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ME486C - Capstone 2: SAE Aero25
January 24, 2025

Engineering Calculations Summary

1. Top Level Design Summary

Our project is to design and build a remote-control (RC) plane to fly at the SAE Aero25 competition on April 4-6, 2025. The plane must carry 67 fluid ounces of liquid water in flight. Thus far, we've designed the plane to have a fuselage made of laser-cut birch. The wings have ribs cut from balsa wood, as well as leading and trailing edges also of balsa. The spars of the wings are carbon fiber.

1.1 CAD

ITEM NO.	PART NUMBER	QTY.
1	Fuselage Panel - Bottom Main	1
2	Fuselage Panel - Side	2
3	Fuselage Panel - Top	1
4	Fuselage Panel - Bottom Tail	6
5	rear assemble	1
6	elevator right	1
7	elevator left	1
8	bottom rib	1
9	rudder rib	5
10	wing right	1
11	wingleft	1

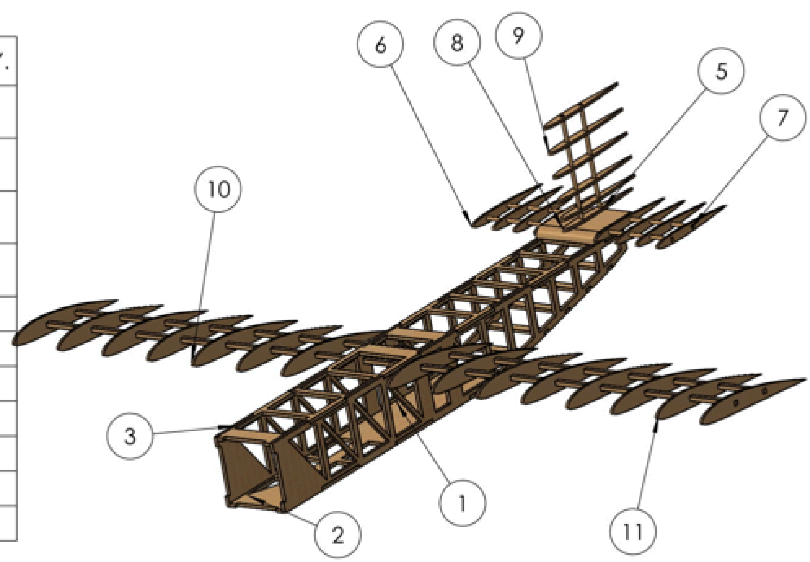

Figure 1: Full Cad Frame

1.1.1 Fuselage

The Fuselage marked by balloons 1-4 is made up of panels that are laser cut out of birch. This allows for the panels to have built in abutments that help solve the problem of weight reduction without compromising the stability of the structure. Using laser cutting allows for precision cuts to have a consistent aerodynamic profile which reduces drag and will help with the short take off goal.

1.1.2 Wings

The wings displayed by balloons 10 and 11 are made of balsa wood ribs and carbon fiber spars. Similar to the fuselage the ribs are laser cut to allow for precision and consistency to have a consistent aspect ratio and aerodynamic profile to increase lift and reduce drag. The spaced ribs and lightweight carbon fiber spars reduce weight while not compromising the airfoil shape.

1.1.3 Stabilizers/ Rudder

This subsystem is shown in balloons 5-9. Similar to the wings, the assembly is made up of balsa wood ribs and spars. However due to the vertical rudder, used for stability in flight, there is an extra piece of balsa wood used to support the stabilizer and rudder. This system is used to house the elevator that allows for increased lift at takeoff and maneuverability.

1.1.4 Landing Gear

The landing gear is a purchased part that we have adjusted to fit our plane. This subsystem meets the requirement of steerable landing gear per the SAE Aero rules. The spacing of the landing gear allows for increased angle of attack due to the short distance from center of gravity.

1.1.5 Electronics

While not shown in the CAD, the electrical diagram highlights how the components will work together per SAE rules. All components are bought and will be assembled and secured in the final fuselage. Velcro with adhesives on the other side will be the primary way to secure smaller electronics while heavy components like the battery and motor being secured with hardware or a designated place for them to minimize movement.

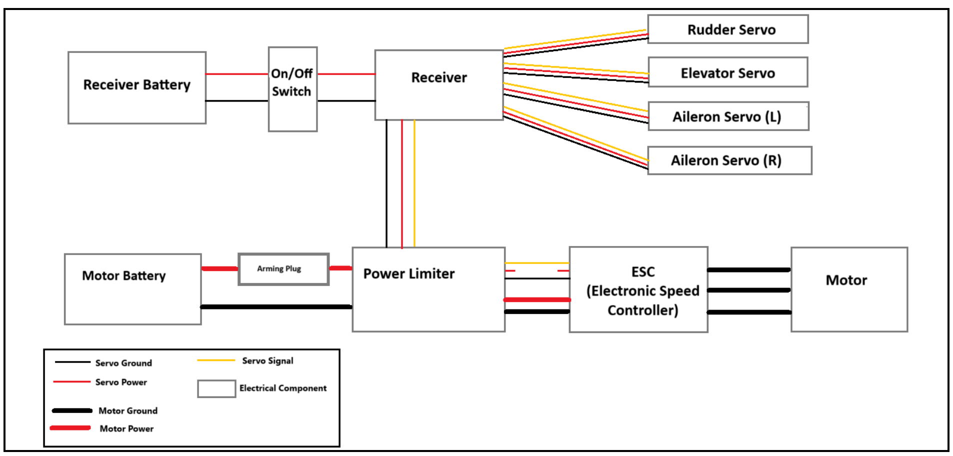

Figure 2: Electrical diagram

1.2 QFD

System QFD

Project: **SAE Aero25**
Date: January 2025

0 = no correlation
3 = minor correlation
6 = average correlation
9 = major correlation

#	Technical Requirement	1	2	3	4	5	6	7	8	9	10
1	Reduce Total Cost										
2	Increase Coefficient of Lift										
3	Decrease Coefficient of Drag		9								
4	Stable Structural Design	6	9	9							
5	Aspect Ratio	3	9	9	9						
6	Efficient Electrical Circuit	9	6	6							
7	Aerodynamic Profile		9	9	6	9					
8	Balanced Weight Distribution		3	3	9	3		9			
9	Reduce Weight	9	6	6	9	3		6	6		
10	Efficient Use of Materials	9		3	3	6	6	6	3	9	

Technical Requirements

#	Customer Needs	Customer Weights	Reduce Total Cost	Increase Coefficient of Lift	Decrease Coefficient of Drag	Stable Structural Design	Aspect Ratio	Efficient Electrical Circuit	Aerodynamic Profile	Balanced Weight Distribution	Reduce Weight	Efficient Use of Materials
1	Stable Flight	7		6	6	9	9	9	9	9		
2	Lightweight	8	6	3	2	6	9	6	9		9	3
3	Cost Effective	6	9				6	3	3		9	9
4	Competition Ready	10		3	2	9	9	9	9	6	3	
5	Short Takeoff	8		9	9	6	6	3	9	9	6	
6	Payload Delivery	8		3	3	3	3		6	9		
7	Structural Integrity	9		6	6	9	6		9	9	3	
8	Reproducibility	10	9	6	3	9		3	9	3		9
9	Safety in Flight Performance	10	6	9	9	9	9	9	9	9	9	3

	Reduce Total Cost	Increase Coefficient of Lift	Decrease Coefficient of Drag	Stable Structural Design	Aspect Ratio	Efficient Electrical Circuit	Aerodynamic Profile	Balanced Weight Distribution	Reduce Weight	Efficient Use of Materials
Technical Requirement Units	$	N/A	N/A	kPa	N/A	W	N/A	N	N	$
Technical Requirement Targets	<$5,000	>0.8	<0.06	196	6<AR<10	450	N/A	N/A	35.6	N/A
Absolute Technical Importance	252	396	348	534	477	363	624	468	321	198
Relative Technical Importance	9	5	7	2	3	6	1	4	8	10

Figure 3: QFD

Figure 3 shows the House of Quality to date. The engineering requirements are to reduce the total cost, increase the coefficient of lift, decrease the coefficient of drag, have a stable structural design, have a reasonable aspect ratio, design an efficient electrical circuit, design a coherent aerodynamic profile, balance the weight in the plane properly, reduce the weight, and efficiently use our materials. The customer need are that the plane flies stably, is lightweight, is cost effective, is competition ready, has a short takeoff, can deliver the water payload, is structurally sound, is reproducible, and is safe in flight,

1.2.1 Engineering Requirements

Engineering Requirements:

The first engineering requirement is to reduce the total cost of the plane. We're working with a limited budget and in order to create a design that can be reproducible, as is listed in the customer requirements, the cost of the plane must be kept as low as possible. Increasing the coefficient of lift and decreasing the coefficient of drag both improve the plane's takeoff distance. An increased coefficient of lift allows the plane to take off in a shorter distance, while the decrease in coefficient of drag lowers the thrust needed to achieve the required lift for takeoff. A stable structural design ensures that the plane does not deform or break at any point in flight. This is ensured by the materials used for each subsystem, as well as the truss design of the plane's fuselage ensuring a resilient design. A high-speed aircraft would require a low aspect ratio if it needed to complete complicated maneuvers, but a higher aspect ratio allows for less drag and produces a higher lift at slower speeds. The aim of this engineering requirement is to find the optimal aspect ratio for the plane. The design also requires an efficient electrical circuit with a power limiter of 450 Watts. All aircrafts in the competition are electrically powered in this way and the electrical circuit ensures the plane will fly safely and in control, with a kill switch should connection between the receiver and transmitter be lost.

The aerodynamic profile of the plane should be designed such that the fuselage does not induce unnecessary drag. It should be designed instead to enhance flight performance, being slim and smooth on the outside. A balanced weight distribution is also needed, as this relates to the structural integrity and safety in flight. The plane will not be able to take off if it is unbalanced, as with a front heavy design it will nosedive and with a back heavy design it will continue to tilt back until the fuselage is perpendicular to the ground instead of parallel. Reduced weight is a goal given by the competition, as a lighter weight plane receives a higher score than a heavy plane. This heavily influences the material choices of this project. Lastly, an efficient use of materials relates to reducing the total cost of the design. New materials will not be bought until needed, so as to make the best use of what we already have as a team.

1.2.2 Customer Requirements

The customer of this project is the Society of Automotive Engineers (SAE). SAE stated their customer requirements clearly in the competition rules. The aircraft must have a steady flight and be in control throughout the entire flight, leading to safety in flight performance. The takeoff is to be shortened as much as possible, as a takeoff in less than 10 feet adds the most points to the total score. The plane must deliver its payload of water, meaning that it takes the water in pre-flight, flies with the water, and disposes the water post-flight. The plane must be competition ready by April 4th and reproducible, meaning that one could follow the final design report of the plane without having questions. The plane must have structural integrity, which relates back to safety in flight of the design. Lastly, the design must be lightweight and cost effective. Both of these requirements heavily affect the material choices of the plane's sub-systems.

2. Summary of Standards, Codes, and Regulations

2.1 Standards

2.1.1 SAE Micro Design Rules

- Aircraft must fit and adhere to the specified competition guidelines.
- Payload requirements: Aircraft must carry and drop a 67 oz water payload with precision.

2.1.2 Safety Standards

- Electrical safety: Compliance with IEEE standards for low-voltage electronics.
- Flight safety: Adherence to FAA model aircraft safety guidelines, including no-fly zones and altitude limits.

2.2 Codes

2.2.1 Structural Safety

- ASME standards for material properties and strength (e.g., for balsa wood and carbon fiber).
- AIAA guidelines for aerodynamic stability and structural analysis.

2.2.2 Electrical Codes

- UL 60950 for safe operation of batteries, motors, and power controllers.
- IEC 62133 for lithium-ion battery safety.

2.3 Regulations

2.3.1 Competition Requirements

- Micro Class payload shall consist of liquid water. Frozen water is prohibited.
- Micro Class aircraft shall have a single Payload Container for carrying liquid water with the following additional requirements:
    1. Payload container shall be fully enclosed with a minimum of two (2) sealable holes.
    2. The first hole shall be on top of the payload container for filling.
    3. The second hole shall be on the bottom of the container and used for unloading liquid water from the payload container.
    4. Payload container must have a minimum volume of 67 fluid ounces. Teams must consider the ability to quickly drain all liquid water as a timed activity.
- Micro Class aircraft are restricted to electric motor propulsion only.
- Micro Class aircraft must use Lithium Polymer batteries. Micro class batteries are allowed to be a maximum of four (4) cells.
- Micro Class aircraft must use a 2021 or newer, 450-watt power limiter from the official supplier.

2.3.2 Team Safety

- OSHA guidelines for shop safety during manufacturing processes (e.g., protective gear and equipment).

3. Summary of Equations and Solutions

3.0.1 Lift and Drag - Chazz Coppa

The load cases for lift and drag were determined based on critical phases of the aircraft's operation where aerodynamic forces were at their maximum or most influential. These conditions include takeoff, taxi, and descent. During takeoff, the aircraft must generate sufficient lift to overcome its total weight within a limited distance. The lift force is critical in determining the shortest possible takeoff distance, while drag must be minimized to reduce the required thrust. During taxi, lift forces must balance the aircraft's weight while maintaining structural integrity under bending moments caused by aerodynamic forces. Lastly, when landing, lift must decrease smoothly to allow controlled descent, while drag is intentionally increased to reduce speed for landing.

$$L = \frac{1}{2}\rho v S C_L$$

L = Lift [N]
ρ = Air Density
v = Velocity
S = Wing Area
C_L = Lift Coefficient

$$D = \frac{1}{2}v^2 S C_D$$

D = Drag
C_D = Drag Coefficient

$$\frac{L}{D} = \frac{C_L}{C_D}$$

$$C_D = C_{D0} + \frac{C_L^2}{\pi e AR}$$

C_{D0} = Zero-Lift Drag Coefficient
e = Oswald Efficiency Factor
AR = Aspect Ratio of Wing

Flight Constraints:

- $\rho = 1.225\ kg/m^3$
- S = $0.2\ m^2$
- b = 1.2 m
- e = 80%
- AR = 7.3

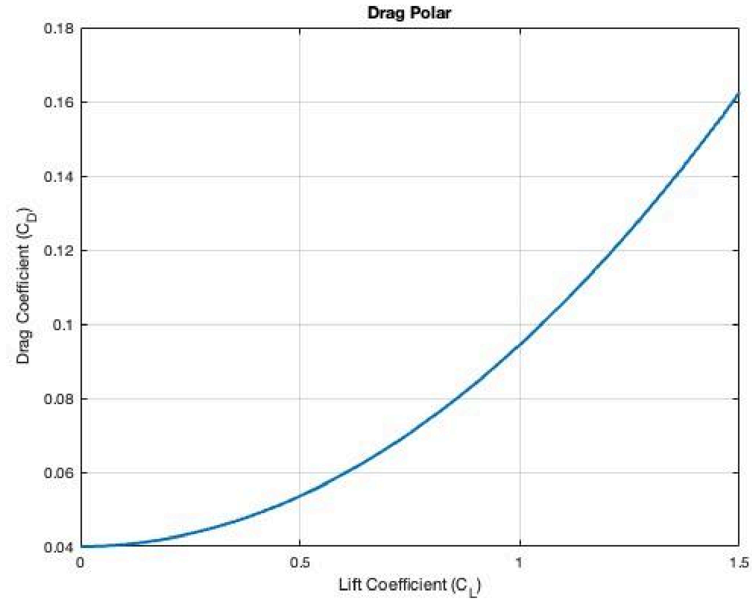

Figure 4: Drag Polar Graph

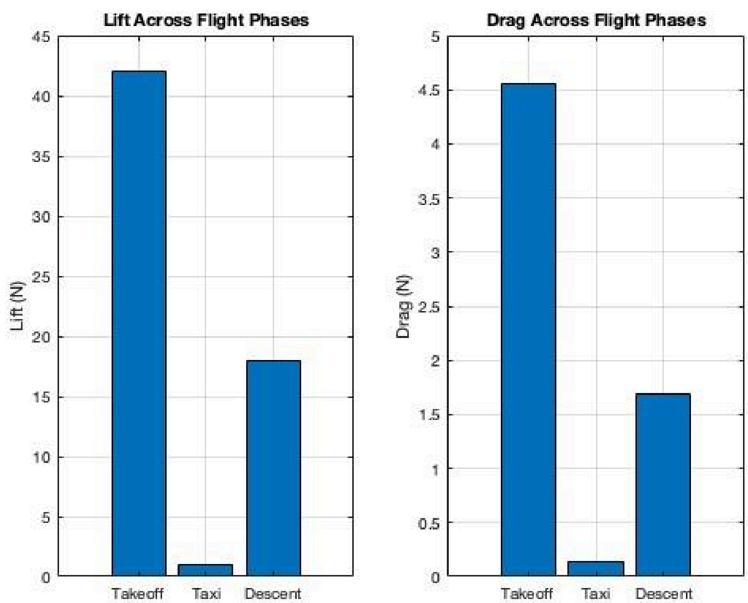

Figure 5: Lift (left) and Drag (right) Across Flight Phases

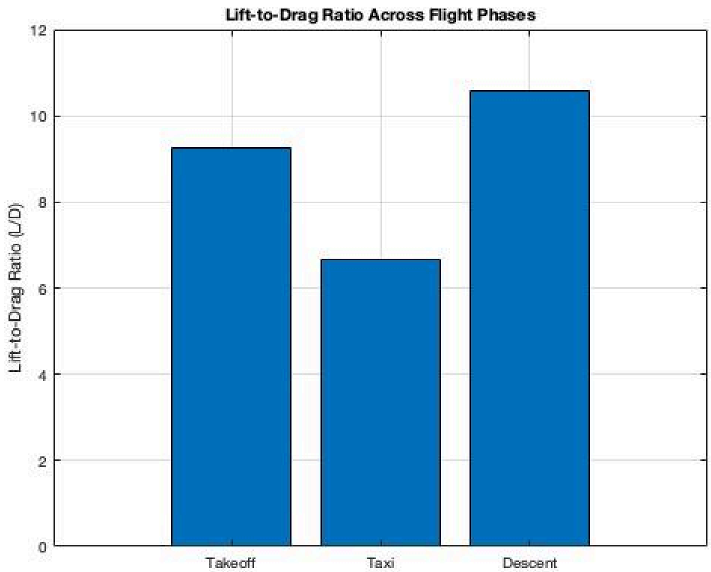

Figure 6: Lift-To-Drag Ratio Across Flight Phases
(reference appendix A!!)

3.0.2 Competition Scoring - Colton Tutrone

In the SAE Aero Design competition, from both the analysis and optimization of aircraft performance, mathematical modeling was crucial. The overall flight score is developed around a few critical equations representing the scoring system of the competition, depending on payload weight, wingspan, and the accuracy of the prediction. These formulae allow teams to quantify how any specific design decision will impact the performance of the aircraft. From these, informed trade-offs between payload capacity, aerodynamic efficiency, and flying stability can be made.

FFS is the key equation in determining the aircraft's performance throughout the flights. It gains its contributions from both FS and WS. FS depends on the payload carried, as well as a PPB to be awarded for correctly predicting the weight of the payload. The Wingspan Score further encourages designs for larger wingspans with the constraint of ensuring flight effectiveness.

These equations have formed a structured approach for teams in pursuit of the maximum competition score by judiciously balancing payload weight, energy constraints, and wingspan. Knowing how these factors mathematically interact will enable teams to optimize their designs for maximum efficiency and performance within strict competition guidelines.

In the Final Flight Score calculation FS1, FS2, and FS3 represent the Flight Scores for three flight attempts, and WS is the Wingspan Score.In the final flight score $W_{payload}$ is the weight of the payload carried by the drone, and PBB represents the Payload Prediction Bonus. Heavier payloads increase the flight score. In the Payload Prediction Bonus, PPB is the predicted payload, which incentivizes accuracy.

$$FSS = Final\ Flight\ Score = FS_1 + FS_2 + FS_3 + WS$$

$$FS = Flight\ Score = \frac{W_{payload}}{w} + PBB$$

$$PBB = Payload\ Prediction\ Bonus = MAX(5 - (W_{payload} - P)^2, 0)$$

$$WS = Wingspan\ Score = 2^{1+\frac{b}{5}}$$

$W_{payload}$ = Regular Boxed Cargo Weight (lbs)
b = Aircraft Wingspan (ft)
P = Predicted Payload

3.0.3 Estimated Weight at Takeoff - Matthew Espinoza

The weight at takeoff is a critical force the plane will experience, as the weight directly impacts the plane's ability to safely lift off the ground and climb effectively. The weight of the plane also plays an important role in Section 3.0.7 to calculate the amount of power needed to operate the aircraft.

$$W_0 = \frac{W_{payload}}{1 - W_e/W_0}$$

$$W_e/W_0 = 0.375$$

$W_0 =$ Weight at Takeoff
W_e = Empty Aircraft Weight
$W_{payload}$ = Weight of Payload

3.0.4 Structure - Jayden Glaus

Structural analysis of the wings' bending moments allows us to choose the material for the wings, most importantly the spars. The possible spar materials have different yield strength and from these calculations, we found the bending moment the spars must be able to withstand.

Assuming
W (Plane Weight) = 4 kg
L (Wingspan) = 1 m
Max 4 times gravity during turns

$$F_{weight} = 4kg * 9.81 \frac{m}{s^2} * 4 = 156\ N$$

$$F_{lift\ per\ wing} = \frac{156\ N}{2} = 78\ N$$

$$Max\ Bending\ Moment = 78\ N * 0.5m = 39\ Nm$$

3.0.4 Estimated Wing Area - Tesla Ventsam

Estimating the wing area of the plane allows us to further our design calculations and decisions, as the wing area is a key detail in the amount of lift the plane can generate, the wingspan, and the aspect ratio.

$$S = \frac{W_0}{W/S}$$

S = wingspan (ft)

$\rho = 0.002377\ slug/ft^3$
$v_{stall} = 7\ mph = 10.267\ ft/s$
$(C_L)_{max} = 2.34$

$$\frac{W}{S} = \frac{1}{2}\rho v_{stall}^{2}(C_L)_{max} = \frac{1}{2}(0.002377)(10.267^2)(2.34) = 0.293\ lb/ft^2$$

Assuming W = 8 lb

$$S = \frac{8}{0.293} = 27.3\ ft^2$$

3.0.5 Aircraft Dimensions from an Estimated Wingspan - Tesla Ventsam

The table below is a reference to the Trainer Design website. These calculations were used as a reference to our own calculations for the respective dimensions of the plane. This table shows all the dimensions in feet.

Calculate the Dimensions of your Plane based on the drawings above: By courtesy of Douglas Morris				
Enter Wingspan	Wing Aspect Ratio	Stab Area % of Wing Area	Elevator Area % of Stab Area	Rudder Area % of Fin Area
4.5	4.9 :1	15%	20%	33%
Click to Calculate				
Avg. Wing Chord	Wing Area	Aileron Length	Aileron Width	Strip Aileron Width
0.92	4.14	1.13	0.23	0.12
Fuselage Length (F)	Fuse ahead of Wing (F1)	Wing TE to Elev. Hinge (F2)	Fuselage Height (F3)	Balance Point from Wing LE
3.38	1.38	2.3	10% 0.34 15% 0.51	25% 0.23 33% 0.3
Stab Area	Avg. Stab Width (S1)	Stab Length (S2)	Elevator Area	Avg. Elevator Width (E1)
0.62	0.45	1.38	0.12	0.09
Fin Area	Avg. Fin Width (R1)	Fin Height (R2)	Rudder Area	Avg. Rudder Width (R3)
0.2	0.26	0.78	0.07	0.09
Clear				

Figure 7: Trainer Design Parameters Calculations

3.0.6 Estimated Takeoff Power - Matthew Espinoza

The competition limits the power used by each team to 450 Watts. Given this restriction, we must keep in mind the power that is required for the plane to take off. If the power used is too high at any given point, we have to reevaluate the materials of the fuselage and wings, the electronics in the plane, and how much power the motor requires.

$$P_R = TV_{inf} = \frac{T}{W}W_0 V_{inf}$$

$\frac{T}{W}$ = Thrust to Weight Ratio = 0.0277
$V_{inf} = 0.7 * V_{liftoff} = 0.7 * (1.1 * v_{stall})$
V_{inf} = 8.59 ft/s
W_0 = Takeoff Weight

$$P_R = 0.0277 * 8 * 8.59 = 1.59\frac{ft*lb}{s} = 2.6\,W$$

3.0.7 Estimated Power/Pound - Matthew Espinoza

Estimated Performance:	From Power/Weight:		To Power/Weight:	
	Watts / lb	Watts / kg	Watts / lb	Watts / kg
May takeoff of ground and do simple maneuvers For cubic loadings up to about 12 oz/cu.ft	35	77	55	121

Figure 8: Trainer Design Power-to-Weight Calculations

Calculated using table provided by Trainer Design

3.0.8 Airfoil/Wing Analysis - Matthew Espinoza

An airfoil and wing analysis mathematically proves the optimal angle of attack for the aircraft design. The ratio of coefficient of lift to coefficient of drag is first plotted against the plane's angle of attack, then the coefficient of drag is plotted against the angle of attack to find the range of angles that best suits the design. This ensures that the plane generates the most lift possible while minimizing the drag created.

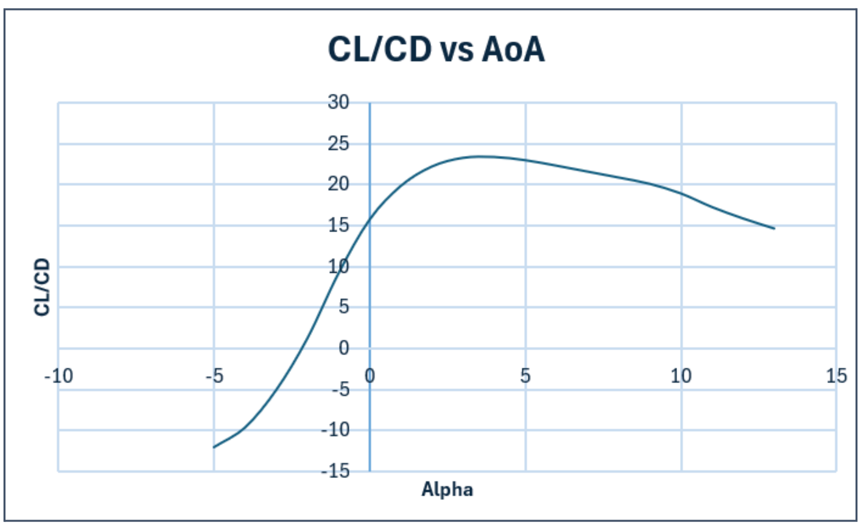

Figure 9: Coefficient of Lift/Coefficient of Drag vs Angle of Attack

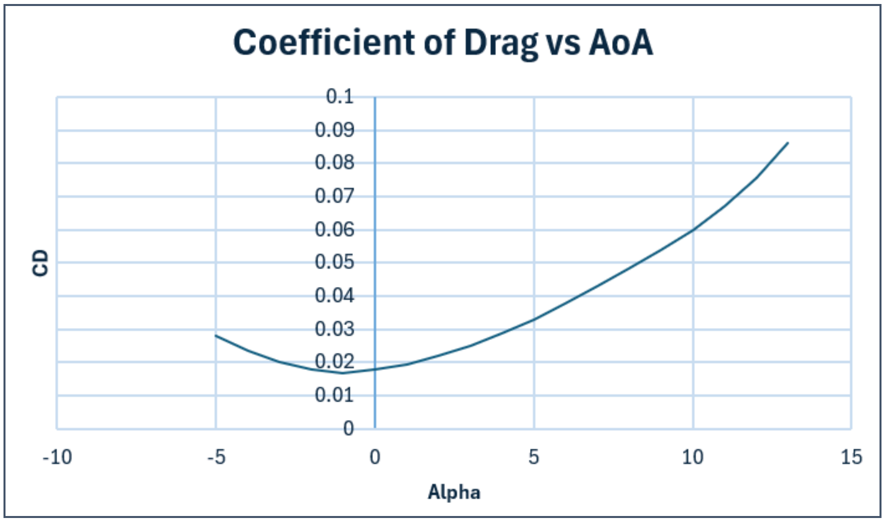

Figure 10: Coefficient of Drag vs Angle of Attack

- Using a NACA 3411 airfoil (Clark-Y)
- Used to find best angle of attack
- Optimizing drag due to power constraints using XFLR5

3.0.9 Reynold's Number - Colton Tutrone

In this design, the Reynolds number plays a crucial role in understanding the aerodynamic flow characteristics over the wing and influencing the design of key features like the wing shape, size, and overall configuration. The Reynolds number is a reflection of the nature of the airflow around the aircraft: the higher the Reynolds number, the flow is by and large turbulent-which enhances lift but can also increase drag-whereas for lower Reynolds numbers, the flow is usually laminar with less drag but possibly unsatisfactory lift characteristics.

The load cases considered in this design focus on those conditions that would impose extreme aerodynamic forces on the aircraft, particularly during critical instances such as takeoff, landing, and at high-speed flight. Worst-case scenarios include the study of lift, drag, and moments at velocities corresponding to high-stress points, such as takeoff speed, where forces are large due to high acceleration, or during rapid maneuvers. Besides, Reynolds was computed using the average chord of 0.11481 m and design velocity of 15 m/s. At such velocity, Reynolds is 116,554. It represents the flow in the turbulence regime, promising better lift, and probably also more serious drag in the flight conditions.
Important Load Cases/Critical Conditions:

Maximum Lift and Drag Forces: These are higher velocities, like in the case of takeoff or fast maneuvers, which result in the most extreme aerodynamic loads. Bending Stresses: Computed from the aerodynamic moments due to lift and drag, these are very important in determining the structural limits of the wing under high-stress conditions.
Flight Velocity and Flow Conditions: The Reynolds number computed for the aircraft is 116,554 to study the behavior in turbulent flow conditions while in operation, which affects the lift and drag characteristics. These load cases ensure that the design can withstand the worst-case aerodynamic and structural forces encountered during normal operations, especially at critical phases of flight like takeoff and landing. The

nature of the flow behavior deduced through Reynolds number analysis allows the optimization of design features for high efficiency in the performance of the drone to meet the competition requirements.

Front Chord Length = 5.9*0.0254 = 0.14986 m
Rear Chord Length = 3.14*0.0254 = 0.07976 m
Average Chord Length = $\frac{0.14986+0.07976}{2}$ = 0.11481 m
Total Estimated Mass: 6.88-7.48 lbs (with payload)

$$V = \left(\frac{2*P*d}{m}\right)^{1/3}$$

P = 450 Watts
d = 3.048 m (10 foot takeoff distance)
m = mass

$$V = 15\, m/s$$

$$Re = \frac{\rho*V*L}{\mu}$$

$\rho = 1.225\, kg/m^3$
V = 15 m/s
L = 0.11481 m (average chord length)
$\mu = 1.81 * 10^{-5}$ Pa*s

$$Re = 116,554$$

3.0.10 Constraint Diagram - Jayden Glaus

Aircraft design constraint diagrams visualize aerodynamic constraints and parameters related to thrust-to-weight and wing loadings. The diagram plots wing loading vs. thrust-to-weight ratio and satisfies all constraints if the point lies above the sustained turn and takeoff curves and to the left of the landing curve.

John D. Anderson Jr. 's book, “Aircraft Performance and Design,” provided theoretical understanding, and the diagram was plotted in MATLAB.

Using the constraints of this competition, the graph showed a minimum thrust-to-weight ratio of 0.5 and a wing loading of about 17 N/m^2 for the shortest takeoff distance, while the landing constraint curve did not affect the results due to the large allowed landing distance.

Takeoff Constraint Curve

$$\frac{T}{W} = \frac{1.31}{g*\rho*C_L*s_g} * \frac{W}{S}$$

Landing Constraint Curve

$$\frac{W}{S} = \frac{\frac{2*s_g*j^2}{g*\rho*C_L*(\frac{T_{rev}}{D}+\frac{D}{W}+\mu(1-\frac{L}{W}))}+\frac{2}{\rho*C_L}+\sqrt{\left(\frac{2*s_g*j^2}{\frac{T_{rev}}{D}+\frac{D}{W}+\mu(1-\frac{L}{W})}+\frac{2}{\rho*C_L}\right)^2-4*\left(\frac{j^2}{g*\rho*C_L*(\frac{T_{rev}}{D}+\frac{D}{W}+\mu(1-\frac{L}{W}))}\right)^2*s_g^2}}{2*\left(\frac{j^2}{g*\rho*C_L*(\frac{T_{rev}}{D}+\frac{D}{W}+\mu(1-\frac{L}{W}))}\right)^2}$$

Sustained Turn Constraint Curve

$$\frac{T}{W} = \frac{2*K*n^2*\frac{W}{S}}{\rho*V^2} + \frac{\rho*V^2*C_D}{2*\frac{W}{S}}$$

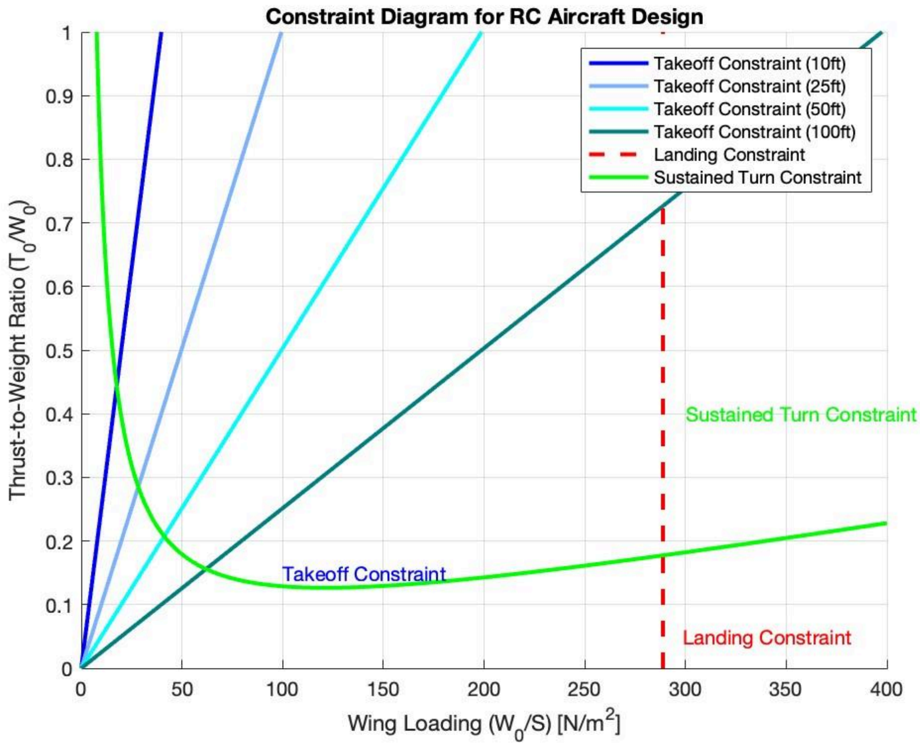

Figure 11: Constraint Diagram

Assuming

C_L = 1.3

C_D = 0.02

Lift = Weight

Max Load Factor = 2

3.0.11 Optimized Parameters - Tesla Ventsam

Section 3.0.5 referenced an online source provided by past RC aircraft engineers to compare our aircraft design to that which was recommended by others. We also used a MATLAB code, shown in Appendix B, to calculate the optimized parameters of the plane. Figure 12 shows the optimization plot function, which calculates feasible aircraft designs and compares these designs against the highest score possible. The code ended its iterations when the difference between the scores calculated was below 0.0001 points. The calculated score used the equations shown in 3.0.2.

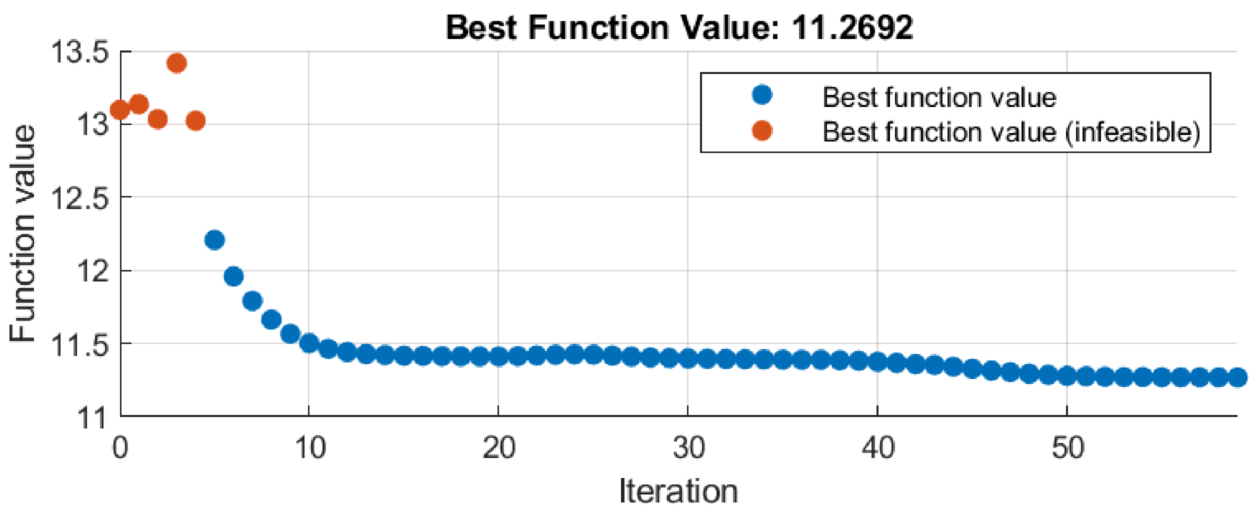

Figure 12: Iterative Score Calculations from Optimization Code

The table below shows the optimized parameters of the plane after running the MATLAB code.

Table 1: Optimized Parameters

Parameter	Length
Wing Half Span	47.511 in
Wing Root Chord	14.199 in
Wing Taper Ratio	28.97
Wing X Location	10.041 in
Horizontal Tail Half Span	14.205 in
Horizontal Tail Chord	8.493 in
Vertical Tail Half Span	10.889 in
Vertical Tail Chord	3.337 in
Length of Fuselage	44.540 in
Cargo Bay X Location	28.973 in

3.0.12 Wing strength Code - Colton Tutrone

The code found in Appendix c is applied to the aerodynamic load and structural stresses of a micro-drone wing, focusing on lift and drag forces, moments, and resultant bending stresses encountered in flight. The analysis simulates the worst-case scenarios that a drone may go through in typical operations, ensuring the resistance of a design to critical conditions.

Key Conditions and Load Cases Analyzed:

Lift and Drag Forces:
Aerodynamic lift and drag forces are determined through basic concepts of the underlying airfoil shape, described by air density, rho; velocity, V; wing area, S; and coefficients of lift, CL, and drag, CD.
Forced Conditions: Worst case for the lift and drag forces normally occurs at high speed when these forces are usually at their maximum, such as in high-speed flight or during rapid maneuvers.
Moment Arms:
The geometry of the wing defines the moment arms for lift and drag. The moment arm for lift acts at half of the wingspan and that of drag at the chord length. Calculations will provide the moments, which could bend the wing.
Moment Calculations:
This code calculates the moments due to lift and drag at various velocities; these are necessary in understanding what is taking place on this wing during flight. It will also find the combined net moment, which represents the overall torque due to both the lift and drag forces.
Bending Stress Analysis:
The code calculates the bending stress at the wing root by dividing the bending moment by the section modulus (Z). This decides the structural integrity of the wing concerning aerodynamic loads, specifically that the wing would not fail under the action of forces.
The stress is then compared to the material's yield strength (in this case, 170 MPa) in order to decide the maximum velocity at which safe operations can be done.
Breaking Point Velocity:
The code calculates the "breaking point" velocity where the bending stress exceeds the material's yield strength-a speed at which structural failure may occur. This critical velocity will form a vital consideration in designing to avoid excessive stresses at high speeds.
Design Considerations:
Worst-Case Load Conditions: Most of the common load cases solved are those during high-speed flight conditions where lift and drag are highest; the design will have to support these peak forces, specifically as regards the wing structure.
Material Strength: The wing material must withstand the bending stress generated during high velocities. The analysis ensures that the maximum stress at any point does not exceed the yield strength of the material, preventing failure during flight.
Operational Limits: The breaking point velocity provides an upper limit for the drone's speed, beyond which the wing could fail. This helps define the operational limits for safe flight.
It therefore summarizes how the code analyzes and ensures that the micro-drone wing can bear the aerodynamic forces and resultant structural stresses under critical flight conditions, like high-speed flight or maneuvering. It further includes all possible load cases on the wings: maximum lift and drag forces, bending moments, material strength among others, with a view to checking the worst conditions that could be faced.

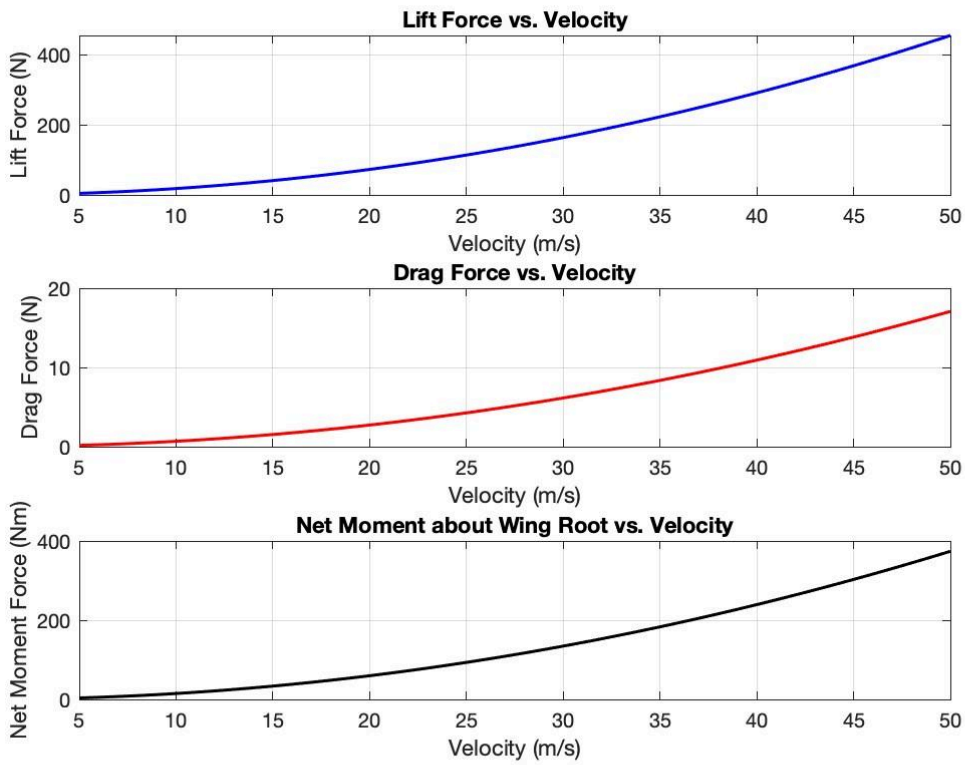

Figure 13: Plots of Forces Produced by WIng Strength Code

3.1 Factor of safety

Part Name	Material	Method of Analysis	Max Stress (MPa)	Yield Strength (MPa)	FoS	Area of Concern (Yes/No)
Battery Circuit	Electrical wires (Copper)	Thermal Analysis (Max Power)	N/A	N/A	XX	No
Wing Spars	Carbon Fiber	FEA + Bending Analysis	120	600	5.0	No
Wing Ribs	Balsa Wood	FEA	1.8	3.5	1.9	Yes

Battery Circuit (450 W)

- Method: Calculate power dissipation in wires and ensure wire gauge supports the current. A 450-watt battery at 12 volts (example) will require 37.5 amps.
- FoS for Electrical Load: Based on wire gauge and thermal dissipation, ensure wires won't overheat or cause failures.
  I=PV=450/12=37.5 amps

Wing Spars

- Material: Carbon fiber (yield strength = 600 MPa).
- Result: Max stress of 120 MPa, resulting in an FoS of 5.0, indicating no structural concerns.

Wing Ribs

- Material: Balsa wood (yield strength = 3.5 MPa).
- Result: Max stress of 1.8 MPa, resulting in an FoS of 1.9, which is close to the failure threshold. This suggests a potential area of concern under higher-than-expected loading conditions.

4. Flow Charts and other Diagrams

4.1 Functional Decomposition

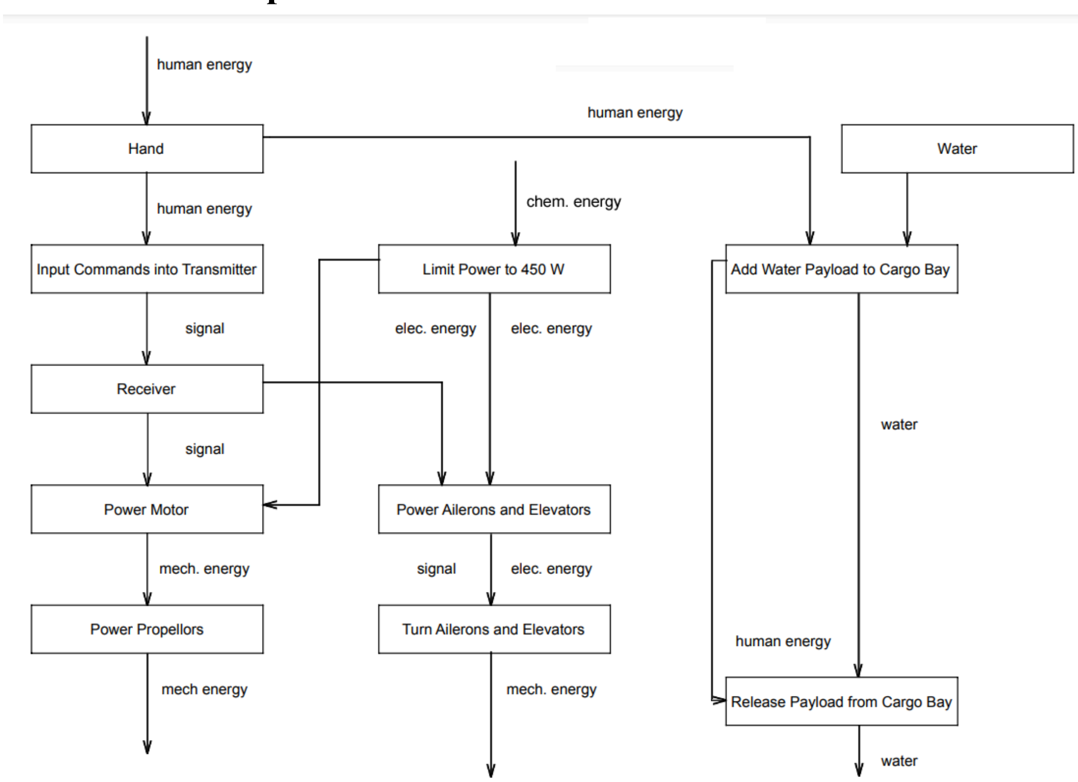

Figure 14: Functional Decomposition

Figure 14 shows the functional decomposition of the plane, explaining the input energies to the system, how these energies interact with one another and the system itself, and the outputs. The input energies are human energy from human control of the transmitter and chemical energy from the batteries in the transmitter and the plane. The transmitter communicates the human control to the receiver in the plane, which then powers the motor and the servos to control the flight path, and the outputs are the plane power in the propulsion system and the aileron and elevator control. There is also a material input and output of the water payload, which the plane will carry in flight.

5. Moving Forward

Moving forward, the team will be focused on the design report for the competition, the building deadlines coming up, and testing the completed aircraft. The entirety of the calculations necessary for the aircraft have been completed. Some calculations for the design report required by the competition must still be completed, namely an aircraft performance prediction which plots the neutral point and static margin on a single graph from an angle of attack of -10 to +15 with the water container half full. This will be started in the following week. The CAD will also need to be completed however only a few parts have yet to be made. The motor mount and nose section, hardware attachment points, and the water tank still need to be fully modeled in CAD. Laser cutting is expected to begin next week beginning with the fuselage panels. Additionally, static finite element analyses (FEAs) will be conducted within the next week to gather factors of safety (FoS) for critical components, ensuring the structural integrity of the design under expected load conditions.

6. Appendix

Appendix A - Lift & Drag Numerical Analysis

```
--- Takeoff Phase ---
Velocity (V) = 15.00 m/s
Lift Coefficient (CL) = 1.50
Lift (L) = 42.01 N
Drag Coefficient (CD) = 0.1624
Drag (D) = 4.55 N
Lift-to-Drag Ratio (L/D) = 9.24
```

```
--- Taxi Phase ---
Velocity (V) = 5.00 m/s
Lift Coefficient (CL) = 0.30
Lift (L) = 0.93 N
Drag Coefficient (CD) = 0.0449
Drag (D) = 0.14 N
Lift-to-Drag Ratio (L/D) = 6.68
```

```
--- Descent Phase ---
Velocity (V) = 12.00 m/s
Lift Coefficient (CL) = 1.00
Lift (L) = 17.92 N
Drag Coefficient (CD) = 0.0944
Drag (D) = 1.69 N
Lift-to-Drag Ratio (L/D) = 10.59
```

Phase	Lift_N	Drag_N	Lift_to_Drag
{'Takeoff'}	42.01	4.5482	9.2366
{'Taxi' }	0.93356	0.13971	6.6821
{'Descent'}	17.924	1.692	10.593

Appendix B - Optimized Parameters Code

Appendix B.1 - Generating Aircraft Initial Parameters

```
function [aircraft, wing, fuselage, hTail, vTail, payload, initialValues] = initializeAircraft

wing.Airfoil.Name = 'Clark Y';                                    % Airfoil Name

wing.Airfoil.ClAlpha = 0.145*180/pi;                              % Airfoil Lift Slope [increase per degree]

wing.Airfoil.Cd0 = 0.04;                                          % Airfoil Zero Lift Drag Coefficient [drag when 0 lift]

wing.Airfoil.Cl0 = 1.2;                                           % Airfoil Lift at 0 AoA [lift coef when angle of attack 0]
```

```
wing.Airfoil.Alpha0L = -5*pi/180;                                           %
Airfoil Zero Lift AoA [angle of attack which makes 0 lift]

wing.Airfoil.K = 0;                                                          %
Airfoil Parasitic Drag efficiency

wing.Airfoil.StallAoA = 15*pi/180;                                           %
Airfoil Aerodynamic Stall Angle of Attack [deg] (max angle of attack)

wing.Density = 4.88243;                                                      %
Mass per unit area [kg/m^2]

wing.HalfSpan = optimvar('b_w','LowerBound',0,'UpperBound',1.63);
% Half Span [m]

initialValues.b_w = 0.815;
% Half Span Initial Value [m]

wing.RootChord = optimvar('cr_w','LowerBound',0);                            %
Root Chord [m] (width at fuselage connection)

initialValues.cr_w = 0.23;                                                   %
Root Chord Initial Value [m]

wing.TaperRatio = optimvar('lambda_w','LowerBound',0,'UpperBound',1);  %
Taper Ratio (LEAVE AT ~1 FOR NOW??)

initialValues.lambda_w = 0.78;                                               %
Taper Ratio Initial Value

wing.MeanChord = (1+wing.TaperRatio)*wing.RootChord/2;                       %
Mean Chord [m]

wing.AspectRatio = 2*wing.HalfSpan/wing.MeanChord;                           %
Aspect Ratio

wing.PlanformArea = (1+wing.TaperRatio)*wing.RootChord*wing.HalfSpan;  %
Planform Area [m^2] (wing area)

wing.XLoc = optimvar('X_w','LowerBound', 0.15);                              %
Placement Location in X-dir [m]

initialValues.X_w = 0.19;                                                    %
X Location Initial Value [m]

wing.Xac = wing.XLoc+0.25*wing.RootChord;                                    %
Aerodynamic Center [m]
```

Fuselage Parameters

```
fuselage.SideLength = 0.127;
% Side Length of Square Fuselage Cross-section [m]

fuselage.Length = optimvar('l_f','LowerBound',0);                        %
Length of Fuselage [m]

initialValues.l_f = 0.9652;
% Length Initial value [m]

fuselage.Fineness = fuselage.Length/sqrt(4/pi*fuselage.SideLength^2);
% Fineness Ratio

fuselage.WettedArea = 4*fuselage.SideLength*fuselage.Length;             %
Wetted/Surface Area [m^2]

fuselage.Density = 0.016;
% Fuselage mass per unit surface area [kg/m^2]

fuselage.Xcg = fuselage.Length/2;                                        %
CG Location in X-dir [m]

fuselage.Volume =
fuselage.Length*fuselage.SideLength^2-fuselage.Length*(fuselage.SideLeng
th-4.66666666667);              % Volume [m^3]
```

Horizontal Tail Parameters

```
hTail.Airfoil.Name = 'NACA 0012';                                        %
Airfoil Name

hTail.Airfoil.ClAlpha = 0.1*180/pi;                                      %
Airfoil Lift Slope [per deg]

hTail.Airfoil.Cd0 = 0.03;                                                %
Airfoil Zero Lift Drag Coefficient

hTail.Airfoil.K = 0;                                                     %
Airfoil Drag efficiency

hTail.Density = 2;                                                       %
Tail mass per unit area [kg/m^2]

hTail.Airfoil.StallAoA = 10.5*pi/180;                                    %
Airfoil Aerodynamic Stall Angle of Attack [deg]

hTail.HalfSpan = optimvar('b_ht','LowerBound',0.2,'UpperBound',0.42);   %
Half Span [m]
```

```
initialValues.b_ht = 0.20828;
% Half Span Initial Value [m]

hTail.Chord = optimvar('c_ht','LowerBound',0.08);                    %
Chord Length [m]

initialValues.c_ht = 0.0889;
% Chord Length Initial Value [m]

hTail.AspectRatio = 2*hTail.HalfSpan/hTail.Chord;                    %
Aspect Ratio

hTail.PlanformArea = 2*hTail.Chord*hTail.HalfSpan;                   %
Planform Area [m^2]

hTail.Xac = fuselage.Length-0.75*hTail.Chord;                        %
Aerodynamic Center [m]

hTail.XLoc = fuselage.Length-hTail.Chord;                            %
Leading Edge Location [m]
```

Vertical Tail Parameters

```
vTail.Airfoil.Name = 'NACA 0012';                                    %
Airfoil Name

vTail.Airfoil.ClAlpha = 0.1*180/pi;                                  %
Airfoil Lift Slope [per deg]

vTail.Airfoil.Cd0 = 0.03;                                            %
Airfoil Zero Lift Drag Coefficient

vTail.Airfoil.K = 0;                                                 %
Airfoil Drag efficiency

vTail.Density = 4.88243;
% Tail mass per unit area [kg/m^2]

vTail.Airfoil.StallAoA = 10.5*pi/180;                                %
Airfoil Aerodynamic Stall Angle of Attack [deg]

vTail.HalfSpan = optimvar('b_vt','LowerBound',0.2,'UpperBound',0.42);  %
Half Span [m]

initialValues.b_vt = 0.3;                                            %
Half Span Initial Value [m]

vTail.Chord = optimvar('c_vt','LowerBound',0.08);                    %
Chord Length [m]
```

```
initialValues.c_vt = 0.12;                                                %
Chord Length Initial Value [m]

vTail.AspectRatio = vTail.HalfSpan/vTail.Chord;                           %
Aspect Ratio

vTail.PlanformArea = vTail.Chord*vTail.HalfSpan;                          %
Planform Area [m^2]

vTail.Xac = fuselage.Length-0.75*vTail.Chord;                             %
Aerodynamic Center [m]

vTail.XLoc = fuselage.Length-vTail.Chord;                                 %
Leading Edge Location [m]
```

Payload Parameters

```
aircraft.Avionics.Mass = 1.5;                                             %
Weight of Electronics [kg]

aircraft.Avionics.Xcg = 0.05;                                             %
CG Location of Electronics [m]

aircraft.Avionics.Length = 0.25;                                          %
Length of fuselage occupied by Electronics [m]

payload.Boxed.Length = optimvar('l_pb','LowerBound',0);                   %
Length of boxed payload [m]

initialValues.l_pb = 0.12446;
% Length of boxed payload Initial Value [m]

payload.Boxed.Height = ...

   optimvar('h_pb','LowerBound',0,'UpperBound',0.20);                     %
Height of boxed payload [m]

initialValues.h_pb = 0.12446;
% Height of boxed payload Initial Value [m]

payload.Boxed.Density = 1000;                                             %
Density of boxed payload [kg/m^3]

payload.XLoc = optimvar('X_p','LowerBound',0.25);                         %
Start Location of cargo bay [m]

initialValues.X_p = 0.4;                                                  %
Start Location Initial Value [m]
```

Thrust Parameters

The competition sets a power limit of 450 W to the motor, limiting static thrust to around 18N.

```
aircraft.MaxThrust = 18.0;                                              %
Max Thrust [N]

aircraft.DynamicThrustQuad = -0.0006;                                   %
Quadratic Term Dynamic Thrust Coefficient [kg*m]
```

General Aircraft Parameters

```
aircraft.RePerLength = 5e5;                                             %
Reynolds Number per unit length [per m]

aircraft.ReferenceArea = wing.PlanformArea;                             %
Reference Area [m^2]

aircraft.StallAoA = wing.Airfoil.StallAoA;                              %
Stall Aerodynamic Angle of Attack [deg]

aircraft.RollingResistance = 0.004;                                     %
Rolling Resistance

aircraft.TakeOffAoA = 5*pi/180;                                         %
Aerodynamic Angle of Attack at Take off [deg]

end
```

Appendix B.2 - Optimizing Parameters

```
[aircraft, wing, fuselage, hTail, vTail, payload, initialValues] = initializeAircraft;

designprob = optimproblem('ObjectiveSense','maximize');

[aircraft, wing, fuselage, hTail, vTail, designprob] = addAerodynamics(aircraft, wing, fuselage,
hTail, vTail, designprob);

[aircraft, wing, fuselage, hTail, vTail, payload, designprob] = addWeightAndSizing(aircraft,
wing, fuselage, hTail, vTail, payload, designprob);

[myAircraft, CruiseState] = createFixedWing(aircraft.Mass, wing, hTail, initialValues);

[aircraft, wing, vTail, designprob] = addStability(aircraft, wing, fuselage, hTail, vTail, payload,
designprob, myAircraft, CruiseState);

[aircraft,designprob] = addPerformance(aircraft, wing.PlanformArea, wing.MeanChord,
designprob);
```

```
emptyAircraftWeight = aircraft.Mass; % Assuming this is the empty weight
takeoffDistance = 10; % Assuming this is the correct field for takeoff distance
wingspan = wing.HalfSpan * 2; % Assuming wingspan is twice the half span
if takeoffDistance < 10
  B = 20;
elseif takeoffDistance >= 10 && takeoffDistance < 25
  B = 15;
elseif takeoffDistance >= 25 && takeoffDistance < 50
  B = 9;
else
  B = 0;
end
weightOfPayload = payload.Boxed.Mass; % Assuming this is the correct field for payload weight
designprob.Objective = 3 * weightOfPayload * (11 / ((emptyAircraftWeight - 1)^4 + 8.9)) + (B - wingspan^1.5);
options = optimoptions(designprob);
options.MaxFunctionEvaluations = Inf;
options.MaxIterations = Inf;
options.StepTolerance = 1e-4;
options.PlotFcn = {'optimplotconstrviolation', 'optimplotfvalconstr'};
options.UseParallel = true;
options.Display = 'off';
[finalValues, maxScore] = solve(designprob, initialValues, 'Options', options);
maxScore
% Assuming that finalValues contains the optimized parameters
disp('Final Optimized Design Parameters (in):');
finalValues=structfun(@(f)39.3701*f, finalValues,'uni',0);
```

```
disp(finalValues);
```

Appendix C - Wing Strength Code

```
% Calculate Lift and Drag Forces
Lift = 0.5 * rho * V.^2 * S * CL;
Drag = 0.5 * rho * V.^2 * S * CD;
% Moment arms for Lift and Drag
lift_arm = wingspan / 2;  % Moment arm for lift (half of
wingspan)
drag_arm = chord;
% Calculate Moments at different velocities
Moment_Lift = Lift * lift_arm;      % Moment due to lift
Moment_Drag = Drag * drag_arm;
Moment_Net = Moment_Lift + Moment_Drag;
% Calculate Section Modulus for Bending Stress Calculation
b = chord;
h = chord;
Z = (b * h^2) / 6;
yield_strength_max = 170;
% Calculate Maximum Bending Moment at Root Using Maximum Lift
M_root = Lift * wingspan / 4;  % Bending moment at wing root
over velocity range
% Calculate Bending Stress at Wing Root
Stress = M_root / Z;
% Find Breaking Point Velocity
break_velocity_idx = find(Stress > yield_strength_max, 1);
break_velocity = V(break_velocity_idx);
mass_lb = 2.43;
```