DJO-1901

Design of an MAV for the 2019 SAE Aero Design Competition

A Major Qualifying Project Report
Submitted to the Faculty of the
WORCESTER POLYTECHNIC INSTITUTE
in Partial Fulfillment of the Requirements for the
Degree of Bachelor of Science
in Aerospace Engineering

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Abstract

The goal of this project was to design a Fixed-Wing Micro Aerial Vehicle in compliance with the rules of the Society of Automotive Engineers 2019 Aero Design Competition. These rules included carrying a payload of 2 inch inner diameter PVC pipe, fitting the unassembled aircraft within a box of dimensions 12.125” x 3.625”x 13.875”, and assembling the aircraft in under 3 minutes. The project team conducted analysis and testing in the areas of aircraft structures, stability and controls, aerodynamics, and propulsion. The final aircraft was constructed with laser-cut plywood and balsa, monokote, carbon fiber, and 3D printed components, weighed 2 pounds unloaded and 3.85 pounds loaded, while still meeting all competition requirements

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Project Advisor	Professor David Olinger
Project Co-Advisor	Professor Raghvendra Cowlagi
Financial Administrator	Barbara Furhman
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Table of Authorship

Section	Authors	Work Completed by
Introduction and Background		
1.1	DC/HC/AF	All
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1.6	DC/HC/ZH	
Design and Analysis		
f2.1	DC/HC/NC/NM/MO	NC/NM
2.2	DC/HC/MO/ZH	DC/ZH
2.3	DC/HC/AF/MO/AW/ZH	MO/AW
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5.5	HC	All
Summary, Conclusions, Recommendations, and Broader Impacts		
6.1	DC	All
6.2	DC/HC/ZH	All
6.3	HC/ZH/MO	ZH
6.4	HC/ZH/MO	MO
Appendix A	N/A	N/A
Appendix B	HC	HC
Appendix C	MO/AW	MO/AW
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Appendix H	MO	MO
Appendix I	ZH	ZH
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1. Introduction

The goal of this Major Qualifying Project (MQP) was to enter a micro class remote controlled (RC) aircraft into the annual aeronautical design competition hosted by the Society of Automotive Engineers (SAE). The Society of Automotive Engineers created this competition to challenge student teams on designing, constructing, and piloting an aircraft that has conflicting design requirements. This gives aeronautical engineering students the ability to apply their theoretical knowledge to a real-world process and become familiar with aircraft design and construction.

The primary goal of the micro class competition is to minimize empty aircraft weight and maximize carried payload weight, while also following rules and restrictions that the SAE set in place. Other goals of this project were governed by constraints in the rules which included size, electronics, and assembly time. Additionally, the aircraft needed to be hand launched and flyable with and without payload.

Eight students, divided into four subgroups, completed this project. The four subgroups were Structures and Integration, Aerodynamics, Propulsion, and Stability and Control. Each group was responsible for detailed research and analysis of its subject. Once the initial analysis was complete, the subgroups all worked together on various parts of the project, regardless of subject, to ensure compatibility.

1.1 Background and Literature Review

The team began background research with the analysis of several previous SAE Micro Air Vehicle (MAV) Design competition entries: Worcester Polytechnic Institute (WPI) entries from 2012 and 2014 as well as the entry from the Military Institute of Science and Technology (MIST) in Bangladesh from 2013 (Blair,J., et al., 2012) (Breault, E., et al., 2014) (Rabbey, M., et al, 2013). The team analyzed these projects closely to assist in understanding the strengths and weaknesses of different MAV designs.

The MIST team concisely laid out the core parameters that a team must consider when designing an MAV, specific to the mission requirements. The MIST team prioritized weight, wing area, and static margin as the most important structural, aerodynamic, and stability aspects for consideration in the SAE competition (Rabbey, M., et al, 2013). The team stressed the importance of iterative design as well as the use of modeling tools such as SolidWorks and ANSYS.

Both 2012 and 2014 MQP teams provided information on aerodynamic and propulsion testing as well as advice on manufacturing considerations for airfoil design. Both teams used a traditional high-wing design for their aircrafts. The SAE rules for 2012 and 2014 were similar with both years requiring the aircraft box having the dimensions of 24" x 18" x 8". The box dimensions were a crucial detail for the team to consider to accomplish the goals of the SAE competition. The 2012 MQP had a final unloaded weight of 0.80 lbs and a loaded weight of 3 lbs (Blair,J., et al., 2012). The 2012 MQP team manufactured the aircraft from balsa wood and carbon fiber spars, and then coated it with UltraCote Lite using Glenn Martin 4 airfoils with a polyhedral wing design (Blair,J., et al., 2012). The aircraft can be seen below in Figure 1. The team finished sixth overall in the SAE Aero Design competition (Blair,J., et al., 2012).

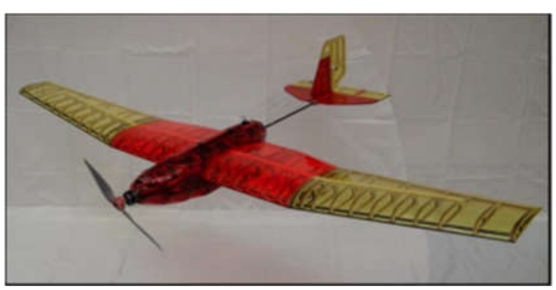

Figure 1. 2012 WPI MQP MAV Aircraft (Blair, J., et al., 2012)

The 2014 MQP team manufactured their aircraft using balsa, aluminum rods, carbon fiber spars, and UltraCote to coat the aircraft (Breault, E., et al., 2014). The team used a Selig 1223 airfoil with a slight dihedral design. The 2014 aircraft was capable of carrying a large payload weight; however, the design needed major modifications last minute therefore the 2014 team did not compete (Breault, E., et al., 2014).

Other reference material included student notes and textbooks for the aerodynamics, aerospace structures, aircraft controls, and aircraft design courses at WPI. The textbooks and class materials provided specific equations and theory for the individual subgroups as well as historical data for design trends (Etkin, B., & Reid, L.D., 1996)(Lennon, A., 1999)(Raymer, D. P., 1992). The aircraft design course notes were helpful in creating a general timeline and layout for the overall design process.

1.2 Project Goals

This MQP aimed to achieve the following goals:

- Design a micro air vehicle (MAV) to compete in the international SAE Aero Design West Competition
  - Minimize empty aircraft weight
  - Maximize payload weight
  - Pack into a box of exterior dimensions 12.125” x 3.625”x 13.875” with a 0.125” wall thickness
  - Reassemble in under 3 minutes
  - Fly the competition course successfully
- Use fundamental theory in areas of Aerodynamics, Structures, Propulsion, and Stability and Controls to design, analyze, and test the various components of the SAE MAV

1.3 Project Design Requirements, Constraints, and Other Considerations

The SAE Aero Design West 2019 competition included a set of rules which guided the team’s goals, design decisions, and stated how the team would obtain points at the competition itself. The full set of competition rules are included in Appendix A. The WPI team followed the rules set for the micro class aircraft. The most confining rules were:

1. All aircraft components must fit into a cardboard box with exterior dimensions of 12.125” x 3.625”x 13.875”, and a minimum wall thickness of 0.125”
2. The payload must be a Schedule 40 White Polyvinyl Chloride (PVC) pipe with a 2 inch inner diameter, with the length depending on how much weight the aircraft can carry.
3. The aircraft must be flight worthy with and without the payload.
4. Only an electric motor is allowed to power the aircraft.
5. The aircraft must be hand launched within 60 seconds, and be able to perform a full 360 degree circuit.
6. The aircraft must be assembled within three minutes.

The micro RC aircraft designed for this competition must follow each of these rules to compete. As shown by the first rule, the aircraft must disassemble to fit within a box of the specified dimensions. This box must contain the entire aircraft and its associated electronics, any payload that the aircraft would carry, and any tools needed to assemble the aircraft. When the PVC payload is in the box with the aircraft and tools, it must remain unobstructed on the inside. This means that the team could not utilize the space within the PVC payload for storing components such as connectors, wires, or carbon fiber rods. Teams could cut the PVC pipe to any length and drill holes in it, but the PVC must otherwise remain unaltered. The battery must also have a separate compartment within the box that is not shared with any other components.

Rules 3 through 6 regard the competition itself. Rule 3 states that the team’s aircraft must be flight worthy regardless of the payload attachment. The rules required that the team complete one flight of three at the competition without the payload. Rule 5 designates the amount of time available for the takeoff launch from the time that the team armed the aircraft, and stated the type of course the aircraft needed to follow. The course involved a straight path of 400 feet, with a 180 degree turn, another straight path of 400 feet, and a final 180 degree turn to a 200-foot landing zone. The final rule stated, rule 6, constrained the complexity of the aircraft assembly process. Since the aircraft and payload needed to fit within a small box, the most difficult constraint was how to disassemble the aircraft so that the team could reassemble it as quickly as possible. Each of these rules played a major role in the design decisions made for the team's competition aircraft.

Before designing an aircraft, the team needed to understand which parameters would affect the score the most. Based on the SAE Aero Design West 2019 rules, the judges would calculate the team's score based on the assembly demonstration and flight rounds. The empty weight of the aircraft and the payload weight contributed to the flight score while the amount of successful flights determined the final flight score. The flight score and assembly demonstration score equations appear below as equations (1.3-1) through equation (1.3-3) where N is the number of successful flight rounds, $W_{payload}$ is the total weight of the PVC pipe payload, W_{empty} is the total empty weight of the aircraft, and *t* is the assembly demonstration time.

$$Final\ Flight\ Score = FSS = 20\,[0.5(\frac{1}{N}\sum_{1}^{N} FS_n + 0.5MAX(FS_n)] + AD \qquad (1.3\text{-}1)$$

Where:

$$FS_n = Flight\ Score_n = \frac{Wpayload}{\sqrt{Wempty}} \qquad (1.3\text{-}2)$$

$$Assembly\ Demonstration = AD = 5(2 - \frac{t}{60})^3 \qquad (1.3\text{-}3)$$

In order to understand the effect of each variable on the final flight score, the team generated a set of graphs in MATLAB to display the effect each variable had. The graphs created by MATLAB appear in Figures 2 through 4 below and the associated MATLAB script appears in Appendix B.

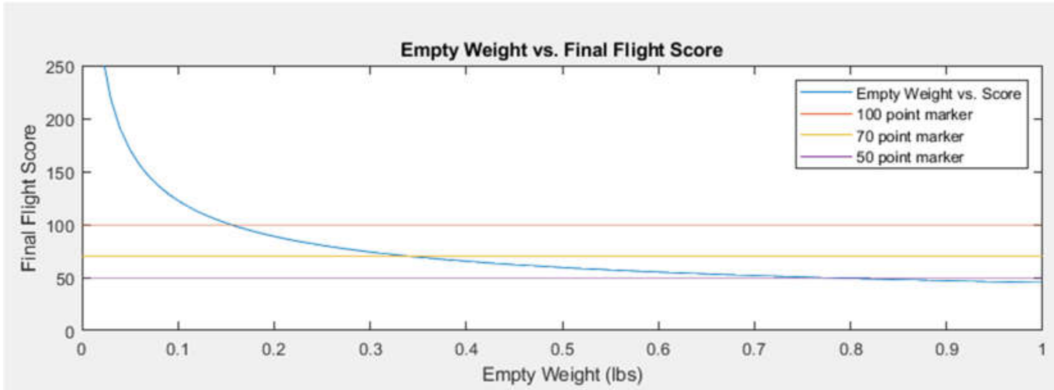

Figure 2. Empty Weight vs Final Flight Score

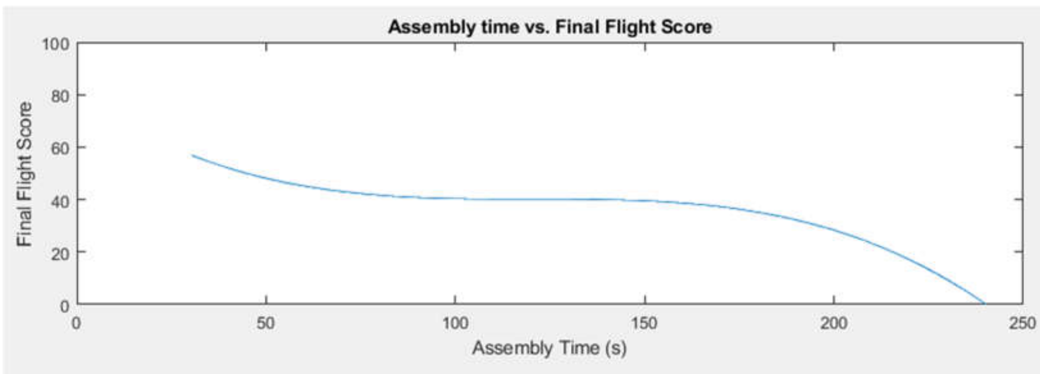

Figure 3. Assembly Time vs Final Flight Score

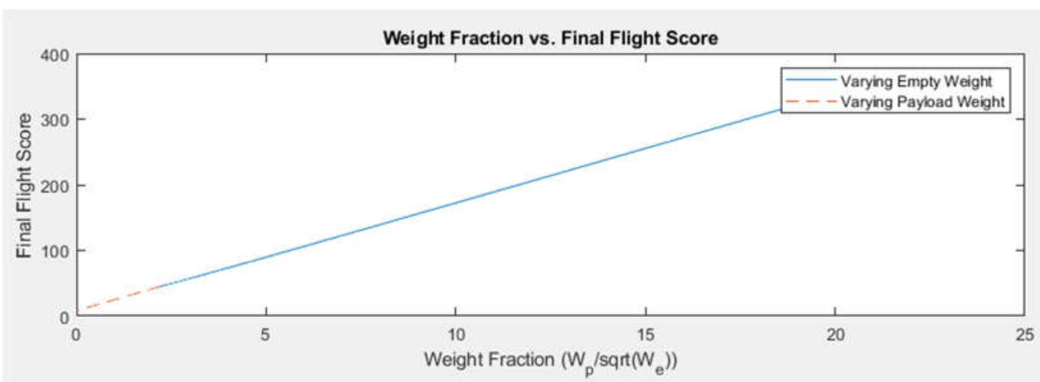

Figure 4. Weight fraction vs Final Flight Score

Figure 2 shows the effect of the empty weight on the final flight score while maintaining a constant payload weight of 2.21 lbs, a constant successful flight number of 3, and a constant assembly time of 60 seconds. Figure 3 shows the effect of the assembly time on the final flight score with a constant empty weight of 1 lb, constant payload weight of 2.21 lbs, and constant successful flight number of 3. Figure 4 shows the combined effect of varying the payload weight and the empty weight. For varying the payload weight, the team assumed a constant empty weight of 1 lb, and for varying the empty weight the team assumed a constant payload weight of 2.21 lbs.

The graph in Figure 4 showed that the empty weight was more important for increasing the final flight score than the payload weight. The team also concluded that the empty weight was more important for increasing the final flight score based on the results in Figure 2. Figure 3 proved to the team that the assembly demonstration should be completed as quickly as possible to maximize the final score. It was interesting to note that there is a plateau in the flight score variance between the assembly times of 100 to 150 seconds, thus providing a "buffer zone" should the assembly time reach that point.

Designing this project required focus on the four subject areas of aerodynamics, structures, propulsion, and stability and controls. These four subject areas were critical to ensuring that the entire aircraft system would perform as desired to satisfy the constraints of the competition. The main constraint on the project was the size of the box for the disassembled aircraft. The dimension constraint affected each subject area differently. For example, aerodynamics was limited by the size of the box because the height dimension constrained the amount of camber on the airfoil. Ideally, the wing sections would be able to stack on top of each other in the box, but this is not possible with a large camber. Another constraint from the competition rules was the limitation on the size and type of battery for powering the aircraft. Due to this constraint, the propulsion system and electronic equipment needed to operate safely using a 3-cell lithium polymer battery. These four critical subject areas dictated the method used for subdividing the project team.

1.4 Project Management

The team divided into four subgroups corresponding to the four critical subject areas. These four subgroups were Structures and Integration, Aerodynamics, Propulsion, and Stability and Controls. The structures and integration group, consisting of Nick Cunha and Nick Manos, was responsible for the structural design, analysis, and integrity of the aircraft, as well as the CAD analysis and modeling. The aerodynamics group, consisting of Diana Celaj and Zachary Hearl, was responsible for the airfoil, wing, and tail design, and lift and drag analysis of the aircraft. The propulsion group, consisting of Michael Oswald and Amelia Wilson, was responsible for the motor, battery, and propeller selection, thrust and drag output of the motor, and necessary wind tunnel testing. The stability and controls group, consisting of Heather Cummings and Adam Frewin, was responsible for the static and dynamic stability of the aircraft, elevator trim analysis, and flight control system. Although there were specific roles and focuses, all subgroups worked cohesively and remained updated on considerations and decisions. This allowed every group member to be aware of the effect any decision had on all subgroups.

1.5 MQP Objectives, Methods and Standards

In order to meet the various objectives listed in section 1.2 and the SAE Aero Design competition rules, the team relied on several software packages to analyze motors, airfoils, stability derivatives, and structural components of the aircraft. Sample results and outputs for the developed tools and software packages are presented in the following sections. More detail is provided on these results in later design and analysis sections.

1.5.1 Tool Development

1.5.1.1 MotoCalc 8

MotoCalc 8 is an analysis tool developed for RC aircraft hobbyists to determine the power plant that would best suit the characteristics of the aircraft they want to design. This tool can compute both static and in-flight predictions on thrust, drag, power, and efficiency for a large database of motors, batteries, and propellers. To operate this tool, users first select either the motor, battery, propeller, speed controller, or gearbox. The software then computes thrust, power draw, current draw voltage, motor RPM, and various efficiencies. Additionally, users can input characteristics of their aircraft, the type of flying they intend to do such as sedate, race, or sailplane, and any restrictions they have on the power plant to determine a suitable motor for the design.

To verify that the thrust outputs given by MotoCalc 8 were accurate, the team sought to replicate the thrust and drag values from wind tunnel testing contained in the 2012 MQP report. Using the information in the 2012 MQP report, the propulsion team determined the theoretical drag and thrust through the use of MotoCalc 8. A sample of the results of the 2012 wind tunnel test and newly calculated theoretical results can be seen in Figure 5. From this figure it appears that MotoCalc 8 is fairly accurate in calculating both static thrust and drag at a given airspeed.

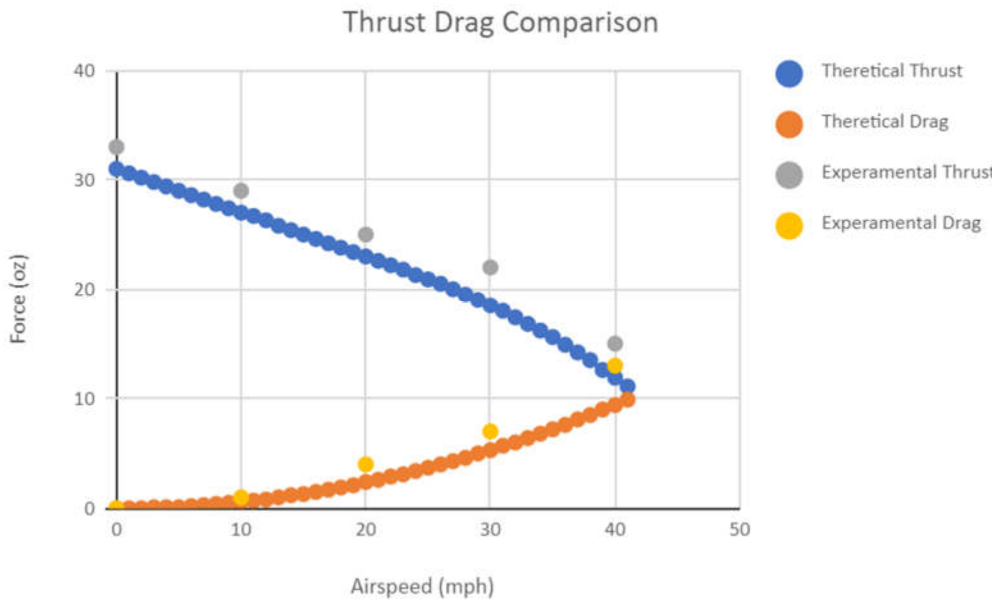

Figure 5. 2012 Sample Wind Tunnel Results vs MotoCalc 8

The necessity of this propulsion tool came from the inaccuracies found from a numerical approach. A general form of the thrust equation as shown by equation 1.5-1 is a function of mass flow rate, velocity before the propeller, velocity after the propeller, air pressure before the propeller, air pressure after the propeller, and the area swept out by the propeller.

$$T = \dot{m}\left(V_{exit} - V_{airspeed}\right) + A\left(P_{exit} - P_{in}\right) \qquad (1.5\text{-}1)$$

Upon simplifying equation 1.5-1 and assuming that pitch speed is equal to the exit velocity and that the exit pressure equals the entrance pressure, Gabriel Written derived the following formula shown in equation 1.5-2 (Written, 2014). In addition, this derivation assumes that RPM is constant at every airspeed.

$$T = 4.392399 * 10^{-8} * RPM * \frac{d^{3.5}}{\sqrt{pitch}}\left(4.23333 * 10^{-4} * RPM * pitch - V_{airspeed}\right) \qquad (1.5\text{-}2)$$

Using this equation to recreate the test data from the 2012 MQP, the propulsion team determined that this equation is accurate for static thrust, but becomes progressively more inaccurate as the airspeed increases due to the initial assumptions. The inaccuracy caused by these initial assumptions is shown in Figure 6. As result, equation 1.5-2 was not used for the project and the team switched to using the MotoCalc 8 tool described previously.

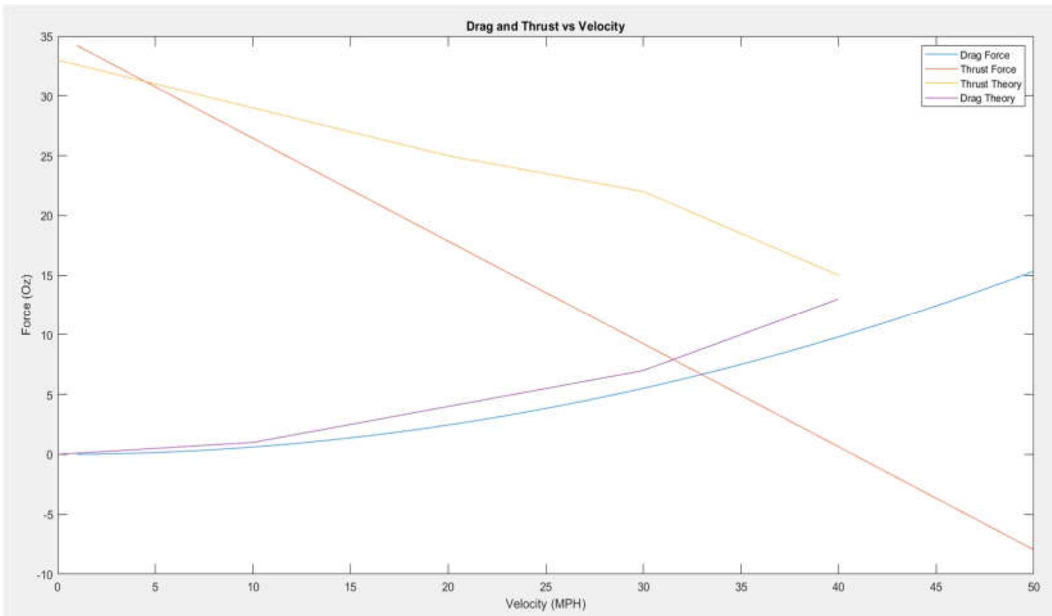

Figure 6. 2012 Sample Wind Tunnel Results vs Derived Thrust Equation

1.5.1.2 eCalc

Like MotoCalc 8, eCalc is an analysis tool developed for RC aircraft hobbyists to optimize their power plant for their aircraft. Unlike MotoCalc 8, eCalc can provide analysis for air vehicles beyond fixed wing aircraft. For example, there are versions of eCalc for airplanes, jets, and multirotor vehicles. In the aircraft variant, the user inputs basic aircraft information such as weight, number of motors, wing area, atmospheric data, battery information, controller or ESC information, motor selection, and propeller selection. The software then outputs information regarding current pull, thrust, power, engine temperature, runtime, and efficiency, to name a few. The software also provides error messages regarding pulling too much current, exceeding the motor's power rating, stalling the propeller, or having too slow of a pitch speed, to name a few.

In addition, the user can also input a cruise speed to get the same output variables for various cruising conditions. A sample of the interface appears in Figure 7.

Figure 7. Sample eCalc Display (eCalc, n.d.)

1.5.1.3 MATLAB

The stability and controls team used MATLAB for many calculations. MATLAB is a matrix based programming platform that allows for complex iterative calculations. This proved useful for analyzing the stability of the aircraft and for generating graphs of dynamic stability characteristics. The team also used MATLAB for trade studies throughout the project involving score optimization, subgroup calculations, and the initial takeoff simulation before it was converted to Python.

1.5.1.4 Solidworks

In order to create scale visual representations of the aircraft with proper dimensioning, the structures subgroup used the computer aided design software Solidworks. Solidworks allowed the team to created solid models of each piece on the aircraft and generate full assemblies to show how the aircraft would look in an unloaded, loaded, and packed configuration. This software was useful for visualizing the design concepts and for generating solid models of parts for 3D printing. The Solidworks models were also used to create dimensioned drawings of the aircraft's final design.

1.5.1.5 XFLR5

XFLR5 is a software package for airfoils, wings, and planes operating at subsonic speeds. The analysis tools are based on lifting-line theory, vortex panel method, and 3D panel method. The software is also capable of performing stability analyses for fixed-wing aircraft. XFLR5 was used to determine many of the aerodynamic coefficients for both airfoils and 3D wings, using its database of NACA airfoils. XFLR5 was also used in conjunction with analyses in MATLAB for alternative methods of determining and cross-checking stability characteristics. Further documentation on XFLR5 can be found at www.xflr5.com.

1.6 MQP Tasks and Timetable

The SAE rules and the time constraints of completing the project led the team to set forth a timeline for major milestones. The project timeline appears in Table 1.

Table 1. Project Timeline

A Term	B Term	C Term	D Term
Completion of initial aerodynamic and structural design	Development and design iterations with a glide test of an initial prototype	Modification of design and successful flight test	Competition and Project Presentation Day

The team dedicated A term to research and design, in which the team went through multiple preliminary design options and specifications. The team analyzed each option based on score, assembly, packing, and feasibility of construction. The team chose a final design at the end of A term to begin developing and testing.

The team updated and refined the designs following more research and tests in B term. The tests in B term involved wind tunnel testing, thrust tests, and a glide test in the final week of the term. Each test allowed the team to modify the original design to render the aircraft flight worthy and optimize score and packing.

In C Term the team completed a powered test of the aircraft to finalize the controls of the aircraft and electronic durability. Following a successful powered test, the team prepared for the assembly and flight demonstrations at the competition.

Based on the deadlines from both the SAE rules and project advisors, the team created a Gantt chart to stay on task. This is presented in Figure 8.

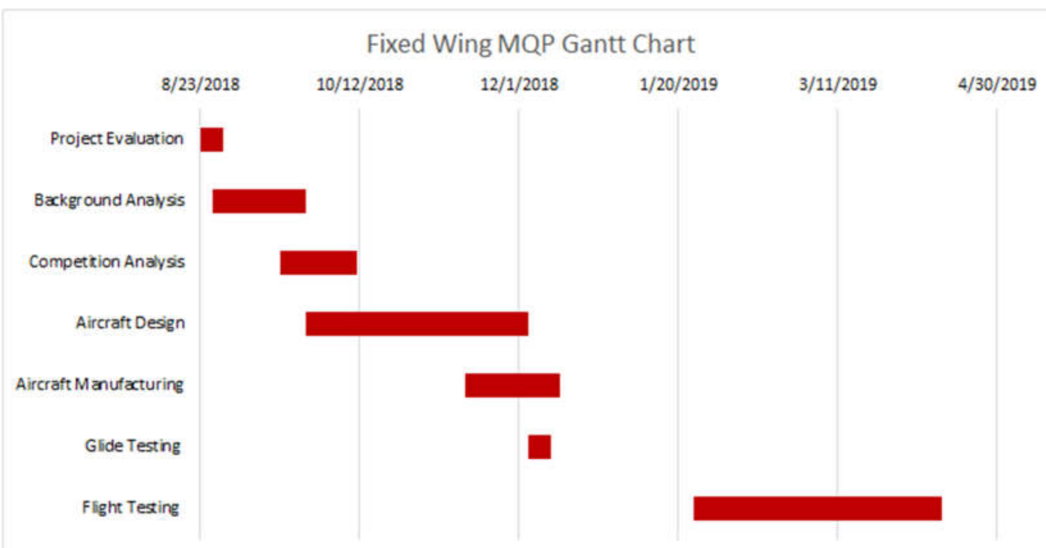

Figure 8. Gantt chart schedule from 8/23/19 until 4/13/19

2. Design and Analysis

2.1 Design Process

The initial design of the aircraft began in A term. Throughout the design process, the team aimed to maximize flight score by creating the lightest aircraft that could carry the most payload. The payload for the 2018-19 competition was 2 inch inner diameter, schedule 40 polyvinyl chloride (PVC) pipe. Teams could cut the PVC pipe to any length and drill holes in it, but the PVC pipe must otherwise remain unaltered. The payload could attach anywhere on the aircraft, but must remain unobstructed while packed. Based on the limitation of 12 x 3.5 x 13.5 inch box, or 567 inches^3 volume, coupled with the additional requirements of the PVC pipe, the team realized how important the volume constraint and packing efficiency would be for the design. With the packing efficiency and flight requirements in mind, a number of designs were proposed for the configuration of the aircraft.

2.1.1 Initial Configuration Design

One flight requirement of the competition stated that the aircraft must successfully fly with and without the payload attached. This meant that the center of gravity (CG) of the aircraft should not shift when the payload was added. Based on this requirement, the team initially looked into two structural designs; one design would carry sections of PVC pipe within the fuselage, and the second design would have sections of PVC pipe mounted externally on the wings of the aircraft. Figure 9 shows the design with the PVC pipe carried within the fuselage. Figure 10 shows the design with the PVC pipe attached under the wings of the aircraft.

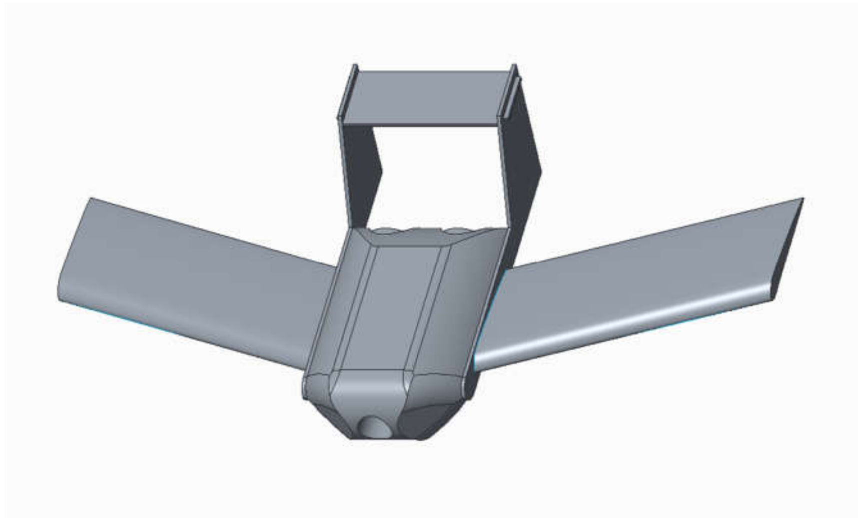

Figure 9. Initial aircraft design with PVC pipe carried inside the fuselage

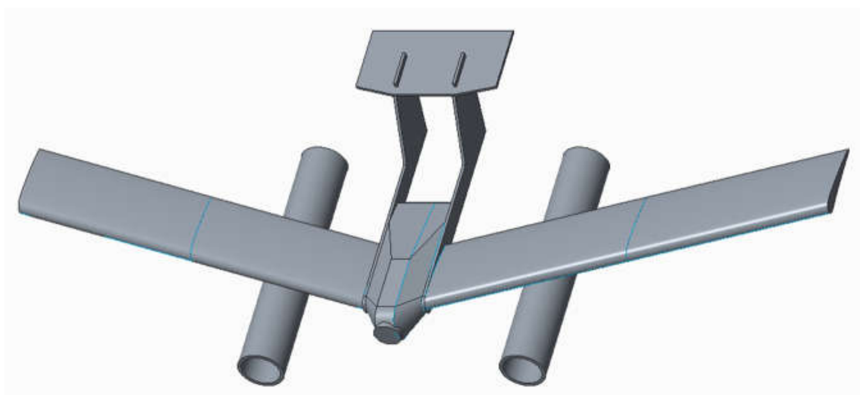

Figure 10. Initial aircraft design with PVC pipe carried under the wings of the aircraft

As referenced earlier, another important rule to recognize was that, based on the SAE 2019 design requirements, the inside of the payload PVC pipes must remain unobstructed while stored inside the box. Because of this requirement, it was important to consider the dimensions of the PVC pipe. The amount of stored payload was the largest determining factor for the maximum size of the aircraft versus the amount of payload the aircraft could carry. Packing configurations for the two proposed models appear in Figures 11 and 12.

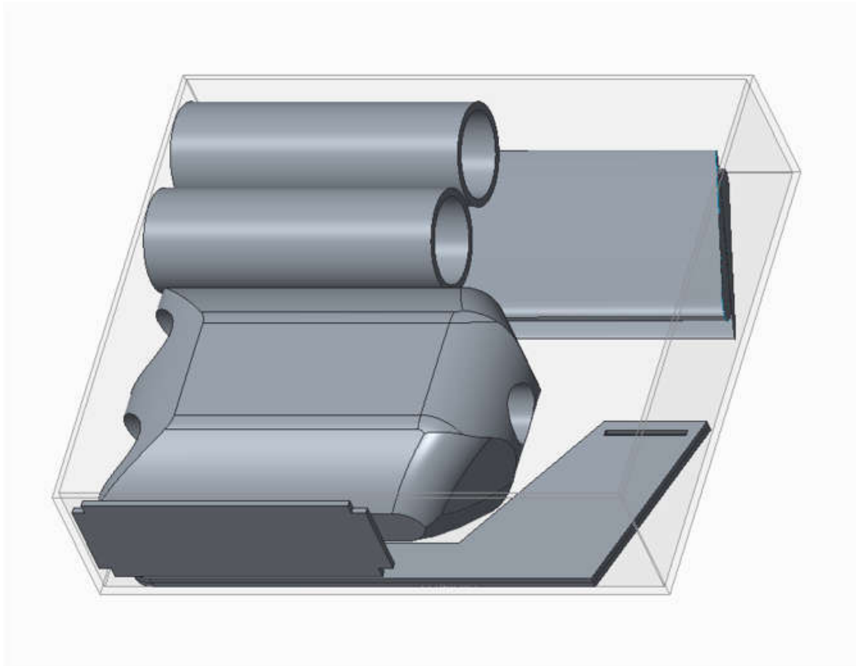

Figure 11. Packing configuration for model with payload PVC pipes carried inside the fuselage

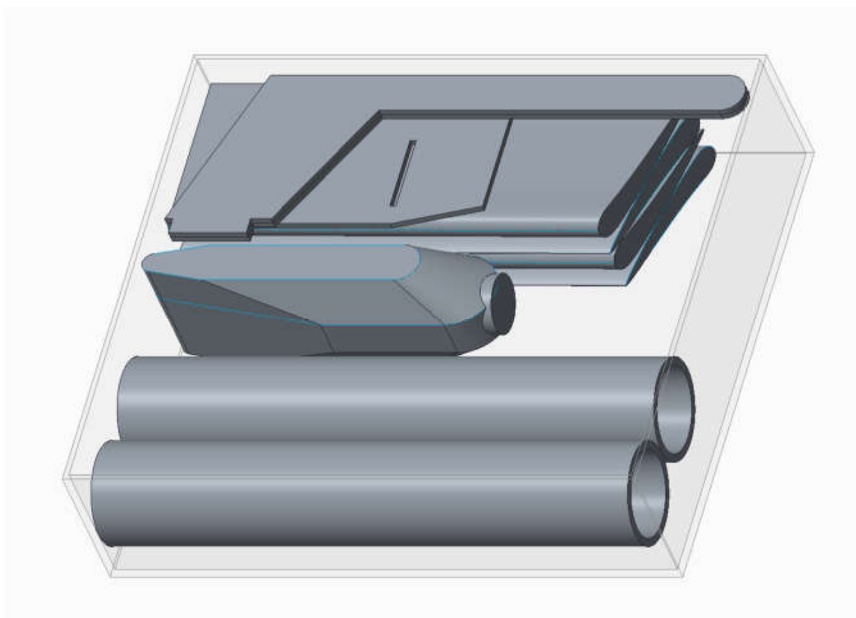

Figure 12. Packing configuration for model with payload PVC pipes carried under the wings

2.1.2 Final Configuration

Following discussion on the original two designs, the team proposed a new design which incorporated features of both the in body and under wing preliminary designs. This design included a skeletal fuselage that the PVC pipe payload could slide over in place of a fuselage skin. This third design was superior due to the fact that the weight of the unloaded aircraft was less than the other two designs, which maximized potential score. The minimal structure of the aircraft allowed for the potential to pack at least two 13 inch sections of PVC pipe in the storage box. Mounts for the battery and electronics also attached to the fuselage skeleton so that the aircraft remained in operation with or without the PVC payload, as required by the competition rules. Figure 13 below shows the final configuration of the aircraft.

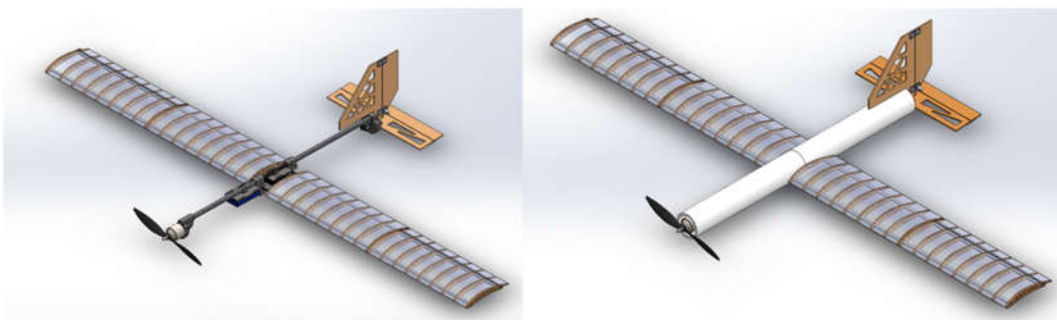

Figure 13. Final unloaded (left) and loaded (right) aircraft design

The fuselage had a single central spine of tubular twill wrap carbon fiber which split into three spines for packing. As seen in the figure above, the PVC pipe slides over the spine and is fixed in place with the wing spars. The front spine assembly consisted of the motor, nosecone, a section of carbon fiber tube, and a joint. The center spine assembly was made of a smaller diameter carbon fiber tube that fits within the front and rear tubes and brackets that both held the wing spars and connected the center to the front and rear assemblies. The rear spine assembly had identical connectors and spine tubing to the front assembly and a rear tail cap that held both the tail control servos and the tail itself. All of the components of the aircraft and their names can be see laid out below in Figure 14 and Table 2 below.

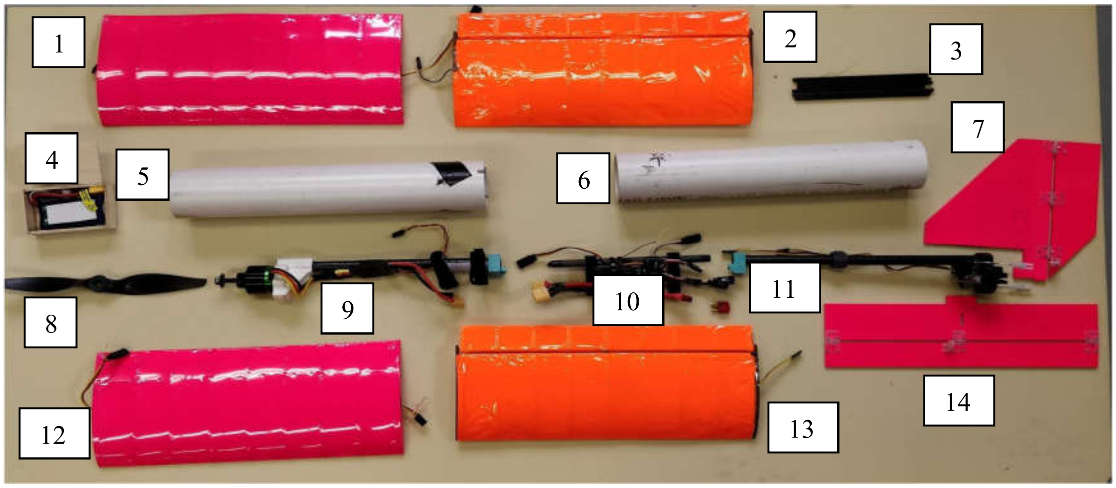

Figure 14. Final Component Layout

Table 2. Component Names

Label number	**Component Name**
1	left inner wing panel
2	left outer wing panel
3	wing connection spars
4	battery and containment box
5	front payload section
6	rear payload section
7	vertical tail
8	propeller
9	front body segment
10	center body segment
11	rear body segment
12	right inner wing panel
13	right outer wing panel
14	horizontal tail

All of the components laid out above fit into the box as required by the competition rules. The final packing configuration is shown below in Figure 15. Table 3 provides a list of the final specifications for the designed aircraft.

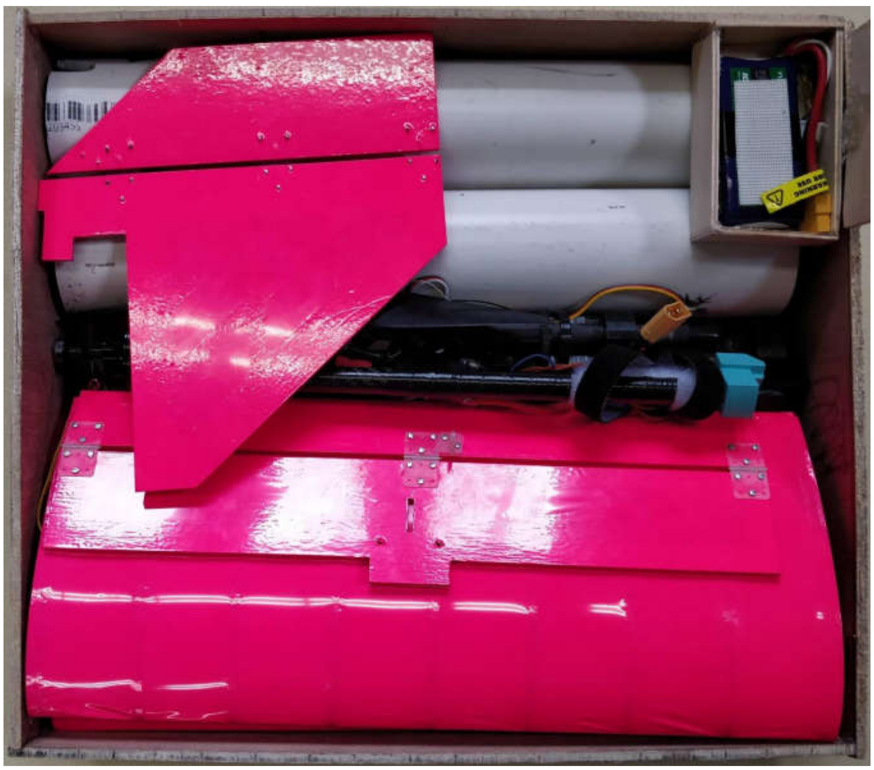

Figure 15. Final Packing Configuration

Table 3. Final Design Specifications

Characteristic	**Value**
Empty Weight	2 lbs
Max Weight	3.85 lbs
Length	30.23 in
Height	7.88 in
Cruise Speed	30 mph
Max Payload Fraction	0.48
Wing	
Airfoil	NACA 9410
Wing Span	4.33 ft
Aspect Ratio	9.45
Wing Area	1.986 sq.ft
Chord	0.4583 ft
Taper Ratio	1
Tail	
Incidence Angle	-4 degrees
VT Volume Ratio	0.03
VT Area	0.2381 sq.ft
HT Area	0.3583 sq.ft
Aspect Ratio	3.87
Propulsion	
Motor	Cobra 2814-12
Propeller	APC-E 9x6
Max Thrust	55 ozf
Battery Life Max Power	1.51 min
Battery Life Cruise Power	4.26 min

2.2 Aerodynamics

In order to select the proper airfoil for the aircraft, the team needed to calculate the required coefficient of lift to carry the desired payload weight. After researching past MQPs and competition winners, the team decided on baseline design specifications for use in preliminary calculations. The team derived these variables from the most effective values for past teams and what seemed attainable for the 2018-2019 design. The values appear in Table 4 below.

Table 4. Base Aircraft Parameters

Empty Weight	1 lb
Total Weight	4 lbs
Cruise Speed	44 ft/s
Maximum Speed	58.7 ft/s
Aspect Ratio	8-9
Maximum Reynolds Number	200,000

The team used an aspect ratio from literature review of past teams as well as an estimated wing span to calculate a preliminary chord length. The equation below was used for this calculation, where AR is the aspect ratio, s is the wingspan and c is the chord length.

$$AR = \frac{s}{c} \qquad (2.2\text{-}1)$$

This resulted in a chord length of 6 inches. However, the team decided that this would result in too large of a wing area for the box that all the components must fit into. The team then decreased and set the final chord length to 5.5 inches.

With this chord length and the desired aspect ratio shown in the table, the team calculated a wing area of 286 square inches. The team calculated a preliminary coefficient of lift of 0.875

using equation 2.2-2 below, where L is the lift required (the total weight of the aircraft), C_L is the coefficient of lift, ρ is the density of air, V is the cruise speed of the aircraft, and A is the wing area.

$$C_L = \frac{2L}{\rho V^2 A} = 0.875 \quad (2.2\text{-}2)$$

2.2.1 Airfoil and Wing Selection

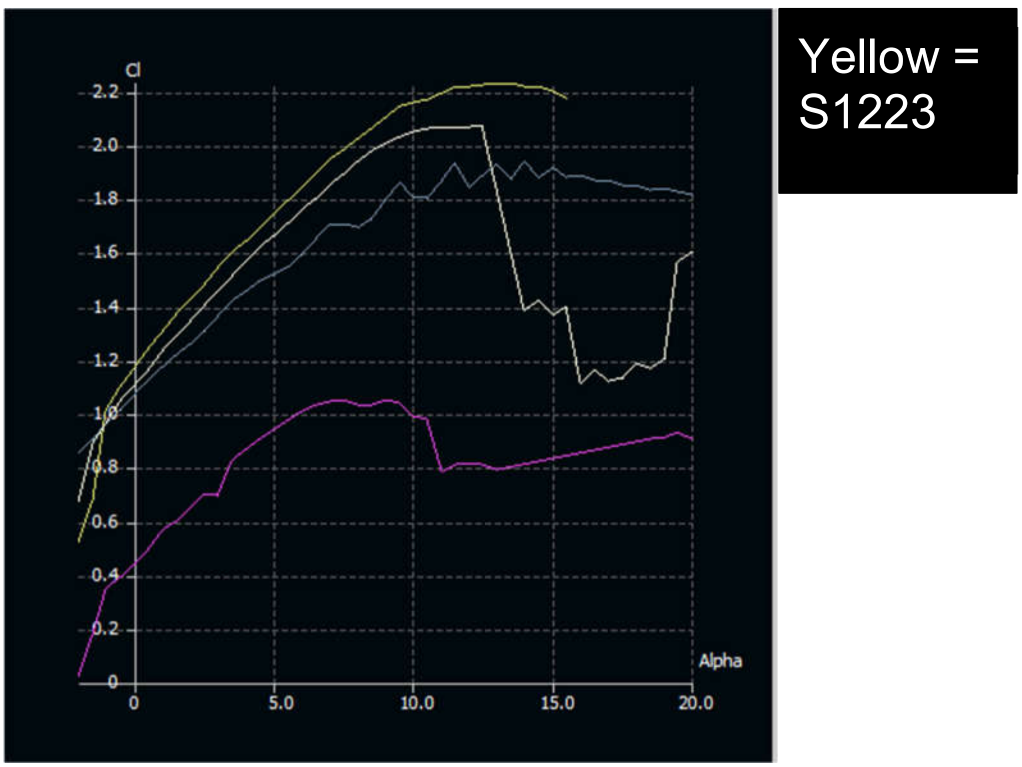

Figure 16. XFLR5 Cl vs alpha(°)

The team used the calculated coefficient of lift to research appropriate airfoils that were viable for MAV and RC vehicles. The aerodynamics team initially chose three airfoils to analyze: S1223, S1223 RTL, and E423. The aerodynamics team also analyzed the Glenn Martin 4, which was used by the 2012 MQP Project, for comparison. The C_L behavior of these four airfoils appears in Figure 16 below as generated by XFLR5.

Although the S1223 and the S1223 RTL displayed the highest coefficient of lift versus angle of attack, these airfoils have trailing edges nearing one-one thousands of an inch which is thin enough to cause manufacturing difficulties and structural concerns. The Glenn Martin 4 had a much thicker shape thus adding weight which the airfoil could not overcome due to its lower coefficient of lift. The thicker shape would restrict the amount of room in the box for payload and the rest of the aircraft. The E423 was ultimately chosen as the preliminary airfoil as it had a high coefficient of lift while being thick enough to feasibly manufacture.

Analysis continued with the E423, however multiple specification changes were made that affected the preliminary airfoil choice. The estimated total payload of the PVC sections reduced the available volume within the box. Additionally, the team calculated the maximum allowable wing sizes to ensure that all other components also fit within the box dimensions. The team continued calculations and analysis with the specifications shown in Table 5.

Table 5. Updated Test Specifications

Characteristic	Value
Total Weight	4.0 lbs
Wing Area	1.96 ft^2 (286 in^2)
Chord	0.458 ft (5.5 in)
Aspect Ratio	9.45
Maximum Reynolds	180343
Cruise Reynolds	123974
Cruise Velocity	44 ft/s
Air Density	0.0765 lbm/ft^3

While the E423 had satisfactory lift characteristics, it posed multiple challenges that would hinder the overall aircraft performance. First, the large camber of the airfoil created a large void between wing sections while packed inside the box, thus reducing total payload capacity and restricting the available space for other necessary components. The space loss could be mitigated with alternative packing considerations but reducing the camber would flatten the airfoil over the payload in the box. Second, the cambered shape of the airfoil produced a large negative moment coefficient, as shown in Figure 17. The large moment coefficient would require a larger incidence angle on the horizontal tail.

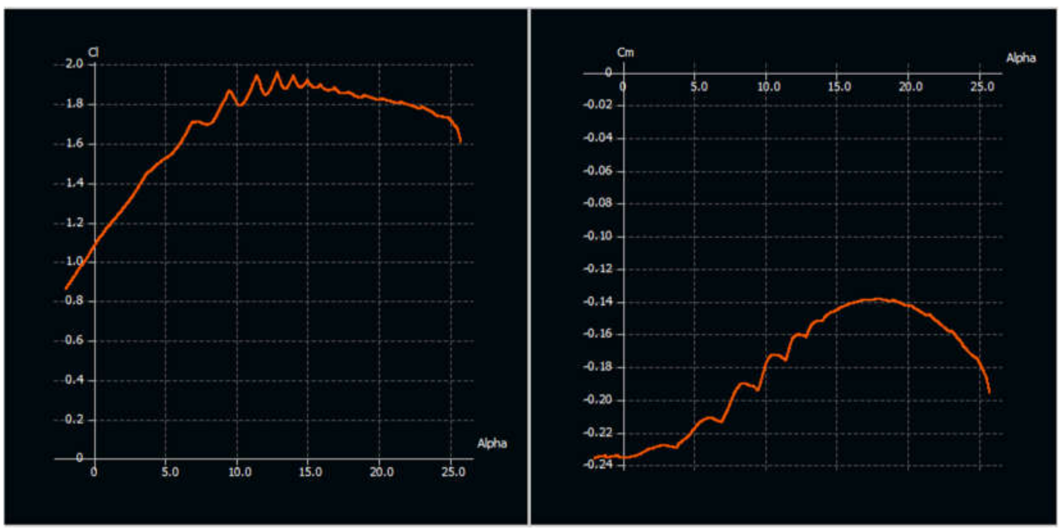

Figure 17. E423 Characteristics at Re=200,000 from XFLR5

Through closer examination of the packing efficiency and performance, the team decided that the camber was too large. The team used the E423 as a baseline for deciding on desirable changes, and chose the NACA-4412, NACA-4410 shown in Figure 18, and the N-22 for further analysis.

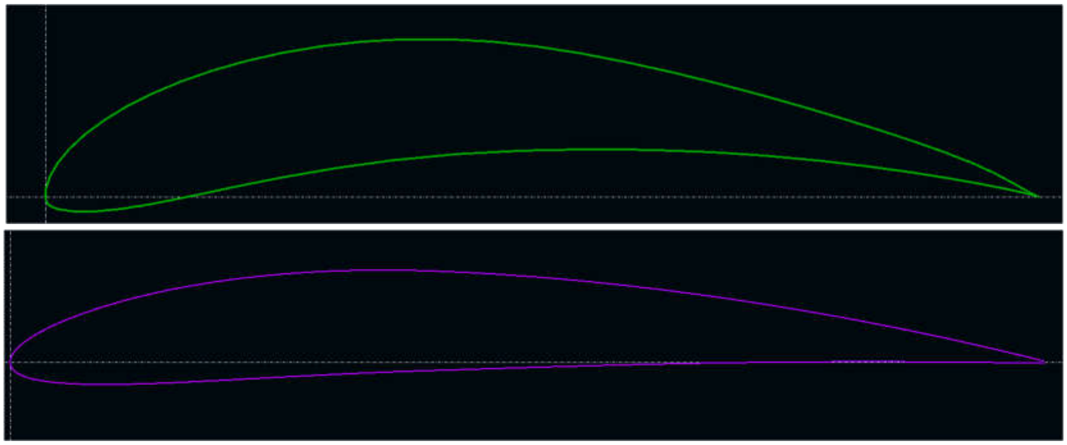

Figure 18. E423 Airfoil (Top) vs NACA 4410 (Bottom)

Table 6 contains data on multiple airfoil properties such as coefficient of lift and drag for various airfoils. The XFLR5 software determined these values using airfoil coordinates and testing them using the variables in table 5. The team used the XFLR5 software to numerically solve for the coefficient of lift, coefficient of drag, and maximum angle of attack. With these values, the lift produced from the wing and the stall velocity were determined using equations 2.2-3 and 2.2-4. The calculated lift used a coefficient of lift value at zero angle of attack, while the stall velocity equation uses the maximum lift coefficient.

Table 6. Airfoil comparison

Airfoil	Cl (alpha=0)	Cl_max	C_d(alpha=0)	Lift Produced by each wing(lbs)	V_stall (ft/s)	Max AoA (Degrees)
Glenn Martin 4	0.443	1.055	0.0165	2.06856807	41.37033287	10
S1223	1.177	2.238	0.0182	5.495947219	28.4043728	15
E423	1.074	1.9	0.0205	5.014993469	30.82750981	15
S1223RTL	1.106	2.078	0.0205	5.164415993	29.4776239	13
NACA 4410	0.46	1.4	0.0112	2.147948786	35.91296211	14
N-22	0.63	1.3	0.011	2.941755946	37.26864207	12

$$L = \frac{1}{2}\rho S V^2 C_l \qquad (2.2\text{-}3)$$

$$V_{stall} = \sqrt{\frac{2Lg}{\rho S C_{l,Max}}} \qquad (2.2\text{-}4)$$

The new wing design for the aircraft allowed for four sections of the NACA 4410 airfoil to fit within the specified box with a total wingspan of 52 inches (13.3 inches for each section) and a chord of 5.5 inches. These wing specifications yield an aspect ratio of 9.45 which was greater than the initial estimate but provided the necessary lift for the expected payload weight. In this configuration, the airfoil lays flat inside the box. Having the wing in this configuration allowed for a maximized wing area using the NACA 4410.

2.2.2 Updated Airfoil and Wing Selection

Issues arose with the previously selected airfoil, the NACA 4410, once testing and analysis began. The NACA 4410 had a maximum coefficient of lift value of 1.4 with the aircraft's wing area at the set cruise Reynolds number. This value gave a stall speed of 24 mph, which would be very difficult to exceed with a hand launch on account of the shape of the aircraft and the necessary angle of attack. It was desired to lower the stall speed to 18 mph or less, which required a maximum coefficient of lift of 2.3. The team made the decision to change the airfoil again because the team decided that cutting payload weight off the aircraft should be a last resort. The team ultimately decided to prioritize the need to produce more lift with a high lift, higher-cambered airfoil and to find an alternative solution to the cambered wings not fitting in the box. The higher-camber airfoil chosen was the NACA 9410, which had a similar thickness to the NACA 4410 and similar camber to the E423. The chosen airfoil can be seen in Figure 19 below.

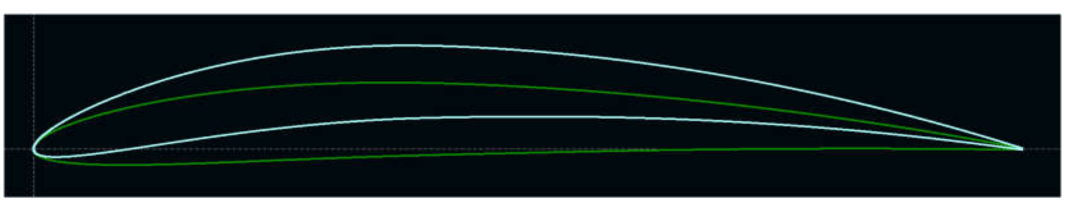

Figure 19. XFLR5 view of NACA 9410 (white) and NACA 4410 (green)

The NACA 9410 produced similar effects to the E423 but was slightly thinner which saved on both weight and space. The coefficient of lift and coefficient of pitching moment characteristics of this airfoil can be seen below in Figure 20.

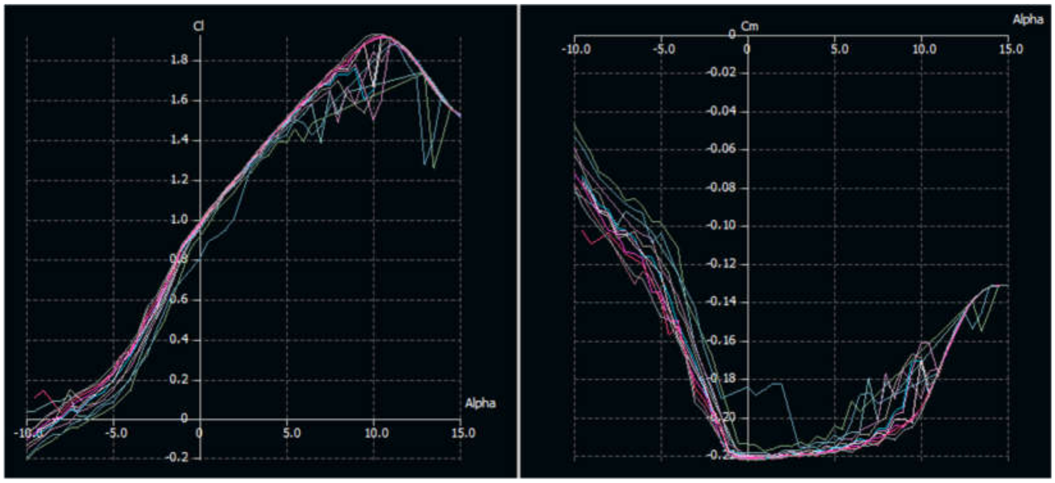

Figure 20. XFLR5 plots of NACA 9410

The NACA 9410 reaches a maximum coefficient of lift of 1.9 at an angle attack of 10°. In order to achieve the needed C_{Lmax}, the team decided to add flaperons for increased lift performance at maximum payload. Flaperons are a combination of ailerons for roll control and flaps for increased lift on takeoff. The addition of flaperons increased the $C_{L\,max}$of the airfoil to 2.17. Equation 2.2-5 below shows the calculation of the necessary lift coefficient change for the use of flaps with $\Delta C_{L\,max\,base}$being 0.8 for a plain flap with a maximum thickness at 10% of the chord. The k coefficients were determined from figures 8.11-8.14 in *High Lift Systems and Maximum Lift Coefficients* notes from the Hamburg University of Applied Sciences (Scholz, D., 2015). The team calculated the new stall speed for the aircraft using flaperons to be 18 mph which satisfied the team's requirement.

$$\Delta C_{L\,max\,flaps} = k_1 k_2 k_3 (\Delta C_{L\,max})_{base} \qquad (2.2\text{-}5)$$

2.3 Propulsion

The next step in the design process involved the power plant selection and choice of associated electrical components for use on the aircraft. The power plant consisted of the motor, propeller, battery, electronic speed controller (ESC), and receiver. A detailed description of the function of each component appears in Appendix C.

2.3.1 Power Plant Selection

For the power plant, the team established a baseline for the weights of the electronic components compared to the overall weight of the aircraft. Table 7 summarizes the weight of the electronic components of the 2012 MQP to estimate the baseline.

Table 7. 2012 Electronic Components

Part	Make/Model	Weight (lbs)
Motor	E-Flite Park Flyer, 1360kV	0.109
Propeller	10x5, with a prop saver	0.012
Servos	Two Hitec 65MG Micro Servo (metal gears)	0.050
Electronic Speed Controller	eRC 25A programmable	0.049
Battery	Tenergy 11.1V 900 mAh 25C	0.141
Transmitter/Receiver	Spektrum DX5e Transmitter with Spektrum AR600 Five-Channel Receiver	Receiver: 0.012

The total weight of the components on the 2012 aircraft was 0.373 lb, which constituted just under half of the 0.881 lb empty weight of the aircraft. The propulsion team considered this ratio while researching electronic components for the 2019 aircraft. The team noticed that the combined weight of the motor and battery for the 2012 team was 0.25 pounds, which was approximately two thirds of the overall empty weight. Therefore, the team began looking into different motors and batteries that minimized the electronics weight while still providing enough power and thrust. Table 8 provides a list of all electrical components used in the design for the 2019 aircraft.

Table 8. Propulsion Components

Component	Use	Quantity	Weight Per Item (lb)
Turnigy Park 480 (1320)	Motor	1	0.231
Turnigy Nano-Tech 1000mAh 3S 40C Lipo	Battery	1	0.193
Aerostar Carbon Fiber Propeller 8x6	Propeller	1	0.048
Talon 35 Castle	ESC	1	0.061
CC-BEC 10A	BEC	1	0.033
S8R	Receiver	1	0.026
HS-40	Rudder/Elevator Servo	2	0.010
D145SW	Aileron Servo	2	0.052
Deans Plug	Arming Plug	1	0.090

Combined, these items have a total weight of 0.72 pounds. This total was 72% of the aircraft's goal weight of 1 pound. Due to the large fraction of weight taken by the electronics, the estimated actual empty weight of the aircraft increased to 1.25 pounds. Despite the attempt to reduce weight, it was impossible to lower the electronic weight to the weight of the 2012 MQP due to the loaded goal weight of 4.26 pounds compared to the 3 pound loaded weight of the 2012 MAV. The larger motor on the aircraft required a larger battery, ESC, and more wiring. In addition, due to the need for flaperons, larger servos were required. In order to determine a more accurate comparison of weights, the team looked into percentage of loaded weights. When compared to the loaded weights, the 2012 MAV electronics made up 12.4% of the total loaded weight. This project's electronics constitute 16.9% of the total loaded weight. This is a 4.5% increase in weight percentage, however the team determined it necessary in order to have a flying aircraft. The following sections discuss the decision-making process for each electronic system.

2.3.2 Motor and Propeller Size Optimization

In order to select an appropriate motor, the team started with the initial assumption that the aircraft would have a cruise speed of 30 mph (44 ft/s) and require a static thrust of 30 to 40 oz. The propulsion team determined these values from analysis of the 2012 MQP team and initial drag values from aerodynamic analysis. The propulsion team researched initial motors and propeller sizes that could be used for the 2019 aircraft using those initial values. Table 9 shows a comparison of potential motors.

Table 9. Motor Comparison (Park 370 Brushless Outrunner Motor, 1360Kv., n.d.) (Park 450 Brushless Outrunner Motor, 890Kv., n.d.) (Park 480 Brushless Outrunner Motor, 1020Kv., n.d.) (Park 480 Brushless Outrunner Motor, 910Kv., n.d.) (Turnigy Park480 Brushless Outrunner 1320kv., n.d.)

Motor	**E-flite Park 480**	**E-flite Park 480**	**E-flite Park 450**	**E-flite Park 370**	**Turnigy Park 480**
Idle Current	1.10A @ 8V	.85A @ 8V	.70A @ 8V	1.00A @ 10V	0.9A @ 11V
Kv	1020	910	890	1360	1320
Speed Control	25A-40A	35A	20A	10A-20A	30A-35A
Voltage	7.2-12	7.2-12	7.2-12	7.2-12	7.4-11.1V
Weight (oz)	3.1	3.1	2.5	1.7	2.8
Propeller	12x6	12x6	11x3.8	10x4.7	8x6/9x5

Using the MotoCalc 8 software, the propulsion team calculated initial thrust data for each of the motor-propeller combinations mentioned in Table 9. This data is presented in Figure 21.

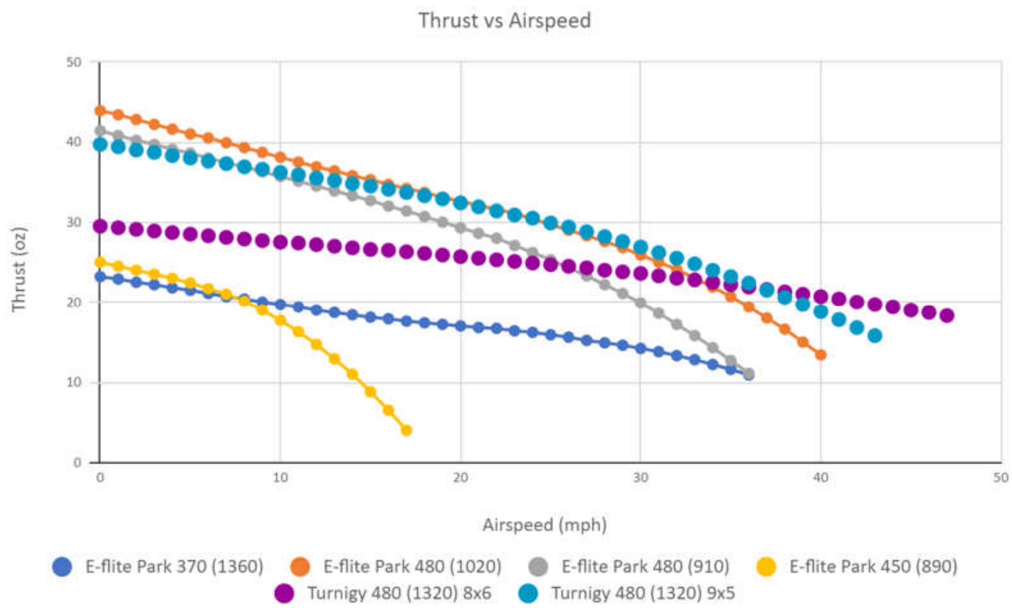

Figure 21. Motor Thrust Comparison

From this data, the team initially disregarded the E-flite Park 450 and the E-flite Park 370 motors due to their low thrust outputs. This left the Park 480 Series motors from Turnigy and E-flite. Ultimately the team chose the Turnigy motor because it weighed less than both of the E-flite motors. Additionally, the E-flite motors required a larger propeller for minimal thrust gains.

The team analyzed two potential propeller sizes for the Turnigy motor: 9x5 and 8x6. In order to compare the two sizes and make a final decision on the motor-propeller combination, the team examined three types of plots for the aircraft's unloaded and loaded flight conditions. Figure 22 provides a comparison of thrust to drag of the aircraft for both propeller sizes.

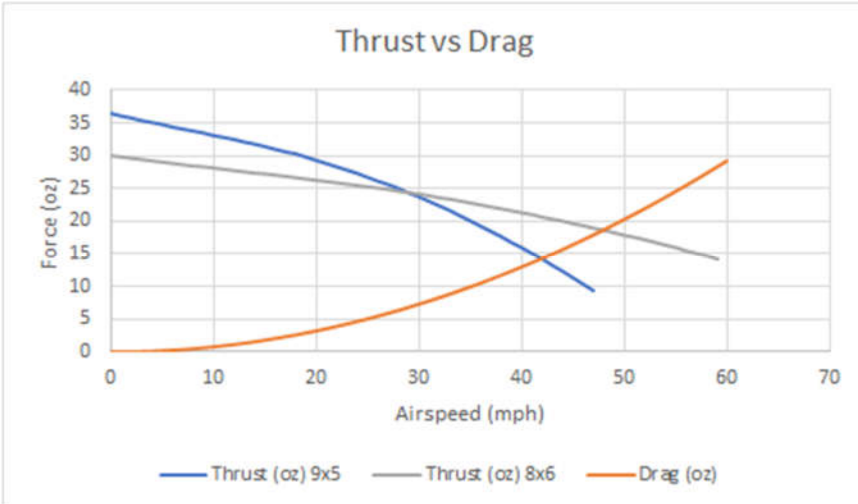

Figure 22. Thrust and Drag at Various Airspeeds

This graph shows that both propeller sizes have enough thrust to overcome the drag produced by the aircraft at the cruise speed of 30mph. This graph also shows that the 9x5 propeller performs better at slower airspeeds whereas the 8x6 propeller performs better at faster airspeeds. The next plots the team considered were the thrust to weight curves for both the loaded and unloaded configurations of the aircraft. These curves are shown in Figures 23 and 24.

Figure 23. Thrust to Weight Loaded at Various Airspeeds

Figure 24. Thrust to Weight Unloaded at Various Airspeeds

Based on the plot above, the team was able to determine the required thrust to weight ratio for the loaded configuration of the aircraft. Once the appropriate thrust to weight ratio was determined, the power to weight ratio could be derived. The power to weight ratio plots for the loaded and unloaded configurations are shown in figures 25 and 26.

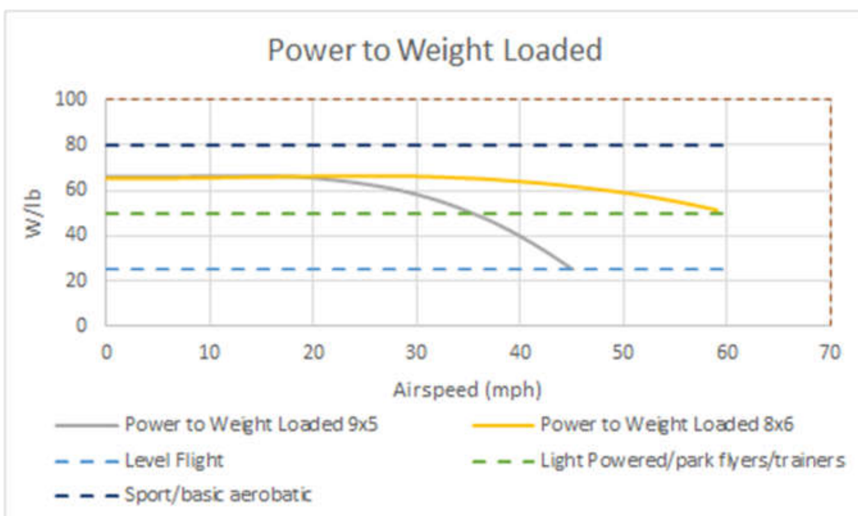

Figure 25. Power to Weight Curves Loaded at Various Airspeeds

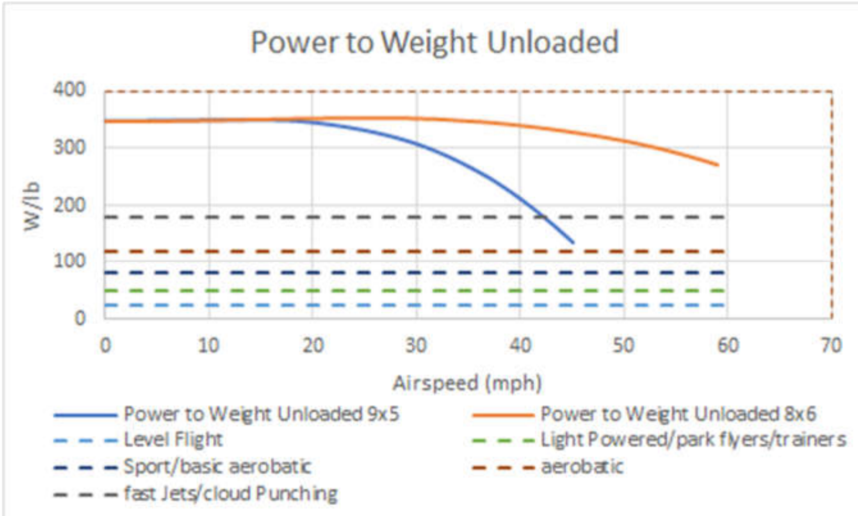

Figure 26. Power to Weight Curves Unloaded at Various Airspeeds

The power to weight curves show how the power output of the Turnigy motor compares with standards for RC aircraft (Carpenter, P., n.d.) (Rule of Thumb for Power to Weight Ratio, 2006). These plots show that in the unloaded configuration, the aircraft would fall into the "fast jet" and "cloud punching" power category for both propellers. However, in the loaded configuration, the aircraft would be classified in the "park flyer" power category with the 9x5 prop, dropping below the power necessary to maintain level flight at higher airspeeds. Based on the above plots, the team decided to use the Turnigy Park 480 1320Kv motor with an 8x6 propeller.

The team researched three possibilities for propellers and propeller attachment. The first option was a traditional attachment with a standard propeller. This setup involves a propeller attached to the motor shaft with a spinner screwed to the shaft to prevent the propeller from sliding off. This option would require the propeller to be detached while in the box, but would be fairly quick to attach during setup. Another option was to use a propeller saver, which involves an elastic ring that holds the propeller on a shortened motor shaft. This setup would also require the propeller to be detached during packing. A benefit of the propeller saver was that in the event of a propeller strike, the elastic ring would allow the propeller to bend upon landing, which would prevent it from breaking. However, the propeller saver does not attach the propeller to the motor shaft as securely as a standard setup and has been known to fail mid-flight (Motor Shaft Modification, n.d.). The last option was a folding propeller, which attached the blades to a hinge. Therefore, when the propeller spins, the centripetal force keeps the blades extended. A benefit of the folding propeller was that it allowed for easier packing in the box because it folds into itself and therefore conserves space. In addition, the blades fold in when power is cut, which reduces drag on landing. This setup, however, is heavier than the other two as the spinner is significantly larger. After compiling research, the team felt that the security and weight saving of the standard spinner was the best option.

In addition to propeller attachment, the propulsion team researched propeller material. The team considered two materials: plastic and carbon fiber. A plastic propeller was easier to balance and significantly cheaper; however, since the team's goal was to minimize weight, the team chose a carbon fiber propeller because it weighed less than the plastic propeller. In addition, a carbon fiber propeller was stronger and stiffer, which helped reduce damage on landing and prevented the propeller from bending. The propulsion team also considered the risks that came from using a carbon fiber propeller. Although the carbon fiber is stronger, it would shatter if broken. As a result, the team ensured that everyone was at a safe distance when the aircraft was powered and the thrower for takeoff wore a hard hat, safety glasses, and a Kevlar glove to prevent injury.

2.3.3 Battery Optimization

The competition rules limited the team's ability to analyze batteries due to the requirement that the team use Lithium Polymer (LiPo) batteries up to three cells (11.1V) and less than 2200mAh. The team needed to analyze three parameters to determine the capabilities of potential batteries. These parameters were the voltage, the mAh, and the C-value. The voltage, or number of cells, determines the RPM of the motor; as RPM increases, so does the thrust output. The team decided to use a three cell battery based on the desire to have a light motor with high thrust outputs. The three cell battery allowed the propeller to spin faster, which ultimately increased the thrust output of the motor. The mAh of the battery is the rating of the total amount of stored energy. Using this the team could calculate the usable battery life from the motor's specific current draw; as the mAh increases, battery life also increases. A comparison of mAh to battery weight can be seen in Figure 27. The information presented is for 11.1V batteries sold by HobbyKing.

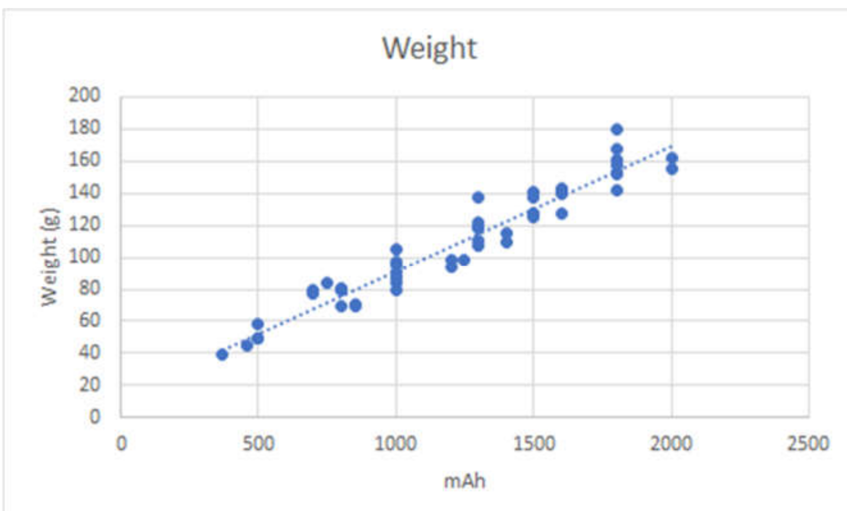

Figure 27. Battery Weight vs mAh (HobbyKing Batteries, n.d.)

As shown in Figure 27, the battery weight increased significantly as the mAh of the battery increased, so it was necessary to accurately determine the capacity of the battery needed in order to reduce weight. The decision on the mAh of the battery came from analysis of the time required to finish the 360 degree circuit. Based on the competition rules, the course consists of a maximum 60 second runup time, flying 400 feet and making a 180 degree turn then flying another 400 feet and making another 180 degree turn to a 200 foot landing zone. Using these dimensions, the data provided by MotoCalc 8 for current and thrust data at various power settings, servo and receiver current pull, and drag data, the team determined that a 1000mAh battery would be sufficient for the design. To come to this determination, the team made a few assumptions. These assumptions were: a 30 second runup time, a cruise speed of 30 mph, a 25 mph ground speed during a climb, no engine power at landing with descent during the final leg, all servos experiencing stall conditions during the entire flight, and a safety factor of 1.2 to prevent draining the battery. From these assumptions the team was able to determine an estimated power setting, and consequently the motor current draw for each leg of the flight. Using this current draw and the time of flight for each leg, the mAh capacity for each leg could be computed using equation (2.3-1).

$$mAh = mA * hour * safety\ factor \qquad (2.3\text{-}1)$$

This equation was used for each leg of the course to determine the total time of flight and mAh required. Using this time of flight and the current draw of servos and receiver, the capacity needed for these components was determined and added to the motor capacity to get the required mAh of the battery. Table 10 provides a depiction of the spreadsheet used to calculate the mAh calculation.

Table 10. Battery Sizing

Leg	Speed (mph)	Distance (ft)	Power Setting	Current Pull (mA)	Time of Flight (sec)	Safty Factor	Time of battery life needed (s)	mAh needed
Control Check				15000	5	1.2	6	25
Runup	-	-		27000	5	1.2	6	45
Takeoff	-	50		27000	10	1.2	12.00	90
Climb	25	200		27000	5.45	1.2	6.55	49.09
Leg 1	30	150		17000	3.41	1.2	4.09	19.32
Turn 1	28	471.24		17000	11.47	1.2	13.77	65.02
Leg 2	30	800		17000	18.18	1.2	21.82	103.03
Turn 2	28	471.24		17000	11.47	1.2	13.77	65.02
Leg 3	30	400		17000	9.09	1.2	10.91	51.52
Loiter	30	100		0	2.27	1.2	2.73	0.00
Land	25	200		0	5.45	1.2	6.55	0.00
Totals	-	**2842.48**	-	-	**86.81**	-	**104.18**	**513.00**
Electronic Battery Needs	Current Pull	mAh needed						
Receiver	120	3.47						
Servos flaparon	5000	144.69						
tail/rudder servos	1100	31.83						
Totals	6220	179.99						
Grand Totals								
Time of battery life	**104.18**							
mAh needed	**693.00**							
mAh of batt	1000							
usable mAh	850							
Remaining	157.00							
Total includes a safty factor of 1.2 for time of flight and max current pull of all servos and motor for entire flight								

2.3.4 Receiver Selection

In order to ensure that the receiver was compatible with the transmitter, the Taranis Q X7, the propulsion team chose to consider FrSky receivers that had a minimum of eight available channels. This led us to look at both the S8R and the X8R 8 channel receivers as shown in Table 11.

Table 11. Receiver Selection. Data from each receiver's manual (FrSky 2.4GHz ACCST X8R Manual, n.d.) (Instruction manual for FrSky S6R&S8R (SxR), n.d.)(FrSky S8R 8/16 Channel Receiver with 3-axis Stabilization., n.d.)

Receiver	FrSky X8R	FrSky S8R
Taranis Compatible	Yes	Yes
Channels Available	8	8
Weight (oz)	0.59	0.43
Voltage (V)	4-10	4-10
Stabilization	None	3-axis stabilization ● Includes a 3-axis gyroscope and accelerometer

Based on the above data, the team chose to use the S8R receiver due to its lighter design and 3-axis stabilization capability. The three axis stabilization worked through the use of an accelerometer and a gyroscope to sense the orientation of the aircraft. Using this information, the receiver provided corrections to the control surfaces based on pilot selected gain values and flight mode.

2.3.5 ESC and BEC Selection

The last major components of the power plant were the ESC and the battery elimination circuit (BEC). An ESC controls the motor power based on an input signal. An appropriately sized ESC must have continuous and burst current ratings that are greater than the motor's current output. Based on the manufacturer's data of the Turnigy Park 480, the maximum current is 28A (Turnigy Park480 Brushless Outrunner 1320kv, n.d.). In order to save weight while still having a safety factor on the ESC, the team elected to use a 35A Castle ESC (TALON 35 AMP ESC, 6S / 25V WITH 7 AMP BEC, n.d.). A BEC is used to convert the voltage from the battery to a lower voltage that can be used by other electronics such as a receiver and servos. . However, the internal BEC of most 35A - 40A ESC's cannot support the 7A continuous current that is required to power the receiver and servos for the aircraft. Due to this, the aircraft used an external BEC to ensure its components were not burned out during use.

2.3.6 Hand Launch Trajectory

In order to ensure that a hand launch was viable for the aircraft, the team created a take-off simulation that plotted the trajectory of the aircraft for variable takeoff characteristics. The requirements for successful take-off were simple:

1. The aircraft does not hit the ground
2. The aircraft reaches cruise velocity before it enters an unrecoverable nose-dive
3. The aircraft does not exceed the stall angle of the wing airfoil

To simulate take-off, the propulsion and stability subgroups collaborated on a Python script which modeled the aircraft as a 2D rigid body and used a Runge Kutta 4 (RK4) scheme to numerically integrate the equations of motion. The thrust was interpolated as a linear function of airspeed from the data shown in Figure 22 above. The full script can be found in Appendix D. Table 12 presents the input variables and the resulting plot appears in Figure 28.

Table 12. Variables used in calculating hand launch trajectory

Motor	Turnigy Park 480 (1320)
Propeller	8X6
Airfoil	NACA 9410
Throw Speed	29.33 ft/s
Throw Angle	10°
Aircraft Weight	4.2 lb

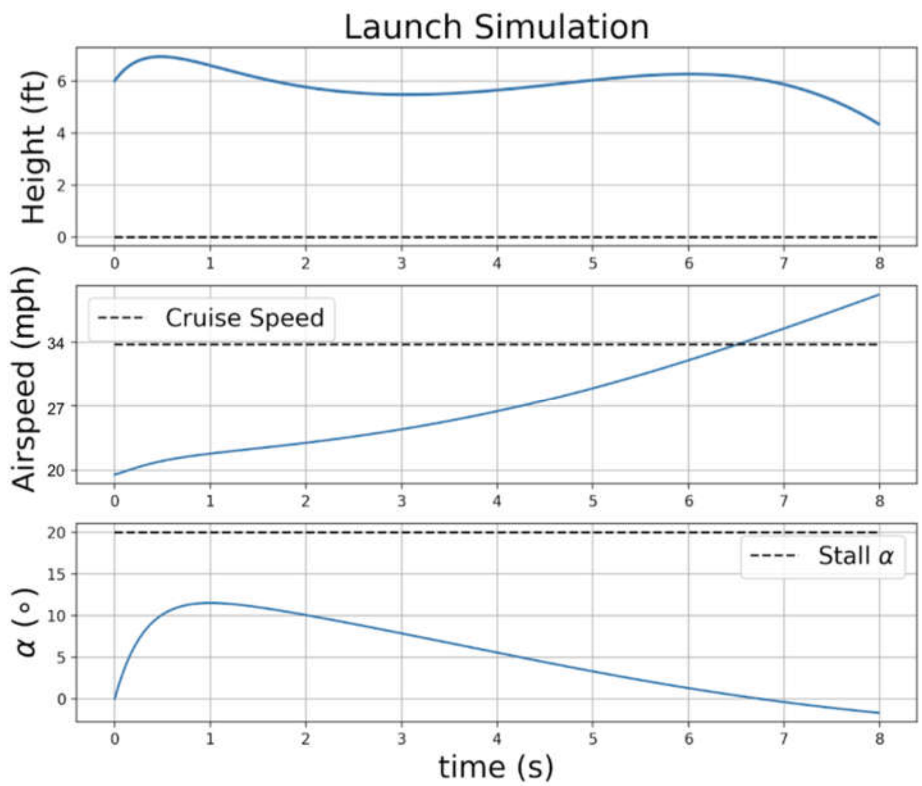

Figure 28. Height, airspeed, and AoA for take-off simulation

2.4 Stability and Controls

While developing the structural design of the aircraft, the stability and controls team stressed the importance of designing the aircraft to be both statically and dynamically stable.

2.4.1 Static Stability

To be statically stable, an aircraft must have a positive static margin, i.e. its center of gravity must be positioned between its aerodynamic center and its neutral point. In order to calculate the static margin, the team defined initial values for the aspect ratios of the wing and horizontal tail $AR_w$and AR_t, the area of the wing and tail $S$and S_T, the wingspan *b*, the wing chord *c*, the distance between the aerodynamic centers of the wing and tail $\underline{l_t}$, the location of the aerodynamic center as a fraction of wing chord h_{ac}, the center of gravity as a fraction of wing chord h, and a flight speed represented as a Mach number M. The stability team then used equations 2.4-1 and 2.4-2 in a MATLAB script to define the lift curve slopes of the wing and tail, which were important parameters for finding the static margin. Appendix E provides all MATLAB scripts and functions for static stability calculations.

$$\text{(For } x = w,\ t\text{)} \quad a_{0x} = \frac{\Delta C_{\square\square}}{\Delta \alpha_x} \qquad (2.4\text{-}1)$$

$$\text{(For } x = w,\ t\text{)} \quad a_x = \frac{a_{0x}}{\sqrt{1 + \frac{a_{0x}}{(\pi(AR_x)e)^2} + \frac{a_{0x}}{\pi(AR_x)e}}} \qquad (2.4\text{-}2)$$

After calculating the lift curve slope of the wing and tail, the stability team calculated the static margin of the aircraft using equations 2.4-3 through 2.4-7. The variable values used in the calculation of the static margin as well as the final calculated neutral point and static margin value are located in Table 13.

$$V_H = \left(\frac{l_t}{\bar{c}}\right)\left(\frac{S_t}{S}\right) \qquad (2.4\text{-}3)$$

$$\frac{d\varepsilon}{d\alpha} = 4.44\sqrt{1 - M^2}\left[\left(\left(\frac{1}{AR_w}\right) - \left(\frac{1}{1 + AR_w^{1.7}}\right)\right)\left(\frac{10 - 3\lambda}{7}\right)\left(\frac{1 - \frac{l_t}{b}}{\left(\frac{2l_t}{b}\right)^{0.33}}\right)\sqrt{\cos(\Lambda)}\right]^{1.19} \qquad (2.4\text{-}4)$$

$$C_{l\alpha} = a_w + a_t\left(\frac{S_t}{S}\right)\left(1 - \frac{d\varepsilon}{d\alpha}\right) \qquad (2.4\text{-}5)$$

$$h_{NP} = h_{ac} + V_H\left(\frac{a_t}{C_{l\alpha}}\right)\left(1 - \frac{d\varepsilon}{d\alpha}\right) \qquad (2.4\text{-}6)$$

$$Static\ Margin = h_{NP} - h \qquad (2.4\text{-}7)$$

Table 13. Geometric parameters for calculating the static margin

Description	Variable	Value (units) (unloaded)
Aspect ratio of wing	AR_w	*9.45*
Aspect ratio of tail	AR_t	*3.87*
Distance between aerodynamic centers	$\underline{l_t}$	*1.2625* (ft)
Wing span	b	*4.333* (ft)
Wing chord	c	*0.4583* (ft)
Wing area	S	*1.986* (ft^2)
Horizontal tail area	S_t	*0.2583* (ft^2)
Mach number	M	*0.027*
Taper ratio	λ	*1*
Wing Sweep angle	Λ	*0*
Location of CG	h	*0.3636*
Location of Aerodynamic Center	h_{ac}	*0.25*
Neutral Point	h_{NP}	*0.4523*
Static Margin	$(h_{NP} - h)$	*0.0575*

The calculated neutral point and static margin show that the aircraft was statically stable. Expressed as distances from the leading edge of the wing, the center of gravity was at 0.167 ft (2 in) and the neutral point was at 0.209 ft (2.515 in). In order to ensure stability between both unloaded and loaded configurations, the aircraft was designed to have the same center of gravity for both configurations, as static stability is independent of total weight assuming all geometric properties remain constant. Figure 29 below shows a visual of the location of the center of gravity and neutral point on a side view of the aircraft.

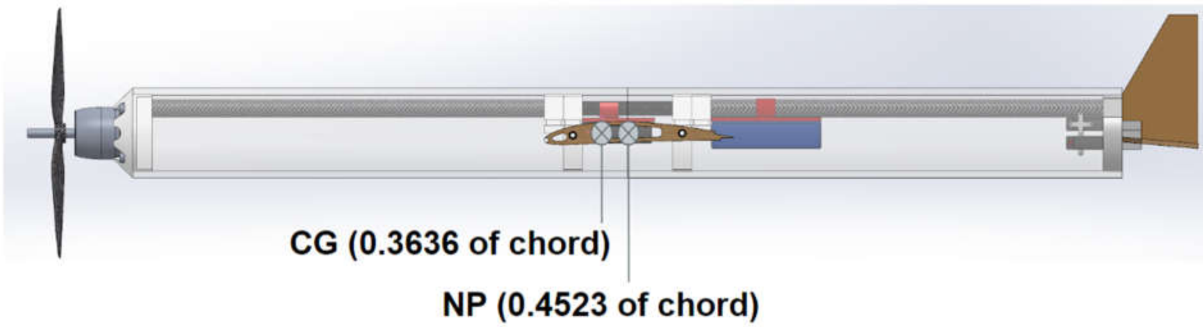

Figure 29. Side view of aircraft with location of CG and neutral point shown

2.4.2 Dynamic Stability

The dynamic stability determined the long-term motion characteristics of the aircraft if a disturbance occurred in flight. There were two types of dynamic stability that the stability team analyzed for the aircraft: longitudinal and lateral stability. Longitudinal stability has two modes: the short period mode and the phugoid mode. Lateral stability has three modes: the roll subsidence, dutch roll, and spiral divergence modes. In both the longitudinal and lateral cases, the team aimed to calculate the stability matrices to determine if the aircraft was dynamically stable. In order for the aircraft to be stable, the real parts of the eigenvalues of the stability matrices must be negative.

To calculate the stability matrices for both longitudinal and lateral stability, the team needed to calculate the non-dimensional and dimensional stability derivatives. Then the stability team calculated the stability matrices in each case and found the eigenvalues of the matrices. Finally, the stability team calculated the settling time of each mode. The settling time defines how long it takes the aircraft to converge to a stable state. Shorter settling times can be beneficial but in the event of a drastic disturbance or an unstable mode, a longer settling time would allow the pilot a larger time frame to make corrections to the flight condition. The team calculated the stability matrices and modes using both the XFLR5 software and MATLAB scripts to ensure that the values were similar. Appendix F shows all equations used in the MATLAB scripts for calculating the stability derivatives and matrices.

2.4.2.1 Longitudinal Stability

To calculate the longitudinal stability of the aircraft, the team used similar geometric values as used in the calculation of the static stability. Added variables for longitudinal dynamic stability compared to static stability were the Oswald efficiency factor e, dynamic pressure ratio η_H, moment of inertia in the y-direction I_y, induced coefficient of drag value C_{D0}, trim angle of attack α_{trim}, aircraft weight W, flight speed u_0, air density at altitude ρ, and gravitational constant g. Equations and variables used in calculating the longitudinal stability of the aircraft can be found in Appendix F.

The method used to calculate the longitudinal stability of the aircraft followed the lecture notes for the AE4723 course at WPI with Professor Cowlagi, Aircraft Dynamics and Control (2017). This method included using the stability derivatives of the aircraft to calculate the longitudinal matrix. Upon developing this matrix, the method required the stability team to find the eigenvalues of the longitudinal matrix. The eigenvalues corresponded to the stability modes which dictate the dynamic response of the aircraft to small disturbances. For the longitudinal stability case, these modes are the phugoid mode and the short period mode. The phugoid mode corresponds to a pitch and speed change of the aircraft with little to no angle of attack variation due to a pitch disturbance. The phugoid mode typically has a longer settling time in the range of 15-25 seconds. The short period mode has a much shorter settling time when it is stable and is typically on the scale of 1-2 seconds. The short period corresponds to angle of attack variation of the aircraft with pitch disturbances.

In order to find the longitudinal stability matrix, the stability team needed to calculate the stability derivatives of the aircraft (Etkin, B., & Reid, L.D., 1996). A combination of XFLR5 analysis and MATLAB calculations were used to determine these stability derivatives. Using a basic model in XFLR5 as shown in Figure 30*,* the team performed a dynamic stability analysis on the simplified aircraft. The model and analysis used the same geometric parameters as stated for the static stability calculations and the new parameters added for the longitudinal calculations. The results of the XFLR5 longitudinal dynamic stability analysis appears in figures 31 and 32*,* showing graphs of the short period and phugoid modes.

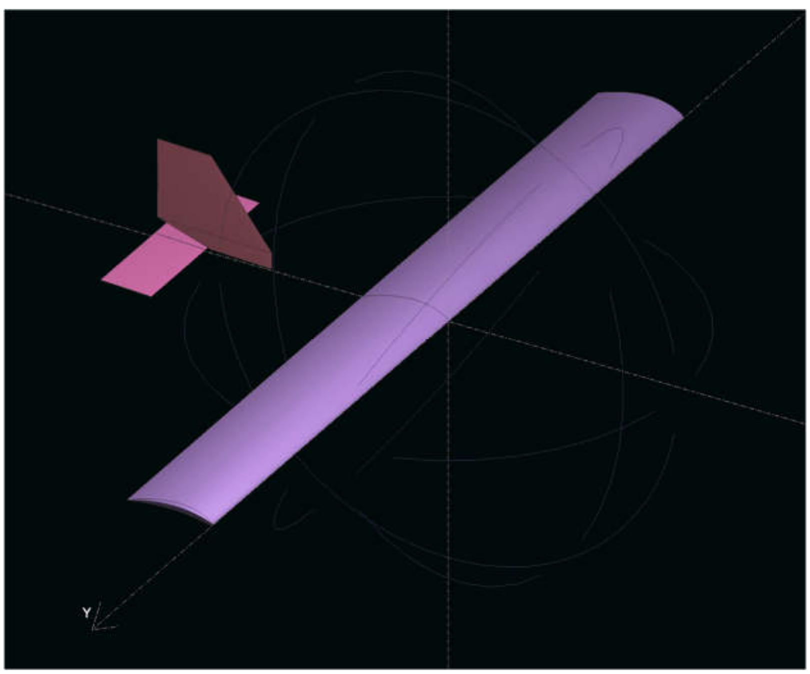

Figure 30. XFLR5 Simplified Aircraft Model

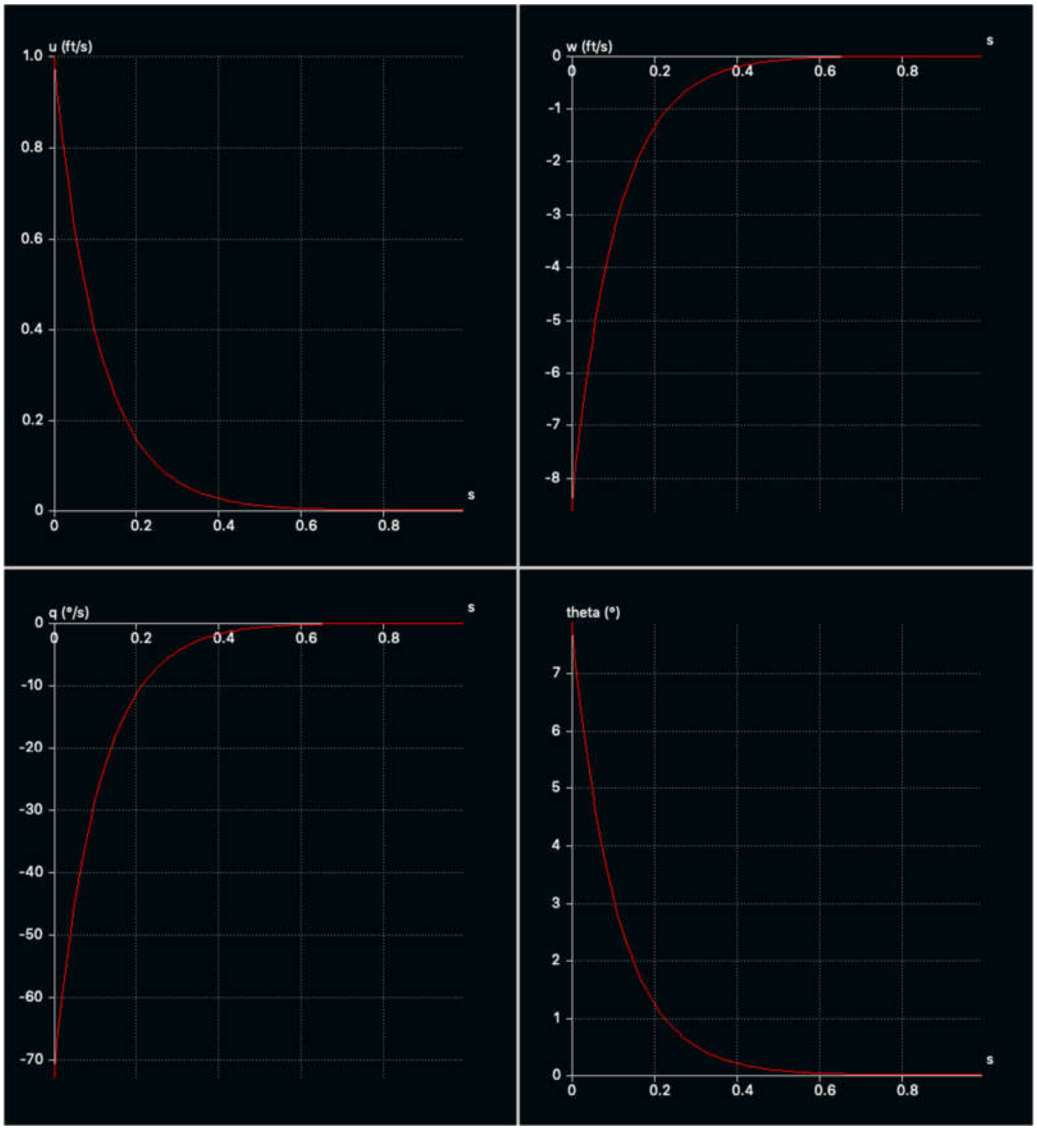

Figure 31. Short Period Mode Response of horizontal airspeed (u), vertical airspeed (w), pitching rate (q), and pitch angle (theta)

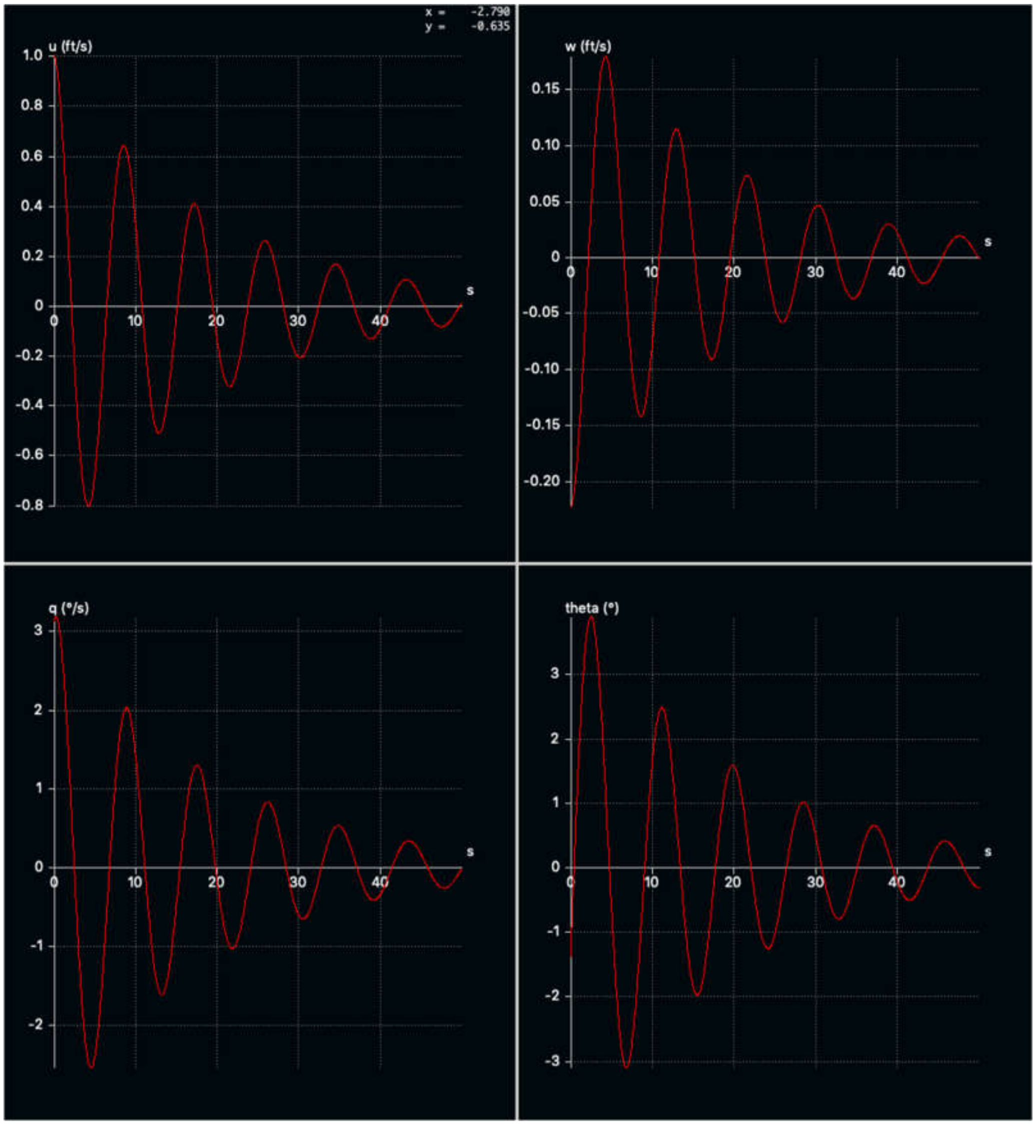

Figure 32. Phugoid Mode Response of horizontal airspeed (u), vertical airspeed (w), pitching rate (q), and pitch (theta)

As shown in the figures for the stability modes, the aircraft was dynamically stable in the longitudinal direction. Both the short period and phugoid mode converged to 0, with the short period settling after 0.58 seconds. The phugoid mode had a much longer settling time than the short period mode with a settling time of 1007 seconds (16.7 minutes). The large settling time of the short period was due to the fact that real portion of the pole was very close to 0. This causes the settling time to dramatically increase as the settling time is defined as $3/$

$abs(Real\ part\ of\ pole)$. While the team agreed that the phugoid mode settling time was too long, due to time constraints, the mode was not altered on the final aircraft.

To ensure that the XFLR5 analysis accurately depicted the dynamic stability characteristics, each stability derivative was calculated through MATLAB scripts as well. A comparison between the XFLR5 longitudinal stability derivatives and the calculated longitudinal stability derivatives appears in table 14 below.

Table 14. Comparison between XFLR5 values and calculated Stability Derivatives

Stability Derivative	**XFLR5 Value**	**MATLAB Value**
$\boldsymbol{C_{Lu}}$	0.00294	.0000111
$\boldsymbol{C_{Lq}}$	6.45387	31.4749
$\boldsymbol{C_{L\alpha}}$	5.13406	5.2074
$\boldsymbol{C_{Mu}}$	0	0
$\boldsymbol{C_{Mq}}$	-7.43582	-6.8216
$\boldsymbol{C_{M\alpha}}$	-0.32401	-0.4167

Upon comparing the values found through XFLR5 and the team's calculated derivative values, the stability team decided that the calculated values were similar to XFLR5, thus making XFLR5 a trustworthy stability tool. The stability team believed that differences between the two calculations were caused by assumptions made for the hand calculated values, where XFLR5 did not make the same assumptions. The largest discrepancy appears in the difference between the values for $\boldsymbol{C_{Lq}}$and C_{Lu}**.** In these cases, the team believed that there was a human calculation error and thus the XFLR5 number was used for further calculations.

2.4.2.2 Lateral Stability

To calculate the lateral stability of the aircraft, the team used similar geometric values as used in the calculation of the static stability and longitudinal stability. Added variables for lateral dynamic stability compared to static and longitudinal stability were: moment of inertia in the x direction I_x, moment of inertia in the z direction I_z, moment of inertia in the x-z direction I_{xz}, the length of the fuselage body l, fuselage side surface area S_{bs}, area of the vertical tail S_V, dihedral of the wing Γ_W, dihedral of the horizontal tail Γ_H, fuselage diameter d, the rolling damping factor RDP, and the X and Z values calculated for the longitudinal stability..

For the lateral stability case, the calculated modes were the roll subsidence, spiral divergence, and dutch roll modes. The roll subsidence mode described the damping of the roll rate with a roll disturbance. A stable roll subsidence will tend towards damping the roll rate to 0 and is not an oscillatory mode like the phugoid and short period. The roll subsidence could be improved by the addition of dihedral to the wings of the aircraft, but the stability team determined that this was unnecessary and would complicate the manufacturing process more than it would help with the roll mode. The spiral divergence mode determined the likeliness of the aircraft to roll into a spiral with a disturbance in a sideslip condition. It is common for the spiral divergence mode to be unstable, and the major determinant in the instability of this mode is the divergence rate. If the divergence rate provides sufficient time for the pilot to correct for the undesirable roll moments, then it is not considered a concern for maintaining stable flight. The final lateral mode was the dutch roll mode. The dutch roll mode is described as an oscillation in the yaw and roll of the aircraft which, when stable, recovers to equilibrium with a typical period in the range of 3-15 seconds.

After analyzing the discrepancies between stability derivatives calculated in MATLAB scripts vs those calculated by XFLR5, the team decided that those calculated by XFLR5 were more reliable. This decision was made based on the lack of literature available on computing certain stability derivatives in a closed form. Most of the literature on the topic suggests experimental determination and verification of many stability derivatives. As the team did not have access to such testing equipment, the decision was made to trust the values calculated by XFLR5, and from this point forward XFLR5 was the sole tool utilized in dynamic stability calculations.

Using the basic model in XFLR5 as shown previously in Figure 30, the team performed a dynamic stability analysis on the simplified aircraft. The model and analysis used the same geometric parameters as stated for the static stability calculations and the new parameters added for the longitudinal and lateral calculations. The results of the XFLR5 lateral dynamic stability analysis appear in figures 33, 34, and 35 showing graphs of the roll subsidence, spiral divergence, and dutch roll modes respectively.

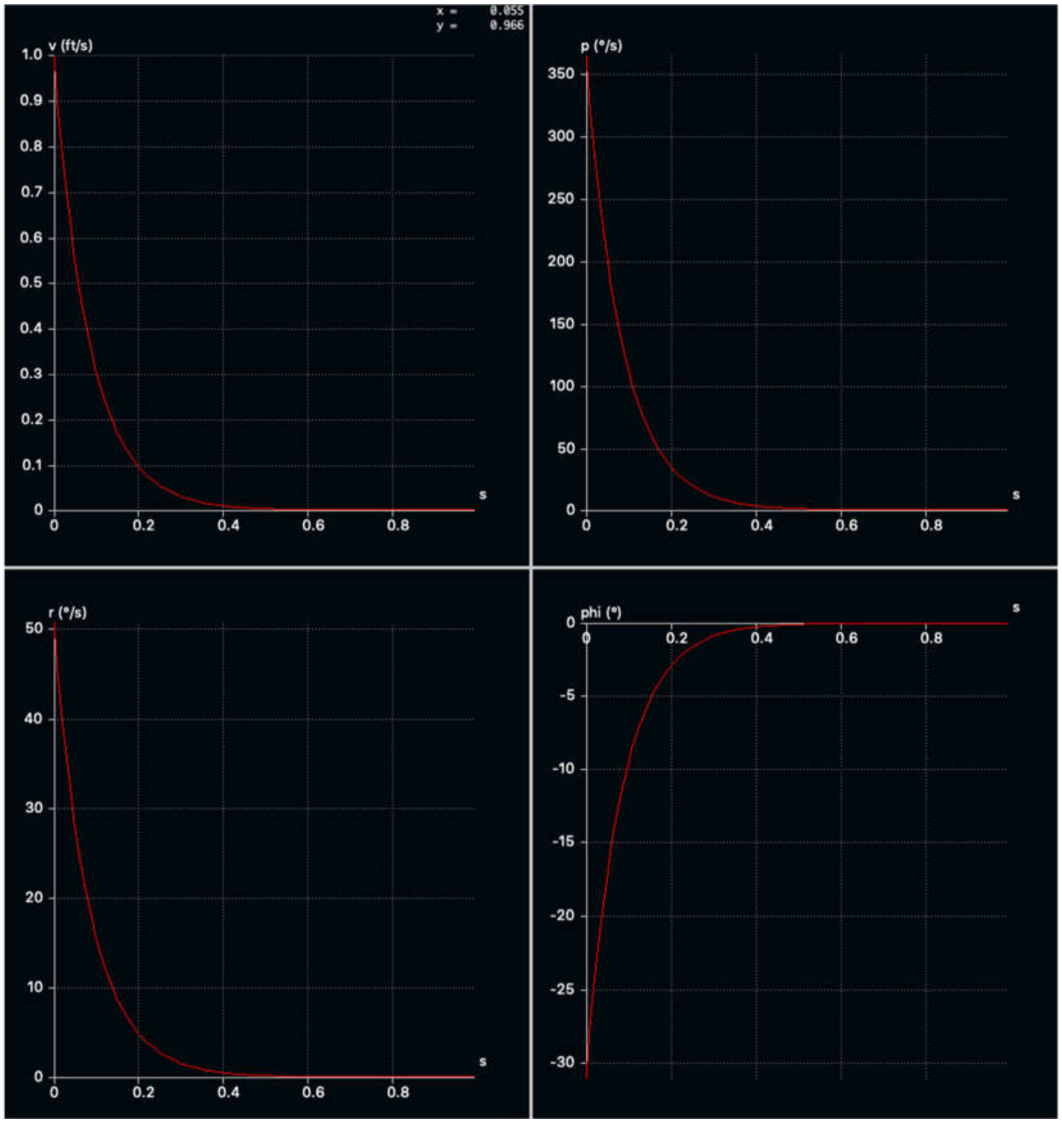

Figure 33. Roll Subsidence Response of sideslip velocity (v), yaw rate (p), roll rate (r), and roll angle (phi)

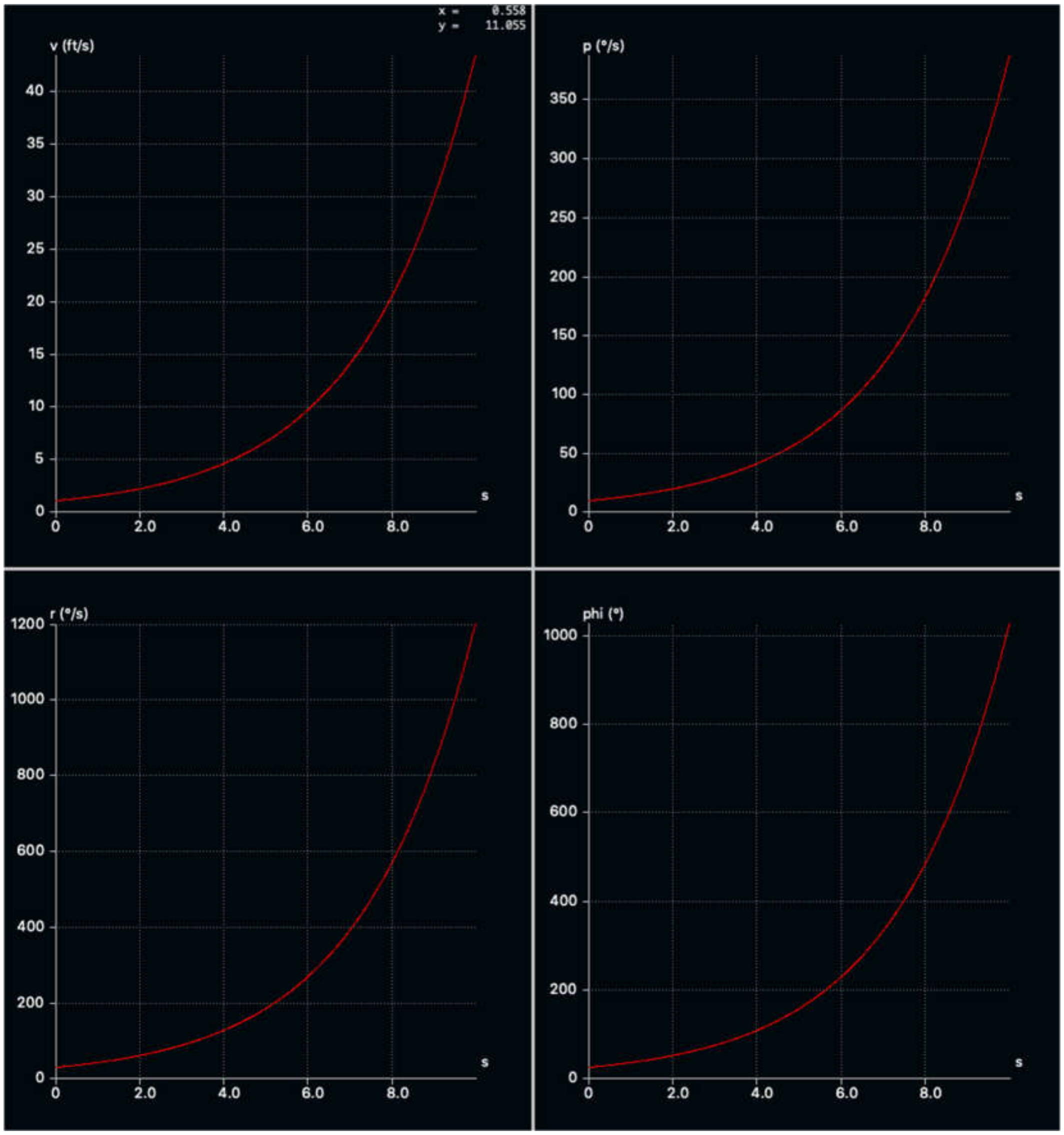

Figure 34. Spiral Divergence Response sideslip velocity (v), yaw rate (p), roll rate (r), and roll angle (phi)

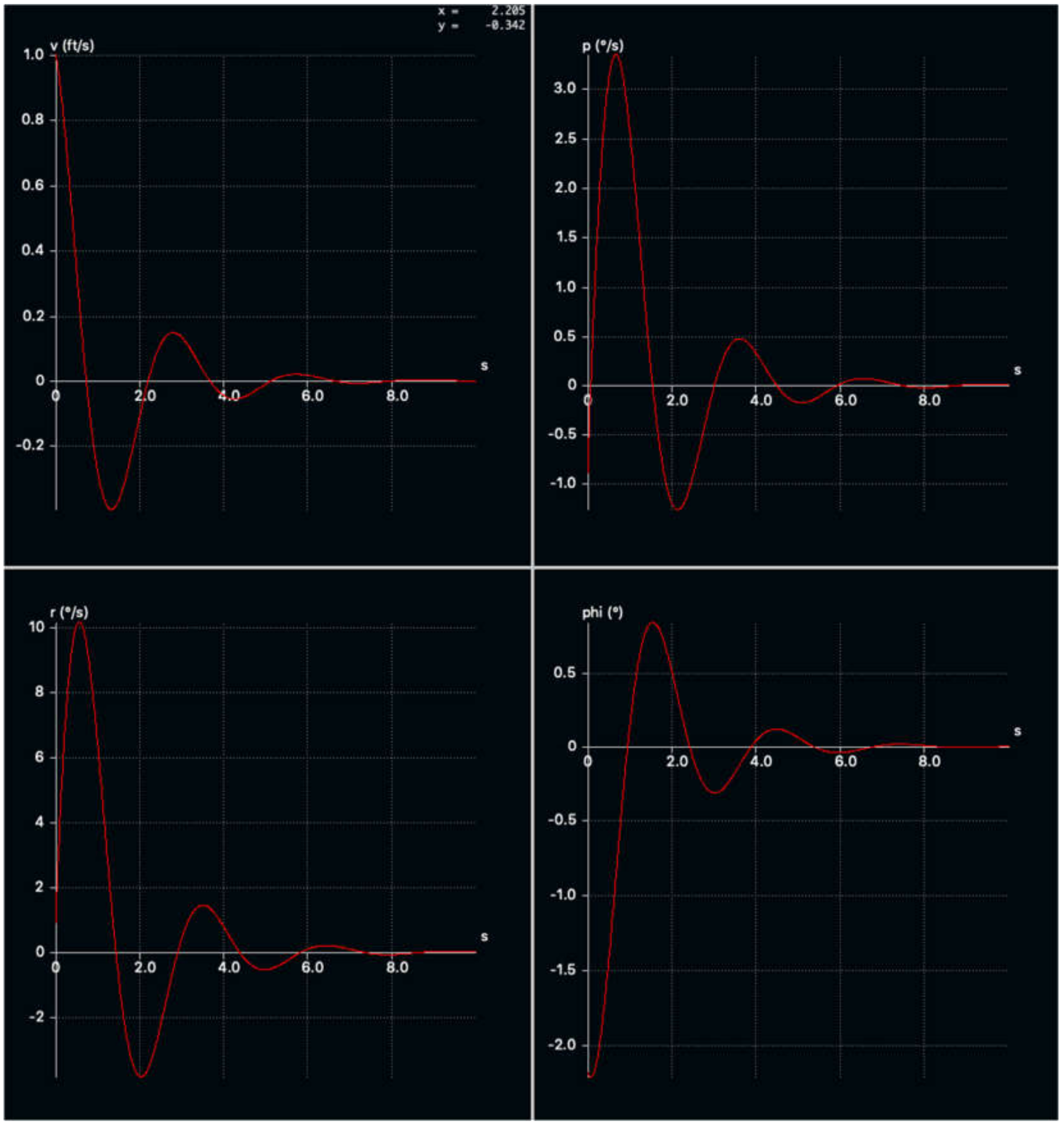

Figure 35. Dutch Roll Response sideslip velocity (v), yaw rate (p), roll rate (r), and roll angle (phi)

As shown in the figures for the lateral stability modes, the aircraft had good settling characteristics for the roll subsidence and dutch roll mode, converging to zero after approximately one second for the roll subsidence and eight seconds for the dutch roll. However, the spiral divergence mode increased exponentially after the same amount of time. It is common for the spiral divergence mode to be unstable, thus the stability team determined that the eight second divergence provided enough time for pilot corrections before the aircraft entered an unrecoverable spiral state.

2.4.3 Trim Analysis for Tail Incidence Angle

In order to counteract the moment of the wing for the aircraft, the horizontal tail must be set at an angle to create an opposite moment. This angle is called the incidence angle. This moment allows the pilot to trim the aircraft to an angle of attack where the net moment is zero. To calculate the incidence angle necessary for the tail, the team first calculated the moment coefficient $C_{M d\alpha}$. This moment coefficient, combined with the aircraft's geometry and lift coefficient at zero angle of attack, allowed the team to calculate an appropriate incidence angle. The following equations, equations 2.4-8 and 2.4-9, show the necessary variables and final outcome for the tail incidence angle. The stability team determined that the tail should be set at an incidence angle of -6.77°.

$$\boldsymbol{C_{md\alpha} = (a_w + a_t \frac{S_t}{S}(1 - \frac{d\varepsilon}{d\alpha}))(h - h_{ac}) - a_t V_H (1 - \frac{d\varepsilon}{d\alpha})} \qquad (2.4\text{-}8)$$

$$i_t = \frac{C_{mac} + C_{l0,w}(h - h_{ac}) + C_{md\alpha}\alpha}{a_t V_H\left(1 - (h - h_{ac})\left(\frac{c}{l_t}\right)\right)} + \varepsilon_0 = -6.77\ degrees \qquad (2.4\text{-}9)$$

2.4.4 Control Surfaces

In addition to determining the stability of the aircraft, the stability and controls team also designed the control surfaces for the aircraft. The control surfaces that the team elected to use included an elevator, rudder, and flaperons. The following sections cover the sizing for each control surface and the subsequent servo sizing and selection for each control surface.

2.4.4.1 Control Surface Sizing

Using various hobby sites and publications (Lab 8 Notes - Basic Aircraft Design Rules, 2006)(Sadraey, M., 2012), the stability subgroup designed the control surfaces using the following estimates of area percentage, as shown in Table 15. The area percentage estimate dictated how much of the wing or tail area should be dedicated to the control surface.

Table 15. Depicting the area percentage required for the control surfaces of the wing and tail

Control Surface	Area Percentage	Resulting Area
Flaperons	10%	$14.3\ in^2$
Elevator	40%	$14.87\ in^2$
Rudder	35%	$4.4\ in^2$

Once the group decided on an area percentage, the team needed to design the specific sizing of each control surface. Using a combination of percentage estimates for chord size and control surface span as well as manufacturing capabilities, the stability team calculated the following dimensions, shown in Table 16, for each control surface. Note that the dimensions have been rounded in order to simplify the measurements when constructing the control surfaces, thus some area values may not be a direct multiplication of the span and chord.

Table 16. Tabulated values for dimensions of the control surfaces

Control Surface	Area	Span	Chord
Flaperon	$14.3\ in^2$	$13\ in$	$1.1\ in$
Elevator	$14.87\ in^2$	$12\ in$	$1.25\ in$
Rudder	$4.4\ in^2$	$4\ in$	$1.1\ in$

Once the stability team determined the dimensions of each control surface, the team determined the necessary sizes for the associated servos.

2.4.4.2 Servo Sizing

For the aircraft, the team used 4 servos: one for each flaperon, the elevator, and the rudder. The team used the following equation, equation 2.4-10, to calculate the torque output required of each servo (Gadd, C., n.d.)(Motor Sizing Calculations, n.d.). The torque equation was taken from Chuck Gadd's discussion on Servo Torque Calculations (n.d). The variable ρ is the air density, $c_{cs}$is the chord of the control surface, $V$is the speed of the aircraft in mph, θ_{cs} is the maximum control surface deflection in degrees, and $\theta_{servo}$is the maximum servo deflection in degrees.

$$T \ = \rho {c_{cs}}^2 V^2 L sin(\ \theta_{cs}) \frac{\tan(\theta_{cs})}{\tan(\theta_{servo})} \qquad (2.4\text{-}10)$$

Using this equation in conjunction with the values for each control surface, the team determined that the rudder and elevator required 6.5 oz/in and 4 oz/in of torque respectively, thus nano servos (HS-40) were chosen. The flaperons required 60 oz/in thus the team chose a specialty thin wing servo with high torque (D145SW). Below is Table 17 outlining the specifications of the two servos selected.

Table 17. Servo Selection. Data from (D145SW, 32-Bit, Wide Voltage, Steel Servo, n.d.) and (HS-40 Economical, Nano, Nylon Gear Servo, n.d.)

Servo	**HS-40**	**D145SW**
Use	Elevator and Rudder	Flaperons
Voltage (V)	4.8-6	4.8-7.4
Torque (oz-in)	9.4-10.5	54-83
Stall Current (A)	0.460	2.5
Weight (oz)	0.17	0.84
Dimensions (in)	0.79 x 0.34 x 0.67	1.18 x 0.39 x 1.45

2.4.5 Elevator Trim Analysis

To ensure the elevator provided sufficient pitch control of the aircraft, the team had to theoretically verify that the selected elevator size would sufficiently alter the pitching moment of the aircraft. When elevator deflection is considered in the pitching moment equations, solving for trim angle of attack, and thereby trim velocity, becomes coupled with solving for a corresponding elevator deflection in the 2x2 matrix equation shown in equation 2.4-11. Control derivatives for the elevator were computed in XFLR5 by examining how lift on the tail changed with elevator deflection and then scaling into pitching moment. Figure 36 shows the iterative calculation of equation 2.4-11 for different inputs of elevator deflection. This figure shows that the elevator could sufficiently change pitching moment and was thus suitable for pitching moment control. It was not important that elevator deflection directly affected total lift, but rather that elevator deflection could induce a strong enough pitching moment to change the angle of attack. This changed the lifting properties and trim angle of attack. The horizontal tail was also an important tool for a pilot to control the stability of the aircraft in case of disturbances in pitch and for the pilot to ascend and descend in flight.

$$\begin{bmatrix} C_{m_\alpha} & C_{m_\delta} \\ C_{L_\alpha} & C_{L_\delta} \end{bmatrix} \begin{bmatrix} \alpha \\ \delta \end{bmatrix} = \begin{bmatrix} -C_{m_0} \\ \frac{W}{QS} - C_{L_0} \end{bmatrix} \quad (2.4\text{-}11)$$

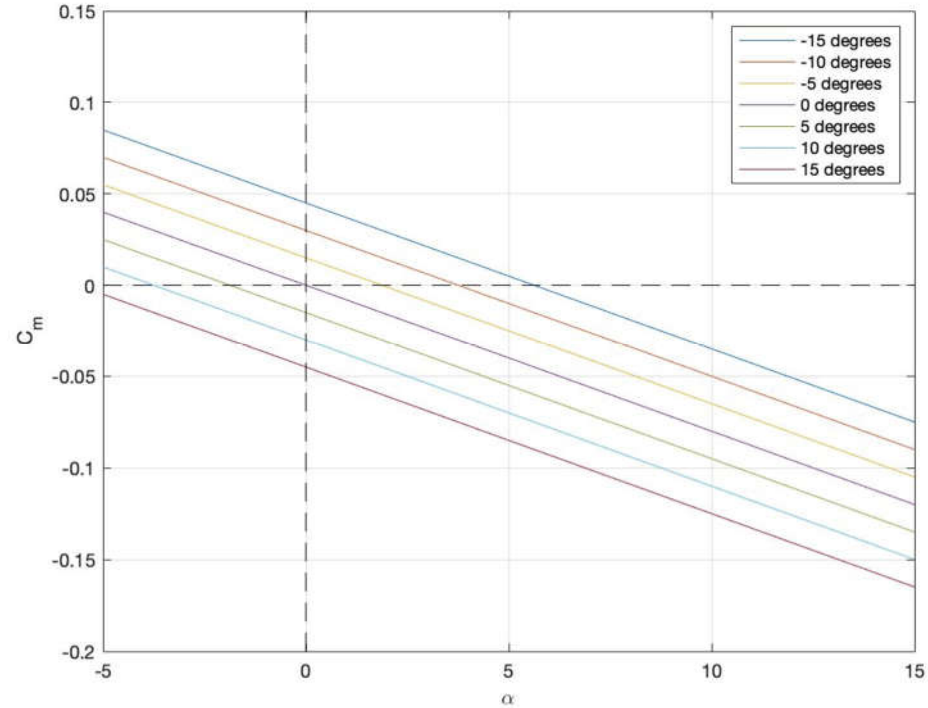

Figure 36. Graph showing the effect of elevator trim angle on the moment coefficient of the aircraft

3. Construction

Upon completing the design and theoretical analysis of the aircraft, the team began construction. The skeletal design and availability of modern fabrication tools such as laser cutters and 3D printers simplified the construction process. The primary structure of the aircraft was composed of prefabricated carbon fiber rods, 3D printed components, and laser cut eighth inch birch plywood and balsa. The minimal structure and heavy use of carbon fiber and low infill 3D printed parts allowed for a lightweight and strong structure for the aircraft. The team created 3D printed parts that had a maximum of 10% infill density with eighth inch thick outer shells. These parts were strong enough in testing to withstand the forces of flight and were lighter in weight than solid machined parts.3D printing also enabled the team to create parts that would be difficult or impossible to CNC machine, such as the tail cap with a built in incidence angle. The nose cone, center connectors, tail cap, and mounts for the battery, receiver, and ESC were all 3D printed. Being able to easily manufacture complex parts was essential to both weight reduction and proper function of the aircraft. The principle of simple manufacturing also extended to the wings and tail. The team made the wing from four 13 inch sections that consisted of nine ribs laser cut from balsa and aircraft plywood. The ribs were glued to prefabricated pultruded carbon fiber spars and coated in a heat adhesive shrink wrap called MonoKote. The MonoKote provided a smooth finish on the airfoil surface and created an efficient, lightweight wing. The four wing sections were secured by a smaller carbon fiber pultruded rod that fit inside the inner diameter of the primary wing spars. The idea behind this design was that the rods would fit tight enough to secure the two wings together.

3.1 Fuselage Construction

The spine of the fuselage was a combination of twill wrap carbon fiber rods and unidirectional carbon fiber rods. Other key components such as the nose cone and tail cap were made from 3D printed polylactic acid (PLA). The team created the full design of the aircraft with the intent of modularity. Some parts that allowed for modularity included the battery, ESC and receiver mounts which had a clip that allowed relocation to adjust the center of gravity. This enabled the team to make changes after construction to fix any unexpected issues. The entire structure of the aircraft without and with payload can be seen in Figures 37 and 38 below.

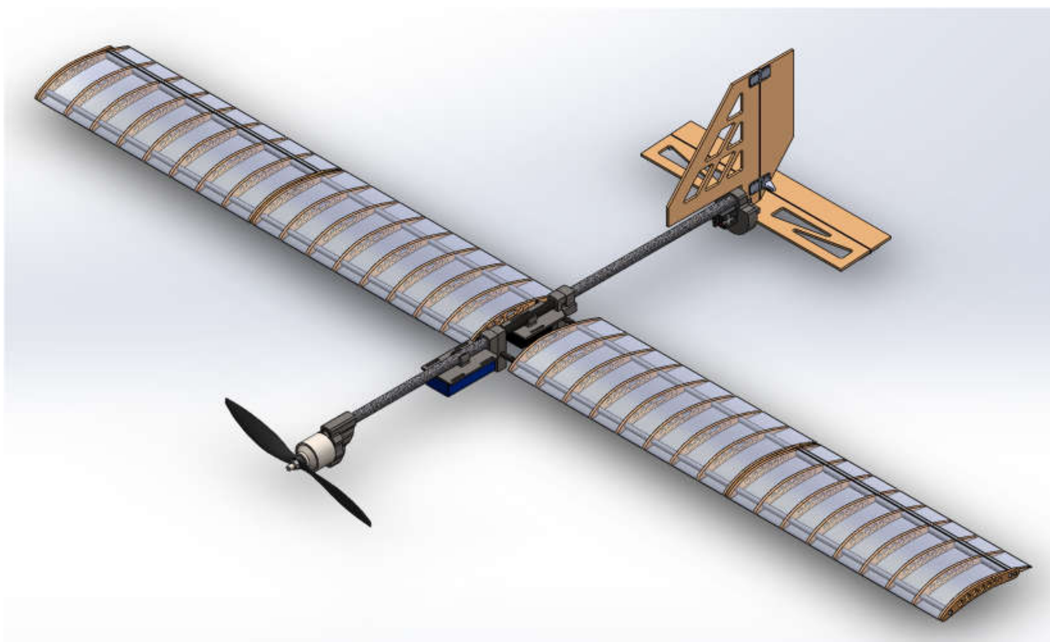

Figure 37. Fully assembled aircraft without payload

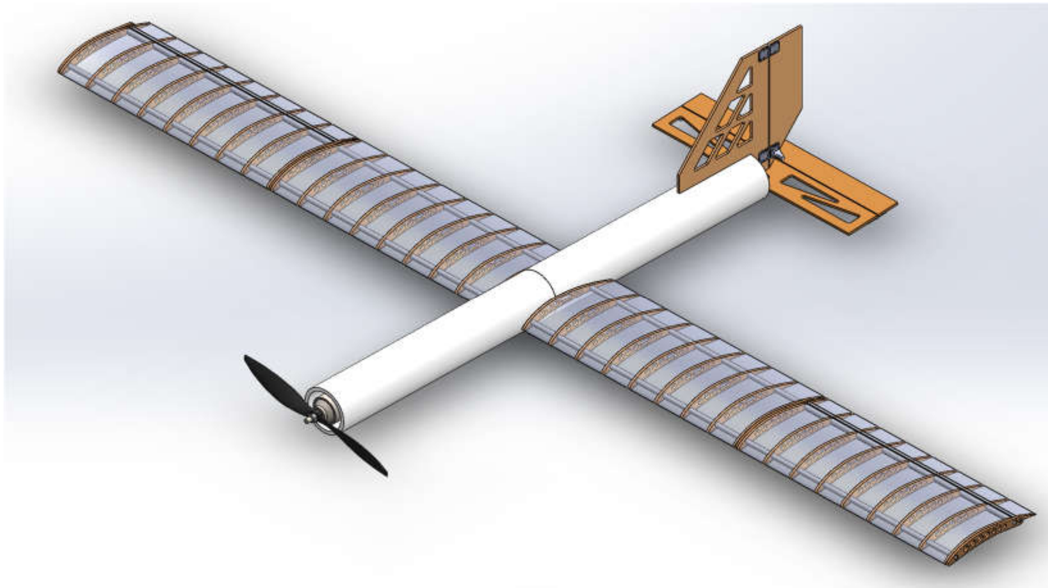

Figure 38. Fully assembled aircraft with payload

The body of the aircraft separates into three sections no longer than 13 inches and each half of the wing separated into two panels that were 13 inches long for packing the aircraft. An exploded view of the aircraft assembly can be seen in Figure 39 below.

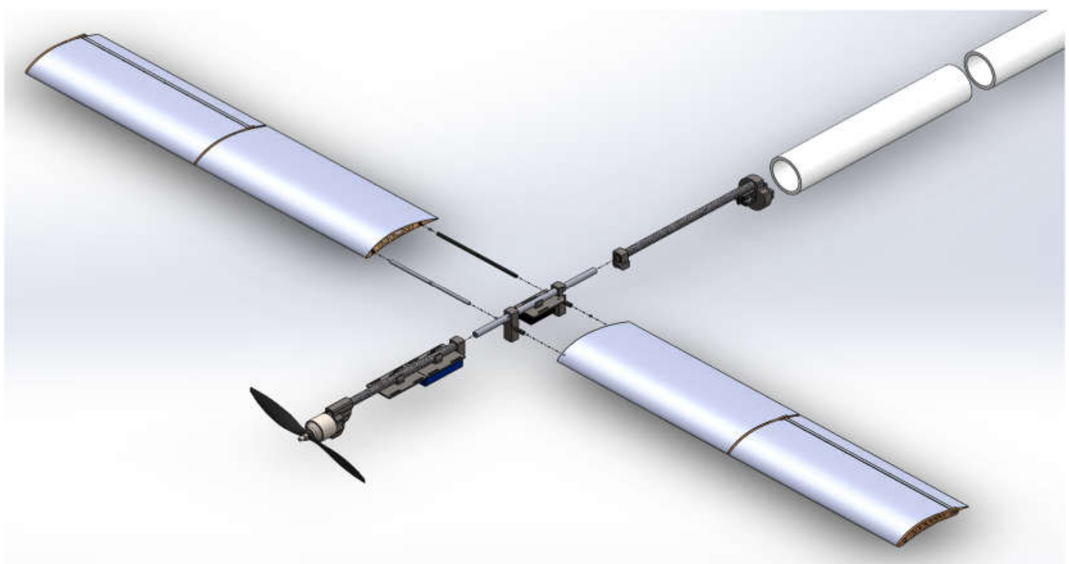

Figure 39. Exploded view of aircraft

Starting from the front of the aircraft, the fuselage consisted of a 3D printed nose cone which secured the motor, a twill wrap carbon fiber rod which served as the spine of the fuselage, and an end connector which featured a male connection end that secured the front section of the fuselage to the center section of the fuselage. The front section of the body assembly can be seen in Figure 40 below.

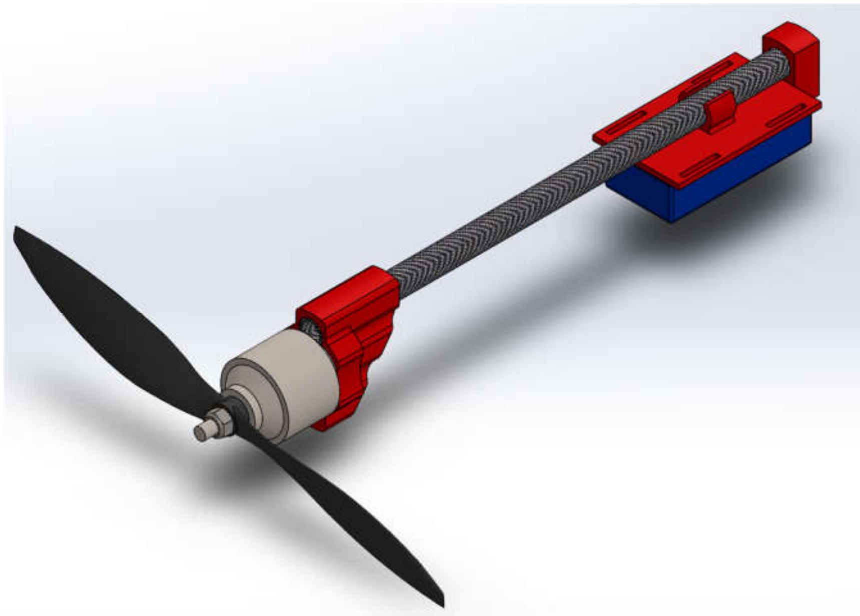

Figure 40. Front section

The battery was mounted to the front section of the aircraft when fully assembled to maintain proper balance of the aircraft. The battery is represented by the blue component in Figure 40 above. The 3D printed components are highlighted in red. The aircraft was designed with a high level of modularity to correct for any possible issues such as the incorrect location of the CG and a higher aircraft weight than expected. The center section of the aircraft fuselage connects the remaining major components together as it joins the front and rear body sections as well as the wings. The model of the center section can be seen below in Figure 41.

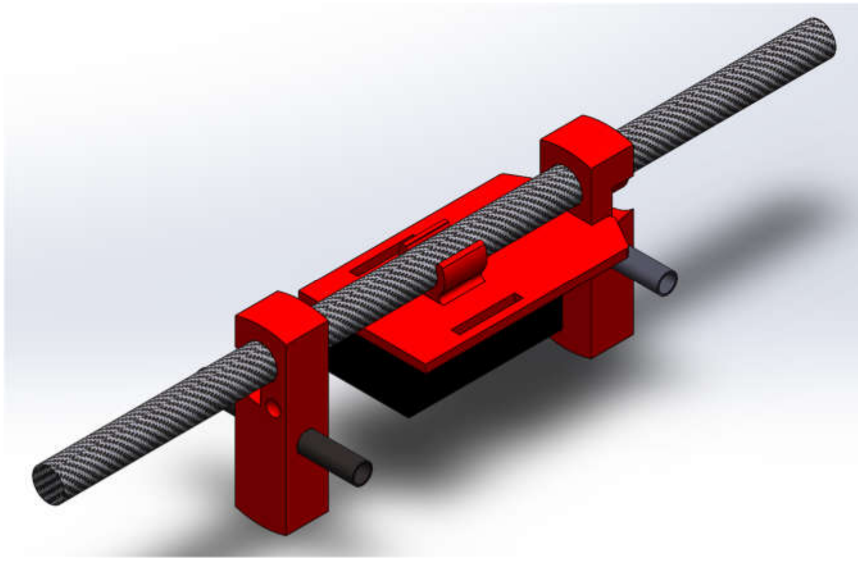

Figure 41. Center section

The center section was designed with a unidirectional carbon fiber rod with two 3D printed center support pieces that served several purposes. The receiver mounted to the 3D printed plate in the middle of the carbon fiber section which allowed wires to run out to every other area of the aircraft. The unidirectional carbon fiber rod was stronger and lighter than the twill wrap rod, but the team found it could not handle tangential loading as well as the twill wrap. The center supports held the PVC pipe in place when it was covering the body. The center supports have carbon fiber tubes that run perpendicular to the center spar that the wing spars run through when the aircraft is assembled. The center connectors also had the female locking grooves for the twist locks that secured the three fuselage components together. The solid spine tube of the center section slid into the tubular rear and front spines with the connectors out of rotational alignment, then the rear and front sections were rotated to lock the sections together. The groove in the female center support and the extrusion on the male connector piece could only separate or attach when the spine of the aircraft was rotated 45 degrees. The connection was retained by a friction fit and small detent. The center support (black) and rear connector (light blue) can be seen in Figure 42 below.

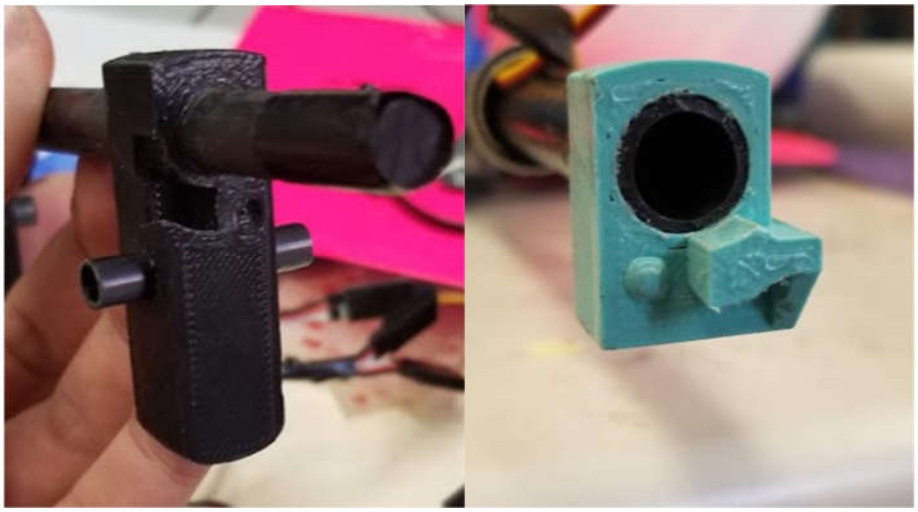

Figure 42. Center support (left) and rear connector (right)

The rear section of the aircraft held the tail and control servos for the elevator and rudder and supported the end of the payload in the loaded configuration. It can be seen in figure 43 below.

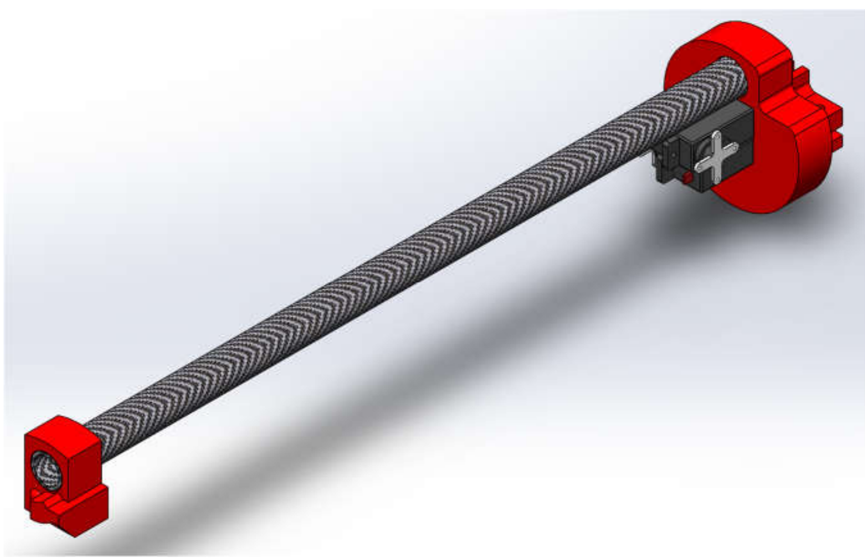

Figure 43. Rear section

3.2 Wing Construction

Two renditions of the wing were constructed for the aircraft: a version without control surfaces, and a final version featuring flaperons. Drawing from previous MQP's aircraft designs, the wing design was based on the construction from the 2012 MQP which featured a similar design made of balsa wood ribs wrapped in Mylar to create the aerodynamic surface. The 2018-2019 wings are a combination of birch aircraft plywood and balsa ribs connected together with a carbon spar. Unlike the 2012 MQP, this year, the team decided to use MonoKote because it has better adhesive properties and is more durable than Mylar. MonoKote is a common material in RC aircraft construction similar to Mylar or heat shrink wrapping. The team attempted to manufacture wings from polystyrene, but based on the shape and size of the selected airfoil, decided to move forward with the aforementioned design.

The spacing between the ribs was chosen to minimize the dips in the MonoKote once the heat shrinking process was complete. An image displaying a solidworks rendition of the skeletal structure of the wing is shown below in Figure 44. A physical version of the wing constructed for flight tests appears in Figure 45 displaying both the MonoKote covering and individual wing ribs.

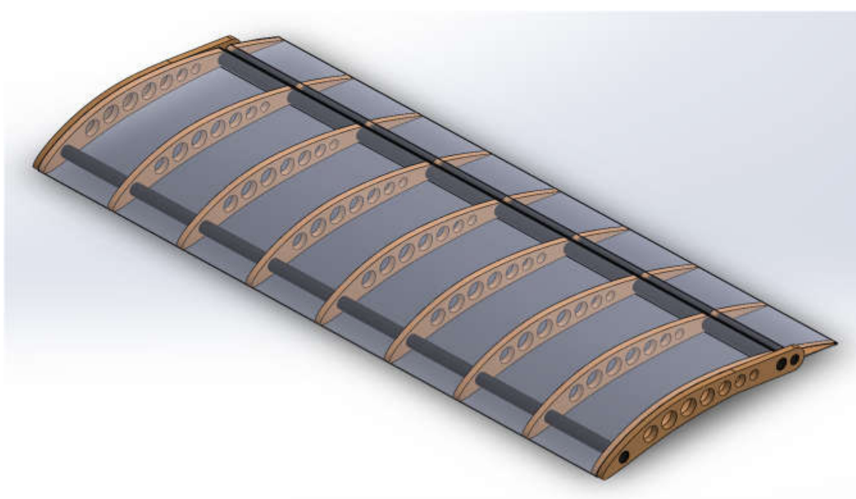

Figure 44. SolidWorks model of wing section with flaperon

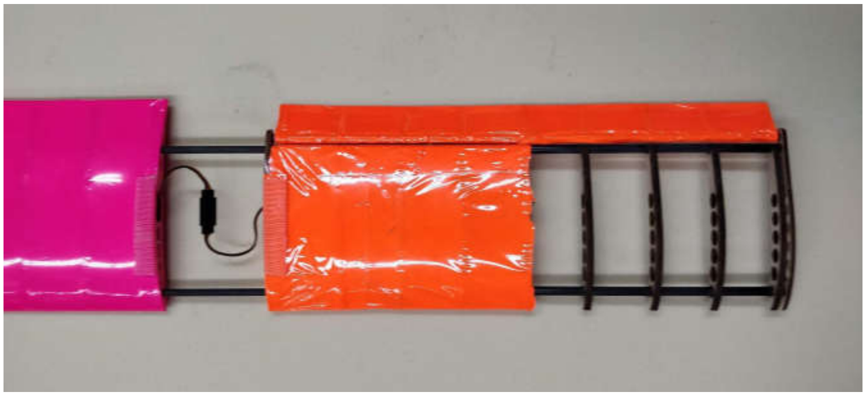

Figure 45. Structure of wing with MonoKote removed

3.3 Tail Construction

The tail was laser cut from a flat sheet of eighth inch birch plywood and covered in MonoKote. Additionally, the tail was mounted at an incidence angle of -4° based on the moment calculation from the stability and controls team. Figure 46 shows the horizontal tail structure used for glide testing.

Figure 46. Structure of the vertical and horizontal tail used for glide testing

The tail cap seen in Figure 47 below held the horizontal and vertical tail at the required -4 degree incidence angle. The tail control surface servos were also glued to the tail cap and rear section of the aircraft spine. The cutouts in the upper left and lower right of the tail cap allowed for the servo control arms to exit the body and attach to the control surfaces.

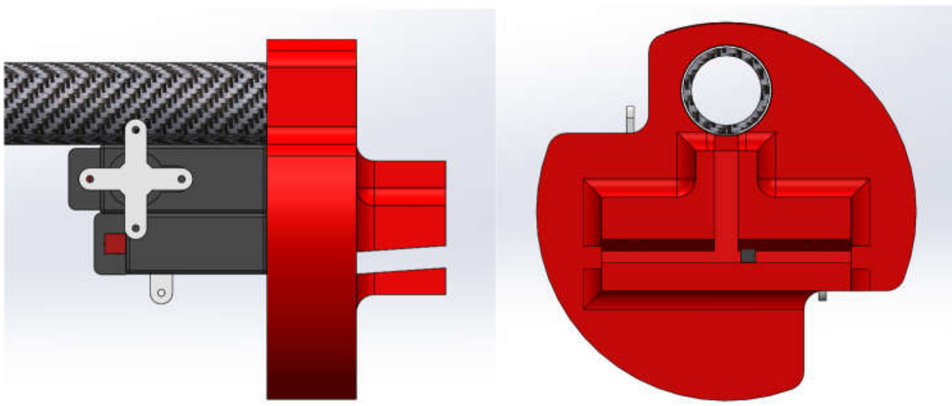

Figure 47. Tail cap side (left) and rear (right) views

The horizontal and vertical tail were both laser cut from plywood and covered in MonoKote for a smooth, consistent surface. The vertical tail had a tab that fit into a complementary slot in the horizontal tail. The assembled vertical and horizontal tail formed an inverted T shape that slid into the tail cap. The tail sub assembly can be seen in Figure 48 below. The T shaped tab that fit into the tail cap is highlighted blue.

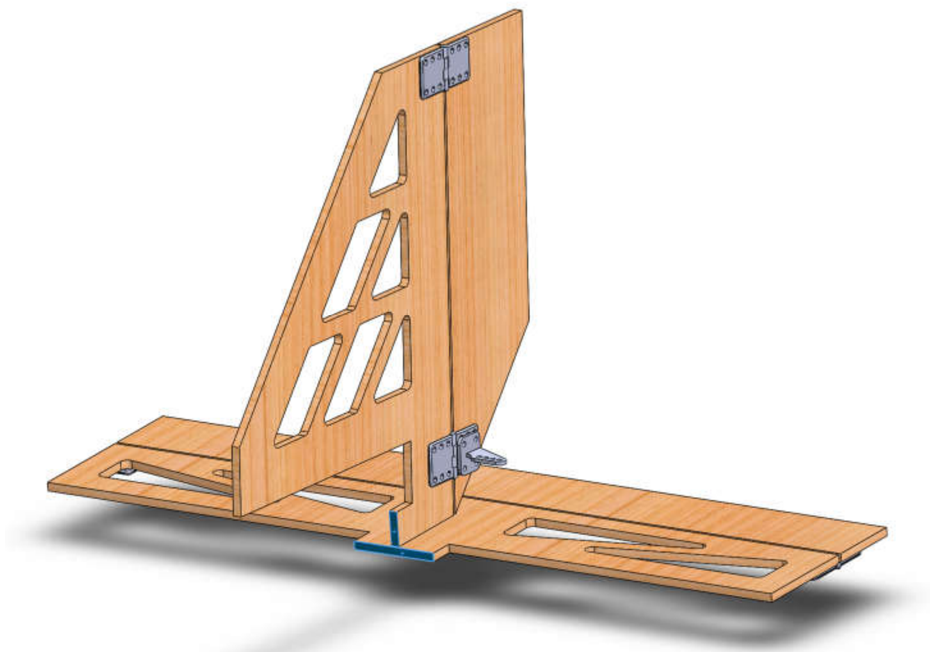

Figure 48. Assembled vertical and horizontal tail.

The tail was then pinned by two nails through precut holes in the locations indicated by the arrows in Figure 49 on the tail cap.

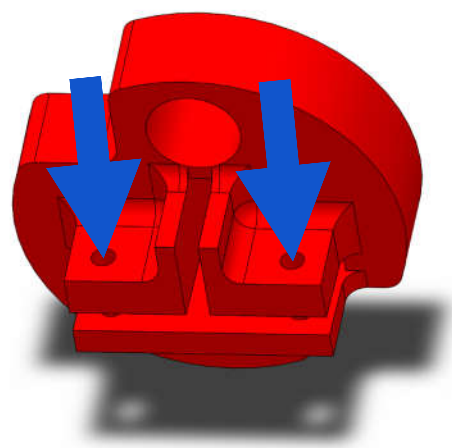

Figure 49. Top view of tail cap

Many of the complex parts on the aircraft were not possible to manufacture without the use of 3D printing. The rapid prototyping lab in the Foisie Innovation Studio allowed the team to create these complex parts quickly with minimal cost.

3.4 Electrical Wiring

The electrical components of the aircraft were wired according to Figure 50. In order to aid in simplifying the assembly of the aircraft, extension wires were used with XT60 clips for all power and ground leads (red and black), bullet connectors for the three motor phase wires (green, orange, and blue), and three pin connectors for the signal leads (purple). This allowed for ideal placement of each electrical component and for body sections to remain wired even when disassembled and packed.

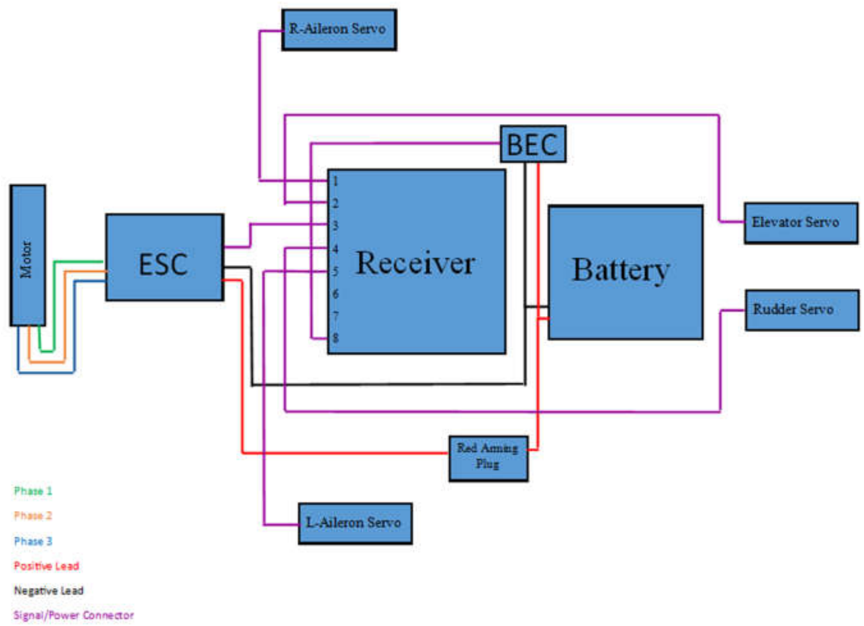

Figure 50. Electrical Wiring Diagram

4. Preliminary Testing

Throughout the iterative design process, the team performed testing on specific components including the airfoil, power plant, and structure. The team also performed hand launch and glide tests to ensure that the aircraft weight would be feasible to hand launch and the aircraft would remain stable throughout flight. The following sections outline these preliminary tests prior to testing the powered aircraft.

4.1 Airfoil Testing

Once the XFLR5 theoretical results were completed for the airfoil, the team sought to replicate the results with numerical data from wind tunnel testing for comparison. Initially, the team used the 8”x 8” open circuit wind tunnel in HL 216 to test a full wing section, but the wing tip effects and aspect ratio did not scale properly to the Reynolds number. In order to counteract these challenges, the team designed a scaled down wing model for the large 24”x 24” recirculating wind tunnel in HL 016 that had a max air velocity of 180 ft/s (122.7 mph). This wind tunnel featured a closed loop design that had temperature control for higher velocity experiments. The tunnel was also equipped with a force balance to measure the axial and normal loads created by an object in the tunnel. To avoid wingtip effects, the scaled wing needed to have at least one chord length of space on either side of the wingtips. With this design consideration, the team maximized the wing size while matching the aspect ratio to get the following dimensions shown in Table 18.

Table 18. NACA 4410/9410 Scaled airfoil parameters

Parameter Name	Scaled Wing Parameters
Aspect Ratio	9.4545
Chord Length	2.0952 (inches)
Wingspan	19.8 (inches)
Reynolds Number	127,293
Operational Velocity	78.4 (mph)

In order to produce results comparable to XFLR5, the wind tunnel had to maintain a speed of 70 mph during testing. The scaled wing was 3D printed into three sections of 75% infill PLA. The wings were joined using carbon fiber rods, hot glue, and duct tape. Once complete, the team followed the procedure of operating the high velocity wind tunnel as seen in Appendix G . The test setup with the scaled down wing appears in Figure 51.

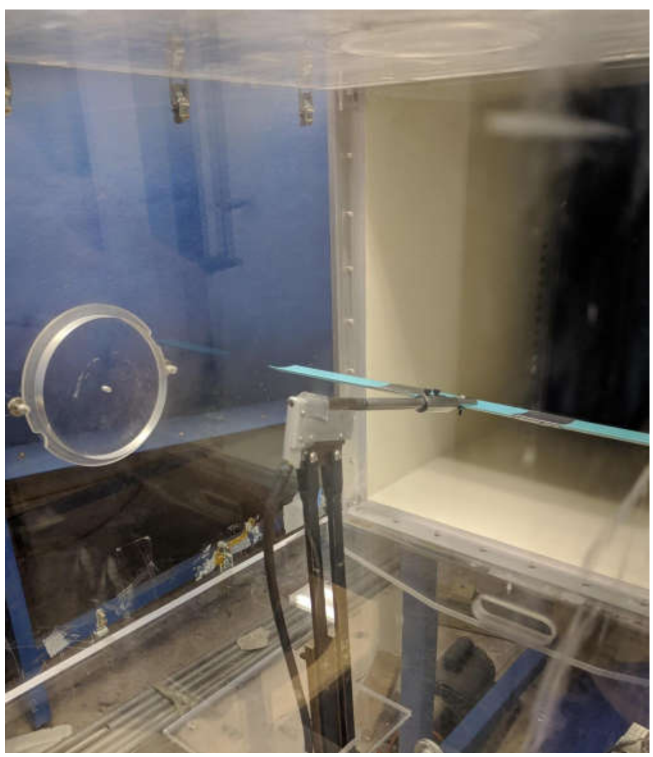

Figure 51. Wind Tunnel Operation with NACA 9410

Testing both the NACA 4410 and 9410 provided the team with adequate results to confirm the test data with the XFLR5 results. Figure 52 shows that the results from XFLR5 and the wind tunnel results produced similar lift curve slopes, but the maximum coefficient of lift values differed. The team speculated this was caused by vibrations in the test chamber and additional wing tip vortices because of the large camber of the airfoil. The wind tunnel tests for the NACA 4410 followed the same wing construction and tunnel procedure as the NACA 9410 and produced more accurate results as seen in Figure 53. The stall angle of attacks were similar for the wind tunnel and XFLR5 curves for the NACA 9410 airfoil.

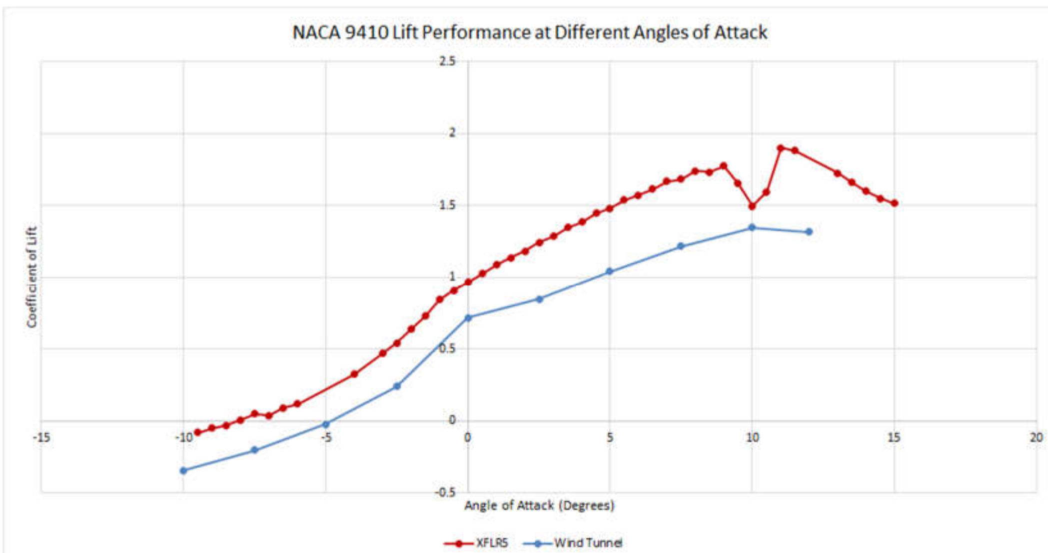

Figure 52. Comparison between XFLR5 and Wind Tunnel data for NACA 9410

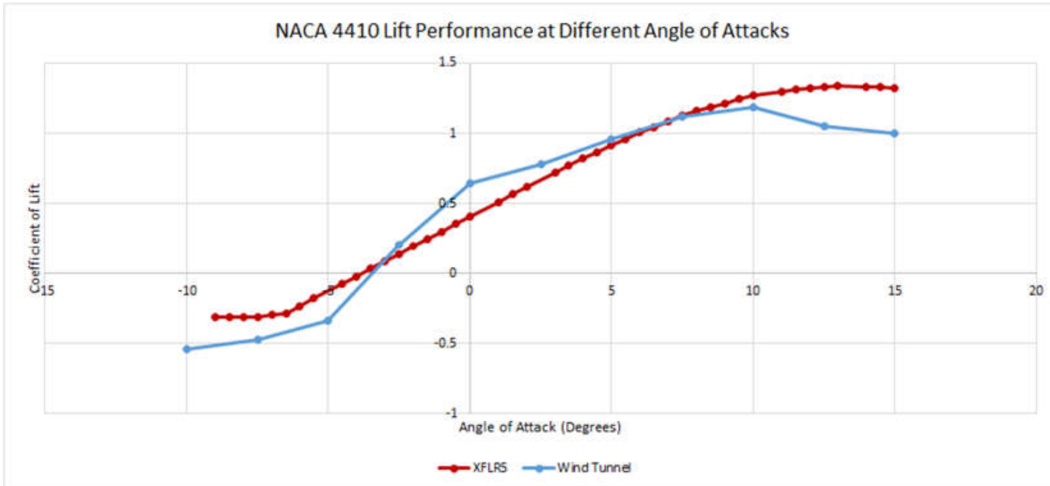

Figure 53. Comparison between XFLR5 and Wind Tunnel data for 4410

The team considered wind tunnel testing a scaled-down version of the aircraft for further analysis and verification. The aircraft was scaled down to be 2.6262 times smaller than the actual aircraft. The aerodynamics team proceeded with the development and assembly; however, the scaled-down aircraft was too weak and structurally unsound to withstand the high speeds of the wind tunnel. Therefore, the team concluded that the wind tunnel testing of the airfoil alone was sufficient.

4.2 Power Plant Testing

In order to verify that the power plant performed as specified by the manufacturer and met the team's performance expectations, the team ran several tests prior to putting it on the aircraft. In order to test the power plant, the team used two pieces of lab equipment. The first was an RC Benchmark 1520 Series thrust testing stand. This testing stand used a strain gage-based testing rig that provided live data on the thrust, current, and voltage of the motor through the use of a computer interface. The testing stand was capable of measuring up to 11 pounds of force, 1,000,000 RPM, 35 V, and 40 A continuous (Series 1520 Thrust Stand, n.d.). Proper use of the test stand can be viewed in Appendix H. The second piece of lab equipment was the closed loop wind tunnel mentioned in section 4.1.

4.2.1 Static Throttle Ramp

For the first test, the throttle was gradually increased from 0% to 100% at a static, non-moving flight condition. Throughout testing, the current, voltage, throttle setting, and thrust were recorded. These values were then compared with data collected from MotoCalc 8, as shown in Figure 55.

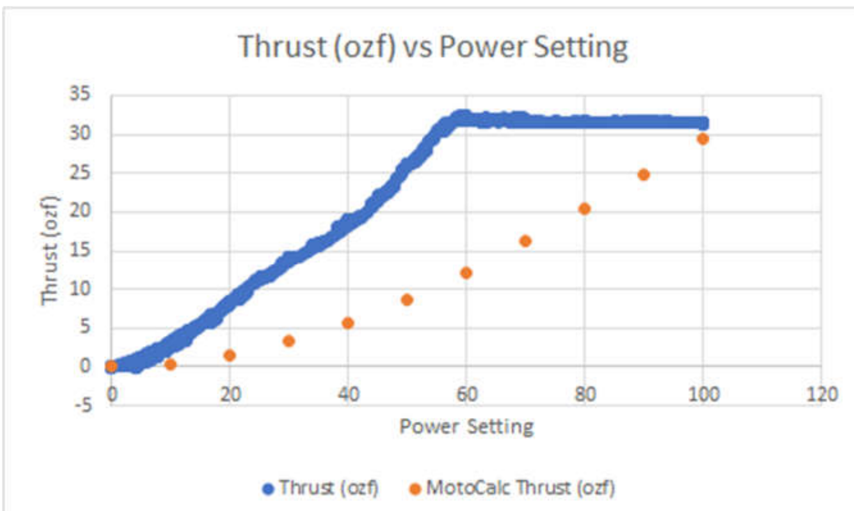

Figure 54. Thrust vs Power Setting

From this graph, it is evident that the motor had a throttle dead zone above 60%, which meant the thrust reached its maximum at 60% power and did not increase beyond that point. However, the MotoCalc 8 data had no throttle dead zone. To correct for this, the team made the 60% power setting equivalent to 100% as shown in Figure 55.

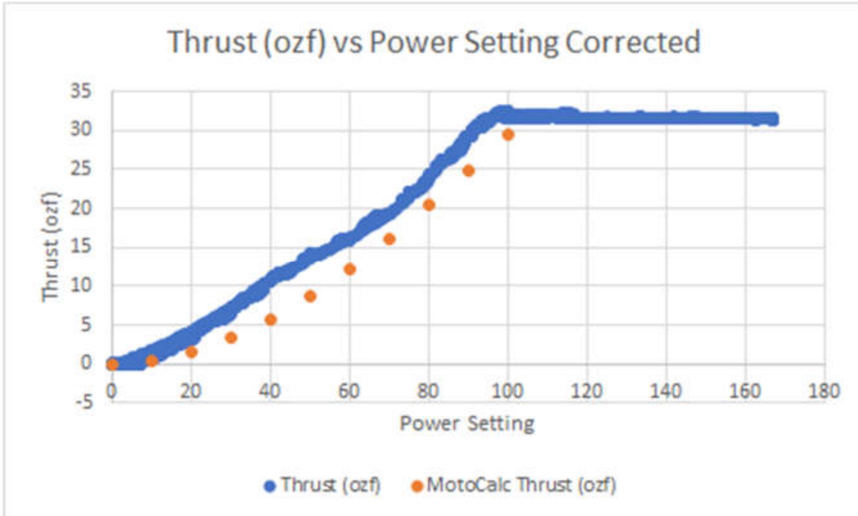

Figure 55. Thrust vs Corrected Power Setting

The corrected data shows that the actual thrust from the motor was equivalent to the data given by MotoCalc 8. In addition to the above curves, the team was also interested in the amount of current the motor drew compared to the thrust output as shown in Figure 56.

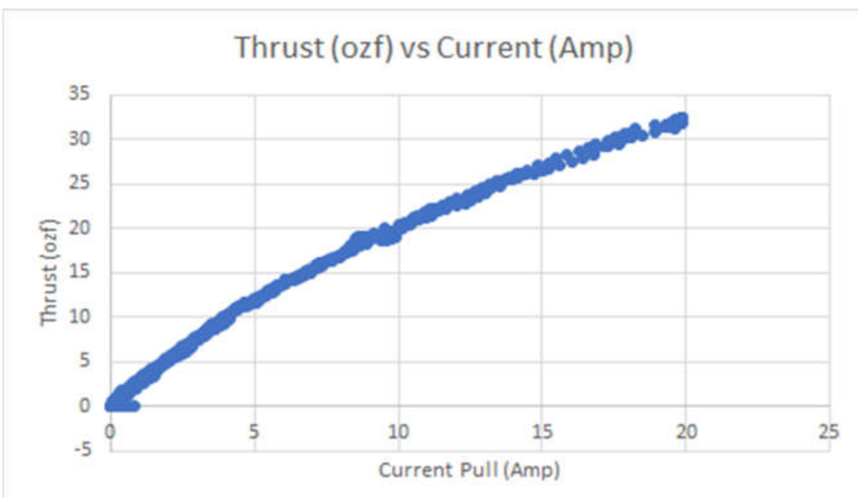

Figure 56. Thrust vs Current graph displaying the current pull at maximum thrust output

4.2.2 Static Battery Life at Maximum Power

For the static battery life test, the team wanted to see how quickly the battery would drain to ensure that the aircraft could complete the course before draining the battery to an unsafe voltage. To test this, the team ran the motor at full power until the motor reached a voltage of 9.4

V, which is 0.4 V above the minimum safe value to run a 3-cell LiPo. The results are shown in Figure 57 and 58.

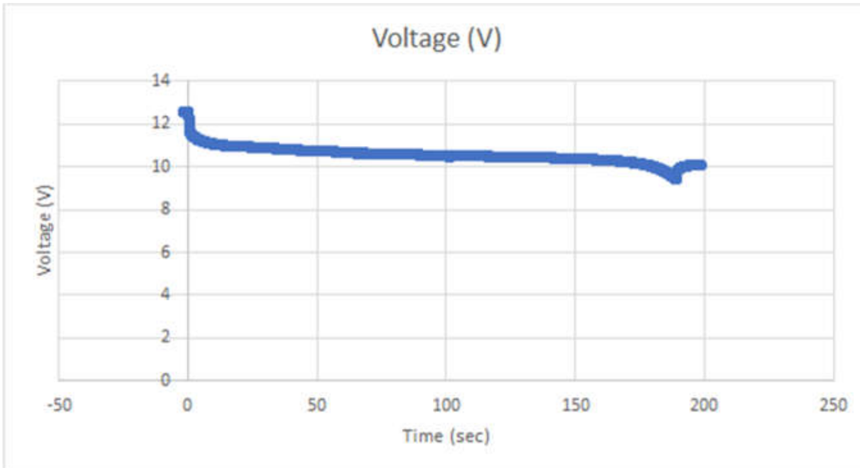

Figure 57. Voltage over time displaying the maximum battery life possible at full power

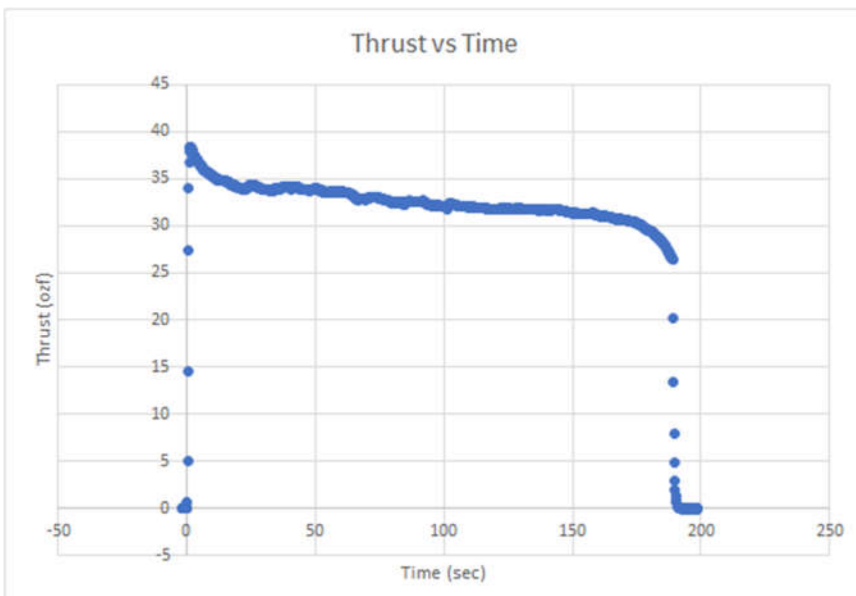

Figure 58. Image highlighting the decrease in thrust with respect to time at maximum power

These plots showed that both the thrust and the voltage at maximum power remained relatively constant with a slight downward trend throughout the life of the battery. However, at the end of the battery's life there was a steep voltage and thrust drop off. At approximately three minutes the battery was unsafe to operate and the RC benchmark software automatically cut power to the motor.

4.3 Structural Loading and Critical Margins

In order to ensure that the materials chosen for construction of the aircraft provided enough durability to withstand the forces of flight and impact, the team conducted two structural loading tests. These were a spar stress test and a wing loading test.

4.3.1. Spar Bending Stress Analysis

The first structural test was the spar stress test. The team calculated the bending moments by estimating the rectangular wings as cantilever beams with uniformly distributed loads. The moment from a uniform distributed load is equal to the moment applied by a point load at the centroid of the distributed load with a magnitude equal to that of the distributed load times its length. The team also wanted the aircraft to withstand maneuvers of up to 3G, which meant each wing should be able to support an equivalent weight of 5.175 lbs. The moment applied to the spars at the root of the wing was determined by equation 4.3-1 below where $w$was the loaded weight of the aircraft, L was the load factor, and b was the wingspan.

$$M = \frac{w\,L}{2} * \frac{b}{4} \qquad (4.3\text{-}1)$$

Substituting values for the team's aircraft and solving the equation gave a maximum moment of 67.5 lbf-in. From the bending moment, the maximum normal stress that any point in the spar experienced was calculated from equation 4.3-2 below. M was the applied moment, y was the distance from the centroid, and $I_{xx}$was the area moment of inertia of the spar.

$$\sigma_{xx} = \frac{-M\,y}{I_{xx}} \qquad (4.3\text{-}2)$$

The pultruded carbon fiber tubes used to attach the root of the wing to the body had an outer diameter of 0.211 inches and an inner diameter of 0.132 inches, the area moment of inertia for a hollow tube of the aforementioned dimensions is $I_{xx} = .0000824\ in^4$. The design used two carbon fiber spars, so the moment each spar needed to support was 33.75 lbf-in. Substituting in these values resulted in a stress of $\sigma_{xx} = 43\ 200\ psi$. The flexural strength of the pultruded carbon fiber tubes used for the aircraft was listed as 200 000 psi, therefore even in a 3G maneuver the spars were only expected to experience 21% of their maximum flexural strength.

4.3.2. Wing Loading Test

The team conducted a wing loading test with a test wing to prove that if the plane underwent a high force maneuver, the structure of the wings would support this additional loading without fracture. The team conducted the test with half the full wing held in place at the root and 5.175 lbs acting on the wing tip (equivalent moment at the root for a 6G maneuver). Figure 59 below shows an image of the loading test on the half section of the wing.

Figure 59. Full wing load test

Despite the large deflection, the wing did not fracture and returned to its original shape after the load was removed. This was a load condition that would never be seen in normal flight because wing loads are distributed across the entire length of the wing and not concentrated at the tip. However, it confirmed that the wing structure would be able to withstand extremely high stress maneuvers.

4.4 Hand Launch Test

For a hand launch test prior to launching the glide prototype of the aircraft, the team acquired a radar gun, courtesy of the WPI Baseball team, to see how fast a member of the team could throw 5 lbs of material. The material was thrown slightly above 20 mph, confirming that the team was able to throw the aircraft above the NACA 9410 stall speed to achieve lift. Glide testing then began once the unpowered aircraft was assembled.

4.5 Glide Test

Prior to assembling a powered aircraft, it was necessary to ensure that the aircraft was stable and durable in a non-powered glide test. The glide test proved the integrity of the stability and controls calculations, showed that it could be hand launched effectively, and proved it would hold up structurally to both normal flight and crashes. The team conducted the first glide test of the aircraft on December 11, 2018. The full glide prototype appears in Figure 60. The glide test procedure can be seen in Appendix I. The aircraft was thrown for multiple distances and from multiple heights, both unloaded and loaded with the two PVC pipes of the body in place. The overall result of the glide test was successful.

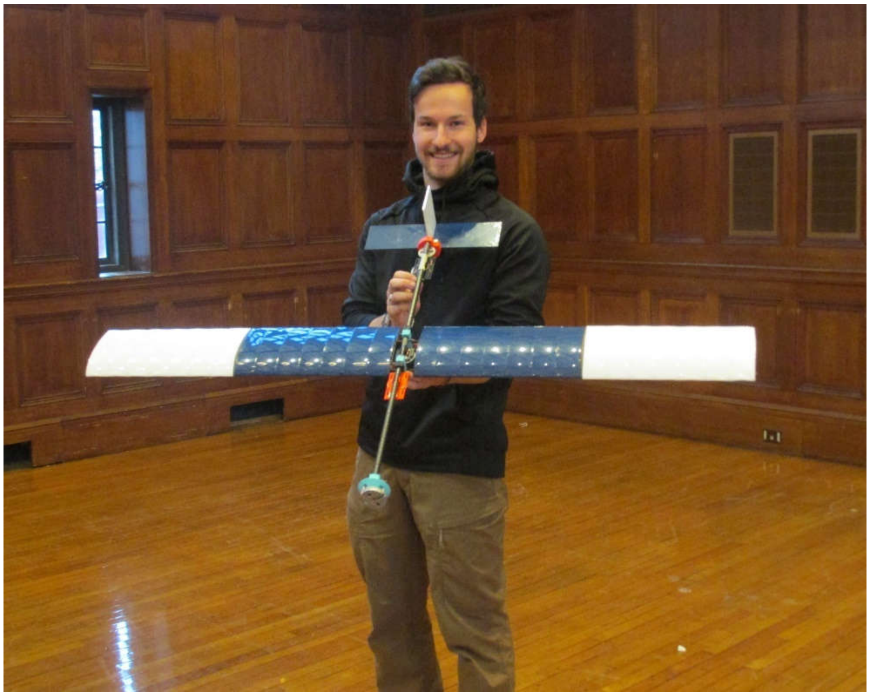

Figure 60. Glide prototype prior to glide test

The aircraft successfully glided for a short period of time over 15-30 feet. The only issue the team encountered with the aircraft during testing was a counterclockwise roll as seen from the rear of the aircraft. Rotational rigidity of the fuselage was an issue as the tail rotated relative to the wings, potentially contributing to the roll instability issues. The plywood used for the horizontal tail was also slightly warped which may have contributed to the roll instability. The aircraft held up well to repeated crashes onto a hardwood floor proving the structural integrity of the airframe. However, the wings became a concern after repeated impact on the floor; the wing spars detached from several of the ribs. Additionally, the point where the wings attached to the fuselage failed after 6 glide test throws, and the nose cone shattered on a hard nose dive impact, as shown in Figure 61. The NACA 9410 airfoil provided sufficient lift to sustain flight while not creating an uncorrectable pitching moment, and the tail incidence angle proved to counteract the moments produced by the airfoil.

Figure 61. Glide prototype after repeated glide tests. Note: wing spars cracked, and nose cone was replaced with duct tape after it shattered

5. Powered Flight Testing and Revised Design

5.1 Powered Flight Testing of Initial Design

After a successful glide test, the team decided to begin powered testing. The team began by testing the unloaded configuration of the aircraft without stability augmentation. Flying under these conditions allowed the team to focus on the stability and overall maneuverability prior to adding payload, as the higher weight required greater thrust. The first few powered flight tests showed that there was a stability issue with the aircraft as it entered multiple flat spins which were unrecoverable. An experienced RC pilot, Jack Tulloch, aided the team in identifying the major causes of the stability issues with the aircraft which were the size of the vertical tail and the center of gravity location. The following table, Table 19, provides an overview of the failed flight tests, their results, and the improvements made to the aircraft following each test.

Table 19. Overview of Powered Flight Tests

Test No.	Date	Configuration	Results	Improvements
1	1/25/19	Unloaded	Flat Spin	Throw Position
2	1/29/19	Unloaded	Flat Spin	New Pilot
3	2/6/19	Unloaded	Flat Spin	New Tail, CG Location

5.2 Improvements on Initial Design

5.2.1 Revised Vertical Tail

The team noticed throughout the initial powered flight tests that there was a major stability issue with the aircraft. The test pilot, Jack Tulloch, confirmed that there may have been an issue with the vertical tail. With this observation, the stability team analyzed the original vertical tail by calculating the vertical tail volume ratio using equation 5.2-1 below where S_v was the area of the vertical tail, l_v was the moment arm of the vertical tail, $S$was the area of the wing, and b was the wingspan.

$$V_v = \frac{S_v l_v}{Sb} \qquad (5.2\text{-}1)$$

Based on historical values found in literature review, the vertical tail volume ratio should have been between 0.02 and 0.05. With the original tail sizing, the stability team found that the vertical tail volume ratio was 0.011, which was too low. The team decided that a volume ratio of 0.03 would provide sufficient area for the vertical tail while maintaining reasonable dimensions for the aircraft packing. The volume ratio of 0.03 required the vertical tail to have an area of 0.2381 ft^2(34.29 in^2). The incidence angle of the tail was also changed from -6.77 degrees to -4 degrees to maintain stability with the new tail size. The stability team calculated the incidence angle using the same method as for the previous tail design. Figure 62 below shows the new dimensions of the tail.

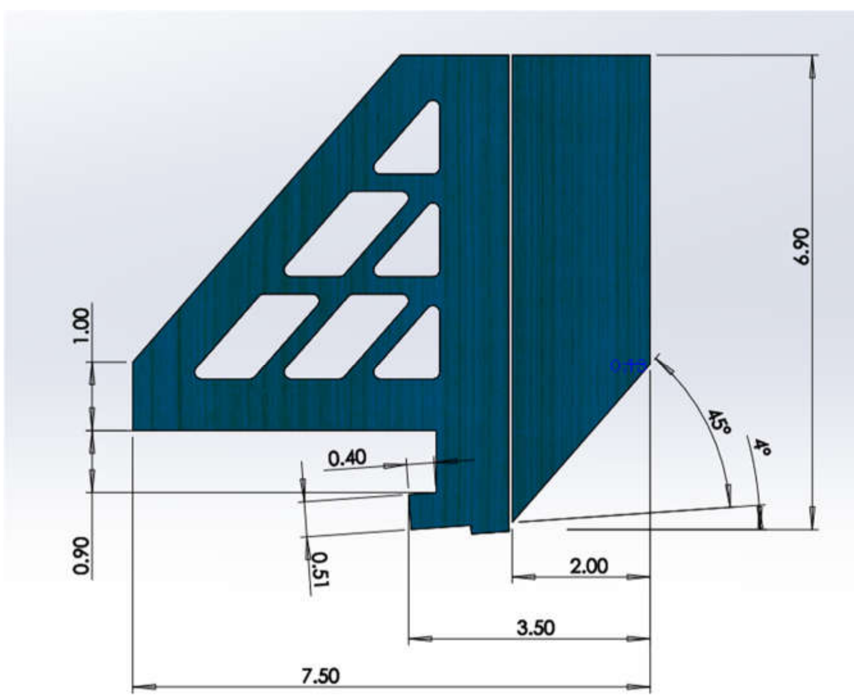

Figure 62. Revised Design Tail Dimensions

The increase in size of the vertical tail also led to an increase in the size of the rudder. The percentage of the vertical tail area dedicated to the rudder was 35%. This percentage was the same as it was for the original tail, however as the overall tail area increased, so did the rudder area. The rudder area increased to 0.083 ft^2 (12 in^2). The area increase of the rudder caused the required torque for the rudder servo to increase. Using the same method that determined the original servo sizing, the stability team determined that the original rudder servo was no longer sufficient. The torque required increased to 19 oz/in. A new servo, the HS-53, was chosen for the rudder with a max torque output of 20.8 oz/in. The stability team determined that the torque output would be sufficient for normal operation of the aircraft.

5.2.2 Center of Gravity

Based on the flight test and the input of the test pilot, the revised design moved the CG slightly further forward on the wing by moving the battery to the front assembly of the aircraft. With the larger tail and change in position of the battery, the new unloaded CG was moved to 27.6% of the chord and the loaded CG was moved to 32.2% of the chord as seen in Figure 63.

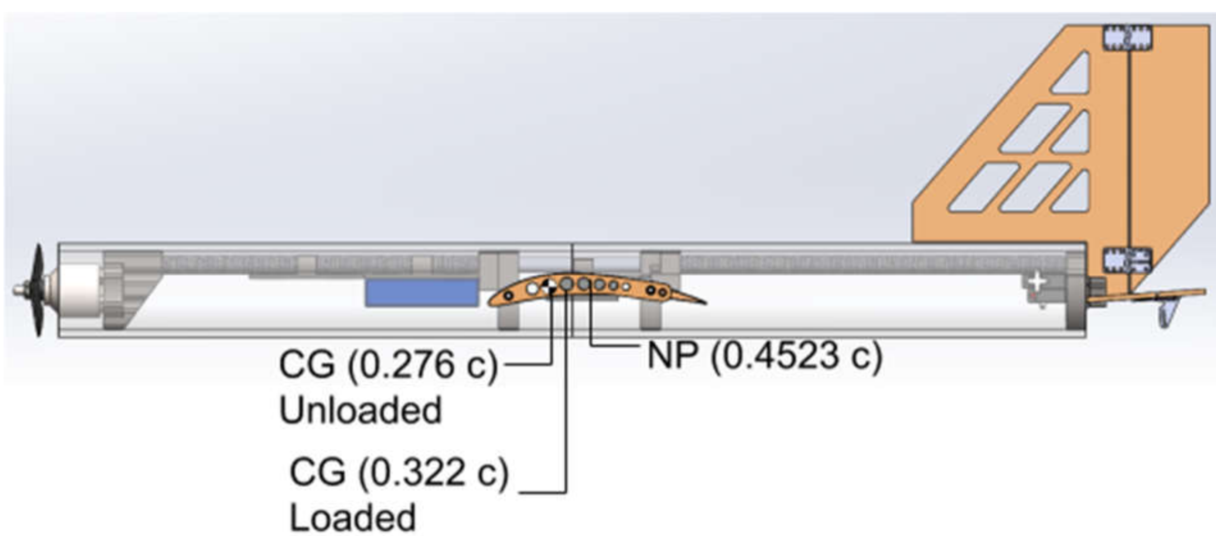

Figure 63. Neutral point location and center of gravity location when loaded and unloaded

5.2.3 Revised Power Plant

During the third powered flight test, the aircraft entered a nose dive into the ground, which sheared the motor shaft of the Turnigy Park 480 1320Kv motor. Unfortunately, the manufacturer no longer had these motors in stock and was unsure when the motor would be available again. Due to this, the team had to find a suitable replacement.

5.2.3.1 Power Plant Selection

When selecting a new brushless motor, the team considered motors with higher thrust values in order to improve the overall performance of the aircraft. Higher thrust would allow the aircraft to fly at faster speeds, improve the takeoff capability while carrying payload, and allow for a larger margin of safety on the required thrust. In order to determine a suitable replacement, the team looked into the previously analyzed motors, as well as several new motors from T-Motor and Cobra. Ultimately, the team decided on the Cobra 2814-12 (1390Kv) motor due to its high thrust output. This motor weighed 0.271 lbs which was approximately 60% more than the previous motor (Cobra C-2814/12 Brushless Motor, Kv=1390, n.d.). This increase in weight helped move the center of gravity forward, which was desired given the results of earlier flight testing. Table 20 below shows a performance comparison of a variety of propeller sizes paired with the Cobra 2814-12.

Table 20. Motor Data (eCalc: Cobra C-2814/12 (1390), n.d.) (Cobra C-2814/12 Motor Propeller Data, n.d.)

Prop	Volt	Amps	Watts	RPM	Pitch Speed	Thrust	eff	Stall Speed	Static Thrust	Cruise Thrust	Mixed Flight Time
7x4	11.1	13.64	151.4	13361	50.6	26.1	4.88	0	24.1	16.1	4.3
7x5	11.1	17.64	195.8	12985	61.5	27.34	3.96	21.1	29.1	21	3.9
7x6	11.1	18.64	206.9	12887	73.2	29.88	4.09	37.9	33.7	23.2	3.6
8x4	11.1	20.78	230.6	12700	48.1	37.46	4.6	0	32.6	21.6	3.2
8x6	11.1	31.14	345.6	11716	66.6	42.89	3.52	26.7	44.1	33.2	2.8
8x8	11.1	39.36	436.9	10961	83	38.06	2.47	54	53.8	31.7	2.6
9x4.5	11.1	28.94	321.3	11941	50.9	52.1	4.6	0	44.6	30.2	2.5
9x6	11.1	33.09	367.3	11528	65.5	50.34	3.88	0	53.8	39.5	2.4
9x7.5	11.1	45.9	509.5	10253	72.8	49.21	2.74	37.3	61.5	43.5	2.2
10x5	11.1	39.26	435.8	10967	51.9	61.34	3.99	0	55.9	38	2.1
10x6	11.1	41.43	459.8	10745	61.1	64.52	3.98	0	61.9	44.2	2
Source	Manufacturer							eCalc			
Lower Limit	9.9	0	NA	NA	50	30	NA	0	30	15	1.5
Upper Limit	12.4	35	450	NA	NA	NA	NA	20	NA	NA	NA

In the table above, the blue cells represent propellers that are too small for the chosen motor, green cells represent values that are within the required operating limits of the other electrical components, yellow cells represent values within the required operating limits but with a limit to the time the motor may run at full power, and red cells represent values not within the required specifications. From this analysis, the propulsion team determined that the two best options for propellers were the 9x4.5 and the 9x6. A benefit to using the 9x4.5 propeller was that it drew less current, which would provide a longer battery life. A benefit of using the 9x6 propeller was its higher pitch speed. Both propellers result in similar thrust outputs. Due to this, both propellers were bought and tested to determine the optimum propeller, which is discussed in section 5.2.3.2. The team also tested a 9x5 propeller previously purchased as it was between the two size propellers purchased. Additionally, the propulsion team decided to switch to a plastic propeller because after initial flight tests, the lower fracture toughness of the plastic propeller was actually preferred in order to preserve the structural integrity of other components of the aircraft such as the motor shaft.

The team also decided to purchase an ESC with a higher current rating in order to ensure that the ESC's current rating was not exceeded during flight. The current listed above in Table 20 was an ideal current draw and did not represent the actual current draw due to various flight maneuvers. The propulsion team recognized that quickly increasing the engine power could cause the burst current to exceed the current limit on the 35A ESC, and some maneuvers could add extra load on the propeller, further increasing the current. With this in mind, the team selected the Lumenier BLHeli_32 51A ESC. This new ESC could handle up to 51A continuous current and 80A burst current. In addition, this ESC did not have an internal BEC, so the team purchased the necessary supplies and had Nick Sorensen create a custom BEC. (Lumenier 51A BLHeli_32 32bit 2-6S w/ Telemetry, n.d.).

5.2.3.2 Power Plant Testing

In order to verify the actual performance of the new motor, various tests were conducted. The first test involved testing the engine at various airspeeds in a closed loop wind tunnel to determine the overall performance of the motor. For this test, the motor was run at full power at increasing airspeeds to compare motor parameters such as thrust, power, current draw, and voltage. Data was collected through the use of the RC benchmark test stand.

The first parameter analyzed from this test was the thrust output compared to the theoretical drag of the aircraft, which is presented in Figure 64.

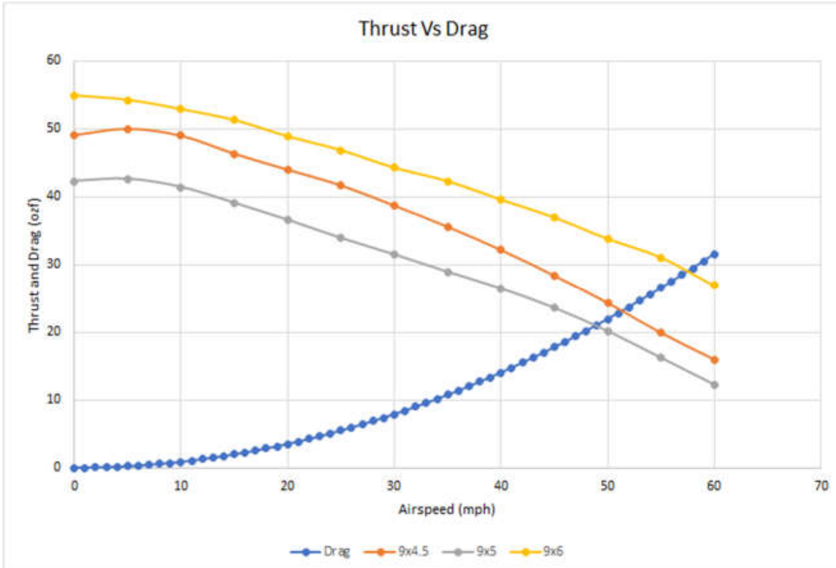

Figure 64. Thrust to Drag Plot

This plot shows that each propeller produced enough thrust to cruise at the target 30 mph speed. Additionally, the 9x6 APC-E propeller produced 55 oz of thrust static, the 9x5 Aerostar propeller produced 42 oz of thrust static and the 9x4.5 APC propeller produced 49oz of thrust. At cruise the 9x6, 9x5, and 9x4.5 propellers produced 44 oz, 31oz, and 39 oz of force respectively. Next the team looked at power to weight ratios, which are presented in Figures 65 and 66.

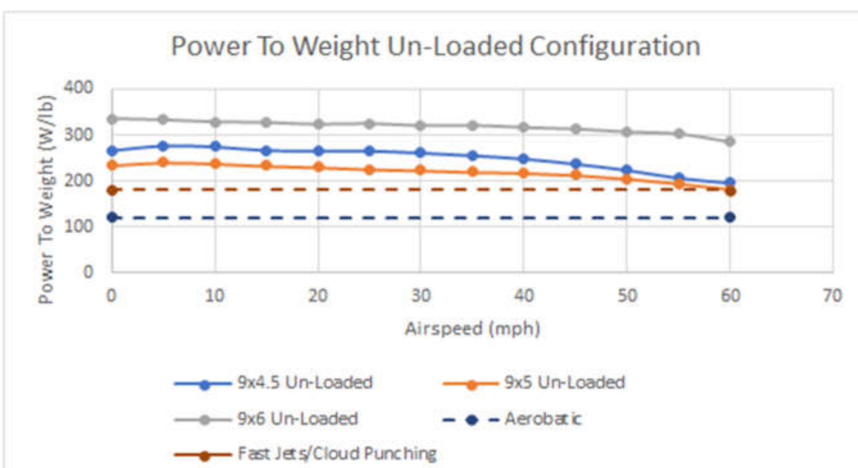

Figure 65. Power to Weight Unloaded

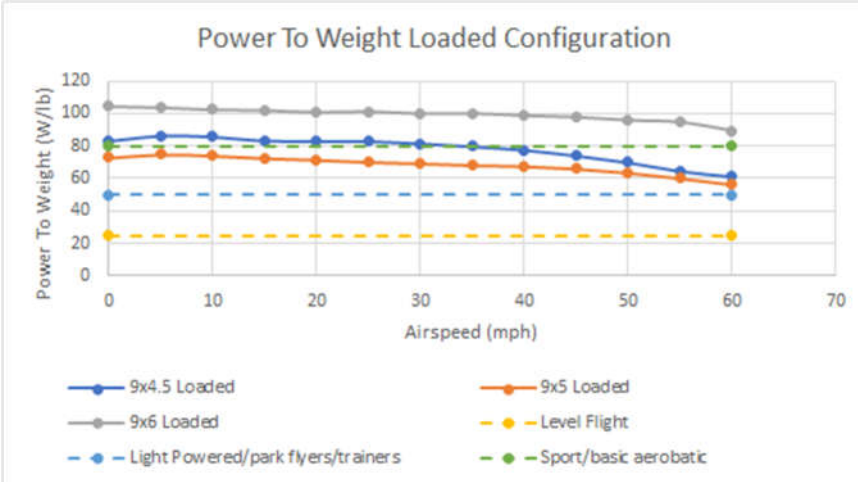

Figure 66. Power to Weight Loaded

These plots showed that in the unloaded configuration, the aircraft was classified as a fast jet/cloud puncher for the 9x6 propeller and aerobatic for the 9x4.5 and 9x5 propellers. In the loaded configuration, the aircraft falls under the park flyers category for the 9x5 propeller and the sport category for the 9x6 propeller. The 9x4.5 propeller falls into the sport category below 30 mph. After 30 mph it is considered a park flyer. These ratings are for the power required only, and the aircraft's ability to actually fly in these categories depends on the structural design and airfoil. The team then looked into the thrust to weight ratios of the aircraft, which are presented in Figures 67 and 68.

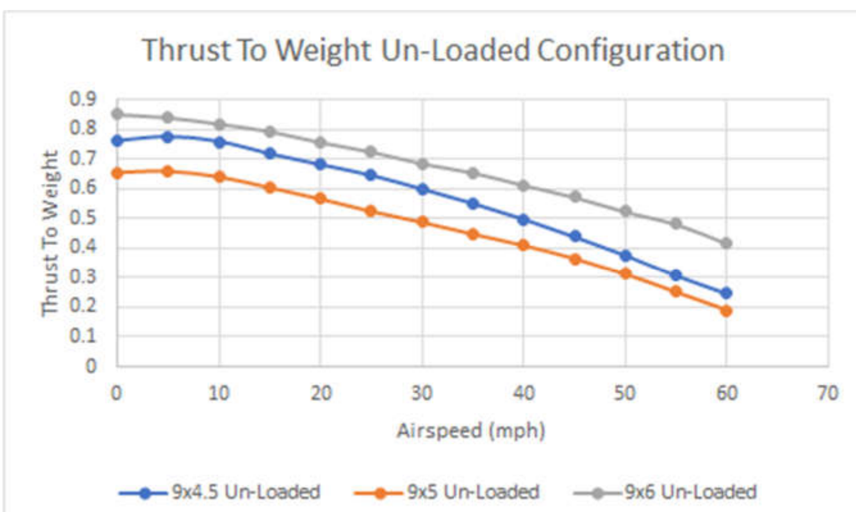

Figure 67. Thrust to Weight Unloaded

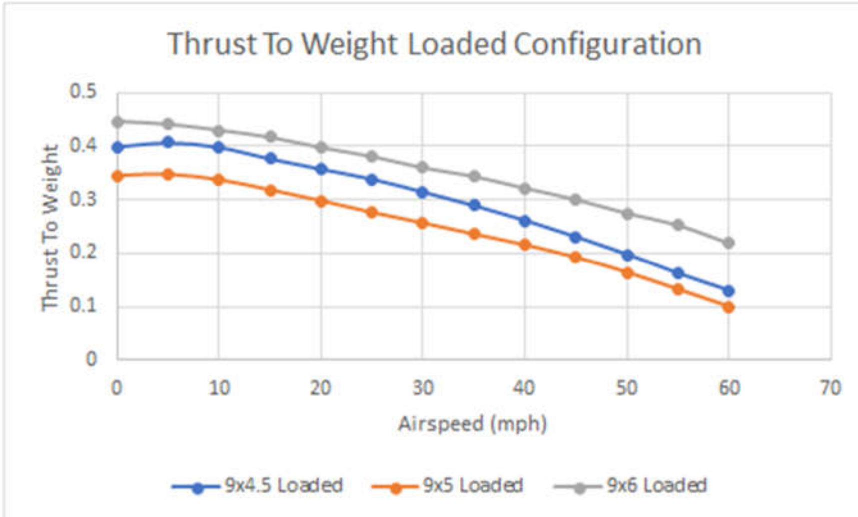

Figure 68. Thrust to Weight Loaded

The above plots show that in the loaded configuration the aircraft had a relatively low thrust to weight ratio. This would limit the turn and climb capabilities of the aircraft; however, it would not hinder its ability to support the added payload.

The next test conducted was a throttle ramp test in order to see the performance of the motor at various throttle settings. In this test, the throttle setting was gradually increased from 0% (ESC signal of 1000) up to 100% (ESC signal of 2000) . This was performed at static conditions and cruise conditions. From the collected data, the propulsion team first examined the thrust at various throttle settings, shown in Figures 69 and 70.

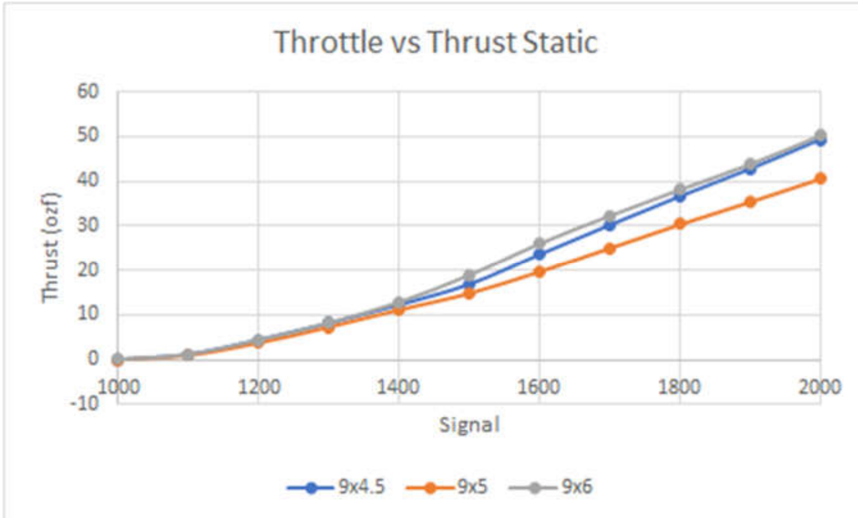

Figure 69. Throttle Level vs Static Thrust

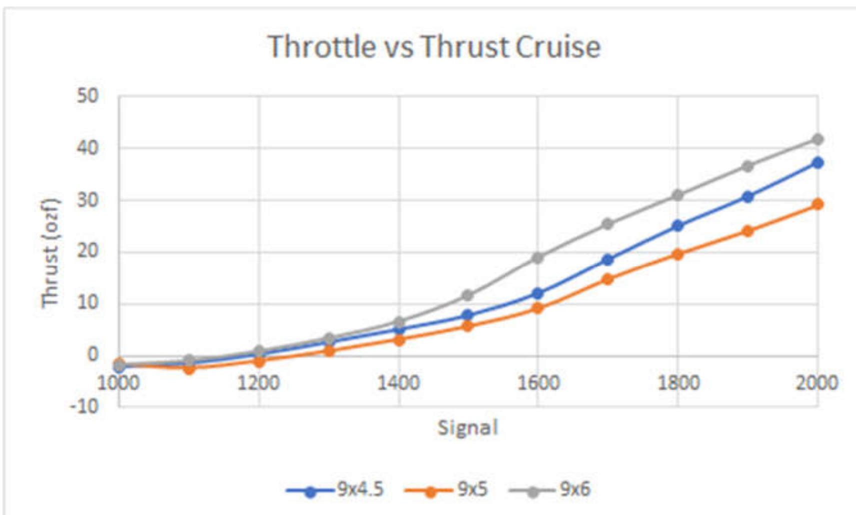

Figure 70. Throttle vs Thrust at Cruise Speed

The propulsion team determined the power setting for cruise from the above plots. From the thrust drag plot in section 5.2.3.1, the required thrust for cruise was approximately 10 oz. Based on Figure 64, at 30 mph, the team achieved a value of 8 oz of drag, and added a safety factor to 20 oz as the cruise power setting to account for wind and unforeseen circumstances. It was then determined that for the 9x4.5, 9x5, and 9x6 propellers the required power settings where 70%, 80%, and 60% respectively. The team then looked into the current draw for the motor at various throttle settings, as in Figures 71 and 72.

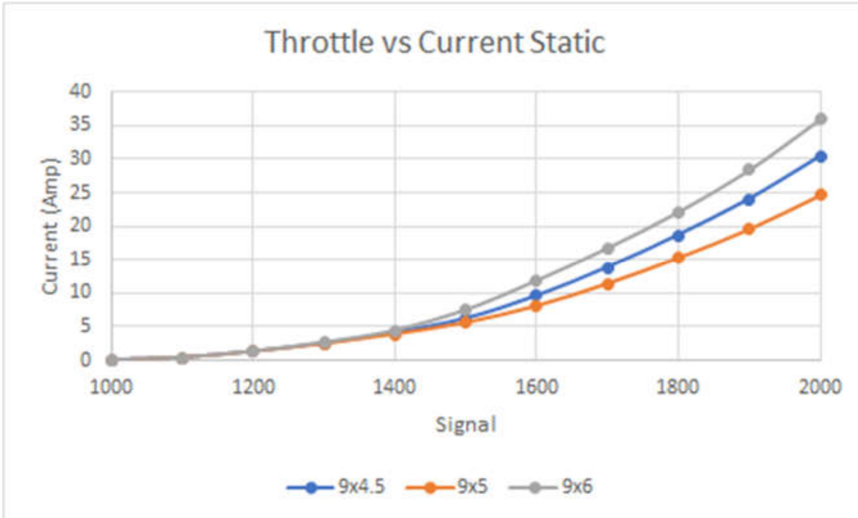

Figure 71. Throttle Level vs Current at Static Conditions

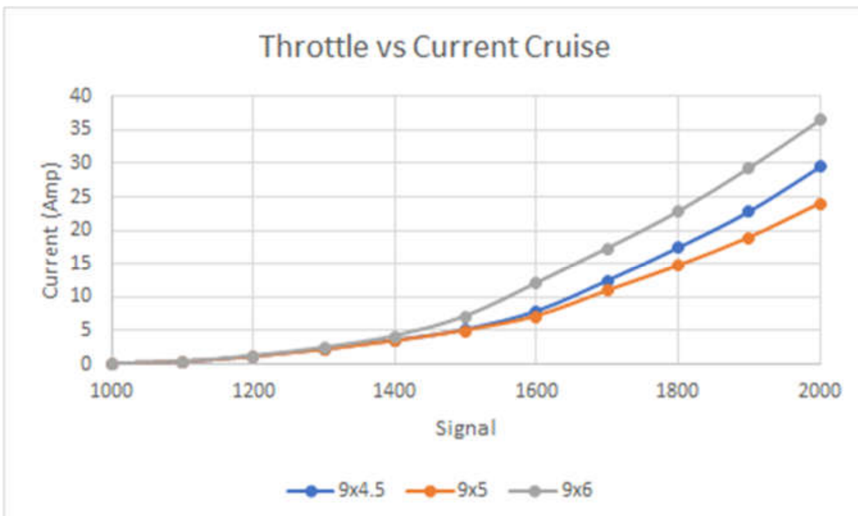

Figure 72. Throttle vs Current at Cruise

This data shows that at maximum power, the 9x6 propeller pulled 36 A in static conditions and 12 A during cruise. The 9x5 propeller at maximum pulled 25A in static conditions and 15 A during cruise. The 9x4.5 propeller at maximum power pulled 30 A in static conditions and 12 A during cruise. Using the max amperage from each category, the propulsion team calculated the new required mAh.

As shown in Table 21, the team determined that the 1000 mAh batteries previously purchased would still be suitable for the aircraft. This table used the same estimated course length and safety factor to calculate the mAh required to complete the course.

Table 21. Updated Battery Analysis

Leg	Speed (mph)	Distance (ft)	Power Setting	Current Pull (mA)	Time of Flight (sec)	Safty Factor	Time of battery life needed (s)	mAh needed
Control Check				15000	5	1.2	6	25
Runup	-	-		36000	5	1.2	6	60
Takeoff	-	50		36000	10	1.2	12.00	120
Climb	25	200		36000	5.45	1.2	6.55	65.45
Leg 1	30	150		15000	3.41	1.2	4.09	17.05
Turn 1	28	471.24		15000	11.47	1.2	13.77	57.37
Leg 2	30	800		15000	18.18	1.2	21.82	90.91
Turn 2	28	471.24		15000	11.47	1.2	13.77	57.37
Leg 3	30	400		15000	9.09	1.2	10.91	45.45
Loiter	30	100		0	2.27	1.2	2.73	0.00
Land	25	200		0	5.45	1.2	6.55	0.00
Totals	-	**2842.48**	-	-	**86.81**	-	**104.18**	**538.61**
Electronic Battery Needs	Current Pull	mAh needed						
Receiver	120	3.47						
Servos flaparon	5000	144.69						
tail/rudder servos	1100	31.83						
Totals	6220	179.99						
Grand Totals								
Time of battery life	**104.18**							
mAh needed	**718.61**							
mAh of batt	1000							
usable mAh	850							
Remaining	131.39							
Total includes a safty factor of 1.2 for time of flight and max current pull of all servos and motor for entire flight								

The last test conducted on the power plant was a battery life test in order to verify that the power plant would be capable of supporting flight throughout the entirety of the competition course. For this test, the team ran the motor at cruise conditions, where the motor was kept with a constant thrust output of 20 oz at an airspeed of 30 mph. The team also ran the motor at static conditions with max power output at 0mph. The control software then automatically shut the engine off once the battery became unsafe to use. This occurred at 10 V for the cruise and static conditions for the 9x4.5 and 9x5 propellers. The max power for the 9x6 propeller had a large voltage dip so the team set the unsafe voltage to 9.7 V.

In order to determine the battery life, the team looked at the ESC signal compared to the time. Once the software hit the cutoff voltage, it dropped the signal to 1000 or 0% throttle. The results of this test appear in Figures 73 and 74 and the total battery life times appear in Table 22.

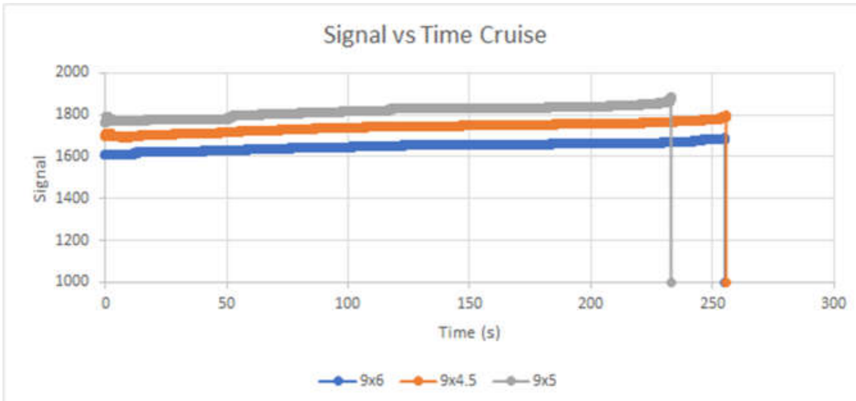

Figure 73. Signal vs Time at Cruise Conditions

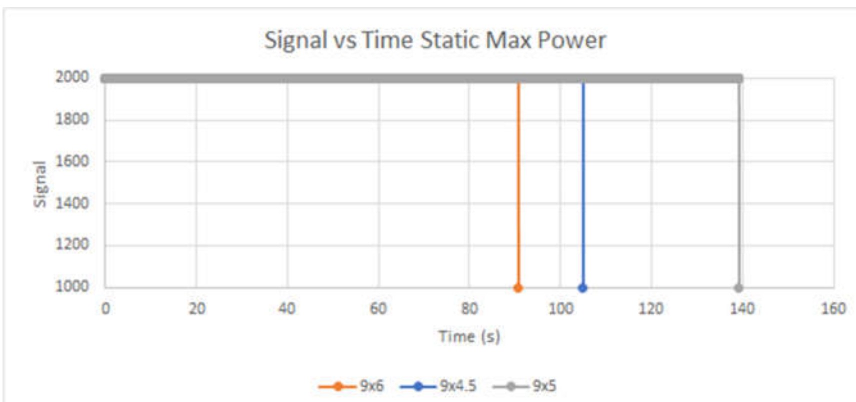

Figure 74. Signal vs Time at Max Power

Table 22. Battery Life

Propeller	Cruise Battery Life (min)	Max Power Battery Life (min)
9x4.5	4.26	1.75
9x5	3.88	2.32
9x6	4.26	1.51

Since the aircraft needed to fly for a maximum of 104 seconds, any of the tested propeller sizes would allow for the required flight time with the 1000 mAh battery.

The last data set the team evaluated was the maximum static thrust output throughout the life of the battery. A graph of this data appears in Figure 75.

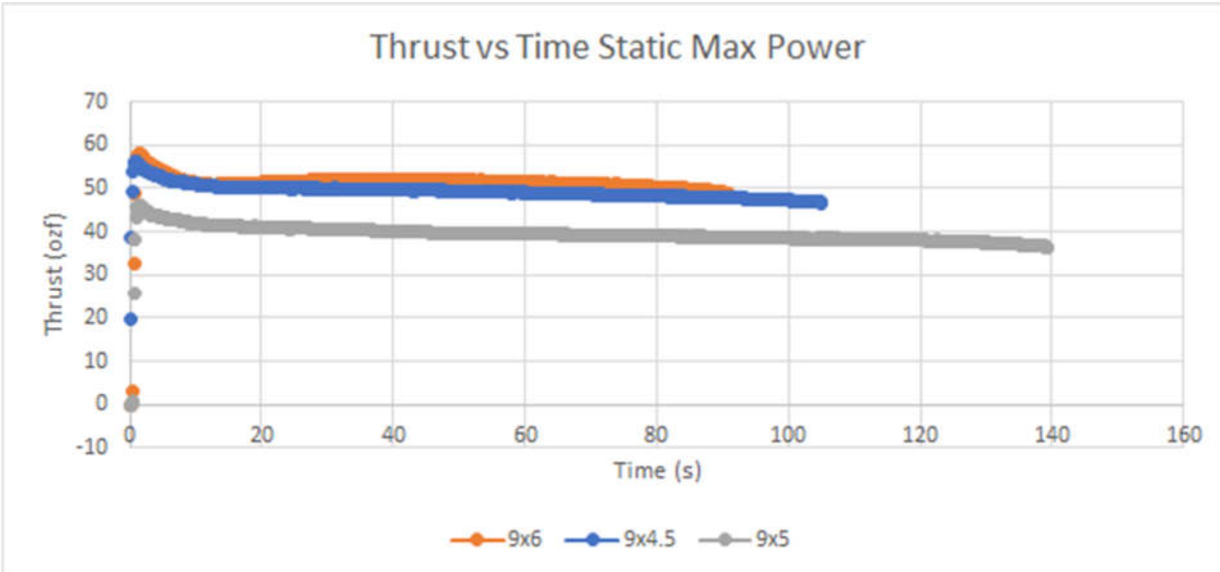

Figure 75. Thrust vs Time at Max Power

Over the lifespan of the battery, the thrust had a small decrease at the beginning of the battery's life span and then remained relatively constant throughout the rest of the battery's life. From analyzing all the above test data, the team decided against the 9x5 propeller. The smaller chord length of the blades resulted in too low of a thrust. Both the 9x4.5 and the 9x6 met all requirements for thrust, power, battery life and current draw. The team ultimately decided upon the 9x4.5 prop due to its slightly lower current pull at the higher power settings.

5.2.4 Revised Structural Components

The initial design of our center connectors included the use of an o-ring to secure the spine sections together however, after initial powered test, the team noticed that motor vibration caused the center connectors to loosen and twist. In flight testing, the team also found that the previous connector design was too flexible and allowed the spine of the aircraft to flex under flight conditions. Due to these concerns, the final design for the center connectors was created as detailed in section 3.1 of the paper. The final rotation lock connectors solved the problems of the o-ring connectors by securely locking all sections of the aircraft together.

The initial aircraft design also had a nose cone that closed off the front of the PVC pipe with the motor mounted to the front of it. The new motor that was chosen was longer than the original, which required designing a new motor mount to ensure the length of the front assembly still fit in the box. The new motor mount recessed the motor into the front payload section to reduce overall length and can be seen below in Figure 76.

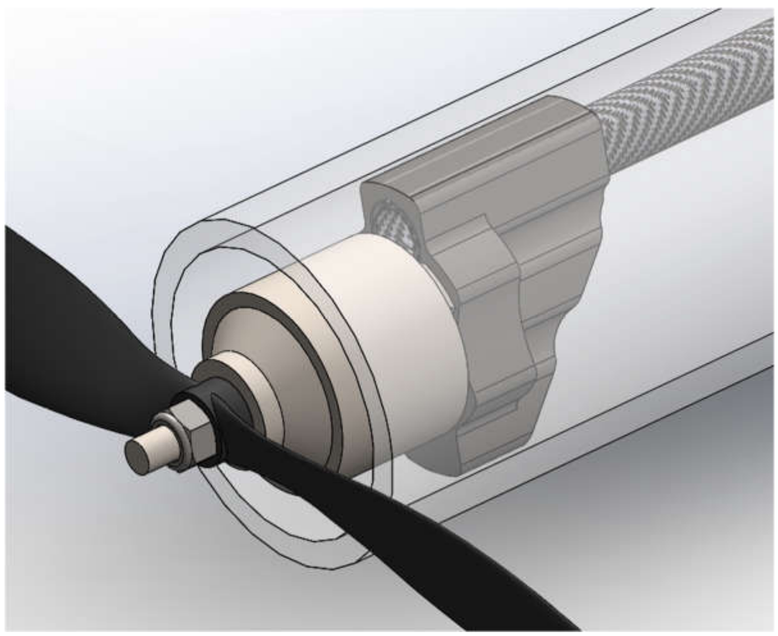

Figure 76. Revised motor mount

The new mount was also better optimized for 3D printing than the initial design as it could be printed without support material on the bed of the 3D printer. During one of the powered flight tests, the original nose cone broke and had to be reprinted. The new design was stronger than the original and more securely attached to the front spine tube.

5.2.5 Revised Packing and Assembly

The final packing lay out correlated well to the SolidWorks models with the four wing sections stacked on one side of the box and the two payload sections on the other with the flat tail pieces laid on top. The packed aircraft can be seen below in Figure 77.

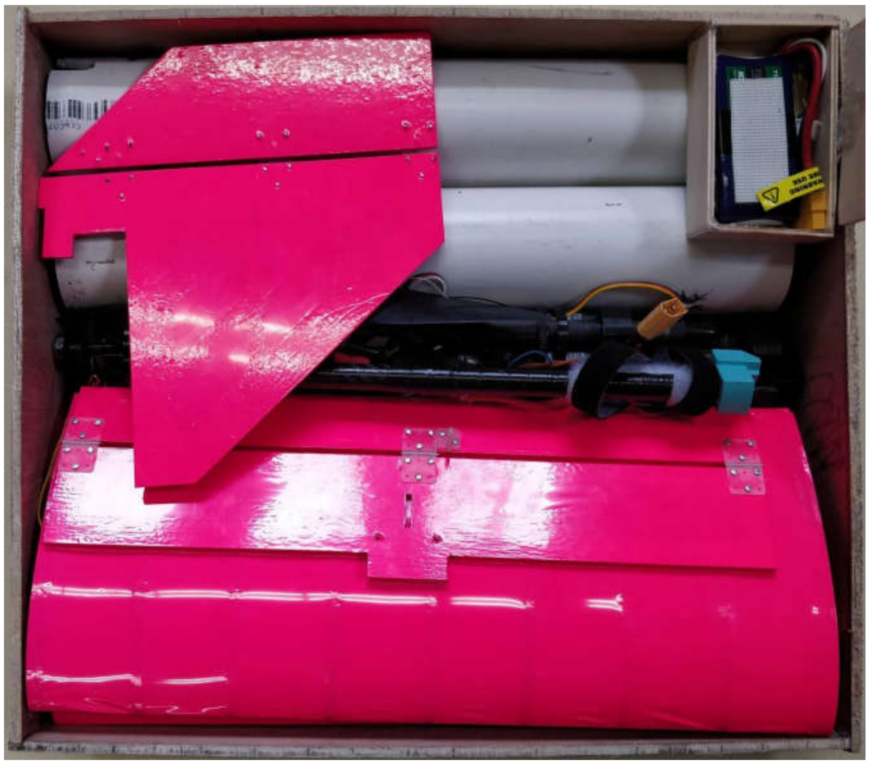

Figure 77. Final Packing Layout of Aircraft with Payload

In practicing the assembly of the aircraft, the team made some refinements to the planned assembly process. The first refinement was to permanently connect the electronics between the three body sections. The fuselage could still be broken into three sections and folded to fit in the box, but it saved time and reduced possible errors by leaving servo connections intact. The team also found that it was easier to slide the PVC pipes on from the front of the aircraft after the fuselage and tail were assembled instead of sliding them on from the back and assembling the tail afterward. The team also divided the assembly process into two sections so the two team members could work in parallel on different tasks. The wings and the body took approximately equal time to assemble so they were assembled in parallel. The two team members then worked together to complete the final step of adding the wings to the fuselage. The entire assembly process took 2 minutes and 30 seconds to complete, which satisfied the goal of assembling the aircraft in less than 3 minutes.

5.3 Powered Flight Testing of Revised Design

Upon completing many revisions to the aircraft based on feedback from the project advisors and test pilot after failed test flights, the team began testing the revised design. Table 23 provides an overview of the flight tests that the team completed after all revisions were made.

Table 23. Powered Flight Tests

Test No.	Date	Configuration	Results	Improvements
4	2/23/19	Unloaded	Flyable but not enough propeller pitch speed and slightly nose heavy	New Propeller, Reduced Nose Weight
5	3/20/19	Unloaded	Successful	No Notable Changes
6	3/20/19	Loaded	Power on stall	Loaded CG, Throw Technique
7	4/2/19	Unloaded	Stability augmentation successful but aircraft difficult to control in high winds	Realignment of Flaps

5.3.1 Unloaded Flight Test

After adjustments to the tail sizing and center of gravity locations and the implementation of the new propulsion system, the aircraft flew successfully in the unloaded configuration. During the flight tests the aircraft exhibited no issues with regard to the hand launch. It was able to climb, cruise, and perform 180 degree turns, which were all of the maneuvers required for the competition course. In addition, the pilot gradually reduced the throttle to test the lifting capabilities of the aircraft. He reduced the throttle to 60% before the aircraft began to lose lifting capability. While flying the aircraft, the pilot noticed that the aircraft was slightly nose heavy and that the pitch speed of the propeller was too slow. He had to fly the aircraft faster than predicted to compensate for the pitch speed and to trim the plane in a nose up configuration to compensate for the slightly tail heavy design. Due to this, the team began looking into weight saving options to reduce the nose weight of the aircraft. To reduce nose weight the team eliminated the need for both the ESC and battery mounting brackets. Instead these two components were re-mounted directly to the carbon fiber rod using velcro straps and electrical tape. To increase the pitch speed the team switched to the 9x6 propeller. Despite these minor issues presented, the major instability of the aircraft was corrected by the modified design. An image from the first successful flight test appears in Figure 78 below.

Figure 78. Image from Successful Flight Test

With the new propeller and no ESC or battery mounts, the team once again tested the airplane in the unloaded configuration. During the fifth flight test, the pilot had much more authority over the aircraft in terms of power due to the increased propeller pitch speed. With this, the pilot was able to maintain control at 50% throttle. During this test, the pilot also tested the flaps of the aircraft. He gradually added in flaps while flying level. This did not cause any noticeable pitching moments on the aircraft and he was able to maintain full control with them down; however, the aircraft did exhibit a roll to the left when the flaps were deployed.

5.3.2 Loaded Flight Test

The next stage of flight tests included adding the PVC pipe payload. Once the payload was added, the aircraft rapidly pitched up and entered into a power on stall resulting in a spin, ultimately nose diving into the ground. This caused the wing to snap and the tail control surfaces to break off. After looking at the video of the flight and the pilot's analysis, it was determined that the most likely cause of the accident was a combination of a tail heavy moments and a bad throw. Having a tail heavy aircraft explained the sharp pitch the aircraft experienced during the flight testing and reduced the responsiveness of the aircraft. In addition, due to the aircraft's design with payload, there was very little grip on the aircraft to throw it. This forced the hand launcher to propel the aircraft forward by pushing on the trailing edge of the wing. After examination of the aircraft, there were cracks in the wing at this location. This could have contributed to the initiation of the upward pitch and the roll that caused the crash. An image of the aircraft prior to hand launch appears in Figure 79 below.

Figure 79. Image from Payload Testing

After the failed payload test a small hole was cut in the PVC in order to allow for the hand launcher to better grip the aircraft for the throw. The team also further analyzed the aircraft CG location in the loaded configuration. The team decided to reduce the amount of payload weight near the tail of the aircraft by cutting slits into the PVC such that the unloaded and loaded CG locations were the same at 27.6% of the chord. The CG locations were confirmed by balancing the aircraft on both wings with two stabilization points located at the expected CG locations.

5.3.3 Stability Test

The final flight test of the aircraft involved a test of the stability augmentation system included in the receiver. The stability features of the aircraft were tested on the ground. To start, the team first set the stability to "level". Then the aircraft was rotated to a random orientation. When this happened, the control surfaces deflected according to the orientation of the aircraft such that it theoretically returned to level. The degree of deflection depended on how far off from level the aircraft was. Next the team tested the "stabilize" feature. For this, the team sharply turned the aircraft to simulate wind gusts or turbulence. While doing this the control surfaces deflected to resist the motion. The magnitude of the deflection varied depending on the input gain. Lastly, the team tested the fail-safe switch. To do this, the team oriented the plane in several orientations and flipped the fail-safe. Once flipped, the control surfaces rapidly deflected in order to return the aircraft to a straight and level flight configuration.

Once ground testing was complete, the team tested the stability augmentation inflight. During this test, there were winds in excess of 17 mph with periods of high gusting, which made the aircraft difficult to maneuver. Despite this, the team elected to test the stability augmentation. For the stability mode, there was not any noticeable difference in how the aircraft responded. However, the aircraft appeared to maintain its level flight path during “level” mode with minor deviations due to wind gusts. The most notable mode was the emergency level switch which successfully corrected the aircraft’s attitude to a straight and level orientation during several tests.

5.4 Final Design Specifications

The final specifications of the aircraft after completing all flight testing appears below in Table 24.

Table 24. Final Design Specifications

Characteristic	**Value**
Empty Weight	2 lbs
Max Weight	3.85 lbs
Length	30.23 in
Height	7.88 in
Cruise Speed	30 mph
Max Payload Fraction	0.48
Wing	
Airfoil	NACA 9410
Wing Span	4.33 ft
Aspect Ratio	9.45
Wing Area	1.986 sq.ft
Chord	0.4583 ft
Taper Ratio	1
Tail	
Incidence Angle	-4 degrees
VT Volume Ratio	0.03
VT Area	0.2381 sq.ft
HT Area	0.3583 sq.ft
Aspect Ratio	3.87
Propulsion	
Motor	Cobra 2814-12
Propeller	APC-E 9x6
Max Thrust	55 ozf
Battery Life Max Power	1.51 min
Battery Life Cruise Power	4.26 min

5.5 Mock Competition Results

Upon completing flight testing and subsequent rebuilds, the team completed a mock competition with the aircraft. The mock competition involved an assembly demonstration, unloaded aircraft flight, and loaded aircraft flight. The scoring scheme for the competition appears in section 1.3. The team successfully completed the assembly demonstration in 2 minutes and 43 seconds which corresponds to an assembly demonstration score of -0.3617. While the team successfully completed the assembly demonstration in the allotted time, the demonstration still received a negative demonstration score. A failed assembly demonstration corresponded to a score penalty of -5 where an assembly demonstration of 2 minutes corresponded to a score of 0.

Because the team did not go to the official competition, the rules were unclear on the scoring scheme for the unloaded flight. Therefore, the team was unable to calculate a flight score for the successful unloaded flight. For the payload flight, the team had not fixed the instability in the roll mode of the aircraft, therefore the aircraft did not complete the course with the payload attached. If the aircraft had completed the course with the payload attached, the team would have received a final flight score of 26.1548. In comparison to the results from the 2018 competition shown in Appendix J, the team believed that the flight score would have allowed the team to rank well at the competition provided the presentation and design scores were high.

6. Summary, Conclusions, Recommendations, and Broader Impacts

6.1 Summary

The team successfully researched, designed, manufactured, and flew a fixed wing micro aerial vehicle with the ability to meet the requirements of the SAE Aero Design West Competition rules. This was achieved by the collaboration between the aerodynamics, propulsion, stability and controls, and structures subgroups to construct the unique aircraft.

All subgroups incorporated creative ideas and knowledge from course work and literature review to create an initial design of the aircraft. The design was focused around the goals that the team hoped to achieve in regards to the competition. From there, the team continued with testing and comparison to theory. Many changes were made to the aircraft specifications and components, however the focus of the design remained the same.

After multiple design iterations, the team was able to construct an aircraft that met competition requirements and was capable of unloaded flight. Although the team was unable to go to the actual competition, the capabilities of the aircraft met expectations and would have been highly competitive.

6.2 Conclusions

The rules supplied by the 2019 SAE Aero West Design Competition put major restrictions on the design of the aircraft. These restrictions posed challenges that all subgroups had to work around. One of the more challenging rules of the competition was fitting the unassembled aircraft into a 12.125” x 3.625”x 13.875” box. This put many limitations on the size of the different components of the aircraft as well as how much payload could be used. It was necessary for all separated parts of the aircraft to be no longer than the longest dimensions of the box. Additionally, the size of the wings had to be reduced as they already took up a large portion of the box. With a smaller wing area, we had to find alternative methods to producing enough lift. The team eventually decided on the addition of flaps to increase the coefficient of lift to what it needed to be.

The competition required the aircraft to be able to fly with and without payload. The plane also had to be fully assembled within three minutes. The team made the body of the aircraft as simple as possible, using carbon fiber rods and 3D printed parts that can easily be put together. This also allowed for easily attaching 2 inch diameter PVC pipe to the plane, meeting the requirement for the plane to be able to carry the specified payload for flight. When unloaded, the plane is easily hand launched because the structure is easy to grab. Alternatively, the loaded aircraft was difficult to throw properly because the PVC pipes did not provide a solid means of gripping the aircraft. A hole was drilled into the PVC pipe to easily throw the aircraft, and meeting the hand launch competition requirement both unloaded and loaded. Once the design of the aircraft was ready, the team was able to begin flight testing. Many issues arose, including the location of the CG as well as an incorrect tail size for stability. Once these were resolved, the aircraft flew unloaded. However, the team was not able to fly the aircraft loaded due to more instability, and did not have time to find the root cause of the issue.

The team was able to meet most of the competition rules, with the exception of flying loaded. The competition rules introduced many challenges, however these challenges were resolved through trial and error and creative work.

6.3 Recommendations for Future Work

Over the course of the design, manufacturing, and flying of the aircraft the team encountered multiple difficulties. The objective of this section is to provide recommendations to future teams for technical and logistical improvements. From research and testing the team found the following areas needed improvements.

The first notable deviation in the final aircraft design from the initial planned design was its empty weight. Initially, the aircraft was set to weigh approximately 1.2 lbs empty. The actual empty weight was 2 lbs which was 67% greater than expected. This meant that the final flight score was much lower because the aircraft was designed to carry more payload than it did. The team advises future teams to focus on reducing the empty weight of the aircraft more than increasing the amount of payload carried. Methods for reducing the weight of the aircraft may include the choice of electronics and the choice of materials for construction, as we found that the electronics onboard weighed more than was initially expected. Another option that could be considered is focusing more on weight fraction of the aircraft versus actual weight. This could allow for a much smaller aircraft, which would more easily fit into the box of the specified dimensions. Focusing more on weight fraction would still allow for a large amount of payload in comparison to the empty weight.

Another structural difficulty was the aircraft assembly time due to the placement and orientation of the PVC payload over the fuselage. The use of the PVC pipe as the fuselage provided a way to keep the aircraft rigid, but it prevented easy assembly for the timed assembly demonstration. By assembling the aircraft without the PVC fuselage but rather a slung or mounted PVC payload, the assembly time would be reduced significantly. The PVC made it difficult to align the wing spars through the connectors inside the fuselage for the attachment of the wings. Mounting or slinging the PVC also could have allowed for a lighter empty aircraft because the fuselage would no longer be restricted to a minimum length of two full sections of PVC. This also could have allowed the aircraft to fit more compactly in the assembly box.

For aerodynamics, the team recommends picking a less cambered airfoil with a lighter overall aircraft. A less cambered airfoil would allow for easier manufacturing of the wing sections as the mylar wrapping would adhere better to the bottom of the airfoil. Using a less cambered airfoil also allows the wing sections to fit more snuggly into the box itself. One possibility would be the use of flat plate airfoils. For the scale of the micro-class aircraft, wing area is more important than the airfoil shape and camber, thus a flat plate mounted at a set angle would provide sufficient lift. The additional wing area would fit in the box if the wing was designed as a flat plate. The aerodynamic analysis of the aircraft would then determine the necessary angle of attack for the airfoil to achieve necessary lift.

For the stability of the aircraft, the team recommends that the lateral stability modes be tested in the wind tunnel. It was difficult to analytically determine the stability derivatives and the subsequent lateral stability matrix; therefore, wind tunnel testing would be important in verifying the XFLR5 stability outputs. Another recommendation is to ensure that the vertical tail is properly sized based on the volume ratio between the vertical tail and the wing. The team found that a major issue for the first few flight tests was that the vertical tail was not the correct size for maintaining lateral stability of the aircraft.

For propulsion, the team recommends that future teams purchase spare components needed for the aircraft from the beginning as it is very easy to break components during testing and components are sometimes shipped broken. Waiting on electronics forced the team to rush flight tests toward the end of B term and beginning of C term. In addition, the team recommends that future teams consider propulsion options that can produce more thrust than what is required as during flight curtain maneuvers will require high thrust to avoid crash beyond what is specified for normal flight. Lastly, the team recommends that students greatly consider the pitch speed of the propeller, this is something not considered until after testing when the motor was not as responsive despite having the required thrust for flight.

For initial testing, the team recommends that future teams complete additional glide tests during B term. This would allow for a better understanding of the aircraft's dynamics and stability before actual flight. It would be beneficial for one of the glide tests to include live control surfaces as well to determine if the control surface sizes provide satisfactory control of the aircraft moments. The use of live control surfaces would also allow for the team to notice trends in the aircraft's flight as it would be able to have a larger glide distance with pitch and roll control.

6.4 Broader Project Impacts

The main goal of this project was to design an aircraft that could carry as much payload as possible while keeping the empty weight of the aircraft as low as possible. Having an aircraft with high payload to weight fractions is important in many civilian and military applications. For instance, both military and civilian organizations use unmanned aircraft for surveillance. One specific example is the use of unmanned aerial vehicles for traffic surveillance in order to get information on traffic patterns and road conditions as well as provide information in event of an emergency (Puri, A., n.d.). Having a high payload fraction would allow for more and heavier equipment to be used onboard the vehicle. This would result in a larger range of data collection, better imagery, and better communication with a central hub. The same concept applies to military surveillance, search and rescue teams, and disaster relief. In addition, the high payload fraction provides some economic benefits for organizations that use micro aerial vehicles for load carrying capabilities. The high payload fraction allows for a larger range on the aircraft and could result in the need for fewer aircraft on a mission, providing large economical savings in power consumption and time of flight.

The second goal of this project was to have an aircraft that could fit inside of a small box and reassemble as quickly as possible. This aspect of this project aids in the convenience and transportability of aircraft. For example, the fast assembly could aid a soldier needing to transport the aircraft across enemy lines, quickly assemble it, and fly it, or an organization needing to transport the aircraft along with other equipment for use at a disaster relief location. Having a fully functional aircraft that could fit into a small volume and be quickly assembled allows for easy storage during transport and keeps the aircraft protected while still being ready to fly within a short period of time.

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Appendix A: 2019 SAE Aero Design Rules

2019 Collegiate Design Series
SAE Aero Design Rules

Version 2019.4

Forward

Welcome to SAE Aero design 2019! Our goal each year is to create and refine a set of competition events with relevant real-world design challenges that students of all engineering disciplines and levels of experience can enjoy. You and your team members will have to apply your engineering knowledge in a design team environment to create an original aircraft design. After the design phase, you and your team must build, test and refine your design and prepare to compete head to head with other teams. Teams validate their efforts by creating a design report, an oral presentation and by demonstrating the flight performance of their aircraft in competition.

SAE Aero Design consists of three competition classes: Regular class, Advanced class and Micro class. Regular class traces its lineage back to the very beginning of Aero Design, when it was created as a heavy lift competition for model aircraft. Regular class continues with the 2017 competition theme of "passengers" and "luggage" as the payload, with some minor rules changes.

For 2019, we have an exciting new theme for Advanced class that will challenge the teams and make Advanced class the most exciting competition class to observe that we have ever created. The new theme involves a subscale demonstration of delivering "colonists" to a designated landing zone using a small autonomous glider carried by the primary aircraft and released in a designated area. The primary aircraft must also deliver "habitats" and "supplies" to the same location by dropping them from the primary aircraft. An additional new challenge for Advanced class: Advanced class is now electric powered, and the aircraft must use a new 750-watt power limiter created specifically for this class. The new Advanced class mission requires teams to create and integrate a set of reliable systems meeting baseline requirements for the class, including the primary aircraft, the autonomous glider, a payload delivery system that accommodates different payloads, and a ground station. A successful Advanced class team will have all elements of the aircraft systems and all team members working together in concert to accomplish this challenging mission.

Micro class continues with the same mission as created for 2018, with some minor rules changes. The low-density industrial material required as payload, aircraft design for the required payload, the aircraft container requirements and mandatory timed assembly demo combine to make Micro class a demanding competition for teams to complete successfully.

The rules committee wants you to know that we closely review the feedback that teams submit on the survey after each Aero Design event and that we do make use of the survey information when we consider and implement new rules and rule changes. Please do read the 2019 rules closely, please read the FAQS for 2019 that we will post in the question and answer forum and please make use of the Aero Design question and answer forum to resolve questions about the rules.

The Aero Design rules committee is excited about the 2019 events and their challenges and we hope that the teams will enjoy this tremendous opportunity to apply their education to real world engineering tasks. Based on history, we expect both 2019 events to sell out quickly, so be sure to *apply as soon as possible. We were able to expand one of the Aero Design venues so that we can accept 85 teams at that location, giving 10 additional teams a chance to compete. If you are waitlisted, please do continue with your design effort, as we were able to accommodate nearly all waitlisted teams over the last few years as other teams dropped out.*

Everyone at SAE Aero Design wishes the best of luck to all teams participating in Aero Design 2019!

Tom Blakeney, SAE Aero Design Rules Committee

Table of Contents

1 COMPETITION REQUIREMENTS

1.1 INTRODUCTION

Official Announcements and Competition Information

The SAE Aero Design competition is intended to provide undergraduate and graduate engineering students with a real-world design challenge. These rules were developed and designed by industry professionals with the focus on educational value and hands-on experience through exposure to today's technical and technology advancement. These rules were designed to compress a typical aircraft development program into one calendar year, taking participants through the system engineering process of breaking down requirements. It will expose participants to the nuances of conceptual design, manufacturing, system integration/test, and sell-off through demonstration.

SAE Aero Design features three classes of competition—Regular, Advanced, and Micro.

- **The Regular Class** is an all-electric class intended to develop a fundamental understanding of aircraft design.
- **The Advanced Class** is an all-electric class designed to inspire future engineers to take a systems approach to problem solving, at the same time, exposing them to explore the possibilities of autonomous flights.
- **The Micro Class** is an all-electric class designed to help students engage in trades between two potentially conflicting requirements, carrying the highest payload fraction possible, while simultaneously pursuing the lowest empty weight possible.

Other SAE Aero Design Competitions:

SAE BRASIL http://www.saebrasil.org.br

1.2 SAE AERO DESIGN RULES AND ORGANIZER AUTHORITY

General Authority

SAE International and the competition organizing bodies reserve the rights to revise the schedule of any competition and/or interpret or modify the competition rules at any time and in any manner, that is, in their sole judgment, required for the efficient and safe operation of the event or the SAE Aero Design series as a whole.

1. *Penalties*
   SAE International and the competition organizing bodies reserve rights to modify the points and/or penalties listed in the various event descriptions; to accurately reflect the operations execution of the events, or any special conditions unique to the site.
2. *Rules Authority*
   The SAE Aero Design Rules are the responsibility of the SAE Aero Design Rules Committee and are issued under the authority of the SAE International University Programs Committee. Official announcements from the SAE Aero Design Rules Committee, SAE International or the other SAE International Organizers shall be

considered part of and have the same validity as these rules. Ambiguities or questions concerning the meaning or intent of these rules will be resolved by the officials, SAE International Rules Committee or SAE International Staff.

3. *Rules Validity*

The SAE Aero Design Rules posted at www.saeaerodesign.com/go/downloads and dated for the calendar year of the competition are the rules in effect for the competition. Rule sets dated for other years are invalid.

4. *Rules Compliance*

By entering an SAE Aero Design competition, the team members, faculty advisors and other personnel of the entering university agree to comply with, and be bound by, the rules and all rules interpretations or procedures issued or announced by SAE International, the SAE Aero Design Rules Committee and other organizing bodies. All team members, faculty advisors and other university representatives are required to cooperate with, and follow all instructions from competition organizers, officials and judges.

5. *Understanding the Rules*

Teams are responsible for reading and understanding the rules in their entirety for the competition in which they are participating. The section and paragraph headings in these rules are provided to facilitate reading: they do not affect the paragraph contents.

6. *Loopholes*

It is virtually impossible for a set of rules to be so comprehensive that it covers all possible questions about the aircraft's design parameters or the conduct of the competition. Please keep in mind that safety remains paramount during any SAE International competition, so any perceived loopholes should be resolved in the direction of increased safety/concept of the competition.

7. *Participating in the Competition*

Teams, team members as individuals, faculty advisors and other representatives of a registered university who are present on-site at a competition are considered to be "participating in the competition" from the time they arrive at the event site until they depart the site at the conclusion of the competition or earlier by withdrawing.

8. *Visa--United States Visas*

Teams requiring visas to enter to the United States are advised to apply at least sixty (60) days prior to the competition. Although most visa applications seem to go through without an unreasonable delay, occasionally teams have had difficulties and in several instances visas were not issued before the competition.

AFFILIATED CDS STUDENT TEAM MEMBERS WILL HAVE THE ABILITY TO PRINT OUT A REGISTRATION CONFIRMATION LETTER FOR THE INDIVIDUAL EVENT(S) THAT THEY ARE ATTENDING. ONCE A STUDENT TEAM MEMBER AFFILIATES THEMSELVES TO THEIR TEAM PROFILE PAGE UNDER THEIR INDIVIDUAL EDIT SECTION, THEY WILL HAVE THE OPPORTUNITY TO PRINT OUT THEIR PERSONALIZED LETTER WITH THE FOLLOWING INFORMATION: STUDENT'S NAME, SCHOOL'S NAME, THE CDS EVENT NAME, OFFICIAL DATES AND LOCATION(S).

9. *Letters of Invitation*

Neither SAE International staff nor any competition organizers are permitted to give advice on visas, customs regulations or vehicle shipping regulations concerning the United States or any other country.

10. *Certificates of Participation*

SAE International and competition organizers do not create any Participation Certificates outside of the auto-generated certificate on your team profile page at sae.org.

Certificates are available as soon as students are affiliated to the current competition's team. Certificates will not be available once that competition year closes.

11. *Violations of Intent*

The violation of the intent of a rule will be considered a violation of the rule itself. Questions about the intent or meaning of a rule may be addressed to the SAE International Officials, Competition Organizers or SAE International Staff.

12. *Right to Impound*

SAE International and the other competition organizing bodies reserve the right to impound any on-site vehicle/aircraft at any time during a competition for inspection and examination by the organizers, officials and technical inspectors.

1.3 SOCIETY MEMBERSHIP AND ELIGIBILITY

1. *Society Membership*

Individual team members must be members of one of the following societies: (1) SAE International or an SAE International affiliate society, (2) ATA, or (3) IMechE or (4) VDI. Proof of membership, such as a membership card, is required at the event. Students who are members of one of the societies listed above are not required to join any of the other societies in order to participate in any SAE competition. Students may join online at: https://www.sae.org/participate/membership/join

Teams are also required to read the articles posted on the SAE Aero Design News Feed (www.saeaerodesign.com/go/news) published by SAE International and the other organizing bodies. Teams must also be familiar with all official announcements concerning the competitions and rule interpretations released by the SAE Aero Design Rules Committee.

2. *Team Pilots*

Team pilots are not required to be students or SAE International members, but all pilots must be current members of the Academy of Model Aeronautics or the Model Aircraft Association of Canada (AMA has an agreement with MAAC). Valid AMA membership cards must be presented at the flying field prior to flying any team's aircraft. Non-US pilots can obtain a discounted AMA Affiliate membership that covers flying activities while in the US by going to the AMA web site and submitting the following form: https://www.modelaircraft.org/files/902.pdf.

1.4 Liability Waiver and Insurance Requirements

All on-site participants and faculty advisors are required to sign a liability waiver. Individual medical and accident insurance coverage is the sole responsibility of the participant.

1.5 Ringers Prohibited

In order to maintain the integrity of a fair competition, the faculty advisor must prohibit ringers. A ringer is someone that has exceptional skills related to the competition (e.g., a professional model builder) that cannot be a legal member of the team but helps the team win points.

1.6 Design and Fabrication

The aircraft must be designed and built by the SAE International student members without direct involvement from professional engineers, radio control model experts, pilots, machinists, or related professionals. The students may use any literature or knowledge related to R/C aircraft design and construction and information from professionals or from professors as long as the information is given as discussion of alternatives with their pros and cons and is acknowledged in the references in the design report. Professionals may not make design decisions, nor contribute to the drawings, the report, or the construction of the aircraft. The faculty advisor must sign the Statement of Compliance given in the Appendix.

1.7 Original Design

Any aircraft presented for competition must be an original design whose configuration is conceived by the student team members. Photographic scaling of an existing model aircraft design is not allowed. Use of major components such as wings, fuselage, or empennage of existing model aircraft kits is prohibited. Use of standard model aircraft hardware such as motor mounts, control horns, and landing gear is allowed.

1.8 Official Languages

The official language of the SAE Aero Design series is English. Document submissions, presentations and discussions in English are acceptable at all competitions in the series.

Team members, judges and officials at Non-U.S. competition events may use their respective national languages for document submissions, presentations and discussions if all the parties involved agree to the use of that language.

1.9 Unique Designs

Universities may enter more than one team in each SAE Aero Design competition, but each entry must be a unique design, significantly different from each other. If the aircraft are not significantly different in the opinion of the rules committee and organizer, then the university will be considered to have only a single entry and only one of the teams and its aircraft will be allowed to participate in the competition. For example, two aircraft with identical wings and fuselages but different empennage would likely not be considered significantly different. For guidance regarding this topic, please submit a rules question at www.saeaerodesign.com.

1.10 AIRCRAFT CLASSIFICATION/DUPLICATE AIRCRAFT

1. *One Team Entry per Class*
   A university is limited to registering one team per class.

2. *Backup Aircraft*
   When a team has an identical aircraft as a back-up, the back-up aircraft must go through inspection with the primary aircraft.

1.11 AIRCRAFT ELIGIBILITY

Aircraft will only be allowed to compete during a ***single academic year***. Aircraft may be entered in both SAE Aero Design East and SAE Aero Design West during the same ***calendar year***, but that same aircraft may not be used in either competition during the following year. Entering the same aircraft in SAE Aero Design West one year and SAE Aero Design East the next year is not allowed.

1.12 REGISTRATION INFORMATION, DEADLINES AND WAITLIST (NEW)

Teams intending to participate in the 2018 SAE Aero Design competitions must register their teams online per the open registration schedule.

Table 1.1 Open Registration Schedule

Event	***Team Limit***	***Start (Open)***	***End (Closed)***
SAE Aero Design East	85 Teams	October 1st, 2018 10:00 am ET	November 12th, 2018 11:59 PM
SAE Aero Design West	75 Teams	October 1st, 2018 10:00 am ET	November 12th, 2018 11:59 PM

The registration fee is non-refundable and failure to meet these deadlines will be considered a failure to qualify for the competition. Separate entry fees are required for the East and West events.

1. *Team/Class/University Policy*
   A university or college can only have one aircraft registered for one class. Auniversity cannot register more than one team per class. The registration fees indicated on the website ($1050) must be paid within 48 hours of registration.

2. *Individual Registration Requirements – ACTION REQUIRED*
   All participating team members and faculty advisors must be sure that they are individually affiliated to their respective school / university on the SAE International website (www.sae.org) Team Profile page.

   If you are not an SAE International member, go to www.sae.org and select the "Membership" link. Students will need to select the "Student Membership" link and then follow the series of questions that are asked Please note all student participants must be SAE International members to participate in the events.

   Faculty members who wish to become SAE International members should choose the "Professional Membership" link. Please note: this is not mandatory for faculty advisors.

All student participants and faculty advisors must affiliate themselves to the appropriate team(s) online.

The "Add New Member" button will allow individuals to access this page and include the necessary credentials. If the individual is already affiliated to the team, simply select the Edit button next to the name. Please be sure this is done separately for each of the events your team has entered.

All students, both domestic and international, must affiliate themselves online prior to the competition.

***NOTE: When your team is registering for a competition, only the student or faculty advisor completing the registration needs to be linked to the school. All other students and faculty can affiliate themselves after registration has been completed; however this must be done prior to two weeks before the competition start date.*

1.13 WAITLIST

Once an event reaches the venue's capacity, all remaining registered team will be asked to be placed on a waitlist. The waitlist is capped at 40 available spaces per event and will close on the same day as registration closes. Once a team withdraws from an event, an SAE International Staff member will inform your team by email (the individual who registered the team to the waitlist) that a spot on the registered teams list has opened. You will have 24 hours to accept or reject the position and an additional 24 hours to have the registration payment completed or process for payment begun. Waitlisted teams are required to submit all documents by the deadlines in order to be considered serious participants and any team that does not submit all documents will be removed from the waitlist.

1.14 POLICY DEADLINE

1. *Failure to meet deadlines*

Teams registering for SAE Aero Design competitions are required to submit a number of documents prior to the competition including a Design Report and Technical Data Sheet that the event judges use to evaluate the team during the competition. When these documents are not submitted our judges cannot properly assess the team. Additionally, teams that do not submit required documents typically do not come to the competition. Teams that do not notify us that they are withdrawing create the following problems:

1. They are included in the static event schedules and judging time is wasted.
2. Their unused registration slot cannot be offered to a team on the waitlist. Additionally, failure to submit the required documents is a clear violation of the rules.

2. *Late Submission Penalty*

Late submission or failure to submit the Design Report will be penalized five (5) points per day. If your required documents are received more than five (5) days late it will be classified as "Not Submitted" and your team will not participate and the automatic withdrawal policy will be in effect.

3. *Automatic Withdrawal Policy*
Failure to submit the required Design Report, Technical Data Sheet, and Drawings within 5 days of the deadline will constitute an automatic withdrawal of your team. Your team will be notified before or on the 4th day of no submission that we have not received your documents and after the 5 days your team's registration will be cancelled and no refund will be given.

1.15 Faculty Advisor

Each team is expected to have a Faculty Advisor appointed by the university. The Faculty Advisor is expected to accompany the team to the competition and will be considered by competition officials to be the official university representative. Faculty Advisors may advise their teams on general engineering and engineering project management theory, but may not design any part of the vehicle nor directly participate in the development of any documentation or presentation. Additionally Faculty Advisors may neither fabricate nor assemble any components nor assist in the preparation, maintenance, or testing of the vehicle. In Brief - Faculty Advisors may not design, build or repair any part of the aircraft. Faculty Advisors that are not eligible student team members may not participate in flight operations during competition weekend except as noted.

1.16 Questions, Complaints and Appeals

1. *Questions*
Any questions or comments about the rules should be brought to the attention of the Rules Committee by submitting a rules question at https://www.saeaerodesign.com.

   General information about hotels and other attractions in the area as well as a schedule of events will be posted on the SAE International website according to the competition in which you are competing: https://www.sae.org/attend/student-events/

2. *Complaints*
Competition officials will be available to listen to complaints regarding errors in scoring, interpretation, or application of the rules during the competition. Competition officials will not be available to listen to complaints regarding the nature, validity, or efficacy of the rules themselves at the competition. In other words, the Organizer will not change the rulebook at the field.

3. *Appeal / Preliminary Review*
A team can only appeal issues related to own-team scoring, judging, venue policies, and/or any official actions. Team Captain(s) and/or faculty advisor must bring the issue to the Organizer's or SAE International staff's attention for an informal preliminary review before filing an official appeal.

   A team cannot file an appeal to cause harm to another team's standing and/or score.

4. *Cause for Appeal*
A team may appeal any rule interpretation, own-team scoring or official actions which the team feel has caused some actual, non-trivial, harm to own-team, or has had a substantive effect on their score.

Teams may not appeal rule interpretations or actions that have not caused them any substantive damage.

5. *Appeal Format*
If a faculty advisor or team captain(s) feel that their issue regarding an official action or rules interpretation was not properly addressed by the event officials, the team may file a formal appeal to the action or rules interpretation with the Appeals Committee.

All appeals must be filed in writing (see Appendix E) to the Organizer by the faculty advisor or team captain only.

All appeals will require the team to post twenty five (25) points as collateral. If the appeal is successful and the action is reversed, the team *will not* forfeit the twenty five (25) collateral points. If the appeal is overruled, the team will forfeit the twenty five (25) collateral points.

All rulings issued by the Appeals Committee are final.

6. *Appeals Period*
All appeals must be submitted within thirty (30) minutes of the end of the flight round or other competition event to which the appeal relates.

7. *Appeals Committee*
When a timely appeal is received, the committee will review in detail the claims. All contentions or issues raised in the formal appeal will be addressed in a timely manner. The consideration in each review is whether the actions in dispute were just and in-line with the intent of the rules. Once the review is completed, a new order will be issued affirming, reversing or modifying the original determination.

All rulings issued by the Appeals Committee are final.

The Appeals Committee must consist of a minimum of three members: the Organizer or delegate, SAE International representative, and either the Chief Steward, the Chief Judge, the Air Boss and/or rule committee member.

1.17 PROFESSIONAL CONDUCT

1. *Unsportsmanlike Conduct*
In the event of unsportsmanlike conduct by team members or that team's faculty advisor, the team will receive a warning from a Competition Official. A second violation will result in expulsion of the team from the competition and loss of any points earned in all aspects of the competition.

2. *Arguments with Officials*
Arguments with or disobedience toward any competition official may result in the team being eliminated from the competition. All members of the team may be immediately escorted from the grounds.

3. *Alcohol and Illegal Material*
Alcoholic beverages, illegal drugs, firearms, weapons, or illegal material of any type are not permitted on the event sites at any time during the competition. Any violations of this rule will result in the immediate expulsion of all members of the

offending school, not just the individual team member in violation. This rule applies to team members and faculty advisors. Any use of illegal drugs or any use of alcohol by an underage person must be reported to the local law enforcement authorities for prosecution.

4. *Organizer's Authority*

The Organizer reserves the exclusive right to revise the schedule of the competition and/or to interpret the competition rules at any time and in any manner which is required for efficient operation or safety of the competition.

5. *Ground Safety and Flight Line Safety Equipment*

1. No open toe shoes allowed. All team participants, including faculty advisors and pilots, will be required to wear CLOSED toe shoes during flight testing and during flight competition.
2. Smoking is prohibited. Smoking is prohibited in all competition areas.
3. All students in all classes involved at the flight line must wear safety glasses.
4. Micro Class must wear hard hats in addition to safety glasses at the flight line.

1.18 SAE TECHNICAL STANDARDS ACCESS

A cooperative program of SAE International's Education Board and Technical Standards Board is making some of SAE International's Technical Standards available to teams registered for any North American CDS competition at no cost. The Technical Standards referenced in the Collegiate Design Series rules, along with other standards with reference value, will be accessible online to registered teams, team members and faculty advisors.

To access, teams can follow these procedures. Once registered, a link to SAE MOBILUS will appear to access the technical standards under "Design Standards" on your team's profile page on sae.org, where all the required onsite team information is added. On SAE MOBILUS, you will have the ability to search standards either by J-number assigned or topic of interest such as brake light.

A list of accessible SAE International Technical Standards can be found in Appendix F.

2 General Aircraft Requirements

2.1 Aircraft Identification

Team number as assigned by SAE International must be visible on both the top and bottom of the wing, and on both sides of the vertical stabilizer or other vertical surface.

1. Aircraft must be identified with the school name and address either on the outside or the inside of the aircraft.
2. Team numbers on Regular aircraft shall be a minimum of 3 inches in height.
3. Team numbers on the Advanced Class primary aircraft shall be a minimum of 3 inches in height. Team numbers on the Advanced Class Colonist Delivery Aircraft (CDA) shall be a minimum of 1 inch in height.
4. Team numbers on Micro Class shall be a minimum of 1 inch in height.
5. The University name must be clearly displayed on the wings or fuselage.
6. The University initials may be substituted in lieu of the University name provided the initials are unique and recognizable.

The assigned aircraft numbers appear next to the school name on the "Registered Teams" page of the SAE Aero Design section of the Collegiate Design Series website at:

SAE Aero East: https://www.sae.org/attend/student-events/sae-aero-design-east

SAE Aero West: https://www.sae.org/attend/student-events/sae-aero-design-west

2.2 Prohibited Aircraft Configuration

Competing designs are limited to fixed wing aircraft only. Lighter-than-air aircraft, rotary wing aircraft such as helicopters or auto-gyros and steerable parafoil aircraft are not allowed to compete.

2.3 Empty CG Design Requirement and Empty CG Markings on Aircraft

All aircraft must meet the following Center of Gravity (CG) related requirements:

1. All aircraft must be flyable at their designated Empty CG position (no payload, ready to fly) on the submitted 2D aircraft drawing.
2. All aircraft must have the fuselage clearly marked on both sides with a classic CG symbol (Figure 2.1) that is a minimum of 0.5 inches in diameter centered at the Empty CG position, per the submitted 2D drawings. (Wing type aircraft may place the two CG markings on the bottom of the wing.)
3. The Empty CG location will be verified during Technical and Safety Inspection.
4. No empty weight flight is required.

Figure 2.1 - Center of Gravity Symbol

2.4 Gross Weight Limit

Aircraft gross take-off weight may not exceed fifty-five (55) pounds.

2.5 Controllability

1. All aircraft must be controllable in flight.
2. If an aircraft is equipped with a wheeled landing gear, the aircraft must have some form of ground steering mechanism for positive directional control during takeoffs and landings. Aircraft may not rely solely on aerodynamic control surfaces for ground steering.

2.6 Radio Control System

The use of a 2.4 GHz radio control system is required for all aircraft. The 2.4 GHz radio control system must have a functional fail-safe system that will reduce the throttle to zero if the radio signal is lost.

2.7 Spinners or Safety Nuts Required

All powered aircraft must utilize either a spinner or a rounded model aircraft type safety nut.

2.8 Metal propellers

Metal propellers are not allowed.

2.9 Lead is prohibited

The use of lead in any portion of any aircraft (payload included) is strictly prohibited.

2.10 Payload Distribution

The payload cannot contribute to the structural integrity of the airframe.

2.11 Aircraft Ballast

Aircraft ballast is allowed with the following conditions:

1. Ballast cannot be used in the closed payload bay or passenger cabin.
2. Ballast stations must be clearly indicated on the 2D drawings.
3. Ballast must be secured so as to avoid shifting or falling off the aircraft and causing a CG problem.
4. Ballast will not be counted as payload.

2.12 Stored Energy Restriction

Aircraft must be powered by the motor on board the aircraft. No other internal and/or external forms of stored potential energy allowed.

2.13 Control Surface Slop

Aircraft control surfaces and linkage must not feature excessive slop. Sloppy control surfaces lead to reduced controllability in mild cases, or control surface flutter in severe cases.

2.14 Servo Sizing

Analysis and/or testing must be described in the Design Report that demonstrates the servos are adequately sized to handle the expected aerodynamic loads during flight.

2.15 Clevis Keepers

All control clevises must have additional mechanical keepers to prevent accidental opening of the control clevis in flight.

2.16 Red Arming Plug

All electric powered aircraft MUST use a discrete and removable red arming plug to arm and disarm the aircraft propulsion system. This red arming plug must be integrated into the electrical circuit between the battery and the electronic speed controller (ESC).

1. The red arming plug must physically be located at 40% to 60% of the aircraft length from the aircraft propeller. This is to allow arming and disarming the aircraft at a safe distance from the propeller.
2. The red arming plug must be located on top of the fuselage or wing and external of the aircraft surface.
3. The location of the red arming plug must be clearly visible.
4. The non-removable portion of the arming plug interface may not have more than one male lead.
5. Disconnecting wiring harnesses to arm and disarm a system will NOT be allowed.

2.17 Repairs, Alterations, and Spares

1. The original design of the aircraft as presented in the written and oral reports must be maintained as the baseline aircraft during the competition.
2. In the event of damage to the aircraft, the aircraft may be repaired provided such repairs do not drastically deviate from the original baseline design. All major repairs must undergo safety inspection before the aircraft is cleared for flight.

2.18 Battery Protection

1. All batteries in the aircraft must be positively secured so that they cannot move under normal flight loads.
2. The battery bay or location in the aircraft must be free of any hardware or other protrusions that could penetrate the battery in the event of a crash.

2.19 Alteration after First Flight

Minor alterations are allowed after the first and subsequent flight attempts.

1. A penalty will be assessed ONLY if 2/3 of the ruling committee (Event Director, Head scoring judge and/or SAE staff judge) agree that there were significant modifications made from the baseline configuration.
2. If the ruling committee determines that the changes are a result of safety-of-flight, the changes will not incur penalty points. Alteration must be reported utilizing Engineering Change Request (ECR) Appendix D.

3 MISSION REQUIREMENTS AND SCORING

3.1 AIR BOSS

The Air Boss is a qualified SAE event official or appointed volunteer that manages the flight line process. Their responsibilities include:

- Ensure the safety of the flight line through maintaining an orderly and controlled runway.
- Be the official of record for the success or failure or the aircraft's flight, including takeoff and landing.
- Declare termination of flight at any time during the attempt.

3.2 PILOT STATION(S)

Pilot area will be defined at pre-competition meeting (Friday Night All-hands). All pilots must fly from designated area.

3.3 ROUND ATTEMPT

Teams are allowed one (1) flight attempt per round.

- **Regular and Advanced Classes:** Without violating other take-off restrictions, a team can have multiple attempts to become airborne within the team's prescribed time limit for each respective class identified in section 3.5.
- **Micro Class:** only one launch attempt is allowed per round.

3.4 MOTOR RUN-UP BEFORE TAKEOFF

Aircraft may be throttled up/run up for takeoff, subject to the following conditions:

- **Advanced Class:** Use of a helper to hold the aircraft for takeoff is allowed. The helper may not push the aircraft on release.
- **Regular Class:** Use of a helper to hold the aircraft is allowed. Main wheels must be placed on the takeoff line for Regular Class. The helper may not push the aircraft upon release.
- **Micro Class:** aircraft must be run up and hand launched within the launch area for Micro.

3.5 AIRCRAFT CONFIGURATION AT LIFTOFF AND DURING THE FLIGHT ATTEMPT

The aircraft must remain intact during a flight attempt to receive full flight score. A flight attempt includes activities at the starting line, the take-off roll, takeoff, flight, landing and recovery after landing.

A 25% deduction from the flight score will be assessed if any of the following items are observed to completely detach from the aircraft during a flight attempt.

- Stickers
- Tape
- Coverings

With the exception of a broken prop during landing, if any other components fall off the during a flight attempt, the flight will be disqualified.

3.6 COMPETITION CIRCUIT REQUIREMENTS

1. During departure and approach to landing, the pilot must not fly the aircraft in a pattern that will allow the aircraft to enter any of the no-fly zones.
2. No aerobatic maneuvers will be allowed at any time during the flight competition in any competition class. This includes but not limited to: loops, figure 8's, Immelmann, all types of rolling maneuvers and inverted flight.
3. Regular and Micro Class aircraft must successfully complete a minimum of one 360° circuit.
4. Advanced Class has no specific flight pattern. (See Advanced Class rules for details concerning the releasable payload drop mission element.)

3.7 TIME LIMITS AND MULTIPLE FLIGHTS ATTEMPTS

1. Multiple takeoff attempts are allowed within the class specific time allotment as long as the aircraft has NOT become airborne during an aborted attempt.
2. If an airborne aircraft returns to the ground after being airborne and is beyond the take-off limits, the flight attempt will be disqualified for that round.

Table 3.1: Flight Attempt Information

Class	**Time Limit (sec)**	**Can make multiple takeoff attempts if:**			**Definition of Takeoff is defined as the point at which:**
		Still within the Time Limit	Bounce within required take-off distance	Bounce outside the required take-off distance	
Regular	120	Yes	Yes	No	The wheels leave the starting line
Advanced	180	Yes	Yes	No	The wheels leave the starting line
Micro	60	No	No	No	The launcher is no longer in contact with the aircraft

3.8 TAKE-OFF

Takeoff direction will be determined by the Air Boss, and will selected to face into the wind if possible.

1. Regular and Advanced Class aircraft must remain on the runway during the takeoff roll.
2. Micro Class must be hand launched from the designated launch area that is a box 8' on each side.
3. Distance requirements are defined in Table 3.2.
4. Making the initial turn before passing the "distance from initial start before turn" requirement will disqualify that flight attempt. (Table 3.2)

Table 3.2: Take-Off information

Class	Take-Off Distance Limits (ft.)	Distance from initial start before turn (ft.)	Description
Regular	200 ft.	400 ft.	Aircraft must be airborne within the prescribed take-off distance.
Advanced	None	None	Aircraft will have the full use of the runway.
Micro	N/A	400 ft.	Team may use the entire launch area per attempt to get the aircraft airborne. Only one (1) launch attempt per round is allowed.

3.9 LANDING

A successful landing is defined as a controlled return to the ground with the aircraft remaining inside the landing zone with positive ground control for that class.

3.10 LANDING REQUIREMENTS

The landing zone is a predetermined fixed area for each class for the purpose of returning a flying aircraft back to the ground. See Table 3.3 for class requirements.

1. The landing zones will be visibly marked at each event site prior to the start of the competition.
2. It is the team and team pilot's responsibility to be aware of the class specific landing zone dimensions at the event site.
3. *All aircraft must remain within their designated landing zone and on the paved runway during landing. Any aircraft that leaves their designated landing zone or the paved runway for any reason during landing are subject to a penalty of 50% of any points earned during the flight prior to landing.*
4. Any flight where the aircraft does not make the initial touch down for landing inside the designated landing zone is disqualified and forfeits all points for that flight.
5. Any landing where the aircraft is not rolling on the ground when it leaves the landing zone (i.e., bouncing into the air as it leaves the landing zone) is disqualified.

6. Touch-and-go landings are not allowed and will be judged as a failed landing attempt.
7. The criteria for being within the landing zone is that no supporting part of the aircraft that is touching the ground can be outside the landing zone. For example, a wing tip or fuselage is allowed to overhang the edge of the landing zone, as long as no supporting part of the aircraft is physically touching anything outside the landing zone.

Table 3.3: Landing Distance Limit

Class	**Landing Distance Limits (ft.)**	**Description**
Regular	Supplied by event	Aircraft must land in the same direction as takeoff and stop within the designated landing zone.
Advanced	Supplied by event	Aircraft must land in the same direction as takeoff and stop within the designated landing zone.
Micro	200 ft.	Aircraft must land in the same direction as takeoff and stop within the designated landing zone.

3.11 GROUNDING AN AIRCRAFT

1. An aircraft will be grounded if it is deemed non-flight-worthy or not in compliance with class rules by any SAE official, event official or a designated technical/safety inspector.
2. Until the non-flight-worthy or out of compliance condition has been addressed and has been cleared by re-inspection, the aircraft will not be allowed to fly in the competition.

3.12 NO-FLY ZONE

Each competition will have venue-specific **no-fly zones**. The no-fly zones will be defined during the all hands briefing at the event and during the pilot's briefings.

1. At no time will an aircraft enter the no-fly zones, whether under controlled flight or uncontrolled.
2. First infraction for crossing into the no-fly zone will result in an invalidated flight attempt and zero points will be awarded for that flight.
3. Second infraction will result in disqualification from the entire event and loss of all points.
4. It is the team and team pilot's responsibility to be aware of the venue-specific no-fly zones and to comply with all venue specific rules.
5. If a team is unable to directionally control their aircraft and it is headed towards or is in a no fly zone, the Judges and/or Flight boss may order the pilot to intentionally crash the aircraft to prevent it from endangering people or property. This safety directive must be followed immediately if so ordered by the officials.

3.13 FLIGHT RULES ANNOUNCEMENT

Flight rules will be explained to all teams before the flight competition begins, either during the pilots' meeting or during activities surrounding the technical inspections and oral presentations.

3.14 FLIGHT RULES VIOLATIONS

1. Violation of any flight rule may result in the team being eliminated from the competition.
2. All members of an eliminated team may be escorted from the grounds.

3.15 LOCAL FIELD RULES

In addition to competition rules, the local flying club may have additional rules in place at the event flying field.

1. Club rules will be obeyed during the flight competition.
2. In the event that club rules conflict with competition rules, it is the responsibility of the team captain and/or faculty advisor to bring attention to the conflict and follow the appeals process to resolve the conflict.

3.16 COMPETITION SCORING

A team's final, overall score is composed of scores in the following categories:

1. Technical Design Report (Design, Written and Drawing)
2. Presentation
3. Flight Score
4. Penalties

Any Penalty Points assessed during the competition will be deducted from a team's overall score.

4 Design Report

The Design Report is the primary means in which a team conveys the story of how their aircraft is the most suited design to accomplish the intended mission. The Design Report should explain the team's thought processes and engineering philosophy that drove them to their conclusions.

Some topics that are important to cover are: selection of the overall vehicle configuration, wing planform design including airfoil selection, drag analysis including three-dimensional drag effects, aircraft stability and control, power plant performance including both static and dynamic thrust, and performance prediction. Other topics as appropriate may be included, see SAE Aero Design Report Guidelines available at www.saeaerodesign.com/go/downloads for additional comments, suggested topics and a suggested outline. For more information regarding performance prediction, a white paper by Leland Nicolai is also available at www.saeaerodesign.com/go/downloads.

4.1 Submission Deadlines

The Technical Design Report, 2D drawing, and supplemental Tech Data Sheet (TDS) must be electronically submitted to www.saeaerodesign.com no later than the date indicated on the Action Deadlines given on the SAE International Website:

https://www.sae.org/attend/student-events

Neither the Organizer nor the SAE International is responsible for any lost or misdirected reports, drawings, or server routing delays. The SAE International will not receive any paper copies of the reports through regular mail or email outside of the emergency submissions email.

4.2 Original Work

The Technical Design Report shall be the team's original work for this competition year. Resubmissions of previous year's design reports will not be accepted. Recitation of previous year's work is acceptable if appropriately cited and credited to the original author. Plagiarism is a forbidden industry and academic practice, all references, quoted text and reused images from any source shall have appropriate citation within thetext and within the Technical Design Report's Table of References providing credit to the original author and editor.

Reports may be checked against previous years submissions to determine if re-use, copying, or other elements of plagiarism are indicated.

For the purposes of the SAE International Aero Design Competition, plagiarism is defined as any of the following:

- Use of information from textbooks, reports, or other published material without proper citation
- Use of sections or work from previous SAE Aero Design competitions withoutproper citation

If plagiarism is detected in the written report, a team will be given 24 hours to make a case to SAE and Rules Committee. If the report and/or case is found to be insufficient, the team will receive zero score for the report. The team will be allowed to compete in all remaining categories of the competition but will not be eligible for awards. SAE also reserves the right to notify the University of the situation.

If plagiarism is detected in the oral presentation, team will receive zero score for the presentation. The team will be allowed to compete in all remaining categories of the competition but will not be eligible for awards. SAE also reserves the right to notify the University of the situation.

The SAE Aero Design Rules Committee & SAE International has the sole discretion to determine whether plagiarism is indicated and the above rules are enacted. The above rules may be implemented at any time before, during, or for up to 6 months after the competition event.

4.3 Technical Design Report Requirements

Technical Design Report will be 50 points (pts) of the competition score as broken down in Table 4.3.1.

1. The Technical Design Report shall not exceed thirty (30) pages, including the certificate of compliance, 2D Drawing, and the Supplemental Datasheet for each class. If the design report exceeds thirty (30) pages, the judges will only score the first thirty (30) pages.
2. The Technical Design Report shall include a Cover Page with Team Name, Team Number, and School Name and Team Member Names.
3. The Technical Design Report shall include a Certificate of Compliance signed by hand by the team’s faculty advisor.
4. The Technical Design Report shall be typewritten and double-spaced. Tables, charts and graphs are exempt from this
5. The report font shall be 12 pt. proportional; or 10 char/in. non-proportional font.
6. The report margins shall be: 1” Left, 0.5” right, 0.5” top, and 0.5” bottom.
7. Each page, except the Cover Page, Certificate of Compliance, 2D Drawing and Technical Data Sheet (TDS) shall include a page number.
8. All report pages shall be ANSI A (81/2 x 11 inches) portrait-format.
9. The Technical Design Report shall include a Table of Contents, Table of Figures, Table of Tables, Table of References and Table of Acronyms.
10. The Technical Design Report shall be single-column text layout.
11. The Technical Design Report shall include one Technical Data Sheet (TDS) appropriate for the team’s competition entrant class.

Table 4.3.1 Technical Design Report

	Page Count	Regular Class	Advanced Class	Micro Class
Cover Page	1	40 pts	40 pts	40 pts
Certificate of Compliance	1			
Design Report	26			
2D Drawing	1	5 pts	5 pts	5 pts
TDS: Payload Prediction	1	5 pts	-	-
TDS: Colonist Delivery Aircraft 2D Drawing	1	-	5 pts	-
TDS: Aircraft Weight Build-Up Schedule (Appendix B)	1	-	-	5 pts
Total	**30**	**50 pts**	**50 pts**	**50 pts**

4.4 2D Drawing Requirements

1. *2D Format and Size*

The 2D drawing must be ANSI B sized page (PDF) format (11 x 17 inches).

1. For teams outside North America that cannot submit an ANSI B size drawings, page format size must be the closest size available to ANSI B.
2. Drawing shall consist of one (1) page.

2. *Markings Required*

The 2D drawing must be clearly marked with:

1. Team number
2. Team name
3. School name

3. *Views Required*

Drawings shall include at a minimum, a standard aeronautical 3-view orthographic projection arranged as described:

1. Left side view, in lower left, with nose pointed left.
2. Top view, above and aligned with the left side view, also with nose pointed left (wing-span break-view permitted).
3. Front view aligned to side view, located in the lower right (projection view non-standard movement as noted by projection view arrows in accordance with ANSI-Y14.5M 1994).
4. (Regular Class Only) Regular Class shall include an additional view, separate from the basic aircraft, illustrating the passenger cabin layout with appropriate dimensions identifying the passenger seating arrangement. Total passenger capacity shall be labeled.

4. *Dimensions Required*

Drawing dimensions and tolerance shall be in English units, decimal notation accordance with ANSI-Y14.5M 1994 to an appropriate level of precision to account for construction tolerances (allowable variation from analyzed prediction to account for fabrication) (i.e. X.X = ± .1 in; X.XX = ± .03 in; X.XXX = ± .010 in).

The minimum required dimensions/tolerances are: Aircraft length, width, and height

5. *Summary Data Required*

The drawing shall contain a summary table of pertinent data to include but not limited to:

1. Wingspan
2. Empty weight
3. Battery(s) capacity
4. Motor make and model
5. Motor KV (micro and Regular Class only)
6. Propeller manufacturer, diameter, and pitch
7. Servo manufacturer, model number and torque specification in ounce-inches for each servo used on the aircraft. Identify servo being used at each position on the aircraft.

6. *Weight and Balance Information*

The 2D drawing shall contain the following weight, balance and stability information:

1. A clearly marked and labeled aircraft datum
2. A weight and balance table containing pertinent aircraft equipment. Each item listed must show its location from the aircraft datum in inches (the moment arm), the force, and resultant moment. See www.saeaerodesign.com/go/downloads for additional information. The minimum list of pertinent equipment includes:
   a. Motor
   b. Battery(s)
   c. Payload
   d. Ballast (if used)
   e. Electronics
3. Aircraft mean aerodynamic cord, stability margin and Center of Gravity (CG) information listed below must be clearly shown on drawing.
   a. Aircraft mean aerodynamic cord
   b. Stability margin for loaded CG and empty CG
   c. Empty CG location (flightworthy)
   d. Fully loaded CG (flightworthy, with payload, if applicable)

4.5 TECH DATA SHEET: PAYLOAD PREDICTION (REGULAR CLASS ONLY)

Regular Class teams must include a total payload prediction curve as part of the technical report. The graph represents an engineering estimate of the aircraft's lift performance based on density altitude.

1. Graph of payload weight shall be linearized over the relevant range.
2. The linear equation shall be in the form of:

$$y = mX + b$$

Y = Payload weight (lbs.)
X = Density Altitude (feet)
m = Slope of the linear line
b = y-intercept.

3. Only one line and one equation may be presented on the graph. This curve may take into account predicted headwind for local conditions, rolling drag, inertia, motor and propeller performance, or any other factors that may affect takeoff performance. All these factors are allowed components of the prediction curve, but only one curve will be allowed; multiple curves to account for varying headwind conditions will not be allowed.
4. The team must provide a brief explanation of how the line was generated in the body of the report. The section of the report containing this information must be noted on the payload prediction curve.
5. Graph axes shall be in English units, decimal notation.

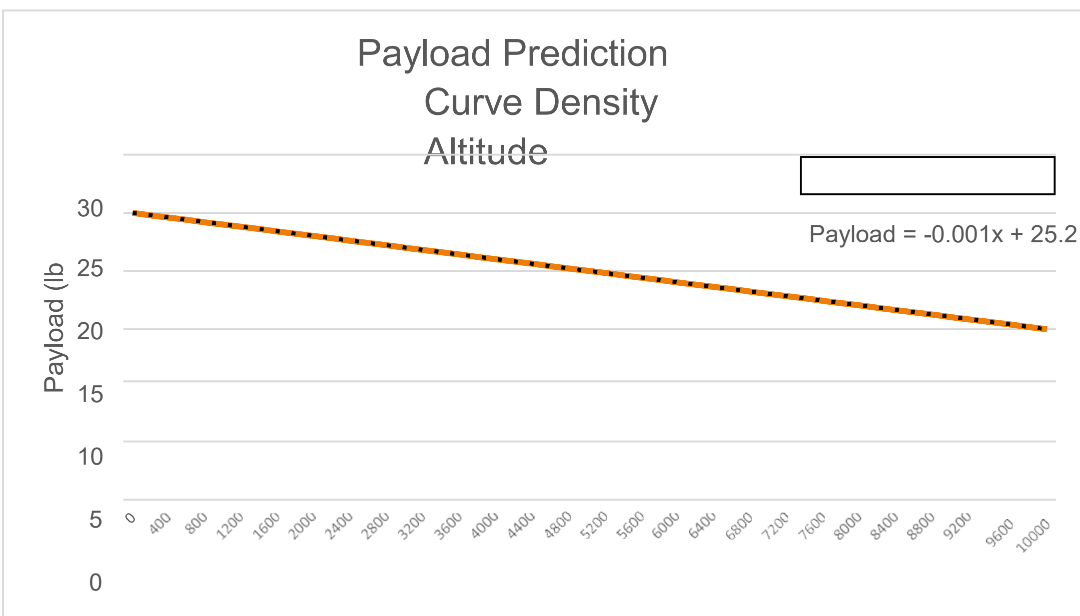


Figure 4.1: Example Regular Class Payload Prediction Curve

4.6 TECH DATA SHEET: COLONIST DELIVERY AIRCRAFT (ADVANCED CLASS ONLY)

An additional 2D drawing must be provided as an Appendix for the Colonist Delivery Aircraft (CDA). This 3-view must be ANSI B sized page (PDF) format (11 x 17 inches) and follow the same requirements as the primary aircraft 2D drawing.

1. Drawing for the Colonist Delivery Aircraft shall identify the location of the loaded CG.
2. Team shall provide a glider polar (airspeed vs sink rate) for the Colonist Delivery Aircraft in fully loaded configuration, with no headwind. An explanation of the glide polar can be found on page 5-8 of the FAA Glider Handbook, Chapter 5: Glide Performance available at the following link.

   https://www.faa.gov/regulations_policies/handbooks_manuals/aircraft/glider_handbook/media/gfh_ch05.pdf

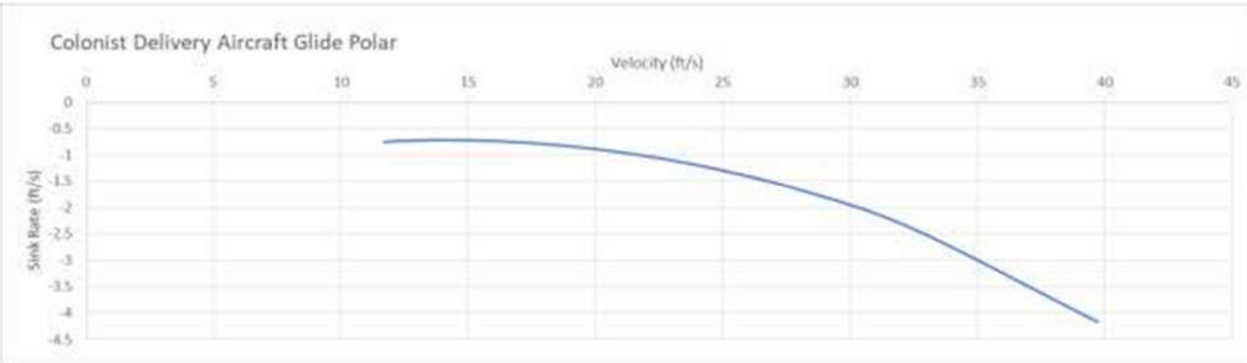


Figure 4.2: Example Advanced Class CDA Glide Polar

3. Team shall provide a list of avionics and equipment for the Colonist Delivery Aircraft.
4. Separate from the basic aircraft drawing, team shall provide a view illustrating the colonist cabin layout with appropriate dimensions identifying the colonist seating arrangement with total colonist capacity labeled.

4.7 TECH DATA SHEET: WEIGHT BUILDUP (MICRO CLASS ONLY)

The Micro Class Weight & Balance Build-up schedule will help teams understand the importance of managing aircraft weight to achieve safety of flight at the desired payload fraction. Each team shall supply a one (1) sheet summary list of aircraft parts that contributes to the overall empty weight of the aircraft (lb.).

A template for the weight buildup can be found in Appendix B.

5 Technical Presentation

Like all professionals, engineers must possess a well-developed ability to synthesize issues and communicate effectively to diverse audiences. The technical portion of the aero-design competition is designed to emphasize the value of an ability to deliver clear, concise and effective oral presentations. Teams can obtain a maximum technical presentation score of fifty (50) points. Presentation score shall be comprised of scores from the presenter's delivery technique and the judges' evaluation of technical content, empirical analysis, and quality visual aide.

5.1 Technical Presentation Requirements

1. Technical presentation shall last ten (10) minutes and be followed by a five (5) minute "Question and Answer" (Q&A) period.
2. Technical presentation shall be delivered in English.
3. Technical presentation shall address, but are not limited to, trade studies performed, design challenges, and manufacturing techniques.
4. Technical presentation is limited to student team members only. Non-team member pilot, faculty advisors, and/or parents can attend the technical presentation but are prohibited from participating in the setup, delivery, and/or the Q&A.
5. Assistance in the use of visual aids is advisable; Film clips, if used, may not exceed one-minute total duration; Film clips may not be accompanied by recorded narration.
6. All Classes shall display their entry aircraft during technical presentation.
7. *Advanced class shall display one at least one (1) of each payload, including the colonist delivery aircraft.*
8. During the presentation and static display setup, the teams shall provide a single sheet (8.5" x 11") marketing/promotion piece to further detail aircraft's feature, capabilities, and unique design attributes.

5.2 TECHNICAL PRESENTATION PROCESS AND PROCEDURES

Each presentation room shall have a lead judge with the responsibility to ensure compliance with competition rules and schedule. Lead judge will identify a timekeeper.

1. With agreement from the speaker, the timekeeper will give the speaker a one (1) minute warning prior to the ten (10) minute limit.
2. If the team exceeds the ten (10) minute limit, the team will be assessed a five (5) point penalty for going over the time limit.
3. The presentation shall be stopped at the eleven (11) minute mark.
4. A team shall have five (5) minutes for Q&A immediately following the presentation. Questions may be asked by any judge on the panel.
5. Any time remaining or exceeding the ten (10) minutes shall be added to or subtracted from the five (5) minute Q&A.
6. Presentation Time Breakdown:

Time (Minutes)	Description
2	Setup presentation, visual aide, and/or static display
10	Perform Technical Presentation
5	Questions & Answers
3	Pack-up presentation and static display

6 Technical Inspection and Aircraft Demonstrations

Technical and Safety inspection of all aircraft will be conducted using the published Technical and Safety Inspection checklists for each class for the current year. The checklists can be found at www.saeaerodesign.com/go/downloads.

Technical and Safety Inspection is the process of checking all aircraft for:

- Compliance with all General aircraft requirements.
- Compliance with all aircraft configuration requirements for their class.
- Overall safety and airworthiness.

All aircraft must pass the Technical and Safety Inspection in order to compete. It is strongly suggested that each team pre-inspect their aircraft and correct any problems using the official inspection checklist before arriving at the competition.

All required Aircraft Demonstrations will be performed at designated locations in the Technical Inspection area.

- **Regular Class** will demonstrate the ability to load and unload their aircraft per the requirements of rule 7.6.
- **Advanced Class** will demonstrate that their aircraft has proven operational ability by providing a video showing the aircraft successfully taking off, dropping a payload and landing per the requirements of rule 8.1.
- **Micro Class** will demonstrate the timed assembly of their aircraft per the requirements of rule 9.6.

6.1 Aircraft Conformance to 2D drawing

During Technical Inspection, the aircraft will be inspected and measured for conformance to the 2D drawing presented in the Design Report.

1. At a minimum, aircraft length, wingspan and height dimensions will be measured and compared to the 2D drawing.
2. All teams must have a hard copy of their design report with them during technical inspection.
3. Aircraft will have their actual empty CG compared to the empty CG presented in the design report 2D drawing.
4. Advanced Class must show longitudinal and lateral C.G. positions or provide a table for each payload configuration

6.2 Failure to report design changes

Failure to report any design changes incorporated after Design Report submission and prior to Technical Check-in will incur a one (1) point penalty for each unreported design change discovered during technical inspection.

6.3 DEVIATIONS FROM 2D DRAWING

Any deviation in construction of the aircraft from the submitted 2D drawing, after submission of the Design Report, must be reported in writing.

1. Each design change must be documented separately using the Engineering Change Request (ECR) – a physical copy of which must be brought to the Technical and safety Inspection
2. Only one design change may be submitted per ECR form.
3. Penalty points for design changes will be assessed in accordance with the penalty guidelines in Appendix D, subject to the judges' final determination.

6.4 SAFETY AND AIRWORTHINESS OF AIRCRAFT

Technical and Safety Inspection will be also be used to assess the general safety and airworthiness aspects of each aircraft by seeking any problems that could cause an aircraft to depart controlled flight. This assessment includes but is not limited to:

1. Unintentional wing warps
2. Control surface alignment
3. Correct control surface response to radio transmitter inputs
4. Structural and mechanical soundness

6.5 INSPECTION OF SPARE AIRCRAFT AND SPARE AIRCRAFT COMPONENTS.

1. All spare aircraft and spare aircraft components (wings, fuselages and tail surfaces) must be presented for inspection.
2. Teams may submit up to two complete aircraft at Technical Inspection on Friday.
3. Additional spare aircraft and parts beyond two sets may be submitted for inspection during the event on Saturday and Sunday.

6.6 AIRCRAFT MUST MEET ALL INSPECTION REQUIREMENTS THROUGHOUT THE COMPETITION.

1. All aircraft must meet all Technical and Safety Inspection requirements throughout the competition.
2. Any official may request that an aircraft be re-inspected if a general, class configuration or safety requirement problem is seen on an aircraft at any time during the event.
3. This includes any errors or omissions made by officials during inspection.

6.7 TECHNICAL AND SAFETY INSPECTION PENALTIES

No points are available to be scored as a result of the Technical and Safety Inspection: teams may only lose points as a result of errors and problems encountered during the inspection process. Any penalties assessed during Technical Inspection will be applied to the overall competition score.

7 Regular Class Design Requirements

The objective of Regular Class is to design an aircraft that can generate revenue by carrying as much payload as possible while observing the power available requirement. Payload will consist of passengers, represented by tennis balls, and Luggage, represented by payload weights, which must be carried on each flight. Accurately predicting the lifting capacity of the aircraft and selecting the appropriate number of passenger seats is an important part of the airplane design.

7.1 Aircraft Dimension Requirement

Regular Class aircraft are limited to a maximum wingspan of 144 inches.

7.2 Material and Equipment Restrictions for Regular Class

1. *Fiber-Reinforced Plastic (FRP)*
The use of Fiber-Reinforced Plastic (FRP) is prohibited on all parts of the aircraft. Exceptions to this rule include: commercially available FRP motor mount, propeller, landing gear and control linkage components. Exploration of alternative materials is encouraged.

2. *Rubber bands*
Elastic material such as rubber bands shall not be used to retain the wing or payloads to the fuselage.

3. *Stability Assistance*
All types of gyroscopic or other stability assistance are prohibited.

7.3 Aircraft System Requirements

1. *Electric Motor Requirements*
The aircraft shall be propelled by a single electric motor (no multiple motors). There are no restrictions on the make or model of the electric motor.

2. *Gear boxes, Drives, and Shafts*
Gearboxes, belt drive systems, and propeller shaft extensions are allowed as long as a one-to-one propeller to motor RPM is maintained. The prop(s) must rotate at motor RPM.

3. *Aircraft Propulsion System Battery*
Regular Class aircraft must be powered by a commercially available Lithium-Polymer battery pack.

1. Required: 6 cell (22.2 volt) Lithium Polymer (Li-Poly/Li-Po) battery pack. Minimum requirements for Li-Po battery: 3000 mAh, 25c
2. Homemade batteries are NOT allowed.

4. *Power Limiter*
All Regular Class aircraft must use a 2015 V2 or newer version 1000 watt power limiter from the official supplier, Neumotors.com.

1. Repair and/or modifications to the limiter are prohibited.
2. The limiter must be fully visible and easy to inspect.

3. Only battery, receiver, speed control, arming plug, and limiter are allowed within the power circuit.
4. The limiter is only available at the follow link:

http://neumotors.cartloom.com/shop/item/24377

This supplier has agreed to ship worldwide to any team.

5. Radio System Battery and Switch

If a separate battery is used for the radio system, the battery pack must have enough capacity to safely drive all the servos in the aircraft, taking into consideration the number of servos and potential current draw from those servos.

1. A battery pack with a minimum capacity of 1000 mAh must be used for the radio system.
2. The battery pack must be a LiPo or LiFE type battery.
3. Battery voltage regulators are allowed.
4. The battery pack must be controlled by a clearly visible and properly mounted on/off switch on the external surface of the aircraft, located at least 12" from the prop.

7.4 PAYLOAD REQUIREMENTS

1. *Types of Payload*

Regular Class payload shall consist of two types; (1) Passengers and (2) Luggage, which must be carried in proportion to one another in the Passenger Cabin and Payload Bay respectively. Both the Passenger Cabin and Payload Bay must be designed for ease of access to both Passengers and Luggage. This will be demonstrated during the oral presentation (Reference Section 7.6 for demonstration details).

2. *Payload Bay Requirements*

Regular Class aircraft shall have a single fully enclosed Payload Bay for carrying Luggage (see section 7.4.2) with the following additional requirements:

1. The Payload Bay shall only contain Luggage.
2. The Payload Bay shall fully enclose the Luggage.
3. The Payload Bay has no restriction on size or shape.
4. Only one Payload Bay is allowed in a Regular Class aircraft.

3. *Luggage and Luggage Support Requirements*

Luggage and Luggage Support Requirements Luggage shall consist of a support assembly and payload plates with the following additional requirements:

1. For weight measurement and scoring, Luggage shall consist of the payload plates and support structure used to retain the weight(s) in the Payload Bay.
2. An average Luggage weight of ½ lb. or more must be carried for each Passenger carried.
3. If the Luggage weight carried does not meet the ½ lb. per passenger requirement, the excess Passengers will be counted as empty seats and the flight scored accordingly.
4. Luggage weight in excess of ¾ lbs. per passenger will not count in Luggage weight scoring.
5. There is no required configuration for the payload plates.
6. Teams must provide their own payload plates.
7. The support assembly must securely bolt together all payload plates by passing one or more bolts through one or more holes in each payload plate. The support assembly must be secured to the aircraft structure via bolts that pass through some or all of the payload plates to ensure that the payload/ luggage cannot shift or move during flight operations. Bolts must be secured with nuts.
8. Tape, Velcro, rubber bands, container systems and friction systems alone may not be used to retain the support assembly and/or payload plates.

4. *Passenger Payload Definition*

The Passenger* payload must consist only of unmodified tennis balls which meet or exceed the minimum size and weight specifications for Type 1 and Type 2 tennis balls as specified by the International Tennis Federation (ITF). Accordingly, the minimum tennis ball weight is 1.975 ounces and minimum ball diameter allowed is 2.57 inches. A list of accepted brands of tennis balls is given at:

http://www.itftennis.com/technical/balls/.

* Teams must provide their own Passengers.

5. *Passenger Cabin Requirements*

Regular Class aircraft must position all Passengers in a single Passenger Cabin.

1. The Passenger Cabin must position all Passengers to be tangent to the same side of a single geometric plane.
2. All Passengers must be constrained to the geometric plane within the Passenger Cabin so that they will not shift or come loose during any portion of a flight.
3. A position designed to hold a Passenger is a Seat. Passenger Seats must be in contiguous positions. Contiguous is defined as being less than 0.25 inches from the adjacent Passenger.
4. Any Passenger not in a seat after a flight will not count as revenue generated by Passengers.

5. Passengers must be in a countable configuration to be scored as a revenue generating Passenger. A countable configuration is defined as when Passengers are clearly visible and can be easily touch-counted.
6. Each Passenger carried in excess of the maximum number (as determined by Luggage weight), will not count as a revenue generating Passenger and will be scored as an empty Seat.

7.5 PASSENGER SEATING REQUIREMENTS

1. Regular Class aircraft must accommodate a minimum of 10 passengers and the required luggage.
2. Regular Class teams must document the number of passenger seats in their design per rule 4.4.3.4 at the time the design report is submitted. This seat number is used to determine passenger count and empty seats for the scoring equation.
3. Teams are allowed to reduce the number of seats in their design one time. This reduction can only be made after design report submission and before technical inspection. An Engineering Change Request defining the new seating number must be submitted on the normal ECR form. There is a one point penalty for each seat removed from the original design submission. After this ECR is submitted, the revised number of seats is then used in the scoring equation.
4. The number of passenger seats may never be increased after Design Report submission.

7.6 REGULAR CLASS PAYLOAD LOADING AND UNLOADING DEMONSTRATION

Technical Presentation for Regular Class shall demonstrate the requirement to quickly load/secure and unload both Passengers and Luggage. This is a timed activity and shall be performed by no more than two (2) members of the team within the following time constraints:

1. One (1) minute to load/secure both Luggage (5 – 7.5lbs.) and 10 passengers for flight.
   - The demonstration will start with the passengers and cargo separate from the aircraft and the aircraft in a flight-ready configuration.
   - The demonstration is considered complete when all required passengers and Luggage are loaded, secured, and the aircraft is put back in a flight-ready configuration.
2. One (1) minute to unload both Luggage (5 – 7.5lbs.) and 10 Passengers.
   - The demonstration will start with all required Passengers and Luggage loaded, secured, and the aircraft in a flight-ready configuration.
   - The demonstration will be considered complete when both the Passengers and Luggage are separate from the aircraft and the aircraft is put back in a flight-ready configuration.

This demonstration will be performed at a designated location in the Technical Inspection area on Friday.

7.7 Regular Class Scoring

In order to participate in the flight portion of the competition, each team is required to have submitted AND received a score for their Design Report and Oral Presentation.

The Final Regular Class Flight Score shall be based upon the Total Revenue earned and Total Penalty deductions received.

Scoring Equation:

$$FFS = \ Final\ Flight\ Score \ = \frac{1}{40\ N}[\sum_{1}^{N} FS]$$

Where:

$$FS = Flight\ Score = \$100P + \$50C - \$100E\ for\ each\ \ flight$$

$$P = Number\ of\ seated\ Passengers\ carried\ on\ a\ \ flight$$

$$C = Luggage\ weight\ (lbs)$$

$$E = Number\ of\ empty\ Seats$$

$$N = Total\ Number\ of\ Flight\ Rounds\ During\ \ Competition$$

Penalty Points

Any penalty points assessed during the competition are now deducted from a team's overall score.

NOTE: Final Flight Score (FFS) less than zero (0) will default to zero (0).

Example.

If a team with 30 seat capacity plane fails to fly a round (via crash/repair or strategic decision not to fly), the team will receive a negative flight score (FS) of -$3000.00.

8 ADVANCED CLASS DESIGN REQUIREMENTS

The objective of the Advanced Class is to design a system that can deliver colonists, habitats, and supplies to the surface of Mars. This class is focused on mission success through understanding of diverse requirements, system-level engineering, and robust execution.

8.1 VIDEO DOCUMENTATION OF PROVEN OPERATIONAL ABILITY FOR ADVANCED CLASS

All Advanced Class teams are required to bring a video documenting the proven operational ability of their Advanced Class aircraft to Technical and Safety Inspection. The hard deadline for video submission is 8AM Saturday Morning.

1. The video must show the following three activities accomplished successfully with their competition aircraft: A takeoff, a successful release of a Colonist Delivery Aircraft (CDA), and a landing of the Primary Aircraft (PA) without damage to the PA. A successful release of the CDA means that the CDA is in a flyable configuration after release. No video is required of the CDA flying or landing after the release from the PA.
2. The video will be reviewed by SAE officials in the Technical Inspection area.
3. Advanced Class aircraft will not be inspected or allowed to compete without the video documentation of proven operational ability.
4. Teams must provide a device to play the video for the officials at a screen size that allows the officials to clearly see both aircrafts.
5. Videos should be no more than 1 minute in length. Edited video will be accepted if the video is of the same flight.

8.2 AIRCRAFT DIMENSION REQUIREMENT

Advanced Class aircraft are limited to a maximum wingspan of 144 inches.

8.3 AIRCRAFT SYSTEM REQUIREMENTS

1. *Electric Motor Requirements*

The Primary Aircraft shall be propelled by one or more electric motors. There are no restrictions on the make or model of the electric motor.

2. *Gear boxes, Drives, and Shafts*

Gearboxes, belt drive systems, and propeller shaft extensions are allowed.

3. *Aircraft Propulsion System Battery*

Advanced Class Primary Aircraft shall be powered by a single commercially available Lithium-Polymer battery pack.

1. Required: 6 cell (22.2 volt) Lithium Polymer (Li-Poly/Li-Po) battery pack. Minimum requirements for Li-Po battery: 3000 mAh, 25c
2. Homemade batteries are NOT allowed.

4. *Power Limiter*

All Advanced Class Primary Aircraft shall use a single 2018 or newer version 750 watt power limiter from the official supplier, Neumotors.com.

1. Repair and/or modifications to the limiter are prohibited.
2. If multiple motors are used, all throttle signals and power must pass through a single limiter.
3. The limiter must be fully visible and easy to inspect.
4. Only battery, receiver, speed control, arming plug, and limiter are allowed within the power circuit.
5. The limiter is only available at the follow link:

https://neumotors.cartloom.com/storefront/category/energy-limiters-telemetry-wattmeters-servo-testers

This supplier has agreed to ship worldwide to any team.

8.4 RADIO SYSTEM BATTERY

The radio system battery pack must have enough capacity to safely drive all the servos in the Primary Aircraft, taking into consideration the number of servos and potential current draw from those servos. If the radio system battery also supplies DAS or other power needs, the radio system battery must be large enough for these power requirements as well.

1. A battery pack with a minimum capacity of 1000 mAh must be used for the radio system.
2. The battery pack must be a LiPo or LiFE type battery.
3. Battery voltage regulators are allowed.
4. The battery pack must be controlled by a clearly visible and properly mounted on/off switch on the external surface of the Primary Aircraft, located at least 12" from the prop.

8.5 RUBBER BANDS

Rubber bands shall not be used to retain the wing to the fuselage.

8.6 COLONIST DELIVERY REQUIREMENTS

Teams are responsible for delivering colonists safely to the surface of the planet through up to 3 unpowered (no on-board propulsion) autonomously guided aircraft.

1. *Colonists*

Colonists must consist only of unmodified table tennis (ping-pong) balls which meet or exceed the minimum size and weight specifications as specified by the International Table Tennis Federation Technical Leaflet T3: The Ball. Accordingly, the minimum ping-pong ball weight is 0.095 ounces and minimum ball diameter allowed is 1.57 inches. Teams must provide their own Colonists.

2. Colonist Delivery Aircraft (CDA)

The following requirements apply to the colonist delivery aircraft:

1. Total weight of each CDA must be less than 9.0 oz.
2. Center of gravity must be clearly marked.
3. If more than one CDA is used, each must be uniquely colored, visible from the ground. Each CDA's transmitter must be marked with the matching color.
4. All Colonists must be carried internally in a single colonist cabin.
5. The colonist cabin must position all Colonists to be tangent to the same side of a single geometric plane.
6. Colonists must be in a countable configuration. A countable configuration is defined as when Colonists are clearly visible and can be easily touch-counted.
7. Three (3) Teladrop® resettable impact shock sensors rated at 50G must be rigidly mounted to the airframe within 1 inch of the CG: 1 Sensor in z-direction, 1 sensor in the y-direction (span), 1 sensor in x-direction (forward/aft). These sensors must be provided by each team and are available at the following links:

https://www.telatemp.com/p/415/50g-teladrop-drop-n-tell-damage-indicator

https://www.shippinglabels.com/teladrop-drop-n-tell-resettable-shock-indicator/sku-l-drop-n-tell-r

8.7 SUPPLY PAYLOAD REQUIREMENTS

There is no limit to the number of Supply Payloads that can be carried by each Primary Aircraft. The following requirements apply to each Supply Payload:

1. Supply Payloads may be mounted internally and/or externally to the airframe.
2. The CG of each mounted Supply Payload shall not exceed a distance of ten (10) inches longitudinally from the Primary Aircraft's EW CG. This shall be measured during technical inspection.
3. All Payloads that are connected (e.g. become tangled during drop) in anyway shall not be counted for score.
4. All Supply Payloads that achieve a scoring drop on the target, or a successful drop from above 100 feet that misses the target, will be inspected and measured at the weigh station after that drop. This is the only time Supply Payloads will be inspected. If any Supply Payload is not in compliance with all related rules, score for that Supply Payload will be zero (0) for that flight attempt.

1. Habitat Module

A Habitat Module must consist of an unmodified Nerf Sports Vortex Aero Howler.

1. Each habitat module shall be marked with the team number measuring 2" in height.
2. If a team elects to make two separate habitat drops during a flight attempt, the habitats for each separate drop shall be a different color.

2. Water Bottle

Water delivery must consist of unopened plastic bottles of water.

1. Allowable water bottles are any commercially available bottle with size of 16.9 FL Oz (500 ml) or 33.8 FL Oz (1 liter).
2. Each water bottle shall have one (1) flexible streamer attached to facilitate payload location and recovery.

1. Each streamer shall be at least 48 inches long and 2 inches wide.
2. Water bottles shall be marked with the team number in two (2) places: on the container, and one end of the streamer with numbers 2-inches in height.
3. Streamers will remain attached to each water bottle at all times. Attached is defined as being able to support the weight of each water bottle.
4. If a team elects to make two independent drops during a flight attempt, the water bottles for each independent drop shall have streamers of a different color.
5. Streamers shall be in a stored configuration prior to the drop and deploy before each water bottle strikes the ground.
6. The original label must be left on the bottle for inspection. If the label is damaged or comes off during a round, the team will not be penalized.

8.8 Static Payload Requirements

1. Static payload shall be in its own payload bay(s).
2. Static payload bay(s) shall be fully enclosed, completely closed off, and physically separated from all releasable payload.
3. Static payload bay(s) shall have no restriction on size or shape.
4. Advanced Class may have multiple Static Payload bays.
5. Teams must be able to unload their Static Payload at the weigh station after their flight in 5 minutes or less.

8.9 Gyroscopic and other stability augmentation

Gyroscopic assist or other forms of stability augmentation are allowed in Advanced Class.

8.10 Autonomous flight

Autonomous flight systems that cause the primary aircraft to navigate without direct pilot control input are prohibited.

Autonomous flight for the Colonist Delivery Aircraft is required, subject to the following rules:

1. At least two degrees of freedom must be controlled by the autonomous system.
2. Teams must have a manual override for control over the Colonist Delivery Aircraft through secondary transmitters. This may be a switch on that transmitter to select between autonomous and manual flight modes.
3. Manual override will be used at the discretion of the Air Boss if there are safety concerns. Any use of the manual override disqualifies the Colonist Delivery Aircraft from successfully scoring points.
4. When switching to manual flight override, the CDA should immediately actuate full up on pitch control. No other manual control or flying will be allowed.

8.11 DATA ACQUISITION SYSTEM (DAS)

Advanced Class Primary Aircraft must have a Data Acquisition System (DAS) that shall record altitude and be used by the payload specialist to guide the primary pilot towards the target zone.

1. Using a ground receiver station, the team must display the real-time altitude of the aircraft to the Payload Specialist and the flight judge
2. Team must automatically record the altitude (ft) at the moment when a CDA or supplies are released from the Primary Aircraft. **A 1 inch indicator must state what type of payload (CDA, water or habitat) was released and provide, in 1 inch font, a single altitude associated with each released payload.**
3. The DAS recording must be performed on the ground station and must support play back for review on demand.
4. Altitude must be measured in feet with display precision of at least one (1) ft. and an accuracy error of less than ten (10) ft.
5. DAS system must use a discrete and removable Red arming plug to apply power to the DAS system. The DAS arming plug must be located on top of the Primary Aircraft 12 inches from the propeller. One Red arming plug can be used for both DAS and FPV.
6. DAS equipment may also have a reset switch, if desired. If a manual reset switch is used, it must be located externally at least 12 inches away from the propeller. A wireless DAS reset system is allowed.
7. DAS systems shall not use the same 2.4 GHz channel as the flight control system, unless the telemetry being used is part of the radio control system being used. A DAS built into the radio control system must meet all DAS rules requirements.

8.12 FIRST PERSON VIEW SYSTEM (FPV)

FPV is **no longer** a requirement for Advanced Class. For teams that wish to use an FPV system for operational reasons, the following conditions apply:

1. Teams will be required to sign up for one of 12 discrete commonly used FPV frequencies. The frequency list will be provided by Aero Design.
2. There will be a frequency sign-up process communicated to teams via the event newsletters
3. If more than 12 Advanced Class teams choose to use a FPV system, some team's frequencies may have more than 1 team using them. Frequency control procedures will be in place at the event to prevent conflicts.
4. The primary pilot must fly visually only(no FPV goggles or ground station reference).
5. FPV systems CANNOT use the same frequency as the flight control system. Use of 2.4 GHz for FPV video is prohibited.
6. The FPV system must use a discrete and removable Red arming plug to apply power to the FPV system. The Red arming plug must be located on top of the aircraft at least 12" from the propeller. One Red arming plug can be used for both DAS and FPV.

8.13 DAS Failures

Any DAS failure during the flight attempt is considered a missed flight attempt and receives zero (0) points.

> *Example: A team has flown four (4) rounds successfully and on the 5th round the Primary Aircraft takes-off successfully, makes a successful drop, but the DAS altitude reading malfunctions. The flight attempt will NOT be considered a qualified flight and the team will receive zero (0) credit for colonist, supplies or static payload for round 5.*

8.14 Payload Specialist

Advanced Class Primary Aircraft must be able to release payloads using a system commanded by the Payload Specialist.

1. Payload Specialist must be a team member and use only the DAS to verbally direct the pilot to the drop zone. Payload Specialists should not count on having line-of-sight to the aircraft.
2. Teams are allowed two (2) drop attempts on the target.
3. The primary pilot cannot have access to or activate any payload release, and the release cannot be connected to the primary pilot's R/C transmitter in any way.
4. The payload release must be manually activated by the Payload Specialist or by an automatic release system that is part of the Primary Aircraft electronics.
5. If an automatic payload release system is used, it must have a manual override controlled by the Payload Specialist.
6. Teams may activate the payload release system using a second 2.4 GHz radio system or some other method based on their DAS or telemetry system.

8.15 Colonist Delivery Aircraft Release Procedures

1. Release of the Colonist Delivery Aircraft must be at least 200 feet away the center of the drop zone, measured parallel to the runway. An SAE official with flags will be stationed up and down wind of the drop zone to indicate when the Primary Aircraft is outside this distance
2. Teams must release the CDA at an altitude of above 50ft but no higher than 100ft.
3. Teams may release a CDA during at any point in their flight pattern, as long as they adhere to 8.15.1 and 8.15.2. See figure below.
4. If any of the mounted shock sensors on the CDA show that they have been tripped upon landing, no points will be awarded for the colonist on that CDA.

8.16 Supply Drop Procedures

1. Teams are allowed two Supply (2) drop attempts on the target.
2. Teams can drop the Supply Payloads in the upwind or downwind directions or both.
3. Dropping direction shall be declared to the Air Boss prior to takeoff and adhered to during flight operations. Stated direction will be recorded on the flight log.
4. Each approach to the target in the direction declared to the flight boss is considered a drop attempt. Teams may drop as many of each payload type as they wish during each attempt.
5. Teams must drop Supply Payloads at an altitude of above 100ft.
6. To receive credit for a flight, the team shall successfully release at least one (1) type of payload (CDA, water, or habitat) at its required altitude.

A successful payload release is defined as the intentional targeted drop of at least one payload (CDA, water, or habitat), while conforming each payload's rules as found in Sections 8.15 and 8.16. The payload does not have to land in the target zone but must meet all requirements after the drop. Altitude at release must be successfully recorded on the DAS and displayed on the ground station.

Example 1:

A team states they will drop in one direction (upwind or downwind). They will get two (2) attempts to position their Primary Aircraft in the proper direction. Each approach to the drop zone in the declared direction will be counted as a drop attempt.

Example 2:

A team states they will drop in both directions; upwind and downwind. Any approach to the drop zone is considered an attempt.

8.17 ADVANCE CLASS SCORING

In order to participate in the flight portion of the competition, each team is required to have submitted AND received a score for both Design Report and Oral Presentation.

The final flight score is based on the total quantity of Colonists and Days of Habitability, derived from the number habitats and amount of water a team successfully delivers across the entire event. Delivery consists of the Colonist Delivery Aircraft or Supply Payloads coming to rest undamaged within a defined 50ft radius circle.

Teams must deliver sufficient habitats and water to support the growing population of their colony. For example, a team that delivers 4 habitats but no water will achieve a Days of Habitability of 0. Additional points are awarded for static payload.

Scoring Equation:

$$FFS = \ Final\ Flight\ Score\ = \frac{N_C \times D}{15N} + \frac{2S_p}{N}$$

Where:

$$D\ = Days\ of\ Habitability = 25\ \left(2^{1-\text{Maximum}\left(\frac{N_C}{8N_H},\frac{N_C}{N_W}\right)}\right)$$

$N = Total\ Number\ Rounds\ During\ the\ Competition$

$N_C\ = Total\ Number\ Colonists\ Delivered\ During\ the\ Competition$

$N_H\ = Total\ Number\ of\ Habitats\ Delivered\ During\ the\ Competition$

$N_W = Total\ Amount\ of\ Water\ (in\ fl\ oz)\ Delivered\ During\ the\ \ Competition$

$S_p = Total\ Static\ Payload\ (lbs)\ Delivered\ During\ the\ \ Competition$

Penalty Points

Any penalty points assessed during the competition are now deducted from a team's overall score.

9 MICRO CLASS DESIGN REQUIREMENTS

The objective of Micro Class is to design light-weight micro UAV style aircraft that can be quickly deployed from a small package and able to carry a large, unwieldy low density payload. For this year, we are basing scores on both the maximum performance of the aircraft and on overall performance throughout the event. The Micro Class assembly demo is now a mandatory timed event. Payload fraction is still a core element of the class.

9.1 AIRCRAFT SYSTEMS REQUIREMENTS

1. *Propulsion Requirements*
   Micro Class aircraft are restricted to electric motor propulsion only.
2. *Propeller and Gearbox*
   Gearboxes on a Micro Class aircraft where the propeller RPM differs from the motor RPM are allowed. Multiple motors, multiple propellers, propeller shrouds, and ducted fans are allowed in Micro Class.
3. *Aircraft Propulsion System Battery*
   Micro Class aircraft must use Lithium Polymer batteries. The maximum size propulsion system battery allowed for Micro Class is a 3 cell 2200mAh lithium polymer battery. Batteries having fewer cells and lower capacity are permitted.
4. *Gyroscopic Assist Allowed*
   Gyroscopic assist and other forms of stability augmentation are allowed in Micro Class.
5. *Aircraft Empty Weight Definition*
   An empty aircraft has completed a successful flight and has the payload removed for weigh-in. All aircraft parts that are not payload, as defined in 9.2, contribute to the empty aircraft weight, including, but not limited to: airframe, receiver, electronics, batteries, hardware, brackets, straps and other associated features.

9.2 PAYLOAD REQUIREMENTS

1. Micro Class aircraft shall use 2 inch diameter, Schedule 40, White Polyvinyl Chloride (PVC) pipe in accordance with ASTM D1785 as payload weights:
   1. Outer diameter: 2.375 inches
   2. Nominal Inner diameter: 2.000 inches
   3. Minimum Wall Thickness: 0.154 inches
   4. Inner diameter (Ref): 2.067 inches MAX
   5. Weight (Ref): 0.680 lbm. /ft.
   6. Color: White
2. When free from the aircraft the inner diameter of the payload pipes shall be free from obstruction.
3. Except being cut to length or having holes drilled in the sidewall of the pipe, the PVC pipes shall be unmodified.
4. Payload support structure for the PVC pipes is NOT included in the payload weight for scoring purposes.

Notes:

- There is no requirement for internal carriage of payload within the aircraft.
- There is no required carriage configuration for the payload.

9.3 MICRO CLASS AIRCRAFT LAUNCH

1. *Hand launched*

The Micro Class aircraft must be launched by hand using an overhand motion.

1. Only one (1) member of the team can enter pre - marked launch zone.
2. The pilot must be outside the pre - marked launch zone during the overhand launch.
3. The aircraft can only be launched by one (1) team member.
4. The aircraft cannot be launched by the pilot.
5. There is no limit on number of steps taken during the overhand launch but the person launching must remain in the launch zone before and after the overhand launch.

2. *Overhand launching violations*

The following actions are not permitted and will invalidate the flight attempt and score for the round

1. Overhand launching the aircraft from any other part of the aircraft other than the fuselage or payload attached to the fuselage
2. Running with the aircraft during launch.
3. Pilot or non-team member launching the aircraft.

9.4 MICRO CLASS AIRCRAFT HAND-LAUNCH SAFETY REQUIREMENTS

Safety gear must be used by the designated team member performing the aircraft toss and any team member assisting with preparing the aircraft inside the launch zone.

Safety gear will consist of:

- Safety glasses
- Hard hat

9.5 AIRCRAFT SYSTEM CONTAINER

1. *Aircraft System Container Requirements*

Micro Class aircraft will fit in an aircraft system container with size limitations. Compliance with the following requirements will be confirmed during technical inspection.

1. The aircraft system container shall be a cardboard box with absolute maximum outside dimensions of 12.125 inches X 3.625 inches X 13.875 inches maximum. Minimum wall thickness of container is 0.125 inches.
2. The fully packed aircraft system container must weigh no more than 10 pounds (lbs.).

3. The aircraft system container must have school name, school address, team name, and team number on a mailing label.

2. *Aircraft System Packaging General Requirements*

The aircraft container must contain the following:

1. The following items must be packaged within the constraints of the aircraft system container:
    - The airframe
    - Propulsion system battery
    - All (maximum) payload to be carried. Teams may not carry any payload that was not stored in the aircraft container.
    - All tools needed for the Assembly Demonstration, except a box-cutter or knife to cut packing tape
2. Any additional components not listed herein required for flight. The propulsion system battery must not be pre-installed in the aircraft.
3. The red arming plug must not be pre-installed in the aircraft.
4. The propulsion system battery must be contained in its own partitioned space in the aircraft system container.
5. The transmitter and any spare parts are not required to be in the aircraft system container.
6. The payload shall not be preinstalled to the aircraft.
7. If the aircraft uses a separate radio system battery, it may be pre-installed in its flight location. If the aircraft uses a radio system battery and a team elects not to pre-install it, then the radio system battery must be contained within its own partitioned space within the aircraft system container.

9.6 TIMED AIRCRAFT ASSEMBLY DEMONSTRATION

1. *Performance*

The Assembly Demonstration will be performed on Friday in the demonstration and inspection area. Timed assembly demonstrations will not be done at the flying site.

The assembly demonstration shall be accomplished under the following constraints:

1. The assembly of the aircraft shall be accomplished within 3 minutes. If 3 minutes is exceeded, then the demonstration will be halted.
2. The assembly demonstration score (AD) will be calculated per section 9.8.
3. The aircraft shall be completely packaged in the aircraft system container. The box will be taped closed with packing tape that will be provided at the assembly demonstration.
4. The assembly demonstration will consist of aircraft removal from the box and installation of the propulsion system battery at a minimum.
5. The use of any type of time-dependent curing adhesives such as glue, super-glue or epoxies are prohibited in the assembly demonstration.

6. Only the tools contained within the aircraft system container at the beginning of the assembly demonstration are permitted for use during the assembly process.
7. All payload carried in the aircraft system container will be installed on the aircraft before the assembly demonstration is considered complete.

2. *Process for Assembly Demonstration:*

1. Two team members tasked with assembling the aircraft will be located at a table in the Technical Inspection and Demonstration area. At this time the fully packaged, non-energized aircraft, with flight battery uninstalled will be in the container with the aircraft system container taped closed with the tape provided. Failure to have the Red Arming Plug removed or the battery uninstalled at this time will result in an assembly demonstration time of three minutes.
2. An official will give the "GO" command to begin the assembly demonstration. Assembly will start with the team opening the aircraft system container. Two officials will record the elapsed assembly time. Teams may use a knife to cut open the aircraft system container as described in rule 9.5.2.1.
3. The prop will be installed for the demo, but shall not be fully tightened to the motor for safety reasons.
4. When the aircraft is fully assembled, with all (maximum) payload from the container installed, with the flight battery installed, the team will give the "DONE" command to signal the officials to stop timing.
5. After the "DONE" command is given by the assembling team, no further assembly may occur.
6. The assembly demonstration is considered complete when all tasks required for flight have been performed with the exception of:
   - Installing the Red Arming Plug.
   - Performing preflight controls check.
7. The officials will inspect the aircraft to confirm aircraft flight ready status and record elapsed time.
8. The aircraft will then be powered up for a flight control and motor function check. If the aircraft checks out as fully ready for flight, the assembly demonstration time is awarded.
9. Disassembling the aircraft during the pre-flight and motor checks will invalidate any assembly time of less than 3 minutes.

9.7 Mission Requirements

1. *Time Limit for Aircraft Launch*

Micro Class aircraft should be assembled prior to entering the launch zone.

1. Each team will have 60 seconds to complete preflight checks, energize the propulsion system, and check the controls and hand-launch the aircraft.
2. Only one takeoff launch attempt is permitted per round.

2. *Aircraft Takeoff and Circuit*

Takeoff for Micro Class is defined as the point at which the aircraft departs the hand of the person throwing the aircraft. Once takeoff occurs, Micro Class aircraft are required to:

1. Remain airborne and fly past the designated turn point before turning approximately 180 degrees in heading.
2. Flying past a second designated turn point, turning 180 degrees in heading.
3. Land in the designated landing zone for Micro **Class**. Micro **Class** aircraft should be prepared to land on either a paved landing zone or an unpaved landing zone.
4. Takeoff direction will be determined by the Air Boss, and normally selected to face into the wind.

9.8 MICRO CLASS FLIGHT SCORING

In order to participate in the flight portion of the competition, each team is required to have submitted AND received a score for both Design Report and Oral Presentation.

Final Flight Score will be calculated in two parts: the average of all flight round scores and the maximum scored for any one flight round.

Scoring Equation:

$$Final\,Flight\,Score = FSS = 20 * [0.5 * (\frac{1}{N}\sum_{1}^{N} FS_n) + 0.5 * MAX(FS_n)] + AD$$

Where:

$$FS_n = Flight\ Score_n = \frac{W_{payload}}{\sqrt{W_{empty}}}$$

The Micro Class Assembly Demo score, either positive or negative, will be added to or deducted from the team's overall score, per rule 3.14.

Assembly Demonstration

$$AD = Assembly\ Demonstration = 5 * (2 - \frac{t}{60})^3$$

$$MAX = Team's\ maximum\ single\ flight\ round\ score$$

$$t = time\ recorded\ in\ seconds$$

$$N = total\ number\ of\ flight\ rounds\ during\ the\ competition$$

Penalty Points:

Any penalty points assessed during the competition will be deducted from the team's overall score.

Appendix B: Score Code

Contents


```
% Cummings, Heather 10/3/2018
clear variables; close all; clc;
```

Variables

```
W_empty   = 1;              %lbs
W_payload = 2.21;           %lbs
t         = 60;             %time in seconds
N         = 3;              %number of rounds completed
```

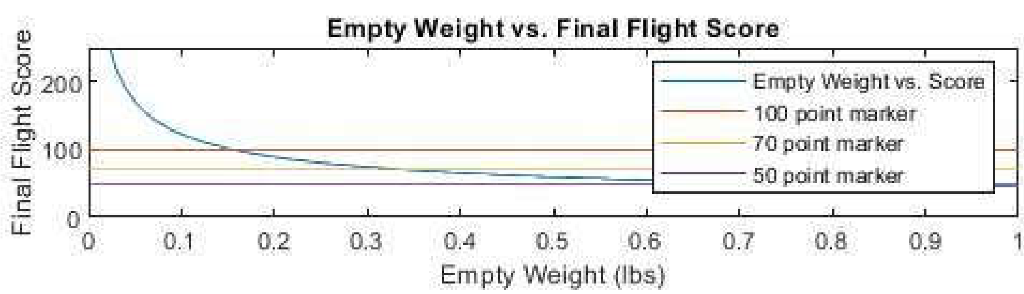
Equations

```
Assembly_Demonstration = 5*(2-(t/60))^3;
Flight_Score_n = W_payload / sqrt(W_empty);
    FS_1 = W_empty/sqrt(W_empty);
    FS_2 = W_payload / sqrt(W_empty);
    FS_3 = W_payload / sqrt(W_empty);
MAX_FS_n = W_payload / sqrt(W_empty);
Final_Flight_Score = 20*(0.5*(1/N*(FS_1+FS_2+FS_3))+0.5*MAX_FS_n)+Assembly_Demonstration;
disp('Expected Final Flight Score')
disp(Final_Flight_Score)
```

```
Expected Final Flight Score
   45.1667
```

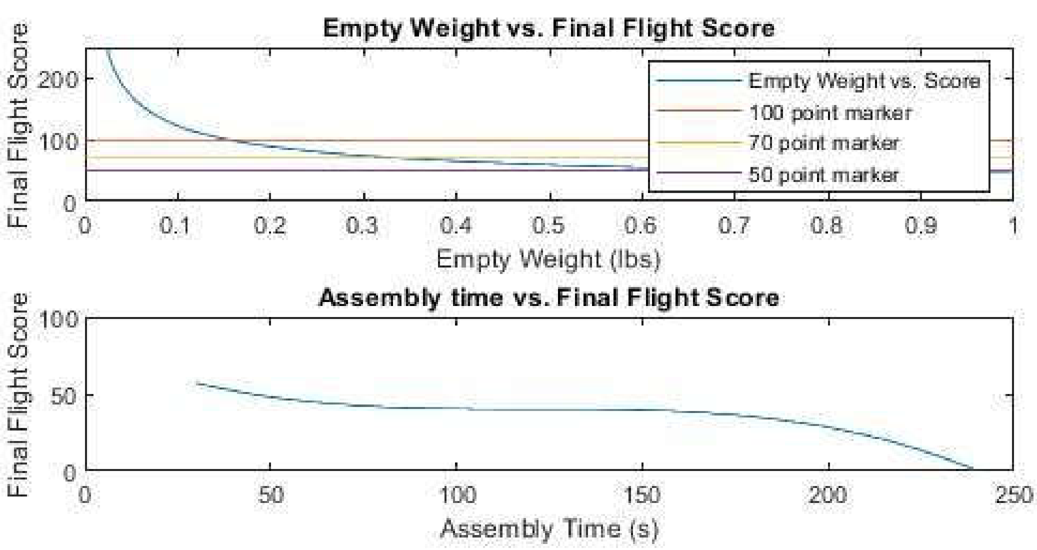
Trade Studies: Empty Weight Reduction

```
FSS = [];
W_empt = [];
for W_empty = 1:100

    W_empty = W_empty*.01;

    Assembly_Demonstration = 5*(2-(t/60))^3;
    Flight_Score_n = W_payload / sqrt(W_empty);
        FS_1 = W_empty/sqrt(W_empty);
        FS_2 = W_payload / sqrt(W_empty);
        FS_3 = W_payload / sqrt(W_empty);

    MAX_FS_n = W_payload / sqrt(W_empty);
    Final_Flight_Score = 20*(0.5*(1/N*(FS_1+FS_2+FS_3))+0.5*MAX_FS_n)+Assembly_Demonstration;

    FSS = [Final_Flight_Score; FSS];
    W_empt = [W_empty; W_empt];
end

subplot(3, 1, 1)
plot(W_empt, FSS); hold on;
title('Empty Weight vs. Final Flight Score')
xlabel('Empty Weight (lbs)')
ylabel('Final Flight Score')
ylim([0 250])
plot([0 1], [100 100]); hold on;
plot([0 1], [70 70]); hold on;
plot([0 1], [50 50]); hold off;
legend('Empty Weight vs. Score', '100 point marker', '70 point marker', '50 point marker')
```



Trade Studies: Assembly Demonstration

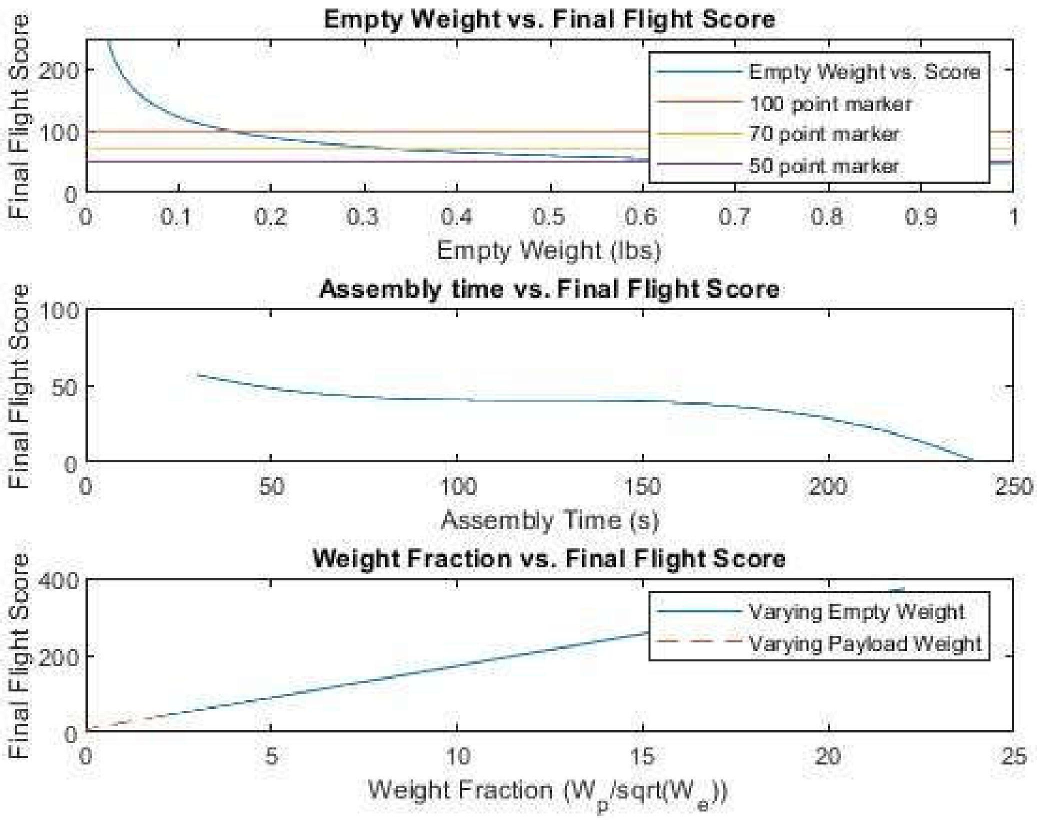
```
FSS_AD = [];
time = [];
for t = 30:360

    W_empty   = 1;    %lbs
    W_payload = 2.21; %lbs
    N         = 3;    %number of rounds completed

    Assembly_Demonstration = 5*(2-(t/60))^3;
    Flight_Score_n = W_payload / sqrt(W_empty);
        FS_1 = W_empty/sqrt(W_empty);
        FS_2 = W_payload / sqrt(W_empty);
        FS_3 = W_payload / sqrt(W_empty);
    MAX_FS_n = W_payload / sqrt(W_empty);
    Final_Flight_Score = 20*(0.5*(1/N*(FS_1+FS_2+FS_3))+0.5*MAX_FS_n)+Assembly_Demonstration;

    FSS_AD = [Final_Flight_Score; FSS_AD];
    time = [t; time];
end

subplot(3, 1, 2)
plot(time, FSS_AD)
ylim([0 100])
title('Assembly time vs. Final Flight Score')
xlabel('Assembly Time (s)')
ylabel('Final Flight Score')
```



Trade Studies: Varying both Weights to find convergence

```
Weight_Fraction1   = [];
Weight_Fraction2 = [];
FlightScore1   = [];
FlightScore2   = [];

for W_empty = 1:100
```

```
    W_empty    = W_empty*.01;
    W_payload = 2.21;
    Flight_Score_n = W_payload / sqrt(W_empty);

    t = 60;
    N = 3;   %number of rounds completed

    Assembly_Demonstration = 5*(2-(t/60))^3;
        FS_1 = W_empty/sqrt(W_empty);
        FS_2 = W_payload / sqrt(W_empty);
        FS_3 = W_payload / sqrt(W_empty);
    MAX_FS_n = W_payload / sqrt(W_empty);
    Final_Flight_Score = 20*(0.5*(1/N*(FS_1+FS_2+FS_3))+0.5*MAX_FS_n)+Assembly_Demonstration;

    Weight_Fraction1    = [Flight_Score_n; Weight_Fraction1];
    FlightScore1    = [Final_Flight_Score; FlightScore1];

end

subplot(3, 1, 3)
plot(Weight_Fraction1, FlightScore1, 'LineStyle', '-'); hold on;

for payload = 0:221

    payload = payload*.01;
    W_empty = 1;
    W_payload = payload;

    Flight_Score_n = W_payload / sqrt(W_empty);

    t = 60;
    N = 3;   %number of rounds completed

    Assembly_Demonstration = 5*(2-(t/60))^3;
        FS_1 = W_empty/sqrt(W_empty);
        FS_2 = W_payload / sqrt(W_empty);
        FS_3 = W_payload / sqrt(W_empty);
    MAX_FS_n = W_payload / sqrt(W_empty);
    Final_Flight_Score = 20*(0.5*(1/N*(FS_1+FS_2+FS_3))+0.5*MAX_FS_n)+Assembly_Demonstration;

    Weight_Fraction2    = [Flight_Score_n; Weight_Fraction2];
    FlightScore2    = [Final_Flight_Score; FlightScore2];

end

plot(Weight_Fraction2, FlightScore2, 'LineStyle', '--'); hold off;
title('Weight Fraction vs. Final Flight Score')
xlabel('Weight Fraction (W_p/sqrt(W_e))')
ylabel('Final Flight Score');
legend('Varying Empty Weight', 'Varying Payload Weight')
```



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Appendix C: Power Plant Overview

The power plant for this project consists of a brushless motor, propeller, electronic speed controller (ESC), battery, servos, transmitter, and receiver. The goal of this appendix is to explain how these individual components work together to power aircraft subsystems and generate propulsive force.

Two of the most important components of the aircraft are the receiver and the transmitter. A receiver controls an aircraft by wirelessly taking inputs from a transmitter and converting the inputs into signals. These signals are interpreted by other components onboard the aircraft and are used to power the motor or move control surfaces. A typical receiver has multiple channels associated with it. Once bound to the transmitter, these channels can be programed to receive inputs from the various control sticks and switches on the controller. Common channel inputs include throttle, left aileron, right aileron, rudder, elevator, flaps, retractable landing gear, control modes, and camera gimbal controls. Some receivers, like the FrSky S8R used in our aircraft, also have 3-axis gyroscope and 3-axis accelerometer sensors built in, so that it is possible to have stability augmentation through 3-axis stabilization. (FrSky S8R 8/16 Channel Receiver with 3-axis Stabilization, n.d.).

A major component of a power plant is the motor. This discussion considers a three-phase direct current (DC) brushless outrunner motor, which is what we used for our aircraft. The three-phase motor has two primary components that work together to spin the propeller, which are the rotor and the stator. The rotor is the outer part of the motor that is free to spin. There are permanent magnets attached to the inside of the rotor that are arranged with alternating poles. The stator is the stationary component inside the motor that has several arms. Each arm of the stator is wrapped with a wire coil called a winding. In a three-phase motor, there are three coils and therefore three independent windings. When charged, these windings create electromagnets. Each of the three independent windings can be individually charged and discharged. The electromagnets on the stator force the permanent magnets on the free rotating rotor to turn at high speeds. Alternating the frequency of charging and discharging varies both the revolutions per minute and power output of the motor.

(Zhao and Yu, 2011).

Despite having the same design concept, each motor is built differently and characterized by several values, such as the operating voltage range and the maximum current. The most important of these characteristics is the KV constant, which represents the unloaded revolutions per minute of the motor operating at 1V. Generally, a higher KV value results in lower torque. This means that a high KV motor can spin a small propeller quickly and a low KV motor can spin a large propeller slowly.

(Understanding Kv Ratings, 2017).

An ESC is used to modulate the charging of the three-phase brushless motor. The function of the ESC is to convert the DC power output of the battery into Pulse Width Modulated (PWM) signals based on the signal input of the receiver. In order to create the PWM signal, the ESC opens and closes switch gates at a frequency determined by the pilot through the receiver. These signals are sent to the three phases of the motor and control the rotation of it. (Zhao and Yu, 2011). It is important to note that each ESC, like the motor, is rated for a specific maximum current and voltage range. With this in mind, when picking an ESC, the maximum current and voltage that an ESC can handle should be greater than or equal to the motor limits. In addition to regulating motor speeds, most ESCs have a built in battery elimination circuit (BEC). The BEC steps down the battery voltage in order to power the receiver and other flight controls such as servos through the signal input wires, which eliminates the need for an independent battery pack. (What are ESC, UBEC and BEC, 2016).

The propeller is attached to the motor shaft and converts the motor’s rotational energy into thrust by using rotating blades. Similar to a wing or tail airfoil, the propeller blades have an angle of attack with respect to the air they rotate through. However, the propeller blades have a variable angle of attack, or twist, which is larger at the center and gradually becomes smaller out to the tips of the blades. This is because the tips of the propeller blades have a much faster velocity than the center of the propeller. By having a twist in the propeller blade, the propeller can produce an equal amount of thrust along the entire blade. Propellers typically have two quantifying characteristics associated with them, which are the diameter and pitch. The diameter measured as the diameter of the area swept out by the propeller as it spins. The pitch is the measurement of how far the propeller moves forward with each revolution. These are typically measured in inches and are noted in the sizing of the propeller. For example, this project considered both 9x5 and 8x6 propellers, which correspond to a 9 inch diameter, 5 inch pitch and 8 inch diameter, 6 inch pitch, respectively. (RC Airplane Propeller Size Guide, n.d.). Optimizing the diameter and pitch of the propeller will ensure that an aircraft has both the thrust and acceleration needed. Typically, for fast accelerations and steep climbs, a lower pitch and large diameter propeller are optimal. For high-speed aircraft, higher pitch and smaller diameter are optimal. (How does prop size/pitch effect a plane?, n.d.).

Another component is the battery, which provides all the electrical power to the various aircraft subsystems. For this discussion we will be only considering lithium polymer (LiPo) batteries. LiPos are characterized by three parameters that describe their power output. The first is the cell count of the battery pack. Each cell has a nominal voltage of 3.7V, a fully charged voltage of 4.2V, and a safe discharge voltage of 3.0V. A typical battery pack consists of individual cells wired in series such that the voltage adds, which means a 3-cell LiPo has a nominal voltage of 11.1V. The second characteristic is the capacity, which is expressed in milliampere hours (mAh). This value is used to determine the battery life given the current pull of the motor and other electrical equipment on board. Battery life is determined by dividing the capacity by the total current pull. For example, if a battery has a 1000mAh capacity, then it can run for one hour with a current pull of 1000mA, or 1A. The last parameter is the discharge constant, or C-value. Multiplying the C-value by the capacity value describes how much current can be safely discharged from the battery at any given time. For example, a C-value of 40 in a 1000mAh battery means a safe discharge 40A at a given time (Instructables, 2017). LiPo batteries have two output connections, the discharge connector and the balance plug. The discharge plug is connected to the ESC and powers the plane. The balance plug is used in conjunction with the discharge plug while charging to ensure that each cell in the LiPo is charged equally. When working with LiPos it is especially important to ensure the battery is not over-worked. This means to not over-discharge the battery or exceed its operation limits, or it may catch fire.

As mentioned earlier, the receiver also has channels to control various control surfaces. These controls are governed by servomotors (servos). Servos rotate small lever arms based on input signals. The input signals that are sent to servos from the receiver are converted into a PWM signals. The duration of the pulse determines the final orientation of the servo arm. This conversion to a PWM signal is done through a control circuit built into the servo. The speed at which the arm travels is also determined by the control circuit through the use of a potentiometer. As the servo arm gets closer to the desired position, the speed of rotation decreases. Once the servo arm has reached its desired position, the servo will remain in this position and resist forces applied to it. The force the servo is able to withstand is given in terms of a torque. A servo will fail if the force on it produces a torque greater than the maximum torque it is able to withstand (Jameco, n.d.). Attaching servo arms to the control surfaces of an aircraft allows the pilot to modify their deflection angles by inputting them into the transmitter.

Appendix D: Takeoff script

```
import numpy as np
import math
import matplotlib.pyplot as plt
import pandas as pd

# Initializing plane parameters

h_cg = 0.3636 # CG
hac = .25 # aerodynamic center
a = 4.796 # wing lift slope
at = 3.625 # tail lift slope
ae = 3 # elevator effectiveness
S = 1.986 # wing area
St = .25833 # tail area
lt = 15.15/12 # distance between aerodynamic centers
c = 5.5/12 # wing chord
de_dalpha = .1919 # downwash derivative

it = -6.77*np.pi/180 # tail incidence
e0 = 0 # initial downwash

alpha_stall = 20 # stall angle, degrees
Cmac = -0.23124 # natural wing moment
Clow = .7401 # wing lift at 0 alpha

W = 4.2 # weight
rho = 0.0023769 # air density
g = 32.26 # gravity accel
m = W/g # mass
Cd0 = .015 # Drag coeff
AR = 9.45 # Aspect Ratio
K = 4/3*1/(math.pi*.9*AR) # induced drag coeff
Jyy = 10 # moment of inertia

def stability_derivs(h, hac, a, at, ae,S, St, l, c, de_dalpha, Cmac, Clow,
it, e0):
    '''
    Computes static stability derivatives for aircraft
    :param h: chord-fraction position of CG
    :param hac: chord-fraction position of aerodynamic center
    :param a: wing lift slope
    :param at: tail lift slope
    :param ae: elevator effectiveness
```

```
    :param S: wing area
    :param St: tail area
    :param l: distance from tail to wing aerodynamic center
    :param c: wing mean chord
    :param de_dalpha: downwash derivative
    :param Cmac: natural wing moment
    :param Clow: lift at 0 alpha
    :param it: tail incidence angle
    :param e0: initial downwash
    :return: Stability derivatives CL_0, CL_alpha, CL_delta, Cm_0, Cm_alpha,
Cm_delta
    '''
    Vh = l/c * St/S
    Cl_0 = Clow + St/S*at*(it-e0)
    Cl_alpha = a + at*(1 - de_dalpha)*St/S
    Cl_delta = St/S * ae
    Cm_0 = Cmac + Clow*(h-hac) - at*(it-e0)*Vh * (1 - (h - hac)*c/l)
    Cm_alpha = (a + at * St/S * (1 - de_dalpha))*(h-hac) - at*Vh*(1-
de_dalpha)
    Cm_delta = Cl_delta *(h-hac) - ae*Vh
    return [Cl_0, Cl_alpha, Cl_delta, Cm_0, Cm_alpha, Cm_delta]

def launch_dot(t, state, params):
    '''

    :param t: time
    :param state: 2 position values, 2 velocity values, 2 angular values
    :param params: additional parameters for state transition
    :return:
    '''
    # State Variables
    x = state[0] # x position, ground-fixed
    z = state[1] # z position, ground-fixed
    thta = state[2] # flight path angle
    u = state[3] # x velocity, body fixed
    w = state[4] # z velocity, body fixed
    q = state[5] # angular velocity of flight path angle, body fixed


    # Stability derivatives
    stability_vals = stability_derivs(h_cg, hac, a, at, ae, S, St, lt, c,
de_dalpha, Cmac, Clow, it, e0)
    Cl_0 = stability_vals[0]
    Cl_alpha = stability_vals[1]
    Cl_delta = stability_vals[2]
```

```
    Cm_0 = stability_vals[3]
    Cm_alpha = stability_vals[4]
    Cm_delta = stability_vals[5]

    # other params
    m = params[0] # mass
    rho = params[1] # air density
    Cd0 = params[2] # constant drag term
    K = params[3] # Induced drag coeff
    Jy = params[4] # moment of inertia
    delta = params[5]

    # Thrust interpolation
    V = math.sqrt(u**2 + w**2)
    T = 1.907 - .01105*V # linear fit from MotoCalc data, regression of ~.98
    alpha = math.atan(w/u) # calculate angle of attack to make sure it does
not exceed stall

    # Force Calcs
    Cl_total = Cl_0 + Cl_alpha * alpha + Cl_delta * delta
    Cm_total = Cm_0 + Cm_alpha * alpha + Cm_delta * delta
    Cd_total = Cd0 + K*Cl_total**2+.08

    L = 1/2*rho*u**2 * S * Cl_total
    M = 1/2*rho*u**2 * S * c * Cm_total
    D = 1/2*rho*u**2 * S * Cd_total

    # summing aerodynamic force
    X = T - D
    Z = -L

    # State derivative
    xdot = u*math.cos(thta) + w*math.sin(thta)
    zdot = -u*math.sin(thta) + w*math.cos(thta)
    thtadot = q
    udot = X/m - g*math.sin(thta) - q*w
    wdot = Z/m + g*math.cos(thta) + q*u
    qdot = M/Jy

    return np.array([xdot, zdot, thtadot, udot, wdot, qdot])


# Script
delta = 0*np.pi/180
```

```
# Passing extra parameters to the ODE
params = [m, rho, Cd0, K, Jyy, delta]

# Cruise speed, where L=W at 0 alpha
v_cruise = math.sqrt(W/(1/2*rho*Clow*S))


# launch velocity and angle
v_launch = 20*1.467
thta0=10*math.pi/180

# initial state
state0 = [0, -6, thta0, v_launch, 0, 0]
# initialize data frame to hold values each loop
state_sim = pd.DataFrame(columns=['x(t)',
'z(t)','theta(t)','u(t)','w(t)','q(t)'])
state = state0
state_sim = state_sim.append({'x(t)':state[0], 'z(t)': -state[1],
'theta(t)': state[2]*180/math.pi,
                              'u(t)': state[3], 'w(t)':state[4], 'q(t)':
state[5]}, ignore_index=True)
dt = 0.01
ts = np.arange(0, 8, dt)
# RK4 integration scheme
for i in range(1, len(ts)):
    t = ts[i]
    k1 = launch_dot(t, state, params)
    k2 = launch_dot(t+ .5*dt, state + .5*dt*k1, params)
    k3 = launch_dot(t + .5*dt, state + .5*dt*k2, params)
    k4 = launch_dot(t + dt, state + dt*k3, params)

    state = state + (dt/6)*(k1 + 2*k2 + 2*k3 + k4)
    state_sim = state_sim.append({'x(t)': state[0], 'z(t)': -state[1],
'theta(t)': state[2]*180/math.pi,
                                  'u(t)': state[3], 'w(t)': state[4],
'q(t)': state[5]}, ignore_index=True)

# calculate airspeed and alpha as function of time
V_sim = np.sqrt(state_sim['u(t)'].values**2 + state_sim['w(t)'].values**2)
alpha_sim = np.arctan(state_sim['w(t)']/state_sim['u(t)'])*180/math.pi

# plots
plt.figure(figsize=(10,8))
plt.subplot(3,1,1)
plt.title('Launch Simulation', fontsize=22)
```

```python
plt.plot(ts, state_sim['z(t)'], lw=2)
plt.plot([0, ts[-1]], [0, 0], 'k--')
plt.grid()
plt.ylabel('Height (ft)',fontsize=20)

plt.subplot(3,1,2)
plt.plot(ts, V_sim)
plt.plot([0, ts[-1]], [v_cruise, v_cruise], 'k--', label='Cruise Speed')
plt.legend(fontsize=16)
plt.grid()
plt.ylabel('Airspeed (ft/s)',fontsize=20)

plt.subplot(3,1,3)
plt.plot(ts, alpha_sim)
plt.ylabel(r'$\alpha$ ($\circ$)',fontsize=20)
plt.plot([0, ts[-1]], [alpha_stall, alpha_stall],'k--', label=r'Stall
$\alpha$')
plt.legend(fontsize=16)
plt.xlabel('time (s)',fontsize=20)
plt.grid()

plt.show()
```

Appendix E: Static Stability MATLAB Code

Contents


```
% Cummings, Heather 9/6/2018
clear variables; close all; clc;
```

Input Variables

Based on aircraft geometry

```
AR_w     = 9.0625;        % nondimensional, aspect ratio of wing
AR_t     = 10/6;          % nondimensional, aspect ratio of horizontal tail
l_bar_t = 13/12;          % ft, distance between ac of wing and ac of tail
b        = 4.927;         % ft, wing span
c        = 5.5/12;        % ft, wing chord
S        = 312/144;       % ft^2, wing area
S_t      = 10*6/144;      % ft^2, horizonatl tail area
h_ac     = 0.25;          % nondmensional, ac locaton as fraction of chord
h        = 0.26;          % nondmensional, cg location as fraction of chord
M        = 0.027;

% Coefficients and factors
C_l0w       = 1.074;
C_lt_1      = 0.2;
C_lt_2      = 0.25;
C_lw_1      = 1.52;
C_lw_2      = 1.82;
e           = 0.9;

% Angle based variables
w_alpha1 = 0.0872665; % 5 degrees, angle of attack
w_alpha2 = 0.174533;  % 10 degrees, angle of attack
t_alpha1 = 0.01745;
t_alpha2 = 0.139626;
lambda    = 1;
Lambda    = 0;
```

Functions to run

look into the lifting of a flat plate to make sure you get the right a_t

```
[a_0t, a_0w, a_t, a_w] = Lift_Curve_Slope(AR_w, AR_t, w_alpha1, w_alpha2, t_alpha1, t_alpha2,
 C_lt_1, C_lt_2, C_lw_1, C_lw_2, e);
a_t = 4;
[h_NP, Static_Margin] = Static_Margin(l_bar_t, b, c, S, S_t, h_ac, h, AR_w, a_t, a_w, Lambda,
 lambda, M)
```

```
function [a_0t, a_0w, a_t, a_w] = Lift_Curve_Slope(AR_w, AR_t, w_alpha1, w_alpha2, t_alpha1,
t_alpha2, C_lt_1, C_lt_2, C_lw_1, C_lw_2, e)
% Cummings, Heather 9/5/2018
% This function calculates the lift curve slope for the tail and wing of a
% micro air vehicle
%{
    a_t      = lift curve slope for tail
    a_w      = lift curve slope for wing
    AR_w     = aspect ratio of wing
    AR_t     = aspect ratio of tail
    w_alpha1 = angle of attack of wing 1
    w_alpha2 = angle of attack of wing 2
    t_alpha1 = angle of attack of tail 1
    t_alpha2 = angle of attack of tail 2
    C_lw_1   = coefficient of lift for angle of attack of wing 1
    C_lw_2   = coefficient of lift for angle of attack of wing 2
    C_lt_1   = coefficient of lift for angle of attack of tail 1
    C_lt_2   = coefficient of lift for angle of attack of tail 1
    e        = span efficiency factor, typically between 0.85 and 0.95,
               (Oswald Efficiency Factor)
%}

%%%%%%% Caluclating Delta_Alpha for the wing and tail %%%%%%%
deltalph_t = t_alpha2 - t_alpha1; % angles must be in radians
deltalph_w = w_alpha2 - w_alpha1; % angles must be in radians

%%%%%%% Caluclating Delta_C_l for the wing and tail %%%%%%%
DeltC_lt = C_lt_2 - C_lt_1;
DeltC_lw = C_lw_2 - C_lw_1;

%%%%%%% Caluclating a_0 for wing and tail %%%%%%%
a_0t = DeltC_lt / deltalph_t;
a_0w = DeltC_lw / deltalph_w;

%%%%%%% Transforming to a Finite wing %%%%%%%
if AR_t <= 4 % aspect ratio of tail
    a_t = a_0t / (sqrt(1 + (a_0t / (pi*e*AR_t))^2) + (a_0t / (pi*e*AR_t)));
elseif AR_t > 4
    a_t = a_0t / (1 + (a_0t / (pi*e*AR_t)));
end

if AR_w <= 4 % aspect ratio of wing
    a_w = a_0w / (sqrt(1 + (a_0w / (pi*e*AR_w))^2) + (a_0w / (pi*e*AR_w)));
elseif AR_w > 4
    a_w = a_0w / (1 + (a_0w / (pi*e*AR_w)));
end

end
```

```matlab
function [h_NP, Static_Margin] = Static_Margin(l_bar_t, b, c, S, S_t, h_ac, h, AR_w, a_t, a_w
, Lambda, lambda, M)
% Cummings, Heather 9/5/2018
% The following script calculates the Neutral point and static margin for
% micro air vehicle designs
%{
    V_h         = horizontal tail volume ratio
    l_bar_t     = distance between aerodynamic centers of wing and tail
    c           = chord length
    S           = wing area
    S_t         = tail area
    AR_w        = Aspect Ratio of hte wing
    C_l0w       = coefficient of lift at 0 angle of attack
    eps_0       = downwash angle for 0 angle of attack
    i_w         = incidence angle of wing
    i_t         = incidence angle of tail
    C_lalph     = coefficient of lift "stability derivative"
    deps_dalph = proportionality constant
    a_t         = tail lift curve slope
    a_w         = tail lift curve slope
    h_ac        = location of aerodynamic center as a fraction of the chord length
    h           = location of the CG as a fraction of chord length
    h_NP        = location of the neutral point as a fraction of the chord length

%}

%%%%%%% Calculating Horizontal Tail Volume Ratio %%%%%%%
V_h = (l_bar_t / c )*(S_t / S)

%%%%%%% Calculating Coefficient of Lift Stability Derivative %%%%%%%
deps_dalph = 4.44*sqrt(1-M^2)*((((1/AR_w)-(1/(1+AR_w^1.7)))*((10-3*lambda)/7)*((1-(l_bar_t/b)
)/(2*l_bar_t/b)^0.33))*sqrt(cos(Lambda)))^1.19

C_lalph = a_w + a_t * (S_t / S)*(1 - (deps_dalph))

%%%%%%% Calculating the nondimensional location of the neutral point %%%%%%%
h_NP = h_ac + V_h * (a_t / C_lalph) * (1 - (deps_dalph));
% achieved by setting C_malpha to 0

%%%%%%% Calculating Static Margin %%%%%%%
Static_Margin = h_NP - h;

end
```

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Appendix F: Dynamic Stability Equations

The following equations were used to calculate the Longitudinal stability derivatives for the SAE Aero Design West 2019 Micro Aerial Vehicle.

List of Variables:

b $\quad wingspan, ft$

c $\quad wing\ chord, ft$

S $\quad wing\ area, ft^2$

S_t $\quad horizontal\ tail\ area, ft^2$

AR_w $\quad aspect\ ratio\ of\ wing$

AR_t $\quad aspect\ ratio\ of\ tail$

W $\quad weight\ of\ aircraft\ with\ payload, lbs$

h $\quad location\ of\ center\ of\ gravity\ as\ a\ fraction\ of\ chord\ length$

h_{ac} $\quad location\ of\ aerodynamic\ center\ of\ wing\ as\ a\ fraction\ of\ chord\ length$

$\underline{l_t}$ $\quad distance\ between\ aerodynamic\ centers\ of\ wing\ and\ horizontal\ tail, ft$

e $\quad oswald\ efficiency\ factor$

I_y $\quad y - direction\ moment\ of\ inertia, slug\ ft^2$

C_{D0} $\quad induced\ coefficient\ of\ drag$

M $\quad mach\ number\ for\ cruise\ condition$

Λ $\quad wing\ sweep\ angle$

ρ $\quad air\ density\ at\ sea\ level, slug/ft^3$

u_0 $\quad velocity, ft/s$

η_H $\quad dynamic\ pressure\ ratio$

V_H $\quad horizontal\ tail\ volume\ ratio$

a_t $\quad lift\ curve\ slope\ of\ tail$

a_w $\quad lift\ curve\ slope\ of\ wing$

Lift and Drag Stability Derivatives

$$a = a_w(1 + \frac{a_t}{a_w}\frac{S_t}{S}(1 - \frac{d\varepsilon}{d\alpha}))$$

$$C_{L\,trim} = C_{W0} = W\ /\ (1/2\,\rho\,u_0{}^2 S)$$

$$C_{Lu} = \frac{M^2}{1 - M^2} C_{L\,trim}$$

$$C_{L\,alpha} = a$$

$$C_{L\,alpha\quad dot} = 2a_t\eta_H V_H(1 - \frac{d\varepsilon}{d\alpha})$$

$$C_{Lq} = \frac{AR_w + 2cos(\Lambda)}{2cos(\Lambda) + AR_w\sqrt{1 - (Mcos(\Lambda))^2}}(1/2\ +\ 2\frac{l_t}{c})a_w + 2a_t\eta_H V_H$$

$$C_{D\ trim} = C_{D0} + \frac{C_{L\ trim}{}^2}{\pi AR_W e}$$
$$C_{Du} = 0$$
$$C_{D\ alpha} = \frac{2C_{L\ trim}}{\pi AR_w e} a$$

Non-Dimensional Stability Derivatives

$$C_{Xu} = -(C_{Du} + 2C_{D\ trim})$$
$$C_{X\ alpha} = -(C_{D\ alpha} - C_{L\ trim})$$
$$C_{Xq} = 0$$
$$C_{X\ alpha\ dot} = 0$$

$$C_{Zu} = -(C_{Lu} + 2C_{L\ trim})$$
$$C_{Z\ alpha} = -(C_{L\ alpha} + C_{D\ trim})$$
$$C_{Zq} = -C_{Lq}$$
$$C_{Z\ alpha\ dot} = -C_{L\ alpha\ dot}$$

$$C_{Mu} = 0$$
$$C_{M\ alpha} = a(h - h_{ac}) - a_t V_H (1 - \frac{d\varepsilon}{d\alpha})$$
$$C_{M\ alpha\ dot} = -2a_t \eta_H \frac{V_H \underline{\underline{l_t}}}{c} \frac{d\varepsilon}{d\alpha}$$
$$C_{Mq} = -2a_t \eta_H \frac{V_H \underline{\underline{l_t}}}{c}$$

Dimensional Stability derivatives

$$X_u = \rho u_0 S C_{W0} sin(\theta_0) + 0.5\rho u_0 S C_{Xu}$$
$$X_w = 0.5\rho u_0 S C_{X\ alpha}$$
$$X_q = 0.25\rho u_0 S c C_{Xq}$$
$$X_{w\ dot} = 0.25\rho S c C_{X\ alpha\ dot}$$

$$Z_u = -\rho u_0 S C_{W0} cos(\theta_0) + 0.5\rho u_0 S C_{Zu}$$
$$Z_w = 0.5\rho u_0 S C_{Z\ alpha}$$
$$Z_q = 0.25\rho u_0 S c C_{Zq}$$
$$Z_{w\ dot} = 0.25\rho S c C_{Z\ alpha\ dot}$$

$$M_u = 0.5\rho u_0 S c C_{Mu}$$
$$M_w = 0.5\rho u_0 S c C_{M\ alpha}$$

$$M_q = 0.25\rho u_0 S c^2 C_{Mq}$$
$$M_{w\ dot} = 0.25\rho S c^2 C_{M\ alpha\ dot}$$

Creation of the Longitudinal A matrix from the Stability Derivatives

X_u/m	X_w/m	0	$-g\cos(\theta_0)$
$Z_u/(m - Z_{w\ dot})$	$Z_w/(m - Z_{w\ dot})$	$(Z_q + mu_0)/(m - Z_{w\ dot})$	$(-mg\sin(\theta_0))/(m - Z_{w\ dot})$
$(1/I_y)(M_u + \frac{M_{w\ dot} Z_u}{m - Z_{w\ dot}})$	$(1/I_y)(M_w + \frac{M_{w\ dot} Z_w}{m - Z_{w\ dot}})$	$(1/I_y)(M_q + \frac{M_{w\ dot}(Z_q + mu_0)}{m - Z_{w\ dot}})$	$(-M_{w\ dot} mg\sin(\theta_0))/(I_y(m - Z_{w\ dot}))$
0	0	1	0

Appendix G: Wind Tunnel Operations

Use of Wind Tunnel:

A wind tunnel is a device used to simulate high speed airflow conditions in a non-moving device. It was used by our team to simulate flight conditions in order to test the aerodynamics of our MAV and the thrust capabilities of our motor at various speeds. For the purposes of this document, the wind tunnel instructions are for a closed-loop, water-cooled wind tunnel. The below instructions and safety precautions should not be used in itself. Refer to the user manual for the specific wind tunnel being used. This document is to supplement that material.

Procedures:

1. Properly secure any testing device and data collecting equipment inside the testing section of the windtunnel. Any connections must be able to withstand the wind velocity of intended use.
2. Close all access points to the test section
3. Turn on the wind tunnel power supply and water supply.
4. Open the compressed air valve and ensure an air pressure of 30 psi.
5. Open the water valves and ensure water is flowing.
6. Turn on the temperature gage located on the control panel and set an internal temperature of the windtunnel. Since the wind tunnel is closed loop, over time the wind friction will cause it to heat up. The water cooling will allow the windtunnel temperature to stabilize at a preset temperature. Note there are limits to this, water cooling can only cool down the tunnel to the water temperature and can only heat up the tunnel through air friction.
7. Select the remote button. Ensure a green light.
8. Select the frequency associated with the wind speed desired. Note, if a high wind speed is desired, it is best to ramp up the wind speed while the wind tunnel is running instead of starting at the high tunnel speed.
9. Hit the run button on the control panel. Wait for the airspeed to stabilize.
10. Repeat steps 7 and 8 for each desired airspeed and test.
11. Once finished test, hit the run button again to stop the windtunnel.
12. Turn off the temperature gage, turn off the wind tunnel power supply and water supply, turn off the compressed air, and shut off the water valves.
13. Remove testing equipment from the windtunnel.

Safety Precautions:

- Never open the tunnel test section while the wind tunnel is in use.
- In the event of any vibrations from the testing equipment, immediately shut down the tunnel.
- In the event that a piece of equipment becomes loose inside the tunnel, immediately shut down the tunnel. Do not attempt to restart the tunnel until all loose debris is cleared.

Appendix H: RC Benchmark Test Stand

Use of Test Stand:
The RC Benchmark Thrust Test Stand 1520 Series is a strain gage based thrust stand for small electric motors producing thrust less than 5 kilograms of force. In addition to measuring thrust, the thrust stand can also measures current, voltage, elapsed time and can test the movement of up to three servos. The below instructions and safety precautions should not be used in itself. Refer to the user manual for the thrust stand and all electrical components. This document is to supplement those materials.

Procedures:
1. Download the RC benchmark testing software from https://docs.rcbenchmark.com/en/dynamometer/software/dynamometer-software-download.html on to your local desktop or laptop.
2. Properly secure the thrust stand to an enclosed surface or windtunnel such that it will not come loose once the motor is powered.
3. Securely mount the motor to the stand using the mounting screws that came with the motor. Ensure that no moving part of the motor is in contact with the testing stand.
4. Securely attach the propeller to the motor.
5. Connect each of the three pole leads to the ESC such that the motor will spin in the appropriate direction once powered. Connect the signal wire from the esc to the esc signal pins on the thrust stand ensuring to not flip the polarity of the esc signal wire. Connect the power leads for the esc to power wires coming from the thrust test stand.
6. If testing servos attach the signal wires from the servos to the servo signal pins on the thrust stand.
7. Connect the USB cable from the test stand to your PC.
8. Ensure that no wires, lose items, or objects will obstruct the propeller once spinning.
9. Connect the battery to the battery leads coming from the test stand. Note that the motor is now getting power. Do not touch the motor or the test stand unless the battery is first removed. Safety glasses and other safety equipment should be worn from this point onwards.
10. Open up the RC benchmark application. On the left side of the screen, select the computer port that the stand is connected to from the drop down menu and click connect.
11. Next, go to the setup menu. In this menu, select the units for thrust and motor speed. If necessary, update the number of poles for the motor. In addition, change the working directory to the location you wish to save test data to. If you wish to use an automated script created to run your test, check the activate experimental scripting mode box.
12. In the utilities menu, if you haven’t already, calibrate the program to your specific thrust stand.
13. In the safety cutoffs menu, update the cutoff values based on your motor, speed controller, and battery. These numbers can be found in the motor, battery and esc

manuals. It is important to set these correctly as in case of exceeding these limits the motor will shut off. If this is done incorrectly, you risk damaging the test stand and or your components.

14. To run the test, you can either create an automated script or run it manually with the slide bars in the manual control tab. Make sure before you start the test that you select the continuous record button to begin the recoding.
15. If during the test one of the safety cutoffs is reached, follow the pop up instructions to reset.
16. Once finished the test, remove the battery prior to touching the test stand. Then remove all electrical components and put away the stand.

Safety Precautions:

- Ensure that components are wired properly and can handle the appropriate amount of current and voltage.
- Ensure that the stand is properly secured and that there is no lose objects around the stand prior to start up.
- In the event of a current overload, voltage overload, or any of the cutoff limits exceeded, immediately power down the motor.
- In the event of vibrations, sparks, motor and or propeller coming loose, or electrical fire, immediately power off both the motor and the stand. In case of fire put out any flames.
- Wear protective equipment such as hard hats and safety glasses while in use.

Appendix I: Glide Test Procedure

Introduction:
The purpose of the glide test is to ensure both static and dynamic stability of the aircraft for future modifications and powered flight testing. The test will be broken into four sections: short hop empty, empty long hop, short hop weighted, and weighted long hop. The goal will be to test the configurations two times each, assuming the aircraft does not break.
Materials:
1. Aircraft
2. Camera
3. Sheet
4. Weights
5. 4 sections of PVC
6. Hard hats
7. Safety glasses

Procedure:
Short hop (height minimal throw):
1. Assemble aircraft
2. Connection check
    a. Back and front rods
    b. Tail
    c. Wings
3. Optional shake test
4. Deploy sheet with at least four people
5. Practice short throws into blanket
    a. Get proper motion of throw
6. Prepare arm launch, ready up with catchers
7. Throw aircraft
8. Catch it using sheet
9. Examine results from camera and visual testing
10. Repeat if not broken

Long hop:
Same procedure as short hop, with the aircraft being thrown from a higher altitude
Weighted throws:
1. Disassemble aircraft
2. Put PVC on
    a. Two pieces of body PVC first
    b. Then four for 5th and 6th rounds respectively
3. Latch and mate all components like before
4. Shake test
5. Follow short and long hop test procedures