CESSNA / ONR STUDENT DESIGN/BUILD/FLY COMPETITION

An AIAA Student Activity

The 2004 Cessna/ONR Student Design/Build/Fly competition was held at the Cessna East Field facility in Wichita Kansas over the weekend of 23-25 April. Twenty six teams from the United States, Canada, Italy and Turkey attended the fly-off portion of the contest. Of the 26 teams attending the fly-off competition, 21 made at least one successful scoring flight attempt, with many teams completing the maximum allowed 5 flights during the two days of competition. A portion of Saturday was consumed by rain, which seems to be a DBF tradition, but Sunday provided perfect flying weather.

The design objective for this years competition was to create an airplane that would fit in a 2 x 4 x 1 foot shipping container and could complete a fire bomber mission and a ferry mission in the minimum amount of time. Each mission was assigned it’s own Degree of Difficulty multiplier factor. The total score for each team is comprised of their flight performance on their best single flight of each mission type, their score on a written report documenting their aircraft design and selection, and a "Rated Aircraft Cost" representing the complexity and manufacturing costs of their design.

New to the competition this year, the Design Engineering Technical Committee sponsored prizes for the top five teams in the form of copies of their Aerospace Design Engineering Guide handbook. This additional support, above and beyond their participation as one of the administering Technical Committees, is greatly appreciated.

The final results showed a close battle between the two teams from Oklahoma State University, with Team Black the final victor. The University of Southern California team placed third . This is only the second time in the contest history the team with the highest paper score, 94.9, has also won the overall competition (Utah State University also had top paper and won the 2000 competition).

The final positions and scores for all of the competing teams are listed in the table below.

More details on the 2004 competition objectives and rules can be found at the contest web site at http://www.ae.uiuc.edu/aiaadbf .

Position	University	Entry Name	Paper Score	RAC	Bomber Score	Ferry Score	Flight Score	Total Score
1	Oklahoma State University	OSU Black	94.90	6.74	11.73	0.37	12.10	170.46
2	Oklahoma State University	OSU Orange	80.53	6.71	11.66	0.29	11.95	143.49
3	University of Southern California	Scquirt	84.85	8.62	10.07	0.00	10.07	99.13
4	University of Illinois Urbana/Champaign	Fiberglass Overcast	93.65	9.51	9.39	0.34	9.73	95.82
5	Cal Poly San Luis Obispo	Moist	91.50	7.84	7.53	0.44	7.98	93.08
6	La Sapienza Rome	Galiieo V	90.80	9.04	3.87	0.31	4.18	42.02
7	Queen's University	Squirt	86.05	6.57	2.74	0.37	3.11	40.65
8	Virginia Tech	Turbulence Syndrome	88.25	15.40	4.58	0.18	4.76	27.30
9	University of Florida	Swamp Stuka	62.23	12.56	5.20	0.21	5.41	26.83
10	Western Michigan University	Yeager - Bomber	70.00	20.27	5.21	0.22	5.42	18.72
11	Mississippi State University	Phyxius	88.10	9.50	1.50	0.38	1.88	17.41
12	University of California Los Angeles	ENGINuity-IGI	64.00	15.45	1.43	0.30	1.73	7.17
13	Clarkson University	Kamikazes	90.00	17.21	0.98	0.24	1.22	6.38
14	Georgia Institute of Technology	Buzzweiser	92.50	7.21	0.00	0.36	0.36	4.64
15	Cal Poly Pomona	Paani Hawaii Jahaz	76.05	16.26	0.47	0.20	0.66	3.10
16	University of California San Diego	Rain of Terror	91.00	9.74	0.00	0.24	0.24	2.21
17	Washinton State University	Spirit of Procrastination	85.50	8.72	0.00	0.17	0.17	1.71
18	Case Western Reserve University	Flying Nemo	80.70	12.30	0.00	0.22	0.22	1.45
19	West Virginia University	Right Flyer	54.50	13.90	0.00	0.27	0.27	1.05
20	Columbia University	The Leaking Lion	67.00	15.80	0.00	0.24	0.24	1.03
21	University of Central Florida	Poseidon's Fury	87.25	12.55	0.00	0.11	0.11	0.80
22	La Sapienza Rome	Michelangelo	84.45	15.04	0.00	0.00	0.00	0.01
23	Istanbul Technical University	ATA-5	77.68	17.87	0.00	0.00	0.00	0.00
24	University of Texax Arlington	Volantis	63.87	17.50	0.00	0.00	0.00	0.00
25	University of Arizonia	Aircat 2004	68.00	18.99	0.00	0.00	0.00	0.00
26	West Virginia University	Sting Ray	33.50	15.82	0.00	0.00	0.00	0.00
27	University of Texax Austin	121.5	88.50	100.00	0.00	0.00	0.00	0.00
28	City College of New York/CUNY	Rhino	87.00	100.00	0.00	0.00	0.00	0.00
29	Middle East Technical University	Anatolian-Craft	83.00	100.00	0.00	0.00	0.00	0.00
30	University of California Los Angeles	Aquanaut	79.50	100.00	0.00	0.00	0.00	0.00
31	Cal Poly Pomona	Death from Above	78.80	100.00	0.00	0.00	0.00	0.00
32	University at Buffalo	Buffalo Bomber	76.00	100.00	0.00	0.00	0.00	0.00
33	Miami University	Swoop-a-Loop	68.45	100.00	0.00	0.00	0.00	0.00
34	United States Military Academy	Boombatz Grandioso	65.15	100.00	0.00	0.00	0.00	0.00

35	University of Colorado	Water Buffalo	64.75	100.00	0.00	0.00	0.00	0.00
36	University at Buffalo	MAX DRAG	59.00	100.00	0.00	0.00	0.00	0.00
37	West Point	Knight Wings	43.85	100.00	0.00	0.00	0.00	0.00

The success of the competition required the efforts of many individuals. Our first thanks goes to the judges who assisted in the operation, technical inspections and scoring of the flight competition; and to the many judges who evaluated and scored the teams written proposal reports. Thanks also go to the Applied Aerodynamics, Aircraft Design, Design Engineering, and Flight Test Technical committees of the AIAA who organized and manage the competition, and the AIAA Foundation for their administrative support. Thanks is also due to the competitions corporate supporters, the Cessna Aircraft Company and the Office of Naval Research. A special thanks goes to Cessna for hosting this years event and providing access to their facilities for both hanger space and the flying field.

Overall the 2004 Cessna/ONR Student Design/Build/Fly competition marked another very successful event, allowing the participating students to mix a highly enlightening educational experience with a good dose of fun. Congratulations to all the teams who participated for your great enthusiasm and achievement.

See you next year - Greg Page: Contest Administrator

Top Report Score

2003 / 2004 AIAA Foundation Cessna / ONR Student Design-Build-Fly Competition

"Black Stallion"

Oklahoma State Black Team

Design Report

March 9, 2004

Oklahoma State University
School of Mechanical and Aerospace Engineering
Stillwater, Oklahoma

Table of Contents

1.0 Executive Summary

The methodology used to design and construct Oklahoma State University Black Team's entry in the 2003/2004 Cessna/ONR student Design/Build/Fly competition is outlined in this report. Two predefined missions, the Fire Bomber and Ferry, must be completed with an optimized aircraft. An overall contest score is computed from the written report score, total flight score, and the rated aircraft cost of the design.

1.1 Design Development

At the beginning of the conceptual design phase, aircraft approximations and figures of merit (FOM) were developed from the mission requirements of both the Fire Bomber Mission (FBM) and the Ferry Mission (FM). Scrutinizing all aspects of the competition, these FOMs and design approximations were used to determine the most critical mission and most important design parameters which would dictate the design of the plane. Later in the conceptual phase, these critical design parameters were investigated within the propulsion, aerodynamic, and structural areas of the aircraft. Numerous plane configurations were produced from different combinations of these design parameters. A comprehensive computer code was developed to analyze the mission profiles and provide sensitivity studies of the figures of merit used for screening. The sensitivity analysis enabled candidate configurations that exhibited the greatest scoring potential. Scoring potential trends illustrated the importance of the fire bomber mission over the ferry mission and showed that small adjustments in payload had significant affect on the total score. Also, the analysis showed possible optimal regions in battery weight, and wing area.

In the preliminary design phase, more detailed analysis and experimental testing was performed. The computer model was updated to analyze a chosen aircraft configuration's mission profile computing flight times, airspeeds, rated aircraft cost, and total scores. Iterations were done in the program to find the optimal airfoil, wing area, wing span, number of battery cells, and motor/batter/propeller combinations. These tests verified many of the figures of merit found in the conceptual design phase and uncovered other major design parameters such as pit times and take-off distance. Several payload systems involving loading and un-loading water were analyzed and tested to find the best way to create head pressure. Two superior payload designs for un-loading the payload emerged: a tank which pivoted down from the fuselage and a fixed tank with a boom that swung down from the fuselage. A gravity and pump feed systems were investigated and tested. Trade-offs between mission effectiveness and rated aircraft cost where justified.

During the detailed design phase, finite element analysis, flight simulation programs, dynamic stability analysis and physical experiments were run to finalize component selection and systems architecture concluded in the preliminary design phase. All aircraft dimensions, control systems, structure, and propulsion system were among the components finalized in this phase. Weight improvements were made through continuous refinement of the molded construction method. Additional analysis and testing aided in power plant selection and performance prediction. Construction drawings were finalized to ensure proper and timely construction. The final result was a well designed and constructed aircraft, which will score well in the competition.

1.2 Design Alternatives

Many aerodynamic, structural, and propulsion configurations were considered and evaluated. The main empennage configurations considered were the conventional tail, V-tail, and canard configurations. A conventional tail was chosen based on stability and construction concerns. Several different airfoil types where considered: high lift airfoil, low drag airfoil and a balanced airfoil. High lift airfoils where eliminated due to high drag possibilities, while low drag airfoils where eliminated due to takeoff concerns. An airfoil that was a compromise between low drag and high lift was chosen. Non-retractable tail dragger landing gear was selected over other possible landing gear designs. With payload being a liquid, the affect of the shift on the pitch and roll axes are serious concerns. Since a shift in CG in the lateral direction is more controllable than in the longitudinal, fuselage was designed to be as wide as possible so that the reservoir could be as short as possible. Also, possible internal geometries that would allow the payload to only shift forward were carefully thought through and considered for possible final configurations. These reservoir considerations were the driving factors of the design of the fuselage. Initially the fuselage was designed to be a lifting body, but evolved into a tear drop shape in light of the payload system requirements. The onboard payload system using the retractable boom was selected while a pump fed ground based system was implemented. A mold based composite construction process was chosen for all aircraft components due to its favorable strength to weight ratio. Graupner motors were selected by the propulsion team due to their superior efficiency. Gearbox and propeller combinations were narrowed for testing.

1.3 Highlights

In order to reduce the number of design parameters, the two missions where analyzed to show the importance of each. Simple mathematical calculation based on the score equations where computed. The contribution of the ferry mission was insignificant, less than 5%. Therefore, design efforts were directed toward optimizing an aircraft for the fire bomber mission.

The optimization program was modified to model the fire bomber mission profile. Through much iteration key aircraft component sizes were selected. Testing, decision matrices, and other analysis confirmed the optimization program's parameters. The program limited the aspect ratio to 8 and a head wing of 10 mph. These design parameters were chosen for stability concerns and increased scoring potential possibilities, respectively. The optimization program helped optimize the battery selection, battery count, airfoil selection, and wing area. In addition, the program allowed for the mission times, speeds, climb rates, and power usage to be predicted.

A static stability analysis showed the optimal placement of the tail surfaces on the aircraft. Placement of the tail could be relatively close to the wing. This would allow for a one-piece fuselage that could fit in the 4 foot long box. Strength and ease of construction are both achieved with the one-piece fuselage.

An extensive analysis was performed to determine the optimal placement of the center of gravity. The worst case shifting of the reservoir's center of gravity forward and aft was found. This enabled the changes in the trim angle of attack for a given elevator deflection to be determined. An aft shift of the center of gravity has a more adverse effect than a forward shift. The farthest forward the center of gravity can be placed the less change it causes for the trim alpha. This analysis predicted the best theoretical center of gravity position and established a tolerance range. Flight testing will be used as the final criteria for the best center of gravity position.

The pump fed ground based payload system evolved into an integrated backpack that would allow one crew member to load all water onto the aircraft. This backpack used two bilge pumps, one small battery, and two quick connects for the bottles. For the onboard payload system, a conformal tank was constructed and connected to a retractable boom via a piece of flexible tubing that doubled as a pinch valve.

The first step in the manufacturing process consisted of cutting and finishing exact plugs of the aircraft's body. The plugs were used to create re-usable molds. Each plug was cut out of foam sheet insulation, and then covered with a thin layer of fiberglass for surface preparation. After the surface had been filled and sanded to a paint quality surface, the plug was fitted into a molding board, which provided a flat surface along the centerline of the part. One half of the mold was laid-up on top of the parting surface and allowed to cure. The second half of the mold was constructed after the parting board was removed, in order to mate together with the first half of the mold. Once the mold was created, the parts were laid up using a balsa core sandwich with fiberglass panels. The mold construction was time-consuming but it enabled exact replicas to be made any part of the aircraft.

Prototype testing served as a valuable tool towards the design of the aircraft. Most importantly, the prototype allowed the propulsion team to find the absolute optimal configuration, while the structures group was able to reduce structural weight through repetitive testing of wing and fuselage laminates.

2.0 Management Summary

Undergraduates and graduates in Mechanical and Aerospace Engineering comprised the design team. To ensure the delivery of a quality aircraft on time, a comprehensive management plan was implemented.

2.1 Team Architecture

In order to maximize the productivity of our team members and ensure a competitive aircraft, the Black Team was divided into three technical groups: propulsion, structures, and aerodynamics/stability and control. Each of the three groups had a lead engineer and specialists who were responsible for the components of the group. To ensure a good line of communication among the technical groups, a chief engineer headed the team. The chief engineer was responsible for overall team direction, communication among the technical groups, and assurance of the availability of necessary items for aircraft construction. Each group lead engineer was responsible for guiding the group in completion of objectives set forth by the chief engineer and the team as a whole. This hierarchical structure allowed a smooth progression of all the aircraft's design and construction phases, while allowing all aspects of the project to be considered. An organizational breakdown can be seen in Figure 2.1.

Figure 2.1: Organizational Breakdown of Team

2.2 Technical Groups

In the end, the aircraft's final configuration and design was a result of the combined efforts of all three technical groups. Each group had a system to develop, and all the systems had to be integrated. Therefore, each component depended on, or drove the design, of another component. Such a structure allowed all members of the team to input ideas on each of the aircraft's systems and overall design. The responsibilities of each of the three technical groups, as well as individual assignments, can be seen in Figure 1 and are described in detail in the subsequent discussion.

The primary function of this group was to take the aircraft configuration and do all necessary aerodynamic design. Initially, the aerodynamics task was to oversee the optimization of the aircraft's design through the use of mathematical models. In addition, they were also responsible for assuring the aircraft's design fell within the scope of the contest rules. Preliminary sizing of the aircraft was completed by the aerodynamics group by using optimization results to confirm that contest objectives were met in the most efficient manner. The aerodynamics team focused on the stability and control aspects of the aircraft, yielding wing and tail sizes, and the appropriate airfoils for each. Another responsibility of the team was to determine the control and actuator sizing. Finally, a prototype was constructed to determine if the design was adequate. Prototype test data was used by the aerodynamics group to refine the design before the final aircraft was constructed.

Responsibilities of the propulsion team involved finding the optimal propulsion system for the aircraft. This task included designing and testing multiple battery, motor, gearbox, and propeller combinations that produced the predicted required thrust. After becoming familiar with the radio-controlled electric propulsion, an optimization program was used to find a reasonable number of combinations for live testing on a dynamometer. Several combinations of propulsion systems were narrowed down using this ground based testing. The group tested the remaining possible combinations on the prototype aircraft. Prototype testing allowed the team to refine the propulsion system and find the configuration which produced the optimal solution for the team's aircraft. The group also investigated tractor versus pusher propellers and placement of the system on the aircraft.

The structures group was responsible for the aircraft's structural design and construction; this entailed fabrication of all aircraft parts and necessary tooling, in addition to successfully integrating all systems. Members of this group were broken into two subgroups in order to accomplish all the goals. One group was responsible for payload system implementation and fuselage and empennage construction while the other was responsible for land based payload systems, landing gear design and test, and wing construction and test. All members of the structures group were involved in the consideration of construction methods, payload handling, and structural analysis and design. During the aircraft's design phase, this group served as the final authority on the feasibility of structurally implementing any designs brought forth by all team members. In addition to implementing the aircraft and systems, this group was responsible for all final construction drawings. The group used the prototype aircraft for structural, construction, and payload system tests.

2.3 Scheduling

The complete design and construction of the aircraft took place over a 3½ month time span. This schedule was accelerated and could not be accomplished without adequate planning and time management. A Gantt chart of the process was built based upon milestones set forth at the beginning of the project. Each phase was dependent upon another phase of the process; therefore the chart had to be explicitly followed to ensure successful completion of all necessary tasks. However, the design was adaptable until the final design freeze. For simplicity, the design report was developed concurrently with each phase of the design process. Figure 2.2 presents the design process to be followed by the team.

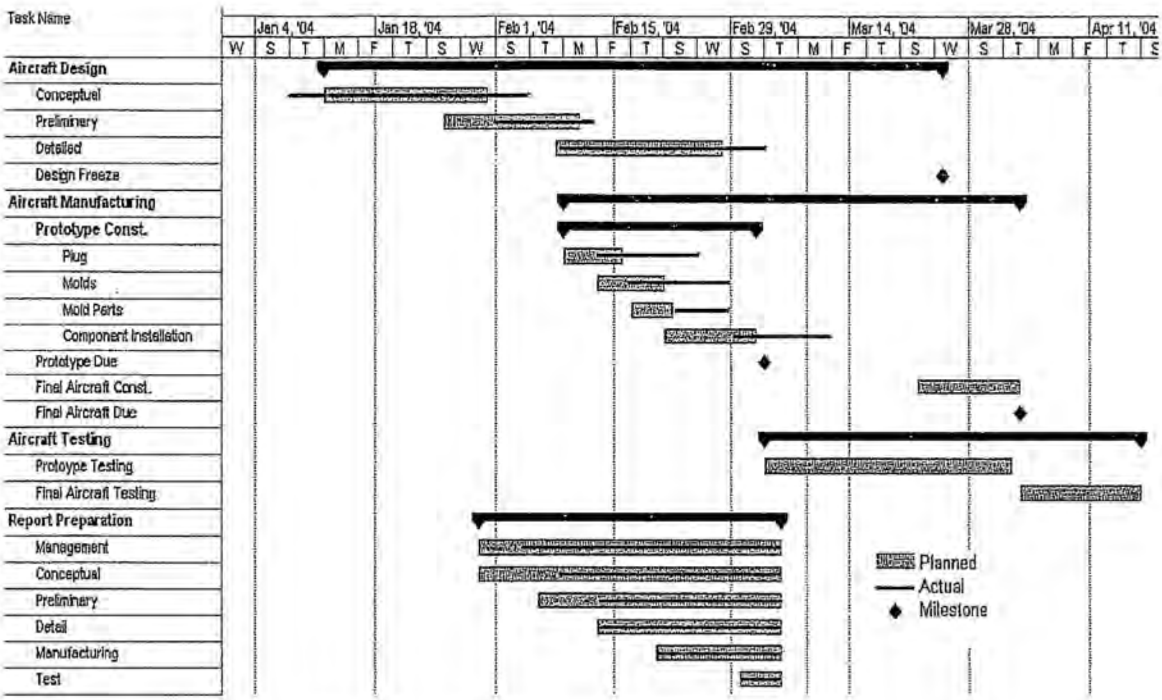


Figure 2.2: Milestone Chart of Design Process

3.0 Conceptual Design

During the conceptual design phase, the team aimed at choosing an aircraft configuration to optimize for performing each mission through an explicit Figure of Merit (FOM) screening process. Multiple tail, gear, and payload configurations were narrowed using the same process.

3.1 Mission Requirements

The purpose of this contest was to design and construct an aircraft that would complete all missions in an efficient manner. The fire bomber mission (FBM) requires an aircraft capable of containing and expelling a payload of up to four liters of water, while the ferry mission (FM) requires the same aircraft to fly the same mission profile without payload. The final aircraft design will complete each mission efficiently while having an optimal rated aircraft cost.

The total flight score for this contest was determined using a weighted function that combined the scores from both missions. It was crucial to determine the importance of each mission during the conceptual design phase. The FBM flight score is a function of 2 times the water weight divided by the total flight time, while the FM flight score is a function of the inverse of the flight time. An initial analysis determined the percentage of each mission's effect on the overall score. The purpose of this analysis was to validate the assumption that the aircraft should be optimized for the fire bomber mission.

Upon inspection of the two mission scores, it can be seen that the FM score will be a small decimal and the FBM score will be a much larger decimal. The total flight score is the summation of the FBM and FM scores. With FBM score being a much larger number than the FM score, it will almost totally control the total flight score. A hypothetical situation was investigated to ensure that this train of thought is correct.

The goal of the design of the aircraft is to maximize the contest score function:

$$\text{Flight Score} = \left(\frac{2 * \text{Payload Weight}}{\text{FBM Time}} + \frac{1}{\text{FM Time}} \right) * \frac{1}{RAC}$$

- Fire Bomber Mission: This mission requires the aircraft to begin empty. As soon as the payload is loaded, the aircraft can takeoff and after which fly 500 feet before making a 180 degree turn. On the 1000 foot long down wind leg the plane is to expel the payload and perform a 360 degree turn. At the end of the down wind leg the plane will make another 180 turn then land the plane and repeat the process again.
- Ferry Mission: The ferry mission consists of four consecutive laps which include the 360 degree circle with an empty aircraft.

In addition to completing each mission, the aircraft must takeoff in a distance of 150 feet and fit in a 4 by 2 by 1 foot box. The design must also aim at reducing Rated Aircraft Cost (RAC), which is used in calculating the final contest score. RAC is a function of aircraft weight, propulsive power, and manufacturing man hours.

The total flight score for this contest was determined using a weighted function that combined the scores from both missions. It was crucial to determine the importance of each mission during the conceptual design phase. The fire bomber flight score is a function of 2 times the water weight divided by the total flight time, while the ferry flight score is a function of the inverse of the flight time. An initial analysis determined the percentage of each mission's effect on the overall score. The purpose of this analysis was to validate the assumption that the aircraft should be optimized for the fire bomber mission.

Upon inspection of the two mission scores, it can be seen that the FM flight score will be a small decimal and the FBM will be a much larger decimal. The total flight score equals the addition of the FBM flight score and the FM score. With FBM flight be a much larger number than the FM flight, it will almost completely contribute to the total flight score. A hypothetical situation was investigated to ensure that this train of thought is correct.

Once it was determined to design for the fire bomber mission, the mission analysis process took several initial concepts under consideration. Time spent on the ground during the fire bomber mission is a crucial factor in producing an optimal total score. Therefore, delivering the water as fast and efficiently as possible was vital to producing a low mission time and ultimately a high total score.

Structural design of the aircraft had to consider wing loading, durability, and integrity of all components, while the aircraft's aerodynamic design had to produce an aircraft capable of flying in all wind conditions. In addition, the wing had to be large enough to allow the aircraft to take off in the prescribed distance.

3.2 Alternative Configurations

Alternative configurations were formulated for the aircraft type, empennage, payload systems, landing gear, and propulsion system. Configuration possibilities and FOMs were analyzed for each section.

3.2.1 Aircraft Type

When comparing multiple alternatives several assumptions were made: the aircraft could operate with one motor, each alternative would have similar wing lifting efficiency and fuselage length, the maximum allowable payload could be carried, and each alternative would perform at the same design wind speed. More detailed analysis would be necessary if one configuration would not have appeared superior based on the FOMs.

- Conventional: A conventional configuration was used as a baseline for comparing the configurations. The performance characteristics would be easily predicted with ample historical data available.
- Flying Wing: A pure tailless flying wing offered lower RAC due to its lack of a tail and a small fuselage. In addition, a flying wing would offer limited structural weight and drag. However, it had poor handling qualities and would require sophisticated augmentation to perform the optimum mission profile.
- Blended-Wing-Body: The blended-wing-body would have the same handling qualities as a conventional configuration, but less drag due to blended intersections and a more streamlined shape. It could also have a higher RAC due to more fuselage volume.
- Canard: A canard design would allow for the horizontal control surface to not detract from the overall lift of the aircraft. This configuration would have good stall characteristics, but be limited during takeoff. Flexible motor setups would easily be implemented.
- Bi-plane: A bi-plane configuration would be able to produce a large amount of lift with smaller wings, however, RAC penalty is very high for multiple wings. A bi-plane would be very similar to a conventional design with respect to flight characteristics.

The FOMs used to screen the different configurations are listed below and then used in Table 3.1 to rank each configuration.

- Takeoff Distance: Aircraft must lift off in 150 ft or it will not receive a flight score.
- Handling Qualities: An aircraft that is difficult to handle during all legs of the mission will have a low scoring potential. A design with the ability to land in a crosswind, handle center of gravity

movements, and be easily trimmed will be able to fly the optimum profile and have the best possible score.

- Drag efficiencies: Configurations should be designed to be fast in order to maximize scoring potential through a reduced mission time. Drag should be minimized in designs to make more efficient use of available power and increase speed potential.
- Rated Aircraft Cost: RAC directly affects the contest score. It is very clear that a lower RAC will produce a higher contest score.

Figure of Merit	Weighting Factor	 Conventional	 Flying Wing	 Blended-Wing-Body	 Canard	 Bi-plane
RAC	0.33	2	3	1	2	1
Takeoff Distance	0.27	2	2	2	1	3
Handling Qualities	0.25	3	1	3	2	2
Drag Efficiencies	0.15	2	3	3	2	1
Total	1	2.25	2.23	2.07	1.73	1.79

Table 3.1: Overall Configuration Weighted Decision Matrix

The concept with the best scoring potential is the conventional configuration. The flying wing configuration is a close second, but the aircraft handling qualities make this configuration less desirable. The blended-wing-body's handling qualities and takeoff distances were as good as the conventional, but its RAC score was low due to the sweeping wing and larger fuselage requirements. The conventional design will minimize the fuselage penalty and be as streamlined and as possible. Such a configuration favors a mid-mounted wing. A mid-mounted wing would allow for increased drag efficiencies and moderate stability while allowing for the best fuselage blend. A mid-mounted wing does not interfere with payload ejection as a low-mounted wing would.

3.2.2 Empennage

Empennage alternatives were narrowed down to four possibilities: a conventional tail, a T-tail, a V-tail, and a cruciform tail.

- Conventional: A conventional configuration provided reasonable stability characteristics while remaining relatively lightweight. The horizontal stabilizer could also be integrated into fuselage easily.
- T-Tail: This tail is heavier than a conventional tail due to the reinforcement necessary to support the horizontal stabilizer. However, it allowed for a smaller horizontal stabilizer since it extended out of the wing wake and propeller wash.

- Cruciform: The cruciform configuration offered a compromise between a T-tail and a conventional tail. It avoided the full weight penalty of the T-tail, while still lifting the horizontal tail and improving its effectiveness.
- V-Tail: A V-tail offered lower RAC penalty and reduced interference drag; however, it introduced stability problems through the adverse roll-yaw coupling of the ruddervators in addition to construction complications.

The FOMs used to screen the different configuration are listed below. The FOMs are then used in Table 3.2 to rank each configuration.

- Weight: The weight of each tail design is counted heavily in the final RAC. To reduce RAC and produce an optimal score the weight must be minimized.
- Construction: The ease of construction for each tail configuration is important to reduce time spent building the aircraft and thus increase the testing time of the aircraft.
- Drag efficiencies: Configurations should be designed to be fast in order to maximize scoring potential through a reduced mission time. Drag should be minimized in designs to make more efficient use of available power and increase speed potential.
- Rated Aircraft Cost: RAC directly affects the contest score. It is very clear that a lower RAC will produce a higher contest score.

					
Figures of Merit	**Weighting Factor**	**Conventional**	**T-Tail**	**Cruciform**	**V-Tail**
RAC	0.3	2	2	2	3
Weight	0.2	2	2	3	2
Construction	0.2	3	2	2	2
Drag Efficiencies	0.3	3	2	2	2
Total	1	2.5	2	2.2	2.3

Table 3.2: Tail Configurations Weighted Decision Matrix

Table 3.2 shows the conventional tail performing the best, although a V-tail scored close due to the lower RAC. However, construction complications and increased weight far outweigh the RAC advantage. Therefore, a conventional tail was chosen.

RAC estimates were generated for four aircraft. A conventional aircraft with a conventional and V-tail and a blended-wing-body with a conventional and V-tail. The RAC values were found to be 6.36, 6.31, 6.42, and 6.47, respectively. Although a conventional aircraft with a V-tail has a lower RAC than one with a conventional tail, the disadvantages brought forth by the V-tail far outweigh the RAC advantage.

3.2.3 Payload

An important aspect of the aircraft design were the payload systems, which involved payload containment, evacuation, and loading. Improper water containment could be catastrophic to the aircraft's stability. In order to maximize flight time the plane must not have to slow down during the downwind leg of the FBM. A payload evacuation system must be designed that empties the plane tank as quickly as possible. Another major factor in flight time is pit time, or time spent loading the water onto the plane. Water loading systems must be selected that decrease pit times. On the following pages, multiple possibilities for each system are investigated and FOMs are used to select the superior designs.

Payload containment systems within the aircraft investigated are discussed below.

- Tubing System: Run 24 feet of 1 inch diameter tubing throughout the aircraft. This would drastically reduce dynamic shift of the water. However, the system would be susceptible to static shift in addition to being very complex and heavy.
- Bladder: A bladder would be used to control the payload. Such a device could keep the payload from shifting dynamically or statically. However, the rules prohibit an elastic bladder and a non-elastic bladder wouldn't do a sufficient job of keeping the water from shifting.
- Multiple Reservoirs: Multiple reservoirs with one way valves would control the payload in dynamic and static shift. The reservoirs would be complex and heavy.
- Single Tank: A single tank would be lightweight and simple. Internal baffling could be employed to aide in water containment. This design could be easily adaptable to achieve a high flow rate.

When developing the weighted decision matrix for selecting a final on-board payload concept, the following FOMs were considered. Table 3.3 presents the on-board payload decision matrix.

- CG Placement: The coincidence or proximity of the payload and aircraft centers of gravity directly affects the stability and controllability of the aircraft, and is therefore greatly important when considering alternatives.
- Static Shifting Control: The slow lateral and longitudinal movement of the payload was considered in each of the alternatives. This is important because such a shift would cause unpredictable movement of the payload center of gravity, and cause the aircraft to become unstable and uncontrollable.
- Dynamic Shifting Control: The rapid movement or "sloshing" of payload must be considered because this produces erratic movements, possibly sending the aircraft into a divergent oscillation.
- Reliability and Durability: The system must perform each time it is called upon, because not performing would result in flight score penalties ranging from three to seven minutes. The system must also be durable because of the amount of required flight testing.
- Payload Retention: All water must be expelled from the aircraft. The inability to accomplish this would result in a time penalty and a smaller volume of water being able to be carried on the second sortie of the mission.

Figures of Merit	Weighting Factor	Tubing	Bladder	Multi. Res.	Single
CG	0.25	2	2	3	3
Static shift	0.2	3	3	2	3
Dynamic shift	0.2	3	3	3	2
Reliability	0.2	1	1	2	3
Payload Retention	0.15	1	1	1	3
Total	1	2.05	2.05	2.3	2.8

Table 3.3: Payload Containment Concepts Decision Matrix

Using Table 3.3, the concept that emerged as the most dominant of the alternatives is the single reservoir concept. A single reservoir allows for better placement and control of the payload center of gravity, which ideally coincides with the empty aircraft's CG. Control of the dynamic payload shift is achieved with a baffling system inside of the tank. This baffling system will be further investigated during preliminary design. A tank without proper baffling could cause catastrophic effects to the plane in flight due to dynamic or static shifting.

Payload evacuation systems investigated to expel water from the aircraft are discussed below.

- Nozzle: A nozzle could be used to provide the optimum exit flow by capitalizing on the low pressure region created at the aft section of the aircraft. One difficulty with this concept was that the flow would become turbulent as soon as the water was released, possibly reducing the effectiveness of the nozzle design.
- Pivoting Tank: The entire tank containing the payload would pivot. Such a tank would allow for a smaller fuselage and sufficient head pressure would be gained. However, the rotational inertia produced by the swinging tank could have adverse effects.
- Boom: A boom would retract from the bottom of the aircraft. The valve and boom actuation could be done with a single servo. Since the boom is empty until it is in place, inertial effects would be minimal.

When developing the weighted decision matrix for selecting a payload expulsion system for the payload, the following FOMs were considered. Table 3.4 summarizes the decision.

- Payload Evacuation Rate: Payload can only be expelled on the downwind leg. If the aircraft must slow down for this, the mission time will be increased, resulting in a lower score.
- Inertia effects: If the payload shifts, its momentum could easily overcome the plane, causing the plane to diverge. With the payload weight being more than the projected plane weight, a small but sudden shift would likely cause a catastrophic failure.

- CG changes: The static margin must not be pushed to the point where the airplane becomes neutrally stable or the aircraft's behavior cannot be predicted. In order to prevent this, the payload evacuation cannot force the CG aft more than three quarters of an inch.
- Drag: The drag created by the water evacuation must be minimal, in order to keep the plane from becoming unstable during the evacuation. A large drag force created by an object in the flow field or spraying water into the stream in an adverse way might cause a moment which would force the plane into a pitch.

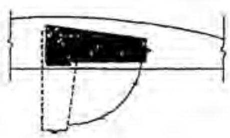
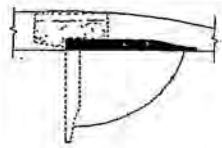
				
Figure of Merit	**Weighting Factor**	**Nozzle**	**Pivoting Tank**	**Boom**
Evacuation rate	0.35	1	2	3
Inertia effects	0.3	3	1	2
CG changes	0.2	3	1	3
Drag	0.15	3	1	2
Total	1	2.3	1.35	2.55

Table 3.4: Expulsion Payload Concept Decision Matrix

From Table 3.4, the optimal configuration is chosen to be a boom. The boom minimizes the center of gravity shift, while eliminating the adverse inertial effects inherent in the pivoting tank concept. Using the boom also provides the most head pressure, resulting in the fastest evacuation times, and reducing the time for which static payload shifting is an issue.

Water Loading significantly affects the overall flight score through the "pit" time that it takes to load and reload the payload onto the aircraft. The following ground based payload concepts were generated to minimize water loading time. All concepts are presented in Figure 3.1 and described below.

- Gravity Feed System: The first consideration for loading the payload onto the plane was a gravity feed. An initial belief was that roughly a six-foot-tall device could hold the two bottles of water and allow gravity to accelerate the loading process. This is a relatively simple device that would provide ease of construction while remaining cost effective.
- Single Pump System: This concept consists of a single pump for both bottles. A manifold would be used to connect tubes from each bottle to the pump intake. The total flow would be discharged through one exit which allows for the use of only one loading orifice on the fuselage.

- Dual Pump System: This system would assign a pump to each bottle, thus doubling the flow rate. Utilizing two loading ports, one for each pump, the time to load the water could ideally be reduced by half. Although the addition of another pump would be costly, the benefits of saving time in the pits would drastically improve the flight score.

FOM's used to screen the alternatives are discussed below and used to complete the decision matrix shown in Table 3.5.

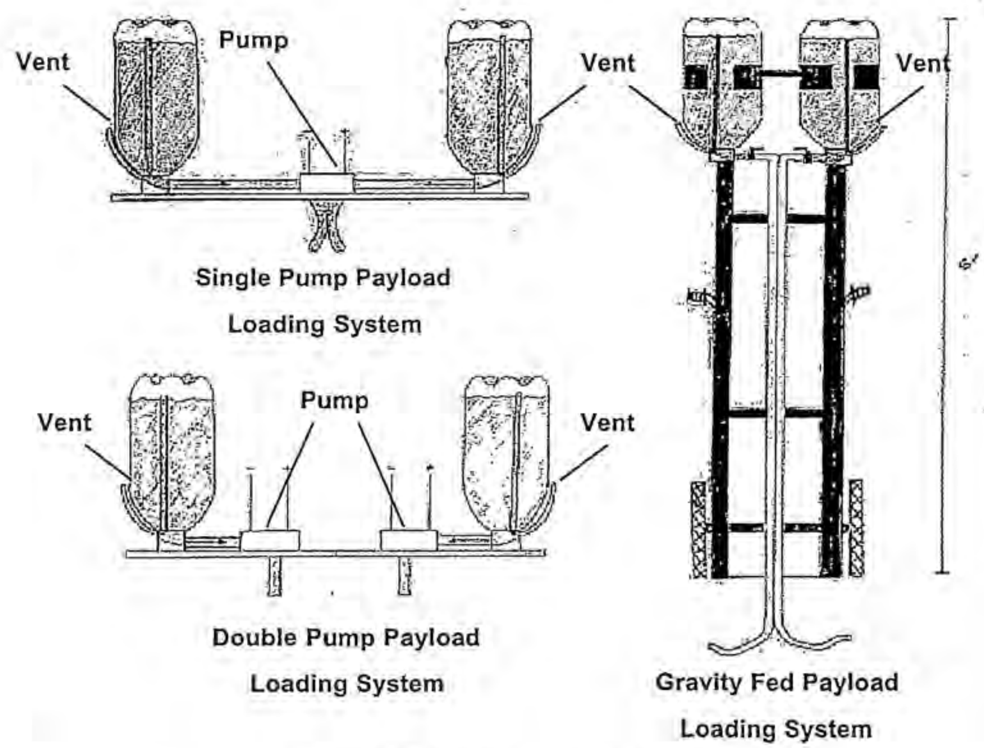


Figure 3.1: Water Loading Concepts

- Expulsion Rate: The flight score is based upon the total mission time, including the time required to load and reload the payload onto the aircraft. Therefore it is imperative to have the quickest possible loading rates in order to increase the flight score.
- Reliability: A system failure will result in a flight score of zero for that attempt. Mild failures causing excessive spillage will also result in time penalties and reduced flight scores.
- Mobility: The ground crew for loading and reloading the aircraft may consist of only three persons, and they must be able to move the apparatus from the staging area to the aircraft and load it as quickly as possible in order to maximize the flight score.
- Simplicity and Ease of Operation: The system must be straightforward and easy to operate in order to reduce preparation and loading times for each sortie.

Figure of Merit	Weighting Factor	Gravity Feed	Single Pump	Dual Pump
Expulsion Rate	0.35	1	2	3
Reliability	0.3	3	2	2
Mobility	0.2	1	2	2
Simplicity	0.15	3	2	1
Total	1	1.9	2	2.2

Table 3.5: Water Loading Concept Decision Matrix

Because the loading and reloading of payload have such a profound effect on the total flight score, it becomes necessary to minimize the time required to do such. The dual pump system is slightly more costly and cumbersome, but because the flow rates are doubled and the actual time required to load the water is halved, the benefits prevail over the drawbacks without question. Utilizing two self-contained pumps will undoubtedly increase the flight score and ultimately the overall team score.

3.2.4 Landing Gear

Many configurations of aircraft landing gear exist, the most feasible options are described below.

- Tricycle gear with bow-type main gear: Such a setup provides adequate ground stability. Using a bow would transfer all of the landing loadings to one point in the fuselage, thus only requiring minimal reinforcement.
- Tricycle gear with independent main gear: Independent struts must be built very strong and aircraft integration must be reinforced greatly, but more freedom is given in gear placement.
- Tail Dragger with Bow-type Main Gear: Tail draggers make use of mains and a small tail wheel. The lack of a large nose wheel will reduce drag and weight. A bow for the mains would give the same advantages mentioned above.
- Tail Dragger with Independent Main Gear: This setup has all characteristics as listed in the tail dragger configuration listed above. All problems associated with independent mains are still present.

The most compatible landing gear alternative was selected using the following FOMs to compare the designs:

- Landing performance: Landing gear would be rendered useless if the plane could not land properly. The gear must allow the aircraft to be stable on the ground shortly after touchdown.
- Weight: Minimum weight is desired in order to reduce Rated Aircraft Cost and maximize aircraft performance.
- Pilot preference: The pilot felt more comfortable landing and taking off if the landing gear was designed similar to what they were accustomed to using.

- Stability: The landing gear must be capable of creating a stable support for the aircraft while it is stationary or moving on the ground. Without stability, the aircraft could topple forward and damage, if not, destroy the propeller.

Figure of Merit	Weighting Factor	Tricycle		Tail Dragger	
		Bow	Independent	Bow	Independent
Landing Performance	0.35	3	3	3	3
Weight	0.3	2	1	3	2
Pilot Preference	0.2	2	2	3	3
Stability	0.15	3	3	1	1
Total	1	2.5	2.2	2.7	2.4

Table 3.6: Landing Gear Evaluation

Table 3.6 indicates a tail dragger with bow type mains is the most viable alternative. Although this design presents some stability issues, the weight advantage and pilot preference make it the most attractive.

3.2.5 Propulsion System

Propulsion system configurations consisting of various motor and battery combinations were investigated. Each configuration is discussed below.

- Double Stack: A Double Stack configuration will provide the most amount of power. The weight of the entire system will be higher due to the multiple motors and battery packs. This extra weight significantly counts against the RAC.
- Double Stack/Shared Pack: The Double Stack/ Shared Pack configuration utilizes a single battery pack to power two motors, thus providing power similar to the Double Stack configuration while saving weight and battery cells.
- Single Stack: The Single Stack configuration minimizes RAC because it uses only one motor and one battery. This system must be properly designed to provide the required power for the aircraft to fly the mission profiles.
- Single Stack/Two Packs: This concept is identical to the Single Stack except that the battery pack is broken up into two smaller packs. Smaller packs will provide for more battery placement options.

These alternatives were compared against the FOMs below to determine the best choice for the battery/motor configuration. Results of this comparison are presented in Table 3.7.

- RAC: The propulsion system has a large effect on RAC, therefore it must be minimized.

- Power Produced: The power produced directly effects aircraft performance. Without ample power, the aircraft will fail in its ultimate mission.
- Weight: Weight is important because total propulsion system and battery weight are heavily penalized in the RAC.
- Efficiency: The ability to produce required performance with the lowest number of batteries for both aircraft performance and RAC.

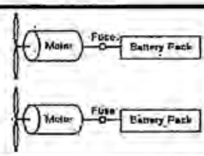
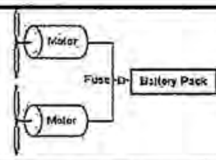
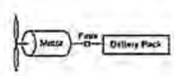
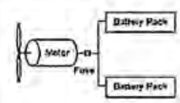
					
Figures of Merit	**Weighting Factor**	**Double Stack**	**Double/Shared Pack**	**Single Stack**	**Single/Two Packs**
RAC	0.35	1	2	3	3
Power Produced	0.3	3	3	1	2
Weight	0.25	1	1	3	2
Efficiency	0.1	1	2	1	2
Total	1	1.6	2.05	2.2	2.35

Table 3.7: Battery/Motor Configuration Decision Matrix

Based on Table 3.7, the single stack/two pack configuration is the best choice for this competition. RAC will be minimized as this is one of the lightest configurations. Using two smaller battery packs will allow the team to place the batteries in the fuselage more easily without sacrificing power. One motor should provide the plane with sufficient power to complete the mission in the desired time.

3.2.6 Structural Concepts

Different structural methods were examined and evaluated.

Fuselage Structure

- Keelson: Building up the fuselage off of a main keelson has potential to produce an overall strong and stiff aircraft, but does not allow for the most desirable shapes and curves for aerodynamic concerns.
- Stringers: A stringer based fuselage allows for more aerodynamic shaping by blending stiff, shaped stringers around a series of bulkheads, creating a more aerodynamic fuselage with smoother cross section transitions.
- Monocoque: A monocoque, or reinforced skin fuselage can produce strong and stiff aircraft through the use of stiff and lightweight composite sandwich structures. The most common way to carry out this technique is through the building of molds, allowing the exact outer shape to be controlled by the mold shape.

The FOMs were selected to evaluate the performance of each method. Evaluation results are presented in Table 3.8.

- Strength/Weight Ratio: The emphasis on weight in the RAC calculations dictates that any structural design chosen must be lightweight, in order to maximize scoring potential.
- Reproducibility: In the event that an aircraft malfunctions or is damaged, it will be necessary to either rebuild or repair, so each method is evaluated on how easily the plane can be fixed or be reproduced.
- Connection Interface: In order to minimize losses, methods were evaluated on their junction characteristics because of the unnecessary drag that can be produced at intersections.
- Dimensionality/Surface Finish: The plane must be as streamlined as possible in order to eliminate drag, so a FOM was chosen to grade each method on surface smoothness and shaping capabilities of each method.
- Construction Ease: An investigation was conducted into the difficulty of each method, and a decision was made based on the audited team skills.

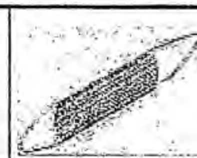
				
Figures of Merit	**Weighting Factor**	**Keelson**	**Stringers**	**Monocoque**
Strength/Weight Ratio	0.25	2	2	3
Reproducibility	0.25	1	1	3
Connection Interface	0.15	1	2	3
Dimensionality/Surface	0.2	1	2	3
Construction Ease	0.15	2	2	1
Total	1	1.4	1.75	2.7

Table 3.8: Fuselage Construction Evaluation

From Table 3.8., the best choice to provide the desired high strength, light-weight, and streamlined aircraft fuselage was the monocoque skin method. This method has the potential for substantial savings in final aircraft weight, which will reduced the required power, shrink the wing size, and result in a higher performance score and higher total scores.

Wing/Empennage Structure

- Foam Core: The first method investigated was foam core composite construction. A CNC foam cutting machine was available, so cutting precise pieces was possible. With proper composite material considerations, sufficient strength for the mission would be easily possible to achieve. The major drawback to this method is the weight of the internal foam core. It is also fairly easy to create replacement parts, with the exception of the foam core surface preparation process.

- Built-Up Construction: Built up construction consists of building a balsa frame and applying Monokote covering to it in order to create the aircraft skin. It was considered mainly because of the advantage that it is extremely lightweight, and fairly simple to build. Another positive aspect of built up construction is that many times, minor damage to the aircraft can be repaired locally. The method is fairly precise due to the use of part templates, which allow for some precision in making the sections of the plane, but in the event of severe failure, a broken part nearly always has to begin from scratch.
- Monocoque: For monocoque construction, the external skin is used as most or all of the structure. Internal structure is generally intended for maintaining shape and providing additional stiffness. This design is particularly desirable for wing structure due to the nature of the tension/compression skin pairs formed by the wing lower and upper skins.

The following is a descriptive list of the FOMs used to evaluate each structural technique for wing/empennage construction. The resulting decision is presented Table 3.9.

- Strength/Weight Ratio: The wing is likely the largest structure on the aircraft, requiring it to be as lightweight as possible and still carry the required loads.
- Bending Strength: The lift force on the wings will impart a bending moment, meaning the most likely failure mode of the wings will be in buckling.
- Reproducibility: Wings are likely to be damaged in the event of a crash during flight testing, so the ease of repair to a damaged wing is evaluated, as well as the ease of reproducing a new wing.

				
Figures of Merit	**Weighting Factor**	**Foam Core**	**Built Up**	**Monocoque**
Strength/Weight	0.4	1	2	3
Bending Strength	0.3	2	2	3
Reproducibility	0.3	1	1	3
Total	1	1.3	1.7	3

Table 3.9: Wing/Empennage Construction Decision Matrix

The monocoque method of providing structure within the skins is again chosen for the wing and empennage surfaces. The combination of a superior strength/weight ratio and the relatively easy reproduction of monocoque structure were the deciding factor in choosing it as the best alternative.

Payload System Structure

The payload system construction method was chosen to be a fiberglass lay-up, with some reinforcement to aid in carrying the spar loading. A monocoque type container will be constructed, so that it will not need internal structure to carry the water, with some stringers internal to the skin for additional stiffness. The monocoque construction required a foam plug with exact dimension be created.

3.3 Conclusions

Many combinations of aircraft type, empennage, payload systems, landing gear, propulsion systems, structural configurations to choose from, the decision matrices, the RAC calculations, and sensitivity studies narrowed down the possibilities to one final conceptual design. The design proving to be the best was the conventional fuselage with a conventional tail, shown in Figure 3.2. This concept showed the most promise for handling qualities and mission performance without compromising RAC. The on-board payload system that best suited this design was the single reservoir with baffling. A boom attachment on the tank would add the necessary head pressure to expel all of the water during the downwind leg. The superior propulsion system was a single motor powered by a split battery pack. The entire design was to be built with a monocoque design, which minimized weight while providing the necessary strength. Tail dragger landing gear with bow-type main gear was chosen and fits well on the aircraft's design.

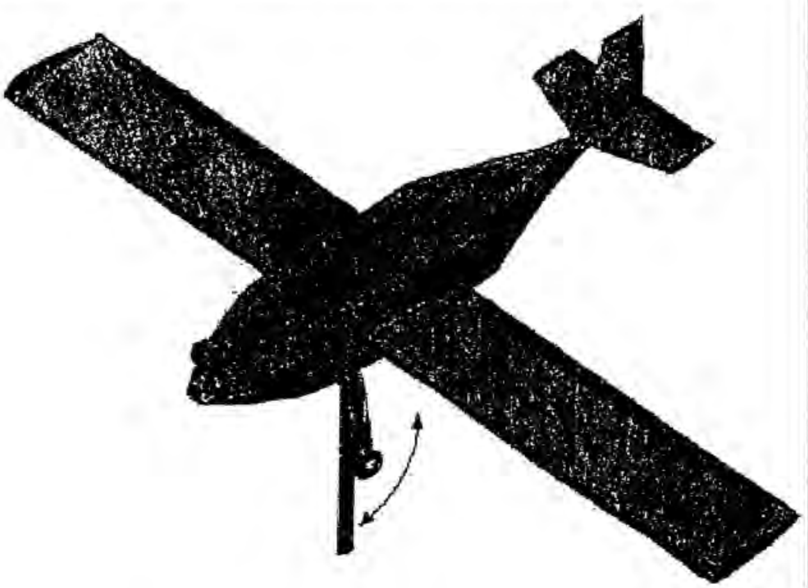


Figure 3.2: Final Conceptual Configuration

4.0 Preliminary Design

The chosen configuration from the conceptual design phase was separated into three groups: aerodynamic, structural, and propulsion groups. Critical design parameters were selected and studied within each group. FOMs were used to find appropriate sizes for many of the design parameters. The mission model program from the conceptual design phase was modeled more accurately and a propulsion performance program was created. These programs optimized the most important design parameters, while the remainder of the design parameters where subsequently analyzed and sized.

4.1 Design Parameters and Sizing Trades Summary

Critical design parameters were selected within each design group. The aerodynamic group investigated wing area, airfoil, wingspan, and fuselage and empennage size, while the structures group investigated payload amount and boom length. The propulsion group investigated motor selection, battery selection and number of cells, propeller pitch and size, and takeoff and cruise power. Many other design parameters existed, but these have the largest effect on RAC, total score, and performance.

- Wing Area: Wing area is crucial for take-off with a short runway. Data found during conceptual design showed the conceptual configuration lifting off at 150 feet with the smallest possible wing area. This trend increases the scoring potential but causes major concerns during take-off. High wing loading allows for faster cruise velocities but longer take-off distances.
- Wingspan: Wingspan has a major effect on wing efficiency and RAC. RAC is minimized for given wing areas as the aspect ratio is lowered, but high aspect ratio wings become more efficient. Also, RAC is minimized with a rectangular wing making elliptical and tapered wings highly penalized. Therefore an RAC/efficiency tradeoff must be made. Construction, fit-in-box, and the ability to pass the wing tip loading test were other considerations.
- Airfoil: Airfoil selection is important because of its direct affect on take-off and cruise. Three low speed airfoils were chosen based on historical data for further analysis: a high lift airfoil, a low drag airfoil, and a more balanced airfoil. The airfoils were evaluated to test the affect each had on the overall score. The high lift airfoil performed well during takeoff due to its high lift coefficient, but its large drag possibilities during cruise was a concern, while the low drag airfoil performed well during cruise due to its low drag coefficients, but its low lift coefficients were a concern during takeoff. The balanced airfoil performed well in both cruise and takeoff situations. More detailed mission models were needed to find the optimal airfoil.
- Fuselage length & Empennage Size: It was desired for the entire plane length to be less than 4 feet long so that it could fit in the box. A one piece fuselage and empennage would benefit the structural integrity and weight of the plane. The empennage size must be sufficient to stabilize the aircraft. As overall aircraft length increases the RAC increases. Drag decreases as the fuselage length increases and empennage size decreases. A compromise must be made to minimize both RAC and drag.

- Battery Selection and Number of Cells: Battery weight has the most effect on RAC. The capacity of the batteries had to be sufficient to perform the mission profile while minimizing the amount of unspent energy. A lower capacity battery would require more cells, thus increasing voltage and RPM. A high capacity battery would complete the mission using fewer cells, thus decreasing voltage and RPM. Minimizing the number of batteries would lower the weight of the system, thus decreasing the RAC.
- Propeller Pitch and Size: Propeller pitch and size impacts the amount of thrust produced a propeller with a high pitch to diameter ratio would be more efficient at higher airspeeds than a low pitch to diameter ratio propeller. The propeller selection had to be based on a trade-off between takeoff and flight performance.
- Takeoff & Cruise Power: Takeoff and cruise power must be optimized to increase the scoring potential of the aircraft. The power generated at takeoff would account for the majority of the available energy from the propulsion system. The remaining amount of energy would be consumed during cruise. The thrust at take-off must not create a current over 40 amps or use too much energy from the batteries causing an insufficient amount of power for cruise. Gear ratios and propeller sizes may be changes to better suit take-off situations or cruise situations.

4.2 Refined Mission Modeling and Optimization Analysis

A mission profile analysis program and propulsion system analysis program were created to optimization the most important design parameters.

4.2.1 Mission Profile Optimization Analysis Program

A mission analysis performance program was written to assist in analyzing each phase of the mission profile to determine the overall result. The program analyzed the entire FBM as best as possible utilizing techniques found in Nelson (1998). The program consists of mathematical models for aerodynamic characteristics, propulsive efficiency, and the weight of a proposed aircraft. The program was written such that various aircraft configurations could be compared quickly and efficiently utilizing aerodynamic modeling characteristics, such as wind speed, taper ratio, sweep angle, and flaps. Once the flight characteristics of the candidate configurations were determined, the performance characteristics for each phase were calculated:

- Time and power required to takeoff within the 150 foot limit.
- Time and power required to climb at a satisfactory rate.
- Time, distance, and g-loading during turning were calculated from weight and stall characteristics.
- Time and distance required during payload evacuation.
- Time and power required during cruise velocities.
- Time and distance required to slow down from cruise velocity to stopping.

The individual components of the mission profile for each configuration were then combined and compared. The overall values from each configuration measured were the times, distances, and energy consumed. The RAC was then calculated and the final flight score was determined from the equation provided by the contest rules.

Certain limitations were placed upon the modeling method used to reduce the number of possibilities. The most important constraint was the aspect ratio. Based on historical data from previous Oklahoma State University configurations a limit of 8 was implemented. A limit of 8 was chosen because, historically, aircraft with an AR higher than 8 were found more difficult to control. Another assumption employed was that the payload would be completely released after completing the 360° turn. Therefore the aircraft was assumed to be carrying the full payload before 360° turn, no payload after the turn, and partial payload while performing the turn. Therefore, it is crucial that the final configuration is able to evacuate all payload during the downwind leg of the mission. The final limitation implemented was a battery usage of 100%. This constraint was utilized to determine the fewest number of battery cells that would produce a useful final score. These limitations kept the predicted performance models within reason.

- Weight model: The weight model was reconstructed to better represent a plane built using monocoque construction. This model was based on historical data and varied structural weight as a function of wing area and estimated structural weight for the fuselage and empennage. The model was made slightly conservative to provide a margin of safety and checked against historical data.
- Drag model: The drag model was reconstructed for the conceptual design according to the drag build-up method found in Raymer (1999). The method gave a close estimate of drag without having exact dimensions. The drag model also included the drag polar for the chosen airfoil and estimated parasite drag for the plane. Proper Reynolds numbers were used for both cruise and take off.
- Propulsion Model: The propulsion model was refined using historical data from a single motor configuration. The new model was reconstructed to remain conservative, yet represent the propulsive system better than the conceptual model.
- Rated Aircraft Cost Model: RAC was updated to better represent the features chosen in the conceptual design.

These models were placed in the program along with the parameters the sensitivity studies showed as optimal. The values included a wind speed of 10 mph and an altitude of 50 feet. The program was adapted so that the aircraft was optimized for overall score. Figure 4.1 below shows the method used for determining optimal configurations within the program.

Sensitivity Studies

Sensitivity studies were done on the critical variables in the mission profile program. These studies analyzed each variables effect on the overall scoring potential of a configuration. Results of these sensitivity studies identified the major contributing factors involved in the design requirements. The conceptual design candidate concepts were more easily judged based on the studies' conclusions. The variables studied were time on the ground (or pit time), wind, wing taper and sweep, the use of flaps, and altitude.

- Time on the ground: Flight time is a direct multiplier in the overall score and therefore crucial to minimize. Pit time, or time on the ground loading the payload, is a critical part of the total flight time. As flight time decreases, minimizing pit time becomes the best way to maximize the overall score. Large score improvements were a result of shorter pit times.
- Wind Velocity: Wind sensitivity test were necessary because of the unpredictable wind conditions. Designing for higher wind speed is a risk. With higher wind, less wing area is necessary for lift off, which improves the overall score. If the wind is below the design criteria, the plane may not be able to lift off in the necessary distance causing an overall score of zero. The sensitivity analysis determined the risk/pay-off levels involved in designing for different wind speeds. Results show designing for a 10 mph wind has relatively low risk due to the traditional windy conditions at the competition.
- Flaps: Flapped airfoils were tested to determine the resulting flight characteristics and scoring potential. Flaps may be able to decrease wing area due to better lift off capabilities causing the weight of a configuration to decrease. However, flaps have an adverse effect on RAC.
- Wing Taper and Sweep: Both wing taper and sweep have adverse effects on RAC causing a design's scoring potential to decrease. Any performance advantages gained from using wing taper and/or sweep would be negated because of RAC. Therefore no wing sweep was considered and a taper ratio of one was used.
- Altitude: The cruise altitude makes large differences in power required from the motor. The lower the plane can fly the less energy it uses to get to that altitude. The plane must fly at a safe altitude, but must not fly so high that it uses too much energy. Historical data for R/C planes showed that an altitude of 50 feet is common.

4.2.2 Propulsion System Analysis Program

Although the propulsion model included in the mission analysis program predicted a fairly accurate overall flight profile, it was not detailed enough to design the complete propulsion system. A separate propulsion analysis program preformed a more detail analysis on the number of batteries, types of batteries, propeller diameter and pitch, and gear ratio for the take-off leg of the mission.

- Battery Model: Manufacturer's data on different types of available batteries were utilized in the propulsion model to predict their performance. Along with different types of batteries, the number, capacities of the cells, and overall efficiency of the battery packs were modeled to produce more realistic performance characteristics.

- Propeller Model: Varying types of propellers were also tested within the propulsion program. Propellers of different diameters and pitches were modeled to determine the thrust, torque, and angular velocities produced. This resulted in a range of optimal propeller sizes to be used during each mission.
- Gear ratio: Different gear ratios were also employed within the propulsion program. The energy consumed by each ratio along with the resulting thrust and energy consumed were evaluated. These tests were performed within the contest constraint of 40 amps. This limitation ensured that the fuse would not be blown during flight.

Many different propulsion configurations were analyzed using the propulsion optimization program that simulated the mission profile. Although the program is very useful in determining the trends associated with changing a given component, the final propulsion system will be determined through numerous static and dynamic tests. Figure 4.1 below shows the method used for determining optimal configurations within the program.

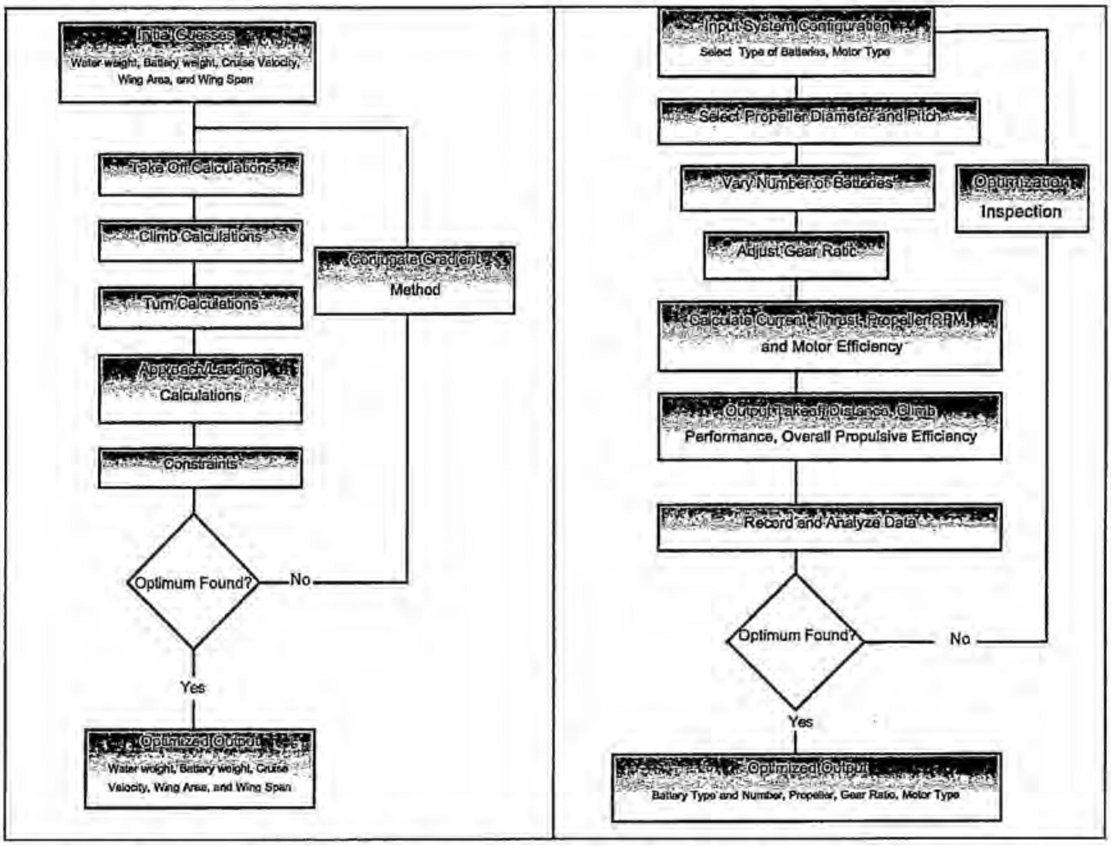


Figure 4.1: Block Diagrams for the Mission Profile and Propulsion Optimization Program

4.3 Optimization

With a base line configuration determined, the optimization program was refined to more accurately model the chosen design.

4.3.1 Mission Profile Program Results

The water weight, wing airfoil, wing area, wing span, and cruise velocity were derived from the mission model optimization program. Overall score was the major driving factor in selecting the best program outputs. The best results allowed lift off in just less than 150 feet and using nearly all of the available energy from the batteries. Performance considerations and stability and control effects not included in the optimization code were another deciding factor. Explained below are the mission profile optimization results.

<u>Water Weight</u>

It was found the maximum amount of payload should be carried. Overall score dropped considerably as payload weight dropped, due to the fact that payload weight affected the single flight score for the FBM significantly.

<u>Wing Airfoil</u>

Using historical data, three low speed airfoil's drag polars were chosen to input into the mission profile optimization program. A high lift airfoil, the Eppler 423; a low drag airfoil, Selig-Donovan (SD) 7032; and a more balanced airfoil, SD7062 were compared. It became clear that the short takeoff distance of 150 feet required an airfoil with a high lift coefficient during take off. The SD7032 was unable to produce scores as high as the other two airfoils. The wing area required for lift off where much larger than the other two airfoils, which increased the RAC considerably. Flaps where added to the SD7032 in an effort to solve this problem but even flapped the airfoil could not produce expectable scores. Therefore the SD 7032 was eliminated as an airfoil choice.

Comparing the scores produced from the E423 and the SD7062, the Eppler prevailed. The SD7062 was then flapped and compared with the E423. The E423 is already a highly cambered wing so the addition of flaps does not create more usable lift due to the large amount of drag created from the flap. The scores produced between the E423 and the flapped SD7062 were to close to be able easily decide. Therefore, other deciding factors were compared.

The deciding factor for airfoil selection was the E423's drag polar. Major concerns with E423 drag potential during cruise made the E423 un-applying. The E423 has a large drag spike near the aircraft's cruise lift coefficient. Therefore the E423 was eliminated as an airfoil choice. Figure 4.2 shows the drag polar and lift curve for the SD7062.

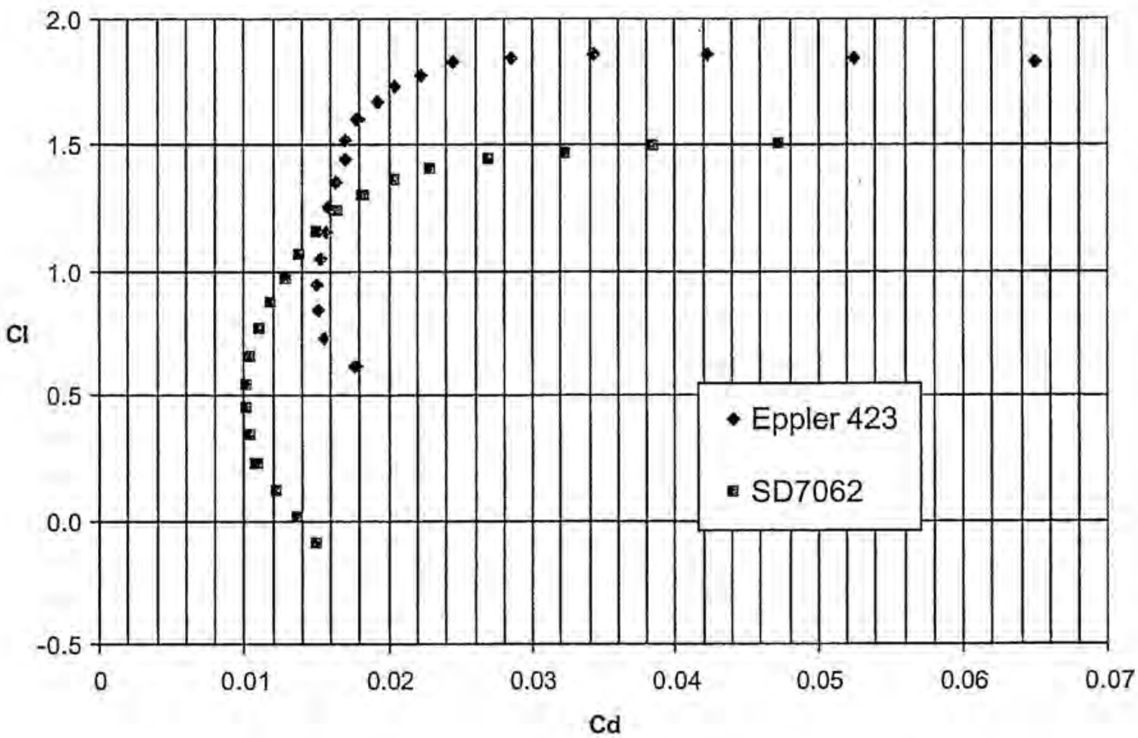


Figure 4.2: Drag Polar and Lift Curve for SD7062 and E423 at a Re of 250,000

Wing Area

The optimization program tended to decrease the wing area as much as possible, since no wing loading constraint was implemented into the program. Concerns with undersized wing areas caused optimal design with larger wing areas to become superior. Therefore the SD7062 was optimized without the use of flaps. Although this increased RAC it provided a margin of safety. The plane could take-off un-flapped in 150 feet with a 10 mph head wind. If the wind where to dip below 10 mph flaps could be deployed and the plane would still be able to take off. Furthermore, flaperons could be used instead of flaps and serve the same purpose without causing a major increase in RAC.

Another constraint not accounted for in the mission optimization program was the decrease in lift due to the disruption of the lift curve over the wing caused by the fuselage. It was assumed the area of the wing covered by the fuselage would be added to the optimized wing area to ensure enough lift is produced. The final range of optimal wing area for the SD7062 was 5.4 to 5.7 square feet.

With the airfoil selected, wing sizing could begin. Two general trends emerged within the optimization code. High scores were found when small wing areas were used with more batteries or large wing areas were used with fewer batteries. The highest scores were found in designs using fewer batteries with smaller wing areas but both configurations had similar scoring potentials. Since the uncertainty range of the optimization code was greater than the scoring range, other deciding factors needed to be examined.

A more detailed study of motor performance during takeoff was necessary in the analysis of the wing. Several optimized combinations were analyzed during takeoff with different propeller sizes, gear box ratios, and batteries. This analysis looked at takeoff distances in different wind conditions, current through the cut-off fuse, and power used during takeoff. All the configurations were re-optimized with the best propulsion system possible in the preliminary phase. The optimization code and propulsion code double-checked one another to ensure that each configuration could successfully takeoff while still maintaining power efficiency and the proper endurance. The score was also monitored as cruise velocity and battery cells varied. The end result showed a more optimized score with smaller wing areas and larger batteries, but this configuration did not have the performance and takeoff margin of safety desired. Figure 4.3 shows the wing area sizing trends produced by the optimization program. The circle on the graph represents the decision made.

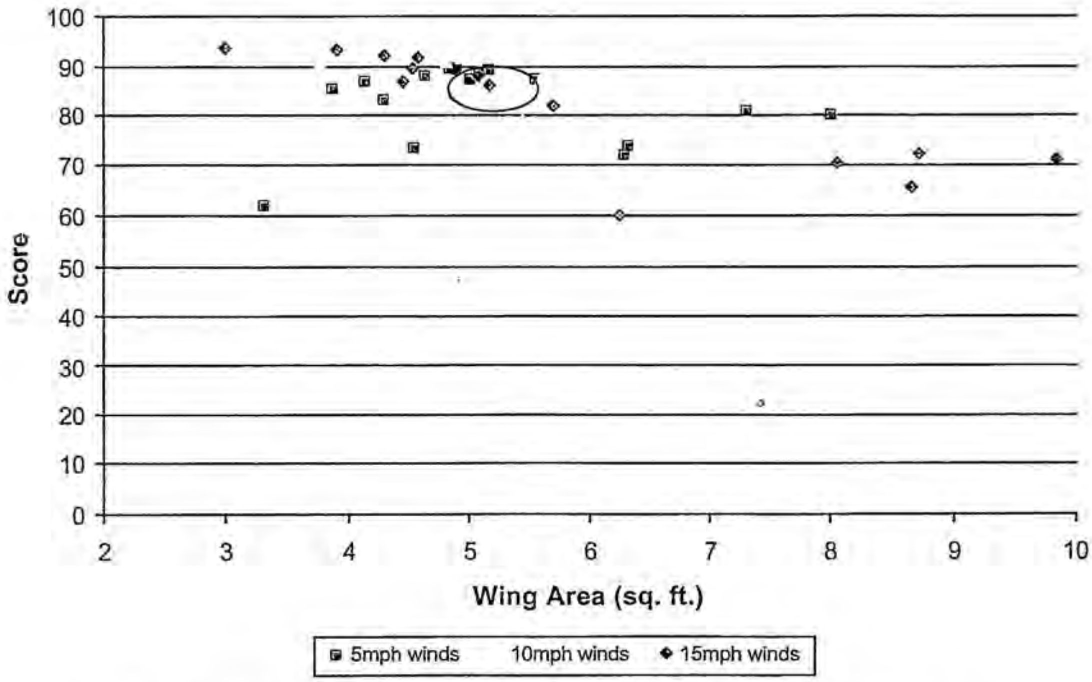


Figure 4.3: Wing Planform Area Optimization Results using an Un-flapped SD7062

<u>Wingspan</u>

Without an aspect ratio limit, the optimization code would drive the aspect ratio too high, which would make the plane nearly uncontrollable. The code includes a maximum aspect ratio constraint of 8 to ensure stability and control is not compromised. As expected the program optimized the wingspan to create an aspect ratio of 8. The range of wingspans considered was 75 to 80 inches.

Cruise Velocity

The optimization program performed a power/speed trade-off to determine the optimal cruise velocity for the design. The energy used at take-off, climb, and cruise must not exceed the batteries capabilities. The program found the optimal power requirements for take-off and cruise, which keep the cruise velocity as high as possible keeping mission times low. Mission time cannot be neglected since it is such a large factor in the scoring function. The optimal cruise velocity that the program outputted ranged from 70 to 80 feet per second.

4.3.2 Propulsion Program Results

The preliminary design phase for the propulsion system involved utilizing a computer program that optimized the motor, gearbox, batteries, and propeller. The program modeled the complete propulsion system throughout the mission profile. The propulsive system components were varied such that the overall propulsion system met or exceeded the aircraft performance requirements estimated for thrust, power, and system efficiency and endurance. This optimization process was an iterative one performed in collaboration with the aerodynamics group. Both groups worked in combination to develop an aircraft model that produced the most desirable score and performance characteristics.

Motor

The propulsion team determined early in the preliminary design phase that Graupner 3300-6 motors would be used. The Graupner motor produced superior results to the Astroflight models. During the optimization process the propeller diameter and pitch, battery cell capacity and number, and gear ratio were held constant, while motor type was varied. Manufacturer provided motor data along with historical test data was varied according to each motor type. The key parameters varied were the voltage constant (Kv) value, internal resistance, and no load current. A direct comparison of the Graupner motors to the Astroflight models can be seen in Figure 4.4 on the following page.

Gearbox and Gear Ratios

An MEC Superbox was used to transmit the power from the motor to the propeller. The pinion and spur gears can be quickly changed to allow gear ratios from 1.18:1 to 3.50:1.

To determine an optimal range of gear ratios, the propeller size, battery type and cell count, and motor were held constant in the optimization program. The gear ratios were then varied and the thrust produced, power required, current drawn, efficiency, and battery endurance were catalogued. The thrust produced and power required from the batteries were matched with the mission profile model requirements for takeoff to obtain realistic results. The current drawn from the batteries was monitored to ensure that the system would not destroy the 40 amp fuse required for safety.

The resulting optimal gear ratios ranged from approximately 1.33 to 2.27. This range provided the most thrust at full throttle while still remaining below the maximum amount of pulled current. At lower gear ratios more thrust is produced at the cost of pulling more current from the batteries.

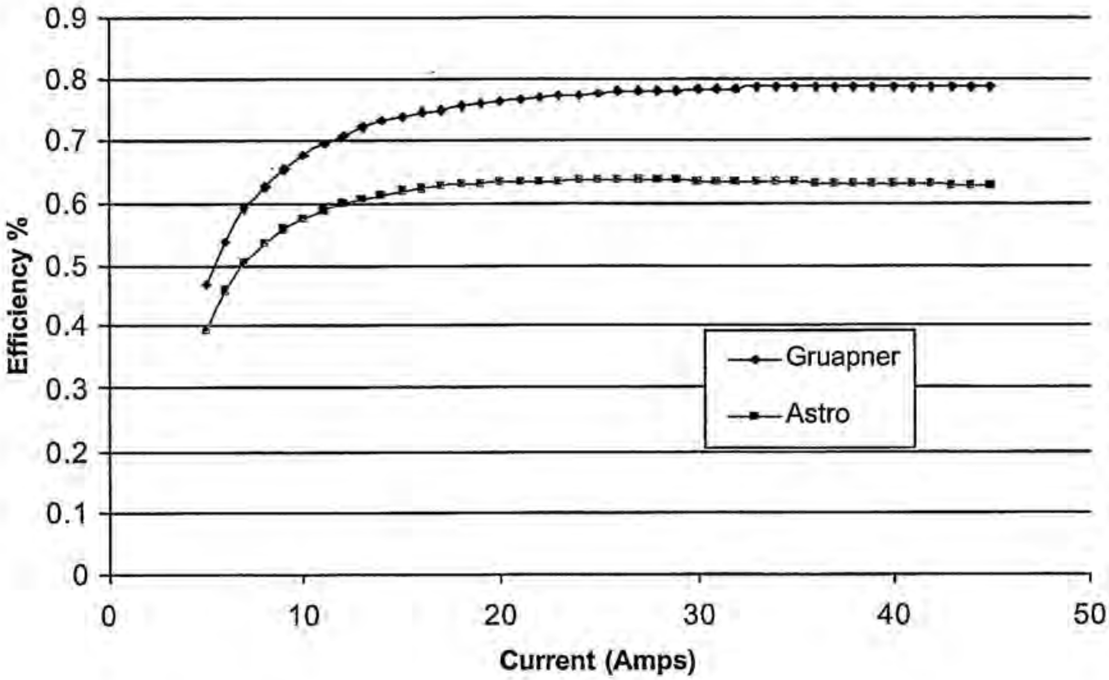


Figure 4.4: Efficiency versus Current for Graupner

Batteries

Battery selection plays a vital role in the optimization of the propulsion system. It is necessary to find the battery which best suit the needs of the propulsion system. The Ni-Cd batteries considered for powering the propulsion system are the CP 1300 SR, SR 1200 Max, SR 900 Max, and SR 2400 Max. Battery information such as the internal resistance, weight, capacity, and capacity to weight ratio can be found in Table 4.1 below.

BATTERY INFORMATION				
Battery Type	***R***	***W***	***Capacity***	***Cap/W***
CP 1300 SR	0.0065	**0.0775**	990	12774.19
SR1200 Max	**0.004**	0.0981	1080	11009.17
SR 900 Max	0.0055	0.083125	810	9744.361
SR 2400 Max	0.0045	0.135	**2100**	**15555.56**

Table 4.1: Battery Information

The first major battery selection consideration is weight; every bit of extra weight penalizes the RAC. The next major consideration is the capacity to weight ratio. It is important to get the most power out from the specific battery weight as possible.

All battery types shown in Table 4.1 were tested in the optimization program. The best performers were the CP 1300 SR and the SR 2400. When optimized the CP 1300 SR gave the aircraft a better flight score. It can be noted that the SR 2400 Max has a much greater capacity, but the extra capacity is unnecessary according the optimization program. The CP 1300 SR is much lower in weight, yet still has enough capacity to produce a better flight score. Therefore the CP 1300 SR was chosen as the optimal battery type.

Utilizing the motor selected and the most favorable range of gear ratios the number of battery cells was varied until the thrust and power requirements for the mission profile were met, resulting in an optimal number of cells from 14 to 18 cells.

Propeller

The main design consideration for propeller selection is sizing, propeller pitch and diameter for the best propulsive efficiency, c_p. A larger diameter propeller would be necessary. Another factor to consider is the propeller pitch-to-diameter, p/d, ratio. Propellers with a low p/d ratio have the greatest c_p at lower airspeeds while higher p/d props have a high c_p at higher airspeeds.

Testing various sized propellers in the optimization program resulted in an important conclusion. The optimal propeller size is dependant on whether or not the aircraft has full payload or no payload. For the FBM, the fully loaded aircraft requires a larger propeller. Optimal propeller sizes range from 16" to 20". For the FM, the empty aircraft requires a smaller diameter propeller. Optimal propeller sizes range from 14" to 18" for the FM.

4.4 Analysis Methods and Sizing

All three technical groups employed separate analysis methods to help refine the respective sizing of the aircraft's design.

4.4.1 Aerodynamic Group

Main aircraft component sizing and power and wing loading are discussed below. Issues surrounding sizes of the components are discussed and the optimal sizes are selected. The components covered are fuselage, empennage, and control sizes.

Fuselage Size

In order to fit the entire fuselage in the box, the length was slightly less than 4 feet long. In addition, the cross section was constrained to a 4" X 8" rectangle due to the RAC penalty. A one piece fuselage and empennage would benefit the structural integrity and minimize weight. The fuselage was modeled in

Pro-E and all internal components where placed inside to ensure proper fit. With all the internal components inserted the fuselage was streamlined as much as possible.

Empennage Size

Horizontal and vertical stabilizer's size and airfoil type were determined. The FOMs for the tail included performance and RAC considerations. The tail had two important tasks: produce the necessary pitching moment at takeoff and landing and trim the airplane in cruise.

- Horizontal Stabilizer: If the projected horizontal tail span exceeded a quarter of the wingspan, the tail was classified as a wing by the RAC and penalized. Therefore, the horizontal tail span was limited to a quarter of the wingspan making the span approximately 19.2 inches. An approximate elevator size had to be determined to ensure the elevator would not be an unreasonable portion of the horizontal surface. The elevator was sized by investigating the worst case scenario, landing. If the elevator had enough control power to trim at landing, it would be sufficient for all other flight regimes. Initially, a NACA 0009 was chosen for the horizontal surface. However, it was found that a reasonable elevator size and deflection could not trim the aircraft at C_{Lmax}. Thus a NACA 2412 was selected to give the necessary down force. Using this airfoil, an elevator having 25% area could trim at landing with a deflection of approximately 25°. An elevator deflection of 1° was necessary to trim at cruise conditions. The horizontal stabilizer was tapered for aesthetic reasons, with a root cord of 8.5" and a tip chord of 5.9".
- Vertical Stabilizer: Using historical data a vertical tail volume of 0.09 was chosen. A NACA 0009 airfoil was selected due to its relatively thin thickness. As done for the elevator, the rudder had to be determined to make sure the vertical tail was large enough. Assuming a 7.5 mph cross-wind component, the rudder was sized and found to be an appropriate percentage of the vertical tail. Therefore, the vertical stabilizer did not need to be increased in size. The rudder spans the entire surface and is 30% of the root chord. The vertical stabilizer was swept for aesthetic reasons.

Control Surface Sizes

- Rudder: The rudder was sized to provide adequate yaw control of the airplane. An analysis spreadsheet was developed to determine the necessary size of the rudder to maintain yaw stability in a high crosswind landing. This spreadsheet calculated the weathercock stability coefficient for the aircraft and the maximum sideslip angle. It then determined the necessary rudder deflection to trim the aircraft at a given rudder size. The optimal size was found to be 30 percent of the vertical tail area, or 18.6 square inches. At this size, the rudder deflection required to trim during landing at a maximum pure crosswind of 7.5 mph is 25 degrees, which allows for a slight factor of safety.
- Elevator: Using the procedure developed in the horizontal stabilizer sizing portion, a reasonable elevator size was chosen. An elevator having 25% chord and a maximum deflection of 25° was sufficient to trim the aircraft during the worst case scenario of landing.

- Flaperons: The flaperon size needs to maximize the efficiency of the wing for takeoff. At this maximum efficiency, it must be able to produce enough lift for takeoff with minimal wind. The flaperons span the entire length of the wing to maximize lift. Several different flapped SD7062 drag polars were investigated. The comparisons are presented in Figure 4.5. It was found that a flap that is 10% of the chord, and flapped 20 degrees produced the optimal lift coefficient. But this flap was not chosen due to concerns with the flex of the skin hinge. Therefore, a flaperon with 15% chord, and 20° deflection was chosen. Flaperons can also be deployed in a way to increase drag, which could be an added benefit when trying to slow the plane down on its decent.

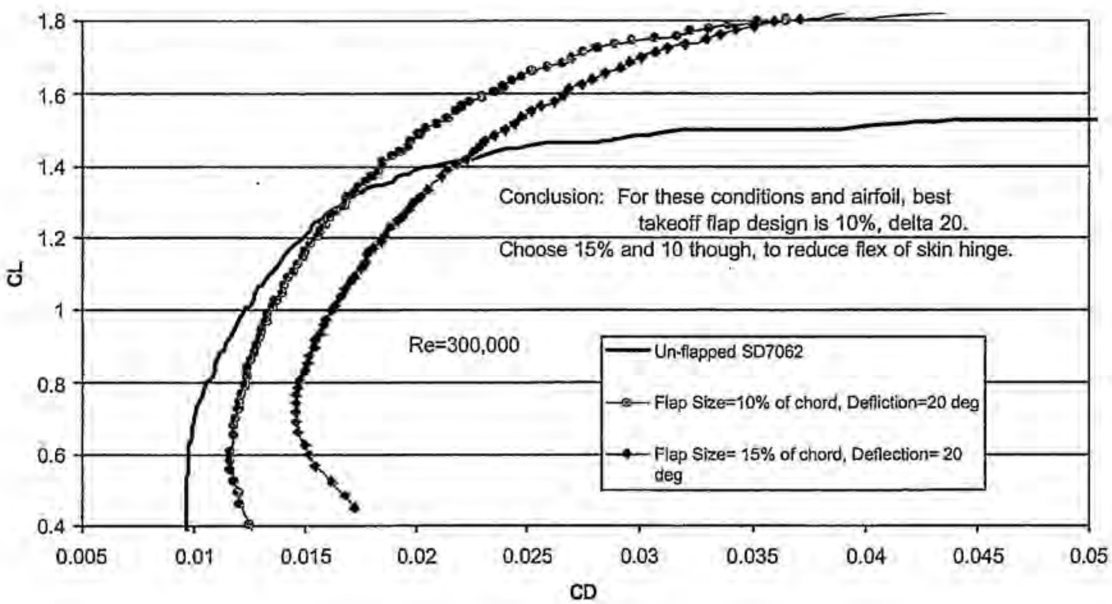


Figure 4.5: Flap Size Determination

Wing and Power Loading

The wing loading for the preliminary configuration was approximately 0.34 oz./in^2 and the power loading was approximately 52 W/lb. The two loading values were a compromise to achieve the correct mission balance. Higher wing loading allowed the plane to penetrate through the wind better, achieve higher cruise velocities, and be less susceptible to gusty conditions because greater changes in pressure differential are required to disturb the plane. However, high wing loadings are less stable at low speeds. To overcome the wing loading disadvantage, a higher power loading was needed to help overcome takeoff and climb requirements.

4.4.2 Structures Analysis

Finite Element Analysis was used to find stress concentration in major components. The results were then used to determine placement of internal components.

Fuselage

The fuselage was modeled using a Pro/Mechanica finite element analysis. This analysis revealed the areas of the fuselage that would need reinforcement by predicting the overall load distribution. The fuselage was modeled under two load sets: a landing impact of four times normal gravity and a turn that generates four times normal gravity. For the static loading set, the fuselage was fixed at the panel where the gear attaches to the fuselage and at the tail dragger wheel attachment. The loads were then applied to the internal surface of the fuselage at four times their actual weight in order to get a ballpark figure of the stress concentration maximums. Two idealizations were made in order for the analysis to run: the filleted edges were removed to simplifying the geometry and the inner and outer skins were modified into a single shell at the midpoints. These simplifications made it possible to run the simulations and generate the visual distributions in a matter of minutes instead of hours. With a maximum local value of 190 kpsi the most likely place for the fuselage to fail is at the thin tail section in buckling. This visual output of stress and strain concentrations enabled some preliminary decisions to be made about the placement of internal structure. It was apparent that bulkheads where necessary to maintain the structural integrity at the main gear bow connection, aft of the payload holding tank, and at the thin section of the tail.

Wing/Empennage

PRO/Mechanica was also utilized for a finite element analysis of the wing structure. The wing was constrained at the root as a cantilevered beam. Figure 14 illustrates the analysis. Note the dramatic deflections at the wingtip and relatively small values for maximum principal stress. This can be attributed to the strength of the tubular carbon spar. The local high stress region at the wing root indicates that it would fail in the first mode of Euler bucking at that region. A further buckling analysis found the bucking load factor of approximately 1.5, meaning that a load 1.5 times greater than the analysis loading would be required to buckle the wing. The empennage was not modeled as a load carrying member, because it would never see any structural loading. It was assumed that the skin strength would be sufficient with the addition of shear webs to provide the stiffness required to maintain the proper shape.

Landing Gear

It was desired that the main landing gear have a slight airfoil shape to minimize drag. A flat section in the landing gear was required to bend an airfoil shaped core around a corner. The main gear would then have relatively sharp angles compared to a true bow type.

Once a composite sandwich structure was decided on, several material test pieces were constructed to determine the optimum lay up. Initially, two samples were constructed for destructive testing. Each test piece was 14" long and approximately 2" wide. Piece 1 used a 1/8" balsa core and 3 ounce fiberglass with a laminate lay-up of $[45$, mono-carbon filament tape, $0, 45, 0_4]_s$. Piece 2 used a 3/16" core and 3 ounce fiberglass lay-up of $[45$, mono-carbon filament tape, $0{,}45, 0_2]_s$. The mono-carbon filament tape was placed near the outside of the lay-up to increase the stiffness of the pieces. Both test pieces were sanded to a tapered point with a rounded leading edge to approximate an airfoil shape, reducing drag as much as possible.

Both samples were tested in the same manner. One end was rigidly attached to a table, and weights were added at a moment arm of ten inches until the piece failed. From this test, the maximum bending moment and the maximum bending stress were found. The deflection was also measured during the tests to determine if the flex in the landing gear would cause the propeller to hit the ground.

The test piece with the thicker core held more weight and deflected less per pound. The strength of the gear can be increased while decreasing the number of layers by increasing the core thickness. The use of a thicker core to increase strength was important so that the overall weight could be reduced. The most critical part of the landing gear was the elbows due to their high stress concentrations. Two molds were built with an elbow to test how the test pieces react to applied loads. The first test used 1/8" balsa wood core covered with 6 layers of 6-ounce fiberglass. Fiberglass de-lamination occurred at the corner so quantitative data was obtained. The desired amount of stiffness was not produced with this lay-up. The second test used a different lay-up scheme but produced similar results. From these early failures, it was decided that a true bow landing gear mold will be used with a large corner radius, in order to reduce the stress concentrations in the corner.

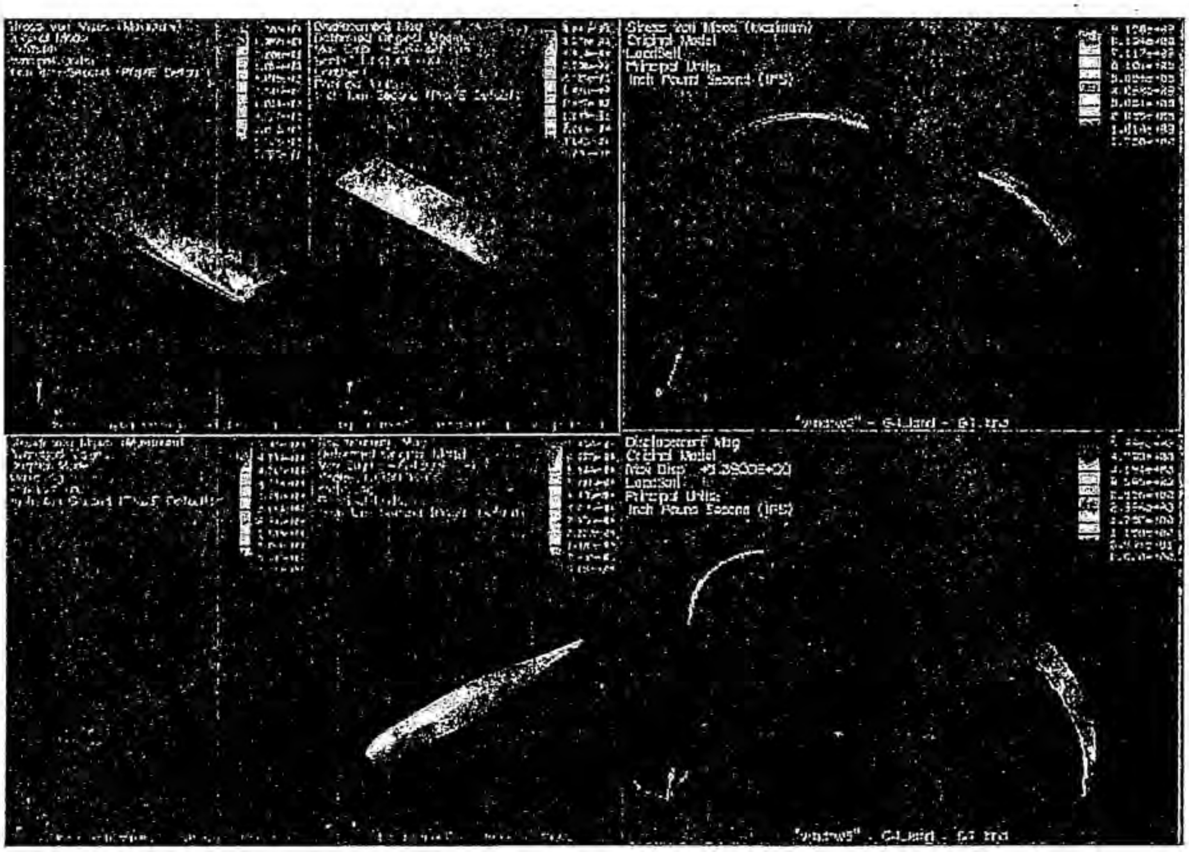


Figure 4.6: Stress and Deflection of Major Structural Components as Predicted by FEA

4.4.3 Payload System Analysis

The payload system required a baffled tank that would carry exactly 4 liters of water and prevent water shifting. The retractable boom was designed to give the best head pressure without causing excess drag of the plane. The boom must ensure no water leaks while retracted. The boom must also be designed to be easily and repeatedly lowered. All these component must be designed in a limited amount of space to allow the fuselage to remain as streamlined as possible.

Tank Sizing

The shape and method of fabrication of the tank was determined. Using the maximum height and width of the inside of the fuselage, an approximate length for the tank was calculated by using a volume slightly greater than four liters. This length along with the maximum fuselage height and width were used to construct a three-dimensional tank in Pro/Engineer. Using Pro/Engineer, the tank was cut and shaped smoothly to fit the inside the fuselage while allowing room for other internal components. The bottom of the tank was change to form a sump to ensure no water could be trapped in the tank. The aft portion of the bottom of the tank was cut to the shape of the boom so that the boom could be flush with the plane. Another design consideration for the tank was the hole through the middle, which allowed the wing's spar to pass through. The tanks final volume was maintained slightly above 4 liters to leave room for baffling.

Baffling

Baffling in the tank was a necessity. Without proper baffling the plane would be uncontrollable. If the water in the tank is free to shift the CG of the plane could change drastically, causing catastrophic results during flight. Listed below are the superior configurations to control the water in the tank.

- Vertical Separation Baffles: Vertical separation baffles were the first solution investigated. They are baffles which only allow travel vertically between parallel horizontal baffles, effectively forming separate compartments. Several orientations of these compartments were investigated, but they generally lacked the desired characteristics, in sloshing or in static settling.
- Horizontal Separation Baffles: Horizontal separation baffles were experimented with, much like those found in an automotive fuel tank. These baffles are usually vertical plates which minimize momentum transfer from compartment to compartment in the tank. Good dynamic movement characteristics are provided, but static shifting characteristics are deficient. Some additional modifications were examined which would assist in the reduction of static shifting, such as one way gates in each baffle allowing static shift toward the plane CG.
- Full Structure Baffles: Several full structure baffles we examined as well, such as a constructed straw box and a honeycomb tank insert. Both tests provided excellent dynamic movement damping, but retained a large amount of water after payload evacuation, due to the baffle's large surface area.

The following FOMs were used to reach a final decision, which is presented by the weight decision matrix in Table 4.2.

- Static Movement: Static movement of the payload is the most serious issue that has to be dealt with in the design. With the small size of the entire aircraft, and the majority of the weight being payload, a shift aft of the payload C.G. of an inch or more could render the aircraft neutrally stable or unstable, and effectively leave the control surfaces useless, resulting in a crash. Also, a neutrally stable aircraft lends itself to pilot induced oscillations(PIO), in which some environmental change or control adjustment sets off an oscillation which cannot be recovered from.
- Dynamic Movement: Dynamic movement of the payload is also a major concern for the design. If the plane has no dynamic damping of the payload movement could trigger a catastrophic PIO.
- Water Retention: The structure must allow for complete payload drainage in order to prevent a penalty to the score. Minimizing surface area not only reduces water retention, but increases the evacuation speed.
- Weight: Weight of this structure must be low for the best score possible.

Figures of Merit	Weighting Factor	Vertical	Full Structure	Horizontal (modified)
Static Shift	0.4	1	2	2
Dynamic Shift	0.2	1	3	3
Water Retention	0.2	3	1	3
Weight	0.2	3	1	3
Total	1	1.8	1.8	2.6

Table 4.2: Baffle System Evaluation

Table 4.2 demonstrates the superior characteristics of the horizontal baffling method. The method of modifying the baffles with one way gates allows for the best payload control. Numerous tests will be performed to produce the optimum baffle and gate sizes and placements. These tests will also ensure no water will be trapped in the tank.

Boom Length

Although the boom is by far the best method for releasing the water, definite challenges that must be considered. The boom must be long enough to provide the necessary head pressure while not creating too much drag.

The tubing that ran from the tank to the end of the nozzle was surgical grade tubing. Surgical tubing provided excellent weight and durability characteristics. The tube diameter from the tank to the end of the boom was 0.75 inches. This allowed for the water to reach the end of the boom faster than smaller tube diameters. The boom was shaped as streamlined as possible around the tube to reduce as much drag as possible. At the end of the boom a 0.5 inch nozzle funneled the water out. A minor pressure drop from the windward side of the boom to the trailing edge of the boom was utilized by putting the nozzle closer to the trailing edge of the boom and facing it slightly aft.

In order to size the length of the boom a spreadsheet using Bernoulli's equation was created, which calculated drain time. The drain time for 4 liters of water was graphed versus the boom lengths. Optimal drain times are achieved at boom lengths from 16 to 18 inches.

Boom Deployment and Retraction Methods

In order to reliably deploy and retract the boom, a tight-fitting, lightweight method was desired. Initially, the boom was to be mounted onto a ball-type valve to allow for the boom to be deployed by the turning of the valve, simultaneously dropping the boom and starting the payload release. This idea was discarded after some tests were made, which found these valves to be bulky and hard to integrate into the fuselage configuration. The other alternative was a pinch-type valve, relying on passive actuation. When the boom is in the stowed position, the hose is pinched, and forced closed. When the boom is deployed, the pinch-arm is rotated off of the tubing, allowing flow through the boom. The boom is deployed by a hinged arm assembly, attached at the midpoint of the boom. This placement reduces the amount of torque required to extend the boom down by giving a greater mechanical advantage. A test apparatus was built to model the fuselage, and several tests were performed to ensure that the design was reliable. Once satisfactory results were obtained, the construction began with the Bernoulli's calculations from the previous section serving as a basis for component sizing.

4.4.4 Propulsion Group

Once analytical testing with the optimization program was completed during the preliminary design phase, live static testing began. The parameters tested included the gearbox, battery cell count and configuration, and propeller dimensions. Actual performance characteristics provided the most realistic method for optimization of the propulsion system. These tests determined the capacity of the batteries, motor efficiency, thrust, and speed. Temperature was monitored during these tests to determine thermal effects on the system performance.

The final propulsion system was tested using a modified dynamometer. Thrust was calculated using a rope and pulley system, weights, and a Detecto AP-4K 4000 g scale. Current, voltage, and power was recorded at different intervals using an Astro Model 101 Super Whatt Meter.

Performance characteristic curves were created from an optimal configuration. Figure 4.8 shows the voltage, current, thrust output, and revelations per second versus time for a Graupner 3300-6 with a 20" propeller, gear ratio of 1.72, and 16 CP 1300 battery cells ran at half throttle.

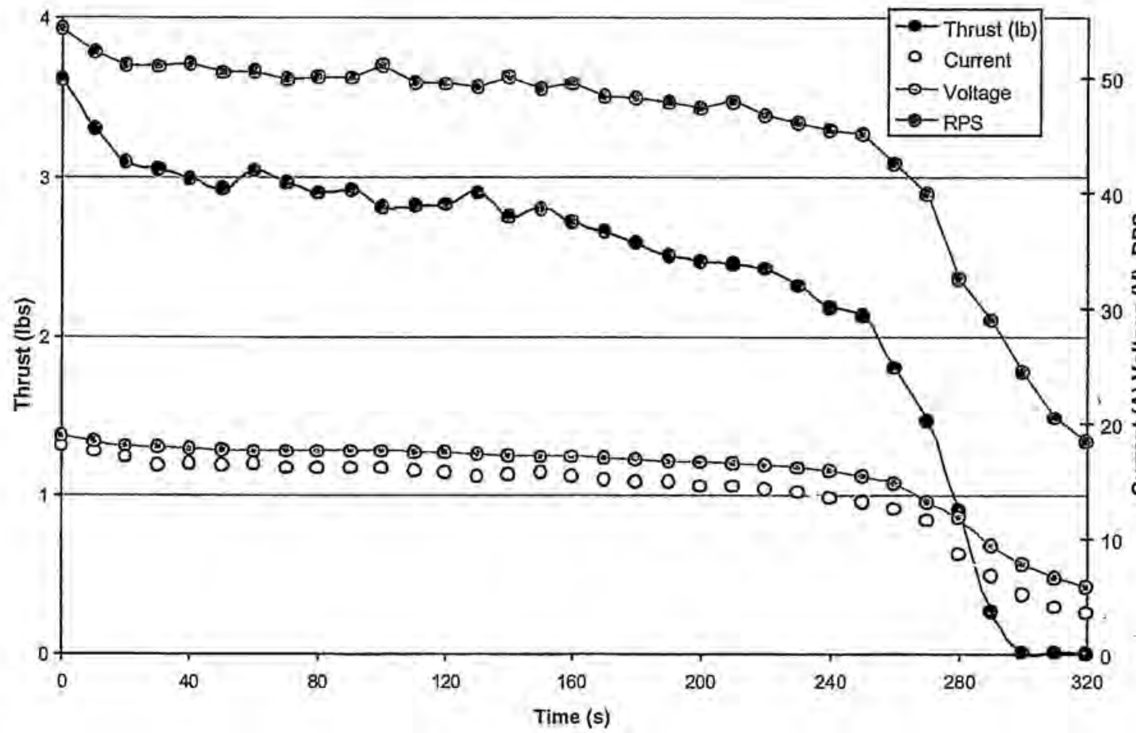


Figure 4.8: Motor/Battery Performance Curve for Graupner 3300-6, 20" Prop., 1.72 gear ratio, and 16 CP 1300 Cells at Half Throttle.

The endurance/performance test shown in Figure 4.8 provides several pieces of information. By integrating the area under the current curve for one battery cycle the total charge in the batteries can be determined. Also the actual internal resistance of the motor can be determined from the no-load voltage of the battery pack and the current pulled.

4.5 Final Aircraft and Predicted performance

At the end of the preliminary phase the optimal ranges for each aircraft component were compiled and compared against one another until a final, optimal configuration was determined. With the final aircraft configuration decided, the predicted performance can be determined. A complete aerodynamic analysis was performed, including a mission simulation, aerodynamic coefficients and derivatives, and static and dynamic stability.

4.5.1 Aircraft Configuration

The final configuration is a conventional aircraft with a mid mounted wing. The fuselage has been streamlined around the interior components creating a low-drag body. The wing was designed with a span of 79.2 inches, a chord of 10.1 inches, and an aspect ratio of 7.8. The wings can vary 17.6 degrees before scraping the runway. The flaperons extend across the full length of the wing span and occupy 15 percent of the chord.

The tail is of a conventional configuration. The horizontal tail was sized with span of 19.2 inches and a taper ratio of 0.7, keeping it within one quarter the wing span. The vertical tail area was sized to approximately 50 square inches. The rudder and elevator are both sized at 30 percent of the vertical and horizontal tail, respectively. The final propulsion system was a Graupner 3300-6 utilizing 16 CP1300SCR battery cells. For the FBM, an APC 20x13E propeller was chosen.

4.5.2 Predicted Performance

The mission performance of the final aircraft was estimated from the performance code used during optimization. The mission was broken into separate stages for analysis. The results of this analysis can be seen below in Table 4.3.

Mission Components	Time (sec.)	Distance (ft.)	Velocity (ft/sec)
Takeoff (loaded)	5.56	143	41.9
Climb (loaded)	7.3	400	60.47
First Turn (loaded)	10.7	162	Acceleration
Acceleration Leg (loaded)	9.25	674	Acceleration
360 turn (partially loaded)	12.7	0	Acceleration
Cruise Leg (un-loaded)	8.66	326	75
Final Turn (un-loaded)	2	162	Deceleration
Deceleration & landing (un-loaded)	4.56	457	Decelerate to 25.46
Ground time for loading	15	0	-
Total time	75.73	-	-

Table 4.3: Time Spent for One Lap During the FBM, Based on 10 mph Head Wind

From Table 4.3, it can be seen that the takeoff distance was 143 feet going into a 10 mph head wind with no flap deflection. To maintain climb velocity, the plane must accelerate to 60.47 fps. The cruise speed was 75 fps. With all the segment times summed, the total time for one lap is 75.73 seconds. This produces a total FBM time of 136.46 seconds or approximately 2 minutes 16 seconds.

4.5.3 Aerodynamic Coefficients and Stability and Control Derivatives

Table 4.4 provides a listing of all the relevant stability and control derivatives for the cruising portion of the mission. Methods employed for this analysis were taken from Nelson (1998).

$C_{L,\alpha}$	5.12	$C_{m,\delta e}$	-0.89	$C_{n,r}$	-0.066
$C_{X,\alpha}$	0.178	$C_{y,\beta}$	-0.277	$C_{l,r}$	0.059
$C_{m,\alpha}$	-1.35	$C_{n,\beta}$	0.0948	$C_{l,\delta a}$	0.368
$C_{Z,\dot{\alpha}}$	-1.37	$C_{l,\beta}$	0	$C_{n,\delta a}$	-0.024
$C_{m,\dot{\alpha}}$	-3.88	$C_{y,p}$	0	$C_{y,\delta e}$	0.147
$C_{Z,q}$	-3.56	$C_{n,p}$	-0.0342	$C_{n,\delta r}$	-0.049
$C_{m,q}$	-10.07	$C_{l,p}$	-0.853	$C_{l,\delta r}$	0.0072
$C_{Z,\delta e}$	-.314	$C_{y,r}$	-0.183		

Table 4.4: Stability Derivatives for Cruise (per radian)

The change in position of the water in the tank affects the overall center of gravity of the plane, altering the handling qualities. A computer modeling analysis was used to calculate the worst case movement forward and aft of the initial center of gravity location. Results of this analysis showed that the payload center of gravity shift caused the overall aircraft CG to move 4.78% (in percentage of the wing chord) forward and 10.38% aft of its initial point. Figure 4.9 presents the CG shift graphically. Such shifts are within the capabilities of elevator control as the static margin is never reduced below 9.3%. These shifts also demonstrated the need for tank baffles and valves to help control the static and dynamic shift of the water payload.

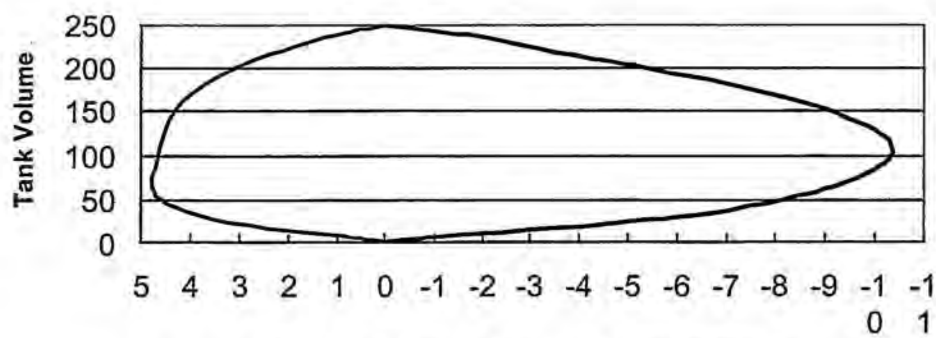


Figure 4.9: Coefficient of Moment Due to Elevator Deflection Versus Alpha and Moment Contributions of Aircraft Components

5.0 Detail Design

The goal of detail design is to finalize the aircraft design and design the methods of construction based on the results from the preliminary phase. Strength and stability requirements were determined and met, while reducing aircraft weight and drag. The final optimization was performed by detailed analysis and design of the plane's aerodynamics, power plant, and structure.

5.1 Component Selection and System Architecture

Table 5.1 presents the final aircraft geometry, weight, systems, and performance configurations.

Geometry	Value	Performance Data	Value
Center of Gravity Location	12 in	C_L max	1.5
Fuselage		L/D max	7.5
Length, (in)	48	**Gross Weight Conditions**	
Width, (in)	8	Maximum Climb Rate, (fps)	7.5
Height (in)	4	Stall Speed, (fps)	24.3
Wing		Maximum Speed, (fps)	71.65
Airfoil	SD 7062	Take-off Field Length, (ft)	110.2
Span, (in)	79.2	**Empty Weight Conditions**	
Area, (ft^2)	5.56	Maximum Climb Rate, (fps)	16.5
Aspect Ratio	7.83	Stall Speed, (fps)	16.9
Incidence Angle, (deg.)	0	Maximum Speed, (fps)	80.1
Flaperon Area Per Wing, (in^2)	72	Take-off Field Length, (ft)	76.7
Horizontal Stabilizer		**Weight Statement**	**Value**
Airfoil	NACA 2408	Payload, (lb)	8.8
Span, (in)	19.2	Manufacturer's Empty Weight (MEW), (lb)	9
Chord at root, (in)	8.43	Gross Weight, (lb)	17.8
Chord at tip, (in)	5.9	**Systems**	**Details**
Volume, (in^3)	0.59	Motor	Graupner 3300-6
Incidence Angle, (deg.)	0	Battery Configuration	16 X CP-1300 SCR
Elevator Area, (in^2)	34.4	Gear Box	MEC Superbox
Vertical Stabilizer		Gear Ratio	1.72:1
Airfoil	NACA 0009	Speed Controller	Astro Flight Model 204
Span, (in)	8.75	Propellor (FBM-nominal)	APC 20x13E
Chord at root, (in)	11.5	Propellor (FM-nominal)	APC 16x12E
Chord at tip, (in)	5.75	Radio	Futaba T8UAP
Volume (in^3)	0.04	Reciever	Futaba FP-R14-8DP
Rudder Area, (in^2)	18.64	Servos	JR DS3421 Premium Digital Servo: Mini

Table 5.1: Final Aircraft Configurations

5.1.1 Propulsion System

The final propulsion configuration can be viewed above in Table 5.1. This configuration required a takeoff thrust of 7.6 pounds and a required power of 684 Watts based the propulsion analysis program. Figure 5.1 below shows a simulated mission profile performed on the dynamometer. The test is based on the power required for each section of the mission profile.

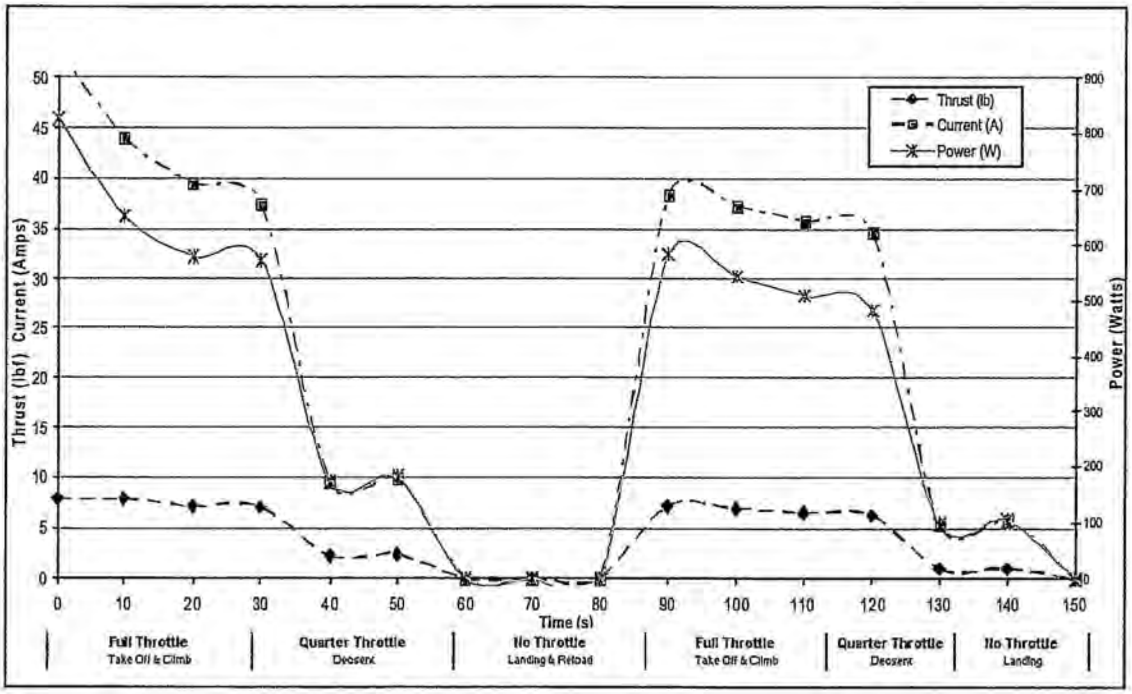


Figure 5.1: Simulated Mission Profile with Thrust, Current, and Power

5.1.2 Structures System

The aircraft's structural design was divided into the following major categories: main wing structure, fuselage, empennage, payload system, and landing gear. The following section will highlight the major components of the aircraft and explain their internal structure. Each of the major components of the structural system can be seen in Figure 5.2.

The internal structure of the wings consists of a standard spar, rib, and shear web configuration. The main spar will be a 0.625" carbon tube 18 inches in length, carrying through the fuselage (including reservoir) and 5 inches into each wing. Ribs will be placed at the root of the wing and at the end of the spar for maximum torsional rigidity. An additional rib will be placed at the end of the wingspan to closeout the wing structure. Shear webs will be placed at approximately 25% and 75% of the chord in order to strengthen the structure in bending. The forward-most shear web will continue the trajectory of the main spar out to the closeout rib. The smaller, aft shear web will enclose the forward edge of the skin-hinged ailerons which run from the middle rib to approximately the close out rib. The aileron servo will be attached to an intermediate rib in order to place it at the point which will cause the least amount of control surface twist.

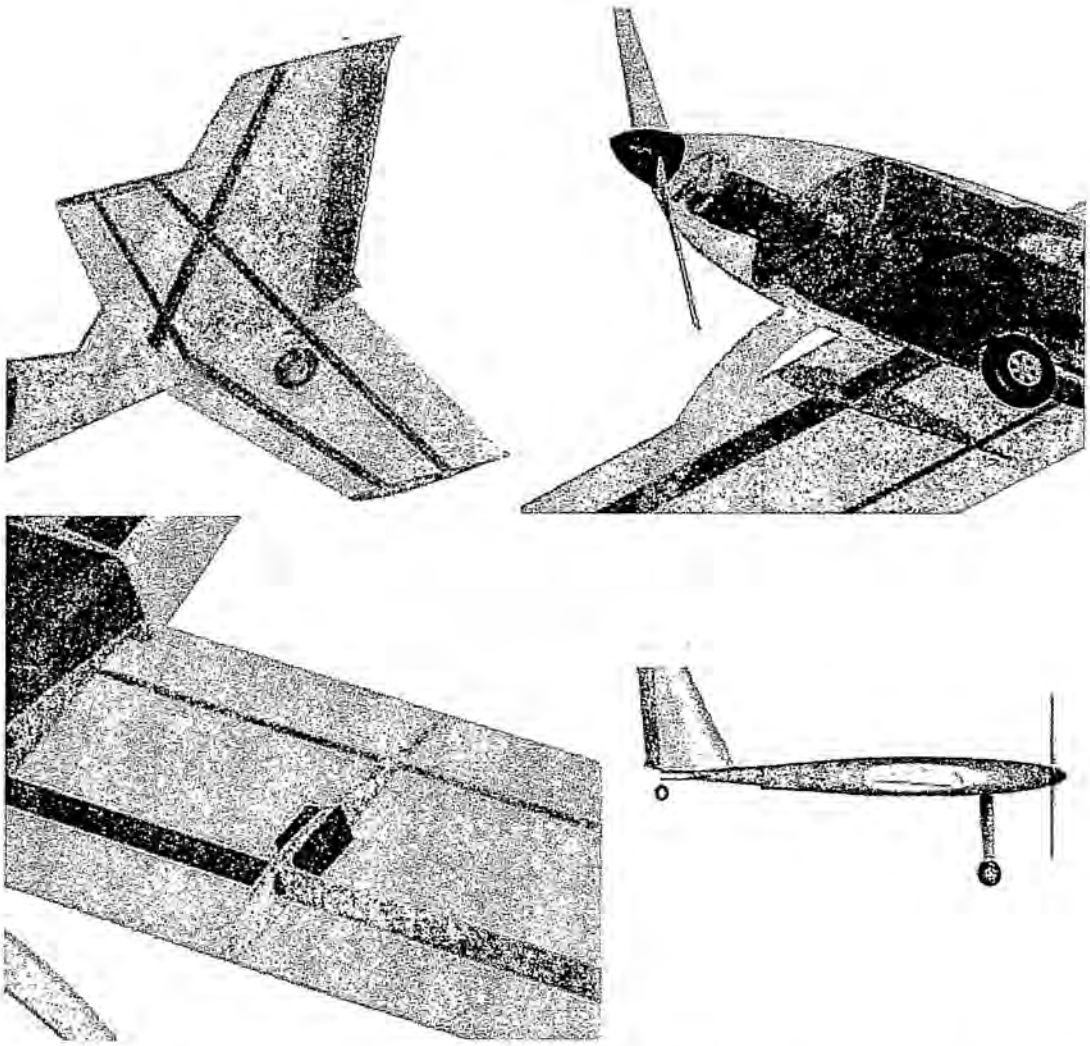


Figure 5.2: Major Structural Structure s (Clockwise from Top Left: Empennage, Nose, Wing, and Side View)

Several bulkheads will reinforce the fuselage skins. The nose detail in Figure 5.2 above shows the forward bulkhead. Note that the front reservoir wall is constrained by the bulkhead, allowing the bulkhead to double as structure and a mounting surface. The speed controller is also attached to this bulkhead. Another bulkhead will be used at the rear surface of the tank, coinciding with the trailing edge of the wings, which serves configuration purposes as well as structural purposes. The servo which controls the payload system will be affixed to the aft side of this bulkhead. The final bulkhead will be placed just aft of the boom opening, protecting the rear internal section of the fuselage. Cutting an opening for the boom in the fuselage dramatically reduces stiffness in that length of the plane. The use of this and the other bulkheads will significantly stiffen the aircraft, and provide torsional rigidity. A firewall bulkhead will also be used to affix the propulsion system at the nose of the plane, but this contributes minimally to the overall stiffness and torsional rigidity of the aircraft.

The empennage structure, Figure 5.2, is similar to that of the other airfoil components of the aircraft. Ribs will be used at the root and closeout of each tail (with the exception of the vertical tail). Two shear webs will be used in each of the components. The vertical tail will also feature a shear web at quarter chord. The elevators and the ailerons will be skin hinged, again using the rear shear web to close the section. The rudder will be hinged on an aluminum torque rod, which will also control the tail dragger wheel.

The landing gear will be constructed via vacuum forming a composite sandwich structure to a tooled mold. The final dimensions of the landing gear are 22" wide at the wheelbase, with a 4" radius in each corner of the bow to reduce stress concentrations and produce a natural shock absorbing effect. The composite sandwich lay-up will consist of 6 layers of 3 ounce fiberglass on each side of a 1/8" balsa core, shaped to a symmetric airfoil. The glass orientations on each side of the core will be $[0,\text{Carbon tape},0,45,0,45,0]_s$. The carbon tape provides additional stiffness with minimal weight addition.

5.1.4 Payload System

A diagram of the payload mechanism can be seen below in the drawing package. The payload mechanism centers on a pinch-type valve. A length of surgical tubing will be pinched between a small, L-shaped aluminum plate (0.050" thick) and a strengthened, elongated portion of the boom. The plate is affixed to two small balsa support blocks mounted on the lower internal skin of the fuselage. These same blocks support a 0.125" aluminum pin on which the boom will pivot. The mechanism will be actuated by a single rearward servo, which will connect to the boom using a simple two-rod linkage. Because the boom pivots more than 90 degrees for stowing purposes, a good seal in the tubing can be assured.

The reservoir is a softened rectangular tank, approximately 13.75" in length by 7.75" in width by 3" in height. The tank was shaped to conform exactly to the interior volume of the fuselage, with the exception of the recess where the boom retracts. The tank is 250 cubic inches in volume, which contains exactly 4 liters of payload and the volume of the incorporated baffling system. The baffling system will consist of 2 baffles which will completely separate the tank into 3 compartments. Each baffle will have a one-way flapper, only allowing payload flow towards the C.G. in the event of flight attitude changes or rough air during payload evacuation.

The ground loading system for the payload consists of a unit that is capable of discharging 2 two liter bottles through two separate nozzles into the reservoir of the aircraft. The system will utilize a pump for each 2 liter bottle to aid in discharging the water. This entire unit will fit into a back pack structure that one member of the ground crew will wear. The water evacuation should take approximately 4 seconds. The backpack will consist of two 1100 gallon/minute pumps, with a special activation switch and a 12-volt flight pack to power the system.

The final structures concept was the fit-in-box criteria specified by the contest rules. One selection of the drawing package illustrates how the aircraft will be disassembled and fit inside the box. The main wing will be detached, along with the landing gear, propeller and nose cone. The empennage structure will be able to stay intact while inside the box.

5.2 Final RAC Calculation

The final predicted RAC of the contest aircraft is presented in Table 5.2.

	Multiplier	Man Hours	Value	Hours	Cost (k$)
Manufacturers Empty Weight	$300				
Aircraft Weight			7.5 lbs		2.25
Rated Engine Power	$1,500				
Battery Weight			1.24 lbs		1.86
Manufacturing Cost	$20				
Wing Area		10 hrs/ft^2	5.56 ft^2	55.6	
Flaperons		5 hrs	1.5	7.5	
Fuselage		20 hrs/ft^3	0.89 ft^3	17.8	
Vertical Tail		10 hrs/surface	1	10	
Horizontal Tail		10 hrs/surface	1	10	
Servos		5 hrs/servo	5	25	
Total MFHR				125.9	2.52
Total RAC					**6.63**

Table 5.2: Rated Aircraft Cost for the Final Aircraft Configuration

5.2.2 Drawing Package

The next five pages of the document present the drawing package. Included in this package is a 3-view of the aircraft, component placement, payload system detail, box-fit detail, and an exploded view of all components.

6.0 Manufacturing Plan

In this section, manufacturing processes explored and used are discussed. In addition, analytical methods were used to present cost and skill required. A final construction timeline is presented.

6.1 Processes Investigated and FOM Screening Process

As previously discussed, all major structural components will be of monocoque construction. Therefore, only processes available to achieve monocoque construction are considered, and the strengths and weaknesses of each discussed.

- Lost Foam Mold: The lost foam method is the easiest of the monocoque methods, because it does not require extensive tooling construction. Once a foam cutout was constructed of each piece, it would be covered in fiberglass, allowed to cure, and then much of the internal foam would be removed. The foam would be removed by cutting tools or by cleaning agents which break down the foam structure, but do not harm the fiberglass. This provides an extremely lightweight structure, but generally lacks the required structural strength unless a significant amount of the foam is left undisturbed.
- Male Molds: Building the aircraft components from a series of male molds was one option investigated. To do so, first, an accurate outer image of each part must be constructed and finished to be used as tooling. This requires either skill in woodworking or a reliable way to harden the outer surface of a foam mold that was cut with a CNC foam cutter. The next step would be to lay up the monocoque skins on each side of the mold in either a dry or wet lay-up, cure as necessary, and bond them together. This would provide sufficient strength, but the best surface of the pieces would be on the inside of the structure, requiring finish work on the outside of the aircraft.
- Female Molds: Building based off of female molds is the most labor intensive method, frontloading the construction schedule with both of the tasks required of foam filled construction and mold construction. Cutouts must be made of the aircraft components, usually from a CNC foam machine. They must be prepared by coating them with fiberglass then sanding and filling the surface pores as necessary. They are then used to construct female tools by coating them with a thick resin-based coat and a thick lay-up of fiberglass to provide dimensionally stable molds. The advantage to this method is it provides finished parts that have the controlled surface on the outside of the component and, if the mold is correctly made, the part has the desired surface.

The following list presents the figures of merit that are used to create the weighted decision matrix in Table 6.1.

- Weight: RAC calculations dictate that any structural design chosen must be lightweight in order to maximize scoring potential. This decision was largely dependent on which method could produce the lightest aircraft structure.

- Strength: The strength of the finished part must be sufficient, because once they are bonded it will be difficult to strengthen or stiffen them additionally.
- Precision: The outer surface sizing and finish must be on the level of a commercially produced aircraft in order to ensure correct tolerances and reduce drag as much as possible.
- Repair Process: The construction method must be either easily repaired or easily repeated, in order to minimize downtime during the flight testing phase.

Figure of Merit	Weighting Factor	Lost Foam	Male Mold	Female Mold
Weight	0.35	3	2	2
Strength	0.25	1	3	3
Precision	0.2	2	2	3
Repair	0.2	2	2	2
Total	1	2.1	2.25	2.45

Table 6.1: Construction Method Weighted Decision Matrix

Using the female molded method of construction will assist in meeting all of the goals of the airplane design. Female molds create a lightweight, strong aircraft which will be easy to repair by creating surplus parts, in addition to having a precise surface. Additionally, it allows the final plane to be constructed quickly. Once the prototype testing is complete, several sets of components can be made for the final plane. Only the best and lightest will be used in competition.

6.2 Component Manufacturing

Detailed descriptions of the process implemented to construct the fuselage, empennage, wing, and landing gear are covered.

6.2.1 Fuselage and Empennage

The fuselage will be constructed of the same skin sandwich panels as the rest of the aircraft. A foam plug of the fuselage was created by first cutting the foam in two dimensions, and then hand sanding to achieve the final shape. After the foam was covered with fiberglass, the horizontal tail was mounted into the correct position on the tail section of the fuselage. This provided several advantages, such as reduction of necessary internal structure, as well as allowing the incidence of the tail to be perfectly fixed in any fuselage piece constructed. The plug was mounted into a parting board cut from plexi-glass, and then coated with a thick gel coating to provide a smooth mold surface. After allowing an appropriate cure time, the gel coat was stiffened with a protective fiberglass shell. This shell also doubled as a parting surface for creating the second mold half. Once the second mold half was created, the parts were allowed to cure for an appropriate time, and then the mold was pulled apart and checked for imperfections. The mold was dimensionally correct, as well as sufficiently stiff to create a non-warped aircraft part.

After the mold was fully cured, it was seasoned with repeated application of tool wax to allow for the easy release of the finished wing. Each half of the mold was laid-up independently, with one layer of fiberglass against the mold surface, then a core of balsa wood, followed by an internal fiberglass skin to complete the sandwich structure. The internal structure of ribs and spars were then bonded to one of the skins. Next, the final step was to lay a bead of structural epoxy around the edge of one half, then align the pieces perfectly and bolt the molds together for the final bonding process. The vertical empennage surface was created in a separate, smaller mold in order to allow for the rudder to be cut from the molded piece. Once cut, the rudder was hinged by a torque rod to allow for rudder actuation. The torque rod lined up with the tail wheel such that the tail wheel could be actuated simultaneously with the rudder.

6.2.2 Wing

The wing mold was constructed using the same steps as described for the fuselage. Once the mold was complete, a sandwich lay-up of fiberglass and balsa core was used for the skins in order to minimize weight. A balsa core was used to provide the necessary stiffness by increasing the moment of inertia of the material through spreading the two skins farther apart. The weight of the fiberglass used was determined by the loading profile of the wing, which is ideally elliptical. At the highest wing-load regions from the root to the end of the spar, a lay-up of two ounce cloth on each side of a 1/16" balsa core was used. From the end of the spar to 75% of the half span, a lay-up of two ounce cloth on the outside and 1 ounce cloth on the inside was used. From this lay-up to the wing tip, one ounce cloth was used on both sides of the core. A full carry-through 5/8" outer diameter carbon tube was used as the spar of the wing, plugging through the fuselage into each wing at the quarter chord. A nylon bolt was used at 75% chord to hold the wing in place and keep the wing from rotating. Balsa shear webs were used in the span direction to prevent wing deformation. Two ribs were used to anchor the spar with one additional intermediate rib between the spar and the closeout rib. The aileron servo was placed on a mounting rib at half span; this placement was chosen to prevent control surface twisting. All of the internal components were mounted prior to the bonding of the top and bottom wing surfaces.

6.2.3 Landing Gear

The landing gear was constructed using a conventional lay up technique. A male mold was constructed from foam using construction drawings and the CNC foam cutter. This mold was then glassed in order to provide a good surface for part. The landing gear was laid-up according to the strength testing which occurred in the preliminary design phase, which led to an eight layer fiberglass gear piece with mono-carbon fiber tape in the outer layer for additional stiffness. The corner radius on the bow is approximately seven inches. This assembly was laid-up wet and then vacuum bagged to ensure that the part was shaped correctly and minimize de-lamination failure.

6.3 Analytic Methods

Analytic methods were employed to aide in the timely and effective construction of the aircraft. These included cost analysis, required skills matrix, and a construction timeline.

6.3.1 Cost

After choosing a construction method and completing the aircraft design, manufacturing and tooling costs for aircraft construction were determined. Propulsion equipment was found to be the major cost, with tooling and construction materials following closely behind. An itemized breakdown of the final construction cost is shown in Table 6.2.

AIRCRAFT CONSTRUCTION			
ITEM	**AMOUNT**	**UNIT COST**	**TOTAL COST**
Construction			
Foam	5	$16/Sheet	$80.00
Tooling/Molded Parts	40	$20/ft2	$800.00
Gear	1	$20	$20.00
Flight Control			
Servos	5	$85	$425.00
Receiver	1	$160	$160.00
Battery	1	$20	$20.00
Miscellaneous	N/A	$50	$50.00
Propulsion System			
Motor	1	$385	$385.00
Batteries	40	$10/Cell	$400.00
Speed Controller	1	$60	$60.00
Miscellaneous	N/A	$50	$50.00
Payload System			
On Board System	1	$40	$40.00
Ground System	1	$125	$125.00
TOTAL			$2,615.00

Table 6.2: Estimated Aircraft Manufacturing Cost

6.3.2 Skills

To effectively use structures personnel during aircraft construction, a skills matrix was developed to assign various tasks. This was necessary to ensure personnel had the required competencies to perform a given task, as a poorly constructed component could be detrimental. In the matrix presented in Table 6.3 the columns represent the required skills, while the rows represent the component being constructed. For this matrix, a task requiring little or no skill was given a score of zero, a task requiring moderate skill was scored one, and a task requiring much skill was scored two. Using the skills matrix, structures personnel were assigned to tasks within their competency.

6.3.3 Schedule

To ensure a timely delivery of the prototype and contest aircraft, a comprehensive construction schedule had to be devised. This schedule, presented in Figure 6.1 was built around predefined milestones and historical time estimates.

	Core Cutting	Fiberglass Layup	Mold/Skin Construction	Control Equipment	Wiring	Modeling
Fuselage/ Tail	1	2	2	2	2	2
Wing	1	2	2	2	2	2
Gear	1	1	1	0	0	1
Payload Systems	1	2	2	2	2	2
Propulsion System	0	0	0	1	2	1

Table 6.3: Required Skills Matrix

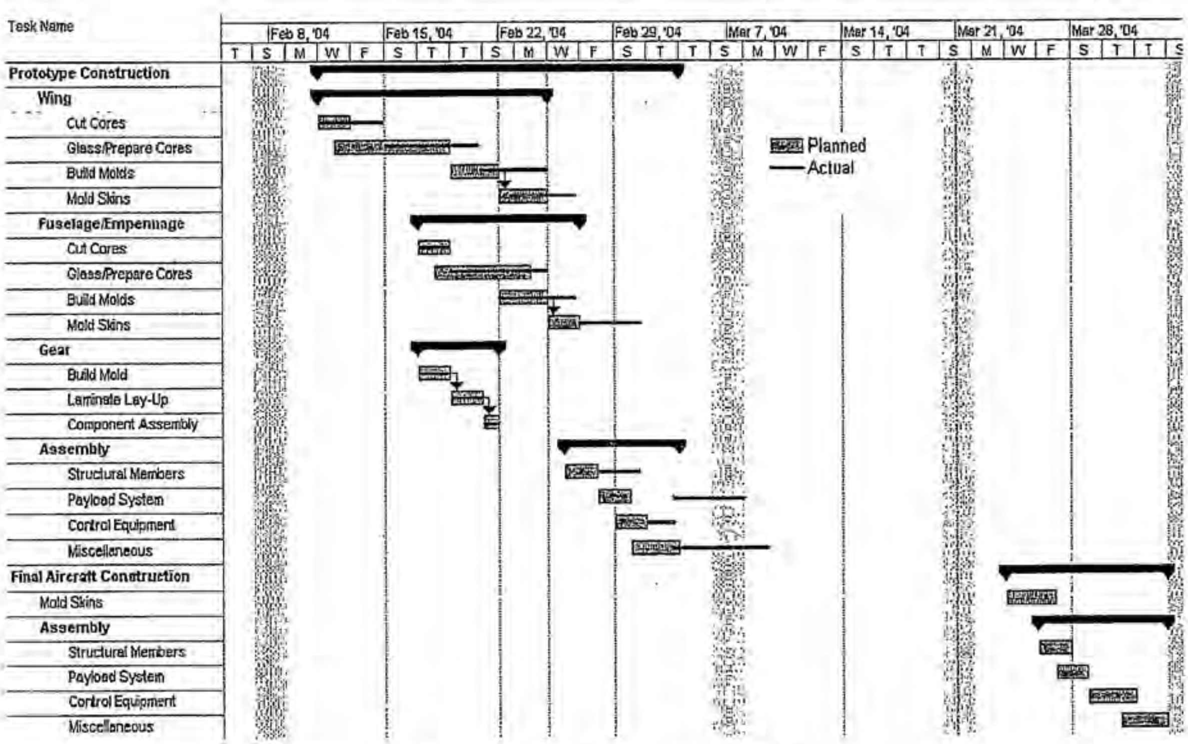


Figure 6.1: Construction Time-line (planned and actual)

7.0 Testing Plan

To validate analytical data and calculations; testing becomes the most important aspect of developing the aircraft's design. New designs can be instantaneously confirmed or invalidated. Due to the importance of testing, a well thought out testing plan must be developed.

7.1 Test Objective and Schedule

Goals were set in order to optimize component design in the payload, propulsion, aircraft structure, and flight testing areas. A detailed schedule was utilized to aide in an effective test procedure. Table 7.1 lists tests preformed, test objectives, and test dates.

Test	Objective	Dates
Payload Stabilization Test	Investigated the effects of dynamic and static movement dampening	1/12 – 2/2
Payload Evacuation Test	Gathered data in order to eject payload as fast as possible	1/12 – 2/16
Motor Test	Collected performance data for motor	1/26 – 2/23
Mold Procedure Test	Allowed for the best construction procedures to be found and practiced	1/12 – 2/9
Wing Structure Test	Determined that wing structure could be undamaged after being lifted by its wingtips	3/22
Final Motor/Battery Test	Optimized the motor and batteries to specific mission requirements	2/23 – 3/3
Ground Crew Test	Optimized for the fastest pit times and allowed the ground crew practice time	3/29 – 4/2
Flight Tests	Fined-tuned aircraft performance and determined usability of stability predictions	3/29 – 4/16

Table 7.1: Test Objectives and Schedule

7.2 Flight Testing Checklists

A thorough and useful execution of flight testing can be accomplished by creating a pre-flight checklist and a flight testing checklist. The pre-flight checklist, seen in Table 7.2 was designed to ensure the plane was airworthy. Structural fatigue or assembly carelessness should be prevented with the use of the pre-flight checklist. The flight test checklist, seen in Table 7.3, was designed to maximize the benefits of flight testing. Specifically in the areas of takeoff, stability and control characteristics, aircraft performance, and competition mission practice.

Ground and takeoff testing will demonstrate aircraft controllability on the ground through rotation velocities and ensure the aircraft will be able to get off the ground in the necessary distance. Different propeller and battery configurations will be tested to ensure the optimal takeoff scenario is reached.

Stability and control characteristics will first be tested with gentle maneuvers in the air. The outcomes of these maneuvers will be a good indication whether or not the analytical data is correct. Next, high speed maneuvers, in addition to stall and tight circle turning, will be used to assess the maneuvering characteristics of the aircraft. These characteristics will be used to fine tune the aircraft's control system. Such things as vertical and horizontal tail volume, control surface sizing, and control surface deflection will be considered.

Pre-Flight Checklist		
Item	**Comments**	**Complete**
Weight and Balance (CG)	CG within operational range	
Motor Mount	Mounting screws tightened, firewall secure	
Control Surfaces		
-Linkages Secure	No endplay	
-Proper Deflection	Check trim points and throw distance	
-Hinge Integrity	Pull on hinge, check for tears	
Landing Gear	Bolts/wheels secure, no delaminations on bow	
Payload System		
-Boom Operational	Full range of motion possible	
-No Water Leaks	Valve, vents, and tank seam	
Structure Sound (Tip Test)	Performed at full gross weight	
Radio Range Test (Including Fail Safe)	Collapsed antennae, motor on and off	

Table 7.2: Pre-Flight Checklist (Used Before Each Flight)

Flight Testing			
Item	**Date**	**Comments or Concerns**	**Results**
Taxi/Ground Handling	3/29		
-Aircraft Tracks Straight			
-Sufficient Maneuvering Control			
First Flight (No Payload)	4/1		
-Sufficient Control in all Axes			
-Aircraft in Control Through all Flight Regimes			
-Stall Characteristics Verified			
-Adequate Control at Takeoff and Landing			
-Boom Operational (Adverse Effects)			
Second Flight (Full Gross Weight)	4/2		
-Repeat First Flight Checklist			
-Adverse CG Effects when Payload Expelled			
-All Water Ejected (Nothing Left in Tank)			
Mission Profile Practice	4/5-4/20		
-Takeoff in Less Than 150 Feet			
-Water Dump Time (FBM only)			
-Landing Distance			
-Pit Time			
-Complete Mission Time			
-Overall Aircraft Control			

Table 7.3: Flight Testing Checklist

Aircraft performance testing will determine the approximate details of the flight. Velocities, takeoff distance, and turning radius will be found and compared to the predicted values. Wingspan, propeller selection, and battery pack configurations will be re-optimized in this stage. In addition to aircraft performance, the payload mechanism will be repeatedly tested and fine tuned for reliability.

With the competition plane optimized, practice for both the pilot and ground crew will be done. Handling and piloting skills are to be refined and perfected through multiple mission simulations.

7.3 Testing Results and Lessons Learned

The series of tests conducted proved to be both, vital and beneficial, to the aircraft's design and operation. Test results were used to refine and strengthen all aspects of the design. Table 7.4 presents a synopsis of results and lessons learned from each of the tests carried out.

Test	Results and Lessons Learned
Payload Stabilization Test	Baffling greatly reduced dynamic water shift, while one way gates helped reduce the static shift. Payload handling is still a very important design aspect.
Payload Evacuation Test	It was found that it will take approximately 12-15 sec. to expel the water. This should be more than sufficient, no further changes necessary.
Motor Test	Graupner motor at least 20% more efficient than Astro throughout flight regime. 16 battery cells more than sufficient to complete FBM and FM.
Mold Procedure Test	Test molds allowed structures group to refine the molding process. This testing allowed the group to produce nearly flawless, dimensionally sound final molds.
Wing Structure Test	Composite laminate used is sufficient to pass wing tip tests, drove down design of laminate to a total wing weight of 1.5 lbs.
Final Motor and Battery Test	A 20 X 13 and 16 X 12 propeller was chosen for the FMB and FM, respectively. Gear ratios of 1.72 and 1.48 were chosen for the FMB and FM, respectively.
Ground Crew Test	The aircraft can be serviced and returned to flight in approximately 15 sec. Aircraft handling issues need to be investigated to reduce pit time.
Flight Tests	Final aircraft testing has yet to be completed.

Table 7.4: Testing Results and Lessons Learned

References

Allen, D.H. and W. E. Haisler, *Introduction to Aerospace Structural Analysis*, John Wiley and Sons, New York, 1985.

Bertin, J. J. and M. L. Smith, *Aerodynamics for Engineers*, 3rd Edition, Prentice Hall, Upper Saddle River, New Jersey, 1998.

Boucher, R. J., *The Electric Motor Handbook*, Astro Flight, 2003.

Dieter, G.E, *Engineering Design*, 3rd Edition, McGraw-Hill, Boston, 2000.

Donald, D., *The Complete Encyclopedia of World Aircraft*, Barnes and Noble Books, New York, 1997.

Jensen, D. T., *A Drag Prediction Methodology for Low Reynolds Number Flight Vehicles*, Notre Dame, Indiana, 1990.

Munson, B. R., D. F. Young, and T. H. Okiishi, *Fundamentals of Fluid Mechanics*, 3rd Edition, John Wiley and Sons, New York, 1998.

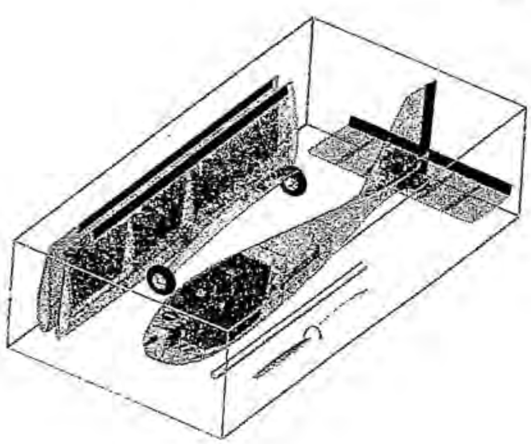
Nelson, R. C., *Flight Stability and Automatic Control*, 2nd Edition, McGraw-Hill, Boston, 1998.

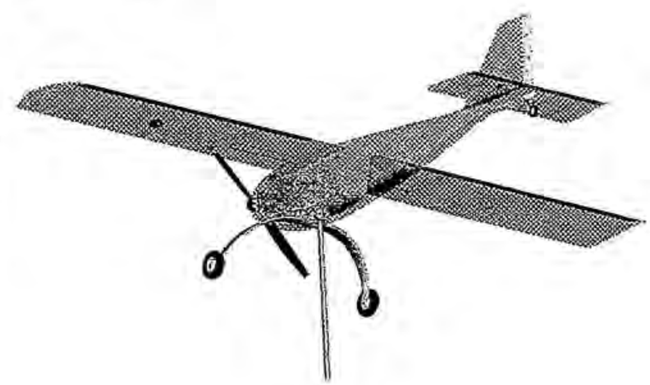
Raymer, D. P., *Aircraft Design: A Conceptual Approach*, 3rd Edition, AIAA, Reston, VA, 1999.

Shigley, J. E. and C. R. Mischke. *Mechanical Engineering Design*, 6th Edition, McGraw-Hill, Boston, 2001.

Cessna / ONR Student Design Build Fly Competition

2003-2004





SPRAY

Oklahoma State University
Orange Team - 2004 DBF

1 Executive Summary

The following section briefly describes each aspect of the report, and lists the major design issues resolved in each section.

1.1 Conceptual Design

In order to guide the conceptual design process, an initial score sensitivity study was performed to determine the effect of design parameters on the scoring potential of a given aircraft configuration, and the sensitivity of the Total Score and Rated Aircraft Cost to parameters such as aircraft empty weight, propulsion battery weight, payload weight, and Fire Bomber and Ferry mission times. The results showed that a competitive airplane must carry the full weight of payload, and that the Ferry Mission was almost completely negligible in comparison to the Fire Bomber mission in their influence on Total Score. Configurations considered during the conceptual design phase included several non-conventional designs including tailless and hybrid designs. Most of the non-conventional designs were considered because of the possibility they would be able to achieve a lower Rated Aircraft Cost. Conceptual design was also performed for payload handling configurations and construction methods. The final configuration chosen was an aircraft of conventional layout capable of carrying the maximum allowable payload weight. The aircraft was to be equipped with a retractable boom to assist in emptying the payload, and was constructed using a fully monocoque composite sandwich construction.

1.2 Preliminary Design

Computer routines were utilized by both the aero and propulsion groups to perform optimization of design parameters and component capacity and configuration. Major design parameters were varied in the code to determine the sensitivity of total score to each of the parameters, and their optimum ranges. Experimental data was gathered by both the propulsion group and the structures group to aid in the design process. Structural experiments were performed to gather information allowing more optimized composite structures. Tests were performed on propulsion system components to validate the accuracy of the computer models used to predict aircraft performance. Propulsion testing results informed the design team in their selection of system components. Payload concepts were integrated into the aircraft configuration. Initial weight and balance models were refined to allow more accurate analysis of design scoring potential.

1.3 Detail Design

During the Detail Design phase, system architecture and component selection, placement, and mounting methods were finalized, and structural details were decided. Tables are included which describe the aircraft configuration, size and placement of major components, and major system architectures. Estimates of aircraft handling characteristics and performance are also provided, along with a projected contest score.

1.4 Manufacturing Plan and Processes

Several manufacturing processes were considered and evaluated in trying to determine the optimum way to achieve the desired aircraft construction scheme. Processes were evaluated in terms of the quality control obtainable, the cost of implementation as it pertained to both money and scheduling issues, and the overall manufacturing capability they provided. The manufacturing section of the report describes the method by which manufacturing processes were chosen independently for each of the major aircraft sub-assemblies

1.5 Testing Summary

The testing summary describes the areas in which testing data has been gathered and the ways in which it has been used. Lessons learned and ways in which the testing data has been utilized are also included. Plans for future testing of both individual components and aircraft flight testing are described.

1.6 Conceptual Design

An initial score analysis was performed to determine the effect of design parameters on the scoring potential of a given aircraft configuration, and the sensitivity of the Total Score to Rated Aircraft Cost and Flight Score. The results were used to define the design mission objectives and establish the Figures of Merit by which aircraft configuration choices would be made. Multiple configurations were evaluated as to their scoring potential before an optimum design was selected. Conceptual design was also performed for payload handling configurations and construction methods.

1.7 Preliminary Design

A computer routine was utilized to perform optimization of design parameter sizing and airfoil selection. Payload concepts were integrated into the aircraft configuration. Initial weight and balance models were refined to allow more accurate analysis of design scoring potential.

1.8 Detail Design

During the Detail Design phase, system architecture and component selection, placement, and mounting methods were finalized, and structural details were decided.

2 Management Summary

The following section describes the composition of the Orange Team and the manner in which responsibilities were divided among personnel. Project planning, scheduling, and organization information is also included in this section, along with a milestone chart showing the planned and actual completion dates for major tasks and events.

2.1 *Design Team Composition and Organization*

The Orange Team was comprised of undergraduate students ranging in age and experience from freshmen to seniors in their final semester of undergraduate coursework. Team members drawn from both the Aerospace and Mechanical Engineering disciplines ensured that the team possessed the broad range of design, optimization, and construction capabilities necessary to field a competitive entry in the contest. Key members of the Design Team were divided into three major technical groups:

- Aerodynamics/Stability and Control (Aero/S&C)
- Propulsion
- Structures

Each group was comprised of a Lead Engineer and two or more Technical Specialists. The Structures Group was further divided into three sub-groups specializing in various aspects of the aircraft's physical construction. The Structures sub-groups consisted of the Wing and Empennage group, the Fuselage and Landing Gear Group, and the Payload Group. The Lead Engineers of each group were directly responsible for operations within their technical area, as well as coordinating the overall project with the other Lead Engineers and the Chief Engineer. The Chief Engineer was responsible for managing and directing the entire project as well as maintaining communication between the technical groups. Figure 2.1 shows the hierarchal organization of key Orange Team personnel, including group assignment and sub-group specialization, where appropriate.

The Aerodynamics/Stability and Control group was responsible for the development of models used to predict scoring potential of various designs and design parameter values. Concurrent to their primary role in developing the aircraft configuration, the Aerodynamics group conducted a sensitivity analysis to determine the degree to which score was affected by each design parameter. The results of the analysis were used to establish initial sizing estimates useful in evaluating the practicality of each concept. After the design concept was chosen, the Aerodynamics group was responsible for analysis and optimization of the planform, including airfoil selection, determination of wing span and area, empennage sizing and geometry, and control surface design.

The Structures Group was responsible for development and integration of both the physical structure and systems of the aircraft. Since the design of the aircraft was largely dependent on how it integrated with the payload system, one of the first tasks of the group was to develop a system for handling the payload. The construction method utilized to build the aircraft was also determined by the structures group. The group was also responsible for necessary testing and research to determine the best method for each portion of the aircraft structure, as well as designing the flight control and payload actuation systems.

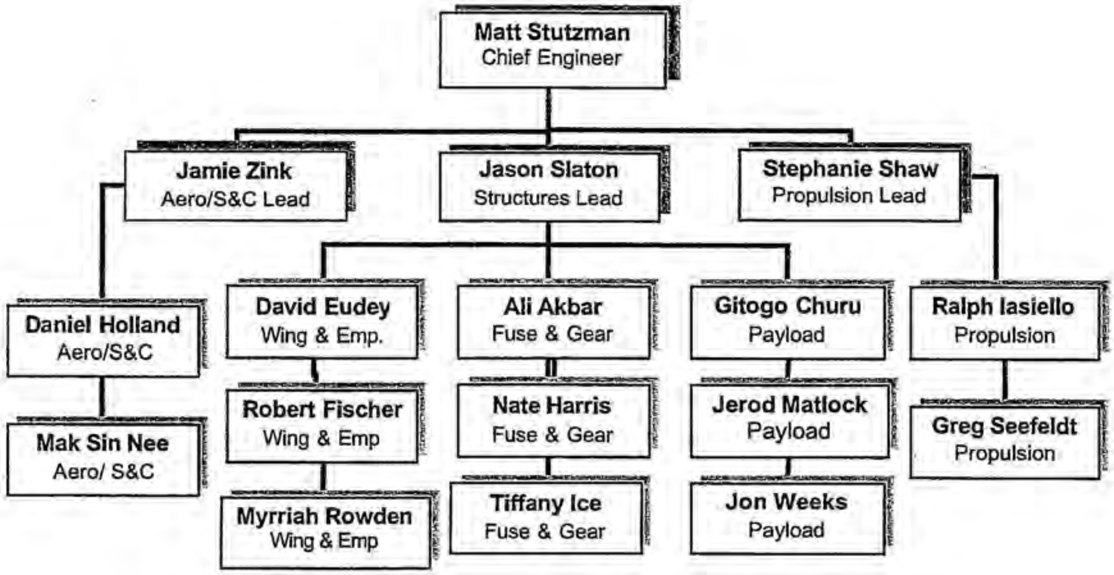


Figure 2.1: Orange Team Organizational Structure

The Propulsion Group focused on determining the best battery/propeller/gearbox combinations for a variety of payload and weather conditions, and developing a scheme in order to obtain maximum efficiency from the available energy in every phase of flight. Much of the Propulsion team's time was spent optimizing the battery pack so that optimal power could be obtained with the lowest weight possible.

2.2 Schedule and Planning Summary

Completing and fielding an entry in the Design/Build/Fly competition necessitated a tight schedule for design, construction, and testing. To ensure that the required tasks were completed in a timely manner, a waterfall chart (Figure 2.2) was created containing projected times for the various aspects of design and manufacturing. The use of this chart along with hard deadlines where required to ensure that the project was completed on time. The following chart shows the major tasks the project was divided into and their projected time frames.

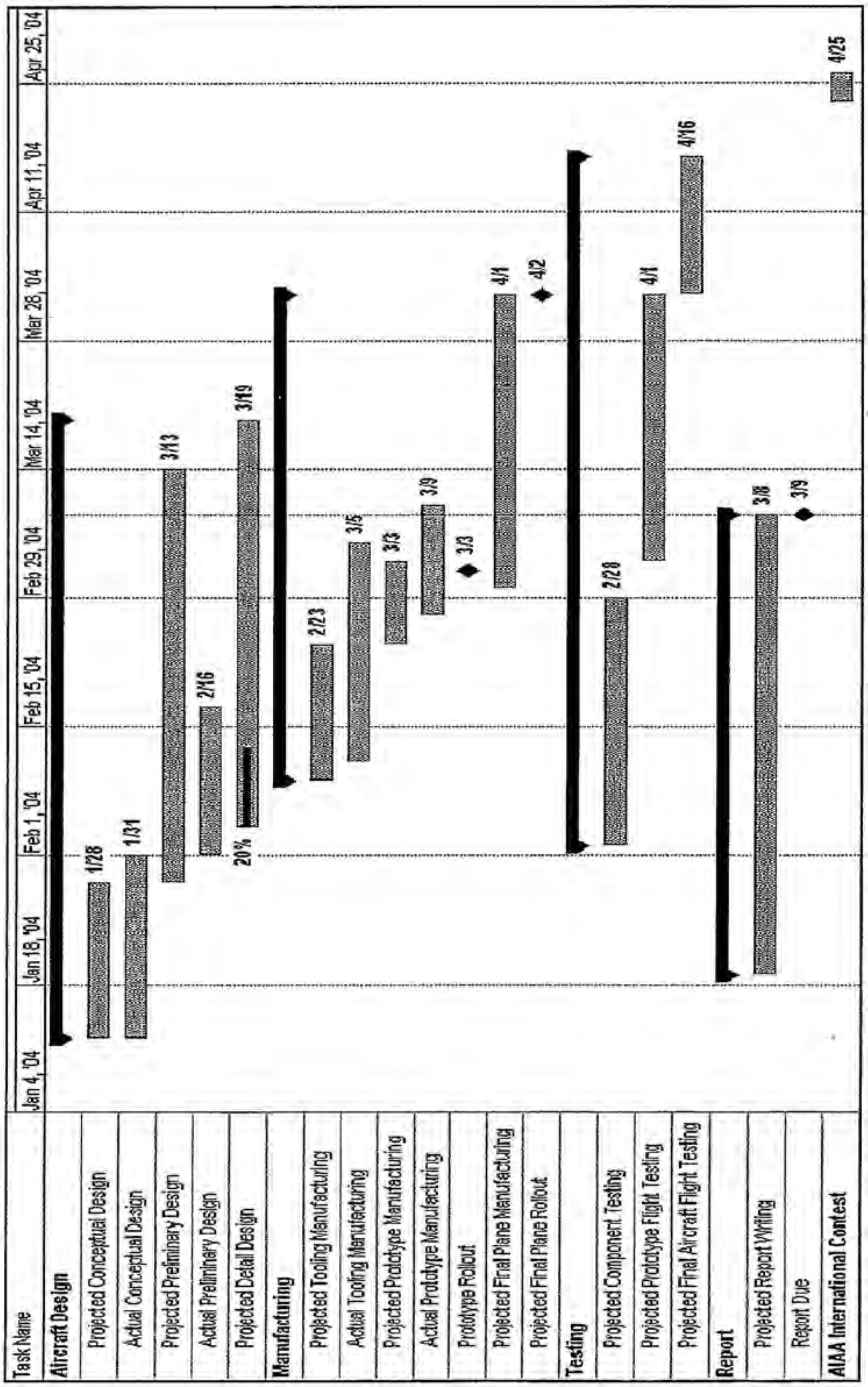


Figure 2.2 : Waterfall Chart with Major Milestones

3 Conceptual Design

The initial step in the design process was to determine the design objectives and the mission for which the aircraft would be optimized. Based on this information, aircraft and system concepts with the possibility of high scoring potential were generated and evaluated. The following section describes the methods by which the design objectives were chosen, the aircraft and component configurations considered, and the resulting overall configuration.

3.1 Mission Requirements

- Secured Storage for Aircraft

The completed aircraft must fit into a storage box that has an interior dimensions of 2-ft wide by 1-ft high by 4-ft long. The aircraft is allowed to have built in folding functions or plug-in joints in order to provide easy access for the entire parts to fit inside the storage box.

- Take-off Limitation

The maximum allowable distance of the runway for take-off for the entire competition is 150 ft. The wheels of the plane must lift off the runway at or before 150 ft from the starting line. If the take-off restriction rule does not meet, flight mission score will result as zero score.

- Fire Bomber Flight Mission

The aircraft is expected to start with empty payload at the beginning of the mission. When the mission started, the teams will load the aircraft with maximum of four 2-liter plastic soda bottles of water. Gravity or pumped loading is allowed to assist the loading as long as it does not pressurize the soda bottles. A time penalty of 1 minute will incur for excessive water spillage during loading.

The aircraft will take-off after loaded with full payload. Once the aircraft reaches the cruise altitude, it will make a U-turn towards the downwind stream. The downwind leg is the only period when the water is allowed to dump from the aircraft payload. The maximum diameter of the dump orifice is limited to 0.5 inch. Therefore, the aircraft needs to slow down in order to have sufficient time to empty the water payload completely before reaching the downwind turn. Late or early dumping will incur a time penalty of 3 minutes per occurrence which including the inadvertent release of water during hard landing. During the cruising period between the upwind and the downwind, the aircraft is also required to make a full 360° turn in the opposite direction of the base. At the downwind turn, the aircraft will make another U-turn and land on the base. Another 3 minutes of time penalty will incur if the water payload of the aircraft is not empty on landing. The aircraft will be reloaded and repeat the similar mission for another lap. If the second lap is not completed, 3 minutes of time penalty will be incurred. The incomplete lap will also count if the 360° turn is not completed. This mission has the difficulty factor of 2.

- Ferry Flight Mission

There will be no water payload for this mission. The aircraft must take-off with no payload and completed all 4 laps before landing on the base. The aircraft is also requiring making a 360° turn for every lap in the opposite direction of the base between the upwind and the downwind period. The difficulty factor for this mission is 1.

- Cost Effectiveness

The total score of the competition is the multiplication of the written report score and the total flight score which is inversely proportionally to the Rated Aircraft Cost (RAC).

3.2 *Mission Analysis*

To design an aircraft capable of achieving the maximum score, it was necessary to know which of the design variables had the most impact on the total score. The aircraft with the minimum RAC might not be the optimum design for the competition, and one of the flight missions may contribute more significantly to the total score than the other. To guide the conceptual design process, a sensitivity analysis was performed to determine which design parameters should be emphasized and which should be relegated to secondary or tertiary importance. Based historical data, a reasonable total flight score and RAC for an aircraft capable of performing the mission were established to obtain a baseline for comparison. Major design parameters were then changed nominally from their baseline values and the total score recalculated. The resulting score trends were then plotted versus the design parameters. Figure 3.1, below, shows the sensitivity of the score to several major design parameters.

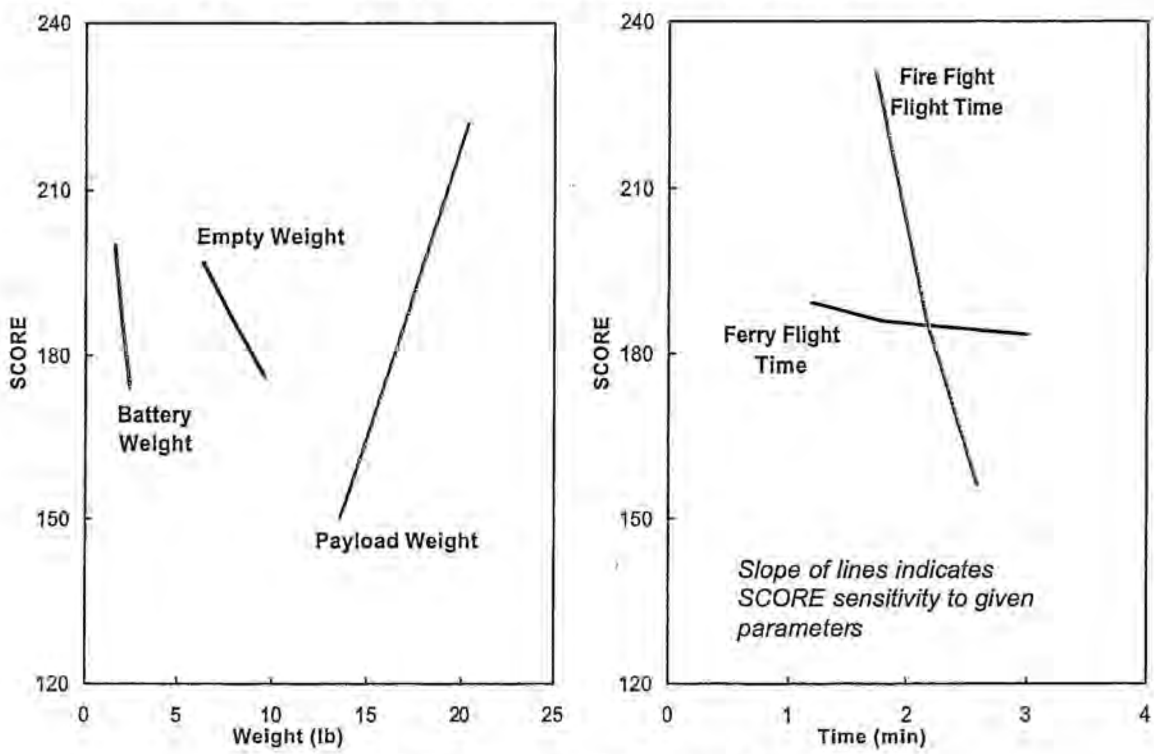


Figure 3.1: Sensitivity Analysis

Note that score has a strong inverse relationship to empty aircraft weight, payload weight and fire bomber mission time, and a strong direct relationship to weight of payload carried. The relationship between ferry mission time and score, however, was very weak. Accordingly, it was decided that the aircraft designed for the contest must carry the full amount of payload while accomplishing the fire bomber mission quickly and efficiently. The aircraft must be as small and light as possible while utilizing the smallest propulsion battery pack consistent with the required performance.

3.3 Alternative Configurations and Figures of Merit

For each aspect of the aircraft design, multiple configuration possibilities were considered. Concepts were creatively generated by thinking of ways to maximize score. All ideas were considered initially, and promising alternatives were then evaluated qualitatively and quantitatively based on the Figures of Merit established for each design aspect. The following section discusses many of the design options assessed by the team, and the final choice of each proposed design when evaluated using the previously mentioned Figures of Merit.

3.3.1 Aircraft Configuration Concepts and Figures of Merit

In order to optimize the overall aircraft design, it was necessary to establish a rubric for evaluating competing concepts. The following Figures of Merit were used to compare and quantify each configuration in terms of its scoring potential.

- Payload System Integration: The single largest contributor to the flight Score was the weight of payload carried during the fire bomber mission. A competitive aircraft must carry the maximum allowable payload and minimize the pit time between flights. Dumping a liquid payload in flight was considered to be the most significant challenge of the competition. Therefore, the payload system was the most critical aspect of the final aircraft design. As a result, an aircraft configuration that optimized the use of internal volume was required, as well as one that did not exacerbate the negative effects on flight handling caused by water slosh.
- Takeoff Performance: A takeoff distance limit of 150 ft. combined with the requirement to lift a relatively heavy payload necessitated an aircraft with a favorable combination of maximum lift coefficient and airframe weight. Inherently stable aircraft configurations with generous control powers and damping coefficients are more easily equipped with high lift devices, and are generally more desirable in this regard than lightly damped aircraft with shorter moment arms and less control authority. In addition, weather conditions at the contest locale required a design that could be reliably flown throughout a wide range of wind speeds and directions.
- Rated Aircraft Cost: Sensitivity studies performed showed that at some typical combinations of Flight Score and RAC, an increase in RAC of a given percentage corresponded to a decrease in score of an equal percentage, meaning that the sensitivity of Total Score to RAC was close to -1;. Therefore, a configuration was desired that would incur the lowest RAC consistent with maintaining the required payload and speed capabilities as well as acceptable handling qualities.

- Handling Qualities: The flight score achieved by the aircraft is dependent to a large degree on the ability of the pilot to fly the mission profile. An aircraft with good handling qualities will enable the pilot to devote more attention to accurately flying the profile and less attention to compensating for deficient behavioral tendencies. During the conceptual design phase, handling qualities characteristic to each of the concepts were discussed and factor weighted. Because of the dynamic nature of the liquid payload, an emphasis was placed on configurations that were able to maintain acceptable flying qualities despite a shift in CG location.

Several aircraft configurations were considered because they supported one or more of the Figures of Merit described above. Many of the configurations were considered based largely on a real or perceived savings in RAC. Some of the configurations that initially appeared to hold promise of significantly lowering RAC became much less attractive upon further investigation of the tradeoffs that would have to be performed. It should also be noted that conceptual design for the payload system was performed concurrently and in conjunction with the conceptual design process for the overall aircraft. Because the two processes ran on parallel courses, some of the aircraft configurations considered would not integrate well with the final payload system configuration, but they did support the payload system configurations being considered at the time. The major configurations that were considered are listed below and are shown in Figure 3.2.

- Conventional Configuration: The conventional configuration featuring a single fuselage with a forward lifting wing and aft-mounted tail forms the standard for comparison. It is simple to analyze and optimize since its characteristics are well known and documented. The conventional configuration allows the use of high lift devices due to its longer moment arms and efficient rate damping, but possibly at the expense of a higher RAC due to a longer fuselage length and requirement for a full complement of tail surfaces and servos.
- Bi-wing Conventional: A biplane offers shorter takeoff distance for a given wingspan, but at the expense of more intersection and wetted area drag, and it also incurs a significant RAC penalty. Handling qualities are similar to the conventional configuration.
- Flying Wing Types: A flying wing design might obtain an RAC savings because of its reduction in number of surfaces and shorter fuselage length. However, the wings would probably need to be swept aft to enhance pitch and yaw stability, which would negate the RAC savings. In addition, because of the lack of a horizontal tail, flaps could not be employed on a flying wing, which would limit the attainable lift coefficient and hence require a larger wing with its accompanying increase in RAC. An alternative wing configuration considered was to sweep the wings forward so that their outboard tips would be forward of the CG and could serve as elevators, allowing the use of flaps. This design was rejected early on due to the instability and lack of pitch damping that would occur as a result of the pitch control surfaces being so close to the CG.
- Tailless Lifting Body: The lifting body concept featured a conventional wing blended into a flat, wide fuselage. The aircraft had no horizontal tail, but featured a hinged pitch control surface on

the aft end of the fuselage. Similar in philosophy to the flying wing, the positive attributes of the lifting body configuration are that it would offer a possibly lower RAC due to a shorter fuselage length, along with the RAC savings inherent in a tailless design, although at the expense of handling qualities and stability.

- Inverted V-tail: A hybrid twin boom design was also considered. The design featured short booms protruding aft from the wing on which an inverted V-tail was mounted. The purpose of the configuration was to minimize RAC by decreasing fuselage length, while still achieving enough longitudinal stability and control power to employ high lift devices. It was determined, however, that in order for the V-tail to be effective in its intended function, its span would have to be greater than 25% of the overall wingspan, incurring a dual wing penalty.
- Canard: A lifting canard configuration offers the possibility of reducing the trim drag and downward lift required on a conventional design to counteract the pitching moment produced by the airfoil and the lift of the wing. With the canard, lift needed for stability from the tail (canard) would not take away from the overall lift of the airplane. This design would also possibly allow more room for water storage between the wing and canard. The advantage of a lifting trim surface, however, is somewhat negated by the fact that it is more difficult to incorporate high lift devices. This is due in part to the fact that the lift from the main wing acts further from the aircraft CG, requiring a larger trim surface to counteract the pitching moment introduced by the flaps, which reintroduces the drag that was supposedly saved. In addition, the CG location close to the rear of the aircraft requires excessively large vertical surfaces for yaw control, and both forward and aft motor locations present issues with CG and propeller ground clearance, respectively.

After qualitatively evaluating each of the proposed aircraft configurations, the designs were assigned a ranking of 1, 0, or -1 in accordance with their support for each Figure of Merit when compared with the baseline design (a conventional aircraft configuration). The Figures of Merit were assigned a weighting factor, and a decision matrix was used to determine the optimum aircraft configuration. Table 3.1 on the next page shows the weighting factors used for the Figures of Merit, and the resulting aircraft configuration.

3.3.2 Payload System Concepts and Figures of Merit

Due to the fact that the payload occupied a volume rather than a fixed shape, and also because of the challenge of gradually releasing a liquid payload in flight, payload system integration was one of the primary considerations made while choosing an aircraft configuration. The Figures of Merit selected for evaluating competing payload system concepts are shown below.

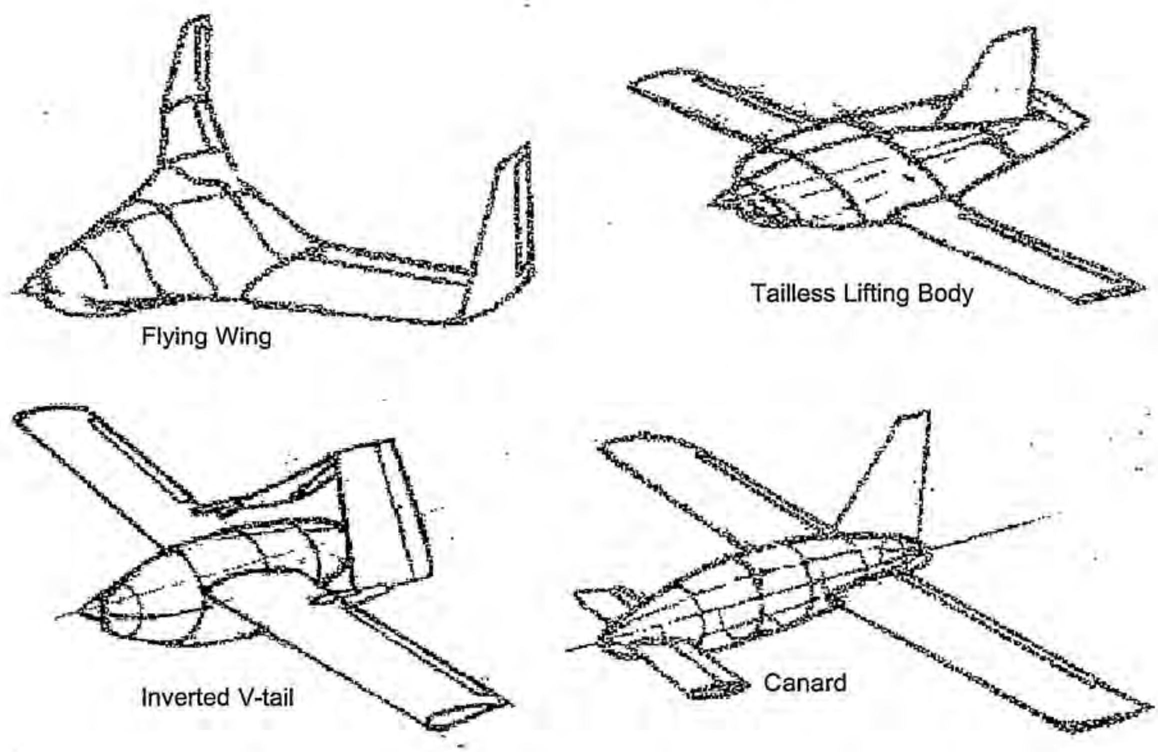


Figure 3.2: Alternative Aircraft Planform Configurations and Rating Results

Table 3.1: Aircraft Configuration Rating Results

Figures of Merit	**Weighting Factor**	**Conventional**	**Biplane**	**Flying Wing**	**Lifting Body**	**Inverted V-tail**	**Canard**
Payload system integration	*0.25*	1	1	0	1	1	1
Takeoff performance	*0.25*	1	1	-1	-1	0	0
Handling qualities	*0.20*	1	0	0	-1	0	1
Rated Aircraft Cost	*0.20*	-1	-1	1	1	0	-1
Overall Weighted Score:	-----	**0.50**	**0.30**	**-0.05**	**0.00**	**0.25**	**0.25**

- Discharge Flow Rate: A high volumetric flow rate was required to ensure that the payload was completely discharged on the downwind leg of the course, as the contest rules required. The fluid nozzle exit velocity, and hence the flow rate, is primarily a function of the pressure difference from the inside of the nozzle to the ambient air. Because of the prohibition against pressurizing the tank to assist in expelling the payload, an alternative method of increasing the pressure difference was required. The contest rules state that the dynamic pressure created by the aircraft flight velocity could be ported to the tank to increase flow rate, but a fluid dynamics analysis using the standard Bernoulli equation showed that the dynamic pressure gain was completely negligible in comparison to what could be obtained by increasing the "head" of the fluid column by even a couple of inches. The results of the Bernoulli analysis can be seen graphically in Figure 3.3, below, which shows the time required to drain a 4-liter reservoir as a function of pressure head when the tank cross sectional area is held constant. The curve showing the time required to drain is highly dependent on pressure head, but unless pressure head is very low, is not strongly dependent on dynamic pressure Each of the payload systems seriously considered incorporated some mechanism for increasing the pressure head during payload release.

- Volumetric Efficiency: The Rated Aircraft Cost function contains a fuselage volume component, calculated by the product of the fuselage frontal area and its length. It was highly desirable to utilize a payload system that minimized the frontal area required for the reservoir and release mechanism, while preserving the required fuselage volume, in order to minimize the RAC.
- Minimal CG Shift During Payload Release: Because of RAC considerations, the aircraft will be built as small as possible, which implies that the payload will be a large fraction of aircraft gross weight. Any shift in the location of the payload center of gravity could possibly cause a significant shift the total aircraft CG. A reservoir configuration that minimized the amount the CG could shift during payload release was required to preserve the longitudinal stability and pitch trim characteristics of the aircraft.
- Manufacturability: In order to minimize cost and manufacturing time and labor, a design simple in both construction and operation is strongly preferred over a complex one.
- Speed and Ease of Loading: As was shown in the sensitivity analysis, mission time for the Fire Bomber task is one of the single largest determinants of total score. A major contributing factor to mission time will be pit time between sorties. The payload system in the aircraft must be able to be loaded easily and efficiently in order to reduce pit time to the absolute minimum.

Multiple payload system configurations (Figure 3.4) were considered in the conceptual design phase, each with a mechanism for increasing the pressure head of the tank over the duration of payload release. Among the schemes considered was a long, narrow reservoir that pivoted about the aircraft CG during payload release to convert the longitudinal dimension into pressure head, This system had many complications, including the adverse effect on flight qualities cause by a large reservoir protruding into the air stream on the top and bottom of the aircraft, the massive change in mass moments of inertia about the longitudinal and lateral axes, and the moment applied on the aircraft to pivot the mass of the reservoir and payload. During the ensuing discussion, however, a retractable boom was proposed which would utilize a tube of the requisite 0.5" diameter hinged at the bottom of the reservoir. During payload release, the boom would swing downward to increase the pressure head of the tank, and thus the volumetric flow rate of the payload. The payload systems in the figure below utilize either a retractable boom or a venting system which maintains tank head above the tank level. Each payload system concept is described in the section below the figure.

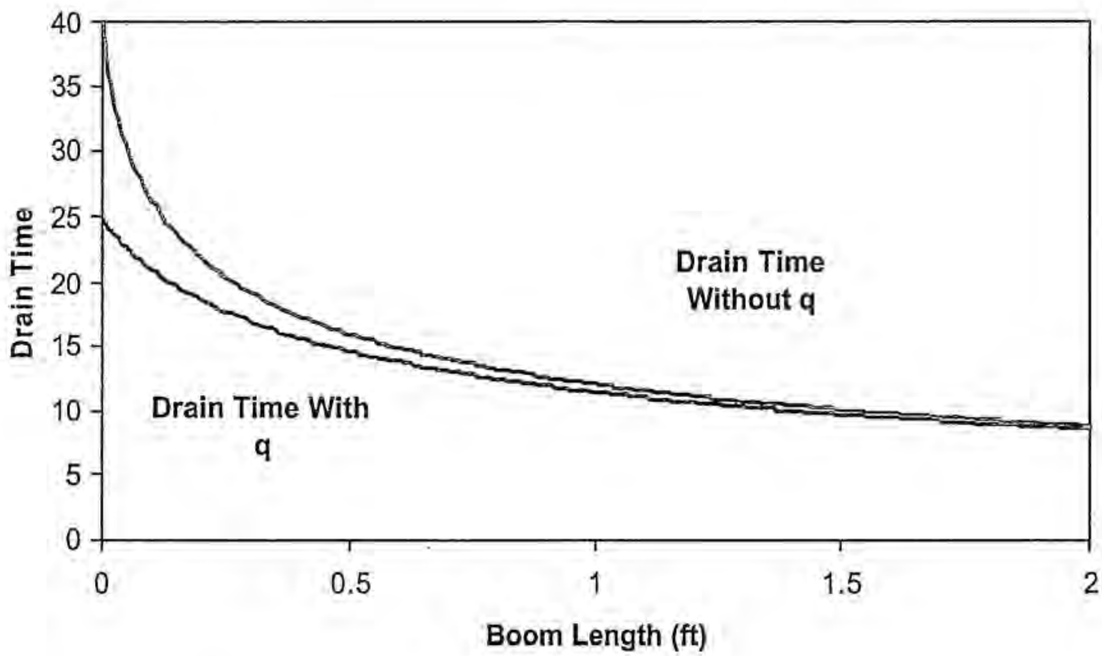


Figure 3.3: Fluid Flow Rate as a Function of Fluid Head and Dynamic Pressure

- Inverted Pyramidal Reservoir with Baffles: An inverted pyramidal tank allows the majority of the volume, and thus weight, to be concentrated at the top of the tank. Depending on the shape of the tank and the slope of the fore and aft tank bulkheads, it may be possible to minimize the CG shift by ensuring that the weight fraction of the payload that can shift significantly is relatively small. The pyramidal reservoir also creates a relatively high head pressure by virtue of its depth. Ease of construction is also an advantage of the pyramidal reservoir. Even complex shapes could easily created using a foam plug overlaid with fiberglass, then hollowed out using acetone to dissolve the foam. Loading would be easy due to the fact that the tank could be loaded through a single point. The reservoir would require some type of baffling to combat dynamic payload shift due to changes in deck angle.
- Multiple Individual Tanks with Check Valves: To further mitigate water slosh and CG shift from what could be achieved with a single baffled reservoir, multiple tanks could be installed. The tanks would be plumbed together with check valves which only allowed water to flow inward toward the center tank. The multiple tank concept retains the head pressure advantages of the

single tank, but would be slightly more difficult to construct, and also more complicated to load, since multiple ports would be required to fill each compartment.

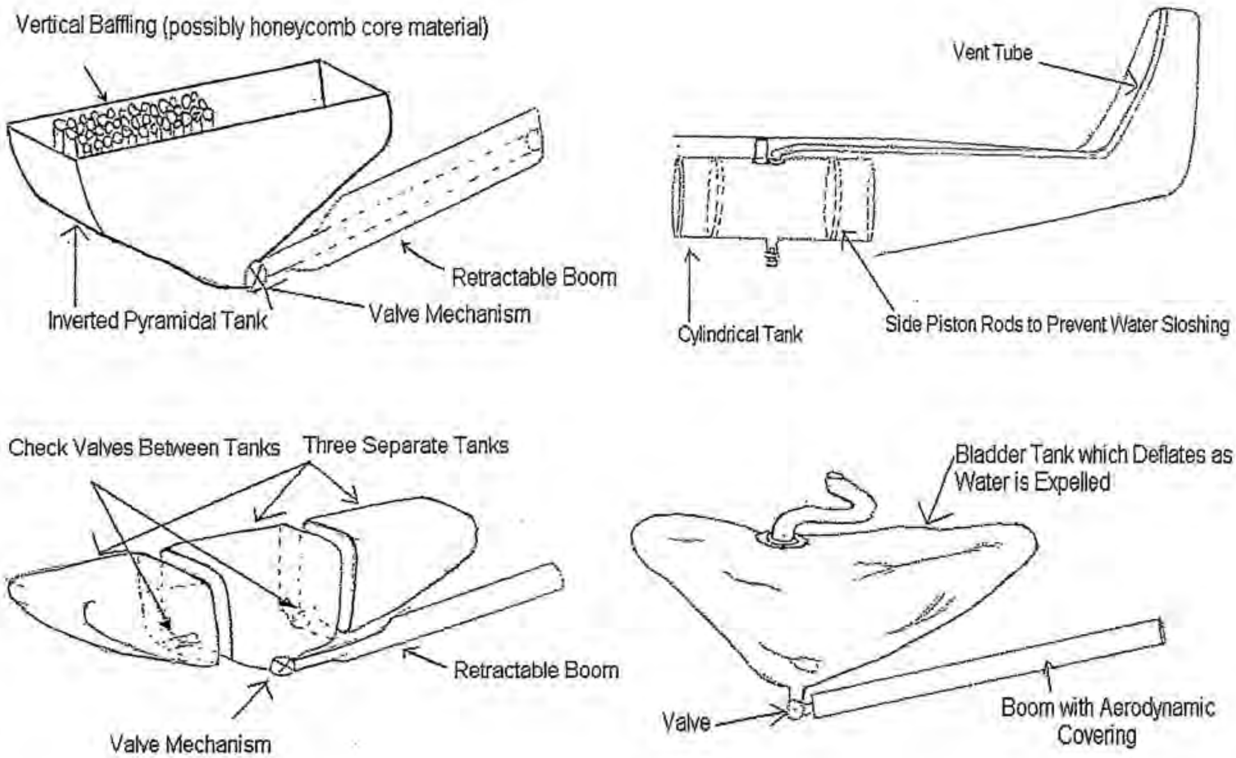


Figure 3.4: Payload System Configuration Concepts

- Variable Volume Cylindrical Reservoir: Another tank concept considered was a long, cylindrical tank equipped with a sealed worm drive running through the center of the tank which operated plungers on each end of the tank. As the payload flowed out of the tank, the worm drive would move the plungers toward the center of the tank, maintaining the tank CG at the original location. The variable volume cylindrical reservoir offers the advantage of minimal CG shift, but at the expense of complexity and weight. A carefully tuned speed would have to be established to collapse the plungers at the same rate that water was flowing from the reservoir, or alternatively, an active control system would have to be designed to regulate the height of water in the cylinder. The cylindrical reservoir also creates some complications with loading; after the payload is released, the plungers would have to be powered back to their expanded position to allow the second sortie's payload to be added. In addition, a cylinder stiff enough to maintain a seal around the plungers would necessarily be heavier than the other concepts.
- Bladder Tank which Deflates as Water is Expelled: A bladder type reservoir offered the advantage of not having to be vented because the tank would collapse as the payload emptied. This also eased manufacture since the tank would expand to fill the available space and did not

have to meet tight shape tolerances. The disadvantage would be that the tank would be hard to restrain, and loading would have to be done through a pressure port. The bladder tank concept was initially conceived with the idea that an elastic bladder might assist with expelling the payload; however, FAQ postings from the contest directors later prohibited elastic bladders.

The nature of the comparisons for the payload systems was such that a numerical decision matrix was not required. The cylindrical reservoir was determined to be non-optimum for several reasons, including the inherent complexity of the variable volume system and the complications associated with loading. In addition, the length of the reservoir would require a longer fuselage which would offset the RAC savings gained from the lower frontal area of the cylindrical reservoir. The bladder tank concept was rejected due to the difficulty of preventing dynamic CG shift (because of a lack of baffling) and also due to the loading complexity. The remaining concepts were considered to be variations on a common theme, and it was decided that the single pyramidal reservoir with a baffling system offered the best compromise of performance, and simplicity. Equipped with a retractable boom, the single reservoir would also provide the highest flow rate of any of the concepts.

3.3.3 Landing Gear Configuration Concepts and Figures of Merit

Proper design of landing gear type and configuration is a key factor in the suitability of the aircraft for its intended mission. Landing gear design holds major implications for aircraft ground handling, structural weight, parasite drag, takeoff and landing performance, and even payload system integration. The optimum landing gear configuration is also dependent on the overall aircraft configuration. Based on these considerations, the Figures of Merit for the Landing Gear system were chosen as follows.

- Compatibility with Aircraft Configuration: As in every other design area, the best landing gear for an aircraft depends on the aircraft. The contest aircraft being designed is highly optimized for its intended mission, so the landing gear system must be well integrated with the rest of the aircraft so as to prevent degradation in utility and performance.
- Weight: As was seen in the mission analysis, empty aircraft weight is a large factor in determining RAC and hence total Score, so a lightweight design is highly desired.
- Drag: Both of the missions in the competition involve a timed course, so speed, and hence drag, should be minimized as much as possible. In addition, lowering drag reduces the amount of power required and hence the battery weight and RAC.
- Ground Handling: The score of the airplane depends on how well the pilot can fly the mission profile, which in turn depends on the aircraft's handling qualities. Ground handling is a major aspect of the operator friendliness of the design.
- Takeoff Performance: Because of the 150 ft. maximum takeoff distance requirement, takeoff is one of the limiting factors when optimizing the aircraft for the mission. A landing gear configuration that offered a better ground stance might make the difference between meeting the 150 ft. requirement or not.

- Ease of Construction: Manufacturability is always a key factor. Proper alignment and strength of the landing gear are important issues that must be able to be accomplished for the design regardless of what is chosen.
- Cost: Material availability and cost are always considerations in any design. The materials used for the landing gear must be available within a reasonable time for a cost that can be met without causing the project budget to be exceeded.

Major landing gear configurations investigated included mostly variations on the common taildragger and tricycle arrangements. Unconventional configurations such as a single main wheel or bicycle type gear with outriggers did not offer any discernible advantages, and thus were eliminated from consideration due to the negative ground handling issues they presented. One of the major considerations in the design of the landing gear was the fact that the aircraft had to fit into the box described in the contest rules. As a result, for most configurations, the landing gear would have to be detached from the aircraft for storage. Because the main wheel location for tricycle landing gear is aft of the CG, and the payload release valve is located roughly on the aircraft CG, tricycle gear configurations must be swept aft from its mounting location to provide clearance for the boom to swing downward from its retracted position. Alternatively, the main gear could be designed as two pieces installed in separate receivers mounted on either side of the fuselage or wing. The section below describes the various configurations considered for the landing gear. Figure 3.5 displays these configurations.

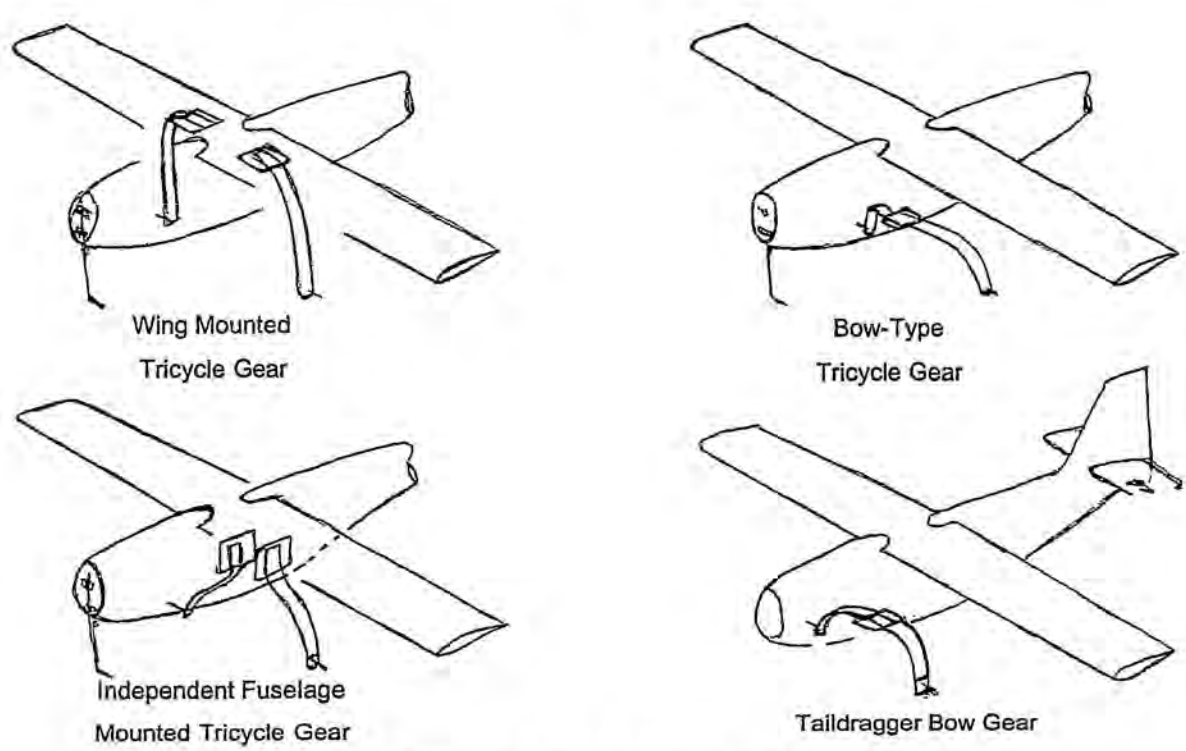


Figure 3.5 Landing Gear Configuration Concepts

- Tricycle Configuration, Bow Gear Swept Aft: The advantages of the tricycle configuration are good ground handling qualities due to the aircraft CG being forward of the main gear contact point, which causes the aircraft to be positively stable on the ground. If the aircraft is landed with a slight crab angle, the aircraft will naturally align itself with the direction of motion. The single piece, bow-type landing gear is simple to build, reliable, and lightweight. Depending on the rake angle, sweeping the bow aft might complicate its manufacture somewhat.
- Tricycle Configuration, Independent Gear Legs Mounted in Fuselage: Because of the need for boom clearance, a design in which the left and right main gear legs were independently mounted in the side of the fuselage was considered. The legs would be mounted in sockets or rails from which they could be removed for storage of the aircraft. The disadvantage would be that the alignment of the two gear legs would be difficult, and the necessity to build two hard points in the fuselage would increase more than the bow type gear.
- Tricycle Configuration, Independent Gear Legs Mounted in Wing: If the landing gear legs were mounted in the wing, they might not need to be removable, since the wing panels must be removable. If the gear legs were permanently mounted they could theoretically be somewhat lighter than if they were removable. The disadvantage with this method is that the gear legs would have to be longer if the aircraft was not a low wing. The required stiffness of the gear legs and hence the weight of the gear would then be increased. Longer gear legs would also be more flexible and possibly have a negative impact on ground handling.
- Taildragger Configuration with Bow-type Gear: The taildragger configuration allows the use of a bow gear without excessive sweep, and employs a single hardpoint installed forward on the fuselage. The lack of a nose gear reduces drag and weight, and the aircraft takeoff performance should be improved, even if only marginally, because the aircraft doesn't have to use the main gear as a fulcrum to raise the nose during rotation. Disadvantages of the taildragger configuration are, of course, that the ground handling qualities are not as good as those of the tricycle gear. Constant corrections must be made to keep the aircraft going in the desired direction on the ground. The bicycle configuration consists of two main wheels, one forward and the other aft of the CG.

Each of the landing gear concepts was qualitatively evaluated and then assigned a numerical ranking based on its support for each of the Figures of Merit. A weighted decision matrix was then created, which showed that the Bow-type Taildragger landing gear was the optimum design for this aircraft. The landing gear decision matrix is shown in Table 3.2, below.

Table 3.2: Landing Gear Decision Matrix

Figures of Merit	Weighting Factor	Taildragger, Bow-type	Tri-Gear, Bow-type	Tri-Gear, Dual Fuselage Mount	Tri-Gear, Wing Mount
Compatibility with Aircraft Configuration	*0.2*	1	0	-1	0
Weight	*0.2*	1	-1	0	0
Drag	*0.2*	1	-1	1	0
Ground Handling	*0.1*	0	-1	-1	1
Takeoff Performance	*0.1*	0	0	0	1
Ease of Construction	*0.05*	0	-1	-1	0
Cost	*0.05*	0	0	0	0
Overall Weighted Score:	-----	**0.60**	**-0.55**	**-0.15**	**0.20**

3.3.4 Propulsion System Configuration Concepts and Figures of Merit

The propulsion system accounts for approximately one third of the aircraft's total empty weight. This large percentage of weight has a significant effect on the center of gravity of the aircraft. It is also important to minimize the weight of the propulsion system because of the impact on RAC. Total score is more sensitive to battery weight than to any other single variable, due to the large impact of battery weight on RAC, accounting for a 3% increase in RAC per quarter pound of battery weight in some typical combinations of flight score and aircraft size and complexity. These two parameters, weight and RAC, as well as five others were chosen as the focus of the propulsion system design and were used to evaluate design alternatives. These Figures of Merit, with the relevance of each, are listed below.

- Weight: Battery weight is the highest contributor to RAC. Using an efficient motor and propulsion system can greatly reduce the number of batteries needed to competitively complete the required missions.
- Rated Aircraft Cost: The location and number of the motor(s) and propeller(s) can potentially increase the RAC if the volume of the plane must be increased in order to compensate for the added motor and battery space.
- Ease of Construction: The location of the motor and propeller must allow for reasonable construction for the Structures team in order to minimize errors during fabrication.
- Available Propeller Sizes: The location of the motor should permit maximum flexibility in choosing the most efficient propeller diameter for varying wind conditions.

- Reliability: The motor must be located in an area that allows it to run dependably, with enhanced motor cooling.

The location of the motor and propeller plays a significant role in the performance of any RC plane. The following alternatives were considered: pylon mount, tractor, pusher, and wing mount.

Using the before mentioned figures of merit, each of the four alternatives were evaluated for location of the propulsion system in a weighted decision matrix shown in Table 3.3. Since weight and RAC are adversely proportional to Score, they were given the largest priority. Aircraft control and drag were also weighted heavily because of their effect on the size of the aircraft and thus the effect RAC. Ease of construction, propeller sizes, and reliability were not weighted as strongly as the other Figures of Merit because they do not have an immediate effect on Score.

Upon evaluation of the advantages and disadvantages of each design using the decision matrix, the tractor configuration was chosen as the most appropriate for the mission. The frontal mounting allows for greater CG control by displacing the weight of the payload. This also allows for an increased propeller diameter which will generate the thrust needed for zero wind take-offs, and also effectively lowers RAC because less wing area will be required.

Table 3.3: Decision Matrix for Propeller and Motor Placement

Figures of Merit	Weighting Factor	Tractor	Pusher	Wing Mount	Pylon Mount
Weight	*0.20*	1	1	-1	0
RAC	*0.20*	1	1	-1	0
Aircraft Control	*0.16*	1	1	1	0
Drag	*0.16*	1	1	0	-1
Ease of Construction	*0.14*	1	1	-1	-1
Available Propeller Sizes	*0.08*	1	-1	0	1
Reliability	*0.06*	0	0	1	0
Overall Weighted Score	-----	**0.94**	**0.78**	**-0.32**	**-0.22**

Placement of the remaining components of the propulsion system was equally as important as the location of the motor and propeller. Position of each propulsion system component was considered based upon CG control and minimizing power losses. These considerations for each component are as follows:

- Battery Pack: The location of the battery pack is vital due to its large contribution to the total aircraft weight. Therefore, the battery pack must be located in the optimum location to maintain CG control. In addition, the battery pack should be located near the motor to reduce the amount of power loss in the wiring. Taking these factors into account, the battery pack will be located in the front of the plane near the motor. If the decision is made to use two battery packs, they will both be located in the front of the plane.

- Gearbox: By using a gearbox, thrust and RPM can be better controlled. Changing the gear ratio ensures that the motor is operating in its most efficient range for the majority of the flight. This saves power and effectively reduces the number of batteries required resulting in a lower RAC score. It is important to choose the best ratio for both the Fire Fight and Ferry Flight missions. Extensive testing during the prototype stage will allow for choosing the most effective gear ratio for the propulsion system. The gearbox will be located directly between the propeller and the motor. The location of the gearbox does not have a significant effect on the RAC score.
- Speed Controller: Speed controllers adjust the throttle setting of the motor based on input from the remote control. Finding the right speed control is not the most important part of the propulsion system, but it still must be optimized for a competitive plane. Two speed controllers will be tested during the Preliminary design phase: the AstroFlight 204D and the Jeti JES 60. The speed controller will be placed in the front of the plane near the rest of the propulsion system.
- Fuse: To protect the motor from excessive current, the fuse must carry the entire motor current. Because the speed controller is a power transformer, the current in the circuit between the speed controller and the motor is not necessarily the same current in the circuit between the battery and the speed controller. For this reason, the fuse must be placed in the motor circuit between the speed controller and the motor.

- **Motor and Battery Combinations**

Several options for battery and motor configurations were considered: single motor/single battery pack, single motor/dual battery pack, dual motor/single battery pack, and dual motor/dual battery pack. As with all design criteria, weight and RAC were held with highest regard when choosing a design. However, several other Figures of Merit were taken into consideration when choosing a motor/battery combination. These Figures of Merit are listed as follows:

- Weight: The motor and battery weight must be kept to a minimum because they will greatly influence both the RAC and the mission times, and thus the overall Score.
- Rated Aircraft Cost (RAC): The battery weight has a huge affect on the RAC score – it is included in the Manufacturers Empty Weight and the Rated Engine Power equations. The number of motors is also a factor in the Rated Engine Power equation.
- Efficiency: The more efficient a system is the fewer batteries that will be needed. This results in a lower RAC.
- Thrust Produced: Enough thrust must be produced for takeoff with a full four liters of payload.
- CG Control: The motor and battery combination must allow for effective control of the center of gravity, which affects stability.

- Ease of Construction: The system must be simple enough that it can be constructed in a reasonable amount of time.

Each of these design criteria were taken into account as Figures of Merit when evaluating the above combinations. Weight and RAC were again given highest weighting factor, however, equally important were efficiency and thrust since they also influence Score. CG control and ease of construction were rated lower because they do not directly influence Score. Results from the analysis can be seen below in Table 3.4.

As shown in Table 3.4, a single motor system is the best for the competition. Not only is this the top design for our aircraft, it is also characteristic of the previously chosen tractor design. The optimum design for the motor/battery combination is one battery pack, however, in order to keep the fuselage cross section small, it may be necessary to split the pack into two smaller packs.

Table 3.4: Decision Matrix for Suggested Motor and Battery Combinations

Figures of Merit	Weighting Factor	Single Motor / Single Battery Pack	Single Motor / Dual Battery Pack	Dual Motor / Single Battery Pack	Dual Motor / Dual Battery Pack
Weight	*0.19*	1	1	-1	-1
RAC	*0.19*	1	1	-1	-1
Efficiency	*0.19*	1	0	-1	-1
Thrust Produced	*0.19*	1	1	1	1
CG Control	*0.12*	0	1	0	0
Ease of Construction	*0.12*	1	1	0	0
Overall Weighted Score:		**0.86**	**0.79**	**-0.37**	**-0.37**

3.3.5 Construction Method Concepts and Figures of Merit

Several primary structural configurations were considered for the aircraft. The goal of the conceptual design phase for structural considerations was to choose a configuration for each aircraft component that would be provide the optimum strength to weight ratio and still be within the manufacturing capabilities of the design lab facilities and team members. The Figures of Merit selected for structural configuration were the same for each of the aircraft components, but different weighting factors led to different construction types in some cases. Figures of Merit for the aircraft structural configuration are listed below.

- Strength to Weight Ratio: Aircraft weight was a major contributing factor to Rated Aircraft Cost and was one of the factors to which total Score showed the highest sensitivity. A structural configuration that attained the required strength while minimizing weight was of the highest importance.

- Formability: Aerodynamically optimum shapes contain a multitude of compound curves. The ideal structural configuration would easily allow the formation of complex shapes.
- Manufacturability: Manufacturability relates to both capabilities and time required. A structural configuration that did not require construction of a large amount of jigs and tooling would be more desirable than one that required excessive amounts of time to prepare for construction of the actual aircraft articles.
- Reparability: Damage to the airplanes during testing is inevitable. If the structure can be repaired and used to gain further data, testing can continue instead of waiting for a new article to be constructed.
- Cost: The design team had a limited budget, which meant that the cost of structural configurations had to be considered, especially those involving exotic and expensive materials.

The aircraft was divided into three structural assemblies so that the optimum construction method could be chosen for each assembly. The three assemblies were the Wing, Fuselage, and Empennage. Listed below are the structural configurations considered for each of the three assemblies, and a description of the configuration, along with its advantages and disadvantages.

- Foam Core with Composite Skin: The foam core method is well known and documented, and involves cutting or carving the desired component shape from a block of insulating foam. The foam is then laminated with one or more layers of composite material to obtain the desired strength. This construction provides nearly unlimited formability and good manufacturability. Jigs are mostly constructed of the cradles the foam is cut from, so little time is required to build tooling. Foam core structures are also quite repairable. Broken structures can usually be epoxied together and possibly reinforced with a small amount of additional composite material. The disadvantage of the foam core method is that the weight is higher than with other methods.
- Conventional Built-up Construction: The built-up method also provides a relatively high strength to weight ratio. It involves building an internal structure from light woods such as balsa and poplar plywood, and skinning it with either balsa or a plastic film such as Monokote®. The built-up method is usually semi-monocoque—structures are usually at least partially sheeted, and gain some of their strength from the plastic film covering. Disadvantages to the built-up method are that complex shapes are more difficult to produce, and careful attention is required during construction to maintain alignment and contour accuracy. In addition, the plastic film requires some experience and skill to properly apply, so manufacturability is somewhat less than for the foam core method. Built-up structures can usually be repaired by reattaching broken structural members with adhesive and replacing the plastic film covering.
- Fully Monocoque Composite Skin Construction: The monocoque stressed-skin construction method provides the highest strength to weight ratio of any of the concepts considered. The primary structural members are the skins, constructed of a thin composite sandwich with a core of balsa. The sandwich skin construction maintains the surface contour over large areas due to its

stiffness. Internal structure is drastically reduced, and consists mostly of shear webs used to connect the upper and lower skins, and other localized structure required to join the components together. Occasional fuselage bulkheads are required to mount components and transmit large compressive and torsional loads. The disadvantage of the monocoque construction is that it requires more tooling than the built-up method. Because the internal structure is so sparse, molds of some sort are usually required to construct a monocoque component. If molds are used, the monocoque method provides more formability than the built-up method, but if molds are not used, the skins must be laid up before they are assembled, resulting in less formability than the built-up method. Reparability of fully monocoque structures is also somewhat lower than for built-up structures because of the highly stressed nature of the skins, and also because of the difficulty involved in maintaining alignment while bonding parts with no structural backing.

The different construction techniques were compared for each structural assembly using a decision matrix with weighted figures of merit. Refer to Section 6 of this report for a discussion on the techniques employed to construct each of these structural compositions.

- Wing Construction: In order to perform a preliminary weight comparison, OSU historical data of construction techniques was consulted. The conventional built-up construction method and fully monocoque balsa and composite construction method are empirically the lightest weight. The strength of the monocoque method, however, is significantly superior to the built-up method.

Table 3.5, below, displays the weighted decision matrix used to choose the wing construction scheme. The most important factor in the decision was the strength-to-weight ratio. While the built-up method is light, its strength is significantly less than monocoque construction. Construction with a foam core does not compare to the other two in terms of this factor, though it is easier to manufacture and repair. The built-up method is slightly lower in cost due to the lack of composite materials.

Table 3.5: Wing Construction Method Figures of Merit

Figures of Merit	Weighting Factor	Foam Core w/ Composite Skin	Conventional Built-Up	Fully Monocoque Composite Skin
Strength to Weight Ratio	*0.50*	-1	0	1
Formability	*0.10*	1	1	1
Manufacturability	*0.20*	1	0	0
Reparability	*0.15*	1	0	0
Cost	*0.05*	0	1	0
Overall Weighted Score:	-----	**-0.15**	**0.15**	**0.60**

- Fuselage Construction: The fuselage structure is required to meet several conflicting criteria. The fuselage needs to hold the payload and all necessary electronic and mechanical equipment. However, the fuselage must also maintain the lowest drag and lightest weight possible.

Additionally, the fuselage must remain structurally sound because all other components will be anchored to it. Aside from the design of the fuselage shape and dimensions, the construction techniques employed also play a large role in achieving the above requirements.

The fuselage construction methods are evaluated in a decision matrix in Table 3.6. The strength-to-weight ratio was still the most important factor in this decision. However, the ability to construct the complex curves of the fuselage plays a very large role. This task is extremely difficult to perform with the conventional built-up method. The same discussion about the foam core method from the wing construction is still applicable here. The ease of using this technique can not justify the poor strength-to-weight ratio.

Table 3.6: Fuselage Construction Method Decision Matrix

Figures of Merit	Weighting Factor	Foam Core w/ Composite Skin	Conventional Built-Up	Fully Monocoque Composite Skin
Strength to Weight Ratio	*0.45*	-1	0	1
Formability	*0.25*	1	-1	1
Manufacturability	*0.10*	1	0	0
Reparability	*0.15*	1	0	0
Cost	*0.05*	0	1	0
Overall Weighted Score:	-----	**0.05**	**-0.20**	**0.70**

- Empennage Construction: Table 3.7, below, explains the rationale behind the construction technique of the empennages using a weighted decision matrix. The horizontal and vertical tails do not require the strength that the wing and fuselage do, as they experience smaller flight loads. Therefore, the overall weight of the built-up method is slightly less than the monocoque composite construction for the devices. Because no complex curves exist in the empennages, formability and manufacturability are not major issues.

The fabrication of the empennage components using the built-up method is easier than dealing with the required molds and composites of the monocoque technique. However, the nature of the vertical stabilizer enabled it to be constructed in the same mold as the fuselage with almost no extra effort. In addition, building both the vertical tail and fuselage pieces together with the monocoque construction method would reduce overall airplane weight by preventing the need for structural reinforcement at their junction. Because the horizontal tail could not be incorporated into the monocoque construction of the fuselage and vertical tail, it would be built separately using the conventional method.

Table 3.7: Empennage Construction Method Decision Matrix

Figures of Merit	Weighting Factor	Foam Core w/ Composite Skin	Conventional Built-Up	Fully Monocoque Composite Skin
Strength to Weight Ratio	*0.25*	-1	0	1
Formability	*0.25*	1	1	1
Manufacturability	*0.30*	1	1	0
Reparability	*0.15*	1	0	0
Cost	*0.05*	0	1	0
Overall Weighted Score:	-----	**0.45**	**0.60**	**0.50**

3.4 Conceptual Design Results

Based on the results of the conceptual design phase for each technical group, the final Orange Team aircraft concept was defined. The final aircraft planform configuration chosen was a conventional layout with a single tractor propeller, high-mounted wing and tail-dragger landing gear. A payload system consisting of an inverted pyramidal reservoir would be installed in the aircraft, with a retractable boom to increase water head during payload release. Both handling qualities and attainable lift coefficient were rated at least equal to every other design, as was the compatibility with the payload handling system. The RAC of the design, while not the absolute lowest possible, is the lowest possible among designs that are able to effectively accomplish the other objectives of handling and performance. The aircraft was to be constructed using a monocoque composite and balsa wood structure. The main landing gear was to be of a composite/balsa sandwich bow type construction, which would create low drag and minimum weight effects. The final aircraft was the one found to possess the optimum combination of qualities enumerated in the Figures of Merit. A sketch of the final Orange Team concept aircraft may be seen in Figure 3.6, below.

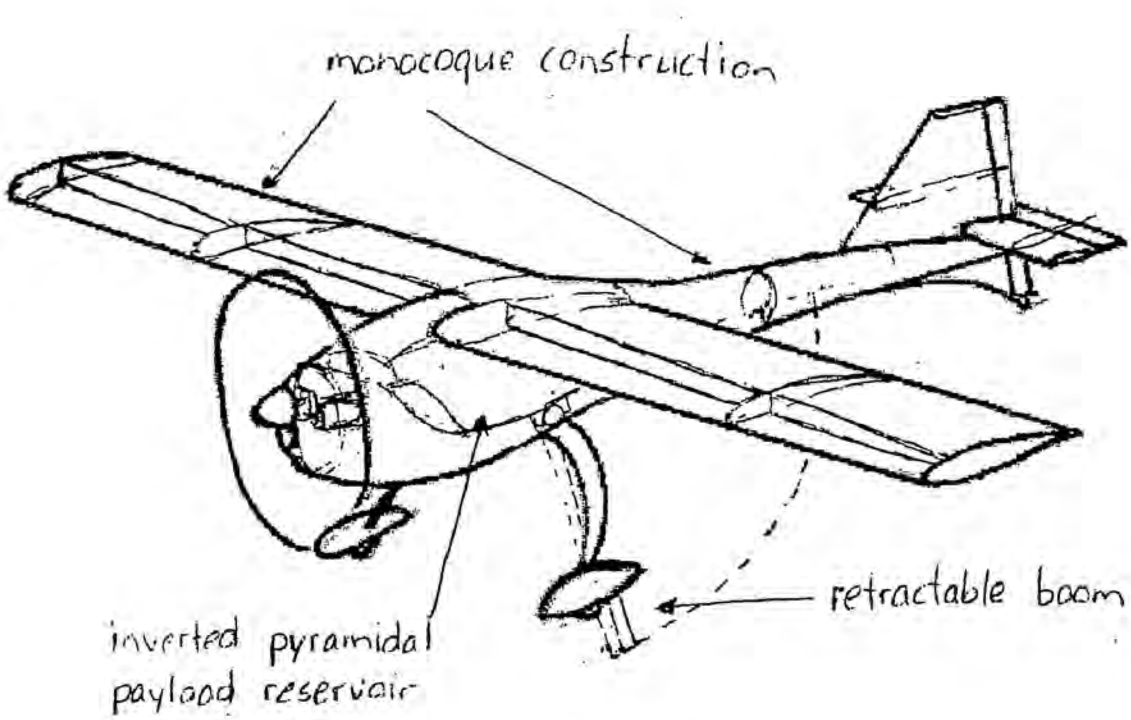


Figure 3.6: Final Aircraft Configuration

4 Preliminary Design

As soon as a final aircraft concept was chosen, the preliminary design process began. During the preliminary design process, the size and configuration of major design parameters were set, structural design schemes and component locations determined, and type, size, and capacity of required components established. Each technical group used unique methods to determine the best way to meet the design goals in their respective area. The following section describes the analytical methods and tools used to optimize the aircraft design in all respects, and the results of the optimization.

4.1 Major Design Parameters and Sizing Trades

Based on the sensitivity analysis performed at the beginning of the conceptual design phase, it was known that several major design parameters were of primary importance in order to optimize the scoring potential of the aircraft. A listing of important design parameters that both directly and indirectly affect the scoring potential of the aircraft are listed in the section below, along with explanations of their importance to the design.

- Wing Area and Aspect Ratio: Required wing area may be one of the single most important design parameters of the optimization. An overly large wing area will increase parasite drag, reducing top speed, and will also increase aircraft empty weight. In addition, a large aircraft that is too lightly loaded may be hard to control in the high winds, and will suffer an unnecessarily high RAC because of the wing area penalty. Conversely, a wing too small for the intended mission may not be able to achieve the 150 ft. takeoff requirement, resulting in a complete loss of flight score. Scoring potential is not as sensitive to aspect ratio as it is to wing area, but an aspect ratio too low or too high can cause either a loss of aerodynamic efficiency or structural problems and tip stalling tendencies, respectively.

- Fuselage Length: Because the aircraft is required to fit into a four foot long box, fuselage length is a major consideration. A fuselage short enough to fit into the box without disassembly can be built lighter than one requiring a removable joint. In addition, a shorter fuselage length will increase scoring potential by reducing RAC if all other variables are held constant. The tradeoff involved with a shorter fuselage is that it is more difficult to obtain the tail volumes and control powers required for stability and control of the aircraft, especially if the wing utilizes high lift devices, which usually create a strong pitching moment which must be counteracted by the tail surfaces.
- Tail Volumes and Sizing: To a somewhat lesser degree, tail volumes are important in the same manner as the wing area. As indicated in the discussion on fuselage length, tail and control volumes must be large enough to provide the required stability, and control power, but not too large so as to create unnecessary parasite drag. In addition, the horizontal tail span must not exceed 25% of the wing span to avoid being counted as a second wing in the RAC function.
- Number and Type of Batteries: Total scoring potential of the aircraft is more sensitive to the number and type of batteries required and the resulting battery weight than any other design parameter. The number of batteries must be sufficient to achieve the power required for takeoff and the endurance to complete the mission, but remain as light as possible in order to minimize RAC. Choosing a propulsion battery requires an analysis of internal resistance, energy density, sizing and packaging concerns. The energy density of the battery must be carefully matched with the mission.
- Propeller/Gearbox Combinations: As part of the propulsion system, propeller and gearbox combinations are vital to the aircraft being able to meet the performance goals. Optimal propeller/gearbox combinations will allow the aircraft to take off and fly the mission with the most efficiency possible. Efficient combinations will allow the motor and/or the battery pack size to be reduced, which will increase score by decreasing weight and RAC. Tradeoffs involved in propeller and gearbox combinations include trading high cruise speed and efficiency for high static thrust.
- Motor Type: The type of motor selected to power the aircraft is probably the single most important decision of preliminary design as it applies to the propulsion system. The motor's capacity and efficiency will determine the size of the battery pack required, which, as noted above, has the largest sensitivity coefficient on RAC of any design parameter. Motor efficiency is probably the single largest factor in the determination of the optimum motor for the aircraft, but tradeoffs involved include motor weight, cost, obtainable power, and availability.

- Structural Configuration: How well the aircraft structure is designed will determine the strength and weight of the aircraft structure. For composite layups, the weight, number and orientation of plies must be correctly optimized in order to achieve the lightest structure possible consistent with the required strength.

4.2 Analysis Tools

To ensure that each aspect of the design was optimized to the furthest extent possible for the intended mission, a variety of analytical, qualitative, and experimental analysis tools were utilized. The following section provides an overview of the tools used in the optimization, along with a brief description of their use and results.

4.2.1 Analytical Methods

An analysis of aircraft design parameters and their impact on the scoring potential of the aircraft was performed using a computer routine. The program calculated aircraft scoring potential based on a multitude of inputs including component efficiencies and aircraft coefficients as well as major design parameters such as battery weight, payload weight, wing area and wing span. Efficiencies and coefficients were set to the level obtainable with available equipment and design practices and held constant for the optimization, while major design parameters were varied to determine their effect on the total score and the sensitivity of the score to each of the parameters. The program searched for local score maxima after initializing itself from random starting points for the major design parameters. By running the program multiple times and plotting the score outputs versus each of the design variables, trends in scoring potential could be determined.

The Propulsion team utilized a computer routine to determine optimum combinations of gear ratios, battery packs, and propellers. In conjunction with the Aerodynamics/Stability group, the Propulsion group ran simulations under varying wind conditions to predict performance of the aircraft under a variety of aircraft configurations and wind conditions. During the Preliminary Design phase, bench testing began on propulsion system components. The resulting data was used to refine the computer models in order to achieve more accurate simulations.

4.2.2 Experimental Methods

As mentioned in the section on analytical methods, the Propulsion group performed a number of experiments during the preliminary design phase to obtain actual results to compare with the manufacturer's predictions for various components. Efficiencies and thrust profiles were developed for motors, battery packs, and propellers.

The Structures group used a combination of historical and experimental data to perform preliminary design. Before much optimization could be done for the gear ratio and propeller, battery type had to be chosen. Once the type of battery was decided, the minimum number of batteries to provide two fully loaded take-offs in 150 feet could be found. It was essential for RAC to use the minimum number of batteries possible. Justification for these decisions will be discussed later in the Detail Design portion of the report.

4.3 Aerodynamic Considerations

One of the aspects of preliminary design with the most broad and far reaching implications for the overall design is the configuration and sizing of the aerodynamic surfaces. Choosing appropriate wing and tail sizes and airfoils is crucial to achieving a design capable of performing the desired mission. The following section describes the manner in which the aerodynamic characteristics of the aircraft were determined, and the results of the analysis.

4.3.1 Airfoil and Wing Area Considerations

The Aero Group exhaustively utilized the numerical performance routine to determine the optimum sizes and configurations for each of the major aircraft components. As described in section 4.2.1, above, the code searched for local maxima when it was initialized with a given set of conditions. The input parameters for the optimization code included airfoil lift/drag polars and wind speed. Simulations were run with wind speeds of 5, 10, 20, and 30 miles per hour, to determine the sensitivity of the optimum wing sizes to wind speed. A range of airfoils was also used, from a highly cambered Eppler 423 to a much thinner Selig-Donovan SD7032. Initial results showed that the optimum wing area would not exceed 6 ft^2. With this knowledge, a stability and control analysis was performed to determine whether acceptable tail volumes could be achieved with a fuselage as short as 4 ft. A 4 ft fuselage was desirable because of the box fit requirement. If the fuselage was required to be longer than 4 ft, a disconnect joint would be required to enable disassembly to fit into the box, which would add complexity and weight. The results of the analysis showed that an acceptable tail volume could be obtained even at the upper ranges of optimum wing areas. This allowed a refinement of the weight model, which resulted in a predicted optimum range of wing areas that was lower than before.

Optimum wing areas predicted by the code ranged from slightly under 5 ft^2 to around 6 ft^2, depending on the airfoil and wind condition chosen. Outputs from the code showed, however, that the scoring potential of the aircraft, while definitely dependent on the wing area, was not extremely sensitive when close to the optimum. A range of areas existed that would provide performance close to the global maximum. Score calculations were also plotted versus cruise power, wing span, and battery weight, and payload weight. The results of the analysis can be seen in Figure 4.1, below. Score is shown plotted versus wing area, wing span, cruise speed, and battery weight. Program outputs were also taken versus weight of payload weight carried, but plots were not necessary because each high scoring configuration carried the maximum amount of payload. Note that each of the plots demonstrates an optimum area for that parameter, but that a small range of optimum values exist.

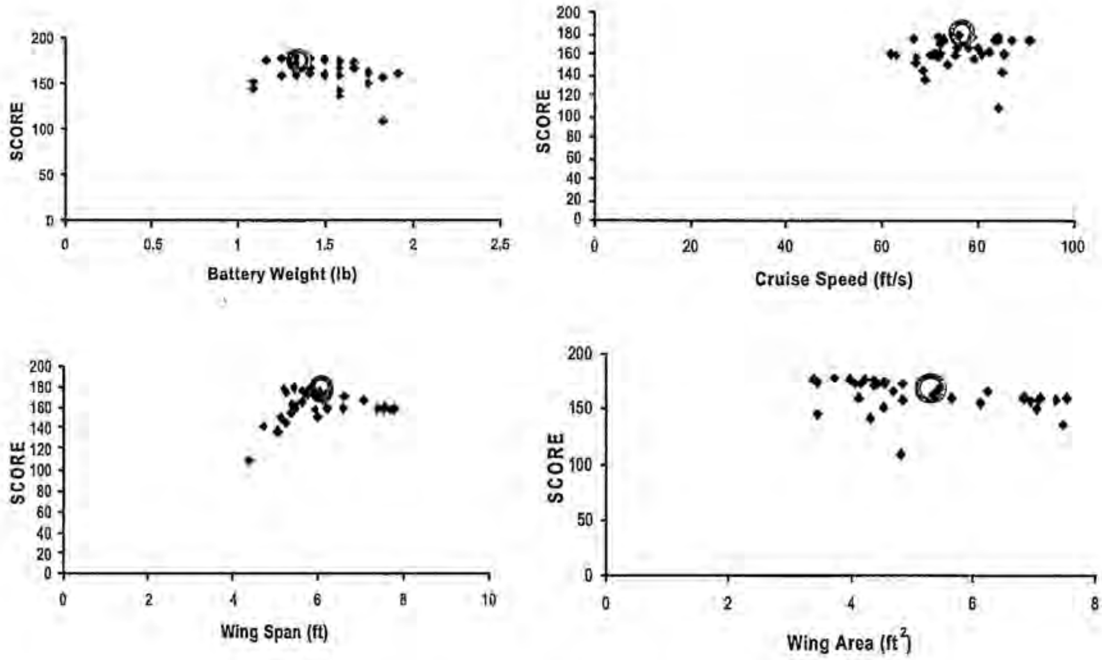


Figure 4.1 Optimization Code Results Run with the SD7062

During the preliminary optimization, the highly cambered Eppler 423 airfoil and the thick SD7062 airfoil seemed to achieve slightly higher scores than the thinner SD7032. The differences were not great enough to determine the optimum airfoil based solely on the merit of the optimization code, but combined with the knowledge that the aircraft would be taking off at a high wing loading during the fire bomber mission led to the elimination of the SD7032 airfoil from consideration. The remaining choices were the SD7062 with or without flaps, and the Eppler 423. The Eppler is designed to operate at high lift coefficients and exhibits a high drag coefficient when operated at a low lift coefficient. In addition, the wind conditions at the contest venue are often gusty during the competition. It was believed that the SD7062 would provide more flexibility than the Eppler in meeting changing weather conditions. In addition, the SD7062 was also more desirable from a manufacturing standpoint because its thick nature and gentle contours make it more easily manufactured than the highly cambered Eppler. In addition, the thickness of the airfoil increased the structural rigidity and strength to weight ratio of the wing as a result. A flapped SD7062 airfoil was chosen for the design. The SD7062 was chosen based on the scoring potential shown in the optimization code, but also because of the Lift/Drag characteristics of the airfoil as predicted by an X-Foil code. Both the flapped and unflapped airfoils exhibit low drag coefficients at their design lift coefficients, and that the drag coefficients do are not strong functions of the lift coefficient until they are well above or below the design lift coefficient. The aircraft was optimized for a 10 mile per hour wind, but contingency plans were made so that the aircraft would be able to complete a mission for which there was no wind as well by using different propeller and gearbox combinations. An optimum area of 5.2

ft^2 was chosen for the wing. This area was felt to be slightly conservative while still not handicapping the airplane from its maximum scoring potential.

4.3.2 Wing Span and Aspect Ratio

The optimization program also plotted wingspan, and hence aspect ratios, but wingspan aspect ratio optimization was performed by use of a Prandtl lifting line analysis which showed that the gain in efficiency reached a point of diminishing returns after an aspect ratio of approximately 8 was reached.

The design aspect ratio is based on the consideration of lift distribution and induced drag. The main wing of the aircraft provides most of the lift force to overcome the total weight of the aircraft during the flight mission. Therefore, high lift is needed for this mission especially during take-off with full payload. The induced drag is related to lift coefficient and aspect ratio as shown in the equation:

$$C_{D\upsilon} = \frac{C_L^2}{\pi \cdot AR}$$

From the above equation, it is obvi ous that the smaller the aspect ratio, the higher is the induced drag. However, if a large aspect ratio is designed to provide large lift coefficient, the wing becomes structurally inefficient and prone to tip stalling. Therefore, the aspect ratio must be optimized to provide the necessary lift coefficient while still keeping a reasonably low induced drag. The parameters are compared with the calculated values for an aspect ratio of 20. Based on the results of the Prandtl lifting line theory, a final aspect ratio of 8 was chosen.

The chosen wing planform was rectangular, mostly because of the nature of the RAC function, which counted wing area as the largest exposed wing chord times the wing span. A rectangular wing planform in the range where the aircraft will be operating would have very little losses, so there was not a reason to use a tapered wing.

4.3.3 Tail Volume Analysis

To obtain a reasonable range of sizes for the horizontal tail, historical data from a variety of sources were reviewed to determine tail volumes characteristic to aircraft with similar types of missions. Although there is no historical data for aircraft of similar size and scale that have similar missions, it was determined that a tail volume of approximately 0.5 should be sufficient to provided the required static stability, even with possible slight shifts in CG during payload release.

As mentioned previously, a key part of the design was the ability to construct the aircraft with a fuselage length of 4 ft. or less. Despite the short moment arm, the required tail volume was achieved with a 4 ft. fuselage and also without violating the 25% horizontal stabilizer rule. (The contest rules state that a horizontal tail with a span greater than 25% is charged as a second wing for RAC purposes).

Based on the pitching moment generated by the wing at cruise, the trim lift coefficient for the tail was calculated. An X-Foil program was used to find a suitable airfoil that would have a low drag when trimmed to the desired lift coefficient. The NACA 2410 was found have the desired characteristics. Using the moment generated by the tail at cruise, the lift needed to sustain static stability was calculated. The airfoil's graph showed the needed lift coefficient located around zero degrees (after taking into account the downwash effect from the wings).

One of the major issues to be considered in the design analysis was the range over which the aircraft CG could shift as the payload emptied, and the extent to which the shift would affect the flying qualities of the aircraft. Basic stability analysis showed that the static margin was great enough that the worst case CG would not cause the aircraft to become negatively stable. Attention then shifted to the change in trim angle of attack that corresponded with a CG shift. The following graph, Figure 4.2, shows trim alpha plotted versus aircraft CG. Also shown on the graph are the aircraft and tank CG as a function of height of water in the tank.

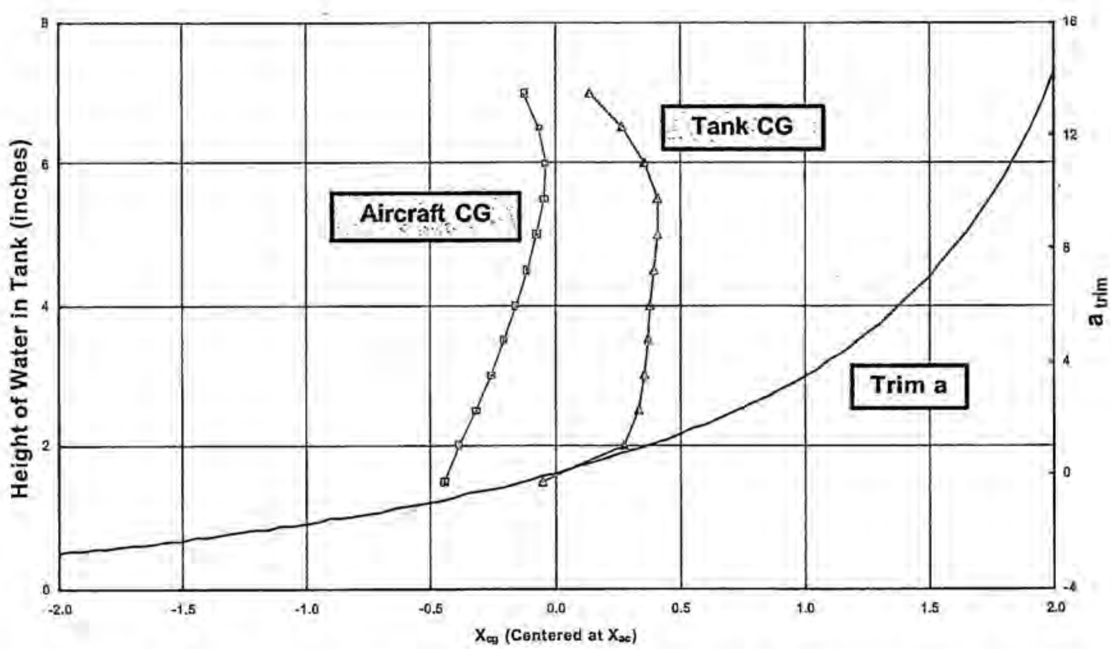


Figure 4.2: Trim alpha as a function of CG, superimposed with CG from Tank Drain Time

- <u>**Vertical Tail Sizing Analysis**</u>

Similar to the horizontal tail, the vertical tail volume was chosen due to historical research and data logs of past airplanes at Oklahoma State University. A tail volume of .05 was decided upon, which corresponded to a vertical tail area of 99 square inches. This tail volume would produce enough stability to maintain a directionally static flight. Reviewing airfoils found in the X-Foil program lead to the decision of using an airfoil that was fairly thin and symmetrical. The optimal airfoil was found to be the NACA 0009. This would allow for the needed static conditions for stability, as well as provide a feasible thickness for construction ease. The rudder sizing for the plane is based on assumptions made for the side wind and cruise velocity. A program was written which solved for the rudder area needed based on a landing β. The assumption for this case was that the plane would experience a worst case scenario β of 31 degrees. At this β a rudder area of 12" was found to be conservatively sized.

- **Elevator sizing**

To size the elevator, the tail was analyzed under the most extreme circumstance; takeoff with full payload. At this moment in flight, the tail must produce the largest counteracting moment to control the pitch of the airplane. The size of the elevator control surface is dependant on the magnitude of the pitching moment that needs to be balanced by the control. This was done by counteracting the pitching moment at the takeoff angle of attack. An elevator chord ratio of 20% was found to provide sufficient elevator control power to achieve maximum lift coefficient.

4.3.4 Aerodynamic Summary

Table 4.1: Aircraft Parameters

Aircraft Parameters		*Wing*	
X_{cg} (ft)	0.1667	$Cl\alpha w$ (rad^{-1})	4.666
X_{ac} (ft)	0.2021	AR	7.74
$d\varepsilon/d\alpha$	0.371	Cl_{ow}	0.45
$Cm_{\alpha f}$	0.000984	b (ft)	6.45
$Cn_{\beta wf}$	0	S (ft^2)	5.375
CL_{α}	7.922	Cm_{acw}	-8.E-02
CL_o	0.49	C_{bar} (ft)	0.8333
Cd_o	0.045	*Vertical Surface*	
V_{cruise} (ft/s)	75	$CL_{\alpha v}$	1.527
V_{to} (ft/s)	50	l_v (ft)	2.5
V_l (ft/s)	50	S_v (ft^2)	0.6934
Horizontal Surface		V_v	0.05
S_t (ft^2)	0.8958	S_r (ft^2)	0.06710
$Cl\alpha t$ (rad^{-1})	3.256	*Mass Inertias*	
V_H	0.5	I_x	4.576
l_t (ft)	2.5	I_y	13.1
τ	0.4	I_z	15.41
Se (ft^2)	0.1875		
b_t (ft)	1.6125		

$Cm_{\delta e}$	-0.783
c_t (ft)	0.5556

4.4 Structures Preliminary Design

Each component of the aircraft was conceived and designed with certain figures of merit in mind. The fuselage, wing, tail, and landing gear were evaluated on the following figures of merit.

- Rated Aircraft Cost: Evaluation of each addition's ability to increase performance versus its RAC.
- Weight/Strength: Weight vs. strength analysis and optimization for each component.
- Ease of Construction: The practicality for the part to be produced efficiently and accurately.
- Aerodynamics: The effect each part has on lift, drag, and aircraft stability.
- Ease of Payload Execution: The structural components needed to allow easy loading and unloading of the payload on the ground and in flight.

4.4.1 Design Parameters and Trade Studies Investigated

During preliminary design, several key points were considered for structural analysis. These design parameters were:

- Selection of correct materials for construction
- Investigation of useable combinations of construction materials
- Weights of respective combinations and materials
- Required strengths of fuselage, wing, tail, and landing gear skins
- Landing gear design loads
- Possible reinforcement needs

During structural analysis, each possible combination of balsa and glass layers and thicknesses was investigated. Each combination was analyzed using the above figures of merit and rated according to its strength, weight, and RAC impact. Using this data assisted in meeting the structural and weight restrictions of each major component.

4.4.2 Wing Structure Preliminary Design

Because the wing was designed to attach to either side of the fuselage, each wing was modeled as a cantilever beam anchored at the fuselage and stretching to the respective wingtip. For simplicity, the lift and drag loads were assumed to act evenly along the wingtips in the proper directions. For the buckling analysis, the wing was modeled as an Euler column with one fixed and one free end. Both loads were resolved to a point load and moved closer to the wingtip to increase the bending moment and, thus, created a conservative analysis.

Some test components were built to experiment with different combinations of glass weight, balsa thickness, and resin saturation. The following table 4.2 shows the data obtained from these trials.

Table 4.2: Experimental Wing Characteristics

Construction Tests	Weight [oz]	Area [in^2]	W/A [oz/in^2]	W/A [lb/ft^2]
Test Wing [0.6oz]	3.104	146	0.0213	0.192
Test Wing [2oz]	5.079	146	0.0348	0.311
Test Wing [2oz]	4.762	146	0.0326	0.292
Test Wing [2oz]	4.162	146	0.0285	0.255
Balsa	2.222	146	0.0152	0.137
Main Wing [2oz]	12.134	275	0.0441	0.395

From the testing performed, two ounce fiberglass was found to be the ideal choice for the wing construction. The best composition consisted of 1/16th balsa sandwiched between two layers of two ounce fiberglass.

The above sample components were also used to test various spar and rib configurations. It is assumed that the skin will carry the majority of the stresses with exception to torsional stresses. The internal shear webs will account for the shear stresses and stabilize against torsion.

Pro/Mechanica was employed to perform a finite element analysis on the wing. The results of this modeling are shown below in Figure 4.3. The top left figure is displacement, upper right is strain, and lower left is Von Mises stress. The bottom right shows the convergence of the analysis. All the Pro/Mechanica analysis was performed at a sixth order, but each converged around the fourth series.

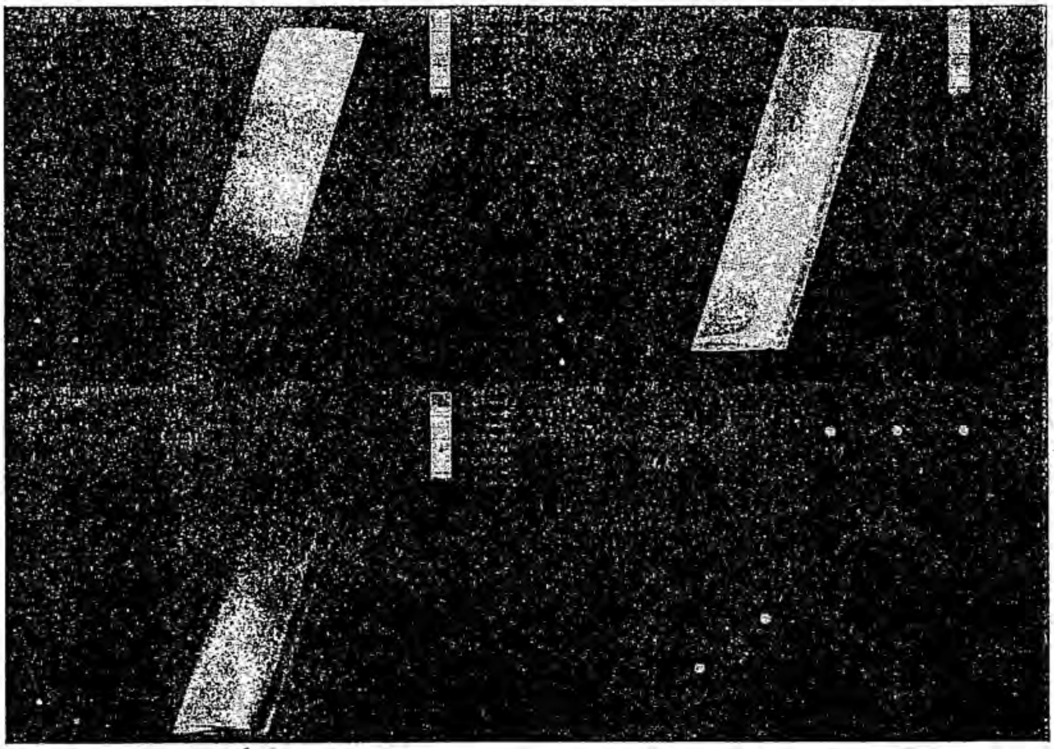


Figure 1.3: Finite Element Analysis for One Wing

One of the test specimens was tested to failure to verify the computational stress analysis. This is shown below in Figure 4.4. The experimental and analytical are consistent. The bottom left experimental picture shows skin buckling at the root. The Von Mises stress concentration from finite element analysis (bottom left picture) also shows this skin buckling at the root edge. The Pro/Mechanica analysis led to a fiberglass make-up that was stronger and thicker at the root and had less material at the tip.

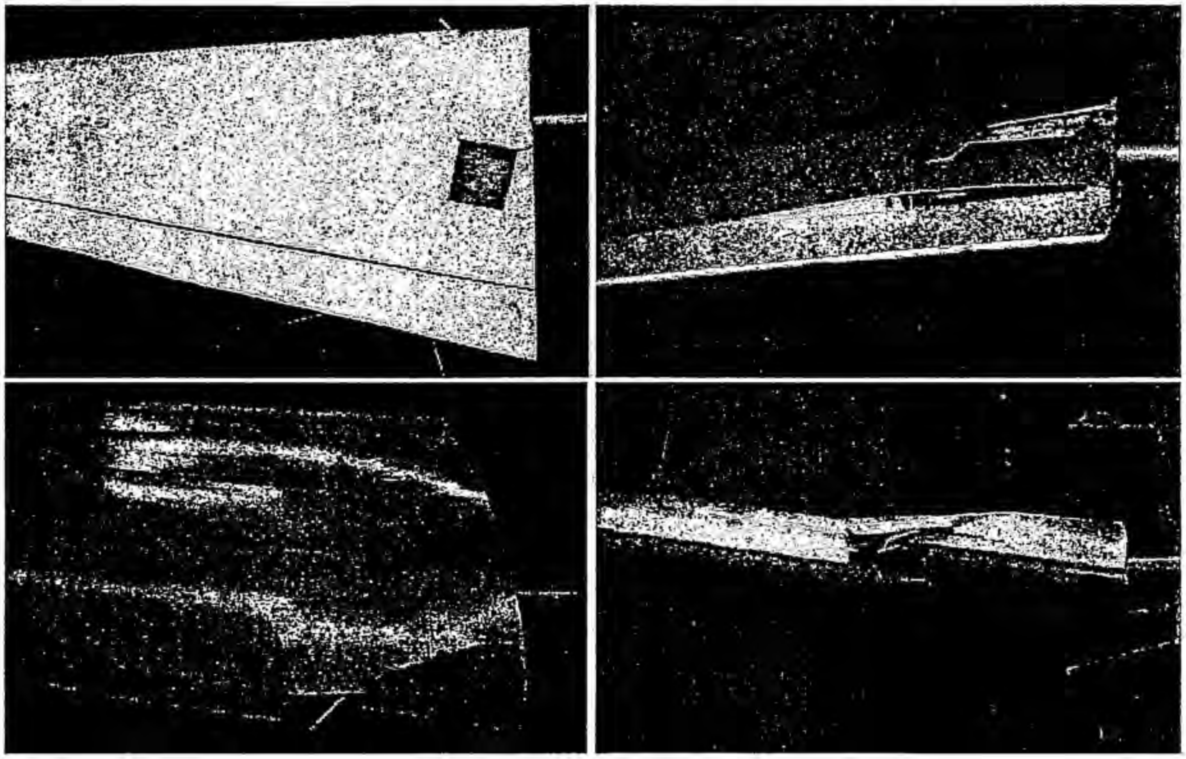


Figure 4.4 Wing Static Tests

4.4.3 Fuselage Structural Design

The fuselage was statically analyzed as a hollow cantilever beam because of the solid mold construction process. Several other simplifications were made in order to perform the analysis. The center of gravity was assumed to be fixed, not allowing for rotation or translation. Dynamic tail forces were not included due to the condition of steady, level flight. The forces applied to the fuselage were modeled as point loads when the aircraft was at full payload capacity. The payload was set at 4 liters of water. Strength comparisons for the fuselage were done with assorted glass and balsa configurations in testing and then optimized for our fuselage. Pro/Mechanica finite element analysis software was used to simulate stresses incurred by the fuselage under a 4-g loading while carrying the maximum payload. This is believed to be the most extreme loading the aircraft will experience in flight.

Due to the payload weight and the method of construction, different areas of the fuselage will carry different loadings. The lower portion will be most influenced by stresses from the payload. The upper portion will carry less loading but will still need to be strong enough to support the anchored wing. In lieu of this, a stronger glass/balsa configuration will be used on the lower fuselage portions while a lighter glass arrangement will be used to construct the upper portions in order to minimize airplane weight.

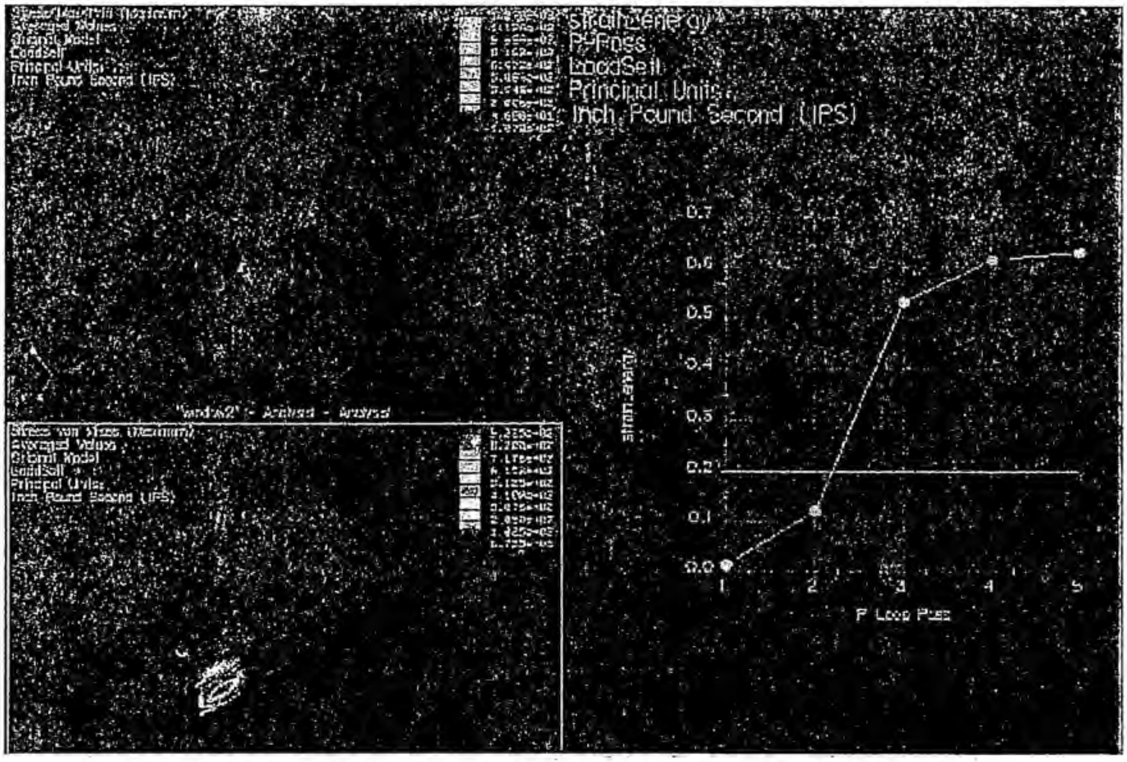


Figure 4.5: Simulation Results for Fuselage Under a 4-g Flight Loading

Finite element analysis was used to estimate the stress distributions resulting from a hard landing. The results may be seen in Figure 4.5. The data obtained from the computational stress analysis was used to better optimize the weight and strength of the individual fuselage areas.

The boom dropping out of the fuselage created some minor design concerns. However, upon further analysis, it was determined that the structure will not lose any critical structural strength due to the boom cutout. It was necessary to place a rubber or semi-stiff plastic brush guard to minimize the airflow into the fuselage from the opening. Figure 4.6 below shows the designed fuselage along with the basic tail structure.

4.4.4 Tail Structural Design

The same analysis was performed for the horizontal tail and vertical stabilizer as the wing analysis. Both the horizontal tail and vertical stabilizer can be treated like a cantilever beam anchored at the fuselage and the same lift and drag assumptions were made. To account for buckling, the tail and stabilizer were then modeled as Euler columns. See Figure 4.7.

Because the bending moment and overall forces will not be as severe on the tail as the wing, the skin will carry most of the load. Due to the reduced loads, it was possible to design a vertical tail without ribs and a horizontal tail composed only of 1/16th balsa.

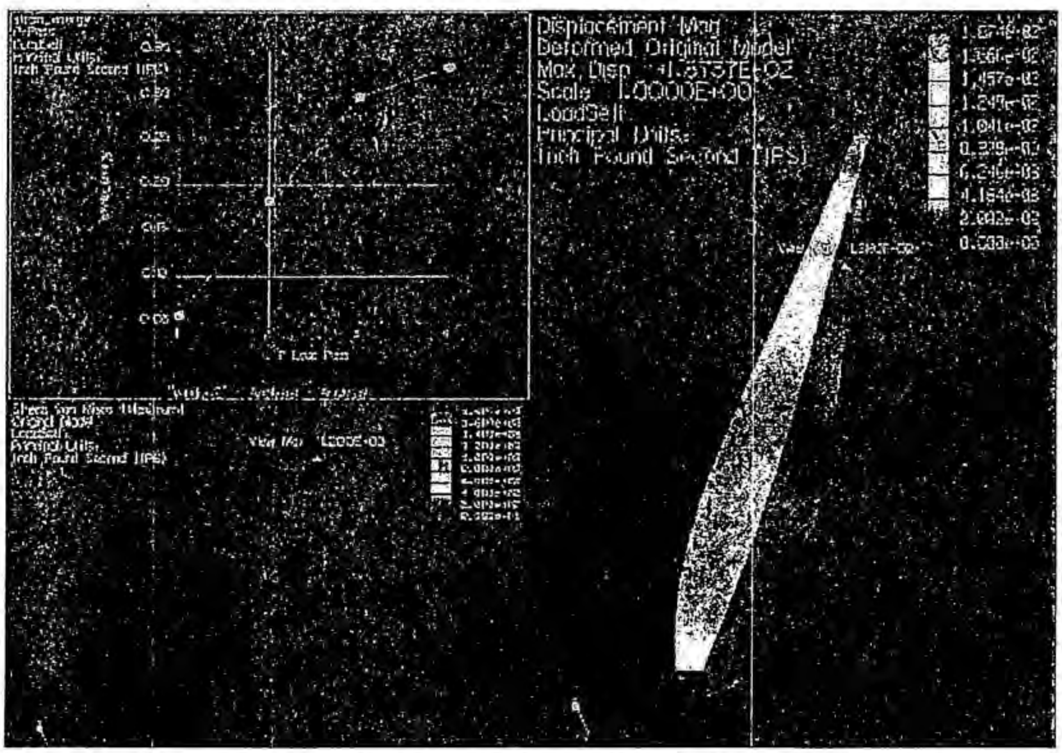


Figure 4.6 : Simulation Results for Fuselage During a 4-g Landing

4.4.5 Landing Gear Structural Design

The tail dragger configuration was chosen for the aircraft. The forward wheels will be located such that a 16-25° angle back from vertical will pass through the forward most and aft most CG points and an angle greater than 25° will pass along the wing plane through the central CG plane to prevent tipping to the left or right. This will prevent the plane from nosing forward or ground looping. The rear wheel was oriented such that it created the greatest lift and propeller clearance while minimizing the aft CG shift of the payload, usually 10-15°.

The preliminary landing gear design sweeps the main gear forward from the CG at an angle of 17.7° and spreads the wheels towards the wings at 42°. When at rest, the plane will sit at a positive angle of attack of 13°.

A Pro/Mechanica model of the landing gear was created to obtain an estimate of the stress and deflection distributions in the landing gear bow. The results of the program are shown in Figure4.8, in which deflection intensity is examined. Also included in the figure is a picture of a landing gear test specimen. This experiment found the deflection to be in the upper bow area next to the fuselage, which is consistent with the finite element analysis.

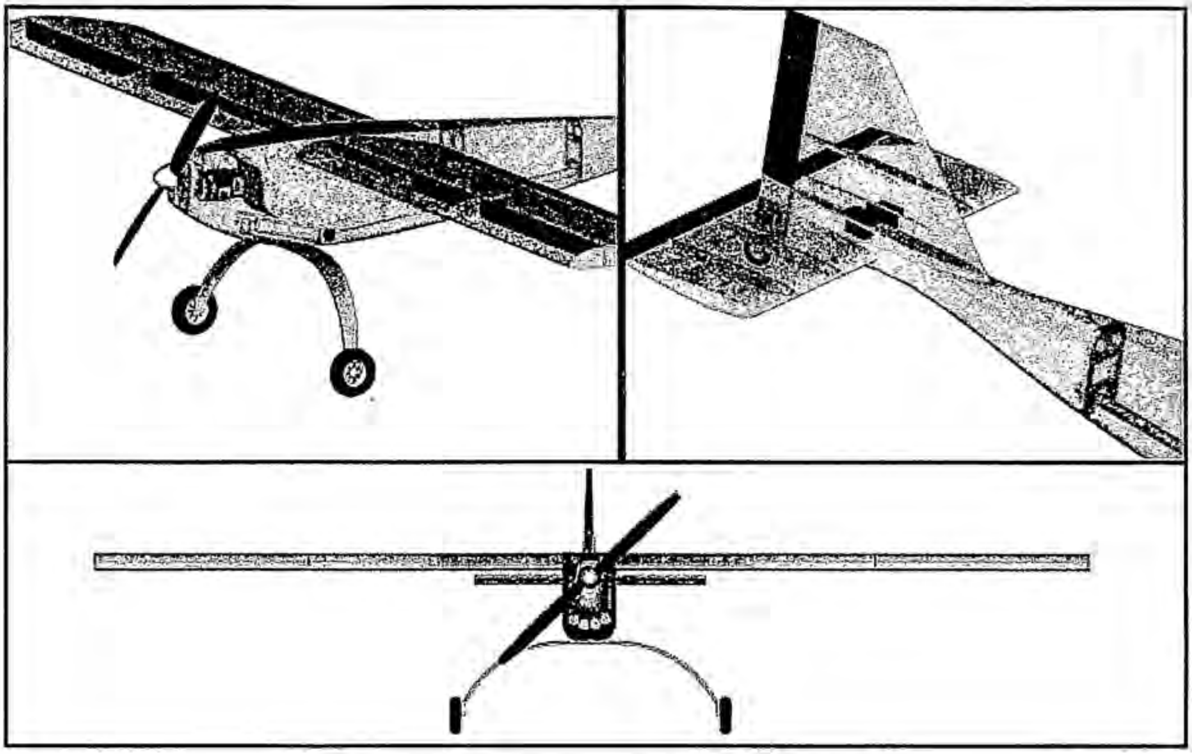


Figure 4.7 Fuselage and Vertical Tail Design

Test gear pieces were constructed and deflection was measured at the failure weight. The plane loaded with water was estimated to weigh a total of 16 pounds. The landing gear was designed with a factor of safety of 1.5, therefore warranting a loading allowance of 24 pounds. An assortment of balsa and fiberglass configurations was used as test specimens. Table 4.3, below, shows the weights of fiberglass, balsa core thicknesses, and carbon fiber mono tape widths used on each test piece. The specimens were built as linear material pieces and were tested at both a 2.5 inch and 10 inch moment arm at various weights up to failure. The final specimen listed was found to meet the weight and loading requirements and represents the gear composition selected.

The landing gear, in a solid spring bow configuration, will be attached to the fuselage using plastic shear bolts to allow disconnection in the event of a crash. A bulkhead will be added inside the fuselage to strengthen the particular attachment area.

The rear landing gear will be streamlined and hidden. The axle and wheel will be built into the base of the tail to reduce drag without inhibiting the aircraft's ability to taxi, takeoff, and land.

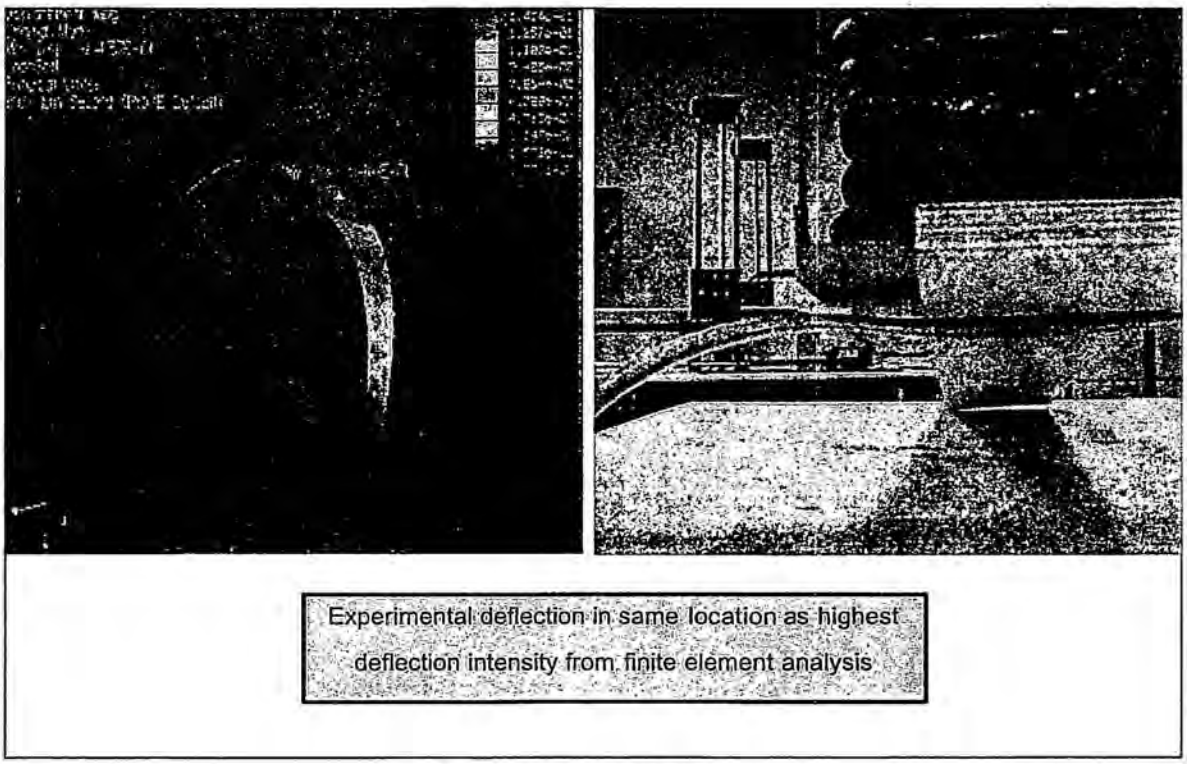


Figure 4.2: Gear Deflection

Table 4.3 Landing Gear Test Specimen Compositions

Glass Wt. [oz]	Glass Fiber Direction	Balsa Core Thickness [in]	Tape Width [in]	Overall Weight [oz]	Failure Wt. [lb] [2.5" Arm]	Deflection [in] [2.5" Arm]	Failure Wt. [lb] [10" Arm]	Deflection [in] [10" Arm]
6	0° [2] 45° [2]	1/4	0.5 [2]	0.92	16.5	1.0	4.8	2.5
3	0° [2] 45° [2]	1/4	0.5 [2]	0.78	9.89	1.0	3.5	2.3
3	0° [4] 45° [2]	1/4	1.0 [2]	1.06	31.5	1.3	7.5	3.5
3	0° [10] 45° [4]	1/8	1.0 [2]	2.15			13.5	3.3
3	0° [6] 45° [4]	3/16	1.0 [2]	2.05			19.1	3.3

4.5 Payload System Preliminary Design

Various aspects of the water payload were investigated, as follows.

1. Problems with changes in the aircraft CG due to water sloth during flight
2. Efficient loading of the plane
3. Efficient water dumping methods

4.5.1 Aircraft CG Movement

As the plane moves, the water in the tank sloshes back and forth causing the center of gravity to shift. Because the payload weighs as much as the unloaded aircraft, any changes in the CG of the payload can potentially destabilize the plane and cause loss of control. To prevent this problem, various designs were investigated by testing different types of baffles and dividers that would restrain the undesirable movement of the water in the tank.

While honeycomb baffling proved very effective, it was discarded as a solution to the water sloshing problem because of its weight, its large volume, and the complexity of making it water-proof. An alternative to honeycomb baffling which was not susceptible to the same issues was designed. Lightweight fiberglass sheets will divide the tank into multiple chambers. A space at the top of the tank will remain un-chambered to allow for a single fill port.

4.5.2 Payload Tank Shape

The water tank is designed to fit in the front quarter of the fuselage with a volume of exactly four liters (245 in^3). The shape of the container will be dependent on the geometry of the fuselage and will have a size of approximately 4 by 8 by 12 inches. Glue will bond the tank snugly to the fuselage walls. The tank will taper slightly to a bottom center point, where it will be fitted with the boom valve.

4.5.3 Payload Release Mechanism

A boom was employed to increase the effective head pressure during payload release. While it is true that the longer the boom, the more pressure, there is a point of diminishing return. At this point, the small increase in head does is not worth the increase in boom weight and length. According to the Bernoulli analysis, this length was approximately 18 inches, which was the design decided upon.

The boom was to be attached to the belly of the fuselage. During take off and landing, the boom was to be aligned so that it lies parallel to the fuselage, and then deploys during dumping. A ball valve was designed to open as the boom deployed and close with the boom in the upright position. A servo and gear assembly will actuate this mechanism. A thin fiberglass shell in the shape of an airfoil will cover the boom to minimize drag.

4.5.4 Payload Ground Handling System

The timed extraction of the four liter liquid payload must be accomplished thoroughly and reliably. An inexpensive plunger-type pump made from plumbing hardware was chosen to meet these needs. The loading system will be managed by a two-person team and will load the vessel with a single stroke cycle. Water from the supply tanks will be quickly drawn into the pump chamber with an upward stroke and will be evacuated during the down stroke through a large hose. Check valves will facilitate the intake and exhaust movements during the cycle. At the entry location atop the aircraft, a conical fitting engages a normally-closed flap valve which serves to prevent spillage during flight. Team members will be able to quickly disconnect the manifold with minimal spillage and will be able to transport the pumping system to the loading area with nominal effort. This concept is shown in Figure 4.9 below.

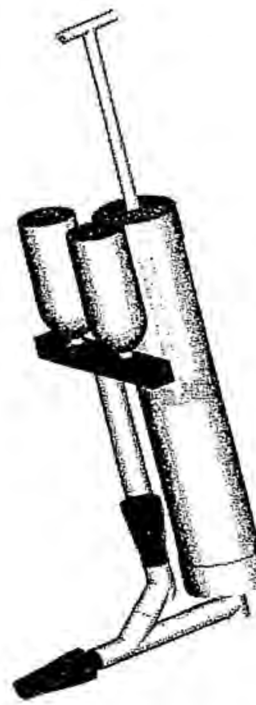


Figure 4.9: Payload Loading System

4.6 Propulsion System Preliminary Design

The propulsion focus during preliminary design was on sizing parameters for the battery packs and propellers. Using an optimization program in conjunction with aerodynamic consideration, optimum power and RAC was found for each wind conditions of 2, 8, 15 and 25 mph winds for take-off.

The program aided in optimizing gear ratio, propeller diameter, pitch to diameter (P/D) ratio, battery size and battery number. By taking into account the weight of the loaded aircraft, parasite drag, and airfoil type, take-off distance could be found by changing the propeller, gear ratio, motor, battery size, and battery number.

4.6.1 Propulsion System Configuration

The propulsion system components needed to be kept as close together as possible in order to minimize power losses. It also needed to be placed in an optimal location for CG control. All parts of the system were located closely to one another, which reduces the amount of power loss in the system. The 40-amp fuse for the quick disarm system was located between the speed controller and motor to avoid overloading the motor.

With the propulsion system components located so closely together, it is crucial to provide a sufficient cooling system. A chin scoop located at the front of the plane just below the nose cone will provide ample cooling for the motor and batteries. A vent will be located behind the nozzle of the water tank allowing for the air to escape.

4.6.2 Motor

Comparison of the AstroFlight Cobalt 40 and Graupner 3300/6 motors proved that although the Cobalt 40 is lighter weight, the efficiency of the Graupner 3300/6 far exceeds that of the AstroFlight motor. This allows for fewer batteries to be used during the competition, which lowers RAC. Not only is the Graupner more efficient, it also creates more thrust than the Cobalt. Thus, the Graupner 3300-6 will be used during the competition.

4.6.3 Battery Number and Size

Battery type was chosen based on several factors. Among the candidates for batteries were Sanyo CP-2400SCR, CP-1300SCR zapped, SR900 Max, and SR650 Max. While each battery provided 1.2 volts, they provided capacities of 2100 milli-amp-hours, 990 mah, 810 mah, and 585 mah, respectively.

Figure 4.10 shows the results of the optimization program for the performance of the propulsion system. These graphs show battery sizing, take-off distance, required thrust, and thrust for various propulsion systems respectively.

Once the size of battery was chosen, the number of batteries needed could be determined. Figure 4.10(b) displays the necessary take-off distance for various wind speeds for 14, 15, or 16 batteries. Because battery weight is so important to the RAC, 16 cells were chosen as the minimum number of cells to power the propulsion system. Though the optimization program shows that 16 cells cannot provide enough power for take-off in 150 feet in the event of no wind, there are options such as lessening payload weight, or overloading the fuse which will provide enough power for take-off in these conditions. Because the Graupner 3300/6 motor can handle upwards of 60 amps, pulling more than 40 amps through the fuse during the few seconds of takeoff exhibits no detrimental repercussions to the motor.

4.6.4 Battery Configuration

The optimum design location of the batteries would have all sixteen together in one pack in order to minimize wire resistance losses. The batteries also needed to be placed as far forward in the fuselage as possible for CG control. It was decided that the batteries would be broken into two smaller packs of eight each due to it not being possible to fit the entire pack in the front of the plane. The two packs will still be placed very close to each other, minimizing loss due to wire resistance.

Endurance and Mission testing were performed on the Sanyo 1300 batteries to ensure that they had enough power to last the two fully loaded take-offs. The second fully loaded take-off was found to be the most critical part of the mission. Results of mission testing can be seen in Figure 4.11. Although not designed to handle more than 30 amps, it was found that the sixteen pack battery would allow for the high current pulled by the Graupner motor during take-off. Consecutive testing proved that the batteries could withstand multiple runs without any significant loss in capacity.

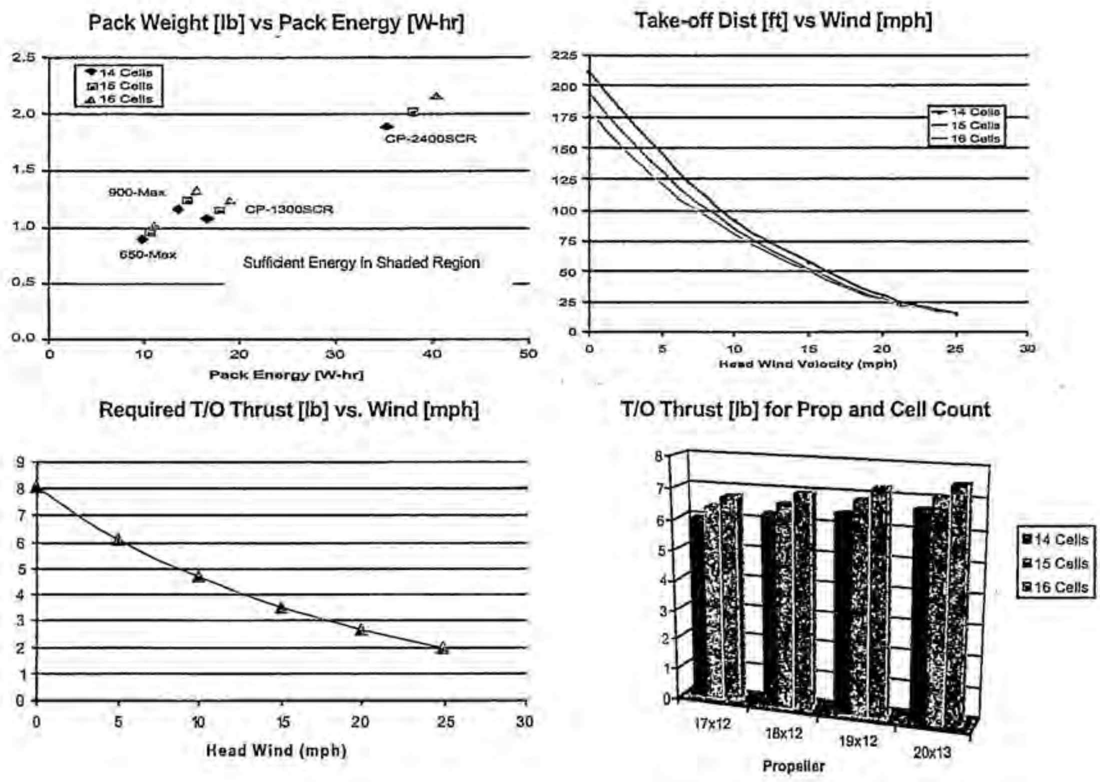


Figure 4.10 : Propulsion System Optimization Results

4.6.5 Propeller and Gearbox Configuration

Several wind and propulsion system scenarios were optimized through simulation programs. Using the program, the propulsion system behavior could be simulated in order to predict the operating conditions for all stages of the flight. Using a flight simulation program a prediction of how the system would perform was created, taking into account all aspects of aircraft weight, aerodynamics, and mission times. Table 4.4 shows several combinations of propulsion system designs to optimize score and efficiency at wind conditions ranging from 2 mph to 25 mph.

Propeller diameter and pitch were optimized using an optimization program. These propellers were then tested on a motor dyno. Initial testing proves that the Bolly 18X12 propeller out performs the APC 18X12 propeller. Therefore, the Bolly propellers will be used during the competition when possible.

Testing between the Motor Electronics Corporation 'Superbox' and the AstroFlight gearbox were conducted based on weight and versatility. The Superbox is much lighter weight and allows multiple gear ratios by allowing the spur gear to be interchanged. A slight modification to the Graupner motor, by way of a spacer, allowed for the Superbox to be firmly mounted to the motor. It was decided to use the Superbox instead of the AstroFlight because of these advantages over the AstroFlight.

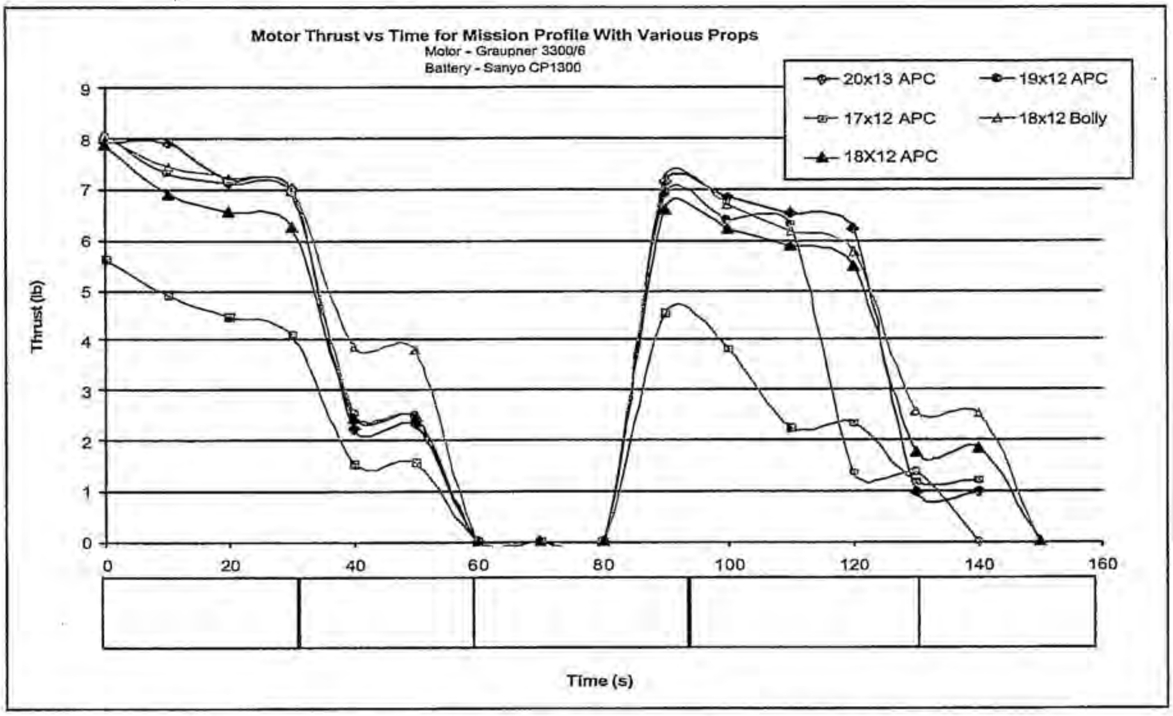


Figure 4.11: Mission Profile Using Optimized Propeller and Gearbox Configurations

These tests show that the following propellers provide enough thrust for both take-offs: APC 20X13E, 19X12E, 18X12E, and Bolly 18X12.

Table 4.4: System Optimization for Various Wind Conditions

Wind Speed (mph)	Battery Number	Propeller Diameter (in)	Propeller Pitch (in)	P/D Ratio	Gear Ratio	Take-off Distance (ft)	Optimum Cruise Velocity (ft/s)
2	16	19	13	.68	1.7	150	78
2	16	18	13	.72	1.5	150	71
8	16	15	13	.87	1.2	138	82
8	16	16	15	.94	1.3	123	81
15	16	15	13	.87	1.8	111	78
15	16	15	14	.93	1.3	75	82
15	16	16	15	.94	1.3	67	81
25	16	15	14	.93	1.3	22	84

25	16	16	15	.94	1.3	18	84
25	16	17	15	.88	1.4	18	84

4.6.6 Speed Controller

After testing of the Jeti JES60 and AstroFlight speed controllers, it was determined that the two were very close in efficiency. Testing was conducted by attaching a thermocouple to the speed controller and measuring temperature change while running a steady current of 40 amps through it. Although the Jeti JES60 is smaller and lighter weight, the motor breaking feature caused difficulties during testing of larger propellers. Turning off the breaking option only allowed for use of half of the radio throttle control. It was decided to use the AstroFlight 204D model because it is more reliable than the Jeti model.

4.6.7 Disarming System Design

The outer housing is 1 ½ inches in diameter and is made of fiberglass. In the center of the circular top is a slot that is only slightly larger than the actual 40 amp fuse. The top surface of the fuse will be flush with the top of the fuse disarming housing as well as the surface of the airplane. The sponge (which acts like a spring) is placed between the plastic circular top and the plastic stationary plate at the upper and lower interface, respectively. The sponge is held into place on the sides by a thin-walled plastic cylinder. Connected to the bottom of the stationary plate are two connectors that will receive the fuse blades when the fuse is inserted. The connectors will be held in by a high strength glue or epoxy. In order to make the insertion of the fuse into the connectors easier, the inlet to each connector has been tapered, which will guide the fuse blades into the connectors. The top of the housing is allowed to move within the cylinder. To insert the fuse, simply place the fuse into the slot and press on the top surface of the fuse until the blades are secure in the connectors. To remove the fuse, press on the circular top being careful not to obstruct the fuse. The top slides past the fuse exposing its sides. The fuse will be gripped with fingers and pulled.

5 Detail Design

After preliminary sizing and component arrangement was complete,

5.1 Configuration Details

Table 5.1: Longitudinal Stability Coefficients Estimation

Longitudinal Stability Coefficients Estimation						
	X-Force Derivatives		**Z-Force Derivatives**		**Pitching Moment Derivatives**	
α	Cx_α	0.342			Cm_α	-1.221
$\dot{\alpha}$			$Cz_{\dot{\alpha}}$	-1.208	$Cm_{\dot{\alpha}}$	-3.624
q			Cz_q	-3.256	Cm_q	-9.768

δ_a			$Cz_{\delta e}$	-0.261	$Cm_{\delta e}$	-0.783
Lateral Stability Coefficients Estimation						
	Y-Force Derivatives		**Yawing Moment Derivatives**		**Rolling Moment Derivatives**	
β	Cy_β	-0.252	Cn_β	0.098	Cl_p	-4.666
p	Cy_p	0	Cn_p	-0.061	Cl_r	0.129
r	Cy_r	0.153	Cn_r	-0.059	$Cl\delta_r$	0.00721
δ_r	$Cy_{\delta r}$	0.052	$Cn_{\delta r}$	-0.020		

5.2 Projected Performance

Table 5.2: Projected Performance Values of Each Segment of the Fire Bomber Flight

Mission Components For Fire Flight	# Times Performed	Time Spent	Distance	Velocity
Takeoff/Loaded	2	4.43 s	86.52 ft	50.95 ft/s
Climb/Loaded	2	6.25 s		54.47 ft/s
Turn 1: Loaded	2	9.6 s		
Acceleration: Unloading	2	10.09 s	721.13 ft	
360° Turn	2	3.6 s	294.88 ft	
Cruise: Unloaded	2	8.7 s		82 ft/s
Turn 2: Unloaded	2	1.8 s	147.44 ft	
Slow Down: Unloaded	2	5.51 s	359.67 ft	
Ground Time	1	30 s		
TOTAL TIME	—	1.33 min		

The predicted score for the fire mission is given below. This assumes a 100% on report score.

$$Score = \frac{DF \cdot Payload\ Weight}{Time \cdot RAC} = \frac{2 \cdot 17.6}{1.33 \cdot 6.758} = 7.83$$

5.3 Structural Details

5.3.1 Wing Details

- **Skin Makeup**

For the main wing, the balsa and fiberglass method of monocoque construction was used. The wing was built using 1/16 inch think competition balsa and fiberglass coating. The glass thickness was transitioned to a lighter ply makeup at approximately 75% of the span from the fuselage. At the inner three-quarters of wingspan, the inner and outer skin of the balsa sandwich was coated with two ounce fiberglass sheets. For the end quarter of the wing span, two layers of 0. 6 ounce fiberglass was used to coat the outside balsa surface and 1 layer of 0.6 ounce was used to coat the inside balsa surface.

- **Internal Structure**

The internal structure of each side of the wing consists of three ribs, three shear webs, and the wing-to-fuselage attachment components. The principal shear web is located at approximately quarter-chord and is constructed of 1/16 inch balsa cut along the end-grain. Another web is located just forward of the flaperon hinge, while the third is located just aft of the hinge. The shear webs are used to distribute the loads, add strength, and maintain flexibility of the wing. The inboard rib is made of plywood, while the three internal ribs and closeout rib are made of 1/16 inch balsa.

- **Connection Method**

The wing is attached to the fuselage by a pin that extends outward from the wing into the fuselage. A connecting carbon fiber rod extending from the wing inserts into a tube that spans the fuselage. The rod runs about three inches spanwise into the wing. A plastic fastener on the aft end of the wing is used to further secure the wing into place. This pin is used for anti-rotation and will also function to "snap" the wing into the fuselage. Clear packing tape additionally binds the wing to the fuselage.

- **Flaperon**

The flaperon combines the function of an aileron and a flap. The flaperon was placed at 10% of the chord length and is attached to the wing by a strip of Kevlar sandwiched between layers of fiberglass.

- **Servo Placement and Design for Flaperon**

The flaperon servos are placed on the middle ribs on each of the wing sides. They are located completely inside the wing to eliminate undesirable drag. The servos are accessible through service hatches on the upper surface of the wing.

5.3.2 Fuselage Details

Because of the RAC measuring the maximum points of the fuselage, the optimized fuselage shape was a square or a rectangular shape to maximize the volume. Since the major concern of evacuating the water from the fuselage during flight was maintaining as much head pressure as possible, a rectangular shape with more depth than width was chosen. However, the corners of the fuselage are rounded, and parts are blended to decrease drag and to create a sleeker shape. In determining the size of the fuselage, the housing of all of the components, especially the motor, batteries, and water tank, were of utmost importance. The fuselage shape was determined from the water tank design; thus, the cross-sectional area and, to some extent, the length of the plane were established by the tank. After building the fuselage shape and size around the tank, the fuselage was lengthened to encompass the motor, batteries, boom, and servos. The propulsion items were placed so as to minimize the wire lengths between components as well as to keep the CG of the plane in the front. A chin scoop was decided upon in order to help the cooling of the closely packed propulsion components. A side hinged engine access door is also located on the top of the fuselage to allow an entrance to the engine components.

5.3.3 Empennage Details

- **Horizontal Tail Details**

The horizontal tail is a balsa wood build-up with a Monokote coating to give it a smooth surface. The balsa wood build-up was chosen because of its light weight properties. Because the horizontal tail is not subjected to as high of stresses as the wing and does not have to be as strong and rigid, the mold method using balsa and fiberglass was not needed. This method can be constructed quickly and easily and does not require a fiberglass coating or mold.

The horizontal tail is attached to the plane through a hole in the fuselage. Glue will be used to permanently connect the tail securely into the fuselage cavity. The fuselage will be reinforced with 1/8" balsa doubler around the juncture. The elevator hinge will be a skin Monokote hinge spanning across the length of the horizontal tail. The servo is located in the interior of the horizontal tail to prevent drag.

- **Vertical Tail Details**

The vertical tail will be constructed with the fuselage. It will be constructed during the molding process of the fuselage, but will transition into lighter glass weight. Competition balsa (1/16 inch thickness) will be sandwiched between layers of 0.6 ounce fiberglass. The hinge rotates using flanges; three flanges are located on the top of the rudder and a connecting pin attaches the top and bottom portions of the rudder together. The tail wheel is also located on the bottom of the rudder. Servos for the rudder are completely enclosed in the interior of the back portion of the plane.

5.3.4 Landing Gear Details

Multiple lay-up variations for the landing gear were tested to failure in order to collect data. A solid spring bow landing gear configuration swept forward 30° was decided upon. A mold for the landing gear configuration was created in order to test the specific design with the maximum weight of the plane. Looking at a front view of the plane, the two main front tires will be located 42° from the centerline and at a 17.7° angle from a line drawn through the CG to a line perpendicular to the plane through the wheel. The front tires have a diameter of 2.5 inches and the rear tire's diameter is 1.25 inches.

5.4 Propulsion System Details

The propulsion team performed several tests on the suggested preliminary design concepts. Results of theses tests are shown below in the Component Selection section.

5.4.1 Component Selection

Each component of the propulsion system was tested for efficiency, endurance, power, and reliability. The results of these tests are as follows:

Battery Pack

The Sanyo CP-1300SCR battery will be used during the competition. Two eight celled packs, connected with a six inch lead, will be placed in the fuselage under the water tank.

Propeller

Depending on the wind speed, a 20X13E, 19X12E, 18X12E, or Bolly 18X12 will be used for the Fire Bomber mission. Smaller diameter propellers such as a 15.5X13 Bolly or a 16X10 will be used for the Ferry Run, when thrust is not as important as speed.

5.4.2 System Architecture

The general setup for various propulsion systems will consist of the same battery configuration, speed controller, fuse, and motor. The propeller and gearbox can be altered prior to each mission to optimize flight at any wind condition.

Table 5.3: Stability Coefficient Characteristics

Longitudinal Stability Coefficients Estimation						
	X-Force Derivatives		**Z-Force Derivatives**		**Pitching Moment Derivatives**	
α	Cx_{α}	0.342			Cm_{α}	-1.221
$\dot{\alpha}$			$Cz_{\alpha*}$	-1.208	$Cm_{\alpha*}$	-3.624
q			Cz_q	-3.256	Cm_q	-9.768
δ_e			$Cz_{\delta e}$	-0.261	$Cm_{\delta e}$	-0.783
Lateral Stability Coefficients Estimation						
	Y-Force Derivatives		**Yawing Moment Derivatives**		**Rolling Moment Derivatives**	
β	Cy_{β}	-0.252	Cn_{β}	0.098	Cl_p	-4.666
p	Cy_p	0	Cn_p	-0.061	Cl_r	0.129
r	Cy_r	0.153	Cn_r	-0.059	$Cl\delta$	0.00721
δ_r	$Cy_{\delta r}$	0.052	$Cn_{\delta r}$	-0.020		

5.5 *RAC Calculation/Weight Statement*

Coefficient	Description	Computation	Value	Total
MEW	Manufacturer's Empty Weigth	300*Weight	8.2 lb	**2460**
REP	Rated Engine Power	(1+.25*(# engines-1)) *Total Battery Weight (lbs)	1 Engine 1.24 lb Battery	**1860**
MFHR	Wings	10 hr/ft^2	10 in Chord 75 in Span	52.08
	Flaperon	5 hr	1.5	7.5

	Fuselage	20 hr/ft^3	43" x 5" x 6.95"	17.295
	Empenage (Vertical)	10 hour	1 Vertical	10
	Empenage (Horizontal)	10 hour	1 Horizontal	10
	Flight Systems	5 hr	5 Servos	25
		Σ WBS hours	20	**2437.5**
RAC	**Rated Aircraft Cost**	**(A*MEW+B*REP+C*MFHR)/1000**		**6.758**

FIGURE 5.2: Rated Aircraft Cost Calculations

5.6 Aircraft Configuration Summary

Given below is a table of all of the final aircraft dimensions.

Table 5.5: Final Plane Parameters

Fuselage		**Weight Statement**	
Length (ft)	4	Airframe (lb)	7.018
Width (ft)	0.4219	Propulsion System (lb)	2.68
Height (ft)	0.5792	Control System (lb)	0.46
Wing		Payload (lb)	8.8
Area (ft^2)	5.2	Empty Weight (lb)	8.258
Span (ft)	6.45	Gross Weight (lb)	17.058
Aspect Ratio	8	**Systems**	
Flaperon Area (ft^2)	1.043	Radio	Futaba T8UAP (PCM)
Airfoil	SD7062 with flaps	Servos	[4] JR DS3421SA
Empennage		Speed Controller	AstroFlight 204D
Horizontal Tail Area (ft^2)	0.9375	Battery Type	Sanyo CP-1300SCR
Horizontal Tail Airfoil	NACA 2410	Battery Configuration	18 cells, 2 packs
Vertical Tail Area (ft^2)	0.6736	Motor	Graupner 3300-6
Vertical Tail Airfoil	NACA 0009	Gear Ratio	1.82 : 1
Elevator Area (in^2)	27	Propeller	18x12 Bolly
Rudder Area (in^2)	9.7	Landing Gear	Bow-type Taildragger

5.7 Drawing Package

6 Manufacturing Plan and Processes

The Figures of Merit used dictated the decisions that were made about the manufacturing processes. The Figures of Merit were chosen to help in distinguishing between the various methods of manufacturing available to the team by highlighting the advantages and disadvantages associated with the different techniques. This process allowed the team to select the manufacturing process that was most ideal for our selected aircraft design, time schedule, and skill level. The Figures of Merit chosen are discussed below in more detail. Available processes are evaluated using these criteria and given a weighting factor.

6.1 Manufacturing Process Figures of Merit

- Required Skills: While a design may be feasible in its conceptual phase, it must also be feasible during its manufacture and construction. The manufacturing process must be within the skill level of the team.
- Accuracy and Surface Finish: This criterion is important in the transition from the prototype to the final phase. It will allow the team to replace or improve parts as needed. Each part must be able to be accurately repeated in the event of a part needing to be replacement.
- Cost: The element of cost is a limiting factor in any design. The manufacturing process selected must be financially feasible and appropriate for the given budget. The materials may vary for the different manufacturing techniques, and the cost of these items were considered.
- Fabrication Time: The ability to meet the deadlines of the project is critical to the success of the airplane. The manufacturing process chosen must be able to be completed within the allotted time. This process must provide adequate time for the prototype and final aircraft to be manufactured while still allowing time for testing and improvements.

6.2 Process Alternatives

During the conceptual design phase, a monocoque construction scheme was chosen which utilized a composite/balsa sandwich structure. Different monocoque manufacturing techniques were considered in order to optimize this type of design.

- Conventional Build Up: This process involves the manufacture of the airplane by building up the individual parts by balsa wood and applying layers of fiberglass. This process does not require the building of molds and produces the lightest parts. While this process has a more rapid construction speed than the molded method, it is lacking in repeatability and the ability to keep true to the original airfoil.
- Layup Over Plug: This process involves the construction of the plug using foam and fiberglass. The skins are layed-up over the plug and then removed. This process achieves an accurate inner mold line, but it produces an unfavorable rough outer mold line surface.

- Full Mold: Individual molds are constructed and used to build the components by laying sheets of fiber glass and balsa wood into the mold. This method is highly repeatable, produces an accurate airfoil, and creates smooth and accurate outer mold line. The construction of the mold is a time intensive process but gives the part the most strength.

6.3 Process Selection

After evaluating the different manufacturing alternatives based on the selected manufacturing Figures of Merit, the full mold process of manufacturing was chosen for the wing, fuselage, and vertical stabilizer portion of the empennage because it provided the highest repeatability, trueness to original dimensions, and an accurate outer mold line. The method of manufacturing chosen for the horizontal component of the empennage was the built-up method because this section does not require as much strength. The built-up process creates a lighter part while saving time and labor. The landing gear required a curved shape that was lightweight but could also support the applied load. The manufacturing method chosen to meet these requirements was laying up fiberglass and shaped balsa over a curved piece of foam and then utilizing a vacuum bag. This process was selected because it did not require the construction of a mold, which saved time, and provided the curvature needed within tolerable dimensions.

Skills Matrix

The skill matrix below shows the important skills necessary for the completion of the project. The number of team members possessing the identified skills are shown below, as well as the importance of each skill, ranging from 0 to 2, on the manufacturing of the wing, fuselage, empennage, and landing gear.

Table 5.6: Skills Matrix

Skill	# of Team Members	Wing	Fuselage	Empennage	Landing Gear
RAC analysis	3	2	2	2	2
CNC Foam Cutting	3	2	2	2	1
Composite Layups	12	2	2	2	2
Woodworking	6	1	1	1	0
Propulsion System Analysis	4	2	2	1	1
CNC Machining	1	0	1	0	0
Technical Writing	5	2	2	2	2
Conventional Balsa Structures	1	0	0	2	0
Pro Engineer Modeling	3	2	2	2	0

6.4 *6.3 Manufacturing Plan*

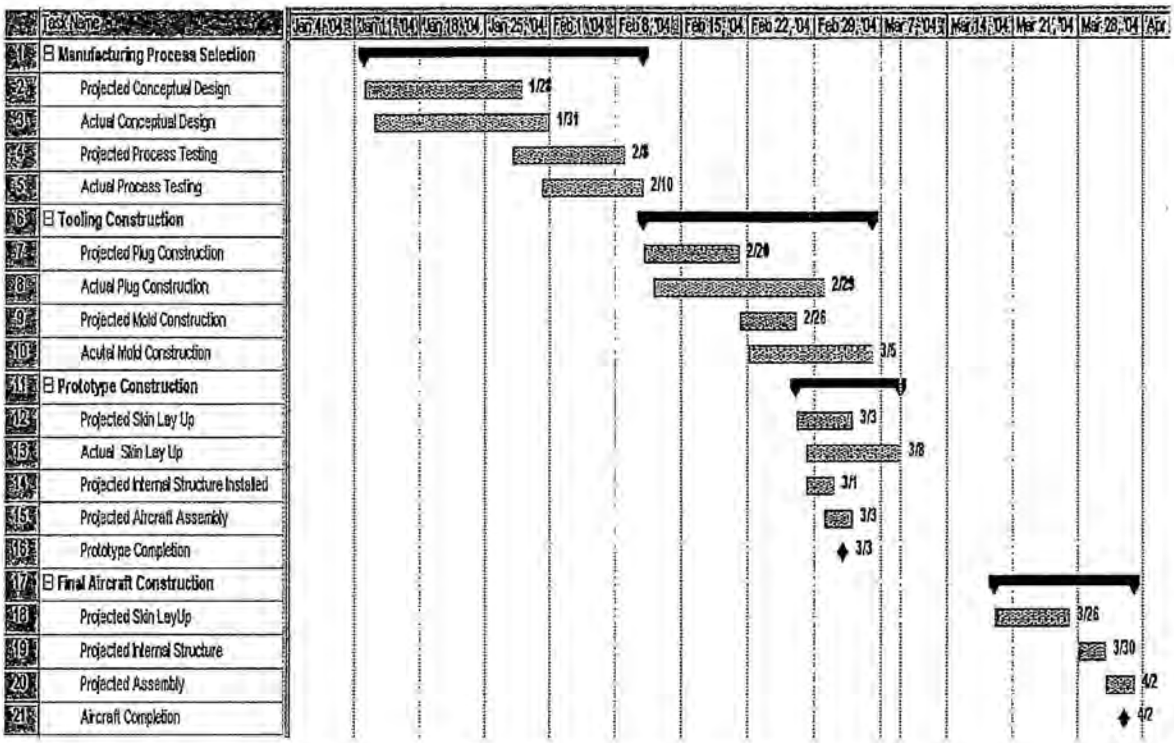


Figure 6.1: Manufacturing Waterfall Timeline

7 Testing Plan

7.1 *Testing Plan*

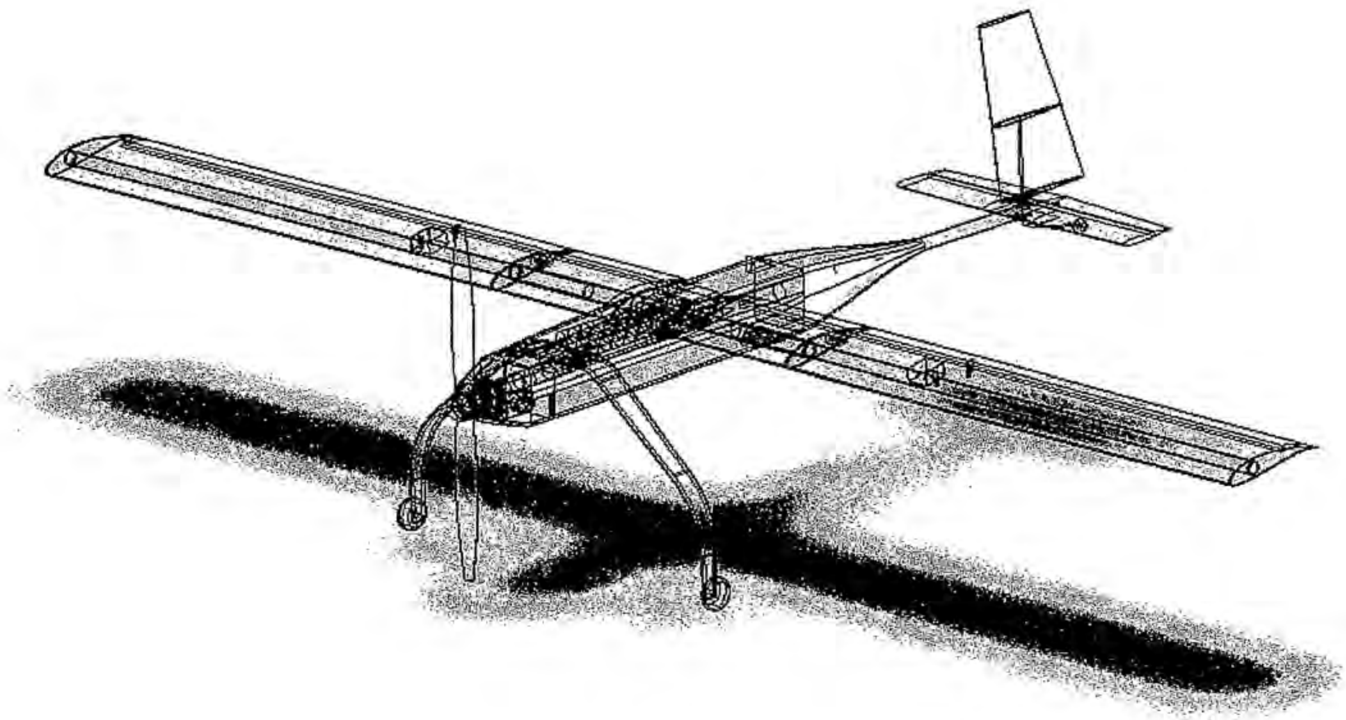
Many design tradeoffs can only be verified during actual flight testing. These tests prove or disprove that all aircraft systems are working together correctly as well as working in harmony with the designer's intent. During these tests safety to testing personnel is paramount precautions are taken when ever the aircraft motor is armed. Also in accordance with safety checklists are developed to insure that the aircraft is flight worthy. The test procedures are designed to step up in complexity culminating in the first full-up mission sortie.

7.2 *Test Cards*

Individual tests of aircraft systems such as the propulsion system and the major structural components carried out before final assembly. These systems are tested again once they are integrated into the prototype.

University of Southern California

SCquirt



AIRCRAFT DESIGN REPORT

2004

TABLE OF CONTENTS

1.0 EXECUTIVE SUMMARY:

The University of Southern California Aero Design Team (ADT) has participated in the Cessna ONR AIAA Design/Build/Fly Competition for several years with consistently high results. The nature of the competition is to design, build and fly an electric, radio-controlled aircraft for specific missions outlined in the official rules. For the 2004 competition, two mission profiles were given: *Firefight* and *Ferry*. Each mission had its own difficulty factor and specified tasks. Both missions had to be flown with the same aircraft and the highest flight score of each contributed to the total flight score. The goal of the Aero Design Team was to develop a solid configuration that would most efficiently complete these two missions. To accomplish this goal, the design focus was divided into three sections: Conceptual Design, Preliminary Design, and Detailed Design.

1. Conceptual Design: During this phase, the team focused primarily on satisfying the requirements of the Firefight Mission. Because of the scoring for the two missions, the firefight mission was found to contribute more to the total score than the ferry mission. When considering the propulsion system, fuselage configuration and possible airfoils, the maximum water payload was assumed. Six different conceptual configurations were analyzed according to a set of ten figures of merit. Broad aspects such as tail configuration, wing configuration, fuselage and payload integration, and estimated cost were the main points of debate for each of the conceptual plane designs. The size of the packing box was also taken into consideration. This parameter however, did not carry as much significance as it did last year due to the change in the rules which eliminated the timed assembly.
2. Preliminary Design: During this phase, the main analytical tool was updated and implemented. An Excel Program called "PlaneSizer" was used last year to assist with the sizing of the plane. Originally based on an older code, PlaneSizer was reformatted by current students to improve the iteration process. This program contained a main page, labeled "Frontpage" which compiled all pertinent inputs and summarized all necessary outputs in one single location. Referencing codes on individual pages separated for different portions of plane design, the Frontpage allowed for concise plane sizing in one central location. It was designed to use different libraries established for different parts such as motors or airfoils and was divided into logical subsections with the ability to perform different trade studies. The code performed the iterations part of the design, outputting the plane's sizing and configuration based on the assigned inputs. By using PlaneSizer in the preliminary design portion, the team was able to establish an estimate of the number of cells, the type of batteries, and motor and propeller size that would produce an optimal

propulsion system for the specified flight mission. With the imbedded iterations, PlaneSizer was also able to select an appropriate airfoil and tail configurations that would satisfy the requirements of the given flight mission. PlaneSizer enabled an efficient downselect of plane configurations and sizings and provided a sufficient base from which the team could further analyze before finalizing the design. A number of lab experiments were performed to verify the accuracy of the calculated results. These test were primarily focused on propulsion system optimization and confirming manufacturer's specifications.

3. Detailed Design: During this phase, the team froze the design layout, sized the plane's components and began some preliminary manufacturing of different parts. Sizes and configurations obtained from this stage were based on initial assumptions and parameters that were iterated through the Excel programs. Detailed CAD drawings were made for all the components of the plane to enable a smooth transition into and completion of the building process.

2.0 MANAGEMENT SUMMARY:

2.1 General Team Structure:

The Aero Design Team consisted of approximately twenty undergraduate students organized into subgroups for different disciplines. Cristina Nichitean was the student captain of the team and was responsible for overall team productivity and coordination. She established weekly and monthly agendas, schedules and deadlines. She was responsible for keeping members on assigned tasks, for running meetings and for overseeing the design and construction phases. The rest of the student members were divided into subgroups, each with a designated team captain, a junior or senior with prior team experience. Team captains and the team leader met once a week to divide up tasks, to consolidate design ideas, to brainstorm, and to solve any problems facing the team at the time. Captains then met on an individual basis with their team members to negotiate and complete the designated tasks. In addition, the entire team, students and advisors, met for three hours a week to communicate and discuss the current state of the plane, to update others on the progress of the week prior and to plan ahead. The team was assisted by a faculty advisor, Dr. Ron Blackwelder and several industry advisors from Aerovironment Inc, Raytheon, and Swift Engineering. The advisors served as a resource of experience and supervision and provided several lectures early in the design process for newer team members. They used their expertise to help with design and construction techniques with which the student members were less familiar.

2.2 Subgroups and Individual Assignments:

The team was divided into nine subgroups, each focusing on a specific portion of the plane or flight procedure. Figure 2.2.1 shows the team organization. Each captain was chosen based on team participation, background experience, and interest. To promote learning, underclassmen and new members volunteered to help with each group. Captains were responsible for organizing their group, communicating and coordinating with the rest of the team and ensuring all deadlines and goals were achieved.

The main computational tool used by the team was the Excel program titled "PlaneSizer." This program contained pages for each of the different subcomponents of the plane design. To support this tool, each captain was also responsible for creating and updating an Excel spreadsheet with the appropriate parameters for his or her specific focus. With the help of advisors, the analysis for each spreadsheet was completed and integrated into the main design tool. The development of these work pages was carried further into the three design phases where intense analysis and optimization were performed to obtain the final configuration of the aircraft. Each group was responsible for planning and building the components under the supervision of more experienced team members and industry advisors. The subgroups were defined as follows:

1. Aerodynamics Group: The main responsibility of this group was to choose the most efficient airfoil for the flight objectives and to estimate the total airplane drag. Initial parameters were set and later changed for optimization. To find the total drag, this group needed to calculate the vortex and airfoil drag using independent variables such as plane geometry and published airfoil data. A program called XFoil was used to investigate the benefits of using modified airfoils. Other responsibilities were to determine the efficiency factor "e", C_{Lmax}, $C_{Lcruise}$, flap deflection, and L/D at different flight conditions.
2. Configuration Group: This group had to determine the overall design and architecture of the plane. In the conceptual and preliminary design phases, the configurator had to integrate all design characteristics established during brainstorm sessions and produce drawings of feasible planes at each meeting. He had to insure that the plane remained within given size parameters, could fit in the packaging box, maintain the proper CG location, and could be readily constructed. The configurator used SolidWorks as the 3D CAD program for creating the drawings.
3. Design Report Group: This group gathered all pertinent information and compile notes from each general meeting and captain's meeting. The report writer was responsible for meeting the requirements and guidelines provided in the 2004 rules and presenting the work of the team in a coherent and technical manner. This group assigned deadlines to each captain who needed to provide documentation of testing, manufacturing techniques and necessary data.
4. Flight Test Group: This group was in charge of assigning and accomplishing specific goals for each test flight as well as collecting necessary data. Prior to each test flight, the flight test captain needed to provide test cards which specifically outlined the objects of the coming flight test and he needed to organize the plane components and all necessary equipment to carry out these objectives. After each test flight, this group was responsible for performing further analysis and presenting summary and conclusions to the rest of the team.
5. Payload Deployment Group: To satisfy the FireFight mission parameters in the 2004 rules, a special team was assigned to analyze possible methods of water deployment. Their focus was on tank configuration, system integration within the plane, water loading, and water dropping. They designed and tested a water tank with various nozzles, analyzed exit positions at various speeds, and developed a system to minimize water loading time.
6. Performance Group: This group's main focus was the PlaneSizer spreadsheet. The members had to integrate and update all inputs and outputs to correspond with this

year's flight missions and guidelines. They had to link all subgroup's pages to one common page so that all the inputs and important outputs could be seen after each iteration and kept within assigned limits. Some of the major parameters listed on this page were Take Off Field Length (TOFL), maximum battery weight, current draw and cruise velocity. The performance group was also responsible for optimizing the plane configuration to produce the best flying score possible. Each aircraft system was then sized accordingly and passed down to the individual groups and if proved feasible, drawn up by the configurator.

7. Propulsion Group: This team was responsible for finding the right combination of batteries, motors, propellers and gearboxes to optimize the propulsion system for this year's mission. A library of different manufacturers and specifications was built and integrated within PlaneSizer. Matching these parameters to model the mission for this year was a critical aspect of performance, hence testing of the top choices of each of these components was done to further complete the analysis. In the past it has been found that the motor and battery data published by the manufacturers was inaccurate at the high power levels needed for this competition. This year however, extensive testing was done on both the battery packs and the motors until dependable data was found and a configuration that would allow the plane to complete the missions was established.
8. Stability and Control Group: This group was responsible for analyzing the various tail configurations proposed during the conceptual design. For this year, there was large debate between a V-tail and a conventional T-tail. They had to create a spreadsheet that contained multiple parameters and equations to analyze the different types of tails. Stability and Control (S&C) was also responsible for establishing an appropriate Static Margin, determining hinge lines, servo locations and sizes, moments for each control surface and moments on the entire plane. Two important factors that were taken into consideration for nearly all S&C analysis were the crosswinds present at this year's competition site and the shift in the plane's center of gravity due to the water drop.
9. Structures Group: This group was responsible for sizing the main structural components of the plane. Initial structural analysis was needed to select between a monocoque fuselage design and the chosen design. The structures team was then in charge of sizing wing spars, the backbone of the plane, and the landing gear. This group had to take into account both static and dynamic loadings. They were also responsible for wing and landing gear attachments and for ensuring that all plane components fit into the packaging box specified in the 2004 rules.

2.3 Configuration and Schedule Control

A schedule of important dates and deadlines was created at the beginning of the year so that time management could be easily visualized. For example, each phase of the design process was assigned a definite period of time to allow for sufficient construction and testing. The schedule took into consideration students' school load, allowed for change where needed and had certain processes overlap for efficiency. Figure 2.3.1 shows the final schedule with planned and actual dates.

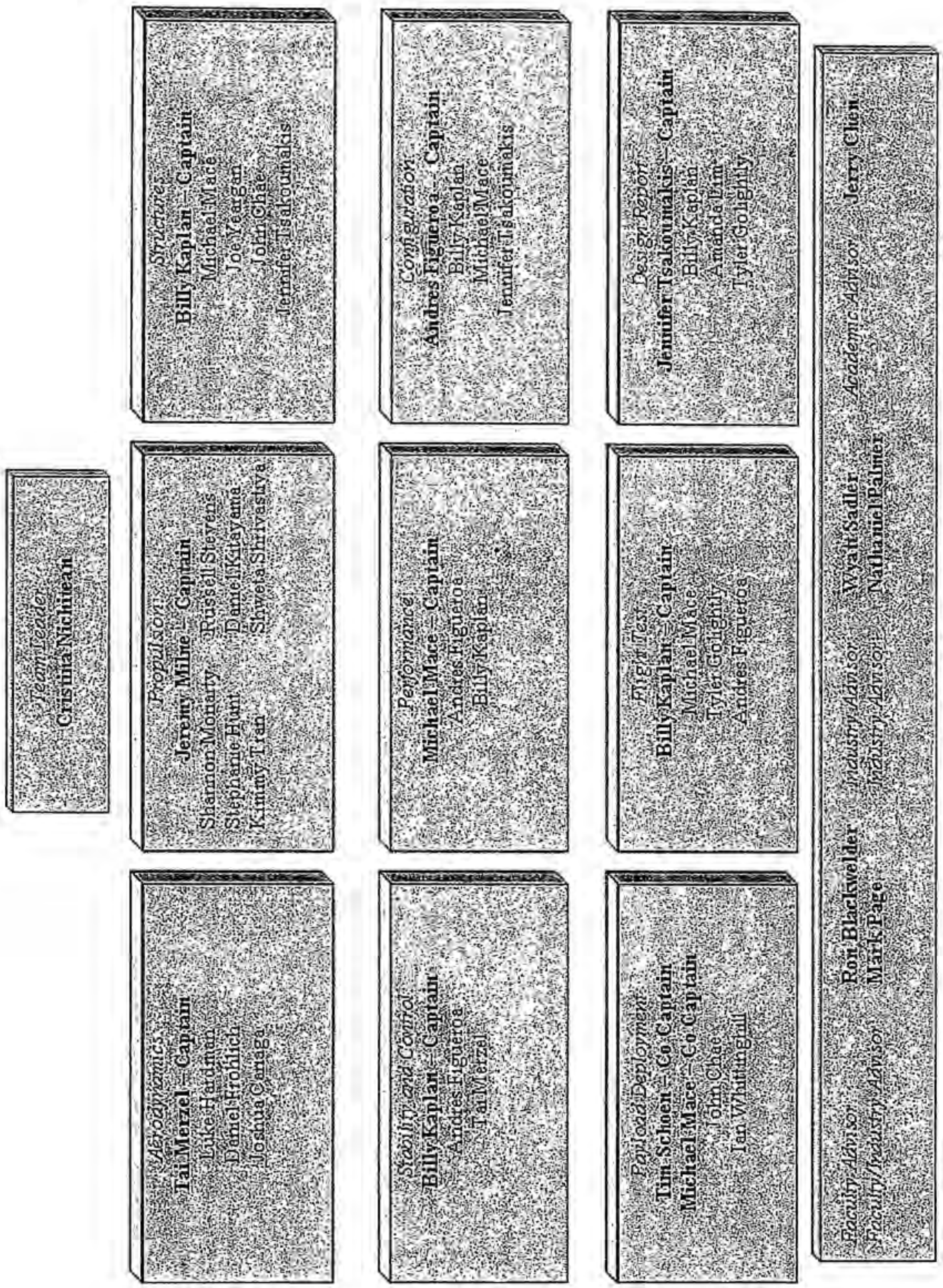


Figure 2.2.1 USC Aero Design Team Structure showing advisors and subgroups.

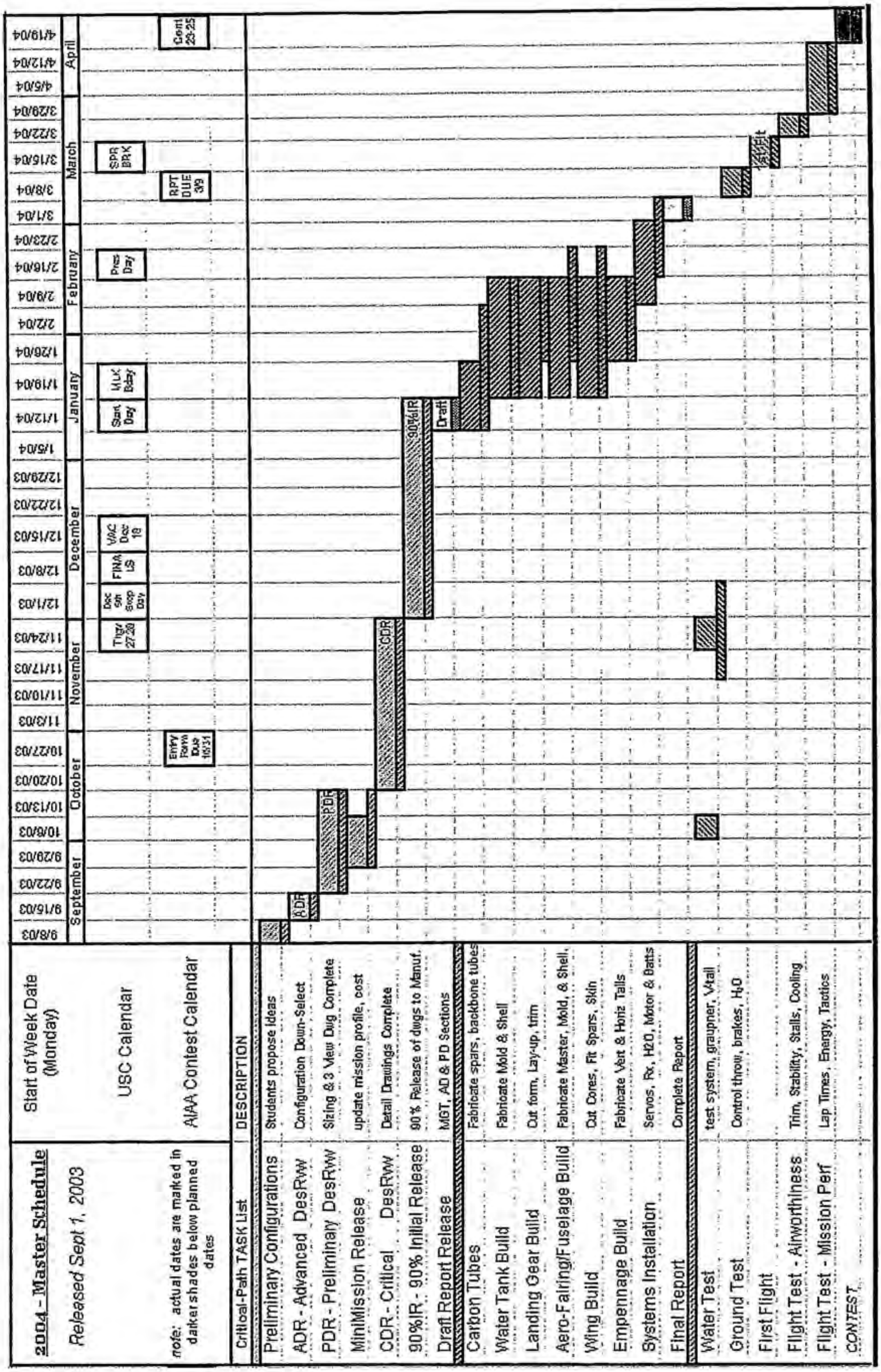


Figure 2.3.1 USC Aero Design Team Schedule 2003/2004

3.0 CONCEPTUAL DESIGN PHASE:

For the 2004 Design/Build/Fly contest, there were two flight missions to accomplish. The *Firefight* mission was designed to be a heavy lift/slow flight mission. It had a difficulty factor of 2.0 and required two water loadings, two water dumps and two landings. Each lap needed to include one 360° turn. The maximum water capacity of the aircraft was four liters and the actual amount carried was factored into the final flight score. The second mission was the *Ferry* mission. It was designed to be a higher speed flight. For this mission the plane had to take off, complete four laps with a 360° turn on each down wind leg, and land. It required no payload and had a difficulty factor of 1.0. As was noted earlier, the driving force behind this year's plane design was meeting the requirements of the *Firefight* mission.

3.1 Key Elements of Mission Requirements

The following conditions were applied to all five configurations under consideration:

1. Box Constraint: The contest rules specify that all components of the plane need to fit inside a 4'x2'x1' box. This drove the configuration to be flexible enough to allow for interlocking parts.
2. Take Off Field Length (TOFL): The rules require the plane to take off within 150'. As a safety factor, the plane was designed to take off within 75% of the total length. This criterion was determined from flight tests conducted in 2000 and was incorporated into the spreadsheet "PlaneSizer" to calculate the performance and design parameters for an optimum solution to the different missions required this year.
3. Number of Motors: Because of the expense that an additional motor would add to the Rated Aircraft Cost, only one motor was assumed for all configurations. From preliminary calculations, one AstroFlight or Graupner motor provided enough power to meet the required TOFL.
4. Battery Weight: All configurations had provisions for the maximum 5lb battery weight limited by this year's rules. This condition ensured that each configuration would have sufficient energy to complete each mission.
5. Water Payload: All configurations were designed with anticipation of carrying the maximum water capacity of 4 liters. Configurations were selected such that the CG of the water tank would be coincident with the unloaded CG of the plane to minimize the in-flight shift during the FireFight Mission.
6. Use of Composite Materials: The high strength-to-weight ratio of composite materials, such as carbon fiber and fiberglass, reduced the weight and provided sufficient strength to withstand the applied loads.

3.2 Alternate Configurations

After an initial brainstorming session, all ideas were compiled into the following five general plane configurations (see figure 3.2.1):

1. Conventional plane with a V-tail: This design was chosen for its simplicity. It was straightforward and familiar to the team members. It would allow for easy water system integration and overall manufacturing. The conventional plane would include two tubes, a main boom and a telescoping tail boom that collapsed into the main boom for packing. The major components attached to the main boom and the tail surfaces attached to the tail boom. This design would provide structural simplicity. The V-tail was chosen to decrease Rated Aircraft Cost.
2. Conventional plane with a conventional tail: This design was chosen for the same reasons as the first plane. A conventional tail was chosen however, despite the added cost. There was debate about the tail construction techniques and about the performance of each configuration in the crosswinds that the plane may encounter at the competition. These points were discussed throughout the entire conceptual design phase and the remainder of the downselect process. Based upon previous experience, the conventional tail was considered to be more crash-worthy than a V-tail. Should the tail be damaged in a hard landing, the 30 minute repair time would be enough to fix a conventional tail where as perfectly realigning the angle of the V-tail could take longer.
3. Biplane with a V-tail: This plane with a V-tail was considered because it would have a low Rated Aircraft Cost (RAC). It would also allow for fairly easy water system integration and with a deeper tank, would possibly allow a faster water dump. However, the manufacturing techniques, the crash worthiness and the assembly process for this plane placed doubt over the possible success of a biplane configuration. With two wings, this plane would be harder to construct and harder to repair.
4. Flying Wing: This configuration was appealing because it did not include a tail. This aspect would provide for easier manufacturing and assembly. Should this configuration be chosen, the team decided it would need a nose gear and downward pointing winglets on the tips of the wing. The landing gear loads would be directed through the wing spar providing stability on a hard landing. A flying wing however would produce poorer flight handling qualities than a plane with a tail due to the crosswind conditions.
5. Blended Wing Body: This configuration was also considered because it has no tail. Without a tail, the blended wing would be easier to manufacture and assemble. However, as with the flying wing, this was a design with which the team has had little experience. With the blended body the hard points and load paths would not be as clearly defined. This configuration would also have a very large sweep angle in the wings which would pose difficulty in making

an angled joiner between the two wing halves. This configuration also has a very low static margin. This would cause difficulty in flight during the Firefight mission due to the CG shift as the water is deployed. Also flight handling would be impaired in crosswinds due to a lack of yaw stability.

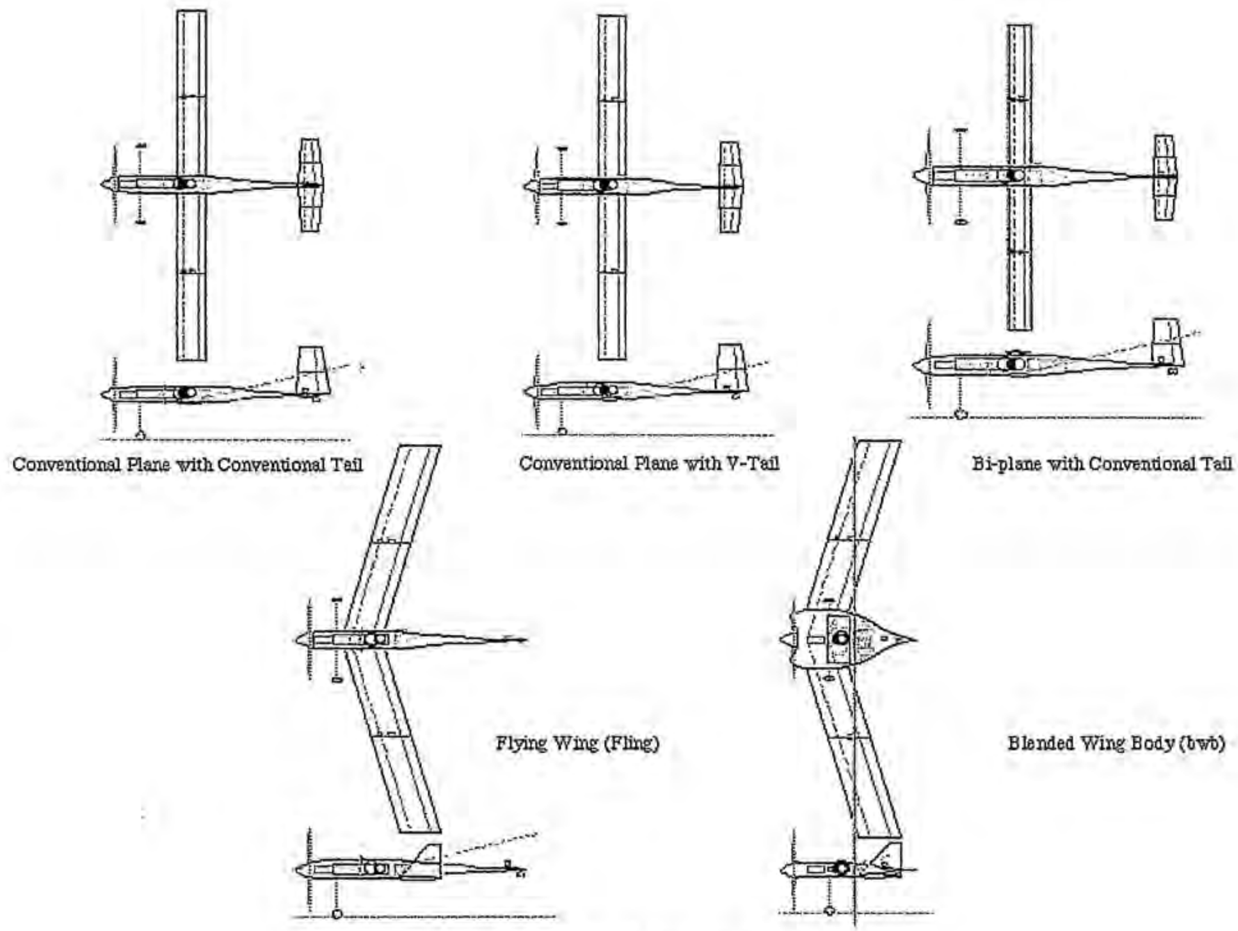


Figure 3.2.1 Conceptual design configurations

3.3 Figures of Merit

Each plane configuration was entered into an Excel program to optimize the two calculated parameters: Drag and Rated Aircraft Cost. The planes were then scored according to 10 Figures of Merit (FOM). First, each FOM was assigned a factor from 1-5 (Column 1 in Table 3.3.1) representing the team's opinion of the relative importance of that FOM in assessing the plane's overall performance in the competition. A factor of 1 was given to a figure that had low importance and a factor of 5 was given to a figure that had high importance. Secondly, for each FOM parameter, each plane configuration was assigned a value from 1-5 representing the quality that configuration would have for the given figure of merit (Columns 3-7 of Table 3.3.1). A value of 1 meant low quality, a value of 5 meant high quality.

The values were then added together and the planes were ranked by total score; the configuration with the highest total score was considered the optimum design. The FOM's were defined as follows:

1. Ease of Manufacturing: This figure of merit was assigned a factor of 1. Manufacturing takes into account the resources and materials available, the techniques required and the feasibility of the team members performing them correctly. With the faculty and industry advisors and experienced team members, it was decided that a plane design should not be heavily influenced by its ability to be manufactured because new construction processes can always be learned.
2. Ease of Assembly: This figure was assigned a factor of 2. It was defined as how readily the plane can fit into the box, how quickly it can be put together from the box and how many components it contains. For this year's contest, assembly time does not contribute to overall flight time so it does not carry much significance for total score. However, it was given a factor of 2 rather than 1 because efficiency at test flights is necessary. The quicker the plane is assembled on the test field the more flights the team can complete before the actual competition.
3. Crash Worthiness: This figure was also assigned a factor of 2. Crash worthiness was defined to include robustness of structure and ease of repair after a crash or hard landing. The teams experience with composite structures has given it confidence that repairs are not that difficult.
4. Structural Simplicity: This figure of merit was assigned a factor of 3. Structural simplicity includes minimizing the number of components and joints as well as incorporating well known load paths.
5. Ground Handling Quality: This figure was also given a factor of 3. Proper steering on the ground prior to take off is essential to minimize mission time. For the FireFight mission, the plane is required to land and refill midway through the mission, so accuracy in landing and stopping is necessary.
6. Experience with Design: This figure was given a factor of 3. Originally this FOM was given a factor of 1, however during the downselect process, the team found that whether or not they had experience with the design was a significant factor in justifying each configuration.
7. Flight Handling Quality: This figure was given a factor of 4. Clearly how the aircraft handles in flight directly affects the overall flight score. Because the competition this year will be held in Wichita, crosswinds will greatly affect plane performance. Stability and control needs to be completely reliable under these conditions.
8. Water System Integration: This figure was also given a factor of 4. The water system is the main part of the FireFight mission and its integration into the structure of the plane and its overall performance is necessary for a good flight score. The team decided that each plane design must allow for a container that would hold four liters of water, must have an access area to expedite the water loading process and provide for quick water dumping.
9. Drag: This figure was given a factor of 5. An Excel model was created to estimate the drag for each of the configurations. Because the drag for all configurations was comparable, this

parameter was normalized to keep small changes in the drag from disproportionately affecting the overall score for each plane.

10. Rated Aircraft Cost: This figure was also given a factor of 5. Both of these last two FOM's play directly into the overall flight score. Rated Aircraft Cost is designed to penalize configurations for complex geometry and large size. Minimizing these parameters improves performance and increases the score since the flight score is divided by the RAC factor. Due to the cost equation in this year's rules, a long and narrow fuselage was needed to minimize the RAC.

MF	FOM's	Configurations				
		V conv	Conv	V-bip	Fling	Bwb
1	**Ease of Manufacturing**	4	4	1	5	5
2	**Ease of Assembly**	4	3	1	5	5
2	**Crash Worthiness**	3	4	1	4	5
3	**Structural Simplicity**	4	4	1	5	4
4	**Flight Handling Qualities**	4	5	4	2	2
3	**Ground Handling Qualities**	4	4	4	5	5
4	**Water System Integration**	5	5	4	5	4
1	**Experience with Design**	5	5	3	2	1
5	**Drag x Cost**	2.2	2.1	1.9	2.1	1.8
5	**RAC**	2.2	2.1	2.4	1.8	1.5
	Total Score	**105.0**	**108.0**	**76.2**	**102.2**	**93.5**

Table 3.3.1 Figures of Merit used for downselect process. "Conv" refers to conventional tail configuration. "V-bip" refers to a biplane with a V-tail. "Fling" refers to a flying wing and "Bwb" is a blended wing body.

4.0 PRELIMINARY DESIGN PHASE:

This phase further developed the conceptual design studies by breaking down the different disciplines and analyzing them thoroughly. Five main design subject areas were examined: Aerodynamics, Propulsion, Stability and Control, Structures, and Water Deployment. Parametric models were created for each area and were integrated into the single analytical tool PlaneSizer. Once PlaneSizer was updated for this year's contest, sizing studies were performed, different missions were tested, and the configurations were optimized.

4.1 Mission Model

The goal of the analytical tools used by the USC Aero Design team was to provide simple preliminary calculations. The use of these tools allowed the team to focus on obtaining results rather than spending time on tedious calculations. As has been cited before, PlaneSizer, was the team's leading mode of analysis. Originally based on an older spreadsheet, PlaneSizer was reconstructed by last year's team and updated for this year's competition. PlaneSizer is a compilation of many Excel Spreadsheets and programs designed to target each major component of plane design and construction. Each team captain was responsible for understanding the current spreadsheet for his design component, updating it to represent this year's parameters, and submitting it to the Performance Group for final inspection and integration. All spreadsheets were then linked together. The result was a complex workbook of different modules categorized by subcomponent and/or task. To facilitate the use of this package, a user friendly "FrontPage" was created to incorporate all the required inputs for the mission profiles as well as the relevant outputs. With all calculations and data on the FrontPage taken from the embedded spreadsheets and pre-established databases, sizing the plane was done easily in a localized fashion. The program ran for several iterations for different variables and produced multiple results that were easily compared. Each sizing iteration began by adjusting the wing area for a target take off field length and then sized the tail for required pitch and yaw stability. Finally, the motor current, voltage, RPMs and thrust were converged to achieve an available rate of climb of 400 fpm This accounted for the increased lift needed during turns. A cruise velocity could then be calculated from the propulsion parameters and used to analyze mission performance. After the different parameters were put in place, the program ran though the sizing process. The flight time and rated aircraft cost for a given design were combined into a single quantity called "Flight Score". Different designs were compared using this quantity.

4.2 Design Parameters and Sizing Trades

4.2.1 *Aerodynamics*

The focus of the aerodynamics group was to ensure that the aircraft would meet all of the main constraints of the competition. They had to make sure the plane would make take off field length, provide sufficient lift, and have minimum drag. These were obtained by studies done in three different subdivisions of this group: drag, airfoils, and wings.

1. Drag: To calculate the total airplane drag, vortex and parasite drags were computed using relations derived from *Fluid Dynamic Lift* and *Fluid Dynamic Drag* by S.F Hoerner. The parasite drag coefficient, C_{do}, was found by inserting numbers from *Fluid Dynamic Drag* into the equation

$$C_{do} = \left(C_{do\ fuselage} + C_{do\ wing} + C_{do\ tails} + C_{do\ landing\ gear}\right)K_{excrescence}$$

where $K_{excrescence}$ is a multiplier for gaps, vents, and antennas. Airfoil polars were taken from a program called Xfoil. Xfoil is a program developed by Dr. Mark Drela of MIT. It employed a panel of calculations for the pressure distribution plus a boundary layer model. It then generated plots for lift and drag coefficients as a function of attack angle for a given Reynold's number. Vortex drag due to lift is accompanied by strong wingtip vortices. This influence is minimized by using high aspect ratio wings to separate the vortices. Elliptical span-loading and winglets also reduce vortex drag. But because most wings do not display a perfectly elliptical span-load, an efficiency factor "e" accounts for the discrepancy. The factor "e" is calculated from the following equation:

$$e_o = e_{wing} K_{wing\ viscous}$$

where e_{wing} is the inviscid vortex drag efficiency factor and $K_{wing\ viscous}$ is the multiplier for viscous drag increase due to angle of attack. For an untapered wing, this factor is 0.89. This was the value used in the drag calculations for this year's plane. Other types of drag considered in the aerodynamic model were junction drag, gap drag, base drag, step drag, trim drag, and profile drag. Junction drag was found at the intersection of two aerodynamic bodies such as the wing and the fuselage. Gap drag was taken at the control surface hinge lines. Base drag occurs at the aft part of the fuselage and is eliminated by bringing the tail cone to a point or blade. The step drag is prevented by using flush brackets. Trim drag, which results from download trimming of the tail, can be minimized with a long tail arm and aft CG position. And the profile drag is caused by the two dimensional airfoil type. The total airplane drag could then be calculated as the sum of the parasite drag components and the vortex drag components, using the following equation.

$$C_D = C_{do} + \frac{C_L^{\,2}}{\pi e_o AR_w K_{trim} K_{junction} K_{gap} K_{base}}$$

These were all analyzed and the calculations were incorporated into the PlaneSizer spreadsheet. Figure 4.2.1 shows a graph of the total airplane drag as calculated by PlaneSizer.

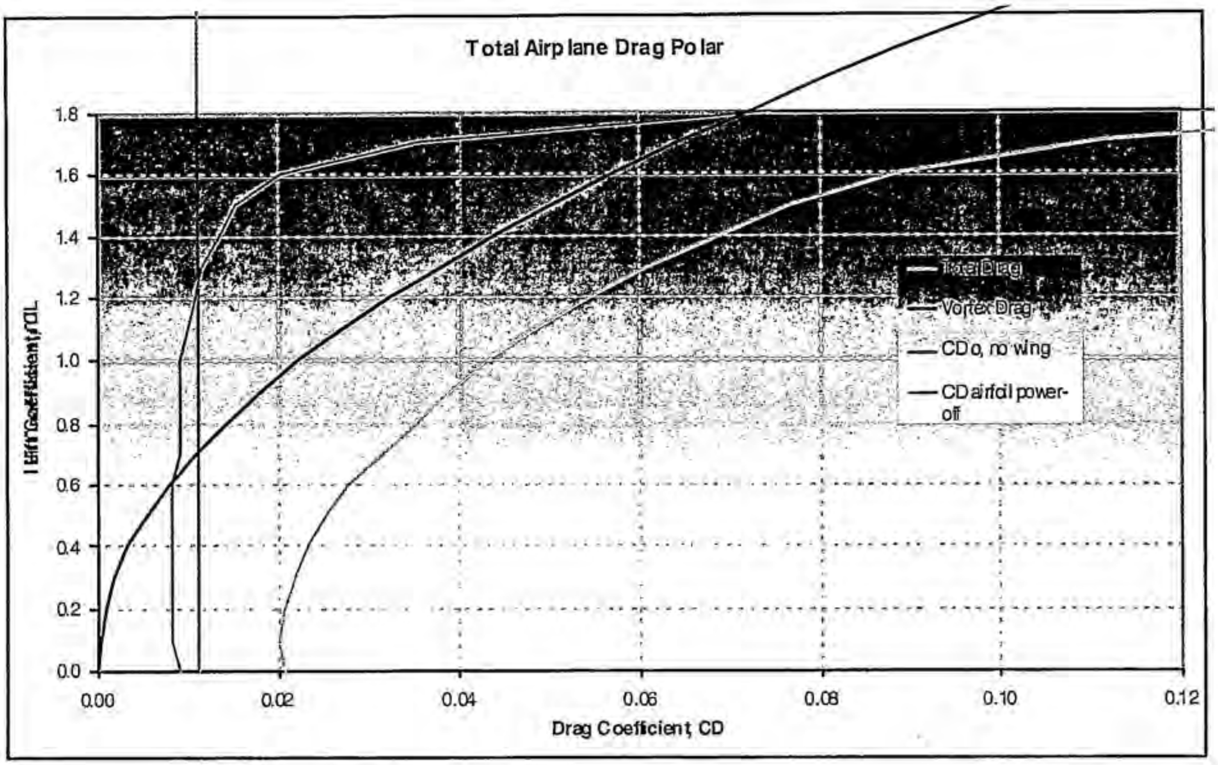


Figure 4.2.1 Example of Total Airplane Drag for the FX-63-137 with 16% increased thickness. Vortex and parasite drag also depicted.

2. Airfoils: The team this year analyzed a variety of airfoil designs that would produce sufficient lift with minimum drag. The chosen airfoil needed to satisfy the following characteristics: high C_{lmax}, low drag at $C_{lcruise}$, acceptable drag on the maneuvers at the approximated C_l, and an appropriate thickness to accommodate wing spar. For example, the LA203-KB, the Eppler 374 and the FX63-137 were specifically tested with Xfoil because they were all airfoils that had a high enough lift coefficient to fulfill the plane needs in this year's FireFight mission. Data on other airfoils, such as the LA203a, SD8020, SD800, RG15, NACA2214, and others with variations in thickness and flaps, was obtained from a library with precalculated data. These airfoils were selected for their high C_{lmax}, their capability to meet take off field length requirements, or their low C_D at cruise velocity. The airfoil chosen was the FX63-137 because it had the lowest drag over the range of lift coefficients anticipated in the contest. The FX was consistently better for both high and low lift coefficients. For structural reasons, a large diameter carbon tube spar was needed to support the wing. Xfoil was used to analyze the airfoils as their thickness was increased from 12% to 16%. Flaps were also added, from 5 degrees up to 10 degrees down to simulate the range that would be used during flight. Figure 4.2.2 shows the total drag polars for different airfoils considered.

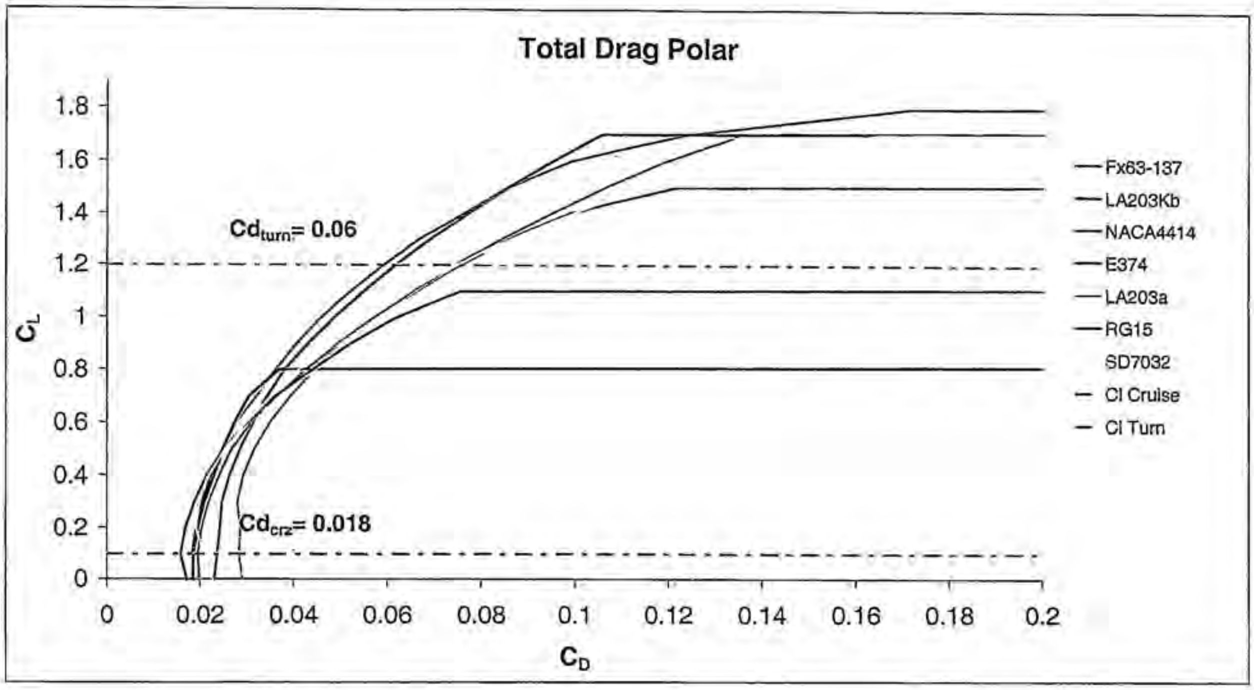


Figure 4.2.2 Graph of C_L v C_D for various airfoils. Dashed lines denote drag coefficients at cruise and turning conditions for the airfoil chosen (Fx63-137)

3. <u>Wing</u>: The two most important parameters for sizing the wing were wingspan and wing area. It was decided to use a non-tapered wing because the increase in difficulty of manufacturing a tapered wing outweighed the slight benefit in drag reduction. Because of the 4'x2'x1' box requirement, it was necessary to have a sectioned wing. In order to minimize weight and complexity, a maximum of two joints in the wing were allowed. This, combined with the box requirement, constrained the wingspan to a maximum of 9 ft. A trade study was performed by varying the wingspan while holding wing area constant and observing the effect on a number of variables such as best flight score, mission time, rated aircraft cost, available/required energy, etc. This study is shown in Figure 4.2.3. It was found that the airplane's performance increased with wingspan up to the limit of 9 ft. However, due to structural limitations in wing spar sizing, it was deemed necessary to reduce the wing span to 8 ft. in order to relieve the bending moment at the center.

   The wing also had to have enough area to take off within the required take off field length when filled with 4 liters of water. Takeoff field length was calculated using the following equation.

$$TOFL = \frac{(TOGW)^2(1.2)^2}{(Takeoff\ Thrust)(1 - MuRoll)(g)C_{L\max}S_w\rho}$$

This formula was incorporated into the Excel tool PlaneSizer such that wing area was always sized to meet TOFL requirements within a safety factor for a given gross weight, C_{Lmax} and propulsion package. Estimated aerodynamic characteristics for the chosen wing (FX 63-137 with 16% t/c and 8 ft wingspan) are summarized in Table 4.2.1.

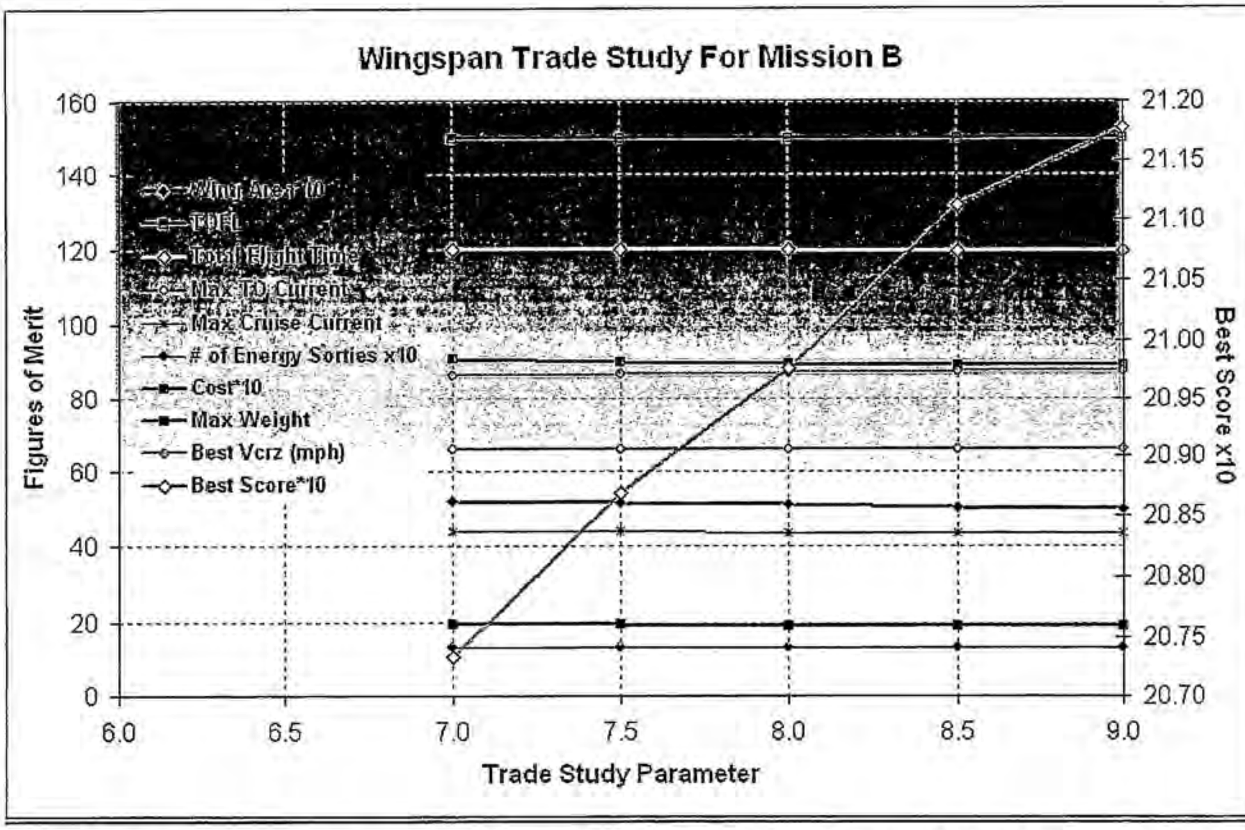


Figure 4.2.3 Trade Study of several Figures of merit vs. Wingspan. Trade Study Parameter is wingspan in dimensions of ft.

Interpolated Drag Values at Cruise				General Aerodynamic Characteristics	
	C_L	C_D	L/D	e	0.94
Heavy Cruise	0.211	0.0236	8.97	C_{lmax}	1.9
Light Cruise	0.105	0.0237	4.41	t/c	0.16
				C_L at V_{cruise}	0.1
Airfoil: FX63-137 16% thickness + flap				C_L in turns	1.2

Table 4.2.1 Estimated aerodynamic characteristics for Fx 63-137 with 8 ft. span

4.2.2 *Propulsion*

The most important factor determining aircraft performance and overall score was the propulsion system consisting of the batteries, motors, gearboxes and propellers. Propulsion was analyzed and optimized by PlaneSizer until a final propulsion configuration was obtained. The spreadsheet calculated the total resistance of the circuit by summing the internal resistance of the individual components such as batteries, cables, speed controller, and motor. An initial assumed current was run through the circuit to calculate a voltage drop across each component in the circuit. The voltage across the motor was used to calculate the motor rpm. The propulsion model then took this rpm and computed the torque required for the chosen propeller and a required current was calculated and compared with the initially assumed current. An iteration process altered the assumed current and then repeated the computation until the required current matched the assumed current. In past years, actual propulsion system performance had been inconsistent with predicted performance. Thus extensive testing of each of the components was performed to verify the manufacturer's specifications and the published performance.

1. Batteries: According to the contest rules, the battery pack was limited to a five pound maximum weight and a 40 amp maximum current draw. The available energy stored in each cell was needed to provide efficient conversion to kinetic energy. Tests conducted by the team indicated that the advertised energy content of the batteries was not available primarily due to the heavy current draw by the motor. This produced temperature effects resulting in increased resistance in the motor, the controller, and the wiring. The tests suggested that the batteries produced only 60% of the manufacturer's listed storage capacity. The analysis was incorporated into PlaneSizer for a library of Sanyo batteries, including five different types: KR-1400AE, RC-2000, RC-2400, CP-1300, and CP-1700. These types were included because of their accessible data, past performances, and because they satisfied the contest constraint of using over the counter NiCad's. Figures 4.2.4 and 4.2.5 show trade studies for battery type and cell count as was calculated through PlaneSizer. The Cp-1300's were chosen because of their availability and their known performance characteristics from previous years. Cell count was set at 28 cells which gave the proper voltage for take off. The battery life gives the needed energy to complete the mission with the smallest amount of weight.

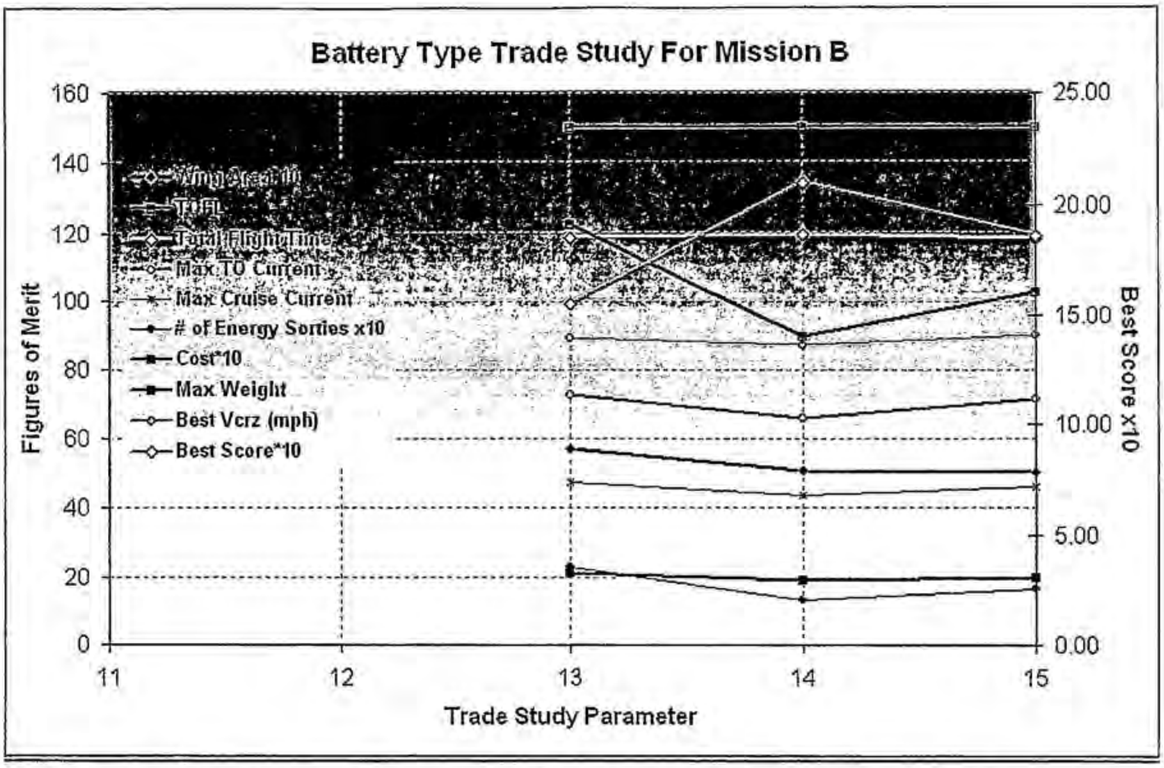


Figure 4.2.4 **Trade study of various Figures of Merit vs. Battery type. Trade Study parameter is battery type where 13, 14, and 15 denote RC-2400's, CP-1300's, and CP-1700's respectively**

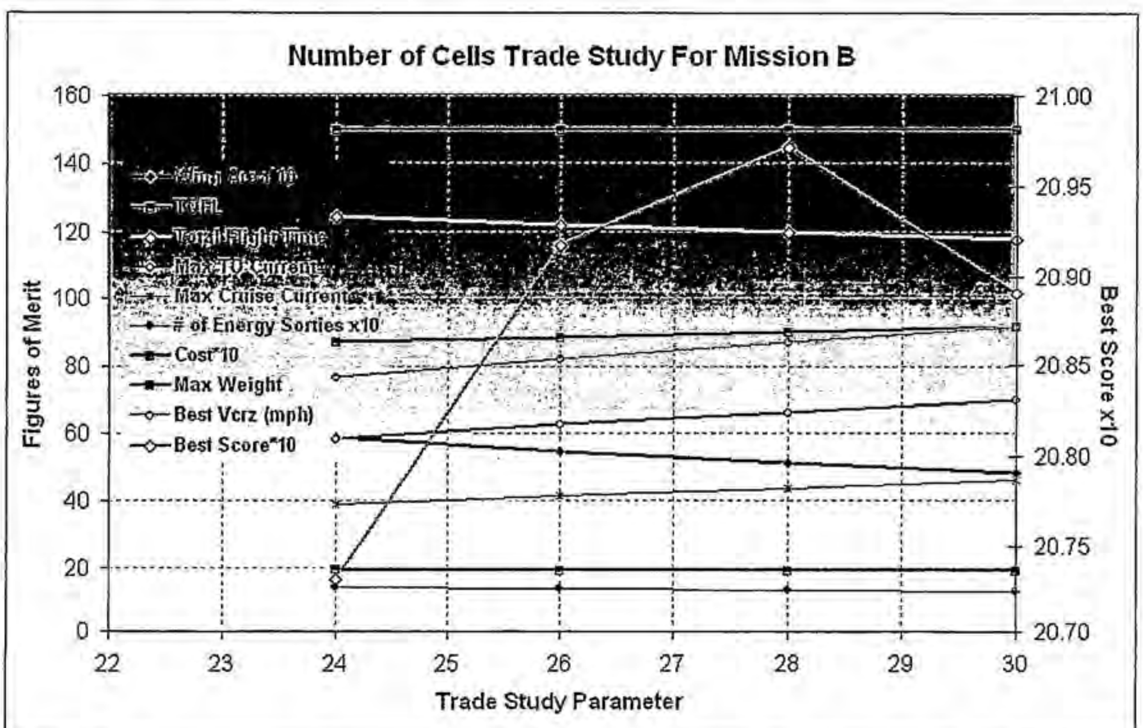


Figure 4.2.5 **Trade Study of various Figure of Merit vs. number of cells assuming batteries are CP-1300's. Trade Study Parameter denotes number of battery cells.**

During the testing of the battery packs, voltage, current, and duration of pack life were recorded. The objectives of the battery tests were to determine if the actual voltage and battery life matched the predicted values and to verify that the battery packs could perform with as predicted with the chosen motor. For these tests, the battery packs were soldered together in the configuration that was to be used in flight. This caused some difficulty because the batteries were lined up end to end rather than side by side and during the soldering process some batteries were destroyed. A customized soldering iron helped to eliminate this problem. The tests confirmed the expected performances calculated by PlaneSizer and 28 CP-1300 batteries, with approximately 33.6V per pack, were chosen.

2. Motors: According to contest rules, either Graupner or AstroFlight models of brushed electric motors had to be used. A library of motors was constructed in PlaneSizer which included parameters such as torque, speed constant and internal resistance for different motors. These parameters were used to establish efficient battery usage and obtain maximum power provided to the propeller. Three motors were considered for this year's plane; the Graupner 3300, the Graupner 930 and the Colbalt 60. Each motor was tested to determine four characteristic parameters: the speed constant, torque constant, no load current and winding resistance. A student designed and student constructed thrust bench was used for all motor tests as shown in Figure 4.2.6. The tests to study possible non-linearities in the motors failed because the thrust bench was limited to torques less than 1ft-lb. However, the propulsion team was able to determine that the Graupner 930-10 was able to handle the current and voltage needed to produce proper torque at take off and was able to endure the mission at full throttle (with the 28 cell battery pack). Data collected included voltage, current, torque and thrust. Data plots of these values vs. time for a typical test are shown in Figure 4.2.7. The chosen Graupner motor had a higher speed constant, was light and efficient. The other two motors considered were considerably heavier. After tests, it was concluded that the motor behaved as predicted and would handle the power levels required of it. However, it was noted that the motor would be running very close to its ultimate capacity when used during flight missions. A Hall-Effect Sensor was used to measure the RPM.

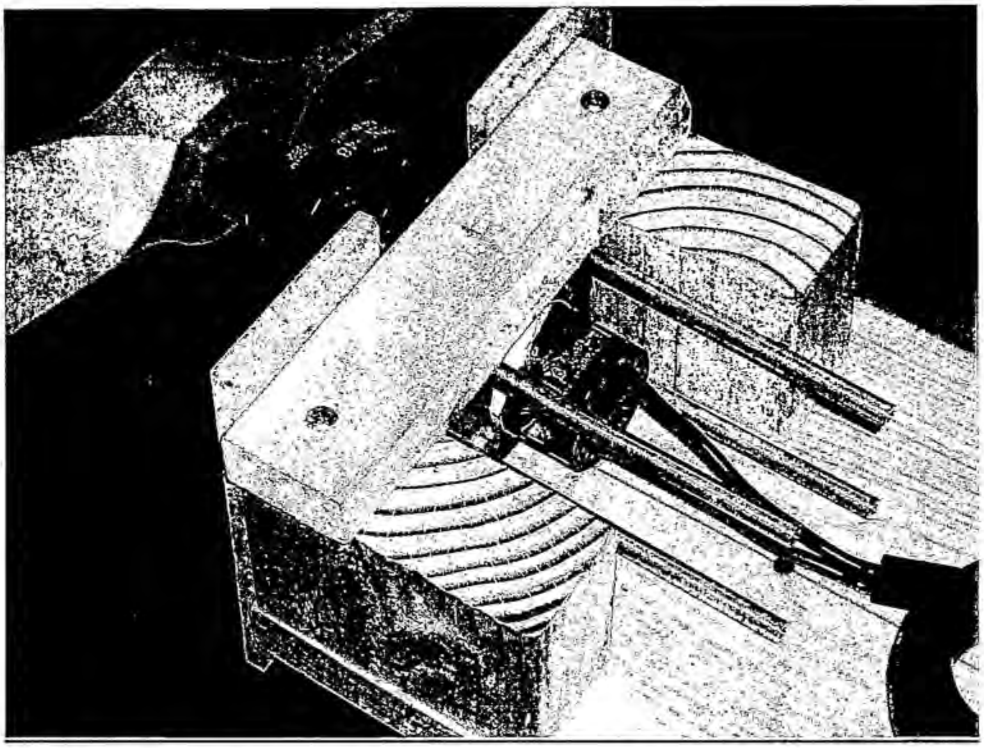


Figure 4.2.6 Thrust bench used to verify analytical propulsion model

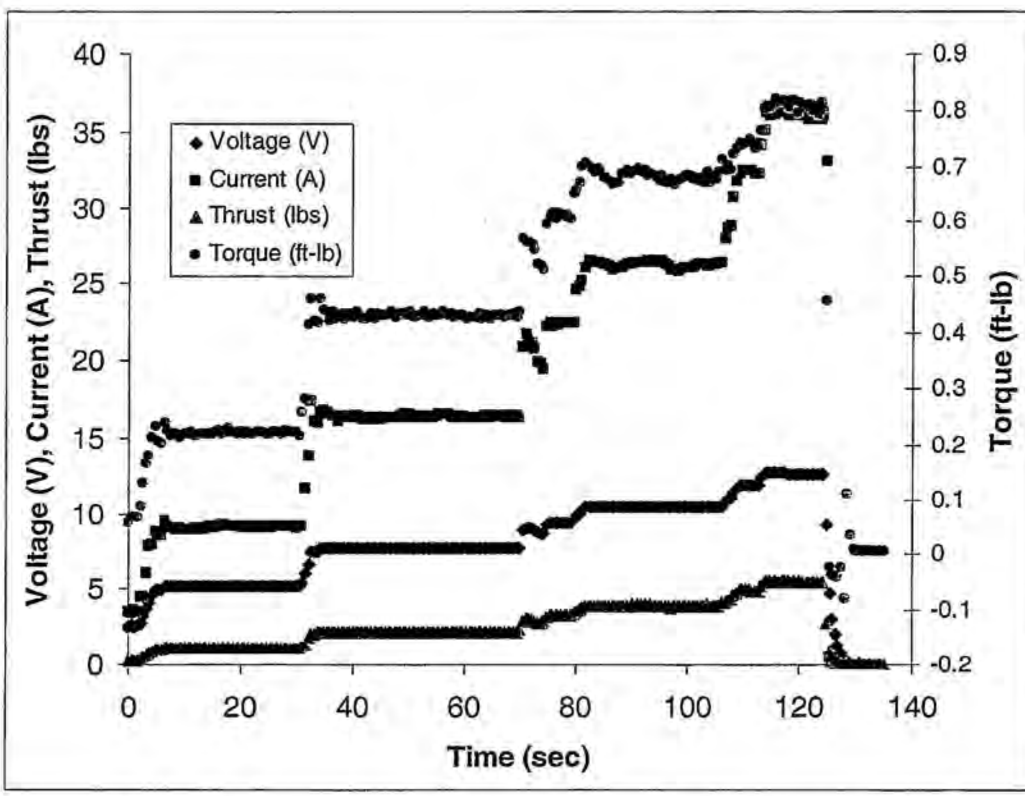


Figure 4.2.7 Time trace of data for Graupner 3300-06 using a power supply and a Bolly carbon 24 x 24 prop

3. Propellers: The propeller diameter and pitch had to be mated properly with the motor and battery pack to ensure the aircraft would meet the take off field length constraint and also to maintain sufficient cruise velocity. Large pitch propellers required more current draw whereas large diameter propellers gave a slower velocity but improved take off field length. Propeller libraries were built and incorporated into PlaneSizer. Trade studies for pitch and diameter were performed as shown in Figures 4.2.8 and 4.2.9. It was found that a 21 in. diameter propeller gave the best flight score without exceeding the 40 amp current limit. The propeller chosen was a wooden Zinger 21-14.
4. Other components: For this year's propulsion model a gear box was necessary. The InnerDemon 3.5:1 was chosen because it allows easy change between gearing ratios. For the speed controller, a Jeti 100 was chosen because it can handle 100A peak current for 30 seconds and 90A current continuously. Fuses were also tested and chosen by the propulsion team. The fuses were tested to see how long they would last at high currents. They were tested at currents greater than takeoff current and for longer than anticipated. The fuses were not a limiting factor in operating current. There was difficulty in providing an accurate measurement of the current and voltage while shorting the battery to the fuse. But by modifying the sensors on the test bench, the team was able to achieve this. The fuses were used throughout propulsion testing to ensure that it would not blow during flight conditions.

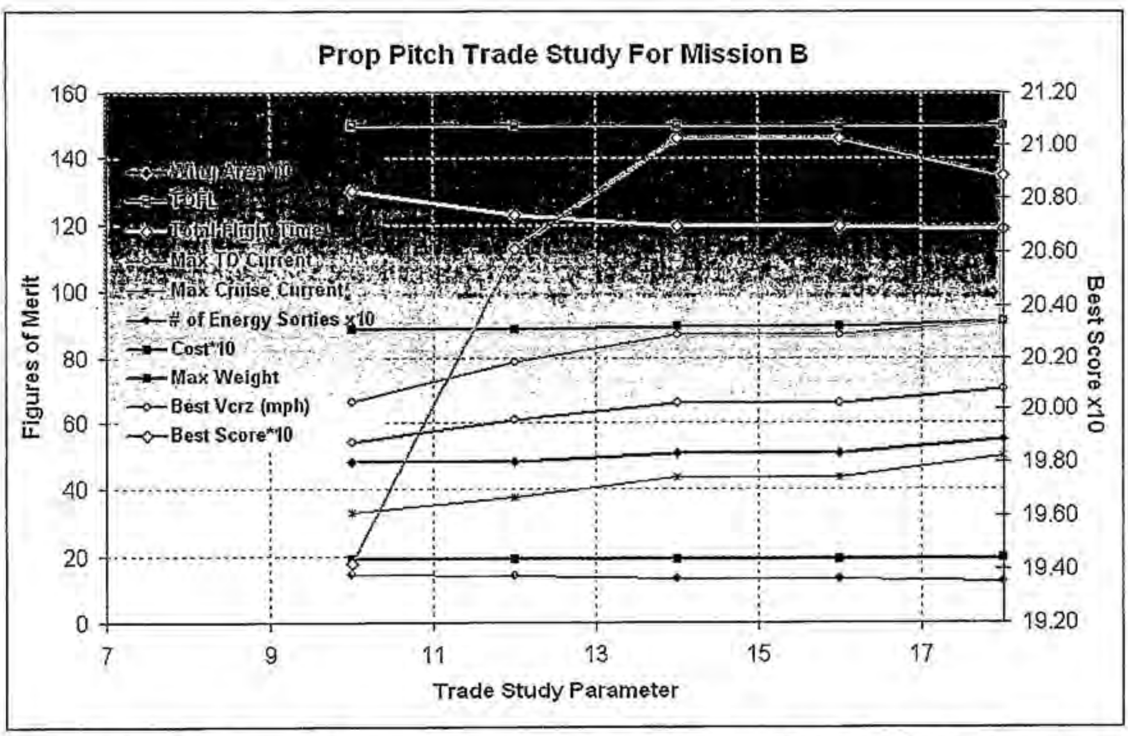


Figure 4.2.8 Trade Study of several Figures of Merit vs. prop pitch. Trade Study Parameter is prop pitch in inches

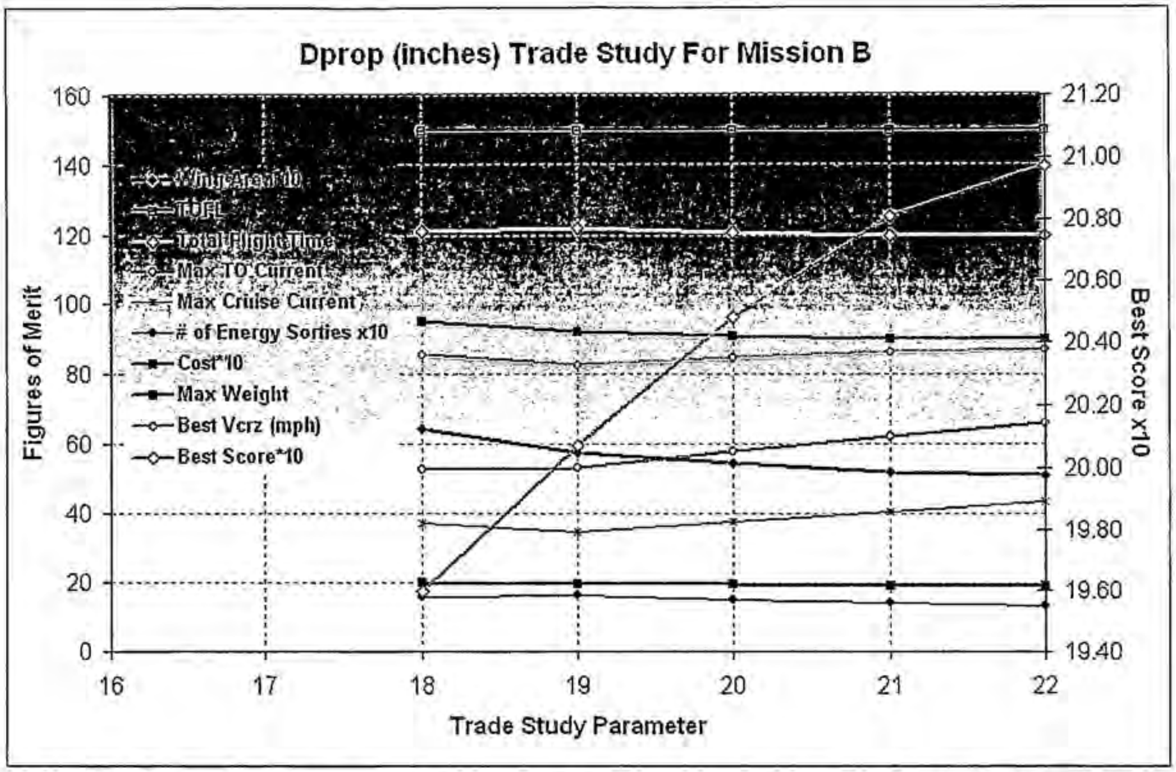


Figure 4.2.9 Trade Study of various Figures of Merit vs. Prop diameter. Trade Study Parameter is prop diameter in inches.

4.2.3 *Stability and Control*

The goal of the stability and control group was to provide a stable and maneuverable plane with the least tail area to minimize drag. It was decided that a conventional horizontal and vertical tail would be more suited to this year's competition than a V-tail because V-tails have diminished cross wind handling, in return for only a 1% reduction in drag. Crosswinds were a major concern for this year's competition, so the conventional tail was picked over the V-tail. Some issues addressed in designing the tail assembly included the tail lever arm, tail aspect ratios, flutter, sweep, as well as the frequency and damping ratios for the plane. Because the tail design was so closely tied in with other systems on the plane, such as the wing and propulsion system, the tail sizing process had to be performed in parallel with the rest of the aircraft. In PlaneSizer, the tail area was varied during each iteration until the required static margin and directional stability ($C_{n\beta}$) were satisfied.

1. <u>*Static Margin Requirement*</u>: The pitch stability was limited by a static margin requirement of 25%. This number was obtained from a test flight in which one of the previous competition's planes center of gravity was shifted away from the c/4 location of the wing until the pilot felt that the plane had insufficient pitch stability to be reliably flown at a competition. By reverse

engineering, it was then determined that the optimal static margin for a DBF plane was approximately 25%. This number was used by PlaneSizer to size the horizontal tail area using a variety of equations from *Aircraft Stability and Control* by Perkins and Hage.

2. _$C_{n\beta}$ Requirement_: The directional stability was limited by a $C_{n\beta}$ requirement of 0.0016/degree. This requirement came from another test flight in which a plane with similar geometry to this year's competition plane was used. In this test, sections of the vertical tail were removed incrementally to reduce the vertical tail area for a number of successive flights until the pilot determined that there was insufficient directional stability to perform well in a competition with severe crosswinds. The tail area was then used to calculate a $C_{n\beta}$ of 0.0016. This number was used by PlaneSizer to size the vertical tail area using equations from *Aircraft Stability and Control* by Perkins and Hage.
3. _Flutter Control_: For crosswind handling, it was determined that a full flying vertical tail would provide more control than a conventional tail with a rudder. A major concern with this full flying tail was aerodynamic flutter, which can occur if the center of mass of the tail is behind the hinge line of the tail. The hinge line was placed at the aerodynamic center of the tail to reduce aerodynamic moment loads on the servo. Therefore, in order to keep the center of mass ahead of the hinge line it was necessary to sweep the vertical tail forward. Tests were done to make sure that this forward swept tail would not result in static divergence. This can occur if a change in the $C_{L\ tail}$ caused a twist in the tail resulting in a significant change in angle of attack. It was determined that with the building methods employed, static divergence was not a major issue. Thus, sweeping the tail forwards was deemed a viable solution to the flutter problem.
4. _Frequency Response and Damping Ratio_: Other important Stability and Control characteristics were the frequency response and the damping ratio for the plane. These parameters would determine how well the plane would return to its initial conditions when perturbed in flight. Ideally, the frequency should be as high as possible, minimizing the time required to return to initial conditions. In reality, frequencies above 1 Hz are deemed acceptable to avoid sluggish handling. The damping ratio specifies how much overshoot the oscillations have and how rapidly the plane will return to its initial conditions. Damping ratios of 70% are considered ideal. Anything above 90% was considered too sluggish, and below 50%, the plane would oscillate excessively. The frequency for SCquirt was found to be 2.13 Hz and the damping ratio was 57% which fell within the acceptable range.
5. _Stall Characteristics_: Stall characteristics of the aircraft were also analyzed to make sure that, in the event of stall, the plane would behave in a predictable fashion. The tail was designed so that it would stall later than the wing to prevent a loss of control in the event of stall. In addition, an un-tapered wing was selected so that the wing roots would stall earlier

than the tips to prevent sudden rolling and provide an upwash at the tail resulting in a pitch down moment for stall recovery.

4.2.4 *Structural Sizing*

The main aircraft structural components were the wing spars, the fuselage's "backbone", the tail boom, the landing gear, and joiners to attach them to each other. Robustness, ease of manufacture and reparability were considered in the design of each of these items. Each component was sized to withstand a variety of extreme cases, such as high G turns, hard landings, wing tip hits, etc.

1. Wing Structures: The wing was composed of 3 sections: a middle 2 ft long section permanently affixed to the fuselage and two outer 3 ft. long sections. Several spar designs were considered, such as I-beams, carbon tube spars, live spars with carbon fiber spar caps, C-beams, and square-beams. For ease of assembly and simplicity of design, carbon tube spars were chosen. Each section contained a carbon tube to carry the loads from the wing to the fuselage. The tubes inside the outer wing sections were smaller than the tube in the inner section so that they could slide together during assembly. It was decided to make the tube for the inner section larger because it would be handling more bending loads than the outer sections. This was because the higher moment of inertia from the larger diameter tubes allowed for more toleration of structural loads due to bending. The overlap between the two tubes of 6 in. was sufficient to transfer the shear loads. The major condition for sizing these tubes was a high G turn. To determine the maximum turning loads the plane would generate, a turning envelope was created. This is shown in Figure 4.2.10. It was found that, at cruise velocity, in order to stay within the boundaries of the thrust limit and stall limit it was necessary to set a structural limit of 10 G's. Hence, the wing spars were sized to withstand an unloaded turn at 10 G's with a 1.5 safety factor. The Shear and Bending moment diagrams for a semi-span under this loading condition are displayed in Figures 4.2.11 and 4.2.12 respectively.

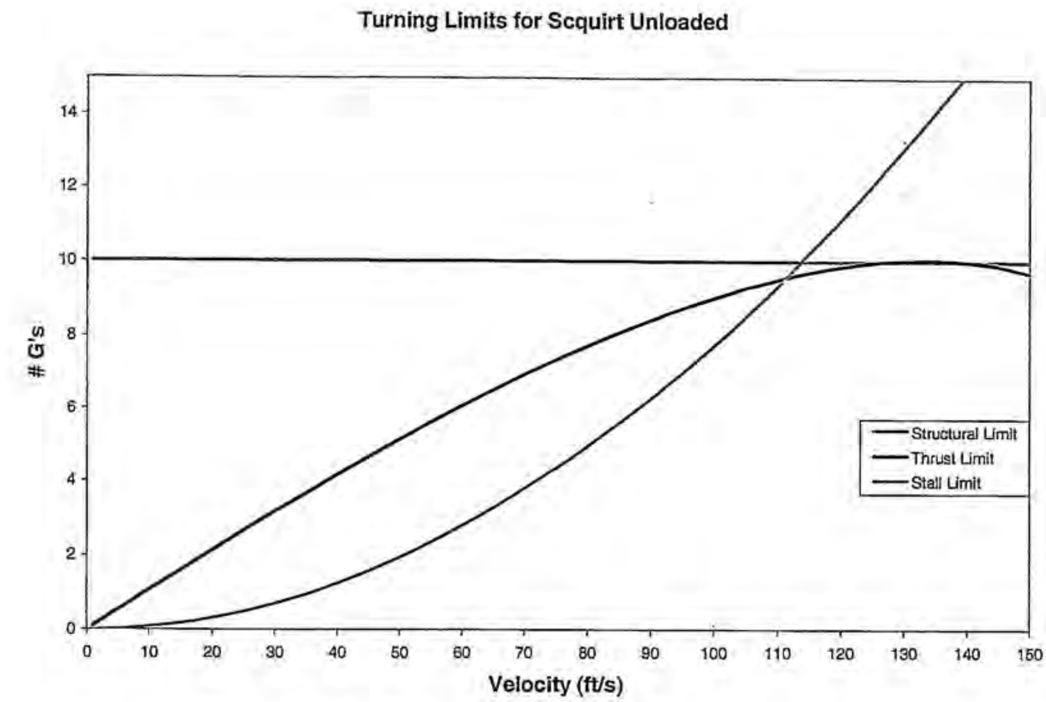


Figure 4.2.10 Turning Envelope for SCquirt (unloaded)

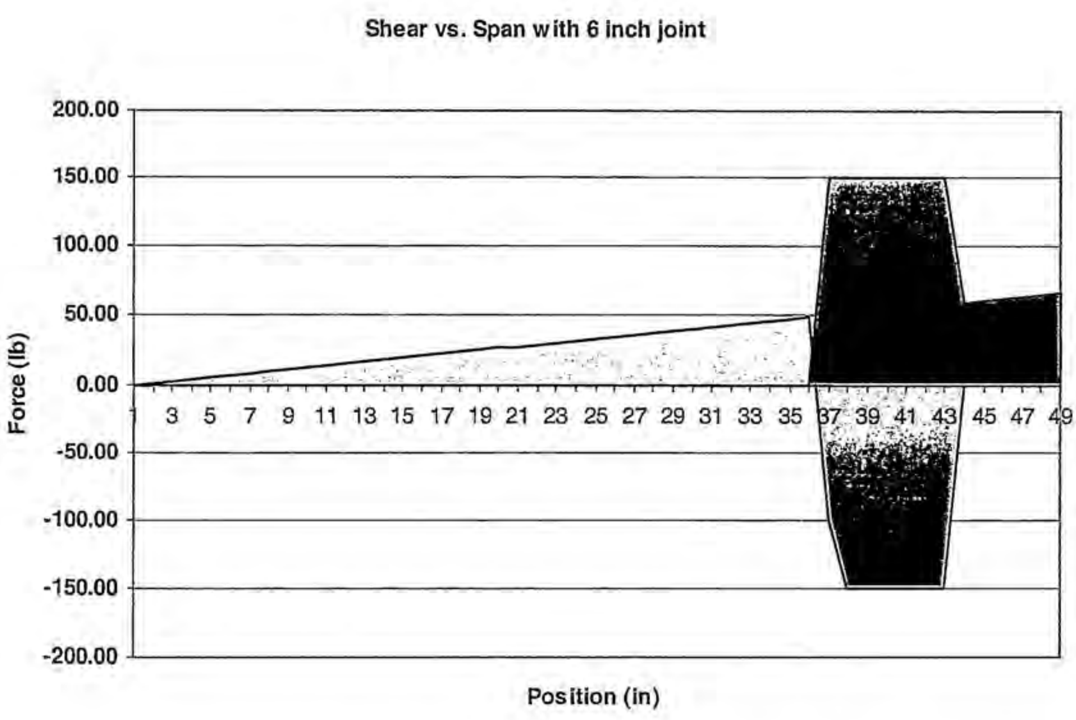


Figure 4.2.11 Shear Loading for semi-span of wing spar in 10G unloaded turn. Red denotes outer wing section, and blue denotes center wing section

Bending Moment vs. Span with 6 inch joint

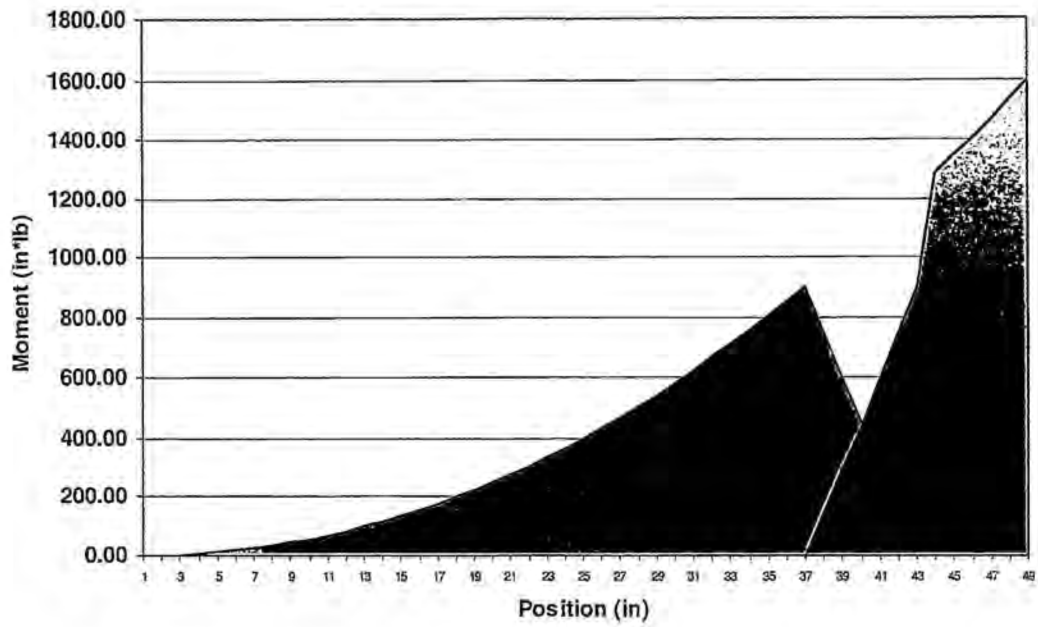


Figure 4.2.12 Shear Loading for semi-span of wing spar in 10G unloaded turn. Red denotes outer wing section, and blue denotes center wing section.

An Excel spreadsheet was created to determine the amount of carbon needed to safely transmit these loads. Other calculations were made to ensure that no skin rupture, shear failure, or core crushing were allowed

2. *Fuselage Structures:* The main structural components of the fuselage were the carbon "backbone" tube and the tail boom. The tail boom had a smaller diameter, so it was able to slide into the backbone during assembly. A 3 in. overlap was left between the two to transfer shear loads. The main condition sizing these structures was a hard landing which would transmit a large load through the landing gear structures. The shear and bending moment diagrams for the backbone and tail boom for this loading case are shown in Figures 4.2.13 and 4.2.14 respectively. The amount of carbon needed was determined using an Excel spreadsheet which calculated the strain energy allowable for a reasonable deflection given a thickness and orientation for the layers of carbon and the values of shear and bending moment from the previous graphs. The layers were bonded at 45° and -45° on the mandrel to withstand the torsional forces on the plane. The backbone was also equipped with carbon caps orientated at 0° for bending strength and hoops orientated at 90° for added stiffness where the motor and tail boom were attached.

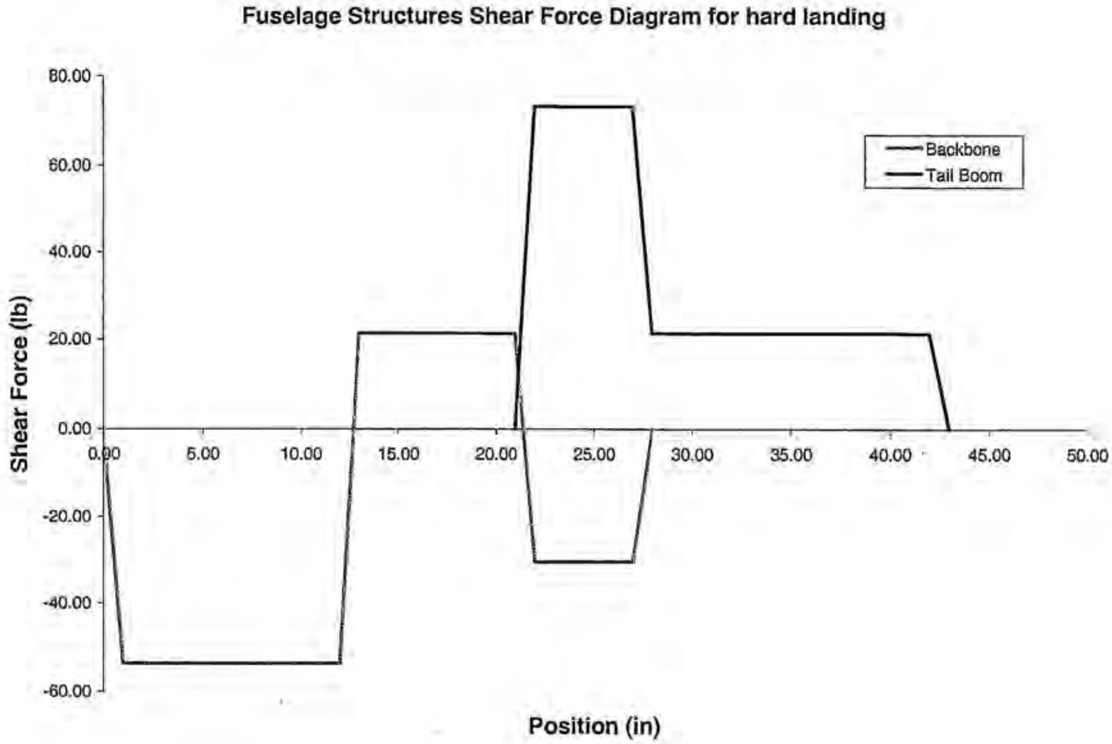


Figure 4.2.13 Shear force diagram for backbone and tail boom during hard landing

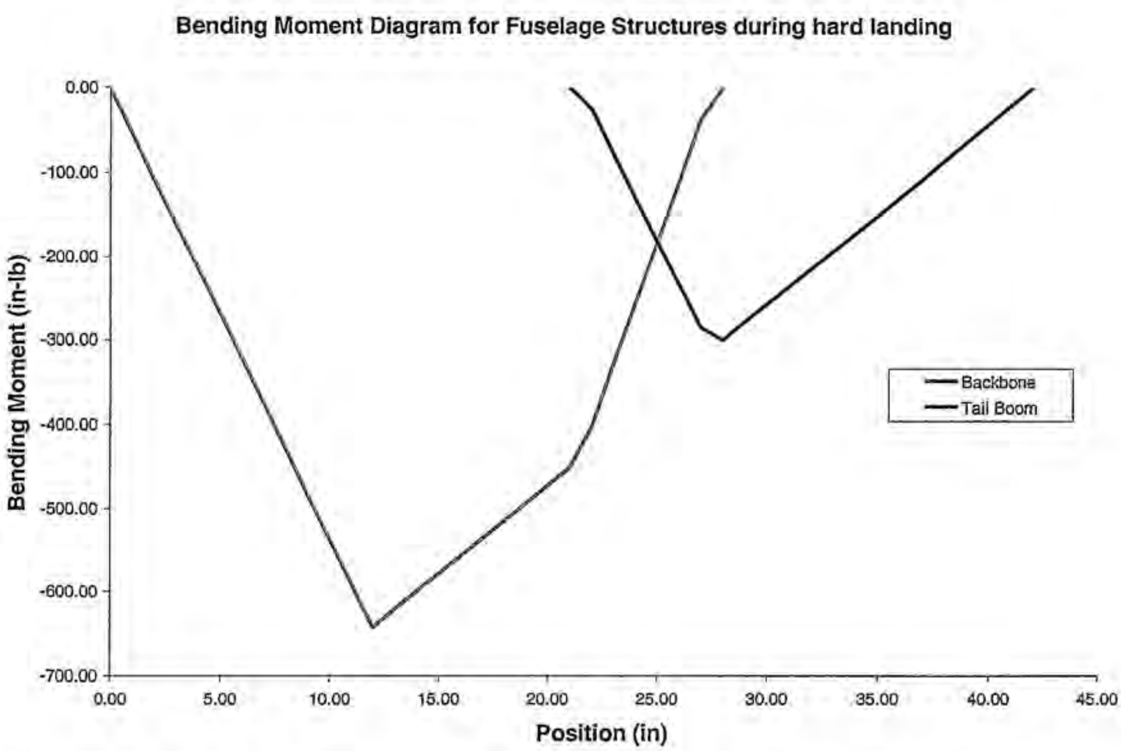


Figure 4.2.14 Bending Moment diagram for backbone and tail boom during hard landing

3. *Landing Gear Structures:* Two types of landing gear were considered: tricycle gear and tail-dragger gear. The structure necessary to incorporate a nose wheel into the airframe would weigh considerably more than that of a tail wheel. Also, having a tail wheel would allow for the ground steering to be tied into the same servo as the rudder, reducing rated aircraft cost. For these reasons, the tail-dragger configuration was chosen. The main gear (forward wheel struts) were sized to give the propeller sufficient ground clearance while allowing 1.5 in. of vertical deflection during a 5G landing. Structural analysis was done for the chosen geometry, and it was determined that 24 layers of unidirectional prepreg carbon fiber were needed to give the required deflection. The bolts attaching the landing gear to the backbone were sized to shear off if the wheels were brought to a sudden stop while the plane was rolling at speeds in excess of 10 ft/sec to prevent tip over.
4. *Wing Spar/Backbone joiner:* An advantage to the backbone/tube spar system was the simplicity of joining them together. The design for this joiner entailed two carbon fiber saddles orientated 90° to each other. A picture of one of these joints is shown in Figure 4.2.15. The two saddles were joined together by four nylon bolts that were sized to shear off during a 1G wingtip strike. That is, a 1G aft wingtip loading would shear the nylon bolts so that the wing would easily separate and sustain minimal damage during a crash. The landing gear was joined to the backbone by similar methods.

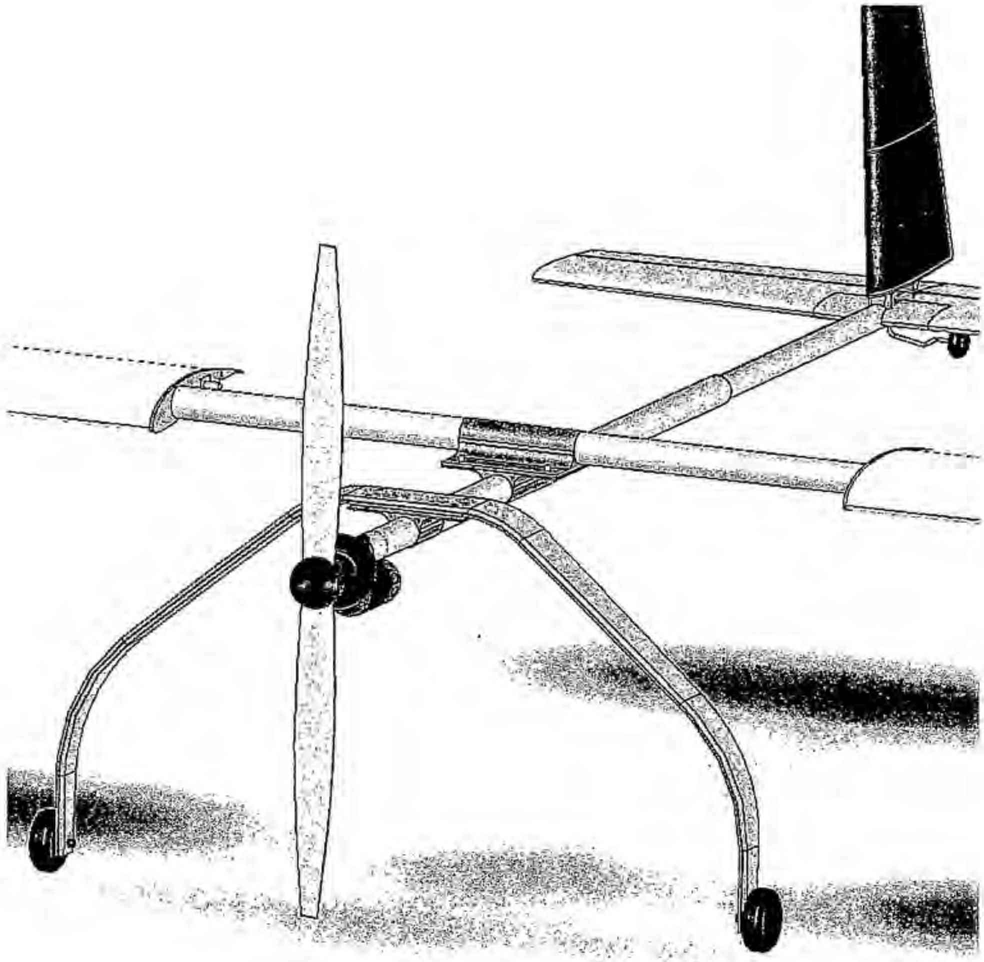


Figure 4.2.15 Joint connecting wing spar and landing gear to backbone

4.2.5 *Water Drop:*

This year, the rules for the FireFight mission called for four liters of water to be dropped out of the airplane while the plane was in flight. All the water needed to be dumped during the downwind leg of the flight and had to exit out of a half inch diameter hole. One of the major challenges of this requirement was to minimize the time it takes the water to exit in order to accomplish a complete draining of the tank during the downwind leg. It was estimated that a downwind leg takes 17-18 seconds to complete. It was decided that a 15 second drop time was most desirable to give a minimum 2 second safety margin. Preliminary tests showed that draining 4 liters of water out of a half inch diameter hole, unaided, took approximately 25 seconds. The goal of the Payload team was to design a tank and a nozzle to create a configuration that would reduce the drop time by 40%.

The tank design was driven by the airplane geometry. Because the cost equations favored a long narrow fuselage, a simple rectangle box was chosen for a tank design. Having square corners maximized volume and having a long rectangle allowed it to fit inside a narrow fuselage. The maximum height and width of the tank was chosen to be four inches. It was assumed that the tank would be made out of wet lay-up carbon cloth but would offer no structural support.

The nozzle design and location was significant in reducing water drop time. A 3-D model of the nozzle was designed in Solid Works and then machined out of plexiglass for testing. The nozzle needed to be thin to fit under the fuselage and offer limited drag. It also needed to be servo activated and designed with a filleted exit hole. Figure 4.2.16 shows a CAD drawing of the nozzle.

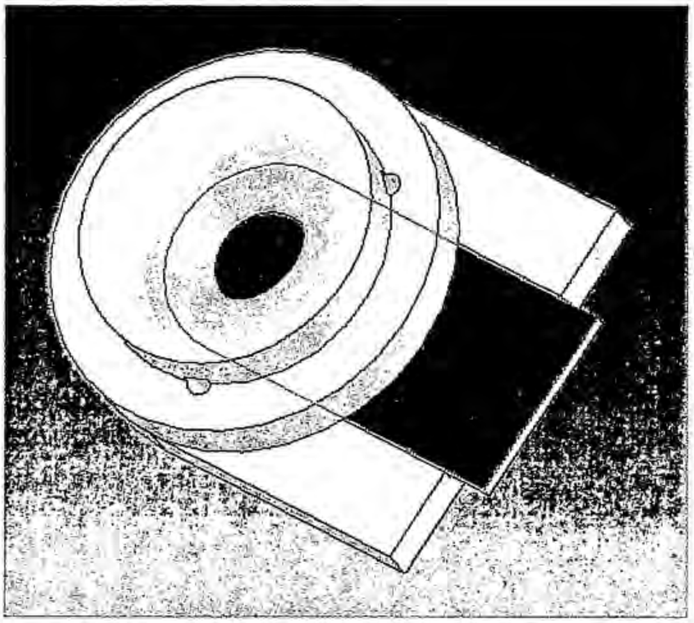


Figure 4.2.16 Knife gate valve and nozzle. The flow would be from upper left to lower right when the knife valve (in green) was opened.

In the preliminary design phase, fluid dynamics, pressure differences, G-loading, and nozzle radius were analyzed and predictions were established. The payload group set up extensive testing to compare theoretical calculations with experimental results. Because the FireFight mission depended so heavily upon the water drop, proper performance was key.

5.0 DETAILED DESIGN PHASE:

This design phase finalized the configuration of the aircraft by refining the analysis performed during the preliminary phase. New assumptions were incorporated within the design that focused on detailed aspects of certain systems. Aircraft components were selected based upon a convergence of analytical results from PlaneSizer and experimentally tested data.

5.1 Component and Systems Architecture Selection

1. Propulsion System: The radio used to drive the propulsion system was a Futaba 9CA model. The speed controller used was a Jeti 100. This model was chosen because it can handle 100A peak current for 30 seconds and 90A current continuously. For this year's propulsion model a gear box was necessary. The gear box was InnerDemon 3.5:1. It allows for easy change between gearing ratios. The chosen motor was the Graupner 930-10 with 28 cells of CP-1700 NiCad batteries, and a 21-14 propeller. This system was optimized in PlaneSizer and experimentally verified on the thrust bench as being the best system for this year's aircraft.
2. Stability and Control: The empennage consisted of a conventional horizontal and vertical tail assembly. The relevant geometric and aerodynamic properties of these tails are given in Table 5.1.1.

Stability Parameters	Horizontal Tail		Vertical Tail	
Tail Area	0.58	ft^2	0.45	ft^2
Tail Span	1.53	ft	0.94	ft
C_{root}	0.42	ft	0.59	ft
C_{tip}	0.34	ft	0.35	ft
Static Margin	25%		---	
Directional Stability ($C_{n\beta}$)	---		0.002	
Pitch Frequency (at cruise)	2.13	Hz	---	
Yaw Frequency (at cruise)	---		1.49	Hz
Pitch Damping Ratio	57%		---	

Table 5.1.1 Stability and Control final parameters computed

The wing control surfaces were two full span flaperons. It was determined that flaperons were the best option due to the rated aircraft cost penalty for multiple control surfaces and servos. The drawback to this configuration was the increased stiffness needed over the area of the control surface to guarantee the same deflection angle at both the root and the tip. To resolve this, the servos were positioned at 40% semi-span to attempt to attain equal loading both inboard and outboard of the servo. The flaperon area was sized to 25% chord and each was driven by a GWS S03N servo rated at 56 oz-in. of torque. This was the lightest servo that satisfied the calculated torque requirement for maximum deflection at cruise velocity plus a 1.3 safety factor. The horizontal tail had an elevator covering 35% of the chord. This was driven by a GWS PARK L mini servo giving 39 oz-in of torque. This servo was sized using the same methods as the wing flaperon servos. The vertical tail was full flying and powered by a GWS S03T servo with 111 oz-in of torque. Extra torque was required because this servo controlled the tail wheel as well as the vertical tail.

The airfoils for both the horizontal and vertical tails were chosen to be symmetric SD8020 at 10% t/c because of their low parasite drag. Much effort was put into assuring that the overall plane CG location stayed constant throughout the water dumping process by carefully positioning the water tank and other components. This assured that the pitch stability remained within the static margin constraint, and made the plane much more reliable to fly. The pitch and yaw frequencies calculated were sufficient to avoid sluggish handling, and the pitch damping ratio, although lower than 70%, was deemed acceptable for test flights.

3. Water Drop: The payload team performed a number of static and dynamic tests, which are further detailed in the testing section. They made multiple prototypes and documented the results from wind tunnel set ups and in flight experiments. The final nozzle configuration was a thin, plexiglass, knife gate valve. It was designed with a discharge coefficient (measure of efficiency of nozzle due to losses from turbulence) of 0.93 and the radius of curvature of the exit hole was equal to the diameter. The custom designed knife gate valve was chosen because commercially available valves were either too heavy or opened too slowly. This gate-valve assembly was able to open quickly and was light enough to be powered by a servo. The final dimensions of the tank were 4"x4"x16" and the tank held the maximum water capacity, 4 liters. A ram air scoop was located at the top of the tank and the nozzle was located on the bottom of the tank near the back. Other possible locations were discussed, such as using a low wing aircraft to utilize the low pressure over the wing to aid the exit flow using the suction pressure from the air passing over the airfoil. However, the aft positioning of the nozzle placed the exiting water away from motor and batteries, and decreased the possibility of water being left in the tank.
4. Landing Gear: Stress analysis was performed on the landing gear, and it was determined that 24 layers of unidirectional carbon were necessary to achieve the required 1.5 in. deflection during a 5G hard landing. Flexibility was set by strut thickness and strength by strut chord. The landing gear was attached to the main backbone using a carbon joint. This joint was designed to be able to slide forward and aft for adjusting the CG location during test flights. The landing gear was affixed to this joint by four 3/8" nylon bolts. These bolts were sized to

shear off in the event that the wheels encountered a rigid object while the plane was rolling in excess of 10 ft/sec to avoid tip over. A steerable tail wheel was attached to the tail boom and controlled by the same servo as the vertical tail.

A pneumatic braking system was installed composed of a pressurized air bottle connected to a servo controlled proportional release valve. The pressure required to activate the brakes was determined by ground testing. The main gear was placed far enough ahead of the CG location to avoid tip over and propeller strikes when the brakes were applied.

5.2 Estimated Mission Performance:

PlaneSizer was used to predict the plane characteristics as well as its flight performance. The different modules built by the group captains calculated the required parameters that allowed for optimization. Final configuration data is tabulated below, including geometry, performance factors and weight statements.

1. Geometry: The estimated geometry based upon the aerodynamic calculations is found in Table 5.2.1. The wingspan was set to 8 feet and the control surfaces were sized as 25% of the chord.
2. Performance Factors: Table 5.2.2 shows the different velocities for each mission. Table 5.2.3 shows the important flight characteristics such as TOFL, maximum current draw and total energy for different missions. Table 5.2.4 shows the lift and drag coefficients during cruise velocity.

Calculated Geometry	Units	Wing	H.Tail	V.Tail
Total Area Sw, SH, Sv	ft2	5.06	0.58	0.45
MAC	ft	0.63	0.38	0.48
Y MAC	ft	2.00	0.37	0.43
Projected Span	ft	8.00	1.53	0.94
Root Chord	ft	0.63	0.42	0.59
Tip Chord	ft	0.63	0.34	0.35
Root Incidence	deg	-3.50	1.00	
Tip Incidence	deg	-3.50	1.00	
Fuselage Height, Width, Length(incl spinner)	ft	0.30	0.30	4.92

Table 5.2.1 Main airplane geometry

Parameter	Units	Mission A (heavy)	Mission B1 (heavy)	Mission B2 (light)	Mission B3 (light)	Mission B4 (light)
Vcruise	fps	130.2	125.0	130.2	125.0	130.2
Vstall	fps	34.1	46.4	34.1	46.4	34.1
Vlift-off	fps	40.9	55.7	40.9	55.7	40.9
Vclimb& Ldg	fps	44.3	60.4	44.3	60.4	44.3

Table 5.2.2 In flight velocity breakdown

Parameter	Unit	Mission A (heavy)	Mission B1 (heavy)	Mission B2 (light)	Mission B3 (light)	Mission B4 (light)
TOGW	lbs	10.3	19.1	10.3	19.1	10.3
TOFL (incl safety pad)	ft	39.0	135.0		150.0	
Total Flight Segment Time	sec	106	37	23	37	23
Total Flight Energy	ft-lbs	86682	15025	14460	15025	14460
Max Cruise Current	A	41.6	43.5	41.6	43.5	41.6
Takeoff Thrust	lbs	9.7	9.7		8.7	
Thrust @ Vcrz	lbs	2.8	3.1	2.8	3.1	2.8

Table 5.2.3 Performance characteristics

Parameter	Unit	Mission A (heavy)	Mission B1 (heavy)	Mission B2 (light)	Mission B3 (light)	Mission B4 (light)
CLmax3-D	n.d.	1.53	1.53	1.53	1.53	1.53
CLcruise	n.d.	0.10	0.21	0.10	0.21	0.10
CDcruise	n.d.	0.024	0.024	0.024	0.024	0.024
L/Dcrz	n.d.	4.41	8.97	4.41	8.97	4.41

Table 5.2.4 Aerodynamic characteristics

5.3 Weights, Balance, and Rated Aircraft Cost Worksheets

1. Rated Aircraft Cost: The RAC has always been a debilitating factor for USC's aero design team. This year considerable effort was focused on reducing this parameter to be more competitive. By investigating alternatives for different propulsion configurations and structural sizing, the team was able to lower the RAC from 11 last year to 8.95 for this year. The fuselage was designed to be long and narrow and fewer battery cells were used. Table 5.3.1 shows the cost break down for the plane.
2. Weight Budget: The predicted weight of the plane was calculated through the model in PlaneSizer using measurements of the aircraft's dimensions from the design and library sheets. The weights and densities of purchased materials were taken from manufacturer's specifications and from lab test data. The lab data was very important because the manufacturer's claimed weights were often inaccurate, especially in composite materials. Lab results were also used to

create data correlations; for example the relation between propeller diameter and propeller weight. Weights for complex shapes were calculated by estimating their volume with simple rectangular approximations and then multiplying by material density. The weight of the outside skin of the wing and the tail was calculated by finding the area of the exposed surfaces and then multiplying by the thickness of the coating material and its density. The spar of the wing was treated as a separate tube and its weight was calculated using its thickness, length and density. The weight build up model is shown in Table 5.3.2 with the break down for the different systems and the Kfudge parameter (weights safety factor). Empty and Heavy Gross weights are given at the bottom of the table.

Cost Table						Total Cost	Rated Cost
Rated Aircraft Cost (AxMEW + BxREP + CxMFHR)/1000		Dims	Manuf. Hrs	Hrs (sub-total)	Cost (sub-total)	**$8,947**	**8.95**
MEW (Manufacturers Empty Weight) (lb.)		10.25		10.25	3076.37		34.4%
REP (Rated Engine Power)	= (1+.25*(#Motors-1))*BattWt			2.26	3390.2		37.9%
	# of Engines	1					
	Total Battery Weight (lbs)	2.26					
MFHR (Manufacturing Man Hours)				124.03	2480.6		27.7%
WBS1.0 (8hr/ftSpan + 8hr/ftMaxChord + 3 hr/control surface)	# of wings	1	58.1			13.0%	
	Wing Span	8.00					
	Max Wing Chord	0.63					
	# of control surfaces	2					
WBS2.0 (10hr/ft of length)	Fuselage Volume (ft)	0.55	10.9			2.4%	
WBS3.0 (10hr/VertWithRudder + 10hr/horizWithElev)	Number of Vertical Surfaces	0	0.0	Biplane Assumes Inter Plane Struts		4.5%	
	Number of Vertical Tails	1	10.0				
	# of Horizontal Tails	1	10.0				
WBS4.0 (5hr/srevo or controller)	# of Servos and Motor Contollers	7	35.0			7.8%	

Fixed Parameters	
A (Manuafcturers Empty Weight Mulitiplier) ($/lb.)	**300**
B (Rated Engine Power Mulitiplier) ($/Watt)	**1500**
C (Manufacturing Cost Mulitiplier) ($/hour)	**20**

Table 5.3.1 Rated Aircraft Cost calculation and results

System	KFudge	Sub-Component	Weight Breakdown	% of Empty Weight
PROPULSION	1.0	Sub-Total (incl Kfudge))	**4.153**	**40.3%**
		Motor(s)	1.069	10.37%
		1 Heatsink	0.138	1.34%
		Battery wt incl. solder & jack	2.260	21.92%
		All Wiring	0.100	0.97%
		1 Speed Controller	0.150	1.45%
		1 Propeller	0.220	2.13%
		1 Spinner & Prop Nut	0.055	0.53%
		1 Motor Mount	0.160	1.56%
WING	1.0	Sub-Total (incl Kfudge))	**1.601**	**15.5%**
		Wing Spar	0.035	0.34%
		Wing Core	0.775	7.52%
		Wing Skin	0.790	7.66%
TAIL & Winglets	1.5	Sub-Total (incl Kfudge))	**0.395**	**3.8%**
		HTail Skin	0.095	0.92%
		HTail Core	0.025	0.24%
		VTail Skin	0.100	0.97%
		VTail Core	0.044	0.42%
		Winglets	0.000	0.00%
RADIO	1.0	Sub-Total (incl Kfudge))	**1.225**	**11.9%**
		Receiver	0.125	1.21%
		Servos	0.600	5.82%
		Battery Pack	0.500	4.85%
LANDING GEAR	1.0	Sub-Total (incl Kfudge))	**2.099**	**20.4%**
		Main Gear Struts & Bolts	1.229	11.92%
		Main Wheels	0.290	2.81%
		MG Axle Hardware	0.072	0.70%
		Nose Wheel or Tail Wheel	0.307	2.98%
		Nose Gear Strut & Mount	0.100	0.97%
		Brakes, Tubing, Air Tank	0.100	0.97%
FUSELAGE	1.20	Sub-Total (incl Kfudge))	**0.839**	**8.1%**
		Fuselage Skin	0.559	5.42%
		Bulkheads	0.140	1.36%
		Backbone	0	0.00%

Airframe Weight =	**4.83**	No systems installed
Flying Empty Weight =	**10.31**	(includes test for actual weight)
Payload Weight (4L water)	**8.82**	
Heavy Gross Weight =	**19.13**	
Light Gross Weight =	**10.31**	

Table 5.3.2 Weight Breakdown in Lbs.

3. <u>Balance Distribution</u>: The CG for both light and heavy planes was set at the same location for ease of handling and flying. A CG balance sheet was created in PlaneSizer and linked to the configurations sheets. Since the plane was geometrically symmetrical about the center of the plane and the vertical CG location had negligible effects on the handling of the plane, the vertical CG calculations were not performed. The individual components of the plane were then moved fore and aft to position the CG so that it aligned with the desired value at 25% of the wing chord. Since the length of the plane was fixed, the batteries and landing gear placement were the items primarily reconfigured to achieve the desired CG.

6.0 MANUFACTURING PLAN:

Plans for building and manufacturing of each component were carefully documented. Detailed procedures and lists of required materials were compiled for each part, with material size, weight, and lay-up configuration listed on component drawing. Because increased weight has such a large affect on airplane performance and rated aircraft cost, great care was taken to ensure all parts were made efficiently and free of excess materials. Three figures of merit were applied to the building procedures to ensure optimal manufacturing.

6.1 Figures of Merit

1. Material Availability: Construction time needed to be minimized to allow for a greater period of time for flight testing. Because of this, materials that were readily available were chosen. Standard graphite, fiberglass, balsa wood, and foam were available through Aircraft Spruce and Specialty Company at any time. Nuts, bolts, and fasteners were ordered through McMaster Carr and were always in stock.
2. Material Cost: Historically, the AeroDesign team receives no university funding for their project. Instead, the team has relied on the support of outside donations. Because of this, using materials with relatively low cost was essential. Rolls of carbon fiber prepreg were donated, valuable scraps of materials were carefully saved, and, if possible, parts were student built instead of purchased. All vendors used by the team were chosen not only for having materials immediately available but for also providing materials within the budget range.
3. Required Level of Workmanship: When constructing a component, efficiency is desirable, but the construction skills of team members needed to be considered. The option of using a monocoque fuselage was considered and dismissed for many reasons; for example, a high level of experience and knowledge of composites is required to create such a part. Instead, carbon composite material was used but simplier parts were designed such as tube spars that were easily wrapped and a square fuselage mold that was lined with a wet lay up. The older team members often lead building tasks while new members gained experience.
4. Repeatability and Crashworthiness: All parts were design so that they could be easily and quickly created, repaired, or replaced. Should a part fail during flight testing, the ability to substitute it with a new one in a short amount of time was essential.

6.2 Analytical Methods Used to Select Final Set of Manufacturing Processes

The manufacturing skill of the team members was considered during the design process to ensure that the team had the capability of building this year's plane design. The building skills of team members were considered ahead of time so it could be determined who would take the lead for each constructed component.

Last year the team received a donation of unidirectional carbon fiber pre-preg. The use of this material was assumed in the design of the structural components. Because many team members gained experience with this material in building last year, the skills, processes and techniques were already understood and were passed on to the newer team members. Tube spars, for example, were first introduced to the team last year. It was estimated that it would take two students about 6 hours to make a tube 40" in length with a 1" diameter. With four spars needing to be built, this was a time consuming process. However, with more experience, the process became more efficient. Using commercially available carbon spars was considered but the specific dimensions and ply orientation needed for structural support could not be matched. The tube analysis and construction proved reliable and efficient based upon last year's plane building and was repeated for this year's construction.

Other components of the plane, such as the wing, landing gear, and fuselage shell, were built with more common methods of construction. Wet lay-ups, female molds and foam cores were used and were easily built based upon experience gained form previous years. This gave an opportunity for these techniques to be taught to new members and allowed for minimal manufacturing time.

6.3 Processes Selected for Component Manufacturing

The manufacturing processes are given for each of the following components.

1. <u>Wing Spar and Backbone:</u> With extensive work and analysis done on shear and moment forces, the backbone and wing spar were both designed to take a maximum crash loading of 5Gs. Tubes were needed for both wing spars, for the main boom and for the backbone of the plane. All were student crafted in the following manner:
    a. <u>Mandrel Preparation:</u> Thin walled aluminum tubes were purchased with outer diameters matching the required inner diameters for each of the tubes. To prepare these tubes, the ends were de-burred in order for the carbon tubes to slide off easily. Next, the aluminum underwent several rounds of wet sanding with 600, 800 and 1000 grit sand paper. The sanding was performed to remove any scratches that would cause impurities in the carbon lay-up or cause difficulties in removing the final tube. After the sanding the mandrels were cleaned with acetone. From this point on, it was crucial to keep the mandrels clean to prevent contamination and bonding problems. The mandrels were then polished with wax and buffed out lightly with cotton rags. This process of wax polishing was repeated three times. Acetone was then used again to clean the mandrels. They were then brought into the exhaust room where

Freekote44 was applied to the mandrels with a lint cloth until a continuous, smooth wet film was apparent. This process required eye protection, respirators and gloves. The mandrels were then heated for 10 minutes at 150°F with a heat gun. The release agent was then buffed out. This procedure was also repeated three times. After the third time, Miller-Stevenson mold release was sprayed on the mandrels. Protection for eyes, lungs and hands were also needed. The mandrels were then heated again for five minutes and buffed out. This was the final step in preparing the mandrels for the carbon lay up. Great care was taken in handling the mandrels to ensure that the carbon tubes would release properly after they were cured.

b. Strip Cutting: Dimensions and layout of the prepreg material were calculated and strips were cut ahead of time. All tubes had four layers for 45° and then 0°caps. The width of the 45's needed to be calculated as well as the length to cover the span of the tube.

c. Strip Layout: Shear strips were first wrapped on the mandrel. The strips were laid out and stretched by hand so that no bubbles or wrinkles were formed. If any imperfections were present during lay up they would only be magnified with each layer applied, drastically weakening the strength of the tube. After the first layer of shear strips was applied, the tube was then wrapped with a 1" stretch tape with a hand-held tension regulator. This procedure helped eliminate any remaining wrinkles. The wrap was kept on for 10 minutes. After removing the tape the remaining three layers of shear ply was applied, alternating +45° and –45°. After the shear plies were added, another layer of stretch tape was applied and then removed. Caps were then applied to the top and bottom of each tube, covering a portion of 60° each. A heat gun was used to increase the adhesive properties of the caps so they would attach better to the mandrel. Hoops at 90° were added last at the hard points of each tube. This was to strengthen the tubes at the joints to prevent crushing or splitting.

d. Curing and Removing: After the tubes were laid up, they were wrapped one more time with stretch tape and placed into an oven to cure at 270°F for approximately six hours. The mandrels were then taken out of the oven and placed in a freezer to allow the aluminum to contract slightly. The carbon tubes were then pulled off, either by hand or with a wench.

2. Joiners: The plane configuration this year required the wing spars and the backbone to be joined at a 90° angle. The joiner used was designed last year under the guidance of industry advisors and proved successful. The design was repeated and improved slightly this year to help the loads from the wing transmit to the mount through compression. This compression was reacted by the mount and bolts through tension. The bolts then transmitted the loads to

the backbone by compression. As designed, the backbone also supported the shear forces. These joiners were made out of pre-preg carbon fiber. The surface of the main boom was sanded down to create a rough surface onto which the joiners would attach. A block of balsa was formed to fit the curvature of the tube and secured with epoxy and micro balloons. Four layers of carbon were laid flat alternating between 0 and 90. Two layers of carbon at 90° were wrapped over the tubes and attached to the flat pieces. The joints were bagged to an aluminum call plate and cooked at 270°F for 6 hours. The flat ply orientations added strength in the direction of the load path and gave a balanced lay-up. The top plies helps to transmit aft wing tip loads from the spar to the mount. Nylon bolts were used as fasteners and were designed to withstand a 5G turn load and to shear off under a 1G lateral tip load. Figure 6.3.1 shows the joiner lay up.

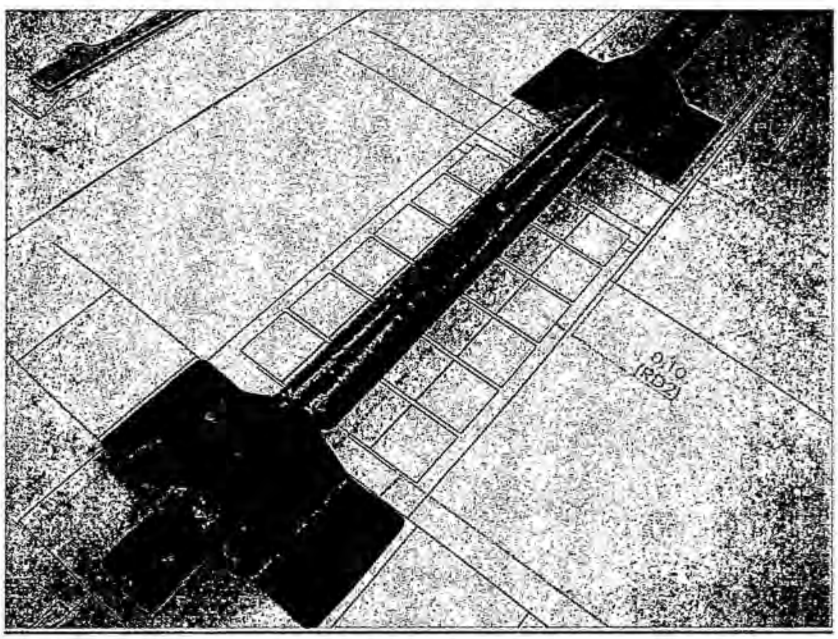


Figure 6.3.1 Joiner lay-up on main backbone tube.

3. Aero-fairings: These were made using a male mold which was hot wired out of foam and sanded. It was covered with clear tape and waxed twice. Carbon strips were taped to the molds to provide rigidity. Two layers of 1.5 oz fiberglass was laid up with reinforced corners. The fairings were bagged with latex and cured.
4. Water Tank: For the water tank, templates were drawn up and the shape of the tank was separated into two sections; the top, essentially being a tank lid, and the bottom, which held the entire volume of water. For the construction of the bottom portion of the tank, a female mold was cut out in tooling foam and covered in monokote. The surface was waxed to act as a release agent so the mold would be reusable. This mold was then covered with two layers

of silk-span, one layer of 3oz fiberglass, and one layer of unidirectional carbon. Bi-directional carbon was used to reinforce the ends of the tank. This tank however, leaked water through sweating. A second tank was made using a male mold. The mold was made with the same tooling foam as the first and this tank had two layers of silk span and one layer of unidirectional carbon. The corners were much cleaner and the part was much lighter than the first version. Figure 6.3.2 shows the bottom of the water tank. It only contained a few point leaks which were eliminated with sealant. Painters plastic spray adhesive was applied to the inside of the tank to make a flexible waterproof liner. The top of the tank was also made from a male mold and constructed with the same lay up schedule as the female molded tank. The top and bottom were sealed together with layers of fiberglass and epoxy with West Systems #410 micro light fairing filler.

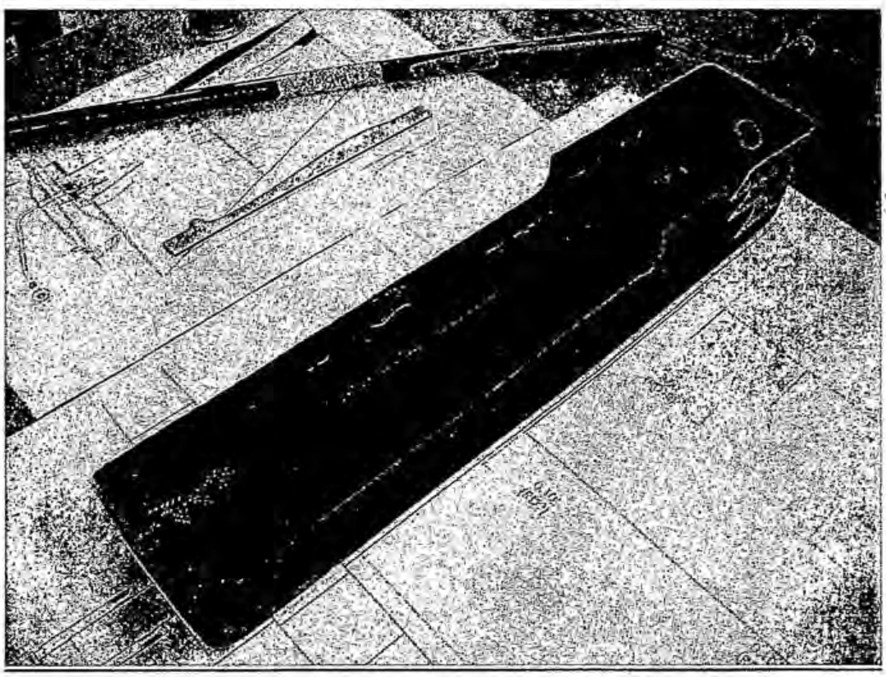


Figure 6.3.2 Bottom half of carbon fiber water tank.

5. Landing Gear: The main gear was made out of pre-preg carbon fiber (see figure 6.3.3). A wooden mold was made out of plywood to fit the drawing templates. An aluminum sheet was placed over this and strips of fibers were laid on top. Twenty seven layers of unidirectional were used with every ninth layer at 90° for shear forces. The gear was then vacuum bagged and cured in an oven at 270°F for six hours. Extra care was taken to prevent delamination at the corners. The tail wheel strut was made with piano wire.

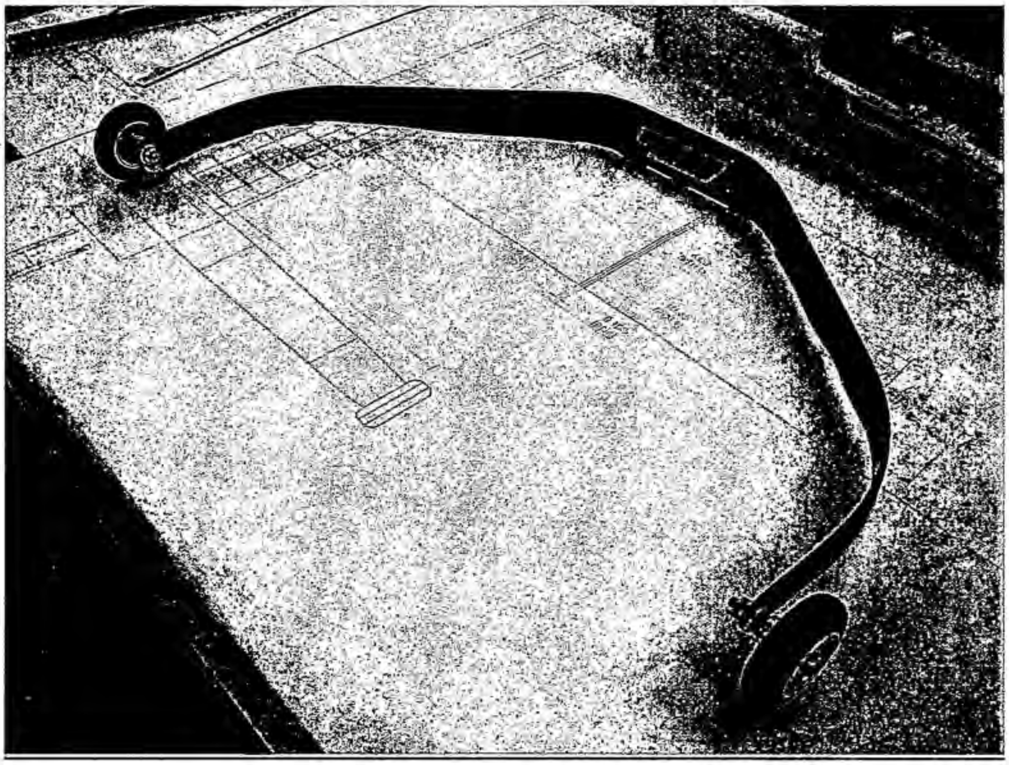


Figure 6.3.3 Picture of finished landing gear.

6. Wing: The wing was constructed out of blue Spyder foam and covered in two layers of 3oz fiberglass. The control surfaces each had a Kevlar hinge. The procedures for wing building were similar to last year's. First the configurator provided actual size drawings of the airfoil. Wooden templates were then made from a laser cutter machine. Ribs were cut from light plywood and the wing cores of the wing were hot wired out of rectangular slabs of foam. Figure 6.3.4 shows the hot wiring process. The cores were then prepared for bagging; first the spar hole was cut and the carbon spars were inserted. A servo wire channel was cut and the ribs were installed. The wings were then covered with two layers of ±45° fiberglass. This was chosen to counter the twisting moments of the wing and to reduce any dimpling of the core. The cores were then vacuum bagged separately. Once the epoxy cured, the hinge line was cut and the control surfaces were wrapped again in fiberglass and bagged separately. They were then reattached and servo holes were cut and servos were installed. Figure 6.3.5 shows a completed wing.

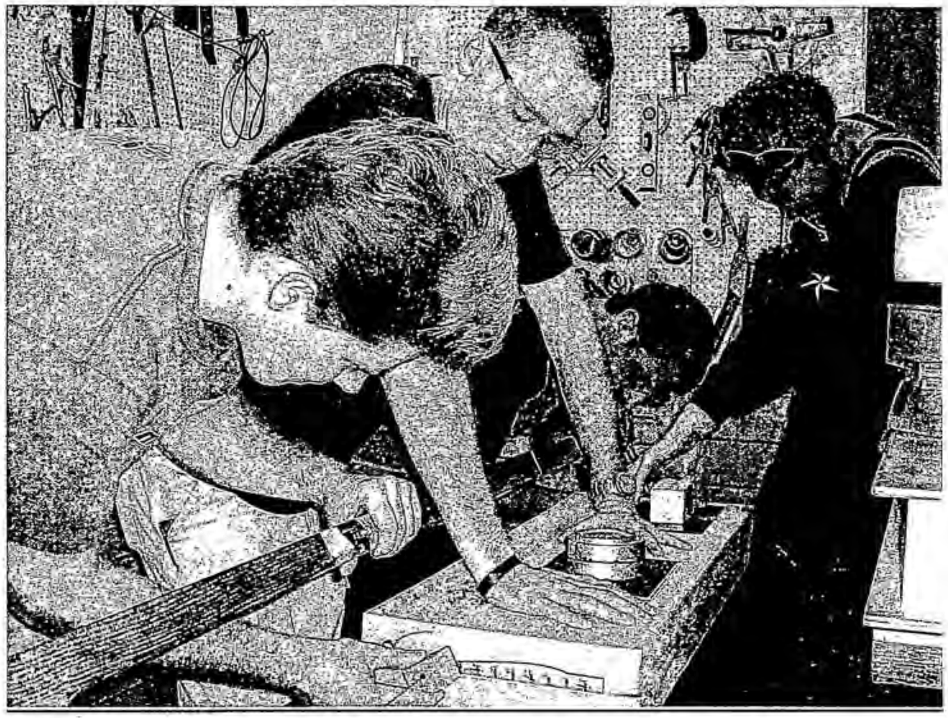


Figure 6.3.4 Team members cutting foam cores for wing.

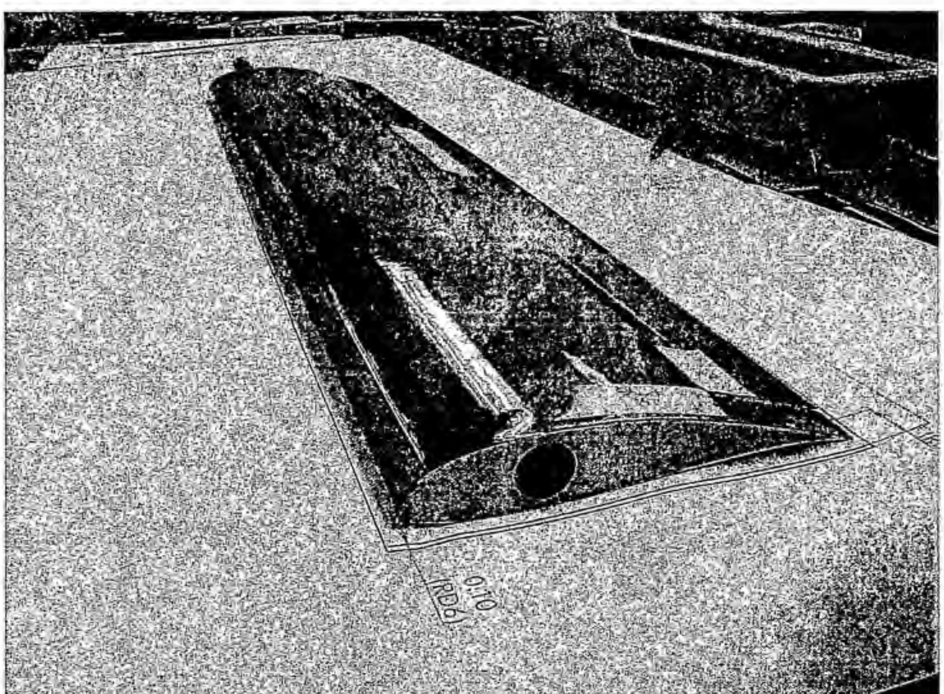


Figure 6.3.5 Picture of completed wing half with carbon spar inserted and covered in fiber glass.

7. Empennage: The building of the tail was divided into two main sets: one for the horizontal tail and one for the vertical tail. A similar process was used for tails as for wings. Drawings and

templates were made and cores were hot wired out of a foam bed with appropriate thickness. For the horizontal and vertical tails, 1.6oz blue foam was used. Ribs for the vertical tail were cut out of light plywood. The vertical tail core was prepared with a channel cut out for a brass tube. A hard point, for the tail wheel wire, and bolting locations were put on the horizontal core. The hinge line was drawn before bagging. The control surface on the horizontal tail was made with a Kevlar live hinge. Each piece was wrapped with one layer of 3oz fiberglass at 45° and carbon caps were laid on top. The pieces were then vacuum bagged separately. Once cured, the hinge line on the horizontal was cut, wing tips were attached, the servo horn hole was cut and the carbon tail wheel attachment was inserted on the vertical portion. Figure 6.3.6 shows a picture of the completed horizontal tail.

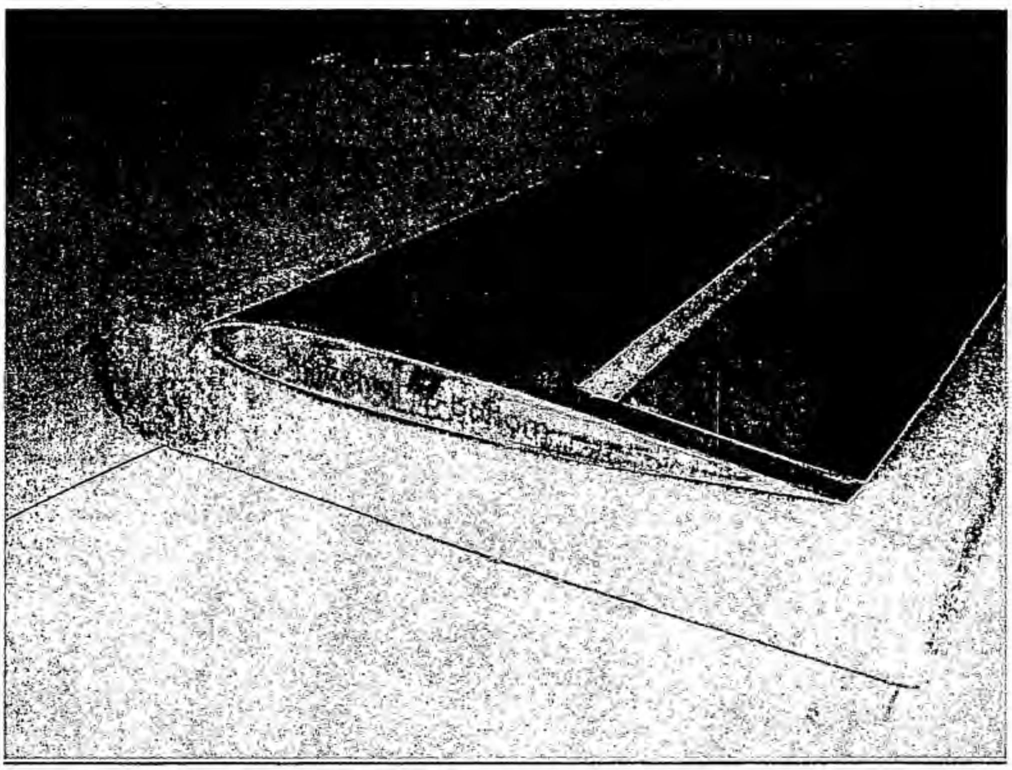


Figure 6.3.6 Completed horizontal tail on foam bed.

6.4 Manufacturing Schedule:

Table 6.4.1 shows the manufacturing schedule. Each part needed to be built was separated and assigned an appropriate allotment of time for component completion.

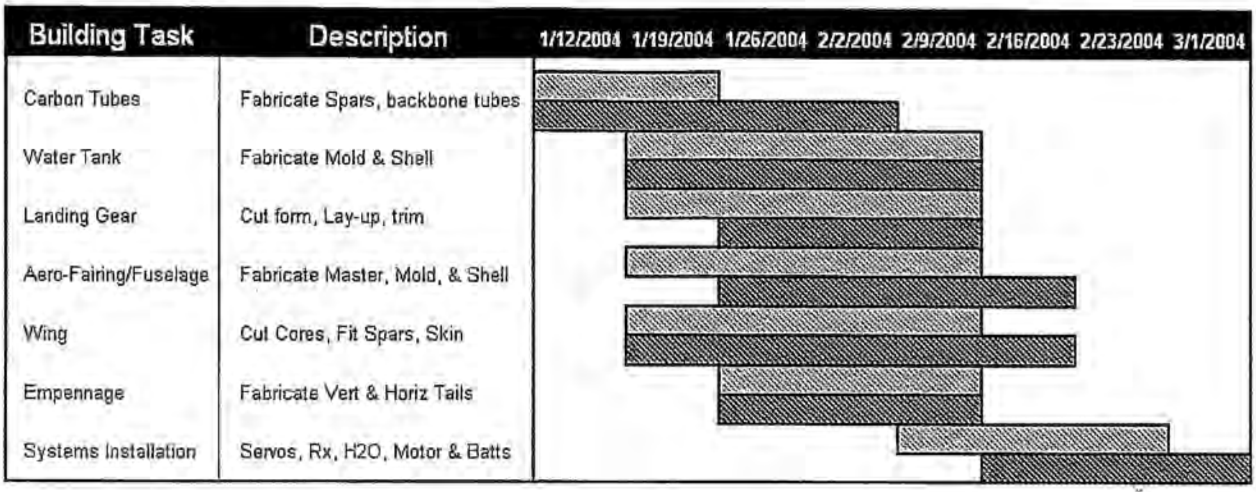


Figure 6.4.1 Detailed building schedule with manufactured parts and dates. Light shaded areas denote planned building, dark shaded areas denote actual building.

7.0 TESTING PLAN:

7.1 Component Testing:

1. Propulsion testing: In past years, manufacturer specifications were unable to be trusted in regards to battery and motor performance. To provide a reliable propulsion system, extensive testing was done on both the motor and battery packs.
   a. Motor Testing: The Graupner 930-10, the Graupner 3300, and the Cobalt 60 were all placed on the thrust bench for testing. The 930-10 was found to be most suitable for this year's plane so more detailed thrust bench testing was conducted for this motor. Figure 7.1.1 shows the motor attached on the thrust bench during one of the tests. The voltage, current, torque, and thrust were all recorded during these tests. The goals of the motor tests were to determine the motor constants and to determine if it would operate at the required power lever for the required time.
   b. Battery Testing: Tests were done on battery type and cell number. CP-1300s, 1700s and 2400s were all tested. Batteries were tested for voltage, current and duration of the pack as a whole. The goals of battery testing were to ensure that the voltage and battery life was what was predicted and to verify that they would perform well with the given propeller and motor. For a battery endurance test, the power coefficient was graphed versus the design advance ratio. From this it was possible to find a propeller that would simulate cruise. This prop simulates cruise by drawing less current. The 22x10 prop was used and proved that the CP1300s gave a minimum of 3.5 minutes of power. A test was also performed to document what happens to the

batteries when the motor was briefly pushed up to full throttle. The amps needed to be monitored to ensure it would not blow the motor. Figure 7.1.2 shows the data obtained from this battery test.

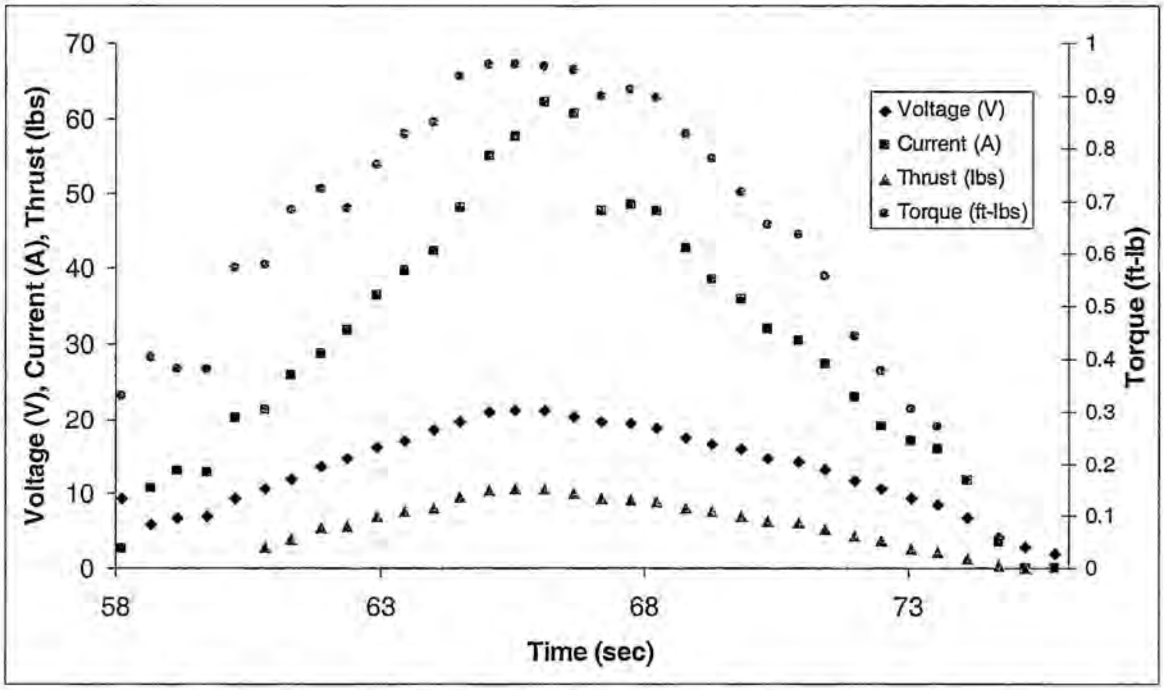


Figure 7.1.1 Time trace of the data for the Graupner 930-10 with the competition prop (zinger 22x14)

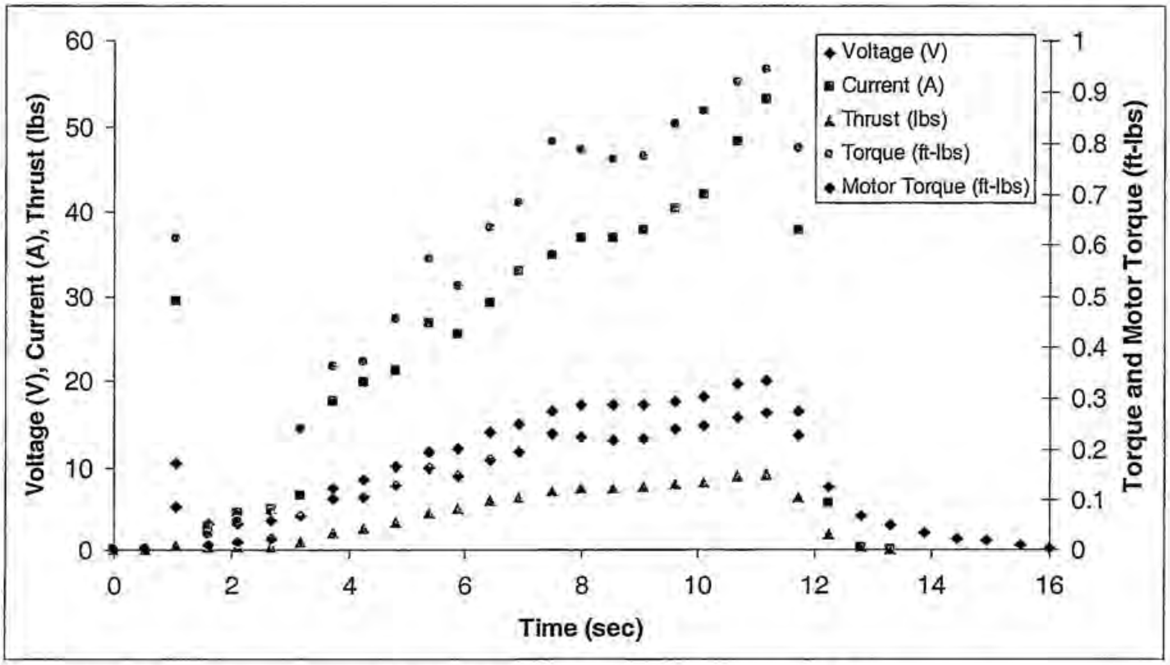


Figure 7.1.2 Time trace for CP 1300's with a 22x14 prop and 3.5 gear ratio.

2. Water Tank: The water tank tests were one of the first tests to be executed. Because the initial water dropping techniques were not dependent upon plane design, they were immediately discussed and analyzed. Water tank testing was divided into two categories: static testing and dynamic testing.

    a. Static Tests: First a plastic tank was constructed. The tank was built 4 inches tall, 4 inches wide and 16 inches long. The tank was sealed using silicone and tubes were built into the top of the tank to allow for pressure readings, refilling, and air inlet. The initial plan called for using pressurized air to create an aerodynamic pressure differential. This plan was deemed impractical due to problems with obtaining a suitable pressure transducer. The air inlet hole was then modified so that it could catch RAM air pressure when the tank was put into a wind tunnel. The tank was placed into the wind tunnel with a 4" hose attached to the end of the nozzle to direct the water out of the tunnel. Because it was not possible to dump water into the free stream air of the wind tunnel (which would be at a lower pressure than room pressure), it was necessary to recreate this lower pressure in the drop zone. This was accomplished by dumping into a four liter beaker that was connected to a tube which had one end perpendicular to the flow in the tunnel, as shown in Figure 7.1.3. This setup was run in the wind tunnel at several different velocities. For each velocity, the pressure in the tank was recorded using a pressure transducer. The air speed was also calculated using a pitot tube and recorded. The valve was manually opened using a servo controller, and the time was taken using a stop watch.

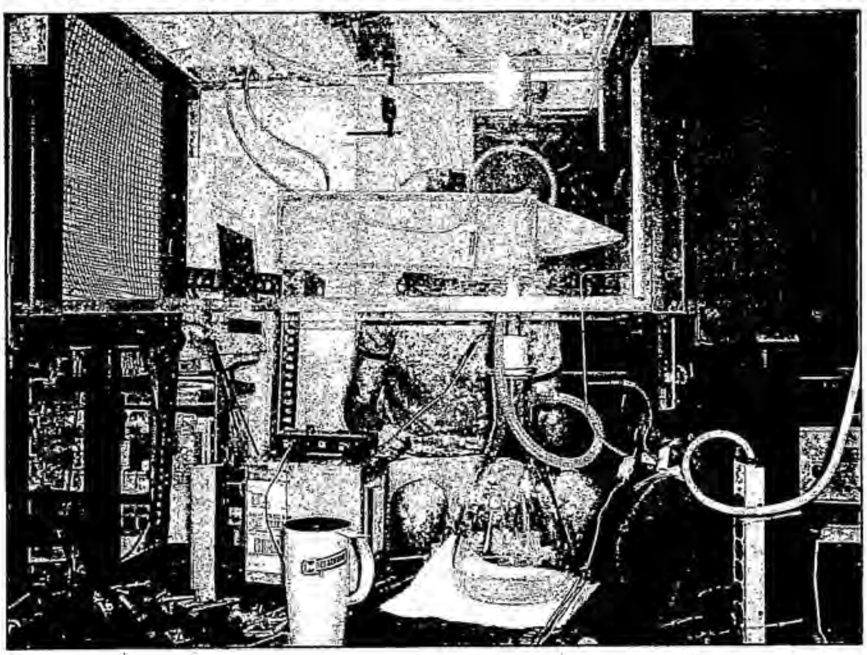


Figure 7.1.3 Wind tunnel set up for static water tank testing

b. Dynamic Tests: In addition to wind tunnel testing, two tests were performed with a tank in flight. A separate tank was constructed and placed into the cargo bay of last year's competition plane. A ram air scoop was placed above the plane and the exit was perpendicular to the flow below the plane. These two tests were run with the plane flying at about 45 mph. The inlet scoop used for test flights was rather crude; a more refined design would probably result in a higher pressure differential due to fewer losses and thus a reduced drop time. It was noted during the test flights that the flow rate out of the nozzle increased significantly during the turns.

c. Results: The data taken from the tank in the wind tunnel is shown in the graph in Figure 7.1.4. The graph shows the pressure and the time to empty the tank for each velocity. The results show, as expected, that as the velocity outside the tank is increased, the pressure differential between the inside of the tank and the free stream increases.

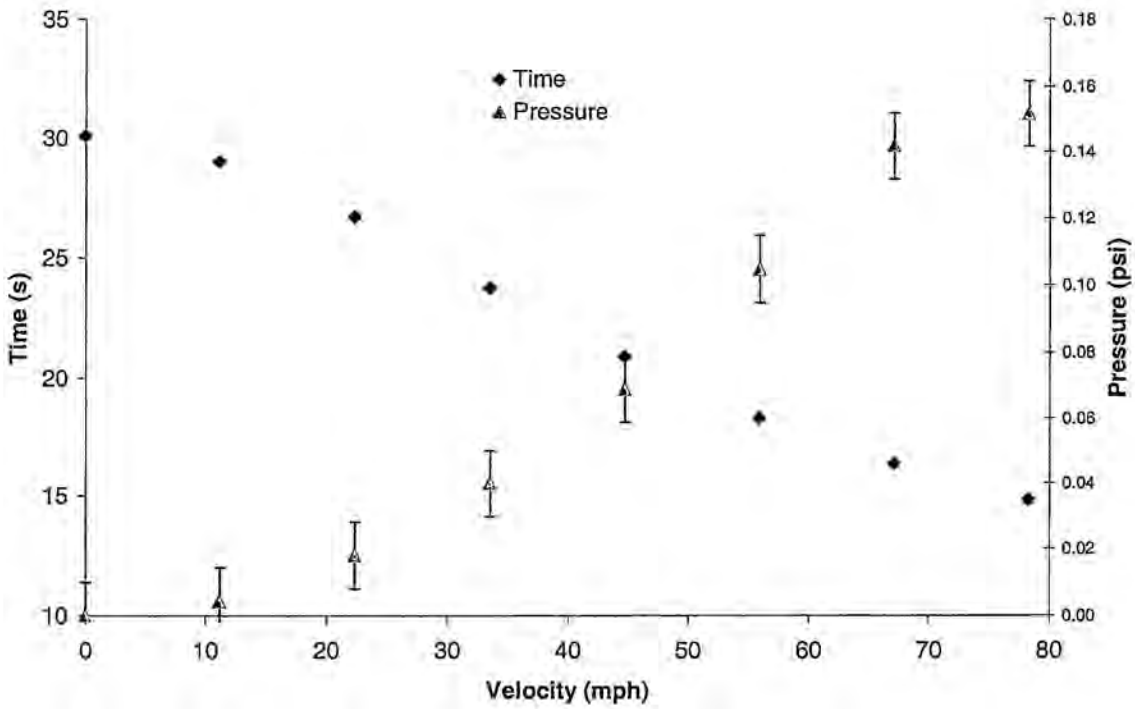


Figure 7.1.4 Graph of pressure and drop time vs. velocity.

The results obtained in the wind tunnel were actually better than predicted. Figure 7.1.5 shows the theoretical drop time (for both the added g-loading and the no g-loading case) and the actual measured drop time vs. pressure. The measured times were less than the theoretical times for the no turn case.

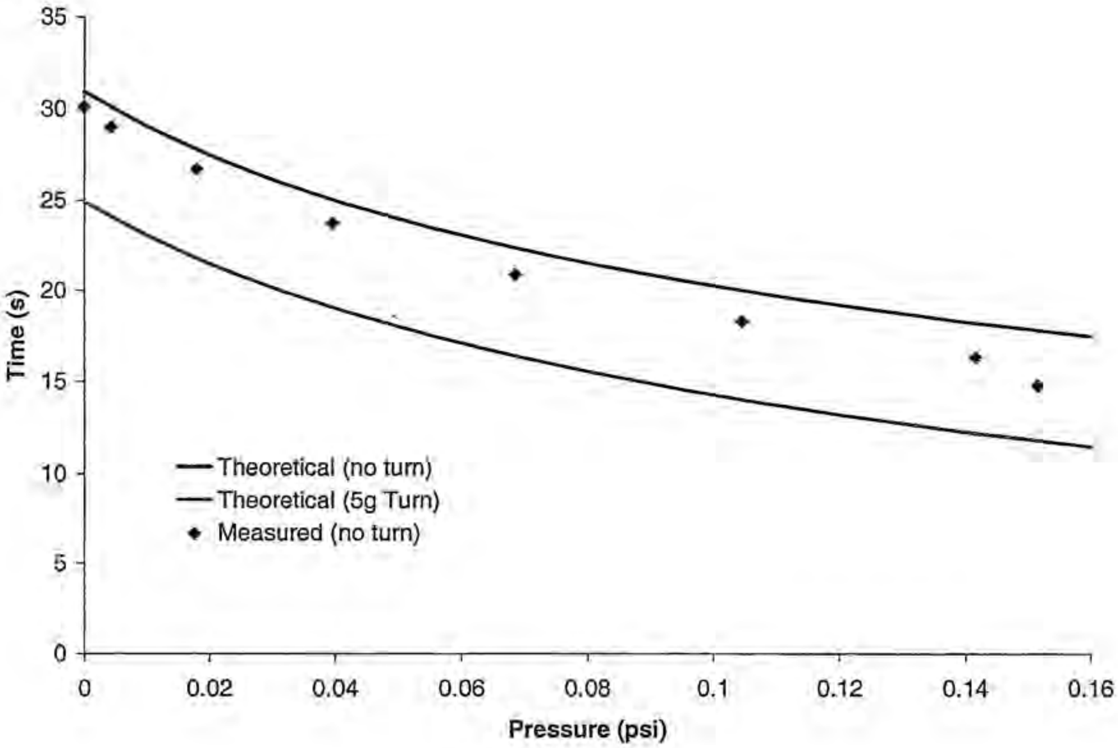


Figure 7.1.5 Theoretical vs. Measured drop times for different pressures.

These results indicate that the competition airplane this year should be able to complete the water drop in under the goal time of 15 seconds.

3. Static Divergence Testing: The full flying vertical tail was swept forward to avoid the possibility of aeroelastic flutter. As a result, there was some concern that the tail would be subject to static divergence. Static divergence occurs if an increase of C_L on the vertical tail as a result of a change in angle of attack causes a significant structural twist in the tail. This twist causes a change in apparent angle of attack, which can cause an even bigger increase in C_L. If this process repeats itself, then the tail could be pushed to the point of structural failure. To analyze this, a tail was taken from a previous competitions airplane. This tail was also swept forward and was built using similar methods to this year's tail. The tail was loaded with weights to simulate the dynamic pressure of a maximum deflection and maximum velocity. The twist in the tail as a result of this loading was negligible. Therefore, it was concluded that static divergence was not a problem given our tail sizes and construction methods.
4. Data Recorder Testing: A new device was purchased this year that allowed the team to collect data while in flight. The USB Flight Data Recorder and Flight Data Recorder Electric

Expander were both manufactured by Eagle Tree Systems. It records the inputs into the servos from the controller and saves the joystick positions in its memory. It has an internal altitude sensor that measures a 1600 foot range, but must be calibrated at ground level at different test sites. A pitot tube measures the stagnation pressure and calculates the dynamic pressure and velocity by comparing that reading with the altimeter's static pressure reading. The electric expander is hooked up to the speed controller and the batteries and measures the voltage and current that passes through them. The RPM sensor works by attaching 2 magnets at a 180 degree difference along a spinning part of the propulsion system and uses a Hall-Effects sensor to determine the RPM from the spinning magnets. The total weight of the recorder, the pitot tube, and the electric expander is .27 pounds. The recorder was located next to the receiver and the pitot tube was attached along the wing, outside the propwash. The electric expander was connected between the speed controller and the batteries. The interface used was provided by the recorder. It records altitude, velocity, receiver battery voltage, and joystick positions. With the addition of the electric expander, it also measures battery voltage, current, watts, amp-hours, and motor RPM. It was ground tested with a Graupner Ultra 3300-6 motor and was verified that the joystick positions work correctly. Figure 7.1.6 and 7.1.7 show outputs from the data recorder. The other recorded data was verified. As of this time, it has yet to be used for in flight tests.

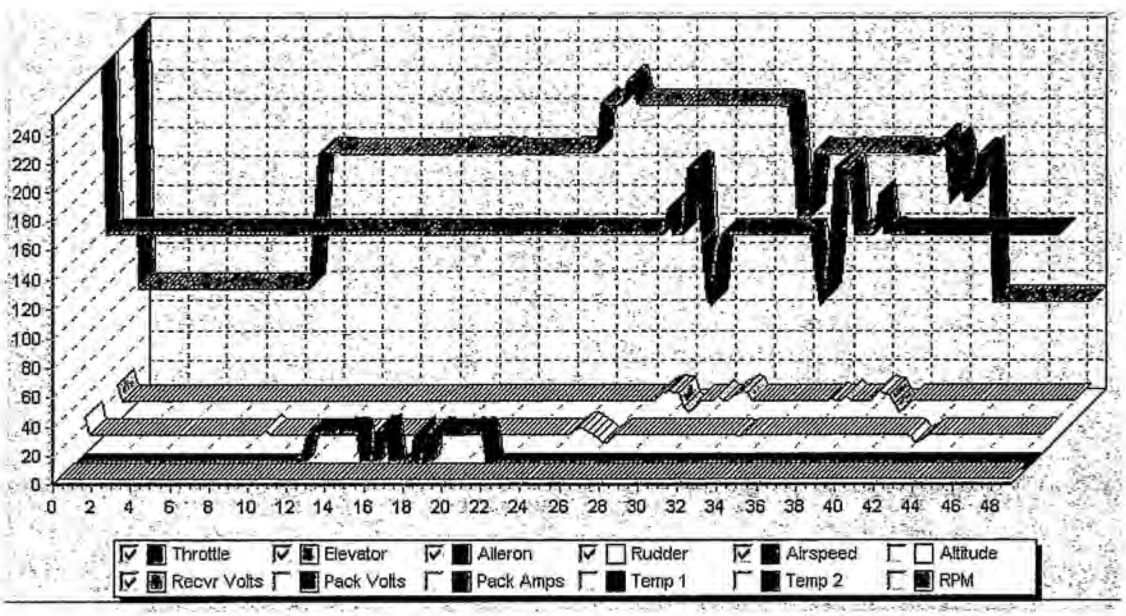


Figure 7.1.6 Flight Data Recorder graph showing joy stick positions and air speed.

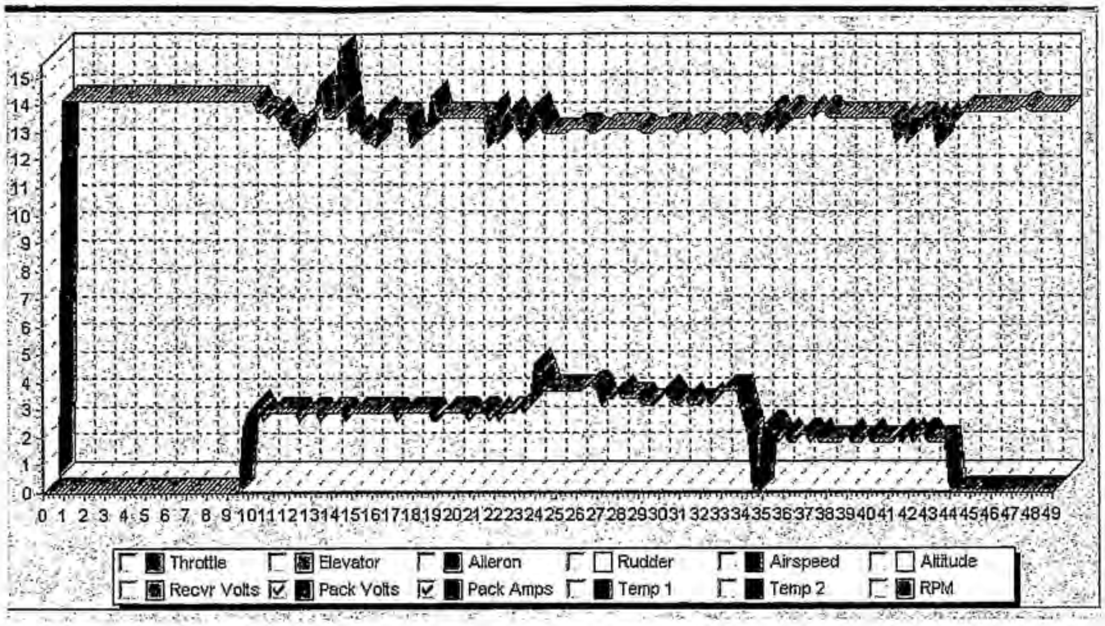


Figure 7.1.7 Flight Data Recorder graph showing voltage and current.

7.2 Flight Test Objectives.

As of report submission date, test flights have not been performed. The first flight is scheduled for March 12. Flight objectives for this first test and all subsequent tests are shown in Table 7.2.1 and 7.2.2.

Flight	Payload?	Primary Objectives	Secondary Objectives
Alpha	No	Determine trim elevator at Vcrz with 0° Flap.	Overall airplane trim and ground handling assessment
Bravo	No	Determine elevator required for stall with 15° Flap	Determine settings for geared flap
Charlie	No	Determine stall characteristics, power on/off	Assess turning performance
Delta	Yes	Analyze overall handling and performance with added weight	Measure cruise Velocity, turning performance

Table 7.2.1 Objectives for first test flight

Further Objectives:
Cruise speed and turning speed
TOFL requirements
Alternate prop pitches
Alternate battery configurations
Valve system
Verification of propulsion system with in flight recording system
Servo loading
Power off landing

Table 7.2.2 Proposed objectives for subsequent test flights

Conclusion:

The 2003/2004 USC AeroDesign Team made a conscious effort to document the design process for this year's plane. Emphasis was placed on reducing RAC and optimizing the propulsion system. Great strides were made on the experimental front as a significant series of component tests were performed. The team is confident in this year's propulsion model due to the correlation between data acquisition and the theoretical values. The water drop element for this year's flight mission opened a new area of study for the team. A basic understanding of fluid dynamics was gained and demonstrated throughout the water tank testing and building. New techniques in design and manufacturing were explored and incorporated with the techniques on which the team has relied in the past. Under the guidance of a great team leader, the labors of the team were efficient, professional and goal oriented. A sincere and dedicated effort was given by both students and advisors throughout the year. Thanks to all those involved in the success of SCquirt.