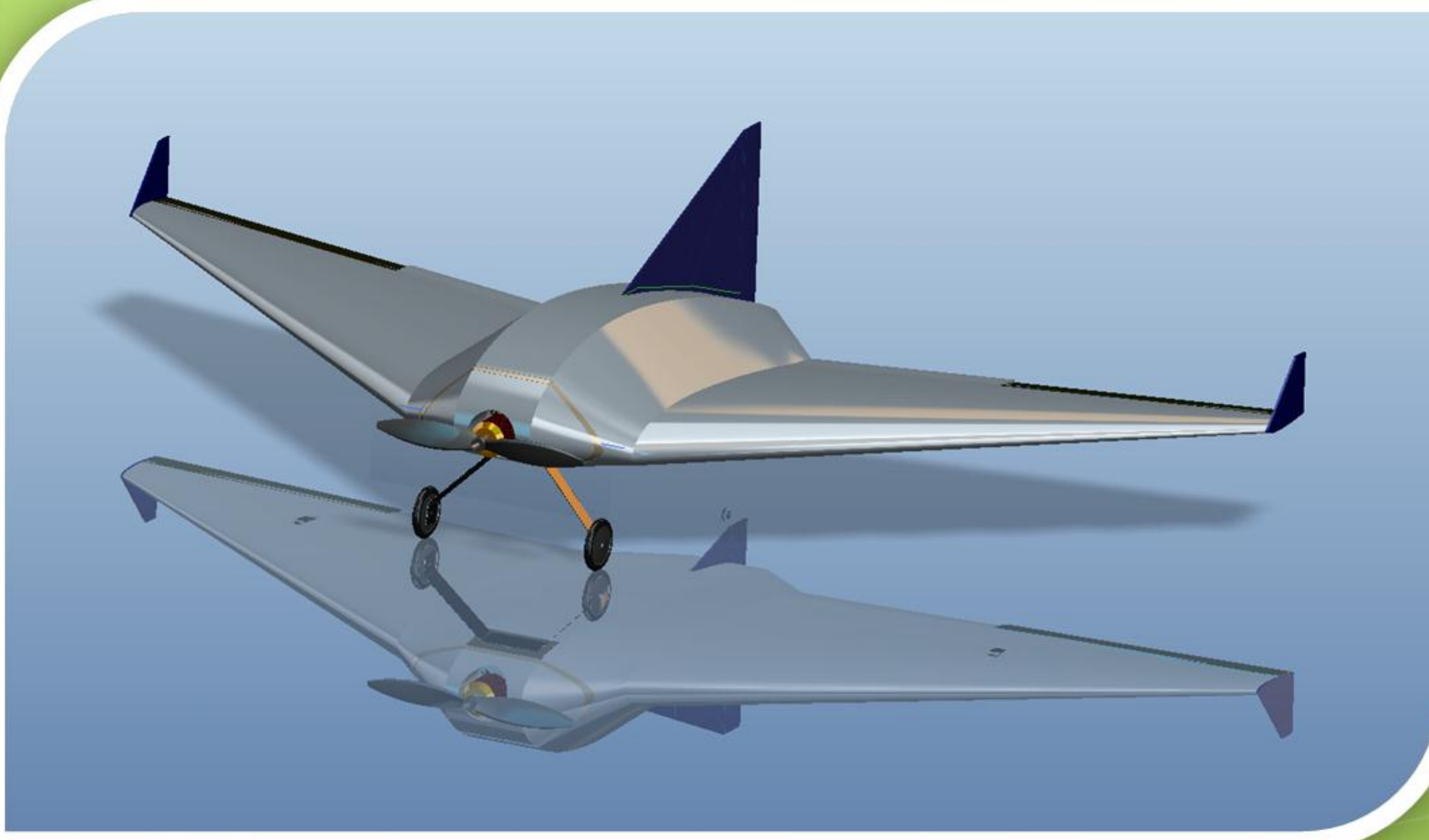
# UNIVERSITY OF NOTRE DAME



# 2012 AIAA Design/Build/Fly Competition

# Table of Contents

# 1.0 Executive Summary

This report details the design, fabrication, and testing of the University of Notre Dame's 2011-2012 Design/Build/Fly (DBF) competition aircraft. The team's primary goal for this year's competition was a first place finish. Having competed for the first time one year previously, the team wished to apply lessons learned from the past to be the best performing team. The aircraft was designed from scratch to be able to execute three specific missions: a speed based flight, a passenger payload flight, and a heavy payload flight.

## 1.1 Mission Requirements

This year's competition, like past competitions, required the aircraft to complete three missions with each mission requiring different performance parameters to be optimized. All three missions required the aircraft to be able to takeoff using no more than 100 feet of runway. Further, total battery weight could not exceed 1.5 pounds and the electronics system had to be safe fused with a 20 amp fuse.

<u>Mission one</u>: Mission one required teams to fly as many laps of the predetermined course as possible in four minutes, with the timer beginning when throttle is advanced for the takeoff roll. The flight score for mission one was a function of the number of laps completed.

<u>Mission two</u>: Mission two required the aircraft to carry 8 aluminum blocks of dimensions 1" x 1" x 5" weighing 3.75 pounds on three laps of the course. The blocks represented passengers, so therefore appropriate spacing around each passenger was required along with an aisle separating passenger rows. Mission two flight score was a function of aircraft flight weight which included the aluminum payload.

<u>Mission three</u>: Mission three required teams to climb to 100 meters altitude as fast as possible with two liters of water as payload. Once 100 meters altitude was reached, the onboard Time End Indicating System automatically dropped the water, signaling the end of the mission timer. Mission three flight score was a function of mission time to climb.

The final team score was then a function of the team's score on this written report as well as the aircraft's RAC weight—the heaviest recorded weight without payload following any successful flight attempt.

## 1.2 Design Process

The team's first step was to perform an exhaustive scoring system analysis to determine the weighting of specific design criteria. The team determined that this year's competition scoring favored a powerful, lightweight aircraft with a high $L/D$ ratio. The aircraft's weight was the largest driver of total score so the team's first goal was to design a lightweight aircraft. Second, the team determined that flight score would benefit from an increased propulsive power by increasing speed for mission one, and decreasing takeoff distance and time to climb for mission three. Weight could not be sacrificed for propulsive power though. Finally, the aircraft had to be able to lift a maximum payload of two liters of water—roughly 4.4 pounds—using less than 100 feet of runway.

Following the scoring system analysis, the team began conceptual design of the aircraft using the design requirements determined from the score analysis as design drivers. This stage involved deciding the aircraft's basic configuration. Following conceptual design, the team entered the preliminary design phase where critical aerodynamic parameters were chosen in addition to giving greater definition to all subsystems. The third design phase—detailed design—involved adding all dimensional quantities to designs as well as choosing electronic & propulsion system components. Following detailed design, the team was able to reasonably predict the aircraft's performance. Finally, the team constructed the aircraft and performance testing began.

## 1.3 Performance Capabilities of the System

The final design was thoroughly tested to ensure consistent high performance of the final design solution. The main results are shown in Table 1.1.

**Table 1.1 – Important Performance Capabilities of Final Aircraft**

| Performance Characteristic | Result |
|---|---|
| L/D max | 24.15 |
| Max Thrust | 8.0 pounds |
| Max Speed | 63.0 mph |
| Empty Flight Weight | 3.936 pounds |
| Max Takeoff Roll Distance | 82 feet |
| Predicted Mission 1 Flight Score | 2.333 |
| Predicted Mission 2 Flight Score | 2.011 |
| Predicted Mission 3 Flight Score | 3.414 |
| Predicted Total Score Range | 321.8 – 371.0 |

The final aircraft did not meet all the original design goals for two main reasons. First, the goals were incredibly lofty. The team set high goals in the hopes that even if they were not met, the aircraft would be in a good position to compete for first place. Second, the models and calculations for the aircraft's performance took many assumptions for granted. For example, the mission model assumed zero wind conditions and a flight typical flight altitude of 50 meters. During typical test flights, however, the aircraft flew in winds up to 17 mph and varied flight altitude too often to accurately model its flight.

# 2.0 Management Summary

The 2011-2012 Notre Dame Design/Build/Fly team consisted of 15 undergraduate aerospace engineering students ranging from freshmen to seniors. As with all teams, success depended on the division of responsibilities into smaller sub-categories to allow for timely completion of project tasks.

## 2.1 Team Organization

The team was divided into five specific groups focused around report & rules compliance, aerodynamics, structures & CAD, propulsion & electronics, and construction & testing. The division of team responsibilities is illustrated in Figure 2.1.

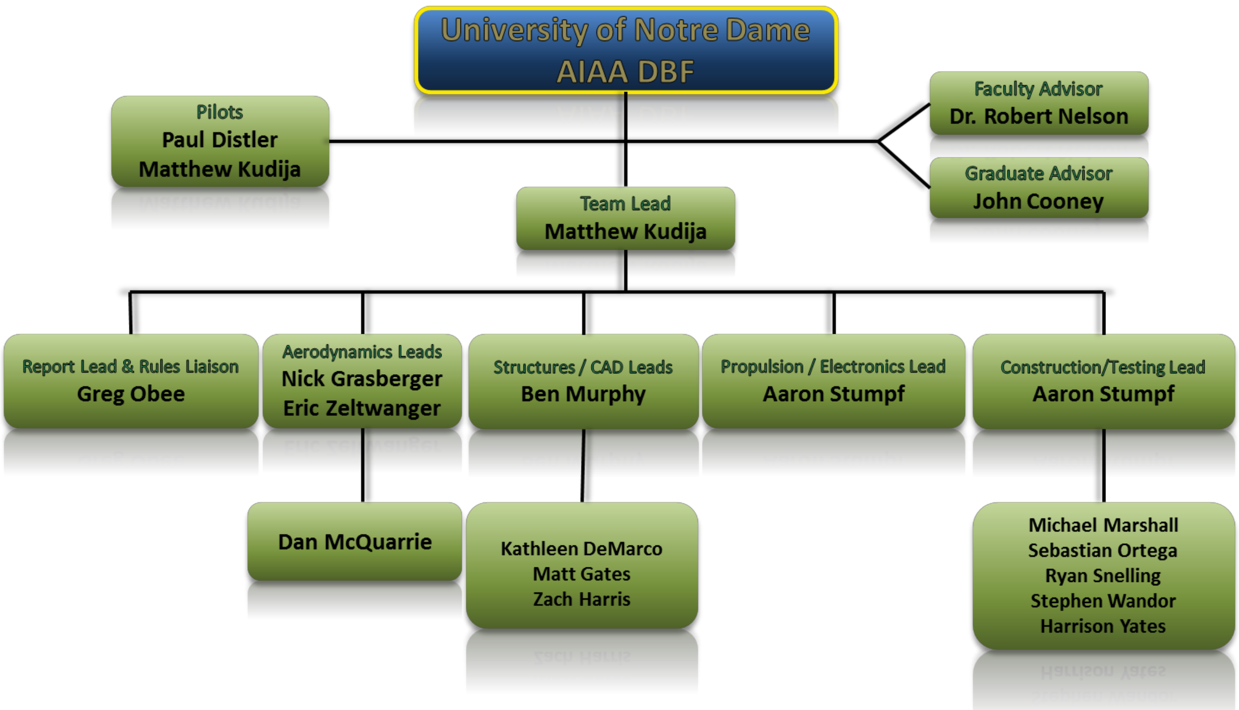



**Figure 2.1 – Team Organizational Structure**

This organizational structure stressed the importance of clear communication and leadership. Overseeing each of the five groups was a group leader who was responsible for organizing and advising those students involved in their respective group. In addition some members had the responsibility of fundraising and handling travel arrangements. Member's responsibilities were as follows:

- **Team Leader –** Maintained entire team's schedule and ensured team members had the necessary tools and skills for completing their assigned tasks. The team leader guided the entire team's efforts.
- **Faculty & Graduate Advisors –** Served as a resource for questions relating to design and acted as official team liaison to the Department of Aerospace & Mechanical Engineering.
- **Pilots –** Performed flight tests of prototype and final aircraft. Responsible for flying the plane at the competition.
- **Report Lead & Rules Liaison –** Assembled the final written report and ensured that final aircraft design met all competition rules and requirements. Lead worked closely with team leader to ensure appropriate progress was made and documented.
- **Aerodynamics Leads & Group –** Led the conceptual and preliminary design and ran performance estimates based on known and calculated parameters. The aerodynamics group was tasked with running stability and control analyses on the design to ensure a well-functioning aircraft was designed.
- **Structures/CAD Lead & Group –** Led the detailed design of all aircraft components utilizing computer aided design software (CAD). The group performed initial weight estimates based off of the CAD models as well as developed the passenger payload carrier and water release mechanism for missions 1 & 2.
- **Propulsion/Electronics Lead & Group –** Selected the propulsion and electronic system components used onboard the aircraft in accordance with all propulsion and electronic system

requirements. Performed ground tests of propulsion systems to determine performance characteristics.

- **Construction/Testing Lead & Group –** Constructed the prototypes and final aircraft. Researched manufacturing techniques and optimized construction process to minimize time and complexity. Extensively tested prototype and final aircraft to ensure designs met all design requirements and team goals.

## 2.2 Milestone Chart

The complexity of the AIAA DBF competition called for an organized and well-thought-out design schedule with the project broken up into smaller, more manageable sub-projects where progress could more easily be measured. The Gantt chart (Figure 2.2) shows the team's predicted progress versus actual progress on these sub-projects. The team's iteration 1.1 design efforts fell behind schedule largely because the team spent more time than originally planned flight testing and gathering data with the first prototype aircraft.

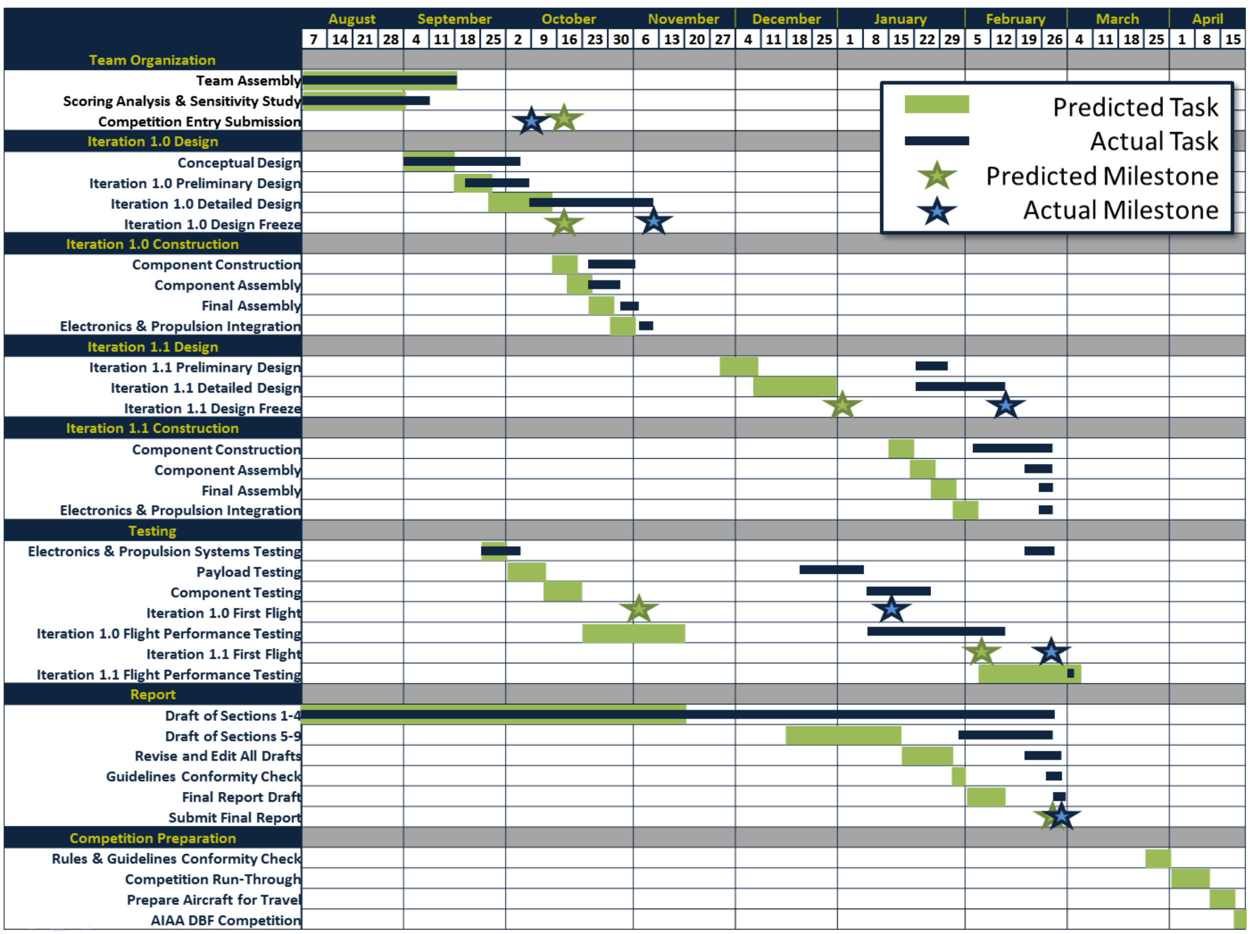



**Figure 2.2 – Design Schedule & Milestone Chart**

# 3.0 Conceptual Design

Before any design work could begin, the 2011-2012 AIAA DBF competition regulations and mission requirements had to be fully understood so that the final design could be assured to be the best solution for the given design criteria. As the first design stage, conceptual design consisted of a competition scoring sensitivity analysis, a breakdown of competition and mission requirements, and a translation of those requirements and scoring analysis into specific design requirements. Basic aircraft and component configurations were then selected based on their ability to satisfy the determined design requirements.

## 3.1 Mission Requirements

The 2011-2012 AIAA DBF competition consisted of three missions which served to test different aspects of the aircraft's capability. Competition regulations, mission specific requirements, and scoring criteria outlined in the contest rules were all carefully analyzed in order to determine each mission's weight in the overall score. A winning design solution will perform exceptionally well in all three missions, thereby receiving the greatest total score while adhering to all competition regulations.

### 3.1.1 Mission Profiles & Score Summary

Course Map – The same course lap is used in all three missions and consists of 7 unique sections, as depicted in figure 3.1:

1. Take-off using no more than 100 feet of runway
2. Climb to safe altitude (Missions 1 & 2) or 100 meters altitude (Mission 3)
3. $180^0$ U-turn 500 feet from the starting line
4. Straight and level flight downwind for 1000 feet
5. $360^0$ turn
6. $180^0$ U-turn
7. Straight and level flight upwind for 500 feet or descent and landing if it is the final lap

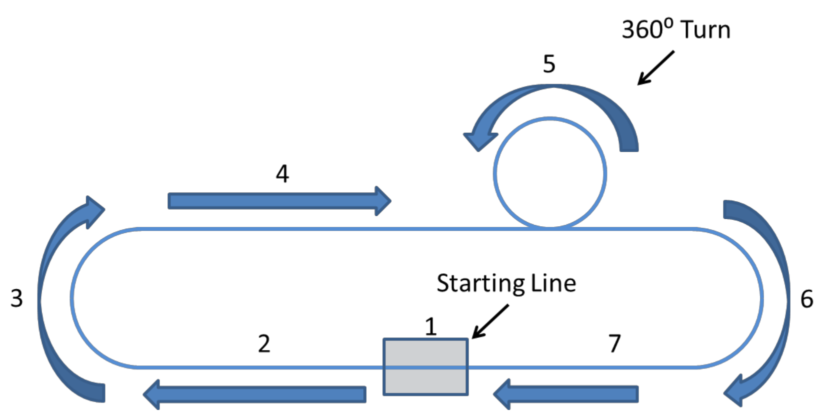



**Figure 3.1 – Mission Flight Course**

Mission 1 : Ferry Flight – All missions begin with the aircraft placed on the runway centerline and require that the aircraft take-off using no more than 100 feet of runway. Mission 1 is a timed flight that measures the maximum number of complete laps that the aircraft can complete within a 4-minute window. The timer begins when the aircraft starts its take-off roll and a lap is counted complete every time the aircraft passes over the start/finish line in the air. The number of complete laps N after 4 minutes is then used to determine the mission 1 flight score M1 using equation 3.1.

$$M1 = 1 + N/6 \qquad \textbf{[Eq. 3.1]}$$

Mission 2 : Passenger Flight – This mission is meant to simulate an aircraft with 8 passengers, where the passengers are simulated by 8 identical aluminum blocks of dimensions 1" x 1" x 5". The total weight of the 8 blocks cannot be less than 3.75 pounds. The blocks must be situated within the aircraft fuselage with the 5" dimension vertical. Furthermore, the blocks must have ½" open space in front and behind each block and 1" separating each column of passengers. No space is necessary between passengers and the outer most wall of the fuselage body, and structure to secure the passenger blocks may be incorporated as necessary in the design. The aircraft must successfully complete 3 laps of the course with the 8-passenger payload. The mission 2 flight score M2 is then determined by equation 3.2 as a function of the flight weight, which is the measured weight of the aircraft with payload intact immediately following the flight.

$$M2 = 1.5 + 3.75/Flight\ Weight\ (lbs) \qquad \textbf{[Eq. 3.2]}$$

Mission 3 : Time to Climb – This mission consists of the aircraft taking off and climbing to 100 meters as quickly as possible can with a Time End Indicating System as simulated cargo. The Time End Indicating System must consist of a water tank filled with 2 liters of water fitted with a servo-operated dump valve. Pressurized water tanks are not allowed and the tank must be vented to the atmosphere. A CAM –f3q device by Soaring Circuits must be used to actuate the water release. The mission 3 flight score M3 is defined by equation 3.3. $T_{avg}$ is the average time to climb of all teams getting a successful score for mission 3. $T_{team}$ is the time from advancing throttle for take-off to when the water plume can be seen exiting the aircraft, signaling that it has reached 100 meters in altitude.

$$M3 = 2 + \sqrt{T_{avg}/T_{team}} \qquad \textbf{[Eq. 3.3]}$$

Overall Score – The overall score for each team is a function of all three flight scores calculated from equations 3.1-3.3, the written report score, and the RAC, where RAC is the maximum empty weight measured after each successful scoring flight. Each team's overall score is defined by equation 3.4.

$$Overall\ Score = \frac{Written\ Report\ Score \times (M1 + M2 + M3)}{\sqrt{RAC}} \qquad \textbf{[Eq. 3.4]}$$

### 3.1.2 Aircraft Design Constraints

Aircraft – This year's theme for the competition is to design a "small passenger aircraft" capable of completing three performance-based missions. The aircraft itself can be of any configuration except lighter-than-air or rotary wing. All take-offs must use no more than 100 feet of runway and be solely powered by the electric motor with no external assistance. Propellers must be commercially produced but can be altered by clipping the tips or painting the blades to balance the propeller. At no time during flight can any component or object be dropped from the aircraft except for the water during mission 3. Finally, the aircraft must be Academy of Model Aeronautics legal which constitutes that the aircraft's take-off gross weight with payload (TOGW) must be less than 55 pounds.

Electronics – Motors must be unmodified over-the-counter model aircraft motors and can be either brushed or brushless. Batteries must be over-the-counter NiCad or NiMH models where the individual cells must be commercially available. The main battery pack may not exceed 1.5 pounds and must power only the propulsion system. The radio receiver must be powered by a separate battery pack. The motors and battery cannot exceed 20 amps current draw and must be safe-proofed by means of a 20 amp fuse.

Payload – All payloads must be carried fully internal to the aircraft's body and cannot shift during flight. Mission 2 payload consists of 8 aluminum blocks which represent passengers. The blocks' dimensions, weight, and placement within the aircraft are given in section 3.1.1. Mission 3 payload consists of 2 liters of water which must be released once the aircraft reaches 100 meters in altitude. The requirements of the water release system are explained in-depth in section 3.1.1.

### 3.1.3 Scoring Sensitivity Analysis

A scoring sensitivity analysis was carried out to prioritize mission and aircraft performance characteristics and thereby maximize overall score. The first step in this analysis was identifying each mission's scoring algorithm relationship to the overall final score. Contrary to past competitions, the mission weighting this year was additive instead of multiplicative. This meant that simply completing all three missions guaranteed 1, 1.5, and 2 'free' flight score points for missions 1, 2, and 3 respectively. However, each mission had an additional opportunity to increase flight score points based on aircraft performance. Figure 3.2 displays each mission's potential for increasing points as a relationship to the mission-specific performance parameter. Each plot's x-axis spans what the design team deemed to be theoretical minimums and maximums of aircraft performance parameters for this year's competition. The red circles represent the team's design goal.

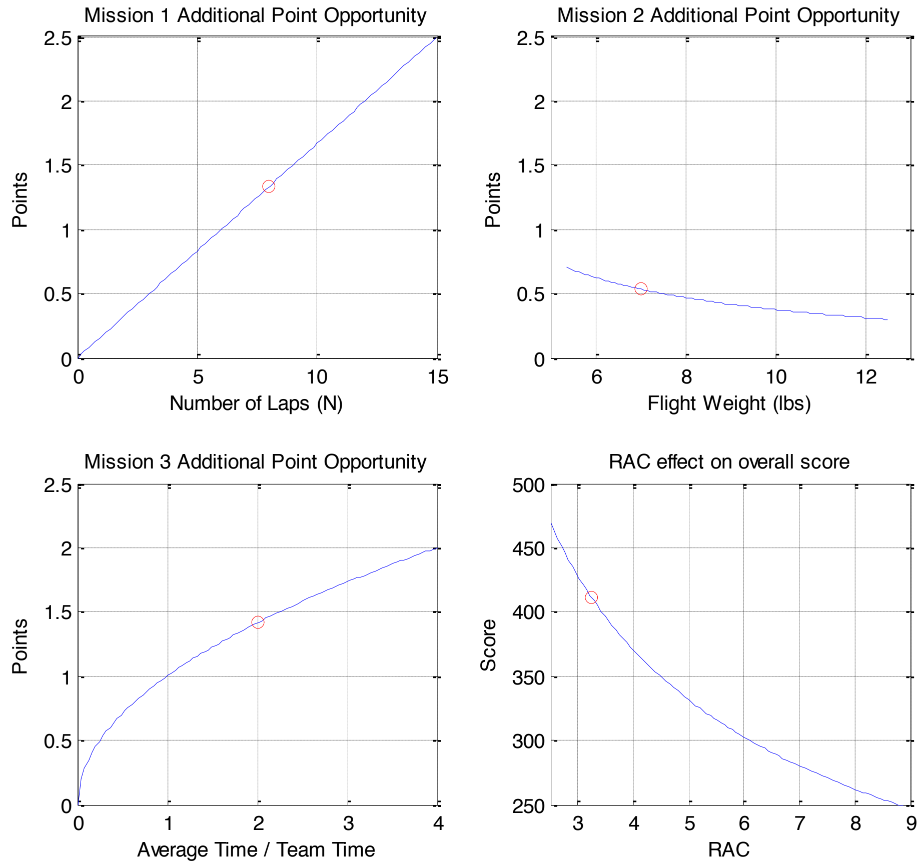



**Figure 3.2 – Performance-Based Additional Point Opportunities**

While mission 2 presents the smallest opportunity for point growth, the performance parameter that is optimized for it—flight weight—has another impact on the overall score in the form of the RAC. For example, adding one pound to the flight weight of the aircraft in mission 2 would only equate to a loss of about 0.05 points for the total flight score—fairly negligible. However, that same pound added to the RAC would equate to a roughly 50 point loss to overall score—quite substantial. Hence, it was quickly determined that the two most important objectives to achieve were to 1) design a lightweight aircraft and 2) complete all missions thereby receiving all 'free' flight score points.

The analysis made it clear that the total overall score—apart from the written report—was dependent on three factors: number of laps completed, aircraft weight, and time to climb. The team desired to find the optimal balance between these three parameters to achieve the greatest score. It was noted that while increasing aircraft speed would mainly increase mission 1 flight score, it would also have a positive effect on the mission 3 flight score by decreasing the time to climb to 100 meters. The team was therefore able to narrow down the design parameters to an optimization study between aircraft weight and aircraft speed. Figure 3.3 shows the relationship between these two parameters based on the aircraft's empty weight and the number of laps completed in mission 1. The plot assumes a team time to climb of half the average time to climb of all teams and a report score of 95.

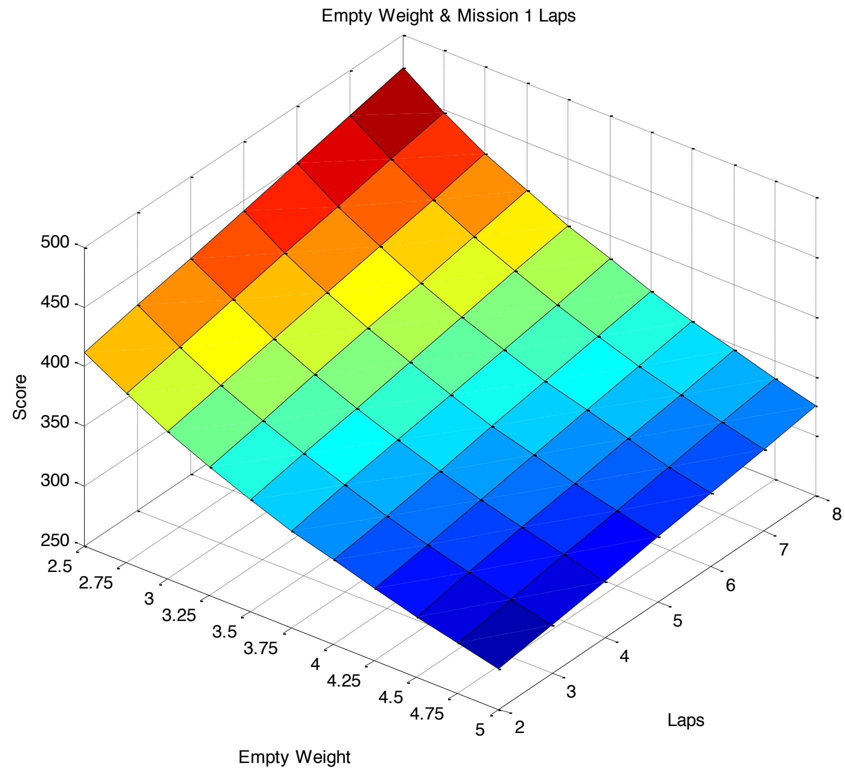



**Figure 3.3 – Scoring Sensitivity Analysis**

From the plot it is evident that empty weight of the aircraft is the overriding parameter for maximizing total score. An aircraft with an empty weight of 4 pounds that completes 8 laps is equivalent to an aircraft with an empty weight of 3 pounds that completes 2 laps. It was the team's engineering judgment that cutting a fourth of the aircraft's empty weight (reducing it from 4 pounds to 3 pounds) would not reduce speed by a fourth (8 laps down to 2 laps). Therefore, it was deemed beneficial to first and foremost design a lightweight aircraft, with speed as a secondary requirement. Based off of these findings, the team set specific design goals in order to help drive conceptual design. The team aimed for a 3.1 pound empty weight aircraft capable of flying 8 laps and climbing 100 meters with full payload in 15 seconds, or half of the estimated average time to climb of all teams. These goals are indicated in figure 3.2 by the red circles. The goals were extremely ambitious and represented numerous design challenges but the team decided to focus on the competition primarily as a learning experience and be open to any and all creative ideas for accomplishing the mission requirements in the most competitive manner possible.

## 3.2 Translating Mission Requirements to Design Requirements

### 3.2.1 Qualitative Design Requirements

The mission requirements and scoring system analysis were then translated into specific design requirements that the aerodynamics and structures groups could use to begin the aircraft's conceptual design. The following design requirements presented in table 3.1 were handed off to the design team.

**Table 3.1 – Translation Into Design Requirements**

| Mission Requirement | Design Requirement |
|---|---|
| Complete maximum number of laps | High thrust, low drag |
| Payload bay of sufficient volume | Large payload bay |
| Water cannot leak | Water-tight tank |
| A high payload fraction | Light construction, high coefficient of lift |
| Fast climb rate | High thrust, high coefficient of lift |
| 100 foot maximum take-off | High thrust, high coefficient of lift |
| 5 minute assembly | Quick connections & accessibility |

The resulting aircraft had to be of extremely lightweight construction while still maintaining enough strength to bear the large load factors of high speed flight. The aircraft had to have plenty of thrust and low drag to allow for high speeds. Finally, the aircraft had to have sufficient lift to allow the aircraft to take-off with the maximum payload of roughly 4.4 pounds.

### 3.2.2 Figures of Merit

To aid the design team in aircraft configuration selection, the results of the scoring system analysis were used to assign weights to design characteristics. The design characteristics used were:

Weight – Empty weight of the aircraft is critically important due to its scoring implications on mission 2 and the RAC. A lightweight design will improve performance in practically every flight characteristic including payload fraction, speed, climb rate, and take-off distance.

Speed – The aircraft's speed is important for a high score in mission 1 and partly mission 3. A high speed design will minimize wetted area $S_{wet}$, to reduce drag, maximize aerodynamic efficiency, and supply sufficient thrust.

Lift & Drag – The aircraft must be capable of taking off in less than 100 feet with 3.75 pounds of payload in mission 2 and roughly 4.4 pounds in mission 3. The wing(s) will need to be optimized to provide all necessary lift while minimizing parasitic drag $C_{D0}$ and wetted surface area $S_{wet}$.

Stability & Control – A lightweight and fast design is of no importance if the aircraft can't be adequately controlled. A stable aircraft design helps to maintain controllability while in flight.

Manufacturability – The aircraft must be able to be manufactured by the team using techniques readily available. Manufacturing certain aircraft components and configurations is simply not in the realm of possibility for the team, and must be purchased commercially, such as electronics hardware.

The five design characteristics were cross-compared in a figure of merit matrix where each characteristic along the left side was rated 1 through 5 on importance compared to the others, where a 5 represents a significantly more important characteristic. When a characteristic is compared to itself it receives a zero and all scores reflected about the main diagonal must equal 5, (i.e. the weight of speed to manufacturability plus the weight of manufacturability to speed must equal 5.) Through this method, the following weights in Table 3.2 were determined. This further confirmed aircraft weight and speed as primary design drivers.

**Table 3.2 – Figure of Merit Matrix**

| | Weight | Speed | Lift & Drag | Stability/Control | Manufacturability | Total | Weight |
|---|---|---|---|---|---|---|---|
| **Weight** | 0 | 3 | 4 | 4 | 5 | 16 | 32% |
| **Speed** | 2 | 0 | 3 | 4 | 4 | 13 | 26% |
| **Lift & Drag** | 1 | 2 | 0 | 4 | 4 | 11 | 22% |
| **Stability/Control** | 1 | 1 | 1 | 0 | 4 | 7 | 14% |
| **Manufacturability** | 0 | 1 | 1 | 1 | 0 | 3 | 6% |
| | | | | | | 50 | 100% |

## 3.3 Configurations Considered & Selection Process

### 3.3.1 Wing Configuration

The wing is the most critical aircraft component not only because it enables the aircraft to fly, but because it also largely determines flight performance. For this reason, it was the first component selected which allowed the basic aircraft form to take shape. Four configurations were considered: monoplane, biplane, tandem wing/canard, and flying wing/blended body. The wing component selection process and weighting is shown in table 3.3.

**Table 3.3 – Wing Configuration FOM**

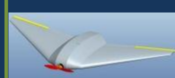
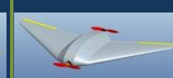
| Wing Configuration | Weight |  Monoplane |  Biplane |  Tandem Wing / Canard |  Flying Wing / Blended Body |
|---|---|---|---|---|---|
| Weight | 32% | 3 | 1 | 2 | 4 |
| Speed | 26% | 3 | 2 | 4 | 5 |
| Lift & Drag | 22% | 3 | 5 | 3 | 5 |
| Stability & Control | 14% | 5 | 5 | 2 | 1 |
| Manufacturability | 6% | 5 | 4 | 4 | 2 |
| Total | 100% | 3.40 | 2.88 | 2.86 | 3.94 |

A flying wing design was determined to be the best configuration in compliance with design requirements. The flying wing received the highest scores for weight and speed and tied with the biplane configuration for the amount of lift and drag produced. The negatives of the design were its poor stability and control characteristics and the difficulty of manufacturing it. Difficulty of manufacturing, however, was not a large cause for concern because the team had sufficient resources at its disposal. Therefore, the only real concern of the team for a flying wing design was the poor stability and control characteristics. If they could be overcome, the result would be a far superior performing aircraft.

### 3.3.2 Motor Configuration

Having narrowed the aircraft design down to a flying wing configuration, the next most critical selection was motor quantity and placement. The configurations considered were a mono tractor, mono pusher, twin tractor, twin pusher, and tractor & pusher combined. The motor selection process and weighting is shown in table 3.4.

**Table 3.4 – Motor Configuration FOM**

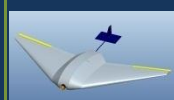
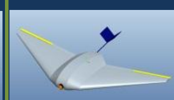
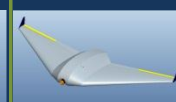
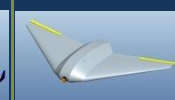
| Motor Configuration | Weight |  Mono Tractor |  Mono Pusher |  Twin Tractor |  Twin Pusher |  Tractor & Pusher |
|---|---|---|---|---|---|---|
| Weight | 32% | 3 | 3 | 1 | 1 | 1 |
| Speed | 26% | 3 | 3 | 5 | 5 | 5 |
| Lift & Drag | 22% | 3 | 3 | 4 | 4 | 4 |
| Stability & Control | 14% | 3 | 2 | 2 | 1 | 2 |
| Manufacturability | 6% | 3 | 3 | 2 | 2 | 2 |
| Total | 100% | 3.00 | 2.86 | 2.90 | 2.76 | 2.90 |

All options were fairly equal in the weighting and selection process so a more detailed trade study was conducted. The positives of a single motor design were that it would be lighter and a more simplistic design, whereas the positives of a dual motor design were that it would be more powerful and therefore fly, take-off, and climb faster. The key question was would the negative impact to the score due to adding a motor and increasing the empty weight be justified by the positive impact to the score due to increased performance? After a detailed weight analysis, it was estimated that a 2$^{nd}$ motor would weigh an additional 0.601 pounds. Assuming a score profile of 8 completed laps, 3.25 pounds empty weight (1 motor), and a time to climb half that of the average, an increase in weight of 0.601 pounds equated to a loss of 33.8 points. In order to make up this score loss due to increased weight, the dual motor design would have to complete 2 additional laps for a total of 10, and climb to 100 meters in 1/4 the average time. These performance increases were deemed impractical and therefore made a single motor design the better option for maximizing total score. The two single motor design options—tractor and pusher—were compared in table 3.4. The mono tractor configuration was deemed the best option because it provided slightly more stability and control over the mono pusher configuration.

### 3.3.3 Empennage Configuration

The empennage of an aircraft significantly influences stability and controllability. In order to choose an appropriate empennage configuration the selection process outlined in table 3.5 was used.

**Table 3.5 – Empennage Configuration FOM**

| Empennage Configuration | Weight |  Conventional Tail on Boom |  V-Tail on Boom |  Single Vertical Stabilizer |  Winglets |  No Tail or Stabilizers |
|---|---|---|---|---|---|---|
| Weight | 45% | 2 | 2 | 4 | 4 | 5 |
| Speed/Drag | 35% | 3 | 3 | 5 | 5 | 5 |
| Stability & Control | 15% | 5 | 4 | 3 | 3 | 1 |
| Manufacturability | 5% | 2 | 2 | 3 | 4 | 5 |
| Total | 100% | 2.80 | 2.65 | 4.15 | 4.20 | 4.40 |

The configurations considered consisted of a conventional tail mounted on an extended boom, a v-tail mounted on a boom, a single, centrally-located vertical stabilizer, winglets, and no tail or stabilizers. The no tail or stabilizer configuration received the highest total score because it was the lightest and negatively affected speed the least. However, the no tail/stabilizer design once again received the worst score for stability and controllability, causing further concerns about aircraft stability and controllability. While a stabilizer-free flying wing aircraft is not unheard of—the U.S.A.F.'s B2 bomber is an excellent example of this design—they are not common due to the increased instability and uncontrollability. Therefore, the team—in their effort to design the lightest and fastest aircraft possible—went ahead with a no tail or stabilizer design but knew that if the design did not provide sufficient stability and controllability, it would be easy to later affix winglets or a single, centrally-located vertical stabilizer.

### 3.3.4 Landing Gear Configuration

Five different landing gear configurations were considered including tricycle gear, skids, retractable gear, tail dragger, and no landing gear. The landing gear selection process and configuration weighting is shown in table 3.6.

### Table 3.6 – Landing Gear Configuration FOM

| Landing Gear Configuration | Weight |  No Landing Gear |  Tricycle Gear |  Skids |  Retractable Gear |  Tail Dragger |
|---|---|---|---|---|---|---|
| Weight | 30% | 5 | 3 | 4 | 1 | 4 |
| Speed/Drag | 20% | 5 | 3 | 3 | 5 | 4 |
| Stability & Control | 10% | 5 | 3 | 3 | 5 | 4 |
| Manufacturability | 5% | 5 | 3 | 3 | 1 | 4 |
| Ground Handling | 35% | 1 | 5 | 2 | 5 | 2 |
| Total | 100% | 3.60 | 3.70 | 2.95 | 3.60 | 3.30 |

Tricycle gear received the highest total score largely due to the improved ground handling of the configuration. Since this year's competition required that the aircraft take-off in under 100 feet, the skid and no landing gear configurations were not feasible. At a maximum take-off weight of around 8 pounds, there would be too much energy loss due to friction. Therefore, the tricycle landing gear configuration was selected for conceptual design, but the possibility of changing the landing gear configuration was left open.

### 3.3.5 Control Surface Configuration

Flying wing aircraft typically have different control surface configurations than conventional aircraft. Flying wing aircraft often use a hybrid control surface called elevons, where one control surface serves the role of elevator and aileron. The control surface configurations considered and the selection process are outlined in table 3.7.

### Table 3.7 – Control Surface Configuration FOM

| Control Surface Configuration | Weight |  2x Ailerons, 2x Elevator, 2x Flaps |  2x Ailerons, 1x Elevator, 2x Flaps |  2x Ailerons, 1x Elevator |  2x Elevons |
|---|---|---|---|---|---|
| Weight | 45% | 1 | 2 | 4 | 5 |
| Speed/Drag | 35% | 3 | 3 | 3 | 3 |
| Stability & Control | 15% | 5 | 4 | 3 | 2 |
| Manufacturability | 5% | 3 | 2 | 3 | 4 |
| Total | 100% | 2.40 | 2.65 | 3.45 | 3.80 |

The chosen configuration consisted of two elevons. The team again took a design approach of starting with a minimalist design which they would then flight test and improve as necessary. If other control surfaces were deemed necessary through flight testing, they could easily be affixed later.

### 3.3.6 Payload Bay & Water Release System Configuration

The team brainstormed different solutions for holding and releasing the two liters of water for mission three and then narrowed the ideas down to the best six ideas. The ideas featured two different tank styles and three different water release mechanisms. Table 3.8 lists the designs considered. Since the water payload system did not restrict the rest of the aircraft design, a decision on which system to use was delayed until detailed design, which is when the design freeze of the entire aircraft occurred. Figure 3.4 shows three of the approaches taken towards the water release mechanism. Picture A is the trap door method where a thin plastic covering would drop out of the bottom of the bottle. Picture B is a pressurized release cap that was later determined to be against competition rules. Picture C shows the valve method, in which the servo would flip the valve open.

**Table 3.8 – Water Payload & Release System Designs Considered**

| Design | Merits & Drawbacks |
|---|---|
| 2-liter bottle with trap door | Lightweight; max flow; difficult to seal |
| 2-liter bottle with puncture seal | Lightweight; easy construction |
| 2-liter bottle with valve | Lightweight; difficult to seal |
| Custom tank with trap door | Mold fit aircraft interior; max flow; heavier; difficult to seal |
| Custom tank with puncture seal | Mold fit aircraft interior; heavier; easy construction |
| Custom tank with valve | Mold fit aircraft interior; heavier; difficult to seal |


**Figure 3.4 – Water Release Conceptual Ideas**

## 3.4 Conceptual Design Summary

Figure 3.5 shows the team's design at the end of the conceptual design phase.

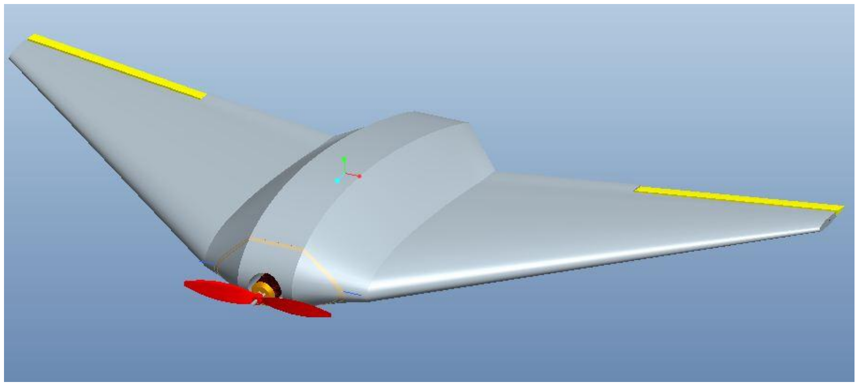


**Figure 3.5 – Conceptual Design**

# 4.0 Preliminary Design

With the major aircraft component configurations selected, the team was then able to refine the basic design and optimize critical design parameters. Throughout the preliminary design process, the team continued to emphasize building a fast, lightweight, and durable aircraft capable of excelling at all three missions.

## 4.1 Design & Analysis Methodology

The preliminary design methodology consisted of first identifying critical design parameters that had significant effect on aircraft flight performance and scoring. Once identified, the critical design parameters were divided into four categories: aerodynamic, propulsion, stability & control, and structural. These four categories of critical design parameters were analyzed thoroughly to ensure that each met all design objectives and requirements. In instances where trade-offs had to be made, the analysis was especially detailed.

#### 4.1.1 Aerodynamic Critical Parameters

- Wing Area – The wing surface area directly affects the amount of lift produced by the wing but also influences surface drag. Therefore, a fine balance must be found and an appropriate wing area selected.

- Wing Span – Similar to the wing surface area, the span and wing aspect ratio heavily influences the aircraft's lift and speed characteristics. An increased wingspan can increase lift and aspect ratio but often requires more structural bracing, thus increasing empty weight.
- Airfoil – The wing airfoil is perhaps the most critical design parameter of an aircraft. It significantly affects all performance characteristics of an aircraft. Furthermore, the aerodynamics involved when dealing with a flying wing aircraft design makes airfoil selection even more important. The team had to select an airfoil which maximized lift, minimized drag, provided sufficient payload volume, and countered the pitching moment due to the flying wing configuration.

#### 4.1.2 Propulsion Critical Parameters

- Motor – Motor selection is based on of the aircraft's estimated weight and speed requirements. Also, this year's rules stated that the motor must be fused to not exceed 20 amps of current draw. This requirement drastically limits propulsion system options. The motor must have enough power to allow the fully-loaded aircraft to takeoff within 100 feet, but also never surpass the 20 amp limiting current draw.
- Batteries – Teams are allowed up to 1.5 pounds of batteries for flight. The team needs to select the optimal battery pack configuration based on the number of individual cells, battery capacity, and voltage. Each mission can have a different battery optimized for that mission's requirements.
- Propeller – A larger propeller provides more thrust but also requires more power from the electrical system causing more current draw. A lightweight, appropriately sized propeller must be chosen to not impose too much current draw on the electrical system. The relationship between propeller pitch and diameter also influences the top speed the aircraft can reach, making that an important consideration for mission 1.

#### 4.1.3 Stability & Control Critical Parameters

- Elevons – Ailerons and elevator control surfaces are often combined into elevons for flying wing aircraft. Any time two control surfaces are replaced with only one, it become especially important to choose an appropriately sized control surface.
- Winglets – Since flying wing aircraft do not have a vertical stabilizer or rudder, winglets are critical to ensure adverse yaw does not occur. A winglet that is too small will not have enough control against yaw whereas a winglet that is too large will add significant drag.

#### 4.1.4 Iterative Design Methodology

When engineering something as complex as an aircraft, it is very difficult to get everything correct the first time through. Having learned this valuable lesson in the Design/Build/Fly competition the previous year, the team decided upon an iterative design approach for the 2011-2012 aircraft design and optimization process. This design methodology was based around making incremental improvements and adjustments to the aircraft over an extended period of time. This broke what was originally an incredibly complex and difficult design challenge into smaller, more attainable challenges. Through the iterative design process the aircraft was incrementally improved during the construction of two prototype aircraft and one final competition-ready aircraft. This methodology and approach gave the team more time to work on the particularly challenging aspects of this year's competition. Figure 4.1 shows the transition from prototype, to iteration 1.2, to final competition-ready aircraft.

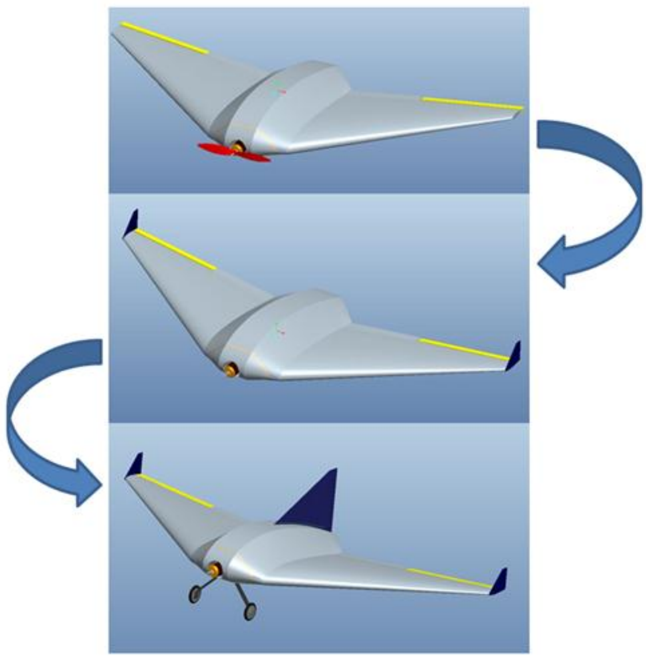


**Figure 4.1 – Iterative Design Methodology**

## 4.2 Mission Model

#### 4.2.1 Description & Capabilities

To better understand the performance requirements of the aircraft, a model of the estimated flight course of the aircraft was constructed in a MATLAB script that took the aircraft through a simulated mission. The goal of this model was to determine the required flight speed of the aircraft to complete the desired number of laps in mission one. However, since the core aircraft geometry must go unchanged between missions per competition rules, the model was then used to predict mission durations for missions two and three. This helped the propulsion and electronics group to select appropriately sized batteries for each mission.

#### 4.2.2 Turns

The team noticed that turns constituted a large portion of the flight course with a total of $720^0$ of turns per course lap. Thus, the team knew that fast turning was a priority and set an appropriate turning radius of 40 feet as the aircraft goal. The team estimated through flight testing of various other remote-controlled aircraft that this radius allowed for the best trade-off between a tight turn without bleeding too much speed. The turn-induced speed bleed-off was estimated from known performance of other remote controlled aircraft to be roughly a 33% loss of speed across the turn. The $720^0$ of turns per lap at a 40 foot radius equaled 502.7 feet of total turns. To factor in the 33% loss of speed across turns, the total turn length of 502.7 feet was multiplied by a weight of 1.33, which yielded a distance of 670.6 feet. While in reality, one lap's worth of turns was still only 502.7 feet in length, the model assumed 670.6 feet to account for the total speed loss across turns.

#### 4.2.3 Straight Course

The course was marked out at 1000 feet long between the two end turns. Thus, there was a total of 2000 feet of straight course. During these sections, the aircraft was assumed to be able to fly at maximum speed after quickly accelerating out of the turns.

#### 4.2.4 Complete Course Model

By the above approximations, one course lap was estimated to be 2502.7 feet in length, and modeled as 2670.6 feet to achieve a more realistic required speed. The team's goal of completing 8 laps in four minutes meant the team had 240 seconds to move 21,364.8 feet. This meant the aircraft had to be able to fly at least 60.7 mph.

#### 4.2.5 Model Uncertainties

The model did not take into account the fact that the timer for mission one begins with the aircraft take-off roll on the ground. This would add considerable time to complete the first lap since the aircraft is starting from a standstill on the ground. By the time the aircraft takes off, climbs, levels out, and accelerates to maximum velocity, it could be upwards of 25-30 seconds. Also, environmental conditions were neglected in the model, which assumed zero wind conditions. Adverse wind conditions could make turning and maintaining the desired course extremely difficult and add significant time. Finally, any altitude the aircraft will lose during the turns was neglected, but will in fact contribute significantly to the overall time it takes to complete a lap since regaining the lost altitude results in slower flight. To account for these uncertainties, the desired speed goal was raised from the model calculated speed of 60.7 mph to a lofty 76.0 mph.

## 4.3 Design Trades & Optimization

### 4.3.1 Wing Design Trade Studies & Optimization

Selection of the overall wing design depended entirely upon the various aerodynamic criteria necessary for competitive performance in all phases of the competition. Following the comprehensive score analysis it was determined that the first mission, in which the aircraft must complete as many laps as possible in four minutes, allowed for the greatest separation among teams in terms of available points. Consequently, the aerodynamic performance of the aircraft was centered primarily upon optimum execution of the first mission, assuming the max payloads in missions 2 and 3 could be carried. *Design of Aircraft* by Thomas C. Corke and *The Elements of Preliminary Aircraft Design* by Roger D. Schaufele were used extensively as aids to aerodynamic design. The texts detail aircraft design from setting preliminary design objectives and proposals through wing design, stability analysis, and estimating final performance characteristics. Included with Corke's text were a series of useful e-files with spreadsheets that help to set key design characteristics based on desired design drivers.

With the emphasis of importance on mission one performance, the design drivers were mostly extracted from the nature of mission one. The high speed and maneuverability requirements of mission one were estimated to be the most stressful on the aircraft, and therefore the foundation for wing design. Additionally, the three missions shared many interconnected performance traits. For example, an increase in motor power not only benefited the mission one score due to increased speed, but also the mission 3 score because of the increased rate of climb. Missions two and three were payload missions which required a max payload of 2 liters of water, or roughly 4.4 pounds, to be lifted to 100 meters altitude. As long as the aircraft was capable of this feat, the design focus could shift to optimizing the aircraft for mission one and in doing so, also increasing mission 2 and 3 performance.

To maximize the performance in this mission and achieve the greatest flight score possible, the turning rate and top speed were chosen as primary wing design drivers. Secondary wing design drivers consisted of being able to quickly take-off and lift the 4.4 water payload to 100 meters.

The first step in the wing design was determining the necessary wing loading to achieve these desired values. Using the e-file spreadsheets supplied with Corke's book, the wing loading was calculated to be roughly 0.8 lb/ft$^2$. The aircraft with maximum payload installed was estimated to weigh 7.65 lb. The subsequently required total wing surface area was just under 5 ft$^2$. However, in the trade study of wing area vs. speed, the analysis determined that top end speed would be achieved with just 0.59 ft$^2$. However, this small of a wing was infeasible, so the trade-off had to be made to create the smallest wing surface area possible that would be still capable of creating 7.65 lb of lift.

# 4.4 Aerodynamic Characteristics

## 4.4.1 Airfoil Selection & Low Reynolds Number Considerations

The combination of a 5 $ft^2$ surface area wing with a central lifting body gave the team confidence that the aircraft would be able to successfully take-off within 100 feet and climb with max payload. Nevertheless, when dealing with a flying wing aircraft, airfoil selection is even more important than with traditional aircraft due to the unique aerodynamic and stability phenomena involved. Of primary concern was selecting an airfoil that performs well at low Reynolds numbers. Airfoil performance can vary significantly depending on the operating Reynolds number, so as a result the team's airfoil search was limited to those deemed appropriate for our flight regime.

The NACA 23015 airfoil was chosen for the wings due to its favorable stall characteristics and fairly uniform drag bucket. Ease of manufacturability also factored into the decision. A simpler geometry was desirable because the outboard wings had to be blended into the main body, which would have been difficult with complex airfoil geometry. Airfoil selection for the main body section required finding an airfoil thick enough to be able to accommodate the aluminum block passengers and 2-liters of water internally. The NACA 4424 airfoil was selected for this purpose. The 24% thickness to chord ratio meant that the aircraft center section was as aerodynamically thick as possible while remaining a traditional airfoil shape, further increasing total lift. While this center-body airfoil was not optimized for low Reynolds numbers, it was considered an excellent compromise since it was still able to considerably contribute to total lift force while housing all payloads. Figure 4.2 shows the difference between the two airfoils chosen for the wing (NACA 23015) and the center body (NACA 4424). Key characteristics to note are the NACA 4424's thickness and the modified NACA 23015's slight reflex to enhance stability of the flying wing planform.

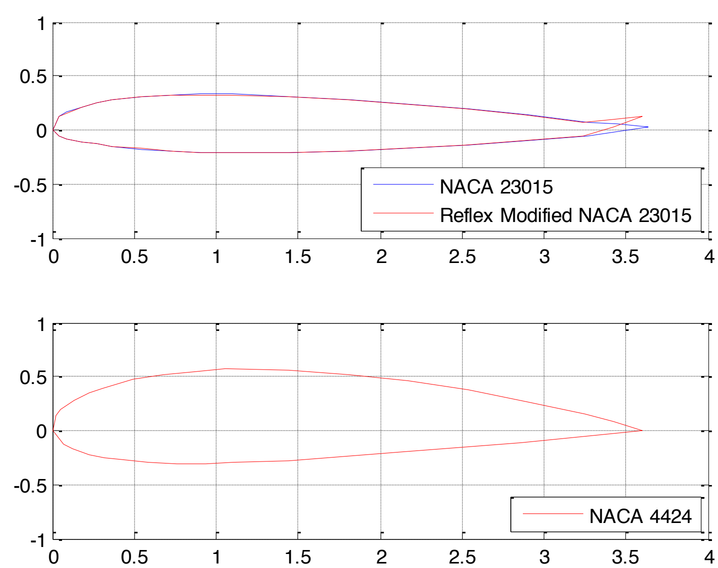



**Figure 4.2 – Airfoils of the Wing (23015) & Center Body (4424)**

The drag polars for both airfoils are shown in Figure 4.3. These figures came from the *Airfoil Investigation Database* online. The NACA 4424 had a larger lift and drag coefficient than the NACA 23015, but was not optimized for speed. The NACA 23015 airfoil, however, was optimized especially for speed.

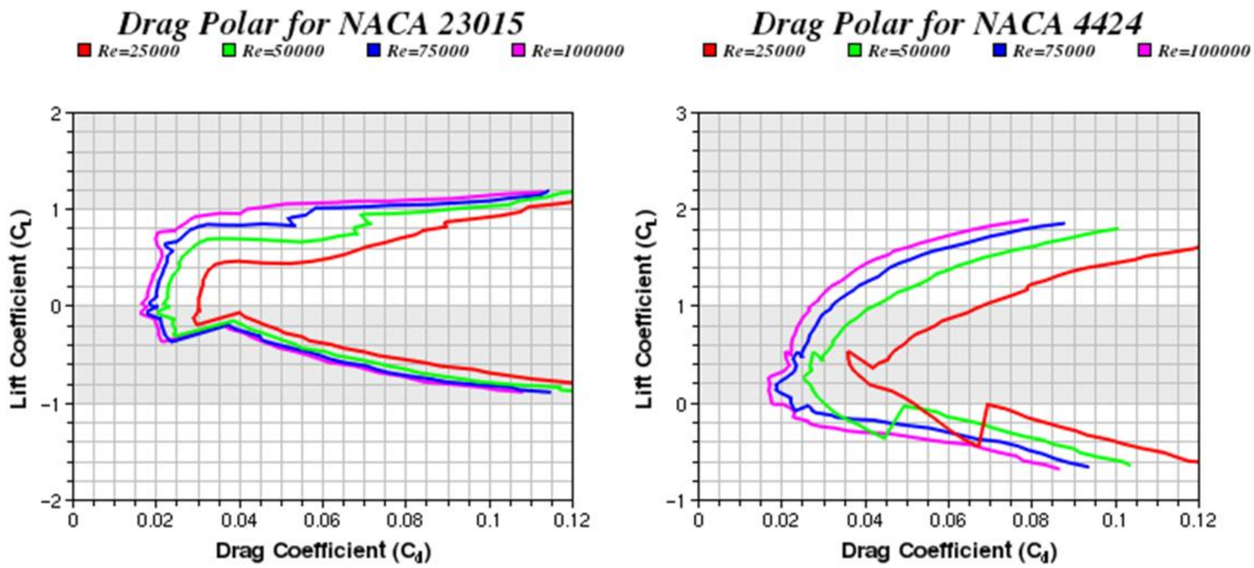



**Figure 4.3 – Airfoil Drag Polars**

#### 4.4.2 Sweep & Taper Analysis

The aerodynamics associated with flying wing aircraft designs are often tricky because there is no empennage or stabilizers to provide directional stability. To help with this, $35^0$ of wing sweep was added along with a taper ratio of 0.18. These values were determined by comparing existing flying wing aircraft designs to match desirable performance characteristics.

#### 4.4.3 Preliminary Drag Estimates

Using Professor Corke's book and analytical spreadsheet, parasitic drag due to frictional forces $C_{do}$ and drag force was found for each component. The induced drag $C_{di}$ was calculated using computational fluid dynamics software. In calculating the values, a Reynolds number of 75,000 was used along with a velocity equal to the maximum expected mission velocity. The surface area of each component was calculated in order to determine the total acting drag force. Table 4.1 and Figure 4.4 show the results of these drag calculations. The entire aircraft design yielded a drag force of 2.67 pounds with the majority of drag due to parasitic drag.

Table 4.1 – Component Drag Estimates

| Drag Estimates | | |
|---|---|---|
| | $C_d$ | Drag Force (lbf) |
| Wing | 0.017 | 1.19 |
| Center-body | 0.015 | 0.61 |
| Landing gear | 0.009 | 0.19 |
| Winglets | 0.007 | 0.13 |
| Vertical Stabilizer | 0.010 | 0.21 |
| Induced Drag $C_{di}$ | 0.011 | 0.34 |
| TOTAL | 0.069 | 2.67 |

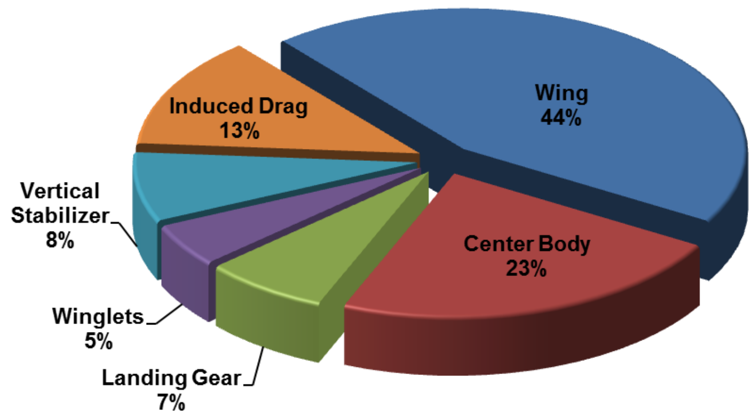



**Figure 4.4 – Estimated Component Drag Force Breakdown**

#### 4.4.4 Preliminary Coefficient of Lift Estimates

The performance characteristics of the NACA 23015 and 4424 airfoils were input into the analytical spreadsheets to calculate the resulting coefficient of lift of the entire aircraft. The benefit of a flying wing aircraft is that its lift to drag ratio is larger than traditional aircraft. This was evident in the lift forces analysis, which yielded a lift to drag ratio of 24.15. The lack of a true fuselage, horizontal stabilizer, and rear empennage lowers the drag forces, leading to an increased ratio. Table 4.2 contains the key lift parameters, including the max coefficient of lift, the zero-lift angle, the coefficient of lift for which drag is a minimum, and total lift force. The main driver in designing the lifting surfaces was that they must be able to lift the 8-pound fully loaded aircraft.

**Table 4.2 – Lift Parameters**

| Lift Parameters | |
|---|---|
| $C_{l_{max}}$ | 1.39 |
| $\alpha_{0_l}$ | -0.8º |
| $C_{d_{min}}$ | 0.40 |
| $L/D$ | 24.15 |
| Lift Force | 8.0 lb |

## 4.5 Stability & Control Characteristics

#### 4.5.1 Static Stability Analysis

Traditional aircraft have a center of gravity that acts around the ¼ chord of the wing which allows for the best stability in flight. For flying wing aircraft, stability is incredibly more sensitive to minor center of gravity shifts and offsets, so setting a desired center of gravity, as well as ensuring the designed aircraft meets that desired center of gravity becomes very difficult for flying wing aircraft.

Upon researching flying wing aircraft stability issues and conditions, it was determined that the aerodynamic center was supposed to be roughly around the ¼ chord. Further, in order to be statically stable, the center of gravity must lie forward of the aerodynamic center. However, since the wing had taper, the mean aerodynamic chord of the wing had to be calculated. Table 4.3 shows the result of this calculation and other stability parameters.

**Table 4.3 – Critical Stability Parameters**

| Stability Parameter | Value |
|---|---|
| Mean Aerodynamic Chord (in) | 13.44 |
| Center Body Chord (in) | 24.00 |
| Center Body Aerodynamic Center (in) | 6.25 aft of nose |
| Aircraft Aerodynamic Center (in) | 10.32 aft of nose |
| Aircraft Center of Gravity without Battery (in) | 14.00 aft of nose |
| Aircraft Center of Gravity with Battery (in) | 7.50 |

Since the center section of the aircraft was also a lifting surface, it had to be analyzed separately from the wing. The center body is one large airfoil with a chord 24 inches long, so its aerodynamic center is at 6.25 inches. This approximation assumed that it was a straight rectangular section, which means that the mean aerodynamic chord is simply equal to the chord of 24 inches. To combine the wing and center body aerodynamic centers into one aircraft aerodynamic center, each section was assigned a weight equivalent to their sizes. The outboard wing area was roughly 70% while the center body was 30%, which yielded an aircraft aerodynamic center at 10.32 inches aft of the nose.

Next, to determine the center of gravity for the aircraft, the CAD model's analysis tools were used to determine that the center of gravity of the empty aircraft model was 14 inches aft of the nose. Based on estimates of how the electronic components, passengers, and all other interior objects could be placed, the center of gravity location could range from about 7-9 inches aft of the nose.

#### 4.5.2 Dynamic Stability Analysis

Because the center of gravity could range from 7-9 inches, the next decision was to determine what type of static margin was desired. The static margin determines the degree to which the aircraft is statically stable. Large static margins require substantial pitch control to achieve steady level flight, but there are no potential stability problems. As the margin is decreased, maneuverability increases and the elevator control requirements decrease, but the aircraft has more potential for unstable flight. Therefore, an appropriate balance had to be struck between maneuverability and stability.

Since the flying wing aircraft design is already considerably more unstable than a traditional aircraft, the team decided to emphasize stability over maneuverability in choosing the plane's static margin. This meant that the center of gravity should be forward of the aerodynamic center. The center of gravity was moved to roughly 7.5 inches aft of the nose by placing the battery closer to the front of the aircraft.

#### 4.5.3 Control Surface Sizing

Elevon Sizing - A large static margin requires larger control surfaces to achieve equivalent control characteristics of a lower static margin aircraft. With this in mind, the aerodynamics group analyzed the required moments to be generated by the elevons. Since the location of the elevons was known, the only variable left in calculating the moments was the forces needed per control surface. Once these forces were calculated the specific surface area of the two elevons were calculated. The calculated dimensions of the elevons were increased slightly from what was required due to the fact that they are acting as both ailerons and the aircraft's elevator. Additionally, a factor of safety of 1.5 was used to set final control surface dimensions. The elevons had a chord of 1.25 inches and a length of 14 inches. The winglets measured 3 inches tall by 3.55 inches at the base and 0.94 inches at the tip.

### 4.6 Propulsion Characteristics

#### 4.6.1 Motor Selection

The score analysis determined that the most critical design requirements were having a fast and lightweight aircraft. Therefore, the team had to select a motor that was powerful enough to achieve high flight velocities, yet light enough to keep the aircraft's empty weight low. To add further complication to motor selection, competition rules stated that the aircraft's electronics and propulsion system cannot draw above 20 amps. This requirement significantly limited the options the team had for the propulsion system.

The first step in selecting a motor was determining the power required from the motor to achieve the desired flight performance of the aircraft. This was done by researching traditional values of power requirements for remote controlled aircraft. The team found that a typical performing remote controlled aircraft requires roughly 75 watts per pound of aircraft. With an estimated maximum flight weight of 8.5 lb, this resulted in a motor power requirement of 637.5 watts. Using the electric power equation $P = I * V$, where the maximum current $I$ is 20 amps, and power is 637.5 watts, the required voltage through the motor was found to be 31.875 volts. Knowing these power, current, and voltage requirements allowed the team to narrow down the possible motor options. Table 4.4 shows the four motors considered and their respective characteristics.

**Table 4.4 – Considered Motors & Characteristics**

| | Weight (ounces) | Max. Current (amps) | Voltage (volts) | RPM/Volt |
|---|---|---|---|---|
| **E-flite 295 Kv Power 110** | 17.5 | 55 | 28.2 - 38.4 | 295 |
| **E-flite 325 Kv Power 90** | 15.8 | 50 | 21.6 - 31.2 | 325 |
| **Rimfire .55 480 Kv** | 9.5 | 45 | 18.5 - 22.2 | 480 |
| **Tacon Big Foot 32 770 Kv** | 7.4 | 43 | 15 | 770 |

While the two E-flite motors best fit the desired current and voltage requirements, they were both considerably heavier than the other two options. The Tacon motor would have led to a slightly underpowered aircraft since its input voltage was so low. Therefore, the Rimfire .55 480 Kv motor was selected to power the aircraft. This motor was plenty powerful enough to lift an 8.5 pound aircraft, but the 20 amp limit required a higher voltage to be pushed through the motor than what it was rated for. To solve this problem, a high voltage electronic speed controller was necessary. The Castle Creations Phoenix ICE High Voltage 60 was chosen for this reason and had a maximum voltage input of 50 volts.

#### 4.6.2 Battery Selection

To maintain the required electric power supply to the motor without surpassing the 20 amp current draw requirement, a high voltage battery pack was required. Power calculations suggested that the battery must supply at least 31.875 volts to meet the aircraft power requirement. Drawing upon previous years' experience, Elite battery cells were selected because they provided excellent milliamp hour capacity to weight ratios which help to keep battery pack weight to a minimum. Table 4.5 shows the different battery pack configurations considered. Important to note, was that battery packs 2 and 3 supplied voltages above the suggested range for the motor. While the motor would still operate at these high voltages, it would be in danger of overheating internal electrical components if left running at too high of a power setting for too long. The required 31.875 V was a conservative estimation since factor of safeties were included in its calculation. Therefore, the programmable speed controller could be used to limit voltage into the motor.

**Table 4.5 – Considered Battery Pack Configurations**

| Battery Cell | # of Cells | Weight (ounces) | Voltage (volts) | Capacity (mAh) |
|---|---|---|---|---|
| 1. Elite 1500mAh | 20 | 14.8 | 24.0 | 1500 |
| 2. Elite 1500mAh | 26 | 19.2 | 31.2 | 1500 |
| 3. Elite 1500mAh | 32 | 23.7 | 38.4 | 1500 |
| 4. Elite 2100mAh | 20 | 22.8 | 24.0 | 2100 |

The battery pack weight limit for this year's competition is 24.0 ounces, or 1.5 pounds, so all battery packs must be below this value. Since the battery is included in obtaining flight weights and RAC weights following each flight, the battery pack weight has a significant impact on total score. Therefore, the lightest battery pack that could last the maximum duration of the mission while providing sufficient power was desired. The first listed battery option is the lightest but also has the lowest voltage output, whereas option 3 is the heaviest but has the highest voltage output. Option 1 was eliminated because of the low voltage and option 3 was considered too heavy.

In order to decide between options 2 and 4, the team considered both batteries' run time. Missions 1, 2, and 3 required a run time of 4.5 minutes, 2.5 minutes, and 2.0 minutes respectively. To estimate battery run times, Castle E-calc propulsion analysis software was used to model the electronic and propulsion system. All system parameters were input to the calculation software, which estimated an approximate run time of 4.2 and 5.9 minutes for options 2 and 4 respectively. The team deemed 4.2 minutes of projected full throttle run time sufficient since the plane's throttle could be cut back to 80-90% maximum thrust for mission one. Therefore, battery option 2 was chosen since it was lighter and provided more voltage. Figure 4.5 shows that at a maximum current of 20 amps, projected propulsion power is 640 watts and the entire propulsion system's efficiency near its maximum.

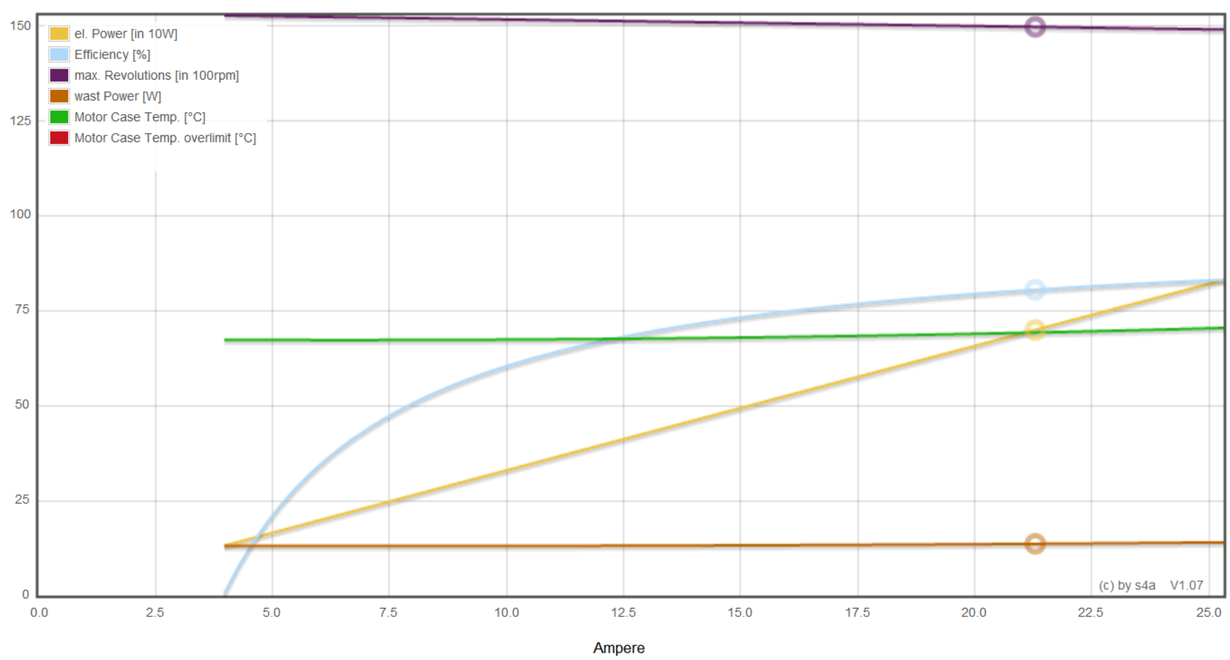



**Figure 4.5 – Predicted Propulsion System Performance**

#### 4.6.3 Propeller Selection

The team considered three different sized propellers: 11x7E, 13x10E, and 15x8E. The 15x8E size propeller would give the greatest thrust but at the penalty of increasing the current through the electronic system, so it was ruled out. The 13x10E was ultimately chosen over the 11x7E because despite the 13x10E propeller increasing electrical current, it increased the thrust over the smaller propeller by a significant portion that made the increased current worthwhile.

## 4.7 Mission Performance Estimates

Using a combination of CAD modeling software (weights), Professor Corke's analytical spreadsheet ($C_l$, $C_{do}$, $V_{max}$, $and\ L/D$) and simple calculations (payload & lap time), the following mission-specific performance estimates shown in Table 4.6 were determined. To determine lap time, the lap distance was estimated and divided by the predicted max velocity.

**Table 4.6 – Mission Performance Estimates**

| Aircraft Parameters | |
|---|---|
| $C_{Lcruise}$ | 1.1 |
| $C_{Lmax}$ | 1.39 |
| $e$ | 0.86 |
| $C_{Do}$ | 0.69 |
| $L/D_{max}$ | 24.15 |
| Max Thrust (lb) | 8.0 |
| Max Speed (mph) | 63.0 |
| Empty Flight Weight (lb) | 3.936 |
| Max Payload (lb) | 4.4 |
| Max Takeoff Weight (lb) | 8.336 |

# 5.0 Detailed Design

Detailed design consisted of refining all subsystems and components to maximize flight score and performance. Finally, after thoroughly optimizing all subsystems, the team worked to integrate all subsystems into the complete design solution.

## 5.1 Final Design Parameters

Table 5.1 shows the finalized component and aircraft design parameters.

**Table 5.1 – Final Design Parameters**

| Wing | | Elevons | |
|---|---|---|---|
| Span (in) | 60 | Length (in) | 14 |
| Airfoil | Modified NACA 23015 | Chord (in) | 1.25 |
| Root Chord (in) | 18.15 | Area (in^2) | 17.5 |
| Tip Chord (in) | 3.55 | Airfoil | Wedge |
| Taper Ratio | 0.18 | **Vertical Stabilizer** | |
| Aspect Ratio | 5 | Height (in) | 10.25 |
| Area ($ft^2$) | 5 | Area ($in^2$) | 67.22 |
| 1/4 Chord Sweep (degrees) | 35 | Airfoil | Flat Plate |
| Max Cl | 1.39 | **Propulsion System** | |
| Cdo | 0.69 | Motor | Rimfire .55 |
| Max L/D | 24.15 | Weight (oz) | 9.5 |
| **Center-Body** | | RPM/V | 480 |
| Width (in) | 4 | Max Current (amp) | 45 |
| Airfoil | NACA 4424 | Propeller | 12x8E |
| Chord (in) | 24 | Electronic Speed Control | Castle Creations HV 60 |
| Height (in) | 6.5 | Servos | Hitec HS81 |
| **Winglets** | | Receiver | Spektrum AR7000 DSM2 |
| Height (in) | 3 | Battery Cells | Elite 1500 mAh |
| Root Chord (in) | 3.55 | # of Cells | 26 |
| Tip Chord (in) | 0.94 | Pack Weight (lb) | 1.2 |
| Airfoil | Flat Plate | Voltage (V) | 31.2 |

### 5.1.1 RAC DISCUSSION

The RAC—the heaviest recorded aircraft weight following any flight attempt with payload removed—was an important variable in the final team score calculation. Since the total flight score multiplied by the written report score is divided by the square root of the RAC value, minimizing RAC resulted in a higher total score. Assuming a 1.5 pound battery is used for all three missions, the team estimated the weights of the aircraft for each mission with payload removed to be 3.9, 4.1, and 4.3 pounds for missions one, two, and three respectively. Therefore, mission three's RAC value would likely be the limiting factor for the RAC value. The team brainstormed methods for reducing the mission three RAC value and determined that since the estimated flight time of mission three was the shortest, a smaller capacity battery could be used, lowering RAC weight and thus, increasing total team score.

## 5.2 Structural Characteristics & Capabilities

For mission 1, the aircraft had to be extremely agile and able to withstand load factors up to 6 corresponding to a 75 degree banked turn, while missions 2 and 3 required a load factor of 3 corresponding to a 60 degree banked turn (each with a 50 % margin of safety). To achieve this, the aircraft had to be as lightweight as possible while maintaining sufficient structural integrity of all key components.

### 5.2.1 Wing

The wing structure of the aircraft was a semi-monocoque design, with a fiberglass composite skin braced by an internal foam core. Following the lay-up of the composites, sections of the internal foam core were carved out to cut weight. The composite skin and internal foam core created an incredibly strong and lightweight wing structure that required no further internal bracing. The wing had to pass a simulated 2.5 G wingtip test during technical inspection, represented of the wing loading that would be experienced during flight, however, because the aircraft was designed to achieve load factors up to 6, the wing tip test was not difficult. Figure 5.1 depicts the aircraft's flight envelope, or V-n diagram for each mission, ranging from a load factor of 6 to -3.5.

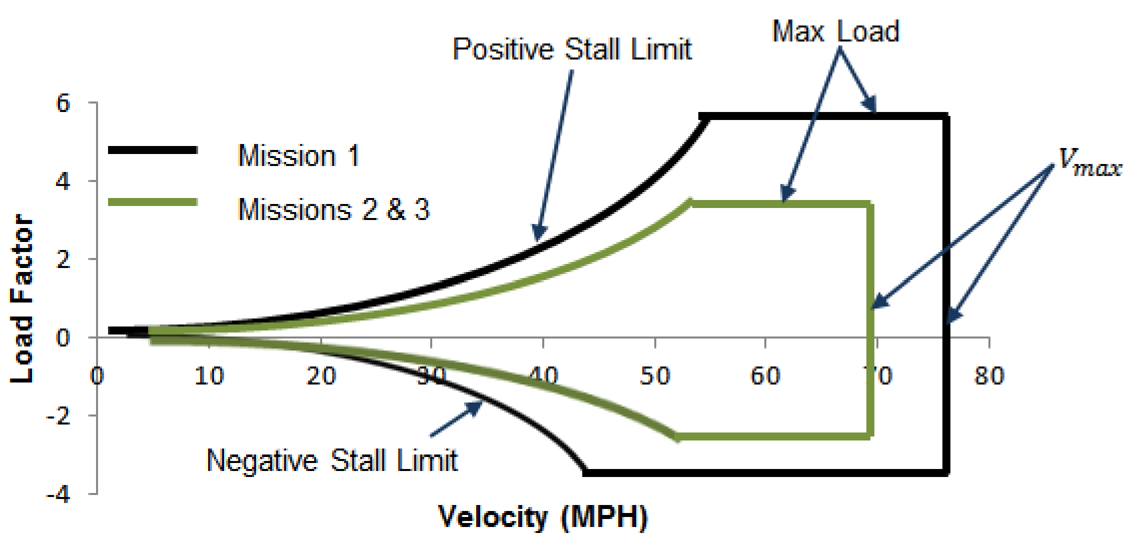



**Figure 5.1 – Flight Envelope**

### 5.2.2 Vertical Stabilizer

The team determined during flight testing that a vertical stabilizer was necessary to reduce the effects of motor toque. The vertical stabilizer was constructed of lightweight balsa wood with minimal materials to reduce weight. The vertical stabilizer is connected to the center-body by sliding in firmly to a notch removed from the aft center-body, and secured with a small amount of epoxy. Figure 5.2 shows the vertical stabilizer balsa wood truss design.

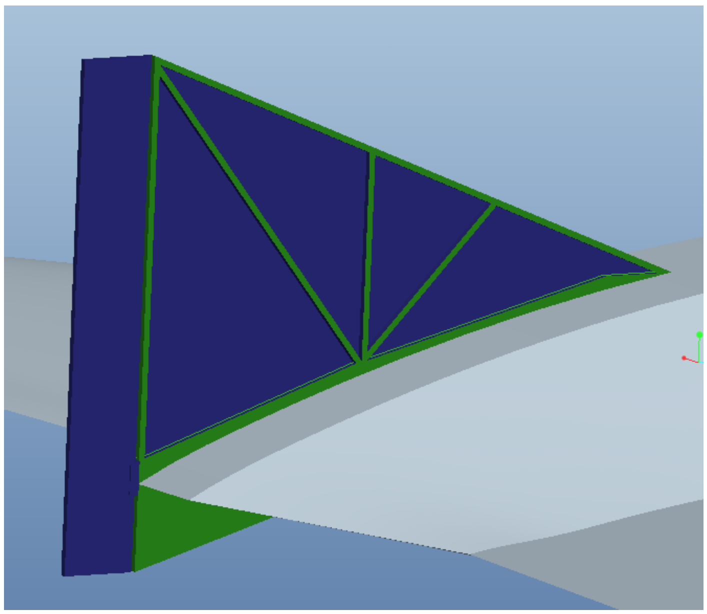


**Figure 5.2 – Vertical Stabilizer**

## 5.3 System & Sub-System Design

### 5.3.1 System Architecture

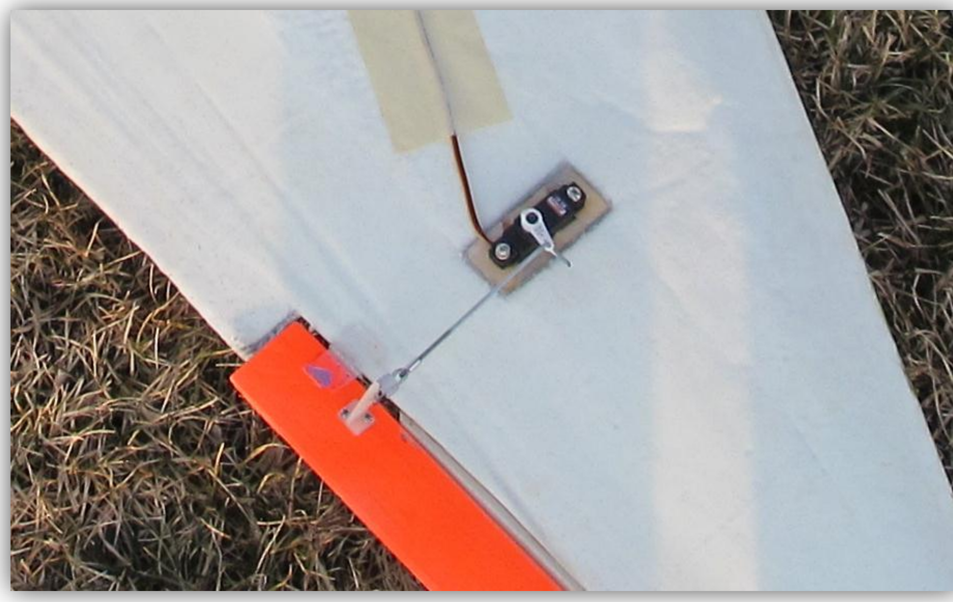
The unique mission set for this year's Design/Build/Fly competition posed a challenge for the design team in building an aircraft balanced for all 3 missions. The payload bay had to be able to hold the 8 simulated passengers as well as the 2 liters of water, all while minimizing weight and keeping assembly time to a reasonable limit. The team decided upon a 'pod' based system architecture. Each mission had a specific payload 'pod' that enabled the aircraft to be quickly made ready for flight while minimizing weight. The individual mission pods are explained in greater detail in the following sections.

### 5.3.2 Component Selection & Integration

<u>Receiver & Controller Selection</u> – The Spektrum AR7000 DSM2 Full Range receiver was selected because of the many controls it afforded the aircraft. The only control surfaces the aircraft had were its elevons and the rudder, which is also attached to the tail landing gear, allowing for steerable ground

control. Having researched flying wing aircraft, such as the Northrop Grumman B-2 bomber, it was determined that differential elevon control added more stability around turns by acting as an airbrake on the inside wing's tip. The Spektrum receiver made also has a failsafe feature allowing the team to program failsafe presets as required.

Servo Selection & Integration With Control Surfaces – Hitec HS81 servos were selected both because the team had good experience with them in the past and they were also extremely lightweight. The entire aircraft was comprised of 4 servos: 1 for each of the two elevons, 1 for the rudder which was attached to the steerable tail wheel, and 1 for the water release mechanism in mission 3. The 2 elevon servos and the 1 rudder servo were imbedded flush into the wing and center body to minimize drag. Figure 5.3 shows the servo set up for an elevon control surface.



**5.3 – Elevon Servo Connection**

The elevons were attached to the wing with plastic hinges. The rudder is attached to the vertical stabilizer in the same way, while the tail landing gear is mounted to the rudder using conventional hardware.

Propulsion System & Integration – Construction and testing on the initial prototype aircraft revealed that the foam and composite nose did not give sufficient structural support for the motor mount. Therefore, a design change was made for aircraft iteration 2 that consisted of an added laser-cut plywood motor mount embedded in the aircraft nose between layers of foam and composite. A similar landing gear support was also added on the underside of the aircraft immediately aft of the motor mount. Figure 5.4 depicts the final design solution for the integrated motor mount and landing gear mount system.

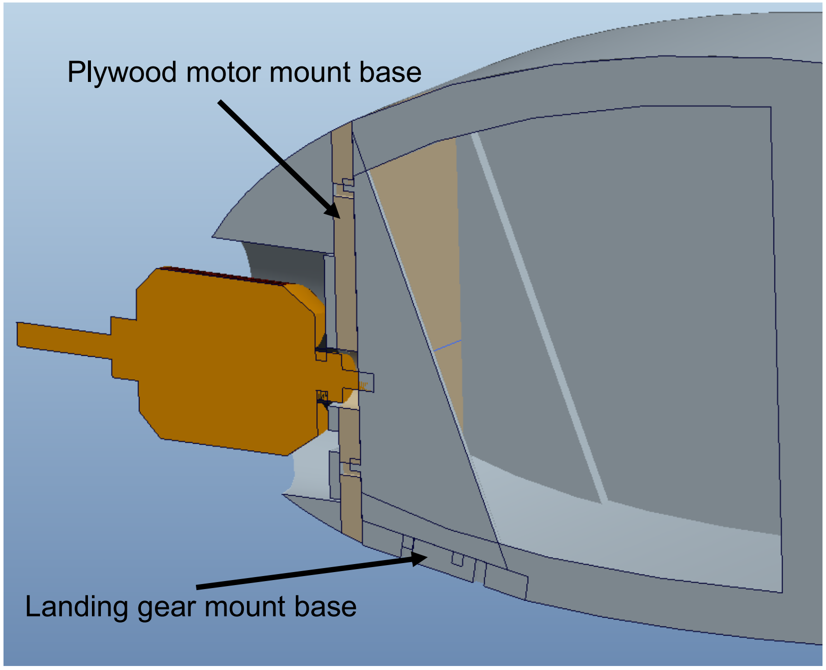



**Figure 5.4 – Plywood Reinforcing**

The internal electronics and battery pack had to be moveable within the payload bay to achieve the desired center of gravity location for each payload configuration. However, the components still had to be secured in place during flight. To solve this problem a strip of Velcro was installed along the bottom floor of the payload bay, and a small piece attached to the bottom of each electronic component. This allowed every electronic component, including the battery, to be easily shifted around by hand, but affixed firmly in place during flight.

Landing Gear Selection & Integration – The team decided upon a tail dragger wheel configuration with the rudder and steerable tail wheel linked together. The next design challenge was finding a lightweight strut that was strong enough to support the 8.5 lb aircraft on a potential hard landing. Carbon composite landing gear struts were selected for their strength to weight properties. Additionally, to cut down on drag, the struts—shown in Figure 5.5—came in symmetric airfoil cross sections.

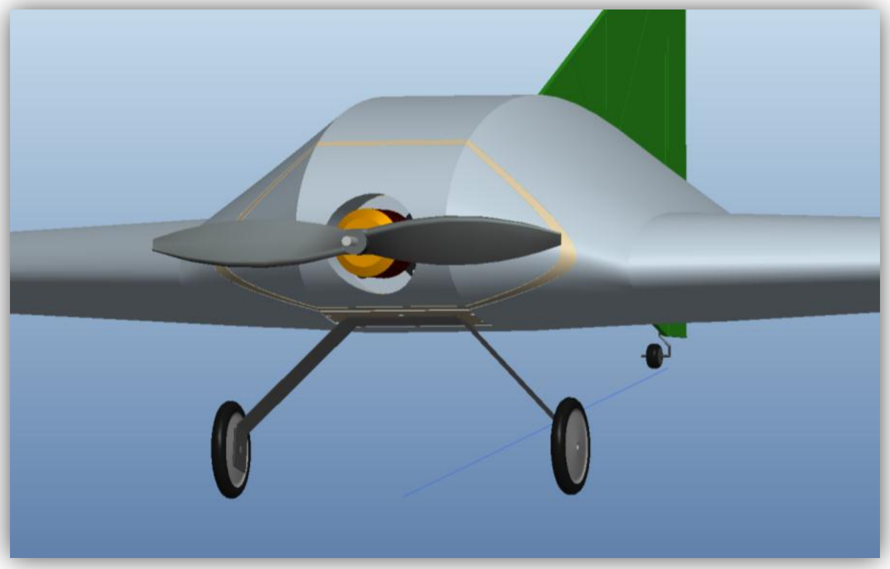


**Figure 5.5 – Landing Gear**

Winglet Integration – The winglets were canted inward towards the aircraft centerline by $5^0$, adding significant stability to the aircraft. The winglets were taped to the wing tips by running a piece of tape through a pre-cut notch in the winglet around the top and bottom surfaces of the wing. Figure 5.6 depicts the (A) winglet design and (B) winglet integration.

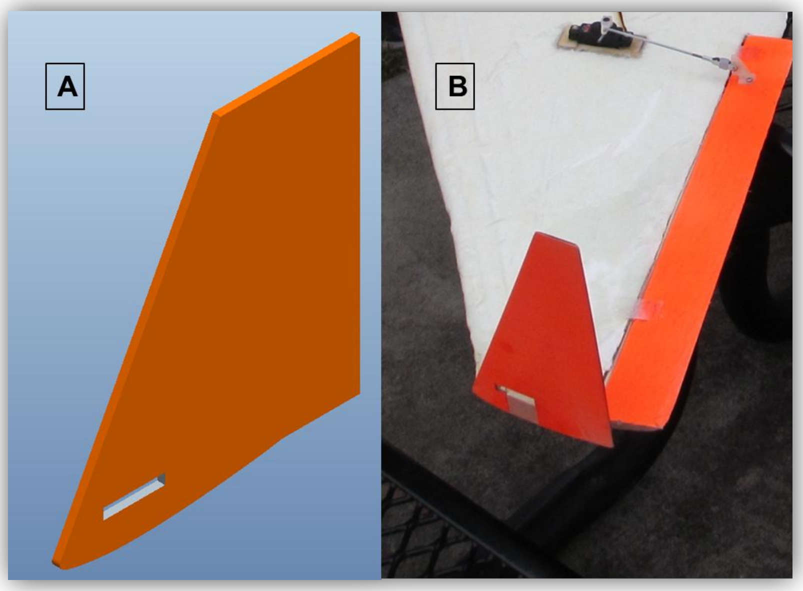



**Figure 5.6 – Winglet Design & Integration**

Water Stowage & Release System Integration – The mission 3 payload 'pod' mentioned in the previous section 5.3.1 was designed around a commercial 2-liter pop bottle. The team looked into numerous possibilities for stowing the water, but the 2-liter bottle was easily the lightest option available. Since the timer for mission 3 stops the clock once they see the stream of water, the team wanted to ensure the stream was a sufficient enough flow to be noticeable at 100 meters up and 500 feet away. To do this a trap-door design was inserted into one side of a 2 liter bottle and then made water tight with sealant. Figure 5.7 shows the designed water release mechanism.

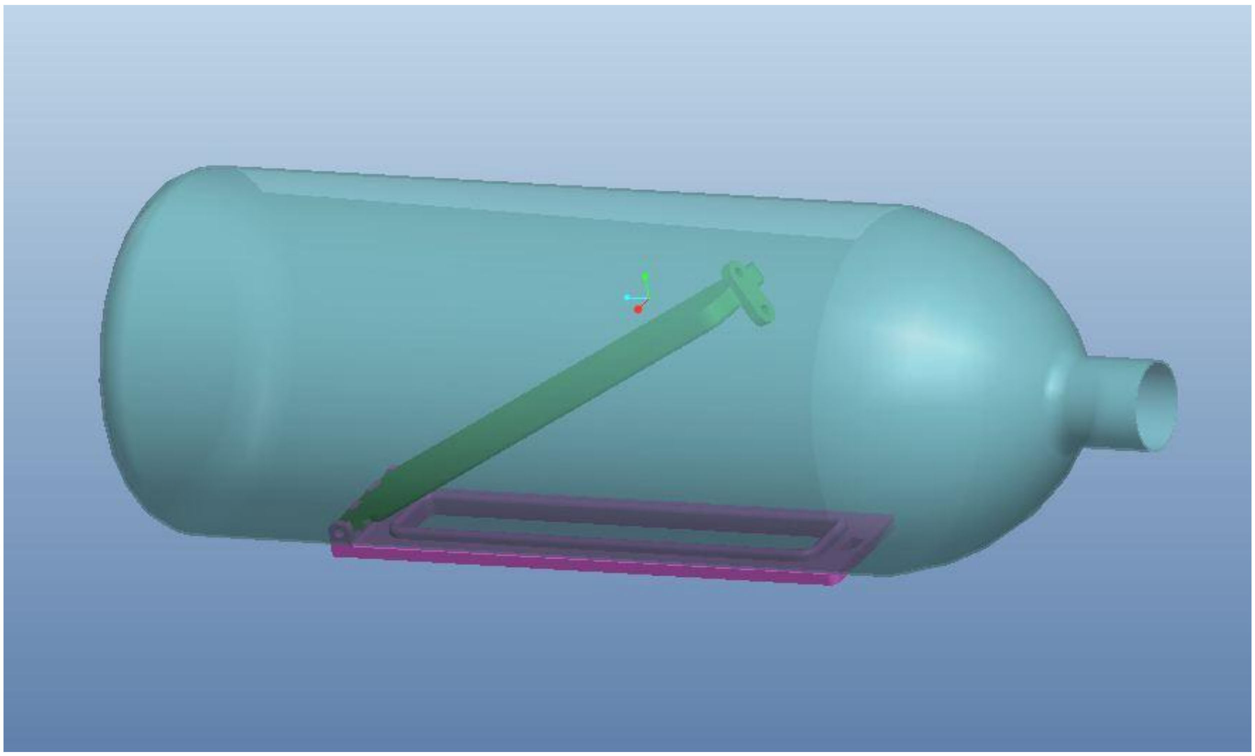


**Figure 5.7 – Water Release Mechanism**

The trap door was flush with the bottom of the aircraft, where a hole was cut out to allow the water through the trap door to drop out of the aircraft. The tank was then vented to outside air pressure via a pitot static tube that acted as a ram air pressurizer to push the water out faster. The trap door pivots on one side and is pinned down on the opposite side. Once the aircraft reaches 100 meters in altitude, the Soaring Circuits CAM sensor activates a servo that pulls the pin from the pinned down side of the trap door. The door, which is spring loaded to open automatically, immediately opens by the force of the spring once the servo pulls the pin. The servo for the water release system will be a part of the mission three payload insert and not installed in the aircraft for missions one and two.

Passenger Payload System & Integration – The competition requirements for the passenger layout require a specific "seating" arrangement of the aluminum blocks complete with aisles and modeled leg room. Figure 5.8 depicts the mission 2 'pod' that drops easily into the payload bay of the aircraft. The passenger pod was constructed of thin lightweight plywood pieces that were laser cut to ensure precise fitting joints. The team researched aluminum densities and found that aluminum alloy 5086 was the least dense, and therefore the best choice for the passengers. This was an important optimization because the team wanted to get the 8 aluminum blocks as close as possible to weighing 3.75 lb. The score for mission three was normalized by 3.75, so any excess aluminum weight above 3.75 lb negatively affected the team's mission 2 flight score.

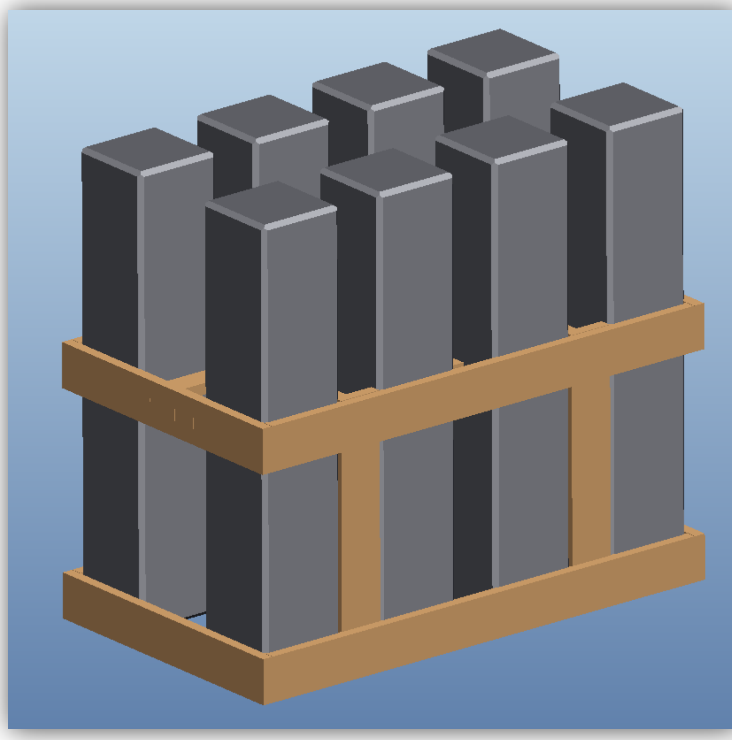


**Figure 5.8 – Passenger Payload Pod**

## 5.4 Weight & Balance

The aircraft's weight is critical to the total competition score so tracking the weight of the aircraft was very important for the team. Figure 5.9 shows how the weight of the aircraft changed throughout all the design phases of the project. As the complexities of the project increased, so did the weight of the design. The prototype aircraft was 1.6 pounds overweight from the initial conceptual design goal. However, the team believes that the difference can be cut down to only 0.5 pounds overweight in the final competition-ready aircraft through more careful and methodical manufacturing techniques in the final aircraft iteration.

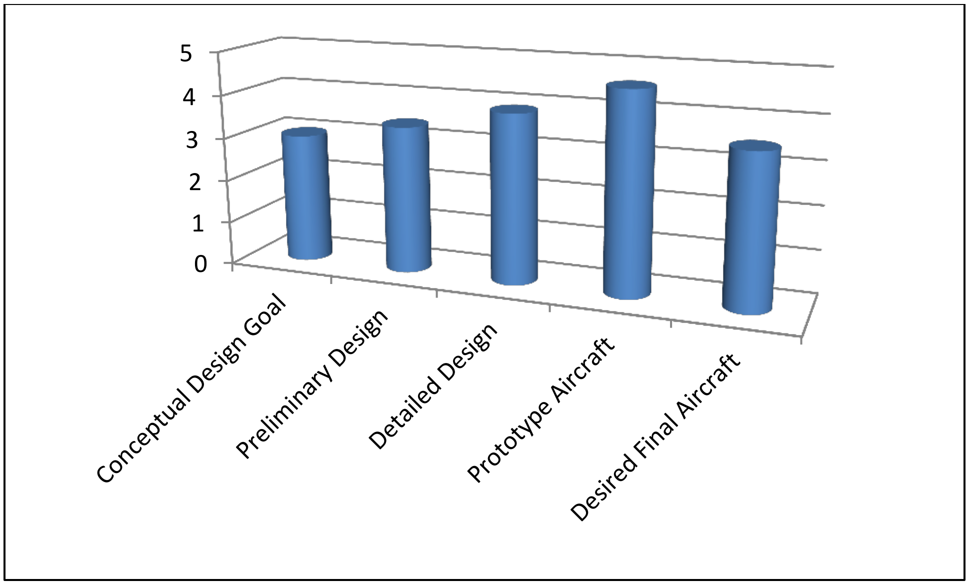



**Figure 5.9 – Weight Tracking Throughout Design Process**

### 5.4.1 Weight & Balance Chart

The balance of the aircraft is also of critical importance in controllability. The desired static margin must be maintained by correctly placing the center of gravity to ensure authoritative, but not unstable, control response. The weights and center of gravity locations of each component included in the aircraft are listed in Table 5.2, where the arm column lists the distances aft of the nose of the aircraft.

**Table 5.2 Weight & Balance Chart**

| Item Description | Weight (lb) | Arm* (in.) | Moment (lb.-in.) |
|---|---|---|---|
| Battery (example) | 1.500 | 4.000 | 6.000 |
| Motor (with prop.) | 0.532 | 1.500 | 0.798 |
| ESC | 0.140 | 3.000 | 0.420 |
| Receiver | 0.036 | 3.000 | 0.108 |
| Receiver Battery | 0.114 | 3.000 | 0.342 |
| Servo 1 | 0.038 | 13.000 | 0.494 |
| Servo 2 | 0.038 | 13.000 | 0.494 |
| Servo 3 | 0.038 | 20.000 | 0.760 |
| Empty Plane | 1.500 | 11.000 | 16.500 |
| **Total Weight:** | **3.936** | **Total Moment:** | **25.916** |

### 5.4.2 Mission Weight Breakdown

The breakdown of weight for each mission is displayed in Figure 5.10.

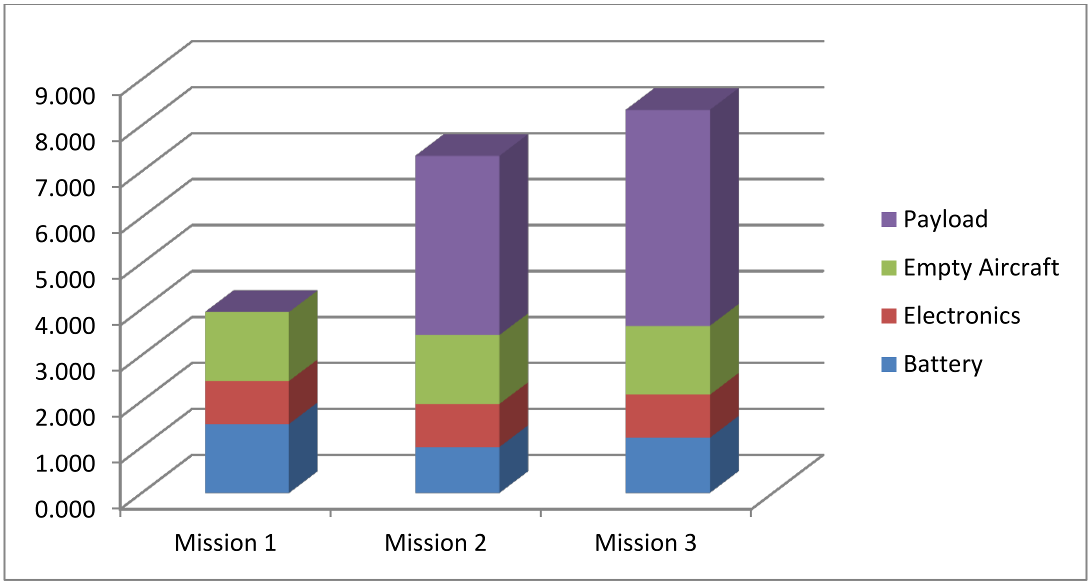



**Figure 5.10 – Mission Weight Breakdown**

## 5.5 Flight & Mission Performance

The final aircraft's flight and mission performance parameters are displayed in Table 5.3. The team created a very competitive aircraft in all aspects of flight and mission performance.

**Table 5.3 – Aircraft & Mission Parameters**

| Aircraft Parameters | |
|---|---|
| $C_{Lcruise}$ | 1.1 |
| $C_{Lmax}$ | 1.39 |
| $e$ | 0.86 |
| $C_{Do}$ | 0.69 |
| $L/D_{max}$ | 24.15 |
| Max Thrust (lb) | 8.0 |
| Max Speed (mph) | 63.0 |
| Empty Flight Weight (lb) | 3.936 |
| Max Payload (lb) | 4.4 |
| Max Takeoff Weight (lb) | 8.336 |

| Mission Parameters | Mission 1 | Mission 2 | Mission 3 |
|---|---|---|---|
| GTOW (lb) | 3.936 | 7.336 | 8.336 |
| Payload (lb) | 0 | 3.9 | 4.7 |
| Takeoff Roll (ft) | 21 | 65 | 82 |
| Climb Rate (ft/s) | 48 | 19 | 12 |
| Cruise Speed (mph) | 61.0 | 47.0 | 42.0 |
| Stall Speed (mph) | 11.2 | 20.5 | 23.1 |
| Max Bank Angle (deg) | 75 | 40 | 40 |
| Turn Rate (deg/s) | 270 | 120 | 120 |
| Load Factor (n) | 6.0 | 3.8 | 3.8 |
| Mission Duration (min) | 4.2 | 2.5 | 2.0 |

### 5.5.1 Mission Performance

In mission one, the team wished to complete 8 laps. In order to accomplish this, it was calculated that the aircraft needed to fly 60.7 mph. Since the final aircraft was predicted to achieve speeds of 63.0 mph, 8 laps was deemed achievable. However, the team felt the positive margin was likely too little to complete a $9^{th}$ lap. Therefore, by Equation 3.1, M1 was predicted to be 2.333.

Mission 2 score was a function of aircraft flight weight. The predicted flight weight of 7.336 pounds yielded a M2 score of 2.011 by Equation 3.2.

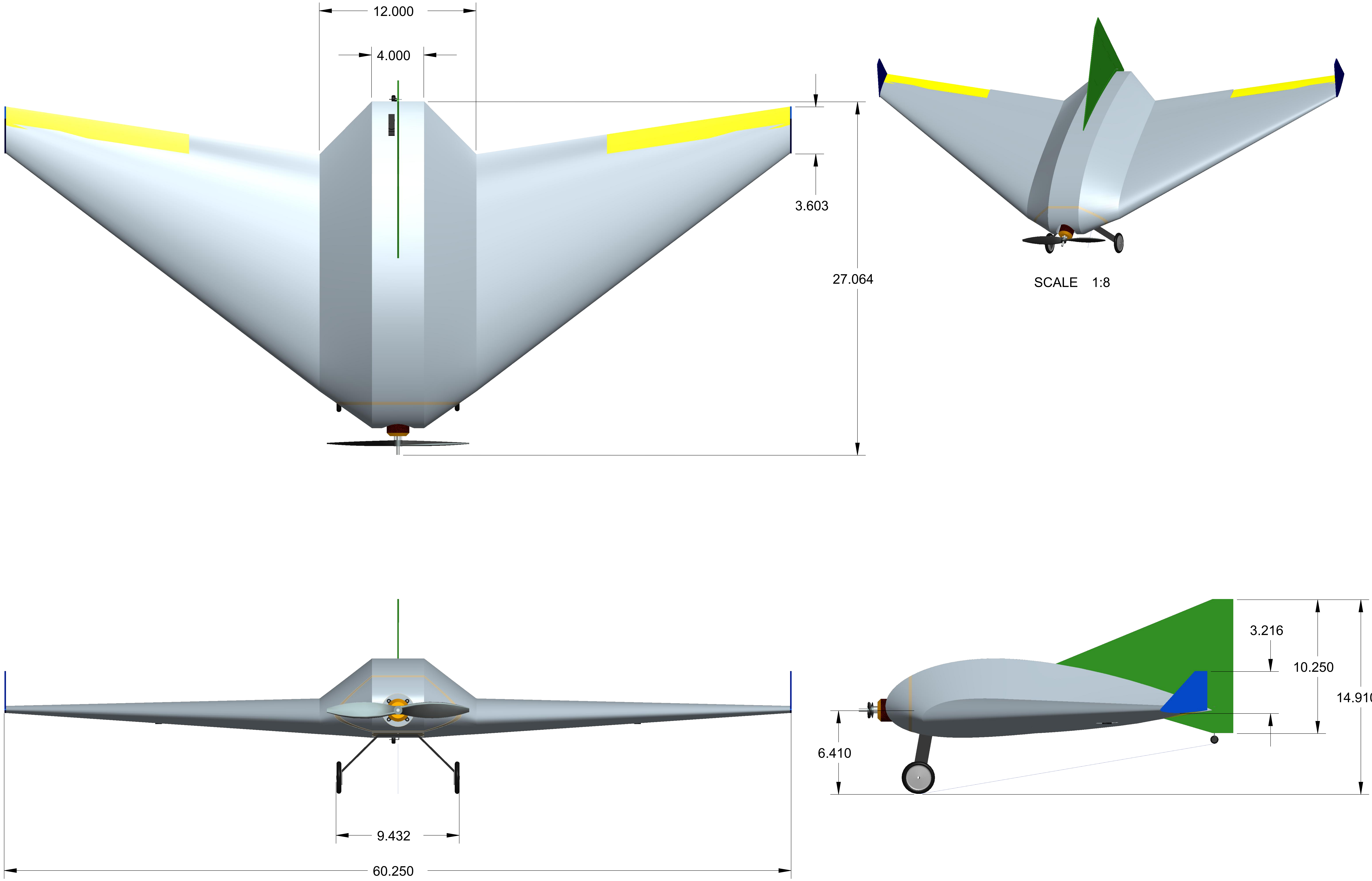
Mission 3 score was a function of the average time to climb of all teams divided by the team's personal time to climb. This ratio was predicted to be roughly equal to 2. M3, therefore, by Equation 3.3 was predicted to be 3.414.

Total predicted flight score was 7.758. With a calculated RAC of roughly 4.2 pounds, the team's total predicted score was found to be 3.786 * Report Score. Assuming a report score in the range of 80 - 100, the total score would be in the range of 303 - 378.6.

## 5.6 Drawing Package

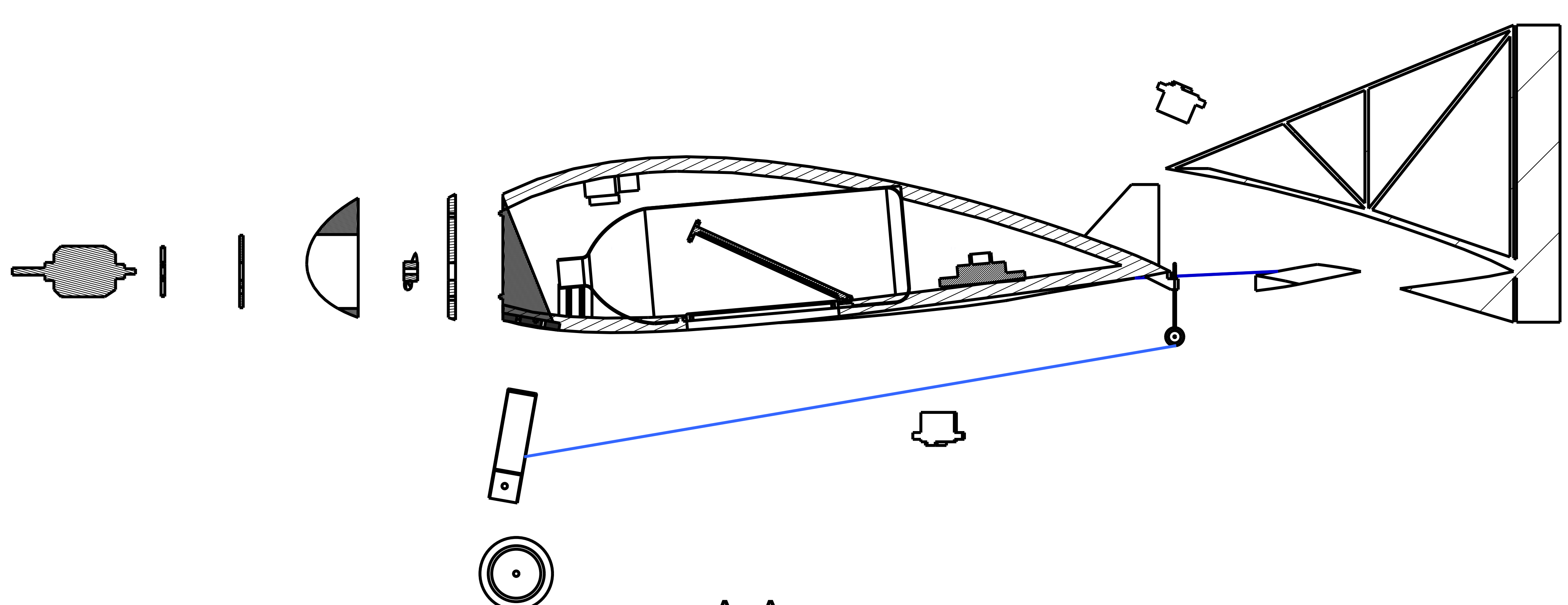
The following four pages illustrate the detailed CAD drawings of the team's aircraft. Drawings included are:

- Three-view drawing and general dimensions
- Structural arrangement exploded view
- Systems layout
- Missions 2 and 3 payload accommodation



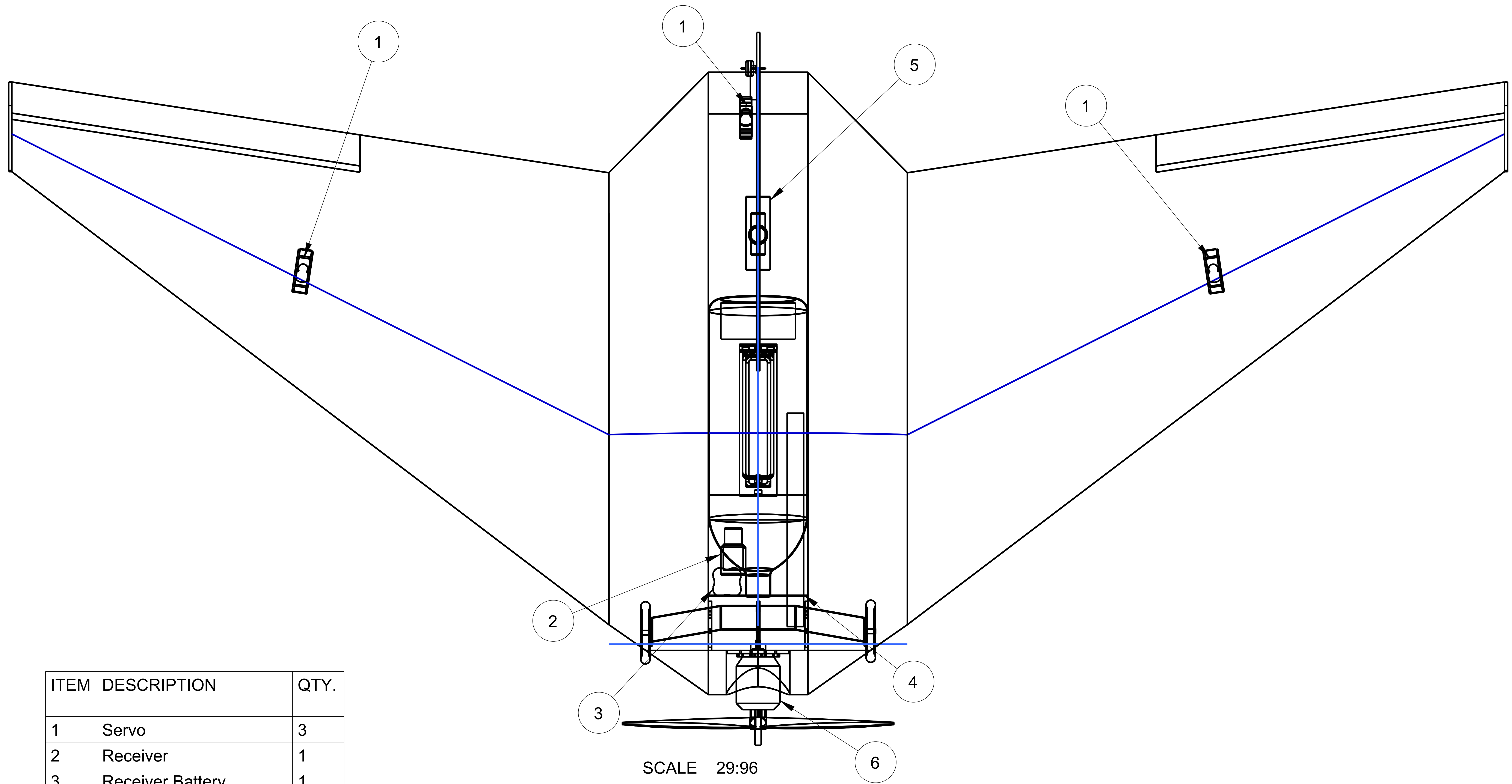

| University of Notre Dame Design/Build/Fly | | |
|---|---|---|
| Drawing Type - ASM | Release - FINAL | Page 43 |
| ALL DIMENSIONS IN INCHES - Drawing # - N00-1 | | Revision- 1 |

SCALE 1:4



SECTION A-A
SCALE 19:96

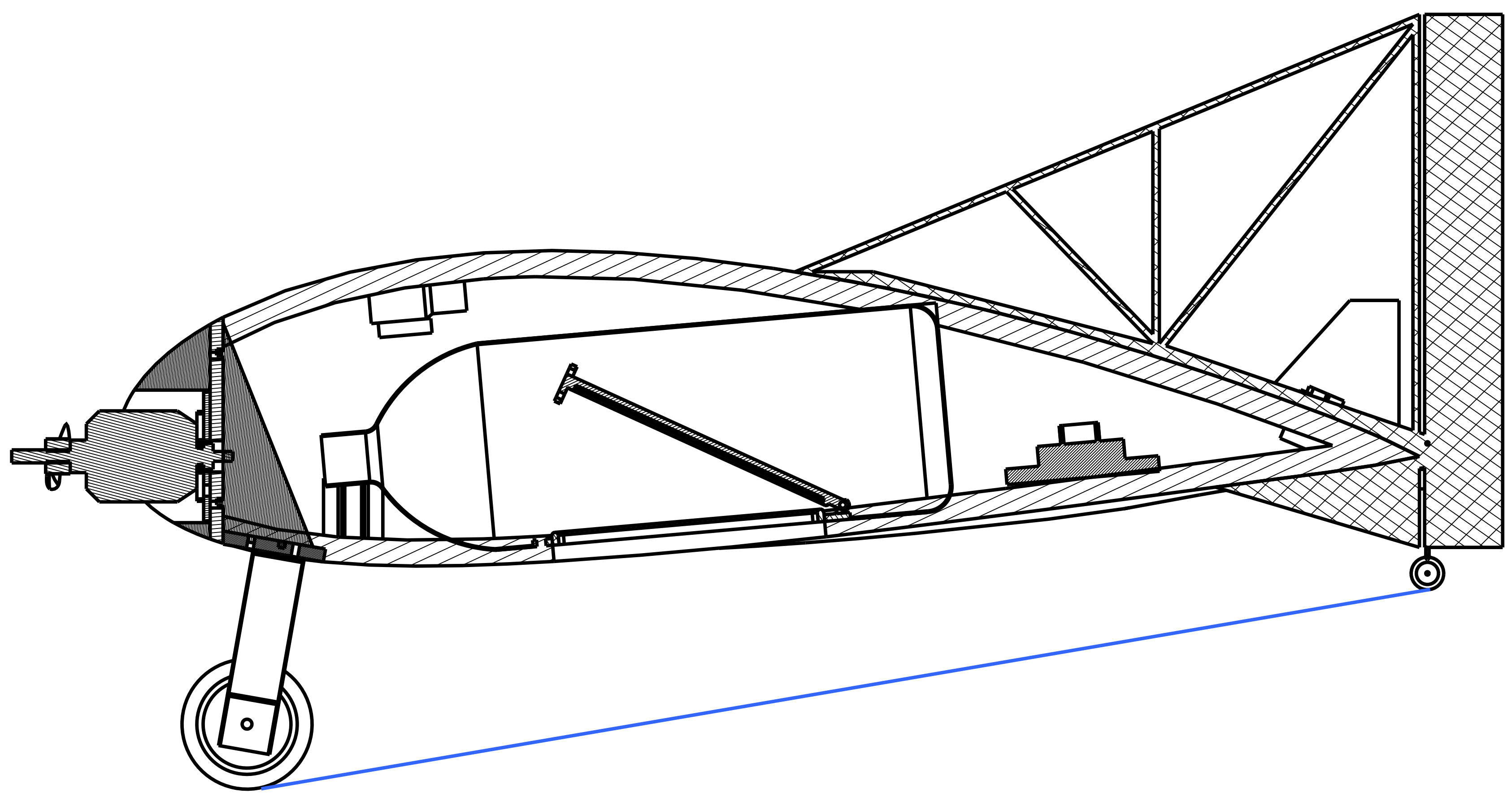
| University of Notre Dame Design/Build/Fly | | |
|---|---|---|
| Drawing Type - EXPLODE | Release - FINAL | Page 44 |
| | Drawing # - N00 | Revision- 1 |



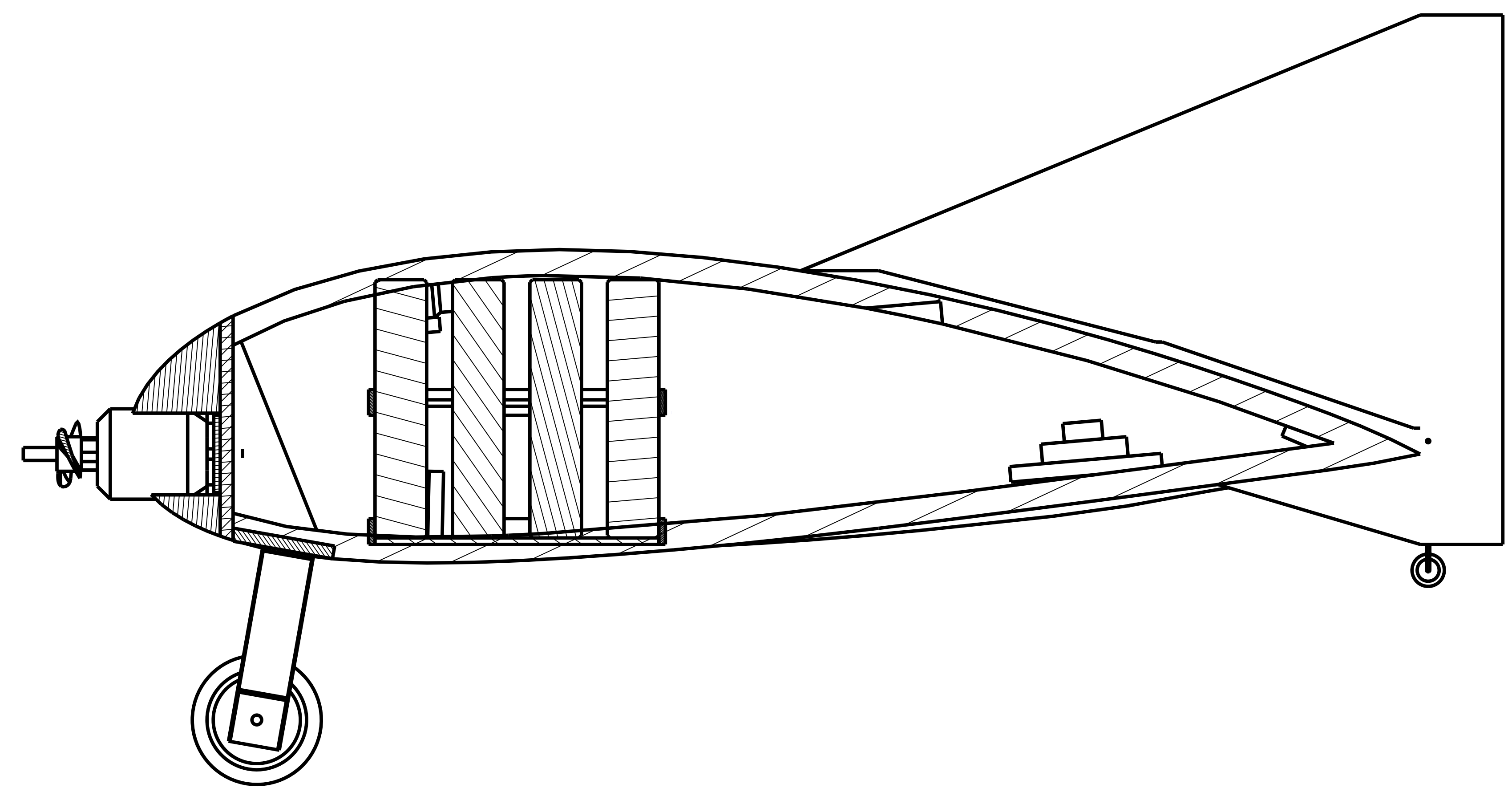

SCALE 29:96

| ITEM | DESCRIPTION | QTY. |
|---|---|---|
| 1 | Servo | 3 |
| 2 | Receiver | 1 |
| 3 | Receiver Battery | 1 |
| 4 | Main Battery | 1 |
| 5 | ESC | 1 |
| 6 | Motor | 1 |

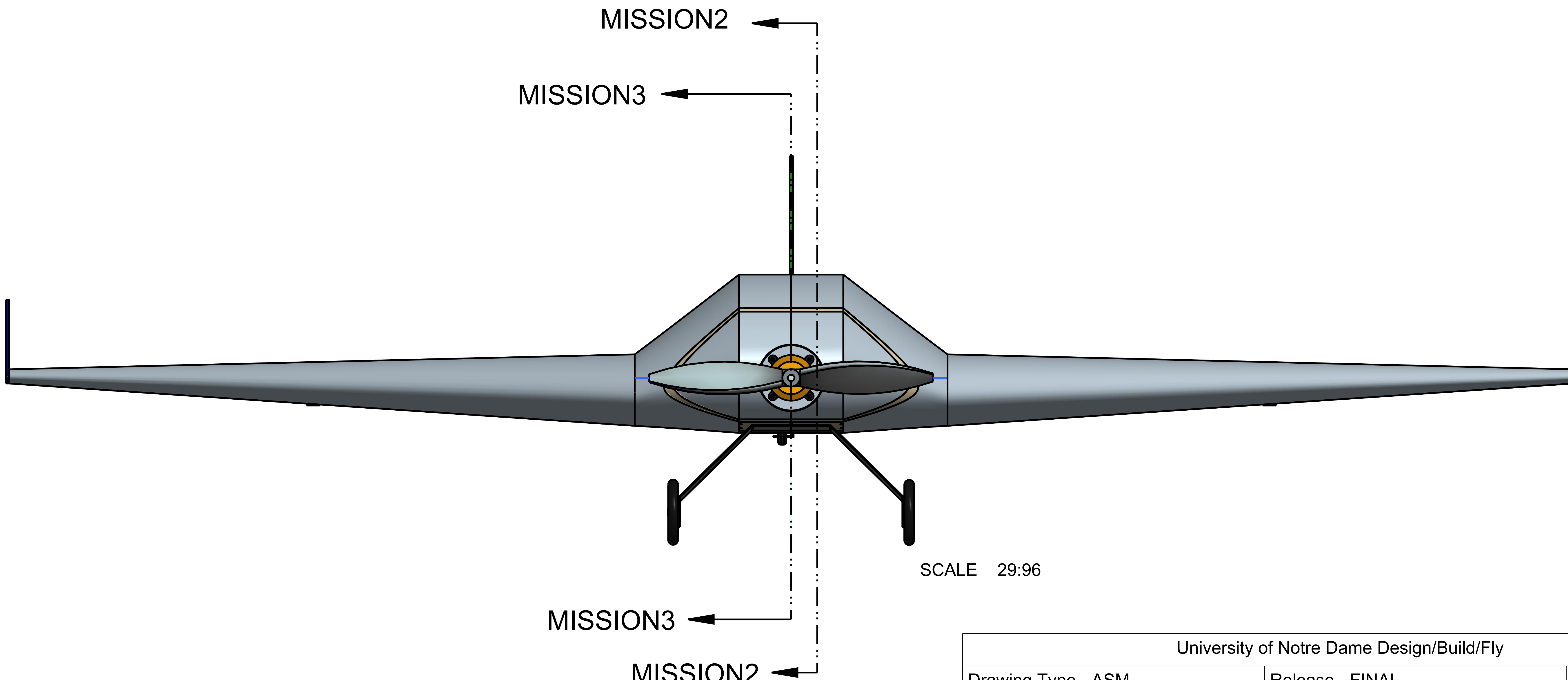
| University of Notre Dame Design/Build/Fly | | |
|---|---|---|
| Drawing Type - SYSTEMS | Release - FINAL | Page 45 |
| | Drawing # - N00 | Revision- 1 |



SECTION MISSION3-MISSION3
SCALE 29:96



SECTION MISSION2-MISSION2
SCALE 29:96




SCALE 29:96

| University of Notre Dame Design/Build/Fly | | |
|---|---|---|
| Drawing Type - ASM | Release - FINAL | Page 46 |
| | Drawing # - N00-1 | Revision- 1 |

# 6.0 Manufacturing Plan & Processes

The construction team followed a plan that broke the aircraft up into systems and subsystems. The team devoted their time and expertise in model aircraft construction to ensuring a seamless and refined final aircraft.

## 6.1 Construction Material/Process Analysis & Selection

During detailed design, the team considered many viable options for the manufacturing of their aircraft. The manufacturing process would define the material used along with the weight and structural strength of the aircraft. Since weight of the final design was so critical, the team dedicated time to achieve the lightest possible aircraft structure. The various methods and processes considered for construction of the aircraft were:

Balsa Wood Build-Up: A balsa wood built-up aircraft is the standard for small scale remote controlled aircraft because of its lightweight characteristics. This construction process is relatively straightforward and the final product is minimalistic and lightweight. One drawback of this method is that a balsa built-up aircraft is not as durable as aircraft manufactured using other techniques meaning that accidents often result in complete destruction, which can pose challenges for flight testing, especially with a flying wing design where proper CG location requires some trial and error. The aircraft also cannot take as high of load factors. The balsa built-up aircraft frame can then be covered with either Monokote, thin plywood, or fiberglass composites.

Rapid Prototyping: Rapid prototyping (also known as 3D printing) machines build computer generated structures from a resin material with exceptional accuracy. The team became interested in manufacturing sub-sections or small components of the aircraft using this method after hearing of a university group in the United Kingdom that had done something similar. However, the cost and weight penalties of making significant airframe components with rapid prototyping were such that is was removed from consideration for any but very small parts.

Composite With Foam Core: Foam core airframe covered in composite (either fiberglass or carbon fiber) gives a structure with exceptional strength to weight characteristics as well as the ability to form complex curves. The manufacturing process is more complicated than other methods, requiring a means to cut the foam to shape, join it as necessary, and then lay-up and cure the composite material. This structure must then be sanded and finished before other components are installed. The team decided that a composite airframe would provide not only the most competitive design, but also a valuable learning experience since the team had not before worked with composites.

## 6.2 Prototype Construction

The prototype aircraft was a test of the two major risks the team decided to take in the design: a flying wing configuration constructed out of composite materials. Successful construction and flight of the prototype would validate these design decisions and provide information as to what design changes were necessary for the competition aircraft.

### 6.2.1 Construction Technique Demonstrator

Figure 6.1 shows the main steps in the construction process.

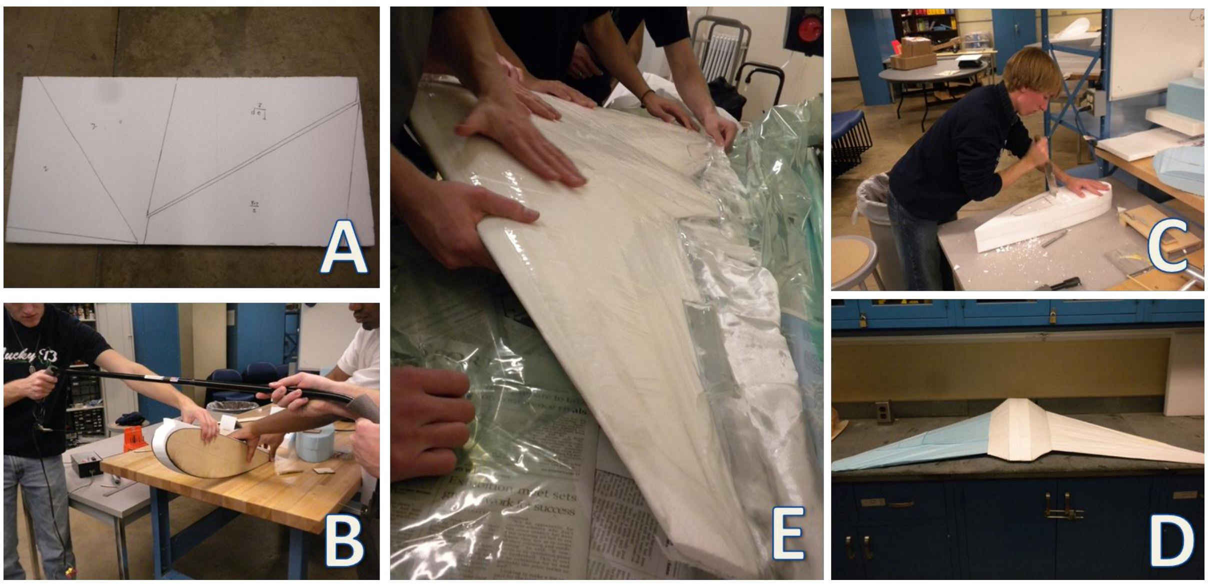



**Figure 6.1 – Construction Process**

The first step in the construction process was (A) the cutting of the foam that forms the core of the airframe. The CAD model of the aircraft was used to laser-cut plywood airfoil cross sections for various positions along the span wise distribution of the aircraft which were used for (B) guides in the foam cutting process. These were attached to foam pieces of the appropriate size, and a manual hotwire then followed the guides to cut out the wing panels, aircraft center body, and center body blends. These foam components were (C) hollowed as necessary and then (D) joined to form the core of the airframe. This structure was (E) covered with fiberglass cloth and allowed to cure overnight. The final airframe structure was then trimmed, sanded, and prepped for installation of other components.

The integration of the final aircraft components started with installation of the laser-cut plywood mounts for the motor, landing gear, and servos. The top and bottom access hatches were cut from the center body section and access hardware was installed. Electronic components, including servos, motor, speed controller, and receiver were installed and secured. The vertical tail was constructed from balsa, covered in MonoKote, and installed in the airframe.  Finally, landing gear, winglets, and control surfaces were installed, giving the completed aircraft. Figure 6.2 shows the internal payload bay without electronics installed.

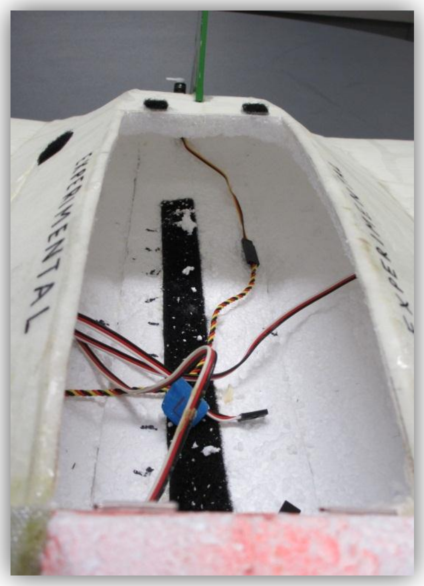


**Figure 6.2 – Payload Bay With and Without Electronics**

The trap door of the water payload mechanism was machined on a CNC router out of high density plastic from a CAD model.  This allowed for the minimal amount of material to be used, creating the lightest design possible.  The passenger payload insert was manufactured using laser-cut plywood.  Figure 6.3 shows the cutting of the pieces for the passenger payload insert.

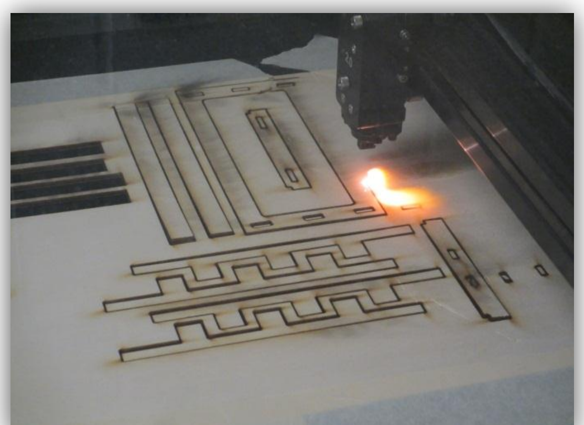


**Figure 6.3 – Passenger Payload Insert Manufacturing**

## 6.3 Manufacturing Schedule & Milestones

The team created a master manufacturing schedule, shown in Figure 6.4, to ensure that appropriate progress with aircraft manufacturing was being made at all times. Apart from a delay in the manufacturing of the mission two and three payload inserts, manufacturing largely stayed on schedule.

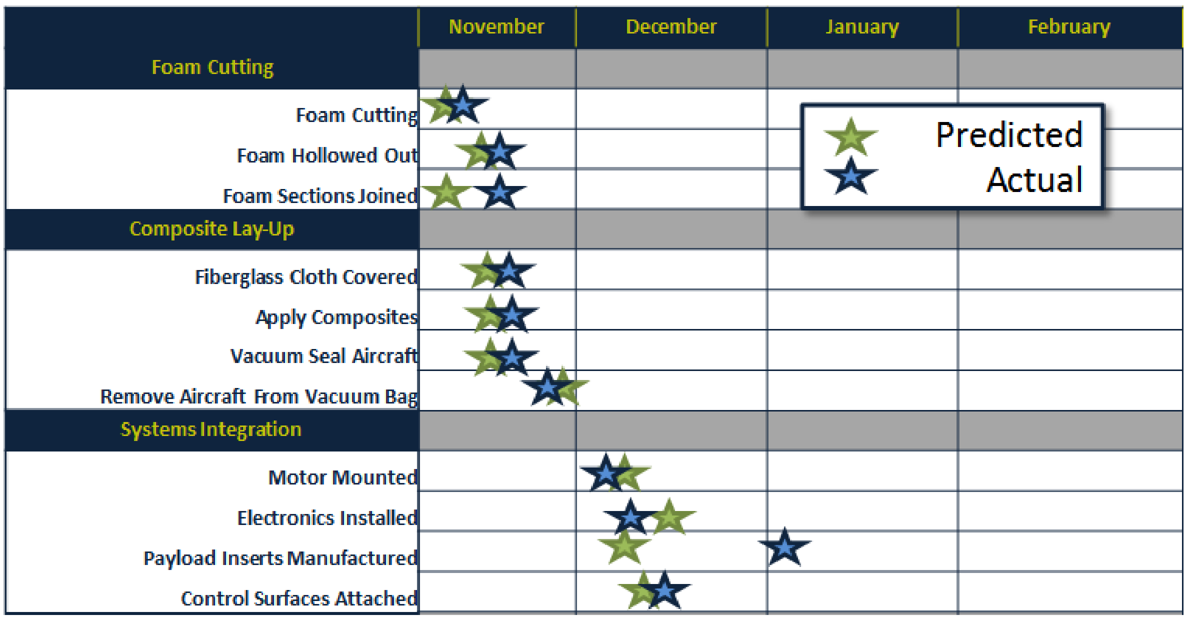



**Figure 6.4 – Manufacturing Schedule & Milestones**

# 7.0 Testing Plan

Testing was critical for the team's iterative design methodology since follow-up aircraft iterations were improved based on test data of the prototype aircraft and sub-systems. Through detailed sub-system and flight testing, the aircraft was refined to perform optimally for each mission. This section details the plan and processes of testing performed on the aircraft, with the results of these tests discussed in section 8.

## 7.1 Test Objectives

### 7.1.1 Propulsion

Propulsion testing was the first testing performed and paved the way for more detailed performance estimates of the aircraft. Propulsion tests consisted of running the entire propulsion system—consisting of the motor, battery, electronic speed control, and receiver—to a testing stand inside of the 5' x 5' subsonic wind tunnel at the Hessert Center for Aerospace Research. The testing plan matrix for the propulsion system is depicted in Table 7.1. The two motors used were the Rimfire 0.55 motor planned for the final design and a Kontronik Fun 480-33 motor used in the previous year's competition. In this way, comparisons could be drawn between the two motors' performance.

**Table 7.1 – Propulsion Testing Plan Matrix**

| Trial | Motor | Battery | Cells | Voltage (V) | Trial Description |
|---|---|---|---|---|---|
| 1 | Rimfire .55 480 Kv | Elite 1500mAh | 12 | 14.4 | Sweep throttle; record amps & thrust |
| 2 | | | 12 | 14.4 | Throttle @ 20 amps; record run time |
| 3 | | | 26 | 31.2 | Sweep throttle; record amps & thrust |
| 4 | | | 26 | 31.2 | Throttle @ 20 amps; record run time |
| 5 | | 2100 mAh | 12 | 14.4 | Sweep throttle; record amps & thrust |
| 6 | | | 12 | 14.4 | Throttle @ 20 amps; record run time |
| 7 | Kontronic Fun 480-33 | Elite 1500mAh | 12 | 14.4 | Sweep throttle; record amps & thrust |
| 8 | | | 12 | 14.4 | Throttle @ 20 amps; record run time |
| 9 | | | 26 | 31.2 | Sweep throttle; record amps & thrust |
| 10 | | | 26 | 31.2 | Throttle @ 20 amps; record run time |
| 11 | | 2100 mAh | 12 | 14.4 | Sweep throttle; record amps & thrust |
| 12 | | | 12 | 14.4 | Throttle @ 20 amps; record run time |

To determine the effect of changing the propeller size, the motor, battery and voltage were held constant while the propeller was changed, causing a change in amps, thrust, and run time. Table 7.2 shows the propeller testing matrix.

**Table 7.2 – Propeller Testing Plan Matrix**

| Trial | Electronics | Propeller | Trial Description |
|---|---|---|---|
| 1 | Motor: Rimfire .55 480 Kv<br>Battery: Elite 1500 mAh<br>26 cells, 31.2 Volts | 11 x 7E | Sweep throttle; record amps & thrust |
| 2 | | | Throttle @ 20 amps; record run time |
| 3 | | 13 x 10E | Sweep throttle; record amps & thrust |
| 4 | | | Throttle @ 20 amps; record run time |
| 5 | | 15 x 8E | Sweep throttle; record amps & thrust |
| 6 | | | Throttle @ 20 amps; record run time |

### 7.1.2 Structures

As a minimum, the aircraft had to satisfy the competition-administered 2.5g wingtip test. The team, however, desired better structural performance than this minimum. Therefore, 4.5 pounds of steel weights were placed atop the already fully-loaded aircraft to perform the wingtip test. The aircraft was capable of sustaining this extreme loading scenario, which proved the aircraft could withstand high load factors in flight.

In addition to the aforementioned structural test, the prototype aircraft survived—relatively unscathed—multiple crashes in which the aircraft hit the ground at high speeds and unusual attitudes. In most cases the aircraft bounced off the ground in one piece and was ready to fly minutes later. The greatest damages came in the form of a few broken propellers which were easily replaced. The team was highly satisfied with the robustness and strength of the fiberglass shell and foam core structure.

### 7.1.3 Flight Performance & Aerodynamics Testing

The main objective of the prototype aircraft flight testing was to verify and validate the aerodynamic models and simulations. The prototype aircraft was the simplest aircraft built and lacked all the systems of the final design. The pilot gained experience in flying the aircraft and his suggestions, along with recorded flight data from an onboard data acquisition system allowed the team to make small design changes to improve responsiveness, performance, and ease of control. Subsequent iterations saw the integration of the landing gear, larger winglets, a temporary vertical tail, increased wing reflex, larger elevon control surfaces, electronic fuse, and the payload inserts for missions two and three.

The pre-flight checklist displayed in Figure 7.1, details the necessary steps to thoroughly and safely preparing the aircraft for flight. Figure 7.2 shows a hand launching of the prototype aircraft during a test flight.

| Propulsion Test Checklist | |
|---|---|
| 1. Batteries Charged | ☐ |
| 2. Receiver off | ☐ |
| 3. All connections secured | ☐ |
| 4. Propeller & Motor Secured | ☐ |
| 5. Data system on | ☐ |
| 6. Fuse inserted | ☐ |
| 7. Battery secured & connected | ☐ |
| 8. Throttle kill switch on | ☐ |
| 9. Receiver on | ☐ |
| 10. Propeller Clear | ☐ |
| 11. Transmitter on | ☐ |
| 12. Throttle test | ☐ |

| Payload | |
|---|---|
| 1. Correct Weight | ☐ |
| 2. Securely fastened | ☐ |
| **Aircraft** | |
| 1. CG correct | ☐ |
| 2. Tight control surface connections | ☐ |
| 3. Landing gear secure | ☐ |
| **Final Checks** | |
| 1. Receiver on | ☐ |
| 2. Transmitter on | ☐ |
| 3. Check control surfaces | ☐ |
| 4. Check wind & weather | ☐ |
| 5. Payload hatch secure | ☐ |
| 6. Aircraft lined up with runway centerline | ☐ |
| 7. Permission for takeoff | ☐ |

**Figure 7.1 – Pre-Flight Checklist**

**Figure 7.2 – Prototype Aircraft Hand Launch**

## 7.2 Master Test Schedule

The team created a master test schedule, shown in Table 7.3, to ensure that appropriate progress with testing was being made at all times.

**Table 7.3 – Master Test Schedule**

| Test | Planned Date | Actual Date |
|---|---|---|
| Fiberglass Composite Manufacturing Test | 24-Sep | 24-Sep |
| Electrical System Power On | 5-Nov | 1-Nov |
| Propulsion Testing | 21-Jan | 21-Feb |
| Structures Testing (Wingtip test) | 21-Jan | 21-Jan |
| Prototype Aircraft First Flight | 21-Jan | 19-Jan |
| Prototype Aircraft Testing | 21-Jan thru 4-Feb | 19-Jan thru 25-Feb |
| Aircraft Iteration 1.2 First Flight | 28-Feb | |
| Aircraft Iteration 1.2 Flight Testing | 28-Feb thru 24-Mar | |
| Payload Testing | 3-Mar | |
| Final Aircraft First Flight | 24-Mar | |
| Final Aircraft Flight Testing | 24-Mar thru 7-April | |
| Mock Competition Tests | 7-Apr | |

# 8.0 Performance Results

The data from all the tests performed on the aircraft and sub-systems was compiled and analyzed. The aircraft was then modified in order to correct any issues that surfaced or to optimize performance where possible. The results of these tests are presented in this section.

## 8.1 Subsystem Performance

For the purpose of simplifying tests and experiments, the aircraft was broken down into its individual subsystems. This allowed for tests to be customized to better evaluate the components in question.

### 8.1.1 Aerodynamics, Stability & Controls Performance

The aircraft was equipped with a data acquisition device that recorded velocity, altitude, and location data during the test flight using GPS signals. The aircraft was flown in a typical course lap to model the mission characteristics. The results of this test are reported in Table 8.1.

**Table 8.1 – Flight Test Results**

| Parameter | Expectation | Test Result | Difference |
|---|---|---|---|
| Empty Flight Weight (lb) | 3.936 | 4.6 | 16.9% |
| Max Takeoff Weight (lb) | 8.336 | 8.336 | 0.0% |
| Max Takeoff Roll (ft) | 82 | 90 | 9.8% |
| Max Climb Rate (ft/s) | 48 | 41 | -14.6% |
| Max Speed (mph) | 63 | 59 | -6.3% |
| Min Stall Speed (mph) | 11.2 | 7.5 | -33.0% |
| Mission One Complete Laps | 8 | 8 | 0.0% |
| Mission Two Duration (min) | 2.5 | 2.6 | 4.0% |
| Mission Three Time to Climb (min) | 0.333 | 0.45 | 35.1% |

Further flight tests were devoted to determining the requirement of vertical surfaces. These tests included 4 flight tests in which flights were made with a) no winglets or vertical stabilizer, b) only winglets, c) only a vertical stabilizer, and d) both winglets and a vertical stabilizer. It was made clear that flying without winglets or a vertical stabilizer was not realistic because the aircraft immediately wanted to yaw when loaded in takeoff configuration due to motor torque and P-factor. The pilot and observers noticed little difference in controllability between flights b and c, but slightly more controllability in flight d where both vertical surfaces were included. The team decided that winglets were slightly more desirable over the central vertical tail because winglets served the double purpose of also reducing drag due to wingtip vortices. The team however had more trouble in deciding whether to go with only winglets or both winglets and a vertical tail. The team decided to perform more follow-up tests in which the aircraft would be loaded with payload to see if that made a difference in controllability for either case. These tests continued past the submission of this written report, so the team decided to include the vertical tail in all official drawings, which then left open the possibility of later removing it from the aircraft if appropriate based on further testing. Figure 8.1 shows the aircraft in flight with both vertical surfaces attached and on the ground with the winglets removed.

**Figure 8.1 – Vertical Surface Flight Testing**

### 8.1.2 Structures Performance

The structures tests consisted of the wingtip 2.5g bend test, the flexing of vertical surfaces, and the bending of the landing gear. The team predicted having a very strong and durable aircraft capable of withstanding large forces. This prediction was verified by the successful completion of the 2.5g bend test, in which there was no noticeable deflection across the 5 foot wing span—quite an incredible feat. The composite structure of the aircraft proved immensely strong, as evidenced by the aircraft's ability to sustain little to no damage in the high speed crash depicted in Figure 8.2.

**Figure 8.2 – Prototype Moments Before High Speed Crash**

### 8.1.3 Propulsion Performance

The team focused much of their optimization effort on finding the most powerful and efficient propulsion system available. To determine the static thrust output of the motor, the team ran the motor in a wind tunnel. The motor was connected to a force balance with strain gauges, and then attached to a computer data logger interface that allowed data to be collected directly into MATLAB. Figure 8.3 shows the experimental test set-up in the wind tunnel.

**Figure 8.3 – Propulsion Experimental Test Set-Up**

The propulsion testing stand was calibrated using a system of known masses attached to string and pulleys which modeled the loads the stand would see under power. A linear curve fit was determined from the calibration data which allowed the team to determine the sensitivity from the calibration data. This sensitivity value was then multiplied by the voltage readings during data collection to yield the equivalent thrust for the voltage in units of pounds force. In this way, the thrust of the propulsion system as a function of percentage of throttle stick advancement was determined for different electronic system set-ups. Figure 8.4 shows the results of this testing. Data points for throttle past 70% for the two trials which used the 26 cell battery were extrapolated due to maximum limits of the thrust stand.

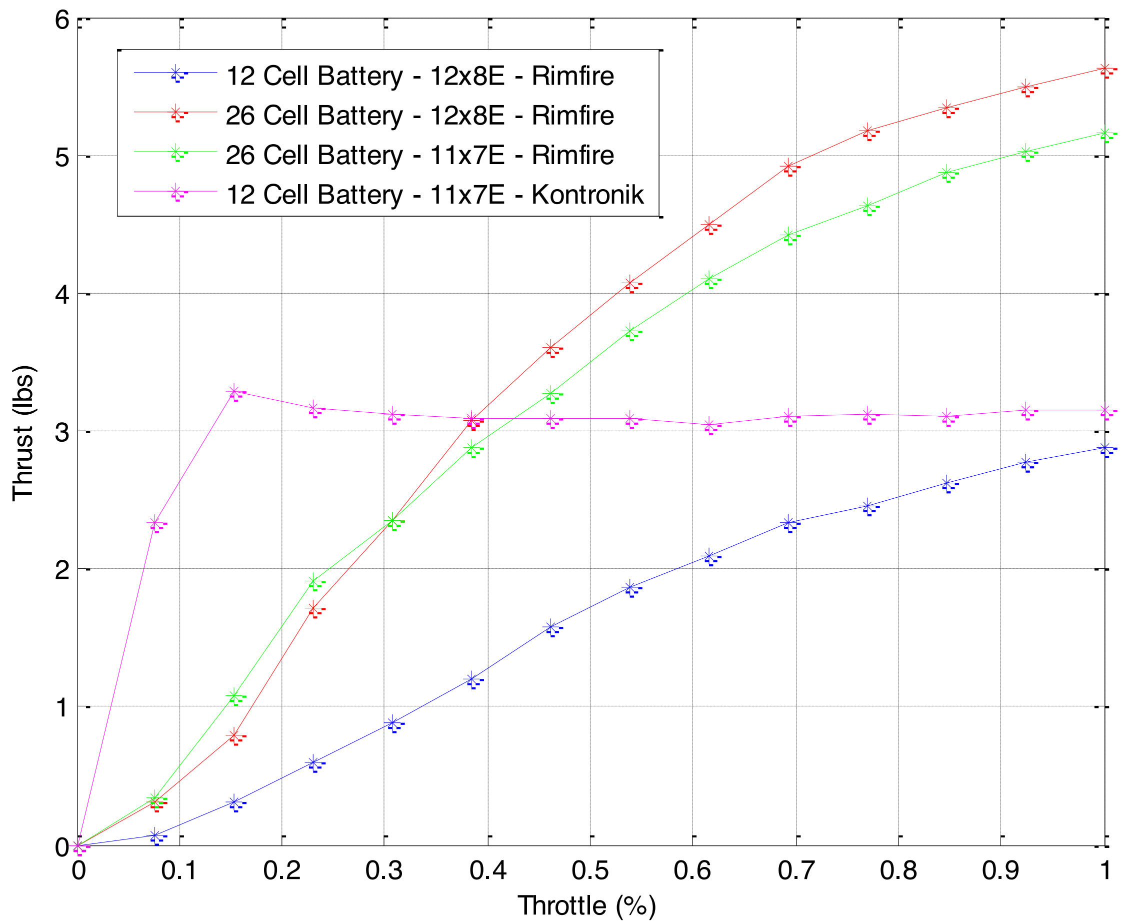



**Figure 8.4 – Propulsion System Test Results**

The blue data was gathered from using the Rimfire .55 480 Kv motor with the Castle Electronics speed controller, 12 cell battery, and a 12 x 8E propeller. The system peaked at 2.9 pounds of thrust and exhibited a steady increase in thrust as the throttle was advanced. Comparing this set-up to the pink data set, which included the team's motor from the previous year, the same battery, and a smaller 11 x 7E propeller yielded that the old motor had a slightly more powerful peak thrust output but was not uniform during throttle advancement. In fact, the thrust reached its peak when the throttle had only been advanced 15%. This oddity was attributed to the speed controller which was paired with the older motor. This speed controller limited the power output to the motor beyond 15% throttle advancement, whereas the new motor and speed controller were smooth in their power advancement. The red and green data sets represent the 12 x 8E and 11 x 7E propeller cases respectively for the new motor, speed controller, and 26 cell battery. The voltage to the motor was increased by a factor of 2.166 due to the higher-voltage 26 cell battery. The maximum thrust output also increased by a factor of roughly 2 from 2.9 pounds to 5.7 pounds. This testing verified the team's projections of the propulsion system's performance, and actually exceeded expectations of 5.0 pounds of thrust by 0.7 pounds.

#### 8.1.4 Payload Accommodation Performance

The water was dropped 500 feet away on the ground to test whether the volumetric flow of the water exiting the aircraft was sufficient to be seen at the furthest point away the aircraft could be from the judges. The test results were affirmative, with all the water exiting the aircraft in an average time of between 2 and 5 seconds. The CAM sensor consistently worked and released the water tank release pin. In comparing altitude data with the onboard GPS data acquisition device it was determined that the CAM consistently released the water at approximately 100 meters above ground. The passenger payload insert fit securely into the center body of the aircraft, ensuring safe transportation of the passenger payload.

### 8.2 Final Aircraft Design

The team was confident that their final aircraft design would be extremely competitive against other aircraft. At the time of this report submission, the team had designed, built, and tested two aircraft iterations, crashed four times of varying severity, had 15 successful flight attempts, and impressed countless campus pedestrians with the aircraft's aerobatic prowess (including loops and axial rolls). Figure 8.5 shows the final aircraft iteration.

**Figure 8.5 – Final Aircraft Design**