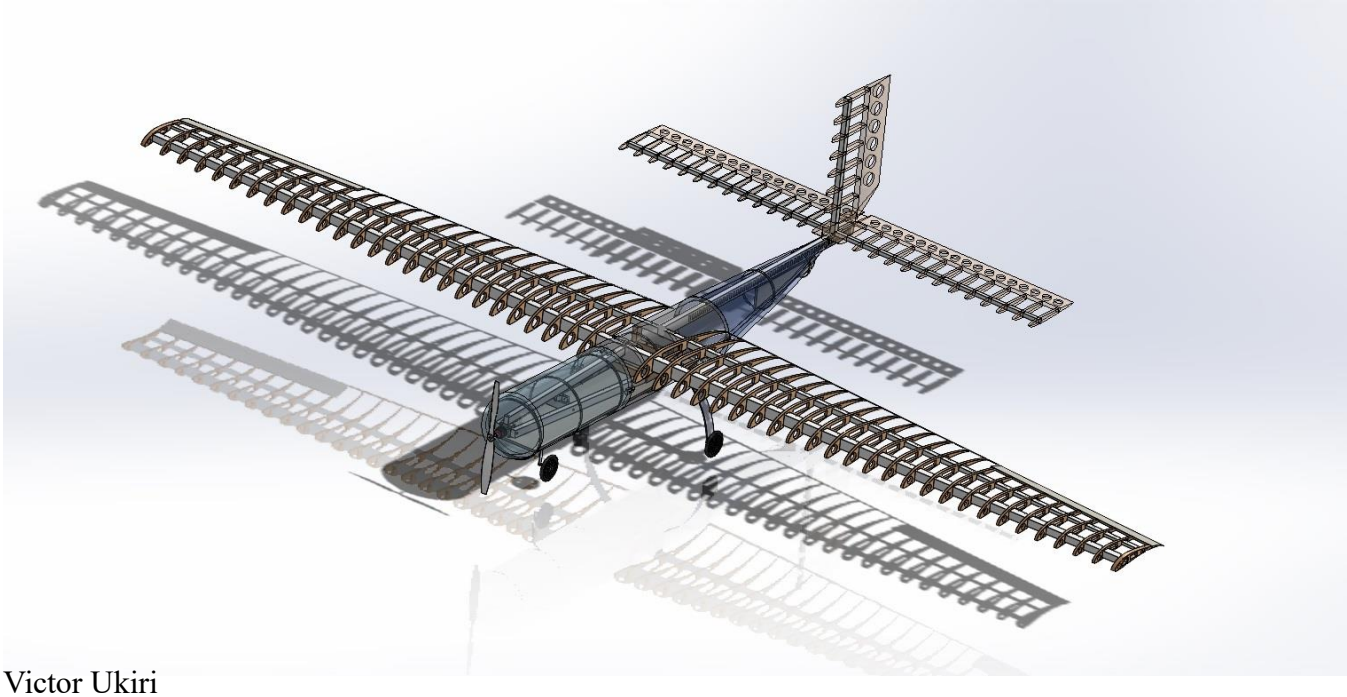
University of Windsor
Faculty of Engineering

LANCER AEROSPORTS



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Team 44

University of Windsor
Faculty of Engineering

APPENDIX A - STATEMENT OF COMPLIANCE

Certification of Qualification

Team Name	LANCER AEROSPORTS	Team Number	44
School	University of Windsor		
Faculty Advisor	Dr. J. Defoe		
Faculty Advisor's Email	jdefoe@uwindsor.ca		

Statement of Compliance

As faculty Adviser:

JD (Initial) I certify that the registered team members are enrolled in collegiate courses.

JD (Initial) I certify that this team has designed and constructed the radio-controlled aircraft in the past nine (9) months with the intention to use this aircraft in the **2023** SAE Aero Design competition, without direct assistance from professional engineers, R/C model experts, and/or related professionals.

JD (Initial) I certify that this year's Design Report has original content written by members of this year's team.

JD (Initial) I certify that all reused content have been properly referenced and is in compliance with the University's plagiarism and reuse policies.

JD (Initial) I certify that the team has used the Aero Design inspection checklist to inspect their aircraft before arrival at Technical Inspection and that the team will present this completed checklist, signed by the Faculty Advisor or Team Captain, to the inspectors before Technical Inspection begins.

Signature of Faculty Advisor	March 6, 2023
	Date
Signature of Team Captain	05/03/2023
	Date

Note: **A copy of this statement needs to be included in your Design Report as page 2 (Reference Section 4.3)**

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Table of Figures

1) Executive Summary

The following report discusses the Lancer Aerosports team's process for designing a Regular Class radio-controlled (RC) aircraft for the SAE Aero 2023 Competition. The mission of the aircraft is to carry the maximum weight possible within the given constraints of maximum aircraft weight, power, and takeoff distance, while also being able to make a turn and return to base within the allowed time limit. The mission comprises several constraint challenges that the team must address, including a 750W power limiter, a runway length of 100 feet, restrictions on materials such as carbon fiber, and minimum and maximum wingspan values of 120 and 216 inches, respectively.

The team's design consists of a maximum wingspan length of 214.66 inches built around the Selig S1223 airfoil. The design features a simple fuselage to minimize drag, ensuring that the fuselage is not a structurally critical member, and that all critical loads, including the payload, are supported by the wings. The payload is to be securely fastened inside the aircraft's fuselage, attached through penetration of the aircraft structure. The team used balsa wood and aluminum materials to reduce the empty weight while maximizing rigidity.

The competition requires that the aircraft’s performance must be predicted up to an acceptable standard. Through aerodynamic, structural, and Finite Element Analysis, the team has predicted takeoff weight, various calculations and schematics have been provided which form the basis on which the team has predicted the aircraft performance.]

2) Schedule Summary

Figure shows the timeline of the project up until the date of the competition (April 13th, 2023).

SAE AERO 2023 — GANTT-CHART

Company Name	Lancer Aerosports 2023									Date:	1/27/2023
Project Lead:	Victor Ukiri	Project Start:	11/10/2022							Revision	1
Pre-Competition				2022			2023				
TASK NAME	Progress	Start Date (DD/MM/YY)	End Date (DD/MM/YY)	Oct	Nov	Dec	Jan	Feb	Mar	Apr	
Phase 1 - Team Establishment	100%	11/10/2022	5/1/2023	Team Establishment							
Phase 2 - SAE AERO 2023 Registration	100%	26/10/2022	25/11/2022	SAE AERO 2023 Registration							
Phase 3 - Capstone Letter of Intent	100%	9/1/2023	3/2/2023				Capstone Letter of Intent				
Phase 4 - Fundraising	60%	13/01/2023	20/01/2023				Fundraising				
Phase 5 - Conceptual Design	100%	15/10/2022	15/01/2023		Conceptual Design						
Phase 6 - Preliminary Design	85%	06/01/2023	10/02/2023				Preliminary Design				
Phase 7 - Design Validation	0%	13/02/2023	10/03/2023					Design Validation			
Phase 8 - Manufacturing	0%	15/02/2023	07/04/2023					Manufacturing			
Phase 9 - Design Improvements	0%	03/03/2023	07/04/2023						Design Improvements		
Phase 10 - Testing	0%	24/03/2023	10/04/2022						Testing		
Phase 11 - Competition	0%	14/04/2022	14/04/2022							Competition	

Figure 1: GANTT Chart of Project Timeline

3) Constraint Analysis

3.1) Propulsion

The competition constraints pose quite a design task considering the aircraft weight and power constraints. Initially, the aircraft must be able to takeoff within 100 ft meaning the propulsion system must be able to provide the necessary thrust to reach the required takeoff speed, given that the power available to the aircraft is limited at 750 W. This means the total aircraft weight is predominantly determined by the power limitation. Maximizing the static thrust available is the best way to reduce the takeoff roll distance, reference 2-page 221provides an equation to estimate the relationship between Hp, and propeller diameter, for $\frac{1\,Hp}{750\,watt}$ a 22-inch two blade propeller was utilized**.** APC is a reliable supply of propellers, and their propeller performance charts were utilized in the analysis. We iterated to find the optimal RPM at our Hp available. We then developed the depreciation of thrust with forward speed graph shown below in Figure 2.

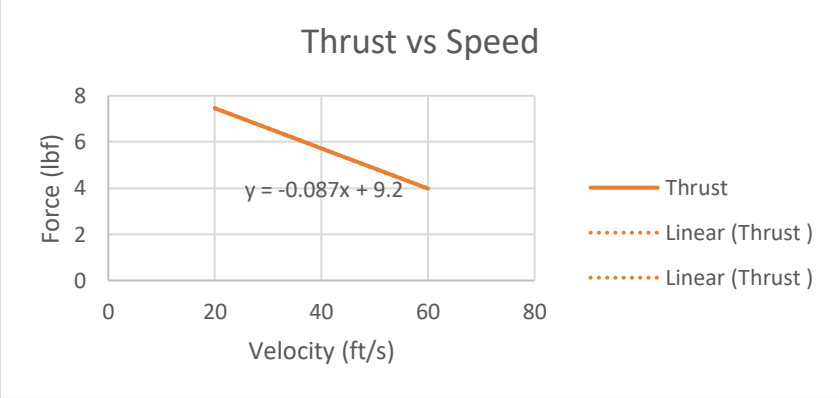

Figure 2: Dynamic Thrust vs Forward speed for APC 22x12WE

3.2) Scoring

The equation below is the scoring formula that will be used in the competition:

$$FFS = Final\ Flight\ Score = FS1 + FS2 + FS3 + WS$$

Equation 1: Flight Score Equation

Where $FS = Flight\ Score = 0.5 * W_{Payload} + PPB$ and

$$PPB = Payload\ Prediction\ Bonus = MAX\ (5 - (Wpayload - P)^2, 0)$$ and

$$WS = Wingspan\ Score = 2^{(1+b/5)}$$

Final flight score is based on the summation of 3 flight scores, and the wingspan score. The larger the wingspan and the heavier the payload, the higher the score. A bonus is applied to flight score based on the accuracy of the amount of payload carried to the payload prediction. As such, we have optimized both parameters within the limits for the design. However, as priority is on 3 successful flights, the aircraft has been designed for reliability ensuring the team would have the opportunity for multiple flights.

3.3) Cost Analysis

The breakdown of predicted costs and expenses for this project is shown below in Figure 3. Materials make 27% of the entire project. Many of the electronics and small parts have already been supplied from past years, which the team is recycling, such as brushless motors, ESCs, wiring, and servos. Due to the team's plane's large wingspan, there is large amount of surface area to cover, and most of the costs go towards building the wings and covering them with protective material. Accommodations, including the cost of the pilot, hotel fees, and other related fees make up 18% of the cost estimate. Transportation for the entire team is estimated to be our largest fee, between flight and petrol costs, making up 55% of the cost estimate.

Figure 3: Predicted Costs

Design Methodology

4.1) Airfoil Selection

Two airfoils were considered. From past years the Selig 1223 (S1223) airfoil was the main contender. This year the team proposed using the Wortman FX75-CL5-140. This was also a high lift low Reynolds number airfoil, that can perform well at low speeds. Airfoil analysis using xflr5 compared the performance of the two airfoils. Results show that the S1223 still outperformed the Wortman when comparing the max lifting coefficient CL. The S1223 airfoil had a max Cl = 2.26 at an angle of attack of 14°, shows the lift coefficient CL vs Angle of attack α (degrees)

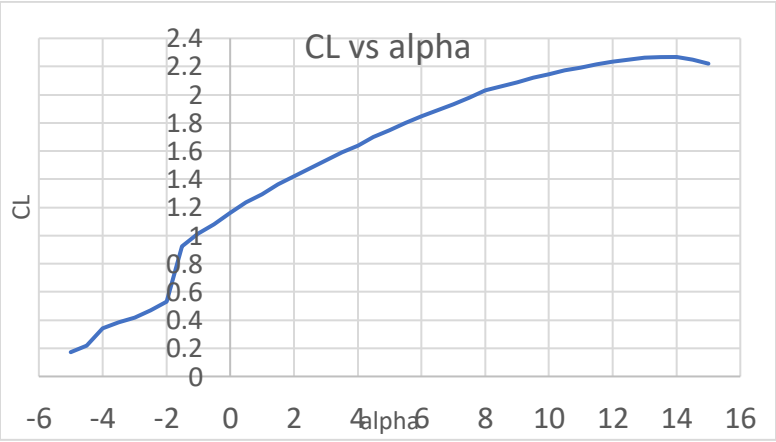

Figure 4: CL vs α data for s1223 from XFLR5

4.2) Lifting Surface

4.2.1) Wing

The team's approach to the wing design is to optimize the performance and maximum possible score. The wingspan was kept at the maximum length possible of 214.66 inches or 17.890 feet with an aspect ratio of 10, a root chord of 27.8 inches, mean chord of 22.27 inches, and taper ratio of 0.5. The aspect ratio was decided because any aspect ratio smaller than 10 would be detrimental to the performance because it would increase the induced drag. At this aspect ratio, the reference area for the wing is 32ft^2.The taper ratio of 0.5 provides an adequate lift distribution to an elliptical wing which is the most optimal design.

Wingspan [in]	Root Chord [in]	Mean Aerodynamic Chord [in]	Taper Ratio (λ)	Aspect Ratio (AR)
214.66	27.8	22.27	0.5	10.0

Table 1: Wing Dimensions

4.2.2) Horizontal and Vertical Tail

The horizontal and vertical were sized using the tail volume coefficient method proposed in reference 5. For general aviation aircraft, it suggests a CHT of 0.7 and CVT of 0.032 for single propeller engine aircraft. For the tail arm of the stabilizers was taken as 60 percent of the fuselage length as suggested by reference, 2 for front propeller aircraft. With this stabilizer dimensions were:

Horizontal tail

Table 1: Horizontal tail

Wingspan [in]	Root Chord [in]	Mean Aerodynamic Chord [in]	Taper Ratio (λ)	Aspect Ratio (AR)
90.11	12.87	12.87	1	7

Table 2: Vertical tail

Wingspan [in]	Root Chord [in]	Mean Aerodynamic Chord [in]	Taper Ratio (λ)	Aspect Ratio (AR)
31.98	15.99	15.99	1	2

Due to the large wingspan of the horizontal stabilizer the downwash effects had to be minimized. For this reason, a large aspect ratio design was utilized for the horizontal stabilizer.

The airfoil used for the stabilizers was a NACA 0009 symmetric airfoil due to its ease to manufacture and specifically when considering the vertical stabilizer, the lack of side force produced when the rudder is not deflected.

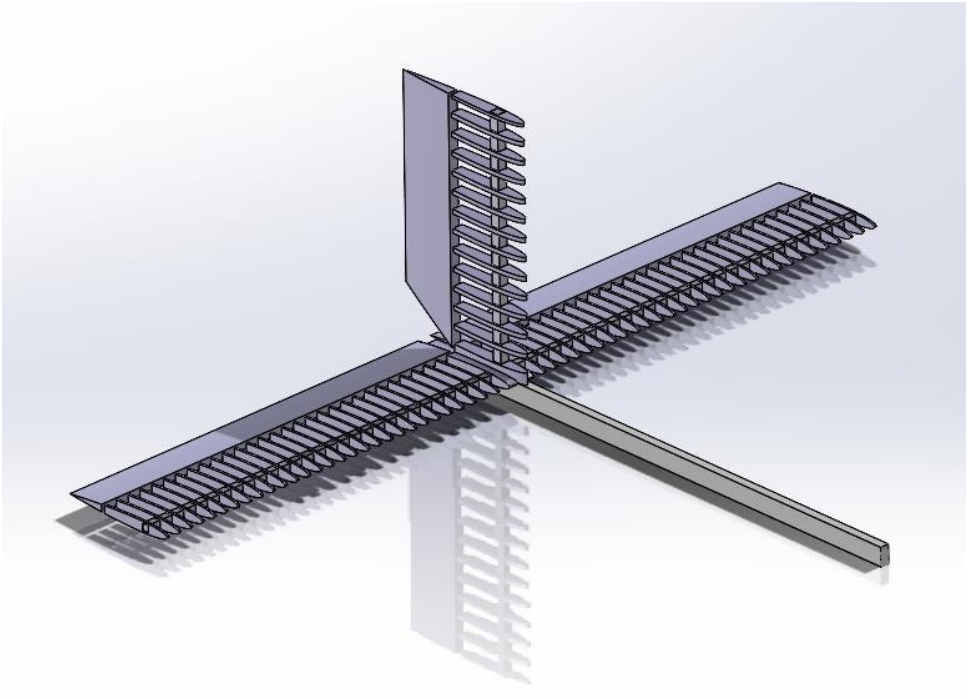

Figure 5: Horizontal Stabilizer, Vertical Stabilizer, and Tail Assembly

4.3) Control Surfaces

To reduce weight and still ensure rigidity. The control surfaces were made using a foam core with balsa wood sheet laying over it. A leading edge of wood on each control surfaces ensured it could be hinged without any concerns.

4.3.1) Ailerons

The aileron was sized by the team to ensure adequate bank capacity and hinged at 75 percent of the wing spar spanning 35 % of the wingspan. The ailerons were cut by RenShape Foam Board using a 3-axis CNC machine to achieve strict tolerances and reduce the manufacturing error.

4.3.2) Elevator

Elevators lined with a leading edge of balsa wood, hinged at 65 percent chord spanning 100 percent of the horizontal stabilizer wingspan.

4.3.3) Rudder

The rudder was fabricated from a polystyrene insulation foam using CNC machining. The material used was found to be flexible, therefore a balsawood frame was created to attach around.

The material used was quite flexible, so a balsawood frame was created to attach around the outside of the rudder and increase rigidity. Like the elevators the rudder is hinged at 65 percent of the chord and spans 100%.

4.4) Fuselage

The approach to designing fuselage was for it to act as an aerodynamic bag. It is to have no significant mass of its own and its only purpose is to make the overall design aerodynamic. The material used for the fuselage will be XPS foam and monocot wrapping. The fuselage length was determined utilizing the relationship between the aircrafts estimated gross takeoff weight (Reference 6 pg. 110) from this relationship the fuselage length was 8.6 ft. A 1 ft diameter was adequate to fit the payload and electronics in the fuselage.

4.5) Landing Gear

A tricycle landing gear configuration was utilized in our design. The weight distribution for the nose gear and main gear was 13% and 87% respectively. The material used for main landing gear was a commercially available carbon fibre frame.

4.6) Power

The heart of the airplane is a HRB 6S 22.2V 5000mAh LiPo battery. It is the most important piece of the airplane because it powers all the servos and electronics on the aircraft. It will be powering a Hacker A50-16L V4 brushless motor. With the constraints on the power available it was crucial to maximize the thrust we could generate.

4.7) Servos

There are six servos within the aircraft. One for each aileron on each side of the wings, one for the elevator, one for the rudder and another for the nose gear. They are tucked into the fuselage or wings and carefully wired through as to not expose too much to the outside to reduce drag. Hatches are carved into the frames of the fuselage and wings to allow ease of access to the electronics.

5) Loads & Environments

5.1) Wing

We considered phases of flight with highest load factor and multiplied the load by a safety factor of 3.

$$L = n * W_{TO} * 3$$

Equation 2 - Load Factor

Here, n represents the load factor in 45°

The reason for the highest safety factor was to consider the wing gust which would have a significant effect on the deflection of our wings.

5.2) Landing Gears

The load applied to each landing gear are as follows,

$$Main\ Gear\ Load = W_{TO} * 0.87 * 3$$

$$Nose\ Gear\ Load = W_{TO} * 0.13 * 3$$

This formula was developed accounting the weight distribution and the safety factor of 3 was used to ensure the landing gears do not buckle in hard landing conditions.

5.3) Environmental Considerations & Assumptions

The performance of the airplane is very dependant on the environmental condition of the competition day. Factors such as air pressure, temperature, wind speed and air density change the performance. An average of high, low and median temperature was taken for Fort Worth, Texas. Weather reports for the

month of April, between the years 2015 and 2022, were used to generate this average and is show n in the table below.

Factor	Average Values
Temperature F	64.9 F
Pressure	30.1 inHg
Windspeed	16.39 ph

6) Structural Loads and Critical Margins

6.1) Wing

The strength of the wing depends on the strength of the spar that supports it. The spars were designed as Tapered hollow beams. The cross section of the beams was decided by considering the maximum allowable deflection of the wing which we the team set as 4°. The cantilever deflection under a distributed load was utilized in analyzing the structural integrity of the design. The deflection helped to find the second moment of inertia of the beam which we utilized to find the dimensions of the cross-section. To verify the results of the numerical analysis Finite Element analysis was conducted various times to on the wing design applying loads using an elliptical distribution along the span of the wing. After performing analysis, the material selected for spars were spruce wood and balsawood for ribs. Based on analysis, tapered beams were optimal for the requirements. Below are some results of the FEA study conducted in SolidWorks.

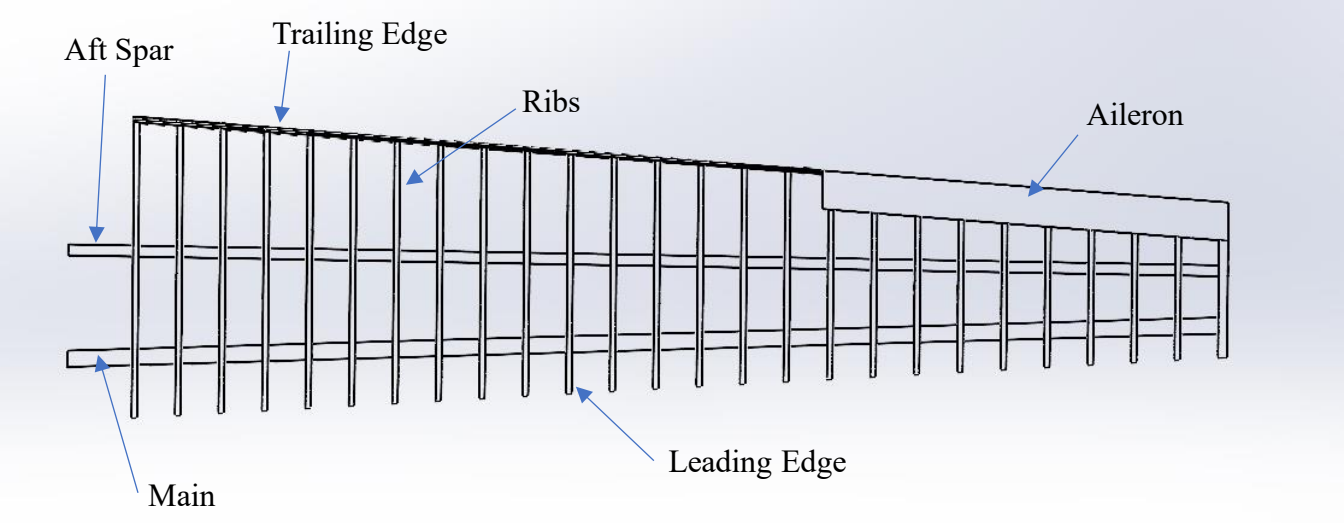

Figure 6: 2D Projection of Wing Assembly

Table 3: Material properties

Parameters/ Constraints	**Values**
Maximum allowable bending stress	(Matweb, 2023) 3300 psi
Max allowable deflection	7.50 in

FEA Results (Wing)	**Values**
Maximum bending stress	3280 psi
Max deflection	6.92 in
Mass of the one-side of the wing (without monocot)	3.99 lb

As we can see in above table, the FEA results are depicting the values within the constraints. Below are the spars dimensions devised from analysis:

Table 4: Main Spar Dimensions

Main Spar Dimensions	
Root	2.45 x 1 in
Tip	1.5 x 1 in
Thickness	0.2 in

Table 5: Second spar dimensions

Second Spar Dimensions	
Root	1.25 x 1 in

Tip	0.5 x 1 in
Thickness	0.2 in

Table 7: Second spar dimensions

FEA Results (Tail)	**Values**
Maximum bending stress	742 psi
Max deflection	1.39 in

Based on the FEA analysis of Tail, the dimensions found optimal for the designs are mentioned below.

Table 8: Vertical and Horizontal Stabilizer spar dimensions

Name	Dimensions	Length	Material
Vertical Stabilizers Spar	0.75 x 0.75 in	27.17in	Spruce wood
Horizontal Stabilizer Spar	1 x 1 in	90.12	Spruce wood
Length of Tail boom	2 x 1 in	46.96 in	Spruce wood

Table 9: Finite Element Analysis of Payload Holder

FEA Results (Payload Holder)	**Values**
Maximum bending stress	2465.64 psi
Max deflection	0.06 in

To determine the payload holder's ability to withstand the weight of the payload, a load 3 times the weight of the payload was applied, and FEA analysis conducted. The analysis revealed that the maximum bending stress reached was 2465.64 psi, which is lower than the calculated yield stress of the material. This indicates that the payload holder can withstand 3 times the weight of the payload without significant deflection.

Figure 10: Finite Element Analysis of Nose Gear

FEA Results (Nose Gear)	**Values**
Maximum bending stress	232.06 psi
Max deflection	0.03 in

To determine the nose gear's ability to endure the motor's thrust and the weight applied to it, a finite element analysis was conducted. The analysis revealed that the maximum bending stress reached was 232.06 psi, which is lower than the calculated yield stress of the material. This indicates that the nose gear can handle both the thrust force and the weight force.

Main gear loading

The team had the main landing gear available. A load of 92 lb was physically applied to main landing gear. With less than 1 in deflection it was determined the main gears could handle load from the aircraft weight adequately.

7) Manufacturing and Joints

The image below shows the connection between the spars. Aluminum sheet is being curved to join the spars. The hex bolts and nuts are used to join the spars together. M12X75 and M21X65 bolts were utilized for joining between spars.

7.1.1) Fuselage Connection

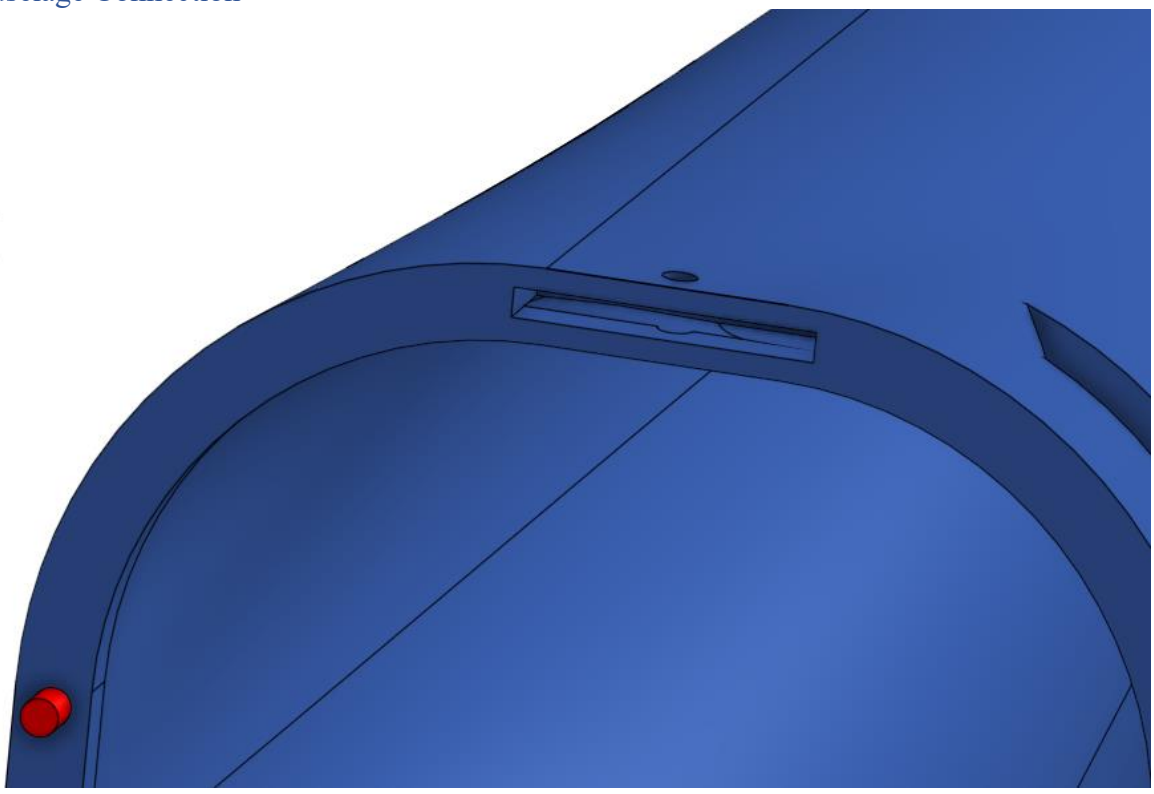

Figure 7: Dowel locator in fuselage connection

To connect the different sections of the fuselage, a combination of two types of connectors will be used. Firstly, two locater dowels will be positioned on either side of each fuselage section. These dowels are

cylindrical pins that fit into matching holes in the adjacent section, helping to accurately align and position the sections.

Secondly, a mortise and tenon linkage will be installed on the top and bottom of each fuselage section. This type of joint provides additional strength and stability to the connection. The mortise, which is a rectangular slot, is cut into the top and bottom of each fuselage section, while the tenon, which is a projecting tongue, is fitted into the mortise of the adjacent section. A hole will then be drilled through the mortise and tenon linkage, enabling a bolt and nut to securely fasten the sections together.

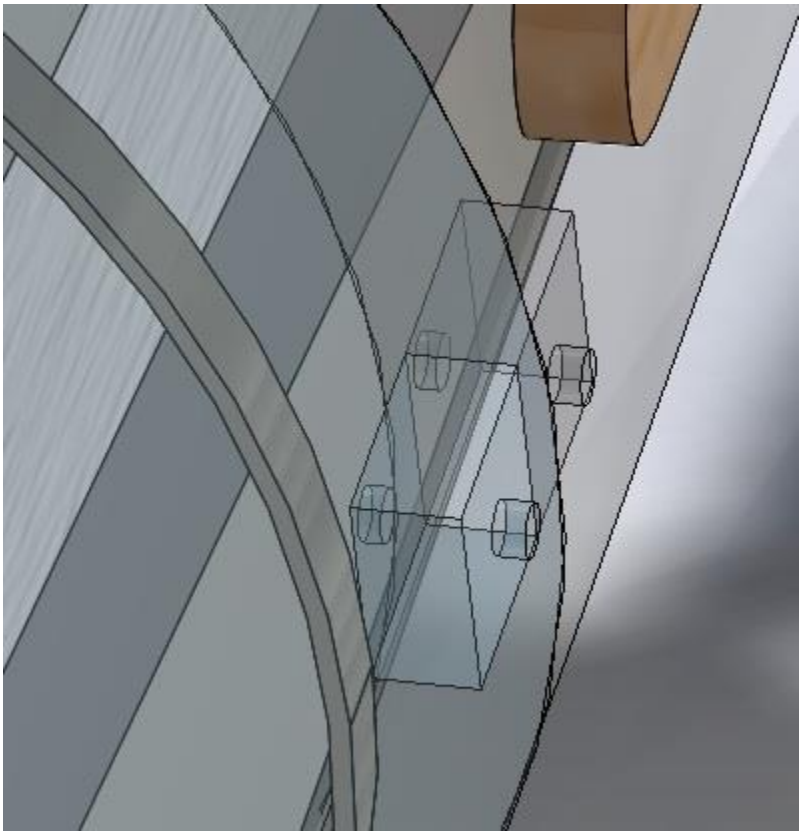

Figure 8: Frame Connection Mechanism for fuselage sections

7.1.2) Payload Holder

The Payload Holder will consist of four key components: two straight spars positioned across the middle, and two U-shaped spars that will attach to them. To secure these spars, each end of the two straight spars will be fitted with two locator dowels inserted into matching holes. In addition, the U-shaped spars will have two locator holes on either end, making it easier to drill screws into the straight

spars and use nuts to secure them in place. At each of the four corners of the U-shaped spars, a hole locator will be positioned to allow for attachment of the main and aft spars to the payload holder. Screws will be used to secure the spars in place, and nuts will then be used to ensure they remain firmly fixed. Two locator holes will be positioned on the front straight spar at a specific distance from the center. These holes will facilitate the connection of the landing gear to the payload holder using screws, which will then be secured in place with nuts.

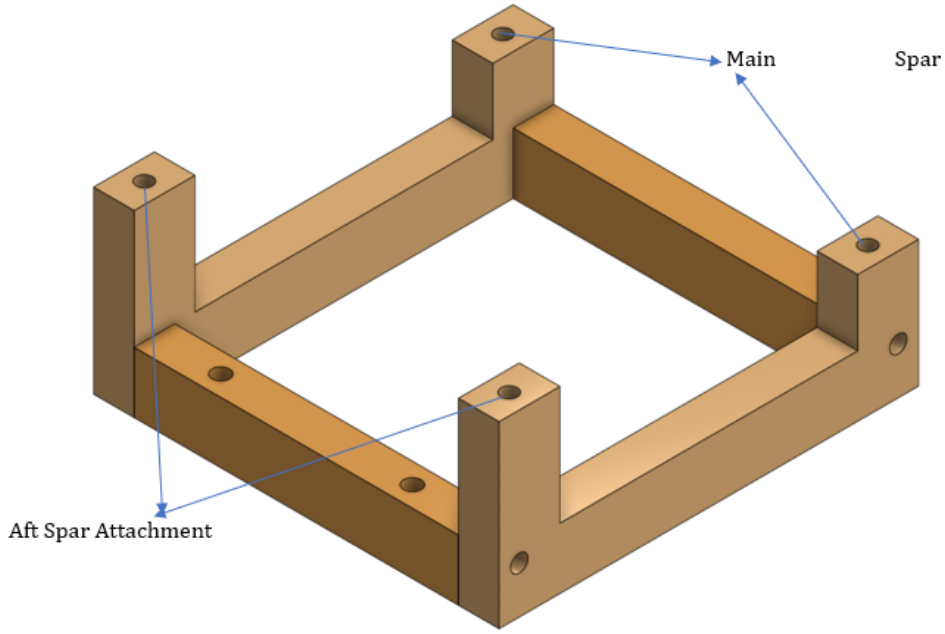

Figure 9: Payload attachment

8) Analysis

8.1) Performance Analysis

8.1.1) 3D drag analysis.

Computing the overall drag equation for the aircraft was crucial to determine the aircraft performance.

Profile drags.

The profile drag was determined referencing methods used in Lee Nicholas white paper for the major aircraft components, wings, stabilizers, fuselage, and landing gear, the overall profile drag $Cd,0 = 0.0401$

Induced drag.

The wingspan of the aircraft is large so tapering the aircrafts wings was crucial in reducing the induced drag. Utilizing Oswald efficiency in place of efficiency factor graph used in Gudmundsson (2013) based solely on taper ratio and Aspect ratio improved the accuracy of the induced drag factor, K, as aircraft components interference is considered.

Viscous drag due to lift.

The viscous drag effects due to lift is significant for estimating the aircrafts overall drag especially when flying at low speeds. As done in Nicolai (2002), the $C_L \; vs \; (C_L - C_{Lmin})^2$ was graphed in excel and the slope was determined as 0.0172.

8.1.2) Phases of flight

Takeoff ground roll

The requirement of the competition is that the airplane must be airborne within 100 ft takeoff distance. To ensure our aircraft achieves this, the wing reference area was maximized with the wingspan constraint and taper ratio constraint.

From (Nicolai & Carichner, 2010), assuming the acceleration as constant the take-off ground roll distance is calculated as,

$$S_g = \frac{0.5 * {V_{TO}}^2}{amean_{70.7}} \text{ , } where \; amean_{70.7} \; is \; the \; acceleration \; at \; 70.7\% \; V_{TO}$$

Equation 3: Ground Roll

The ground roll was modelled using a MATLAB that first determined the required takeoff velocity and then calculated the mean acceleration at 70.7% of V_{TO}. With the information, we available from the propeller vendor charts the dynamic thrust at 70.7% was determined and used in the calculations.

Calculating required takeoff velocity and thrust as (Nicolai & Carichner, 2022):

$$V_{TO} = 1.2 \, V_{stall} = 1.2 \sqrt{\frac{W_{TO}}{S_{ref}} \frac{2}{\rho \, C_{L_{max}}}}$$

Equation 4: Required Takeoff Speed

C_{Lmax} (Figure 4: CL vs α data for s1223 from XFLR5) (Figure 5, C_LVs α data from s1223 from XFLR5) (for takeoff is taken to be at 80% of maximum available C_{Lmax} (Nicolai & Carichner, 2022) , the takeoff weight is measured to be 35 lbs, reference area as 32 ft^2, and the density to be as 0.002377 $slug/ft^3$ After performing the calculations, the Sg was 78 ft, which was within competition constraints.

Climb, Cruise, Bank

The climb cruise and bank segment were simulated in AVL. The necessary climb angle to reach our desired altitude of 100ft at the initial start before determined as 10°. For the cruise segment, we utilized AVL to determine the optimal cruise CL that met our flight requirements, the incidence angle of the wing was set to ensure the angle between the fuselage reference line, and the ground line remained as close to zero during cruise. For banked turns we utilized AVL simulations constraining the velocity as the constant cruise velocity of 38 ft/s for each bank, this enabled us to get the necessary control surface deflections and CL to perform each banking maneuver.

Stability Analysis

To ensure dynamic and static stability in all flight modes AVL's Eigen mode analysis was utilized. Meeting the criteria for static stability during all phases of flight the aircraft requires the CG to be between the AC and the neutral point. After confirming the static stability criterion, the dynamic stability of the aircraft was analysed. To be dynamically stable the aircraft must show a tendency to correct itself to its trim state after a perturbation.

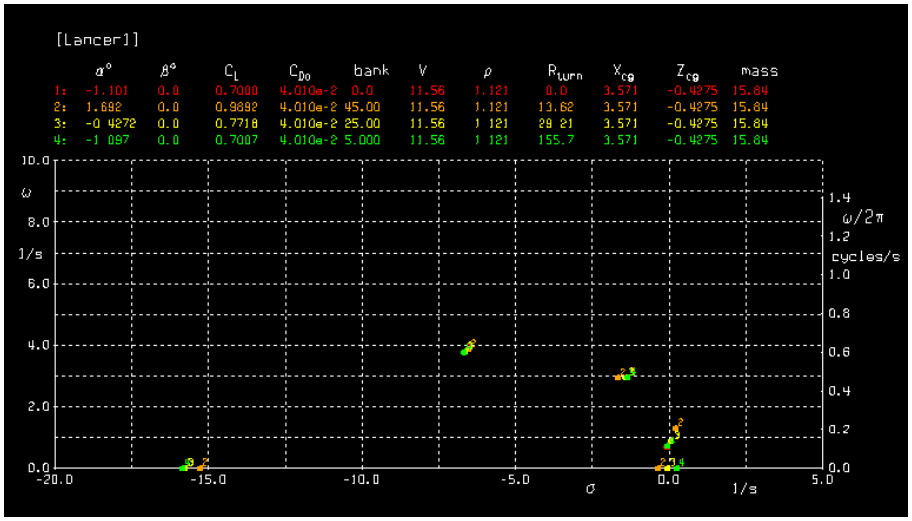

Figure 10: AVL eigen mode plots

The aircraft was stable in all modes except Spiral mode and Phugoid mode. For spiral mode, it is critically un-damped and unstable, but it takes a long-time post perturbation for the aircraft to lose control, the real sigma values were also much less than 1 for all flight phases which reassured adequate time to control the unstable spiral. The Phugoid mode was stable for climb and cruise but for bank angles greater than 5 °, it was unstable. For the unstable Phugoid modes during banking maneuvers we had to ensure the real values of the modes were <1 and time period with the unstable modes were much larger than the time it takes to perform the turns. After confirming this in the Eigen mode animation, we were confident the aircraft would not go out of control during a bank and meets the dynamic stability criterion for all other phases of flight.

9) Payload Prediction

The payload prediction curve considers the effects of a change in density, dynamic thrust, lift and, drag etc. These factors have a direct effect on our payload carrying capacity. Modelling the ground roll following steps outlined in (Nicolai & Carichner, 2010), the variation in densities at different altitudes affected the required takeoff speed causing it to increase, while decreasing the mean acceleration. The equation representing the linearized payload prediction curve is,

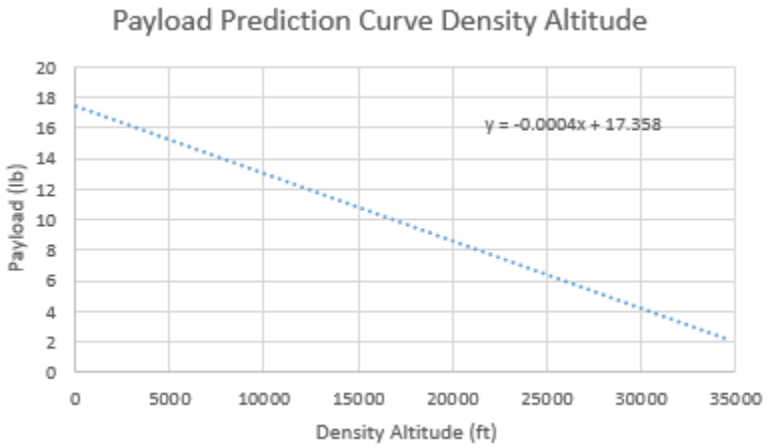
Payload Prediction Curve Density Altitude
y = -0.0004x + 17.358
Payload (lb)
20
18
16
14
12
10
8
6
4
2
0
0
5000
10000
15000
20000
25000
30000
35000
Density Altitude (ft)

Table of Acronyms

$C_{T;}$ Coefficient of thrust $C_{L;}$ Coefficient of lift $C_{D;}$ Coefficient of drag

$C_{P;}$Coefficient of power

FS; Flight score.

Cd,0; Profile drag

P: Number of seated passengers carried on a flight C; Luggage weight (lbs.)

E; Number of empty seats

N; Total number of flight rounds during competition α; Angle of attack

ρ; Density in at sea level in imperial units

AR; *Aspect ratio*

λ; *Taper ratio*

HT*; Horizontal tail*

VT; *Vertical tail*

VHT; *Horizontal tail volume* VVT; *Vertical tail volume* CG; *Center of gravity*

M.A.C.; *Mean aerodynamic chord* ESC; *Electronic speed control Kp; Propeller sizing constant PBHP; Engine Power (in BHP)*

D; *Diameter*

n; Loading factor at bank angle of 45°

L; Lift force

CNC; Computer Numerical Control

V_{LOF}; Take-off velocity

S_g; Desired take-off roll distance

AC- Aerodynamic Centre

V_{TO}- Takeoff Velocity

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[4] Nicolai, L. (2010). *Fundamentals of aircraft and Airship Design: Aircraft Design; volume I*.

[5] Dr. Leland M. Nicolai, Estimating R/C Model Aerodynamics and Performance. June 2002.

[6] USA.com, "Fort Worth, TX Weather," n.d. [Online]. Available: http://www.usa.com/fort-worth-tx-weather.htm.

Appendix

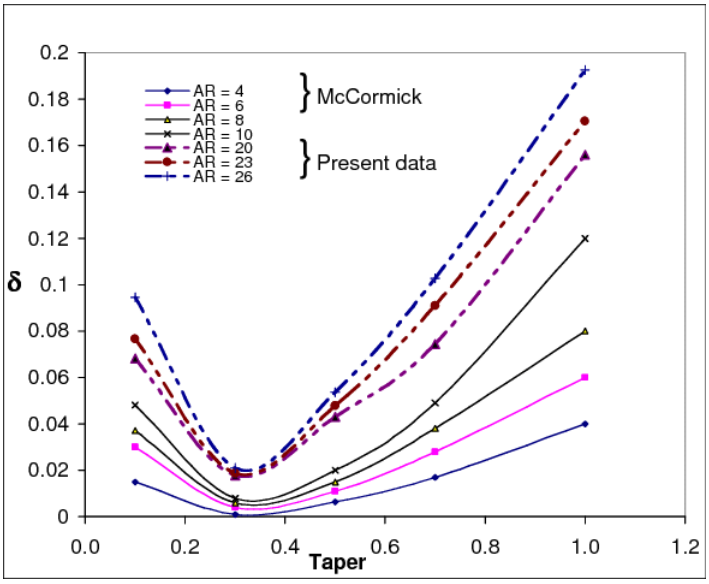

Figure 11: Taper

Equations (10.23–10.25) provide an estimate of the propeller diameter as a function of horsepower (Ref. 28). The propeller diameters obtained from these equations should be compared to the maximum diameters obtained from tip-speed considerations, and the smaller of the two values used for initial layout.

$$\text{Two blade:} \quad d = 22\sqrt[4]{\text{Hp}} \qquad (10.23)$$

$$\text{Three blade:} \quad d = 18\sqrt[4]{\text{Hp}} \qquad (10.24)$$

Propellor Vendor charts

4000 RPM

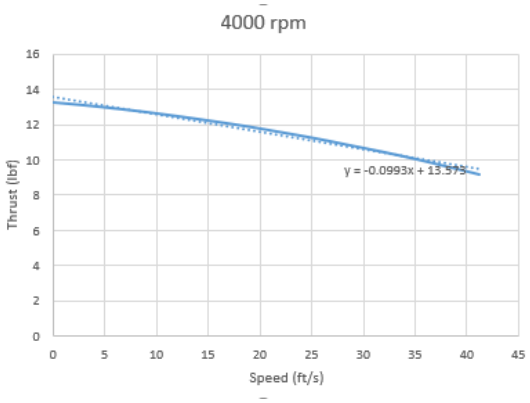

Figure 12: Thrust vs Speed at 4000rpm.

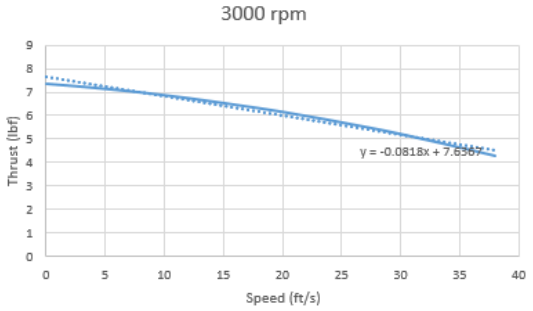

Figure 13: Thrust vs Speed at 3000rpm.

Iteration process for 3300 rpm and static thrust available

Power	Thrust		Power available	
0.457	5.1		0.75	
1.191	10.48			
			dynamic thrust	7.24760218
Power	Thrust		Power available	
0.68	7.374		1	
1.71	13.26			
			Static thrust	9.202660194
Power	Rpm		Power available	
0.68	3000		1	
1.71	4000			
			Operating rpm	3310.679612
Rpm	Slope		RPM	
3000	-0.082		3310.679612	
4000	-0.099			
			Slope of Thrust	-0.087236893

General aviation (one-engine propeller)	0.7	0.032

Tail volume coefficients from Reference 5

```
% optimal fuselage length
clear;clc;
w0= 35;

%Historical values for a and c
a=3.68;
c=0.23;

%Fuselage length formula
%f_length=a*w0^c;
f_length=8.5965;

%fprintf('Fuselage length is %f ft\n',f_length)   %legth display change in ft

%flight velocity  (keep constant when you want to see how Fl affects drag
%on stabilizers)
CL_max=2.025; %airfoil dependent from excel file
%i=20;
rho=0.0023769;
mu=3.677*10^(-7);
ar_w=10;


wings = [];
flightscore = [];
emptyw = [];
a= zeros([1 7]);
 for i = 25:1:32
     wingsp = ((i*ar_w)*0.5);
     disp(wingsp)
     fv=1.2*(2*w0/((i)*rho*0.8*CL_max))^(0.5); %Factor in 1.2 from airship design
     %fv=40;
     q=0.5*rho*(fv^2);   %dynamic pressureFrom fundamentals of airship design page 73

%fprintf('Take-off velocity is %f ft/s\n',fv)

%Drag portion

%fuselage drag
    %d_fuse=0.33333*f_length; %FR assumed to be 3
     d_fuse=1;
%d_fuse=f_length/3;
     Sref_fuse=6.816; %Reference area of fuselage calculated from planform
     Swet_fuse=28;
     FR=f_length/d_fuse; %Fineness ratio
%FR=3;
     FFf=1+(60/((FR)^(3)))+0.0025*FR; %Form factor
     re_fuse=rho*fv*(f_length/mu);   %Fuselage reynolds number
     cft_fuse=0.074/((re_fuse)^(0.2)); %Skin friction of fuselage
     Cdmin_fuse=FFf*cft_fuse*Swet_fuse/1;
%fprintf('cdmin fuse is %f\n',Cdmin_fuse)
```

```
%Wing drag

    tr_w=0.5;   %Taper ratio
    bw_w=(ar_w*(1))^0.5;   %Wingspan based on constant aspect ratio and wingarea
%disp(bw_w)
%disp(bw_w)
    Swet_wing=2*i; %Double the reference wing area
    cr_w=(2*i/bw_w)/(1+tr_w); %root chord wing
    mac_w=(2/3)*cr_w*((1+tr_w+tr_w^2)/(1+tr_w)); %Mac wing
    t_c=0.121; %Airfoil dependent
    FFw=(1+2*(t_c)+100*((t_c)^4))*1.05; %Form factor of wing check for Airfoil dependence
    re_w=rho*fv*mac_w/mu; %Reynolds number of wing based on chord
    %disp(re_w)
      %Add laminar cft
    cft_wing=0.074/(re_w^0.2);  %Turbulent skin friction wing (check for turbulence
    Cdmin_wing=FFw*cft_wing*Swet_wing/i;
    %fprintf('Cdmin wing %f\n',Cdmin_wing)

    %Horizontal stabilizer drag

    ar_h=7;      %Aspect ratio
    tr_h=1;      %Taper ratio
    cht=0.7;       %Volume coefficient
    lht=0.6*f_length;  %Tail arm
    Sht=cht*mac_w*i/lht;  %Reference area
    Swet_h=2*Sht;
    bw_h=(ar_h*(Sht))^0.5;
    %cr_h=(2*Sht/bw_h)/(1+tr_h); %root chord horizontal stabilizer
    cr_h=bw_h/ar_h;
    %disp(cr_h)
    mac_h=(2/3)*cr_h*((1+tr_h+tr_h^2)/(1+tr_h)); %Mac wing
    t_ch=0.09; %Airfoil
    FFh=(1+1.2*(t_ch)+100*((t_ch)^4))*1.05; %Form factor
    re_h=rho*fv*mac_h/mu;  %Reynolds number of wing
    %disp(re_h)
    cfl_h=1.328/(re_h^0.5);  %laminar
    Cdmin_h=FFh*cfl_h*Swet_h/Sht;
    %fprintf('Cdmin h %f\n',Cdmin_h)

    %Vertical Stabilizer drag

    ar_v=2;                    %  Vertical tail aspect rati0
    tr_v=1;                   %Taper ratio
    cvt=0.032;                    %Tail volu,e coefficient
    lvt=lht;            %Tail arm
    Svt=cvt*bw_w*i/lvt;           %Reference area
    Swet_v=2*Svt;                 %Wetted area
    bw_v=(ar_v*(Svt))^0.5;        %Wing span
    %cr_v=(2*Svt/bw_v)/(1+tr_v); %root chord horizontal stabilizer
    cr_v=bw_v/ar_v;
    %disp(cr_v)
    mac_v=(2/3)*cr_v*((1+tr_v+tr_v^2)/(1+tr_v)); %Mac wing
    t_cv=0.09; %Airfoil dependent tc ratio
    FFv=(1+1.2*(t_cv)+100*((t_cv)^4))*1.05; %Some airfoil dependent values are present
    re_v=rho*fv*mac_v/mu; %Stabilizer reynolds number
    %disp(re_v)
    cfl_v=1.328/(re_v^0.5);  %laminar
    Cdmin_v=FFv*cfl_v*Swet_v/Svt;
    %fprintf('Cdmin v %f\n',Cdmin_v)
```

```
%Landing gear drag
Cd_gear=1.01;  %single strut based off white paper reference 3
g_area=(1.76*4.15)/(12^2); %frontal area of landing gear
Cdmin_gear=(3*1.01*g_area)/(i);


%Engine drag
cd_engine=0.25; %from white paper, for cowled engine
e_area=22*1.38/144; %frontal area of engine based on propeller area, hud diameter of 1.38 in, prop diameter of 22
cdmin_engine=cd_engine*e_area/i;

%Total Cdmin
Cdmin_total=Cdmin_fuse+Cdmin_wing+Cdmin_h+Cdmin_v+Cdmin_gear+cdmin_engine;
%fprintf('Total Cdmin is %f\n',Cdmin_total)

%Complete drag equation

%Oswald efficiency factor
e0=1.78*(1-0.045*(ar_w^0.68))-0.64;
K=1/(pi()*ar_w*e0); %Induced drag factor utilizing oswald efficiency factor

g=32.22;   %acceleration due to gravity in imperial units
KK=0.0172;  %from airfoil excel graph
CL=0.7; %airfoil dependent correct for 3d finite;
Cl_min=1.2382; %airfoil dependent
Clg=0.7; %3d wing dependent


CD=Cdmin_total+(K*(CL^2))+KK*(CL-Cl_min)^2; %not yet added landing gear effect
CDg=Cdmin_total+(K*(CLg^2))+KK*(CLg-Cl_min)^2; % CD during ground roll
% AIRCRAFT PERFORMANCE (Level unaccelarated flight)
Power_required=(CD*(w0/CL)*sqrt((2*w0)/(rho*CL*i)))/550;  %Fundamentals of airship page 73
disp(Power_required)

%Thrust_required=CD*q*i;
Thrust_required=(Cdmin_total*q*i)+((K*(w0^2))/(q*i));
disp(Thrust_required)
%fprintf('Total CD is %f\n',CD)
Cl_needed=w0/(q*i);

%Take off Roll

T=7.46; %Dynamic thrust at 70.7 percent Vt0
prop_efficiency=0.6;
hp=1.006;

%T=(550*prop_efficiency*hp)/(2*(sqrt(w0/(i*rho*0.8*Cl_max))));
Fc=0.03;     %Formula a=(g/W)[(T-D)-Fc(W-L)]; Sg=(Fv^2)/(2amean)
L=(0.5)*(rho)*((0.707*fv)^2)*i*Clg;
area_drag=pi()*((d_fuse^2)/4);
%D=(0.5)*rho*((0.707*fv)^2)*area_drag*(CD); %make change use frontal fuselage area
D=(0.5)*rho*((0.707*fv)^2)*i*(CDg);

amean=(g/w0)*((T-D)-Fc*(w0-L));

Sg=(fv^2)/(2*amean);
%fprintf('The take-off ground roll is %f ft\n',Sg)
```

```
%Rotation distance
Sr=0.5*fv;
%Transition distance, Str
R=(fv^2)/((0.15)*g); %rotation distance
climb_rate=fv*((T-D)/w0); %Rate of climb
theta_cl=asin(((T-D)/w0));
%disp(theta_cl)
Str=R*sin(theta_cl); %transition distance
%fprintf('The transition distance is %f ft\n',Str)

%Climb distance
Htr=10;
Scl=(35-Htr)/tan(theta_cl);
climb_angle=theta_cl*180/pi;
%fprintf('Climb distance is %f ft\n',Scl)

%Total take off distance
takeoffdist_total=Sg+Sr;
%fprintf('Total takeoff distance is %f ft\n',takeoffdist_total)

%Time during take off
g_time=fv/amean;  %Ground roll time
tr_time=(Str+(Scl/cos(theta_cl)))/(fv); %Transition distance time
total_time=g_time+tr_time;
%fprintf("Total time required for take-off is %f seconds\n",total_time)

%Landing Analysis (time req clear 50ft, free roll then brake to stop)
%NOTE Vs equal fv as w0 does not change and no flaps are used
Vs=fv/1.2; %Stall speed
V50=1.3*Vs;
Vtd=1.15*Vs;

%Air distance,Sa
Sa=(L/D)*((((V50^2)-(Vtd^2))/(2*g))+50);
%fprintf('Air distance is %f ft\n',Sa)

%Free roll distance, Sfr
L_over_d=L/D;
%Braking distance,Sb
CLG=Clg; %airfoil dependent and landing confif dependent
Sb=(w0/(g*Fc*rho*i*((CD/Fc)-CLG)))*log(1+(rho/2)*(i/w0)*((CD/Fc)-CLG)*(Vtd^2));
%fprintf('Braking distance is %f ft\n',Sb)
%empty weight calculation
    ref_density = 1.967;
    aa_w = 0.2207; %airfoil dependent
    aa_hw = 0.035019; %airfoil dependent
    aa_vw = 0.094546; %airfoil dependent
    wvol = aa_w*bw_w;
    hwvol = aa_hw*bw_h;
    vwvol = aa_vw*bw_v;
    m_w = ref_density*wvol;
    m_hw = ref_density*hwvol;
    m_vw = ref_density*vwvol;
    wm_total = (m_vw + m_w + m_hw)*1.2;
    payload = w0 - 12 - wm_total;
    if (wingsp <= 18) && (takeoffdist_total <= 100) && (Sb <= 400)
        flightscore(i)= 2*(1+(wingsp/5)) + payload/2;
        wings(i) = wingsp;
        emptyw(i) = wm_total;
        emptyw = wm_total;
        disp("Constraint was met")
```

```
        disp(T)
        fprintf("Wing area is %.2f ft^2\n", i)
        fprintf("Take-off roll distance is  %.2f ft\n", Sg)
        fprintf("Total takeoff distance is %.2f ft\n", takeoffdist_total)
        fprintf("Breaking distance %.2f ft\n ", Sb)
        fprintf("Empty weight is %.2f lb\n ",emptyw)
    else
        disp("Constraint was not met")
        disp(T)
        fprintf("Wing area is %.2f ft^2\n", i)
        fprintf("Take-off roll distance is  %.2f ft\n", Sg)
        fprintf("Total takeoff distance is %.2f ft\n", takeoffdist_total)
        fprintf("Breaking distance %.2f ft\n ", Sb)
        fprintf("Empty weight is %.2f lb\n ",emptyw)
    end
end
for j = 1:length(emptyw)
    if  emptyw(j) ~= 0 %emptyw ~= 0
        fprintf("Empty weight is %.2f lb\n  Wing Span %.2f ft\n  Flightscore is %.2f", emptyw(j),wingsp(j),flightscore(i));
        %fprintf("Empty weight is %.2f lb\n  Wing Span %.2f ft\n  Flightscore is %.2f", emptyw,wingsp,flightscore);
    end
end
```

Figure 14: MATLAB Code

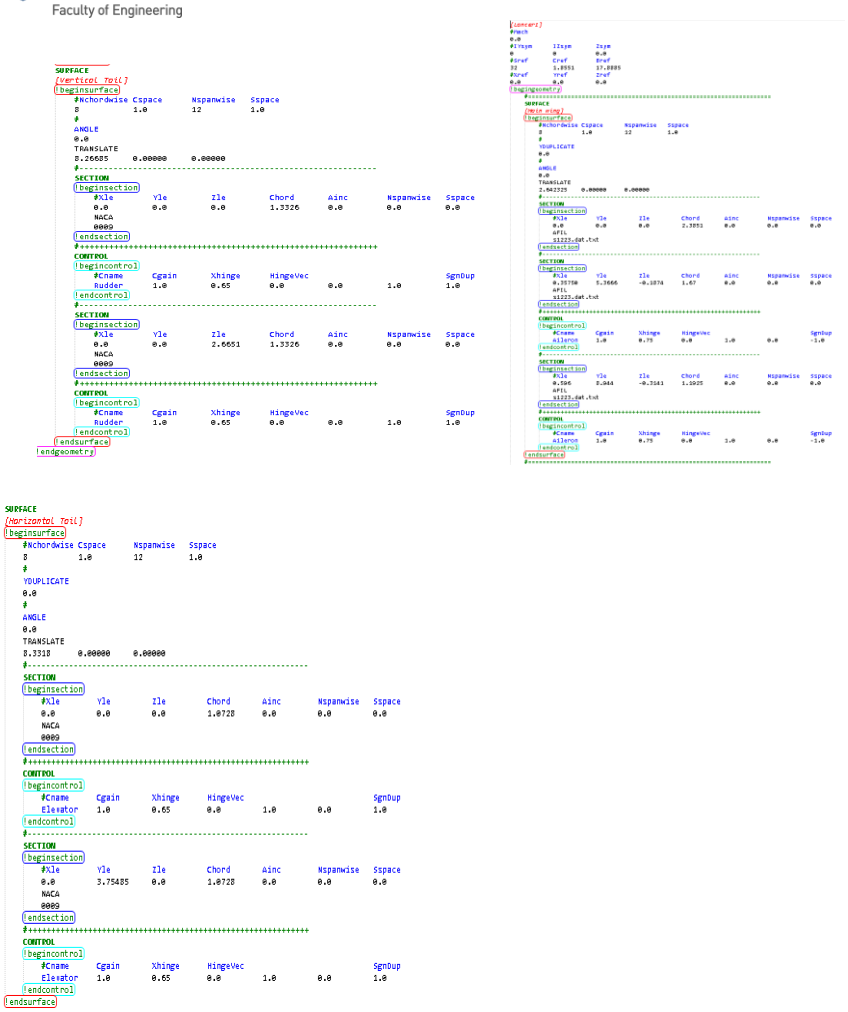

Figure 15: AVL Simulation for wing, horizontal tail, and vertical tail

```
Run case  1:  TrimCruise

alpha         ->  CL          =  0.700000
beta          ->  beta        =   0.00000
pb/2V         ->  pb/2V       =   0.00000
qc/2V         ->  qc/2V       =   0.00000
rb/2V         ->  rb/2V       =   0.00000
Aileron       ->  Cl roll mom =   0.00000
Elevator      ->  Cm pitchmom =   0.00000
Rudder        ->  Rudder      =   0.00000

alpha      =  -1.10122     deg
beta       =   0.00000     deg
pb/2V      = -0.970869E-15
qc/2V      =   0.00000
rb/2V      =  0.319761E-16
CL         =   0.700000
CDo        =  0.401000E-01
bank       =   0.00000     deg
elevation  =   0.00000     deg
heading    =   0.00000     deg
Mach       =   0.00000
velocity   =   11.5556     m/s
density    =   1.12060     kg/m^3
grav.acc.  =   9.81000     m/s^2
turn_rad.  =   0.00000     m
load_fac.  =   1.00000
X_cg       =   3.57127     Lunit
Y_cg       =  0.120516E-07 Lunit
Z_cg       = -0.427481     Lunit
mass       =   15.8399     kg
Ixx        =   6.88271     kg-m^2
Iyy        =   3.91601     kg-m^2
Izz        =   9.87342     kg-m^2
Ixy        =  0.145612     kg-m^2
Iyz        = -0.758136E-08 kg-m^2
Izx        = -0.391923     kg-m^2
visc CL_a  =   0.00000
visc CL_u  =   0.00000
visc CM_a  =   0.00000
visc CM_u  =   0.00000
```

```
-----------------------------------------------
Run case  2:  45deg bank

alpha         ->  CL          =  0.989201
beta          ->  beta        =   0.00000
pb/2V         ->  pb/2V       =   0.00000
qc/2V         ->  qc/2V       =  0.146754E-01
rb/2V         ->  rb/2V       =  0.141513
Aileron       ->  Cl roll mom =   0.00000
Elevator      ->  Cm pitchmom =   0.00000
Rudder        ->  Cn yaw  mom =   0.00000

alpha      =  -4.40678     deg
beta       =   0.00000     deg
pb/2V      = -0.529542E-02
qc/2V      =  0.714699E-02
rb/2V      = -0.687139E-01
CL         =   0.989201
CDo        =  0.401000E-01
bank       =   45.0000     deg
elevation  =   0.00000     deg
heading    =   0.00000     deg
Mach       =   0.00000
velocity   =   11.5600     m/s
density    =   1.12060     kg/m^3
grav.acc.  =   9.81000     m/s^2
turn_rad.  =   13.6222     m
load_fac.  =   1.41421
X_cg       =   3.57127     Lunit
Y_cg       =  0.120516E-07 Lunit
Z_cg       = -0.427481     Lunit
mass       =   15.8399     kg
Ixx        =   6.88271     kg-m^2
Iyy        =   3.91601     kg-m^2
Izz        =   9.87342     kg-m^2
Ixy        =  0.145612     kg-m^2
Iyz        = -0.758136E-08 kg-m^2
Izx        = -0.391923     kg-m^2
visc CL_a  =   0.00000
visc CL_u  =   0.00000
visc CM_a  =   0.00000
visc CM_u  =   0.00000
```

```
-----------------------------------------------
Run case  3:  25deg bank

alpha         ->  CL          =  0.771781
beta          ->  beta        =   0.00000
pb/2V         ->  pb/2V       =   0.00000
qc/2V         ->  qc/2V       =  0.409003E-02
rb/2V         ->  rb/2V       =  0.845786E-01
Aileron       ->  Cl roll mom =   0.00000
Elevator      ->  Cm pitchmom =   0.00000
Rudder        ->  Cn yaw  mom =   0.00000

alpha      =  -5.16951     deg
beta       =   0.00000     deg
pb/2V      = -0.370910E-02
qc/2V      =  0.199066E-02
rb/2V      = -0.409979E-01
CL         =   0.771781
CDo        =  0.401000E-01
bank       =   25.0000     deg
elevation  =   0.00000     deg
heading    =   0.00000     deg
Mach       =   0.00000
velocity   =   11.5600     m/s
density    =   1.12060     kg/m^3
grav.acc.  =   9.81000     m/s^2
turn_rad.  =   29.2129     m
load_fac.  =   1.10338
X_cg       =   3.57127     Lunit
Y_cg       =  0.120516E-07 Lunit
Z_cg       = -0.427481     Lunit
mass       =   15.8399     kg
Ixx        =   6.88271     kg-m^2
Iyy        =   3.91601     kg-m^2
Izz        =   9.87342     kg-m^2
Ixy        =  0.145612     kg-m^2
Iyz        = -0.758136E-08 kg-m^2
Izx        = -0.391923     kg-m^2
visc CL_a  =   0.00000
visc CL_u  =   0.00000
visc CM_a  =   0.00000
visc CM_u  =   0.00000
```

```
Run case  4:  5deg bank

alpha         ->  CL          =  0.700748
beta          ->  beta        =   0.00000
pb/2V         ->  pb/2V       =   0.00000
qc/2V         ->  qc/2V       =  0.158254E-03
rb/2V         ->  rb/2V       =  0.174425E-01
Aileron       ->  Cl roll mom =   0.00000
Elevator      ->  Cm pitchmom =   0.00000
Rudder        ->  Cn yaw  mom =   0.00000

alpha      =  -5.16951     deg
beta       =   0.00000     deg
pb/2V      = -0.370910E-02
qc/2V      =  0.199066E-02
rb/2V      = -0.409979E-01
CL         =   0.700748
CDo        =  0.401000E-01
bank       =   5.00000     deg
elevation  =   0.00000     deg
heading    =   0.00000     deg
Mach       =   0.00000
velocity   =   11.5600     m/s
density    =   1.12060     kg/m^3
grav.acc.  =   9.81000     m/s^2
turn_rad.  =   155.702     m
load_fac.  =   1.00382
X_cg       =   3.57127     Lunit
Y_cg       =  0.120516E-07 Lunit
Z_cg       = -0.427481     Lunit
mass       =   15.8399     kg
Ixx        =   6.88271     kg-m^2
Iyy        =   3.91601     kg-m^2
Izz        =   9.87342     kg-m^2
Ixy        =  0.145612     kg-m^2
Iyz        = -0.758136E-08 kg-m^2
Izx        = -0.391923     kg-m^2
visc CL_a  =   0.00000
visc CL_u  =   0.00000
visc CM_a  =   0.00000
visc CM_u  =   0.00000
```

```
#[Lancer 1]
#Victor     Ukiri,      Date:2022/12/28
#x,y,z      cordinate   system      matches     AVL         default
Lunit       =           0.3048      m
Munit       =           0.4535      kg
Tunit       =           1.0         s
g           =           9.81
rho         =           1.1206      #at         xkm         above       sea         level,      value       matches     value       from        density     calc.
#fuselage   length      =           1           m
#mass       x           y           z           Ixx         Iyy         Izz
1.1390      1.33785     0.0         -0.50       0           0           0           ! fuselage pod
1.1390      3.8757      0.0         -0.50       0           0           0           ! fuselage pod
1.3         6.250       0.0         -0.25       0           0           0           ! fuselage pod
4.515       3.37        3.974       0.0         0           0           0           ! Right wing
4.515       3.37        -3.974      0.0         0           0           0           ! Left wing
3.0         3.9         0.0         -1.626      0           0           0           ! Main gear
1.5         1.2         0.0         -1.626      0           0           0           !Nose gear
0.3757      8.60        1.877425    0.0         0           0           0           ! Right hor stab
0.3757      8.60        -1.877425   0.0         0           0           0           ! left hor stab
0.5888      8.60        0.0         1.33255     0           0           0           ! Vertical stabilizer
14.2        3.55        0.0         -0.50       0           0           0           ! Payload
#Electronics
0.8         0.1667      0.0         0.0         0           0           0           ! Motor
0.68        1           0.0         0.0         0           0           0           ! propulsionbattery
0.21        2           0.0         0.0         0           0           0           ! reciever battery
```

Figure 16: Mass File