5e6 2e6 3e6 4e6 5e6],wing_alpha_range,wing_CL_range,wing_airfoil_name);

tail_alpha_range=-5:0.1:10;
tail_CL_range=-0.06:0.01:1.1;
tail_airfoil=ParseXFOILData_alpha(1,tail_airfoil_file,
[1e5 2e5 4e5 6e5 8e5 1e6 1.5e6 2e6 3e6 4e6 5e6],tail_alpha_range,tail_CL_range,tail_airfoil_name);

SHOWPLOTS=1;
CD.induced=NaN*ones(NCL,1);
D.induced=NaN*ones(NCL,1);
CD.fuse=NaN*ones(NCL,1);
D.fuse=NaN*ones(NCL,1);
D.tot=NaN*ones(NCL,1);
q=zeros(NCL,1);
va=zeros(NCL,1);
e=NaN*ones(NCL,1);

for i=1:NCL
  q(i)=aircraft.m*atmosphere.g/(aircraft.S*CL(i));
  va(i)=sqrt(2*q(i)/atmosphere.rho);

  % parasite drag
  D.par(i)=q(i)*aircraft.fpar;

  % aerosurface drag
  [CD.w_prof(i),CD.w_ind(i),CD.h_prof(i),CD.h_ind(i)]=ComputeAeroSurfaceDrag(SHOWPLOTS,va(i),surfacefile(i),wing_airfoil,tail_airfoil);
  D.w_prof(i)=q(i)*aircraft.S*CD.w_prof(i);
D.w_ind(i)=q(i)*aircraft.S*CD.w_ind(i);
e(i)=(CL(i)^2/(pi*aircraft.AR))/CD.w_ind(i); % compute span efficiency

D.h_prof(i)=q(i)*aircraft.S*CD.h_prof(i);
D.h_ind(i)=q(i)*aircraft.S*CD.h_ind(i);

D.tot(i)=D.par(i)+D.w_ind(i)+D.w_prof(i)+D.h_prof(i)+D.h_ind(i);
CD.tot(i)=D.tot(i)/(q(i)*aircraft.S);
end

figure();
subplot(2,1,1)
hold on;
plot(va,D.par,va,D.w_ind,va,D.w_prof,va,D.h_ind,va,D.h_prof,va,D.tot)
h11 = line([aircraft.Vs aircraft.Vs],[-5 lim]);
set(h11,'color','b');
h12 = line([Vmax Vmax],[-5 lim]);
set(h12,'color','m');
hold off;
axis([aircraft.Vs-10 lim1 -5 lim1]);
legend('parasite','wing induced','wing profile','HTail induced','HTail profile','Total','Stall Speed','Max Speed')
ylabel('Drag force (lbs)')
xlabel('v_a (ft/s)')

subplot(2,1,2)
hold on;
plot(va,va.*D.tot*4.45*.3/1000)
limit = refline([0 Pava]);
set(limit,'Color','r');
h12 = line([Vmax Vmax],[0 Pava]);
set(h12,'color','g');
hold off;
axis([aircraft.Vs-10 lim1 0 20]);
ylabel('Power (kW)')
xlabel('v_a (ft/s)')
legend('Power Required','Power Available','Max Speed')

% compute drag coefficient fit as CD=CD0 + k*CL^2
A=[ones(NCL-2,1) transpose(CL(1:NCL-2)).^2];
CDpoly=(A'*A)\A'*CD.tot(1:NCL-2);%

fprintf('---------------------------------------------------
')
fprintf('----------

UAV Characteristics

Wing Area = %.2f ft^2 \n',aircraft.S)
fprintf('Projected Span, b = %.2f ft \n',aircraft.b)
fprintf('Reference Chord Length, c = %.2f ft \n',aircraft.c)
fprintf('Aspect Ratio = %.2f \n',aircraft.AR)
fprintf('L/D = %.2f \n',LD)
fprintf('Wing Loading = %.2f lb/ft^2 \n',WS)
fprintf('Battery Weight Ratio = %.2f \n',Wb)
fprintf('Payload Weight = %.2f lb \n',Wp)
fprintf('Battery Weight = %.2f lb \n',Wb*aircraft.W)
fprintf('Empty Weight = %.2f lb \n',We*aircraft.W)
fprintf('Total Takeoff Weight = %.2f lb \n',aircraft.W)
fprintf('Thrust to Weight Ratio = %.2f \n',TW)
fprintf('Stall Speed = %.2f ft/s \n',aircraft.Vs)
fprintf('Cruise Speed = %.2f ft/s \n',aircraft.Vc)
fprintf('Cruise Life Coefficieny = %.2f \n',CLc)
fprintf('Power Generated = %.2f kw \n',Pgen)
fprintf('Power Available = %.2f kw \n',Pava)
fprintf('Reynolds Number = %.2f \n',Re1)
fprintf('Cd_o = %.4f \n',CDpoly(1))
fprintf('Oswald efficiency = %g \n',1/(pi*aircraft.AR*CDpoly(2)))

fprintf('-----------------------------------------------------------\n')
fprintf('-----------------------------------------------------------\n')

%%%% FLIGHT ENVELOPE
%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%%
Dto = 18.34; % Drag at sea level for takeoff
Pmin_sea = 1.055; % kw

h = [0 5000 10000 15000 20000 25000 30000 35000 40000 45000 50000 60000 70000 80000 90000 100000];
d = (10^-4).*[23.77 20.48 17.56 14.96 12.67 10.66 8.91 7.38 5.87 4.62 3.64 2.26 1.39 0.86 0.56 0.33];
Drag = [18.34 18.45 18.56 18.7 19 19.1 19.2 19.45 19.8 20.3 20.9 22.3 24.1 27.1 30 32];
n = length(h);

% h vs. v plot
Vstall = zeros(n,1);
for i=1:n
    if i == 1
        Vstall(i) = 37;
    else
        Vstall(i) = (Vstall(1)/(sqrt(d(i)/d(1))));
    end
end
Vcruise = 1.2*Vstall;
\[ H = [0 \ 5000 \ 10000 \ 15000 \ 20000 \ 25000 \ 30000 \ 35000 \ 40000 \ 45000 \ 50000 \ 60000 \ 70000 \ 80000 \ 90000 \ 100000]; \]
\[ Vpmax = [113.5 \ 118.2 \ 123.5 \ 129.1 \ 135.8 \ 143.4 \ 151.8 \ 161.9 \ 175.3 \ 191.45 \ 209.8 \ 243.1 \ 273.7 \ 295.1 \ 309 \ 210]; \]

\[ habs = 94000; \]

```matlab
figure();
hold on;
plot(Vstall,h,Vpmax,H)
hlabs = refline([0 habs]);
set(hlabs,'Color','r')
hold off;
xlabel('Velocity (ft/s)')
ylabel('Altitude (ft)')
legend('Stall Constraint','Power Constraint','Absolute Ceiling')
```

%-------------------------------------------------------------
% h vs. h_dot plot

\[ Pexc = Pava-Pmin_sea; \]
\[ W = aircraft.W*4.448; \quad \% \text{Newtons} \]
\[ hdot_o = Pexc*1000*3.28*60/W; \quad \% \text{ft/m} \]

\[ h = 0:1:habs; \]
\[ hdot = hdot_o*(1-(h/habs)); \]
\[ hoper = (1-(100/hdot_o))*habs; \quad \% \text{Operational} \]

```matlab
figure();
hold on;
plot(hdot,h)
hlabs = refline([0 habs]);
set(hlabs,'Color','r')
h11 = line([0 100],[hoper hoper]);
set(h11,'color','g')
h11 = line([100 100],[0 hoper]);
set(h11,'color','g')
hold off;
xlabel('hdot (ft/min)')
ylabel('Altitude (ft)')
legend('Climb Rate','Absolue Ceiling','Max Operation Ceiling')
```

%-------------------------------------------------------------
% h vs. time to climb plot

\[ h=0:500:habs; \]
\[ n = length(h); \]
\[ tc = zeros(n,1); \]

```matlab
for i = 1:n-1
    if i==1
        tc(i) = ((habs/hdot_o)*log((habs-h(i))/(habs-h(i+1))))/60;
    else
        tc(i) = tc(i-1)+((habs/hdot_o)*log((habs-h(i))/(habs-h(i+1))))/60;
    end
end
```
\[ H = 0:500:500; \]
\[ \text{nn} = \text{length}(H); \]
\[ \text{TC} = 0; \]
\[ \text{for } i = 1:\text{nn}-1 \]
\[ \quad \text{if } i==1; \]
\[ \quad \quad \text{TC} = \frac{(\text{habs}/\text{hdot}_o) \log((\text{habs}-H(i))/(\text{habs}-H(i+1))))}{60}; \]
\[ \quad \text{else} \]
\[ \quad \quad \text{TC} = \text{TC} +\frac{(\text{habs}/\text{hdot}_o) \log((\text{habs}-H(i))/(\text{habs}-H(i+1))))}{60}; \]
\[ \quad \text{end} \]
\[ \text{end} \]
\[ H = 0:500:61500; \]
\[ \text{nn} = \text{length}(H); \]
\[ \text{TC1} = 0; \]
\[ \text{for } i = 1:\text{nn}-1 \]
\[ \quad \text{if } i==1; \]
\[ \quad \quad \text{TC1} = \frac{(\text{habs}/\text{hdot}_o) \log((\text{habs}-H(i))/(\text{habs}-H(i+1))))}{60}; \]
\[ \quad \text{else} \]
\[ \quad \quad \text{TC1} = \text{TC1} +\frac{(\text{habs}/\text{hdot}_o) \log((\text{habs}-H(i))/(\text{habs}-H(i+1))))}{60}; \]
\[ \quad \text{end} \]
\[ \text{end} \]

figure()
hold on;
axis([0 14 0 100000])
plot(tc, h)
hlabs = reline([0 habs]);
set(hlabs, 'Color', 'r')
h11 = line([0 TC],[hoper hoper]);
set(h11, 'color', 'm')
h13 = line([0 TC1],[61000 61000]);
set(h13, 'color', 'g')
h12 = line([TC TC],[0 hoper]);
set(h12, 'color', 'm')
h14 = line([TC1 TC1],[0 61000]);
set(h14, 'color', 'g')
xlabel('t_c (hours)')
ylabel('Altitude (ft)')
legend('Time to Climb','Absolute Ceiling','Max Operational Ceiling','61,000 ft')
hold off;

fprintf('Flight Envelope \n')
fprintf('____________________ \n')
fprintf('Absolute Ceiling = %.1f ft\n',habs)
fprintf('Operational Ceiling = %.1f ft\n',hoper)
fprintf('Rate of Climb at sea level = %.2f ft/min \n',hdot_o)
fprintf('Time to Climb to 61,000 ft = %.2f hours \n',TC1)
fprintf('Time to Climb to Max Operational Ceiling = %.2f hours \n',TC)

%===================================================================
%%
%====================================================================
%' WING STRUCTURE ANALYSIS

clc, clear

%---------------------------------------------------------------------------------------------------
%%% READ IN DATA FROM A FILE
%---------------------------------------------------------------------------------------------------

surfacefile = 'Cl_data';
fid=fopen(surfacefile);

for i=1:6, fgets(fid); end

%%% Wing
tline = fgets(fid);
fprintf('Analyzing %s
',tline)
tline = fgets(fid);
NS = sscanf(tline(37:44), '%d');

for i=1:12, fgets(fid); end % skip 12 lines

y = zeros(NS,1);
c = zeros(NS,1);
A = zeros(NS,1);
c1 = zeros(NS,1);
cd_i = zeros(NS,1);

for i=1:NS
  tline=fgets(fid);
  D=sscanf(tline,'%f');
  y(i)=D(2);
  c(i)=D(3);
  A(i)=D(4);
  c1(i)=D(8);
  cd_i(i)=D(9);
end

%---------------------------------------------------------------------------------------------------
%%% %---------------------------------------------------------------------------------------------------

% SEA LEVEL
rho = 23.77e-4; % slug/ft^3
Vc = 44.4; % ft/s

% CARBON FIBER PROPERTIES
E = 200*145037.7377; % GPa to Psi
density = 1.4*((2.54^3)*0.00220462)*(12^3); % g/cc to lbs/ft^3
T = (1/32)/12; % inches to ft, thickness about 4 to 5 plies

Bweight = 180; % Battery weight, lbs
Warea = 557.5; % ft^2
Span = 2*y(NS); % ft
Motor = 4; % lbs, 0 10 and 25 ft
SC = 0.157; % lbs/ft^2, Solar cell weight
Tail Area = 45; % ft^2
Fuse_Area = 144; % ft^2
Vtail Area = 17.5; % ft^2
Fw = 250; % lbs, payload weight

% CALCULATING WEIGHTS
Tail weight = Tail_Area*T*density; %lbs
Fuse weight = Fuse_Area*T*density; %lbs
Vtail weight = Vtail_Area*T*density; %lbs
wing weight = Warea*T*density; %lbs
Scell weight = SC*Warea; %lbs

% DISTRIBUTING WEIGHT ALONG THE WING

Wd = wing_weight/Warea; %lbs/ft^2, wing weight distribution

% CALCULATING AREA WHERE BATTERY WEIGHT IS DISTRIBUTED
AREA = 0;
for i = 20:NS
    AREA = AREA + A(i);
end

% DISTRIBUTING WEIGHT
Bwi = zeros(NS,1);
SCi = zeros(NS,1);
Wdi = zeros(NS,1);
Pwi = zeros(NS,1);

for i = 1:NS
    Wdi(i) = Wd*A(i);
    SCi(i) = SC*A(i);
    Pwi(i) = ((Pw-200)/Warea)*A(i);
    if i >= 20;
        Bwi(i) = (Bweight/(2*AREA))*A(i);
    end
end

vol = zeros(NS,1);

for ijk = 1:2;

end
for i = 1:NS
    Spi(i) = vol(i)*density/(12^3);
    if (i == 7 || i == 17) % location of motor according to y(i)
        W(i) = Wdi(i)+Bwi(i)+SCi(i)+Spi(i)+Pwi(i)+Motor;
    else
        W(i) = Wdi(i)+Bwi(i)+SCi(i)+Spi(i)+Pwi(i);
    end
end

% LIFT DISTRIBUTION ALONG THE WING
Li = zeros(NS,1);
for i = 1:NS
    if i <=33
        Li(i) = ((1/2).*rho*(Vc^2).*A(i)*cl(i));
    else
        Li(i) = cosd(15)*((1/2).*rho*(Vc^2).*A(i)*cl(i));
    end
end

% CALCULATING BENDING MOMENT
Mi = zeros(NS,1);
MI = zeros(NS,1);
for i = 1:NS
    I = y(i);
    for j = i:NS
        if y(j) == I;
            Mi(i) = (Li(j)-W(j))*(y(j)-I);
        else
            Mi(i) = MI(j-1) + (Li(j)-W(j))*(y(j)-I);
        end
        MI(j) = Mi(i);
    end
end

% SPAR SIZING
vol = zeros(NS,1);
volume = 0;

% Finding Ix
Ix = zeros(NS,1);
h = zeros(NS,1);

t = 0.15; % inches
w = 2.15; % inches
I = 0;

for i = 1:NS
    if i <=33
h(i) = .036*c(i)*12; %inches, assuming 3.6% thickness of mean chord
Ix(i) = ((w*h(i)^3)-(((h(i)-2*t)^3)*(w-t)))/12; %in^4
else
   I = I+1;
   h(i) = 0.16*12; %inches, assuming 16% thickness of mean chord
   Ix(i) = ((w*h(i)^3)-(((h(i)-2*t)^3)*(w-t)))/12; %in^4
end
if i == 1;
   vol(i) = (((h(i)-2*t)*t)+(w*2*t))*(y(i)*12);
else
   vol(i) = (((h(i)-2*t)*t)+(w*2*t))*((y(i)-y(i-1))*12);
end
volume = volume + vol(i);
Weight = density*volume/(12^3); %lbs
end

% WING DEFLECTION

EI = zeros(NS,1);
for i = 1:NS;
   EI(i) = E*Ix(i);
end
% Moment Equation
N = 4; % Ployfit order
[P,S] = polyfit(y,Mi,N);
def = zeros(NS,1);
x1 = zeros(NS,1);
x2 = zeros(NS,1);
x3 = zeros(NS,1);
x4 = zeros(NS,1);
x5 = zeros(NS,1);
for i = 1:NS
   x1(i) = (((P(1)/((N+1)*(N+2)))*(y(i)^((N+2))));
x2(i) = (((P(2)/((N)*(N+1)))*(y(i)^((N+1))));
x3(i) = (((P(3)/((N-1)*(N)))*(y(i)^((N))));
x4(i) = (((P(4)/((N-2)*(N-1)))*(y(i)^((N-1))));
x5(i) = (((P(5)/((1)*(2)))*(y(i)^((2))));
def(i) = 12*(x1(i)+x2(i)+x3(i)+x4(i)+x5(i))/EI(i);
end
Y = zeros(2*NS,1);
DEF = zeros(2*NS,1);
% % Distributing points along the WHOLE wing
for i = 1:(2*NS)
   if i <= NS
      Y(i) = -1*y(NS-i+1);
      DEF(i) = def(NS-i+1);
   else
      Y(i) = y(i-NS);
DEF(i) = def(i-NS);
end
end

%---------------------------------------------------------------------
% STRESS CALCULATION

Sigma = zeros(NS,1);
z = .5*h;
for i = 1:NS
    Sigma(i) = (Mi(i)*12*z(i)/Ix(i));  % Psi
end
end

%---------------------------------------------------------------------
% OUTPUTS

Landing_gear = 40;  % total
Addones = 15;  %lbs, formers in the fuse
TOTAL = Addones+wing_weight+Fuse_weight+Tail_weight+Vtail_weight+Scell_weight+Bweight +2*Weight+Pw+Landing_gear+4*Motor;
fprintf('Weight of the Wing Skin is = %.2f lbs.\n',wing_weight)
fprintf('Weight of the Fuselage Skin is = %.2f lbs.\n',Fuse_weight)
fprintf('Weight of the Horizontal Stab. Skin is = %.2f lbs.\n',Tail_weight)
fprintf('Weight of the Vertical Stab. Skin is = %.2f lbs.\n',Vtail_weight)
fprintf('Weight of the Solar Cell is = %.2f lbs.\n',Scell_weight)
fprintf('Weight of the Battery is = %.2f lbs.\n',Bweight)
fprintf('Weight of the Wing Spar is = %.2f lbs.\n',2*Weight)
fprintf('Weight allocated for Payload is = %.2f lbs.\n',Pw)
fprintf('Weight allocated for 4 Motor is = %.2f lbs.\n',4*Motor)
fprintf('Weight allocated for Fuse formers is = %.2f lbs.\n',Addones)
fprintf('Weight of the Landing Gear system is = %.2f lbs.\n',Landing_gear)

% PLOTS

figure(1);
plot (y,W,'b','-1*y,W,'b')
axis([-65 65 0 12])
xlabel('Span (ft)')
ylabel('Weight (lbs)')

figure(2);
plot (-1*y,LI,'b','-1*y,LI-W,'r',y,LI,'b',y,LI-W,'r')
axis([-65 65 0 15])
xlabel('Span (ft)')
ylabel('Lift (lbs)')
legend('w/o Weight','w/ Weight')

figure(3);
plot (y,Mi)
xlabel('Span (ft)')
ylabel('Moment (ft*lbs)')

figure(4);
plot (Y,DEF,'b',Y,2*DEF,'g',Y,3*DEF,'m',Y,4*DEF,'r')
xlabel('Span (ft)')
ylabel('Deflection (ft)')
legend('1g Gust','2g Gust','3g Gust','4g Gust')

figure(5);
plot (y,Sigma)
xlabel('Span (ft)')
ylabel('Stress (Psi)')

%==========================================================================
%%
READ IN DATA FROM A FILE FOR HT
%==========================================================================

%surfacefile = 'Cl_dataHT'; % Cruise Condition
%surfacefile = 'Cl_dataHTT'; % Max Cl
surfacefile = 'Cl_dataHTTT'; % Loq Cl
fid=fopen(surfacefile); 
for i=1:6, fgets(fid); end

%%% Wing
%fprintf('Analyzing %s
',tline)
tline = fgets(fid);
NS = sscanf(tline(37:44),'%d');

for i=1:12, fgets(fid); end % skip 12 lines

y = zeros(NS,1);
c = zeros(NS,1);
A = zeros(NS,1);
cl = zeros(NS,1);
cd_i = zeros(NS,1);

for i=1:NS
    tline=fgets(fid);
    D=sscanf(tline,'%f');
    y(i)=D(2);
    c(i)=D(3);
    A(i)=D(4);
    cl(i)=D(8);
    cd_i(i)=D(9);
end

%%% DISTRIBUTING WEIGHT ALONG THE HT
%==========================================================================
Wt = Tail_weight/Tail_Area; % lbs/ft^2, wing weight distribution

% DISTRIBUTING WEIGHT
Wti = zeros(NS,1);

for i = 1:NS
    Wti(i) = Wt*A(i);
end

vol = zeros(NS,1);

for ijk = 1:2;
% -------------
% DISTRIBUTING WEIGHTS ALONG THE HT

Spi = zeros(NS,1);
W = zeros(NS,1);

for i = 1:NS
    Spi(i) = vol(i)*density/(12^3);
    W(i) = Spi(i)+Wti(i);
end

% ---------------
% LIFT DISTRIBUTION ALONG THE WING
Li = zeros(NS,1);

for i = 1:NS
    Li(i) =((1/2).*rho*(Vc^2).*A(i)*cl(i));
end

% ---------------
% CALCULATING BENDING MOMENT
Mi = zeros(NS,1);
MI = zeros(NS,1);

for i = 1:NS
    I = y(i);
    for j = i:NS
        if y(j) == I;
            Mi(i) = (Li(j)-W(j))*(y(j)-I);
        else
            Mi(i) = MI(j-1) + (Li(j)-W(j))*(y(j)-I);
        end
        MI(j) = Mi(i);
    end
end

% ---------------
% SPAR SIZING

vol = zeros(NS,1);
volume = 0;

% Finding Ix
Ix = zeros(NS,1);  
h = zeros(NS,1);  

% D-TUBE  
t = 0.035;  % inches  
w = 1;  % inches  
I = 0;

for i = 1:NS  
    h(i) = .1*c(i)*12;  %inches, assuming 3.6% thickness of mean chord  
    Ix(i) = ((w*h(i)^3)-(((h(i)-2*t)^2)*w))/12;  %in^4  
    if i == 1;  
        vol(i) = (((h(i)-2*t)*t)+(w*2*t))*(y(i)*12);  
    else  
        vol(i) = (((h(i)-2*t)*t)+(w*2*t))*)((y(i)-y(i-1)))*12;  
    end  
    volume = volume + vol(i);  
    WeightHT = density*volume/(12^3);  %lbs  
end  

% WING DEFLECTION  
EI = zeros(NS,1);  

for i = 1:NS;  
    EI(i) = E*Ix(i);  
end  

% Moment Equation  
N = 4;  % Ployfit order  
[P,S] = polyfit(y,Mi,N);  

def = zeros(NS,1);  
x1 = zeros(NS,1);  
x2 = zeros(NS,1);  
x3 = zeros(NS,1);  
x4 = zeros(NS,1);  
x5 = zeros(NS,1);  

for i = 1:NS  
    x1(i) = ((P(1)/(N+1)*(N+2)))*(y(i)^2);  
    x2(i) = ((P(2)/(N+1)))*(y(i)^3);  
    x3(i) = ((P(3)/(N-1)))*(y(i)^4);  
    x4(i) = ((P(4)/(N-2)))*(y(i)^5);  
    x5(i) = ((P(5)/(2)))*(y(i)^6);  
    def(i) = 12*(x1(i)+x2(i)+x3(i)+x4(i)+x5(i))$/EI(i);  
end  

Y = zeros(2*NS,1);  
DEF = zeros(2*NS,1);  

% Distributing points along the WHOLE wing  
for i = 1:(2*NS)  
    if i <= NS  
        ...
Y(i) = -1*y(NS-i+1);
DEF(i) = def(NS-i+1)*12;
else
  Y(i) = y(i-NS);
  DEF(i) = def(i-NS)*12;
end

% STRESS CALCULATION

 Sigma = zeros(NS,1);
z = .5*h;

for i = 1:NS
  Sigma(i) = (Mi(i)*12*z(i)/Ix(i)); % Psi
end

figure(6);
plot (y,W,'b',-1*y,W,'b')
xlabel('Span (ft)')
ylabel('Weight (lbs)')

figure(7);
plot (-1*y, Li, 'b', -1*y, Li-W, 'r', y, Li, 'b', y, Li-W, 'r')
xlabel('Span (ft)')
ylabel('Lift (lbs)')
legend('w/o Weight','w/ Weight')

figure(8);
plot (y,Mi)
xlabel('Span (ft)')
ylabel('Moment (ft*lbs)')

figure(9);
plot (Y,DEF,'b',Y,2*DEF,'g',Y,3*DEF,'m',Y,4*DEF,'r')
xlabel('Span (ft)')
ylabel('Deflection (in)')
legend('1g Gust','2g Gust','3g Gust','4g Gust')

figure(10);
plot (y, Sigma)
xlabel('Span (ft)')
ylabel('Stress (Psi)')

fprintf('Weight of the Tail Spar is = %.2f lbs.
',2*WeightHT)

%==========================================================================
%==========================================================================
%==========================================================================
%%%                READ IN DATA FROM A FILE FOR VT
%==========================================================================

%surfacefile = 'Cl_dataVT';  % Cruise condition
surfacefile = 'Cl_dataVT';  % Max Cl
fid=fopen(surfacefile);

for i=1:6, fgets(fid); end

%%% Wing
tline = fgets(fid);
%fprintf('Analyzing %s
',tline)
tline = fgets(fid);
NS = sscanf(tline(37:44),'%d');

for i=1:12, fgets(fid); end % skip 12 lines

y = zeros(NS,1);
c = zeros(NS,1);
A = zeros(NS,1);
cl = zeros(NS,1);
ct_i = zeros(NS,1);

for i=1:NS
tline=fgets(fid);
D=sscanf(tline,'%f');
y(i)=(7/NS)*i;
c(i)=D(3);
A(i)=D(4);
cl(i)=D(8);
ct_i(i)=D(9);
end

%==========================================================================
%%%                DISTRIBUTING WEIGHT ALONG THE VT
%==========================================================================

Wv = Vtail_weight/Vtail_Area; %lbs/ft^2, wing weight distribution

% DISTRIBUTING WEIGHT
Wvi = zeros(NS,1);

for i = 1:NS
    Wvi(i) = Wv*A(i);
end

vol = zeros(NS,1);

for ijk = 1:2;
% % DISTRIBUTING WEIGHTS ALONG THE HT
Spi = zeros(NS,1);
W = zeros(NS,1);

for i = 1:NS
    Spi(i) = vol(i)*density/(12^3);
    W(i) = Spi(i)+Wvi(i);
end

% -------------------------------
% LIFT DISTRIBUTION ALONG THE WING
Li = zeros(NS,1);
for i = 1:NS
    Li(i) = ((1/2).*rho*(Vc^2).*A(i).*cl(i));
end

% -------------------------------
% CALCULATING BENDING MOMENT
Mi = zeros(NS,1);
MI = zeros(NS,1);
for i = 1:NS
    I = y(i);
    for j = i:NS
        if y(j) == I;
            Mi(i) = (Li(j)-W(j))*(y(j)-I);
        else
            Mi(i) = MI(j-1) + (Li(j)-W(j))*(y(j)-I);
        end
        MI(j) = Mi(i);
    end
end

% -------------------------------
% SPAR SIZING
vol = zeros(NS,1);
volume = 0;

% Finding Ix
Ix = zeros(NS,1);
h = zeros(NS,1);

% D-TUBE
h = 0.035;  % inches
w = 1;  % inches
I = 0;

for i = 1:NS
    h(i) = .1*c(i)*12;  % inches, assuming 3.6% thickness of mean chord
    Ix(i) = ((w*h(i)^3)-(((h(i)-2*t)^3)*(w-t)))/12;  % in^4
    if i == 1;
        vol(i) = (((h(i)-2*t)*t)+(w*2*t))*(y(i)*12);
    else
        vol(i) = (((h(i)-2*t)*t)+(w*2*t))*((y(i)-y(i-1))*12);
    end
volume = volume + vol(i);
    WeightVT = density*volume/(12^3); %lbs
end

%-------------------------------------------------------------------------------------
% WING DEFLECTION

EI = zeros(NS,1);
for i = 1:NS;
    EI(i) = E*Ix(i);
end

% Moment Equation
N = 4; % Ployfit order
[P,S] = polyfit(y,Mi,N);
def = zeros(NS,1);
x1 = zeros(NS,1);
x2 = zeros(NS,1);
x3 = zeros(NS,1);
x4 = zeros(NS,1);
x5 = zeros(NS,1);
for i = 1:NS
    x1(i) = ((P(1)/((N+1)*(N+2)))*(y(i)^(N+2)));
    x2(i) = ((P(2)/((N+1)))*(y(i)^(N+1)));
    x3(i) = ((P(3)/(N*(N+1)))*(y(i)^N));
    x4(i) = ((P(4)/((N-1)*(N-1)))*(y(i)^N));
    x5(i) = ((P(5)/((1)*(2)))*(y(i)^2));
    def(i) = 12*(x1(i)+x2(i)+x3(i)+x4(i)+x5(i))/EI(i);
end

%-------------------------------------------------------------------------------------
% STRESS CALCULATION

Sigma = zeros(NS,1);
z = .5*h;
for i = 1:NS
    Sigma(i) = (Mi(i)*12*z(i)/Ix(i)); % Psi
end

figure(11);
plot (W,y,'b')
ylabel('Span (ft)')
xlabel('Weight (lbs)')

figure(12);
plot (Li,y,'b',Li-W,y,'r')
ylabel('Span (ft)')
xlabel('Lift (lbs)')
legend('w/o Weight','w/ Weight')
figure(13);
plot (Mi,y)
ylabel('Span (ft)')
xlabel('Moment (ft*lbs)')

figure(14);
plot (def,y,'b',2*def,y,'g',3*def,y,'m',4*def,y,'r')
ylabel('Span (ft)')
xlabel('Deflection (in)')
legend('1g Gust','2g Gust','3g Gust','4g Gust')

figure(15);
plot (Sigma,y)
ylabel('Span (ft)')
xlabel('Stress (Psi)')

fprintf('Weight of the Ver_Tail Spar is = %.2f lbs.
',WeightVT)

fprintf('Total Weight of UAV = = %.2f lbs.
',TOTAL+WeightHT+WeightVT)