Cessna Aircraft Company
Raytheon Missile Systems
AIAA Foundation

The 2010 Cessna/Raytheon Missile Systems Design/Build/Fly Competition Flyoff was held at Cessna Field in Wichita, KS on the weekend of April 16-18, 2010. This was the 14th year the competition was held, and participation continued to increase from past years. A total of 69 teams submitted written reports to be judged. 65 teams attended the flyoff all of which completed the technical inspection and 62 teams made at least one flight attempt. More than 600 students, faculty, and guests were present. Good weather allowed for non-stop flights each day with a total of 179 flights during the weekend. 46 Teams were able to obtain a successful scoring flight. Overall, the teams were better prepared for the competition than ever before, which was reflected in the number and quality of the written reports, teams attending the flyoff, completing tech and flying the missions. A historical perspective of participation is shown below.

The primary design objective for this year was to accommodate random payloads of mixed size softballs and bats. A delivery flight was first required, where the airplane was flown with no payload. As usual, the total score is the product of the flight score and written report score. More details on the mission requirements can be found at the competition website: http://www.ae.uiuc.edu/aiaadbf

First Place was Oklahoma State University Team Orange, Second Place was Oklahoma State University Team Black and Third Place was Purdue University B'Euler Up. The top three teams had a very close competition. Oklahoma State University (OSU) Orange was ahead after all three teams completed the third mission. They then re-flew Mission 2 and dropped their ball loading time from 16 seconds down to 12 seconds to stretch their lead. Purdue and OSU Black then retried their mission 2 but unfortunately both had problems. A full listing of the results is included below

We owe our thanks for the success of the DBF competition to the efforts of many volunteers from Cessna Aircraft, Raytheon Missile Systems, and the AIAA sponsoring technical committees: Applied Aerodynamics, Aircraft Design, Flight Test, and Design Engineering. These volunteers collectively set the rules for the contest, publicize the event, gather entries, judge the written reports, and organize the flyoff. Thanks also go to the Corporate Sponsors: Raytheon Missile Systems and Cessna Aircraft Company, and also to the AIAA Foundation for their financial support. Special thanks go to Cessna Aircraft for hosting the flyoff this year.

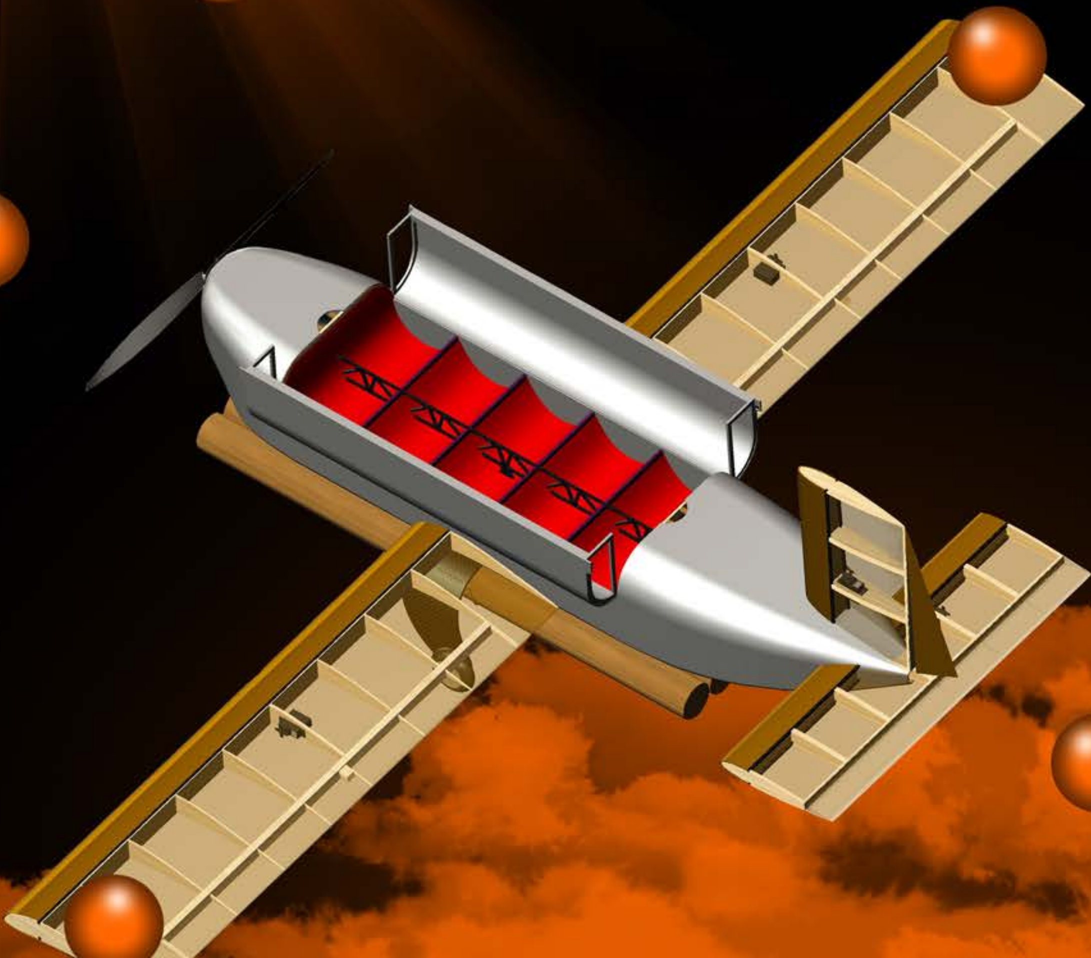
Finally, this event would not be nearly as successful without the hard work and enthusiasm from all the students and advisors. If it weren't for you, we wouldn't keep doing it.

David Levy and Tom Zickuhr
For the DBF Governing Committee

Oklahoma State University

ORANGE Team

"Orange Seduction"



Design/Build/Fly 2010

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1.0 EXECUTIVE SUMMARY

The challenge presented to the team for the 2009-2010 contest consisted of three missions. The first, a ferry mission, required the aircraft to fly two laps around the competition course, with the score determined completion time and overall weight of the aircraft system, including the case. The second mission consisted of a timed loading of a random mix of 11" and 12" softballs into the interior of the aircraft, followed by three untimed circuits of the course. Finally, the third mission consisted of three timed laps being flown while the aircraft carried up to five ―bats" on the exterior of the aircraft. To address these challenges, the team designed an aircraft capable of balancing the requirements posed by each mission, primarily high speed and maneuverability to complete the first and third missions, short loading time to accommodate the second, and overall low system weight for all missions.

To improve the aircraft's performance in the first mission, great effort was made to minimize the turning radius of the vehicle, in order to reduce the distance the plane had to fly while completing the timed laps. Also, the structure of the vehicle was kept to an absolute minimum to reduce weight. This was accomplished not only through the use of lightweight materials, but also by designing existing structural elements to perform multiple roles. For example, the central bulkhead of the interior payload bay, while acting as part of the softball loading system, was also built to provide the majority of the bending strength to the vehicle. It was integrally connected to the wing mounting, landing gear, and lateral bulkhead systems of the internal structure.

Because the key to success in the second mission was having the shortest possible loading time for the internal payload, the team designed a robust system capable of quickly and efficiently storing the softballs inside the aircraft with a minimum of excess structure. The system consisted of dorsal clamshell doors which opened to expose a two-by-five ball storage grid. The doors acted as a funnel, helping to guide the balls into their final positions. The grid walls acted as bulkheads, providing structural strength as well as securing the payload, again demonstrating the effort taken to make the internal structure multipurpose. The grid was also draped with ripstop sewn to create a hammock for the internal payload, which in turn provided space below for the batteries. Because the entirety of the loading system was fixed in the aircraft, no additional mechanisms were needed in the case. This primarily reduced loading time, eliminating unnecessary operations, but it also reduced the structure and thus weight of the case itself. All of these elements combined to yield an expected total loading time of less than eight seconds.

The features designed into the aircraft for the first mission were also used to boost the score from the third mission, which again demanded that the aircraft be able to quickly complete three laps around the course. The main exception was that for the final mission, an external payload of bats had to be carried. After conducting a score sensitivity analysis, the team designed the plane to carry five bats in order to maximize the mission three flight score. These bats were arranged side by side in a single row under the belly of the fuselage, with their centers of gravity (cgs) located at the cg of the aircraft. The bats were secured to the plane fore and aft by straps connected to the payload bay bulkheads; this system was found to be the lightest and simplest possible, once again taking advantage of existing structural elements to minimize system weight.

The combined features of the airplane allowed it to perform well for every mission. In mission 1, the airplane was able to cruise and maneuver at high speeds, thereby decreasing lap time and directly increasing score. For mission 2, the aircraft was able to be loaded in 8 seconds, again improving the score. Because of the way the bats were loaded for mission 3, the aircraft was still able to handle well at high velocities, which once again increased the score. In conclusion, the choices made for the aircraft's configuration and the design of its components caused the aircraft to stay competitive for all missions.

2.0 MANAGEMENT SUMMARY

2.1 Project Management

The personnel of Orange Team were separated into four divisions: Aerodynamics, Propulsion, Structures, and Computer Aided Drafting (CAD). Each group was managed by a Team Lead, and the activities of the team as a whole were managed by the Chief Engineer. A breakdown of the members of each group is shown below.

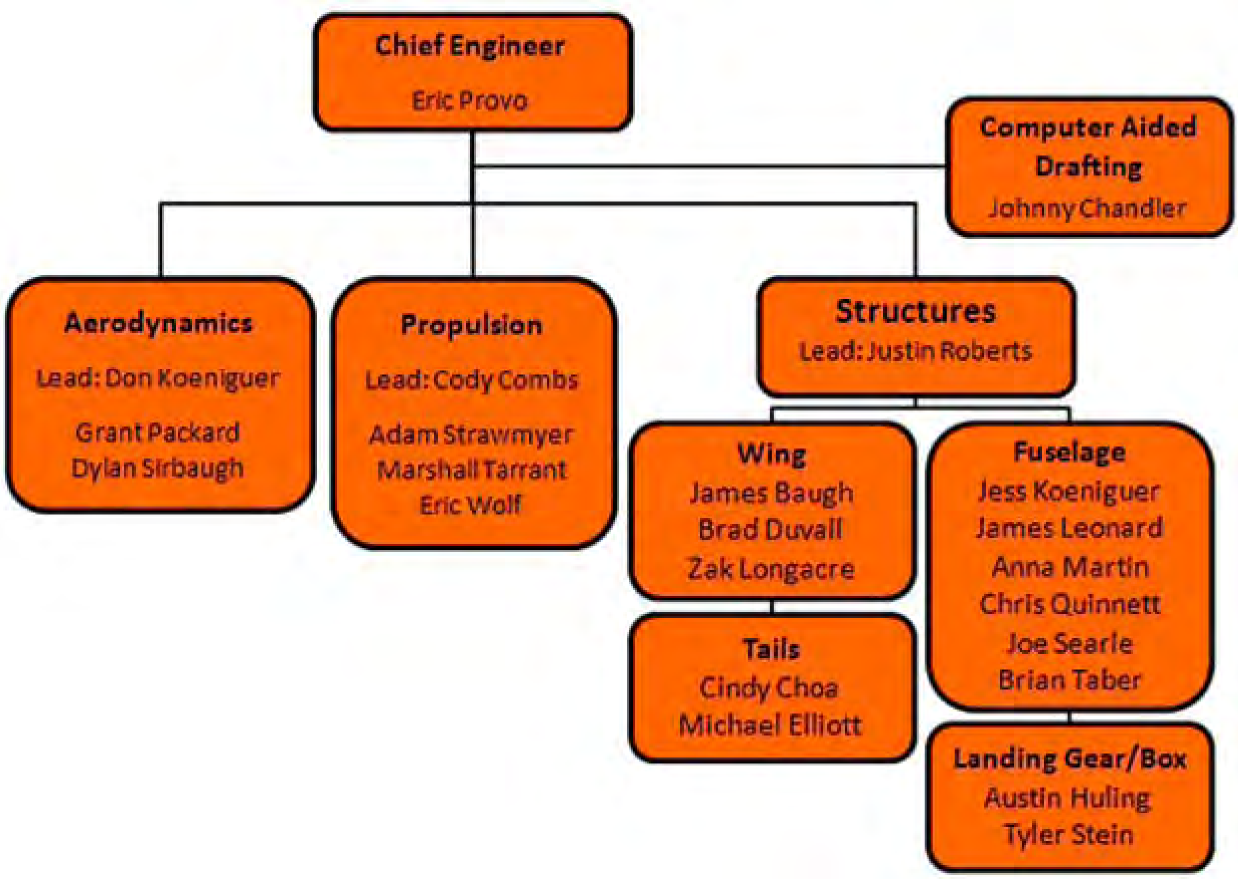


Figure 1: Orange Team Personnel Flow Chart

Aerodynamics was responsible for the overall external configuration of the aircraft, including wing design, tail design, and airfoil selection. They performed in-depth analysis and optimization on the aircraft geometry, determining its performance, stability, and control characteristics. Propulsion used the results from Aerodynamics to determine the motor placement, propeller size, and necessary power for the aircraft. Afterwards, they performed their own optimization and tests to find the best model of motor and electrical components for the aircraft. Structures designed the overall configuration and basic components first, concentrating on a few important areas and using the data from Aerodynamics and Propulsion. They then focused more on the details of their designs, testing, and optimizing. Finally, the teams worked together to produce the best aircraft for the missions. Tests of the final design were done, and a few small changes were made. Throughout the entire process, the components were drawn up by the CAD lead, and the Chief Engineer made sure the sub-teams communicated with each other and the schedule was followed.

2.2 Milestone Chart

In order to ensure that the team stayed on schedule throughout the semester, a Gantt chart detailing all of the major project activities was created. This schedule reflects several important milestones, primarily a series of five progress reports marking the end of the Conceptual and Preliminary/Final design phases and Rollout of both the prototype and final competition aircraft.

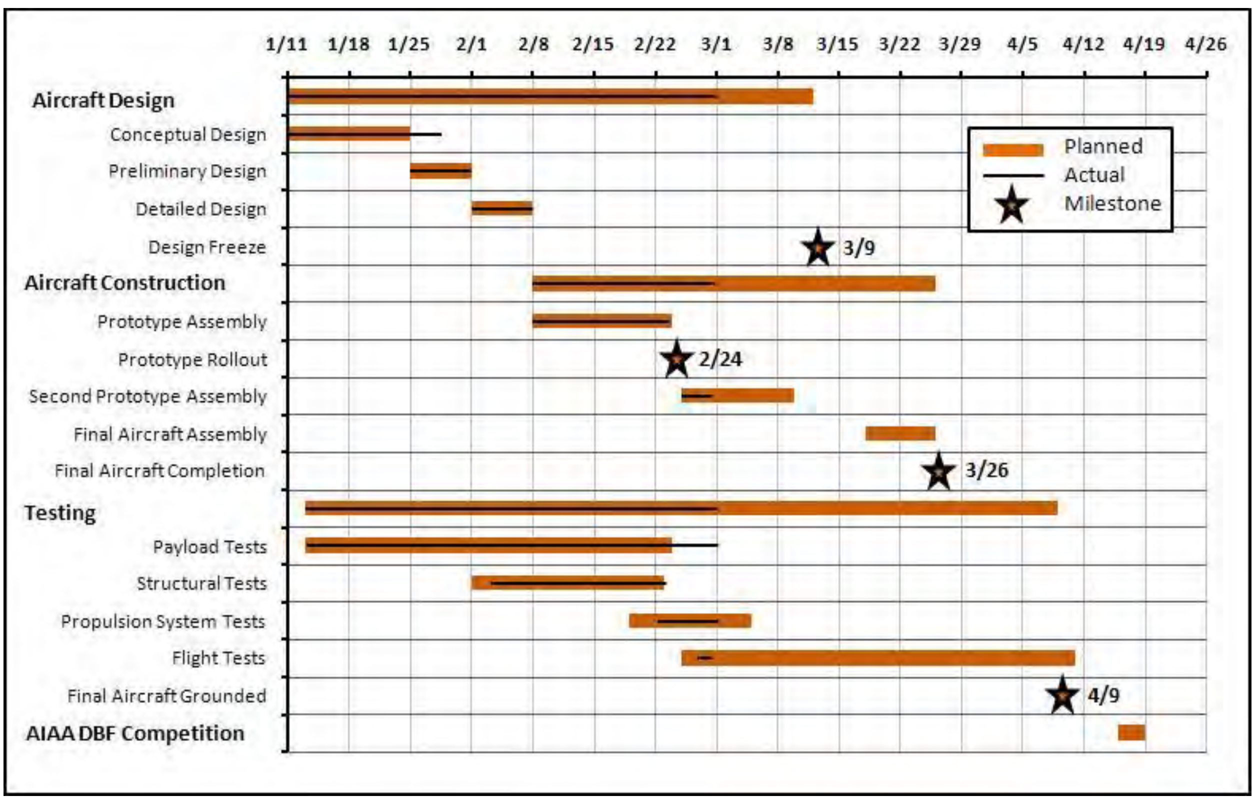


Figure 2: Project Gantt Chart

3.0 CONCEPTUAL DESIGN

Determining mission requirements and design constraints was the starting point of the conceptual design phase, which provided the missions' goals and limits to carry out a sensitivity analysis. Configurations were then selected from figures of merit (FOMs) that emerged from the sensitivity analysis.

3.1 Mission Requirements

The AIAA Design/Build/Fly Competition for 2010 consisted of three missions and the design report. Mission 1 was scored out of 50, while mission 2, mission 3, and the design report were each scored out of 100. The equation for a team's total score is:

Total Score = Report Score * (Mission 1 Score + Mission 2 Score + Mission 3 Score)

This equation, the general rules, and each individual mission's scoring equation were carefully studied in order to find and properly weigh FOMs to carry out an effective sensitivity analysis.

3.1.1 General Mission Requirements and Concerns

- All of the hardware for a team's flight (aircraft, tools, transmitter, etc.) must be able to fit into a case no larger than 2'x2'x4'.
- The case may not use Velcro, tape, or magnets in any way and may not be torn or significantly damaged or the team forfeits that flight attempt.
- Individual battery packs may not weigh more than 4 pounds.
- Payloads must be properly secured on aircraft to prevent cg drift in flight.
- For a legal takeoff, the plane must take off the ground and stay off the ground in 100 feet or less
- The plane must successfully land for the team to receive a score for the flight attempt.
- The heaviest weight from all of a team's successful flight attempts for **any** mission will be the one used for scores with weight as a factor.

3.1.2 Ground Crew/Assembly Crew

After entering the staging area, the team assembles and flight checks their aircraft prior to being called to the flight line. The assembly and checkout must be completed in less than 5 minutes.

Mission 1 does not require any extra assembly time beyond the initial 5 minutes.

For mission 2, the ground crew starts out in the bullpen with the case next to the team. The plane is fully assembled and set at the start line of the runway. When the time starts, member A runs to the plane and open the doors. Member B opens the case while member C grabs the bag of balls, then stays with the case to make sure it does not blow away. Member C runs to the plane and dumps the balls in, then runs back to the case and puts the empty bag inside so that member B can close the case. Member A sorts the balls in the payload bay, then closes the plane and runs back to the bullpen. The time will end when the team has finished loading the aircraft, closed the case, returned to the designated loading crew area, and calls ―Stop".

For mission 3, the loading of the bats is not timed, but the team assumed it was included in the allotted 5 minute assembly time. After the wings are attached to the aircraft, the bats are placed in the pegs attached to the underside of the plane and are fastened in place with a strap. The plane is then placed at the start line of the runway.

3.1.3 Mission 1: Ferry Flight

Mission 1 consists of a two lap ferry flight around the course. The flight score is given by the equation: $M_1=T_1*W_1*50$. The variable W_1 is the lightest fully loaded case weight of any team that completed the mission divided by the heaviest recorded weight of the completed flight attempts by the team in question. The variable T_1 is the best time any team had for mission 1 divided by the best time of the team in question. Flight time starts when the plane advances the throttle and ends when the plane flies over the finish line to complete the second lap.

3.1.4 Mission 2: Softball Flight

Mission 2 begins with a random assignment of six to ten softballs with mixed diameters of 11" and 12". This is followed by a timed loading of the softballs into the aircraft from the case and then a successful 3 lap flight with the softball payload. The score for this mission is given by: $M_2=T_2*W_2*100$. The variable W_2 is the lightest fully loaded case weight for any team that completed the mission divided by the heaviest recorded weight for any of the completed flight attempts of the team in question. The variable T_2 is the best time any team had for loading the softballs divided by the best time of the team in question. Time will start with the team in the loading area, the case closed with the softball bag inside, and the aircraft on the runway. Time ends when the team calls ―Stop" after the case is shut with the ball bag inside, and the case, team members, and aircraft are all back in their respective starting positions. The 3 lap flight following the softball loading must be successful in order to receive a score.

3.1.5 Mission 3: Bat Flight

Mission 3 consists of a timed 3 lap flight with a payload of bats, the number of which is chosen by the team. The team may choose 1 to 5 bats for a flight attempt. The score for this mission is given by $M_3=T_3*B*100$. The variable B is the number of bats successfully carried by the team in question divided by the most bats any team successfully carried. The variable T_3 is the fastest flight time of any team's successful mission 3 attempt divided by the time of the team in question. The time starts at the beginning of takeoff and ends when he plane flies over the finish line to complete the third lap.

3.2 Design Requirements Definition

The following are the requirements for the different missions. These requirements were based on what most affected the score.

- **Mission 1: Ferry Flight –** Minimize aircraft weight to improve ferry time.
- **Mission 2: Softball Flight** – Minimize loading time for the softballs while properly securing the balls in a lightweight grid system
- **Mission 3**: **Bat Flight** – Design a way to safely carry the bats without adding too much weight.
- **Case Design** – create a sturdy, lightweight case to carry and protect the aircraft, as well as hold softballs for mission 2.

Sensitivity Analysis

By using the mission score equations, a sensitivity analysis was created using Excel. By creating some initial benchmarks based on OSU airplanes that have performed similar missions, the team was able to determine how much a particular scoring parameter affected the score for that mission. The team assumed a best weight for mission 1 and 3 to be 10 pounds, with the best total time in mission 1 to be 60 seconds. A loading time of 4 seconds was used as the best score and 5 bats were carried to determine the score for mission 3. The topographical charts below show the final score a team would obtain based on weight and time or number of bats carried.

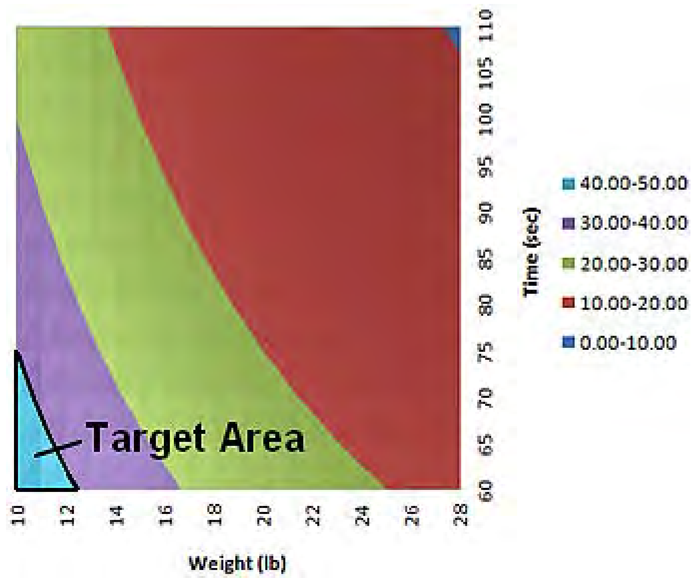


Figure 3: Mission 1 Score Analysis

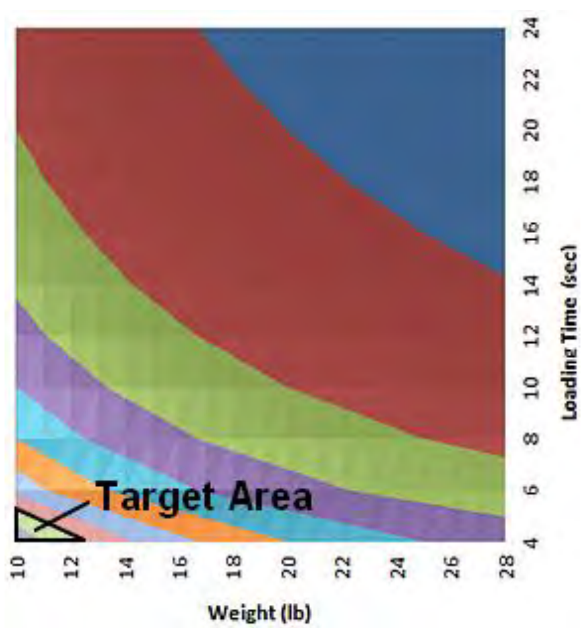


Figure 4: Mission 2 Score Analysis

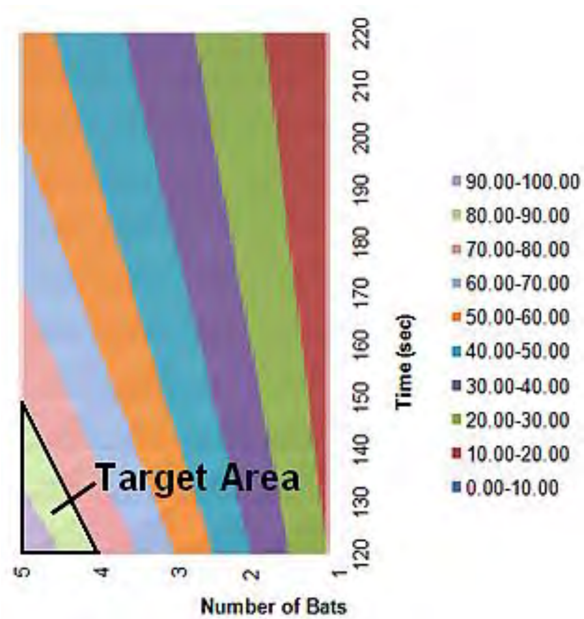


Figure 5: Mission 3 Score Analysis

From the mission 1 analysis, to be within the top 20% of the scores, the plane needed to be less than 12 pounds and fly faster than 75 seconds total. For mission 2, loading time needed to be less than 6 seconds with the same weight. For mission 3, the plane needed to carry the full amount of bats to be competitive and fly within 150 seconds to score higher than 80 points.

The score functions for these missions produced three dimensional graphs, but a two dimensional version was used to illustrate what impacted the team‘s score the most, which was overall weight and mission 2‘s loading time. This is seen in how the target area of mission 2 is smaller than the target areas of missions 1 and 3. The team also determined, through the analysis, that the score was far more sensitive to weight than flight time. Shedding a pound of weight was roughly equivalent to allowing the team to fly 20-25 seconds slower for a given score. This revelation drove Aerodynamics to insist that the group apply all available measures to reduce the weight of the entire system in addition to pursuing design concepts that would minimize weight even at the cost of some flight speed. As a result the team designed a loading system that would not only minimize mission 2‘s loading time, but be extremely lightweight.

3.3 Solutions, Configurations, and Results

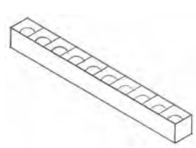
3.3.1 Fuselage

Internal Payload Bay

Due to the sensitivity analysis that identified weight and mission 2‘s loading time as critical score defining attributes, it was decided the aircraft would be built around the payload design. This led the team to test a number of payload designs by building cardboard mock-ups to determine the best system.

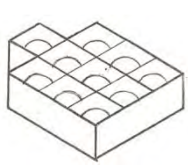
The design of the interior was driven by mission 2, in which six to ten balls had to be held in a grid pattern after being speed loaded. The first decision made was to have the payload apparatus stay inside the fuselage instead of being removable because the rules stated that all parts of the plane must be in the plane when the timing for mission 2 began, negating any speed benefits from the removable designs.

Wanting to maximize the score, the team determined the loading system needed to secure the payload as quickly as possible. Several loading systems were discussed, investigated, and optimized.



Single Stack — Placing the balls in one line along the fuselage in order to keep a ‗grid” pattern

- This would make the balls harder to load, so it was removed from the design option list.



3x3+1 Grid – 3x3 grid with an extra compartment in the middle in case ten balls were assigned

- This made the fuselage too wide and bulky, thus heavier, so it was also removed.

Although the team could remove some design concepts through conversation, other could only be scrutinized with actual tests and real life data. Thus, mock-ups were created and timed trials were conducted. Table 1 shows the times for each test using ten balls dumped in the same manner they would be in mission 2. Each design was tested four times with the same three people loading for consistency.

Trial Run	 **2x5 Grid**	 **Fabric**
1	3.06	4.47
2	3.38	3.28
3	4.19	4.75
4	3.84	3.94
Average Time (s)	3.62	4.11

Table 1: Timed Trials of Payload Housing Systems

As the times show, the 2x5 grid performed faster than the suspended fabric. Also, the fabric design would have called for a sturdier door system since the elastic fabric would need to stretch from the doors to the spine to adequately wrap around the balls. The fabric would have also blocked the bottom of the fuselage from access without some sort of complicated (and heavy) latching system, preventing the team from getting to where the batteries were to be placed. It was also discovered that the grid system provided a great deal of internal support for the fuselage, allowing the outer skin to be lighter and weaker, as the spine going down the middle of the payload bay would support the grid ribs and give extra strength to the fuselage itself.

Once the grid system was decided on, the shape of the spine became a consideration because the shape could affect sorting time. Four spines were tested using fuselage bays mocked up with cardboard: trough, concave, slant top, and flat top. As done for the payload housing systems, each was tested four times with the same three people.

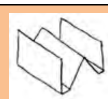
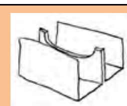
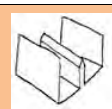
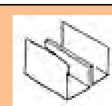
Trial Run	 Trough	 Concave	 Slant Top	 Flat Top
1	4.16	3.50	4.03	2
2	4.29	3.75	4.94	2.79
3	3.87	3.50	4.7	2.5
4	3.5	2.44	3.59	2
Average	3.96	3.30	4.32	2.3225

Table 2: Timed Trials of Spine Designs

Although the flat top clearly beat all the other designs, it was believed, at the time, that the concave design was the best choice overall, as the raised portions of the spine would provide a place for the doors' fasteners to be secured. The team later changed this decision when the mechanisms for the doors were moved to the other side of the bulkheads.

Since the ribs integrated with the spine, they would also brace the spine against the outer skin and make the plane more structurally stable. At the ends of the grid system, bulkheads would be fixed in order to transfer bending loads to the fuselage and keep the end softballs from rolling around, as shown in Figure 6. After more testing, the whole system could be hollowed out to save weight.

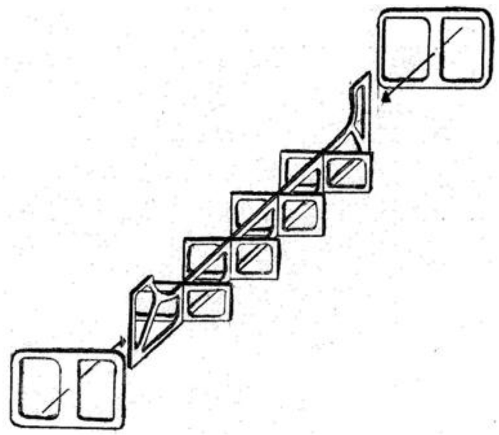


Figure 6: Spine Assembly Sketch

The wings' main spars would transfer lift loads into the center spine via a key cut into the spine. The main landing gear would also transfer the load it experienced to the spine through the bottom of the fuselage. This would allow the spine to act as the central load bearing member for the aircraft and would serve the purpose of both strengthening the airframe as well as allowing other unloaded parts of the aircraft, such as the doors, to be weakened.

The payload floor was the next major design point of the fuselage. The group decided to design a false floor, because not only did the floor need to support the balls, but it also needed to provide room for batteries and possibly the wing spar. (The wings were being designed at this time as well, and they are covered in section 3.3.5.) The following types of floor were investigated.

- **Suspended Mesh** – String or wire suspended under the ribs that could be reached through to access components underneath
- **Hammock** – Fabric placed over the grid that could be easily removed to adjust components under the floor
- **Double Divots –** Individual spaces for each ball built into the floor that could hold the balls and keep space open underneath where components could be stored
- **Bubble Wrap –** Wrapping components in bubble wrap in place of a floor so the softballs do not damage the components
- **Foam Cubes –** Placed on the actual floor of the fuselage to raise the balls above the component height (used as the baseline)

The conceptual designs for the payload floor were compared using the decision matrix shown in Table 3. This compared each design option based on their weight, structural support, and the access each design allowed for aircraft components around or underneath the softballs. Based on this, the hammock was chosen.

FOMs	Suspended Wire	Hammock	Double Divots	Bubble Wrap	Foam Cubes
Weight	0	0	-1	0	0
Structure	-1	1	0	0	0
Access	1	1	-1	0	0
Total	0	2	-1	0	0

Table 3: Payload Floor Decision Matrix

Exterior

Since the time taken to load the softballs into the aircraft directly affected the score of mission 2, a great deal of emphasis was placed on reducing loading time. Due to ease, speed, and structural design, a top loading system with clamshell doors was selected, as the curved double doors acted as a funnel for the payload.

A number of different latching methods were considered, and most were eliminated for weight, complexity, time to manipulate, or reliability. After elimination, three ideas remained, as described below.

- **Kitchen Latch –** A simple cabinet door latch that secures itself with springs
- **Loop-Pin –** A piece on each door comes together and is secured with a pin through the loops
- **C-Clasp –** A dowel piece that has a c-shaped clasp fit snugly to it

As Table 4 shows, the C-clasp idea was chosen because of its simplicity and weight.

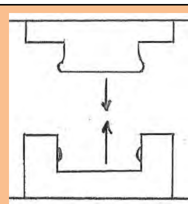
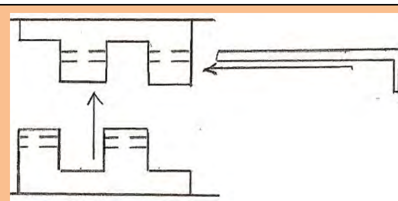
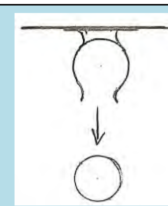
	 Kitchen Latch	 Loop-Pin	 C-Clasp
Speed	0	-1	1
Simplicity	0	-1	1
Weight	0	-1	0
Total	0	-3	2

Table 4: Payload Door Retention Decision Matrix

3.3.2 Bat Configuration and Retention

Configuration

The placement of the bats on the aircraft played a very important part in the aircraft design and configuration, so many options were examined. A list of conceptual configurations of the bats and a short description of each are shown in Table 5.

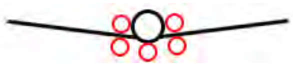
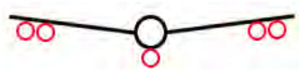
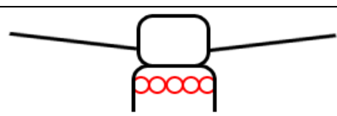
	1) 2 bats, low-mounted under inner portion of each wing, 1 bat underneath midline of fuselage
	2) 1 bat low-mounted under inner portion of each wing 1 bat high-mounted over inner portion of each wing 1 bat underneath midline of fuselage
	3) 5 bats around the circumference of fuselage
	4) 1 bat low-mounted under outer portion of each wing 1 bat low-mounted under inner portion of each wing 1 bat underneath midline of fuselage
	5) 2 bats low-mounted under outer portion of each wing 1 bat underneath midline of fuselage
	6) 2 bats high-mounted over inner portion of each wing 1 bat underneath midline of fuselage
	7) 5 bats mounted between landing gear

Table 5: Bat Configurations

It was decided the best configuration for mission 3 would be number seven, with the five bats mounted between the landing gear under the span of the fuselage. This was a simple concept that did not interfere with other subsystems with the exception of the landing gear, and it used the least amount of extra material to secure the bats. This was accomplished by using structure that already had to be strong rather than having to strengthen another part of the aircraft for just one mission, thus saving weight.

Another key element in mission 3 was making sure the bats were secured to the aircraft. Retention concepts were developed, keeping in mind that connection points would be made on or near the forward and aft bulkhead.

A list of conceptual retention systems can be seen in Table 6 with a short description of each.

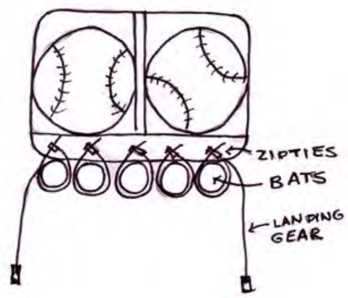
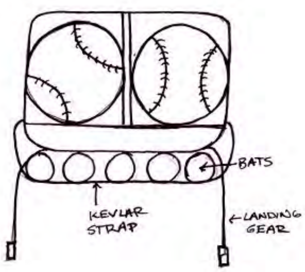
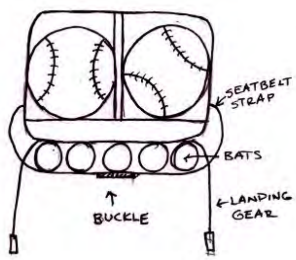
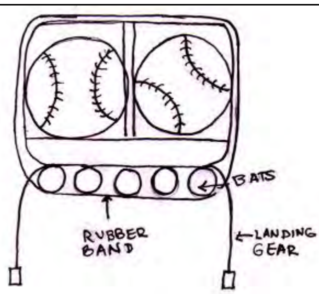
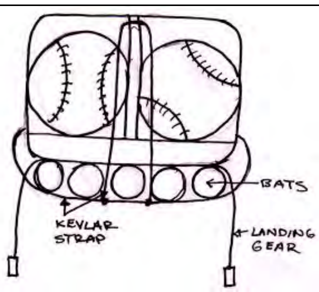
	1) Zip ties secure each individual bat Requires 10 zip ties, 5 for forward bulkhead and 5 for aft bulkhead Requires 20 additional holes to the bottom of the fuselage to secure zip ties
	2) 2 Kevlar straps secure all 5 bats One end is fixed to one side of fuselage; other is fastened by a hook on the other side Kevlar straps mount to each bulkhead
	3) 2 seatbelt straps secure all 5 bats Both ends fix to each side of the fuselage A buckle fastens the straps together and tightens
	4) 2 industrial strength rubber bands secure all 5 bats Rubber bands maneuver around the empennage and propeller of aircraft
	5) 2 Kevlar straps and 2 rubber bands/strings secure all 5 bats Kevlar straps fix to the fuselage on one end and the other end secures to a hook Rubber bands/strings eliminate any slack

Table 6: Bat Retention Configurations

It was decided configuration 5 would be the best option in order to stay within the rules for mission 3. The Kevlar strap would support the load of the 5 bats, with the addition of the string, increasing reliability.

3.3.3 Wing

Wing Planform and Size

Design of the wing was driven largely by two factors, the first being the efficient generation of lift per unit weight and span, and the second being the constraint of the case. Several different planform shapes were considered during conceptual design and a typical straight-wing design was chosen due to weight, ease of manufacturing, and aileron necessity.

In order to fit in the case, the wings either had to have a 4 ft span or come apart. The team ran several calculations and found that a 4 ft span was not long enough to efficiently and suitably run the missions. Thus, the wing was designed to not only be detached from the fuselage, but also break into two pieces. The following section discusses how the wing connections were designed.

Wing Mounting

In order to design the best mounting system for the competition's requirements, five different mounting systems were developed and reviewed by the team.

- **Friction Fit** – Uses a male and female end to connect one wing to each side of the fuselage
- **Two L Channel with Pins** – The main spar of each wing forms an L, and is pinned together inside the fuselage
- **C Channel with Block** – The wing spars meet at the spine of the aircraft where a block is mounted that fits snugly in the C channel (used as the baseline)
- **Two Flats with Pins** – A single spar for each wing passes through the spine and overlaps the other wing's spar, then are pinned together
- **Double C Channel** – The main spar for each wing is a C channel that fits into a larger C channel that is fixed inside the aircraft

As can be seen in Table 7, the friction fit was chosen as the best fit for the FOMs.

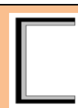
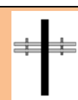
FOMs	 Double C Channel	 C Channel with Block	 Two Flats with Pins	 Two L Channels with Pins	 Friction Fit
Weight	1	0	1	0	1
Strength	-1	0	0	0	1
Manufacturability	-1	0	1	-1	-1
Assembly Speed	0	0	-1	-1	1
Total	-1	0	1	-2	2

Table 7: Wing Mounting System Decision Matrix

3.3.4 Propulsion Systems

Motor Configuration

Several different motor configurations were explored and investigated for the aircraft, and a simple decision matrix was used to evaluate the different options. Each of the various arrangements is listed below.

- **Single Tractor** – A single propeller on the nose of the plane (used as the baseline)
- **Single Pusher** – A single propeller on the rear of the plane
- **Double Tractor** – Two propellers on the wings ―pulling” the plane
- **Tractor and Pusher** – ―Push-pull” method, combines the single tractor and pusher

Propulsion came to the conclusion that the conventional single tractor propeller was the best choice for designing an aircraft to optimally perform the given mission requirements. It was the simplest and lightest configuration with none of the drawbacks the others had.

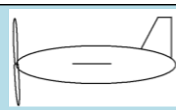
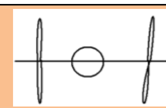
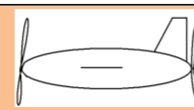
FOMs	 Single Tractor	 Single Pusher	 Double Tractor	 Tractor and Pusher
Weight	0	0	-1	-1
Takeoff Performance	0	-1	1	0
Stability	0	0	-1	0
Drag	0	0	-1	0
Complexity	0	-1	-1	-1
Total	0	-2	-3	-2

Table 8: Motor Configuration Decision Matrix

Cooling

Two methods of cooling the motor, batteries, and radio gear were considered for the aircraft: unpowered cooling and powered cooling.

- Powered cooling would consist of fans and conduits that actively push air across the internal components to promote increased cooling by convection.
- Unpowered cooling would use apertures in the fuselage to cool the components by simple ―free” convection.

Unpowered cooling was selected due to the weight savings from having no extra cooling components. In addition, Structures intended to use simple openings in the fuselage in order to open the payload bay doors. These openings could double as cooling ports, increasing the aircraft‘s structural efficiency.

3.3.5 Airplane Configuration

The missions drove the airplane configuration from the very beginning. As a group, the requirements that garnered the most points were discussed. It was decided that weight, payload compliance, structural ease, and stability were the most important FOMs for the aircraft‘s overall configuration.

In the conceptual phase of the project, every plane option that was not rotary wing or lighter than air was considered. From there, the team narrowed the choices down to four basic configurations, which are shown in Table 9.

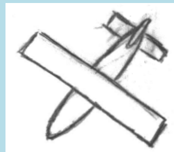
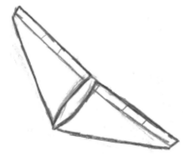
			
Biplane	**Conventional**	**Flying Wing**	**Canard**

Table 9: Configuration Options

Due to the design of the payload system, the biplane was removed from the options, as the double wing design conflicted with the payload doors. The biplane would also mean more weight, which would hurt the team's overall score. The flying wing was determined unsuitable because positioning six to ten balls within the fuselage and placing five bats on the underside was unrealistic. The canard design was then examined and compared to the conventional design. The team collectively decided the building complexities and the increased takeoff distance required by placing a canard was not worth the lifting surface in front of the wings. In the end, the conventional design was selected.

Due to the cargo bay doors and the payload, the only way to connect the wings was through or underneath the cargo bay. Connecting the wings through the cargo bay added extra complexity and took away cargo space needed for both the softballs and the batteries. So a false floor (discussed in 3.3.1) was added to the fuselage which would contain the wing box and connect the low wing.

The next decision on the external configuration of the airplane was the placement of the propeller. There were many options after the tractor propeller configuration was decided. The first priority was the ground clearance the propeller would need. As a standard, the team left one inch between the prop and the ground. There were two ways to keep the ground clearance above the one inch mark. The team could add length to the landing gear or could mount the propeller high on the airplane. Lengthening the landing gear added more weight than shifting the propeller higher, so moving the propeller to the highest part of the fuselage was the logical choice.

Structural Weight Goal

Since a large portion of the flight score was based on the aircraft's weight, it was essential that the plane be designed and made as light as possible while still maintaining structural integrity during optimum flight. Looking at the designs of the past and this year's requirements, the team decided to strive to construct a plane as light as — preferably lighter than — the plane built by the 2008 OSU Black Team. This was a reasonable and reachable goal, as this year's payload weight and volume was nearly half that of 2008's competition. Therefore, with a maximum payload weight of 6.25 lbs (5 bats), the team expected the aircraft to have an approximate empty weight of 3.5 lbs and a GTOW around 11 lbs. With such a light airplane, it was important that the designed cg be the same for all three missions, which resulted in the payload system previously discussed. The plane was designed to have a constant cg within the wing's chord in order to avoid having the cg longitudinally shift for different missions.

3.3.6 Landing Gear

The first and most important decision facing the team in terms of the landing gear was selecting the configuration. Whichever configuration was chosen needed to be light, strong, and easy to handle. The team considered four configurations: taildragger, tricycle, bicycle, and quadricycle.

- **Taildragger –** Main two-wheeled gear with one smaller wheel at the tail
- **Tricycle –** Main two wheeled gear with a smaller gear at the nose (used as baseline)
- **Bicycle –** Two main single gears in a line with smaller gear on each wing
- **Quadricycle –** Two sets of main two wheeled gear

As can be seen in Table 10, the tricycle and quadricycle tied for first place, but due to weight and ground handling, it was determined that tricycle was the best.

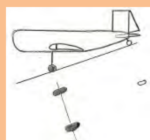
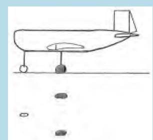
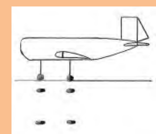
FOMs	 Taildragger	 Tricycle	 Bicycle	 Quadricycle
Ground Handling	-1	0	-1	-1
Strength	0	0	-1	1
Weight	0	0	1	-1
Mission Compliance	0	0	-1	1
Total	-1	0	-2	0

Table 10: Landing Gear Decision Matrix

3.3.7 Tail and Stability

For more takeoff and landing control, the team wanted the tail to be placed in the prop wash. There were several combinations that could achieve this result, but the team also knew tail effectiveness, ground clearance of the tail, and structure would drive the choice.

After these priorities were decided upon, the next step was to decide which tail type to use. Many options were discussed initially, but the team decided the most competitive options were a conventional tail, a T-tail, a cruciform tail, an H-tail, and a V-tail, which are shown in Table 11. The team used the conventional tail design as a benchmark for the other options.

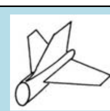
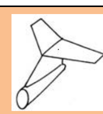
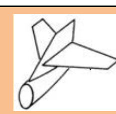
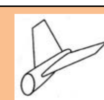
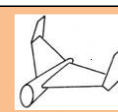
	 Conventional	 T-Tail	 Cruciform	 V-Tail	 H-Tail
Structure	0	-1	-1	-1	-1
Effectiveness	0	-1	-1	-1	0
Ground Clearance	0	0	-1	-1	0
Weight	0	-1	0	1	-1
Total	0	-3	-3	-2	-2

Table 11: Tail Decision Matrix

The final configuration chosen was a conventional tail. The next stage was to decide the placement of the tail. Two requirements drove this decision. The first requirement was to have the tail as far away from the cg of the airplane as possible. The second requirement was for the horizontal stabilizer to be within the prop wash, yet outside of the wing vortex.

3.3.8 Case

This year's contest called for a case to hold the disassembled airplane pre-flight and, for mission 2, hold the softballs. In order to maximize score, the case needed to be as small and light as possible, yet still give easy and quick access to the softballs. The initial configuration looked at was to make a completely rigid case that conformed to the shape of the airplane. This case would have a back door which could fall open quickly and allow for the removal of the softballs from the floor of the case.

After some consideration, however, the team decided that by making the main landing gear of the plane removable, the plane could be tilted in the case, making the case smaller. The team also decided to make a rigid frame covered in fabric, instead of one with rigid walls, to further reduce weight.

3.4 Final Conceptual System Selection

After the above in-depth selection process, team decided the overall configuration of the aircraft's design would have the following:

- Interior payload held in a grid pattern with a low spine and draped with a hammock to minimize loading time and provide space beneath payload
- Straight, detachable wings for easy and secure assembly
- A conventional airplane configuration to best hold the payload bay
- Tricycle landing gear configuration which could hold the bats for mission 3
- Case made from a frame draped with fabric to reduce weight

Team Orange selected this configuration to maximize the aircraft's possible flight score. Figure 7 shows the conceptual design solution in the assembled configuration. Figure 8 shows the aircraft disassembled and within its case.

Figure 7: Conceptual Configuration

Figure 8: Final Conceptual Design within Case

4.0 PRELIMINARY DESIGN

Preliminary design covers propulsion, aerodynamics, and structures. Each section covers several different aspects of the design, and discusses models that were created in order to show how the team optimized the designs proposed during the conceptual design phase.

4.1 Design and Analysis Methodology

Keeping in mind that the weight of the system and the loading time were what most affected score, each part of the airplane was analyzed by its own method. These methods are described in general below, then expanded upon in the following sections.

Airfoil – Using a multivariable analysis, comparisons were made to find the best type of airfoil for the missions, then further comparisons were made for that type.

Tail – The tail was sized using stability and control analyses, while keeping the overall size capable of fitting in the case.

Motor – Motors were compared using manufacturers' data in order to find the lightest motor that could give the required thrust.

Propeller – Propellers were tested in order to find the correct pitch and diameter in order to take off in less than 100 feet.

Batteries – Batteries were surveyed and tested in order to find the highest capacity per unit weight.

Payload Bay – Different grid types were built and tested in order to find the highest strength per unit weight, allowing the balls to be dropped in without damage to the system.

Wing Mounting – The spar joining system was designed by comparing possible weights to strengths, then building and testing.

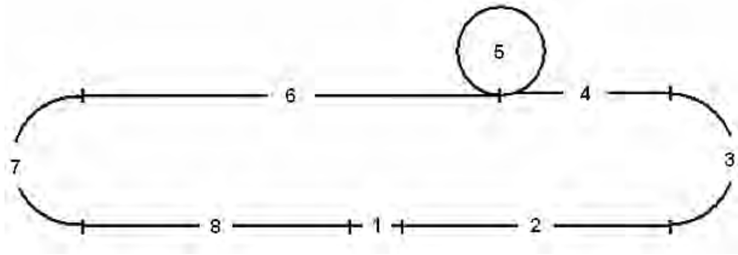
Case – The case was designed by considering different materials for each part, then choosing the ones that gave the highest strength for the lowest weight.

4.2 Mission Modeling and Optimization Analysis

This section discusses the processes and programs that Aerodynamics and Propulsions used to optimize the performance parameters of the aircraft to best fit the missions.

4.2.1 Aerodynamics Mission Optimization

Aerodynamics used a Mission Profile Optimization Program (MPOP) in order to determine a rough estimate of scores for different configurations. The program considered aerodynamic characteristics, weight estimates, and propulsive efficiency in order to model the mission profile in its entirety. It ran the aircraft through the mission course shown in Figure 9. Power required, energy used and g-Forces were all used as constraints as well as the physical constraints chosen by the team in order to fit the plane in the case. Drag was built up using models given in Raymer (1999) while the weight estimates were determined by Structures. Propulsion determined the propulsive efficiency, and experimental data was used for the battery and power estimates.



1	Takeoff	Full power, Flaps on
2	Climb	Full power, Flaps on
3	Turn 1	Full power, Flaps on
4	Accelerate	Full power, Flaps off
5	Turns 2 & 3	Full power, Flaps off
6	Cruise	Cruise power, Flaps off
7	Turn 4	Cruise power, Flaps off
8	Slow	Power off, Flaps on

Figure 9: Mission Course Profile

Using simultaneous variables, the program computed a score based on wing span, wing area, cruise velocities and batteries. With this, Aerodynamics found the highest scoring airplane. Through many iterations, the team was able to graph the data based on score to find optimum wing areas, wing spans and battery weights.

4.2.2 Propulsion Optimization

In tandem with the MPOP, Propulsion used a program to optimize a propulsion system to match the optimized mission performance. Airplane characteristics taken from the MPOP's winning plane, such as airfoil, wingspan, wing area, drag, and battery type and quantity were entered into Propulsion's optimization code. Parameters of various motors and propellers, along with different throttle percentages and gearbox ratios, were also inputted in order to match the performance outputs from the MPOP. The propulsion program then outputted mission performance data such as takeoff distance, current drawn, power consumption, and cruise conditions, for a range of possible wind speeds. This process narrowed down which propeller, motor, and gearbox ratio would best work for the design of the aircraft.

4.3 Design and Sizing Trade-offs

4.3.1 Aerodynamic Trade-offs

Using the MPOP, Aerodynamics examined three different types of airfoils: a high lift (Eppler 422), a medium lift (MH 114), and a low lift (SD 7034). This ensured that Aerodynamics found the best type for this year's competition. With these airfoils and MPOP, data for each airplane design was obtained and translated into a score. Below are charts showing the optimum wing span, wing area and battery weight for each airfoil used.

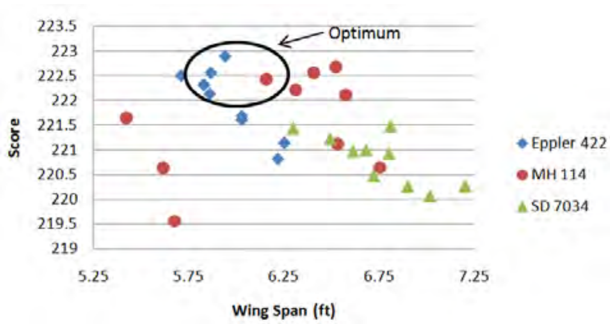


Figure 10: Wing Span Trade Study

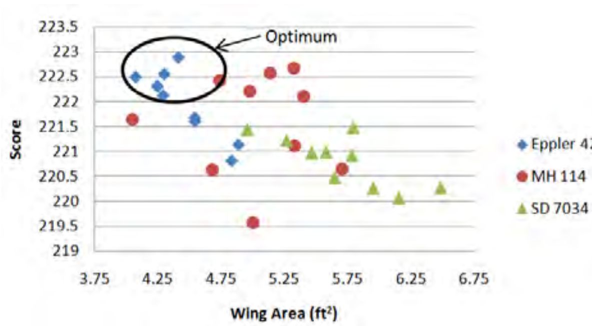


Figure 11: Wing Area Trade Study

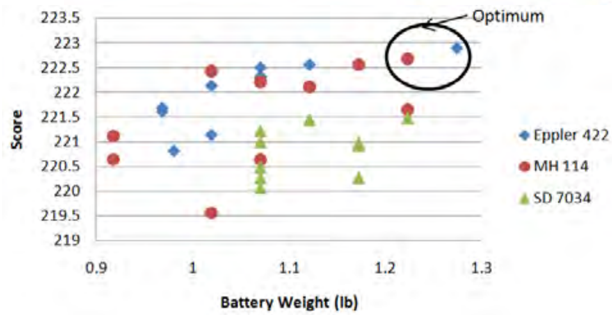


Figure 12: Battery Weight Trade Study

According to the graphs above, the high lift airfoils tended to give the highest score output. Using this data, Aerodynamics looked at various high lift airfoils, while using the MH114 and SD 7034 to continue in the aircraft optimization.

Airfoil Selection and Survey

By using the optimization discussed above, the team looked at several high lift airfoils. By further refining the search using thickness, camber, and flatness, the airfoils were narrowed down to twenty-eight for further comparison. With the narrowed list of airfoils, Aerodynamics divided the task evenly and ran a total of 6,300 optimization attempts to find the best possible configuration.

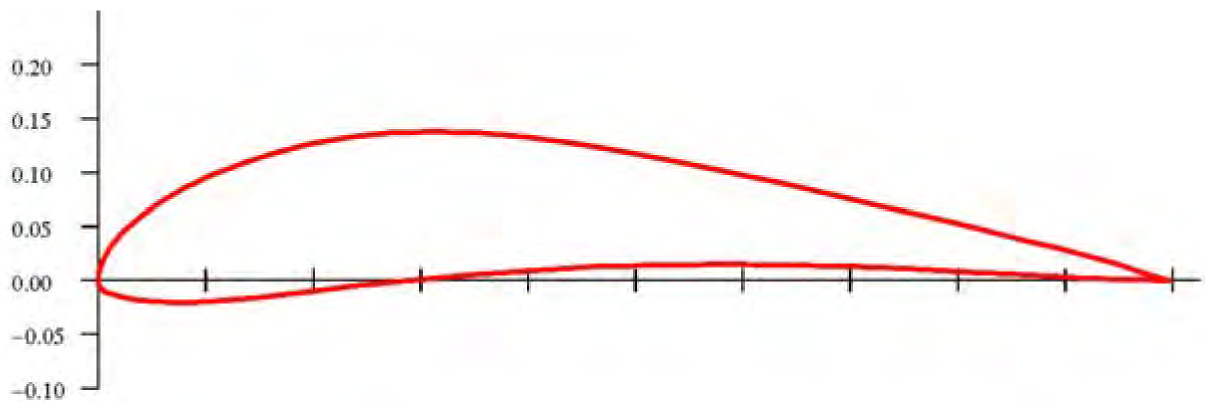


Figure 13: Eppler 422

From the optimization programs, the Eppler 422 airfoil was selected due to the highest score/weight ratio of the airfoils tested. The team then sized the wing that gave the airplane the highest score for the three missions. Using the optimization tradeoffs, the team arrived at a wing planform area of 4.5 ft^2. The wingspan was determined to be 6 ft with a chord of 9 inches giving the airplane an aspect ratio of 8. The wing design was then run through the MPOP at various wind speeds to ensure that the plane could reach the takeoff requirement from 0 to 25 mph of wind.

4.3.2 Propulsion System Trade-offs

A propulsion optimization program was used to find a propulsion system that would match the required performance that the MPOP outputted. The parameters from selected motors and propellers were input into the propulsion program and it in turn outputted flight data such as takeoff distance, endurance, power consumption, and cruise velocity. In this way the propulsion program could find and confirm propulsion systems that would match up to the required performance found with the MPOP.

Battery Selection

Using the MPOP, batteries of different make and type were inputted. Some of the batteries tested were: the Elite 1500, 2000, 4300 and GP 2000 batteries. The program took the battery type‘s information, applied it to the power requirements for the mission, and then outputted how many batteries it would take to fly the mission and the mission score possible. The battery that scored the highest for the three missions combined was the Elite 1500. According to the MPOP, 26 of them would be needed to successfully fly the missions, weighing in at 1.326 pounds. The table below shows the score result of each, taking the weight and power trade into account.

Battery	Total Flight Score
Elite 1500	222
Elite 2000	216
Elite 3300	215
Elite 4000	211
GP 2000	211

Table 12: Battery Flight Score Impact

Motor Selection

To select a motor to meet the takeoff and cruise endurance criteria set by the score optimizer, a range of motors was tested with the propulsion program. Three inputs were required: the motor resistance, Rm, the no load current, Io, and its voltage constant, Kv. The gearbox ratio could also be changed to try to make the motors mission compatible. The number of motors to test was reduced by first throwing out motors that were unreasonably heavy (over 10 ounces), had too low of a thrust (under 5 pounds), or lacked market availability.

In order to be considered, the motor had to enable the airplane to accomplish both a 100 foot takeoff in less than 5 mph winds and complete 3 laps in winds of at least 20 mph for mission 3 with a full payload. The motors found to meet these requirements in the propulsion program were in the 1100 to 1600 Kv range with a 6.7:1 gearbox. The gears were also chosen to be metal so they would not be stripped at the high torque needed to spin the 18 and 19 inch propellers considered. The most competitive motors are shown in Table 13.

Motor	Weight(oz)	Kv	Io(A)	R(ohm)	P(W)	T(lb)	Ibatt(A)
Neu 1110-3Y	4	1512	.45	.05	477.5	6.91	23.37
Neu 1112-3Y	4.7	1175	.35	.06	359.4	5.74	15.09
Neu 1509-2.5Y	7.5	1450	1	.04	460.7	6.74	22.46
Neu 1905-1.5Y	6	1350	1.4	.019	437.8	6.52	20.72

Table 13: Motor Trade

Of the motors, the 1905-1.5Y was chosen due to its low current draw as well as its competitive power and thrust. Its current draw with the 26 batteries was 5 amps under the limit, leaving room for extra power if real-life tests of the system showed more power was needed for flight maneuvering and takeoff. This potential for extra power left room for several propeller sizes as well if issues later called for a change. Even though it had the second highest weight of the other motors, this versatility led the team to choose it over the others.

Propeller Selection

Propellers of many different diameters and pitches were researched and subsequently tested in the propulsion program. A 20” propeller diameter maximum was imposed early on due to the design height of the plane combined with the required 1” of clearance between the ground and propeller. The team did not want to make the nose gear any longer than it needed to be due to case constraints and the weight increase.

For each diameter, several pitches were tested at different wind speeds in order to collect data. This data was used to create efficiency plots which were compared to find the closest to the optimum for the missions. The 19x10 inch propeller performed the best, but even with the motor mounted at the top of the fuselage, this size propeller would not allow for our mandated 1” of clearance between the propeller and the ground for the aircraft‘s current design. In lieu of this constraint, it was decided that the next best propeller, the 18x10 inch propeller, would be used. The 18x10 inch propeller, although second best, gave the power and performance necessary while conforming to size constraints placed on the aircraft and case. Propeller testing is discussed in more detail in section 8?

4.4 Analysis Methods and Sizing

<u>4.4.1 Tail Sizing</u>

Airfoil Selection

The NACA 0014 airfoil was selected for both horizontal and vertical tail due to the symmetric profile and ease of construction. It is a common airfoil for this use and thick enough to make construction fairly simple.

Horizontal Stabilizer Sizing

The team used a moment balance across the aircraft to determine the necessary down force provided by the tail to rotate the nose during takeoff. Without accounting for prop wash, the team calculated a tail area of 121 in^2 to achieve takeoff rotation. In addition, the team sized the tail to provide a larger static margin of 19%. This gave an area of 140 in^2, well in excess of the minimal requirements. This yielded a span of 20” and chord of 7”. To trim the aircraft, the tail was set at -2 degree incidence.

Elevator Sizing

The sizing of the elevator was performed by using a function in MathCAD based on an equation from Nelson‘s *Aircraft Stability and Automatic Control*. The team had the difficult task of sizing the elevator not only for adequate pitch control, but trimmed at an angle of attack that would not stall the wing. This was done to allow the pilot to do maximum performance turns without fear of stalling the aircraft.

For ease of construction, the span of the elevator was locked at 20”, the span of the horizontal stabilizer, and only the chord changed as a function of changing τ. The maximum angle of deflection was determined to be 17.5^{o}.

Vertical Stabilizer Sizing

Using the tail volume coefficient, and knowing the distance from the quarter-chord of the vertical stabilizer and the cg of the aircraft, the tail area was found to be .518 ft^2, or 74.59 in^2. As stated in the conceptual design of the case, in order to cut down on its overall size, a Mooney tail design was selected. Also, the top of the case was constrained, so the maximum height above the plane was ten inches. Since the bottom of the airplane sat on the bottom of the case, the max takeoff roll angle also made the trailing edge sweep angle on the tail. This made the dimensions of the tail to be 10” in height, with a top width of 6.6” and a bottom width of 9.3”.

Rudder Sizing

Using the same optimization program used by Propulsion, the maximum wind the plane could fly in was 25 mph, which came out to 36.65 ft/s. Using the same program, the landing velocity was approximated at 25 ft/s, which was close to the benchmark landing velocity of other aircraft in years past.

Given these two velocities, the side wash angle was found to be 52.83^{o}. Using the moment coefficient for a rudder, the vertical control surface was sized at 8 in^2 to counteract the crosswind. Then, using the part of the tail where the break occurs to the slope, the rudder was sized as a triangle to fit in the case. The rudder‘s exact dimensions were 7 inches tall with a maximum width of 2.25 inches.

<u>4.4.2 Grid System</u>

In order to perform mission 2, the inside of the fuselage was designed to hold ten softballs. According to the rules, the softballs must be restrained in a grid pattern during flight, and score for the second mission was highly based on the loading time for the softballs. This meant that not only must the payload bay be light, but it must also be able to prevent the balls from moving.

The group looked at several types of designs before selecting a simple grid pattern. With this decision, work was done on creating the best form of this type of grid.

This was where weight and strength considerations came in. The group wanted the spine and ribs to be as light as possible while able to endure ball impacts during loading. After studying trade-offs based on removing material from the grid system, the group decided to design a truss system. A well designed truss system would give the best strength, yet allow the team to remove the most material, thus saving weight

The grid itself was spaced to hold the softballs, one to each grid slot. The truss design for each part was then modeled and tested in SolidWorks. First, the spine was developed. Four main designs were considered and tested by modeling them in SolidWorks and then simulating balls dropping on the weakest points. Then the factor of safety (FOS), by which the truss held, could be evaluated and compared. Anything over a 1.5 FOS but under 5 was desirable. Each design was modeled in 1/16” balsa with each link being 1/8” wide.

Many different truss designs were examined, and after the analysis was complete, the group selected the best truss, seen in Figure 14. As the appearance of the design suggests, the top portion of the design provides more impact strength. Two balls dropped on the weakest point returned a FOS of 1.51. The spine's thickness was increased to 1/8" instead of 1/16" because of impact failures. The 1" wide key slot for the wing spar was inserted into the bottom center of the spine. The curved inner corners were added so the CNC machine could carve out the holes.

Figure 14: Truss With Best Key

Ribs were then designed to fit into the grid, first using triangle shapes much like the spine before simply hollowing out the center to save weight. A number of ribs with several different combinations of materials and designs were tested by dropping softballs on them. After analyzing the results, the final design was selected and made from carbon fiber and is shown in Figure 15.

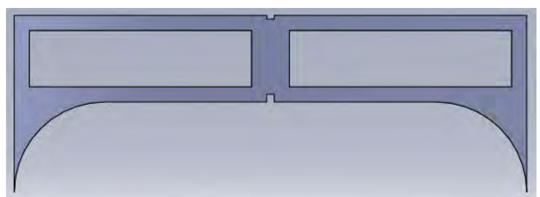


Figure 15: Final Rib

This grid design was not only part of the retaining system for the softballs but helped keep the fuselage rigid. The final dimensions of the spine were 21" by 3 3/8".

As discussed in section 3.0, suspended fabric was chosen to reinforce the truss and create the payload floor. The suspended floor provided space for the batteries and the wing spar, and prevented the balls from hitting these components or the bottom of the fuselage. After researching many types of fabric, rip-stop polyester was chosen, the same type of fabric used in kites and parachutes, which was known to be able to take large sudden loads.

4.4.3 Landing Gear

The landing gear design had to be able to accommodate each mission payload, as the length of the bats in mission 3 conflicted with a centered nose gear. The design selected in the conceptual design phase, a tricycle configuration, provided stable take-offs and landings and adequate ground handling.

Nose Gear

One large problem experienced by teams throughout the last few years was breaking of the nose gear during landing. This often occurred when the gear flexed in a lateral direction (port/starboard). The Euler buckling equation is $F = \frac{\pi^2 EI}{(KL)^2}$. F is the critical load, E is the modulus of elasticity, I is the area moment of inertia, K is the effective length factor, and L is the length. This equation shows that if the length was doubled, the allowed critical load was cut to a fourth. This relation between length and critical load justified the need for the nose gear to be as short as possible. One easy way to decrease the nose gear length was to move it back along the fuselage since the bottom of the plane was closer to the ground. Moving the nose gear back interfered with the bat restraint system, however the team provided a gap between the bats to allow the nose gear a space to fit.

Taking the Euler equation into consideration, the nose gear took on the shape of a mushroom, laid up with carbon fiber and Kevlar to support roughly 20% of the aircraft's weight and to absorb the landing forces like a spring.

Main Gear

The main gear was spaced slightly aft of the cg of the plane, so that the nose gear supported about 20% of the aircraft weight. If the main gear was moved further aft, takeoff roll would increase. The main gear needed to be relatively easy to attach and remove in case of failure. It also needed to have a backwards rake so that it was perpendicular to the ground when landing; the angle needed was approximately 15^{o}. The height of the team's landing gear depended mainly on the size of the propeller being used on the aircraft. The gear was sized so the tip of the propeller was one inch off the ground for clearance during takeoff.

Carbon fiber composites were used to make the bow-shaped landing gear to better take the impact of landing. Its exact layup and shape was refined through tests.

4.4.4 Case Trade Study

The scores on each mission were largely based on the weight of the system, including the airplane, radio controller, and the case. Each team's score used the lightest score from any team as a reference. Thus, if a team did not have the lightest configuration, the team would only receive a fraction of the total score. For this reason, all optimization on the case was done due to weight.

As discussed in the section 4.0, it was determined that removing the main landing gear would tilt the tail downward so the bottom of the fuselage rested on the floor of the case, making the case's height 14 inches. This vastly decreased the system weight and caused the team to reanalyze the overall layout and design of the case and the plane.

Several different materials were analyzed to find one that was both light and strong enough to use for the floor and supports. These materials are presented in Table 14.

Floor Material	Density (g/in^3)	Walls and Roof Material	Density (g/in^3)
Foam Core Board	2.195	**Balsa Wood**	1.179
Plastic Board	2.973	**Single Glass Balsa**	2.326
1/8" Plywood	7.205	**Double Glass Balsa**	3.472
Cardboard	2.978	**Wire Hanger**	.8333 (g/in)
		1/4" Carbon Fiber Tubing	.9449 (g/in)

Table 14: Material Weights

The configuration of the floor was decided based on maximum possible weight of the contents of the case that would be supported by the floor and the maximum possible dimensions of the case. The optimization of the case floor was then performed by reducing the amount of weight of the contents actually coming in contact with the floor and then reducing the overall dimensions of the container to house the components as compactly as possible. By reducing the amount of weight on the floor, the bending stress was reduced, lowering the need of material in each member, and thus reducing overall weight. The primary weight that would act on the floor included the plane, each wing, and the radio receiver. Figure 16 shows several floor configurations believed to offer max strength and least weight for the case. In all of the configurations, the wings would be suspended on the walls with two straps each to secure them. The detached landing main gear would be suspended from the roof of the case straddling the fuselage of the plane.

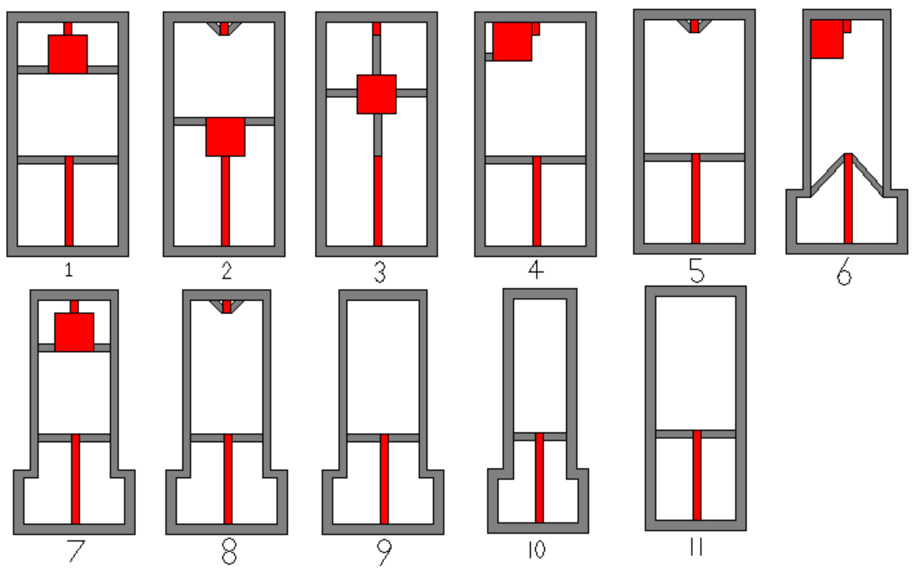


Figure 16: Foam Floor Optimization

In configurations 1, 2, 3, 4, 6, and 7, the weight of the receiver and the weight of the aircraft were supported by the nose gear and the bottom of the tail, which was then supported by the floor. Configurations 5, 8, 9, 10, and 11 were considered with the receiver off the floor, leaving only the weight of the aircraft on the floor. In an attempt to remove as much weight off the floor as possible, the nose of the aircraft was suspended from the roof with a strap running under the plane just behind the nose gear. Configurations 9, 10, and 11 used this idea.

The T shapes, configurations 6, 7 8, and 9, were believed to reduce the amount of material used to cross brace the areas where the weight of the contents were supported. However, since the final dimensions of the plane were not known, they were quite oversized. Once the dimensions were finalized, the case was changed to configuration 10. Later, the team decided to have the 18” propeller always attached, which led the team to make a rectangular shaped case that was 20.5”x48”, represented by configuration 11. In each configuration, the members were 2” wide. This was an attempt to provide enough material to support all of the weight and prevent major types of failure from bending loads, axial loads, and buckling. The data used to decide the best configuration is presented in Table 15. Configuration 11 was lighter than all other configurations except configuration 10, which could not be used because the propeller must remain attached.

Configuration	Total Length	Area	Volume	Weight (g)	Weight (oz)
10	152.5”	305 in^2	62.525 in^3	137.242 g	4.84103 oz
11	161.5”	323 in^2	66.215 in^3	145.342 g	5.12673 oz

Table 15: Foam Floor Configuration Data

In order to further reduce weight, the walls and roof of the case were designed as a frame with fabric covering, like a tent. These walls must be rigid enough to support the weight of the wings and the weight of the front of the fuselage, and the roof must be able to support the weight of the main landing gear and the radio controller. Table 16 shows different materials for the frame that would be able to support these loads. The posts ran along each edge of the case, excluding the floor, which required four 14” wall posts from the floor to the roof, two 20.5” posts and two 48” posts for the area of the roof. The fabric used for the cover of the case was lightweight ripstop fabric. The fabric wrapped completely around the foam floor and the frame. The weights shown in Table 16 are considered per square yard of material and are considered to be worst case.

Tent Posts	Metal Hanger	.25” square carbon tube	.25” square carbon tube cut longitudinally
Total Length	193”	193”	193”
Weight (oz)	5.67316 oz	6.43269 oz	3.21635 oz

Table 16: Frame Material Data

Tent Material	1.9 oz/yd^2	1.4 oz/yd^2	1.1 oz/yd^2
Area	3886 in^2	3886 in^2	3886 in^2
Weight (oz)	5.69707 oz	4.19784 oz	3.2983 oz

Table 17: Ripstop Data

The case was designed to open from the top, with the whole roof folding back like a door along one of the long edges of the case. Since the case needed quick and easy access to the softballs for mission 2, only a minimal part of the door would be opened during this mission. This would be big enough to remove a bag with ten softballs in it. Fabric snaps kept the area of the door closed that the team did not want to open during mission 2 while easy to open hooks secured the second part of the door. The total weight of the case was estimated at 13 oz, including hardware and epoxy, and the final case configuration was a rectangular cube of 20.5”x48”x14”.

4.5 Lift, Drag, and Stability Characteristics

After the configuration was decided, the plane was tested at a range of wind conditions to see how it performed. This study ranged from a no wind condition all the way up to 25 mph of wind. In the no wind condition, the main concern was ensuring that the aircraft would be able to take off in the required 100 ft. For this case, the fully loaded plane was able to takeoff and leave the ground in 66 ft with the original numbers that the performance optimizer suggested for cruise velocity. In order to maximize score, the team increased the cruise velocity numbers until the aircraft's takeoff was just under 100 feet and power required was still less than power availiable. When the plane was run though a increasing range of wind speeds, 25 mph was found to be the highest wind speed the plane could complete the mission in. In this case, the plane took off in a very short distance but the cruise speed was severely lowered due to the need for much more energy to penetrate through the strong winds. The Cl and Cd graphs for the Eppler 422 are shown in Figure 17.

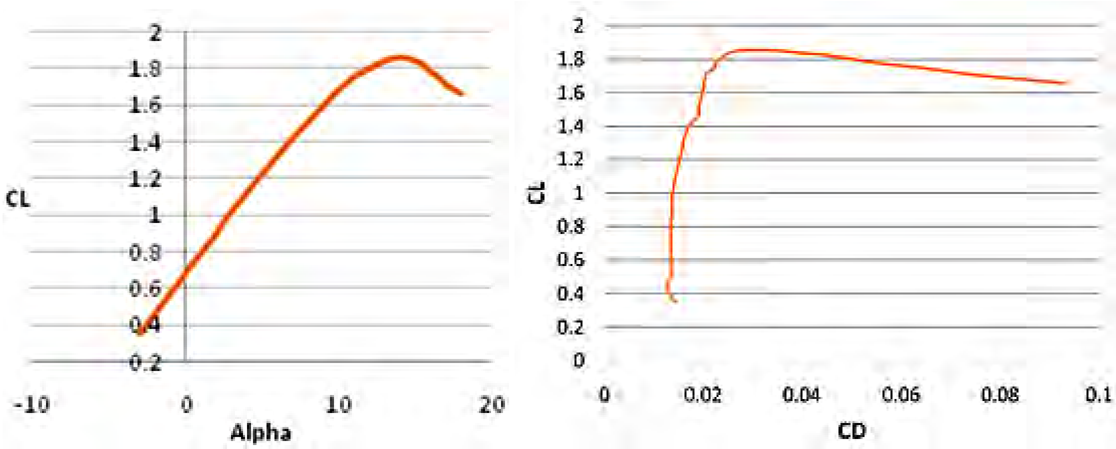


Figure 17: Lift Curve and Drag Polar

Drag Characteristics

In order to accurately predict the aircraft's flight characteristics, a drag model was made. The team put together a basic preliminary model of how each component of the current configuration contributed to drag. The only portion of drag left out of the analysis was the induced drag created by the wing during flight. The actual values calculated were the drag areas of each component or the D/q value. The total drag for mission 1 was 1.78 lbs, while for mission 3 it went up to 2.03 lbs. Figure 18 shows pie graphs displaying how each component compares to the overall drag produced for each external configuration.

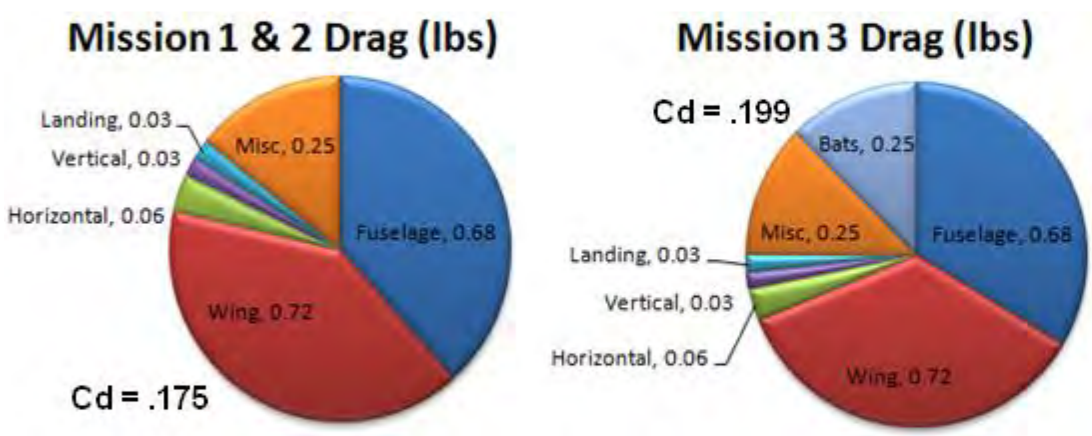


Figure 18: Drag Breakdowns

Stability Characteristics

Using the equations for C_m, the team was able to determine the overall stability of the aircraft. Using the moment breakdown for the fuselage, the team found the coefficient to be near zero and unaffected by alpha. The overall coefficient vs. alpha is shown in Figure 19.

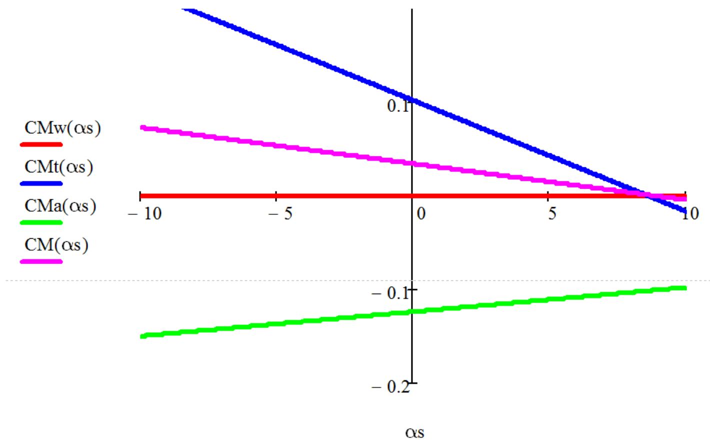


Figure 19: Cm vs Alpha

Figure 19 shows that the team's plane design is statically stable. The neutral point was designed at 0.44 of the chord length, giving the plane a 19% static margin. While this may seem high, this static margin was desired by the team's pilot for better control of the airplane.

Once this was determined, a study on dynamic stability began. Stability coefficients were calculated and then used to find over 25 stability derivatives. These derivatives were then used to predict the dynamic aircraft modes using eigenvalues. Figure 20 shows the real response (η), frequency/imaginary response (ω), damping value (ζ), and time to half for each mode.

Aircraft Mode Eigenvalues				
Mode	**η**	**ω**	**ζ**	**$t_{1/2}$**
Short Period	-4.38	6.07	0.721	0.16
Phugoid	-0.029	0.65	0.044	23.79
Roll	-0.006	0	-	108
Dutch Roll	-0.22	0.49	0.44	3.13
Spiral	-2.28	0	-	0.3

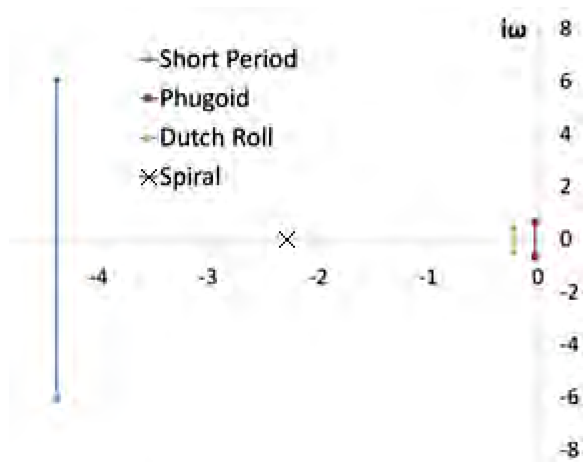


Figure 20: Eigenvalues of Aircraft

As seen in Figure 20, all dynamic modes of the aircraft were stable, so the sizing of all of the stabilizers was proven to be correct.

4.6 Aircraft Mission Performance

Using the MPOP, the team was able to get basic time estimates for the missions used to configure the aircraft. In both the missions, high performance turns were used to decrease the overall completion time of the course. The team used the distances calculated by the program, and based on the estimated cruise and turn velocities, the total time breakdown was calculated, as shown in Table 18.

Mission 1 Profile (5 mph Wind)				
Mission Phase		**Velocity (ft/s)**	**Distance (ft)**	**Time (sec)**
Lap 1	Takeoff	0-43.6	12.152	1.0
	Climb	54.2	149.8	2.8
	Cruise	65.0	350.2	5.4
	180^0 Turn 1	65.0	97.2	1.5
	360^0 Turn	54.2	149.8	2.8
	Cruise	65.0	1000.0	15.4
	180^0 Turn 2	65.0	97.2	1.5
Lap 2	Cruise	65.0	1000.0	15.4
	180^0 Turn 1	65.0	97.2	1.5
	360^0 Turn	54.2	149.8	2.8
	Cruise	65.0	1000.0	15.4
	180^0 Turn 2	65.0	97.2	1.5
	Cruise	65.0	500.0	7.7
	Added Time for Acceleration/Deceleration and Error			5.2
Total				**79.9**
Score (out of 50)				**45.0**

Table 18: Mission 1 Profile

Using MPOP again, the team substituted in the new cruise velocity for the mission carrying the bats. The overall time increased as expected, due to how the cruise velocity dropped and the turns could not be completed as efficiently. The total time breakdown is shown in Table 21.

Mission 3 Profile (5 mph Wind)				
Mission Phase		**Velocity (ft/s)**	**Distance (ft)**	**Time (sec)**
Lap 1	Takeoff	0-43.6	65.7	3.2
	Climb	46.6	307.6	6.6
	Cruise	60	192.4	3.2
	180^0 Turn 1	60	219.0	3.6
	Cruise	60	1000	16.7
	360^0 Turn	46.6	307.6	6.6
	180^0 Turn 2	60	219.0	3.6

Lap 2	Cruise	60	2000	33.4
	180^0 Turn x2	60	438.0	7.2
	360^0 Turn	46.6	307.6	6.6
Lap 3	Cruise	60	2000	33.4
	180^0 Turn x2	60	438.0	7.2
	360^0 Turn	46.6	307.6	6.6
	Added Time for Acceleration/Deceleration and Error			7.0
Total				**144.9**
Score (out of 100)				**91.8**

Table 19: Mission 3 Profile

Again, the team ran the mission profile for mission 2, but this time the profile was done for the best and worst case scenarios. Since this mission's score does not depend on time of flight, the plane could be flown for maximum efficiency if needed, yet the profile was done to find the fastest completion time to show the team what to expect.

Mission 2 Profile (10 Balls) (5 mph Wind)				
Mission Phase		**Velocity (ft/s)**	**Distance (ft)**	**Time (sec)**
Lap 1	Takeoff	0-43.6	39.0	2.2
	Climb	49.3	316.0	4.7
	Cruise	60.6	184.0	3.0
	180^0 Turn 1	60.6	169.9	2.8
	Cruise	60.6	1000	16.5
	360^0 Turn	49.3	316.0	4.7
	180^0 Turn 2	60.6	169.9	2.8
Lap 2	Cruise	60.6	2000	33.0
	180^0 Turn x2	60.6	338.4	5.6
	360^0 Turn	49.3	316.0	4.7
Lap 3	Cruise	60.6	2000	33.0
	180^0 Turn x2	60.6	338.4	5.6
	360^0 Turn	49.3	316.0	4.7
	Added Time for Acceleration/Deceleration and Error			7.0
Total				**137.7**

Table 20: Mission 2 Profile Maximum Weight

Mission 2 Profile (6 Balls) (5 mph Wind)				
Mission Phase		**Velocity (ft/s)**	**Distance (ft)**	**Time (sec)**
Lap 1	Takeoff	0-43.6	25.3	2.2
	Climb	49.3	267.8	3.7
	Cruise	60.7	232.2	3.8
	180^0 Turn 1	60.7	139.0	2.3
	Cruise	60.7	1000	16.5
	360^0 Turn	49.3	267.8	3.7
	180^0 Turn 2	60.7	139.0	2.3
Lap 2	Cruise	60.7	2000	33.0
	180^0 Turn x2	60.7	278.0	4.6
	360^0 Turn	46.6	267.8	3.7
Lap 3	Cruise	60.7	2000	33.0
	180^0 Turn x2	60.7	278.0	4.6
	360^0 Turn	46.6	267.8	3.7
	Added Time for Acceleration/Deceleration and Error			7.0
Total				**131.1**

Table 21: Mission 2 Profile Minimum Weight

As Table 20 and Table 21 show, the difference in payload weight does have an impact on the flight time, 6.6 seconds — providing further proof of how important weight is to flight time and score.

5.0 DETAIL DESIGN

In this phase of the design, the group finalized and optimized all of the components for the system previously discussed. Predictions were made for mission and flight performance based on the final design, and building began.

5.1 Dimensional Parameters

Table 22 shows the finalized dimensional parameters for the general structural systems, propulsion, and electrical systems.

Wing	
Span	6'
Chord	9"
Aspect Ratio	8
Wing Area	4.5 sq. ft
Airfoil	EPPLER 422
Static Margin	19%
Aileron	
Span	25.3'
Chord	1.8"
δ_a	20°

Horizontal Stabilizer	
Airfoil	NACA 0014
Span	20"
Chord	9"
Area	140 sq. in.
Incidence	-2°
Elevator	
Span	20"
Chord	1.75"
δ_e	17.5°

Vertical Stabilizer	
Airfoil	NACA 0014
Span	10"
Root Chord	9.3"
Tip Chord	6.6"
Rudder	
Span	8"
Max Chord	3"
Min Chord	.5"
δ_r	20°

Fuselage	
Length	45.5"
Width	8.75"
Height	5.5"
GTOW (est.)	12.6 lb

Electrical System	
Speed Controller	Jazz 55-10-32
Radio Receiver	Spektrum AR9000
Number of Servos	5
Servo Type	Futaba S3102

Motor	
Type	Neu 1905-1.5Y
Weight	6 oz
Kv	1350 rpm/v
I_o	1.4 A
R	0.019 Ω
P_m	437.8 W
Thrust	6.52 lb
I_{batt}	20.72 A
Propeller	18 x 10

Batteries	
Type	Elite 1500
Capacity	1000 mAh
R	0.015 Ω
V	1.2 v
I_{max}	25 A
Number of Cells	26
Pack Capacity	1000 mAh
R_{pack}	0.39 Ω
V_{pack}	31.2 v
$I_{max, pack}$	25 A

Table 22: General Aircraft Dimensions and Parameters

5.2 Structural Characteristics and Capabilities

These are the overall structural characteristics and general capabilities of the team's airplane. Further details follow in section 5.3. Weight of the system affected the score a great deal, so the aircraft and case were made as small and light as possible. The main landing gear was made to be removable, allowing the case to be made smaller, thus removing unneeded height. For clearance, the nose gear was positioned between the third and fourth bats.

The tail of the aircraft was also uniquely designed to allow it to fit snugly into the case. Although the tail is forward swept, it carries out all functions a normal tail would, but its design saves space and weight.

For the inner structure of the aircraft, weight was further saved by making the spine of the aircraft act as the center of the grid system for the payload. This also allowed for the landing gear and the center spar of the wing to attach to it, providing a load path to the strongest point in the airplane. The grid's ribs attached to the walls of the fuselage, providing further support, while a cloth hammock draped over the whole grid, creating a ‗floor' above the bottom of the fuselage where the batteries were placed and took a majority of the payload's impact during loading. The hammock was sewn by hand with rip-stop polyester. This lightweight material was also used in covering the case, further lowering the system's overall weight.

These designs combined to produce a system that could compete effectively for the best score. Every change made was to reduce weight, gain a better loading time, or make everything integrate better.

5.3 System Design, Component Selection and Integration

5.3.1 Grid System

The final grid design is shown in Figure 21.

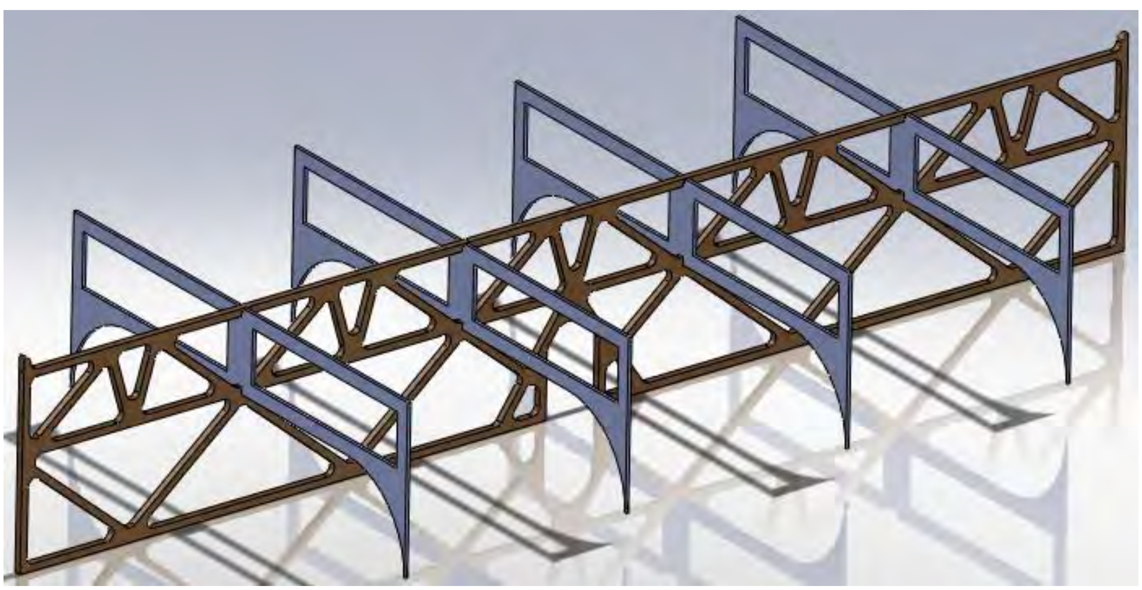


Figure 21: Final Grid Design

The hammock was sewn into one entire piece, as shown in Figure 22, and draped over the grid. To help keep the hammock's shape during loading, narrow plastic tubing was cut and sewn where the hammock folded over the top edge of the ribs.

Figure 22: Ball Hammock

As Figure 23 shows, the hammock leaves enough space below it to be used by batteries, the wing box, landing gear fasteners, and wires for the elevator, rudder, and ailerons.

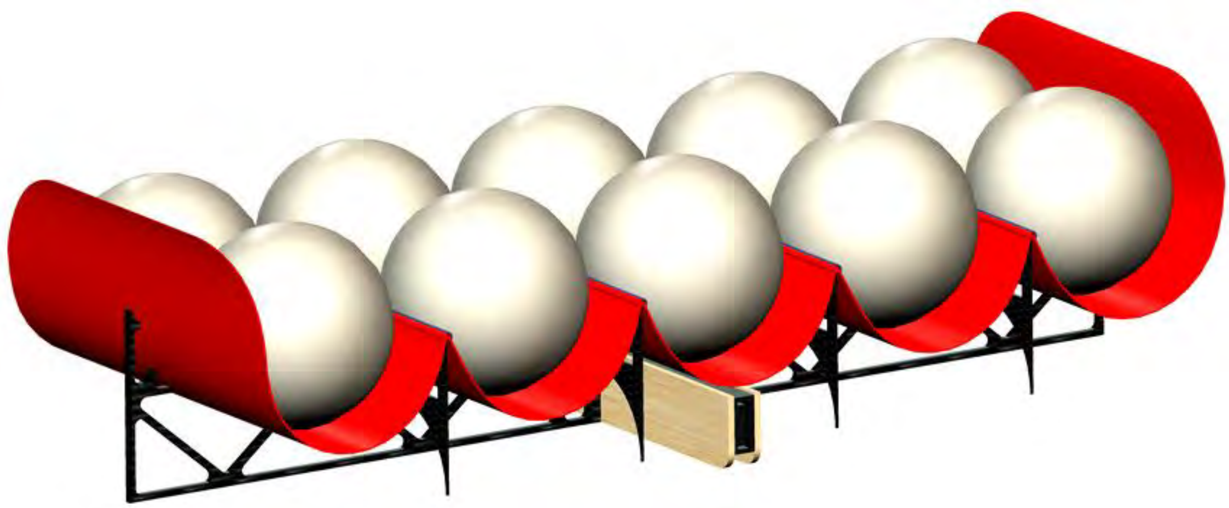


Figure 23: Hammock Space

5.3.2 Clamshell Doors

A two door, clamshell design was selected because of its ability to perform multiple roles. The doors not only enclosed the payload bay, but also helped secure the softballs in the grid system and acted as a funnel to assist in speedy loading. The doors were hinged at the cg of the softballs with a lightweight fabric hinge. Bulkheads were at the fore and aft of each door. This kept the doors rigid and provided a front and back barrier to keep the softballs from missing the payload bay during loading. Each door was secured at the fore and aft payload bay bulkheads by a simple formed c-shaped clasp. This provided a strong, lightweight, and reliable latching system to secure the softballs. The payload doors had lightening holes cut into then to lose weight.

The doors were cut from the mold of the fuselage. This allowed the exact shape desired to be more easily obtained.

5.3.3 Wing Mounting System

The final major design consideration was the wing connection system. As discussed in 3.3.3, a single spar with a friction fit was determined to be the best method to secure the wings. The friction fit was the best method because the forces which would try to pull the wings from the fuselage were small, allowing for the absence of bolts or pins. The male end of the connection system was in the wing and this was attached to two ribs for strength. Kevlar string was wrapped around the spar configuration to reinforce the wing box. With this configuration, and the overall design of the wings, the team predicted that they would hold GTOW at 2.5 g's without breaking. Testing is discussed in sections 7 and 8.

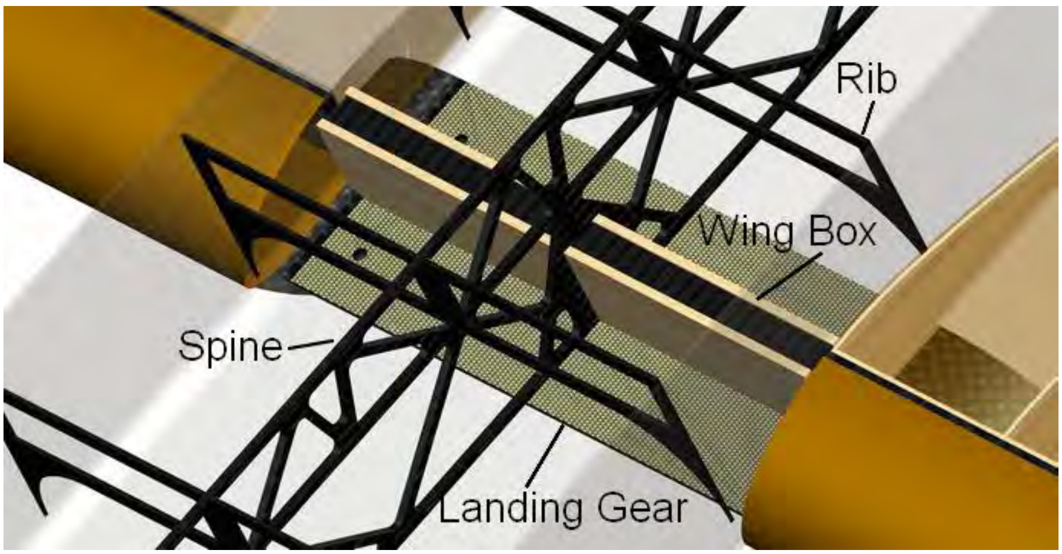


Figure 24: Connection System

5.3.4 Tail

Since it was decided that the landing gear would be removable for the sake of saving weight on the case, a forward swept tail configuration was decided upon. Using the determined sizing dimensions and referencing benchmark forward swept tail aircrafts, the CAD drawings were generated as seen in Figure 25. Refer to Table 22 for general dimensions.

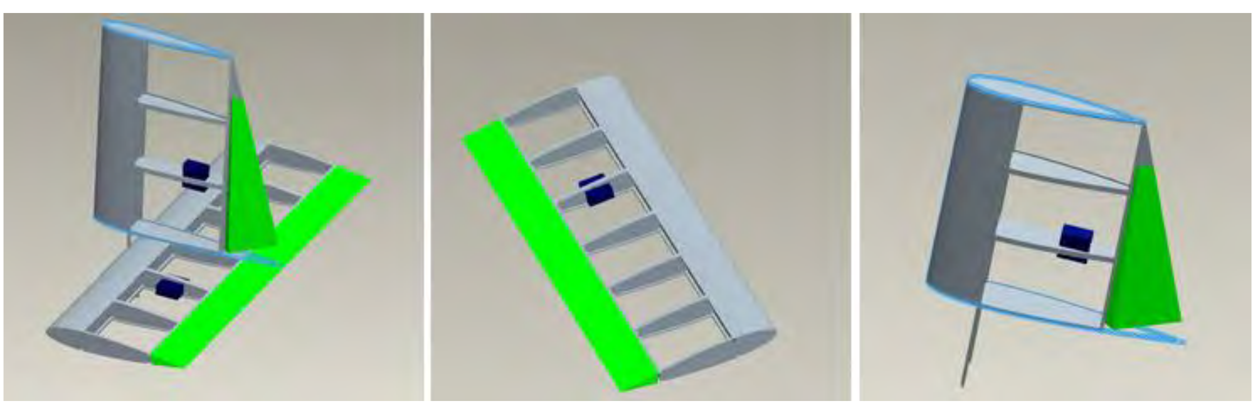


Figure 25: CAD Models of Empennage (Full assembly, Horizontal Tail, Vertical Tail)

In Figure 25, the dark blue blocks represent the servo placement. It was decided that the servos would be mounted in the tail for both the horizontal and vertical components. Placing the servos in the horizontal and vertical stabilizer rather than in the fuselage would allow for a shorter metal pushrod, which would decrease the overall weight of the structure.

A D-channel ran around the entire leading edge of the horizontal and vertical stabilizer. This provided extra torsional rigidity as well as extra structure to the overall empennage.

5.3.5 Landing Gear

A basic tricycle configuration for the landing gear was selected for its structural efficiency and ground handling. The main gear was designed as a bow gear, because this served multiple purposes that could be utilized. Since the landing gear was designed to be strong and durable, the team used its structure to carry the external payload. The gear was wide enough to carry five ―bats" side-by-side and just tall enough to ensure successful tail ground clearance on takeoff.

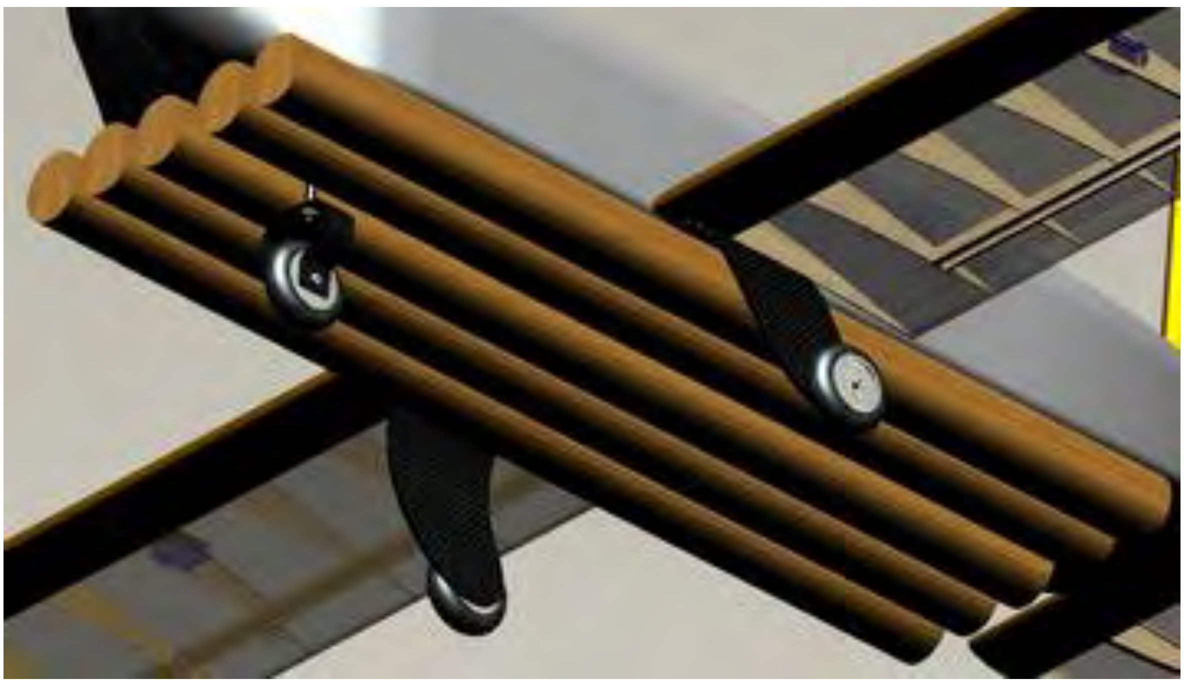


Figure 26: Landing Gear with Bats

The bow gear was fastened to the fuselage using four small bolts, two on either side of the center bat. The bats lay in a line with a small (1/4"-1/2") gap between the third and fourth bats to enable the nose gear rod to pass between. The two bolts that lie within the gap will be replaced with threaded hooks during the third mission. Small notches cut in the main gear kept Kevlar string in place, and the string wrapped under the bats sat in the hooks to tightly fetter the bats and landing gear to the fuselage. The bow gear was made from composite. With this design, the team estimated that the landing gear could and should be made to endure GTOW loading conditions when the airplane was landing at a ten degree angle. If the gear could hold up to this, then it would be able to hold up to normal landing conditions.

The nose gear was slightly off-center to allow it to pass between the third and fourth bats during mission 3. A mushroom gear was chosen for its light weight, energy absorption, and easy customization and optimization. The mushroom gear was attached to a thin carbon rod, and made of composites. . The team designed the nose gear to hold a predicted weight of 20 percent of the aircraft under landing conditions.

5.3.6 Case

The final case design consisted of a foam core board base with a frame made of cut carbon tubing. This frame supported walls and a ceiling made from ripstop polyester. When the plane was in the case, the main landing gear was removed so that the tail of the fuselage rested on the floor. The nose was suspended from the ceiling so that the nose gear did not rest on the floor. The radio gear was stored in a pocket on the ceiling of the case and the detachable wings were held by straps to the longitudinal walls, also not touching the floor. This let the floor become optimized so that it only had to support the carbon fiber frame and the tail section. Figure 27 shows the unfolded case with the positions of the fasteners.

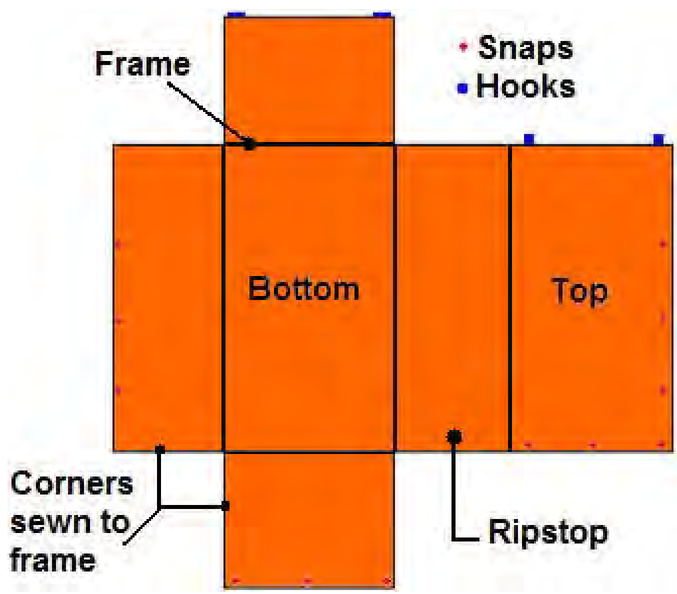


Figure 27: Unfolded Case

5.4 Weight and Balance

The total weight of the airplane, without payload, was estimated to be 6 lbs. For weight analysis, a weight and balance table was computed based on estimates for the aircraft components, both for worst case scenario and empty payload. Every measurement was made from the nose of the plane, not including the propeller.

		CG		
		X	Y	Z
Mass Wing	1.13	1.5	0	0
Mass Bat 1	0	1.5	-4	-0.25
Mass Bat 2	0	1.5	-2	-0.25
Mass Bat 3	0	1.5	0	-0.25
Mass Bat 4	0	1.5	2	-0.25
Mass Bat 5	0	1.5	4	-0.25
Mass Receiver	0.02	3	0	0
Mass HT	0.31	3.33	0	0
Mass VT	0.33	3.33	0	0.5
Mass Motor	0.38	0.13	0	0
Mass Fuselage	1.17	1.92	0	0
Mass Batteries	1.33	0.62	0	0
Total Mass	4.66			
		X	Y	Z
	Total CG	1.50	0	0.04

		CG		
		X	Y	Z
Mass Wing	1.13	1.5	0	0
Mass Bat 1	1.25	1.5	-4	-0.25
Mass Bat 2	1.25	1.5	-2	-0.25
Mass Bat 3	1.25	1.5	0	-0.25
Mass Bat 4	1.25	1.5	2	-0.25
Mass Bat 5	1.25	1.5	4	-0.25
Mass Receiver	0.02	3	0	0
Mass HT	0.31	3.33	0	0
Mass VT	0.33	3.33	0	0.5
Mass Motor	0.38	0.13	0	0
Mass Fuselage	1.17	1.92	0	0
Mass Batteries	1.33	0.62	0	0
Total Mass	10.91			
		X	Y	Z
	Total CG	1.50	0	-0.13

Table 23: Mission 1 and 3 Weight and Balance

For mission 2, the only cg change due to the softballs was in the y-direction when an odd number of balls was placed in the compartment. In the case where seven or nine balls were loaded, the cg changed by the weight of one softball.

		CG		
		X	Y	Z
Mass Wing	1.13	1.5	0	0
Mass Balls	0.4	1.5	0.125	0
Mass Receiver	0.02	3	0	0
Mass HT	0.31	3.33	0	0
Mass VT	0.33	3.33	0	0.5
Mass Motor	0.38	0.13	0	0
Mass Fuselage	1.17	1.92	0	0
Mass Batteries	1.33	0.62	0	0
Total Mass	5.06			
		X	Y	Z
	Total CG	1.50	0.01	0.03

Table 24: Mission 2 Weight and Balance

A diagram showing how the balls would be placed for each scenario is shown in Figure 28 to illustrate how to obtain the minimal cg change of the aircraft. As the weight difference between the 11" and 12" balls was only .3 oz, it was deemed acceptable to treat them the same in this evaluation.

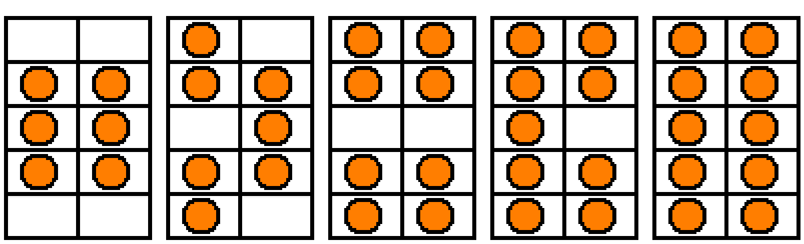


Figure 28: Ball Placement in Grid to Ensure Constant CG

5.5 Flight Performance Parameters

Once the airplane‘s design was finalized, some basic flight parameters were calculated for the missions with timed flights; see Table 25.

Flight Parameters					
Aircraft Parameters		Mission Parameters	Mission 1	Mission 2	Mission 3
C_{L0}	.4861	Max Climb Rate (ft/s)	9.212	8.274	7.257
C_{LMAX}	1.3249	Stall Speed (ft/s)	29.08	37.06	41.55
e	.85	Cruise Speed (ft/s)	65	60.6	60
C_{DO}	.0321	Take-Off Distance (ft)	12.152	39.0	65.717
		Maximum Speed (ft/s)	85	85	85
		Max G-Load	4.356	2.296	1.89
		Turn Rate (deg/s)	120	64.3	50

Table 25: Flight Parameters

5.6 Mission Performance

After the final design of the aircraft was completed, a reevaluation of the aircraft‘s predicted performance was done.

5.6.1 Pre-Mission Assembly

The assembly of the aircraft requires very little time to complete. The wings must be bolted on, main landing gear must be attached, and the wiring must be connected to the wings. With the team‘s design of these different components, the process should take less than 2.5 minutes, only half of the time limit for assembly. This leaves time for unexpected problems should they arise.

5.6.2 Mission 1 – Ferry Flight

The main focus on this mission is performing the 2 laps as fast as possible. Due to the lack of payload, there is no need to worry about takeoff distance or a lack of power to climb. The plane will easily be able to fly at its maximum velocity, the only limiting factor being the constraints of the aircrafts‘ turning radius. The total flight time for this mission was estimated to be 80 seconds in 5 mph wind speeds, assuming maximum velocity of 65 ft/s with time added for decelerating and accelerating back up to maximum speed before and after turns. This produced a score of 45 out of the possible 50 points.

5.6.3 Mission 2 –Softball Payload Flight

The ground crew can only improve their running speed so much, so the primary concern of this mission was cutting down the team‘s softball loading speed. With the hatch‘s clamshell design and the grid system‘s lowered spine, the team has consistently shown that the time to load the softballs into the plane, once the teammate arrives at the plane, will be 1.5 to 2 seconds. The total loading time (running to the aircraft, loading payload, and returning) was estimated to be 8 seconds.

The other critical mission goal is having enough power to make the three laps with the heavy softball payload. The plane must have enough battery power to take off in less than 100 ft with the payload and also complete the three laps. Time is not an issue during flight, so the plane can be flown at its most efficient speed to conserve power without risk of reducing score. Assuming we would achieve the quickest loading time, the score received would be 95.8 out of 100 points.

5.6.4 Mission 3 – Bat Payload Flight

In order to be competitive in this mission, the team must fly with all five bats. Flight time is crucial in this mission but a balance of speed and endurance must be met in order to carry this payload. Not only do the 5 bats weigh more than the full softball payload, but they will also cause a significant increase in drag due to being mounted on the outside of the aircraft. In 5 mph winds, it is estimated that the plane will be able to complete the mission in 145 seconds. The score achieved was 91.8 out of 100 points.

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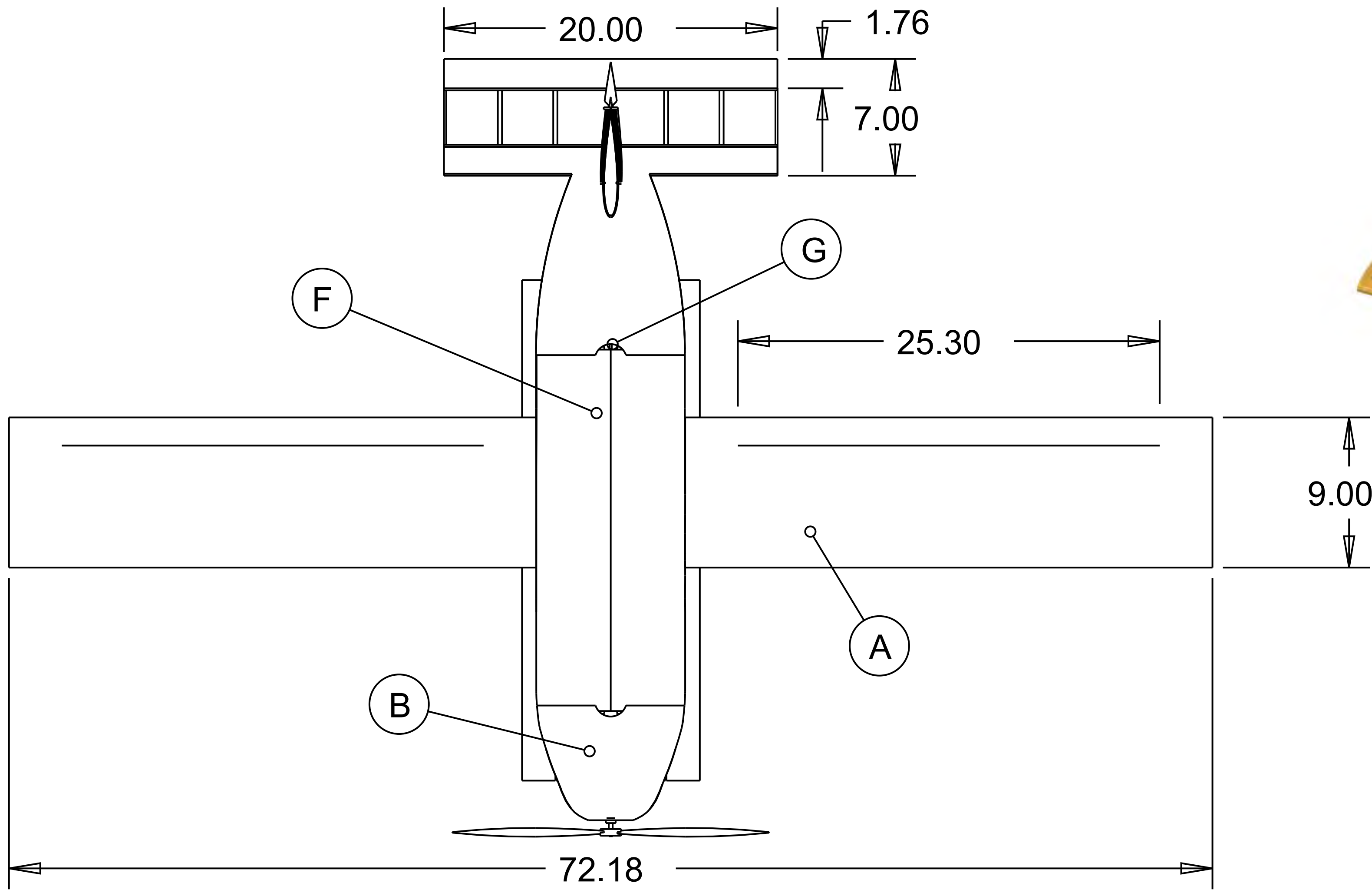
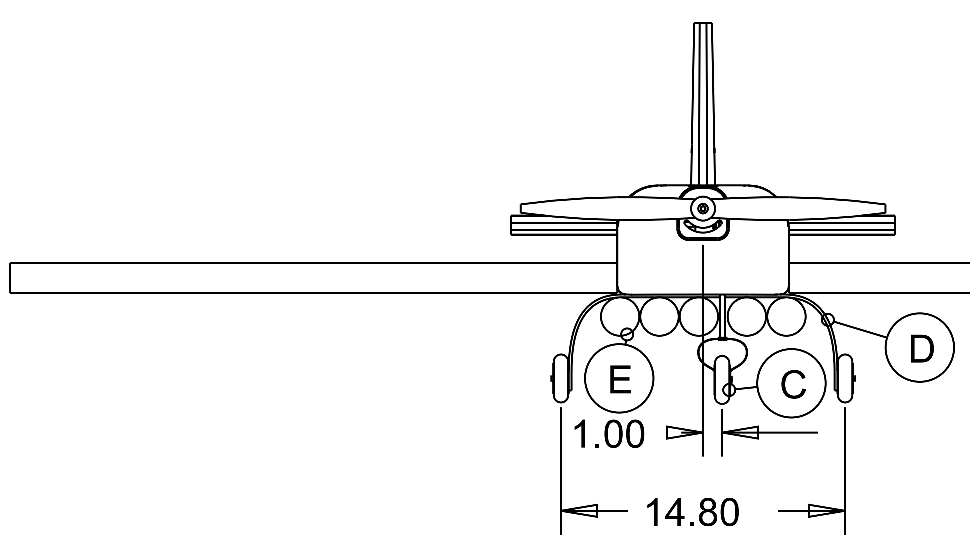
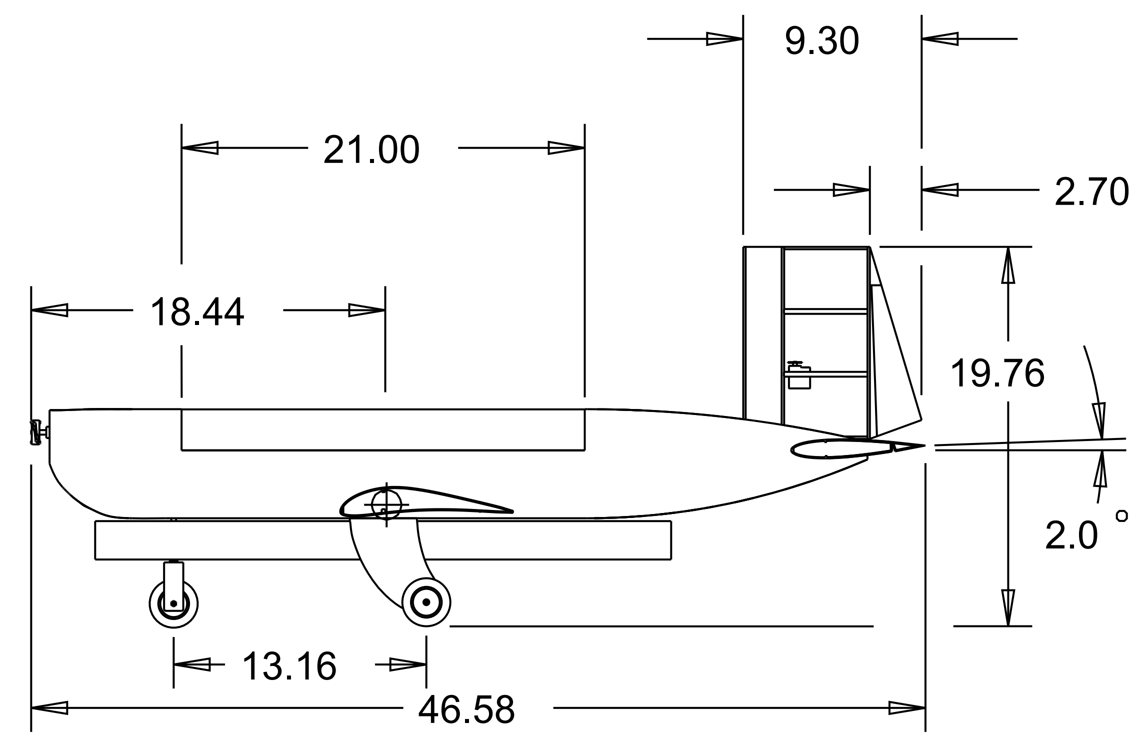
Orange Seduction	02/24/2010	SHEET 1







PRIMARY COMPONENTS					
A	MAIN WING	D	MAIN LANDING GEAR	G	FINGER WELL/EXHAUST 2X
B	FUSELAGE	E	EXTERNAL PAYLOAD		
C	NOSE GEAR	F	CARGO DOOR 2X		

NOTE:
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CESSNA - RAYTHEON - AIAA DESIGN/BUILD/FLY 2010

DOCUMENT TITLE

AIRCRAFT 3 - VIEW

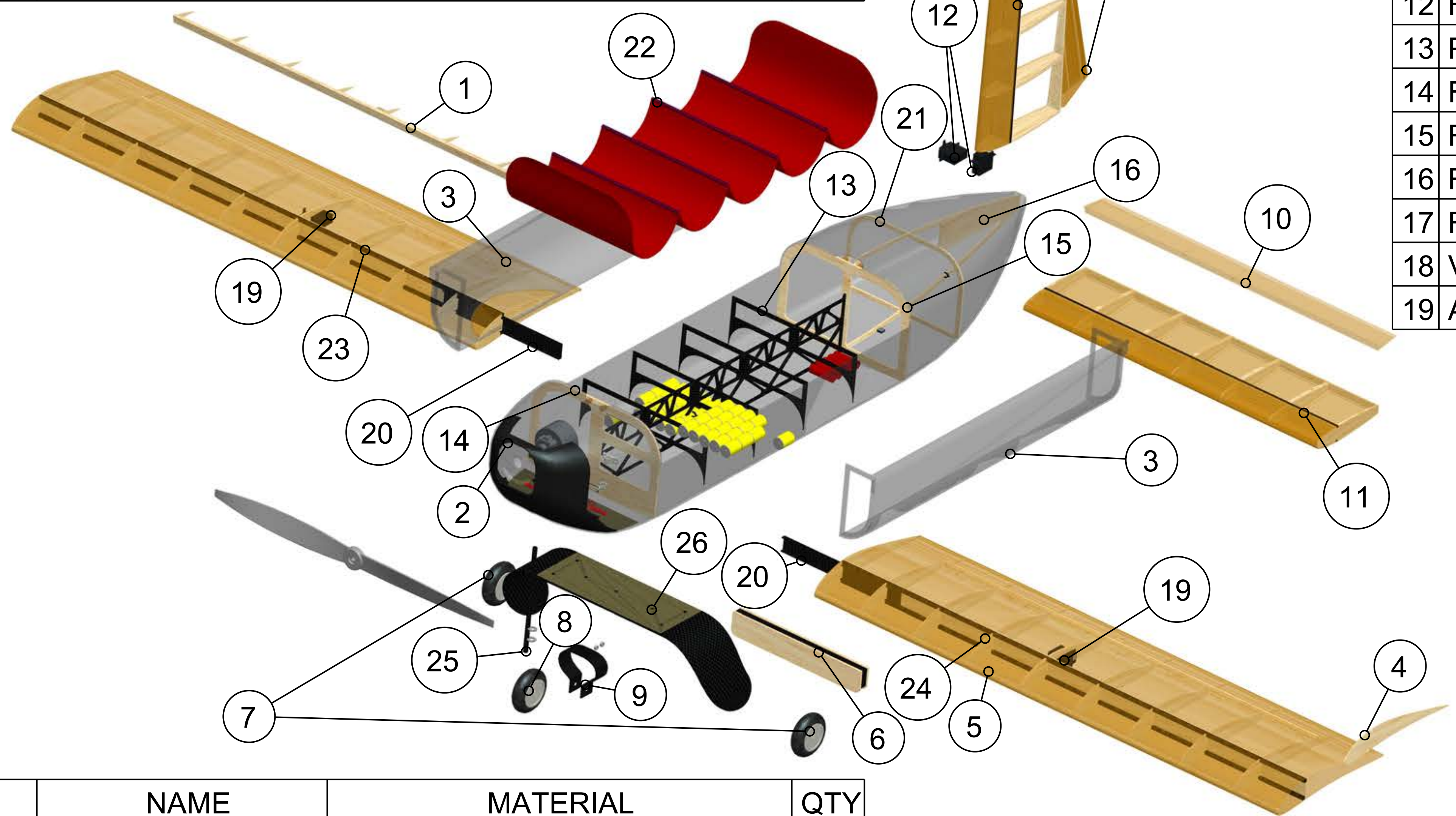
DRAWN BY	SIZE	APPROVAL DATE	REPORT TITLE	REV #
JOHNNY CHANDLER	B	02/24/2010	DRAWING PACKAGE	A
CHIEF ENGINEER: ERIC PROVO	SCALE	1/10		PAGE 1 OF 5

OKLAHOMA STATE UNIVERSITY TEAM ORANGE

Orange Seduction | 02/24/2010 | SHEET 2

	NAME	MATERIAL	QTY
1	AILERON	COMPOSITE REINF. BALSA	2
2	MOTOR MOUNT	LIGHT PLYWOOD	1
3	CLAMSHELL DOOR	COMPOSITE REINF. BALSA	2
4	AILERON RIB	BALSA	20
5	AILERON D-CHNL.	BALSA	2
6	WING BOX	CARBON FIBER & LIGHT PLY.	1
7	3" MAIN WHEELS	RUBBER & PLASTIC	2

	NAME	MATERIAL	QTY
8	NOSE WHEEL	RUBBER & PLASTIC	1
9	SHOCK ABSORBER	CARBON COMPOSITE	1
10	ELEVATOR	COMPOSITE REINF. BALSA	1
11	H.S. SHEAR WEB	CARBON/BALSA COMPOSITE	1
12	H/V STAB SERVOS	FUTABA S3102	2
13	PAYLOAD GRID	CARBON COMPOSITE	1
14	F MAIN BLKHD	LIGHT PLYWOOD	1
15	R MAIN BLKHD	LIGHT PLYWOOD	1
16	REAR SPINE	LIGHT PLYWOOD	1
17	RUDDER	MONOCOTE-COATED BALSA	1
18	V. S. SHEAR WEB	CARBON/BALSA COMPOSITE	2
19	AILERON SERVO	FUTABA S3102	2



	NAME	MATERIAL	QTY
20	MAIN SPAR	CARBON COMPOSITE	2
21	REAR BULKHEAD	LIGHT PLYWOOD	1
22	PLD HAMMOCK	RIP-STOP NYLON	1
23	ALRN. STRINGER	CARBON/BALSA COMPOSITE	4
24	ALRN. SHR. WEB	BALSA	16
25	NOSE GEAR	CARBON COMPOSITE	1
26	MAIN GEAR	CARBON COMPOSITE	1

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DOCUMENT TITLE
STRUCTURAL ARRANGEMENT

DRAWN BY JOHNNY CHANDLER
CHIEF ENGINEER ERIC PROVO
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APPROVAL DATE 02/24/2010
REPORT TITLE DRAWING PACKAGE
REV # A
SCALE 1/10

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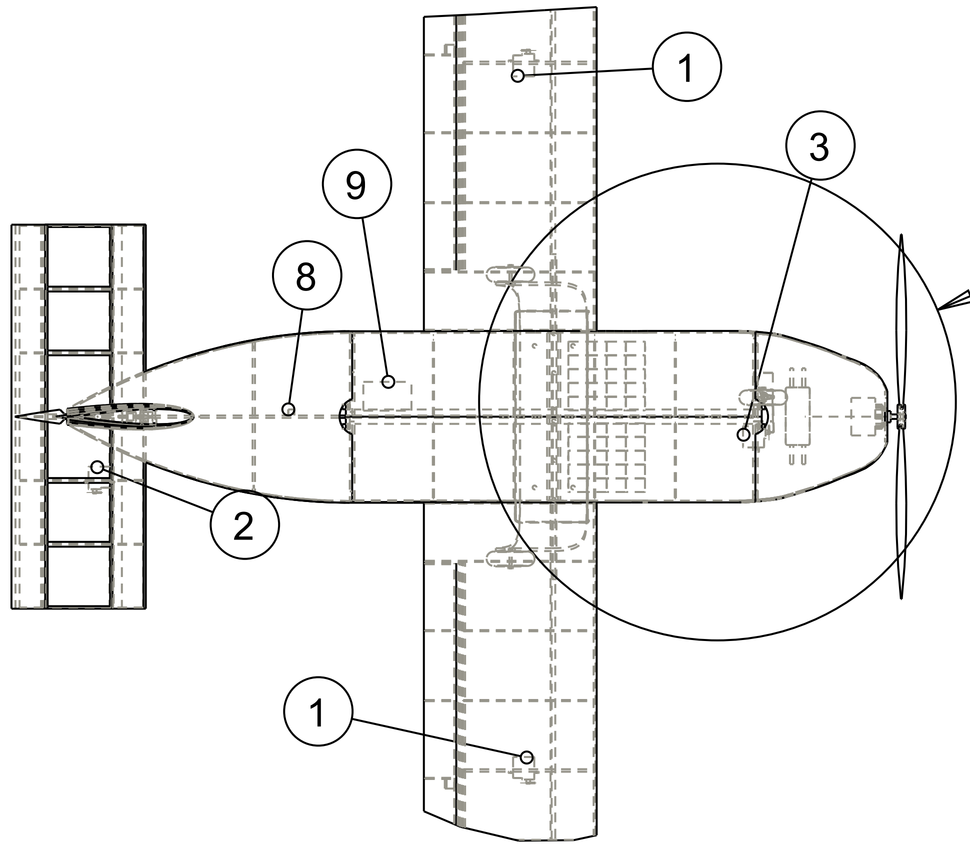

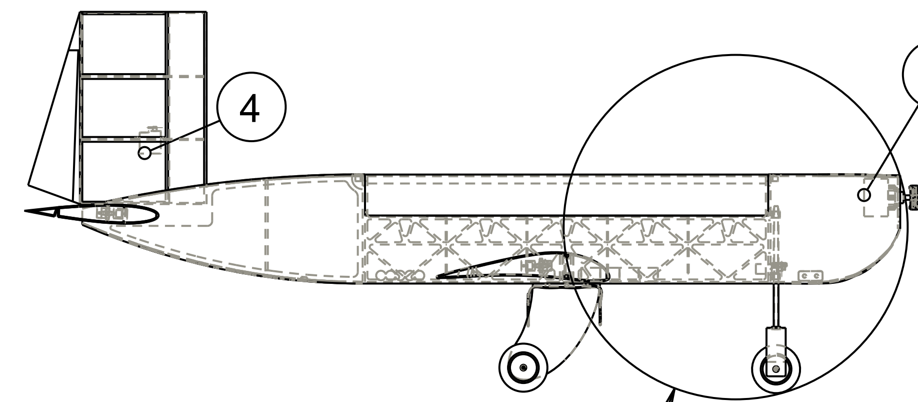

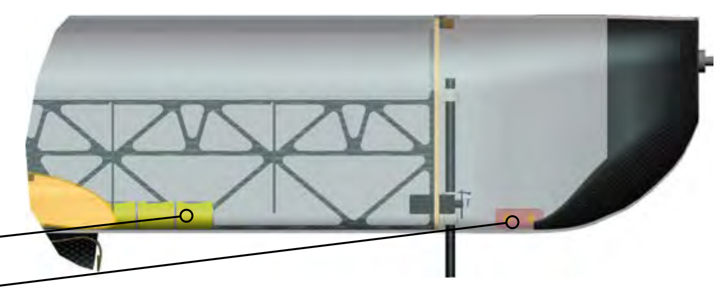

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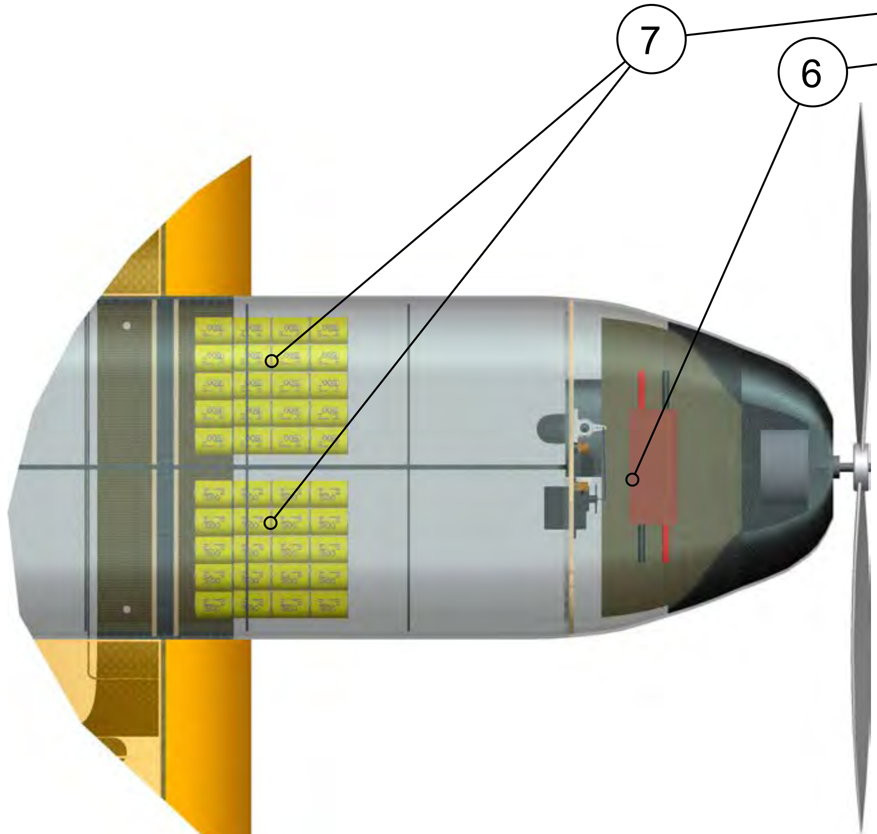

DETAIL A
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	ITEM	MANUF.	MODEL	QTY
1	AILERON SERVO	FUTABA	S3102	2
2	ELEVATOR SERVO	FUTABA	S3102	1
3	NOSE GEAR SERVO	FUTABA	S3102	1
4	RUDDER SERVO	FUTABA	S3102	1
5	MOTOR	NEU	1905/1.5Y	1
6	SPEED CONTROLLER	JAZZ	55-10-32	1
7	MAIN BATTERIES	ELITE	1500	26
8	RECEIVER	SPEKTRUM	AR9000	1
9	RECEIVER BATTERY	KAN	400mA	1

NOTE:
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DOCUMENT TITLE
SYSTEMS LAYOUT / LOCATION

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SIZE B | APPROVAL DATE 02/24/2010 | REPORT TITLE DETAILED DRAWING | REV # A

SCALE 1/10 | SHEET 3 OF 5

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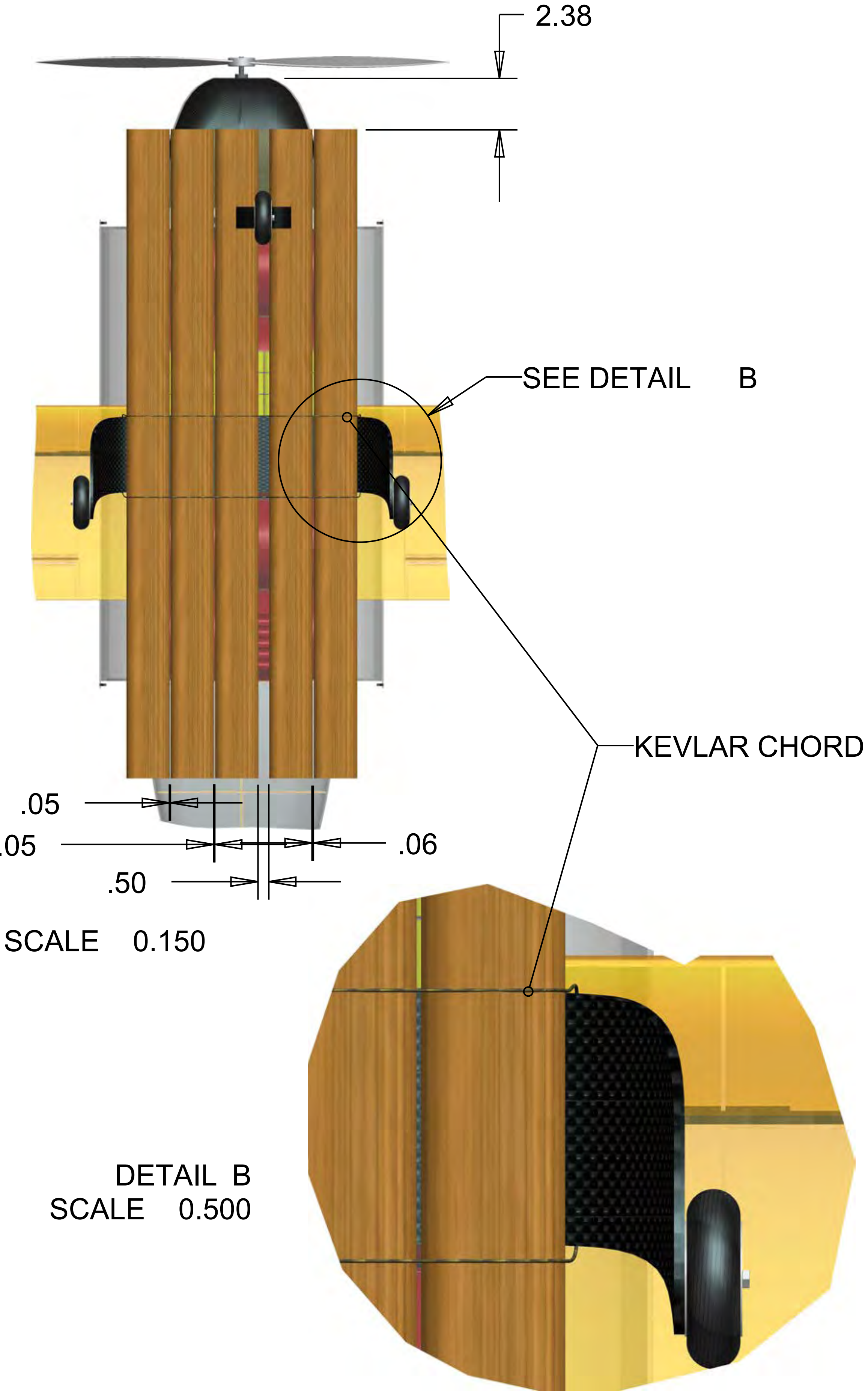
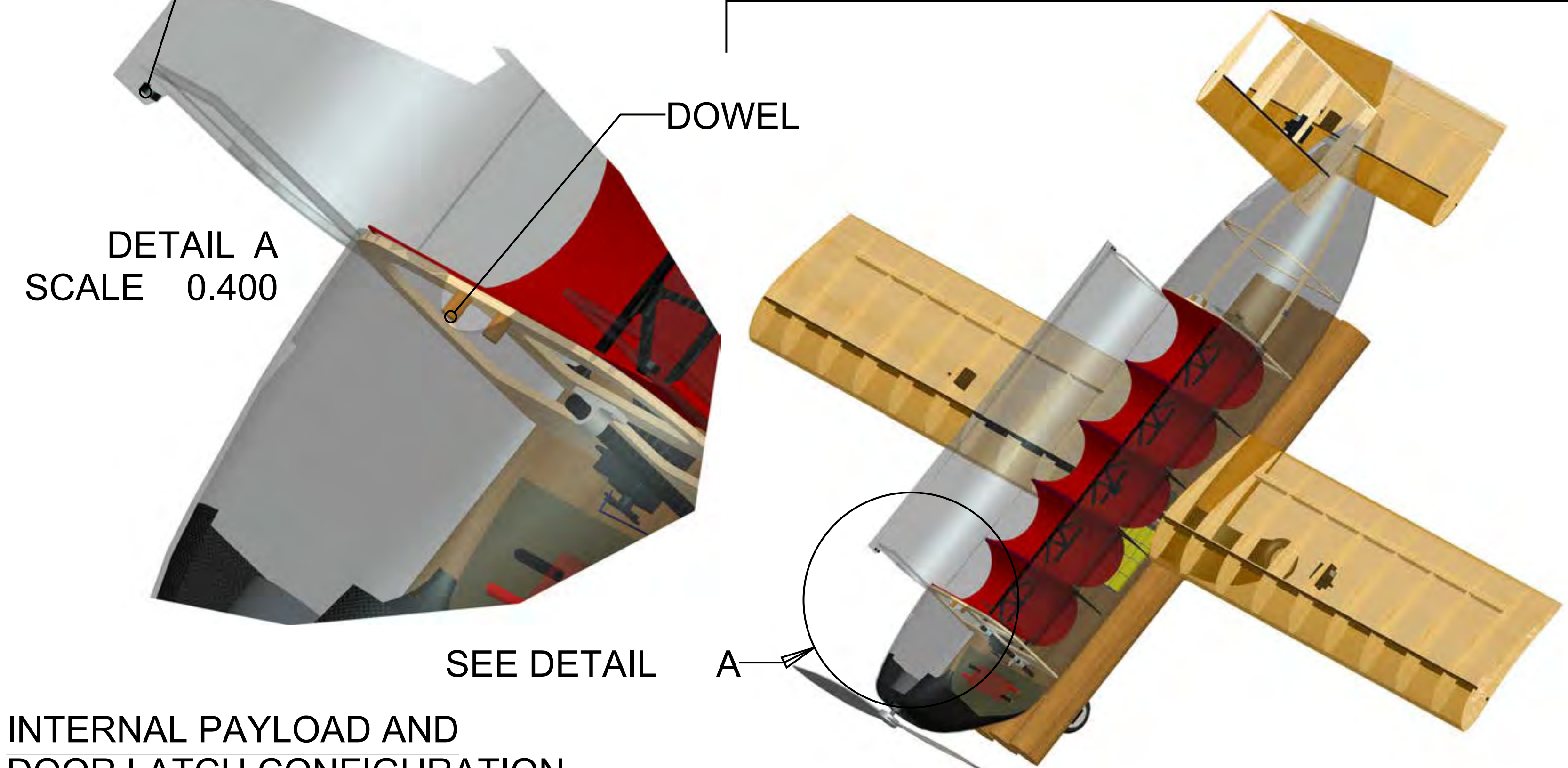
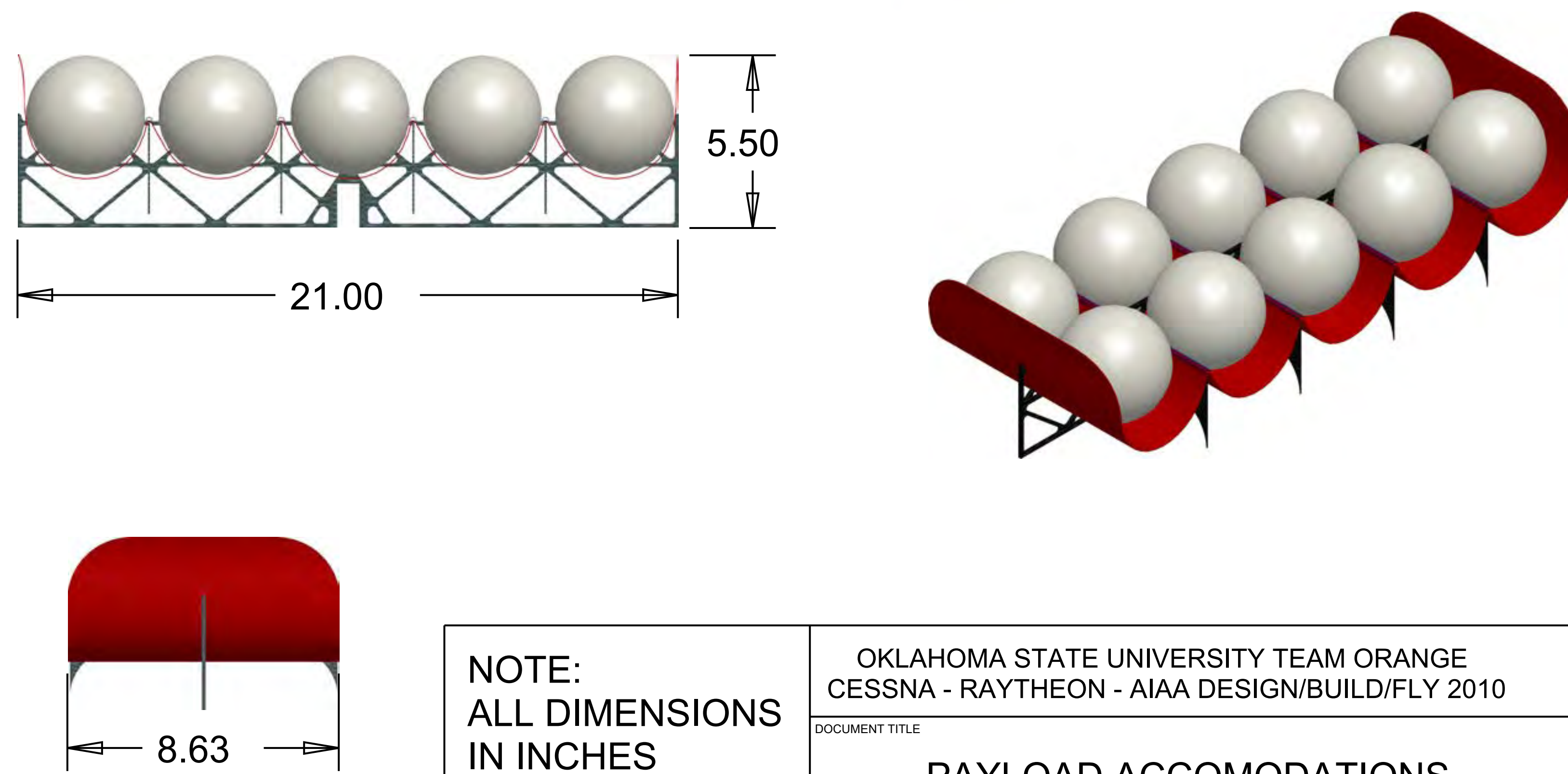
Orange Seduction | 02/24/2010 | SHEET 4



EXTERNAL PAYLOAD CONFIGURATION



INTERNAL PAYLOAD AND
DOOR LATCH CONFIGURATION



CRADLE ASSEMBLY

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DOCUMENT TITLE
PAYLOAD ACCOMODATIONS

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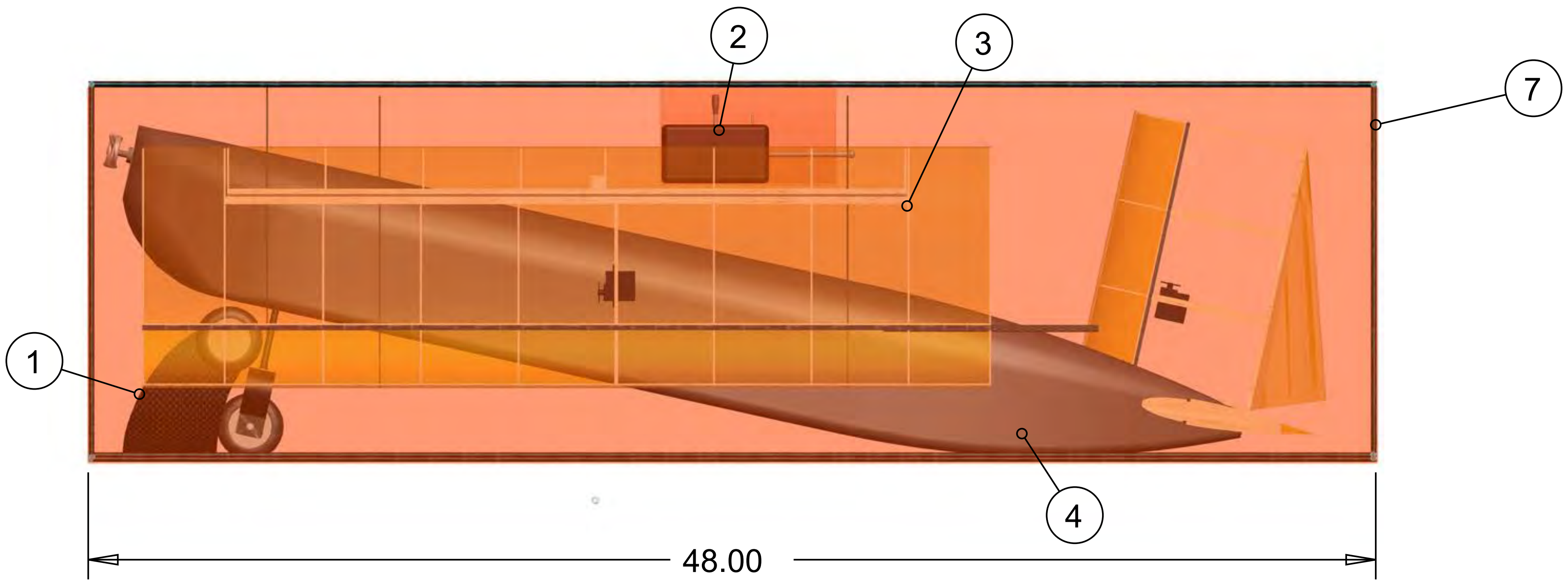
SCALE 0.170 | SHEET 4 OF 5

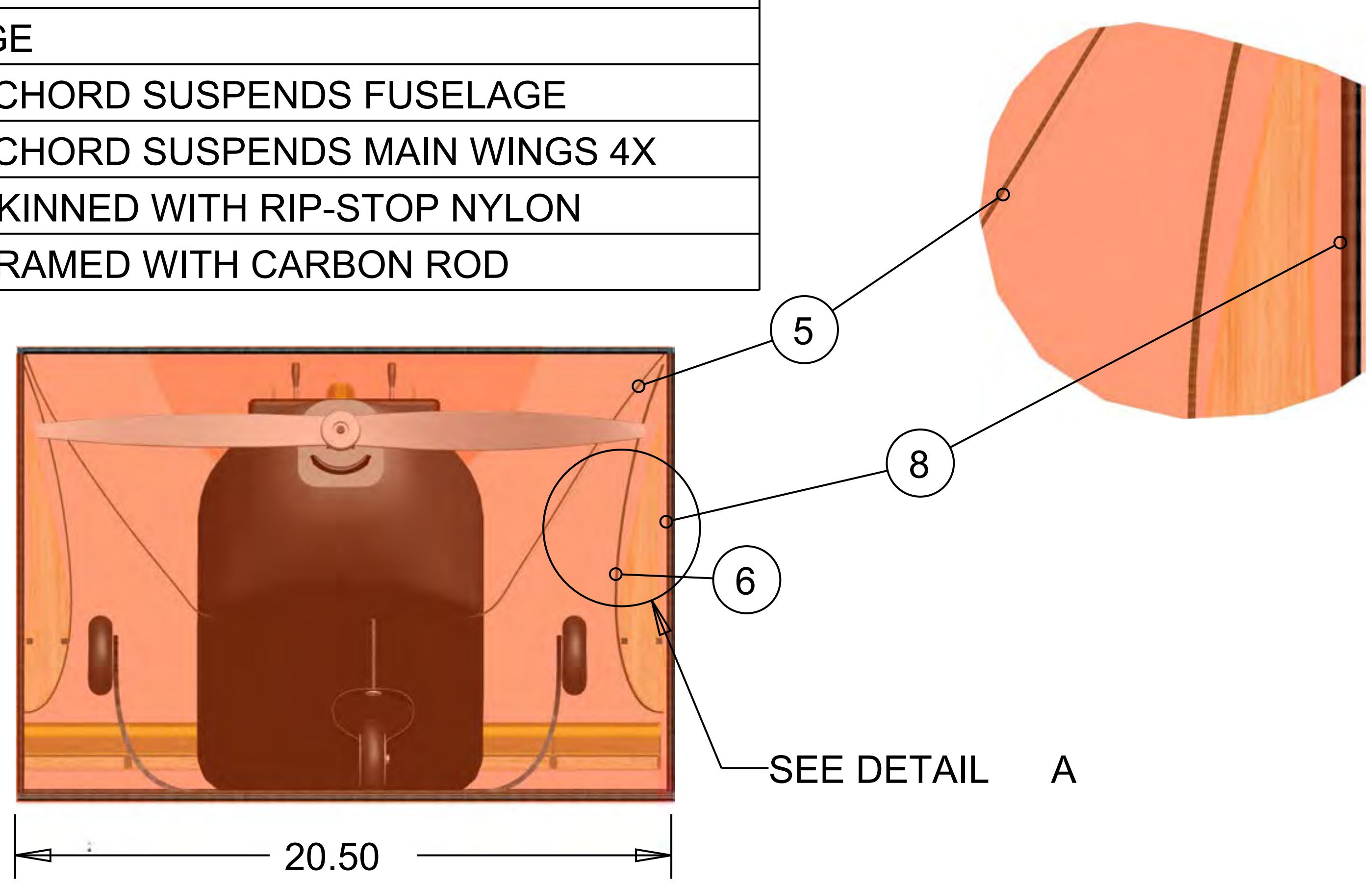
OKLAHOMA STATE UNIVERSITY TEAM ORANGE

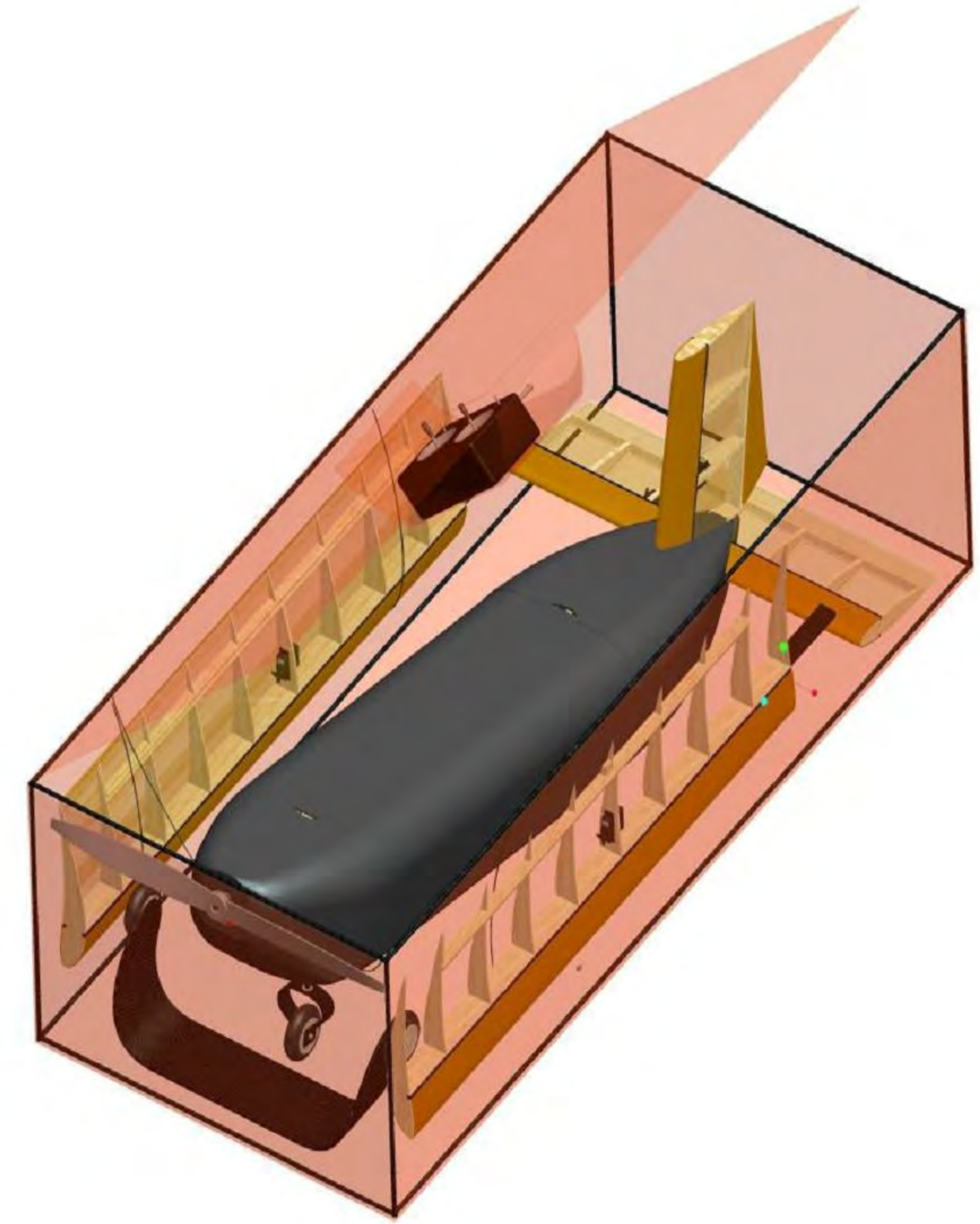
Orange Seduction | 02/24/2010 | SHEET 5



	DESCRIPTION
1	MAIN LANDING GEAR
2	TRANSMITTER STORED IN POUCH
3	MAIN WING 2X
4	FUSELAGE
5	KEVLAR CHORD SUSPENDS FUSELAGE
6	KEVLAR CHORD SUSPENDS MAIN WINGS 4X
7	CRATE SKINNED WITH RIP-STOP NYLON
8	CRATE FRAMED WITH CARBON ROD

DETAIL A
SCALE 0.500





NOTE:
ALL DIMENSIONS IN INCHES

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DOCUMENT TITLE: CASE DESIGN	
DRAWN BY: JOHNNY CHANDLER	SIZE: B
APPROVAL DATE: 02/24/2010	REPORT TITLE: DETAILED DRAWING
REV #: A	CHIEF ENGINEER: ERIC PROVO
SCALE: 0.170	SHEET 5 OF 5

6.0 MANUFACTURING PLAN & PROCESSES

6.1 Investigation & Selection of Major Components & Assemblies

Looking into several different methods, the team selected the processes that would provide the desired characteristics for that particular part of the design. The manufacturing processes used are described below.

6.1.1 Fuselage

Three different methods were looked into for creating the fuselage. They are described below.

- **Balsa Build Up** – create a fuselage framework and then shape the fuselage skin around it
- **Mold Method** – create foam model of the fuselage, and mold gypsum around it. This gypsum mold is used to create a fiberglass-balsa composite fuselage
- **Lost Foam Core** – similar to foam core method used for wings, except the internal foam is removed for component mounting

The mold method was chosen because of its ease of use and success in past years. This year's plane was made with a top-bottom mold so that components could be placed in the bottom part of the fuselage. These components could then be adjusted to the correct placement, then the top of the fuselage could be placed on top, sealing everything in. The other methods did not allow for this while keeping the fuselage strong.

6.1.2 Wing and Tail

In order to build the wing and tail, two methods were looked into.

- **Foam Core –** the cross section is cut out of foam and covered with balsa or composite
- **Composite/Balsa Buildup –** lay down the spars and glue the ribs and internal components in before covering with Monokote

Because of weight considerations, the group chose the composite/balsa buildup. After Aerodynamics determined the airfoil used on the wings, the structures wing group decided to begin manufacturing a multitude of wings with various attributes that could be tested independently as well as collectively. The changing configurations included modifications in rib spacing, rib fiber-glassing, spar configuration and wing attachment system. The first decision the group made was to build everything as light as possible and only increase weight and strength when the current configuration was determined to fail. This method of design and testing will ensure the lightest wings possible are used to perform the required tasks.

The components of the wing were constructed using the materials listed in Table 26.

Component	Material
Spars	3/16" x 3/16" spruce
D-tube	1/16" balsa
Leading Edge	1/4" x 1/4" balsa
Rib Caps	1/16" x 1/8" balsa
Shear Web	1/16" balsa
Ribs	1/16" balsa
Skin	Monokote

Table 26: Materials List for Wing

In order to accurately construct every part needed for the wing, an inventory list with specific directions to make each part was created and used throughout the construction of every wing. A basic kit containing every part in the inventory list was used to minimize time making parts once construction had started. To construct the ribs, the individual parts were printed out from CAD and hand cut out of 1/16" balsa. To maintain proper spacing of the ribs, a CAD drawing of the wing assembly was printed out to scale and the wing was directly constructed on a sheet of wax paper above the drawing. Spruce spars covered in carbon tow were laid through each rib after having been properly spaced and CA'ed in place. Glassed shear webs and leading edge braces were applied to the spaces between ribs. Trailing edge, partial ribs, rib caps, and the D-channel were adhered to the ribs in their respective order. Monokote was then applied over the entire structure.

Each wing side's male C-channel was inserted and secured to the fuselage via friction through the female C-channel.

6.1.3 Landing Gear

Using a composite layup construction method, a bow gear and mushroom nose gear were created from high density foam molds. These molds were laid up with balsa, carbon fiber, and Kevlar, creating the necessary rigidity and yield strength properties for each gear.

6.1.4 Payload Grid

With balsa-ply and carbon fiber, the grid was formed using the composite layup method and put together with epoxy. The ripstop was sewn and placed over the grid and secured with thread. This system was then laid into the fuselage. The ends of the ripstop were secured to the fore and aft bulkheads with short hooks.

6.1.5 Case

With carbon fiber rods to act as the frame, ripstop was sewn and tightly secured to the rods to act as the floor, walls, and lid. Snaps and hooks were added to the lid, fastening it closed to the rest of the case.

6.2 Milestone Chart

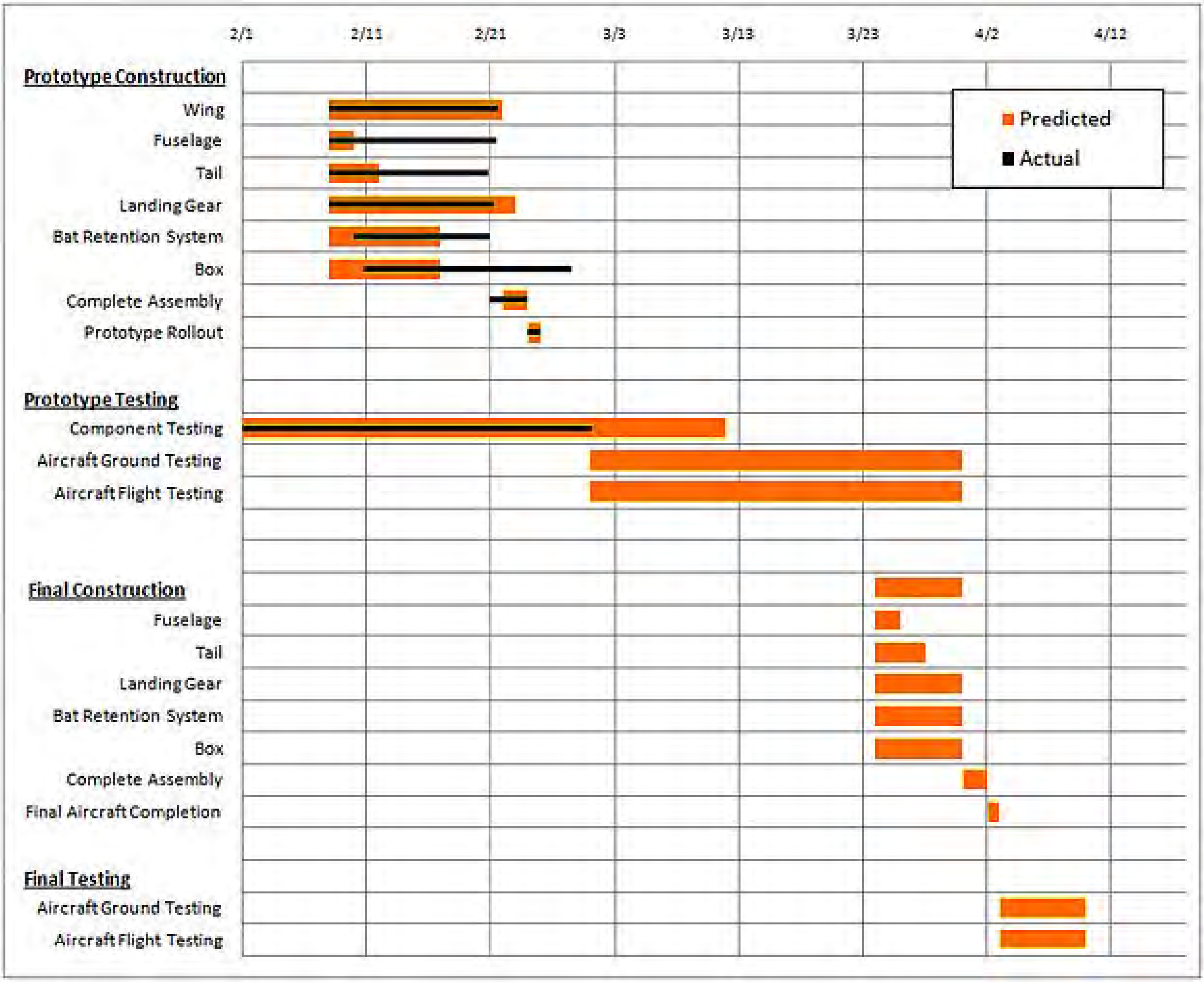


Figure 29: Milestone Chart

7.0 TESTING PLAN

To improve and ensure the validity of component and system designs, several tests were conducted. These tests examined the physical properties of the parts and assemblies, allowing the team to verify material ability.

7.1 Objectives

For every part and system, tests were carried out with a set plan of action and purpose, ensuring the optimum design was being utilized. From the performance results, the team was able to further improve the overall capabilities of the aircraft.

<u>7.1.1 Component</u>

Wing

The objective of the wing testing was to see how the wing acted under bending. The wing was first submitted to an estimated GTOW of 15 lb, then subjected to up to 5 g's to simulate turns. The 5 g loading was one to load the wing until it broke, so the team knew what sort of factor of safety was on the wing.

Landing Gear

The landing gear test was designed to simulate the plane coming in at different angles to make sure the gear did not break on impact. The objective was to load the main gear with estimated GTOW and simulate different landing velocities to see if it could hold up to most landings. The nose gear was also tested, but with only 20 percent GTOW, since this gear only had to support that much.

7.1.2 Propulsion Testing

In order to ensure that the performance of the actual components of the propulsion system matched the performance of the code used to size them, several tests were performed on both the individual components and the propulsion system as a whole before actual flight tests could be attempted. The objectives for these tests are outlined in this section.

Battery Testing

The batteries were tested to ensure earlier estimates in the aircraft's optimizations were accurate and to help the team make battery packs with enough battery capacity to finish each mission. This was done by cycling the batteries several times. Then each was tested by fully charging and discharging the battery with a CBA (Computerized Battery Analyzer), which measures the voltage against time with a constant current draw.

Propeller Testing

The Oklahoma State University wind tunnel setup was used to run propellers of interest with a constant power source and motor at different values of J to output the thrust, torque, and input power. This data was used to create the efficiency curves that were used to choose the optimum propeller for the missions.

Motor Testing

The motor and gearbox were tested using the chosen propeller in the Oklahoma State University wind tunnel with a constant power source. This validated the predicted performance and found any unexpected losses to inform the team what changes were needed.

System Testing

After all of the propulsion system components had been individually tested and all issues removed, the full system was tested in the wind tunnel. The overall efficiency of the system and the full battery life were determined at various conditions, and the cooling of the components was also evaluated. The full system test objective was to see how the components worked together, verifying the optimization done was appropriate for the real world application.

7.1.3 Flight Test

With all of the systems individually tested, the flight test needed to be conducted to ensure all of the systems were able to coexist and work collectively to make the aircraft carry out the missions. Carrying out the flight tests provided the team with the aircraft's overall performance and data to compare to the theoretical values. Starting with simple maneuvers, the team's pilot became familiar with the aircraft's handling and was able to inform the team where improvements could be made. With these flight tests, a plan was laid and carried out, efficiently gathering data for the team to analyze and use to further improve the aircraft's overall design.

7.1.4 Payload Loading

As the total score was greatly affected by the speed of the ground crew, tryouts were conducted to select the best individuals to participate in mission 2. Once the individuals were chosen, practices were carried out to find the best methods and to allow the crew to develop a sound routine.

7.2 Master Test Schedule

Test	Objective	Start Date	End Date
Materials	Verify adequate strength needed	1/25	2/8
Propeller	Determine actual performance	1/26	1/28
Motor	Determine actual performance	2/19	2/23
Battery	Determine actual performance	2/3	2/18
Wing	Ensure wing strength	2/8	2/18
Tail	Test aerodynamic loads	2/8	2/17
Landing Gear	Verify landing gear durability	2/8	2/17
Bat Retention System	Check strength and reliability	1/25	2/17
Box	Make lightest weight box possible	2/8	3/8
Assembling the Plane in 5 Minutes	Decrease the time to assemble	2/24	4/9
Servo Testing	Determine if all controls work	2/24	4/9
Timed Loading of Softballs	Decrease the time to load	2/24	4/9
Prototype Testing	Check flight characteristics	3/7	3/28
Final Aircraft Testing	Check flight characteristics	3/28	4/9

Figure 30: Master Test Schedule

7.3 Flight Test Check List

Unless otherwise stated, all test flight objectives included a takeoff within 100 ft and a successful landing. Successful landings were defined as controlled landings with no damage to the aircraft. For the softball test mission flights, the payload started with 6 balls, and then progressed to 10 one ball at a time after each successful test. For every flight test, Propulsion measured the voltage remaining in the batteries to see how much the aircraft used during that test and to make sure the aircraft had enough power for the next flight.

FLIGHT TEST PLAN			
Flight Number	Special Flight Designation	Payload	Objectives
1	Maiden Flight	Empty	Successful Takeoff, no matter the distance Fly straight and level for at least 200'
2	Ferry Mission	Empty	1-lap, DBF Mission Profile (no max performance turns)
3		Empty	3-laps, DBF Mission Profile (no max performance turns)
4	Softball Mission	Softballs	1-lap, DBF Mission Profile (no max performance turns)
5		Softballs	3-laps, DBF Mission Profile (no max performance turns)
6	Bat Misson	Bats	Fly straight and level for 100'
7		Bats	1-lap, DBF Mission Profile (no max performance turns)
8		Bats	3-laps, DBF Mission Profile (no max performance turns)
9	Stall Test #1	Empty	Attempt 4 max performance turns Did aircraft stall? } If no: keep elevator deflection } If yes: decrease elevator deflection
10	Stall Test #2	Bats	Attempt 4 max performance turns Did aircraft stall? } If no: keep elevator deflection } If yes: decrease elevator deflection
11	Time Trial #1	Empty	3-laps, DBF Mission Profile (max performance turns)
12	Time Trial #2	Bats	3-laps, DBF Mission Profile (max performance turns)

Figure 31: Flight Test Plan

Time Trial flights were repeated until the pilot was confident that he could not improve the times. The pilot was also asked the following questions:

1.) On a scale of 1 to 10 (10 being the worst) were you satisfied with control response?
2.) On what axis was the control response the least satisfactory?
    a. Do you feel that this needs to be addressed?
    b. Would you prefer greater control surface area or more deflection of the control surface?
3.) How flyable was the aircraft? Did you feel as if you were always fighting some adverse moment or force to keep it flying, and if so—what is it?
4.) What improvements would you suggest that would make the aircraft easier for you to fly?

8.0 PERFORMANCE RESULTS

Once the tests were completed, the data was quickly compiled and analyzed. Designs were adjusted and changed where improvements needed to be made. The following section reveals the results of these tests and explains the changes that came about because of them.

8.1 Subsystems

<u>8.1.1 Component</u>

Wing

To demonstrate the wing's ability to withstand the loads experienced while maneuvering at maximum gross weight, the wing was subjected to a cantilever bending test to determine the bending moments and stress applied to the spars.

First, prototype wings without the C-channel joiner were tested using the three point bending test, but they did not provide accurate results for the team's final design. The next test wing had the C-channel built into it and a half-span of 31 inches. By testing it while cantilevered (using the male C-channel extension as the connection point) the team found that the wing failed at an estimated GTOW of 15 pounds while at 5 g's, which was much larger than the predicted GTOW at 2.5 g's as specified in section 5.3.

Wing Type	Test Loading
No C-Channel	Failed
C-Channel	Passed GTOW at 2.5 g's, Failed at 5 g's
Structural Requirement	GTOW at 2.5 g's

Table 27: Wing Test Data

Landing Gear

The main landing gear was bolted to a rigid testing board loaded with the approximate weight of the fully loaded aircraft (12 lbs). Then the testing rig was held level and dropped from various heights equivalent to impacts of different landing velocities using an approach angle of 10 degrees (3 degrees is usual). The gear was also tested for impacts slightly off-axis. The nose gear testing apparatus consisted of a board horizontally oriented to attach the landing gear to. The board, carrying 20% of the airplanes weight, was aligned with a vertical guide rail. The board and gear were then dropped from various heights.

Multiple tests were performed for both the main gear and the nose gear. Table 28 shows the tests run on the various types of landing gear, and how each design failed. The second 14 inch bow gear withstood three drops at 9 inches before failing in tension on the lower surface at the fuselage side. The mushroom gear survived, undamaged after 3 drops from 9 inches. This was what the team predicted for both landing gears in terms of survivability, which far exceeded the normal requirements for landing gear.

	**3” Drop (landing velocity 20 ft/s)**	**5” Drop (landing velocity 30 ft/s)**	**9” Drop (landing velocity 40 ft/s)**
14 in Bow Gear #1	Structurally sound	Structurally sound	Cracked during 2nd drop, failure at fuselage side during 4th drop
14 in Bow Gear #2	Structurally sound	Failure at the fuselage side	
14 in Bow Gear #3	Structurally sound	Structurally sound	Structurally sound after several drops with/without bats
16 in Bow Gear #1	Structurally sound	Structurally sound	Failure at fuselage side during 2nd drop
Mushroom Gear	Structurally sound	Structurally sound	Structurally sound after 3 drops

Table 28: Landing Gear Test

8.1.2 Propulsion

Batteries

The Elite 1500 batteries were tested to determine individual battery capacity. Most of the higher performing batteries held a charge at or slightly above 1200 milliamp hours. A survey of several battery tests is shown below in Figure 32.

The batteries found to have the highest capacity were made into the battery pack that would power the motor. The battery pack was subsequently tested several times with the computerized battery analyzer to ensure that it was consistently holding an appropriate amount of charge. As shown in Figure 33, the pack consistently carried at or around 1300 milliamp hours. This was considerably more capacity than what was originally predicted.

This higher than expected capacity bodes well for actual flight performance, where losses in capacity occur due to high current draw. Even at the estimate of a 15% loss in capacity during actual flight, this still puts the battery pack capacity well higher than what was first expected.

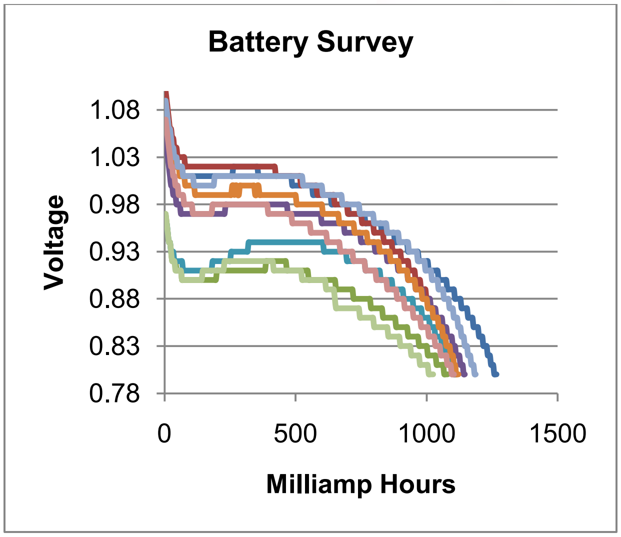


Figure 32: Battery Survey

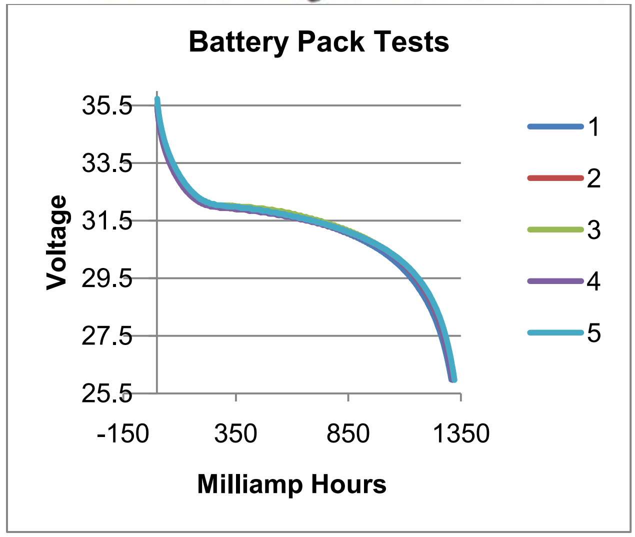


Figure 33: Battery Pack Tests

Propeller

In order to choose a propeller, in house wind tunnel testing was done on a variety of propellers. Figure 38 shows the plotted performance characteristics of some of the main propellers that were tested.

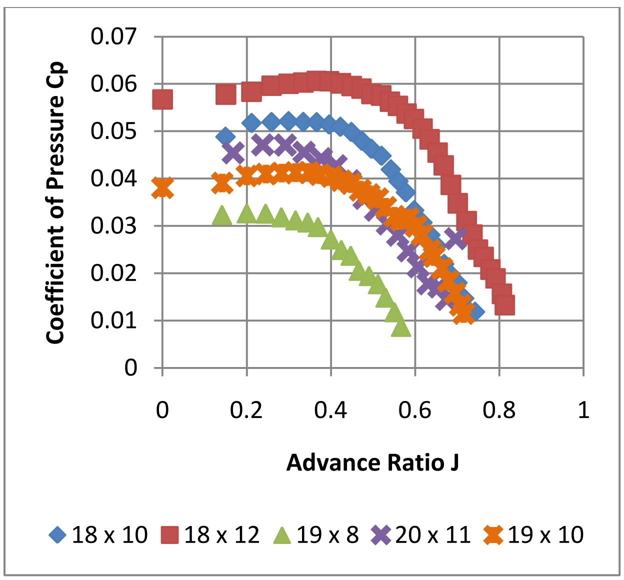


Figure 34: Cp vs J

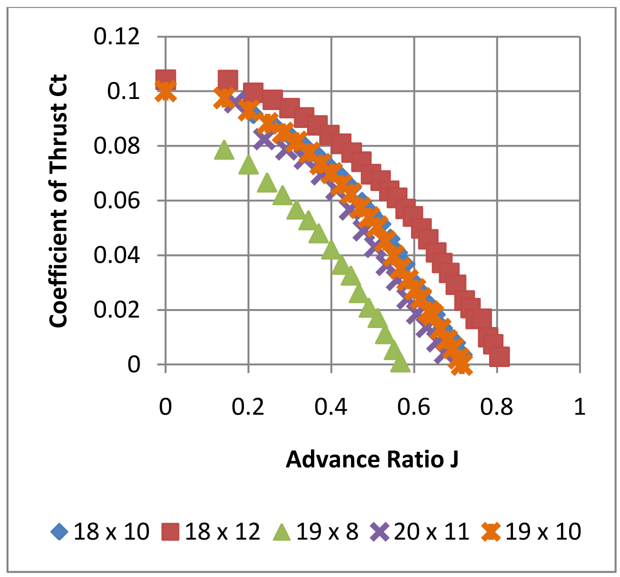


Figure 35: Ct vs J

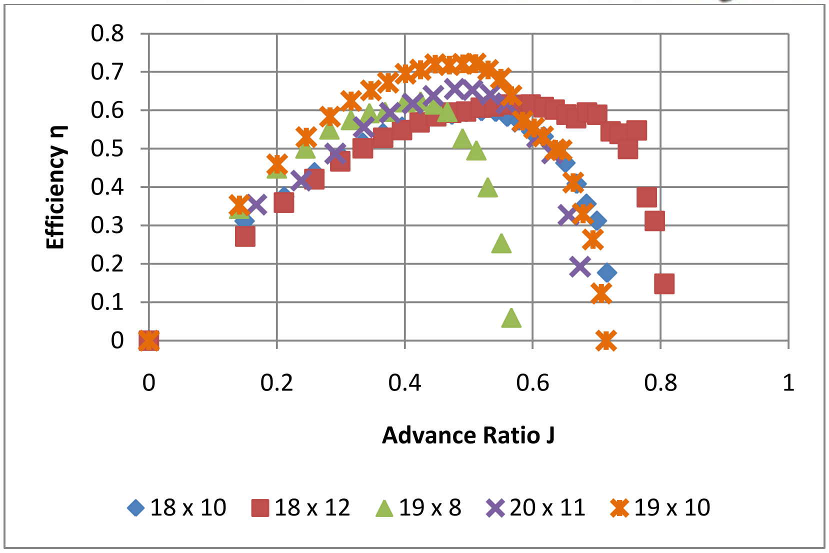


Figure 36: Efficiency vs. Advance Ratio

Based on the optimization program, the plane would operate on a J range from .35 to .63. However, if the best propeller, the 19x10, was chosen the bow gear would have to be taller and therefore heavier to accommodate the large diameter. So the next best propeller for this plane was chosen, the 18x10. The 18x10 peaks at 0.5 and has good efficiency across the advance ratio range required for our aircraft.

Propulsion System

Static motor tests for all three missions were performed on a test rig using the propeller and batteries that would be used on the aircraft. The duration of the tests were based on the mission time and power estimates from the MPOP program and each mission was simulated from beginning to end with appropriate changes in thrust for takeoff, turns, cruise, and landing. Each simulated mission was successfully completed with battery capacity to spare and with the motor outputting the required thrusts at current draws only slightly higher than originally predicted.

8.2 Flight Testing

After the subsystems were tested, the plane was put together and made flight ready. The plane was weighed before flight to compare actual weight to predicted weight. The actual weight of the airplane was 5.3 lbs, much lower than the predicted 6.35 lbs. Using the flight test plan, outlined in Figure 31, the test objectives were met in many wind conditions with careful weight progression and execution.

Mission	Average Actual Flight Time (sec)	Predicted Flight Time (sec)	Percent Difference
1	69	79.9	14.60 %
2 (6 balls)	130.5	131.1	0.46 %
2 (10 balls)	136	137.7	1.24 %
3	142.5	144.9	1.71 %

Table 29: Actual Flight Times vs. Predicted

Mission 1 was the first test mission flown due to the lack of a payload, which lowered the risk for flight mishaps. Takeoff distance was no issue as the plane took off in 10 feet. After doing those flight tests, the completion time ranged from 63 to 75 seconds.

Mission 2 was the second flown test mission. The team began by loading 6 balls and then increasing to the full ten. The team told the pilot to ensure completion instead of speed as flight time does not matter, and was able to complete the mission runs. The flight time for six balls ranged from 120 to 141 seconds, while the flight time with ten balls ranged from 130 to 142 seconds.

Mission 3 was the last mission to be completed. Unlike mission 2, time was crucial to this mission so the team had the pilot push the aircraft harder to obtain faster flights. The test flight times ranged from 140 to 146 seconds, which was a decrease in estimated flight time due to the lower weight of the aircraft. The takeoff distance was still below 100 feet.

The actual times were better than the predicted for a number of reasons. When the team was making their predictions and calculating estimates (through codes and other numerical methods), they were being conservative, at times assuming worst case scenario. This resulted in the predictions being a little higher than the actual results.

Figure 37: Prototype Coming in for Landing

2009/2010

Cessna/Raytheon AIAA Design/Build/Fly Competition

OSU Black

Oklahoma State University

TABLE OF CONTENTS

1.0 EXECUTIVE SUMMARY

This report documents the efforts of the Oklahoma State University (OSU) Black Team to produce an unmanned aerial vehicle system that will be competitive in the 2009/2010 AIAA/Cessna/Raytheon student Design/Build/Fly competition. All aspects of the design maximize the scoring function. The competition score is made up of both a written report and a flight score.

Flight score[1] is determined by the system performance in three different missions: a two-lap ferry flight, a timed loading and three-lap flight carrying softballs internally, and a three-lap flight where up to five bats are carried externally. Important parameters for score are system weight (aircraft, case, transmitter, and assembly tools), aircraft speed, and softball loading time. Important mission constraints also include a five minute aircraft assembly time, 100-foot takeoff distance, a maximum box size of 2 x 2 x 4 feet, and a maximum of four pounds batteries for the propulsion system. The score function is most sensitive to softball loading time, thus design began by considering this requirement. All the documented design decisions minimize loading time for Mission 2, provide low drag to promote high aircraft speed in Mission 1 and 3, and encourage high structural efficiency.

The optimized aircraft integrates a two by five payload grid structure into a low-wing monoplane tail-dragger configuration for the best possible loading time, lightest airframe, and fastest flight speed. This, combined with a lightweight case and no required assembly tools, attains the optimum flight score.

Analysis showed that turning flight comprises a significant portion of the mission course. The aircraft structure was designed for this and is capable of withstanding high g-loads for rapid turns. The wing uses a MH 114 airfoil with a span of 78 inches and an area of 787 square inches. The large span, required for a high rate of turn, requires a two-piece detachable design that is manufactured by conventional built-up construction for low weight and load carrying ability. Primary wing structure consists of four spruce/carbon composite spar caps. The fuselage contains an “egg-crate” structural pattern for fast loading time and for restraining the internal payloads while also serving as an integral part of the fuselage structure. The fuselage is manufactured by molding a lightweight fiberglass with a balsa core. The carbon fiber main landing gear serves as a hard point for mounting the external bat payloads. Because the main gear, by necessity, is the strongest part of the aircraft, it is ideal to serve double duty as a mounting location for the bats. The tail-dragger configuration provides a low-drag solution and is well-suited to meet the short takeoff requirement. The propulsion system consists of an 18x10 propeller driven by a Neu 1110/3Y motor and 6.7:1 gearbox and powered by 26 Elite 1500 battery cells.

Predicted aircraft performance capabilities for Missions 1 / 2 / 3 are as follows: takeoff distances of 20 / 58 / 96 feet, wing loading of 1.3 / 2.0 / 2.4, thrust-to-weight ratios of 0.9 / 0.56 / 0.47, and stall speeds of 17 / 22 / 24 mph. Predicted mission performance capabilities for Missions 1 / 3 are as follows: cruise speed of 47 / 31 mph and flight times of 62 / 118 seconds. Total system weight for Mission 1 and 2 is 9.33 pounds, loading time for Mission 2 is 7 seconds, and Mission 3 payload is 5 bats.

2.0 MANAGEMENT SUMMARY

Given the limited time and resources available to the team, an efficient management of personnel and work must be coordinated and planned. As such, the team composition and schedule is presented below.

2.1 Team Organization

The 2010 OSU Black Team consists of 28 undergraduate mechanical and aerospace engineering students ranging from freshmen to seniors. In order to provide effective distribution of effort among the many tasks needed to compete, the team is broken into four technical groups as shown below in Figure 2.1.

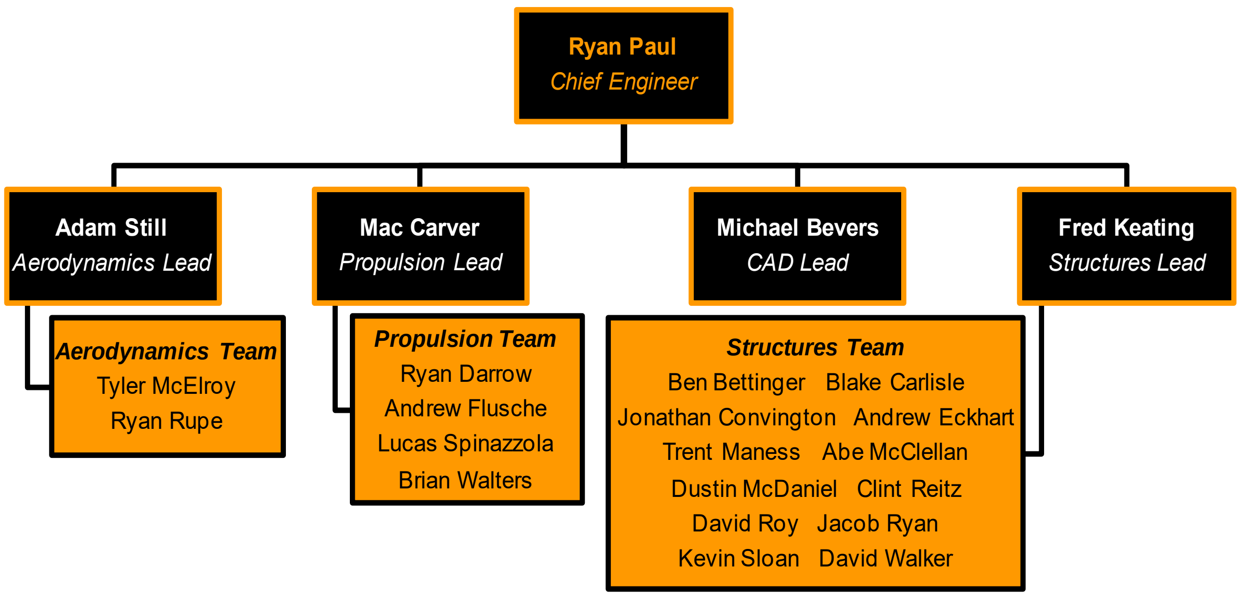


Figure 2.1 — Team Organizational Hierarchy

The team is headed by a chief engineer—supported by four technical leads—who assumed responsibility for the management and coordination of the individual groups. Primary responsibilities for the groups are as follows:

- **Aerodynamics**: Analyzes and selects configurations to optimize the mission score, ensures adequate stability and control, and predictions of flight performance parameters
- **Structures**: Tests and selects materials for each component of the system (case and aircraft), plans the manufacturing processes, and fabricates and integrates the components and subsystems into the final system solution.
- **CAD**: Creates and compiles drawings and models of parts, components, sub-systems, and assemblies for concept visualization, design optimization, and construction plans.
- **Propulsion**: Analyzes power requirements for various mission profiles, designs and optimizes propulsion and control systems, and researches, selects, and tests the propulsion system components.

2.2 Milestone Chart

This project is undertaken in a single, sixteen-week semester at OSU. Planning and the effective use of time are thus critical, as other competitive teams are usually in the flight test phase when our team is just beginning the conceptual phase of the project. An overall schedule and milestone chart is given in Figure 2.2.

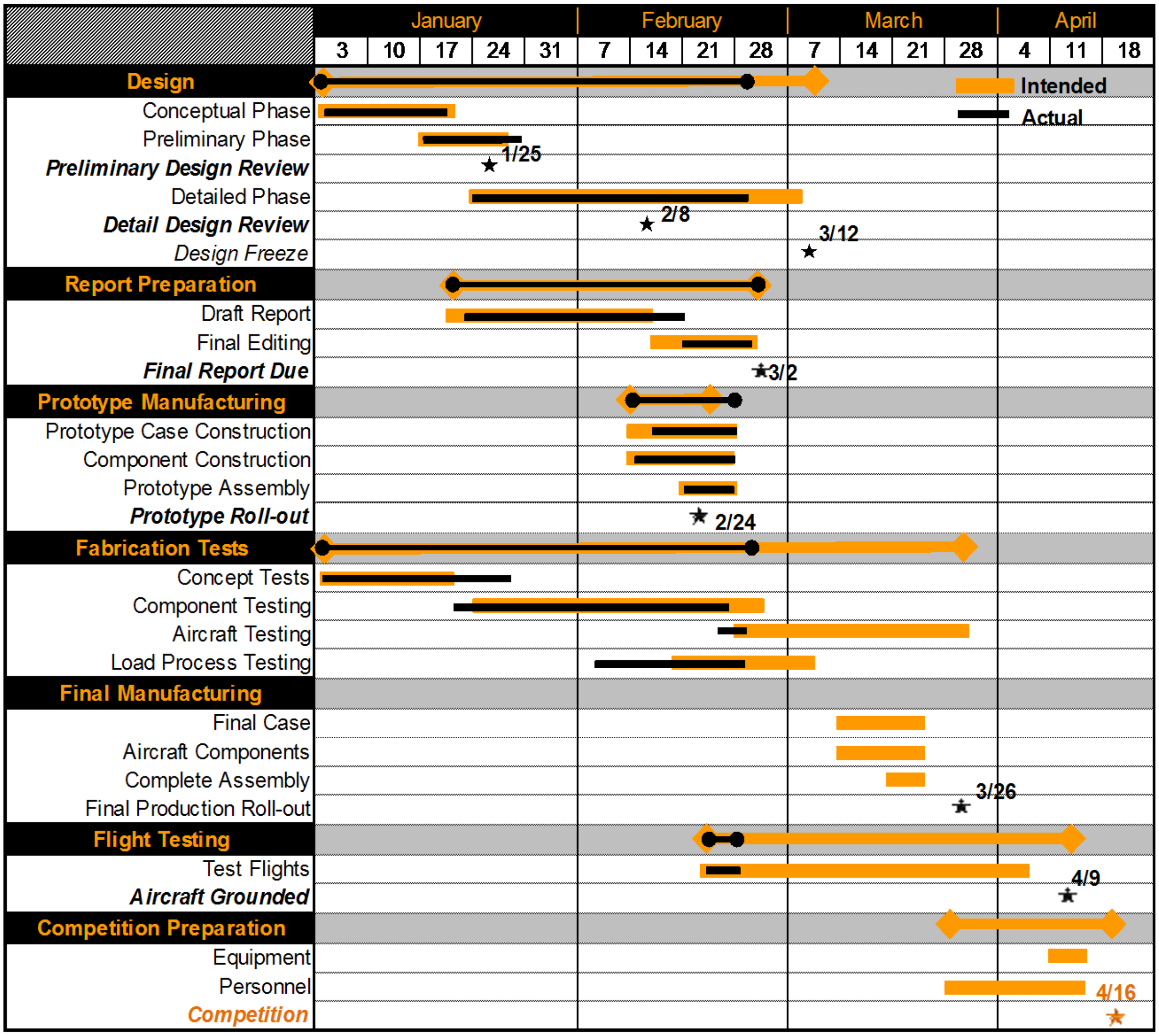


Figure 2.2 — Team Schedule and Milestone Chart

3.0 CONCEPTUAL DESIGN

Conceptual design begins with a study of the contest rules. The rules allow the team to identify design constraints and mission requirements. The key system requirements are then defined as figures of merit (FOM) and used to weigh different concepts against each other.

3.1 Mission Requirements

Total contest score is determined by both the written report and the flight score, as shown below.

$$\text{Total Score} = \text{Flight Score} * \text{Report Score} \qquad \textbf{(Eq. 3.1)}$$

The flight score is the sum of the score achieved in each of the three missions. The missions must be flown in order, and no score is given for partially completed missions. This ensures that a competitive design is optimized for each mission at the contest. General specifications and mission requirements are outlined below:

- The aircraft, transmitter, and all necessary tools must fit in a storage case with maximum external dimensions of 2 x 2 x 4 feet.
- The aircraft must be capable of carrying six to ten 11- or 12-inch circumference ASA Fast Pitch Girl's Softballs. The softballs must be stored internal to the aircraft and must remain secured.
- Up to five bats must be carried external to the aircraft. The bats will be 2-inch nominal diameter, not tapered, between 26 and 30 inches in length, and weigh between 16 and 20 ounces. The contest supplied bats will have a 3/16-inch hole at the center of gravity that each team must use to secure the bats. Additional restraints must be used to secure the bats in all six degrees of freedom.
- A maximum takeoff distance of 100 feet.
- A maximum battery weight of 4 pounds with the current draw limited to 40 amperes via a fuse.

3.2.1 Mission One Requirements

Mission 1 is a two-lap ferry flight with no payload. The score (M_1) is dependent on the system weight and the flight time as shown below in Equation 3.2.

$$M_1 = \frac{t_{ref}}{t_{team}} * \frac{W_{ref}}{W_{team}} * 50 \qquad \textbf{(Eq. 3.2)}$$

Flight time begins when the throttle is advanced for takeoff, and ends when the aircraft passes over the finish line after two laps. A successful landing must follow for the flight to be scored. The maximum score for Mission 1 is 50 points. Note that system weight (W_{team}) is normalized with respect to the lightest weight recorded for any team that completes Mission 1 (W_{ref}). The weight recorded for each team is not simply the weight at the start of Mission 1, but the heaviest weight recorded upon entering the staging area for any of the three missions. Flight time (t_{team}) is normalized by the fastest time for any which successfully completes Mission 1 (t_{ref}).

3.2.2 Mission Two Requirements

Mission 2 requires the aircraft to carry six to ten ASA Girl's Fast Pitch 11-inch or 12-inch circumference softballs (3.5-inch or 3.8-inch diameter) for three laps. The number of softballs to be carried is randomly determined by rolling two die before beginning the mission. The payload must be carried internally in a grid pattern without moving. Based on available specifications of the balls, the maximum payload weight for this mission is 4.01 pounds. The score function for Mission 2 (M_2) is dependent on the time required to load the balls in the aircraft (t_{team}) and the system weight (W_{team}), represented in Equation 3.3.

$$M_2 = \frac{t_{ref}}{t_{team}} * \frac{W_{ref}}{W_{team}} * 100 \qquad \textbf{(Eq. 3.3)}$$

The maximum score for Mission 2 is 100 points. As with Mission 1, the individual team weight and time are normalized against the lightest competition system weight (W_{ref}) and fastest loading time (t_{ref}). Mission 2 time begins with no more than three team members located 10 to 20 feet away from their closed aircraft. The appropriate number of softballs must be located in a plastic bag inside the team's closed case. Once time starts, the team must get the softballs to their aircraft and secure them inside as quickly as possible. Time stops only once all loading crew members return to the start position with the plastic bag located inside the closed case and the crew has announced "Stop!" After the aircraft is loaded, the flight portion must be completed successfully. No additional team interaction is allowed except for installing the safety fuse and positioning the aircraft on the runway.

3.2.3 Mission Three Requirements

Mission 3 requires teams to fly three laps with an external payload. The external payload is up to five contest-supplied bats; each team determines how many bats to fly. Mission 3 score (M_3) depends on the number of bats ($N_{bats,team}$) carried as well as flight time (t_{team}), shown below in Equation 3.4.

$$M_3 = \frac{t_{ref}}{t_{team}} * \frac{N_{bats,team}}{N_{bats,ref}} * 100 \qquad \textbf{(Eq. 3.4)}$$

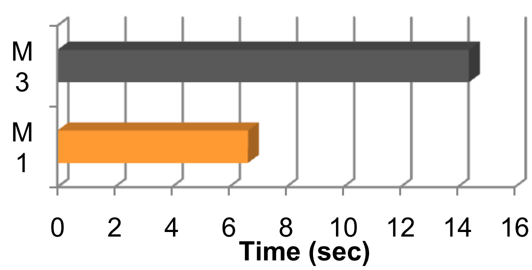
The time for Mission 3 begins when the throttle is advanced for takeoff and ends after the aircraft passes over the finish line in-flight after three laps. A successful landing must follow in order for a flight score to be recorded. The individual team flight time is normalized against the contest best (t_{ref}) and well as the highest number of bats carried successfully by any team ($N_{bats,ref}$).The maximum score for Mission 3 is 100 points.

3.2 Translation of Mission Requirements into Design Requirements

A score sensitivity study was conducted to evaluate the effect of mission scoring parameters on the overall flight score. This analysis was undertaken using assumptions based on historical data from past DBF competitions. The assumptions are as follows, shown in Figure 3.1a.

Total Course Distance	3500 ft
Mission 1 Ave. Lap Time	30s
Mission 3 Ave. Lap Time	45 s
Mission 3 Ave. Lap Speed	80 ft/s
Number of Bats	5
Best Loading Time	8 s

a)



b)

Figure 3.1 — (a) Benchmark Data and Assumed Parameters for Score Analysis; (b) Flight Time Equivalent to 1 Second of Loading Time per Unit Score

The Mission 2 score is based on rapid payload loading, whereas Missions 1 and 3 scores are dependent on flight time. Using the assumptions above, the sensitivity of the mission score to loading time and flight time were determined. Figure 3.1b shows that in order to have the same effect on score as only one second of loading time, each lap in Mission 1 must be flown about 6 seconds faster than the average ferry flight speed, and each lap in Mission 3 must be flown about 14 seconds faster than the average loaded flight speed. Thus, the overall contest score is more sensitive to the loading time than either of the flight times.

The scoring function for Mission 3 is weighted with respect to the team who carries the most bats. A score sensitivity analysis of the Mission 3 scoring function was performed to determine the optimal number of bats to carry. Based on the assumptions listed in Figure 3.1b, the Mission 3 scoring function reduces to Equation 3.5.

$$M_3 = \frac{130}{t} * \frac{N}{5} * 100 = 2600 * \frac{N}{t} \qquad \textbf{(Eq. 3.5)}$$

The number of bats, N, is a discrete parameter and the flight time, t, is a continuous variable**.** The graphical results are represented in Figure 3.2.

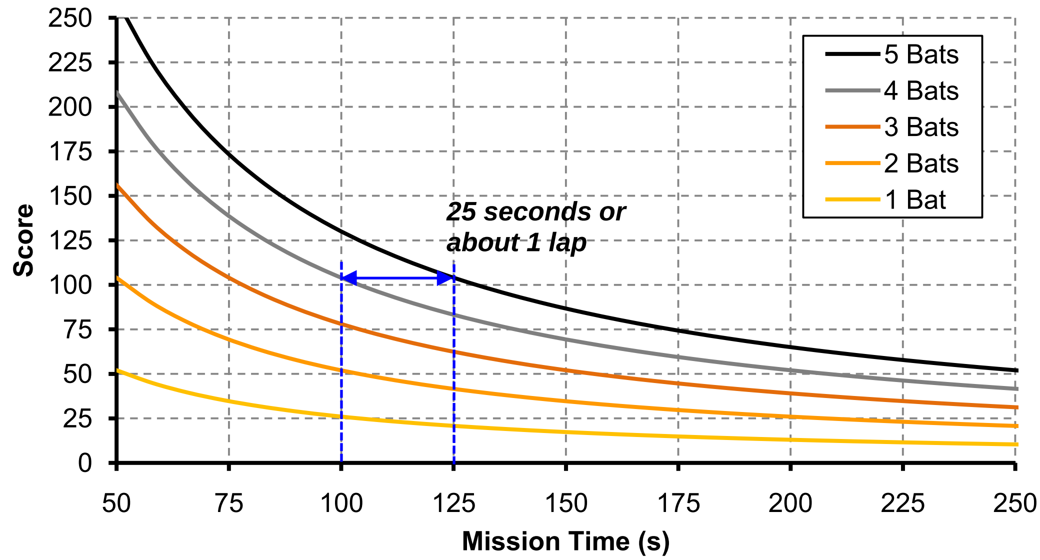


Figure 3.2 — Flight Score as a Function of Mission Flight Time for Various Bat Loads

Figure 3.2 indicates that one bat is worth approximately 25 seconds of flight time, which is equivalent to about 70 percent of one lap. To be competitive, five bats should be carried.

With the score analysis completed, it was concluded that the best design solution for the contest must be developed with the following objectives in mind:

- **Mission 1:** Minimize the aircraft drag when in the empty configuration to fly the laps as fast as possible.
- **Mission 2:** Optimize the aircraft, case, and loading scheme to minimize the loading time.
- **Mission 3:** Choose an aircraft configuration that will allow the carriage of all five bats to maximize the flight score.

- **Overall**: Since weight factors into two of the three mission scores, a lightweight aircraft and case solution should be developed. The aircraft and case must be structurally efficient and have high strength-to-weight ratios while maintaining a configuration that can be loaded rapidly.

3.3 Review of Considered Solution Concepts and Configurations

Ideas for aircraft configurations were placed into a morphological chart (Table 3.3).

Configuration	Alternative Solutions				
Wing	Monoplane	Biplane	Tandem	Blended	Elliptical
Tail	Conventional	Cruciform	T-Tail	H-Tail	V-Tail
Motor Placement	Pusher	Multi-Engine	Tractor	Pod	Dual Inline
Landing Gear	Tricycle	Bicycle	Tail-Dragger	Tip-Dragger	
Box	Dual Lid	Top Opening	Side Opening	Midline Opening	
Ball Loading	Inserts	Hand Load	Speed Loader	Carry Tray	
Bat Configuration	Wing Mount	Fuselage Mount	Landing Gear Mount	Outrigger Mount	

Table 3.3 — Morphological Chart of Proposed Configuration Alternatives

The morphological chart allowed for easy identification, organization, and comparison of alternatives. Some ideas were quickly discounted on the premise that they would not fulfill mission requirements. The candidate solutions for each configuration were then compared qualitatively against each other (with the exception of the internal payload loading method; this was gauged based on testing) for various weighted figures of merit (FOMs) that were determined to be critical to each respective sub-system's performance. For each configuration and FOM, a score of +1, 0 or -1 was given to indicate, respectively, superior performance, adequate performance, or inferior performance.

3.3.1 Internal Payload Loading

Contest score is very sensitive to internal payload loading time. Thus, it was decided to make this the starting point of the design process.

<u>Alternatives</u>

- **Hand-Placed**: Softballs are carried to the aircraft by the ground crew and manually placed into the payload area.
- **Mechanically Assisted Placement**: The payload is loaded by a mechanism integrated in the aircraft capable of being stowed. The crew feeds the balls into the mechanism.
- **Installable Insert**: Softballs are loaded into an insert immediately after they are removed from the plastic bag inside the aircraft case. The insert is carried over to the aircraft and loaded.

Due to the importance placed on this aspect of the overall design, selection of a solution was determined by physically testing the alternative concepts with mock-ups that closely simulated the above configurations. The results of the tests are shown in Figure 3.4:

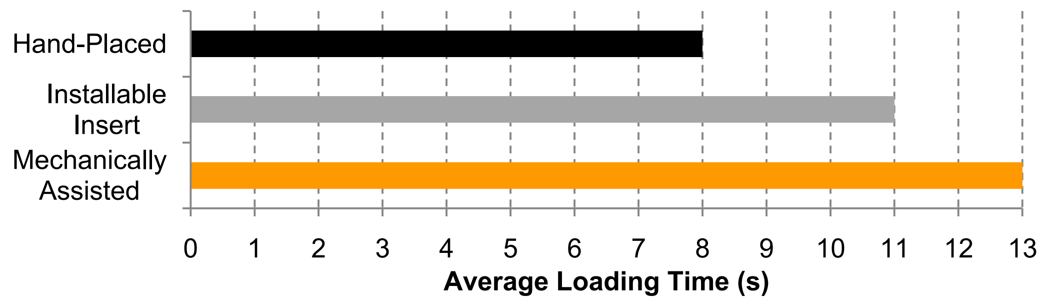


Figure 3.4 — Results of Timed Trials for Loading Method Concepts

The fastest loading times occurred with the direct hand-placed payload loading. Hand placement lacked a "bottleneck" in the overall loading process since multiple calls could be loaded by several people at once. Additionally, hand placement allows for the fewest steps possible among the other alternatives and lacks the need for added equipment and extraneous system weight. Therefore, hand placement was deemed the loading scheme of choice.

3.3.2 Internal Payload Configuration

The ideal solution for the internal payload configuration would be lightweight, have a low drag profile, and be easily integrated into the fuselage structure.

Alternatives

- **Single Layer, 2-wide:** Softballs are arranged in a single vertical layer with 5 rows of 2 balls each.
- **Single Layer, 3-wide:** Payload is arranged in a single vertical layer with a combinations of 2 rows of 3 balls each and 2 rows of 2 balls each
- **Single Layer, 4-wide:** Arrangement similar to a "billiard rack" setup, with 4 rows of diminishing size.
- **Vertical Stack:** Softballs are arranged in a single row, stacked two high in the vertical direction.

Figures of Merit

- **System Weight:** Weight is a factor in Mission 1 and 2. As such, any payload configuration that can also serve as part of the load-bearing structure can help reduce system weight.
- **Loading Time:** Loading speed is critical to the Mission 2 score and is therefore weighed heavily among the other FOMs.
- **Drag:** Mission 1 is an empty ferry flight. The ferry flight must be flown as quickly as possible, so a lower drag profile can increase performance and, therefore, flight time. A wider payload configuration will result in a larger drag profile.
- **CG Balancing:** Increasing the ease of properly balancing the internal payload results in a smaller shift of the center of gravity thereby decreasing the need for a large (and heavier) tail.

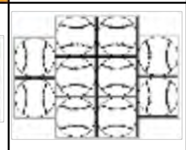
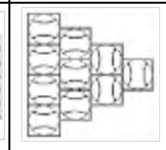
Configuration		Single Layer, 2-wide	Single Layer, 3-wide	Single Layer, 4-wide	Vertical Stack
					
FOM	Weight Factor	Scoring			
System Weight	40	0	0	0	-1
Loading Time	40	1	1	1	-1
Drag	10	1	-1	-1	0
CG Balancing	10	0	1	-1	0
Totals	100	50	40	20	-80

Table 3.5 — Weighted Decision Matrix for Fuselage/Internal Payload Configuration

Table 3.5 shows that the single layer, two-wide arrangement was selected due primarily to the best drag profile it offers as well as being inherently easy to allow integration into the fuselage structure.

Two concepts were considered in which to implement the single layer, two-wide arrangement.

- **Orthogonal:** Payload restraints form a square grid pattern around the softballs.
- **Diamond:** Payload restraints form a diamond shaped grid around the softballs.

Like the selection for the fuselage/payload arrangement, the selection for the payload restraint configuration was largely driven by integrating the grid pattern into the fuselage structure, thereby minimizing system weight. Based upon this criterion, the diamond pattern was deemed inferior in terms of structural versatility as well as added construction complexity. Therefore, the orthogonal configuration was chosen.

3.3.3 General Aircraft Configuration

With the internal payload configuration determined, the general configuration of the aircraft was selected. The configuration would have to be capable of being designed around the payload without compromising other important considerations, especially system weight and performance.

Alternatives

- **Monoplane:** Conventional aircraft with a simple fuselage and empennage and a single wing. Comparatively easy to design, manufacture, and fly.
- **Biplane:** Essentially a dual-wing aircraft, the effective increase in wing area can give higher payload capacity but at the cost of added weight and drag.
- **Twin Boom:** Features two fuselage sections to facilitate loading from two directions simultaneously.
- **Blended Wing Body:** Modern configuration containing no fuselage section. Theoretically, it is an aerodynamically superior aircraft but has inherent stability and manufacturing challenges.

Figures of Merit

In addition to system weight and drag, the general aircraft configuration selection considered several other FOMs.

- **Takeoff Distance:** A configuration that inherently has improved takeoff performance can help meeting the 100-foot mission requirement.
- **Storage Size:** By decreasing the volume needed to store the plane, the case can be made smaller and lighter, decreasing total system weight.
- **Stability and Control:** A competitive aircraft should ideally be as stable and controllable as possible to ensure consistent completion of the missions and high mission performance.

Configuration		*Monoplane*	*Biplane*	*Twin Boom*	*BWB*
					
FOM	***Weight Factor***	***Scoring***			
System Weight	40	0	-1	-1	0
Drag	20	0	-1	0	1
T/O Distance	20	0	1	-1	0
Storage Size	10	1	0	0	-1
S & C	10	0	0	-1	-1
Totals	**100**	**10**	**-40**	**-70**	**0**

Table 3.6 — Weighted Decision Matrix for General Aircraft Configuration

After considering the above options in Table 3.6, the monoplane configuration was determined to be best suited toward not only easily accommodating the internal payload, but also meeting mission requirements and achieving competitive performance. Additionally, this configuration concentrates all load paths in one place for high structural efficiency, offers lower drag than a biplane or twin boom configuration, and has no components that interfere with loading.

3.3.4 External payload Configuration

Alternatives

Three primary external payload configurations were considered.

- **Wing-mounted Payload**: Payloads would be placed under the wing near the root. Hard points would be incorporated into the wing at mounting locations. A bending moment reduction would be realized in flight.
- **Gear-mounted Clamp**: The payloads would be mounted under the landing gear as close as possible to the centerline of the fuselage. To secure the payload, a clamp would be used
- **Gear-mounted Hard Point**: One bat is mounted on the centerline of the fuselage. The remaining four bats are attached outboard directly onto the main landing gear.

Figures of Merit

As in previous concept selections, system weight and drag were selected as FOMs. In addition to those, manufacturing complexity was considered in order to select a concept that could be constructed relatively quickly to focus manufacturing resources to the primary aircraft structure.

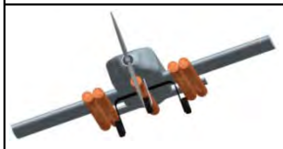
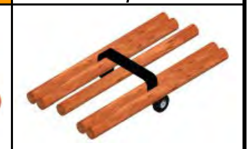
Configuration		Wing-mounted	Gear-mounted, Clamp	Gear-mounted, Hardpoint
				
FOM	**Weight Factor**		**Scoring**	
System Weight	40	-1	1	1
Drag	40	-1	0	-1
Mfg. Complexity	20	0	1	1
Totals	**100**	**-80**	**60**	**20**

Table 3.7 — Weighted Decision Matrix for External Payload Configuration

<u>Configuration</u>

After considering the three configurations, the gear-mounted clamp configuration was chosen (Table 3.7). Since the main gear is already a strong and reinforced structure, it serves as an ideal point to attach the external payload, eliminating the need to reinforce other portions of aircraft upon which to mount hard points. This configuration is the most structurally efficient while being easy to manufacture.

3.3.5 Landing Gear

With the external payload configuration established, the landing gear options were made clear with only two configurations: a tricycle and tail-dragger design.

<u>Figures of Merit</u>

Important FOMs such as system weight and takeoff performance were considered. The landing gear selection also considered two other FOMs.

- **Ground Handling:** How well the aircraft maneuvers on the ground during the takeoff roll can have an effect on takeoff distance itself. Additionally, longer takeoff distance results in an overall longer flight time.
- **Payload Interference:** The external payload should always remain free from contact with the ground during both takeoff roll and rotation. A ground strike not only increases takeoff distance and time, but could also damage the payload or aircraft itself.

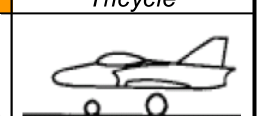
Configuration		Tail-dragger	Tricycle
			
FOM	**Weight Factor**	**Scoring**	
System Weight	40	1	0
Ground Handling	30	-1	1
T/O Performance	20	1	0
Payload Interference	10	0	-1
Totals	**100**	**30**	**20**

Table 3.8 — Weight Decision Matrix for Landing Gear Configuration

3.3.6 Empennage

Many empennage configurations were researched based upon prior competition entries and team experience. Three primary configurations were chosen from several options to best suit the aircraft weight, stability, and case sizing requirements.

<u>Alternatives</u>

- **Conventional**: The conventional tail generally contributes to a lower aircraft weight and provides good performance and control due to being in the prop wash.
- **Cruciform**: A cruciform tail improves control effectiveness during stall by removing the horizontal tail from the wing's "shadow," but loses the prop wash advantages.
- **V-tail:** The V-tail provides a means to provide an equal control volume as the conventional or cruciform tail, but with a lower height which conserves case space and weight.

<u>Figures of Merit</u>

In addition to system weight and manufacturing complexity, two other FOMs were introduced.

- **Height**: Vertical height consumes case volume which contributes to system weight.
- **Control Effectiveness:** By placing the tail directly within the prop wash, tail efficiency is improved, allowing for size and weight reduction and better takeoff performance.

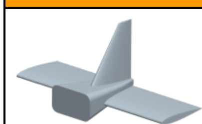
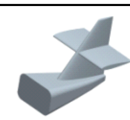
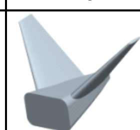
Configuration		*Conventional*	*Cruciform*	*V-Tail*
				
FOM	*Weight Factor*	*Scoring*		
System Weight	40	0	-1	-1
Control Effectiveness	30	0	0	1
Height	20	0	-1	-1
Mfg. Complexity	10	1	1	0
Totals	**100**	**10**	**-50**	**-30**

Table 3.9 — Weighted Decision Matrix for Empennage Configuration

As seen in Table 3.9, the conventional tail was determined to be the best configuration based on comparatively low weight, ease of manufacture, and the benefits of prop wash placement.

3.3.7 Engine Placement

<u>Alternatives</u>

Four alternatives were investigated when deciding the overall propulsion system configuration. These alternatives are detailed below.

- **Single Tractor**: A single propeller and motor is located at the nose of the plane, allowing the propulsion system to operate efficiently in an undisturbed airstream, while also keeping system weight low.

- **Single Pusher**: This single propeller/motor configuration can also keep weight to a minimum. However, placing the propeller aft of the fuselage creates problems with propeller efficiency, tail effectiveness, as well as takeoff performance.
- **Dual Tractor**: Placing a propeller on each wing will increase both aircraft speed and reduce in-flight bending moment on the wing. However, overall system weight increases.
- **Tractor/Pusher Inline**: This configuration will place propellers both forward and aft of the fuselage along the centerline.

Figures of Merit

The important FOMs of system weight and takeoff performance are considered here as well as two others:

- **Thrust:** Propulsion system must efficiently meet or exceed aircraft thrust requirements.
- **Payload Interference**: Propellers cannot interfere with either the placement of external payload or loading of internal payload.

Configuration		*Single Tractor*	*Single Pusher*	*Dual Tractor*	*Tractor/Pusher Inline*
FOM	***Weight Factor***	***Scoring***			
System Weight	40	0	0	-1	-1
T/O Performance	20	0	0	1	1
Thrust	20	1	-1	0	0
Payload Interference	10	0	0	-1	-1
Aircraft Torquing	10	0	0	-1	0
Totals	**100**	**20**	**-20**	**-40**	**-30**

Table 3.10 — Weighted Decision Matrix for Propulsion System Configuration

From Table 3.10, the single tractor was chosen as the most effective propulsion configuration. It is the simplest and most lightweight alternative. It will also offer the highest propeller efficiency and control effectiveness during takeoff.

3.3.8 Case Design

Alternatives

Several alternatives were evaluated in order to select the case that will allow the team to load the aircraft the fastest and be manufactured in the most lightweight manner.

- **Top Opening:** The top face of the box opens to allow access. The opening can be a single or dual hatch design.
- **Side Opening:** One of the side faces opens to reveal the contents.
- **"Treasure Chest":** The top face of this case opens in addition to half of a side face.
- **Triangular:** This configuration features three faces to eliminate some volume and weight.

<u>Figures of Merit</u>

As with other concepts, system weight and loading time were the FOMs. The decision matrix is below in Table 3.11.

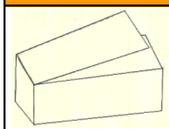
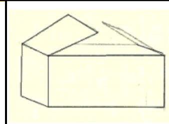
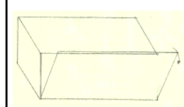
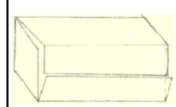
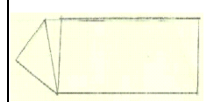
Configuration		*Top Opening, Single Hatch*	*Top Opening, Dual Hatch*	*Side Opening*	*"Treasure Chest"*	*Triangular*
						
FOM	***Weight Factor***			***Scoring***		
Loading Time	50	1	0	-1	0	-1
System Weight	50	1	1	0	-1	0
Totals	**100**	**100**	**50**	**-50**	**-50**	**-50**

Table 3.11 — Weighted Decision Matrix for Case Concepts

Loading time trials were conducted with prototype cases in each configuration. The top opening, single hatch case resulted in the fastest loading time. The "treasure chest" configuration was nearly as fast, but re-latching this case proved more difficult than the top opening one. Additionally, as seen Figure 3.12, the top opening design uses less material and contributes to an overall lower system weight.

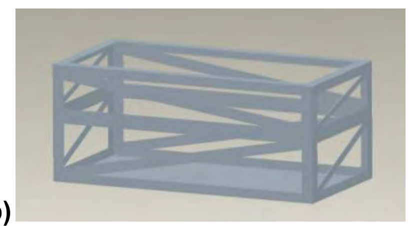
a)  **b)** 

Figure 3.12 — Final Case Concepts: (a) Top Opening and (b) "Treasure Chest"

3.4 Summary of Conceptual Design

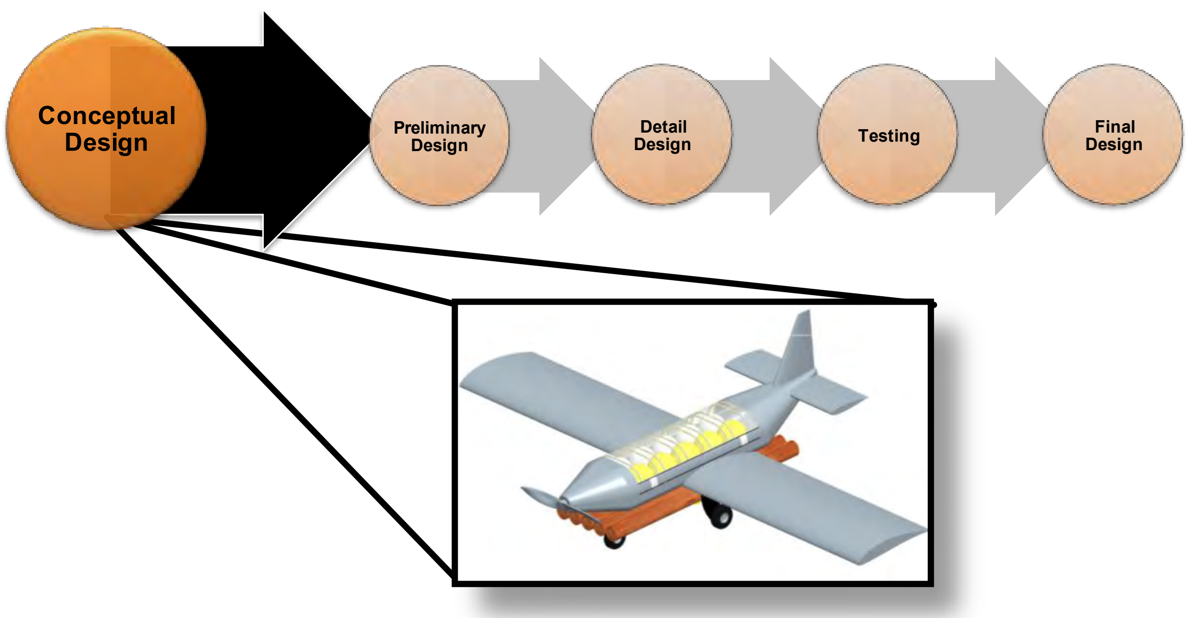


Figure 3.13 — Progress of Overall Design Process

4.0 PRELIMINARY DESIGN

With the general aircraft configuration selected from the conceptual phase, the aerodynamics, propulsion and structures groups began preliminary design within their respective areas of specialty. The first step during this phase was initial sizing of major components. Much of this was based on past designs with comparable mission requirements. An optimization program was used to develop trade studies in order to identify the highest-scoring design solutions. Performance predictions were then made based on the optimization configuration.

4.1 Design and Analysis Methodology

The preliminary phase was focused on taking the configuration developed in the conceptual phase and defining it quantitatively in terms of basic characteristics such as wing area, wing span, and propulsion capability. The goal was to develop a design that would *achieve the highest score possible.*

The analysis was performed primarily with the aid of a unified Multi-Disciplinary Optimization (MDO) program. The program itself was partitioned into two separate codes: one primarily for aerodynamic analysis and one for propulsion analysis. Since these programs were distinct and capable of being run independent of the other, it was important to maintain good communication between the two groups. Output generated by the aerodynamics team would be used as input for propulsion and vice versa. As numerical values regarding flight performance became available and eventually more accurate, the structures team was able to begin their portion of analysis as well. Structures relied mostly upon physical testing supplemented with simple analysis and interaction with the CAD lead. Several iterations of MDO analysis and physical testing were necessary to be sure the team as a whole was converging toward a uniform and optimum solution.

4.2 Mission Model

The Mission 1 and 3 flight profile (Figure 4.1) was analyzed by breaking it into five major components: take-off, climb, turn 1, cruise turn, and cruise. Mission 2 is based on loading time rather than flight time and it was modeled using results from timed trials.

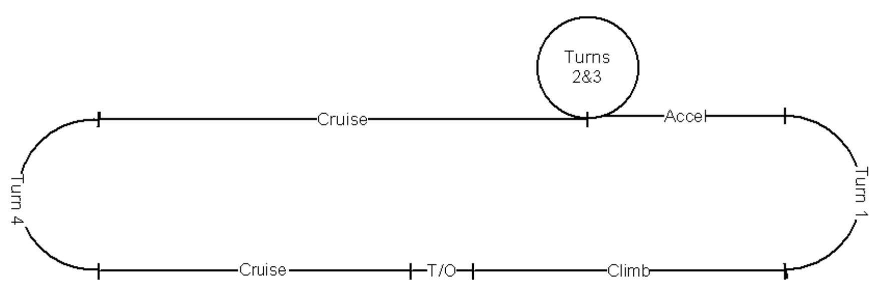


Figure 4.1 — Layout of Mission Course

Takeoff

Takeoff time was calculated by definite integration of Newton's Second Law based on the thrust curve, mission weight, rolling friction, and aerodynamic lift and drag of the aircraft in takeoff configuration.

Integration is carried out from 0 to ($V_{T/O} - V_{wind}$), where V_{wind} was initially assumed to be zero. Distances were determined by computing the definite integral of the aircraft's velocity function. The model assumes full power during takeoff with efficiency losses from the batteries, motor, gearbox, and propeller to generate realistic takeoff thrust. Rolling friction coefficient (μ) was assumed to be 0.05 while $C_{L_{TO}}$ was approximated using 95 percent of the wing airfoil maximum sectional lift coefficient with flaps. Drag from the fuselage and wing were calculated using empirical methods described in Raymer[2] and Hoerner[3] as well as airfoil and wing properties, respectively.

Climb

Rate of climb is calculated based on available power and mission weight. Full power is assumed for climb, and time to climb is simply calculated based on rate of climb to an operating height of 100 feet.

Turn 1

The first turn in each mission is encountered below maximum cruise velocity and taken at 180 degrees. Maximum *g*-loads are calculated using maximum flapped C_L at stall angle of attack. Turn rate is calculated assuming a level crosswind turn at full power, climb velocity and maximum *g*-load.

Cruise Turn

All subsequent turns are analyzed as 180-degree turns at full power and maximum *g*-load assuming maximum C_L at stall angle of attack for a non-flapped configuration. The turning analysis was changed afterward by assuming a flapped configuration to improve mission performance. In total, there are 7 cruise turns for Mission 1 and 11 turns for Mission 3. Time and distance for each were calculated assuming instantaneous turn rates.

Cruise

Cruise speed was calculated assuming level flight over the 2,000 feet of straight-away in each lap. 85 to 90 percent of power is assumed during cruise to account for power requirements in variable winds and the maximum power capability of motor. Maximum dash speeds above the cruise velocities can be maintained for short periods and are calculated using maximum motor power capability accounting for propeller, motor, gearbox, battery, and power transmission efficiencies.

4.3 Optimization Tools and Methodology

The MDO program was used find optimum aircraft configurations. It predicts optimum aircraft configuration parameters such as battery weight, wing span, planform area, and cruise speed using an iterative approach that concurrently analyzes aerodynamics, performance, propulsion, and structural requirements. A flowchart representing our basic utilization of the MDO program is shown in Figure 4.2.

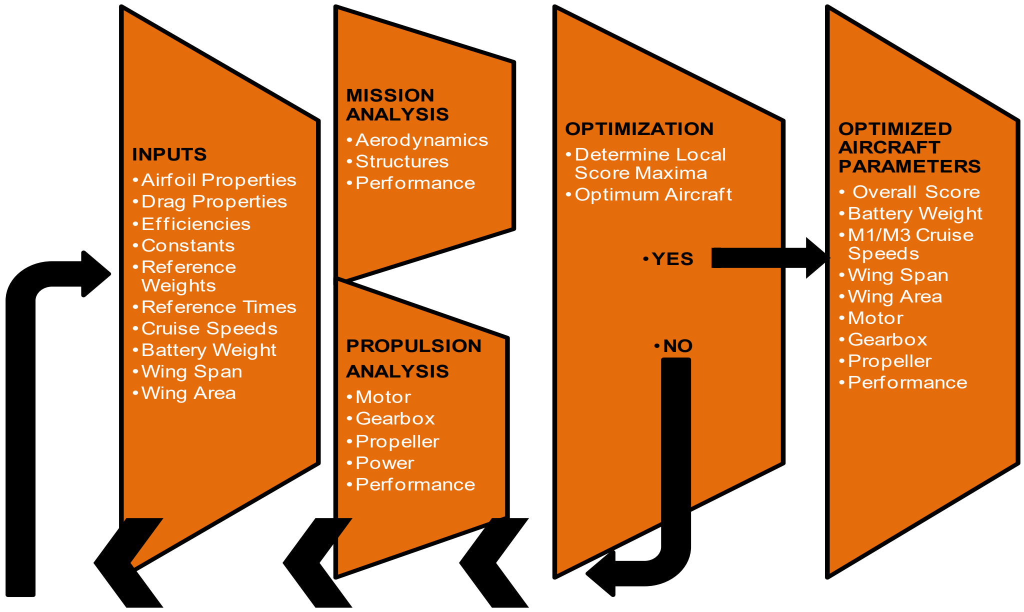


Figure 4.2 — MDO Flowchart

Inputs

Program inputs were required to establish baseline criteria and compute overall score. These criteria included airfoil properties such as sectional lift and drag curves, estimates for drag, motor, and gearbox efficiencies, and gravitational and friction constants. Mission cruise speed, power requirements, span, and aspect ratio constraints were established to maintain realistic solutions. The propulsion analysis program inputs included motor, propeller, and battery characteristics and aircraft aerodynamic parameters. From these inputs, takeoff performance, cruise conditions, energy consumption, and endurance capabilities could all be calculated. Furthermore, efficiencies and flight conditions for each mission segment could be examined in a way to design an aircraft that would achieve the highest contest scores. Wind inputs were also utilized to determine each aircraft's performance at different environmental conditions. Conservative estimates for Mission 1 and Mission 2 reference weight ($W_{1,ref}$ and $W_{2,ref}$), Mission 2 load times (t_{ref} and t_{team}), and Mission 1 and 3 flight times (t_{ref} and t_{team}) were entered based on historical competition data. These baseline results were used by the MDO code to calculate and compare the score of alternative configurations in order to produce the highest scoring designs.

Mission and Propulsion Analysis

The models used in the program include aerodynamic, propulsive, structural, and performance relationships. Aerodynamic modeling is performed using a combination of lift theory using sectional airfoil data and drag formulas based on methods used by Raymer[2]. The structural model is empirically based on historic in-house data and includes conservative estimates for both propulsion and case weight. Along with the aerodynamic analysis, a propulsion program was utilized in order to optimize components of the propulsion system. The aerodynamics and propulsion teams conducted an iterative process with both programs to arrive at the optimal aircraft and power plant configuration. Only Mission 1 and 3 were analyzed and modeled in the programs; the Mission 2 payload weighs less than the Mission 3 payload and induces less drag since it is carried internally. Therefore, Mission 2 is ignored in the program since it adds no value as a strategic design point.

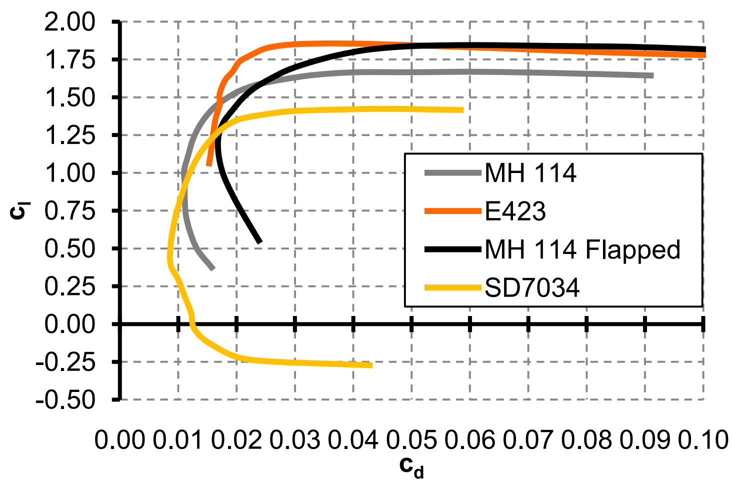
Optimization

Optimization outputs were based on localized maximum scores generated for each aircraft configuration. Several outputs were generated and used to determine trends for each input scenario. Optimized outputs were used alongside sound engineering experience and judgment to determine final aircraft geometry and power requirements.

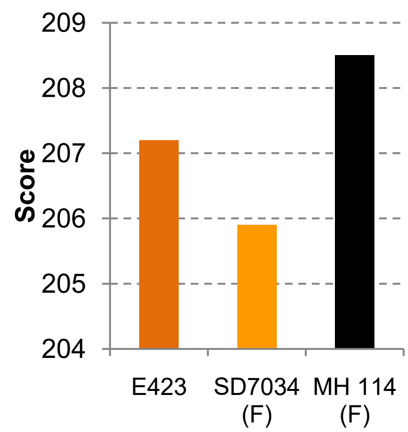
4.4 Design and Sizing Trades

4.4.1 Aerodynamics

Airfoil: Several airfoils were initially investigated which would provide low drag, high lift, easy manufacturing, and good stall qualities. The Martin Heppeler (MH) 114, SD 7032, SD 7034, and the Eppler 423 were used as baselines for comparison with other airfoils because of good performance in prior OSU DBF aircraft. Airfoil data was generated using Profili Pro[4] and used in the MDO program (Figure 4.3a). After multiple iterations, the Eppler 423, SD 7034, and MH 114 were found to be the best. These airfoils were each run flapped and un-flapped (Figure 4.3b).



a)



b)

Figure 4.3 — (a) Drag Polars of Airfoil Trade Study; (b) Score Results of Airfoils

The score for each of the airfoils falls within the range of programs uncertainty of ± 2 points. The Eppler 423 was discarded based on the highly cambered geometry which is difficult to manufacture and requires more tail volume for stability and thus more drag. The SD 7034 required higher wing spans (see Figure 4.4) for comparable scores, and is thinner than the MH 114 leading to a wing that is heavier and less stiff. The flapped MH 114 was chosen based on the higher performance shown in the optimization, high lift generation, good stall qualities, and suitable geometry for manufacturing.

Two-Piece Wing: A trade study was performed early in the preliminary design phase to determine the feasibility of creating a single-piece wing with a 4-foot span. This wing would need a lighter structure, further decreasing system weight, but might incur a severe penalty in flight performance resulting in a slower lap speed. Each of the candidate airfoils were input into the MDO code, with the wing span was constrained to less than 4 feet, and was run several times. The program was then constrained to less than 8 feet based on case size restrictions and run again multiple times. The score outputs were compared to determine which wing size performed better. The results below in Figure 4.4 show that a span of less than 4 feet would typically have a lower score. Therefore, a two-piece wing with a larger span was chosen to be further investigated and accurately sized.

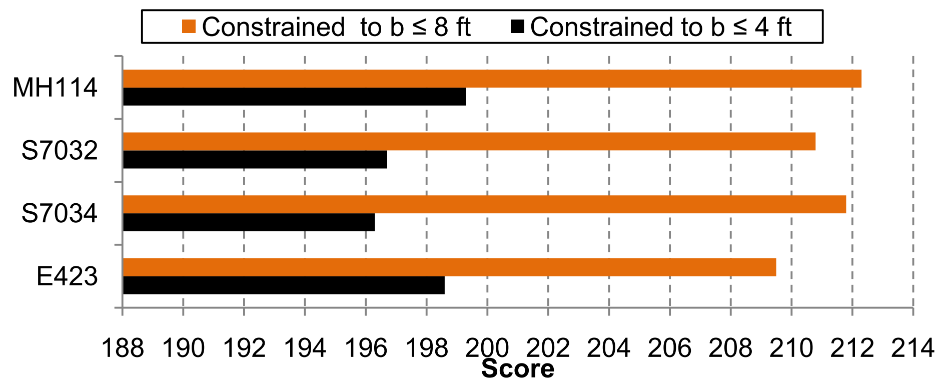


Figure 4.4 — Comparative Flight Score for Candidate Airfoils Due to Wing Span Constraints

Wing Span: The three airfoils discussed previously were used in the MDO program to generate several optimized aircraft solutions. The program outputs were plotted (Figure 4.5a) to show a wide scatter in wing span with score. However, the data for MH 114 can be seen to cluster more within the 6.5- to 7-foot range for best scores and the trend for wing planform area shows a similar scatter. Upon further investigation, it was noted that in most cases the optimum configuration was constrained by an aspect ratio of 8, which was set in the program based on wing stall/spin concerns during turning, a characteristic of high-aspect ratio wings. The high aspect ratio is necessary to provide enough performance to overcome the added drag created by the five externally loaded bats of Mission 3. Therefore, a wing span of 6.5 feet was chosen at the lower end of the cluster range to save case volume.

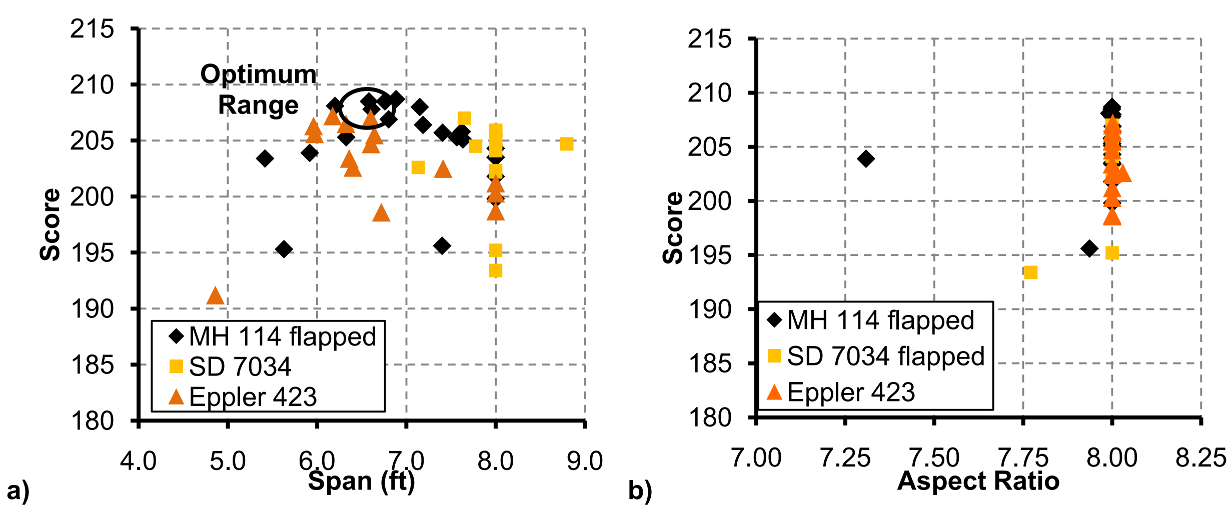


Figure 4.5 — (a) Flight Score for Varying Wing Span; (b) Flight Score for Varying Aspect Ratio

Wing Area: The inherent spacing of the payload design as well as structural wing strength required the wing chord to be 10 inches. The wing area was established based on a simple rectangular geometry using a 6.5-foot span and a 10-inch chord. The slight taper at the wing ends was added to further increase the aspect ratio by reducing planform area to 5.33 square feet.

Tail: The tail was sized to provide static stability and dynamic control. The initial horizontal tail sizing was performed by using the recommended tail volume by Raymer[2]. A horizontal tail volume of 0.50 was selected, giving a surface area 1.33 square feet. This tail size with a conventional elevator provides sufficient pitching moment and a relatively high static margin for the aircraft. The vertical tail was also sized in a similar manner using a suggested tail volume of 0.03 giving a planform area of approximately 0.6 square feet. Considerations for the vertical stabilizer height were made to fit the aircraft into the smallest case possible and reduce overall system weight. The rudder was initially over-sized by utilizing a control horn to provide sufficient control during ground handling.

Horizontal Tail Airfoil: Initially, a symmetric airfoil NACA 0009 was chosen for the horizontal stabilizer, but it required an incidence of -4 degrees for proper stability. Since small incidence angles are difficult to produce at this scale, a cambered airfoil, the NACA 4409, was selected based on a similar sectional lift coefficient at 0 degrees as the NACA 0009 at -4 degrees (Figure 4.6). The resulting horizontal cross section is an inverted NACA 4409 at an incidence of 0 degrees which provides sufficient stability and less drag.

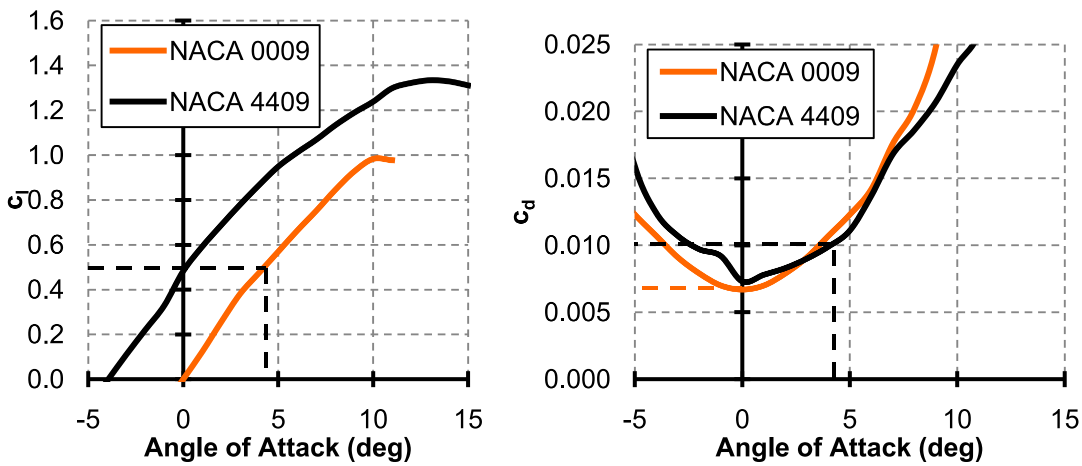
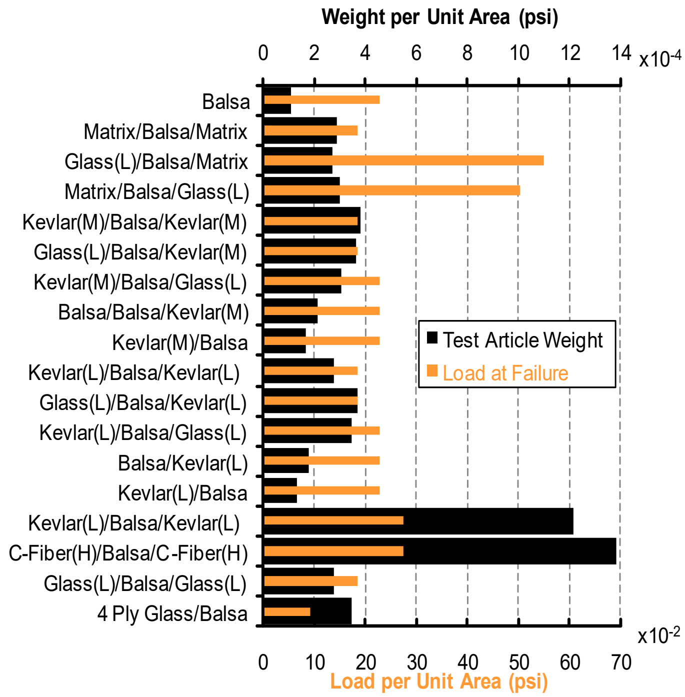


Figure 4.6 — (a) Sectional Lift and (b) Drag Curves for Candidate Horizontal Tail Airfoils

4.4.2 Structures

Payload: The aircraft was designed entirely around the payload bay of the aircraft to maximize structural efficiency and decrease overall mission weight. Testing several different softball loading configurations, it was determined that the individual ball spaces, created by intersecting channels, would incorporate the payload structure into the overall aircraft structure most efficiently, resulting in the lowest weight. Several ball-securing options were examined with considerations to the mission rules, weight, and structural efficiency. The sections securing the softballs also served as fuselage bulkheads by curving halfway up the sidewalls and mating with similar upper sections embedded in the hatch. Keelsons running down the center the length of the fuselage divide the payload grid in half and provide additional longitudinal stiffness to the fuselage.

Material Selection: Materials for fuselage, wing, and case construction were used based on historical experience from prior OSU DBF aircraft and predicted flight loads generated by the aerodynamics team. Figure 4.7 shows the trade study of the strength tests of different material against the material weight. To design the most structurally efficient aircraft, it only needs to be strong enough to accomplish the required missions with the least amount of weight possible. Therefore, the aircraft only needs to withstand the maximum aerodynamic loads expected to occur with a very small factor of safety.



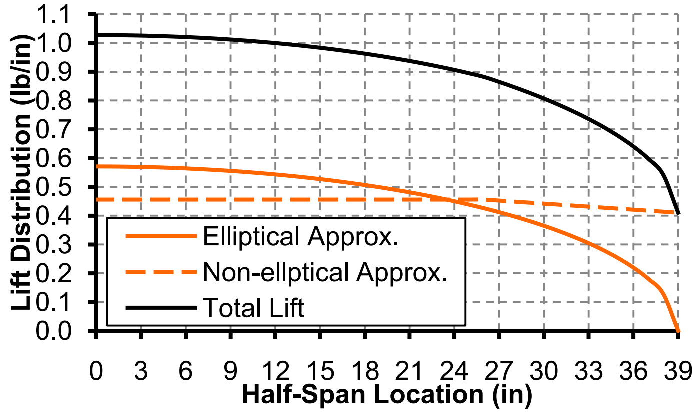
Material/Lay-Up	Strength-to-Weight Ratio
Balsa	302
Matrix/Balsa/Matrix	391
Glass(L)/Balsa/Matrix	246
Matrix/Balsa/Glass(L)	275
Kevlar(M)/Balsa/Kevlar(M)	414
Glass(L)/Balsa/Kevlar(M)	397
Kevlar(M)/Balsa/Glass(L)	334
Balsa/Balsa/Kevlar(M)	291
Kevlar(M)/Balsa	226
Kevlar(L)/Balsa/Kevlar(L)	300
Glass(L)/Balsa/Kevlar(L)	400
Kevlar(L)/Balsa/Glass(L)	377
Balsa/Kevlar(L)	242
Kevlar(L)/Balsa	176
Kevlar(L)/Balsa/Kevlar(L)	603
C-Fiber(H)/Balsa/C-Fiber(H)	627
Glass(L)/Balsa/Glass(L)	374
4 Ply Glass/Balsa	377

Figure 4.7 — (a) Unit Weight and Strength Characteristics of Lay-Ups; (b) Strength-to-Weight Ratios of Lay-Ups

The primary materials chosen for the bulk of the fuselage and tail were fiberglass-reinforced balsa to provide high strength and stiffness and low weight. Carbon fiber laminate was chosen to reinforce high-load bearing structures such as wing carry through and some spar sections.

Fuselage: The main fuselage body size was determined by the payload structure size. The nose and tail sections were determined by conically fairing the main body to increase aerodynamic efficiency. Additionally, the nose length was sized based on the motor and gearbox space requirements. It includes a slightly positive inclination from the centerline to maximize ground clearance for the propeller. The tail cone was extended to provide maximum moment arm length for the tail. The fuselage structure is strengthened by the payload section in addition to the material properties.

Wing: In order to size the wing structure, estimates of lifting loads were generated. Based on initial weight predictions, the wing lift distribution was determined as seen in Figure 4.8a. Mission analysis indicated that a 6-*g* load would be encountered during the high-speed turns of Mission 1 (§4.5.3). The expected root moments were then determined based on the lift distribution and dynamic loads (Figure 4.8b).



Case	Dynamic Load (g's)	Root Moment (lb-ft)
Tip Test	0	18.6
Mission 1	6	28.6
Mission 3	3	20.1

Figure 4.8 — (a) Half-Span Lift Distribution; (b) Estimated Bending Moment Due to Turning Loads

Structural tests indicated a need for two wing spars in order to provide adequate torsional rigidity. Basic bending and torsion tests (§8.1.1) were conducted on generic wing sections with different material properties for the maximum predicted aerodynamic loads. The tests focused on structural weight and rigidity. A very stiff wing would provide more than adequate rigidity at the expense of added structural weight. Therefore, tradeoffs were made to reduce the weight by scaling back the rigidity of the wing to only withstand a minimal amount of stress beyond our predicted flight loads. The main spar was chosen to be constructed from carbon-fiber reinforced balsa to give acceptable bending rigidity and low weight.

Landing Gear: The landing gear length was sized to place the aircraft at an initial angle of 12 degrees to the horizontal. This angle is the maximum stall angle of the wing and provides the best lift for short takeoff. Additionally, as suggested by Raymer[2], the gear's wheel hub is placed at a forward station relative to the aircraft's center of gravity, 20 degrees off vertical of its position to ensure static stability while on the ground

4.4.3 Propulsion and Power

Motor and Gearbox: Based upon the aerodynamic analysis performed with the MDO code, a "worst-case" motor power requirement of 475 watts was found for a fully loaded aircraft with no wind conditions during cruise. For takeoff, the requirement was 720 watts. However, the power for takeoff is only needed for a few seconds and the maximum rated power of a motor can be briefly exceeded provided adequate cooling is available. Brushless motors in the power range of 300 to 600 watts with low voltage constants were investigated as options to fulfill the power requirements. A trade study was performed in which each motor was paired with an 18x10 propeller and an optimum gearbox was used based upon the value of the voltage constant. Results of maximum cruise motor power required, takeoff distance, and cruise efficiency are shown in Table 4.9 below.

Manufacturer	Model	Weight (g)	Prop	P_{max} (W)	T/O Distance (ft)	η_{cruise}
Medusa	MR-028-056-0900	160	18x10	417	97	0.503
Medusa	MR-036-050-0810-5	230	18x10	569	100	0.524
Neu	1902/3Y	124.738	18x10	300	92	0.497
Neu	1110/3Y	113.398	18x10	500	99	0.522
Neu	1112/2.5Y	133.24	18x10	500	104	0.53
Neu	1905/3Y	181.437	18x10	600	100	0.501
Neu	1506/3Y	170.097	18x10	700	91	0.489

Table 4.9 — Motor Trade Study Results

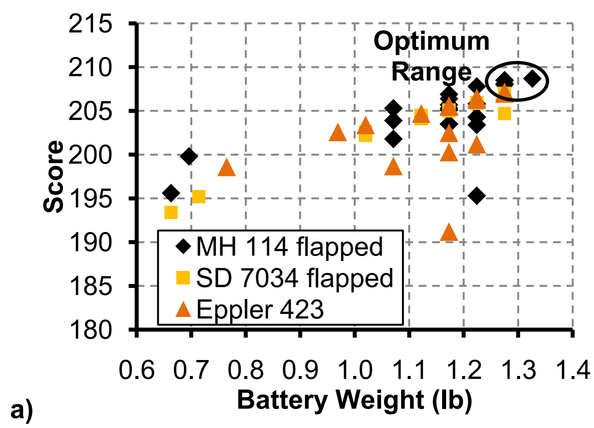
The motor selection was conducted based on the trade study results. A large emphasis was placed on system weight, so the Medusa motors and the overpowered Neu motors were not selected. Also, the Neu 1902 series was removed from consideration because of heating concerns caused by motor overloading during cruise. Due to its significantly lighter weight, power capabilities, and high cruise efficiency, the Neu 1110/3Y motor was selected.

Propeller Pitch and Diameter: The selection of the propeller required a careful balance between both pitch and diameter. By using previously compiled propeller data along with the MDO program, various propellers were studied. Propeller diameter was limited to 18 inches due to the high current drawn by the motor for larger diameters. With the diameter set at 18 inches, propeller pitch was decreased until the specified take-off distance of 100 feet was met. Using the highest pitch possible while still meeting take-off requirements will allow the aircraft to cruise at maximum velocity and efficiency.

4.5 Mission Performance

4.5.1 Mission Cruise Speed and Battery Weight

The MDO program outputs show a similar trend for both battery weight and mission cruise speed. Increasing battery weight effectively increases power and thus increases speed capacity of the aircraft. The results in Figure 4.10 show that an aircraft with a high energy density will achieve a faster lap time increasing the overall score for Mission 1 and 3.



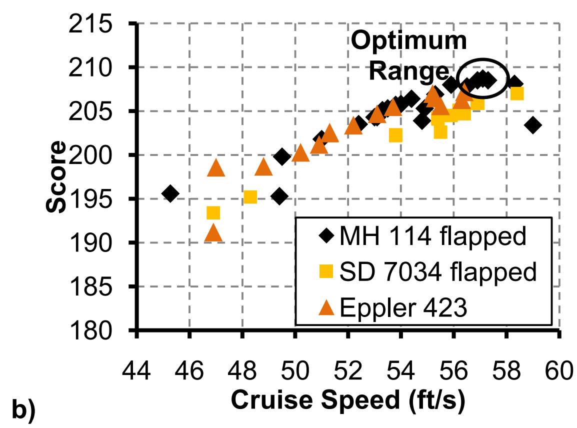


Figure 4.10 — (a) Score for Varying Battery Weights; (b) Score for Varying Cruise Speeds

Separate optimization runs were made using the described aircraft configuration to determine the best power requirement. Several batteries were tested including the Elite 1500, 2000, 3300, and 4000 series as well as the GP 2000. Performance data for these batteries was found through testing performed at OSU. The results show that an aircraft with a high energy density will achieve a faster lap time increasing the overall score for Mission 1 and 3. 26 Elite 1500 cells (1000 mAh, 25 A maximum, 0.051 pounds, 19,608 mAh/lb each) provide the highest scoring aircraft. The total power required is 720 watts with 1.33 pounds of batteries.

4.5.2 Turn Performance

Additional mission performance analysis of the optimum configured aircraft was conducted using an input only variation of the MDO code. This study showed that power requirements during the turns were severely limiting Mission 1 and 3 cruise speeds, battery usage, and thus score. By allowing for an altitude loss in the turn, turning speed could be increased, reducing flight time during Mission 1 and Mission 3, increasing battery use and overall score, shown below in Figure 4.11.

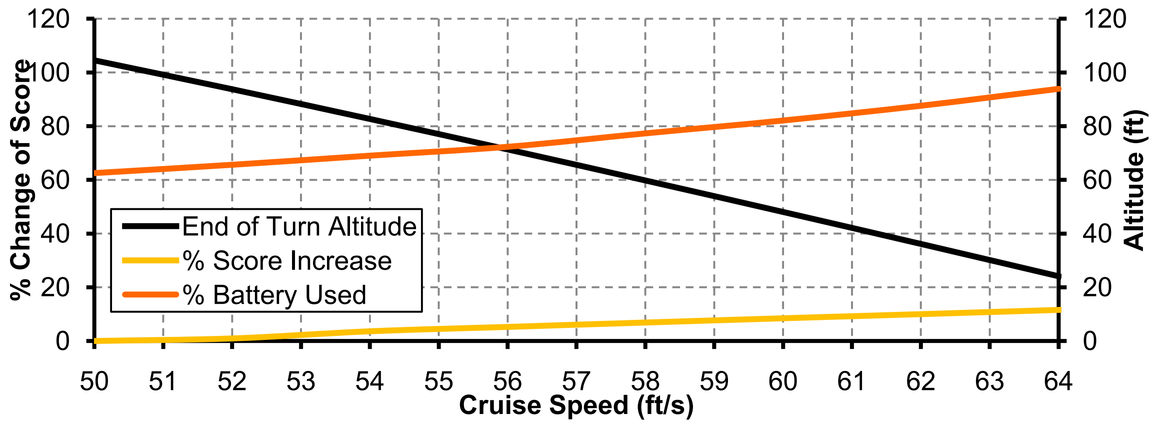


Figure 4.11 — Percent Change of Flight Score and Change in End of Turn Altitude as Functions of Cruise Speed

4.5.3 Flapped Turn

Increasing turning performance with the use of flaps during turns was also investigated. The input variation of the MDO program was modified to simulate flaps during the turn. By using flaps, the effective lift coefficient is increased. This results in tighter turns, a shorter lap time, and a score increase of 5 to 10 percent (Figure 4.12a). The trade study also reinforces the decision to use the flapped MH 114 airfoil, which enables the use of flaps on takeoff and in turns, and flaps off during the straight-aways to maximize cruise speed. Another method will be investigated during flight testing which uses a combined vertical climb and turn maneuver, but both will use flaps during the turning portions. One drawback that the study reveals is the high *g*-loads encountered during the turns. Therefore, the aircraft must be capable of sustaining higher *g*-loads. However, the high loads increase flight score, as seen in Figure 4.12b.

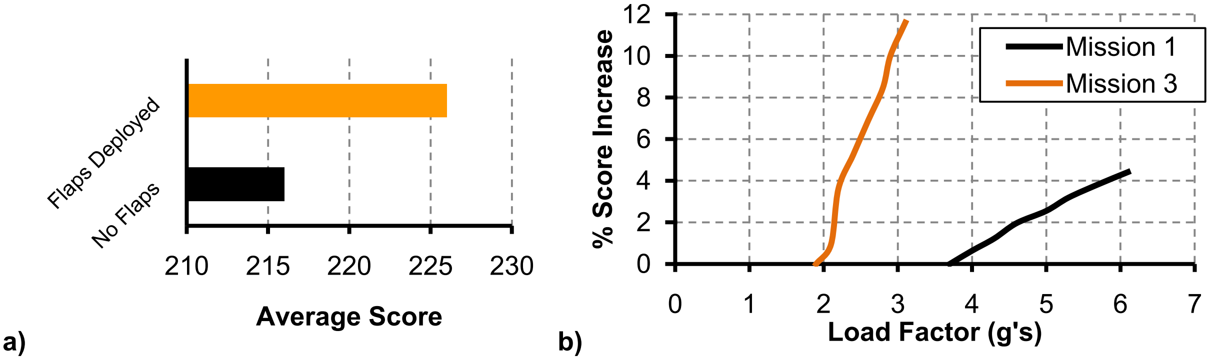


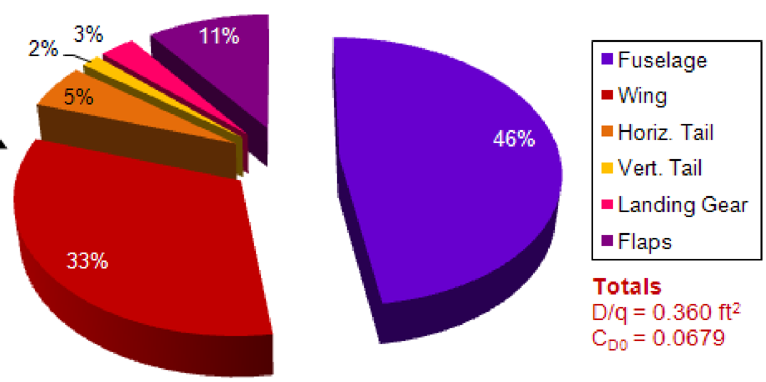
Figure 4.12 — (a) Averaged Effect of Flaps on Flight Score; (b) Change in Flight Score as a Function of Load Factor

4.6 Lift, Drag, and Stability Characteristics

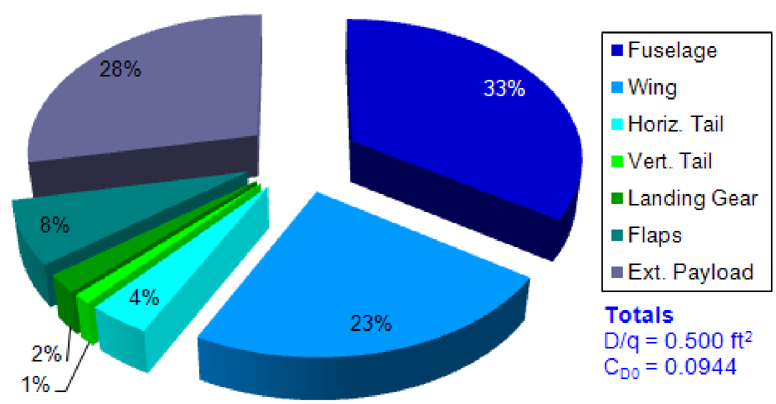
4.6.1 Aerodynamic Qualities

Estimations for the total drag profile of the aircraft were made by using the method of equivalent skin friction, as suggested by Raymer[2]. Drag predictions for each component of the aircraft were intentionally kept conservative, ensuring that the aircraft would be designed to fly effectively with the worst-case drag scenario. Drag estimations for the external payload included the bats, the holding device, and interference drag using data and methods from Hoerner[3]. The breakdown of the total parasite drag of the aircraft components is represented in the Figure 4.13.

Mission 1 & 2 Drag



Component	D/q (ft^2)	C_{D0}
Fuselage	0.171	0.0323
Wing	0.117	0.0220
Horiz. Tail	0.0191	0.00361
Vert. Tail	0.00546	0.00103
Landing Gear	0.0105	0.00199
Flaps	0.0366	0.00690
Ext. Payload	0.141	0.0266
Mission 1 Total	**0.360**	**0.0679**
Mission 2 Total	**0.360**	**0.0679**
Mission 3 Total	**0.500**	**0.0944**



Mission 3 Drag

Figure 4.13—Drag Build-Up by Aircraft Component for "Clean" Mission Configuration (*Top*) and "Dirty" Mission Configuration (*Lower*)

The overall lift of the aircraft is due to a combination of the wing lift and the lift generated by the tail. Since the tail generally creates a negative lift force, the lift coefficient provided by the wing must be large enough to satisfy the takeoff requirements. The combined lift curves of the aircraft are seen in Figure 4.14 below.

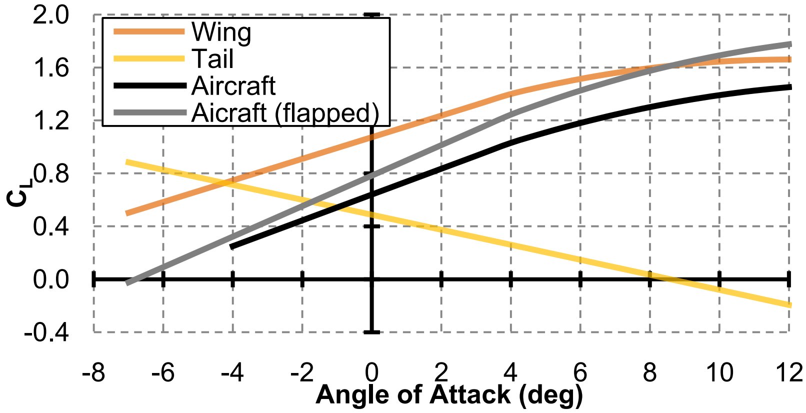
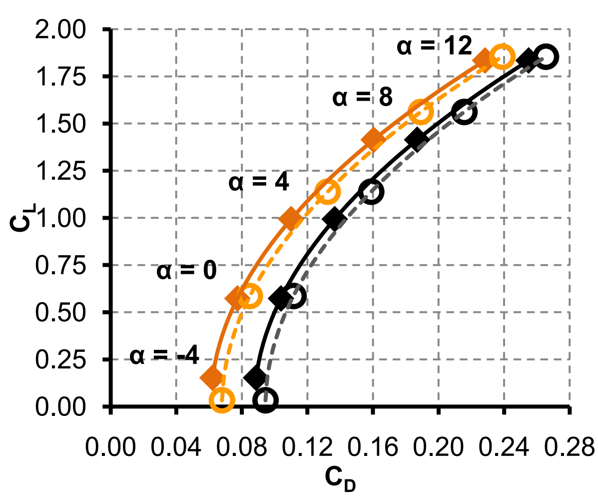
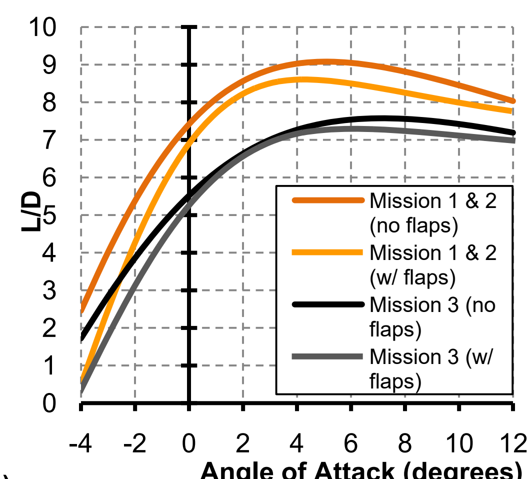


Figure 4.14 — Total Aircraft Lift Curve Build-Up

This range of lift coefficients can now be combined with the zero-lift drag coefficients to generate total aircraft drag polars across the various missions and aircraft configurations. The use of flaps during a mission causes significant changes to the aerodynamic characteristics of the aircraft. These effects are shown in Figure 4.15a. Flaps will be utilized on takeoff and through the first turn of the course to provide additional lift to reach cruising altitude faster where the aircraft can accelerate to its optimum speed. Flaps will also be lowered as the aircraft enters the 360-degree turn to increase the lift and drag, allowing the turn rate of the aircraft to increase as well as minimizing the turn radius. This will allow the aircraft to complete the 360-degree turns much faster, resulting in lower mission time and a higher score. The lift-to-drag ratio of the aircraft are plotted in Figure 4.15b to determine the optimum angle of attack at which to fly the vehicle for the different missions and configurations of the aircraft.



a)

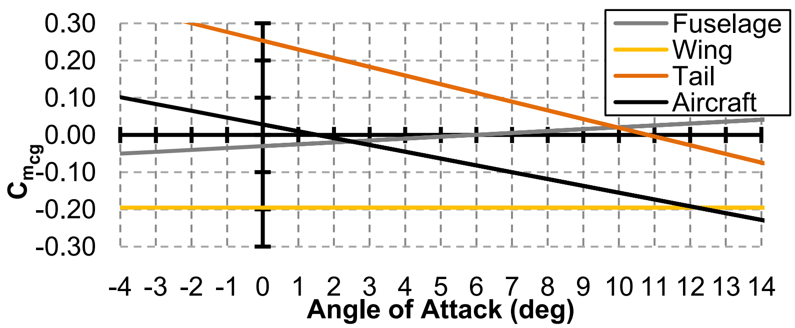


b)

Figure 4.15 — (a) Total Aircraft Drag Polar for Various Mission Configurations; (b) Total Aircraft Lift-to-Drag Ratios for Various Mission Configurations [*Refer to Legend of (b) for Both Figures*]

4.6.2 *Stability Characteristics*

Initial stability and control analysis was performed to ensure static stability of the aircraft and sufficient static margin. By creating a simple moment curve diagram for each component and the aircraft, the static stability of the aircraft was established. Figure 4.16a below shows the static stability of the aircraft and the contributions of each major component to the aircraft moment. The neutral point and static margin of the aircraft were determined as shown in Figure 4.16b.



x_{ac}/c	0.250
x_{cg}/c	0.250
x_{NP}/c	0.633
S.M.	0.383

Figure 4.16 — (a) Pitch Moment Build-Up; (b) Static Margin and Related Static Pitch Stability Data

The aircraft is designed to easily change the position of the batteries to keep the center of gravity aligned with the center of the payload. The wing's quarter-chord is then located at the aircraft's center of gravity, eliminating any change in pitching moment produced by the lift of the wing during flight. The neutral point is at 63.3 percent of the chord length, giving the aircraft a static margin of approximately 38.3 percent. The relatively high static margin is typical based from similar benchmarked aircraft.

The change in pitching moment with a range of elevator deflections can be seen in Figure 4.17, giving the amount of control input required to maneuver the aircraft in pitch. From the plot, the trim point of the aircraft is shown to be at one degree with no deflection of the elevator. This indicates that the aircraft should require little to no adjustment for straight and level flight.

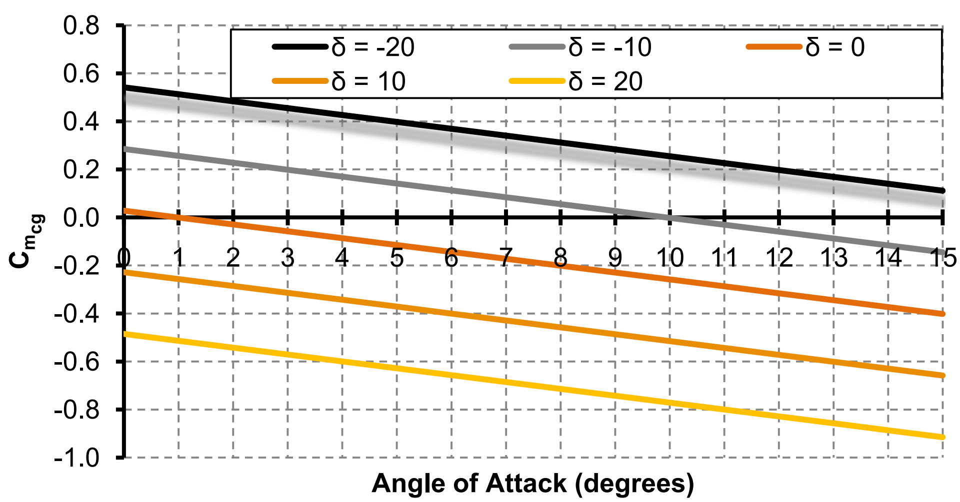


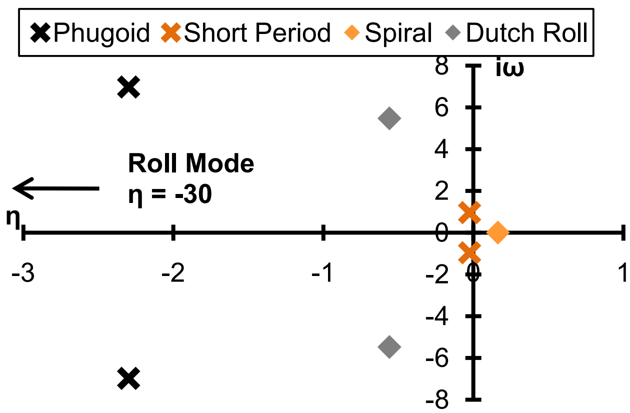
Figure 4.17 — Pitch Moment Curves for Varying Elevator Deflection Angles

Once the static stability had been established, an analysis on dynamic stability was performed. The stability derivatives were calculated for both longitudinal and lateral motion at maximum payload and drag conditions. The stability derivatives were used to calculate the stability coefficients to predict the aircraft flight modes using methods suggested by Nelson[5]. Table 4.18 below shows the coefficients for both the longitudinal and lateral motion of the aircraft.

X-Force		Z-Force		Pitching Moment		Y-Force		Yaw Moment		Rolling Moment	
C_{Xu}	-0.120	C_{Zu}	-1.598	C_{mu}	0.00	$C_{y\beta}$	-0.450	$C_{n\beta}$	0.242	$C_{l\beta}$	0.000
$C_{X\alpha}$	0.464	$C_{Z\alpha}$	-4.230	$C_{m\alpha}$	-1.642	C_{yp}	0.000	C_{np}	-0.100	C_{lp}	-0.715
$C_{X\dot{\alpha}}$	0.000	$C_{Z\dot{\alpha}}$	-2.014	$C_{m\dot{\alpha}}$	-3.827	C_{yr}	0.231	C_{nr}	-0.0812	C_{lr}	0.236
C_{Xq}	0.000	C_{Zq}	-5.879	C_{mq}	-11.17	$C_{y\delta a}$	0.000	$C_{n\delta a}$	-0.192	$C_{l\delta a}$	0.436
$C_{X\delta e}$	0.000	$C_{Z\delta e}$	-0.453	$C_{m\delta e}$	-0.900	$C_{y\delta r}$	0.123	$C_{n\delta r}$	-0.0792	$C_{l\delta r}$	0.0189

Table 4.18 — Important Longitudinal and Lateral Stability Coefficients

From the stability derivatives, the root plot of Figure 4.19a was generated showing the real response (η) and the frequency/imaginary response (ω) of each mode of flight.



Longitudinal			
	iω	ζ	λ
Phugoid	0.9861	0.07005	-2.298 ± 6.977i
Short Period	1.382	-0.1816	-0.0251 ± 0.960i
Lateral			
	iω	ζ	λ
Spiral	0.0	1.0	0.164
Dutch Roll	5.409	0.0966	-0.560 ± 5.476i
Roll	0.0	1.0	-29.94

Figure 4.19 — (a) Root Locus Plot and (b) Eigen Values for Longitudinal/Lateral Modes of Dynamic Stability

Figure 4.19b shows that the aircraft is stable in all modes of flight with the exception of spiral mode, which is a typical flight dynamic quality. The spiral mode root is very small, yielding a relatively long time to double. Spiral instability is not a significant issue and is easily corrected with control input.

4.6.3 Control Surface Sizing

The control surface sizing was based in part on historical OSU designs, pilot preferences, and control requirements from analysis. The ailerons and rudder were sized large and allowed deflection to +10 / -20 degrees and ± 20 degrees respectively. The elevator was sized to give control during takeoff with deflection of +10 / -20 degrees. The elevator was limited to -10 degrees (up) to prevent stalling of the aircraft. The ailerons are limited to 10 degrees down deflection to limit the wing from stalling at low angles of attack based on the highly cambered nature of the airfoil section.

4.7 Aircraft Mission Performance

Aircraft mission performance was reviewed to predict overall mission score and aircraft handling. The results of the trade studies found during optimization showed that both the aircraft and the piloting of the aircraft must be optimized to achieve the best result.

4.7.1 Pre-Mission: Softball Loading

Softball loading time is a critical parameter for mission two. In the MDO code, the team's best estimate of a rapid loading time, including the approach and retreat from the aircraft at twenty feet was made at 8 seconds. These estimates were made by loading softballs into a simulated version of our configuration using the competition setup. Training and practice were performed by a designated loading crew to continually make improvements in these times (§8.2).

4.7.2 Mission 1: Ferry Flight

The critical performance parameters of Mission 1 is weight and lap speed. Weight is restricted to the aircraft construction, but the lap speed can be improved by optimized handling of the aircraft around the turns. The trade study from optimization showed that battery usage was not being optimized because of the power required during turns, and that a severe altitude loss would occur for an increase in flight velocity. Possible alternatives are to use an altitude turn maneuver to account for the altitude loss, but further testing must be accomplished to demonstrate feasibility. The propulsion optimization program was used to calculate the Mission 1 times, speeds, distances, and energy consumption during the phases of the flight. The results of the minimum speed mission and maximum speed are shown in Table 4.20.

Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)	Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)
Lap 1	*Takeoff*	0 - 33	10	0.8	**Lap 2**	*Cruise*	64	500	10.2
	Climb	43	595	2.6		*Turn 5*	64	90	1.4
	Turn 1	43	170	5.5		*Cruise*	64	250	10.2
	Turn 2 & 3	64	180	2.8		*Turn 6 & 7*	64	180	2.8
	Cruise	64	750	10.2		*Cruise*	64	750	10.2
	Turn 4	64	90	1.4		*Turn 8*	64	90	1.4
	Cruise	64	500	10.2		*Cruise*	64	400	10.2
			Totals		**Distance**	**2294**			
					Time	**33.5**			

Table 4.20 — Initial Estimation of Mission 1 Profile (5 MPH Wind Speed)

4.7.3 Mission 2: Softball Payload Flight

Mission 2 loading time is the most critical component in optimization. The aircraft flight profile is not considered as the Mission 3 payload is heavier and requires more power because of the drag induced, therefore if the aircraft is able to complete Mission 3 it will be able to complete Mission 2. However, the aircraft may be configured with a lighter battery weight to complete the mission. The choice to fly at a lighter weight would only be beneficial if Mission 1 turning performance cannot be improved and a lighter battery weight could be used for it as well considering that both $W_{1,ref}$ and $W_{2,ref}$ should be equal.

4.7.4 Mission 3: Bat Payload Flight

Mission 3 parameters are similar to Mission 1 in that turning performance and speed are critical parameters to be optimized. However, in addition to flying fast the aircraft must be able to transport 5 bats. The number of bats to be flown is optional, but during conceptual design it was decided that carrying five bats was optimal (§3.2). The same turning technique will be tested for Mission 3 carrying the payload as was tested in Mission 1. The propulsion optimization program was used to calculate the performance capabilities of the aircraft for Mission 3. It was calculated that the aircraft could complete the mission in 134 seconds, as shown in Table 4.21.

Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)	Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)
Lap 1	*Takeoff*	0 - 33	52	2.8	Lap 2				
	Climb	43	499	6.1		*Cruise*	63	500	7.7
	Turn 1	43	329	7.6		*Turn 5*	63	188	3.0
	Cruise	63	500	7.7		*Cruise*	63	500	7.7
	Turn 2 & 3	63	377	6.0		*Turn 6 & 7*	63	377	6.0
	Cruise	63	500	7.7		*Cruise*	63	500	7.7
	Turn 4	63	188	3.0		*Turn 8*	63	188	3.0
	Cruise	63	500	7.7		*Cruise*	63	500	7.7

Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)
Lap 3	*Cruise*	63	500	7.7
	Turn 9	63	188	3.0
	Cruise	63	500	7.7
	Turn 10 & 11	63	377	6.0
	Cruise	63	500	7.7
	Turn 12	63	188	3.0
	Cruise	63	400	7.7
Totals			8352	134.2

Table 4.21 — Initial Estimation of Mission 3 Profile (5 MPH Wind Speed)

4.8 Summary of Preliminary Design

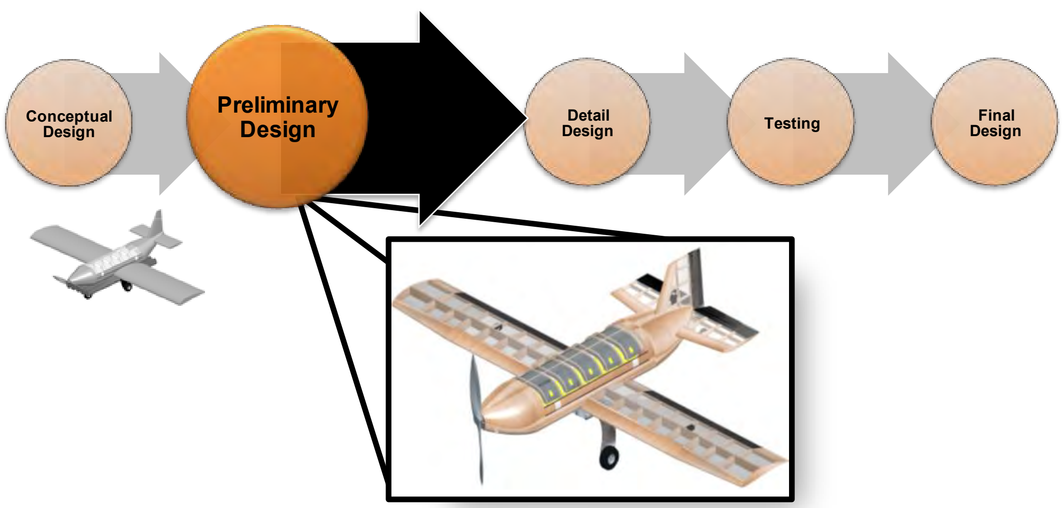


Figure 4.22 —Progress of Overall Design Process

5.0 DETAIL DESIGN

Upon the completion of the preliminary design phase, work began on optimizing and integrating sub-systems. Manufacturing processes are planned to ensure correct execution of all fabricated parts of the design. Detailed aircraft and mission performance analysis are conducted and finalized in order to improve flight score wherever possible.

5.1 Dimensional Parameters

Fuselage	
Length (in)	40.25
Width (in)	8.75
Height (in)	5.0
Wing	
Airfoil	MH 114
Span (in)	78.0
Root Chord (in)	10.0
Tip Chord (in)	9.0
Area (in^2)	768
Aspect Ratio	7.9
Incidence Angle (deg.)	0.0
Flaperon Area (in^2)	104
Max. Thickness (in)	1.3
Case	
Length (ft)	24
Width (ft)	48
Height (ft)	14
Empty Weight (lb)	3.02
Transmitter (lb)	1.06
Overall Aircraft Size	
Length (in)	43.75
Width (in)	78
Height (in)	17

Horizontal Stabilizer	
Airfoil	NACA 4409 (inverted)
Span (in)	24.0
Root Chor	
Tip Chor	
Tail Vol	
Incidence (deg.	
Elevator Ar	
Vert	
Airfoi	
Span (	
Root Chor	
Tip Chor	
Tail Vol	
Rudder Ar	
Aircraft W	
Airframe	
Propulsio	
Controls	
Max. En Weight	
Max. Payload (lb)	6.25
Max. Gross Weight (lb)	11.5
CG Location (in)	17.8
Total System Weight	**9.33**

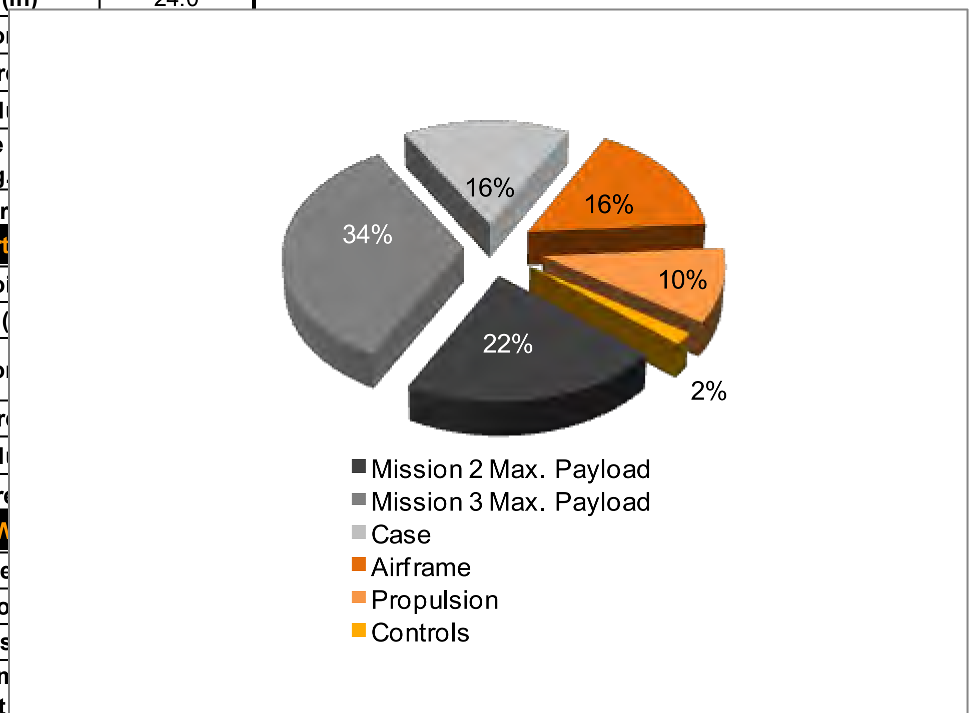


Figure 5.1 — (a) Major Dimensional Parameters; (b) Total System Weight Breakdown

5.2 Structural Characteristics and Capabilities

Tight, high-g turns are required for Mission 1 and Mission 3 strategies. As such, the wing and all other flight-critical structures were designed to handle up to 6 *g*'s in the Mission 1 configuration and 3 *g*'s in the Mission 3 configuration. Testing on the bat-holder mechanism was performed to assure a 3-*g* load would not allow motion along a bat degree of freedom. The softball securing mechanism is not expected to see high g-loads, but the system was tested up to 2 *g*'s in any direction to assure payloads could not shift in-flight. The landing gear must be capable of supporting a wide range of aircraft weights during a hard landing. Drop tests were performed by modeling the Mission 3 aircraft weight before installation on the aircraft (§8.1.2). The V-n diagrams (Figure 5.2) show the structural limitations for the final aircraft design to be considered during both flight testing and competition (§8.0).

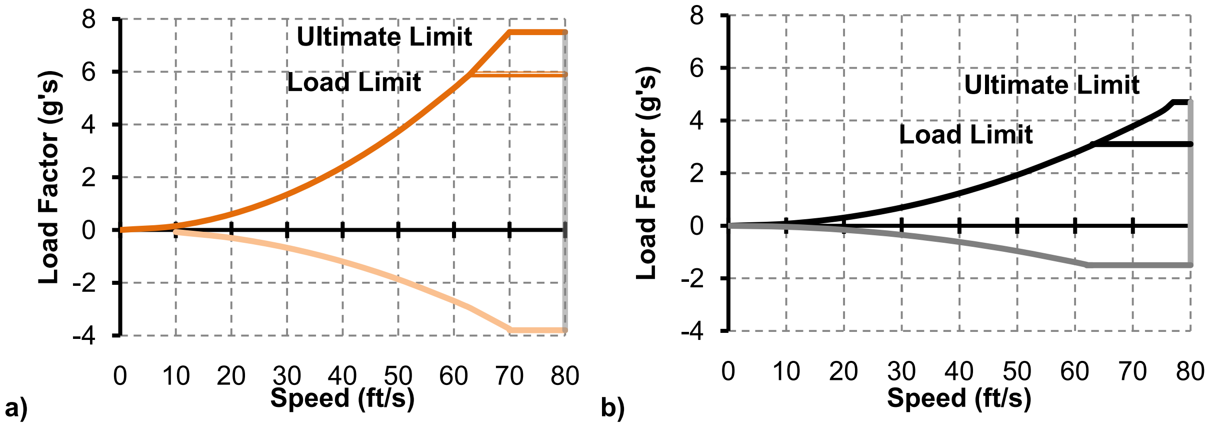


Figure 5.2 — V-n Diagram for (a) Mission 1 and (b) Mission 3

The case is capable of supporting the unassembled aircraft weight, radio transmitter, and all assembly tools without deformation. Furthermore, to accommodate the requirement that the balls be dumped from the bag into the case, a section of its floor as well as the hinge area was strengthened to accommodate quick opening and closing during loading without damaging the case.

5.3 Component and Sub-System Selection and Integration

5.3.1 Fuselage

The fuselage was designed around the internal payload with quick loading, aerodynamics, electronics integration, and structural integrity in mind. The desirable softball arrangement became a 2x5 grid after considering aerodynamics and loading time. The central fuselage structure became a grid spaced to allow for the softball payload. Through the center of this grid run two keelsons that span the entire length of the fuselage to carry longitudinal bending loads. Propulsion batteries fit snuggly between the keelson structures, and can be moved fore and aft to facilitate center of gravity motion, if necessary. Additionally, two stringers run the full fuselage length directly under the top-opening hatch to provide structural integrity. Out from the central payload grid, the fuselage has full bulkheads. The motor is secured to the first bulkhead. The speed controller is mounted in the nose of the aircraft near the motor. The receiver and its battery pack are mounted in the tail cone to isolate them from the high current generated by the motor and propulsion batteries and to minimize wire length running to the tail surfaces. Figure 5.3a shows the fuselage structural members and propulsion batteries.

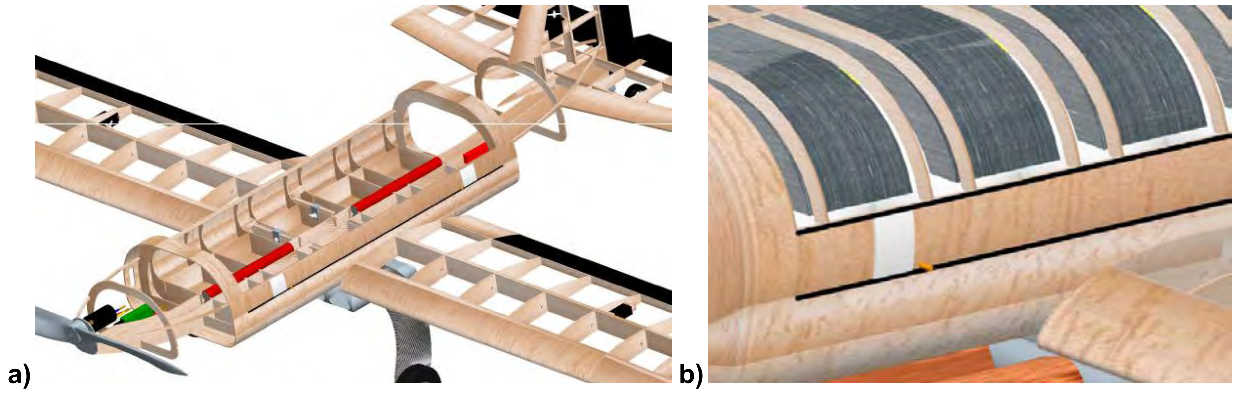


Figure 5.3 — (a) Internal Fuselage Structure; (b) Hatch Securing Mechanism

Consideration was given to the hatch construction and securing scheme. The hatch was designed as a semi-rigid structure consisting of its own bulkheads that penetrate the areas between the balls to restrict motion. Foam on the top of the hatch provides secures the two different sizes of softball. The hatch is capable of securing the softball payloads during flight maneuvers. The hatch is held to the fuselage by two elastic bands stretched over a pin, shown in Figure 5.3b above.

5.3.2 Two-Piece Wing

A two-piece wing was selected to meet the size restraint imposed by the case and still allow the optimized span of 6.5 feet to be used. The wing attachment became a critical item to design, as the attachment must be able to carry the loads through the fuselage and allow for the assembly within the five minutes of allotted time. The grid pattern of the fuselage also influenced the wing attachment method. For maximum structural efficiency, the wing spars were designed to be inserted into the fuselage. Furthermore, this design is self-aligning, and is secured with two clamps and bolts installed through the wing spars and the adjacent bulkheads, as shown in Figure 5.4a below. A lightweight joiner was designed to allow for additional wing area and without adding more weight than necessary.

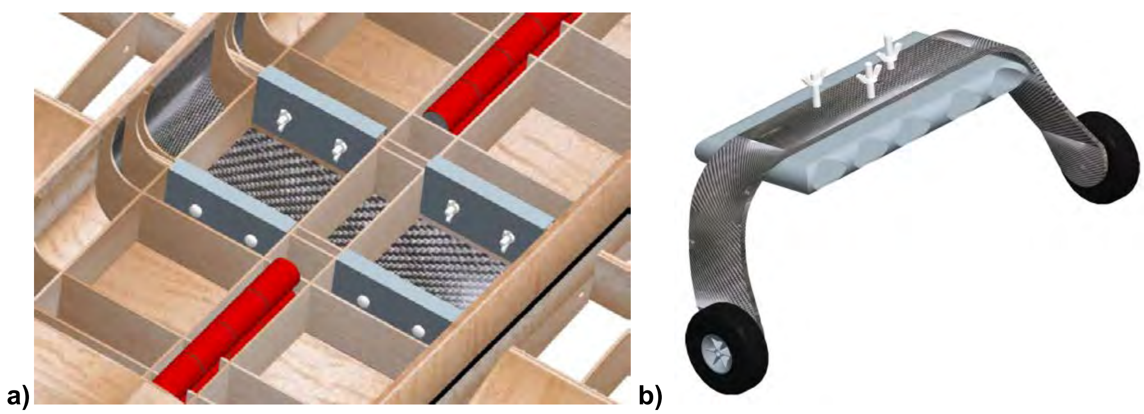


Figure 5.4 — (a) Wing Attachment Mechanism; (b) Main Landing Gear and Bat Mounting Tray

5.3.3 Landing Gear and Bat Mounting Scheme

A tail wheel configuration was selected to accommodate the external payload arrangement. Additionally, the deck angle allows for short takeoff and lower drag. The primary features of the landing gear are the main bow gear, the bat mount, and the steerable tail wheel. The main gear is constructed of carbon fiber and includes an attachment for the bats. The tail-wheel is supported by a bracket on the underside of the vertical stabilizer but is also directly connected to the rudder for directional control on the ground. The bat attachment consists of two molded plates that are drawn together with three bolts to fully secure the bats in all six degrees of freedom. The main landing gear and bat mounting assemblies are shown above in Figure 5.4b. The main gear was designed to be detachable in order to minimize the case size. The gear attaches to the fuselage in the wing carry-through area, which concentrates the load paths to a single area. The gear attaches to the fuselage via three nylon bolts and wing nuts.

5.3.4 Case

The smallest case possible was designed to minimize system weight. The size and shape of the case was determined by the overall configuration of the aircraft. The fuselage length necessitated using the full 48-inch length dimension, and the optimal horizontal tail volume required the full 24-inch width dimension. With a removable main landing gear, the case is required to be only 14 inches high. Test articles showed that the case weight scaled linearly with each of these dimensions, so a reduction was beneficial. The case configurations, both open and closed, are shown in Figure 5.5.

Figure 5.5 — Loaded Case Configuration

It was critical for the case to maintain its shape and be able to support the weight of the aircraft and essential mission components. Furthermore, the case is an essential part of the loading process; it contains a mount for the aircraft to rest upon when the main gear is detached. This mount is also used as a “fence” into which the softballs are dumped during the timed loading. Containing the balls in one location was found to be critical for a rapid loading time.

5.3.5 Propulsion and Electrical Components

Electrical and propulsion sub-systems such as motor, batteries, propeller, gearbox, speed controller, and servos were selected to satisfy the design requirements while maintaining a light weight. The propulsion system consists of an 18x10 propeller powered by a Neu 1110-1512/3Y motor through a 6.7:1 Neu P32 gearbox. A lightweight speed controller was selected based on historical performance, and a total of 5 servos were sized and selected to power control surface deflections based on hinge moment requirements summarized below in Table 5.6.

Control Surface	lb-ft	oz-in
Aileron	0.235	45.2
Elevator	0.198	38
Rudder/Steering	0.189	36.4

Table 5.6 Control Surface Hinge Moment Requirements

Weight is conserved by using two lightweight servos for the split elevator rather than one heavier one, and by combining steering capability and rudder control using one servo inherent to the tail-dragger design. A summary of the propulsion and electrical sub-system component selection is shown below in Table 5.7.

Component	Description
Motor	Neu 1110-1512/3Y
Battery	26 Elite 1500
Speed Controller	Kontronik Jazz 55-6-32
Receiver	JR Model R921 9 channel 2.46 GHz
Transmitter	Spektrum Dx6i 6 channel 2.4 GHz DSM Spread
Aileron Servos	2 Futaba S 3102 Metal Gear Micro Servo
Rudder/Steering Servo	Futaba S 3102 Metal Gear Micro Servo
Elevator Servos	2 Futaba S 3110M Metal Gear Micro Servo

Table 5.7 — Summary of Propulsion and Electrical System Components

5.4 Weight and Balance

The final aircraft design is intended to have the wing's quarter-chord at the center of gravity. As such, it is important to ensure that these two locations coincide for all the missions since the aircraft's static stability does not account for center of gravity shift. The weight distribution and aircraft balancing for all missions are represented in Tables 5.8, 5.9, and 5.10. Note that the moment arm for each component is referenced from station zero at the forward tip of the fuselage.

Component		Weight (lb)	Arm (in)	Moment (lb-in)
Structure	*Fuselage*	1.2	21.0	24.5
	Wing	0.6	17.8	10.1
	Horizontal Tail	0.2	38.3	5.9
	Vertical Tail	0.1	39.0	3.9
Propulsion	*Motor/Gearbox*	0.4	0.5	0.2
	Speed Controller	0.1	4.9	0.4
	Batteries	1.3	16.3	20.8
	Propeller	0.1	-0.4	-0.1
Avionics	*Receiver*	0.0	32.0	1.2
	Receiver Batteries	0.1	32.0	2.5
Landing Gear	*Main Gear*	0.4	17.8	7.0
	Tail Wheel	0.1	38.3	3.1
Total	***Aircraft***	**4.5**	**—**	**79.5**
	Center of Gravity (in.)		**17.8**	

Table 5.8 — Weight and Balance for Mission 1 (Empty) Aircraft Configuration

Component		Weight (lb)	Arm (in)	Moment (lb-in)
Aircraft	*Empty*	4.5	17.8	79.5
Payload	*Ten 12-in Softballs*	4.0	17.8	71.2
Total	***Aircraft***	**8.5**	—	**150.8**
	Center of Gravity (in.)		**17.8**	

Table 5.9 — Weight and Balance for Mission 2 Aircraft Configuration

Component		Weight (lb)	Arm (in)	Moment (lb-in)
Aircraft	*Empty*	4.5	17.8	79.5
Payload	*Five 20-oz Bats*	6.3	17.8	110.9
Total	***Aircraft***	**10.7**	—	**190.5**
	Center of Gravity (in.)		**17.8**	

Table 5.10 — Weight and Balance for Mission 3 Aircraft Configuration

5.5 Predicted Flight Performance

The final design performance parameters are predicted below in Table 5.11 using methods prescribed by Anderson[6,7] and will be compared to the actual parameters determined from the results of flight testing (§8.2).

Aerodynamic Parameters								
	C_{Lmax}	C_{Lmax} (flapped)	Oswald Eff. Factor	$C_{D,0}$	$(L/D)_{max}$	Min. Glide Angle (deg)	GTOW (lb)	W/S (lb/ft^2)
Mission 1	1.45	1.73	0.81	0.068	9	6	6.7	1.3
Mission 2	1.45	1.73	0.81	0.068	9	6	10.7	2
Mission 3	1.45	1.73	0.81	0.094	7.5	8	12.9	2.4
Performance Parameters								
	(T_A/W)	Max. Climb Rate (ft/min)	Max. Takeoff Distance (ft)	Cruise Speed (ft/s)	Stall Speed (ft/s)	Max. Load Factor (g's)	Turn Rate (deg/s)	Max. Bank Angle (deg)
Mission 1	0.9	1284	20	69	25	6	158	80
Mission 2	0.56	696	58	69	32	4	104	71
Mission 3	0.47	474	96	63	35	3	83	71

Table 5.11 — Summary of Predicted Aircraft Performance Characteristics

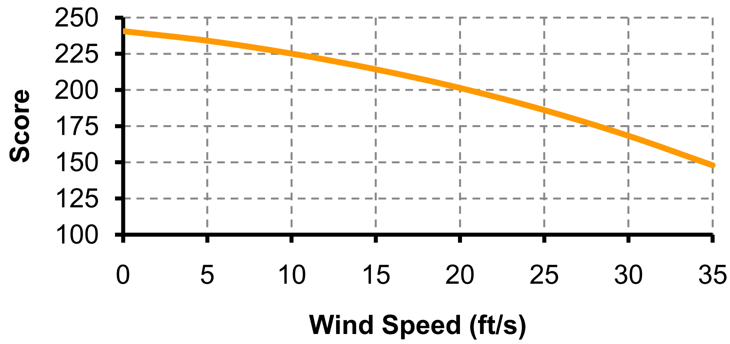
5.6 Predicted Mission Performance

The parameters of the final design were used to predict the overall performance and final score of the design.

5.6.1 Score Prediction

Final score predictions were based on baseline assumptions (Figure 5.12b). A determination of final score outcomes were based on the baseline aircraft and wind speed. However, the wind speed encountered on the day of the competition will very likely be different than those assumed. Figure 5.12a shows the effect of wind speed on flight score.

a)



b)

Baseline Aircraft	
$W_{1,ref}$ (lb)	9.5
$W_{2,ref}$ (lb)	9.5
$t_{1,ref}$ (s)	65
$t_{2,ref}$ (s)	7
$t_{3,ref}$ (s)	125
N_{ref}	5

Figure 5.12 — (a) Flight Score as a Function of Wind Speed; (b) Baseline Assumptions for Score Prediction

5.6.2 Mission1: Ferry Flight

Mission 1 flight performance is determined using a maximum cruise speed of 47 mph while the turns are taken at a slower speed and the maximum load factor, reducing turn radius to decrease course time as shown below. Even at the maximum wind condition of 30 mph, battery consumption is still only 62.4 percent, takeoff distance is predicted to be 20 feet with zero wind speed, and Mission 1 score is 48 points. The predicted Mission 1 performance profile is shown below in Table 5.13.

Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)	Energy Use (% Battery)	Current (mA)	Battery Charge (mAh)	Battery Charge (%)
Lap 1	*Takeoff*	0 - 30	20	1.1	0.8	24.4	7.5	0.6
	Climb	43	530	2.6	1.8	24.4	17.4	1.5
	Turn 1	43	170	5.5	0.6	24.4	37.3	3.1
	Cruise	72	500	6.6	4.5	19.2	35.4	2.9
	Turn 2 & 3	64	134	2.0	2.4	24.4	13.6	1.1
	Cruise	72	500	6.6	4.5	19.2	35.4	2.9
	Turn 4	64	67	1.0	1.2	24.4	6.8	0.6
Lap 2	*Cruise*	72	500	6.6	4.5	19.2	35.4	2.9
	Cruise	72	500	6.9	4.5	19.2	37.0	3.1
	Turn 5	64	67	1.0	1.2	24.4	6.8	0.6
	Cruise	72	500	6.9	4.5	19.2	37.0	3.1
	Turn 6 & 7	64	134	2.0	2.4	24.4	13.6	1.1
	Cruise	72	500	6.9	4.5	19.2	37.0	3.1
	Turn 8	64	67	1.0	1.2	24.4	6.8	0.6
	Cruise	72	400	5.6	4.5	19.2	29.6	2.5
Total		**—**	**4589**	**62**	**43**	**—**	**357**	**30**

Table 5.13 — Mission 1 Profile (Zero Wind Speed)

5.6.3 Mission 2: Softball Payload

The main focus of Mission 2 was softball loading time and system weight. The flight portion is not timed, and will be flown similarly to Mission 3. After optimizing the case for space and weight, the final result is a 3-pound case. This improves the overall system weight by 66 percent from the initial estimation of 14.7 pounds. Mission 2 score is estimated to be 88 points.

5.6.4 Mission 3: External Bat Payload

Mission 3 design requirements retain the necessity to carry 5 bats externally. The aircraft will have the capability to carry this payload cruising at 47 mph. To reduce lap time and increase score, the aircraft will slow to maximum corner velocity through the turns at 43 mph. Final handling around the turns will be compared to flight test results to determine the fastest and safest route to maximize score. Mission 3 score is estimated to be 96 points. The predicted Mission 3 performance profile is shown in Table 5.14.

Mission Phase		Velocity (ft/s)	Distance (ft)	Time (s)	Energy Use (% Battery)	Current (mA)	Battery Charge (mAh)	Battery Charge (%)
Lap 1	*Takeoff*	0 - 40	96	3.6	0.77	24.4	24.3	2.0
	Climb	43	454	6.1	1.83	24.4	41.1	3.4
	Turn 1	43	330	7.6	2.64	24.4	51.5	4.3
	Cruise	68	500	6.3	4.42	21.4	37.6	3.1
	Turn 2 & 3	63	270	4.4	5.83	24.4	29.8	2.5
	Cruise	68	500	6.3	4.42	21.4	37.6	3.1
	Turn 4	63	135	2.2	2.92	24.4	14.9	1.2
	Cruise	68	500	6.3	4.42	21.4	37.6	3.1
Lap 2	*Cruise*	68	500	7.4	4.42	21.4	43.7	3.6
	Turn 5	63	135	2.2	2.92	24.4	14.9	1.2
	Cruise	68	500	7.4	4.42	21.4	43.7	3.6
	Turn 6 & 7	63	270	4.4	5.83	24.4	29.8	2.5
	Cruise	68	500	7.4	4.42	21.4	43.7	3.6
	Turn 8	63	135	2.2	2.92	24.4	14.9	1.2
	Cruise	68	500	7.4	4.42	21.4	43.7	3.6
Lap 3	*Cruise*	68	500	7.4	4.42	21.4	43.7	3.6
	Turn 9	63	135	2.2	2.92	24.4	14.9	1.2
	Cruise	68	500	7.4	4.42	21.4	43.7	3.6
	Turn 10 & 11	63	270	4.4	5.83	24.4	29.8	2.5
	Cruise	68	500	7.4	4.42	21.4	43.7	3.6
	Turn 12	63	135	2.2	2.92	24.4	14.9	1.2
	Cruise	68	400	5.9	4.42	21.4	35.0	2.9
Total		**—**	**7766**	**118**	**86**	**—**	**735**	**61.2**

Table 5.14 — Mission 3 Profile (Zero Wind Speed)

For this mission performance analysis, the total flight score for all three missions is predicted to be 241 points.

5.7 Summary of Detail Design

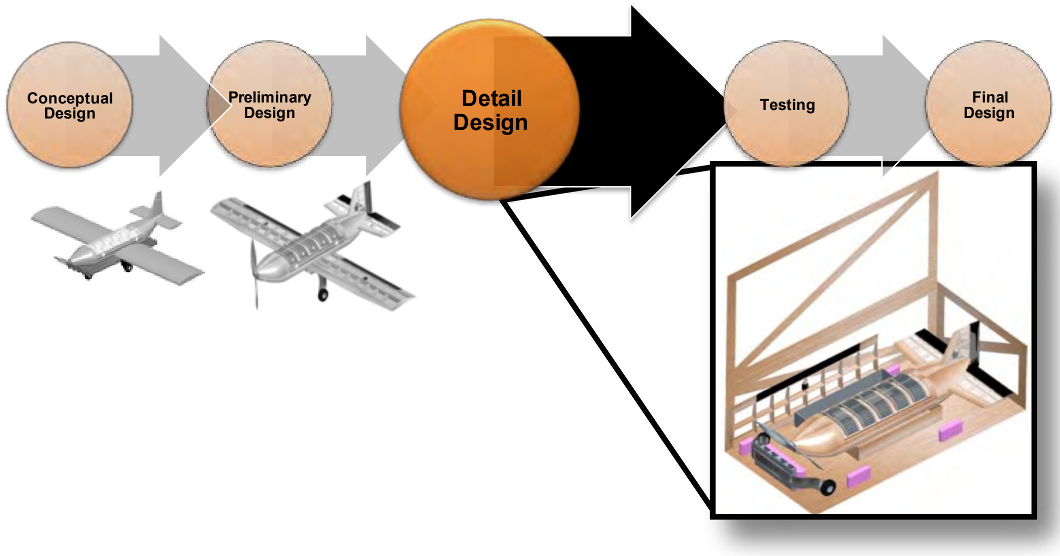


Figure 5.15 — Summary of Overall Design Process

5.8 Drawing Package

The drawing package consists of dimensioned 3-view, structural arrangement, sub-systems layout, and payload accommodation drawings.

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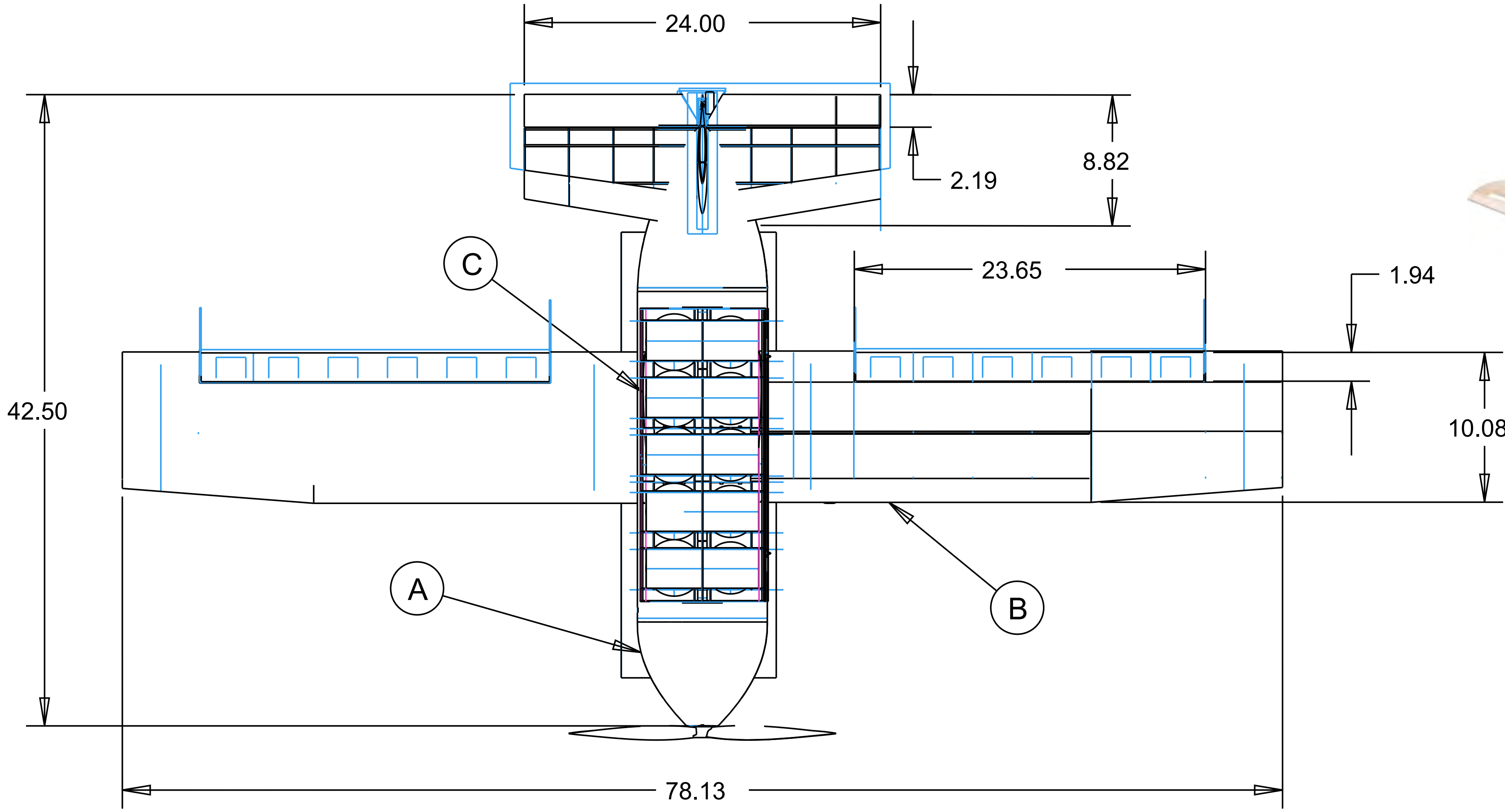

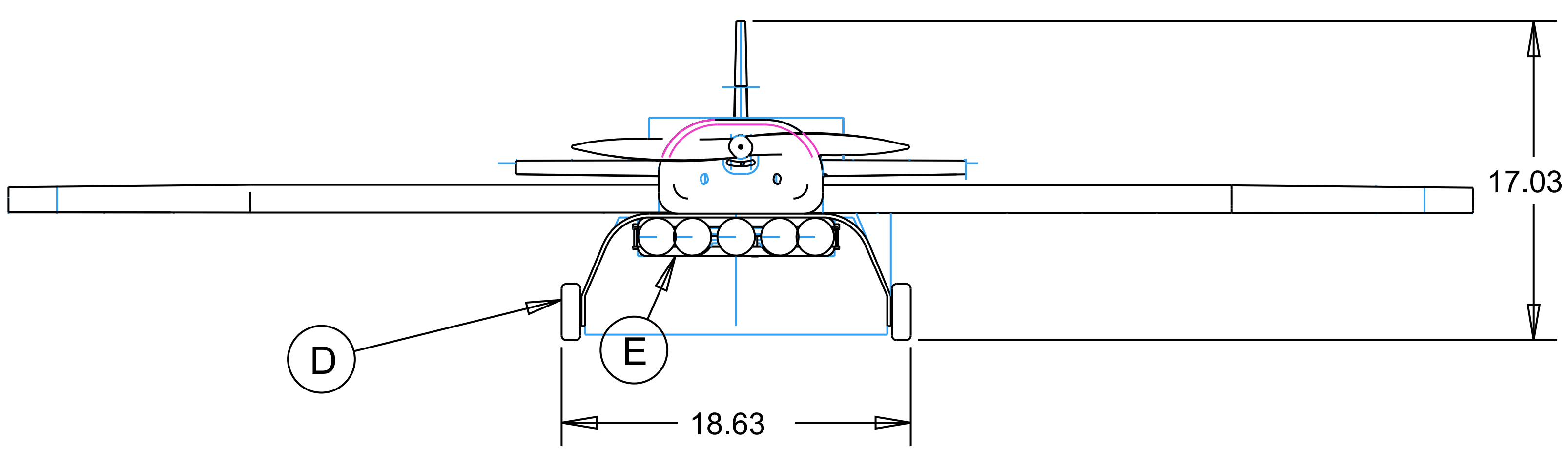

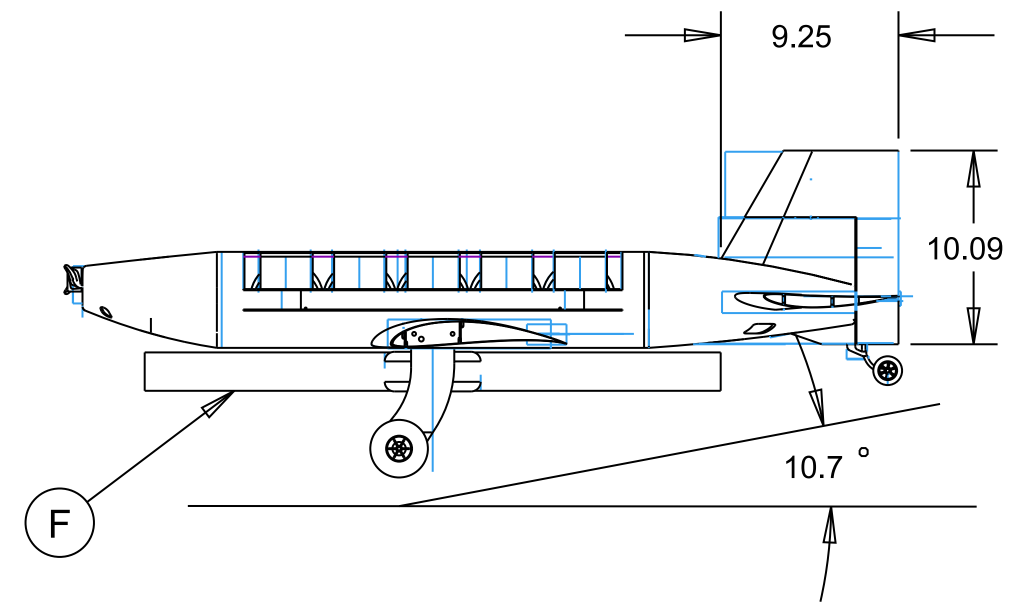

PRIMARY COMPONENTS			
A	FUSELAGE	E	EXT. PAYLOAD MOUNT
B	WING ASSEMBLY (2)	F	EXTERIOR PAYLOAD (REF)
C	INT. PAYLOAD HATCH	G	TAIL GEAR
D	MAIN LANDING GEAR		

NOTE:

ALL DIMENSIONS ARE GIVEN IN INCHES

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CESSNA-RAYTHEON-AIAA DESIGN/BUILD/FLY 2010

DOCUMENT TITLE:

AIRCRAFT 3-VIEW

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CHIEF ENGINEER: RYAN PAUL	SCALE: 1:10		SHEET: 1 of 5	

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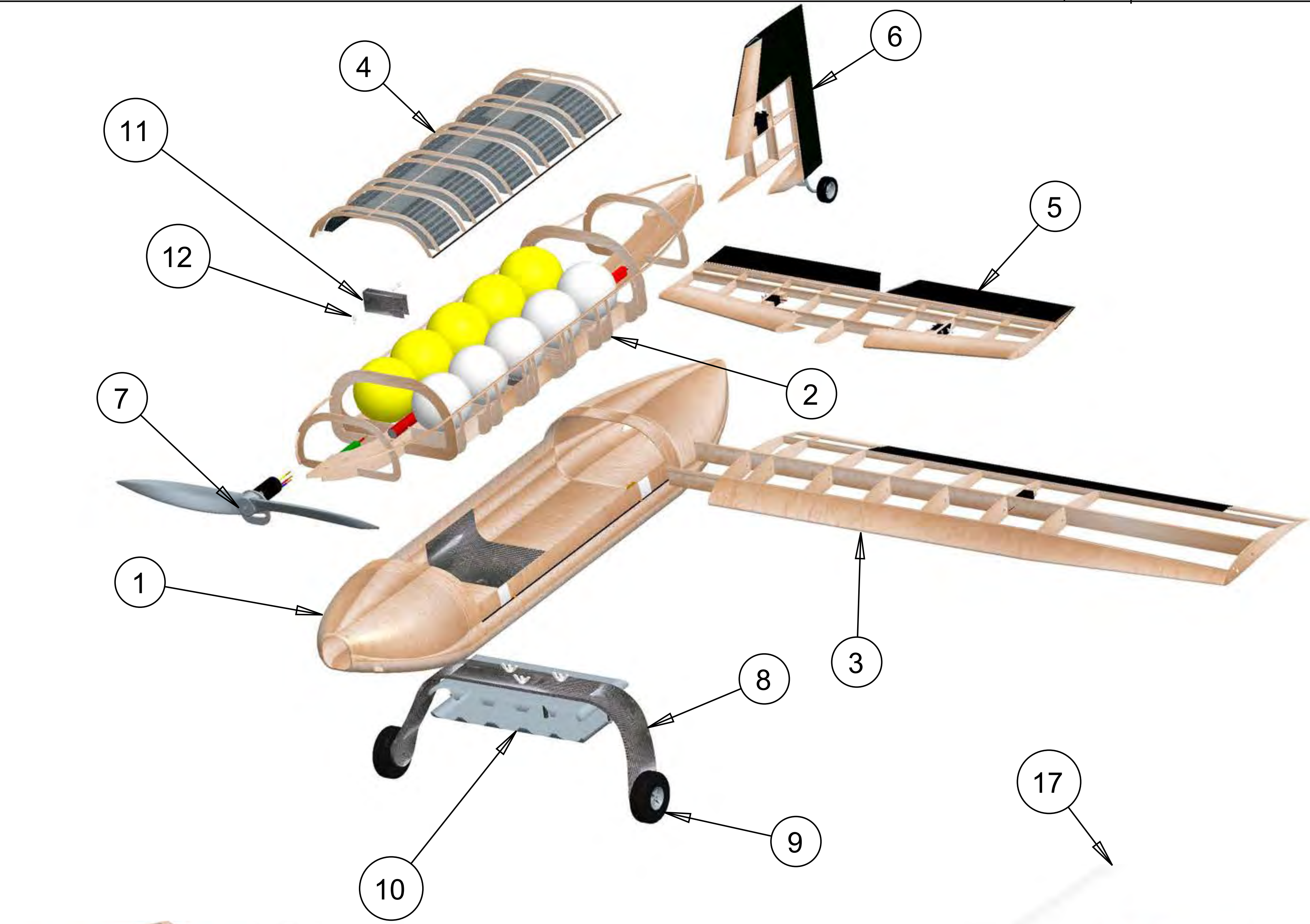
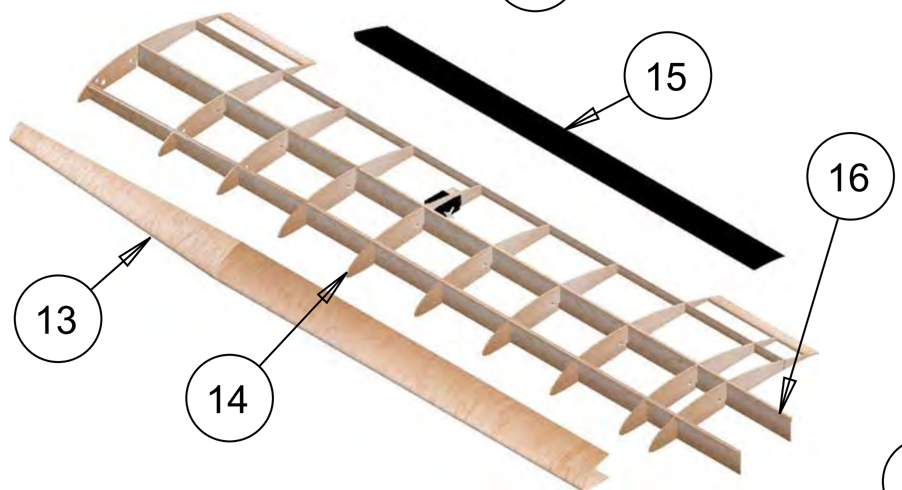


P/N	QTY	NAME	MATERIAL
1	1	FUSELAGE SKIN	COMPOSITE REINF. BALSA
2	1	FUSELAGE STRUCT.	COMPOSITE REINF. BALSA
3	2	WING ASSEMBLY	BALSA
4	1	PAYLOAD HATCH	BALSA
5	1	HORIZ. TAIL	BALSA
6	1	VERT. TAIL	BALSA
7	1	MOTOR BULKHEAD	CARBON FIBER COMPOSITE
8	1	MAIN GEAR	CARBON FIBER COMPOSITE
9	2	MAIN WHEEL	
10	1	EXT. PAYLOAD MECH.	CARBON FIBER COMPOSITE
11	4	C-CHANNEL	CARBON FIBER COMPOSITE
12	8	HARDWARE	NYLON
13	1	LEADING EDGE	BALSA
14	10	WING RIB	BALSA
15	1	AILERON	BALSA
16	2	SHEAR SPAR & WEB	COMPOSITE REINF. BALSA
17	2	HATCH SKIN	MONOCOAT
18	3	HATCH STRUCTURE	BALSA
19	5	FOAM INSERTS	.25" THICK
20	12	HATCH RIBS	BALSA



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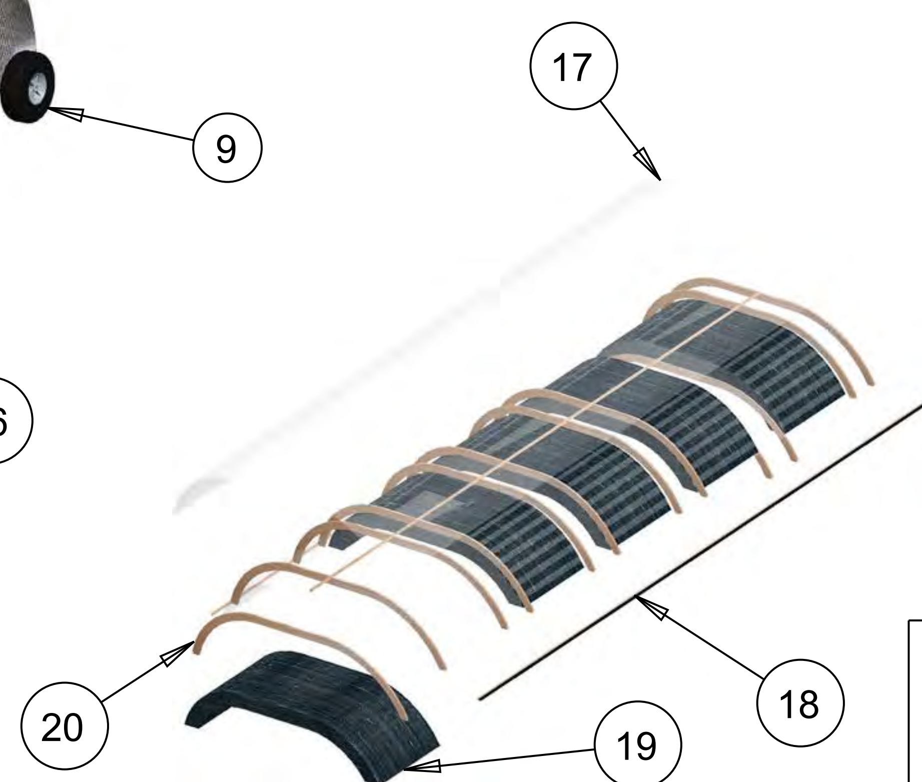

PAYLOAD HATCH DETAIL

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STRUCTURAL ARRANGEMENT

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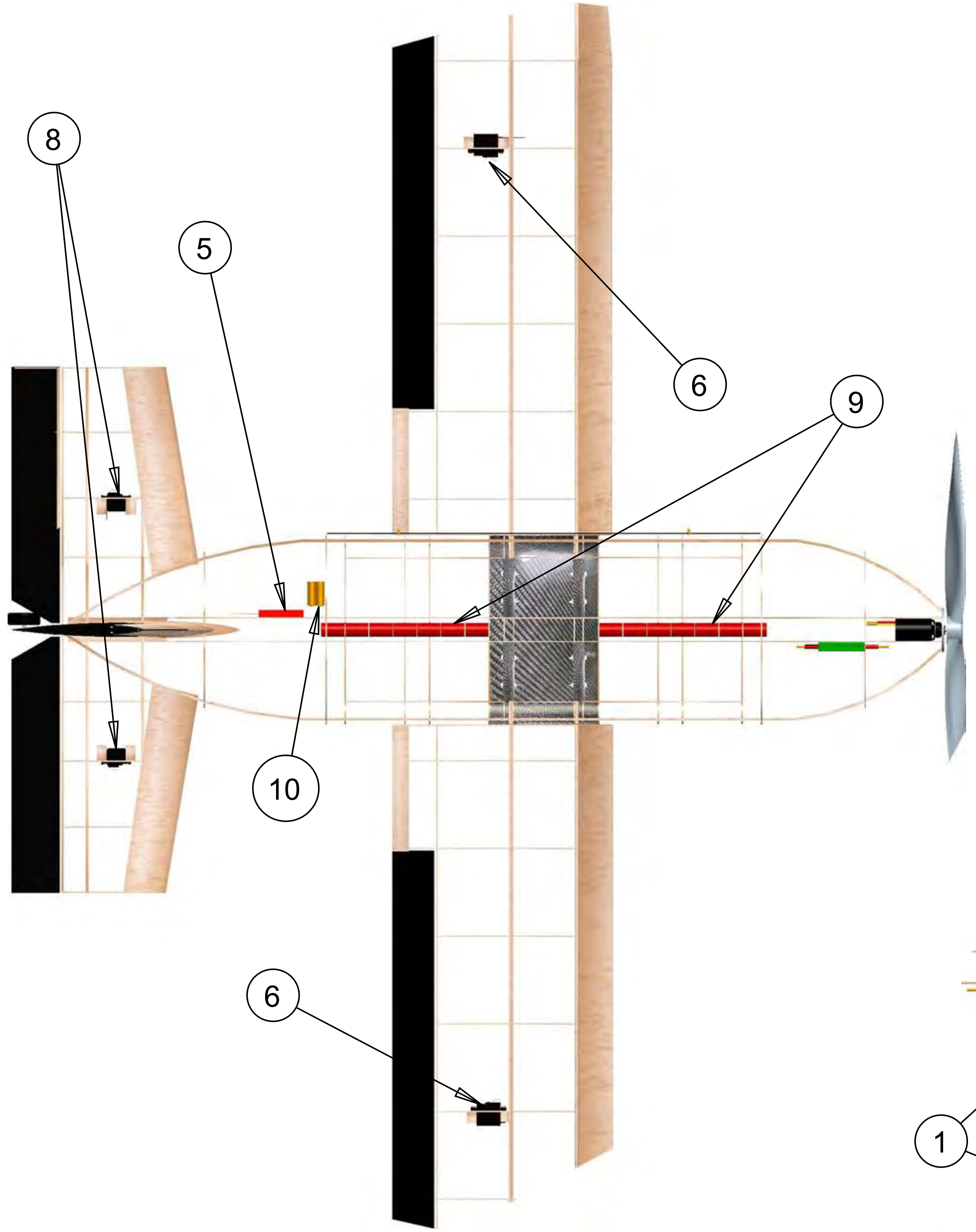
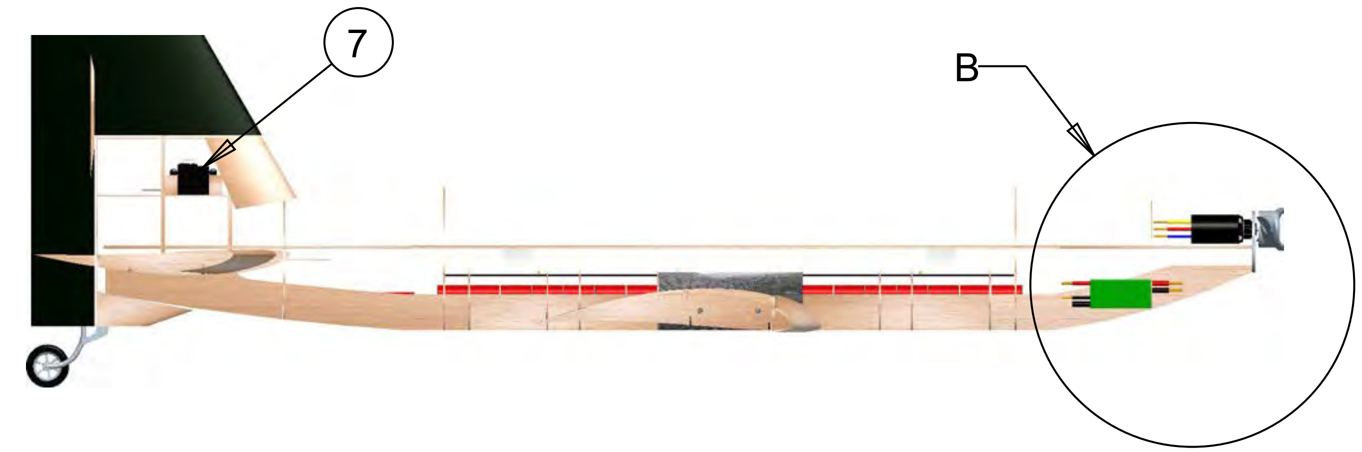
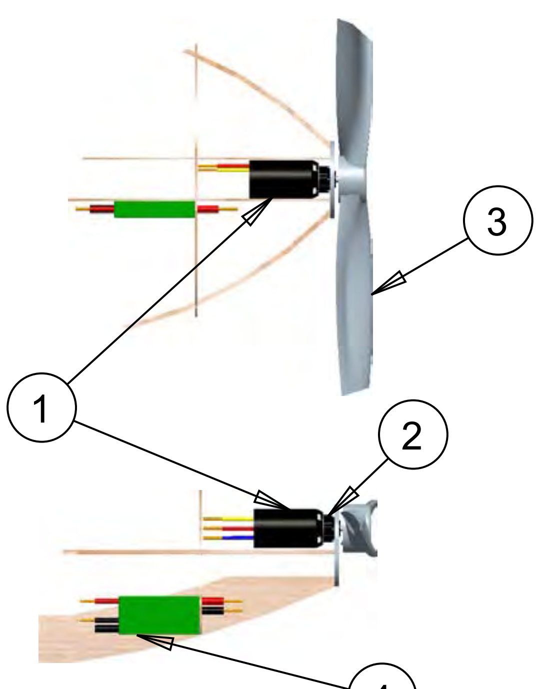




DETAIL A: SIDE VIEW



DETAIL B

1:5 SCALE

P/N	QTY	ITEM	VENDOR INFO
1	1	MOTOR	NEU MODEL 1110-1512/34
2	1	GEARBOX	NEU MODEL P32 6.7:1
3	1	PROPELLER	APC MODEL 18 x 12
4	1	SPEED CONTROLLER	KONTRONIK JAZZ 55-6-32
5	1	RECEIVER	JR MODEL 921
6	2	WING SERVO	FUTABA MODEL S3102
7	1	VERT. TAIL SERVO	FUTABA MODEL S3102
8	2	HORIZ. TAIL SERVO	FUTABA MODEL S3110
9	26	2/3A BATTERY	ELITE MODEL 1500
10	4	RECEIVER BATTERY	SPECTRUM MODEL 1100 mAh

NOTE:

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CESSNA-RAYTHEON-AIAA DESIGN/BUILD/FLY 2010

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SYSTEMS LAYOUT

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CHIEF ENGINEER: RYAN PAUL

SIZE: B

APPROVAL DATE: 2-16-2010

REPORT TITLE: DRAWING PACKAGE

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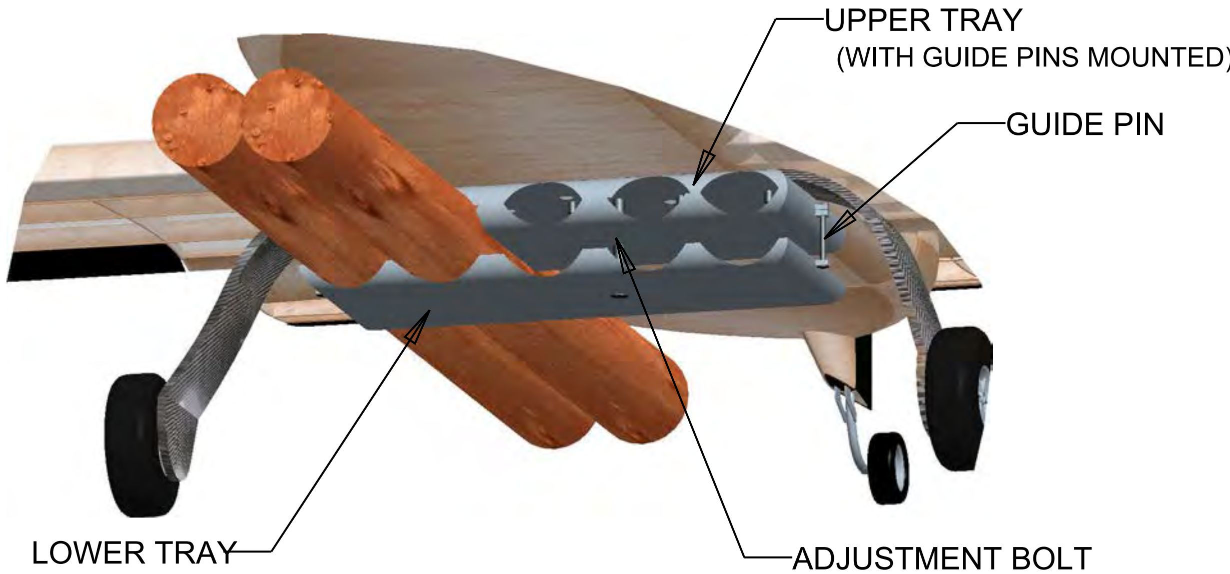
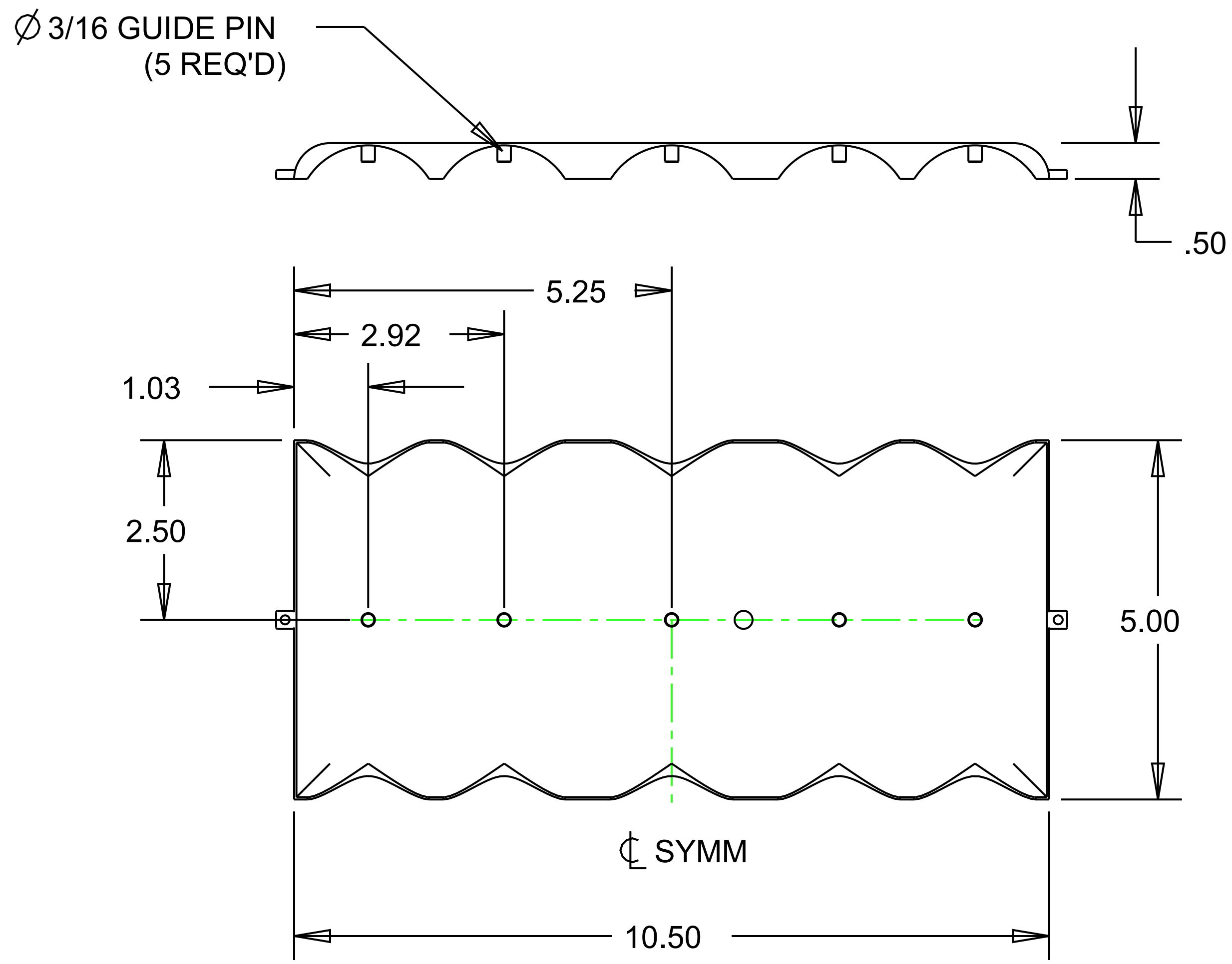
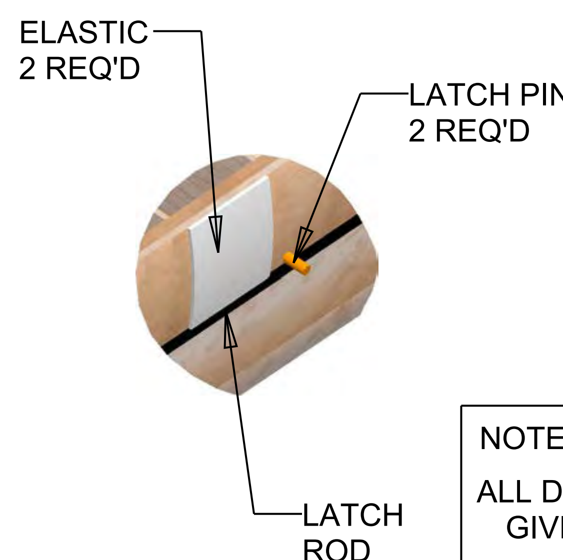


EXTERIOR PAYLOAD PLACEMENT/MECHANISM
(SHOWN WITH TRAYS IN LOADING POSITION)



UPPER TRAY DESIGN
(SAME AS LOWER EXCEPT FOR ATTACHED PINS SHOWN)



DETAIL A
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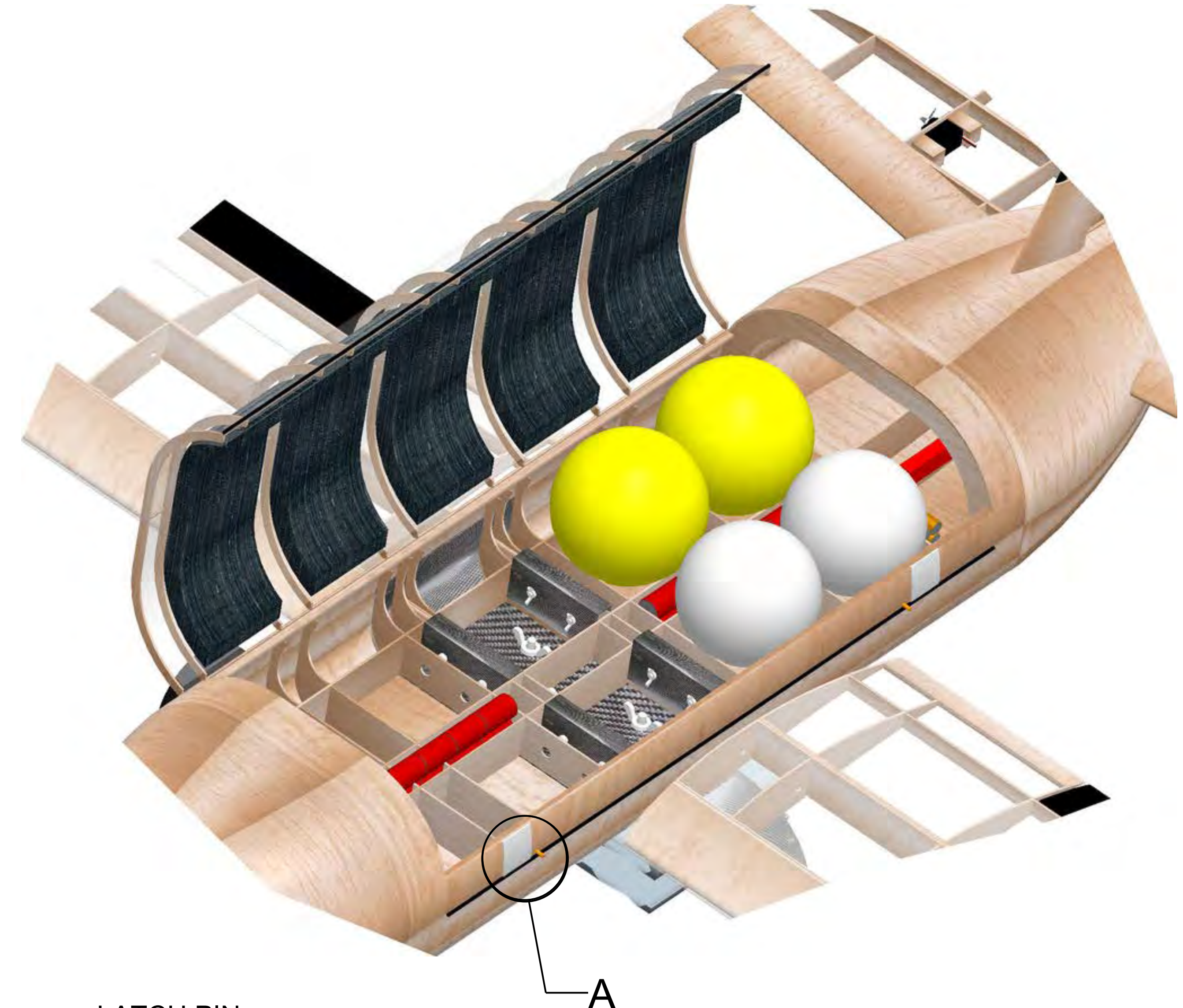

INTERIOR PAYLOAD COMPARTMENT/PLACEMENT

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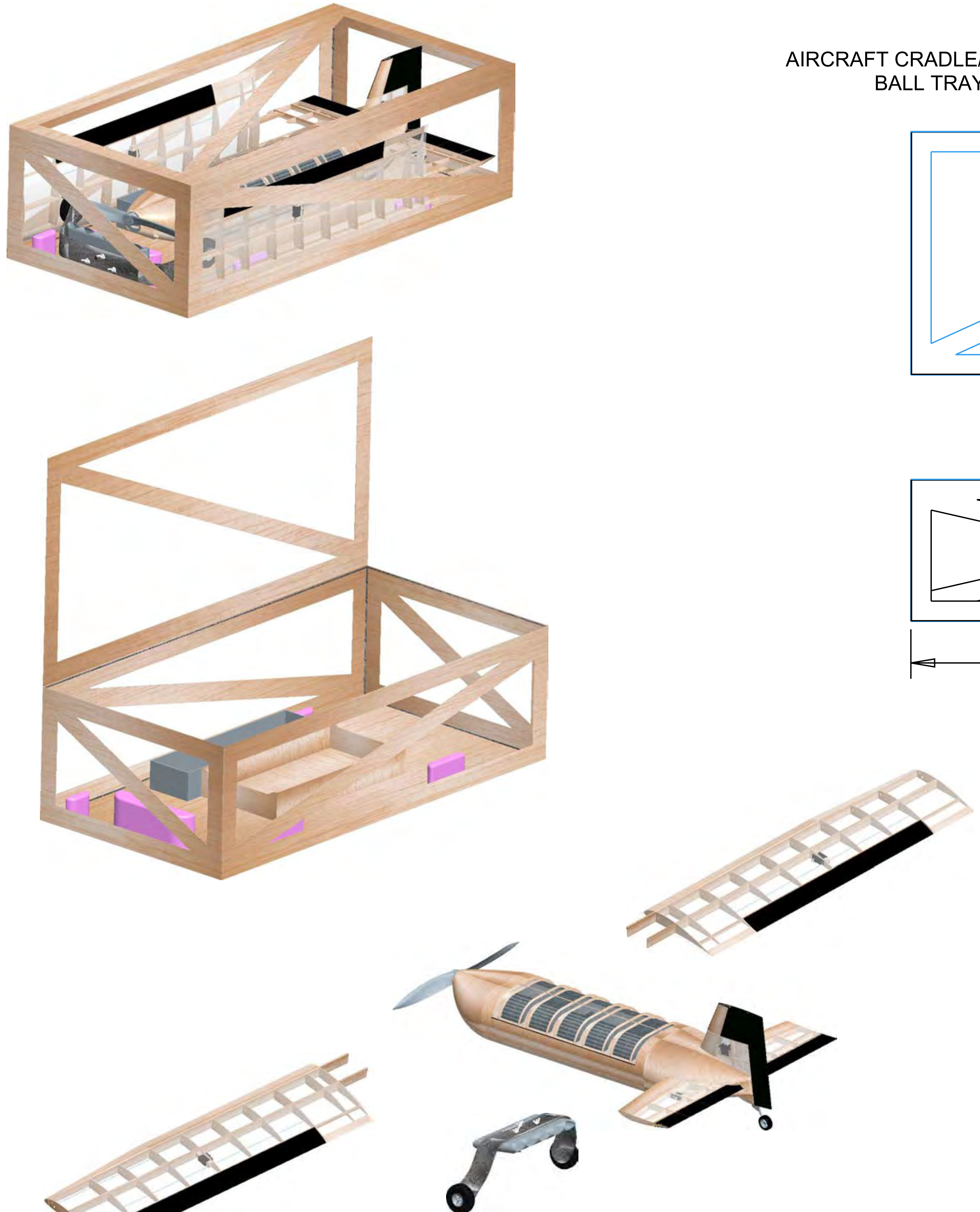
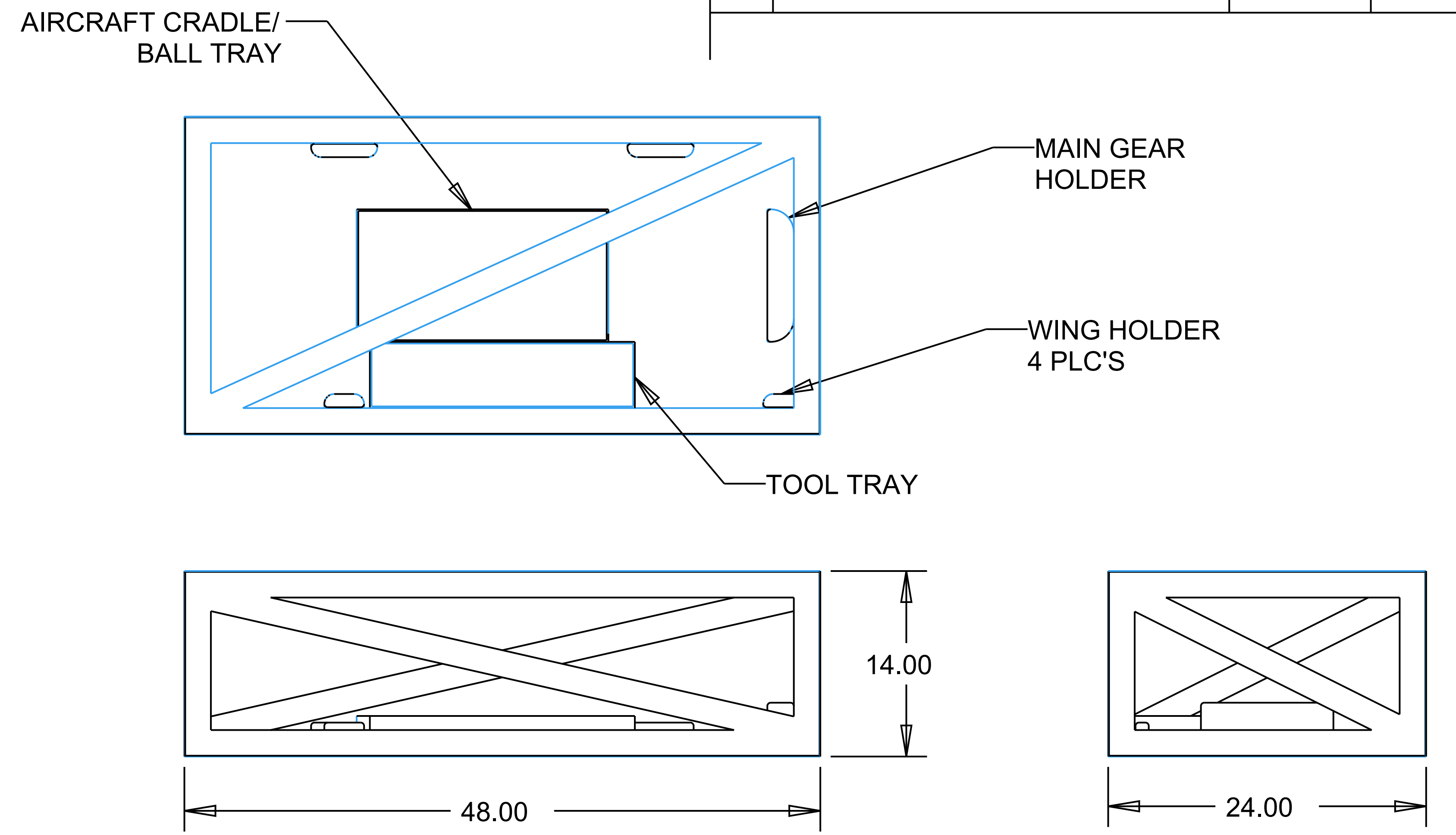
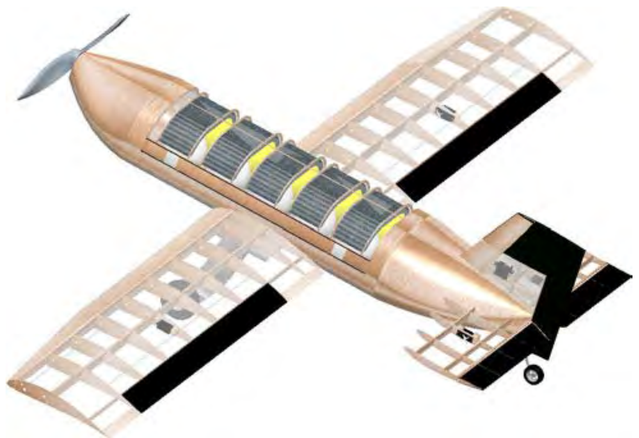
CHIEF ENGINEER: RYAN PAUL	SCALE: 1:4		SHEET 4 of 5	



ASSEMBLY PROGRESSION



BOX ASSEMBLY



NOTE:

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6.0 MANUFACTURING PLAN & PROCESSES

6.1 Investigation & Selection of Major Components

Several methods were considered for construction of major components. The mission requirements represented a dichotomy to the team; keeping weight down was vital, but score optimization required an aircraft that could handle high-*g* turns; structural efficiency was paramount.

6.1.1 Fuselage

Three alternatives were considered for the build-up of the fuselage. The options are outlined below:

- **Lost-Foam Method**: A solid foam version of the fuselage is constructed and covered in fiber-glass; the foam is then removed. This may result in a very lightweight part, but no permanent tool-ing is available.
- **Truss Build-Up Method**: The fuselage is constructed in a truss like manner out of rectangular pieces of balsa and spruce. The frame is then skinned.
- **Mold Method**: A foam plug the desired shape of the fuselage is created. The plug's invert is created into a mold which is used to layup the part.

Configuration		*Lost-Foam*	*Truss Build-Up*	*Mold Method*
FOM	***Weight Factor***		***Scoring***	
System Weight	50	-1	1	1
Mfg. Complexity	25	1	-1	-1
Mfg. Cost	25	0	-1	0
Totals	**100**	**-25**	**0**	**25**

Table 6.1 — Weighted Decision Matrix for Fuselage Fabrication Method

As shown in Table 6.1, the mold method was chosen to manufacture the fuselage. This decision was made after considering weight, construction speed, and construction skill required.

6.1.2 Wing and Tail Sections

Three methods were considered for the construction of the wing:

- **Built-Up:** Spars are constructed from lightweight composite materials and bonded to balsa ribs and shear webs. The assembly is covered in MonoKote^{TM}.
- **Foam Core:** Foam is cut to the desired wing size and shape. Fiberglass is laid over the foam to provide rigidity.
- **Molded Wing:** Full molds are built for the wing. The top half and bottom half of the wing are laid up individually. Spars, ribs, and a shear web are added to half the skin in the mold. The other skin half is then joined by closing the two molds.

The built-up method was chosen to manufacture the wings and tail surfaces. This decision was made af-ter considering weight, construction speed, and construction skill required show in Table 6.2 below:

Configuration		Built-Up	Foam Core	Molded
FOM	**Weight Factor**	**Scoring**		
System Weight	50	1	-1	1
Mfg. Complexity	25	0	1	-1
Mfg. Cost	25	1	0	-1
Totals	**100**	**75**	**-25**	**0**

Table 6.2 — Weighted Decision Matrix for Wing Construction Method

6.1.3 Case

A composite lay-up method was selected to manufacture the box. Individual panels were constructed and attached to a carbon fiber frame. Cutouts were made from the panels to reduce weight. Internal structures were built to hold the aircraft, landing gear, transmitter, assembly tools, and softballs during loading.

6.2 Manufacturing Processes

6.2.1 Fuselage

The fuselage was constructed from molded composites. Mold making (Figure 6.3a) was performed by forming a foam plug to the desired fuselage shape, and inverting this plug into a female mold using a casting resin. The mold making process was very labor-intensive, but it produced accurate, strong, and lightweight structures. Additionally, after the tooling was produced it was used over and over without re-pair. The mold halves were laid up individually using composite materials with a balsa core to produce the fuselage skins (Figure 6.3b). Internal bulkheads were bonded to one fuselage half, and then the other skin was joined.

Figure 6.3 — (a) Fabrication of Fuselage Molds; (b) Fuselage Skin Lay-Up in Molds

6.2.2 Wing

The wing is built in two halves. Each half consists of a front and rear spar that use spruce to carry compression and carbon fiber to carry tension. Balsa ribs and balsa sheeting form the leading edge. The wing was built on a jig for alignment. Each rib was cut on an NC Mill to keep the desired airfoil shape. The spars and shear webs were laid up individually before inserting ribs at the appropriate locations (Figure 6.4). MonoKote™ was then applied to the wings to provide a smooth aerodynamic surface. On each wing half, the spars extend out of the wing 5 inches. The front and rear wing spars are accepted into a fuselage c-channel. The C-channel runs the full fuselage width in order to promote structural efficiency.

Figure 6.4 — (a) Initial Lay Out of Wing Structure Parts; (b) Assembly of Wing Structure

6.2.3 Landing Gear

The landing gear was constructed via a layup from a male mold. The mold was constructed out of high density foam which provides a very smooth surface. The final construction layup of the main bow gear consisted of a balsa core inside layers of carbon fiber. One challenge was molding the landing gear with the swept forward shape demanded by the tail-dragger configuration of our aircraft. To accommodate this shape, the balsa core was preformed into the desired geometry before being sandwiched in carbon fiber. The outline of the core was then cut out in carbon fiber. The tail-wheel was constructed from a steel wire. The wire was shaped and inserted through a bracket on the fuselage bottom for structural support and then formed to pass into the rudder to provide directional control.

6.2.4 Case

The case was constructed to be as light as possible. A study was performed to decide what the lightest case material would prove to be (§4.4.2). The lightest box was constructed as a carbon frame and fiberglass reinforced balsa with cut-outs covered by MonoKote™. The carbon frame, in cross-section, was designed as a C-channel for stiffness. After laying up the carbon frame, the balsa sidewalls were attached and holes covered with MonoKote™.

6.3 Milestone Chart

In Figure 6.5 below, the chart/schedule shows the planned and actual completion of key milestones for the first prototype construction from February 8 to February 24.

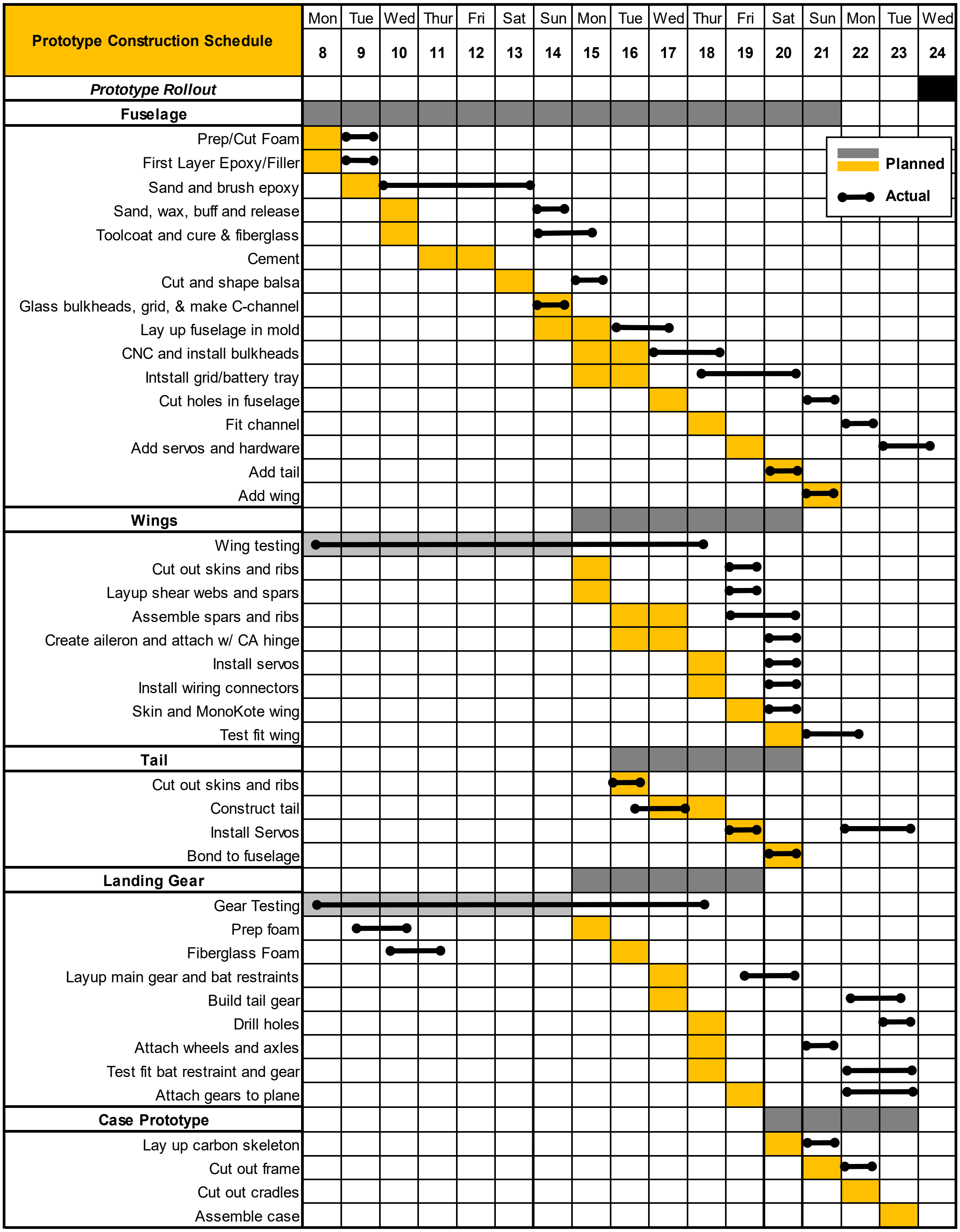

Figure 6.5 — Schedule for Prototype Construction

7.0 TESTING PLAN

Testing of components and sub-assemblies was performed, in part, during conceptual, preliminary and detail design process to ensure functionality of each part and the integrated system, in addition to flight testing following the design phase. These tests were completed to validate design choices made from analysis and to better predict and refine actual performance.

7.1 Test Objectives

Test objectives were described within the test plan to ensure that testing was accomplished to meet the specific goal of creating the highest competition scoring aircraft. The objectives included testing of components, propulsion system, system, softball loading, and flight testing.

7.1.1 Structural Components

Objectives were developed for the aircraft structural component testing of the wing, landing gear, and bat attachment and are given below in Table 7.1.

Wing

Wing Structural Test Objective Checklist			
Test	*Objective*	*Description*	*Test Results/Comments*
3 Point Bending	Determine max. moment sustainable with spar config.	2 ft span; weigh test piece; load to failure	
	Determine max. moment sustainable with varying spar cap sizes/materials	2 ft span; weigh test piece; load to failure	
	Determine the max.moment sustainable with varying shear web materials	2 ft span; weigh test piece; load to failure	
Cantiliver attachment	Determine the max.g-load sustainable	Load at 18 in. from root; weigh wing; load to failure; vary rib spacing, material	
	Determine the maximum g-load sustainable at attachment point	Loaded at 18 in. from root; weigh wing; load to failure; vary attachmen configuration, hardware, materials	
Tip test	Simulate a 2.5-g tip test for competition requirement	Weight aircraft with max. payload; lift aircraft by wing tips at quarter-chord	

Table 7.1 — Checklist for Wing Test and Objectives

Landing Gear

Vertical Drop Test Objectives

1. Determine maximum "worst case" load capabilities for landing gear designs.
2. Determine failure modes to influence redesign.

Pivot Beam Drop Test Objective

1. Determine load capabilities for simulated landings at incidences and rotations.

Bat Attachment

Six Degree of Freedom Test Objective

1. Determine capability of design to restrain 5 bats in six-degrees of freedom.

Structural Efficiency Test Objective

2. Determine weight and stiffness to optimize design

7.1.2 Propulsion System

In order to validate propulsion components and optimize the propulsion system for the aircraft, individual test objectives were outlined for the propeller, motor, and batteries. The objectives are summarized in Table 7.2

Propulsion Test Objective Checklist			
Test	*Objective*	*Description*	*Test Results/Comments*
Battery Cycle Charge/Discharge	Determine actual charge capacity of batteries	Use Computerized Battery Tester (CBT); fully charge; discharge at set current level; monitor voltage output, capacity	
	Measure actual voltage, current, and resistance of batteries		
Propeller Test	Determine capabilities for several propeller types, pitch, and diameter	Constant motor and power supply; wind tunnel test at varying dynamic pressure; measure output torque, thrust, RPM, voltage and current	
Motor Test	Determine actual motor performance with selected propeller	Wind tunnel test at varying dynamic pressure; measure output torque, thrust, RPM, voltage, current; observe heat output	
Motor/Propeller System Test	Determine actual performance of selected motor/gearbox/propeller system	Wind tunnel test at varying dynamic pressure or flight test; measure output torque, thrust, RPM, voltage, current; observe heat output	

Table 7.2 — Checklist for Propulsion Tests and Objectives

7.1.3 System

With the aircraft and case assembled, tests were performed to discern the overall capability of the system to fulfill mission requirements as a competitive design. The objectives for system tests are summarized below.

<u>Case</u>

Structural Integrity Verification Objective

1. Determine the capability of the case to carry heaviest mission capacity.
2. Verify exterior material durability and integrity.

<u>Loading</u>

Loading Time Trials Objective

1. Conduct practice time trials to optimize loading scheme, loading time, and teamwork.

<u>Aircraft</u>

Tip Test Objective

1. Determine the capability of the aircraft to withstand a simulated 2.5-*g* load tip test at maximum payload capacity.
2. Determine the actual center of gravity of the aircraft for each mission payload configuration.

Flight Test Objectives

1. Given in Table 7.3.

Flight Test Objective Checklist			
Test	*Objective*	*Description*	*Test Results/Comments*
Taxi Test	Determine taxi handling characteristics	Perform powered taxi to test ground handling characteristics	
Flight Control Check	Set flight control deflections/differentials to design	Check control surface deflections, limits, coordination	
Empty Flight	Determine capability of flight and stability	Check for t/o < 100 ft, 2 lap maximum, check for handling/control, stall deflections, turn loads	
Weighted Flight	Determine ability to fly carrying weight	Incremental weight addition, fly and check for handling, control, t/o < 100 ft	
Softball Payload Flight	Determine aircraft mission 2 capability	flights include 6-10 softball payloads, check: C.G., t/o < 100ft, control, flight speeds, turn loads	
Bat Payload Flight	Determine aircraft mission 3 capability	5 bats externally mounted, check assembly time, CG location, t/o < 100ft, flight speeds, turn loads	
Extreme Limit Flight	Determine aircraft limits	5 bat payload, full power turns, full power cruise, max speeds and turns, push to failure	
Course Handling Test	Practice course handling to optimize flight time	Flight in M1 and M3 config., test level, vertical, combined turn maneuvers for fastest lap speed	

Figure 7.3 — Checklist for Flight Tests and Objectives

7.1.4 Pre-Flight Checklist

A pre-flight checklist (Figure 7.4) was created to prevent unnecessary error, ensure safety, and improve mission performance during the flight tests.

PRE-FLIGHT CHECKLIST	
Aircraft Structural Integrity	
Visually inspect each part for damage	
☐ Wing	☐ Control Surfaces
☐ Fuselage	☐ Landing Gear
☐ Horizontal Stabilizer	☐ Hatch
☐ Vertical Stabilizer	☐ Payload
Avionics and Controls	
Ensure all wires are firmly connected and component perform as needed	
☐ Servo Wiring	☐ Zero All Servos
☐ Avionic Power Test	☐ Receiver Battery Charge
☐ Range Test	☐ Receiver Batter Temp.
☐ Servo Test	☐ Failsafe
Propulsion	
Check all power/charge levels and ensure motor peforms as desired	
☐ Battery Charge	☐ Tach Motor RPM
☐ Battery Temperature	☐ Verify Motor Power
☐ Motor Wiring	
Final Checks	
Alert all in the area that the test will begin	
☐ Ground Crew	☐ Pilot and Spotter
☐ Final Visual Inspection	

Figure 7.4 — Pre-Flight Checklist

7.2 Master Test Schedule

The complete test program was formatted in a Gantt chart (Figure 7.5) to show testing and timeline progression to better manage the critical schedule.

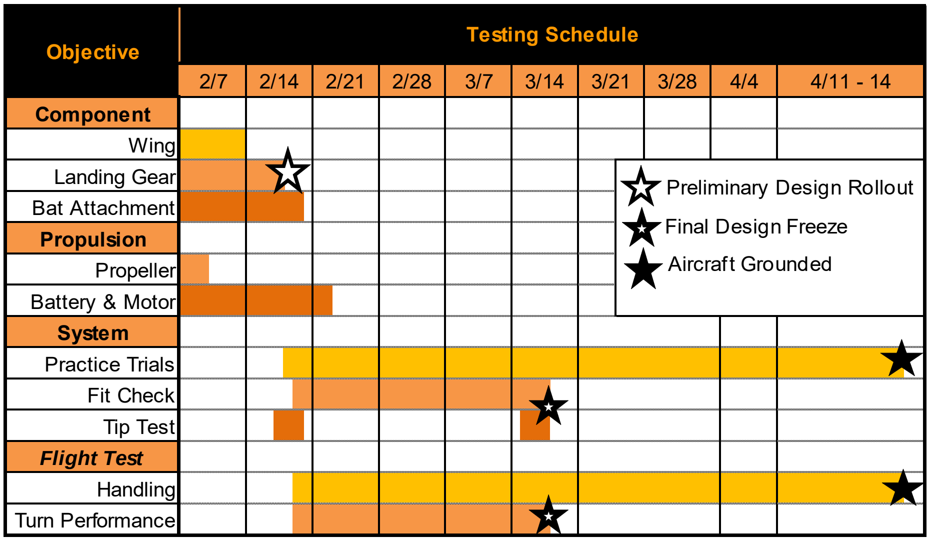

Figure 7.5 — Testing Schedule

8.0 PERFORMANCE RESULTS

Once testing was complete, the results analyzed and used to make and corrections or adjustments to the respective components or sub-systems if necessary. The results of those tests and the associated design changes, if they were required, are detailed below.

8.1 Performance of Sub-Systems

8.1.1 Wing

Tests were conducted at first using wing section mock-ups and subsequently underwent three-point bending tests (Figure 8.1a) to determine their ultimate load capacity as well as to verify material properties of the wing. The best-performing wing section test was able to sustain 55 pounds of applied force, equivalent to a 25 pound-foot moment before failing due to delaminated shear webs (Figure 8.1b)

Figure 8.1 — (a) Wing Section Mock-Up Undergoing Bending Test; (b) Failure of the Wing Section

Then, further tests were performed using full wings under a cantilever loading scheme to simulate flight conditions (Figure 8.2a)

Figure 8.2 — (a) Full Wing Cantilever Test; (b) Subsequent Failure of the Wing

This full-scale wing was loaded to 25 pounds at a distance of 18 inches from the root, imparting an equivalent moment of 38 pound-feet before failing due to buckling of the main spar very near the root (Figure 8.2b). This moment is well within the maximum expected root moment of 28.6 pound-feet (§4.4.2). The test results indicate that the wing's strength was sufficient for all flight conditions. Continued testing focused on increasing structural efficiency by reducing wing weight.

8.1.2 Landing Gear

Prototypes for the main landing gear were made from carbon fiber composite and an aluminum alloy. After initial testing, the carbon composite gear was selected. The carbon gear allowed for easy implementation bat-mounting mechanism, but also had to be designed to support a "hard landing" when the aircraft is fully loaded. This load scenario was determined to be ground impact at a velocity 20 percent greater than the stall speed along a steep glide slope of 10 degrees. It was calculated that an equivalent load condition could be simulated by dropping a loaded gear from a height of one foot. Early test articles showed failure from poor energy absorption. Acceptable results were achieved only after using a bow shape in the gear. Once this was implemented, the gears were capable of handling much more load. For both tests of thick and thin lay-ups, the gear was capable of withstanding a weight of 12.5 pounds before failing. The load distribution within the gear is shown in Figure 8.3b.

a) b)

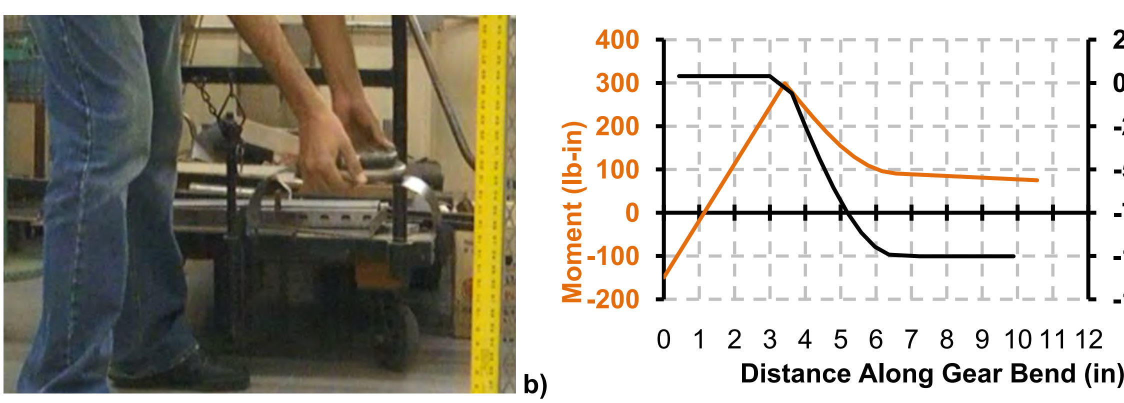

Figure 8.3 — (a) Drop Test of Unbowed Gear Prototype; (b) Axial Loads and Bending Moment Within Gear Along Its Shape

8.1.3 Propulsion

Batteries

Batteries were initially cycled through using a defined charge/discharge method. For each discharge, the cells were connected to the CBA discharger and capacity/voltage charts were generated (Figure 8.4) and were used to create optimized battery packs. Once these packs were constructed, they could then be analyzed as a system using the CBA. Due to the high voltage caused by discharging 26 cells in series and limited power capabilities of the CBA, a very low current had to be used. Battery packs were discharged at 35.1 V and 2 amps. Capacity numbers gathered during these discharges were reduced by 10 percent to account for the fact that actual flight currents will be much higher than the test current.

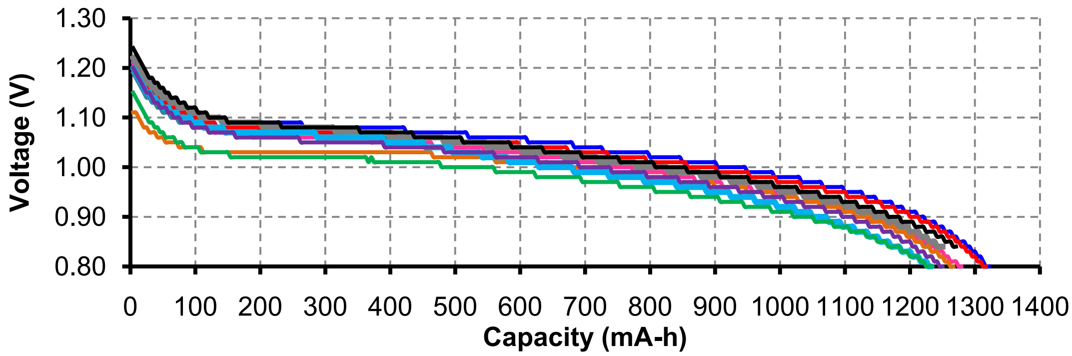

Figure 8.4 — Test Results for Battery Discharging Cycles

Flight testing was conducted to test the battery output during typical aircraft operation. Figure 8.5 shows the payload and flight duration for three initial test flights. The primary goal was to test battery usage to ensure the aircraft will successfully complete the missions without losing power. The first test flight was flown with an empty payload to simulate Mission 1. Three laps around the course were completed at low throttle to establish trim and proper handling of the aircraft in flight. Upon complete of the test, the capacity propulsion batteries were measured. Two additional test flights were flown with heavier payloads to measure battery usage for different flight times to simulate the required missions (Figure 8.5).

Test	Payload	Duration (s)	Battery Charge (mAh)	Battery Use (%)
Flight 1	Empty	144	310	25.8
Flight 2	6 Softballs	187	450	37.5
Flight 3	10 Softballs	125	460	38.3

Table 8.5 — Battery Charge Consumption for First Test Session

The total available charge of the 26-cell battery system is over 1200 mAh, well more than what was required for the test flights. Only 2 laps were flown in Flight Test 3. For a full mission, it can be estimated between 650 to 700 mAh of capacity would have been used.

Propeller

A wide variety of propellers were examined during wind tunnel testing. Figures 8.6a, b, and c below show performance characteristics of the optimum propellers for the aircraft for three propellers of various diameter and pitch.

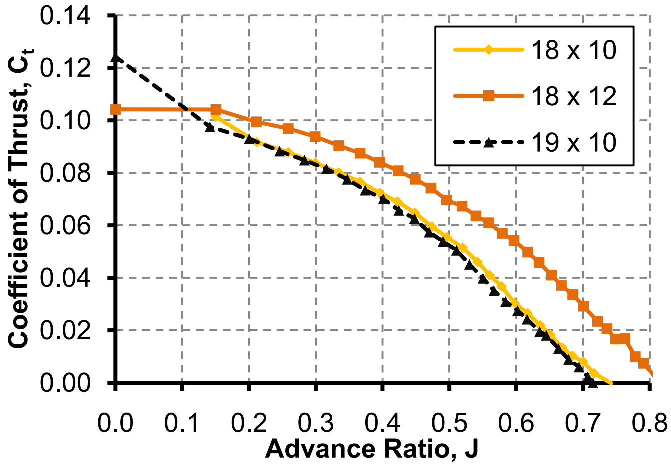

Figure 8.6a — Thrust Coefficient as a Function of Advance Ratio

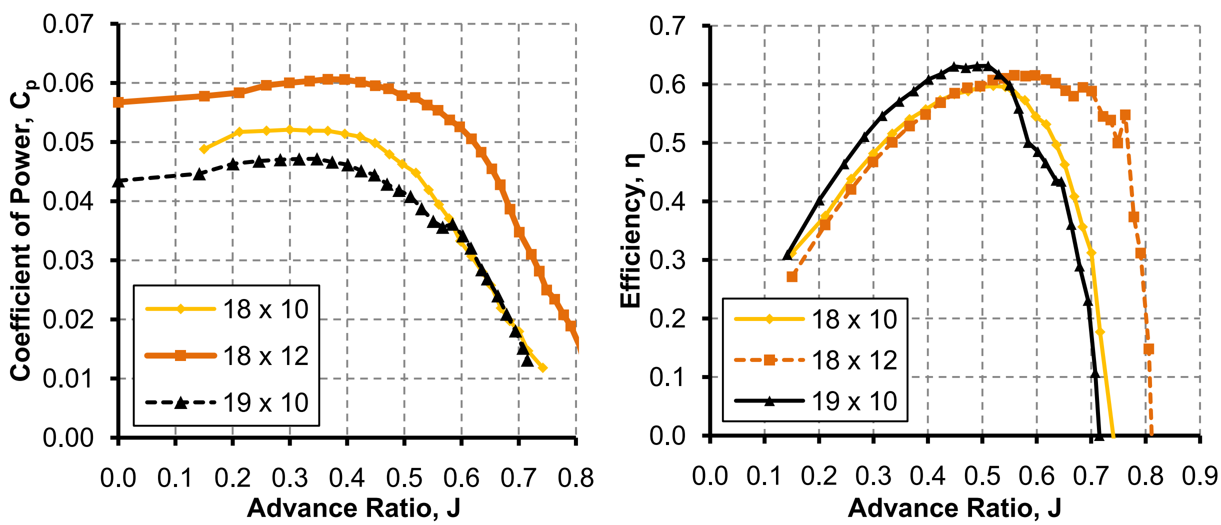

Figure 8.6b and c — (b) Coefficient of Power and (c) Propeller Efficiency as Functions of Advance Ratio

8.2 Performance of Total Aircraft System

Flight and ground testing of the prototype aircraft was conducting to verify and validate the predicted performance of the aircraft and make modifications as necessary. As of the date of this report, flight tests performed indicated a need for structural changes to the aircraft based on pilot inputs concerning stability and control. The structural changes needed include: increasing vertical tail size for directional stability, decreasing rudder size for control, increasing aileron and wing stiffness due to perceived aero elastic effects, and minor material and hardware changes to landing gear axles and wing carry-through. Actual take-off performance for Mission 1 and Mission 2 with a 10-softball payload at 10 mph wind conditions during testing was compared to predicted estimates for the detail design as shown in Table 8.7.

Take Off Performance Comparisons			
Mission	Predicted (ft)	Actual (ft)	% Difference
1	19	15	-21
2	22	30	36

Table 8.7 Predicted vs. Actual Take-Off Performance

System energy consumption was analyzed after flying simulated Mission 1 and 2 courses. The flights were carried out at 30 percent throttle over 2 laps in a 10 mph wind, which is one lap short of Mission 2, but results show 73 and 62 percent battery remaining (§8.1.3). Conservative extrapolation of the results considering 90 percent throttle and an additional lap for Mission 2 gives 47 and 34 percent battery remaining for Mission 1 and Mission 2 respectively. This is a 30 and 50 percent reduction compared to predicted values for Mission 1 and 2. Deviation from predicted results can be attributed to gusting wind conditions and additional drag due to manufacturing errors. Further flight testing will be conducted to simulate Mission 3 requirements. Table 8.8 shows the results of the performance tests for each mission and the outcomes and any modifications that must be made to increase the mission score.

Aircraft Performance Tests		Test Results	Outcomes
General Aircraft Tests	C.G. Test	Aircraft C.G. located at the wing quarter cord	C.G. location as predicted for stability
	Taxi Test	Ground Handling qualities are good	No modifications necessary
	Control Deflection	Control surface deflections checked before each flight	Controls deflected in correct position with control input
Mission 1 Test Flight	Empty Payload Ferry Flight	Aircraft takeoff distance: 10 feet into a 10 mph headwind	Well under 100 foot takeoff distance limit
		Required Elevator Trim slightly higher than predicted1 degree up	Elevator Trim Set at 3 degrees up
		Aileron trim was set to 0	As predicted
		Roll control was observed to be sluggish in the turns	Increase the differential between the ailerons greatly improved the performance
		144 second flight-Battery Discharge: 310 mAh	Sufficient battery charge to Complete Mission 1
Mission 2 Flight Test	6 Softball Payload Flight	Aircraft takeoff distance: 45 feet into a 10 mph headwind	Well under 100 foot takeoff distance limit
		Directional instability and tail wag (Rudder Horn adversely effected by air resistance causing instability)	Reduce the size of the rudder horn and position behind rudder hinge moment
			Increase Rudder Longitudinal Length to increase vertical tail planform area
			Ventral Fin under the fuselage to increase vertical tail volume
		187 second flight-Battery Discharge: 450 mAh	Sufficient battery charge to Complete Mission 2
Mission 2 Flight Test	10 Softball Payload Flight	Aircraft takeoff distance: 60 feet into a 10 mph headwind	Well under 100 foot takeoff distance limit
		Directional instability and tail wag (Rudder Horn adversely effected by air resistance causing instability)	Reduce the size of the rudder horn and position behind rudder hinge moment
			Increase Rudder Longitudinal Length to increase vertical tail planform area
			Ventral Fin under the fuselage to increase vertical tail volume
		125 second flight (2-lap) -Battery Discharge: 460 mAh	Sufficient battery charge to Complete Mission 2
Mission 3 Flight Test	5 Bats Payload Flight	Aircraft takeoff distance:75 feet into a 10 mph headwind	Well under 100 foot takeoff distance limit
		Required Elevator Trim slightly higher than predicted1 degree up	Elevator Trim Set at 4 degrees up
		202 second flight-Battery Dischargh: 570 mAh	Sufficient battery charge to Complete Mission 3

Table 8.8 — Summary of First Flight Test Session Results and Design Change Outcomes

Several loading trials were conducted to decrease the loading time for Mission 2. Figure 8.9 shows the progression as loading time was improved from the predicted 8 seconds to 7 seconds.

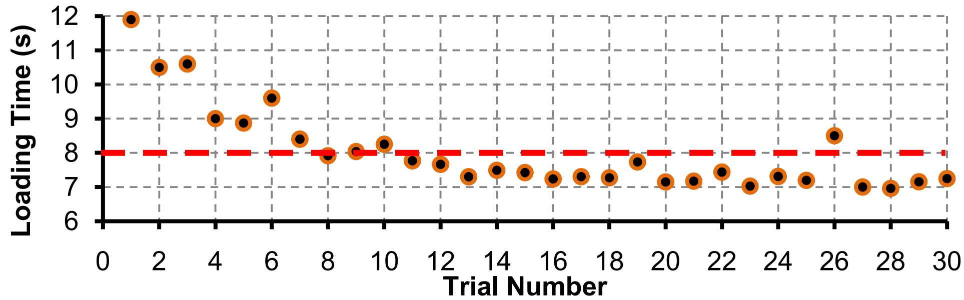

Figure 8.9 — Softball Loading Time Trial Results Against Predicted Time

In Figure 8.10, the overall progress of the system design as of the testing phase is given. The final system design will be the end result of the lessons learned from the tests yet to be completed. It will be the culmination of all the work conducted by the OSU Black Team and their entry for the DBF contest.

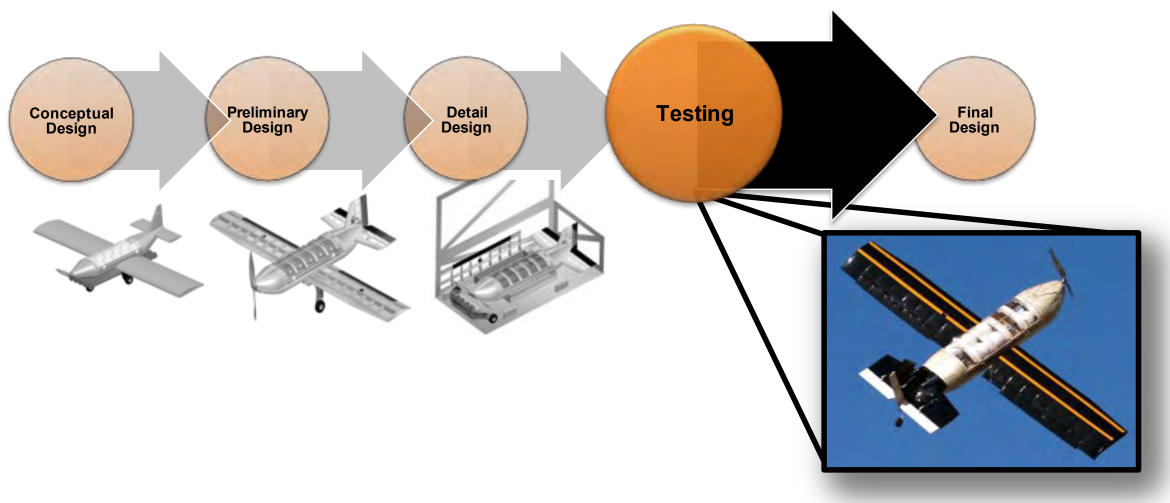

Figure 8.10 — Progress of Overall Design Process

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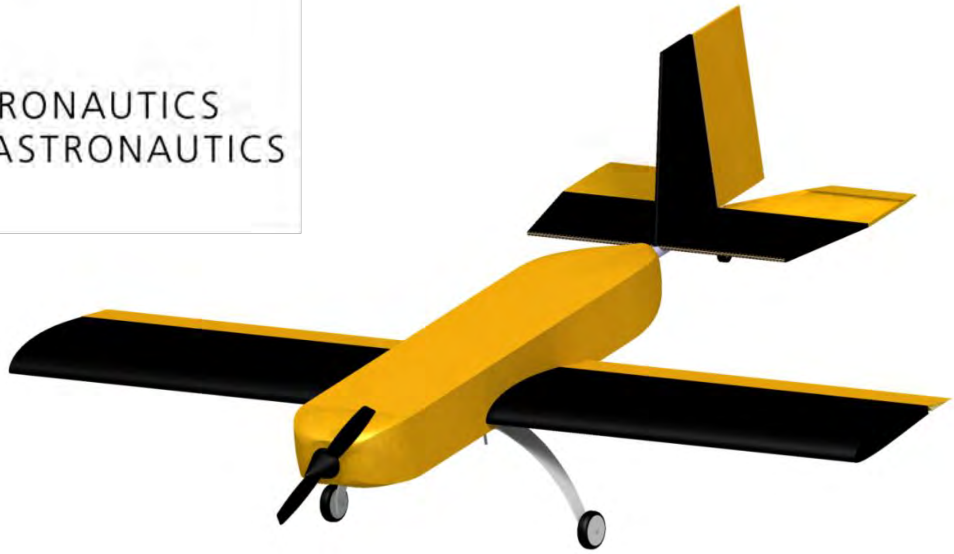

Team B'Euler Up

2010 AIAA Cessna/Raytheon Student Design/Build/Fly Competition

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1.0 EXECUTIVE SUMMARY

This report details the design, testing, and manufacturing processes conducted by the Purdue University B‟Euler Up Team in preparation for the AIAA/Cessna/RMS Student Design/Build/Fly Competition. The primary objective is to maximize the total score, which is comprised of a written report score and a total flight score. The flight score consists of three missions to be completed in succession: a ferry flight and two payload flights. The ferry mission (Mission 1) requires the empty aircraft to fly two laps as quickly as possible, whereas the payload missions involve carrying a maximum of 10 softballs (Mission 2) and up to 5 bats (Mission 3). Each mission rewards a low system weight. Mission 1 and Mission 3 scores also consider flight time, whereas the Mission 2 score includes payload loading time. Therefore, a successful aircraft must be lightweight, fast, and quick to load. The aircraft must also takeoff in less than 100‟, and fit inside a 2‟ x 2‟ x 4‟ dimension enclosure case when unassembled. Score analysis shows that system weight is the most important factor of the flight score, making it the most critical design variable.

A spiral design approach, as seen in Figure 1.1, was used to systematically design and optimize an aircraft to yield the highest overall score. The first design spiral was accomplished in four stages: conceptual design, preliminary design, manufacturing, and testing. During the conceptual design, the competition rules and requirements were analyzed using structured design methods. Concept generation tools were used to identify several possible aircraft configurations. An exercise in trade studies was used to further develop two concepts. Lessons learned from this exercise were used to create the conceptual design for the aircraft that would be sized. Preliminary design focused on refining the aircraft on the basis of simulated mission performance and scores. The aircraft was built and tested, both in flight and in the wind tunnel. This testing revealed that aircraft performance was comparable to initial predictions, however, the prototype was much heavier than expected, and the propulsion system exceeded the 40 Amp current limit imposed by the contest rules. The second design spiral focused on system optimization through extensive wind tunnel, propulsion, and structural testing to reduce weight and modify the propulsion system to keep within the mandated current limits.

The final design is a low wing monoplane with a conventional tail and tractor style propulsion system. The wings were sized to a 5.5‟ wingspan, 11” chord with a Selig Donovan 7062 airfoil to meet takeoff distance requirements. Propulsion analyses lead to the selection of a Neu 1107/2Y brushless motor and 18 Elite 2200 4/5SC battery cells. This design minimizes propulsion weight while providing the needed thrust and endurance to quickly complete each mission. The fuselage integrates a hinged, quick-latching lid to enable a minimal loading time of the payload into the fuselage. The tail is mounted to a boom and was designed to minimize weight. The monoplane and conventional tail design is a well-established, stable aircraft that is relatively easy to manufacture and repair. This design is also a proven platform for the efficiency, maneuverability, speed, and lifting requirements for each of the missions. The combination of these design features produces the best solution to the mission requirements.

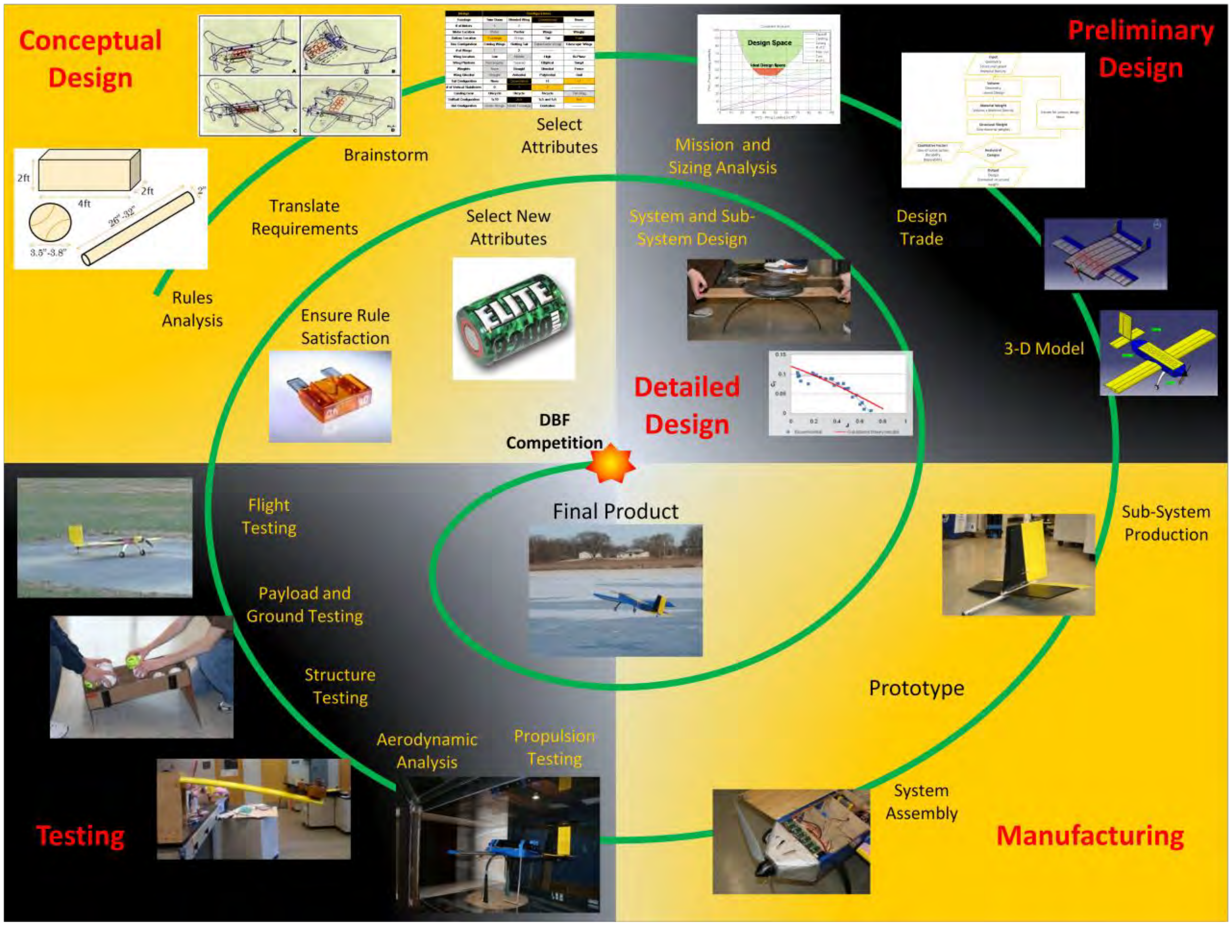


Figure 1.1: Spiral Design

This design is optimized for minimal weight, loading time, and mission flight time. Flight testing has shown that this design will result in an estimated flight score of 208.78 out of a possible 250. The performance capabilities are documented in Table 1.1.

Table 1.1: Performance Capabilities

Performance Capabilities					
Mission 1		Mission 2		Mission 3	
				Straight Velocity (ft/s)	65
Straight Velocity (ft/s)	80			Turn Velocity (ft/s)	55
Turn Velocity (ft/s)	70			Flight Time (sec)	147
Flight Time (sec)	79	Loading Time (sec)	11	Number of Bats	5
Gross Flight Weight (lbs)	4.92	Gross Flight Weight (lbs)	9.30	Gross Flight Weight (lbs)	11.17
System Weight (lbs)	6.55	System Weight (lbs)	6.55	System Weight (lbs)	6.55
Estimated Score	**34.54**	**Estimated Score**	**76.34**	**Estimated Score**	**97.90**
				TOTAL SCORE	**208.78**

2.0 MANAGEMENT SUMMARY

Team B'Euler Up consisted of engineering students with a wide range of backgrounds and a common interest in aeronautics. Members independently analyzed the Design/Build/Fly (DBF) rules and scoring factors in order to brainstorm aircraft configurations. During preliminary aircraft sizing, members were separated into sub-teams to promote a variety of innovative and competitive design solutions.

2.1 Design Team Organization

After the aircraft configuration selection, team members were assigned to sub-teams in the areas of aerodynamics, structures/CAD, propulsion, and report. Each sub-team was assigned a team leader to create a management hierarchy for efficient organization. Figure 2.1 lists the team members and their primary areas of responsibility.

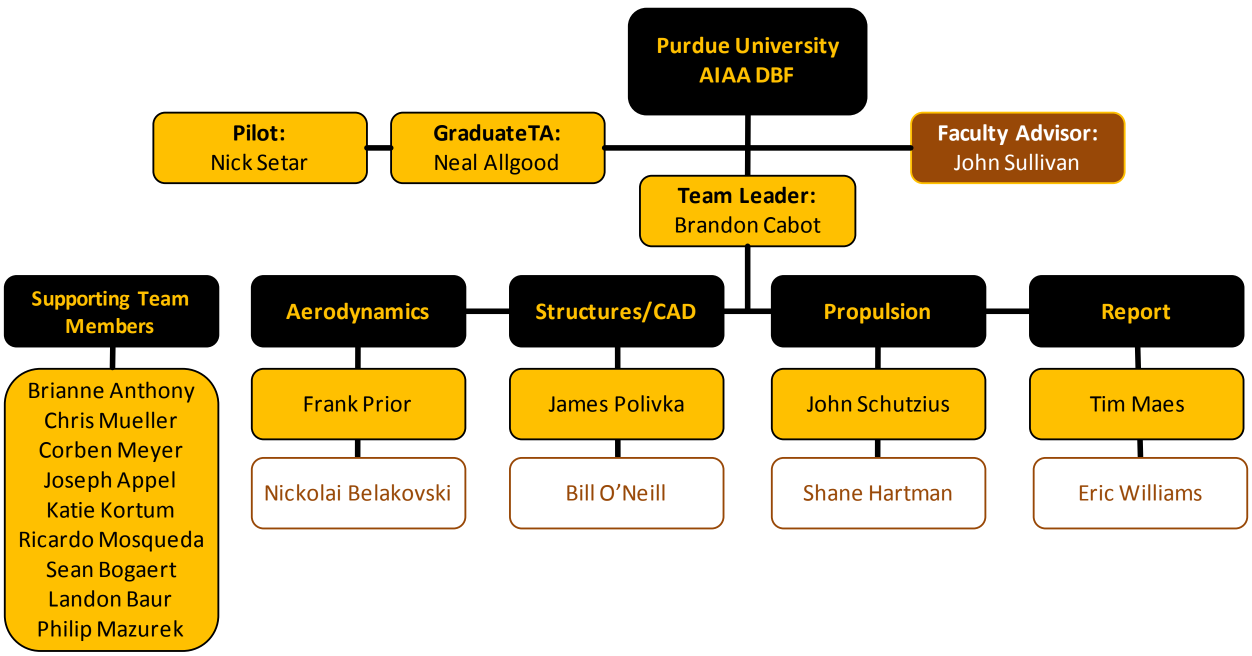


Figure 2.1: Team Organization Chart

Very few members of the team had direct experience with remote controlled aircraft and building techniques so the more experienced members were required to pass on knowledge to other team members. By working between groups and sharing the knowledge and skills available, the team joined together to build a successful aircraft.

2.2 Milestone Chart

In order to keep such a complex and demanding project on schedule, a Gantt chart was used to map out the planned and actual schedules and milestones. The milestone charts can be seen in Figure 2.2 and Figure 2.3 on the next page.

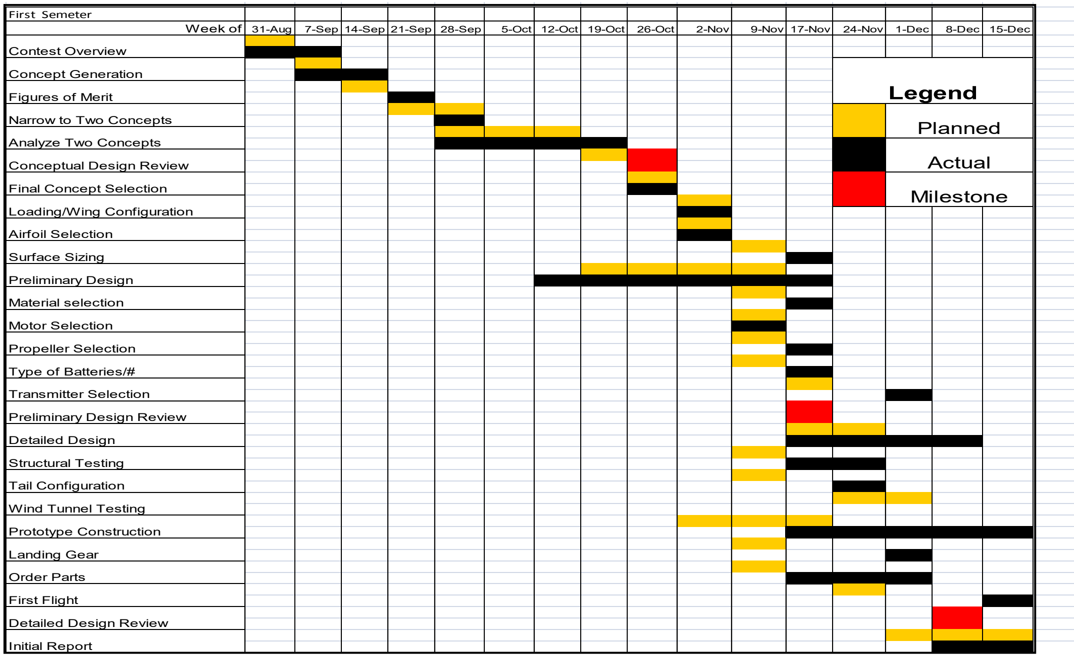


Figure 2.2: Fall Semester Milestone Chart

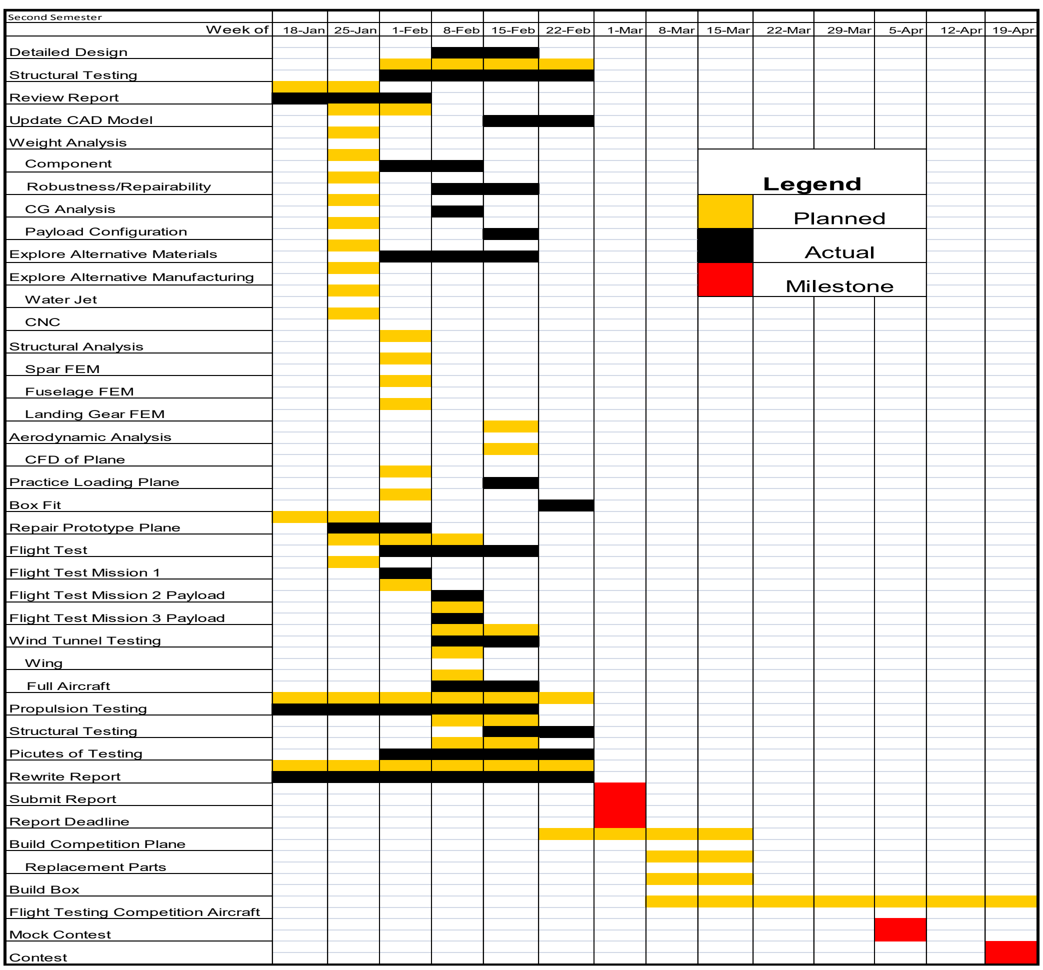


Figure 2.3: Spring Semester Milestone Chart

3.0 CONCEPTUAL DESIGN

Conceptual design began by determining and analyzing mission requirements and design constraints. The design goals were determined to perform a figure of merit analysis and trade study.

3.1 Mission Requirements

The mission requirements and aircraft constraints are described in the contest rules for the 2010 AIAA DBF competition[1]. Adherence to contest rules is necessary for eligibility in the competition. The overall goal of the competition is to successfully complete three missions, each of which rewards quickness and lightness.

3.1.1 Aircraft

The aircraft may be any configuration the team desires, with the exception of rotary wing or lighter-than-air. The propulsion system must consist of commercially available propellers, unmodified over-the-counter brushed or brushless electric-powered motor(s), and a power supply of NiCd or NiMH batteries. The maximum battery weight is 4 lbs and the maximum current draw is 40 Amp. All flight hardware, before assembly, must fit in a case no larger than 2‟ x 2‟ x 4‟. The case must include all equipment the team brings to the staging area, including the aircraft, radio transmitter, safety fuse(s), and tools required for assembly.

3.1.2 Payloads

Two separate payloads must be carried by the competition aircraft; one must be internal, and the other must be external. The internal payload is comprised of 6 to 10 ASA Girls Fast Pitch 11” and 12” softballs. The softballs must be arranged internally in a grid pattern, and cannot be staggered or overlapped. The external payload consists of one to 5 bats; which will be between 26” and 30” long, weigh between 16 oz and 20 oz, and have a diameter of 2”. A hole will be placed at the center of gravity (CG) of each bat to restrain motion in the given direction and the team will have to create locking mechanisms for other directions of motion.

3.1.3 Mission Profiles

There are three missions that each team will attempt to complete with a maximum of five total attempts. The aircraft must be assembled and inspected within a 5 minute period before any mission may be attempted.

Each flight must follow the flight course, as shown in Figure 3.1. Takeoff must occur within the first 100‟ of the runway, the upwind leg is 500‟ long, and the downwind leg is 1000‟ long including a 360° turn, and a final leg of 500‟. Each mission must be successfully completed in order to advance to the next mission.

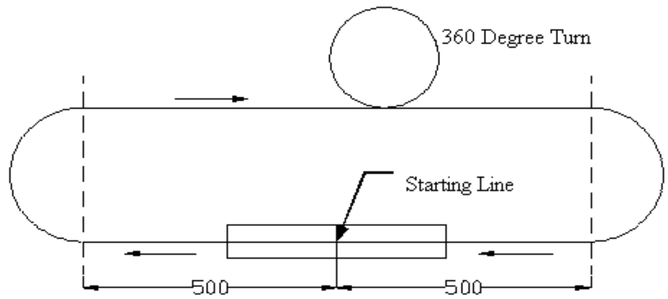


Figure 3.1: Course Layout

Mission 1 Ferry Flight:

Mission 1 is an empty ferry flight consisting of two laps. Teams takeoff and complete the two laps as quickly as possible. Flight time is from throttle up to completion of the second lap. Second lap is complete when the aircraft passes over the start/finish line while still in the air and a successful landing must be completed for the attempt to be counted as successful.

Mission 2 Internal Payload Flight:

Mission 2 is a three lap flight with a random selection of 6 to 10 softballs internally stored in the aircraft. Teams will select the required number of softballs from a bag containing random amounts of 11” and 12” softballs, and place them inside the case. The only timed portion for Mission 2 is loading time; which begins with the aircraft on the flight line and the payload inside the case. Loading time ends once the payload is loaded into the aircraft, the case is closed, and the loading crew returns to a designated area.

Mission 3 External Payload Flight:

Mission 3 is a three lap flight with externally mounted bats. Teams choose how many bats are to be used for the flight. The bats will vary in length and weight and will be selected randomly. The loading of the bats is not timed. Flight time is from throttle up to completion of the third lap.

3.1.4 Scoring

Mission 1 Score (50 points max) = T1*W*50

- T1 = (Reference Time) / (Team Time)
  - Reference Time: lowest flight time for any team that successfully completes Mission 1.
  - Team Time: the flight time for Mission 1 for the team being scored.
- W1 = (Reference Weight) / (Team Weight)
  - Reference Weight: the lightest system weight of the any team that completes Mission 1.
  - Team Weight: the heaviest system weight recorded for any flight attempt throughout the contest.

Mission 2 Score (100 points max) = T2*W2*100

- T2 = (Reference Time) / (Team Time)
  - Reference Time: lowest loading time for any team that completes Mission 2 successfully.
  - Team Time: the loading time for Mission 2 for the team being scored.

- W2 = (Reference Weight) / (Team Weight)
  - Reference Weight: the lightest system weight of any team that successfully completes Mission 2.
  - Team Weight: the heaviest system weight recorded for any flight attempt throughout the contest.

Mission 3 Score (100 points max) = T3*B3*100

- T3 = (Reference Time) / (Team Time)
  - Reference Time: lowest flight time for any team that successfully completes Mission 3.
  - Team Time: the flight time for Mission 3 for the team being scored.
- B3 = (Team Number of Bats) / (Reference Number of Bats)
  - Team Number of Bats: the number of bats carried by the team being scored.
  - Reference Number of Bats: the highest number of bats carried by any team.

Total Flight Score (250 points max) = Mission 1 Score + Mission 2 Score + Mission 3 Score

Total Score (25000 points max) = Written Report Score (100 points max) * Total Flight Score

3.2 Translating Mission Requirements into Design Requirements

Initial scoring analysis was performed through a sensitivity study analyzing mission scoring parameters. In order to complete the sensitivity study, reference estimates of scoring parameters needed to be made. Reference estimates of flight time, loading time, and system weight were made based on historical data from well scoring 2007, 2008, and 2009 DBF teams. These average scores, found in Table 3.1, were assumed to receive the maximum score of 250. From these control values each parameter was analyzed for its sensitivity in relation to the total flight score.

Table 3.1: Sensitivity Analysis

Scoring Parameter	Reference	FT1 + 10%	LT + 10%	FT3 + 10%	W + 10%
Flight Time M1 (s) - FT1	65	71.5	-	-	-
Loading Time M2 (s) - LT	10	-	11	-	-
Flight Time M3 (s) - FT3	150	-	-	165	-
System Weight (lb) - W	5.5	-	-	-	6.05
Total Score	250	245	241	241	236
% Change in Flight Score	0	-1.82	-3.64	-3.64	-5.45

Figure 3.2, on the next page, shows the change in total score plotted against a percent change in each scoring parameter. This shows system weight as the most sensitive scoring factor. Upon examination of the sensitivity analysis, multiple conclusions were drawn. If flight time is viewed as the combination of both Mission 1 and Mission 3, flight time is an equally sensitive scoring factor as system weight. Additionally, loading time and Mission 3 flight time have an equal effect on score.

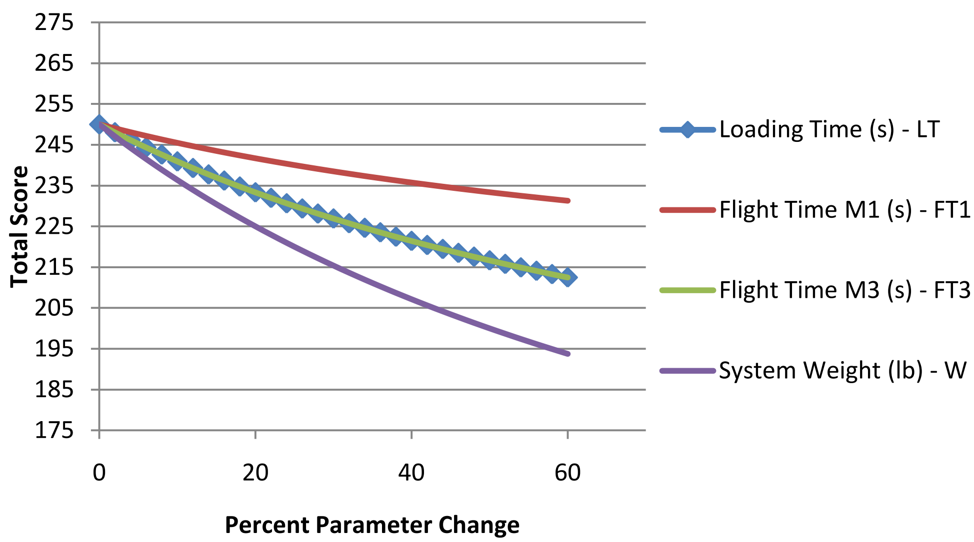


Figure 3.2: Sensitivity Study Plot

After completing the score sensitivity study, the number of bats to be carried still needed to be decided. This decision required a trade study between flight speed and the increased scoring potential with more bats. This scoring analysis required thrust and drag estimations and was therefore completed in the preliminary design section.

3.2.1 Translating Mission Requirement to Design Requirements Summary

Scoring Analysis and historical DBF research led to the conclusion that the design solution should involve the following design objectives and requirements:

- Aircraft Assembly – Create a design solution to fit within the given case dimensions and to be assembled in under 5 minutes.
- Mission 1: Ferry Flight – Create a design solution to successfully and timely complete an unloaded flight.
- Mission 2: Internal Payload Flight – Create a design solution that minimizes the loading time of an internal softball payload and can successfully complete a three lap flight.
- Mission 3: External Payload Flight – Create a design solution that can successfully carry the optimum number of external bats while minimizing an increase in system weight from Mission 2.
- Manufacturing – Produce a lightweight aircraft (<5 lbs based on historical data) and case solution to minimize system weight. Design an easily manufacturable and repairable aircraft.

3.3 Concept and Configuration Generation

After analyzing the mission requirements, the first step towards designing a solution is to consider all of the possible configurations and concepts.

3.3.1 Brainstorming

Success throughout the entire design process relied on careful and thorough conceptual design. The House of Quality (HOQ) and objective tree are similar design tools and by completing them both, new characteristics and ideas can be discovered. The objective tree in Figure 3.3 shows how the mission requirements can be broken down into design requirements and characteristics.

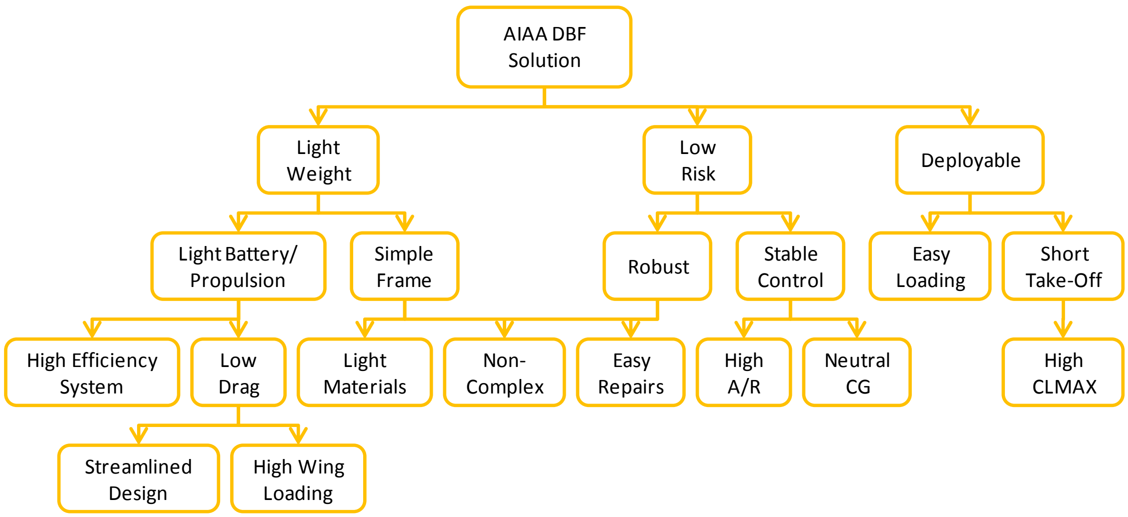


Figure 3.3: Objective Tree

A HOQ was constructed to further translate the mission requirements into design requirements. As seen below in Figure 3.4, the importance and relationship of each characteristic was measured and ranked from least important to most important.

Direction of Improvement			+	+	0	+	-	0	0	+	0
Customer Requirements (What) / Design Requirements (How)		Importance	Aspect Ratio	Lift/Drag	Power/Weight	Propulsion Efficiency	Empty Weight	Wing Loading	Takeoff Distance	Vcruise	Vstall
Mission 1	High Speed	0.18	7	8	9	6	6	2	4	9	4
	Manuverable	0.08	6	2	5	9	7	7	1	3	7
Mission 2	Load Time	0.14	1	0	0	0	1	0	0	0	0
	Endurance	0.12	6	9	8	9	8	4	2	8	2
Mission 3	High Lift	0.08	6	6	8	5	6	8	5	2	3
	System Weight	0.14	3	0	9	1	9	4	7	2	3
General	Stable	0.06	9	6	1	0	4	6	1	1	8
	Robust	0.12	3	1	2	0	5	4	2	5	8
	Risk	0.08	5	1	3	1	3	2	2	1	6
Absolute Importance			10	12	13	12	13	7	5	11	6

Most Important	
Moderatly Important	
Somewhat Important	
Least Important	

Figure 3.4: House of Quality

Using the HOQ analysis, the objective tree and design requirements from the scoring analysis, members of the team independently created conceptual design sketches, as shown on the next page in Figure 3.5. These sketches identified various plausible aircraft configurations.

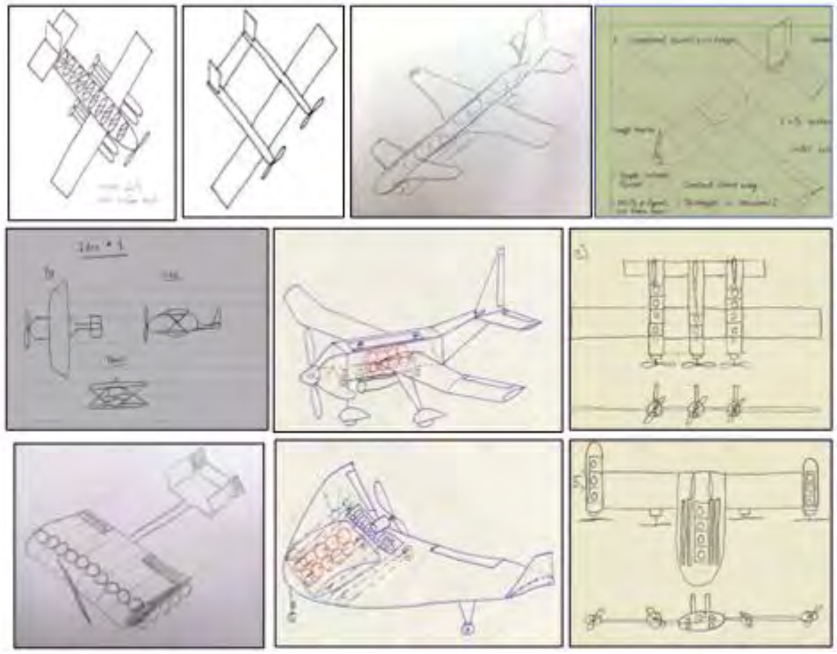


Figure 3.5: Conceptual Design Configurations

To further optimize the potential for conceptual design and to familiarize newer members with the design process, the team was divided into two groups to create unique design solutions. The Black and Gold groups‟ selections can be seen in Figure 3.6 with silver denoting mutual selections and gold and black representing the corresponding group selections.

Design	Configurations			
Fuselage	Twin Boom	Blended Wing	Conventional	Boom
# of Motors	1	2	--------------	--------------
Motor Location	Puller	Pusher	Wings	Wingtip
Battery Location	Fuselage	Wings	Tail	Front
Box Configuration	Folding Wings	Slotting Tail	Detachable Wings	Telescopic Wings
# of Wings	1	2	--------------	--------------
Wing Location	Low	Middle	High	Bi-Plane
Wing Planform	Rectangular	Tapered	Elliptical	Swept
Winglets	None	Straight	Blended	Fence
Wing Dihedral	Straight	Anhedral	Polyhedral	Gull
Tail Configuration	None	Conventional	H	U
# of Vertical Stabiliziers	0	1	2	--------------
Landing Gear	Unicycle	Bicycle	Tricycle	Tail Drag
Softball Configuration	1x10	2x5	1x5 and 1x5	5x2
Bat Configuration	Under Wings	Under Fuselage	Centerline	--------------

Figure 3.6: Team’s Morphological Matrix

Ultimately, the groups developed two separate design solutions as shown, on the next page, in Figure 3.7. The Black group‟s configuration involved a conventional wing design with a conventional tail. The Gold group‟s design was a lifting body with a U-Tail. Both had detachable wings for case fit, tail dragger landing gear, single tractor engines, hinge-based softball loading mechanisms, rectangular wings, and bat mounting systems below the fuselage.

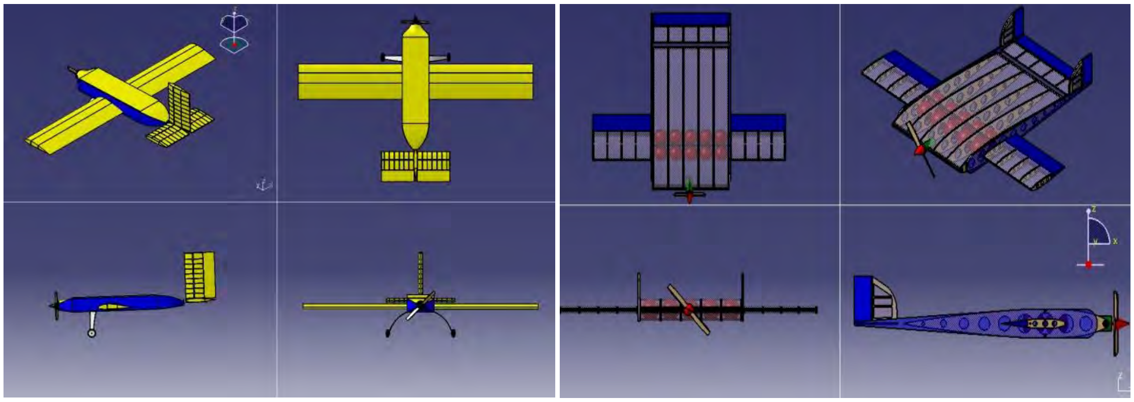


Figure 3.7: Black Group Design and Gold Group Design

Team B‟Euler Up‟s final aircraft design decision would be based on the lessons learned from the Black and Gold group‟s design process. The familiarity of manufacturing of the Black group‟s design was an advantage; however, issues with CG due to payload location could become a factor. The Gold group‟s design of a lifting body would have large amounts of internal payload storage, but historically has severe pitch stability issues. Additionally, ease of manufacturing was a consideration.

Equipped with the lessons from the Black and Gold Group exercise, the whole team worked together to discover the optimum design solution using Figure of Merit (FOM) analysis, numerical selection matrices, and trade studies.

3.3.2 Figures of Merit

Six design figures were chosen based on the results of mission modeling, scoring analysis, and experience. The FOMs and their descriptions are listed below in order of importance:

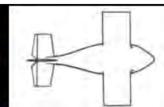
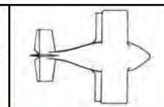
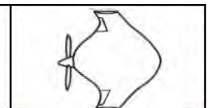
1. **Speed:** The main focus of the weighting scheme was speed (both flight speed and loading speed) because it encompassed three of the four scoring parameters. Speed is a factor in all missions.
2. **System Weight:** System weight was ranked second due to its high correlation to flight score, however, the system weight is only considered in scoring of Mission 1 and Mission 2.
3. **Reliability:** A successful flight must be completed in order to score for any mission. Therefore a reliable and consistent aircraft is essential.
4. **Range:** The design solution must be able to carry the payloads for the duration of the mission; therefore the range of the design is necessary to be considered in concept generation.
5. **Stability:** An unstable aircraft can be difficult or impossible to fly; therefore stability was an important design criterion.
6. **Durability:** Based on the limited manufacturing experience of the team, the aircraft needed to be robust in order to withstand unforeseen mishaps.

3.3.3 Concept Weighting Process

Based on the selected FOMs, decision matrices were used to numerically evaluate possible design concepts. Each FOM was given a numerical value for weight and each configuration was rated on a scale from one to four with higher numbers better meeting the FOMs. Only feasible configuration options were considered. In addition to the FOMs, a trade study was performed to expand upon the weighting within the numerical decision matrices.

3.3.3.1 Wing Configuration

Table 3.2: Wing Configuration FOM

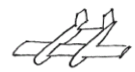
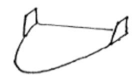
Team B'Euler Up				
Merit	**Weight**	**Mono-Plane**	**Bi-Plane**	**Blended Wing**
Speed	23	3	2	3
System Weight	17	3	2	2
Reliability	17	3	3	2
Range	16	3	3	3
Stability	14	3	4	2
Durability	13	3	3	2
Total	**100**	**300**	**274**	**239**

1. Mono-plane: A simple proven design with minimal complexity but might not be able to produce the amount of lift necessary while remaining within a reasonable size.
2. Bi-plane: Reduces wing loading; however it increases complexity, leads to challenges with case fit, but is historically very stable.
3. Blended Wing: Increases difficulties with manufacturing and field repair in the event of a crash during the competition. Can provide a large amount of lift without a large wingspan, but can have potential pitch stability issues.

A mono-plane was selected to eliminate complexity and issues with case fit. Asymmetric loads are not a factor since the number of bats to be flown is chosen by the team and can be designed around the optimal number given in the scoring analysis. A detachable wing design was chosen because of the simplification of construction and repairability.

3.3.3.2 Fuselage Configuration

Table 3.3: Fuselage Configuration FOM

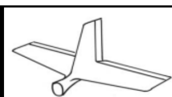
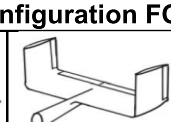
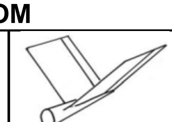
Team B'Euler Up				
Merit	**Weight**	**Boom**	**Twin Boom**	**Wing**
Speed	23	3	3	3
System Weight	17	3	2	2
Reliability	17	3	3	2
Range	16	3	3	3
Stability	14	3	4	3
Durability	13	3	3	2
Total	**100**	**300**	**297**	**253**

1. Single Boom: Reduces risk and complexity, however, issues with CG are a factor. More slender than other concepts, the single boom minimizes drag and frontal area.
2. Twin Boom: Distributes the internal load of softballs, and reduce the clutter of the bats as external payloads. However, it adds weight and complexity. Case fit could be an issue depending on how the aircraft brakes down.
3. Blended Wing Body: Requires thick inner camber to house internal payload. Increases complexity in manufacturing, as well as difficulties with on-site repair. Stability could potentially be an issue.

The single boom was chosen to reduce risk and complexity. It is a stable design and proven from historical conventional configurations. A single motor was selected to minimize complexity and risk along with a tractor configuration to fit with the fuselage configuration.

3.3.3.3 Tail Configuration

Table 3.4: Tail Configuration FOM

Team B'Euler Up					
Merit	**Weight**	**Convention**	**U-Tail**	**V-Tail**	**T-Tail**
Speed	23	3	3	3	3
System Weight	17	3	2	4	2
Reliability	17	3	3	2	3
Range	16	3	3	3	3
Stability	14	3	4	2	3
Durability	13	3	3	3	2
Total	**100**	**300**	**297**	**284**	**267**

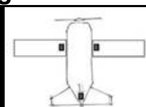
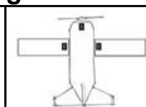
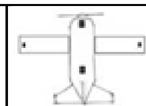
1. Conventional Tail: Structurally sound due to direct mounting to the boom of the horizontal and vertical tail. The effectiveness of the vertical tail is large because interference with the horizontal tail effectively increase its aspect ratio. Ideal for mounting to boom/fuselage.

2. U – Tail: Increases effectiveness of horizontal tail due to placement of vertical tail. Shorter vertical surfaces help with case-fit issues. Often used with propeller aircraft to reduce the yawing moment from propeller slipstream on the vertical tail. Requires more complex control linkages.
3. V – Tail: Combines the horizontal and vertical tail functions. Reduces control authority in combined yaw and pitch maneuvers.
4. T – Tail: Increases the effectiveness of the vertical tail and removes horizontal tail from downwash during climb. Poses structural issues.

In order to determine the best options for controlling the aircraft, the aerodynamics team looked at various tail configurations. No configurations had significant benefits over the conventional tail; therefore a conventional tail was selected for the design solution. A detachable tail configuration was chosen to better meet the case fit requirements.

3.3.3.4 Landing Gear Configuration

Table 3.5: Landing Gear Configuration FOM

Team B'Euler Up				
Figure of	**Weight**	**Tail**	**Tricycle**	**Bicycle**
Speed	23	3	3	3
System Weight	17	3	2	3
Reliability	17	3	3	2
Range	16	3	2	3
Stability	14	3	3	2
Durability	13	3	3	2
Total	**100**	**300**	**264**	**254**

1. Tail Dragger: Reduces taxi stability and handling characteristics, but results in a light weight low drag gear configuration. Ground clearance with bats may pose an issue.
2. Tricycle: Improves taxi handling characteristics and stability, but the nose gear increases drag. Provides bat storage beneath aircraft.
3. Bicycle: Poses difficult ground handling characteristics.

A tail dragger design was selected due to ease of manufacturing, low weight, and low drag. The ground clearance issue was resolved by increasing the length of the tail wheel.

3.4 Conceptual Aircraft Summary

The final design solution included a conventional aircraft configuration with a low mounted detachable mono-plane, detachable tail boom, conventional tail, tail dragger landing gear configuration, and a hinged lid fuselage for softball loading. This design solution is proven with predictable handling qualities. The final design configuration can be seen, in Figure 3.8, on the next page.

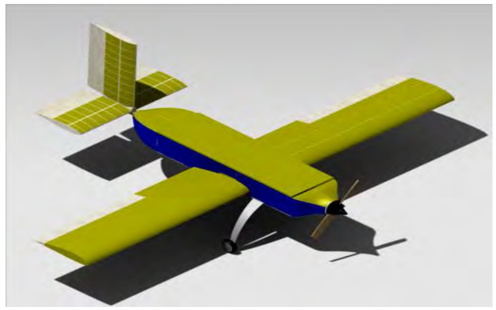


Figure 3.8: Final Design Solution

4.0 PRELIMINARY DESIGN

This section of the design process uses analytical aircraft weight and mission simulation models in order to size the aircraft for maximum score. Team B'Euler Up was able to estimate flight performance and design an optimized aircraft based on mission requirements. Preliminary design was accomplished in several phases, as depicted in Figure 4.1.

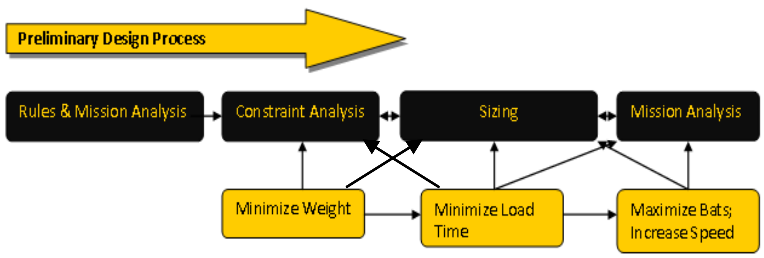


Figure 4.1: Preliminary Design Methodology

4.1 Modeling Methods

Models were created for analyzing the design constraints and mission requirements. The results of these models were used as initial values for modeling the aerodynamics, stability and control, structures, and propulsion aspects of the aircraft.

4.1.1 Constraint Model

Design requirements were translated to a bounded design space through the creation of a constraint analysis model, shown in Figure 4.2. This model used aircraft and mission specific parameters, such as aspect ratio and payload weight, as inputs to equations for speed and power requirements.

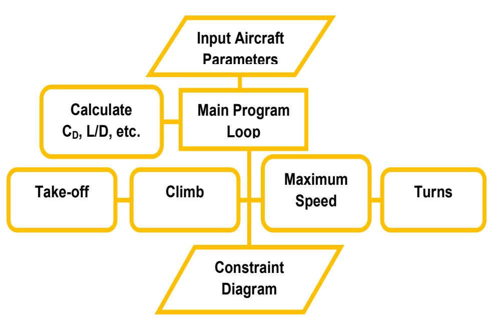
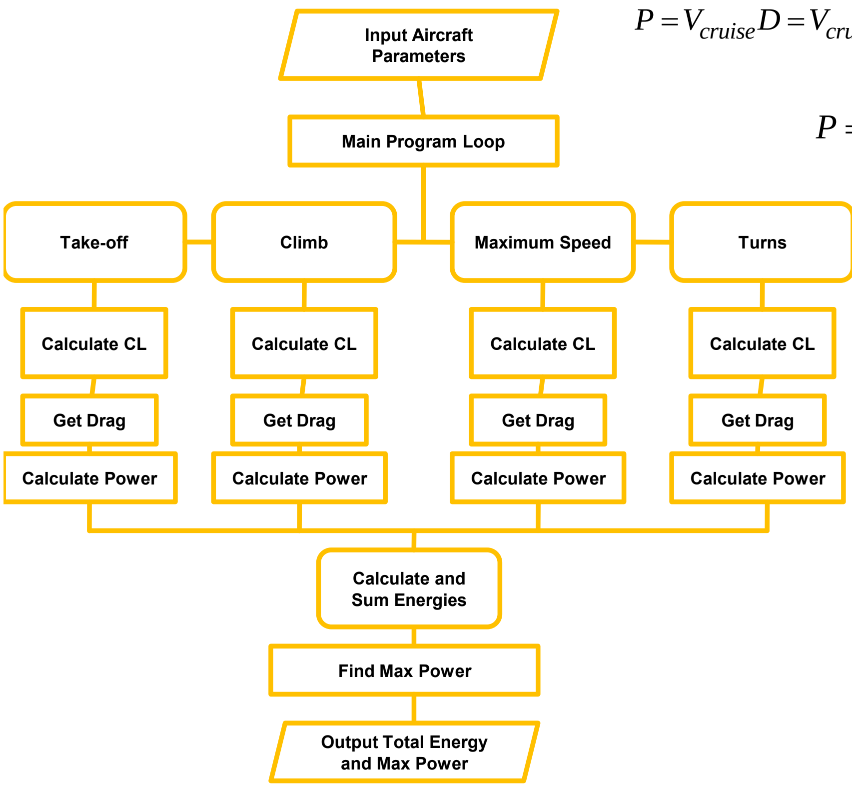


Figure 4.2: Mission Constraint Analysis

4.1.2 Mission Model

The mission model was used to estimate power and energy requirements for the aircraft which could then be used in turn to estimate battery and motor weights. In order to calculate power and energy requirements, each mission was broken into four legs: takeoff, climb, turn, and cruise. The minimum power required was calculated by simulating the mission for each point in the design space. Subsequently, instantaneous power requirement was integrated over time to obtain the optimal energy requirement.

Figure 4.3, illustrates the process[2] used to produce the energy and power requirements.



$$P = V_{cruise} D = V_{cruise}\left(C_{D_e} qS + \frac{k(nW)^2}{qS}\right)$$

$$P = V\left(\frac{1.44W^2}{\rho S C_{L_{\max}} g} - D\right)$$

$$P = V\left(\frac{ROC * W}{V} + D\right)$$

Figure 4.3: Flowchart of Mission Modeling Process

4.1.3 Aerodynamic Model

For preliminary sizing of the aircraft, an aerodynamic analysis was conducted on the conceptual design aircraft configuration. The aerodynamics model[3], as shown in

Figure 4.4, shows how the aerodynamic force coefficients were calculated. Initially, the input parameters to the calculations were based on steady level flight at an estimated competitive cruise velocity. Using these values, the coefficient of lift that was required could be calculated. From the lift values, decisions were made on the airfoil choice and wing size needed. For the sizing of the wing, induced drag effects had to be considered, which resulted in increasing the aspect ratio of the wing. However, a trade-off exists between aspect ratio and the strength of the wing structure. A higher aspect ratio increases wing efficiency yet decreases the chord. This decreases the wing thickness which reduces the available area for the structural spar. To circumvent this issue, the chord was kept constant while the wingspan was increased. To complete the calculation of aircraft drag, the parasite drag was calculated for each component of the aircraft separately and summed for accuracy.

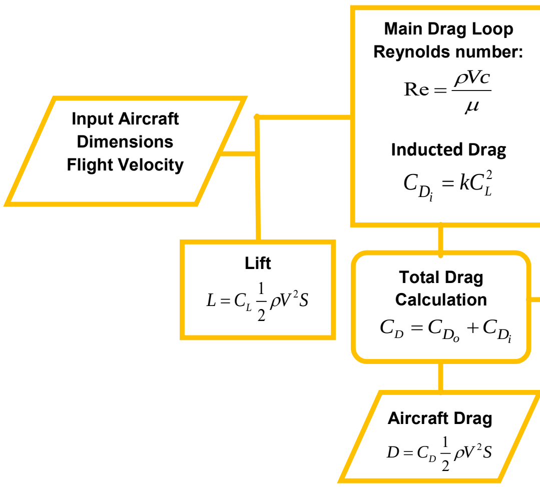


Figure 4.4: Aerodynamics Sizing Model

Inputs to the model, such as C_{D_0} estimates, were based on a combination of historical data and airfoil analysis. These values were needed to thoroughly assess the sizing trade-offs inherent in wing sizing and fuselage shape with respect to aerodynamics.

4.1.4 Stability and Control Model

The stability and control model focused on tail sizing and optimizing the weight of the control surfaces and tail boom system. The tail volume coefficients method was employed to calculate the necessary size of the control surfaces, and then material properties were used to determine the weight of the tail boom system. Figure 4.5 illustrates this sizing model. Through this iterative process, the tail system designed would be one of the lowest possible weight and size while still giving adequate control of the aircraft.

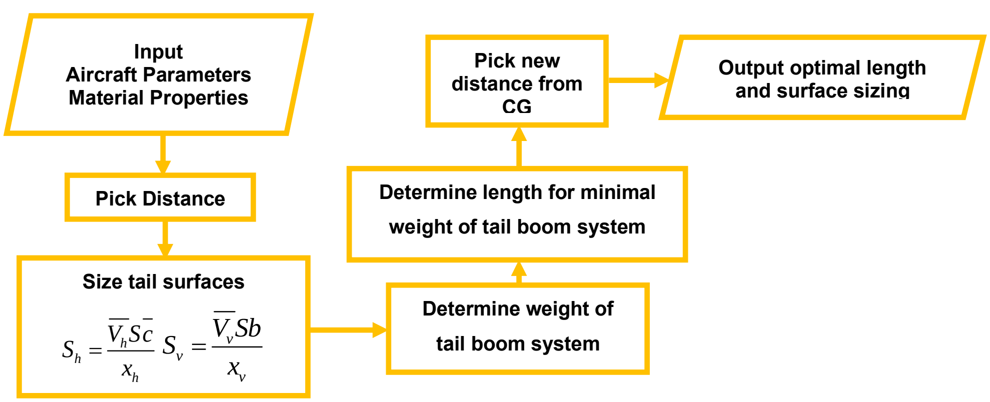


Figure 4.5: Stability and Control Sizing Model

4.1.5 Structural Model

Preliminary structural modeling minimized aircraft weight by analyzing different materials and structural layouts to find the lowest possible weight. An algorithm was created to approximate the weight of various aircraft components. Preliminary sizing, such as: aspect ratio, span, and fuselage dimensions were used to find the amount of material required. Material properties, such as density and relative strength, are used for weight estimates. The structural model, shown in Figure 4.6, was used to estimate weight and select a design for various components, such as the wing, fuselage structure and spar.

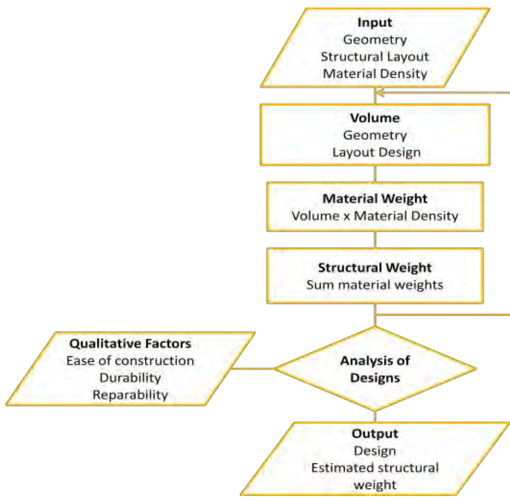


Figure 4.6: Structures Sizing Model

4.1.6 Propulsion Model

The power and energy requirements from the mission model were used to determine motor size. Empirical data was used to estimate the power to weight ratio for the preliminary propulsion configuration. Weights and powers from available motors as well as data for propellers, gearboxes, and batteries were used to determine the best propulsion system configuration. The flowchart, Figure 4.7, demonstrates the logic process.

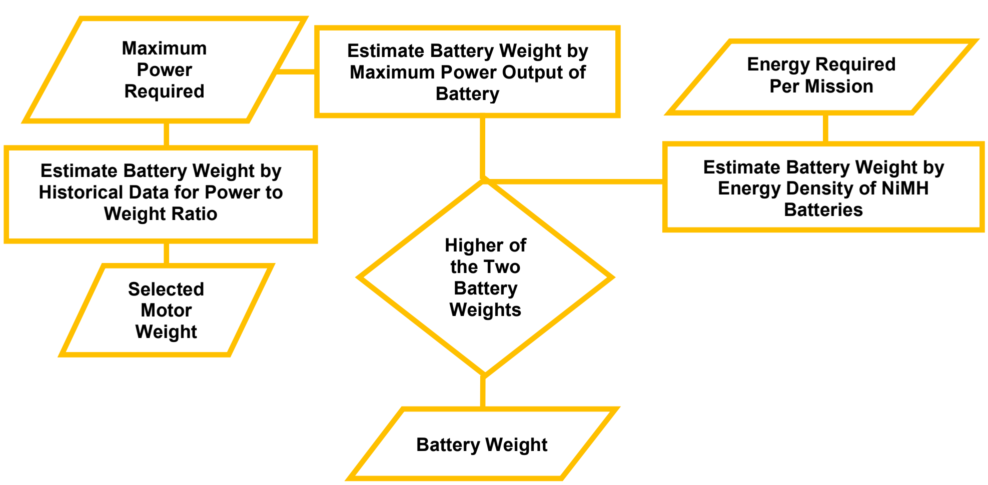


Figure 4.7: Propulsion Sizing Model

Propulsion model results consisted of system weight, motor type, gearbox, propeller, and the battery configuration that would result in the highest mission scores.

4.2 Sizing Trades

In this section the results from the modeling processes are described, and specific details on how the models were used are provided. Through the processes described herein, preliminary sizing for the aircraft was completed. Results are tabulated at the end of the preliminary design section.

4.2.1 Constraint Model Results

The constraint model, depicted earlier, took into account the mission requirements in order to produce a design space based on power loading and wing loading. The resulting ideal design space is displayed in Figure 4.8.

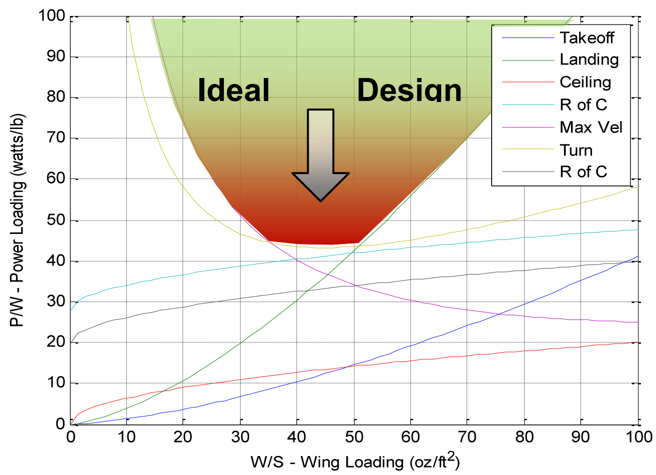


Figure 4.8: Constraint Analysis Graph with Design Space

Based on the shape of the constraint curves, the ideal design space occurs near minimal power loading. Thus, a specific power and wing loading coupling of 45 watt/lbf and 53 oz/ft^2 were used in the weight and scoring calculation program. These programs determine the aircraft weight and effectively fly each mission to compute a mission score. These tools were used to obtain initial sizing based on Mission 3, since an aircraft optimized for Mission 3 would be more than capable of accomplishing Mission 1 and Mission 2. The calculated values for initial sizing are displayed in Table 4.1.

Table 4.1: Sizing Results Summary

Summary	
Aspect Ratio	6.55
Wing Surface Area (ft^2)	5.50
Wingspan (ft)	6.00
Gross Take Off Weight (lbf)	13.96
Payload Weight (lbf)	6.25
Empty Weight (lbf)	5.58
Total Battery Weight (lbf)	2.13
Total Energy Available (J)	115020

4.2.2 Mission Model Results

Using the outputs from constraint analysis in mission modeling, total power requirements were calculated for each leg of the mission. The results for Mission 3 were employed for initial sizing since it had the most demanding flight requirements. Summing the energy requirements gave the team starting values for propulsion sizing, which will be covered in the propulsion results section. The output of the analysis is shown in Table 4.2.

Table 4.2: Energy Usage Breakdown by Flight Leg

Energy Use Breakdown by Flight Leg						
Leg	Power Required (Watt)	Velocity (ft/s)	Time (s)	Distance (ft)	Energy Used (J)	Energy Excess (J)
Take-off	577	51	2	80	1281	113739
Climb	598	34	9	301	6569	108451
Cruise Type #1	275	45	66	3000	24771	90249
Cruise Type #2	832	80	38	3000	55985	59035
Loiter	253	35	0	0	55985	59035
Turn Type #1	510	59	35	2081	73974	41046
Turn Type #2	583	65	41	2679	97986	17034

The optimal number of bats for Mission 3 was determined by using the constraint model to provide an estimated system weight for each scenario from one to 5 bats. With this system weight, the mission model determined a flight time, and therefore a score from the conceptual design scoring analysis. Figure 4.9 shows the maximum score for Mission 3 was obtained by carrying 5 bats, even though the flight time was slower due to the increased payload.

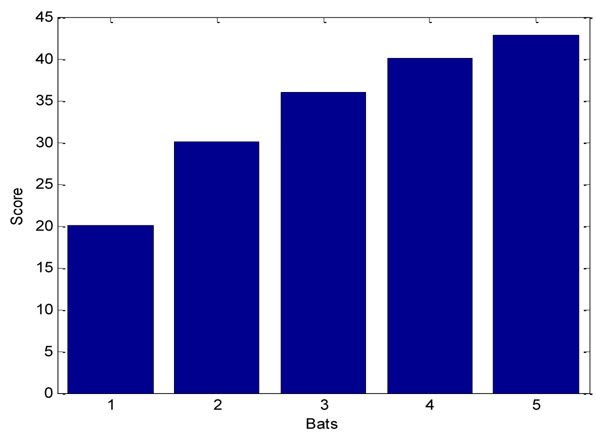


Figure 4.9: Mission 3 Score vs. Number of Bats Carried

4.2.2 Aerodynamics Results - Lift & Drag

Following the aerodynamics model, an analysis was performed on over 100 airfoils to determine their lift and drag characteristics at competitive flight velocities.

Seven airfoil types were selected for further comparison, including NACA, Gottingen, Eppler, Clark, Drela, Selig, and Selig-Donovan designs. To gather theoretical values for each design, a subsonic viscous analysis was completed in XFOIL (a computational package for two dimensional viscous analyses) over a range of angles of attack. This provided lift and drag coefficients as well as drag polars and lift coefficient curves for each airfoil. In order to determine the Reynolds numbers at which XFOIL would run, accurate flying conditions at the competition location are needed. These values constitute the first assumptions necessary for aerodynamics analysis. Table 4.3 displays all the assumed flight conditions:

Table 4.3: Assumptions for XFOIL Analysis

Assumed Flight Conditions	
Elevation of Wichita, KS (ft)	1300
Flight Altitude (ft)	1500
Cruise Flight Velocity (ft/s)	80
Reynolds Number	541900
Air Density (slug/ ft^2)	0.0023
Viscosity (lb-s/ft^2)	3.75E-07

With the Reynolds number based on these flight conditions, each airfoil was evaluated independently. Figure 4.10, shown on the next page, shows drag polars for several NACA-type designs in the desired lift coefficient range of 0.75 to 1.3.

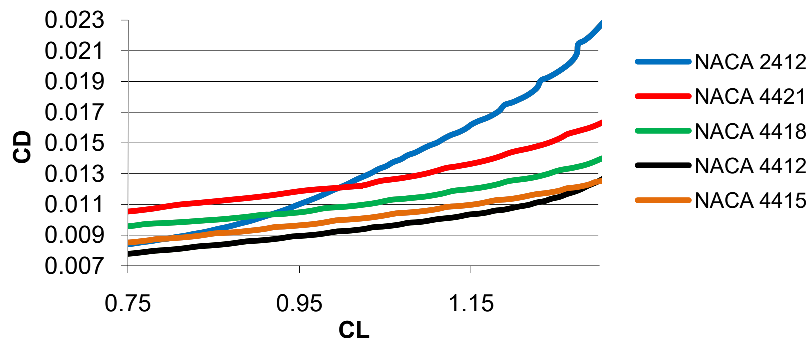


Figure 4.10: Best NACA Drag Polars

Many of the NACA airfoils have either relatively high drag values or low stall angles throughout the desired C_l range. However, the NACA 4412 has reasonable drag coefficient values and was decided to be the most favorable of the NACA series. The analysis also tested various Selig-Donovan type designs, the results for which are displayed in Figure 4.11.

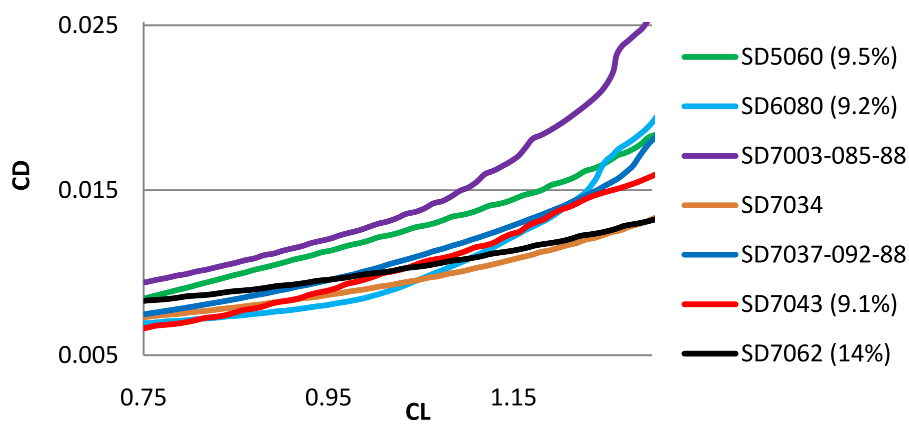


Figure 4.11: Selig-Donovan Drag Polar

These airfoils were very impressive in their range of lift coefficient values and low drag coefficient values over the entire domain. For a thorough comparison, all of the airfoils with the most favorable characteristics, across the various airfoil types, were plotted together to determine the best fit for the aircraft. The main consideration of the airfoil choice was a low drag coefficient. Figure 4.12 shows the drag coefficients for the best airfoils selected from the previous analysis plotted against a wide range of lift coefficient values.

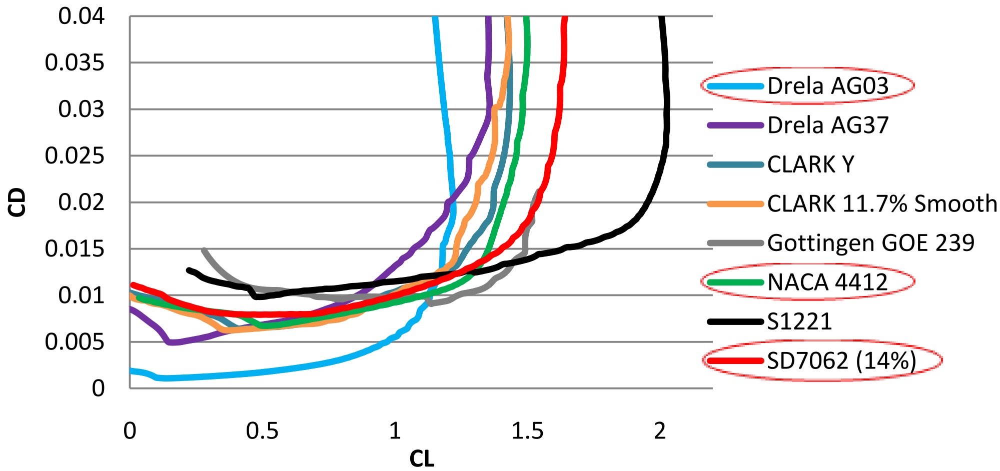


Figure 4.12: Airfoil Drag Polar Comparison for Lowest C_d

From this comparison, the three airfoils with the lowest C_d over a range of Cl from 0.75 to 1.3 were then selected and analyzed in detail. The Drela AG03 consistently produces the least drag for a range of C_l values from 0 to 1.1. The stall angle is at 10.8º, which is fairly high, but the airfoil does not create large enough C_l values without an incidence angle. In addition, a maximum thickness of 5% of the chord means the wing would need a large chord length to be structurally feasible. A one foot chord length implies a maximum thickness of 0.6", which would be difficult to strengthen sufficiently.

The NACA 4412 consistently produced low drag for a range of C_l values from 0 to 1.3. Its stall angle is 14.3º and the airfoil‟s C_l range is higher than the Drela AG03‟s. Ultimately the NACA 4412 is a better selection than the Drela AG03 despite having larger drag values.

The Selig-Donovan 7062 has consistently low C_d values through the range of C_l values. The stall angle is at 16.2º, and it has a higher C_l with a negligible increase in C_d as compared to the other airfoils. It also has low drag at high C_l values allowing for high speeds and bank maneuvers. This airfoil is also a proven success with past DBF teams. Because of the low drag, high stall angle, and proven performance, the Selig-Donovan 7062 became the choice for the first competition prototype. Figure 4.13, on the next page, displays the airfoil geometry.

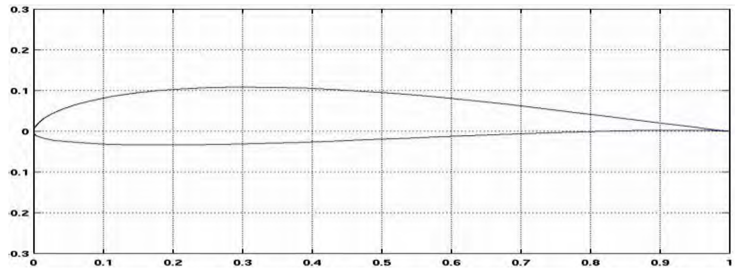


Figure 4.13: Selig-Donovan 7062 Profile

The Selig-Donovan 7062 requires precise manufacturing, especially in the sharp trailing edge, in order for the airfoil to be efficient. This slows down production and repair time, but it was deemed that the desired airfoil characteristics were worth the additional effort required.

4.2.3 Stability and Control Results

After performing a historical analysis of past DBF aircrafts, the sizing model was used to determine an optimal control surface size and tail boom length based on weight. Figure 4.14 shows the results of this process.

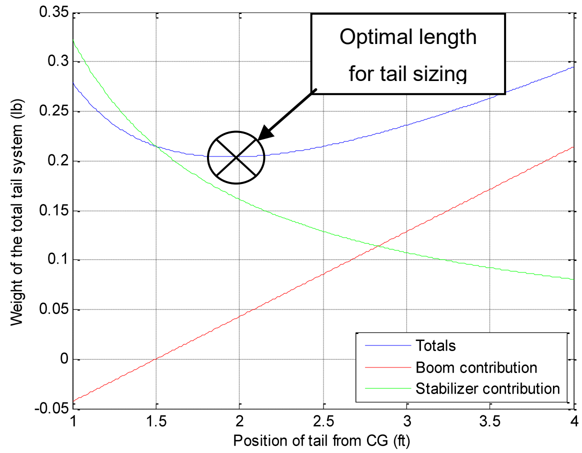


Figure 4.14: Weight of the Tail Components vs. Distance of Tail from CG

Given the materials used in tail construction, the optimal length for the aerodynamic center of the tail from the CG of the aircraft was found to be 2‟ while the total weight was estimated at roughly 0.2 lbs. Since the tail boom length estimate is given as a range of distances from the CG, the weight estimate provided by the analysis is an average and not exact. However, because the contribution of the tail boom to the weight of the system is linear, it will not affect the optimal length for minimization.

For sizing the percent chord for elevator, rudder, and ailerons, suggestions from Roskam[4] were used along with a historical analysis. Table 4.4 summarizes the results of tail sizing and control surface sizing.

Table 4.4: Preliminary Tail Sizing

Parameter	Value
Optimal Length from CG (ft)	1.94
Horizontal Tail Surface Area (ft^2)	2.04
Vertical Tail Surface Area (ft^2)	0.95
Chord for HT (ft)	1.04
Chord for VT (ft)	0.95
% Chord for Elevator	45
% Chord for Rudder	40
% Chord for Aileron	20
% Span for Aileron	40

Once the tail and boom length were sized, the static margin was calculated to determine stability characteristics. Using the equation listed below, the aircraft"s aerodynamic center was calculated as a percent of the chord. The static margin for the aircraft was calculated as 32% of the chord, or a distance of 3.55 inches from the center of gravity back to the aerodynamic center. This positive static margin means that the aircraft exhibits static longitudinal stability. The equation below was used to calculate the position of the aerodynamic center.

$$\bar{x}_{ac_A} = \frac{\bar{x}_{ac_{wf}} + \frac{C_{L_{\alpha_h}}}{C_{L_{\alpha_{wf}}}} \eta_h \frac{S_h}{S} \bar{x}_{ac_h} (1 - \frac{\delta\varepsilon}{\delta\alpha})}{1 + \frac{C_{L_{\alpha_h}}}{C_{L_{\alpha_{wf}}}} \eta_h \frac{S_h}{S} (1 - \frac{\delta\varepsilon}{\delta\alpha})}$$

S_h, S: Horizontal Tail Area and Wing area, respectively

$\bar{x}_{ac_{wf}}$: Non-dimensional wing and fuselage aerodynamic center position

$\frac{C_{L_{\alpha_h}}}{C_{L_{\alpha_{wf}}}}$: Ratio of C_{L_α} for the horizontal tail over C_{L_α} for the vertical tail

η_h: Dynamic Pressure Ratio

$\bar{x}_{ac_{wf}}$: Non-dimensional horizontal tail aerodynamic center position

$\frac{\delta\varepsilon}{\delta\alpha}$: Change in downwash angle per angle of attack

To ensure the center of gravity stays within CG limits of the aircraft with different payload configurations, an analysis of softball placement was completed.

The team considered the optimal placement of softballs for differing payload assignments with respect to CG movement. In order to determine the best softball locations, an analysis was done using Microsoft Excel and geometric information about the design. The distance between the estimated CG of the empty aircraft and the CG of the different payload configurations should be minimized to lessen the impact on static margin. The softballs are positioned so close to the center of gravity along the pitch axis, a moment created from asymmetric loading about the roll axis would be negligible. This allows the control characteristics of the loaded aircraft to remain similar to those of the unloaded aircraft. Table 4.5 shows the position of the center of gravity for the softballs with each possible configuration. Figure 4.15 shows the loaded position of each softball configuration.

Table 4.5: CG of Softballs with Each Configuration

Position of Center of Gravity			
Number of Balls	Aft of Spar (in)	Left to Right (in)	Mass at CG (oz)
10	2	0	70
9	2	0.22	63
8	2	0	56
7	2	0.29	49
6	2	0	42

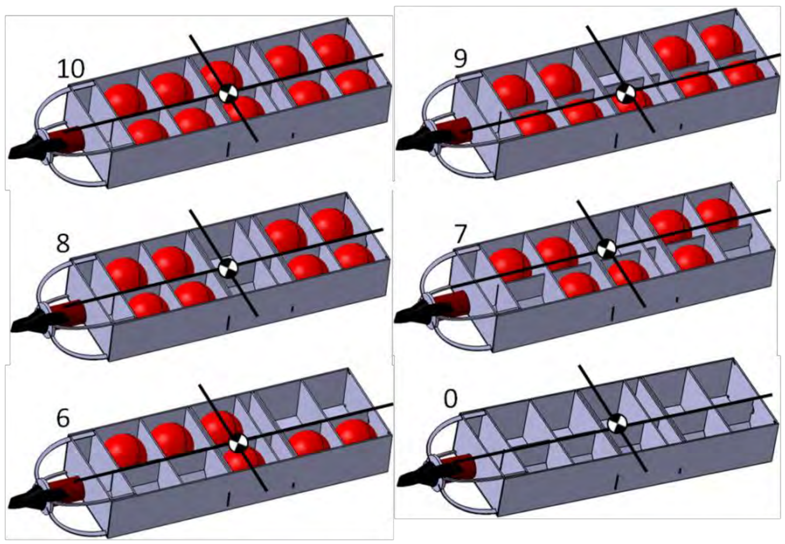


Figure 4.15: Softball Placement and CG Location

4.2.4 Structural Results

Several designs were considered for the wing structure. Analysis was done on eight concepts, and a design was selected based on the output from the model. Weight was calculated by taking the expected dimensions of the wing (chord, span and thickness) and estimating the amount of material by volume needed for construction. The volume was multiplied by density to give the weight estimate. A built-up balsa wing proved to be the best solution. Table 4.6 summarizes the weight of each design.

Table 4.6: Wing Structure Analysis Results

Design Idea	Weight (oz)
Foam and Fiberglass	11.2704
Foam and Balsa	16.76
Balsa with Foam Leading Edge	7.8896
Balsa with Balsa Sheeted Leading Edge	9.2944
Strictly Built-up Balsa	6.7984
Solid Foam	27.3648
Cored out Foam	1.49824
All composite	10.3856

Similar analysis was done for the spar design and the main fuselage structure. The size of the fuselage was based on the payload size and weight, and the size of the spar was based on the expected wing loading and simple beam theory. A T-frame primary fuselage structure built out of fiberglass reinforced balsa was found to be most desirable. Table 4.7 summarizes the results for the fuselage design, respectively.

Table 4.7: Primary Fuselage Structure Analysis Results

Design Idea	Weight (oz)
Plywood Box	33
Fiberglass Reinforcred Balsa T-frame Box	14
Plywood and Fabric Box	18
Composite	24

The spar design used a variation of the design model used for the wing and fuselage. The design focused on the stresses anticipated in the spar. The sizes used to determine the weight for each design were calculated from the maximum stresses using the Euler-Bernoulli Beam Theory[5] and the properties of the materials available. The lightest materials with the highest ultimate stresses were chosen for each element of the structure. The ultimate stress of each material was required to be greater than the maximum predicted stress experienced by that element. Table 4.8 summarizes the results of the spar analysis with the weights and highest percent maximum stress in the design.

Table 4.8: Spar Design Analysis Results

Design Idea	Weight (oz)	% Ultimate Stress
Foam with 2 Plywood Shear Webs	18	64
Fiberglass Covered Balsa Shear web	3.5	80
Square Carbon Tube	3	46
Round Carbon Tube	5	75

The preliminary spar design using the Euler-Bernoulli Beam Theory showed that the carbon fiber rod provided the highest ultimate stress and the lowest weight for any spar design, with a maximum stress of 126 ksi, or just under 50% the manufacturer‟s published ultimate stress for the material. During beam testing, delamination of the carbon rod occurred at 25% of the expected load. This was due to the shear stress and the strength of the epoxy not being accounted for in the simple analysis. After the discovery of the issues with the carbon fiber rod, it was determined a revised study of the wing structure was necessary. To gain flight test data, a hollowed out foam wing with a foam spar and plywood spar caps was constructed for the prototype. This was the heaviest of the studied spar designs, but it was the easiest to construct. The limited time to flight test and man hours required to redesign the balsa ribs to accommodate a new spar design facilitated the need for this quick wing design. Further analysis of the spar designs showed that the fiberglass covered balsa shear web was the best choice.

4.2.5 Propulsion Results

Designing the propulsion system was carried out by selecting a combination of motor, battery, and gearbox such that the system provides the power and endurance necessary for flight while minimizing weight. Starting with the motor selection, power loading from constraint analysis and the energy calculations from mission modeling showed that 600 Watts of continuous power were required to fly the most propulsion critical mission, Mission 3.

Using this power requirement, four Neu motors analyzed as candidates for the propulsion system. These four motors were chosen do to their availability to the team and their successful use by past winning teams. The manufacturer specifications[6] were used in conjunction with basic energy calculations to determine the relevant quantities, which are summarized in Table 4.9.

Table 4.9: Motor Selection Analysis

Motors					
Motor	K_V (RPM/V)	Weight (oz)	I_O (Amps)	R_m (Ω)	Cont. Watts
Neu 1107/2Y	3100	3.7	0.9	0.019	300
Neu 1110/2Y	2250	4	1.3	0.024	500
Neu 1112/2Y	1750	4.7	1.5	0.027	600
Neu 1509/1.5Y	2400	7.5	2	0.007	750

When taking into consideration the 600 Watt power requirement, only the Neu 1509 and Neu 1112 were realistic candidates based on the motor ratings. Among those two, the 1509 had a significantly lower resistance (implying higher efficiency) which led to its selection as the motor for the propulsion system.

In order to determine the best battery pack, three Elite series packs were compared on power to weight ratio and energy density. Only Elites NiMH batteries were compared because historical data suggested they had the highest power to weight ratio among manufacturers. Table 4.10 shows the Elite 1500A with the highest power to weight ratio and a relatively high energy density.

Table 4.10: Battery Cell Power to Weight Ratios and Energy Densities

Battery	Power to Weight (Watts/lb)	Weight (oz)
Elite 1500A	625	117
Elite 2200	380	91
Elite 5000SC	290	134

As a result, the 1500A series was selected as the battery source for the propulsion system. Using the energy requirements from the required endurance and range in mission modeling, it was determined that 18 cells would be required to have enough power to successfully fly Mission 3. Actual measurements on an assembled 18 cell Elite 1500 battery pack yielded a Mission 3 battery weight of 1.3 lbs.

With these estimates, a propeller could be selected. Utilizing features built into the commercially available software, MotoCalc, the analysis in Table 4.11 was conducted on the complete propulsion system under flight conditions.

Table 4.11 Propulsion System Comparison

Motor	Battery	Gear Box	Propeller Size	Rating
Neu 1509/1.5y-2400	Panasonic 1500	6.7:1	20 x 12	1.000
Neu 1509/1.5y-2400	GP3700	5.3:1	19 x 14	0.999
Neu 1509/1.5y-2400	GP3700	5.3:1	18 x 14	0.994
Neu 1509/1.5y-2400	Gold Peak 1500	6.7:1	20 x 10	0.991

The final propulsion system selected consisted of the Neu 1509/1.5Y-2400 motor with a 6.7:1 gearbox and a 20” x 12” propeller. Figure 4.16 shows that this combination provided the highest motor efficiency and overall efficiency.

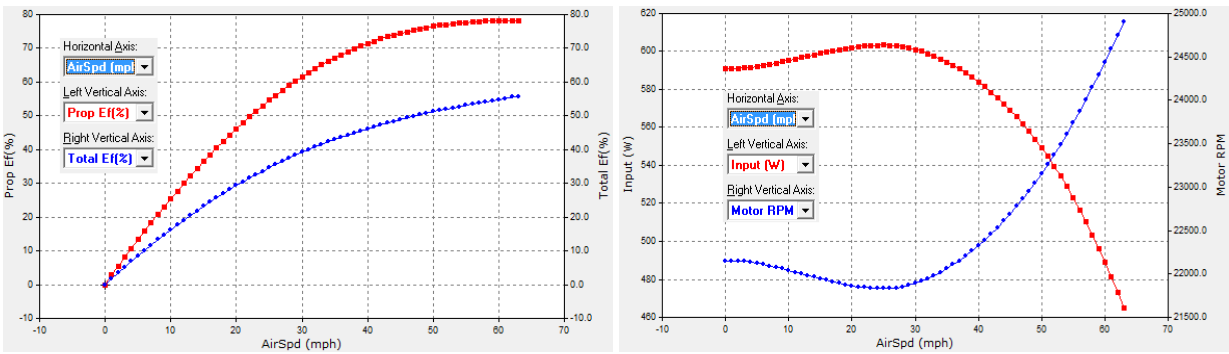


Figure 4.16: (a) Airspeed vs. Propeller Efficiencies (b) Airspeed vs. Total Input Wattage and RPM

From Figure 4.16, it is shown that the max power consumed by the propulsion system occurs at 24,600 RPM and 604 watts. At this power level, MotoCalc analysis showed that the voltage was 20.8. From simple calculations, it was found that 28.8 Amps of current are drawn at this power setting, well below the 40 amp limit of the speed controller.

The MotoCalc analysis is not expected to be extremely accurate to actual propulsion traits, but it provides a good starting point for further analysis to be carried out in the detailed design.

4.3 Aircraft Parameters and Estimated Mission Performance

The preliminary design analysis resulted in the first round sizing of the aircraft. Some sizes would still change in detail design, but not significantly. Table 4.12, shown on the next page, lists the parameters of the aircraft that were determined in preliminary design, as well as the updated estimates for scoring.

Table 4.12: Estimated Aircraft Parameters and Scoring

Aircraft Specs	
Wing Span (ft)	6.00
Wing Chord (ft)	0.92
Span Horizontal Tail (ft)	1.96
Chord Horizontal Tail (ft)	1.04
Span Vertical Tail (ft)	1.00
Chord Vertical Tail (ft)	0.95
Empty Weight (lbs)	5.58
Battery Weight (lbs)	2.13
Estimated Box Weight (lbs)	1.13
Transmitter & Tools (lbs)	0.50
System Weight (lbs)	9.34
10 Softballs (lbs)	4.38
5 Bats (lbs)	6.25

Reference Values		
Mission 1	Flight Time (sec)	65
Mission 2	Loading Time (sec)	10
Mission 3	Flight Time (sec)	150
	Number of Bats	5
System Weight (lbs)		5.5
Mission Specs		
Mission 1	Straight Velocity (ft/s)	80
	Turn Velocity (ft/s)	70
	Flight Time (sec)	79
	Gross Flight Weight (lbs)	7.71
	System Weight (lbs)	9.34
	Score	24.24
Mission 2	Loading Time (sec)	11
	Gross Flight Weight (lbs)	12.09
	System Weight (lbs)	9.34
	Score	53.56
Mission 3	Straight Velocity (ft/s)	65
	Turn Velocity (ft/s)	55
	Flight Time (sec)	147
	Number of Bats	5
	Gross Flight Weight (lbs)	13.96
	System Weight (lbs)	9.34
	Score	102.04
TOTAL SCORE	**179.84**	

4.4 First Spiral Conclusion

During the preliminary design process the aircraft was sized and a prototype was built and tested. Results of prototype testing are summarized in the performance results section. Key findings include an oversized propulsion system that drew more than the regulated 40 Amps, and an overweight aircraft. The excessive current draw can clearly be seen in the performance results section. In the second spiral, efforts were focused on further optimization of the propulsion system while weight was reduced by redesigning the fuselage and reconstructing the wings with more advanced construction techniques.

5.0 DETAIL DESIGN

The prototype functioned as a proof of concept and showed that the proposed design was stable and flew well. The goals of the detail design were to further refine the aircraft by reducing weight and correcting the propulsion issues discovered during flight testing of the prototype. Detail design was accomplished using the methodology as depicted in Figure 5.1.

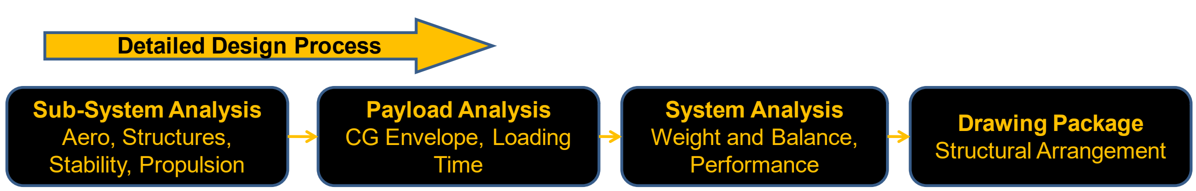


Figure 5.1: Detailed Design Methodology

5.1 Dimensional Parameters

The final dimensional parameters of the aircraft design are listed in Table 5.1. This table is a compilation of the dimensions of the aircraft wings, fuselage, aerodynamic control surfaces, and other vital aircraft elements.

Table 5.1: Dimensional Values

Wing		Horizontal Stabilizer	
Airfoil	SD 7062	Airfoil	Flat Plate
C_{root} (ft)	0.92	C_{root} (ft)	1.05
C_{tip} (ft)	0.92	C_{tip} (ft)	1.05
B (ft)	5.5	B_H (ft)	1.96
S (ft^2)	5.5	S_H (ft^2)	2.06
AR	6.55	AR_H	1.86
Fuselage		**Vertical Stabilizer**	
Length (ft)	3.25	Airfoil	Flat Plate
Width (ft)	0.69	C_{root} (ft)	1.07
Tail Boom Length (ft)	2.42	C_{tip} (ft)	0.82
Propeller Size	20" x 12"	B_V (ft)	1
		S_V (ft^2)	0.95
		AR_V	1.06

5.2 Structural Characteristics and Capabilities

While the structural characteristics of the prototype proved to be sufficient for all missions, the design was further analyzed and improved to lower the system weight. The structures team identified four key areas that could be improved to reduce structural weight: wing, landing gear, tail, and fuselage. This was accomplished through more in depth computational analysis and experimental testing. The structural testing details are given later in the testing section.

5.2.1 Wing

The spar was designed to take a simulated 2.5 G tip test and flight load of 40 lbs lift. This gives a maximum bending moment at the root of 30 ft-lbs. The fiberglass balsa sheer web design was analyzed and tested with different fiberglass thicknesses to find the optimal combination. Table 5.2 shows designs considered and tested. The load to weight ratio was used to determine the best design.

Table 5.2: Spar Design Considered

Spar Design	Ultimate Load (lbs)	Weight (lbs)	Load to Weight
Fiberglass Covered Balsa (Single Layer)	27.6	0.126	6.81
Fiberglass Covered Balsa (Double Layer)	46.6	0.220	6.61
Fiberglass Covered Balsa (Hybrid)	46.6	0.173	8.40

A hybrid balsa fiberglass design with half double/half single layer fiberglass was able to meet the desired load requirement, while keeping a small weight. The spar will be constructed of 1/16" thick balsa sheet that is 1.5 in. tall coated in lightweight fiberglass. This spar design is capable of holding a 46.6 lbs ultimate load. The failure mode of this type of spar is buckling. To prevent buckling, small balsa stringers would run along the top and bottom of the spar to constrain the edges of the sheet. This creates a clamped edge constraint and increases the critical buckling load. The ribs also help to prevent buckling.

The structure of the wing itself is a simple balsa rib and stringer design with MonoKote covering, as per the analysis in section 4.3.3. The spar runs the length of the wing and connects the ribs. Balsa stringers also run the length of the wing to help maintain shape between the ribs and add strength to the wing. A larger longeron is added along the trailing edge in front of the aileron to provide a hinge mount and to provide an extra attachment point to the fuselage to prevent twisting at the attachment point. The wing is mounted to the fuselage by inserting the spar and rear longeron into custom fit slots. The spar is pinned to the fuselage via a 1/16" PTO pin.

5.2.2 Landing Gear

The landing gear chosen is a commercially available carbon fiber main landing gear designed for aircraft of similar weight. Both aluminum and composite gear were considered and tested. Both were able to withstand a 35 lbs load simulating a 3 G landing force without failing, but the composite gear was chosen because it was 5.5 oz lighter than the 10.4 oz aluminum gear. The landing gear was attached to a solid hardwood plate integrated into the floor of the fuselage to distribute the landing load on the fuselage. The tail wheel was attached to a custom rod that ran through the tail boom and actuated by the rudder. This proved to be lightweight and effective during taxi testing.

5.2.3 Tail

The tail was designed to be as lightweight as possible while providing an adequate amount of control surface area to keep the aircraft dynamically stable. The horizontal and vertical tails were preliminarily designed to be NACA 0012 airfoils, built up with the same rib and stringer construction used in the wings. However, further aerodynamic analysis of the aircraft showed that the aerodynamic difference between the airfoil design and a flat plate design was negligible at the operating range of Reynolds numbers. Therefore, it was decided to use a flat plate airfoil for the horizontal and vertical tails. This decision was based on the ease of construction in manufacturing a flat plate rather than the rib and stringer construction. The diminished thickness of the vertical tail raised concerns about the structural mounting of the vertical tail to the boom. This was resolved by carving out the leading edge of the vertical tail and attaching a small carbon fiber rod for strength and rigidity. The structural integrity of the tail was validated by wind tunnel tests at maximum flying speed and control surface deflection.

5.2.4 Fuselage

The fuselage was designed to support the payload and the forces from the propulsion system and the tail control surfaces. The preliminary design of the fuselage consisted of a central spine, floor, outer walls, and bulkheads to compartmentalize the softballs. This was constructed from 1/16 „" thick birch plywood. Plywood was initially chosen due to its rigidity and durability. For the detailed design, different materials were considered to lighten the structure. Thinner plywood, balsa, and fiberglass covered balsa were considered, but based on the trends found in Table 5.2; 1/16"" balsa sandwiched with a single layer of fiberglass was chosen to create the central spine and floor while the walls and dividers were made from regular balsa. The single piece spine allowed for transfer of all the forces along the fuselage without weaknesses associated with joints. To limit the weakness of joints, all connections in the fuselage were made using tabbed finger joint construction.

5.3 System Design, Component Selection, Integration, and Architecture

The designed aircraft contains many parts that must be integrated in a secure and simple way. Due to case fit concerns, the tail and wings were designed to be removable from the fuselage. Because assembly time is not a scoring factor but must be held to under 5 minutes, the goal of the wing and tail attachments was to be lightweight and simple. Spar boxes were designed and implemented into the fuselage to allow for easy plug-in wings and pins were chosen to keep the spars of the wings locked into the spar box. The tail boom attachment used a single nylon bolt and thumb screw which could easily be accomplished in under 5 minutes. This solution is a solid locking mechanism that holds the sections together without excessive assembly or construction time.

To document the aircraft architecture, Figure 5.3 was created to show the system design and integration. This figure is color coded to match components with a corresponding subgroup which aided in team organization and was used to assign component responsibilities.

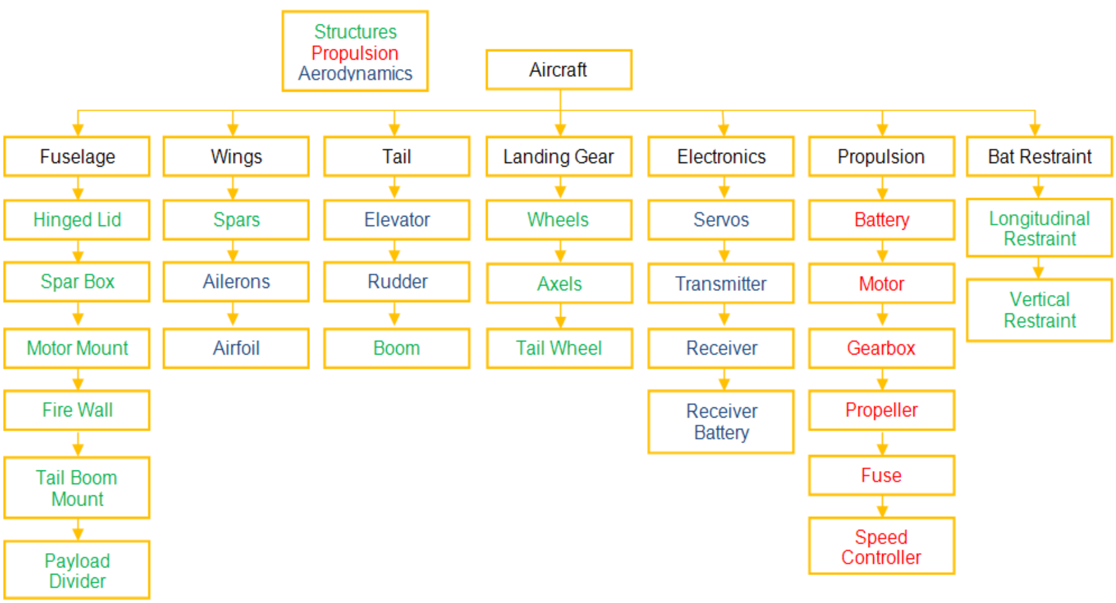


Figure 5.2: Aircraft Architecture Tree

5.4 Propulsion System Analysis

One of the primary goals of the detailed design was propulsion optimization to keep the current draw under the regulated 40 Amps. Optimization of the aircraft propulsion system, shown on the next page in Figure 5.3, required consideration of the combination of motor, battery, gearbox, and propeller.

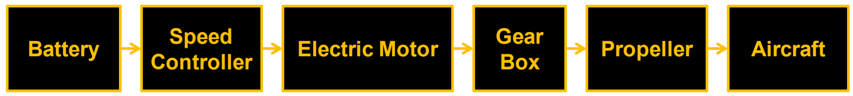


Figure 5.3: Propulsion System Diagram

5.4.1 Propulsion Sizing

Battery voltage and current draw dictate the RPM of the motor, the RPM of the motor/gearbox dictates the current drawn by the propeller, and the current drawn by the propeller alters the voltage of the battery. In order to determine the best propulsion system configuration, a computer program was used to iterate through several combinations of motors, gearboxes, batteries, and propellers. The first section of the computer program determined the power required to fly the aircraft over a range of cruise speeds from 0 to 100 ft/s. The second program used the motor manufacturer‟s specifications for internal motor resistance, no load current draw, and RPM per volt (K_v) rating to determine the current draw and motor RPM at the power required for the desired cruise speed. The third program used the motor current draw and RPM to calculate an advance ratio for different propeller diameters. This advance ratio was used to find a matched coefficient of thrust and power for a range of propeller diameters and pitches. The matched coefficient of thrust was then used to find thrust produced. The propeller thrust and power coefficient curves were obtained through wind tunnel testing and calculations obtained from Goldstein‟s propeller theory, as summarized in the performance section. The iteration process is outlined in Figure 5.4 below.

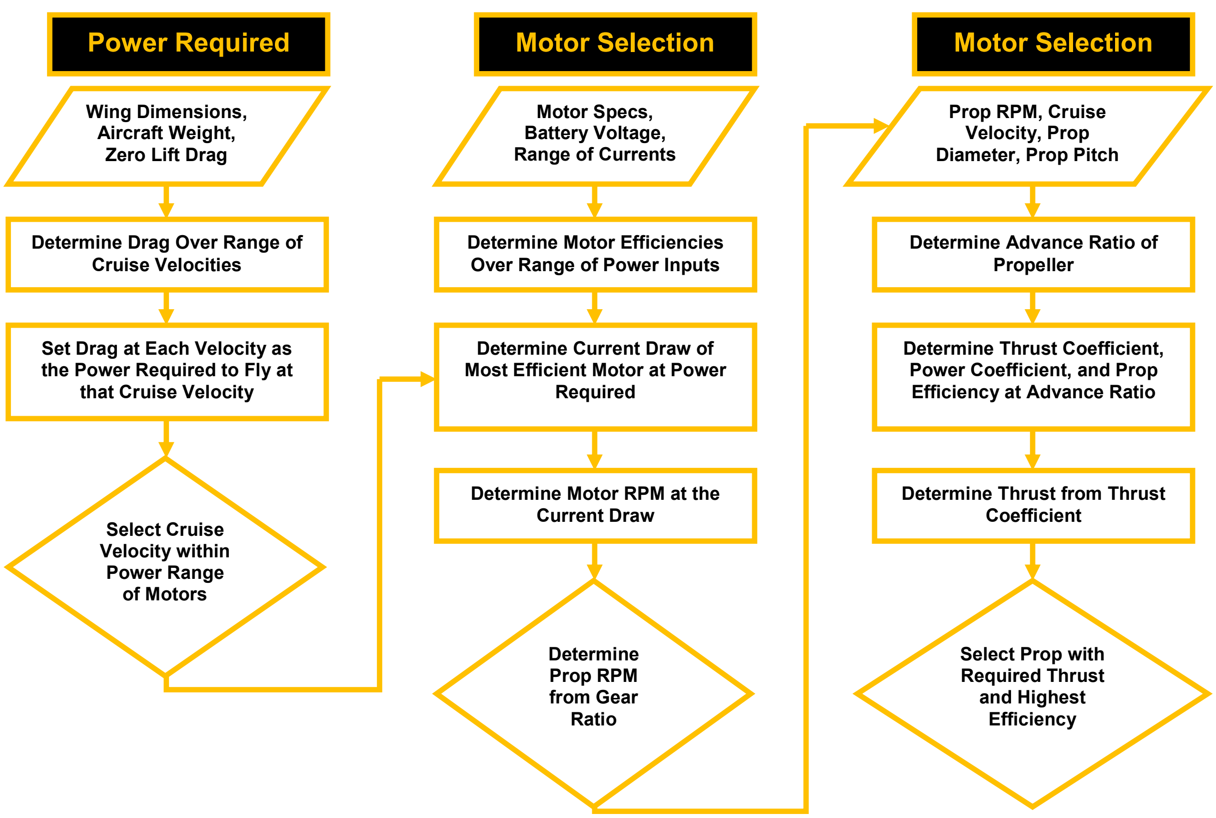


Figure 5.4: Motor Selection Iteration Flowchart

Upon completion of the iteration process, each combination of motor and propeller that met the design criterion was analyzed for final efficiency and system weight. These potential combinations are shown in Table 5.3. The motor selected for all missions was the Neu 1107/2Y with the Neu Box Planetary 6.7:1 gearbox. This motor was not only the lightest motor tested, it was also the most efficient motor over the power range required to fly each mission. Preliminary analysis had shown that this motor was not rated for the high level of power expected by the propulsion system, yet motor temperature analysis from wind tunnel testing showed that this motor would be sufficient for the relatively low flight times of this competition.

It was determined that a different propeller was optimal for each mission. Mission 1 and Mission 2 are best suited by an APC 16” x 12” propeller. This combination produces 4.17 lbs of dynamic thrust at 80 ft/s and requires 381.8 Watts of power. Calculations showed that Mission 3 would require an APC 18” x 10” propeller. This combination produces 7.08 lbs of dynamic thrust at 65 ft/s and requires 371.1 Watts of power. The power required for Mission 3 is less than that of Mission 1 and Mission 2 because of the decreased flight speed and propeller pitch.

The battery selected for use was an 18 cell pack of Elite 2200 cells. The pack was selected after experimentally testing different packs and finding that the listed milliamp hour capacity for the batteries was not representative of the milliamp hours that could be drawn from each cell. From experimentation, it was found that on average, only 2/3 of the rated milliamp hour capacity could be drawn from each cell. From this finding, it was determined that a pack of 21.6 volts made of cells with a 2200 mAh capacity was the optimal pack to fulfill the power requirements of all three missions. These performance calculations were validated through extensive wind tunnel testing. Table 5.4 shows the propulsion system selections for all missions.

Table 5.3: Available Configurations

Motors				
Motor	K_V (RPM/V)	Weight (oz)	I_O (Amps)	R_m (Ω)
Neu 1107/2Y	3100	3.7	0.9	0.019
Neu 1110/2Y	2250	4	1.3	0.024
Neu 1112/2Y	1750	4.7	1.5	0.027
Neu 1509/1.5Y	2400	7.5	2	0.007

Propellers	
Diameter (in)	Pitch (in)
14	10, 12
16	8, 10, 12
18	8, 10, 12
20	8, 10, 12

Gear Boxes
Neu Planetary 4.2:1
Neu Planetary 6.7:1

Table 5.4: Selected Configurations

Team B'Euler Up	Selected System Configurations			
	Motor	Propeller	Gear Box	Battery
Mission 1 & 2	Neu 1107/2Y	16X12	Neu Planetary 6.7:1	18 Elite 2200
Mission 3	Neu 1107/2Y	18X10	Neu Planetary 6.7:1	18 Elite 2200

5.5 Weight and Balance

Weight and balance of the prototype and competition aircraft were computed for several loading conditions. These conditions are based on the final measurements before the first flight. The loading conditions are representative of the mission profiles. The weights of each aircraft and their components, as well as the center of gravity for each payload configuration are shown in Table 5.5. The CG and moment arms are measured from forward most point on the aircraft.

Table 5.5: Aircraft Weight and Balance

Weight Build-Up (Prototype)		
Components	Weight (lb)	Arm (in)
Fuselage	2.31	31.00
Batteries	1.89	7.30
Landing Gear	0.48	11.80
Wings	1.47	19.60
Tail	0.58	46.00
Tail Boom	0.06	42.50
Receiver	0.06	19.60
Servos	0.22	36.00
Wires, Bolts	0.20	19.00
Speed Controller	0.10	7.50
Propeller and Adapter	0.30	1.00
Motor and Gearbox	0.66	3.70
Flight Weight	8.30	
CG (in)	19.94	

Weight Build-Up (Competition)		
Components	Weight (lb)	Arm (in)
Fuselage	0.75	18.50
Batteries	1.14	6.88
Landing Gear	0.48	14.68
Wings	0.78	16.00
Tail	0.47	46.60
Tail Boom	0.03	37.96
Receiver	0.02	20.92
Servos	0.22	32.92
Wires, Bolts	0.27	18
Speed Controller	0.08	6.40
Propeller and Adapter	0.30	1.00
Motor and Gearbox	0.38	1.60
Flight Weight	4.92	
CG (in)	15.90	

Pay Configurations (Competition)		
Payload Option	Weight (lb)	CG (in)
0 Softballs	0.0	15.9
6 Softballs	2.6	16.6
7 Softballs	3.1	16.7
8 Softballs	3.5	16.8
9 Softballs	3.9	16.8
10 Softballs	4.4	16.9
5 Bats	6.3	15.2

In addition to the weight analysis, a trim diagram was created to supplement the stability calculations based in the preliminary design. Using the aircraft geometry, the moment about the quarter chord, Cm_{cg}, was calculated as a function of the coefficient of lift C_L and the elevator deflection δ_e. Figure 5.5 illustrates the longitudinal stability characteristics of the aircraft when the center of gravity is shifted from its current position stated in the static margin in preliminary design.

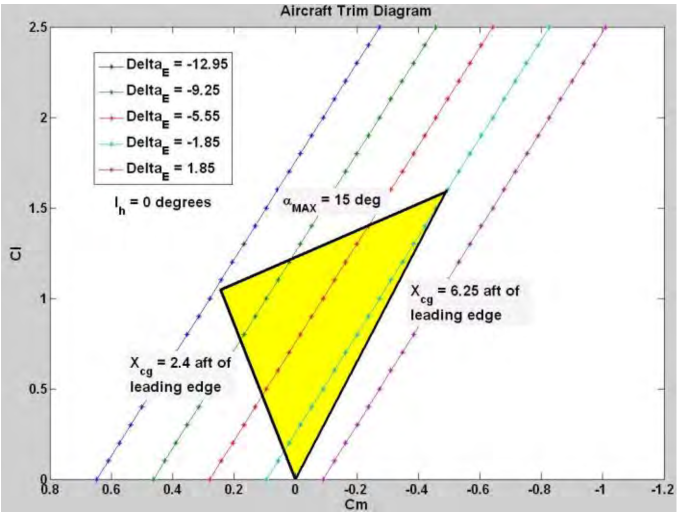


Figure 5.5: Trim Diagram

The lines emanating from the origin outlined in black are the lines where the moment coefficient is equal to zero for its respective center of gravity. The two black boundary lines, from left to right, correspond to the moment when the static margin is 40% and 5% respectively. The trim triangle is bounded above by a line of constant angle of attack. This angle of attack correlates to the stall angle of attack, which is 15 degrees.

By examining the trim triangle, the trim characteristics become evident. The tail on the aircraft is oriented such that there is a zero degree incidence from the wing angle of attack. It can be seen from the diagram that when the position of the center of gravity is father forward, the elevator needs to be deflected more in the upward direction than when the center of gravity is farther back on the aircraft near the aerodynamic center. The current position of the center of gravity is where the static margin is at 32%. This space is inside the trim space since the two boundaries correspond to 40% and 5% static margin. This diagram confirms that the aircraft has been designed in the intended stability range.

5.6 Case Design

The objective for the case design was minimizing weight while maintaining adequate structural integrity to support the aircraft during transit to the flight line. Since no impact loading or drop tests are required, the case was designed for a goal weight of 32 oz. To determine the construction material for this case, several common design materials were considered for weight. Each construction method uses a truss structure covered in hobby covering. Table 5.6 shows the comparative weights for the truss design using different construction materials. Figure 5.6 shows an illustration of this construction technique.

Table 5.6: Case Truss Weight Estimation

Truss Building Material	Weight (oz)
Balsa Wood	10.04
Carbon Fiber Rods	12.85
Corrugated Cardboard	19.68
Corrugated Plastic	30.40

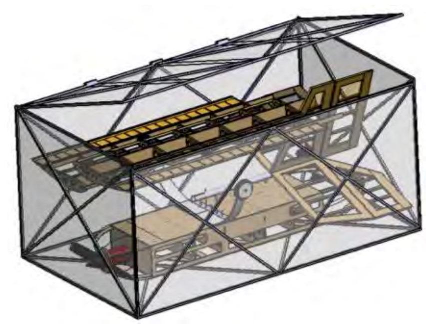


Figure 5.6: Picture of Case

The balsa wood truss technique is the lightest option and was therefore chosen as the construction method for the aircraft enclosure case. This construction technique is also much less expensive and easier to manufacture and repair than the next lightest option; carbon fiber.

In order to quickly unload the softballs, the case will use lightweight snap-type cabinet clasps to secure the lid. This will allow for quick opening and closing of the lid and will add a minimal amount of weight. With the addition of covering, latches, adhesive, the projected case weight is 18 oz, well under the 32 oz goal.

5.7 Flight Performance Parameters

Dimensional parameters and mission profile predictions for the final competition aircraft are summarized in Table 5.7.

Table 5.7: Aircraft and Mission Characteristics

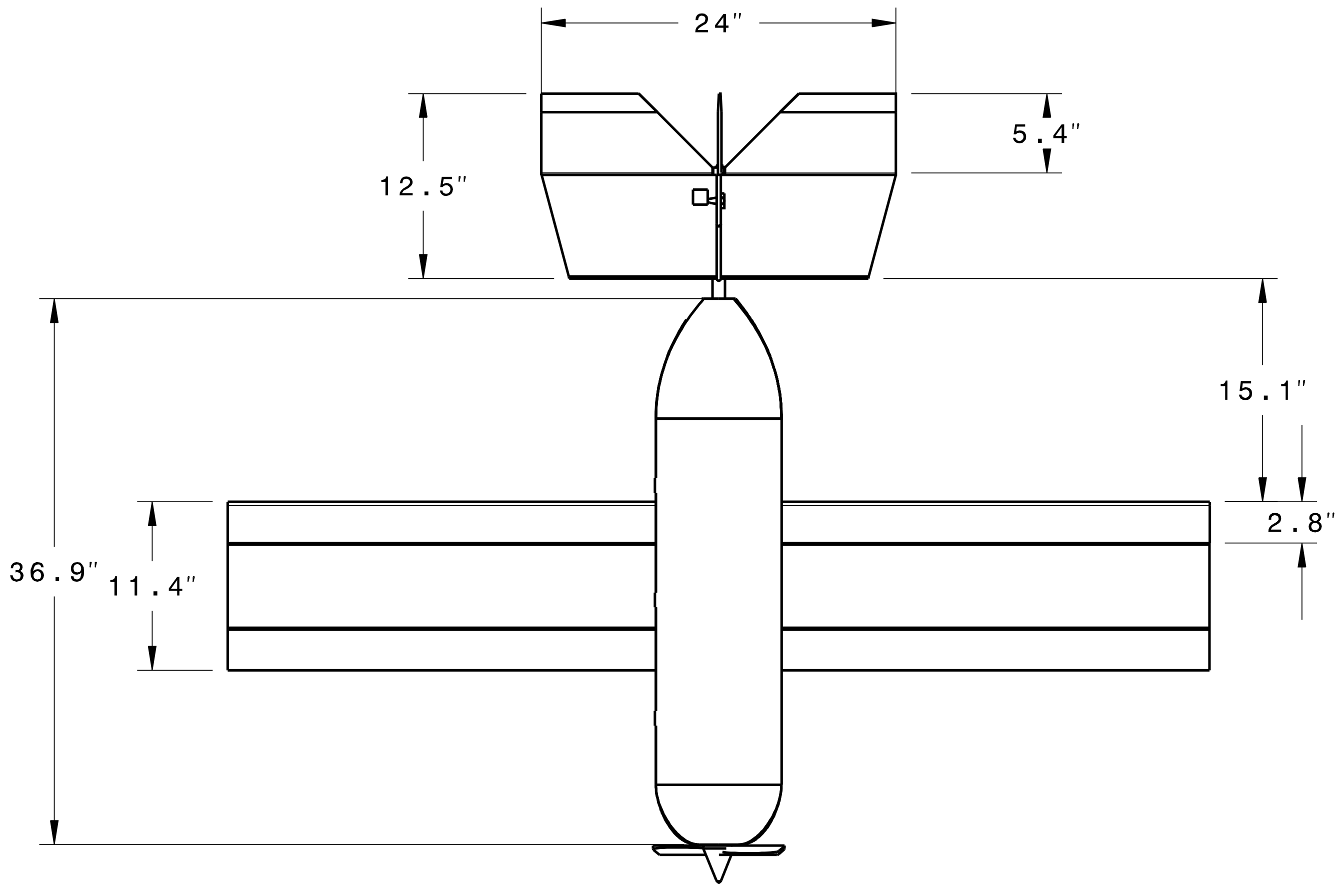
Wing	
Airfoil	SD-7062
Mean Chord (ft)	0.92
Span (ft)	5.50
Area (ft^2)	5.50
AR	6.55
Fuselage	
Length (ft)	3.25
Width (ft)	0.69
Height (ft)	0.35
Tail Geometry (Horizontal)	
Airfoil	Flat Plate
Chord (ft)	1.05
Span (ft)	1.75
Area (ft^2)	0.92
AR	3.34
Tail Geometry (Vertical)	
Airfoil	Flat Plate
Chord (ft)	0.95
Span (ft)	1.00
Area (ft^2)	0.88
AR	1.5

Predicted Performance			
C_L Max	1.50		
L/D Max	101.44		
Weight			
Parameter	Mission 1	Mission 2	Mission 3
Electronics (lbs)	0.12		
Batteries (lbs)	1.14		
Fuselage (lbs)	0.75		
Wings (lbs)	0.78		
Tail (lbs)	0.47		
Landing Gear (lbs)	0.48		
Propeller (lbs)	0.30		
Motor (lbs)	0.38		
System Weight (lbs)	4.92		
Box Weight (lbs)	1.13		
Payload Weight (lbs)	-	4.38	6.25
Gross Flight Weight (lbs)	4.92	9.30	11.17
Systems			
Propeller	Mission 1 & 2	Mission 3	
	APC 16" x 12"	APC 18" x 10"	
Batteries	18 Elite 2200		
Servos	4 Futaba S3150		
Gearbox	6.7:1		
Receiver	Futaba FP-R148DP 8-Channel		
RX Battery	JR 4.8V 700mAh		
Speed Control	Castle Creations Phoneix 45HV		
Motor	Neu 1107/2γ		

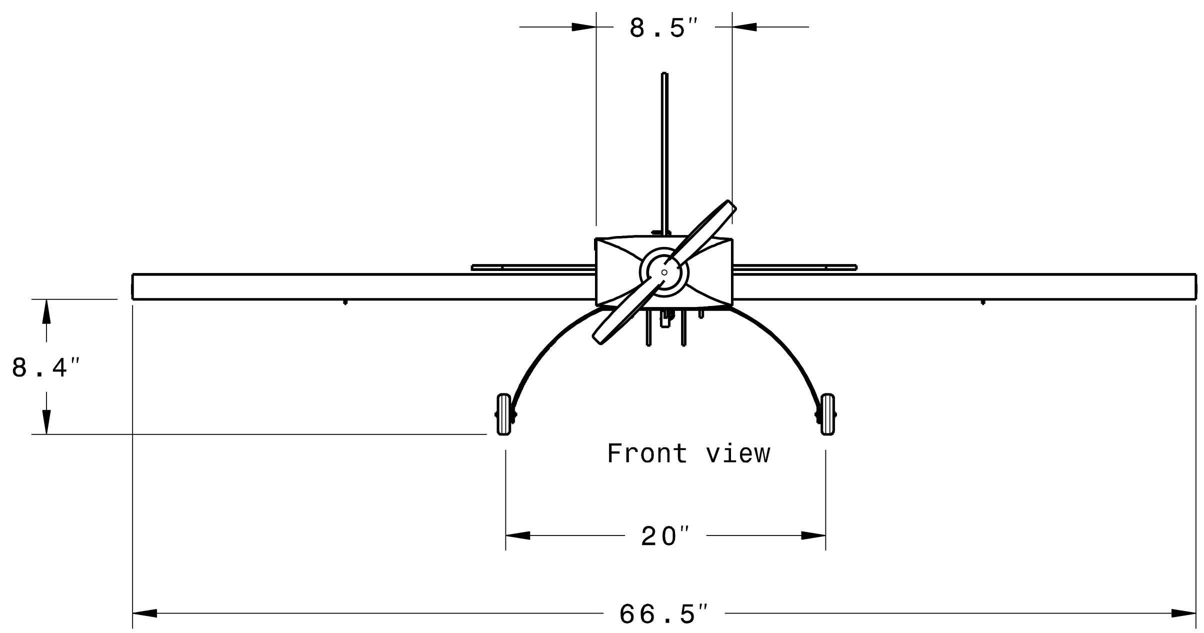
Reference Values		
Mission 1	Flight Time (sec)	65
Mission 2	Loading Time (sec)	10
Mission 3	Flight Time (sec)	150
	Number of Bats	5
System Weight (lbs)		5.5
Mission Specs		
Mission 1	Straight Velocity (ft/s)	80
	Turn Velocity (ft/s)	70
	Flight Time (sec)	79
	Gross Flight Weight (lbs)	4.92
	System Weight (lbs)	6.55
	Score	34.54
Mission 2	Loading Time (sec)	11
	Gross Flight Weight (lbs)	9.30
	System Weight (lbs)	6.55
	Score	76.34
Mission 3	Straight Velocity (ft/s)	65
	Turn Velocity (ft/s)	55
	Flight Time (sec)	147
	Number of Bats	5
	Gross Flight Weight (lbs)	11.17
	System Weight (lbs)	6.55
	Score	97.90
TOTAL SCORE	**208.78**	

5.8 Drawing Package

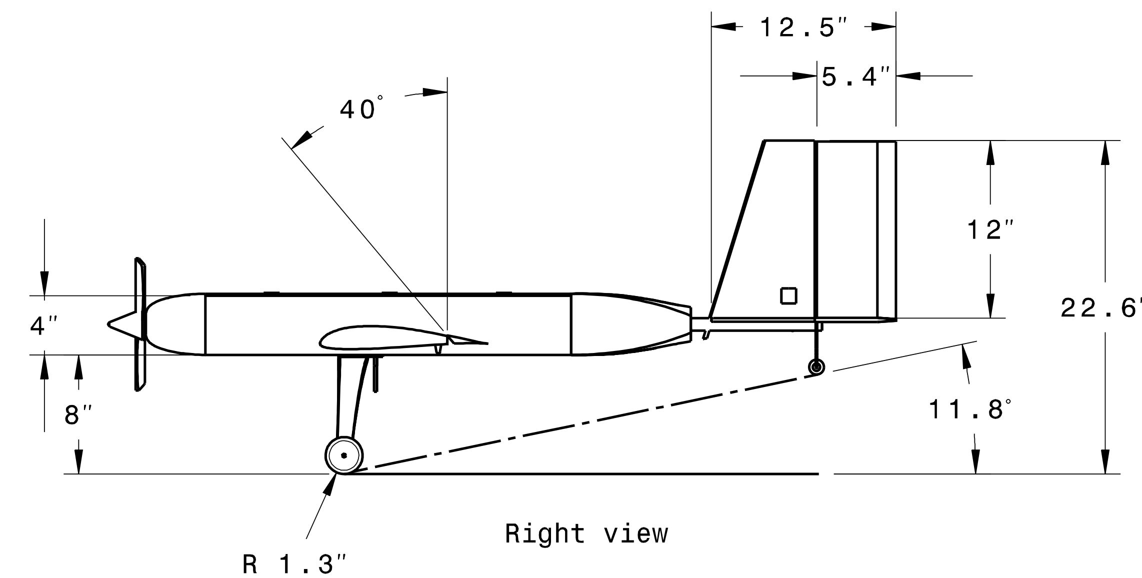
The drawing package was completed by compiling all of the design parameters and selected component configurations from the previous sections. CATIA was the CAD package used for designing components and configurations. Each part was modeled and added to an assembly of the fuselage, wings, or tail. The three parts were put together to form one model that accurately represents the design that Team B‟Euler Up chose. These drawings were used by the team, for not only visualization purposes, but also the manufacturing of parts in the CNC machine. Utilization of a CAD package allows for verification of component clearances and a source of dimensioned part drawings for manufacturing.



Top view

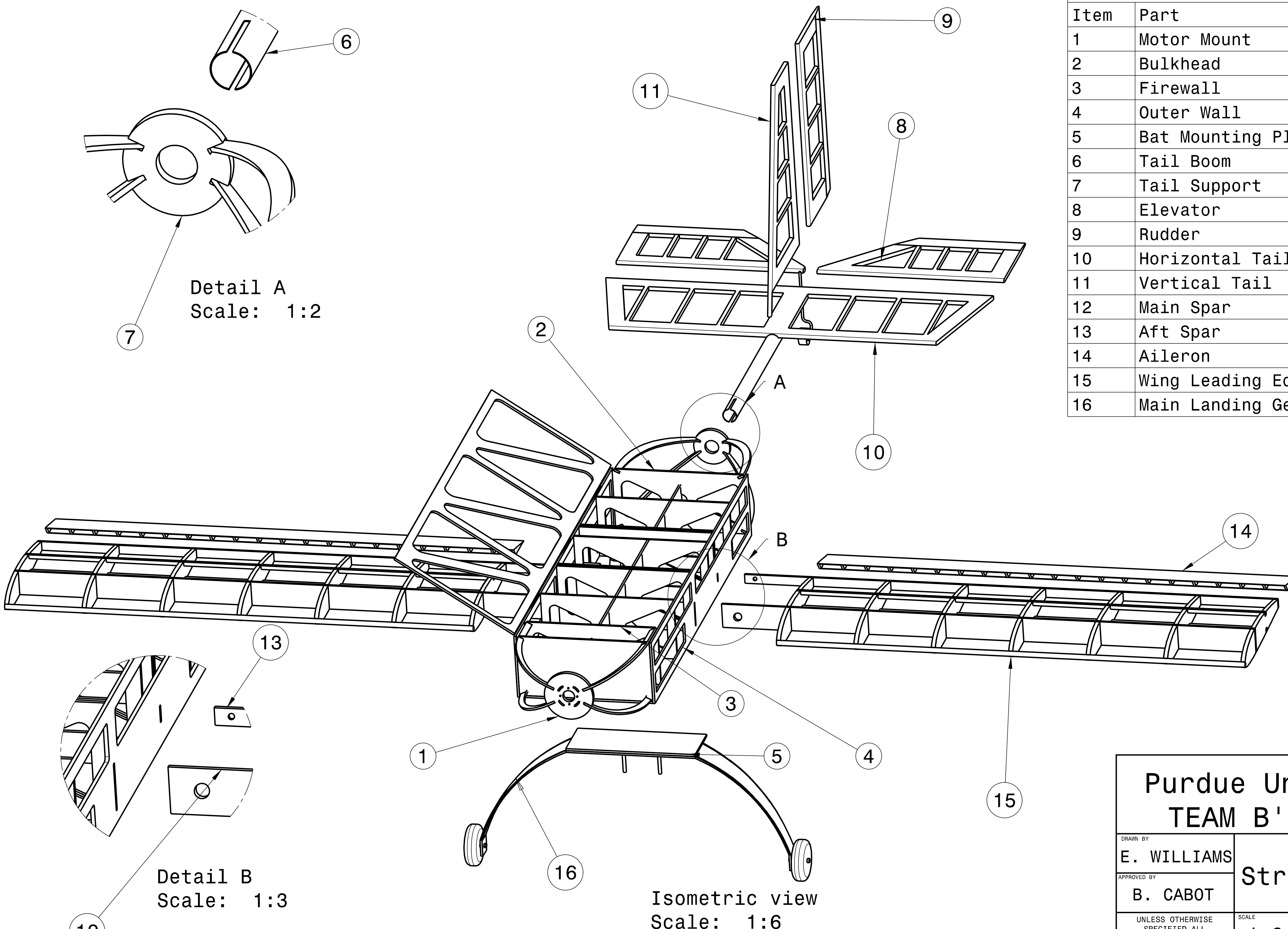


Front view



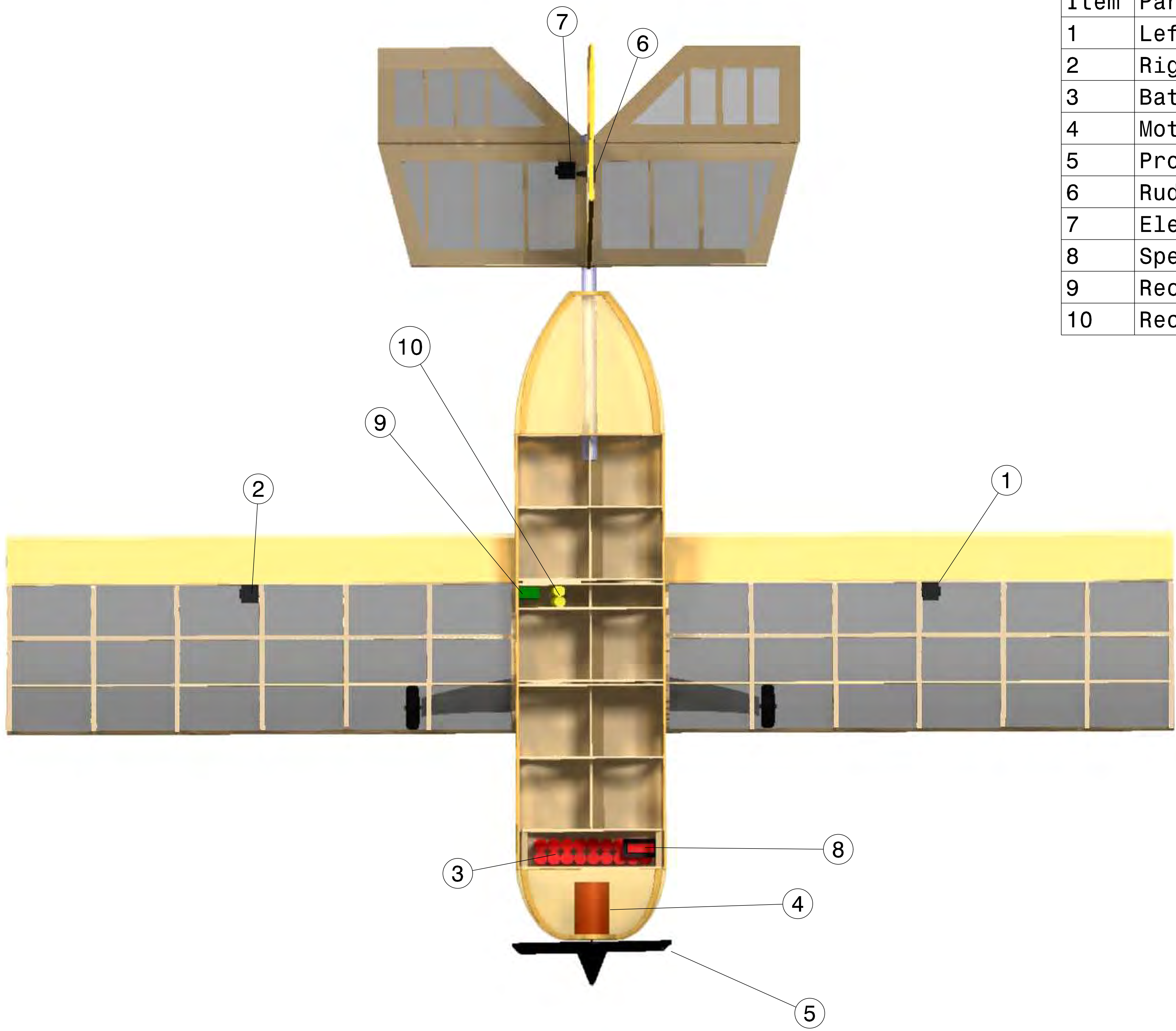
Right view

Purdue University TEAM B'EULER UP			
DRAWN BY E. WILLIAMS	Aircraft Three-View		
APPROVED BY B. CABOT			
UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE GIVEN IN INCHES ANGLE TOLERANCE: 30' DIMENSION TOLERANCE: .03	SCALE 1:12	DRAWING NO. 2010-001	REVISION A
	DATE OF APPROVAL 02/25/2009	REPORT PAGE 40 OF 57	PAGE 1 OF 4



Engineering Bill of Materials			
Item	Part	Material	Qty.
1	Motor Mount	1/4" Plywood	1
2	Bulkhead	Fiberglass/Balsa	5
3	Firewall	Fiberglass/Balsa	1
4	Outer Wall	Fiberglass/Balsa	2
5	Bat Mounting Plate	1/8" Plywood	1
6	Tail Boom	Aluminum	1
7	Tail Support	1/8" Plywood	1
8	Elevator	Balsa	1
9	Rudder	Balsa	1
10	Horizontal Tail	Balsa	1
11	Vertical Tail	Balsa	1
12	Main Spar	Fiberglass/Balsa	2
13	Aft Spar	Fiberglass/Balsa	2
14	Aileron	Balsa	2
15	Wing Leading Edge	Balsa	2
16	Main Landing Gear	Carbon Fiber	1

Purdue University TEAM B'EULER UP				
DRAWN BY E. WILLIAMS	Structural Arrangement			
APPROVED BY B. CABOT				
UNLESS OTHERWISE SPECIFIED ALL DIMENSIONS ARE GIVEN IN INCHES ANGLE TOLERANCE: 30' DIMENSION TOLERANCE: .03	SCALE 1:6	DRAWING NO. 2010-001		REVISION A
	DATE OF APPROVAL 02/25/2009	REPORT PAGE 41 OF 57	PAGE 2 OF 4	



System Component List		
Item	Part	Source
1	Left Aileron Servo	Futaba 53150
2	Right Aileron Servo	Futaba 53150
3	Battery Pack	18 Elite 2200 Cells
4	Motor	Neu 1107/2
5	Propeller	APC 16x12 / APC 18x10
6	Rudder Servo	Futaba 53150
7	Elevator Servo	Futaba 53150
8	Speed Controller	Castle Phoenix 45HV
9	Receiver	Futaba FP-R148DP 8-Channel
10	Receiver Pack	JR 4.8V 700 mAh

Purdue University
TEAM B'EULER UP

DRAWN BY
E. WILLIAMS

APPROVED BY
B. CABOT

Systems Layout

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ANGLE TOLERANCE: 30'
DIMENSION TOLERANCE: .03

SCALE
1:12

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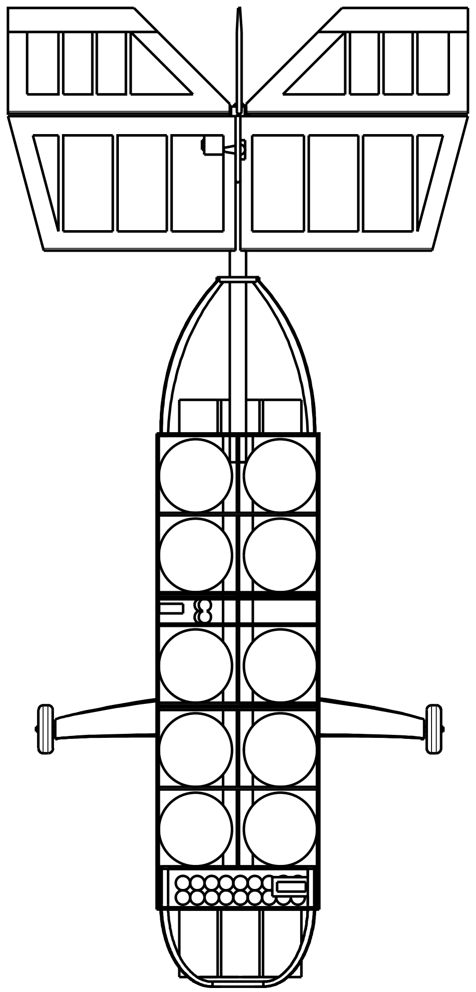
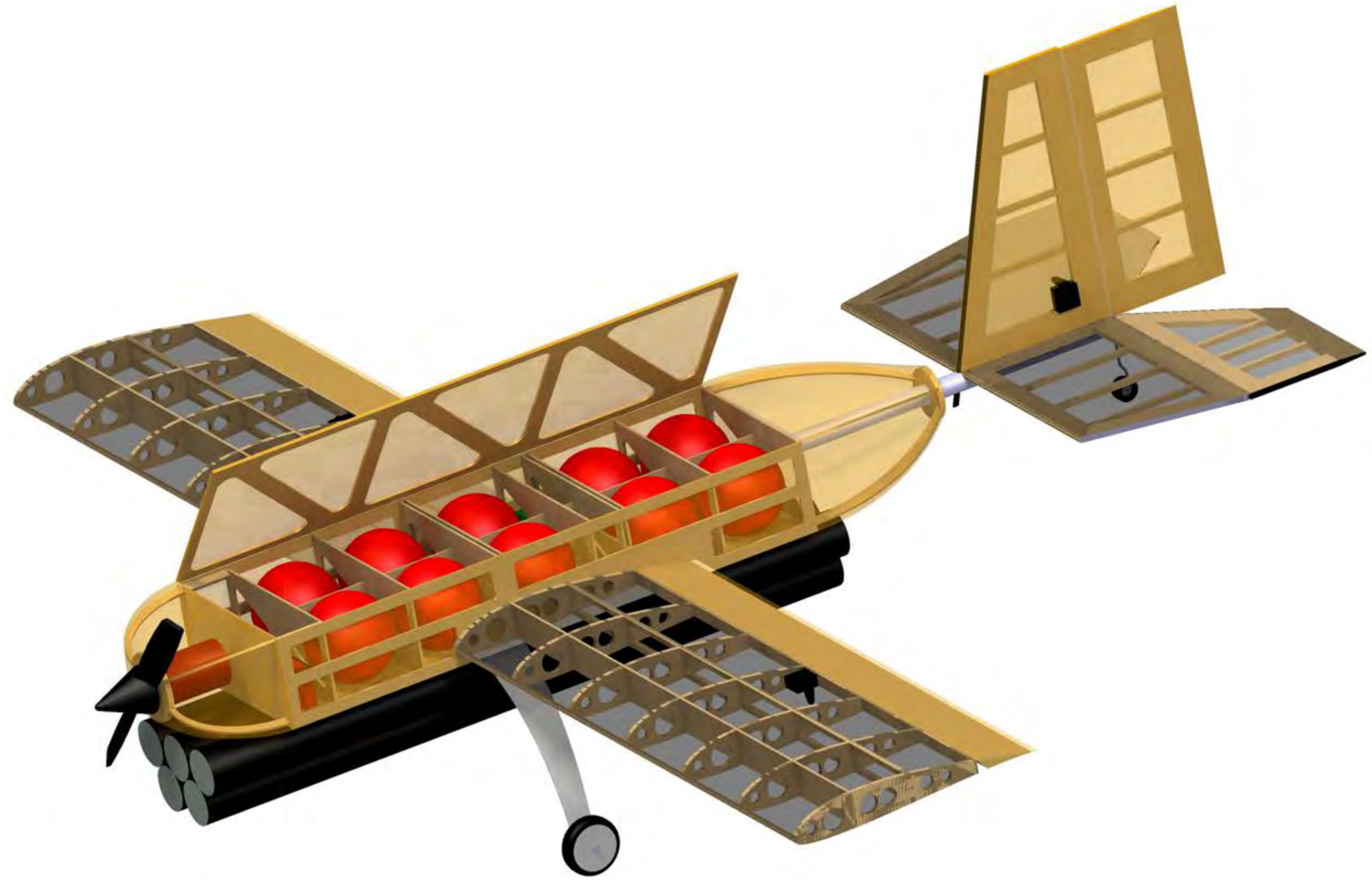
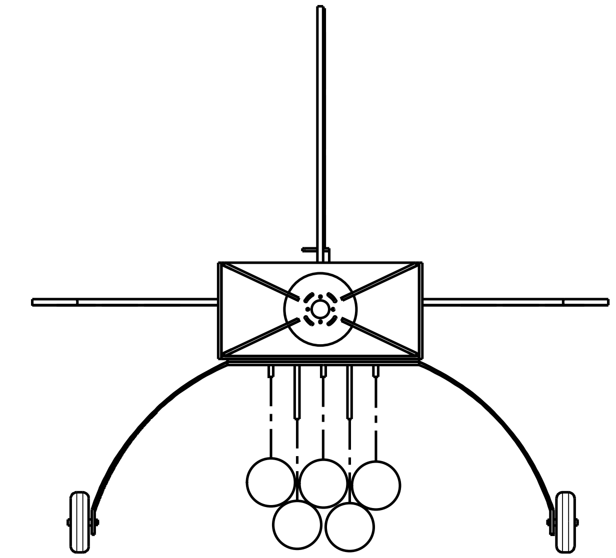
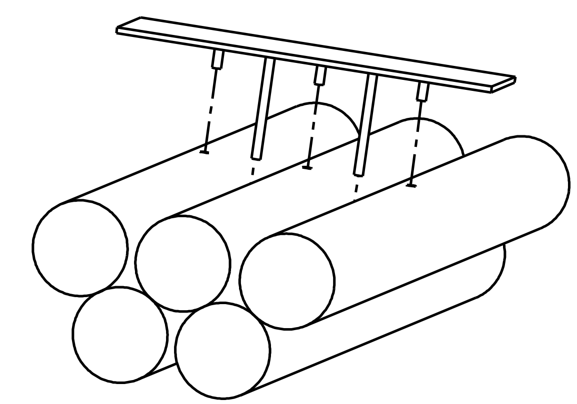


Top view with Balls Loaded
Scale: 1:10

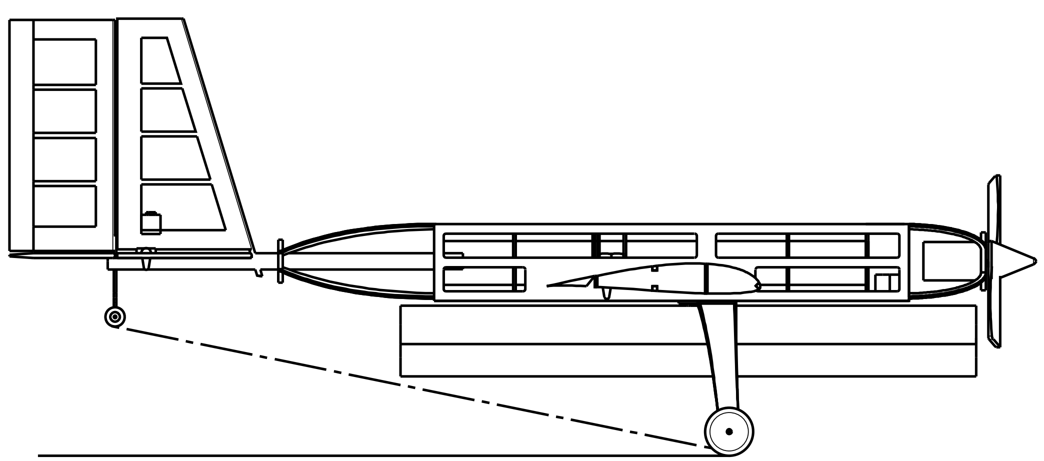




Bat Mounting Front View
Scale: 1:8



Bat Mounting
Isometric view
Scale: 1:4



Bat Clearance View
Scale: 1:10

Purdue University
TEAM B'EULER UP

DRAWN BY
E. WILLIAMS

APPROVED BY
B. CABOT

Payload Accomodation

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ANGLE TOLERANCE: 30'
DIMENSION TOLERANCE: .03

SCALE
1:7

DRAWING NO.
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6.0 MANUFACTURING PLAN & PROCESSES

The manufacturing process used in the construction of the final design was a combination of conventional model aircraft construction techniques and newer techniques in an effort to produce the lightest airframe possible. Construction methods of previously successful teams were investigated and implemented to determine the best processes that results in the lightest airframe.

6.1 Process and Material Selection

To optimize the aircraft‟s weight, it was necessary to explore the advantages and disadvantages of several different building processes and building materials. This analysis would dictate the design of critical components so that they are as strong and as light as possible.

6.1.1 Manufacturing Methods and Materials

Several options were considered for the manufacturing process for the wings and the fuselage.

- **White Foam with Fiberglass**: White Styrofoam is cut by a hot-wire cutter to the desired shape and then covered with fiberglass and resin in a wet lay-up. The fiberglass skin carries the majority of the loads of the structure and is a common method for the manufacturing of wings.
- **Pink Foam Cored**: This method utilizes pink extruded polystyrene foam to create its structure. To minimize weight, any non-structural area is cored. The wing is then covered with MonoKote. This method has the advantage of easy construction and durability.
- **Balsa Build-Up:** This method can be light weight with a high strength to weight ratio. Wood frame components can be precisely manufactured from CAD drawings using a CNC water-jet. Duplicate parts can be easily cut, facilitating a relatively simple field repair.
- **Molded Composite:** A female mold, made out of medium density fiberboard, can be manufactured from CAD drawings. Then composite laminate is laid in the mold, vacuum bagged, and allowed to cure. This method requires many steps; however the final product can be very light and strong.

6.1.2 Manufacturing Methods and Materials Figure of Merit

The following four FOMs were employed to select a proper manufacturing method. Each FOM was given a numerical value for weight and each configuration was rated on a scale from one to four with higher numbers better meeting the FOMs.

- **Weight**: Minimizing the weight of components is the key to a high flight score, so this was rated as the most important FOM.
- **Ease of Construction:** The amount of time involved in manufacturing and the expertise necessary to build the component should be taken into consideration.
- **Reparability:** During the competition, field repair must be conducted efficiently and quickly.
- **Durability:** The aircraft must have the ability to fly a minimum of five flights.

Table 6.1: Manufacturing Methods and Materials Figure of Merit

Materials Figure of Merit							
	Figure of Merit	Weight (%)	White Foam w/ Fiberglass	Pink Foam Cored	Balsa Build-Up	Molded Composite
Wings & Tail	Weight	55	2	1	4	2
	Ease of Manufacturing	20	3	4	3	2
	Repairability	15	3	2	2	4
	Durability	10	3	2	1	3
	Total	100	245	185	320	240
	Figure of Merit	Weight (%)	White Foam w/ Fiberglass	Pink Foam Cored	Balsa Build-Up	Molded Composite
Fuselage	Weight	55	2	2	4	2
	Ease of Manufacturing	20	3	4	3	2
	Repairability	15	2	2	3	3
	Durability	10	2	2	2	4
	Total	100	220	240	345	235

6.2 Manufacturing Process of Major Components

The manufacturing process of each main component has been chosen to facilitate the ease of construction and reparability.

6.2.1 Fuselage Construction

A more conventional balsa wood-frame and MonoKote construction was selected to manufacture the fuselage. This method allowed for more team involvement as less experience is necessary for this type of construction. A 3-D drafting model was created and used to cut out the parts on a CNC water-jet. This technique insured a high degree of precision on the parts. Joints that were expected to see higher stresses, such as at the motor mount, were attached using epoxy for more strength while other joints were bonded with cyanoacrylate glue.

The fuselage contained a spar box to hold the wings in place. This spar box was built up using 1/8” balsa fiberglass sandwich structure because of the high stress concentrations in this area. To prevent the wings from sliding out of the spar box, a pin was passed through both the spar box and each wing‟s spar.

6.2.2 Wing Construction

The main spar was also a balsa fiberglass sandwich structure. A 1/16” balsa sheet was coated with a light-weight fiberglass weave. A second layer of fiberglass was added to the inboard half of the spar. This was to insure that the root of the spar sufficiently strong to handle the increased bending moment at that location.

The wing structure was constructed out of balsa, again using the water-jet to cut the pieces. 1/8” thick balsa was used for the ribs, while 1/4” square balsa rods were used for stringers along the top and bottom of the wing, as well as along the leading edge. The leading edge was sanded to the appropriate contour. A specially designed jig was used to hold the wing in place and ensure that all the pieces were properly aligned during assembly. A second spar was used at the trailing edge of the wing where the aileron is attached. This second spar was made using balsa construction. The wing was covered using MonoKote.

The aileron was built up by sheeting balsa ribs with 1/64” balsa. Hinge tape was used to attach the ailerons along the top surface. These techniques are well established practices in RC aircraft construction.

6.2.3 Tail Construction

The tail is constructed using balsa. 1/4” balsa stock was cut to form the outline of the surface shape and featured a truss structure to add strength and support the MonoKote covering. A carbon fiber rod was used along the leading edge of the vertical stabilizer and attached through the tail boom to ensure structural integrity.

6.2.4 Landing Gear Construction

A pre-fabricated carbon fiber landing gear was used to minimize assembly time and reduce weight. The landing gear was bolted into the fuselage with a 3/32” plywood mounting piece.

6.2.5 Payload Restraint

The payload of bats was constrained by attaching a piece of 3/32” plywood to the bottom of the aircraft. This section had five carbon fiber rods to hold the maximum amount of bats in place horizontally. Two zip-ties encircled the bats which were attached through the fuselage to hold them in place. This was perceived to be the lightest possible option.

6.2.6 Assembled Prototype

Figure 6.1: Assembled Prototype

6.3 Manufacturing Schedule

A detailed manufacturing schedule was created to assist with the organization of manufacturing processes. This schedule can be view below in Figure 6.2.

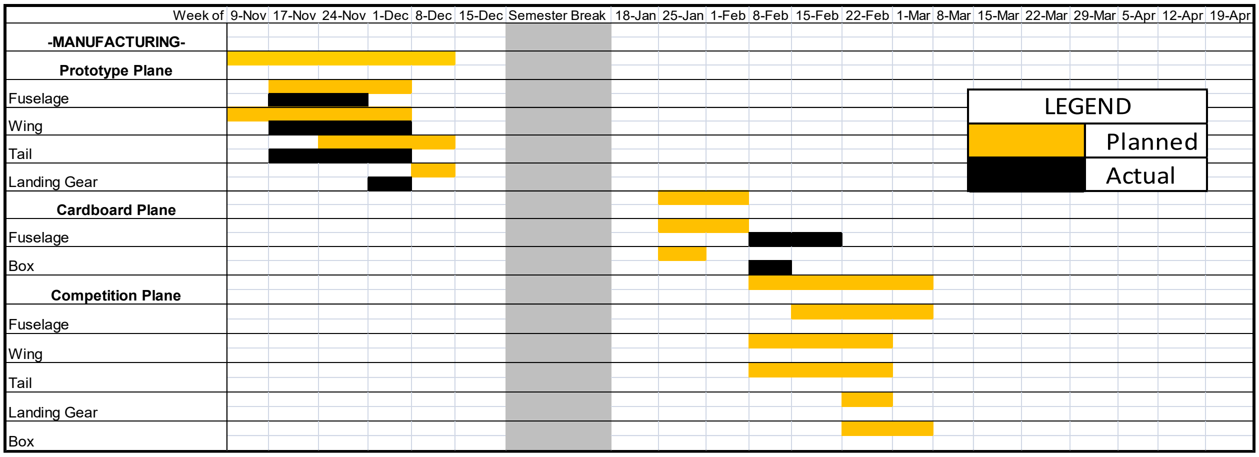


Figure 6.2: Manufacturing Schedule

7.0 TESTING PLAN

Based on experiences learned in the first design spiral, testing was considered an integral part in verifying the spiral design process. Six areas were explored: structural, aerodynamics, propulsion, payload, ground, and flight testing. The objectives were to verify the structural integrity of the airframe and landing gear, the aerodynamic characteristics of the integrated assembly, the motor and propeller combination, deployment methods, functionality of all aircraft systems, and the flight model. The competition mission was practiced by the pilot and ground crew for familiarity. The testing milestone chart is given in Figure 7.1, with the planned schedule in gold and the actual schedule in black.

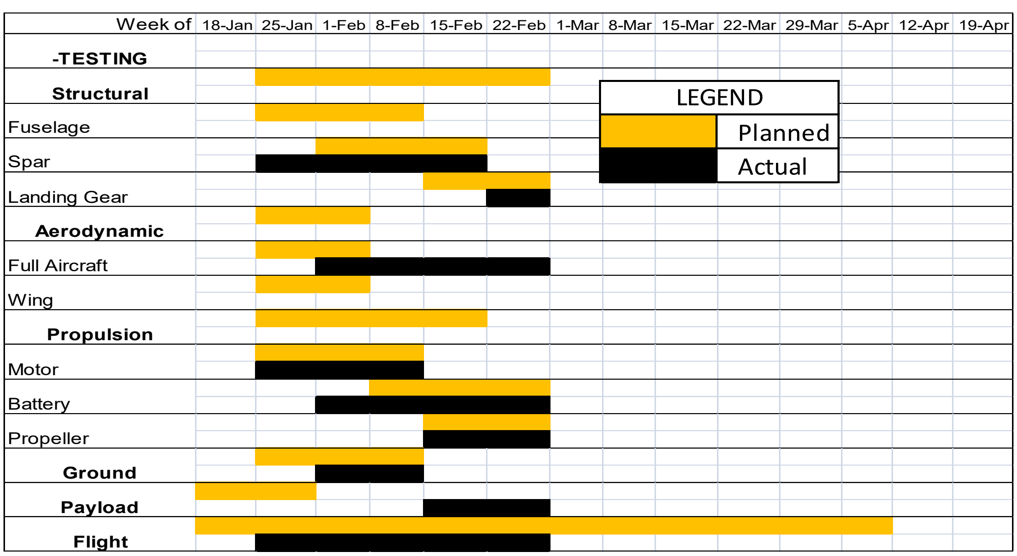


Figure 7.1: Testing Milestone Chart

7.1 Structural Testing

Structural testing was an integral part in confirming the analysis completed in the detailed design.

7.1.1 Wing Spar Box

To verify the structural integrity of the aircraft, tests were performed on key structural components. The wing spars were tested to ensure they would be able to handle the loads expected during flight and the tip test required during technical inspection. A mock-up of the wing spar was created at one third the length and was tested to failure. To test the wing spar box, one end was fixed and a load was applied to the free end, as shown in Figure 7.7.2. Incremental loading of 5 lbs was applied until failure. This experiment concluded a maximum load of 83 ft-lbs corresponding to roughly 2.5 Gs.

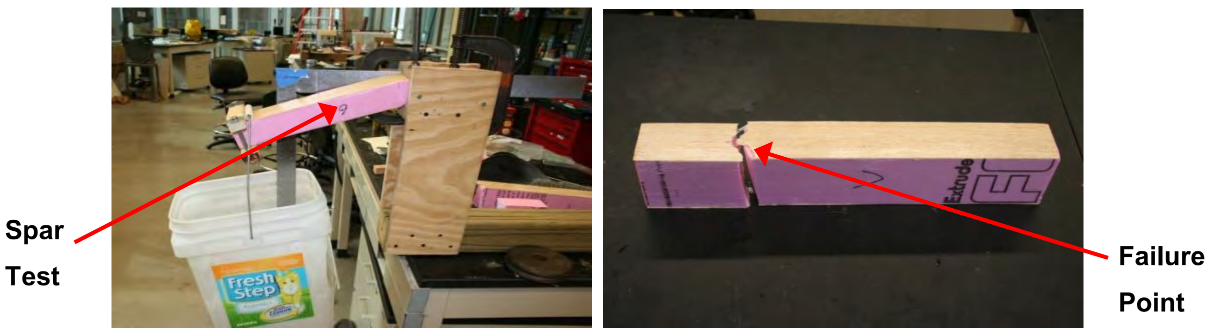


Figure 7.7.2: Spar Testing (a) During Test and (b) After Failure

7.1.2 Landing Gear

The selected carbon fiber landing gear was tested to ensure that it could withstand the loads expected in the event of a rough landing. The landing gear was held upright and weight was added until a 3 G load was reached. It was determined that the ability to handle a 3 G load would meet and exceed any load that would be experienced in flight. The landing gear handled a 35 lbs load without fracture. This test concluded that the carbon fiber landing gear selected for use was within design requirements.

Figure 7.3: Landing Gear Under 3 G Simulated Load

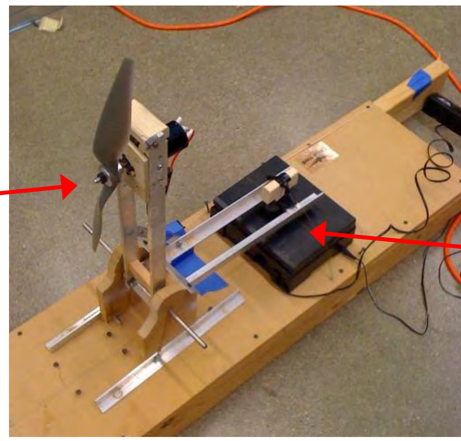
7.2 Aerodynamic Testing

Wind tunnel testing was used to evaluate the aerodynamic characteristics of the integrated assembly, including drag polar and lift coefficient versus angle of attack. These are then compared to the results of our preliminary constraint analysis. Results are provided in the performance section.

7.3 Propulsion Testing

The propulsion testing consisted of a static motor endurance test and dynamic motor tests for several propulsion configurations. The dynamic motor test simulated Mission 1, Mission 2, and Mission 3. Throughout each test, the RPMs, temperature, wattage, voltage, current, throttle position and thrust were recorded using an EagleTree data logging system. Figure 7.4 shows the propulsion system test stand.

Propeller



Digital Scale

Figure 7.4: Propulsion System Test Stand

Propulsion wind tunnel and flight testing was conducted to determine thrust and power data for propulsion system optimization. Obtaining experimental data allowed the team to evaluate how well the historical data and flight codes modeled the performance of the propulsion system, and then further optimize the system. Results are provided in the performance section.

7.4 Payload Testing

Ground mission tests involved testing the payload loading mechanisms, refining the payload compartment, and allowing the ground crew to practice for the competition. The payload restrain mechanisms will be refined to minimize the time required. During these ground tests the competition ground crew will be selected based on time trials. Results are provided in the performance section.

7.5 Ground Testing

Ground testing was conducted in order to ensure the functionality of all aircraft systems. The receiver was connected to all control servos and the propulsion system. The pilot deflected all surfaces to ensure no interference between the surfaces and all controls were properly connected. Several different weights and throttle settings were tested to cover the operating range of the aircraft and pilot‟s comments on handling were recorded.

7.6 Flight Testing

Flight testing was divided into two sections: initial evaluations and mission simulation. An onboard data acquisition system was implemented for all flights to diagnose problems that may occur and to evaluate actual versus predicted performance. A flight testing checklist, as shown in Figure 7.5, was used to ensure that proper pre-flight and post-flight procedures are performed.

Purdue B'Euler Up Flight Test Card

Flight Test #	4
Date	1/10/10 4pm
Mission Objectives:	Simulate Mission 1

Pilot	Nick
Pilot Comments:	

Batteries

	Transmitter	Receiver	Propulsion
Secure	√	√	√
Voltage Pre-Flight (V)	10.0	5.0	16.7

Weather

Temperature (F)	41
Wind Velocity (MPH)	3
Wind Direction	S/SW
Sky Conditions	Clear

Startup Procedure

CG Check	√
Transmitter Programming	√
Extend Transmitter	√
Transmitter **ON**	√
Receiver **ON**	√
Motor **ON**	√
Static Test	√

Shutdown Procedure

	OFF
Batteries	√
Receiver	√
Transmitter	√

	Transmitter	Receiver	Propulsion
Battery Heat	Cool	Cool	Cool
Battery Voltage (V)	9.7	4.8	15.1

Radio

Range Check	√
Directional Check	√

Figure 7.5: Flight Test Card

The initial evaluation portion of flight testing focused on performance, handling, and control evaluations. The first flight consisted of evaluating the controllability of the aircraft as well as making any necessary trims to the control surfaces. The next several flights allowed the pilot to evaluate the handling of the aircraft in several flight regimes from high speed cruise all the way down to the stall. The final portion of the first phase made use of the on board data acquisition system to determine aircraft drag and power.

Once any major control issues were resolved, mission simulation began. The first mission simulation was Mission 1. Afterwards, the aircraft was loaded with 1, 2, 3, 6, and 10 softballs (each on separate flight) in order to evaluate control aspects of the aircraft with the respective loads. The flight path of Mission 2 was not simulated since the mission is not timed.

Due to limitations in resources at the field, only 3 bats could be attached to the aircraft for Mission 3 testing, as opposed to the planned 5. The test was solely to evaluate control characteristics of the aircraft under the given load.

8.0 PERFORMANCE RESULTS

To examine the actual performance of the aircraft, testing was done on the propulsion system and plane aerodynamics in addition to multiple flight tests. These tests are critical in the final confirmation of the detailed design analysis and expected flight characteristics.

8.1 Subsystem Performance

To optimize the propulsion system and validate calculations and predictions made, testing was conducted on the motor, battery, and propeller. This testing would dictate the final configuration of the propulsion system and provide empirical data to back up theoretical predictions.

8.1.1 Propulsion Testing

To supplement the computer analysis discussed in detailed design with propeller performance characteristics, a series of wind tunnel tests were conducted on different propellers within the diameter range of 12” to 20”. The experimental setup is shown in Figure 8.1. All of the propellers were tested on an AXI 4130 motor in Purdue‟s Boeing wind tunnel. The measurement system used consisted of the wind tunnel force balance (measuring thrust and drag), the Eagle Tree Data Logger (measuring RPM), and the Transducer Technologies torque sensor. Tests were conducted at different advance ratios, which is a function of forward velocity and propeller RPM. Experimental results were then used to evaluate the non-dimensional propeller parameters, namely advance ratio, coefficient of thrust, coefficient of power, and propeller efficiency.

Experimental results were then validated against Goldstein‟s Propeller Theory[7], which gives an exact solution for propellers with finite blades operating under light loading. Goldstein‟s theory predicts propeller performance based on blade element theory, a bound vortex model (accounts for circulation distribution), and the vortex model (for induced velocity). In this approach, the propeller is stripped into a number of sections and at each section beta (zero lift line angle), chord, and airfoil characteristics are specified. Forces and power at each section are evaluated; total force and power are then obtained by integrating these sectional forces and power from root to tip.

Figure 8.1 through Figure 8.3, on the next page, summarize the experimental and Goldstein‟s theory results for an 18” x 10” propeller. It can be observed from these figures that Goldstein‟s theory accurately predicts and validates the obtained experimental results.

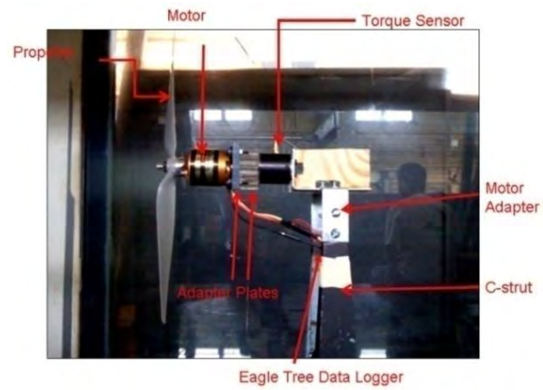


Figure 8.1: Experimental Setup of Propeller

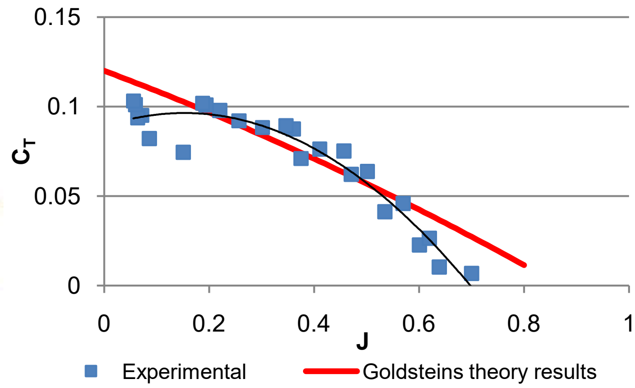


Figure 8.2: C_T vs. J (18" x 10" Propeller)

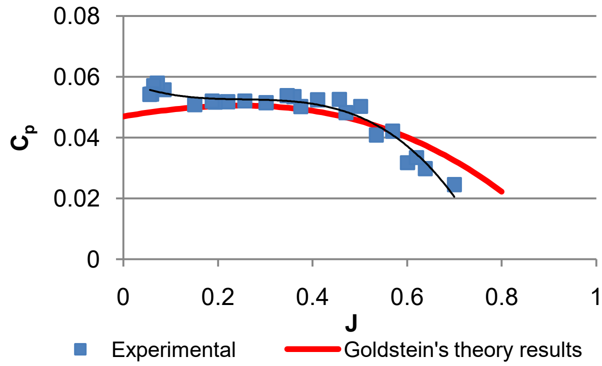


Figure 8.3: C_P vs. J (18" x 10" Propeller)

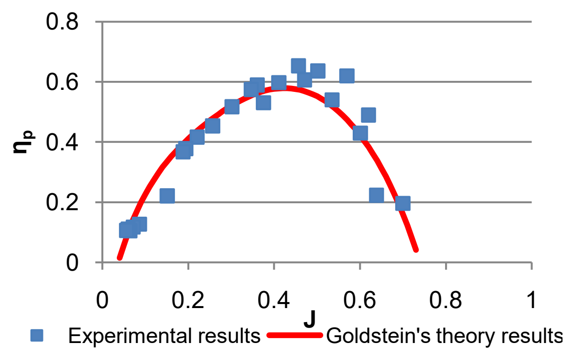


Figure 8.4: η_P vs. J (18" x 10" Propeller)

Analytical propulsion analysis was verified with full scale wind tunnel testing of the aircraft. Mission 3 was simulated by setting the aircraft with 5 bats at zero degrees angle of attack on a lift/drag balance.

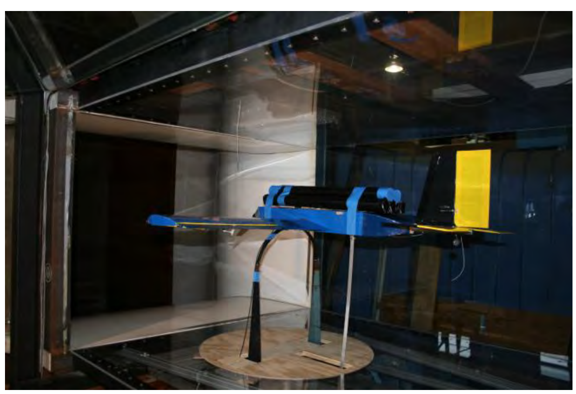


Figure 8.5: Mission 3 Propulsion Wind tunnel Test

The wind tunnel was set to a flow speed of 65 ft/s and the motor was accelerated until the drag reading on the lift drag balance was zero. The current draw, propeller RPM, and power required were recorded during the test run using an Eagle Tree Data Logger.

Table 8.1: Comparison of Analytical and Experimental Propulsion Data

Team B'Euler Up	Mission 3 Propulsion Comparison		
	Analytical	Experimental	Percent Difference (%)
Current (A)	23.43	33.38	29.81
Power (Watts)	506.09	470.67	-7.53
RPM	6206.49	5485.82	-13.14
Thrust (lbf)	7.08	4.91	-44.20

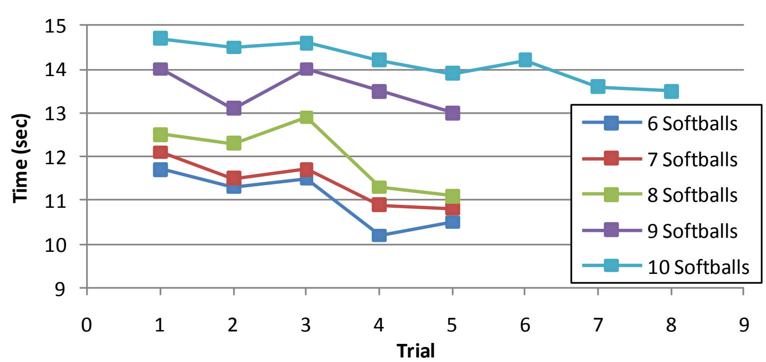
The predicted thrust was lower than the actual thrust value and the current draw was higher than expected. However, the wind tunnel tests showed that the selected battery pack was still able to fly Mission 3 with the increased power requirements. The experimental propeller RPM was used to recalculate the propeller efficiencies at the tested flight speed and the 18" x 10" propeller remained the best choice for Mission 3.

8.1.2 Battery Testing

Initial flight tests were conducted using Elite 1500 cells and it was found that they did not have adequate amp-hour capacity. Propulsion tests were conducted in the wind tunnel and it was found that cells rated at 2200 mAh would be required. Mission analysis showed that a 19.2 volt pack would be necessary to meet the power required while keeping the current draw under 40 amps.

8.1.3 Payload Testing

Payload loading time for Mission 2 is a large part of the overall score, so the team organized a ground crew to practice loading the aircraft. The ground testing setup was similar to what is to be expected at the competition. The ground crew started 10‟ behind the payload, which was separated 20‟ from a cardboard mock-up of the aircraft. A cardboard mock-up of the aircraft was built so team members were able to practice loading the payload configurations and to highlight any unforeseen problems that may need to be analyzed prior to final construction. The time required for the ground crew to pick up the required payload, load the aircraft, and return to the start line was recorded. Figure 8.6 compares the loading times for each payload combination.



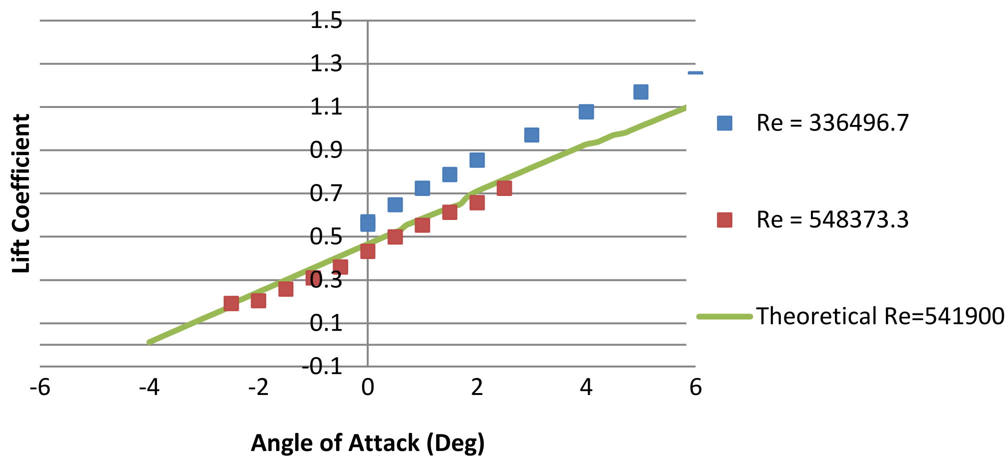
The ground crew required more trials for combinations that were especially challenging, especially ones which had more payload items to load in the aircraft. By perfecting loading techniques the ground crew was able to reduce the loading times of each payload combination. The individuals on the ground crew developed specific roles, carrying the payload or opening the restraint hatch. Contingency plans were developed in the event that the payloads are dropped in the process of the loading. The selection of the ground crew was based on results of the tryouts and consists of three people plus one alternate.

8.2 Aircraft Performance

Performance predictions of the entire aircraft were validated using extensive wind tunnel and flight tests.

8.2.1 Wind Tunnel Testing

The completed aircraft was brought to Purdue‟s Boeing Wind Tunnel in early February to evaluate the aerodynamic characteristics of the integrated assembly and in order to have data for which historical data could be compared. The wind tunnel was run at two speeds so that two different flow regimes (based on Reynolds number) could be analyzed. Reducing the wind tunnel data resulted in Figure 8.7.



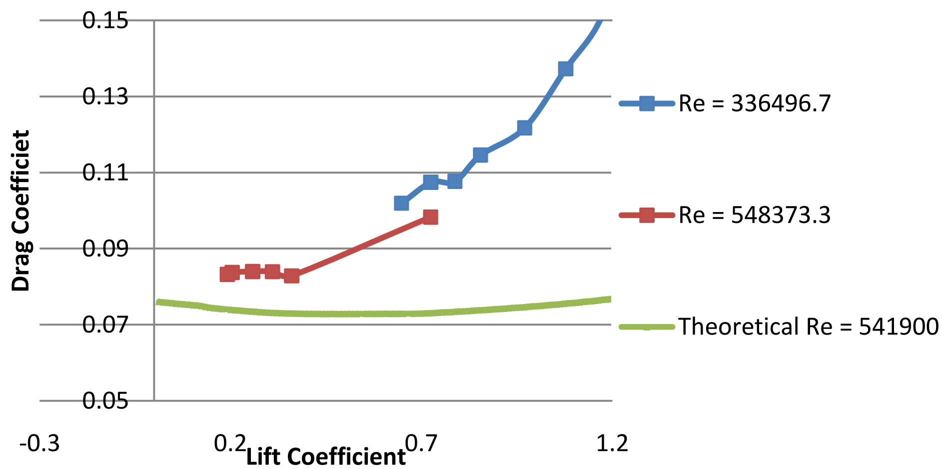


Figure 8.7: (a) Lift Coefficient vs. AOA (b) Drag Polar

These curves indicate how accurate our initial aerodynamics assumptions were. Using the preliminary aerodynamics method located in section 4.2, the lift coefficient for a wing using the SD 7062 airfoil was modeled at the aircraft‟s current wingspan of 5.5‟. More than an adequate amount lift was found at a span of 6‟, so it was determined that decreasing the span to 5.5‟ was acceptable.

Running at a similar Reynolds number to the wind tunnel test, the coefficient of lift versus alpha plot was compared to the wind tunnel data. In Figure 8.7a, the green theoretical data set matches up with the experimental lift curve at a Reynolds number of 548373. Also, the max C_l found through the wind tunnel test was about 1.5 and C_{D0} was found to be about 0.08, whereas the max C_l and C_{D0} values assumed when performing constraint analyses were 1.5 and 0.065, respectively.

Table 8.1: Lift Comparison

Lift Comparison (0° Angle of Attack)		
Airfoil	Wingspan (ft)	Lift (lbs)
SD7062 Theoretical	5.5	17.55
SD7062 Wind Tunnel	5.5	17.51

In addition, the estimated lift values and actual wind tunnel lift values were compared to judge the accuracy of the aerodynamic calculations. The theoretical values for the lift of the aircraft were calculated using the aerodynamic model in the Preliminary Design. In Table 8.1, the theoretical and wind tunnel values of lift at 80ft/s and zero degrees angle of attack are compared. This comparison helps validate the mathematical model created and used in the preliminary design aerodynamics section.

8.3 Flight Performance

The prototype aircraft was completed by the middle of December in time for a maiden flight test before the winter break. Unfortunately, during the first flight test a pilot error led to a crash after 30 sec of flight time. The front of the aircraft was destroyed, the tail broke off and the right wing spar cracked completely at the root.

Repairs were undertaken, taking into account discoveries made during the brief testing. The most significant change made was a new tail with a shorter boom length, as the initial tail gave too much control. Also, the MonoKote on the wings went from solid color all around to blue on top and yellow on bottom, so the pilot could better see the orientation of the aircraft. The repaired aircraft has made over a dozen successful flight tests to date.

One of the most important discoveries made from the flight testing was the excessive current draw of the prototype. An EagleTree onboard data logger recorded the volts and amps drawn during the flight tests. The data gathered from one of the tests simulating Mission 3 is presented in Figure 8.8 on the next page.

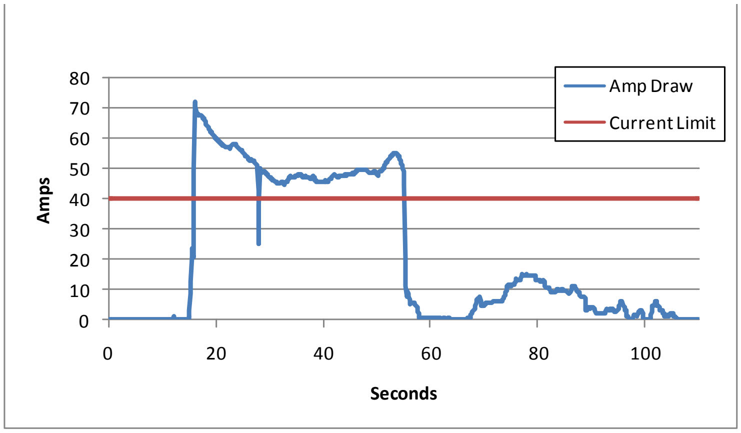


Figure 8.8: Current Draw vs. Time for Mission 3 Analysis

This figure clearly shows the current spike at throttle up and the current drop during a stall test around 60 seconds. The contours of this plot show that the propulsion system used for the prototype far exceeded the 40 Amp limit imposed by the rules. However, after the stall test, the aircraft continued to fly well at a lower power setting (between 5 and 15 Amps).

Other results and observations from flight testing are listed in Table 8.2. Also listed are any comments relevant to the flight and any pertinent data gathered.

Table 8.2: Aircraft Performance Evaluation

Flight Number	Description	Comments and/or Solutions
Flight #1	First test flight	Aircraft crashed after 30 seconds
Flight #2	First flight of repaired aircraft	Pilot trimmed the aircraft and performed stall testing.
Flight #3	Familiarization flight	Pilot tested the plane further, performing various maneuvers to test controllability.
Flight #4	Propulsion testing	First flight with NiMH batteries. Previous flights used Lithium Polymer packs.
Flight #5	Mission 1	Completed in 71 seconds.
Flights #6-8	1-3 softballs all placed aft	Pilot tested handling qualities of the plane with modified C.G.
Flight #9	6 softballs far aft (idealistic mission 2 simulation)	
Flgiht #10	Familiarization flight (new day of flight tests)	During the flight, MonoKote covering the left wing ripped off. This was repaired in the field with masking tape.
Flight #11	10 softballs (worst case mission 2 simulation)	Aircraft flew mission nominally
Flight #12	3 bats (NOT a mission 3 simulation)	Only 3 bats could be mounted. The flight tested handling charasteristics of the aircraft with the given load.

8.4 Performance Results Summary

Table 8.2 summarizes several key performance parameters. Testing results have demonstrated that the overall aircraft design is on a steady pace for continuous improvement and refinement.

Table 8.2: Performance Evaluation Summary

Criterion	Predicted	Tested	Method
Ferry Flight Takeoff Distance (ft)	12	10	Flight Testing
Payload Flight Takeoff Distance (ft)	71	62	Flight Testing
Mission 1 & 2 (Watts of Power)	300	381.8	Wind Tunnel Testing
Mission 3 (Watts of Power)	300	371.1	Wind Tunnel Testing
C_{lmax}	1.5	1.5	Wind Tunnel Testing
C_{D0}	0.065	0.08	Wind Tunnel Testing

9.0 REFERENCES

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