AIAA/Cessna/Raytheon
Design/Build/Fly
2010-2011

AIR HERCULES

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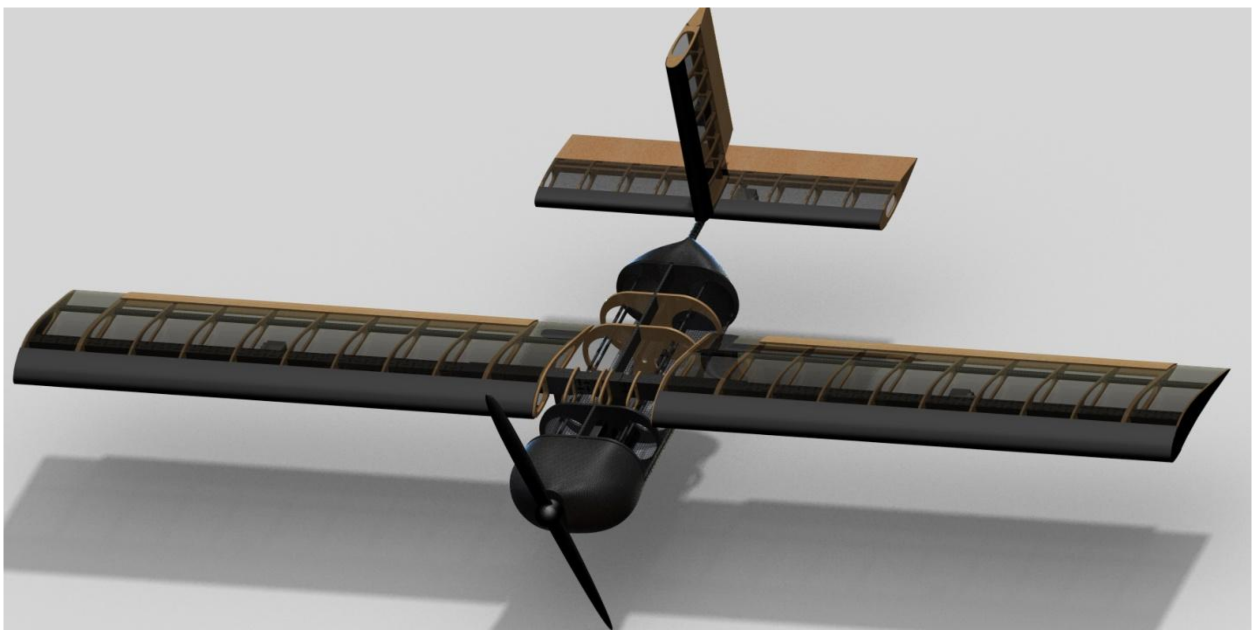

Aircraft Design Report

Florida A&M University - Florida State University
College of Engineering

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1.0 EXECUTIVE SUMMARY

This report provides a detailed overview and elaborate explanation of the design, testing, and fabrication methods employed by the FAMU-FSU College of Engineering AirHERCULES team in order to achieve the highest score for the 2010-2011 AIAA/Cessna Aircraft Company/Raytheon Missile Systems - Student Design/Build/Fly Competition. The title of this year's competition is called "Soldier Portable UAV" and all aircraft parameters are designed to satisfy the mission profile while optimizing the score obtained in each of the three missions. This competition is unique to previous years due the requirement of a hand launched vehicle and the carry-on suitcase packaging constraint of specified total dimensions.

The first mission is a "dash to critical target" and is purely a measure of speed, maneuverability, and endurance. The aircraft must traverse as many laps as possible within a four minute time period and the score is relative to other team's performances. The aircraft was designed to be lightweight, very maneuverable, and possess a high top speed while maintaining structural and flight integrity. This was achieved by utilizing high strength-to-weight ratio materials such as carbon fiber and Sitka Spruce and designing the structure shape to better withstand encountered loads.

Mission two is titled "Ammo Re-Supply" and requires the aircraft to carry a steel bar payload internal to the aircraft mold lines. The aircraft must complete three laps and successfully land to achieve a score for this mission. The score is dependent on the payload-to-weight ratio and is dependent on the payload-to-weight ratio. The steel bar has some minimum size constraints but the weight is decided upon by the team. The aircraft was designed to satisfy this mission by utilizing carbon fiber bulkheads to provide lightweight structure to the fuselage while also constraining the steel bar in the payload bay. The propulsion system was selected based on optimal recovery from hand launch in which the aircraft quickly attains stall speed. A large thrust to weight ratio was selected as the optimal solution along with a high capacity battery for endurance. Based on several iterations to maximize potential score, the optimal payload-to-weight ratio is 1.5.

Mission three is similar to mission two in that the aircraft must complete three laps while carrying a payload. The payload in this case consists of golf balls and the number of balls carried is decided by the team. This mission demands for high volume and weight capacity so an efficient packing method must be employed to reduce drag. Another method to reduce drag is utilizing a nose and rear fairing and forgoing the landing gear. The number of golf balls selected by the team was 16 based on several scoring analysis iterations.

2.0 MANAGEMENT SUMMARY

Team *Air HERCULES* consists of twelve mechanical and electrical engineering students from the Florida A&M University-Florida State University College of Engineering. Six senior design students lead the team – four mechanical and two electrical – and the remaining six students are underclassmen members of the AIAA Student Branch. The organizational structure of the team features the senior members with both executive and design roles, to ensure proper management of both the design process and administrative tasks.

2.1 DESIGN TEAM ORGANIZATION

The design architecture was broken down into sub-systems in the areas of aerodynamics, electronics/controls, materials/fabrication, and propulsion, each led by senior members of the team. This project is advised by Dr. Joe Yeol, a post-doctoral researcher from Columbia University, and Dr. Chiang Shih, the Chair of the Mechanical Engineering Department at Florida State University.

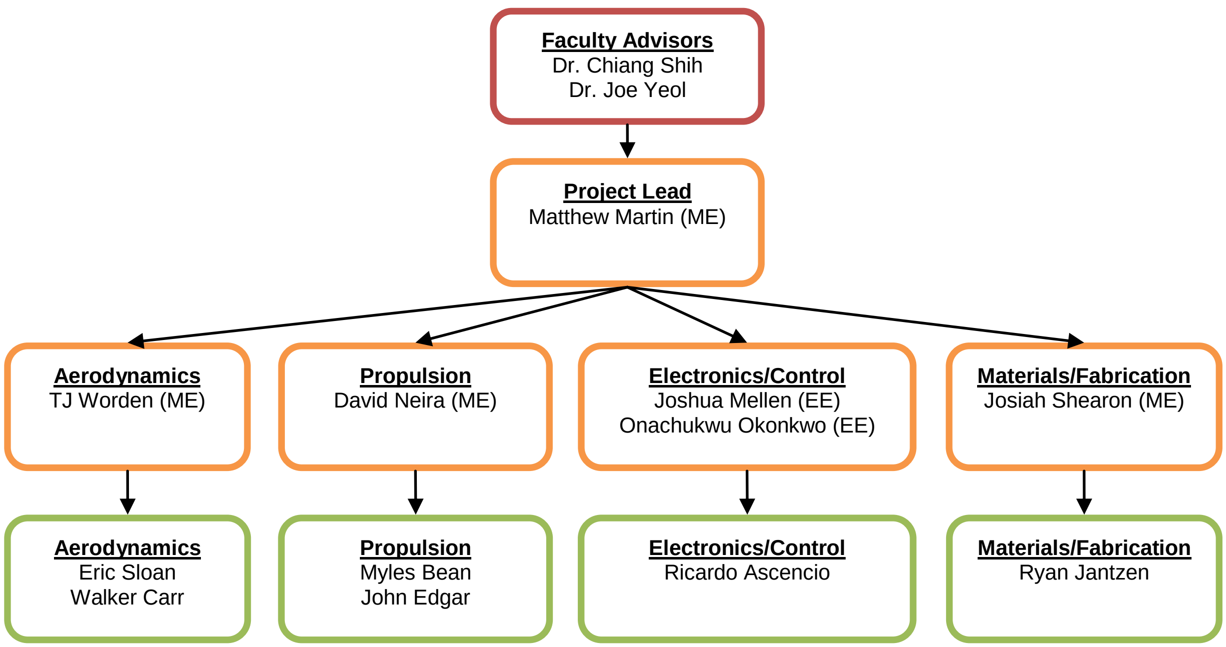


Figure 2.1: Team Organization

The project lead is responsible for the overall management and organization of the design team and ensures deadlines are met. Aerodynamics is responsible for the overall external configuration of the aircraft, including wing design, tail design, and airfoil selection. In-depth analysis and optimization of the aircraft geometry, flight performance, stability, and control characteristics are achieved. The Electronics/Controls team is responsible for the electronic systems and their configuration for optimum performance including the batteries, transmitter, receiver, and ESCs. The Materials team selects and fabricates the materials best suited for each component. The Propulsion team analyzes, designs, and integrates the propulsive components such as the motor and propeller.

2.2 Milestone Chart

In order to ensure the team progressed through the design process in an efficient and timely manner, a Gantt chart was developed with milestones and projected progress goals. Although all effort was made to adhere to specific deadlines, the necessity to extend some phases of the project resulted in slight discrepancies from the projected timeline.

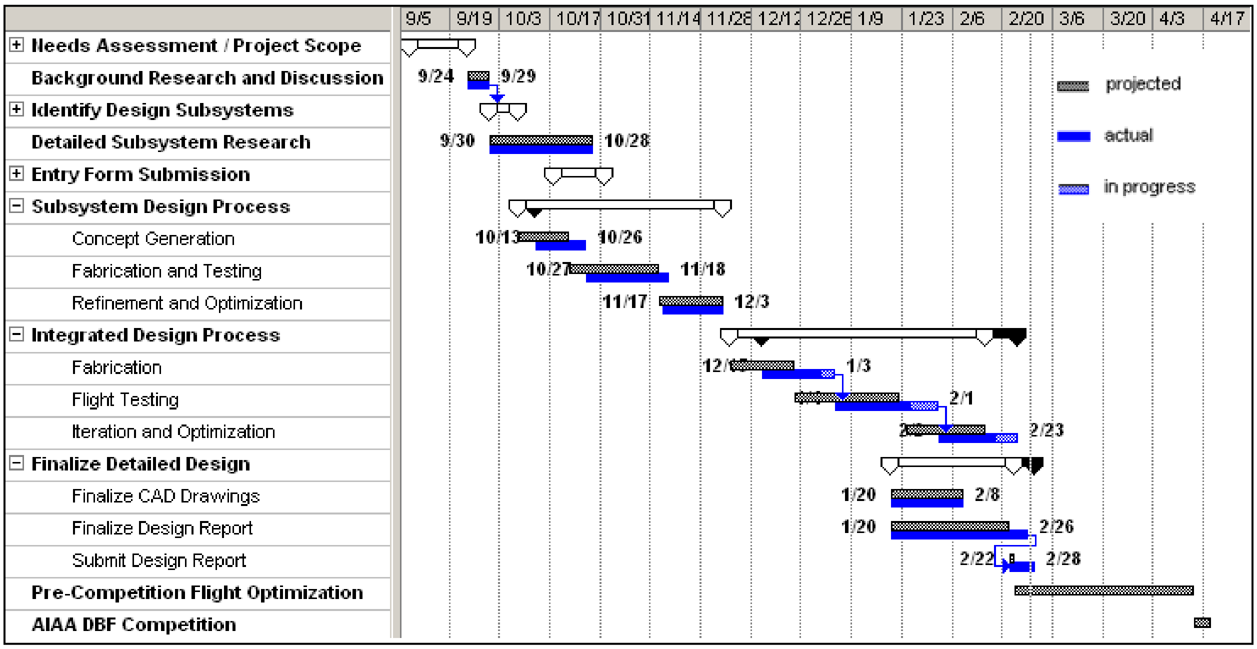


Figure 2.2: Milestone Chart

3.0 Conceptual Design

In the conceptual design phase, a needs assessment is performed to determine the design constraints and overall direction of the project. These needs are then translated to product specifications, which represent the design requirements needed to achieve a successful and competitive design. Figures of Merit (FOM) analyses of the main design elements lead to the selection of the optimum aircraft configuration.

3.1 Competition Requirements

Several design constraints and specifications are imposed by this year's contest rules, as listed below:

- Can be of any configuration other than rotary wing or lighter than air.
- Must be propeller driven and electric powered by NiCad or NiMH batteries.
- Maximum propulsion battery weight of 3/4 lb.

- Maximum current draw of 20amps.
- Disassembled aircraft and required assembly tools must fit in a commercially produced carry on suitcase of maximum linear dimensions totaling 45 inches; no single dimension can exceed 22 inches.
- Must be hand launched.
- All payloads must be secured and fully contained within the aircraft's mold lines.

3.2 Mission Profile

The total flight score equation is $FlightScore = \frac{M_1 + M_2 + M_3}{\sqrt{RAC}}$ where M_n represents the nth mission flight score, and *RAC* represents the maximum empty weight of the aircraft for any of the three missions. For ease of analysis, the *RAC* can be factored into each individual mission scoring equation to see the direct impact of the empty weight on the mission score, as detailed below. The flight course for all missions is shown in the following figure.

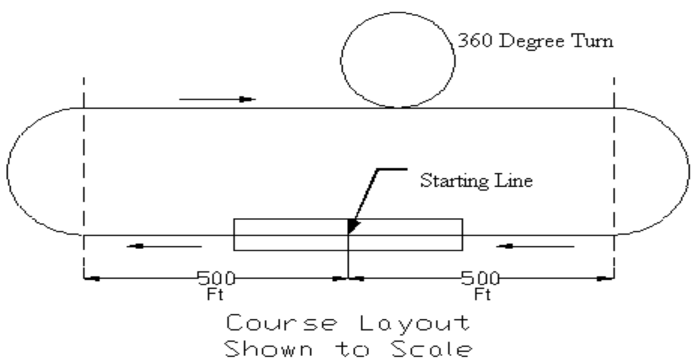


Figure 3.1: Flight course for all three missions as provided by the competition rules.

3.2.1 Mission One: Dash to Critical Target

In Mission One, the aircraft has four minutes to fly as many complete laps around the flight course as possible without a designated payload. Time starts as soon as the craft leaves the launcher's hand during the first launch attempt. The scoring equation for Mission One is $M_1 = \frac{N_{laps}}{N_{laps\_\max}} \times \frac{1}{\sqrt{RAC}}$ where N_{laps} is the number of laps completed by a specific team and N_{laps_max} is the maximum number of laps completed by any team. In order to maximize the Mission One score, the aircraft is required to feature a lightweight and fast design with high maneuverability.

3.2.2 Mission Two: Ammo Re-Supply

In Mission Two, the aircraft must be loaded with one or more steel bar payloads provided by the team and fly three complete laps around the course. The payload(s) must be a minimum of three inches in width and four inches in length, with a variable thickness which the team selects. The scoring equation for

Mission Two is $M_2 = 3\times\left(\frac{P}{P+EW}\right)\times\frac{1}{\sqrt{RAC}}$ where P is the chosen payload and EW is the empty weight of the aircraft for Mission Two. To achieve a high score for Mission Two, the aircraft must have a high payload capacity while maintaining a minimal empty weight.

3.2.3 Mission Three: Medical Supply Mission

In Mission Three, the aircraft must be loaded with a team specified number of golf balls and again fly three complete laps around the flight course. The scoring equation for Mission Three is $M_3 = 2\times\left(\frac{N_{balls}}{N_{balls\_max}}\right)\times\frac{1}{\sqrt{RAC}}$ where N_{balls} is the number of golf balls carried, and N_{balls_max} is the maximum number of golf balls carried by any team. To achieve a high score for Mission Three, not only does the aircraft have to have a high payload capacity, but it must also feature a cargo bay or storage area large enough to internally store as many golf balls as capable given the vehicles lift characteristics.

3.3 Mission Scoring Analysis

To translate the needs into product specifications, a scoring sensitivity study was conducted to determine the significance of each design parameter on the overall flight score. Initially, assumptions and ranges for certain parameters were made, which was accomplished by researching performance results from past Design/Build/Fly competitions, as well as reviewing the current capabilities of similar aerial vehicles. For this sensitivity study, a best empty weight of 1 pound and a maximum payload to weight ratio of 3:1 were assumed.

First, each mission score was analyzed individually to determine the maximum achievable score and the specific design and performance requirements needed to score within the top 20% of that mission. It was decided that scoring in the top 20% would lead to satisfactory performance in the competition. The following figures depict 2D contour plots of the variation of mission score versus the scoring parameters for each mission.

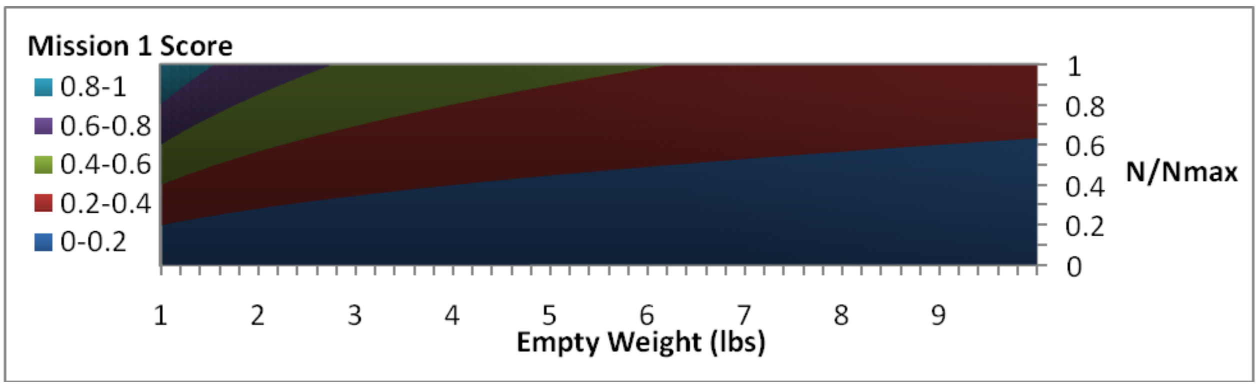


Figure 3.2: Mission 1 Score Vs. Empty Weight and Course Lap Ratio

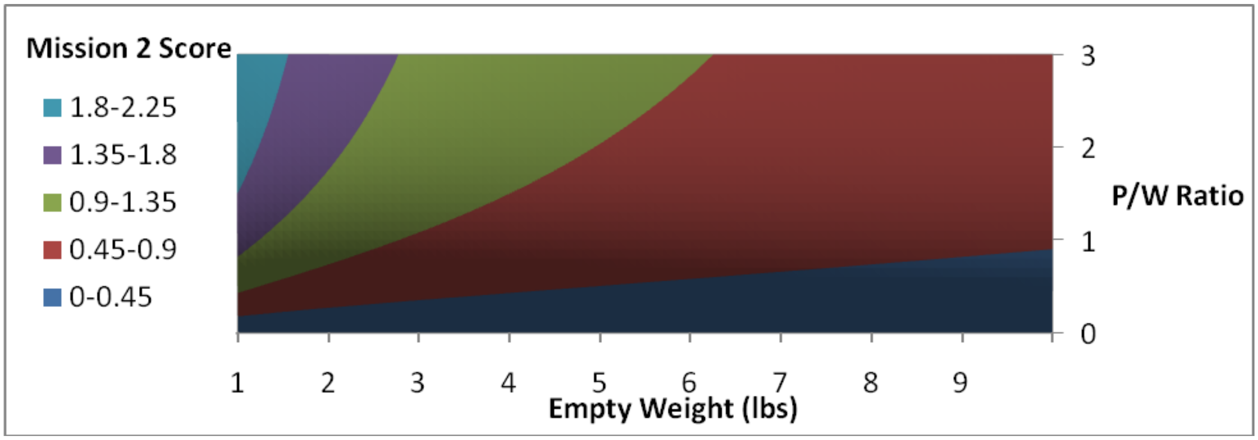


Figure 3.3: Mission 2 Score Vs. Empty Weight and Steel Bar Payload to Weight Ratio

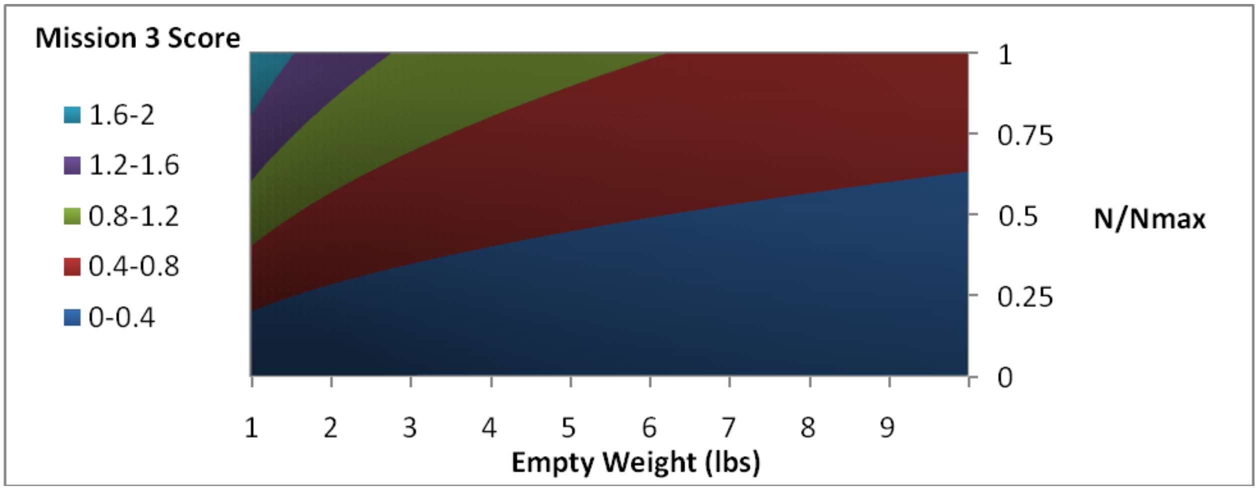


Figure 3.4: Mission 3 Score Vs. Empty Weight and Golf ball Payload Ratio

From these figures it is evident that in order to score in the top 20% for all three missions, the aircraft must have an empty weight of no more than approximately 1.6 pounds. In Mission One it is necessary to achieve at least 80% of the maximum number of laps flown by any team, in Mission Two the aircraft must have a minimum payload to weight ratio of 1.5, and in Mission Three the aircraft must carry at least 75% of the maximum number of golf balls. These graphs also show that Mission 2 is the most crucial mission, as it has the highest possible maximum score.

Next, a percent change study of the total flight score as a function of each mission's scoring parameter was performed to determine the quantifiable impact of each parameter. The following figure depicts the impact of the RAC and the other scoring parameters on the total flight score. Nominal values were set at 50% of the assumed maximum value for each parameter. It can be seen that the RAC has the most significant impact on the total flight score as a small change in the RAC results in a considerable change in the flight score.

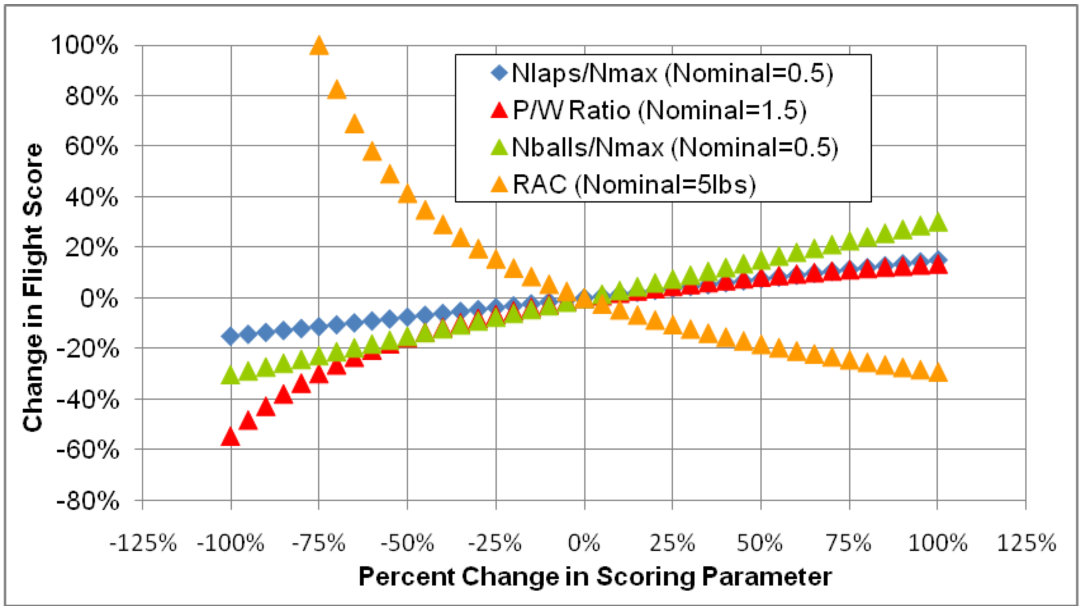


Figure 3.5: Percent Change of Total Flight Score Vs. Percent Change of Scoring Parameter

3.4 DESIGN REQUIREMENTS

Based on the competition requirements and the scoring analysis, the design requirements for the aircraft were determined and are displayed in the following table:

Table 3.1: Aircraft Design Requirements

Specification	Value
Empty Weight	< 1.6 lbs
Payload to Weight Ratio	> 1.5
Propulsion Battery Weight	< 0.75 lbs
Maximum Current Draw	20 amps
Maximum Linear Dimension	22 in
Assembly Time	< 5 minutes

3.5 CONCEPT GENERATION AND SELECTION

3.5.1 Figures of Merit

The analysis of each primary aircraft subassembly was evaluated by using figures of merit that were deemed pertinent to the performance of the aircraft. The explanation of each is listed below.

- **Weight -** The desired overall aircraft weight must be below 1.6 pounds. With that said, the weight of each component is very important and must be minimized.
- **Drag -** Drag opposes our thrust force generated by the motor which determines the amount of energy must be drawn from the batteries. This is another very important figure that must be minimized.
- **Lift -** There must be sufficient lift to sustain flight with the maximum desired payload.

- **Stability -** The aircraft must carry out each required task reliably with very little performance fluctuation.
- **Maneuverability -** There must be effective control of the aircraft such that each mission can be performed with very little energy consumption or trouble.
- **Launch Recovery -** Post launch There must be sufficient lift at low velocity to recover quickly from the launch and enough thrust to approach the stall speed before significant altitude is lost.
- **Portability -** The aircraft must be disassembled and packed into its carry bag with little effort.
- **Ergonomics -** The aircraft must conform to the launcher's hand to comfortably achieve, during launch, an initial velocity that allows the aircraft to quickly reach the designed stall speed for its given payload.
- **Durability -** The aircraft must sustain light to moderate handling and the occasional rough landing.
- **Storage Capacity -** The payload must securely store within the fuselage of the aircraft. It is required that the aircraft hold a maximum payload volume for a given design.
- **Complexity -** All required assembly must be completed with the available expertise.
- **Manufacturability -** All manufacturing must be completed with the available facilities

3.5.2 Fuselage Configuration

When designing the fuselage configuration, three concepts were considered:

- **Single Boom** - One fuselage body extends throughout the aircraft.
- **Double Boom** - Two similar fuselage bodies extend throughout the aircraft.
- **Blended Body** - Incorporates the wing and fuselage into a single body.

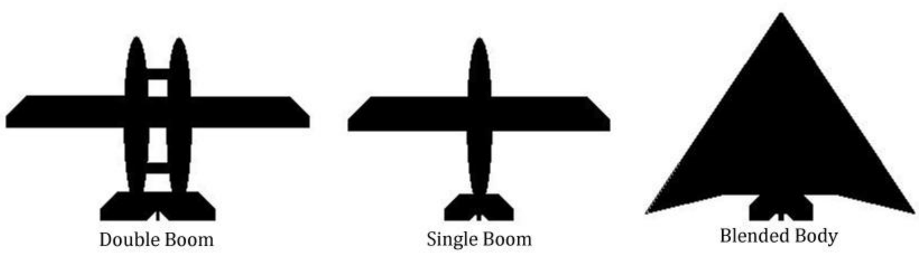


Figure 3.6: Fuselage Configurations

There are several critical parameters to consider when selecting a fuselage configuration: weight, drag, ergonomics, durability, and capacity. Each of these criteria was evaluated with respect to each design concept to determine the most effective option. The results are presented in Table 3.2.

Table 3.2: Fuselage Configuration Decision Matrix

Figure of Merit	Weighting Factor	Double Boom	Single Boom	Blended Body
Weight	0.35	1	3	5
Drag	0.20	2	4	5
Ergonomics	0.15	2	5	1
Durability	0.10	3	4	5
Storage Capacity	0.20	5	4	1
Total	**1.00**	**2.35**	**3.80**	**3.60**

This matrix illustrates that the most efficient fuselage configuration is the single boom layout. The advantages of this design reside primarily in its simplicity, ease of launch, and storage capacity while maintaining minimal drag.

3.5.3 Wing Configuration

The Contest rules state that there can be no rotary wing or lighter than air vehicles. Thusly, our choices are limited to fixed wing designs. The configurations that were considered are visually represented in Figure 8 and described below:

- **Monoplane** - A single wing that is positioned perpendicular to the fuselage. Very light, simple design with minimal drag.
- **Biplane** - Two wings that are stacked one above the other. Provides more lift than the monoplane given all other dimensions are the same. However, this configuration is heavy and induces significant drag.
- **Canard** - Two wings positioned in parallel, the smaller of the two leading the main wings. Gives the potential to provide more lift and better control characteristics. Unfortunately, it is very easy to design a poor performing and unsafe canard aircraft.
- **Delta** - Single wing with a linear increase in wing span as it progresses down the fuselage. Provides structural rigidity and high storage volume. Many of the benefits provided by this configuration are divulged at supersonic speeds; which is inapplicable to this design project.
- **Flying Wing** - Single wing aircraft with integrated body. In its ideal form, this configuration is the most aerodynamically efficient of all that were considered. With that said, it is also the most unstable and difficult to efficiently apply a payload.

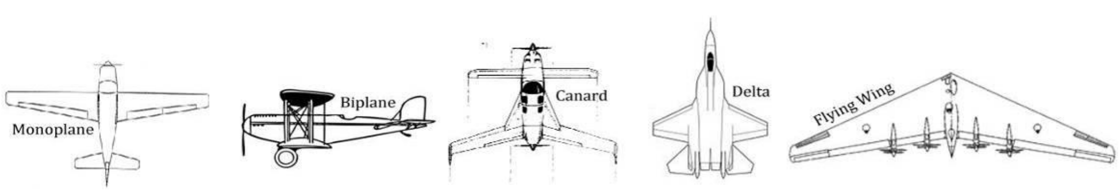


Figure 3.7: Wing Configuration

These wing configurations each were weighted with regard to their perceived performance. Then they were inserted into a decision matrix against several Factors of Merit (FOM) that were deemed paramount to performance of the aircraft.

Table 3.3: Wing Configuration Decision Matrix

Figure of Merit	Weighting Factor	Monoplane	Biplane	Canard	Delta Wing	Flying Wing
Weight	0.15	4	1	3	4	1
Drag	0.20	4	2	2	1	3
Lift	0.30	3	5	4	3	4
Portability	0.10	5	2	2	4	1
Stability	0.15	4	5	3	3	5
Manufacturability	0.10	5	4	2	3	1
Total	**1.00**	**3.90**	**3.40**	**2.90**	**2.85**	**2.90**

The selected wing configuration from the previous decision matrix was a monoplane. This design configuration is expected to be the lightest, most easily manufactured, and subject to the least induced drag. Over all, the monoplane is the most widely tested and accepted flight format of all the configurations that were considered; confirming that this configuration is a solid performer and a good selection.

3.5.4 Tail Configuration

The selected tail must possess a series of qualities that will render a stable aircraft that is light weight with low drag properties. There are many tail designs to choose from. It was necessary to analyze the properties of several of these choices and weigh their strengths and weaknesses to arrive at a final selected tail configuration. The tail configurations that were considered are depicted in Figure 9 and described below:

- **Conventional -** Vertical stabilizer is mounted to the fuselage with a 90^{0} offset from the horizontal stabilizers which are mounted to the same general location. Provides little drag and good control. May not be optimal for specific mission parameters requiring extreme stability such as aerial surveillance.
- **V-Tail** - Two angled fins that extend from the tail with trailing 'ruddervators' that control both pitch and yaw control functions simultaneously. Reduces wetted area of the rear control system which in turn reduces induced drag. Requires more complex control system and induces higher stresses on the fuselage and tail.
- **Twin Tail** - Tail with two vertical stabilizers. Provides better control with respect to yaw. The increased vertical control surface area also induces additional drag.
- **T-Tail** - Composed of horizontal stabilizers that are mounted at the tip of a single vertical stabilizer which is connected to the fuselage. Provides a benefit for rear engine aircraft which limits flow interference to the propeller. Induces additional stress on vertical stabilizer which must be accounted for in the final design.

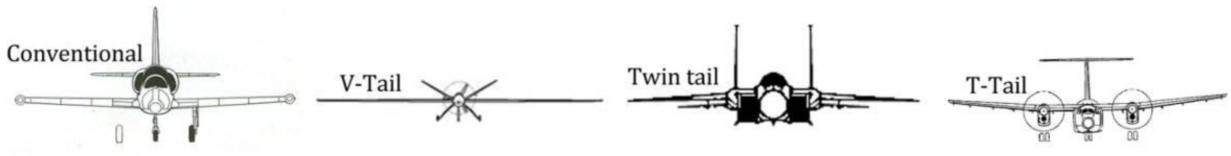


Figure 3.8: Tail Configuration

These tail configurations each were weighted with regard to their perceived performance. Then they were inserted into a decision matrix against several Factors of Merit (FOM) that were deemed paramount to performance of the aircraft.

Table 3.4: Tail Configuration Decision Matrix

Figure of Merit	Weighting Factor	Conventional	V-Tail	Twin Tail	T-Tail
Weight	0.15	3	4	3	2
Drag	0.20	4	5	3	3
Stability	0.35	5	2	3	3
Maneuverability	0.20	5	2	4	4
Manufacturability	0.10	4	2	3	3
Total	**1.00**	**4.40**	**2.90**	**3.20**	**3.05**

The selected tail configuration pulled from the above decision matrix was the conventional tail design. This design, just like the wing configuration, is tried and true. The conventional tail configuration is the most stable, maneuverable, and most easily manufactured for our purposes.

3.5.5 Propeller Layout

For the aircraft propeller layout, five propeller/motor configurations were examined.

- **Single Tractor** – A single propeller is placed in front of the fuselage. The motor is mounted behind the propeller and faces forward giving an appearance that the aircraft is "pulled" through the air.
- **Single pusher** – A single propeller is situated at the rear of the fuselage. Motor is mounted forward of the propeller facing the rear giving an appearance that the aircraft is "pushed" through the air.
- **Double tractor** – Two propellers are placed in front of the wings. The motors are mounted behind the propeller pulling the aircraft.
- **Tractor/pusher** – This configuration uses two propellers. One pulling and the other pushing the aircraft through the air.
- **Ducted fan** – Propulsion configuration where a fan is mounted within a cylindrical duct.

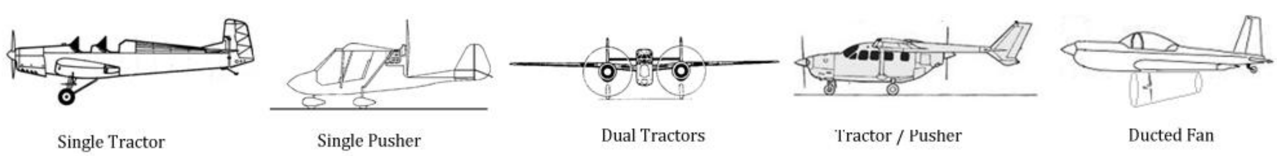


Figure 3.9: Propeller Layout

The most important FOMs to consider are weight, efficiency, launch recovery and complexity. Each of these is evaluated among several configurations in the following decision matrix:

Table 3.5: Propeller Layout Decision Matrix

Figure of Merit	Weighting Factor	Single Tractor	Single Pusher	Dual Tractors	Tractor / Pusher	Ducted Fan
Weight	0.30	5	5	3	3	1
Efficiency	0.30	4	3	3	2	5
Launch Recovery	0.30	3	2	5	4	4
Complexity	0.10	5	4	2	2	1
Total	**1.00**	**4.10**	**3.40**	**3.50**	**2.90**	**3.10**

This matrix indicates that the best propulsion system configuration is with the single tractor propeller. This design will keep the weight and complexity of the motor and propeller to a minimum while maintaining a good efficiency of the motor and a good recovery time from the launch.

3.5.6 Landing Platform

For the landing platform, five designs were considered:

- **Single Wheel** – One wheel located at the center of gravity for the aircraft. This design is simple and lightweight; however, it may not be strong enough support the entire weight of the aircraft. It would also be very unstable when landing.
- **Bicycle** – Two wheels are centered along the longitudinal axis of the body. Distributes load through two shafts. The landing would be unstable.
- **Tricycle** – A single wheel is located toward the nose of the aircraft and two wheels are located toward the rear of the aircraft on the same rotational axis. This is a very stable design but it is relatively heavy and will induce more drag.
- **Tail Dragger** – Two wheels located toward the nose of the aircraft and a single wheel located toward the rear. The front wheels are on longer shafts which cause the nose to point upward and the tail to "drag". This is a stable design but the majority of the load would be supported by the smaller tail wheel. This may cause durability issues.
- **No Landing Gear –** No protruding landing platform is used. Instead, the aircraft is designed to endure the impact of landing through a fortified undercarriage. This will increase the weight of the fuselage, however will also reduce drag.

The following matrix describes the design criteria for selecting the landing platform and the respective scores of each design.

Table 3.6: Landing Gear Decision Matrix

Figure of Merit	Weighting Factor	Single Wheel	Tricycle	Tail Dragger	Bicycle	No Landing Gear
Weight	0.30	4	3	3	2	3
Drag	0.10	4	4	3	3	5
Durability	0.15	2	5	4	4	2
Stability	0.10	1	5	3	3	4
Assembly	0.15	4	3	3	2	5
Ergonomics	0.20	4	3	2	1	5
Total	**1.00**	**3.40**	**3.60**	**2.95**	**2.30**	**3.85**

As can be seen from the above decision matrix, landing gear on such a small, hand launched aircraft was deemed cumbersome and unnecessary given the proper structural reinforcements are implemented. The primary benefits of this selection are decreased drag and assembly time while it very ergonomic for the user.

3.5.7 Golf Ball Payload Configuration

For the golf ball payload configuration, four layouts and storing methods were inspected:

- **Lattice / Box** – Balls are arranged in a lattice structure where crevices between adjacent balls secure those in contact. This lattice would be contained in a box where the dimensions are designed so that the balls fit tightly in each direction. This design is the most efficient use of space but the container is relatively heavy.
- **Row / Tube** – Balls are arranged in a single row contained by a tube. A series of row / tube configurations can be used to accommodate more golf balls. This design would minimize vibration and utilize space efficiently. The tube would be somewhat heavy.
- **Grid / Matrix** – Golf balls are arranged in a grid in which each ball is secured in its own slot. This design would ensure that the payload is secure but the matrix would be inefficient for material and space usage.
- **Disordered / Netting** – Golf balls are randomly placed into a netting material and secured by tying the open end. The material would be very lightweight but the golf balls would be difficult to secure.

The decision matrix below describes the design criteria for securing and arranging the golf ball payload and the respective scores for each design.

Table 3.7: Golf Ball Payload Decision Matrix

Figure of Merit	Weighting Factor	Lattice / Box	Row / Tube	Grid / Matrix	Disordered / Netting
Weight	0.25	3	4	1	5
Ball Density	0.15	5	3	2	1
Ergonomics	0.10	2	5	2	3
Stability	0.35	4	5	5	1
Complexity	0.15	4	5	2	3
Total	**1.00**	**3.70**	**3.95**	**2.60**	**2.20**

As can be seen from Table 3.7 above, the tube design obtained the best score primarily due to weight, simplicity, and efficiency of design.

The conceptual design is summarized and displayed in Figure 3.10 below. The aircraft is a single boom monoplane with a conventional tail configuration. The propulsion system consists of a single tractor propeller and motor and the landing system consists purely of a reinforced fuselage platform. Regarding the payload configuration, the golf balls will be oriented in a row orientation within the fuselage constrained by tubes. The steel bar will also be constrained within the fuselage by interior bulkheads. The preliminary design phase will determine more specific subsystem components and aircraft sizing.

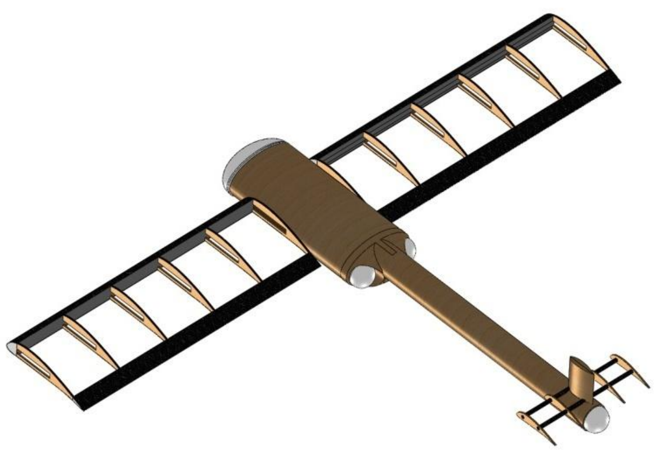


Figure 3.10: Aircraft Conceptual Design Configuration

4.0 PRELIMINARY DESIGN

In the preliminary design phase, critical design parameters associated to the areas of aerodynamics, power systems, propulsion, and structures were determined. Trade studies related to these critical design parameters evaluated the trade-offs between each aspect's design alternatives. A mission model was also developed. This model allowed for the analysis and optimization of the design by providing a prediction of the aircraft's flight performance and mission performance.

4.1 Critical Design Parameters

Aerodynamics/Stability

- **Airfoil Selection** – The airfoil selection process is crucial to ensure the aircraft has the desired performance throughout all flight segments. The complexities imposed by the hand-launch require that both the wing and tail airfoils be versatile and effective, given the uncertainty of the flow conditions over them (laminar or turbulent). Also, the airfoils must provide adequate lift and stability while maintaining maximum aerodynamic efficiency.
- **Wing Geometry** – The most critical geometric aspect is the wing area. Wing area will affect certain performance parameters such as the lift, maximum speed, stall speed, and take off capabilities. The aspect ratio is also a key geometric consideration as it will influence the aircraft's aerodynamic efficiency and stability characteristics.
- **Tail Geometry** – The aircraft's stability characteristics will be most influenced by the geometry of the horizontal and vertical stabilizers. Both stabilizers must provide adequate opposition to disturbances influencing the aircraft's state.
- **Control Surfaces** – The control surfaces must feature proper size and placement to effectively maneuver the aircraft.

Power and Propulsion Systems

- **Motor -** The selected motor must be lightweight and capable of rotating the propeller at the desired rpm for the selected propeller. The motor will comply with the competition requirements that the motor will not pull any more than 20 amps.
- **Battery Pack -** The selected battery will be lightweight (less than 12 ounces) and provide the necessary power to the motor in order to run the desired propeller at a sustained RPM for a specific period of time. The selected minimum amount of flight time at full power is 4 minutes as determined by mission 1. The target current draw for mission 2 and 3 are reduced through power management studies.
- **Propeller** - The propeller must be large enough to provide the minimum thrust values and have the required pitch to maintain speed in order to negotiate through the course within the desired amount of time while overcoming headwind.

Structures

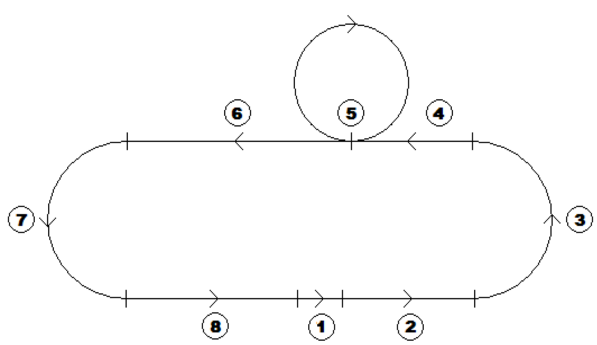
- **Material Selection** – The materials that are selected to compose the structures of the UAV are required to possess the necessary strength for proper load distribution and desired deflection while being as light as possible.
- **Structural Members** – The goal of structural member design is to effectively utilize the strengths of each individual material and properly implement each member such that the aircraft structure possesses sufficient strength to complete each mission while minimizing overall structure weight.

- **Subsystem Attachments** – The aircraft must easily be assembled/disassembled to allow it to fit into a carry on suitcase. The proper attachments are necessary to allow for such an operation. Each attachment must both be lightweight and strong while securing the joining subassemblies while attached.
- **Landing Platform** – This system must be lightweight while providing the proper landing impact sustainability and damping to the entire aircraft. The required materials used for this aircraft will be abrasion resistant and very tough. The structure supporting this platform must be able to withstand a significant amount of impact without deformation and assist in stabilizing the aircraft during the landing sequence.

4.2 Mission Model

In order to produce the most efficient aircraft to satisfy the mission profile, the course layout needs to be thoroughly analyzed and key subsystems must be optimally selected. Understanding of how the course breaks down into various phases and knowledge of what each phase requires allows the designer to make more informed decisions regarding subsystem optimization. The optimization process utilizes several techniques, including the development of an optimization algorithm and trade studies of various components.

There are essentially eight segments of the flight course, each with unique demands. The figure below demonstrates how these phases are assessed and where they occur on the course. The ability to accurately predict the circumstances encountered throughout the mission reduces error in design parameters such as propulsion battery weight, motor size, propeller size, overall aircraft dimensions.



	Mission Phase	Throttle Setting
1	Hand Launch	Full Throttle
2	Acceleration	Full Throttle
3	Turn 1	Full Throttle
4	Cruise	75% Throttle
5	360 Loop	Full Throttle
6	Cruise	75% Throttle
7	Turn 2	Full Throttle
8	Landing	Half Throttle

Figure 4.1: Mission Course Segments

4.3 Hand Launch Analysis

The most unique and critical aspect to this year's competition is the requirement for a hand-launched system. This condition affects every aspect of design from propulsion system to overall dimensions. Some important considerations include human capacity, ergonomics, recovery time (time to reach stall speed), and launch techniques.

Our team decided that this is where our design process would begin and started by developing a test to reveal optimal launch strategies. The main target of our test was to determine the effect of projectile weight on launch speed and the maximum weight that a human can comfortably handle. Comparing the trends for various gripping and launch techniques would divulge the best approach.

4.3.1 Testing Scheme

We developed a throwing test-bed which was designed to simulate the gripping portion of an aircraft five inches in width; five inches was chosen based on the carry-on suitcase dimensions. The platform also had an attachment for weights to be added or removed. We tested weights in one pound increments from one pound to eight pounds and varied the grip for each. Two main gripping techniques were investigated: Type 1 (holding only the sides of the platform), and Type 2 (holding the sides and supporting the rear using the index finger). The main objective was to determine the effectiveness of utilizing a supporting structure for faster launch speeds.

A high speed camera was implemented to determine launch speed using the sampling frequency and the time required to cross place-markers. The sampling frequency was 300 Hz and each place-marker was located 20 inches apart. Through projectile motion evaluation, the desired launching angle was determined to be 45 degrees; however, this varied between 35 and 45 degrees. Each weight was thrown 5 times for each grip type and averaged for statistical significance. The distance traveled and the time of flight were both recorded to validate the launch speed data using kinematics. The data was then plotted to obtain a governing equation describing the relationship between weight and launch speed. This equation was used to predict launch speeds for a larger weight range as shown in the figure below.

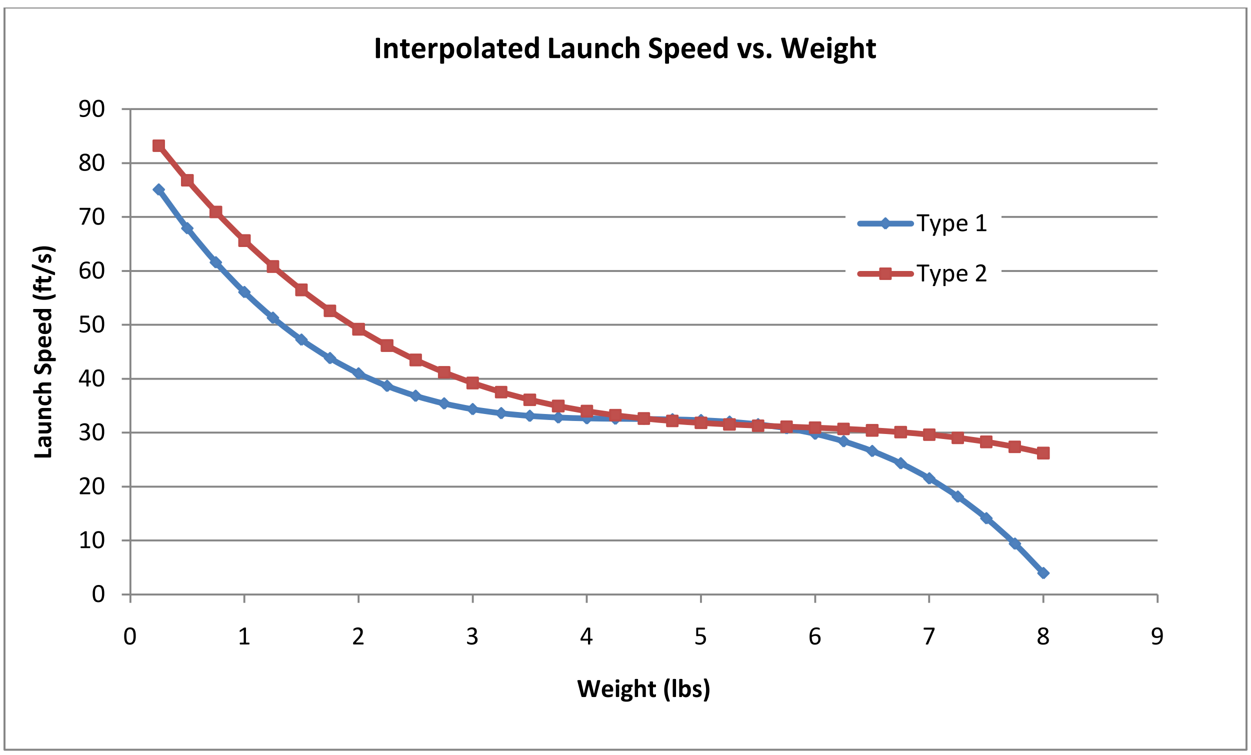


Figure 4.2: Affect of aircraft weight and grip (Types 1 and 2) on launch speed.

4.3.2 <u>Testing Conclusions</u>

Based on the Hand Launch Testing, it was determined that using the index finger for support (Type 2 grip) was more advantageous for getting the aircraft up to speed. It was also determined that the maximum aircraft weight that a human can throw with significant velocity (~30ft/s) is around six pounds. In order to further reduce the range of possibilities, more analysis would need to be conducted regarding recovery time (time to reach stall speed). This parameter is heavily dependent on aircraft weight and the thrust generated by the propulsion system.

4.4 INITIAL SIZING

This year's competition rules are unique in comparison to previous years' rules. Although each aircraft must conform to the same suitcase dimensions, significant variability in overall size may occur due to the fact that each team must independently size their payloads. Certain teams may strive to maximize size in order to have higher payload capabilities, while others may sacrifice payload capacity for a smaller RAC. The results of the hand launch testing coupled with both the traditional sizing methods described by Raymer and some unique methods developed by our team allowed for initial sizing of the aircraft to be performed.

First, the desired aircraft weight was determined by taking into account the mission performance requirements, the properties of each aircraft component, the ergonomics associated with hand-launch, and empirical performance estimations based on past DBF designs. To explore the trade-offs between payload capacity and overall system weight, the mass and geometric properties of each aircraft component (including payloads) and the mass properties of traditional materials used in RC aircraft fabrication were considered. Through several iterations, it was decided that a maximum system weight of 3 pounds would be an ideal target. This weight was determined to be the most favorable because it provides the best ratio between the estimated empty weight of the aircraft (including components) and the estimated payload capacity.

The scoring analysis revealed that a minimum payload to weight ratio of 1.5 is required, which led to the conclusion that at least 15 golf-balls must fit internal to the fuselage (along with all other components). The sizes of the motor, battery, control hardware, and other aircraft components were approximated with a factor of safety to determine the necessary fuselage volume.

It was assumed that the minimum speed of the aircraft throughout all phases of flight would occur during take-off. At 3 pounds, the hand launch tests reveal a minimum initial launch speed of roughly 40 ft/s. Therefore, the wings were sized using this speed so as to completely avoid stall. The lift coefficient of a mid-lift range airfoil was used for initial sizing to provide a conservative estimate of the necessary wing area. Also, it was initially assumed that the half-wing span would be limited solely by the maximum linear dimension of the suitcase to yield the highest aspect ratio and thus the most aerodynamically efficiency wing. The results of the initial sizing methods are listed in Table 4.1.

Table 4.1: Initial Aircraft Sizing

Specification	Value
Total Weight	3 lbs
Fuselage Volume	46.3 in^3
Wing Area	406 in^2
Effective Wing Span	44 in
Aspect Ratio	4.55

4.5 SUBSYSTEM TRADE STUDIES AND SIZING OPTIMIZATION

4.5.1 Optimization Algorithm

MatLab and C-programming software were used to develop an algorithm that accepts a set of inputs regarding critical aircraft parameters, analyzes those values using related equations and subsystem modules, evaluates the flight performance and mission scores, and iterates to determine the optimum configuration. Standard equations for lift, drag, thrust, power consumption, and other related parameters were used to evaluate a given aircraft using solutions to simultaneous equations. The scoring relationships to these equations were established through a breakdown of the critical parameters for each mission (Mission 1 – speed, maneuverability, endurance; Mission 2 – lift, thrust, weight capacity, endurance; Mission 3 – same as 2 with volumetric capacity). Figure 4.3 is a flowchart which describes the algorithms' functionality.

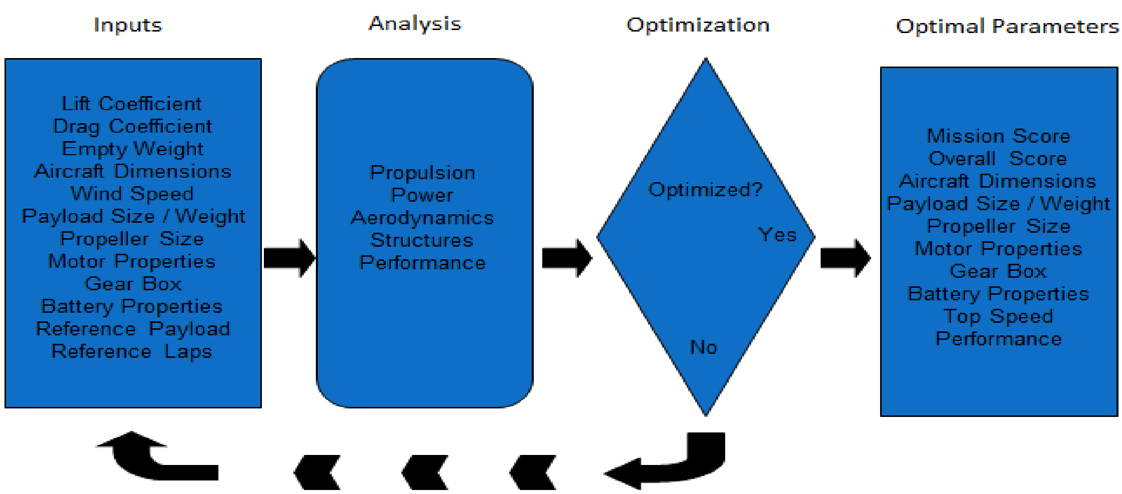


Figure 4.3: Optimization Algorithm Flowchart

4.5.2 Power and Propulsion Systems

Batteries

As per the competition rules, we are required to use either Nickel Cadmium (NiCad) or Nickel Metal Hydride (NiMH) batteries. NiMH batteries have a higher energy density and do not suffer from the Memory Effect that NiCad batteries do. For this reason, we selected NiMH batteries as the best candidate for the specifications.

Table 4.2: Battery Type Comparison

Specification	Battery Type	
	NiCad	NiMH
Nominal Cell Voltage	1.2V	1.2V
Energy Density	50-150	140-300
Cycle Durability	2000	1500
Full Discharge Required	Y	N
Internal Resistance	Very Low	Low
Toxicity	Medium	Low
Volatile	N	N

The specifications also outline that the propulsion batteries must be fused and separate from other electrical and electromechanical systems. The propulsion batteries will provide power to the motor controller and the motor itself, while a separate battery pack will provide power to the receiver, servos and any other control subsystems.

Propulsion Battery Pack

The propulsion battery pack must supply high voltage per unit weight in order to minimize the required current draw by the motor. With this in mind, the battery cells will be oriented in series to maximize the battery pack voltage but must be composed of cells with the appropriate electric charge. Lastly, the battery pack must be composed of several individual cells that are oriented in a desired configuration that will allow for easy installation and removal. The smaller the cells, the simpler it will be to arrange them within the fuselage. Several NiMH battery cells were considered. Each battery cell was inputted into the Optimization Algorithm and the results were compared. Due to the 20 amp fuse limitation, the optimization algorithm was run with an 18 amp continuous draw with a 2 amp safety margin. The results are listed below:

Table 4.3: Optimization Algorithm Results

Specification	Battery Type			
	Elite 3300	Elite 2000	Elite 1700	Elite 1500
Electric Charge	3300 mAh	2000 mAh	1700 mAh	1500 mAh
Weight	1.93 oz	1.16 oz	1.0 oz	0.81 oz
Dimensions (LxD)	1.7" x 0.91"	1.67 "x 0.66"	2" x 0.55"	1.13" x 0.66"
Maximum Cell Number	6	10	11	14
EST Pack Weight	11.58 oz	11.6 oz	11 oz	11.32 oz
EST Pack Voltage	7.2	12	13.2	16.8
EST Flight Time (Min)	11	6.67	5.6	5

Table 4.3 above reveals the specifications for four batteries, each with different electric charge. The most important parameters considered when evaluating the battery pack include weight, size, pack voltage, and electric charge. The batteries which possess a higher current capacity, electric charge, typically have higher weight and lower pack voltage. Batteries with lower current capacity are lightweight with high pack voltage but have limited flight time.

Motor

The motor selection was based on several desired characteristics: lightweight, high efficiency, and high power. All the motors considered should comply with the 20 Amp maximum current draw requirement specified in the competition rules. Five motors which complied with this constraint were evaluated in Table 4.4 below:

Table 4.4: Motor Trade Study

Motor	Weight (oz)	KV	Power (W)	Max Current Draw (Amps)	R (ohm)	Io (Amps)
Himax HA2025-4200	2.82	4200	175	15	.075	0.75
Himax HA2015-4100	2.20	4100	110	10	.141	.80
Himax HG2015-5400	2.26	5400	110	20	.120	1.0
Neu 1105/3Y	4.41	3300	350	20	0.039	0.65
Neu 1105/3.5Y	4.41	3000	350	18	0.055	0.45

For the motor selection, the main concern was the weight of the system. The Neu motors provide higher power than the Himax motors, but they require more battery capacity which will increase system weight. The Himax HA2015-4100 and Himax HG2015-5400 motors are extremely lightweight but provide limited power. The Himax HA2025-4200 motor possesses moderate characteristics of both power and weight when compared to the other motors under consideration.

Propeller

Once the battery and motor were selected, the propeller size could be evaluated. Propeller dimensions are characterized by diameter and pitch (displayed as DIAMETER X PITCH), which are the primary variations in propeller types. Larger diameter propellers typically generate higher thrust but also consume more power. Pitch refers to the angle or twist of the blade. A larger pitch value generally results in a higher top speed but also puts more load on the motor, resulting in higher power consumption.

There are three main factors used to assess propeller effectiveness: thrust coefficient, power coefficient, and propeller efficiency. Each of these factors is evaluated with respect to the advance ratio, which is essentially a comparison of linear aircraft velocity to propeller blade velocity. It is desirable for the advance ratio to be larger as this indicates that less propeller rotations are needed to move the aircraft a specified velocity.

The following comparisons were generated using manufacturer provided data since the facilities for dynamic thrust testing were not available to the team. These comparisons were used to perform trade studies on each of the following propellers: 12x8, 12x10, 10x7, 9x7.5, and 8x6. Actual data for static thrust would be confirmed using in-house testing.

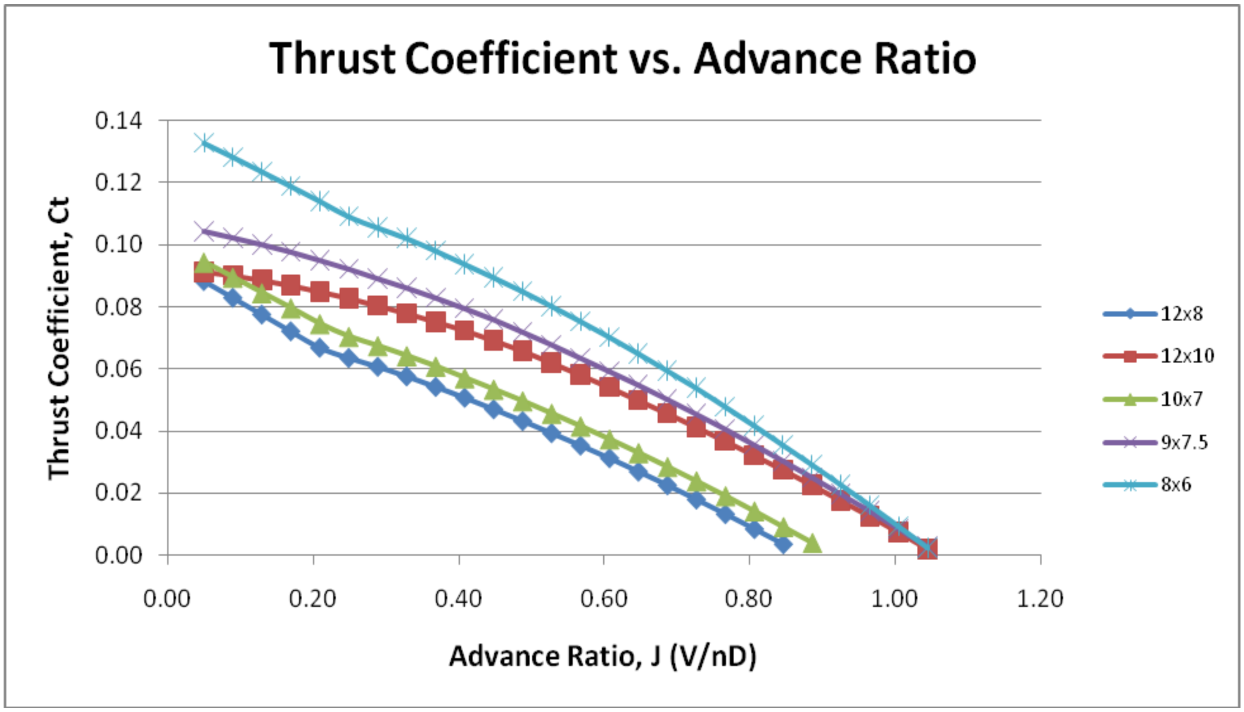


Figure 4.4: Thrust coefficient comparison for various propellers

The first parameter employed in propeller trade studies was the thrust coefficient, C_T. The thrust coefficient is an indicator of the amount of potential thrust and is defined as:

$$C_T = \frac{T}{\rho \cdot n^2 \cdot D^4}$$

where T is the generated thrust, ρ is the fluid density, n is the propeller frequency, and D is the propeller diameter. Optimal propellers possess a higher thrust coefficient for larger advance ratios as this indicates there is more thrust potential at higher speeds. Figure 4.4 above shows the comparison of each propeller under consideration. The 8x6 propeller exhibits the best thrust coefficient characteristics while the 12x8 appears to be the most unsatisfactory.

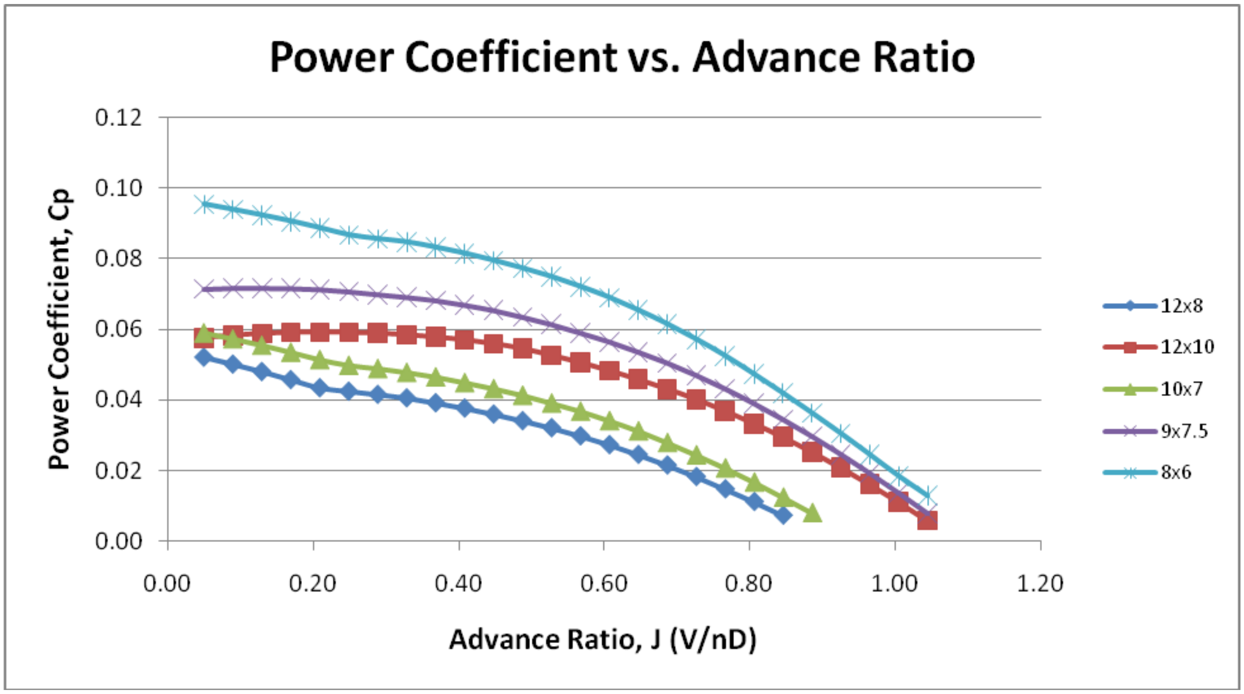


Figure 4.5: Power coefficient comparison for various propellers

The next parameter utilized in the propeller trade studies is the power coefficient. The power coefficient denotes how much power is consumed when in operation and is defined as:

$$C_P = \frac{P}{\rho \cdot n^3 \cdot D^5}$$

where P is the consumed power, ρ is the fluid density, n is the propeller frequency, and D is the propeller diameter. It is desirable for a propeller to have a lower power coefficient as this signifies less power will be consumed during flight. This will allow for the battery to have a lower electric charge and will reduce overall weight of the aircraft. Figure 4.5 above displays the relationship of power coefficient and the advance ratio for each propeller. In this case, the 12x8 propeller possesses optimal characteristics while the 8x6 seems to consume the most power.

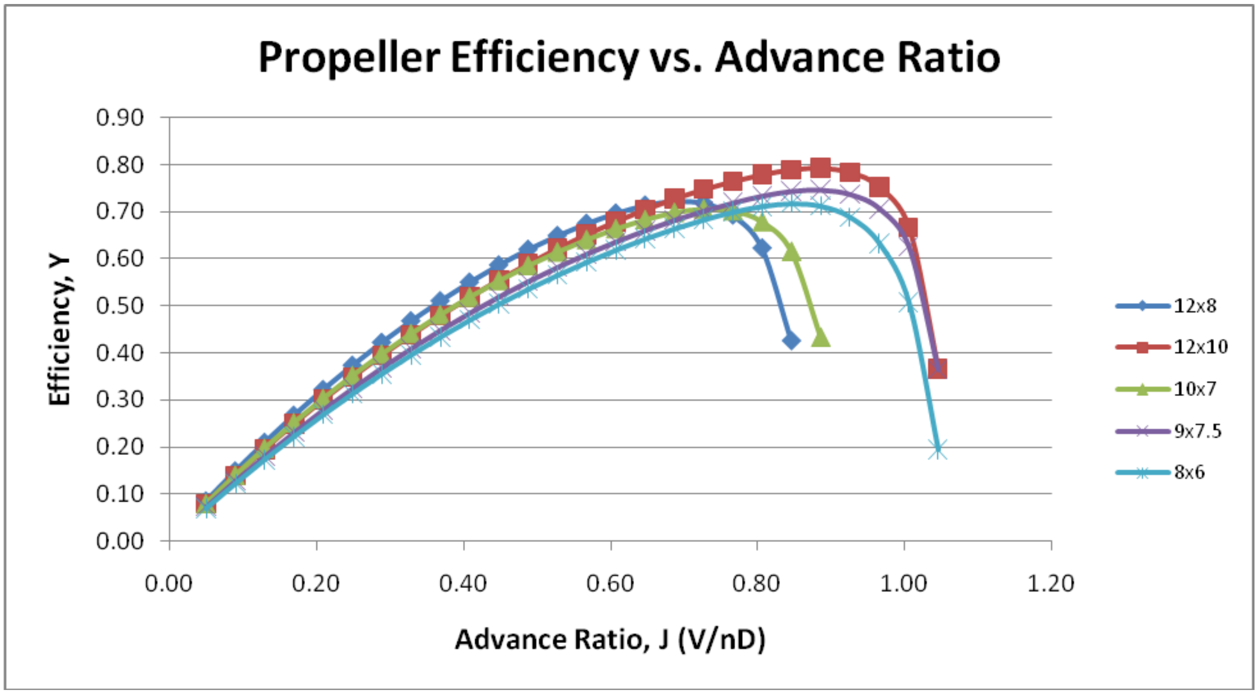


Figure 4.6: Propeller efficiency comparison for various propellers

The most important parameter regarding propeller evaluation is the propeller efficiency. This efficiency is an indicator of the amount of thrust generated per unit of consumed power. The equation which describes propeller efficiency can be shown as:

$$\gamma = J \cdot \frac{C_T}{C_P}$$

where J is the advance ratio (V/nD), C_T is the thrust coefficient, and C_P is the power coefficient. Ideally, the best propeller would have the highest efficiency at a larger advance ratio. Figure 4.6 above displays the efficiency of each propeller. The 12x10 propeller appears to be the most favorable solution as it is the most efficient and at a relatively high advance ratio. The 10x7 seems to be the worst solution because it has the lowest efficiency at a relatively low advance ratio. It is also important to consider that the larger the diameter and pitch of the propeller, the more load it will put on the motor. This will cause the motor to draw more power and will reduce the flight time.

The battery, motor, and propeller data were evaluated in conjunction with the optimization code to determine the optimal solution for the aircraft. Several tests were conducted to determine the effect of battery pack voltage, motor power, and propeller size on thrust generation and flight time. The batteries considered ranged from 7.2 – 16.8 Volts and 1500 – 3300 mAh; the tested motors varied in power from 110 – 350 Watts; and the propeller sizes ranged from 8x6 – 12x10. Each configuration was assessed by thrust per unit weight and potential mission score.

Through several iterations of tests and implementation of the optimization algorithm, the final power and propulsion system was decided. The battery pack would consist of ten Elite 1500 NiMH batteries, possessing a total of 12V and weighing approximately a half pound. The motor selected was the Himax

HA2025-4200 which has a 175 Watt capacity and weighs 2.82 ounces. The optimal propeller was determined to be the 12x10 which boasts the highest efficiency of all propellers considered.

Control System

To meet the desired specifications, our electronics must be able to power and control all the electrical and electromechanical systems in order to sustain flight for the entire duration of each flight mission with maximum current draw.

Servo Battery Pack

The servos draw very little power to operate when compared to the motor, allowing for fewer and less powerful batteries to be used as most of the empty weight for the aircraft will consist of batteries. They require a minimum voltage of 5V, and under strain, approximately draw 1A of current from the batteries. Due to the lower current draw, a lower capacity battery was chosen: the KAN 400 mAh 2/3 AAA. The battery pack will contain five cells for a combined nominal voltage of 6V. An additional cell could be added if the voltage is insufficient after a short amount of time (due to the battery's voltage drain). For a 400 mAh capacity battery supplying power to 4 servos under continuous strain, the battery can theoretically last 6 minutes.

Fuse

As per the competition rules, a 20A ATO Blade Fuse will be used to restrict the current draw from the batteries and protect the electrical and electromechanical systems from surges.

Servo Controller

There are two systems that could be used to control the servos. A general purpose microcontroller could be used to interpret the signal from a receiver and retranslate them to the servos (in the case of the fail safe maneuver where the receiver loses signal with the transmitter for 3 seconds or longer). This design would be ideal for weight, however, the work involved in programming the controller is great, as there's very little information on what kind of signals are sent from the receiver to the servos. Another issue would be programming the controller to interpret and make use of the data sent to and received from the gyroscope.

For this competition, a more simplified approach can be used as more complex designs can lead to setbacks in the design and issues when flying the plane. A transmitter/receiver was selected such that the Fail Safe was already incorporated and programmable. In addition, the receiver would need to be able to make use of a gyroscope. There are many transmitters/receivers like this on the market, especially for video telemetry for projects such as solar gliders design to gather intelligence from specific locations. The Aurora 9 Transmitter w/ Optima 7 Receiver combination was selected for our design for its optimal cost and having all the necessary tools to control our aircraft and meet the competition's specifications.

Motor Controller

Table 4.5: Motor Controller Trade Study

Figures of Merit	Motor Controller		
	CC Thunderbird 18	CC Phoenix 25	Atlas Black 20
Weight	0.6 oz	0.6 oz	0.625 oz
Size	1.32x0.33x0.90 in	1.08x0.91x0.16 in	1.875x0.875x0.375 in
Continuous Amp	18	25	20

For the chosen motor, the Castle Creations Phoenix 25 was recommended for best compatibility with our motor, and as thus, was chosen. The Atlas Black 20 is capable of powering the motor but it is rather borderline with the requirements. If cost was more of an issue, the Atlas Black 20 might have been chosen. The advantage of using the Phoenix motor controller is that it is lighter and even smaller than the Atlas Black controller. The motor controller will also be powered separately from the receiver (red wire from the receiver disconnected).

4.5.3 Aerodynamics

Airfoil Selection

To begin the wing design process, several airfoils are compared and analyzed to determine those that best exemplify the desired characteristics as per the design requirements. Due to the time constraints imposed by this competition and the teams' limited experience with aerodynamic study, it was decided that only pre-existing airfoils would be nominated for analysis (i.e. no radical new designs would be developed by the team). Historical and acquired information from similar aerial vehicles provided a basis for the airfoils considered. Literature resources and a 2D panel method solver, JavaFoil, were used to compare the drag polars (C_l vs. C_d), lift curves (C_l vs. α), and moment coefficients of each airfoil. Performance parameters such as the design lift coefficient (C_l), maximum lift coefficient (C_{lmax}), and maximum stall angle (α_{max}) were used in the optimization code to determine which airfoil results in the best mission performance.

Due to the sizing constraints and low takeoff speed at hand launch, it was realized that a high lift airfoil would be required to reduce the necessary wing area. The hand launch also imposes uncertainty in regards to the initial flow characteristics over the airfoil. Since laminar flow cannot be guaranteed, an airfoil that possesses a gradual reduction in lift during stall is preferred so changes in flight characteristics can be quickly resolved. The maximum aerodynamic efficiency occurs when the airfoil is operating at its design lift coefficient, and therefore the ideal airfoil must produce adequate lift while operating at its design lift coefficient and expected cruise velocities.

The following figures depict a comparison of the drag polars, lift curves, and moment coefficients for various considered airfoils at the approximate expected Reynolds number of 200,000.

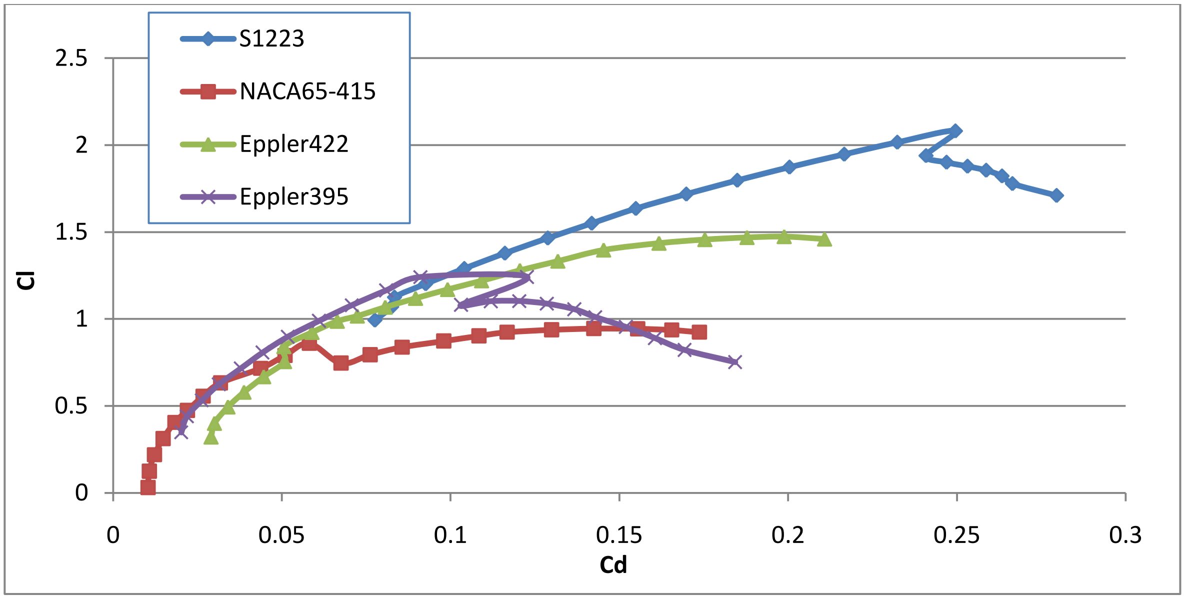


Figure 4.7: Drag Polars for Various Considered Airfoils

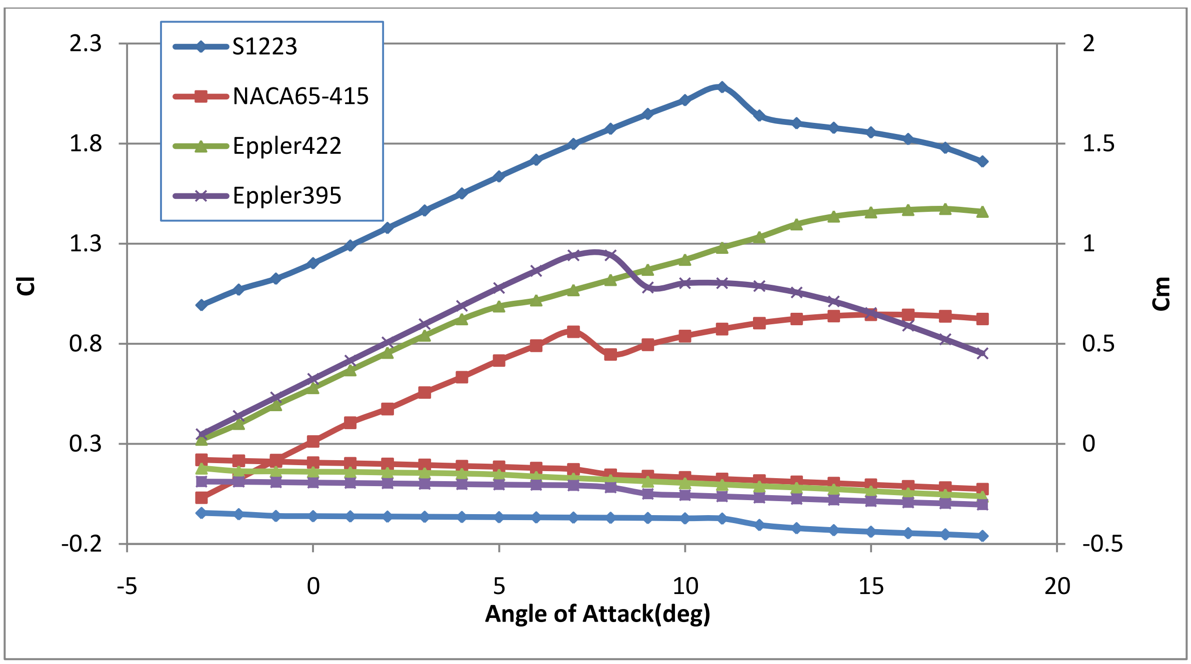


Figure 4.8: Lift Curves and Moment Coefficients for Various Considered Airfoils

The S1223 and other similar airfoils produce an extremely high amount of lift, however the relatively high moment coefficient necessitates large tail surfaces to maintain stability, which lead to an increase in drag and weight. Both the NACA65-415-type and Eppler 395-type airfoils feature low drag and avoid the laminar bucket, as desired. Conversely, the NACA65-415-type airfoils don't produce high enough lift and the Eppler 395-type airfoils feature a low stall angle. Through several iterations, it was determined that the Eppler 422 airfoil best exemplifies the desired design and performance characteristics. The following table and figure depict the section geometry and airfoil performance at a design mach number of 0.04.

Table 4.6: Eppler 422 Performance Characteristics

Specification	Value
Max Cl	1.474
Stall Angle	17°
Max L/D	16.62
Cl at Max L/D	0.841
Angle at Max L/D	3°

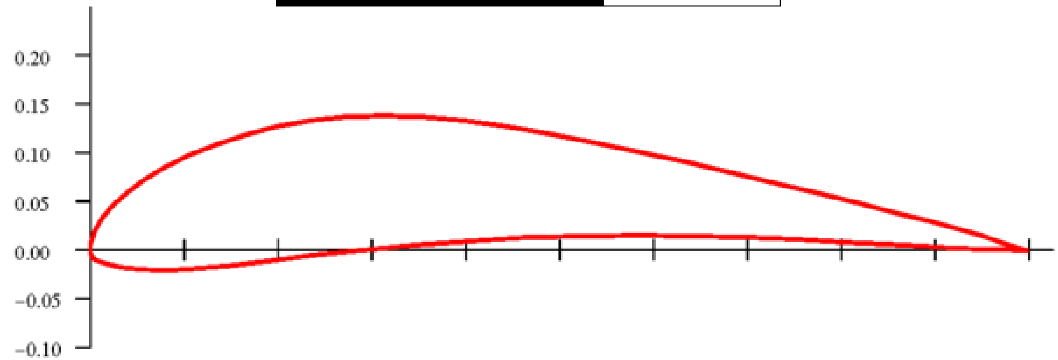


Figure 4.9: Eppler 422 Section Geometry

Wing Sizing

With the airfoil section finalized, trade-offs between the wing's key geometric parameters were conducted to determine the optimum dimensions. Using the section's performance parameters, the wing area, span, chord, and aspect ratio were investigated using the mission scoring optimization code as discussed further in the Mission Scoring Prediction section. As assumed in the initial sizing, it was confirmed that maximizing the wingspan would result in the best overall performance by increasing the aerodynamic efficiency of the wing. A ½ inch tolerance was used to ensure the wings would fit inside the suitcase, which resulted in a half-span of 21.5 inches and a total span of 43 inches. The optimum wing area to provide the necessary lift while minimizing drag was found to be 299.3 in^2, resulting in a chord length of 6.96 inches and an effective aspect ratio of 6.17.

Empennage Design

The primary goal of the tail design is to determine the proper size for both the horizontal and vertical stabilizers to counteract the pitching and yawing disturbances experienced during flight. This is usually achieved with the use of a symmetric airfoil. Through a similar analysis as was used in the wing airfoil selection, several airfoils were considered for the empennage. Since flow characteristics vary minutely for symmetric airfoils, the optimum section was chosen to produce the least amount of drag while still featuring an adequate size for simplistic fabrication. A NACA-0012 airfoil was chosen for both the horizontal and vertical stabilizers. The airfoil section and drag polar at Re=200,000 and Mach=0.04 can be seen in the following figure.

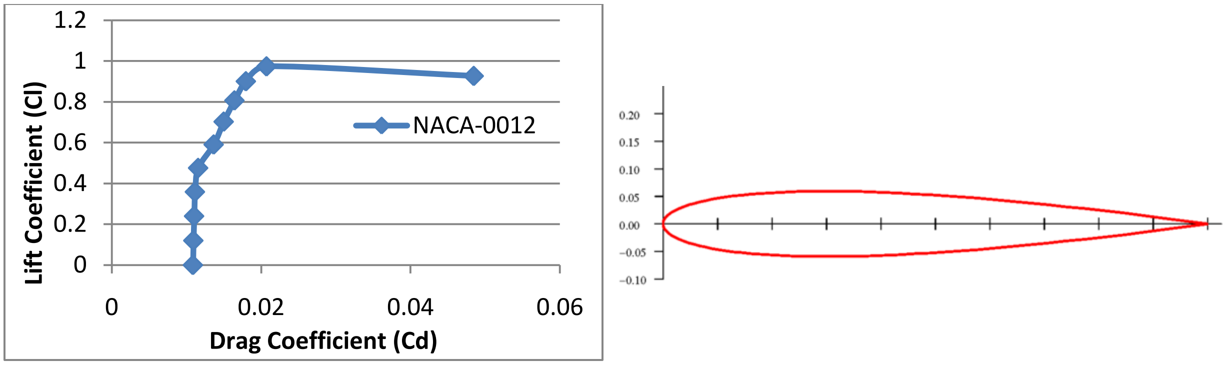


Figure 4.10: NACA-0012 Drag Polar and Shape

Tail sizing was achieved using classical methods based on the tail volume coefficient, which relates the wing's span, chord, area, and effective moment arm. By investigating the aircraft's static longitudinal and lateral stability characteristics, values of 0.04 for the vertical stabilizer and 0.7 for the horizontal stabilizer were determined to provide adequate stability while minimizing weight and drag. These results are discussed further in the stability and control section. The following table documents the vertical and horizontal stabilizer sizes.

Table 4.7: Empennage Size

	Area	Span	Chord	Aspect Ratio
Horizontal Stabilizer	64.73 in^2	14.29 in	4.53 in	3.15
Vertical Stabilizer	34.09 in^2	7.41 in	4.6 in	1.61

Control Surfaces

The size and shape of the control surfaces were designed to provide the desired changes in aircraft attitude, while maintaining ease of fabrication. The area for the rudder, elevator, and ailerons was initially determined using historical data and then optimized using the Tornado Vortex Lattice Method MATLAB Simulation as described in the Stability and Control section. The elevator is designed to span the entire length of the horizontal stabilizer to allow for simplistic integration. In order to enable adequate clearance for elevator deflection, the rudder spans from the top of the vertical stabilizer down to the second to last rib. The ailerons span from the wing's second-most inner rib to the second-most outer rib. A summary of the control surface sizes is provided in the subsequent table.

Table 4.8: Control Surface Sizing

	Area	Span	Chord
Ailerons	19.26 in^2	17.04 in	1.13 in
Elevator	29.15 in^2	14.29 in	2.04 in
Rudder	11.76 in^2	6.22 in	1.89 in

Stability and Control

The stability and control characteristics of the aircraft were explored and optimized using a MATLAB Code called Tornado, which uses the Vortex Lattice Method to simulate the forces and moments imposed during flight. The aircraft’s wing, tail, and control surface geometry was modeled and then analyzed through multiple simulated flight scenarios.

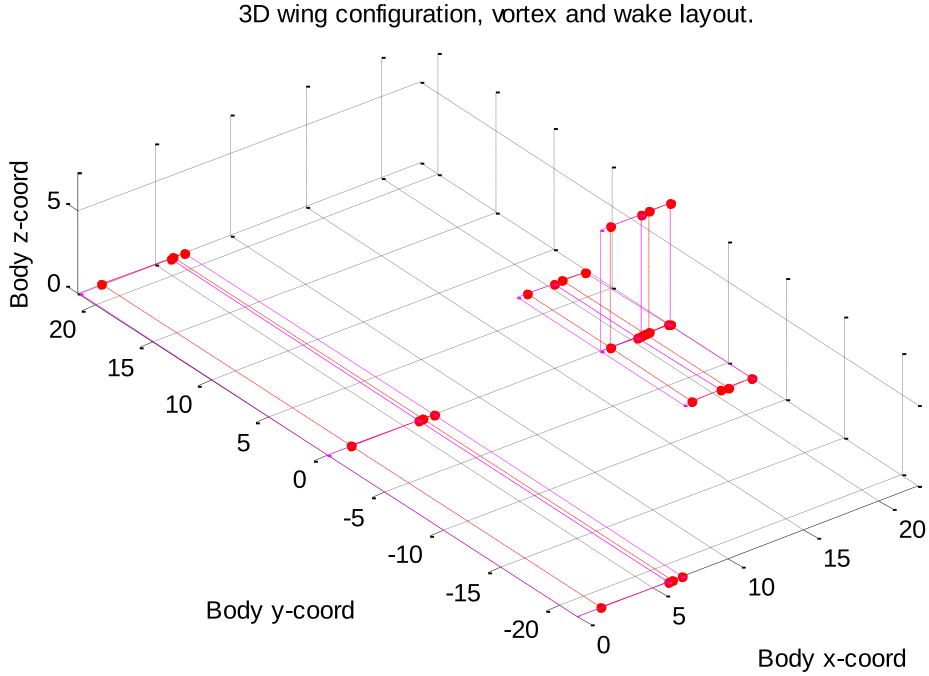


Figure 4.11: Wing, Tail, and Control Surface Lofting in MATLAB

Analysis of the aircraft’s longitudinal stability ensured that the horizontal stabilizer and control surfaces effectively counter forces and are capable of overcoming the downwash induced by the flow field of the wing through various angles of attack. The stability coefficients are plotted against angle of attack in the following figure, and the negative pitching moment (Cm) signifies the neutral point is behind the center of gravity; thus the aircraft is statically stable. The static margin was found to be 18%, which although high, provides the necessary handling qualities to minimize the chance of catastrophe during both take-off and landing. Also, it is noted that the effects of rolling (Cl) and yawing (Cn) moments are extremely low, as expected due to the symmetry of the tail airfoils. The lateral and directional stability were confirmed in the same manner, by analyzing the effectiveness of both stabilizers and control surfaces to induced rolling and yawing moments over a varying range of incidence angles. The lift and drag coefficients also depict the aircraft’s total lift and drag as a result of combining all lifting surfaces. These values are evidently higher than those estimated during the wing sizing, however they do neglect the fuselage components.

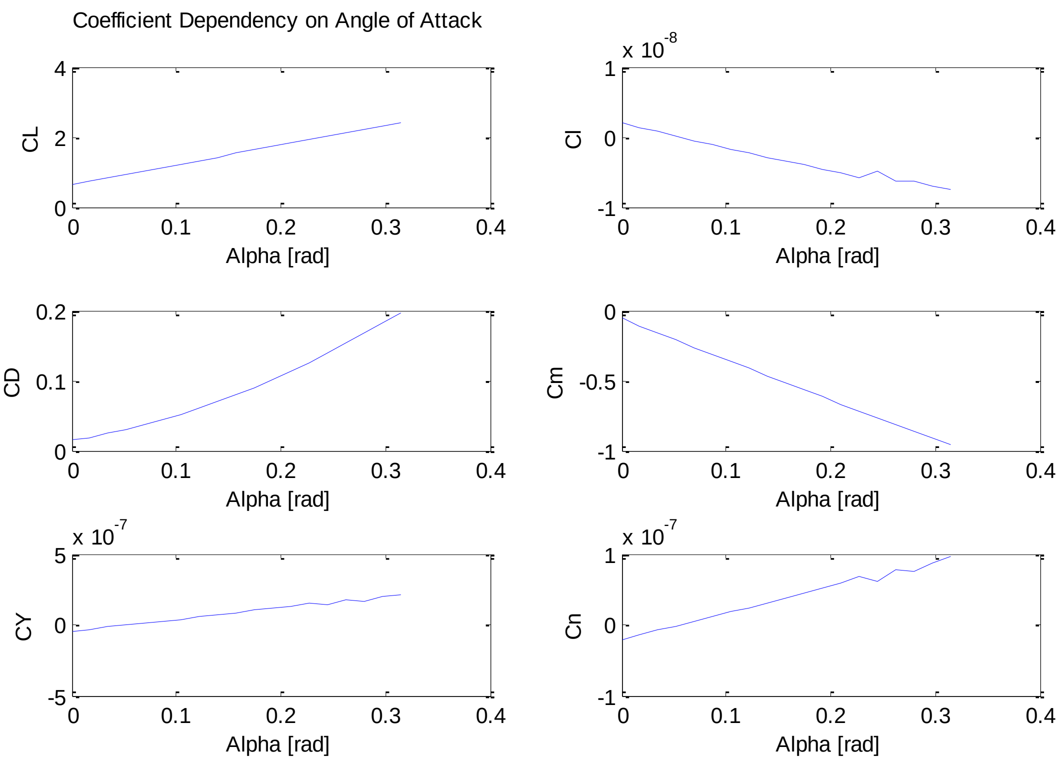


Figure 4.12: Stability Coefficients for Varying AOA (α)

4.5.4 Structures

Materials Selection

As per the mission scoring analysis results, the selected materials must be as lightweight as possible while featuring a high enough strength to withstand the impact of a hard landing. The material selection process involves utilizing the appropriate material selection charts with regard to materials strength and weight. The modulus for utilization of the chart in Figure 4.13 was derived with respect to the two material properties of interest, yield strength and density. The two most practical materials were found to be carbon fiber (CFRP) and wood, as they both feature adequate yield strength in comparison to density. Considering the significant impact that weight has on the competition scoring, minimizing this parameter was of the utmost importance.

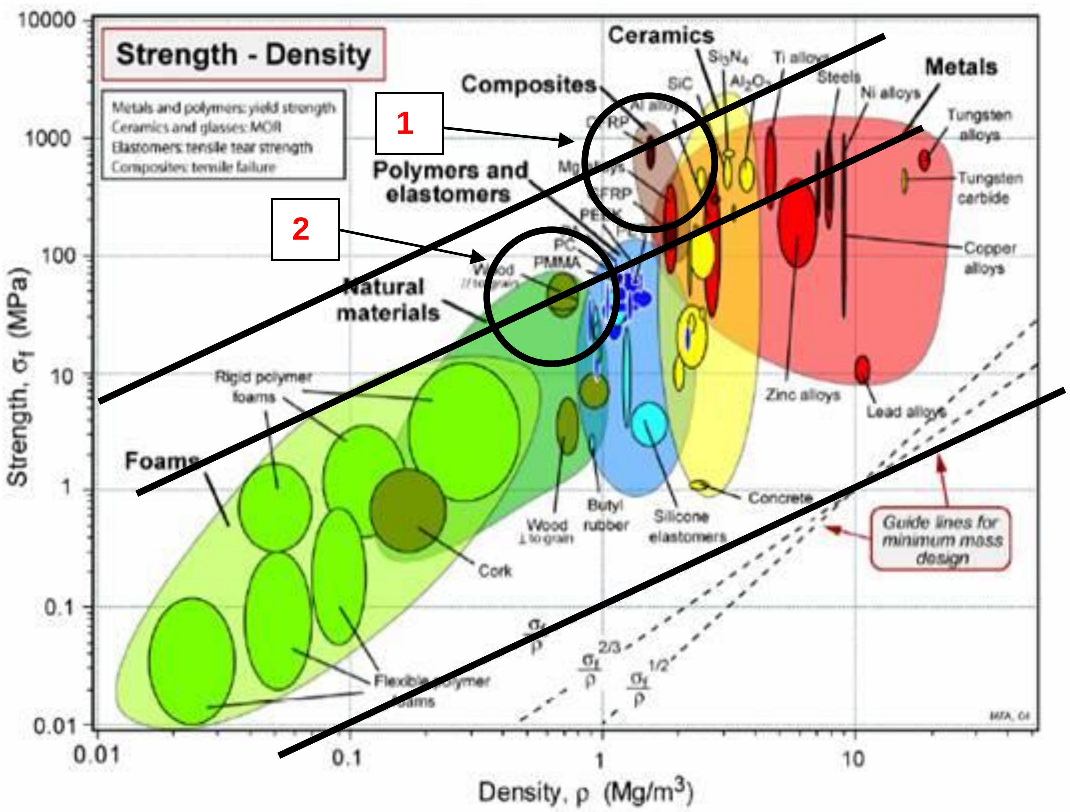


Figure 4.13: Material Strength and Density Chart

Structures Study

As previously stated, the optimal materials to be selected are both wood and carbon fiber. With intent to minimize weight, research was first performed on the lightest available materials. It was discovered that the three most commonly available woods are Balsa, Beechwood, and Sitka Spruce. There was also a large amount of data available for these woods, allowing for the most accurate modeling of structures while utilizing one or more of these three choices.

Table 4.9: Wood Property Comparison

Property	Material		
	Balsa	Beechwood	Sitka Spruce
Density (kg/m3)	130	398	387
Yield Strength (MPa)	25	32	37.8
Young's Modulus (MPa)	6	10	11.2

Considering the limited dimensions of the UAV, there is not much available space for structures. Acknowledging that Balsa wood is the least dense, and considering the other property values displayed above, it would be the best choice of the three materials. However Balsa wood will require a larger geometry in order to equalize the load bearing capacity of the other woods; which will simply not suffice in a cramped space. With this consideration, it was decided that Sitka Spruce possesses the optimal property values for the aircraft structures. To continue the trade study, structure options with regard to carbon fiber were then explored and analyzed. Up to this point, Sitka Spruce was the best choice and was used as the benchmark to compare other material options. Considering the strength and stiffness that carbon fiber offers, it is an ideal material. However, in comparison to wood, carbon fiber is a heavy material and must be used sparingly. Common structural practices utilize carbon fiber with consideration to structural shape and material hybridization with other lighter elements in order to increase structural efficiency. Considering the limited composite manufacturing experience of the team, intricate shapes were not an option, though shaping of the composite structures for rigidity was considered whenever possible. In order to minimize the weight of the carbon fiber hybrid structure, structural foam was modeled with carbon fiber as a sandwich structure. The effective density, Modulus of Elasticity, and Yield Strength were generated and compared with relation to core thickness and the number of sheets of carbon fiber that was used on either side of the sandwich panel.

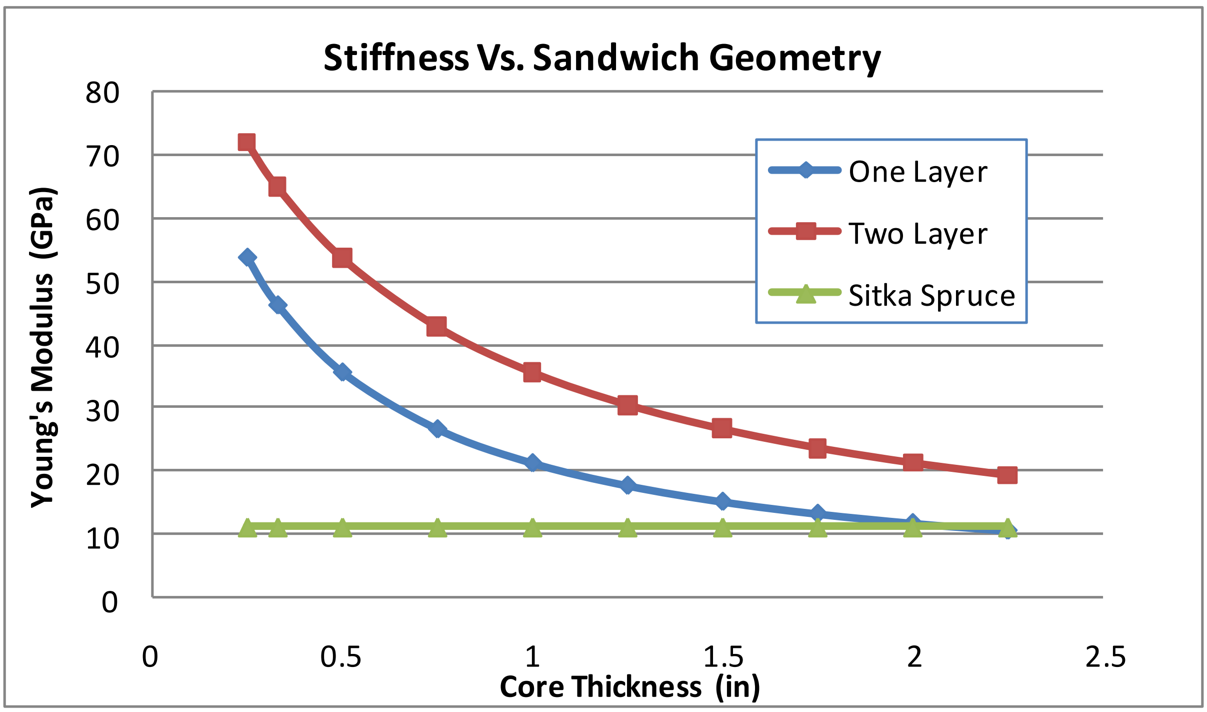


Figure 4.14: Material Young's Modulus Comparison

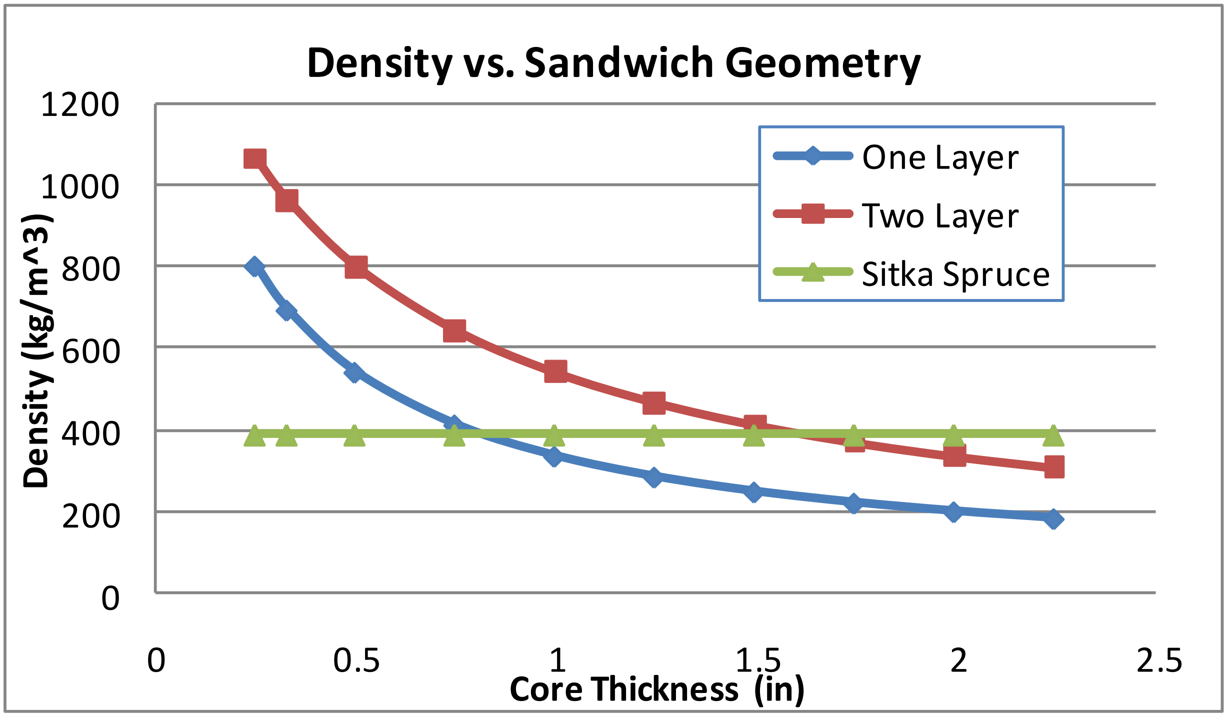


Figure 4.15: Material Density Comparison

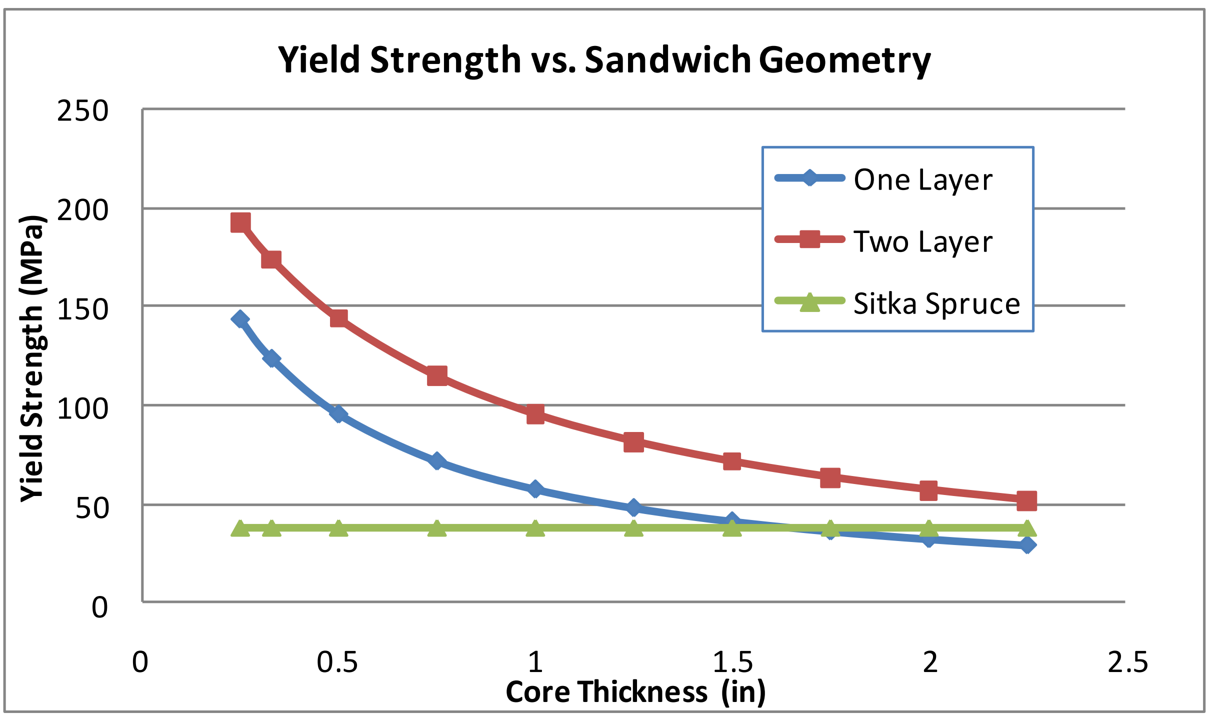


Figure 4.16: Material Yield Strength Comparison

As can be seen in the above graphs, with a core thickness of 0.75 inches the effective density of the carbon fiber / foam core sandwich is equal to that of Sitka Spruce while both the effective failure strength and effective Young's Modulus are twice that of Sitka Spruce. Due to these results, the desired material will be carbon fiber / foam core structures. The limiting factors for foam core structures are that they are sensitive to the orientation of the incident load and require more space than a stronger more dense non-hybrid materials to sustain an equal load. It was decided that a combination of simply shaped carbon fiber, CFRP foam core, and Sitka Spruce will be utilized in the construction of this aircraft.

Payload Configuration and Fuselage Structure

Steel bar payload

The competition requires that the steel bar payload be designed and supplied by each team. The steel bar must also be contained within the mold lines of the body while disassembled and stored in the suitcase. The steel bar must be simple to install and rigidly affixed within the fuselage while in flight as the center of gravity must not shift at any point in time.

Golf ball payload

The golf balls are supplied at the competition using USGA legal balls. The USGA rules state that the diameter of the golf balls cannot be less than 1.68 inches and the weight can be no more than 1.62 ounces. Intuitively the golf balls will be designed with minimal diameter to combat wind resistance and maximum weight to maximize momentum as to overcome wind resistance, however some variability in golf ball diameter must be allowed.

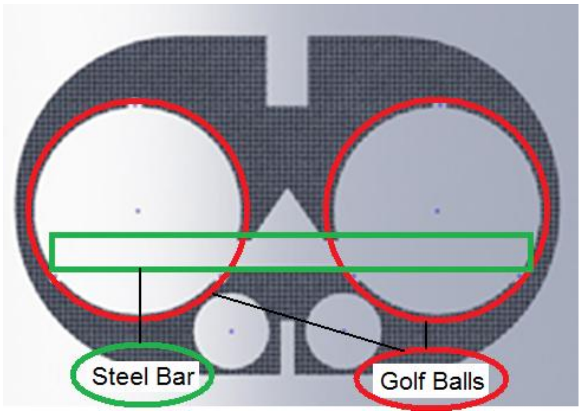


Figure 4.17: Payload Configuration Within Bulkhead

Initially, the preferred design involved storing the steel bar payload within the cross-section of the wing. This was thought to allow minimal structure reinforcement of the fuselage as the weight would be distributed along the wing. This design was abandoned as it would be difficult to store the payload within the wing while the aircraft is stored in the suitcase. The next and final choice involves utilization of a bulkhead system which provides structure to the aircraft but also allows for fixation of internal components and payloads.

4.6 Performance Prediction

Once the critical subsystem components were selected, it was possible to predict the aircraft's mission performance. The performance evaluation included recovery from hand launch, flight capabilities, and mission score.

One of the most critical aspects of the competition is the hand launch requirement and the aircraft's ability to quickly recover and reach stall speed. Utilizing the hand launch testing results which were previously discussed, it was determined that a human could throw an object weighing three pounds or less approximately 10 m/s. Since the aircraft was initially designed to weigh a total of three pounds when fully loaded, this value could be used to estimate initial launch speed. The team decided to incorporate a safety factor of 1.25 in order to account of error or inconsistencies. This reduced the estimated launch speed to 8 m/s for all predictions.

Figure 4.18 below illustrates how critical aircraft parameters adjust over time to the hand launch in Mission 1. For this mission, the aircraft empty weight was calculated to be two pounds. The parameters being evaluated include thrust, drag, aircraft velocity, lift, and distance traveled over the recovery span. The standard equations for lift and drag were implemented along with initial wing and fuselage sizing to calculate specific values and their variation over time. Dynamic thrust curves were used to interpolate how the thrust varies with increasing velocity. To determine aircraft velocity and displacement, Newton's Second Law and standard kinematics were employed. Each parameter was assessed simultaneously over half-second increments to estimate the time required for the aircraft to reach stall speed and to reach top speed.

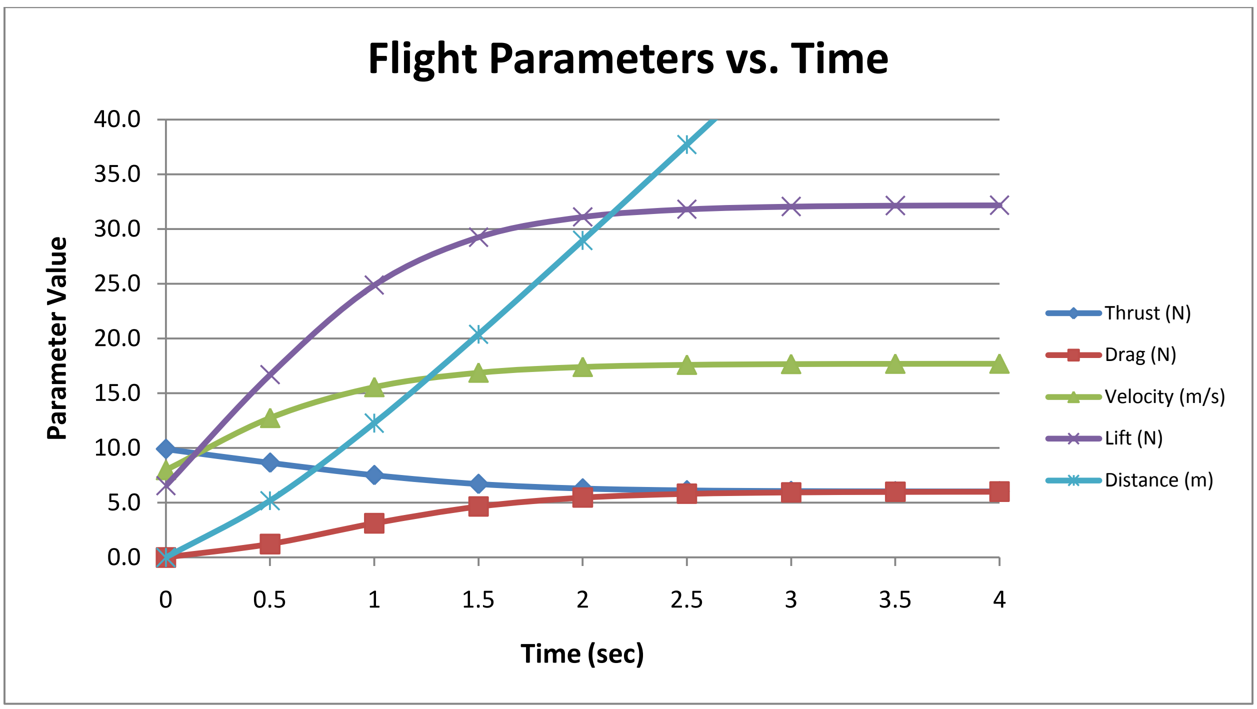


Figure 4.18: Mission 1 Recovery Model

As shown in the above figure, the aircraft has nearly obtained stall speed, calculated to be 8.4 m/s, immediately after launch. It only takes a tenth of a second for the aircraft to recover which is due to the relatively low weight of the aircraft and the high launch speed. Figure 4.18 also shows that the aircraft will reach top speed around 2.5 seconds after launch. This occurs when the thrust and drag have reached equilibrium.

Figure 4.19 below demonstrates similar trends in critical aircraft parameters during hand launch recovery for missions 2 and 3. These two missions were modeled the same since the total aircraft weight for both missions was designed to be equal at three pounds. Thrust, drag, aircraft velocity, and lift were all examined using the same methods discussed in the previous section. Aircraft displacement was also considered but the scale was too large for this figure.

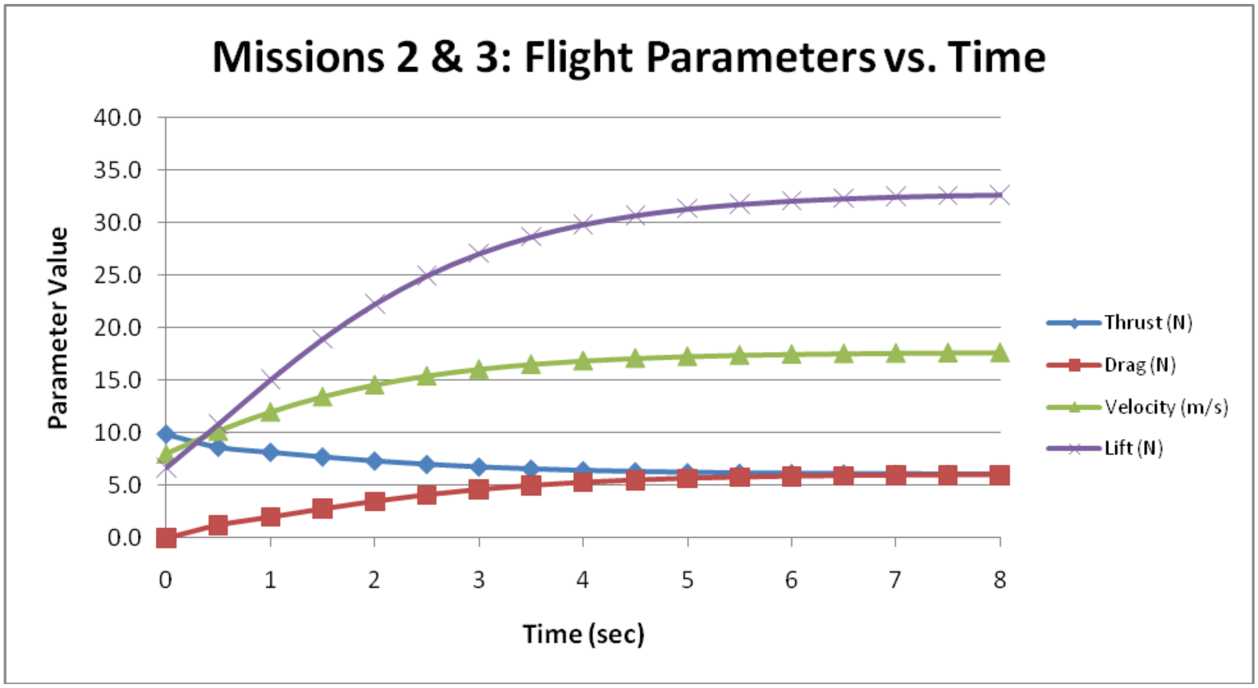


Figure 4.19: Missions 2 & 3 Recovery Model

As can be seen from the figure above, the hand launch recovery time is much more significant. Unlike mission 1, the aircraft did not obtain stall speed immediately after launch. For missions 2 and 3, the stall speed was determined to be approximately 13 m/s. This velocity is obtained in about one second after launch. This is sufficient for recovery time as the aircraft will have over two seconds before it hits the ground. The figure also shows that it will take approximately six seconds to reach top speed after launch.

After the launch recovery phase was assessed, the full mission predictions could be performed. Utilizing top speed data, course layout, turn rate, wind speed approximations, and adjustments in throttle settings, a reasonably accurate estimation could be made regarding mission flight time, number of completed laps, individual mission scores, and overall competition score. Estimates of maximum completed laps in mission 1 and maximum amount of golf balls carried were determined conservatively to be 6 and 30, respectively. These values were tabulated and are displayed in table 4.10 below.

Table 4.10: Predicted Mission Performance

Parameter	Mission 1	Mission 2	Mission 3
Reference Maximums	6 laps	-	30 golf balls
Empty Weight	2.00 lbs	2.00 lbs	2.00 lbs
Recovery Time	0.1 sec	1.0 sec	1.0 sec
Time to Top Speed	2.5 sec	6.0 sec	6.0 sec
Top Speed	58.7 ft/s	58.7 ft/s	58.7 ft/s
Turn Rate	78.6 deg/s	62.9 deg/s	62.9 deg/s
# of Completed Laps	4	3	3
Mission Flight Time	240 sec	182 sec	182 sec
Payload	-	1.00 lbs	10 golf balls
Total Weight	2.00 lbs	3.00 lbs	3.00 lbs
Mission Score	0.67	1.00	0.67
Overall Score		**1.65**	

5.0 Detail Design

5.1 Aircraft Dimensional Parameters

Table 5.1: Aircraft Dimensional Parameters

Overall Aircraft		Fuselage	
Total Length	23.83 in	Length	15.56 in
Total Width	47.98 in	Width	5.02 in
Total Height	10.24 in	Height	3.08 in

Wing		Horizontal Tail		Vertical Tail	
Airfoil	Eppler 422	Airfoil	NACA-0012	Airfoil	NACA-0012
Area	299.28 in^2	Area	64.73 in^2	Area	34.09 in^2
Span	48 in	Span	14.29 in	Span	7.41 in
Chord	6.96 in	Chord	4.53 in	Chord	4.6 in
Aspect Ratio	6.18	Aspect Ratio	3.15	Aspect Ratio	1.61
Ailerons		**Elevator**		**Rudder**	
Area	19.26 in^2	Area	29.15 in^2	Area	11.76 in^2
Span	17.04 in	Span	14.29 in	Span	6.22 in
Chord	1.13 in	Chord	2.04 in	Chord	1.89 in

Table 5.2: Component Specifications

Electrical System	
Transmitter	Aurora 9
Receiver	Optima 7 Ch Receiver
Servos	4 Power HD D65
Motor Controller	Castle Creations Phoenix 25A

Batteries		
Type	Receiver	Propulsion
Brand	KAN 400	Elite 1500
Capacity	400 mAh	1500 mAh
V	1.2 V	1.2 V
Imax	20 A	20 A
Number of Cells	5	10
Vpack	6 V	12 V
Weight	1.48 oz	8.10 oz

Motor	
Type	Himax HA2025-4200
Weight	2.82 oz
Kv	4200 rpm/v
Io	0.75 A
R	0.75 Ω
Power	175 W
Thrust	2.23 lb
Current Draw	15 A
Gear Ratio	6.6:1
Propeller	12 x 10

5.2 Structural Characteristics and Integration

The fuselage structure was designed to be lightweight while acting as a structural web that provides both airframe rigidity and support for the electronic components and internal payloads for each mission.

5.2.1 Structural System Integration

The goal behind the structure generation is to utilize all internal components to their fullest potential. This structural system integrates a series of bulkheads, tie rods, payload supports, landing platform, and the motor to compose a rigid aircraft fuselage. The bulk heads are faced toward the front of the aircraft in order to maintain the desired fuselage cross-sectional shape. Six of these bulkheads span the length of the aircraft which are joined by an upper and lower spine and six carbon fiber tie rod tubes. Four of the bulkheads, all six of the carbon fiber rods, and the spines are all composed of carbon fiber; two of the low load bearing bulkheads are made of Sitka Spruce. Joining the upper and lower spines at the front is the motor and heat sink. The motor is joined to the spines by small but high strength bolts which both fixes the motor to the aircraft and provides additional benefit utilizing the aluminum heat sink as a structural member. The landing platform is composed of a continuous sheet of carbon fiber which is fused to a thin layer of foam that is fixed to the lower section of each bulkhead and the lower spine.

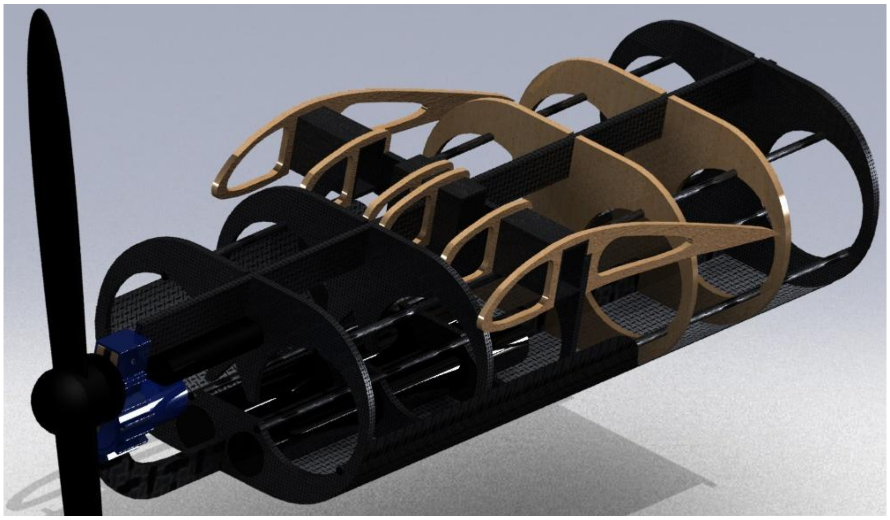


Figure 5.1: Aircraft Fuselage Structure

Wing Attachment

The wing attachment design allows for simple and quick wing installation. The wings are attached by series of three hollow carbon fiber couplers that are permanently fused to the primary bulkhead within the fuselage. In order to conserve space within the suitcase, a removable foam core beam is tightly slid into position between the three fuselage wing couplers. Once the removable beam is in position, the wings can slide into position flush with the fuselage. The seam of the wing is then taped both to prevent mishaps caused by axial forces along the wing and to comply with competition rules.

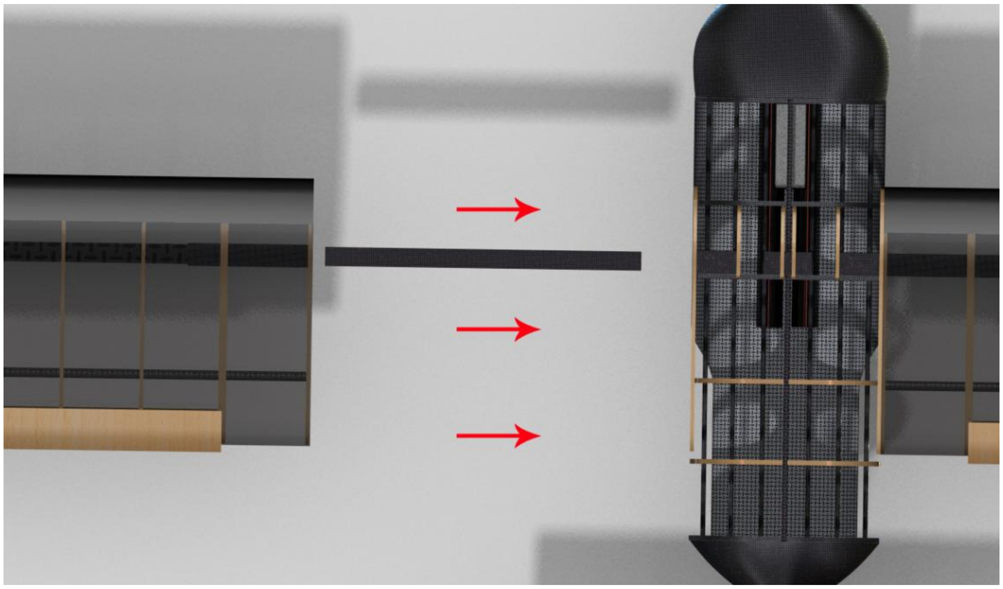


Figure 5.2: Wing Attachment

Payload Bay

The payload bay is composed of the bulkhead and tie rod system. The bulkheads allow for passage and securing of each payload item while the tie rods also constrain the payload to the desired location.

Golf ball payload

The golf balls are loaded from the rear of the aircraft into two rows that are oriented from front to back. The rear fairing and tail are removed and the balls are dropped in one at a time along the track that is composed of three carbon fiber tie rods per hole. A total of 16 golf balls are held within the payload bay. Once completely full, the fairing will be reinstalled and the balls will then be secure and ready for flight.

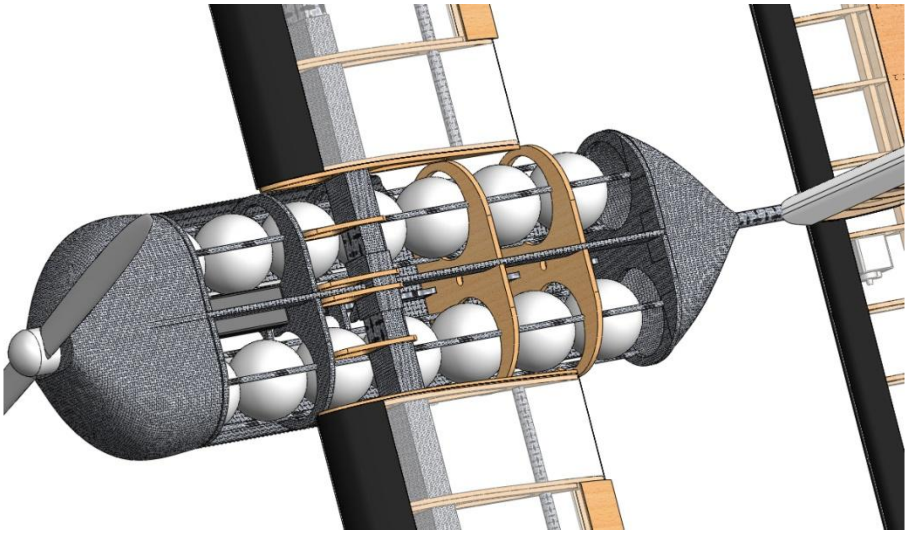


Figure 5.3: Golf Ball Configuration

Steel bar payload

The steel bar payload was designed to comply with mission dimensional requirements and is 4.25" wide, 5" long, and 0.25" thick. The steel bar does not need to be installed quickly as the mission requirement is that the bar must be preinstalled prior to approaching the staging box. Therefore, the steel bar payload was designed to be installed with the propeller, nose cone, and motor disassembled and detached from the fuselage. The leading bulkheads have a slot for the bar to tightly slide into place. Once the steel bar is inserted, the components can be reinstalled, thus securing the steel bar from moving forward. The steel bar is depicted below and highlighted in red.

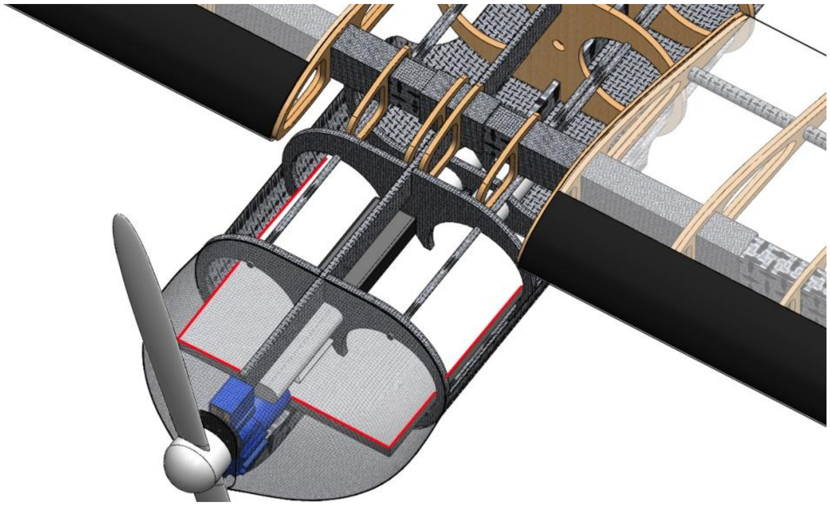


Figure 5.4: Steel Bar Configuration

5.2.2 Suitcase Component Layout

The layout of the disassembled aircraft in the suitcase can be seen in the following figure. This configuration allows for optimum placement of each component while ensuring a very minimal potential for damage. The suitcase dimensions are 21”x15”x9”.

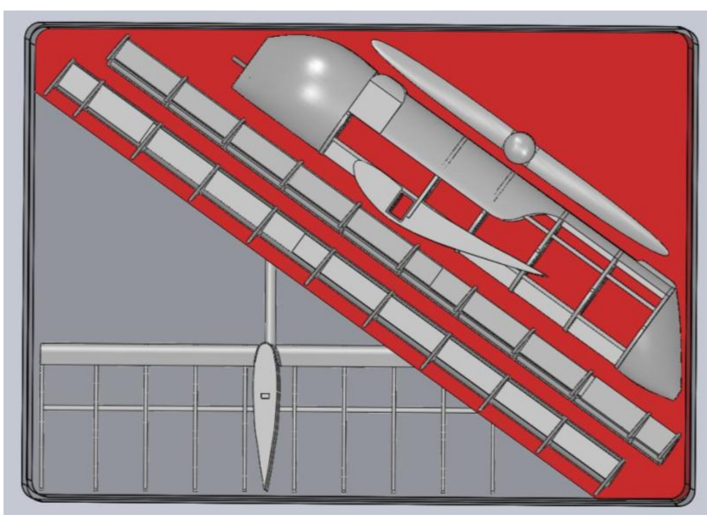


Figure 5.5: Component Suitcase Configuration

5.2.3 Control System

The control system was designed around the restriction that the propulsion system had to be powered separately from the other control systems. That is, two batteries are used instead of one and a Battery Eliminator Circuit (BEC). The diagram below shows how all the components were connected together.

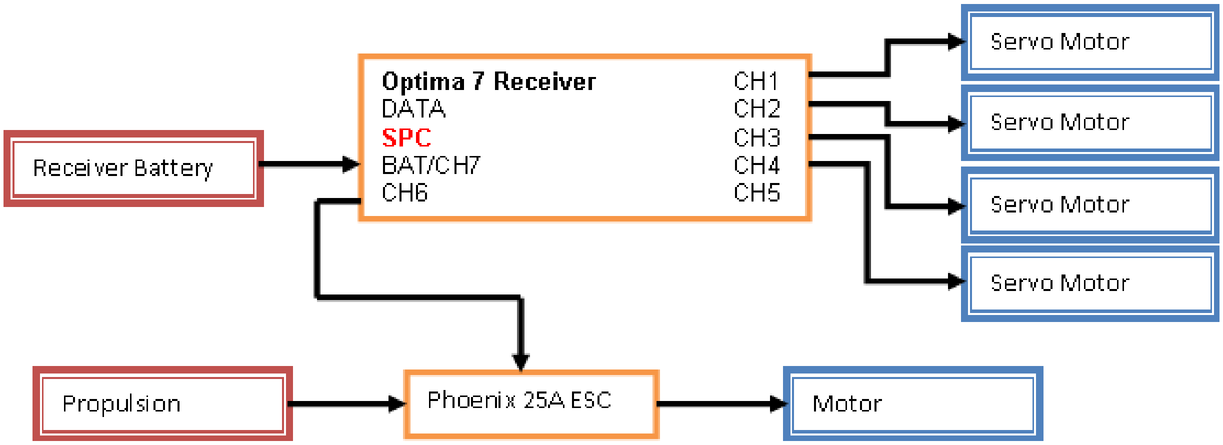


Figure 5.1: Control System Diagram

5.3 Weight and Balance

The aircraft was designed to maintain the same center of gravity for all three missions. This allows for more predictable balance and control characteristics regardless of the designed payload. As can be seen in Figure 5.6 below, the center of gravity is located at 33% of the chord behind the leading edge of the wing, which is approximately the same location of the center of lift.

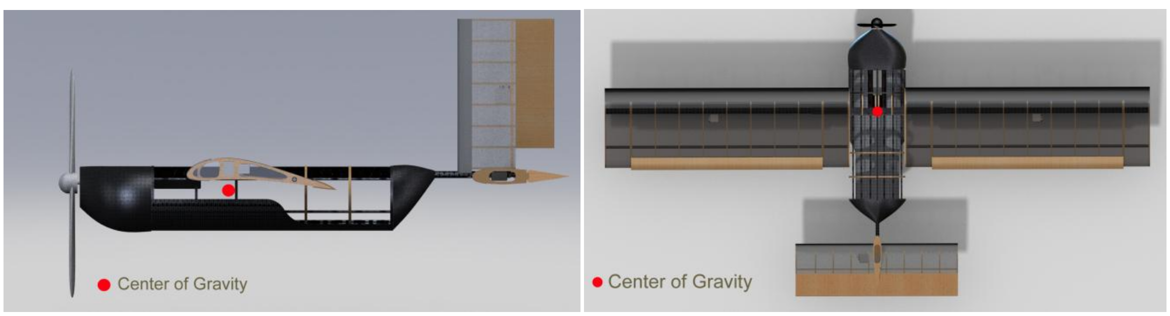


Figure 5.6: Aircraft Center of Gravity

5.4 Flight Performance Parameters and Mission Performance

Through simple testing schemes and changes to the design, the aircraft weight was reduced by 0.4 pounds. This was possible because the aircraft structure was initially designed conservatively in order to properly withstand impact during landing. Lighter materials were used in certain locations in which structure wasn't critical. Weight reducing cuts were also made in certain locations in order to minimize weight.

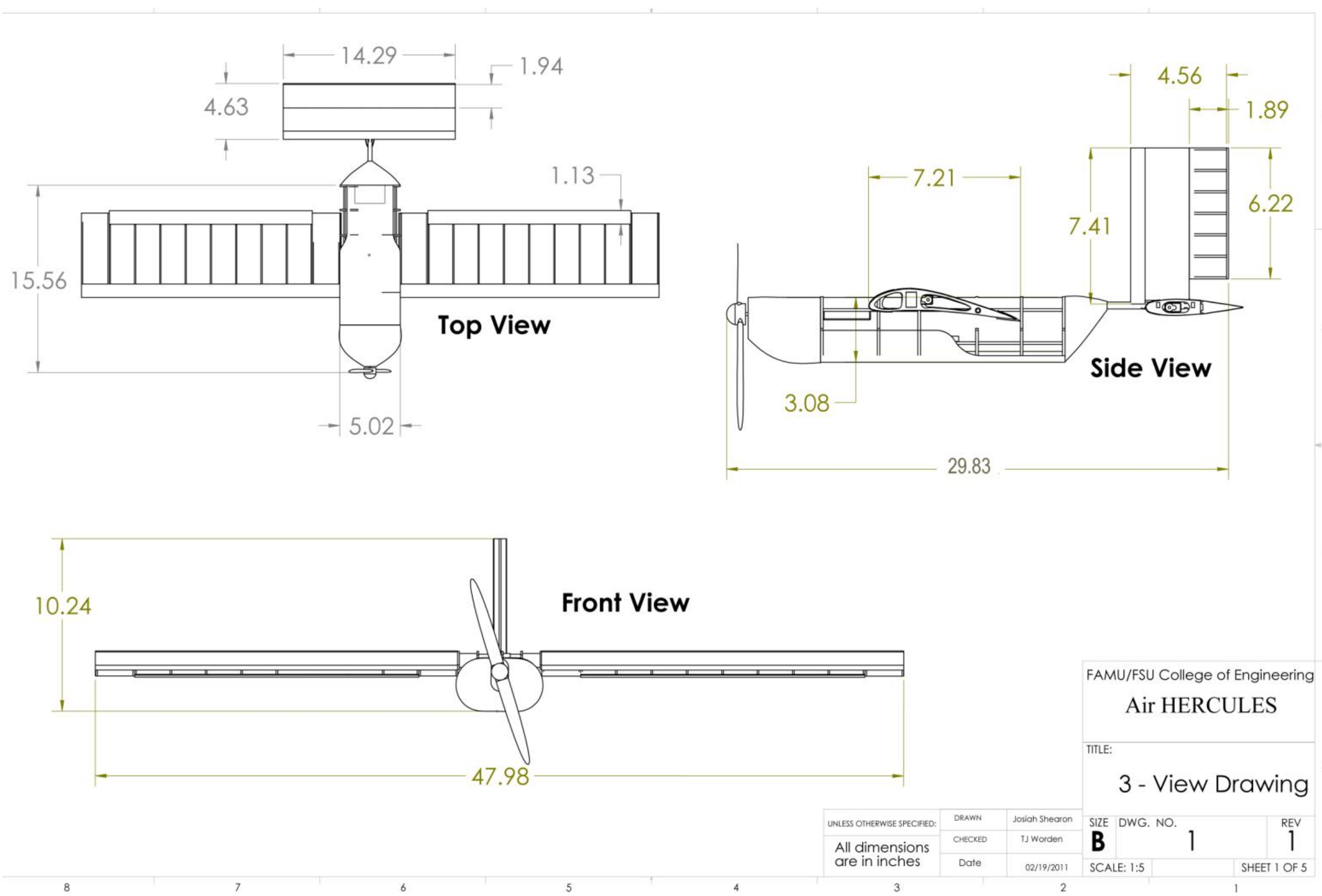
Table 5.3 below outlines the flight parameters and performance prediction for each mission. As can be seen, the mission score improved by 40% when compared to the preliminary design performance assessment. This can be contributed to the reduction in empty weight which also allowed for more payload.

Table 5.3: Flight Parameters and Mission Performance

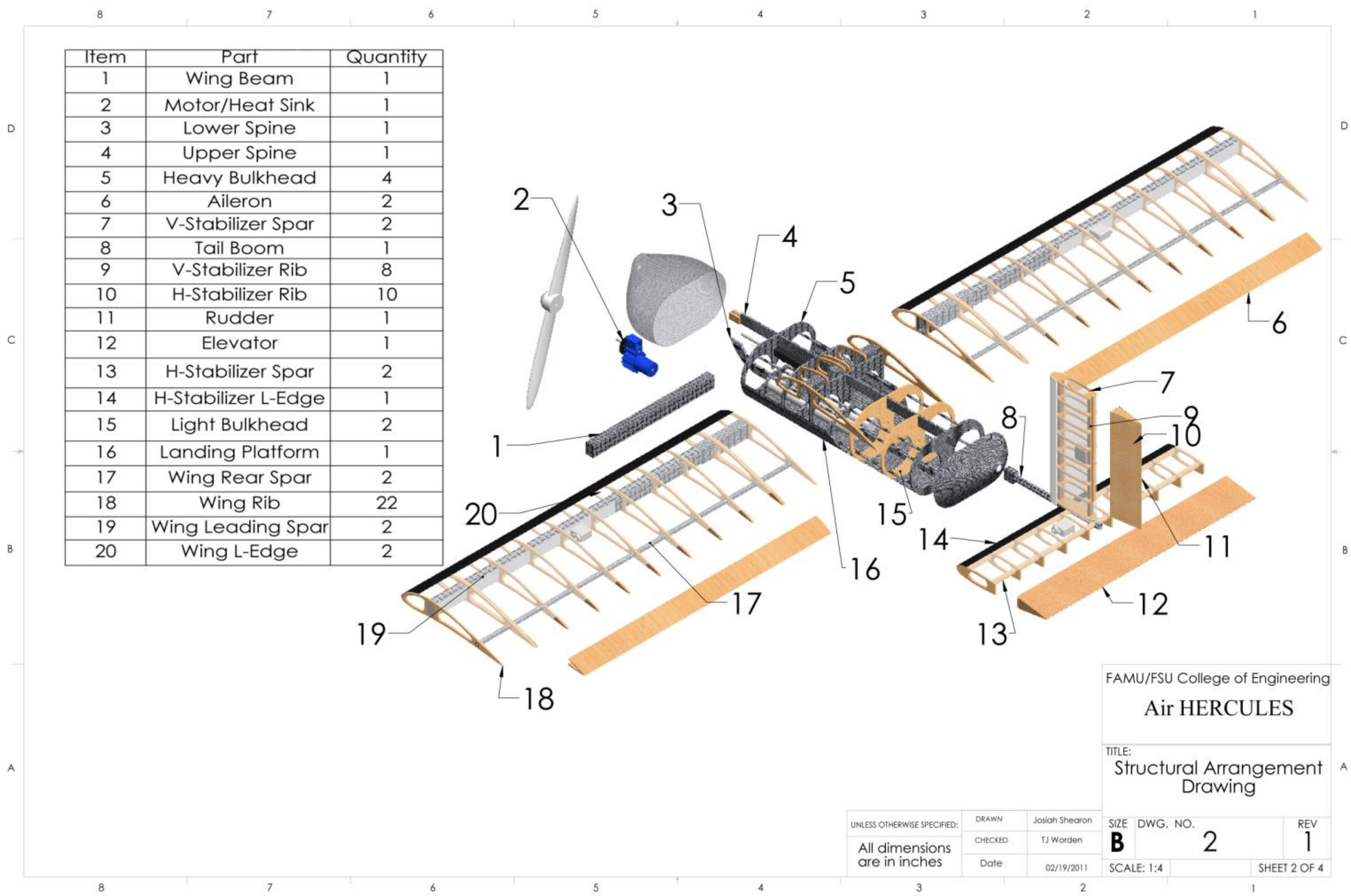
Parameter	Mission 1	Mission 2	Mission 3
Reference Maximums	6 laps	-	30 golf balls
Empty Weight	1.60 lbs	1.60 lbs	1.60 lbs
Recovery Time	-	1.0 sec	1.0 sec
Time to Top Speed	3.0 sec	6.0 sec	6.0 sec
Top Speed	58.7 ft/s	58.7 ft/s	58.7 ft/s
Turn Rate	86.7 deg/s	62.9 deg/s	62.9 deg/s
# of Completed Laps	4	3	3
Mission Flight Time	240 sec	182 sec	182 sec
Payload	-	1.40 lbs	16 golf balls
Total Weight	1.60 lbs	3.00 lbs	3.00 lbs
Mission Score	0.67	1.40	0.93
Overall Score	**2.37**		

5.5 DRAWING PACKAGE

5.5.1 3 View

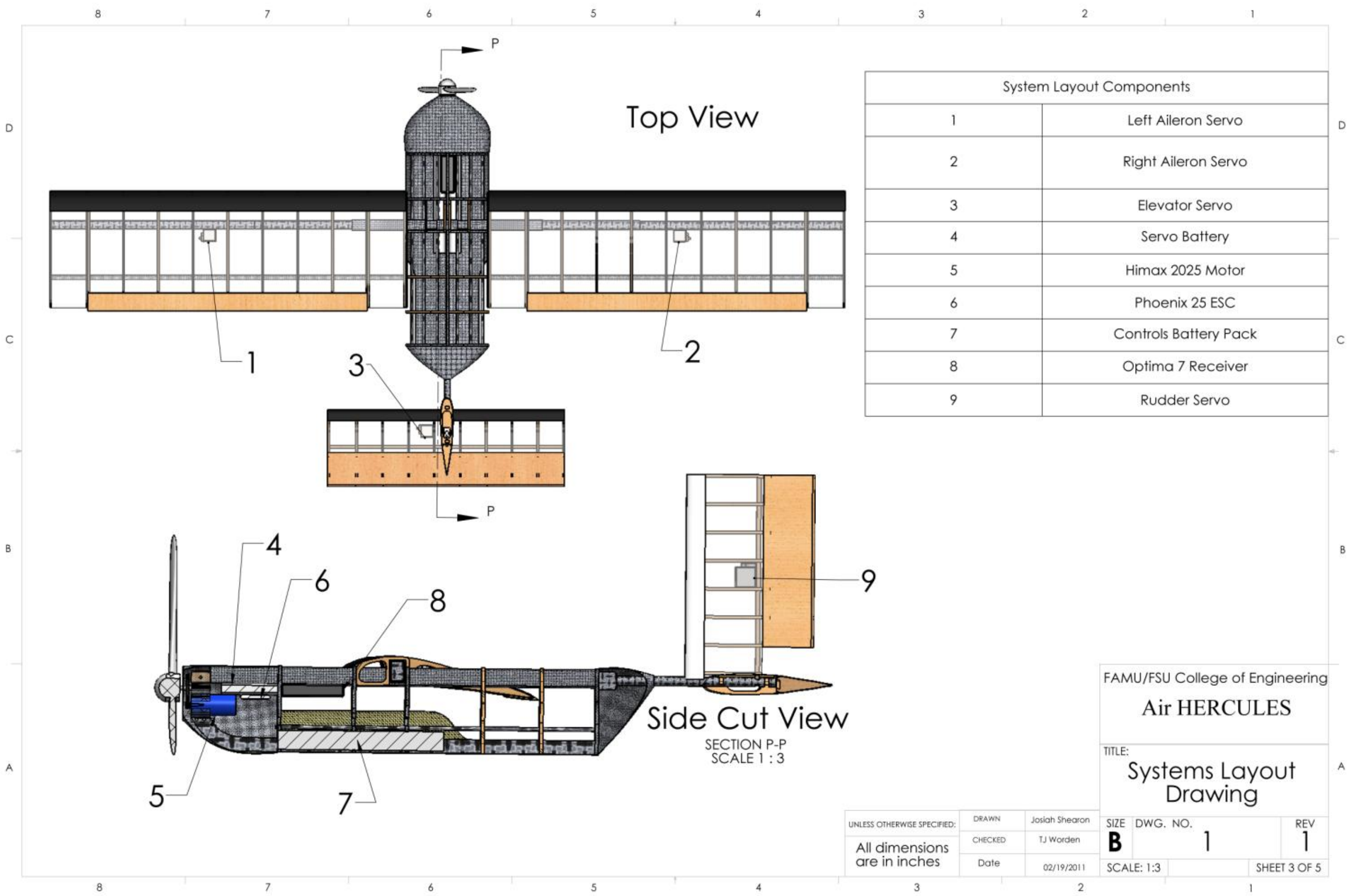


5.5.2 Structural Arrangement

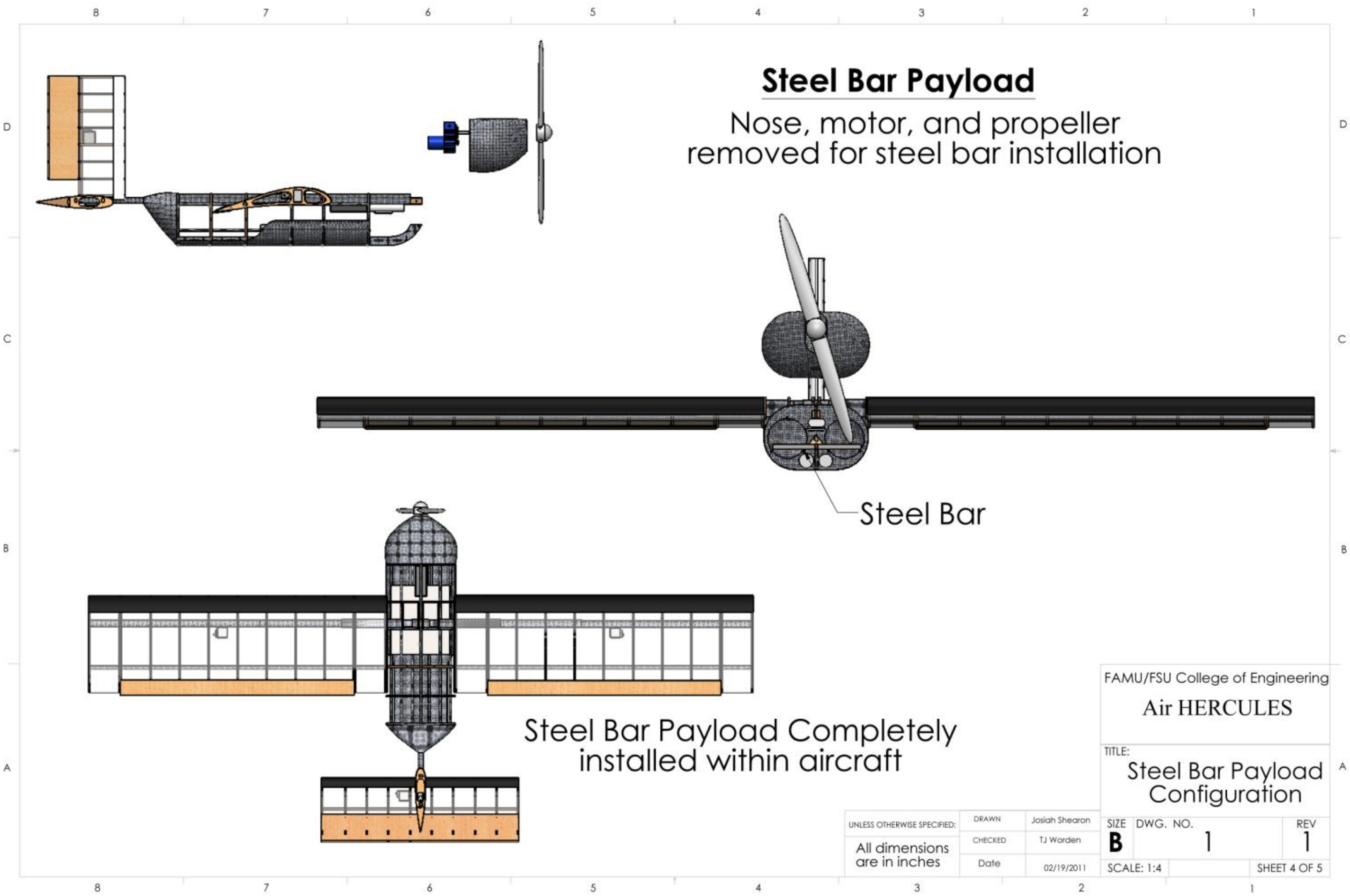


Item	Part	Quantity
1	Wing Beam	1
2	Motor/Heat Sink	1
3	Lower Spine	1
4	Upper Spine	1
5	Heavy Bulkhead	4
6	Aileron	2
7	V-Stabilizer Spar	2
8	Tail Boom	1
9	V-Stabilizer Rib	8
10	H-Stabilizer Rib	10
11	Rudder	1
12	Elevator	1
13	H-Stabilizer Spar	2
14	H-Stabilizer L-Edge	1
15	Light Bulkhead	2
16	Landing Platform	1
17	Wing Rear Spar	2
18	Wing Rib	22
19	Wing Leading Spar	2
20	Wing L-Edge	2

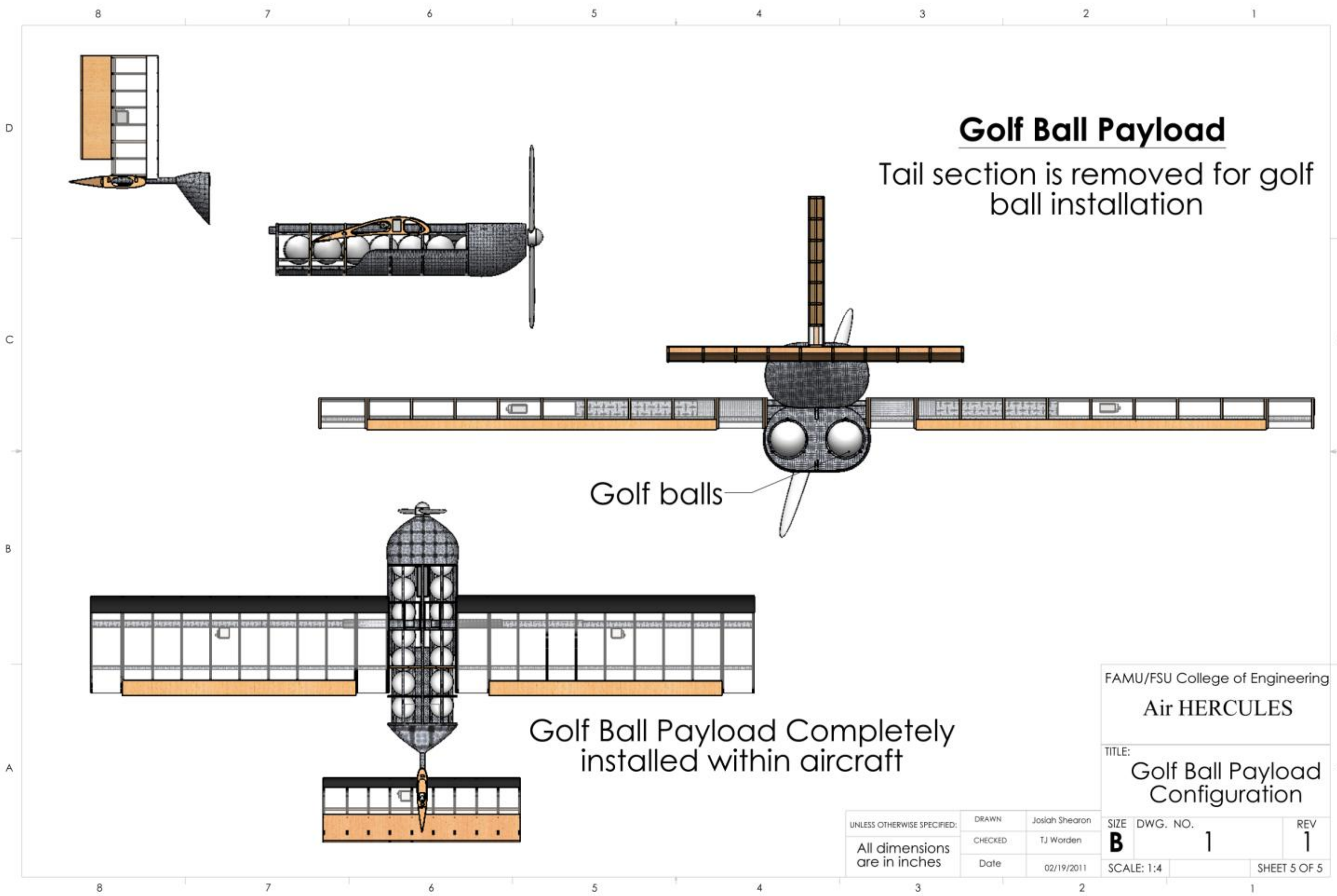
5.5.3 <u>System Layout</u>



5.5.4 **Steel Bar Payload**



5.5.5 Golf Ball Payload



6.0 Manufacturing Process

The manufacturing processes selected were explored in great detail in order to generate the most durable and precise final products. Fortunately, the design team was provided access to the precision manufacturing equipment at the High Performance Materials Institute, which allowed for the use of cutting-edge fabrication techniques without the need to invest in expensive equipment. To this end, CAD and CNC tools were utilized for mold and bulkhead fabrication. Carbon fiber and foam are two major materials utilized in this aircraft design, and access to HPMI provided access to the necessary fabrication tools to create these structures. Three primary methods of carbon fiber fabrication were considered.

- **Ambient pressure wet layup** – Wet-layup fabrication is achieved by placing the carbon fiber and other materials into a mold and pouring resin onto the surface. The excess resin is removed manually and the part is allowed is cure.
- **Wet layup W/ vacuum bagging** – This process takes one step further than the ambient pressure wet layup. After the carbon fiber is set up and the excess resin is manually removed an airtight bag and other materials are added to the layup and vacuum is applied. Excess resin is drawn out of the fibers and the part is allowed to cure.
- **Vacuum Infusion** – Vacuum infusion is similar to the vacuum bagging process with the exception of the resin transfer to the carbon fiber cloth. The carbon fiber is set in place dry and bagged. Once sealed, vacuum is applied to the bag and all of the contained air is sucked out. Once all leaks are accounted for and removed, the vacancies in the bag is allowed to draw the resin through the layup and the carbon fiber is infused with only as much resin as is needed.

Though all three of the previous processes will render an acceptable product, the lightest and strongest structures are generated using the vacuum infusion process. Due to this, the selected carbon fiber manufacturing technique was vacuum infusion.

6.1 Aerodynamic Components

Three main aerodynamics components need to be manufactured by the team: the wings, the fairings, and the control surfaces. Both the main wings and the tail wings were designed to be composed of a carbon fiber foam core spar, Sitka Spruce airfoil ribs, leading and trailing edge supports, and Microlite heat activated shrink wrap. The carbon fiber foam core spar will act as the main structural member supporting the load and, for the main wings, will extend into the fuselage for attachment. The Sitka Spruce airfoil ribs are then attached to the spar two inches apart using lightweight epoxy. These ribs will have weight reducing cuts where there is no need for support structure. The leading and trailing edge support is made of lightweight cardstock. The purpose of this component is to maintain the shape and structural integrity of the leading and trailing edges. Finally, the wing frame is wrapped in Microlite to create a consistent airfoil shape. Microlite is a thin plastic film which shrinks and activates an adhesive when introduced to sufficient heat.

The methodology for manufacturing the fuselage fairings utilizes the vacuum infusion technique. A female mold was fabricated out of machinable wax using a CNC machine which was then used in vacuum infusion to create the fairing shape out of a carbon fiber sheet. When the fairing has cured, it is removed from the mold and assembled to the nose and rear of the aircraft using pin connectors.

The final components manufactured for aerodynamics was the control surfaces. These surfaces were first sized and modeled in CAD then fabricated out of thin Balsa wood sheets. The ailerons, elevator, and rudder were all similarly constructed. The shape was determined by the trailing edge of each airfoil shape and the chord position of the surface. Balsa wood was oriented at the proper angles, adhered using epoxy, and enclosed with Microlite. The control surfaces were attached to the wing and tail using plastic hinges which extended into both sections.

6.2 Manufacturing Milestone Chart

In order to remain in line with project requirements and timeline, a milestone chart was assembled detailing the manufacturing order and processes. Figure 6.1 below displays this timeline in more detail.

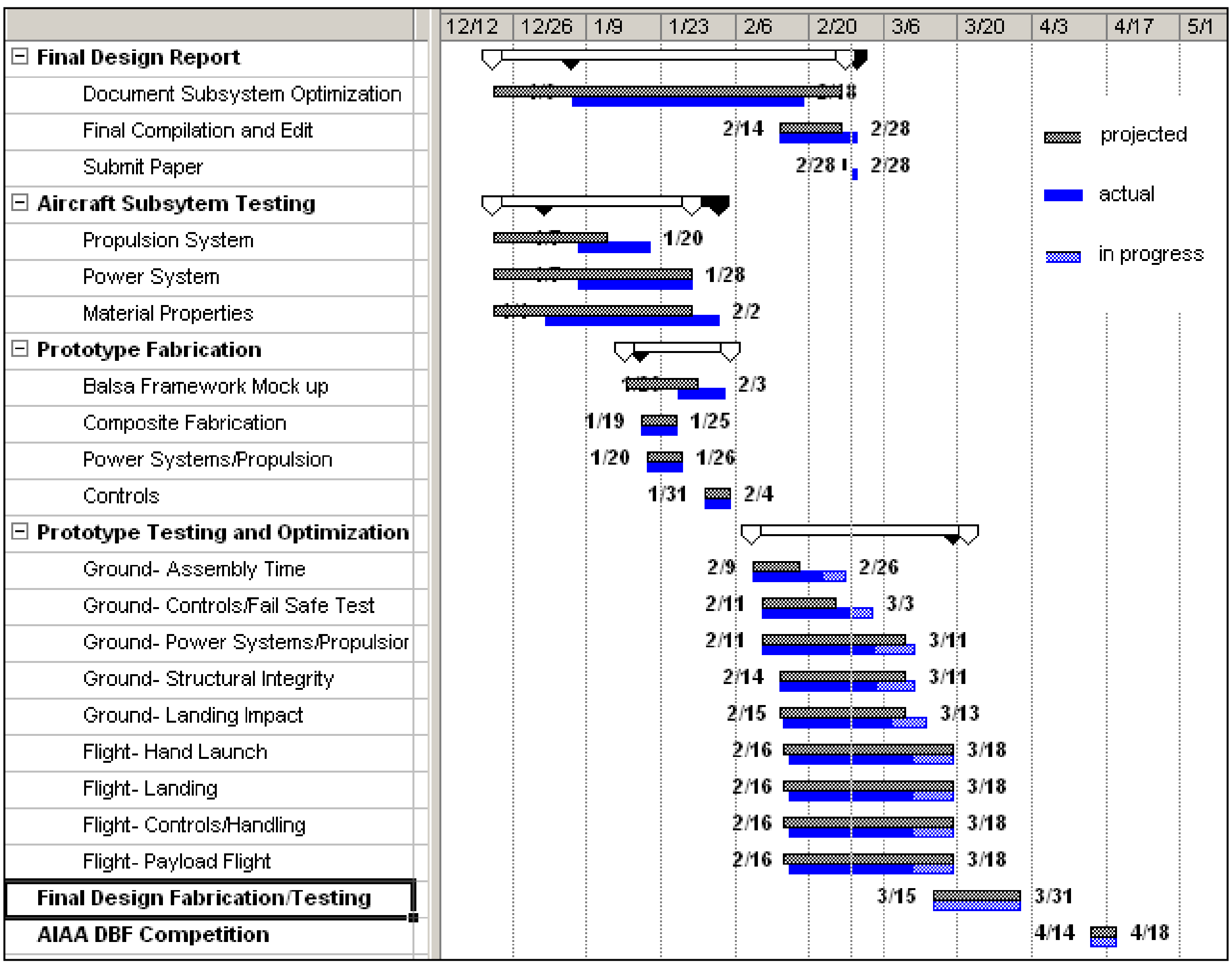


Figure 6.1: Manufacturing Schedule

7.0 TESTING PLAN

Throughout the design process, several tests were conducted to verify predictions and approximations. These tests allowed the team to make necessary changes to the aircraft design in order to finalize an optimal and practical solution.

7.1 SUBSYSTEM TESTING

Power and Propulsion System

Numerous tests were implemented on both the servo and propulsion battery packs in order to discern realistic flight times and necessary time for charging. The batteries were connected to the Himax motor and the transmitter was set to full throttle for the time required to fully drain the battery. While the battery was powering the motor and propeller, data was recorded for current draw, output power, and thrust generated, among many other parameters. These tests confirmed the estimated flight endurance time of five minutes with power drastically dropping after that time.

Aerodynamics

Critical aerodynamic components such as the wings, the fuselage fairings, and control surfaces were tested to ensure sufficient functionality. The wing cross-section was placed in a wind tunnel at various wind speeds to determine lift and drag forces at various angles of attack. Smokewire flow visualization was qualitatively used to determine the point of boundary layer separation at the critical angle of attack.

The fuselage fairings were tested in a similar manner. A scaled blunt body was placed in a wind tunnel without fairings and drag was measured. Then the fairings were mounted and drag force was again measured. The two were compared and it was easily discernible that the fairings drastically improved aerodynamics by reducing drag nearly 50%.

The control surfaces were tested in ambient conditions to evaluate system functionality. The transmitter was programmed at several different sensitivity settings to determine which settings were most appropriate. Further testing in a wind tunnel and in flight would be needed to completely evaluate the control surfaces.

Structures

Several structural members were tested to ensure the structural integrity of the aircraft. The landing platform was of main concern due to the no landing gear approach. The fuselage was design to be impact and abrasion resistant but drop and skid tests were performed to validate design intentions. Wooden prototypes were first configured and dropped at various heights, with and without payload. Balsa wood was the lightest material considered but also the weakest. The objective of this prototype was to determine what loads the material could handle. Skid testing was executed by sliding the undercarriage of the fuselage on a cement surface similar to an aircraft runway. Carbon fiber was the material used for this platform and proved extremely abrasion resistant, even when fully loaded.

The wing spars were also tested for structural capacity. It is required that wings be able to withstand and 2.5g load and this will be simulated by holding the aircraft up at the wing tips. Initial testing of the wing spars was conducted by fixing a wing section to a surface and adding weights incrementally. This was performed to the point of failure to gain a better understanding of the spar strength and areas in need of improvement.

System Packaging and Assembly / Disassembly

In this year's competition, it is critical that the aircraft be able to be packaged into a carry-on suitcase of specified dimension when disassembled. The UAV must then be assembled in under five minutes and be ready to fly. Several tests were executed in which the team packaged the aircraft inside the suitcase, expose the case to loads reasonable for general purpose handling, remove the components, assemble the aircraft, and prepare for launch. The most critical lesson learned from this testing methodology was that the suitcase design was deforming some of the Microlite covering on the wings by poking small holes or stretching the material. This was resolved by lining the interior with impact resistant foam.

7.2 Continued Testing Schedule

Table 7.1 below outlines the testing schedule for 2011. It involves every aspect of the design and its purpose was to keep the team on track and to validate design predictions.

Table 7.1: Testing Schedule

Test	Motivation	Start date	End date
Propulsion System	Performance verification	1/4	1/20
Power System	Performance verification	1/4	1/28
Material Properties	Durability and Strenght	1/2	2/2
Ground Assembly Time	Decrease assembly time	2/9	2/26
Ground Controls	Preflight certification	2/11	3/3
Ground Power Systems/ Propulsion	Preflight certification	2/11	3/11
Ground Structural Integrity	Preflight certification	2/14	3/11
Flight Hand Launched	Recovery Assessment	2/15	3/13
Flight Landing	Structural Assessment	2/16	3/18
Flight Controls and handling	Handling Evaluation	2/16	3/18
Flight Payload Flight	Capacity Assessment	2/16	3/18
Final Aircraft Testing	Finalize System	3/15	3/31

System integration testing was conducted on the aircraft to assess compatibility and overall effectiveness of the design. Ground testing was first implemented which includes controls testing, landing impact testing, propulsion evaluation, and assembly. Flight tests were also performed which involves hand launch recovery testing, handling and controls, landing assessment, payload testing, and mission performance evaluation. Both ground and flight test were executed throughout the remaining weeks until competition in order to produce the most flight-ready aircraft for competition. The following were answered during the testing evaluation for each sub-system.

Controls and Electronics

- Do all control surfaces perform as desired?
- Are transmitter adjustments needed for throttle and control sensitivity?
- Fail-Safe Maneuver Check
- Does the aircraft handle as expected? What is maximum control surface deflection and turn rate before stall?

Aerodynamics

- Do the wings produce sufficient lift for all 3 missions?
- Does the tail produce adequate stability and control?

Power and Propulsion System

- Does the propulsion system produce necessary and expected thrust?
- Verify flight endurance time
- Is top speed as expected?
- Does transmitter throttle settings very thrust as expected?
- Does the aircraft recover from hand launch?

Structures

- Can the landing platform withstand the impact during landing phase? Abrasion?
- Can the wing spars support a 2.5g load? Can they be picked up from tip without deforming?
- Does the payload bay adequately support golf balls and steel bar payloads?
- Do the bulkheads withstand landing impact force without fracture?
- Is the propeller damaged?

Each of the previous check-list items were thoroughly analyzed through an iterative process until the desired product was fully developed. If a problem or malfunction was encountered, the necessary steps were taken to resolve the issue effectively while maintaining aircraft integrity. Several changes to the aircraft were expected due to these testing procedures.

7.3 Changes Due To Testing

The results of the continued testing divulged that many changes were necessary to the prototype in order to increase flight effectiveness and reliability.

- **Center of gravity**

  The first concern was the center of gravity. There were errors in the prototype development process which created problems with weight distribution resulting in inaccurate center of gravity placement. The aircraft was tail heavy and caused difficulty in maneuvering the aircraft during initial flight testing. The cause of the error was located and corrected.

- **Control Surface Hardware**

  During normal flight in gusty conditions, the prototype was difficult to maneuver. It was discovered that the cause for this maneuverability problem was loose hinges on the control surfaces, weak tie rods from the servo to the control surface, and low torque availability from the servo motors. In response to these problems, new, more robust, control surface hinges, tie rods, and servos were installed.

- **Tail Boom**

  Another concern with regard to stability and control was the tail boom length. The shorter the tail, the more responsive the yaw and pitch controls are. It was shown that these controls were too responsive/touchy for the pilot to reliably maneuver with the original designed boom length. The boom length was extended by 8 inches. This change resulted in a much smaller learning curve for the pilot and much more reliable aircraft maneuverability.

After each of these concerns were addressed, a series of final aircraft flight and landing tests were repeatedly performed. Small changes were made in response to different concerns that were raised during the flight tests. Details of these flight tests are explored in the next section.

8.0 PERFORMANCE RESULTS

8.1 SUBSYSTEM PERFORMANCE

The testing discussed in the previous section was implemented and the results and performance are discussed here. The first testing involved the propulsion system in which the thrust values provided by the manufacturer were confirmed. The propulsion and servo battery packs were also tested to determine the endurance limits at various throttle settings and mission strategies. An Eagle Tree data logger was implemented to record several parameters in real-time such as current draw, motor power and rpm, airspeed behind the propeller, and thrust generation. Results from this testing are included in section 4.5. Figure 8.1 below reveals the testing apparatus for the propulsion system.

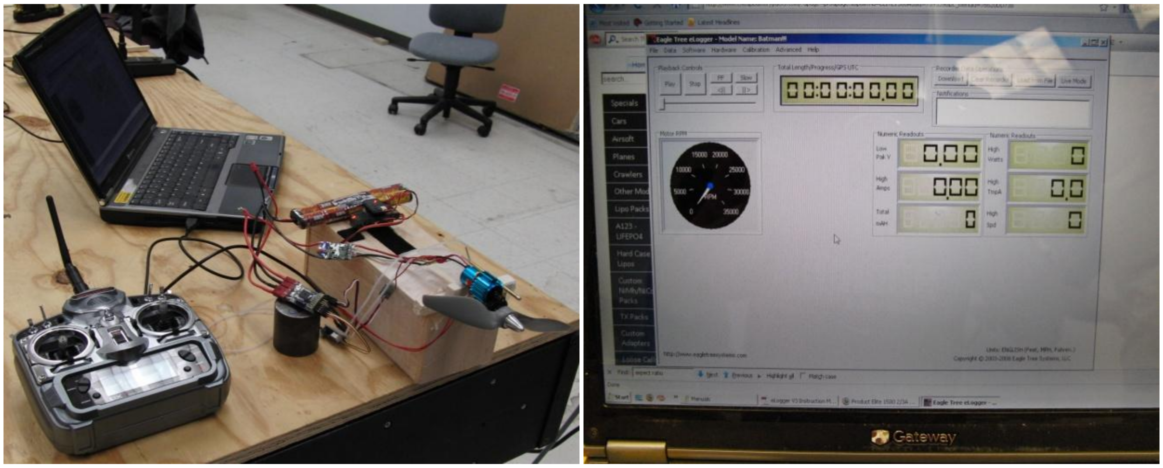


Figure 8.1: Propulsion system testing setup (left) and data logger interface (right)

Other completed testing includes structural assessment of various aircraft components. The aircraft wings were evaluated by fixing the extended spars and placing weights at the tip of the wing. The purpose of this test was to simulate a 2.5g loading on the wing and to verify it will pass safety inspection when the aircraft is lifted from the wing tips. Weights were placed in 100 gram increments until failure occurred at 300 grams. This proved that the current design would not be able to withstand the required load and would have to be re-designed. Figure 8.2 below depicts the wing testing methodology and the results of deformation.

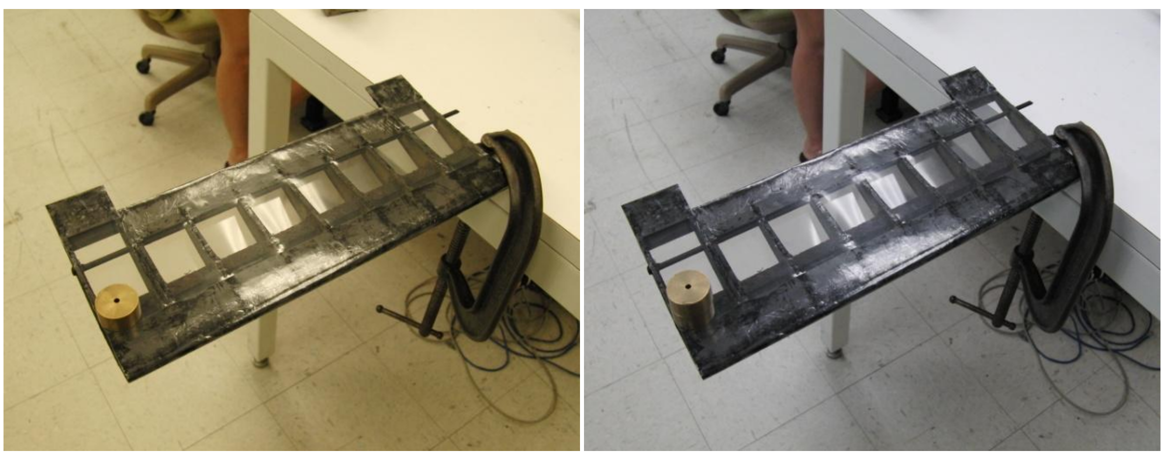


Figure 8.2: Wing spar structure testing progression. 100g load (top left), 200g Load (top right), 300g load (bottom left), fully deformed (bottom right)

Another structural test which was performed involved drop testing of the fuselage and payload bay when fully loaded with golf balls. Two separate prototypes were tested: a balsa wood frame and a primarily carbon fiber frame. The fuselage framework was fully assembled and 16 golf balls were secured within the body for impact analysis. The system was dropped in one-foot increments starting at a one foot height from the cemented landing surface. The weight of the system was made to be three pounds in order to realistically simulate in-flight conditions. After each drop, the landing platform, the bulkheads, the payload bay dowels, the motor and other load supporting structures were inspected for damages.

The balsa wood prototype was destroyed on the first test from a one foot drop height. The fracture occurred along the grain of the wood which was oriented vertically. It was learned from this design that the grain of the wooden bulkheads would need to be oriented horizontally in order to better resist impact forces.

For the carbon fiber body, the first sign of damage did not occur until the aircraft was dropped from five feet. The Sitka Spruce bulkhead developed a fracture along one of the grain lines which eventually broke off after further testing. It should be noted that the wood grains on this prototype were oriented horizontally and proved to be much stronger. Abrasion testing was also conducted to evaluate the effectiveness of the landing platform. No damages were incurred during this testing.

It was concluded from the drop tests that the balsa wood body, while extremely lightweight, was not structurally adequate to support the impact loads experienced during landing. It was also concluded that the carbon fiber body was overly designed for supporting impact loads. This development would allow for the aircraft to be designed lighter and more testing is still needed to determine what system will be both structurally sufficient and lightest weight. This allowed the utilization of thinner carbon fiber bulkheads as well as upper and lower spines, resulting in a much lighter fuselage. Figures 8.3 and 8.4 below display the methodology and results of the drop tests.

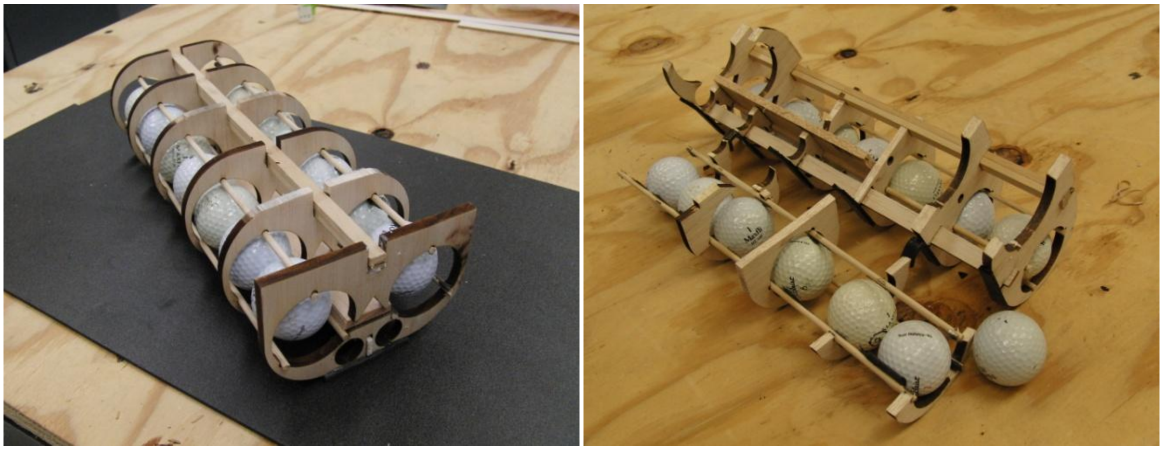


Figure 8.3: Balsa wood fuselage payload bay prototype drop testing results

Figure 8.4: Landing Platform Drop Testing – 1 ft drop (left) and 2 ft drop (right)

Figure 8.5: Landing Platform Drop Testing – 5 ft drop (left) and wooden bulkhead fracture (right)

8.2 Aircraft Performance

The first phase of each mission is the hand launch and this system was designed and optimized through several iterations to fully satisfy this condition. Flight testing began by testing this phase to verify design predictions. Shorty before being launched, the propulsion system was set to full throttle. When the aircraft was unloaded to simulate mission 1, the UAV got immediately in the air and appeared to have zero recovery time. This confirmed predictions and simulations in the design phases. For missions 2 and 3, the aircraft was weighed down by the payload. In this case, the UAV did have a noticeable recovery time; however, after a slight loss in altitude, the system reach stall speed and began to ascend. This took approximately one second as predicted earlier in the design phase.

Other important parameters which were assessed included the handling of the aircraft, the flight time, ability to overcome wind, and the landing phase. The aircraft control surface hardware appeared to be slightly smaller than needed for the desired maneuverability. This didn't prove to be a major issue but was addressed in later modifications as shown in section 7.3.

It was estimated that the aircraft would complete four laps in mission 1 and would take three minutes to complete three laps in missions 2 and 3. This approximation assumed a 5 mph wind speed and on the day of testing, the wind was near 15 mph. This obviously affected the aircraft's performance but it successfully completed 5 full laps in mission 1 and required 5 minutes to complete the three laps in the second two missions utilizing slow, methodical, and careful flight maneuvering. The battery lasted throughout all flight tests and only required a 20 minute charge time at an accelerated charge rate.

The final flight assessment involved the landing tests. When the aircraft approached the runway, the throttle was reduced to 0% in order to minimize propeller interference with the ground. This reduced the speed and the lift drastically and the UAV began to glide but lose altitude relatively quickly. The aircraft lightly contacted the ground, bounced three times, and slid 60 feet until it stopped. The aircraft did not sustain any damage to the landing platform, bulkheads, propeller, or payload bay other than the expected light scraping of the landing gear due to sliding on the concrete. The following series of flight tests continued in a similar manner providing performance data and hints of adjustments to make for further optimization.

8.3 Competition Performance

Once the aircraft was fully optimized, it was packed into the carry on suit case and transported with the design team and pilot to Tucson, AZ where the competition was held on April 15, 2011. The competition requirements stated that the aircraft must be pulled from its carry on suit case, assembled, and flight ready in under 5 minutes. Each of the three missions held this requirement. Throughout the three missions, the aircraft was indeed unpacked and assembled as designed in under 5 minutes without issue. The three missions were performed as follows:

After successful assembly the aircraft was taken to the runway in order to perform Mission 1 on April 15th, 2011. The flight weight of this mission was 2 pounds. The aircraft was thrown into the oncoming wind which increases the relative wind speed to the aircraft and assists in launching as it reduces the required relative ground speed to the aircraft. The aircraft quickly climbed to an altitude of ~150 feet. The aircraft needed to perform as many laps as possible in 4 minutes, after this it was allowed to land. To the all the team members it was thought that the aircraft must perform the maximum number of laps and land in 4 minutes. In actuality the aircraft was allowed to fly for the 4 minutes and land at its leisure after the mission was over. Due to this misunderstanding, the aircraft throttled down and slowed to land toward the end of the four minutes; before the 4 minutes were up, pilot was quickly informed that he could fly through the finish line and land at his leisure. As he was informed of this he quickly throttled back to 100% only to be 5 seconds short of 5 laps. The Mission 1 requirement was fulfilled with the total lap number of 4. The aircraft resituated to land, cut the throttle, and glided to land, bounced once, and slid 80 feet. Mission 1 was a complete success.

Mission 2 was performed the next day on April 16th, 2011, the conditions for this flight were less windy, however the wind direction was blowing from the direction of the crowd. Safety requirements of the competition require that the aircraft not be launched in the direction of the crowd. It was decided that the aircraft would be launched in the direction of the run way with the wind flowing perpendicular to the path of the aircraft. The flight weight of the aircraft at this time totaled to 4 pounds. During launch recovery, due to the cross wind, the left wing stalled out and caused the aircraft to roll slightly. Fortunately the aircraft had previously ascended enough so that the minor loss of altitude due to this left wing stall did not cause a problem. The aircraft recovered gracefully and continued upon its intended trajectory. The aircraft climbed again to ~150 feet and flew the required 3 laps. After the required number of laps were performed the aircraft throttled down to 0% and landed, without bouncing this time, and slid down the runway for ~50 feet. Mission 2 was a complete success just as was Mission 1.

Mission 3 was performed on April 16th, 2011 just as Mission 2. The golf balls required by Mission 3 were loaded to the designed number of 16; bringing the total vehicle weight to 3.7 pounds. The wind had changed directions favorably, allowing the aircraft to be launched into the oncoming wind. The launch was flawless just as all three required laps. The aircraft came in very quickly, throttled to 0% and slid ~120 feet to a stop. Mission 3 was a complete success just as Missions 1 and 2.

The competition was a complete success. The aircraft performed exactly as it was designed without a single design flaw showing through to the performance results. The FAMU/FSU DBF team placed 18 out of 82 worldwide teams. The lessons learned by this year's DBF team will be passed on to next year's team and hopefully give an edge to compete and place higher in the rankings of the great experience that is the AIAA Design/Build/Fly Competition.

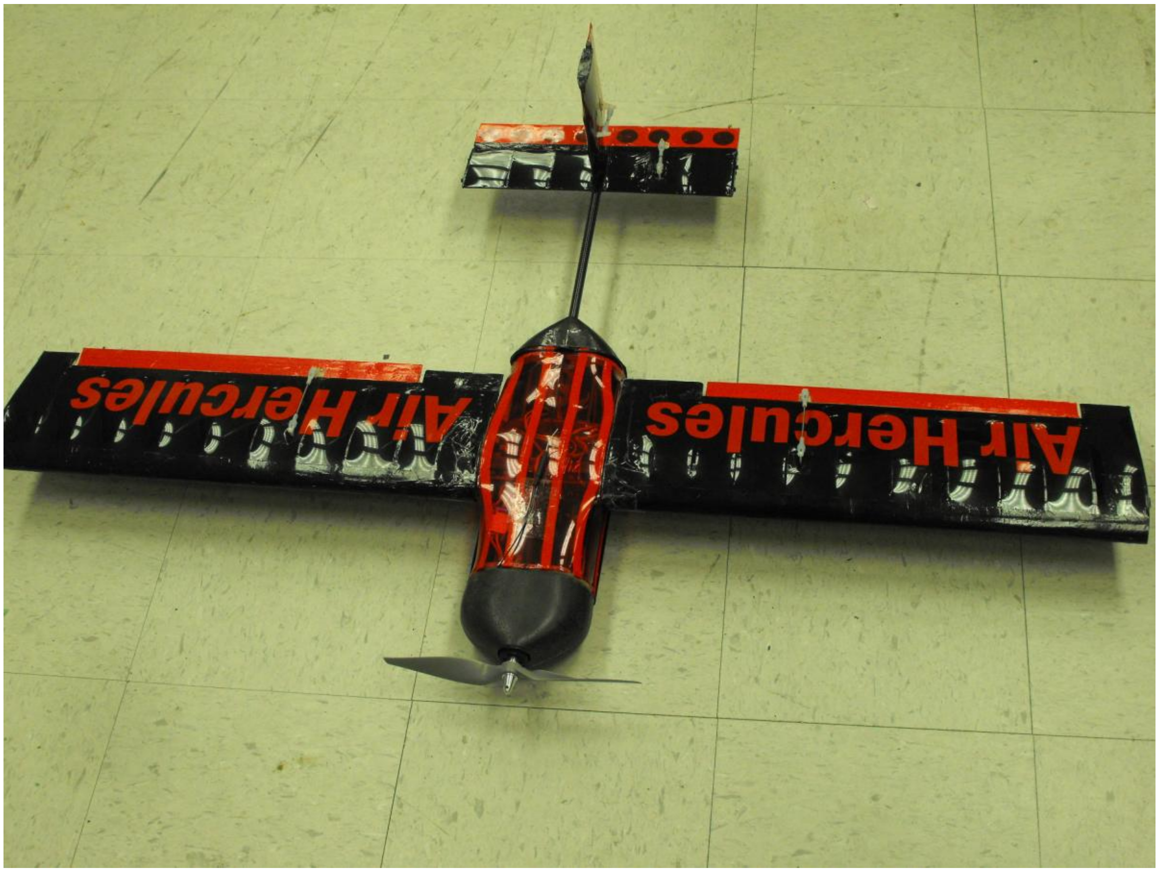


Figure 8.6: Soldier Portable UAV

Figure 8.7: FAMU/FSU AIAA DBF Team, Pilot, and UAV

9.0 SAFETY REVIEW

Safety was considered as a top priority throughout the project. The main concern related to the testing and manufacturing processes. The propulsion system was the first subsystem tested and also had the highest risks to safety. The testing apparatus consisted of different batteries, motors, and propellers mounted to a table with data logging equipment.

NiMH batteries were used because of their low hazard but were still handled with caution. T-plug connectors were implemented in order to avoid plugging into devices with reversed polarity. Charging the batteries was a simple process but, if handled improperly, could result in a fire or similar hazard. It was decided that at least two people had to be present when charging the batteries to avoid misuse.

The motor and propeller were harmless on their own; however, when assembled together with the battery pack they posed a serious threat. The motor had the capacity to rotate at 30,000 rpm which would spin the propeller up to 7,000 rpm! Several precautionary measures were used when testing the propulsion system. First, the apparatus was mounted securely to the table. This was done with the transmitter off and battery unplugged. Once it was determined that the constraint was adequate, the battery was connected and then the transmitter was turned on. The person holding the transmitter would yell, “Clear prop!” before switching it on. After everyone was a safe distance away from the system, the testing would begin. It should be noted that there was an instance where a team member was injured by the propeller due to failure to follow safety protocol. The injury was minor and no other instances occurred.

During the manufacturing process, several potentially dangerous tools were used, including a laser cutter, water jet cutter, hot wire, drill press, dremel, hand held drill, razors, and CNC machines. To avoid any injuries, extreme caution and care were used when handling these devices. Eye protection was required along with masks if necessary. For the water jet and CNC machines, supervisors handled the work. Safety was also an issue when working with composites. Gloves and coats were worn at all times and chemicals were handled properly. All of these precautionary measures led to no injuries throughout the manufacturing process.

Regarding the design of the aircraft, safety was also considered. Since the UAV would be hand launched, it was designed so that the propeller had adequate clearance from the launcher. A 20 Amp fuse was wired in series between the battery and motor in order to cut the circuit when not in use. The fuse was placed in an easily accessible and safe position behind the propeller. It was also critical that the design have adequate structure for handling so that no parts become dislodged under high loads which could result in catastrophic failure. Locktite was used on all nuts and bolts to ensure that all components were secure. The design, itself, was made to be robust and reliable so any handler knew what to expect and could easily notice a malfunction.

All of the measures were taken to ensure the safety of the design team and all who were in proximity of the vehicle.

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