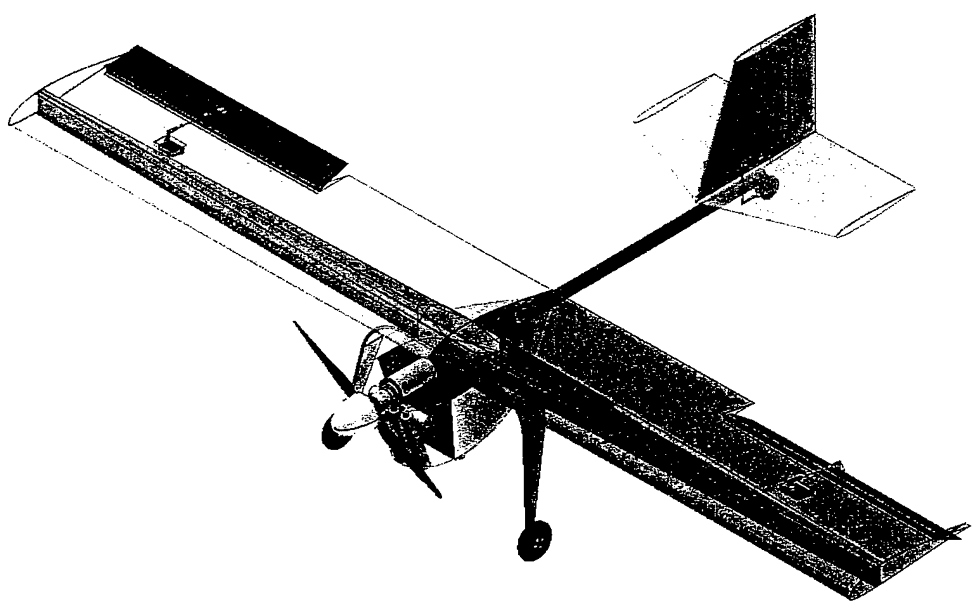
University of California

San Diego

Design Report

Submitted March 10, 2004

Rain of Terror



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1.0 Executive Summary

This report outlines the steps taken by the student members of the American Institute for Aeronautics and Astronautics (AIAA) at the University of California, San Diego, (UCSD) to design and construct an unmanned remote aircraft to compete in the 2003/2004 Design/Build/Fly (DBF) competition. The objective of the competition was to design an aircraft that operates at optimum performance while meeting all competition and mission requirements set forth by the Cessna/ONR DBF Committee. This aircraft must be able to complete two missions. The first mission (fire fight mission) calls for the aircraft to deploy four liter of water and complete a 360 degree turn during the "downwind" stretch of the course twice with a touchdown between laps in order to refill the water tank. During the second mission (ferry mission), the aircraft must take-off, complete four laps, and land. On all laps flown for this mission, the aircraft must complete a 360 degree turn in the direction opposite of the base and final turns on the downwind leg of each lap.

1.1 Conceptual Design

The conceptual design phase began by first analyzing the competition rules and requirements. The rules stipulated that the aircraft must complete two possible missions. Once the mission objectives were understood the team was divided into three technical groups: aerodynamics, structures, and propulsion. Each group produced figures of merit to represent mission objectives for each component of the aircraft that fell within their technical area. Concepts and ideas were studied in terms of feasibility, adherence to contest and mission requirements, conformance with all technical areas, and scoring potential.

The aerodynamic group sought an aircraft configuration that had an optimal Rated Aircraft Cost (RAC), low manufacturing cost, stability, and aerodynamic efficiency. During the conceptual design, various aircraft configurations were considered: the bi-plane, canard, conventional, and flying wing. Moreover, the aerodynamic group was responsible for the general shape and placement of the wing in regards to the fuselage along with researching various tail designs.

Ensuring the structural integrity of the aircraft was a crucial part of the structural conceptual design process. The structures group took into consideration ease of manufacture, RAC, strength, and weight savings when deciding upon the structural components of the aircraft. Various design concepts for the fuselage, wing, and tail structures along with the Water Deployment System (WDS) and landing gears were discussed.

The propulsion group examined motor and battery combinations to find the optimal configuration, which maximizes thrust while minimizing the rated engine power. Four motor/battery configurations were considered in regards to their effects on the RAC, battery and motor weight, thrust, and current draw.

The conceptual design phase produced a primary aircraft configuration consisting of a conventional monoplane with a rectangular high-wing and T-tail with a WDS that released the payload from the bottom of the fuselage.

In search of the most efficient method by which to deploy the water payload two alternative designs were seriously considered: the first involved a hose that could be rolled on a spool internal to the fuselage, during payload deployment the hose would be released and fall, and the second design involved a using a venture to speed the flow going over the water exit in order to create low pressure over the water exit holes, increasing the flow rate.

1.2 Preliminary Design

With a complete conceptual aircraft configuration, the preliminary design phase determined the design parameters, and trade studies were done to optimize the parameters.

The aerodynamic group performed trade studies to determine the wingspan, chord length, airfoil, and tail area that would optimize the design. Three wingspan configurations were considered along with various chord lengths and tail dimensions. Moreover, the aerodynamic group discussed and researched numerous airfoils before coming to a decision.

The structural preliminary design included determining the wing, tail, and fuselage structures, along with the motor mount and landing gear configurations. The aircraft's weight, applied loads and location of the center of gravity affected the overall structural design and fuselage layout. The structural group built a prototype of the selected WDS in order to determine the minimum amount of force required to propel the payload from the bottom of the fuselage.

From the information gained by the aerodynamic and structural groups, propulsion requirements were determined by utilizing the stipulated maximum take off distance and aircraft weight. The minimum required thrust to complete the missions was calculated, and the optimum propulsion system configuration was found.

The preliminary design determined the sizing of the wing and tail geometry, primary structure dimensions, and propulsion configuration. The wingspan and chord length were calculated to be eight feet and 12 inches, respectively. The S4083 airfoil was selected for the wing, while the NACA0009 was chosen for the tail. The most efficient shape for the spar was found to take the shape of an I-beam with a chord-wise linear taper. The fuselage was shaped after a NACA0035 airfoil and constructed in three parts/modules, which provides adequate aerodynamic streamlines that reduced drag and gave ample room for the payload. The aircraft's takeoff weight was estimated to be approximately 19.8 pounds and the center of gravity was determined to be 2.5 inches behind the leading edge of the wing. An 18 inch diameter propeller with a 16 inch pitch powered by the Graupner 3300-7 motor and 18 Sanyo 1300 mAh cells compose the propulsion system.

1.3 Detail Design

With the information gained from the preliminary design phase, a foundation was provided for finalizing component and system architecture selection.

The aerodynamics group analyzed the mission. The total theoretical flight scores and RAC costs were calculated determining the desired flight mode and predicted possible scoring potential. Take off distance and aircraft stability were also calculated for optimization.

The structural group finalized the component integration and architecture of the structural design. Local strength requirements were established through finite element analysis to be implemented during manufacturing and incorporated into the engineering drawings. The boom mounting and wing joining details were determined and incorporated into the final design.

The propulsion system was finalized and static tests were performed in order to ensure optimal performance. Wire routing and servo placement for control surface movement was determined and updated into the drawings.

The final configuration of the airplane was a high-wing, conventional t-tail with a pulling motor configuration. The high-wing configuration required no dihedral for stability. The rectangular wing geometry included a span of 8 feet, a chord of 10 inches for an aspect ratio of 9.6. The main wing used the S4083 airfoil and the T-tail is comprised of the NACA0009 airfoil for both the vertical and horizontal stabilizers. The semi-monocoque fuselage housed the payload that is deployed using a servo controlled cylindrical tapered valve; gravity and an increased pressure differential, caused by the placement of a hole at the top of the water tank, were utilized to expel the water from the aircraft. A carbon fiber central platform absorbs loads and connects the payload, boom and empennage, wings, and motor mount. All structural members are composed of fiberglass, carbon fiber, and plywood. With known materials and component sizes, the CG was calculated to be placed 0.0625 inches in front of the quarter chord. An 18 inch diameter propeller with a 16 inch pitch powered by the Graupner 3300-7 motor and 18 Sanyo 1300 mAh cells compose the propulsion system.

The results of the three design phases produced an aircraft that completes the required mission with the highest scoring potential.

2.0 Management Summary

2.1 Architecture of Design Team

The Rain of Terror (ROT) Team consists of nine undergraduate and one graduate UCSD students studying aerospace engineering. Before work commenced, three technical groups were formed: aerodynamics, structures, and propulsion (Figure 2.1). A project manager supervised each technical group, delegating specific tasks in the areas of interest to each of their group members. In so doing, the project managers were responsible for directing and managing their groups' progress in their specialized areas. Skills and contributions of team members are listed in Figure 2.2. A project manager monitored the performance and advancement of the overall team; making sure each group had all the necessary information to complete their assigned tasks. Together the project manager and the lead engineering students kept the team on task and assured the efficient progress and development of the project.

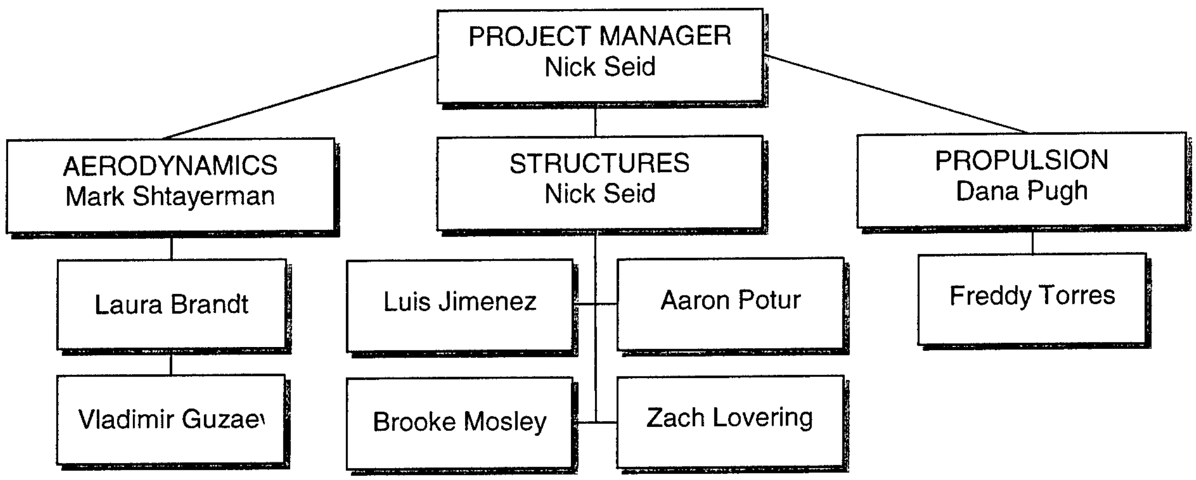


Figure 2.1 Rain of Terror Design Personnel

The consolidation of each groups' efforts into one solid finalized design took a great deal of collaboration and compromise. Since the design concepts of one group places restraints on the other groups, weekly meetings were set up to discuss the integration of the different modules. During these weekly meetings, compatibility issues were resolved, new ideas were generated, and updated assignments were given. This management style allowed for a reliable, secure, and steadfast design process, while considering all aspects of aerodynamics, structures, and propulsion.

Team Members	Major	Year	Auto Desk Inventor	AutoCAD	MSC Patron	Technical Writing	Machining	Fabrication
Laura Brandt	AE	SR	X	X	X	X		X
Vladimir Guzaev	AE	JR		X				X
Luis Jimenez	AE	SR	X	X	X			X
Zach Lovering	AE	FR	X	X				X
Brooke Mosley	AE	JR	X	X				X
Aaron Potur	AE	SR	X	X	X			X
Dana Pugh	AE	SR	X	X	X	X		X
Mark Schlocker	ME	GR	X	X	X			X
Nic Seid	AE	SR	X	X	X	X	X	X
Mark Shtayerman	AE	GR	X	X	X	X	X	X
Freddy Torres	AE	SO		X	X	X	X	X

Table 2.1 Rain of Terror Members' Skills and Contributions

2.2 Aerodynamics

The aerodynamic group was responsible for the shape of the aircraft. The group's primary focus was the analysis of forces acting on the body of the aircraft. These forces were a result of the relative motion between the body and the air (relative wind). In order to accomplish this goal, the aerodynamic group generated a static stability model, based on initial weight estimations, to determine the aircraft lift requirements. Figures of merit and competition regulations were taken into account while performing lift surface sizing. Airfoil selection along with wing, tail, and control surface sizing were conducted with the information gathered from this model. Control surface deflection and servo sizing were also studied using this model to determine control stability and maneuverability. Further aircraft configurations were examined to optimize take off distance, lift-to-drag (L/D) ratio, cruise, and velocity.

During weekly meetings, the structures and propulsion groups conveyed their estimates concerning aircraft weight, sizing, and power requirements, giving rise to the production of updated lift requirements and the refinement of aerodynamic components.

2.3 Structures

The structural group was accountable for determining the structural requirements to support the loads applied on the aircraft; these requirements consisted of designing the component configuration of the aircraft, selection of the materials, and the manufacturing techniques needed to produce the aircraft. Various design considerations for the Water Deployment System (WDS), landing gear, and wing joiner were evaluated in regards to figures of merit and competition regulations in order to obtain an optimum design. The group's analysis focused heavily on hand calculations and CAD modeling to implement their design ideas. Information was exchanged during the weekly meetings with the aerodynamic and propulsion groups, which aided in the decision of the primary structural components of the fuselage, wing, tail, landing gear, WDS, and other aircraft related components.

2.4 Propulsion

The propulsion group was primarily responsible for providing enough energy and thrust to complete the mission profiles. The group was responsible for analyzing different types of motor, propeller, and battery combinations that would provide the most thrust with an efficient Rated Aircraft Cost (RAC). Figures of Merit (FOM) and competition regulations greatly impacted the propulsion group's criteria. The competition guidelines limited the propulsion group's selection of technical equipment: motors, batteries, and propeller selection. Analytical and experimental means were developed to achieve the best combination of the propulsion systems' three main components in order to create an efficient and stable aircraft. Every change in the aerodynamics or structure of the aircraft called for an adjustment in the propulsion system, which called for collaboration and compromise between the groups during the weekly meetings.

2.5 Task Scheduling

In order to meet the deadline a project milestone chart was created. Following this guideline allowed for a quick and efficient design and manufacturing process. Between mid-September and early October, the team set a tentative schedule of completion dates of each phase of the aircraft development. The chart below (Figure 2.2) depicts the intended and actual dates of completion of eight milestones: design team assembly, conceptual design, preliminary design, detail design, design review, report, aircraft construction, and aircraft testing:

- Assembly of Design Team: The returning 2002-03 DBF members collaborated on how many new members the project would need to compete in this year's competition. During the first week of classes, a meeting was held and a new team began to form. The final team had 8 members.
- Conceptual Design: During the conceptual design phase mission requirements were defined, alternative configuration concepts were discussed, and a final design was constructed.
- Preliminary: With the completion of the conceptual design, an investigation into design parameters was initiated.
- Detail Design: The detail design process was composed of component and system architecture selection.
- Design Review: A presentation of the aircraft's detailed designs, plans for fabrication, and a cost break down to Faculty Advisor.
- Report: The process of the conceptual, preliminary, and detail designed phases were recorded in order to maintain an organized and efficient project progression.

- Construction: The manufacturing process was the most time consuming and lengthy. Therefore, a substantial amount of time was devoted to this phase of the aircraft development.
- Flight Testing: Having completed construction, test flights were undergone to ensure flyability, structural integrity, and stability. A sufficient amount of time was given to the flight test in order to give our pilot enough time to become comfortable with the aircraft and to fix any problems encountered.

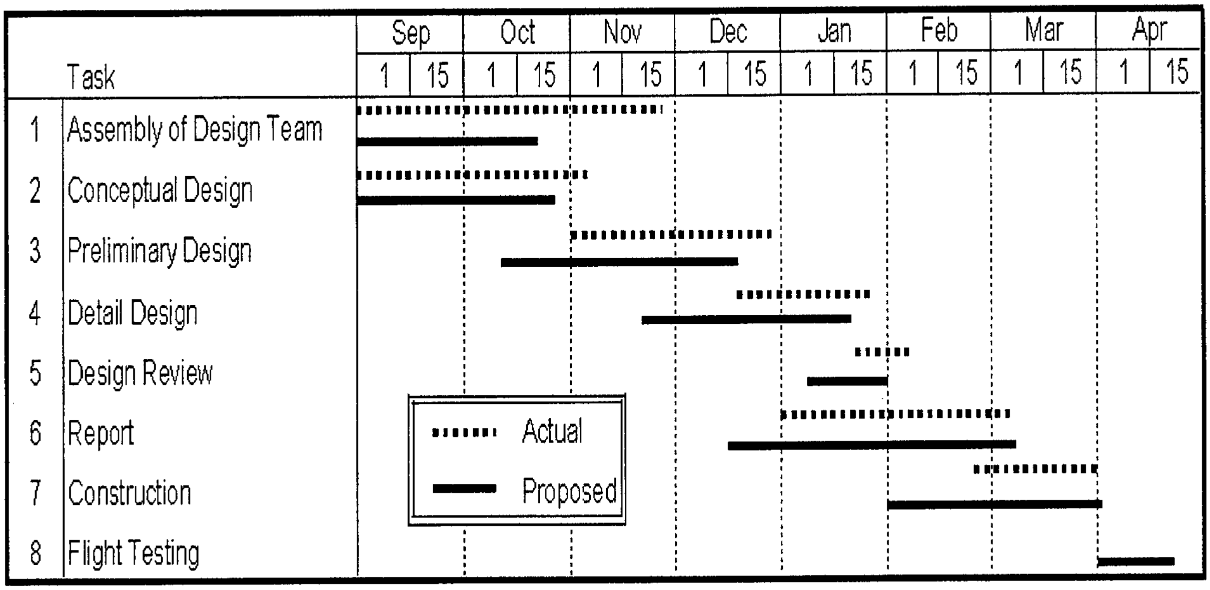


Figure 2.2 Rain of Terror Team Milestone Chart

3.0 Conceptual Design

Four factors dominated the design of the aircraft: competition requirements, aerodynamic effects, structural integrity, and propulsion systems. The competition rules, mission profiles, and Rated Aircraft Cost (RAC) guided the conceptual design phase. The aircraft was considered within three technical areas: aerodynamics, structures, and propulsion. During the conceptual design, the aerodynamic group considered various aircraft configurations, wing shapes, and tail designs while also determining the placement of the wing in regards to the fuselage. The structural group focused on ensuring the structural integrity of the aircraft. They analyzed numerous design possibilities for the fuselage, wing, and tail structures along with the Water Deployment System (WDS) and landing gear. The propulsion group examined four motors and battery combinations to find the optimal propulsion configuration. Within each group, ideas were generated, concepts were collaborated, and compromises were made in order to optimize the design and fulfill mission requirements in order to achieve the best possible score.

3.1 Competition Parameters and Aircraft Configuration

The conceptual design phase commenced with a study of the competition requirements. Within each technical area, a focus was placed on the parameters set by the competition. These constraints affected the aircraft weight and size and in turn affected the RAC.

3.1.1 Aircraft Requirements

The Cessna/ONR Student Design/Build/Fly Competition Committee provided guidelines for the overall size and weight of the aircraft. The contest rules dictated that the unassembled aircraft must fit in to a 2-ft wide by 1-foot high by 4-ft long box. Using solid modeling analysis, maximum dimensions were determined by placing each component into a 3-D model of the box. These dimensions limited the ultimate aircraft size. Dimensions were estimated so that the maximum wing length was four feet, the maximum fuselage height was under one foot and width less than two feet, and the vertical tail size was less than one foot tall. Moreover, the competition required the aircraft to be less than 55 pounds, which limits the choice of component selection.

3.1.2 Mission Profiles

The aircraft must complete two missions. During each lap of the sortie, the aircraft must complete a 360 degree turn on the downwind leg of each lap flown.

A. Fire Fight; Difficulty Factor 2.0. The aircraft takes off with its payload (water), releases the payload at the downwind leg, and return to land; it is then reloaded, takes off, releases its' payload again, and returns for its final landing (Figure 3.1). The maximum allowable water capacity is four liters.

B.

Phase	Description
1	Take-off and climb to altitude
2	Turn 180 degrees (Begin Releasing Payload at end of Turn)
3a	Fly to center of field
3b	Complete a 360 degree turn
3c	Fly to end of field
4	Turn 180 degrees (Stop Releasing Payload at beginning of Turn)
5	Return to Runway and Land
6	Reload
7	Take-off and climb to altitude
8	Turn 180 degrees (Begin Releasing Payload at end of Turn)
9a	Fly to center of field
9b	Complete a 360 degree turn
9c	Fly to end of field
10	Turn 180 degrees (Stop Releasing Payload at beginning of Turn)
11	Return to Runway and Land

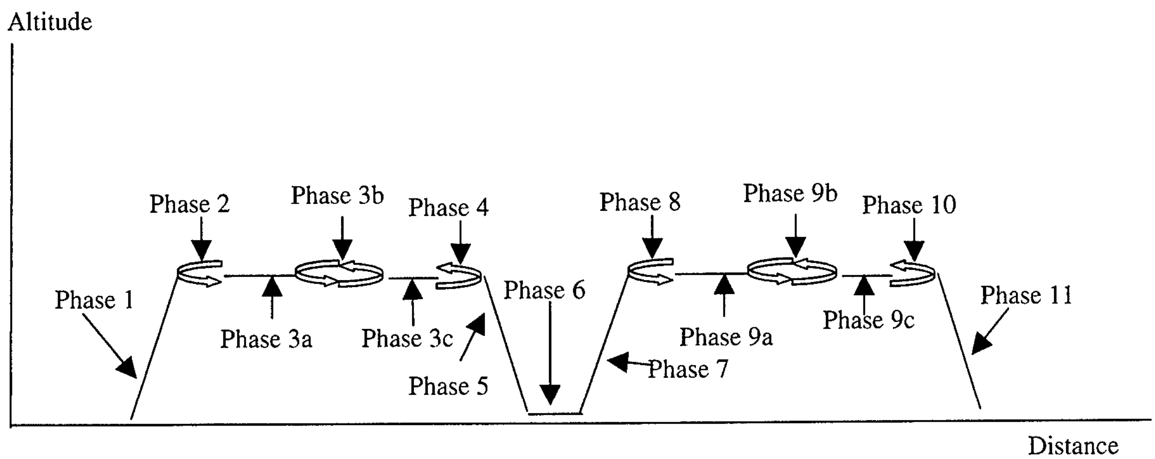


Figure 3.1 Fire Fighter Mission Profile.

C. Ferry; Difficulty Factor 1.0. The aircraft is required to take-off, complete four laps and land. During each lap the aircraft must complete a 360 degree turn in the direction opposite of the base on the downwind transition (Figure 3.2).

Phase	Description

1	Take-off and climb to altitude
2	Complete Four Laps Consisting of: a. Turn 180 degrees at upwind end of field b. Fly to center of field c. Complete a 360 degree turn d. Fly to downwind end of field e. Turn 180 degrees and fly towards upwind end of field
3	Return to Runway and Land

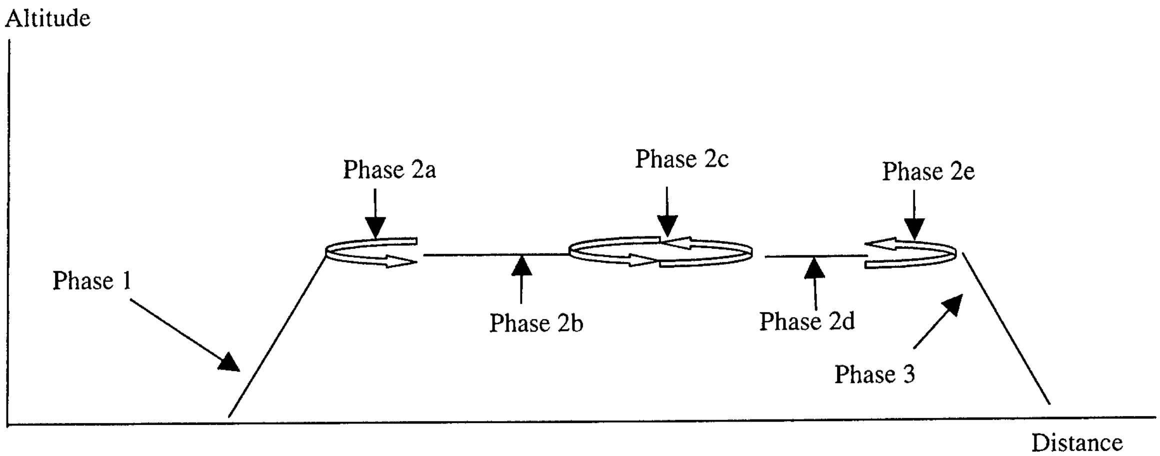


Figure 3.2 Ferry Mission Profile.

An evaluation of several different components was undergone in order to determine the mission that offered the optimum scoring potential. This was done in order to select the mission that the aircraft would be designed around. Because of the different degrees of difficulty, each mission offered a different scoring potential. For instance, mission A required the aircraft to overcome extra weight because of the four liters of water carried in its WDS; while mission B required the aircraft to fly three additional laps. Furthermore, the power requirements for mission A were substantially higher due to the fact that the aircraft must experience two take-offs. Through a complex analysis of comparing the aircraft configurations required by each mission, scoring potential, and RAC, it was found that mission A (fire fight) provided the best opportunity for obtaining points. Therefore, the team decided to design the aircraft around mission A and used mission B as the fall back mission.

3.1.3 Rated Aircraft Cost

The RAC was a mathematical cost function relating all of the aircraft design parameters. The battery and aircraft weight, numbers of wings, size and span, number of servos and control surfaces, fuselage length and tail size all contributed to the RAC. The Rated Aircraft Cost was defined by the following equation:

RAC = ($300*MEW + $1500*REP + $20*MFHR) / 1000

- MEW = Manufactures Empty Weight. (Actual airframe weight without batteries or payload)
- REP = Rated Engine Power. (Defined as (.25*(# Engines)-1) *Total Battery Weight)
- MFHR = Manufacturing Man Hours. This parameter was the sum of the following:

1. Wing span multiplied by chord length: 10 hr/ft^2.
2. Fuselage length x Width x Height: 20 hr/ft^3
3. Each vertical surface with active control: 10 hr/surface
4. Each vertical surface with no active control: 5 hr/surface
5. Horizontal stabilizer, no more than 25% of the greatest span: 10 hr/surface
6. Flight systems: 5 hr/servo or motor controller

In order to achieve a high competition score, the RAC must be minimized. Therefore, the wingspan and chord length, fuselage size, and propulsion systems were limited.

3.2 Aerodynamics

The aerodynamics primary components were assessed after the design parameters of the missions were established and the mission with the maximum score potential was chosen. By evaluating past team experiences and the competition guidelines, suitable Figures of Merit (FOM) were established for initial aerodynamic analysis. For each aircraft component, alternative configuration concepts were placed in FOM matrices. The FOM's considered were: RAC, ease of manufacture, stability, and aerodynamic efficiency.

- RAC effects: Since RAC is the heavily influences the final score, the RAC changes need to be evaluated for each design change.
- Weight: Any weight changes associated with a specific configuration will affect flight characteristics and battery requirements thus increasing RAC score.
- Ease of Manufacture: Manufacturing takes materials, time, and resources, a limiting factor.
- Stability: Any configurations that will render the aircraft unstable will be heavily penalized because of the increased risk of loss of control and possible destruction of the aircraft. Therefore, this FOM was most heavily weighted because any loss of stability may result in failure to meet mission requirements
- Aerodynamic Efficiency: The aerodynamic efficiency accounts for lift, drag and performance of the aircraft, which is responsible for flight time.

The FOMs were compared using a comparative system consisting of +, 0, and – (good, satisfactory, and unsatisfactory, respectively). The RAC, Weight, Ease of Manufacture, and Aerodynamic Efficiency were all given a weight of one, while the stability of the aircraft was given a weight of two because of its overall effect on the mission.

3.2.1 Aircraft Configuration

Various aircraft shapes were researched, including: bi-wing, canard, conventional, and a flying wing. These aircraft shapes were compared in the FOM matrix in Table 3.1. This FOM matrix exposed the advantages and disadvantages of each design concept. The following are the descriptions of each design:

- Bi-plane: A bi-plane generates extra lift, however, it also generally weighs more, includes more work hours, and the servos associated with a second wing greatly increases RAC score.
- Canard: A canard has built-in anti-stall characteristics. However, it requires additional thrust to take off because the main wing cannot reach its maximum angle of attack.
- Conventional: A conventional aircraft has a standard fuselage, tail, and wings, along with well-documented flying capabilities.
- Flying wing: A flying wing has great weight saving advantages; however, the lack of tail surfaces renders the plane unstable. This concept was easily eliminated.

(R) – Rejected (S) – Selected	Figures of Merit (x weighting)					
	Ease of Manu-facture (x1)	RAC effects (x1)	Stability (x2)	Weight savings (x1)	Sum of Ratings	Decision
Bi-Plane	-	-	0	-	-3	R
Canard	-	-	+	-	-1	R
Conventional	0	0	0	0	0	S
Flying Wing	-	+	-	+	-1	R

Table 3.1 Figures of Merit for Wing Configurations

Given the relative ease of manufacture, stability, and proven performance, the conventional configuration was chosen, having an RAC of 8.2. The bi-plane had a weight penalty of 2.5 lbs and RAC of 9.3 due to additional wing and extra battery weight to power the plane. Canard design was rejected because of the slight penalty in RAC (0.2) and increased runway distance due to stability characteristics. The flying wing was abandoned, because it was found a highly risky design because of stability, even though the RAC was 7.0.

3.2.2 Wing Placement

Once the shape of the aircraft was decided, the wing placement relative to the fuselage was taken into consideration. High, mid, and low wings with dihedrals were considered and evaluated using the FOM in Table 3.2.

- High: A high wing has better roll stability because of the placement of center of gravity in relation to the wing. While the high wing does not interfere with the payload deployment mechanisms, it does, however, require the use of large landing gear.
- Mid: A mid wing is very difficult to construct. This is because they require a transfer of loads through the middle of the fuselage, which would require strengthening the structure and hence adding weight. Furthermore, blending the fuselage body to the wing, to reduce drag, will be very difficult because of the nature of the payload and relative size of the fuselage to the wing thickness.

- Low: A low wing will require dihedral for roll stability. Furthermore, the landing gear is smaller, lighter, and easier to attach and manufacture. Therefore, the landing gear is relatively small in comparison to the landing gear required for a high wing and, thus, lower weight and drag. However the wing spar will interfere with the WDS thus increasing the complexity of the aircraft.

(R) – Rejected (S) – Selected	Figures of Merit (x weighting)					
	Ease of Manufacture (x1)	RAC effects (x1)	Stability (x2)	Weight savings (x1)	Sum of Ratings	Decision
High	+	0	0	0	+1	S
Mid	-	-	0	-	-3	R
Low with Dihedral	-	0	0	0	-1	R

Table 3.2 Figures of Merit for Wing Placement

The mid and low wing placements were discarded because of the difficulty to manufacture, resulting in larger RAC scores. Due to the ease of manufacture, stability, weight savings, and the RAC effects, an upper wing configuration seemed ideal. The high wing placement allowed for one central structure that combined all dynamic loads during aircraft operation. Additionally, with the superstructure of the aircraft being above the payload, the water release valve mechanism and servo can be easily installed over the CG of the plane, thus providing stability during the deployment of the payload.

3.2.3 Wing Shape

With high wing aircraft in mind, several designs for the wing shape were taken into consideration: elliptical, rectangular, tapered and swept wings. When comparing these different wing designs an additional FOM was included, aerodynamic efficiency (Table 3.3).

- Elliptical: An elliptical wing has the highest efficiency factor, thereby, having the lowest drag. However, the elliptical wing has a complex shape, which is difficult to manufacture.
- Rectangular: A rectangular wing is the easiest to manufacture and has the lowest RAC score per wing area. However, the efficiency factor is low and, therefore, the drag is high.
- Taper: A taper wing is more efficient than the rectangular wing; however, a larger root cord is required to achieve the same wing area thus increasing the RAC.
- Sweep: A sweep wing provides efficiency benefits at high speeds. However, the aircraft will not experience the benefits of the sweep due to being a low speed. Furthermore, manufacturing a sweep wing has significant difficulties, the wing joiner and spar are very difficult to manufacture.

(R) – Rejected (S) – Selected	Figures of Merit (x weighting)						
	Ease of Manufacture (x1)	RAC effects (x1)	Stability (x2)	Weight savings (x1)	Aerodynamic Efficiency (x1)	Sum of Ratings	Decision
Elliptical	-	-	0	0	+	-1	R
Rectangular	+	0	0	0	0	+1	S
Tapered	0	-	0	0	+	0	R
Swept	-	-	+	0	-	-1	R

Table 3.3 Figures of Merit for Wing Shape

A rectangular wing was chosen because of the ease of manufacture and the RAC score. This was because, an elliptical wing was almost impossible to manufacture without the proper machinery, while a tapered wing was only 4% more efficient than a rectangular wing given an aspect ratio of 10 (Abbot 17). The tapered wing was also disregarded because it had a higher RAC by 0.1. The swept wing was eliminated because of the difficulty in manufacture and lack of efficiency benefits.

3.2.4 Tail Design

In order to stabilize the aircraft in the pitch and yaw directions, designs for a tail section were considered: V tail, H tail, and T tail. A tail was crucial for the stability of the aircraft. Therefore, the stability FOM was considered the most substantial when it came to selection (Table 3.4).

- V tail: This tail provides stability; however, it is difficult to manufacture compared to other designs.
- H tail: This design is ideal for twin fuselage planes, because it provides additional stiffness to the structure; however, extra controlled vertical surface and extra weight must be considered.
- T tail: This is a conventional tail section design that is highly stable and easy to manufacture.

(R) – Rejected (S) – Selected	Figures of Merit (x weighting)					
	Ease of Manufacture (x1)	RAC effects (x1)	Stability (x2)	Weight savings (x1)	Sum of Ratings	Decision
V – Tail	-	+	0	0	0	R
H – Tail	0	-	0	-	-2	R
T – Tail	+	0	0	0	+1	S

Table 3.4 Figures of Merit for Tail Sections

A T-tail was selected, because it proved stability and ease of manufacture. A V-tail design was rejected because of the increased complexity of manufacturing, despite the fact that it had a lower RAC (0.15). The H-tail design was discarded because of the penalty of 0.21 in RAC and considered unnecessary since our configuration has only one fuselage body.

3.3 Structures

Ensuring the structural integrity of the aircraft was a crucial part of the design process. Therefore, the structural conceptual design phase began with setting up FOM to ensure a high-quality aircraft. The FOM for the fuselage, wing, and tail structures along with the payload delivery mechanism and landing gear configuration were ease of manufacture, RAC, strength, and weight savings.

- Ease of Manufacture: This FOM deals with the manufacturing feasibility and difficulty level. This FOM is important because time, materials and resources are limited.
- RAC Effects: The RAC is an important aspect of the overall score.
- Strength: Each component is evaluated in terms of how well it will behave given general loading conditions. The wings and tail are often subjected to harsh handling conditions and must therefore be able to withstand heavy loads at maximum g-loading. Additionally, the wings must survive 2.5 g loading of the plane. Therefore, this FOM was heavily weighted because if the structure fails then the mission may be lost.
- Weight Savings: Any weight change associated with a specific configuration will affect flight characteristics. A low weight aircraft is desirable because weight affects the amount of lift, thrust required, and overall RAC. Therefore, this FOM was also heavily weighted because of its effects on the overall mission.
- Durability: The aircraft must repeatedly be able to withstand heavy loads.

The structural FOMs were similarly compared using a comparative system consisting of +, 0, and – (good, satisfactory, and unsatisfactory, respectively). The Ease of Manufacture, RAC, and Durability were all given a weight of one, while Strength and Weight Savings were given a weight of two because of their overall effect on the mission.

3.3.1 Wing and Tail Structures

The wing and tail structural design required strength, rigidity, and durability. Three different ideas were considered for the internal structure of the wing and tail: white foam core, blue foam core, and balsa.

	Figure of Merit (x weighting)						
(R) - Rejected (S) - Selected	Ease of Manufacture (x1)	RAC Effects (x1)	Strength (x2)	Durability (x1)	Weight Savings (x2)	Sum of Ratings	Decision
White Foam Core	+	+	-	-	+	1	S
Blue Foam Core	+	0	0	-	0	0	R
Balsa Frame	-	-	+	+	-	-1	R

Table 3.5 Figures of Merit for Wing and Tail Structure

White foam core was selected for both the wing and tail sections because of the ease of manufacture, RAC, and weight. Compared to white foam, a balsa wood structure seemed impractical. The manufacturability of a balsa frame structure is very complex and time consuming; the RAC, strength, weight, and durability costs are also high. Furthermore, white foam is less dense and lighter than blue foam, yielding weight savings and a lower RAC value. White foam is less durable and weaker than blue foam. However, for the predicted loads these components undergo, the white foam sufficed.

3.3.2 Wing Spar

The aircrafts spar must be able to withstand two loading scenarios:

- A wingtip test: Assumed to simulate a ±2.5g point load.
- Aerodynamic loads: during flight at the aircrafts maximum loaded weight. Loads assumed to be a maximum of 5g's.

Spars can be very difficult to make properly, therefore manufacturing needs were analyzed carefully. An FOM chart was created in attempt to quantify the options considered. The following are FOM's necessary for an effective analysis (Table 3.6).

- I-Beam: Works well in bending of one axis, and fairly in the other. Is not good for torsional loads.
- Box Beam: Work well in two axis and fairly with torsional loads.
- Cylindrical Tube: Bending is the same in two axis and works well with torsional loads.

(R) - Rejected (S) - Selected	Figure of Merit (x weighting)						
	Ease of Manufacture (x1)	RAC Effects (x1)	Strength (x2)	Durability (x1)	Weight Savings (x2)	Sum of Ratings	Decision
I-Beam spar	+	+	0	0	+	+4	S
Box Beam	0	0	0	0	0	0	R
Cylindrical Tube	-	0	+	+	0	+2	R

Table 3.6 Figures of Merit for Wing and Spar

The I-Beam was selected because of its overall high ranking in Ease of Manufacture, RAC, and Weight Savings. The Box Beam and Cylindrical Tube designs were discarded because they did not rank as well as the I-Beam.

3.3.3 Fuselage Structure

The initial conceptual phase of the fuselage structure was centered on the carrying and delivering of the payload. Therefore, an additional FOM was considered, storage capacity. The aircraft must be able to house the payload, batteries, and flight control servos. Two concepts concerning the fuselage dominated the conceptual design phase: a semi-monocoque fuselage and a thin keel fuselage.

- Semi-monocoque fuselage: This design would house the payload and release it using a cylindrical valve at the bottom of the fuselage.
- Carbon Fiber Fuselage: A thin keel would connect the wings, boom, motor and payload to one structural member.

(R) - Rejected (S) - Selected	Figure of Merit (x weighting)							
	Ease of Manufacture (x1)	RAC Effects (x1)	Strength (x2)	Weight Savings (x2)	Storage Capability (x1)	Durability (x1)	Sum of Ratings	Decision
Semi-Monocoque	+	+	0	+	+	0	+5	S
Thin Keel	0	-	0	-	0	+	-2	R

Table 3.7 Figures of Merit for Fuselage Structure

The semi-monocoque fuselage structure proved to be the better design. The keel design was heavy and difficult to fabricate. In contrast, the conventional semi-monocoque design provided a balance between the strength, weight and manufacturability. Also, this type of fuselage provided storage room for batteries and motor controller while the keel design does not inherently have extra space.

3.3.4 Water Deployment System

The Fire Fight mission required that the aircraft release water from the fuselage on two occasions during its flight via radio-control. The aircraft consists of an internal tank from which the water was released. Three valve mechanisms were considered for deploying the water. A brief description of the three valve mechanisms is outlined below.

- Cylindrical Valve: This concept entailed having the water flow out of the tank through a valve which moves a small cylinder from the path of the water, allowing it to exit the tank.
- Gate Valve: Another idea was to have the valve move a small gate out of the water's path, allowing the water to exit through the space left void by the gate.
- Ball Valve: This idea requires that the flow of water be prevented by placing a small ball in its path of travel. The amount of water allowed to flow out is determined by how much the ball is removed from the water's exit path.

Each Water Deployment System (WDS) valve concept was considered using the structural figures of merit (Table 3.7); however, additional figures of merit were also utilized: reliability, required servos, and testing data.

- Reliability: Reliability was the most important design criterion for the AWD. The water must be released flawlessly every time. Therefore, this FOM was weighed twice as much as the others.
- Servos Required: Additional servos are expensive and add complexity to the design. It was desirable to keep the number of servos to a minimum and keep the design simple in order to keep the weight and RAC low.

Figure of Merit (x weighting)

(R) – Rejected (S) – Selected	Ease of Manufacture (x1)	RAC Effects (x1)	Strength (x2)	Weight Savings (x2)	Reliability (x2)	Servos Required (x1)	Sum of Ratings	Decision
Cylindrical Valve	0	+	0	0	+	+	+4	S
Gate Valve	+	0	0	0	-	+	-1	R
Ball Valve	-	-	-	0	+	-	-3	R

Fig. 3.8 Figures of Merit for Water Deployment System

The cylindrical valve WDS method was selected because of its overall high ranking in RAC, Reliability and Servos Required (requires only one servo). The Gate and Ball Valves were rejected because of there low rankings and the fact that the cylindrical valve ranked so high.

3.3.5 Landing Gear

Three landing gear configurations were studied in the conceptual phase of the design: tricycle, tail wheel and quad.

- Tricycle design: The tricycle design has one strut and tire under the nose of the aircraft and two other struts and tire configurations under each wing.
- Tail Wheel (tail dragger): A tail wheel design is comprised of a strut and tire being placed under each wing and a wheel under the tail boom. This design allows for a wider wheel base and ground directional control.
- Quad: A quad configuration is composed of four tires under the fuselage. This design is unstable at landing and increases weight and drag.

	Figure of Merit (x weighting)						
(R) - Rejected (S) - Selected	Ease of Manufacture (x1)	RAC Effects (x1)	Strength (x2)	Durability (x1)	Weight Savings (x2)	Sum of Ratings	Decision
Tricycle	+	0	0	0	0	+1	R
Tail Wheel	+	0	0	0	+	+3	S
Quad	0	0	0	0	-	-2	R

Table 3.9 Figure of Merit for Landing Gear Configurations

The tail dragger configuration was selected because of the ease of manufacture and weight. The quad landing gear was rejected because of its weight and ease of manufacture. The tricycle landing gear was a possibility; however, it posed problems with the WDS which is addressed in the preliminary design.

3.4 Propulsion

The goal of the propulsion conceptual design phase was to maximize thrust while minimizing rated engine power (REP). The FOM considered were: RAC, battery and motor weight, thrust, and motor current draw.

- RAC: The RAC is a substantial part of the overall score and was therefore, weighted very heavily by the propulsion group.
- Battery Weight: The Battery weight greatly affects the RAC and flight performance and was therefore, weighed heavily.
- Motor Weight: The motor weight is also important to minimize because each addition of a motor directly adds to the manufacture's empty weight (MEW) and REP.
- Thrust: Thrust is important because it is the force required to overcome the drag. Therefore, the thrust was considered a main priority and received a heavy weight
- Current Draw: Motor current draw is imperative to not draw too much power and thus burning up the fuse or the engine thus causing a plane crush.

The structural FOMs were similarly compared using a comparative system consisting of +, 0, and – (good, satisfactory, and unsatisfactory, respectively). The Motor Weight and Current Draw were given a weight of one, while the Battery Weight and Thrust were given a weight of two because of their overall effect on the mission. Because of the large effect, the propulsion system had on the RAC score, the RAC was weighed as three.

Four motor battery configurations were analyzed to optimize performance: 1 motor/1 (1.5 pound) battery pack, 1 motor / 1 (2 pound) battery pack, 2 motors / 1 (2.5 pound) battery pack, and 2 motors / 2 (1 pound) battery packs.

(R) – Rejected (S) – Selection	Figures of Merit (x weight)						
	RAC (x3)	Thrust (x2)	Current Draw (x1)	Motor Weight (x1)	Battery Weight (x2)	Sum of Rating	Decision
1 Motor / 1 (1.5lbs) Battery Pack	+	-	+	+	+	+5	S
1 Motor / 1 (2 lbs) Large Battery Pack	+	0	0	+	0	+4	R
2 Motors / 2 (1lbs) Battery Packs	-	+	+	-	0	-1	R
2 Motors / 1 (2.5lbs) Large Battery Pack	0	0	-	-	0	-2	R

Table 3.10 Propulsion FOM

The setups with two motors received low RAC scores (over 9.5). The RAC is penalized for multiple motors and additional motor weight. Therefore, the systems with more than one motor were discarded.

As seen in Table 3.9, the single motor propulsion systems give favorable RAC values (8.1 and 8.8 respectively for 1.5 pound and 2 pound battery packs), low current draw, low motor weight, and reduced battery weight. From previous experience and competitions, a 1.5 pound battery pack is estimated to be the most efficient and to provide the power required to fulfill the mission. However, in order to formally decide between the two remaining systems, the thrust must be calculated. This calculation was performed in the preliminary and detail sections.

3.5 Final Aircraft Configuration

With the collaboration of concepts from the aerodynamic, structural, and propulsion technical groups, a final aircraft configuration was produced. The final design had a conventional configuration with a rectangular high white foam wing and T-tail. The fuselage structure was a carbon fiber configuration and a tail dragger configuration was chosen for the landing gear. The propulsion system was comprised of a single motor with a 1.5 pound battery pack.

3.5.1 **Figures of Merit for Final Ranking Charts**

Aerodynamic FOM Chart

(R) – Rejected (S) – Selected	Ease of Manu-facture (x1)	RAC effects (x1)	Stability (x2)	Weight savings (x1)	Aerodynamic Efficiency (x1)	Sum of Ratings	Decision
Bi-Plane	-	-	0	-		-3	R
Canard	-	-	+	-		-1	R
Conventional	0	0	0	0		0	**S**
Flying Wing	-	+	-	+		-1	R
High	+	0	0	0		+1	**S**
Mid	-	-	0	-		-3	R
Low with Dihedral	-	0	0	0		-1	R
Elliptical	-	-	0	0	+	-1	R
Rectangular	+	0	0	0	0	+1	**S**
Tapered	0	-	0	0	+	0	R
Swept	-	-	+	0	-	-1	R

Table 3.11 Aerodynamic Figures of Merit Chart

Structural FOM Chart

I – Rejected (S) – Selected	Ease of Manu-facture (x1)	RAC effects (x1)	Strength (x2)	Weight savings (x1)	Storage Capability (x1)	Durability (x1)	Sum of Ratings	Decision
White Foam Core	+	+	-	+		-	1	**S**
Blue Foam Core	+	0	0	0		-	0	R
Balsa Frame	-	-	+	-		+	-1	R
I-Beam spar	+	+	0	+		0	+4	**S**
Box Beam	0	0	0	0		0	0	R
Cylindrical Tube	-	0	+	0		+	+2	R
Semi-Monocoque	+	+	0	+	+	0	+5	**S**
Thin Keel	0	-	0	-	0	+	-2	R
Tricycle	+	0	0	0		0	+1	R
Tail Wheel	+	0	0	+		0	+3	**S**
Quad	0	0	0	-		0	-2	R

Table 3.12 Structural Figures of Merit Chart

Propulsion FOM Chart

(R) – Rejected (S) – Selection	RAC (x3)	Thrust (x2)	Current Draw (x1)	Motor Weight (x1)	Battery Weight (x2)	Sum of Rating	Decision
1 Motor / 1 (1.5lbs) Battery Pack	+	-	+	+	+	+5	**S**
1 Motor / 1 (2 lbs) Large Battery Pack	+	0	0	+	0	+4	R
2 Motors / 2 (1lbs) Battery Packs	-	+	+	-	0	-1	R
2 Motors / 1 (2.5lbs) Large Battery Pack	0	0	-	-	0	-2	R

Table 3.13 Propulsion Figures of Merit Chart

4.0 Preliminary Design

With a completed general aircraft configuration, the preliminary design phase focused on design parameters and trade studies. The aerodynamic group performed trade studies to determine the wingspan, chord length, airfoil, and tail area that would optimize the design. While, the structural group determined the configuration of the wing, tail, and fuselage, along with the motor mount and landing gear. The Water Deployment System (WDS) was designed and components chosen to perform at maximum efficiency. Moreover, the propulsion group calculated the optimum propulsion system configuration by determining the minimum thrust required to complete the missions.

4.1 Aerodynamics

Once the conceptual design phase was completed, the parameters obtained by the generated primary aircraft configuration were used in the preliminary design phase to further narrow and refine the design. Trade studies were conducted to determine the wingspan, chord length, airfoil, and tail area that would optimize the design.

4.1.1 Wingspan

During the conceptual phase, the wingspan was limited by the competition rule that the aircraft must fit into a 4x2x1 foot box. Therefore, each wing section length was limited to maximum of 4 feet. Furthermore, the competition rules stipulated that the maximum take off distance was 150 feet. In order to minimize thrust and still take off within the required distance, a large wingspan was desired. Optimizing software was written in MATLAB to determine the range for the best wingspan as a function of RAC. The program determined that the optimal wing area of 8 ft^2 and a wingspan range of 7 to 9 ft minimized the RAC for the aircraft of 19.8 lbs.

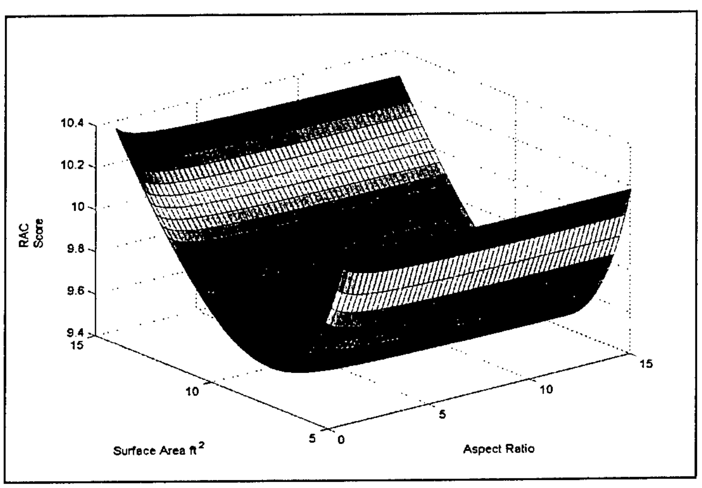


Figure 4.1 RAC Score v Surface Area and Aspect Ratio

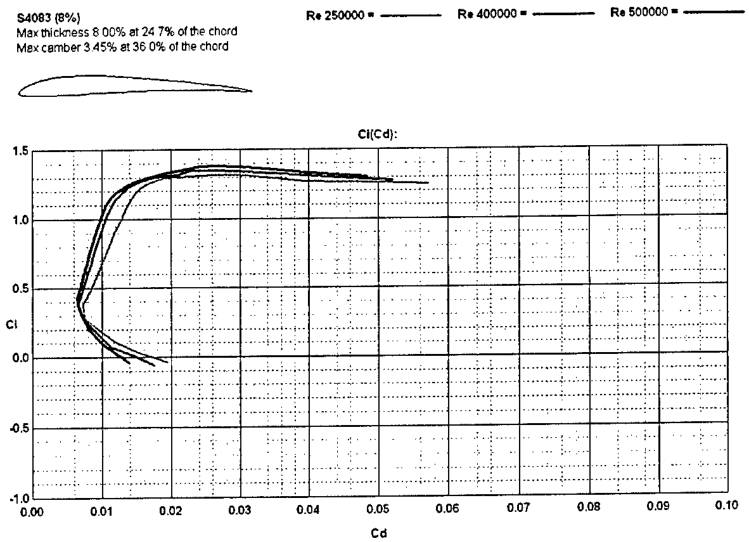
4.1.2 Wing Chord Length

Provided a fixed wing area of 8 ft^2 the most aerodynamically efficient wing would have a very large aspect ratio (Figure 4.1). However, the span is limited to the box dimension of 4 foot sections. To simplify the manufacturability of the wings, two equal sections were considered thus limiting the span of the aircraft to 8ft with the chord length of 12 inches.

4.1.3 Airfoil Selection

In order to optimize the takeoff distance and maximize the lift to drag ratio of the wing, numerous airfoils were considered. Profili software was used as a tool to narrow the airfoil selection. Using this software, theoretical coefficients of lift and drag for the Reynolds number of 250,000 and 500,000 were calculated for 50 airfoils. In order to verify computer generated theoretical calculations, a comparison was done with actual wind tunnel results from research performed by Michel Selig. Calculations were performed to determine the total take off distance and the lift to drag ratio. The selection of the airfoil was a process of reducing the take off distance and optimizing the lift to drag ratio, thereby minimizing required thrust.

Remaining airfoils were of the class of medium high lift with fairly low drag with lift coefficient of 1.3 to 1.6 and of parasitic drag of 0.005 to 0.010. An airfoil was sought that provided a high lift to drag ratio (for the whole plane), a low stall speed and a fairly high cruise speed to be able to minimize power at takeoff and achieve the fastest time. Airfoils such as SD7062, GOE-769, E580, and S4083 were considered. S4083 was determined the best compromise between the high cruise speed (52.3) and low stall speed (39.6 ft/s) with a high lift to drag ratio of 15.3. Other airfoils provided lower stall speed but also lower lift to drag ratios meaning that the overall drag would be higher.



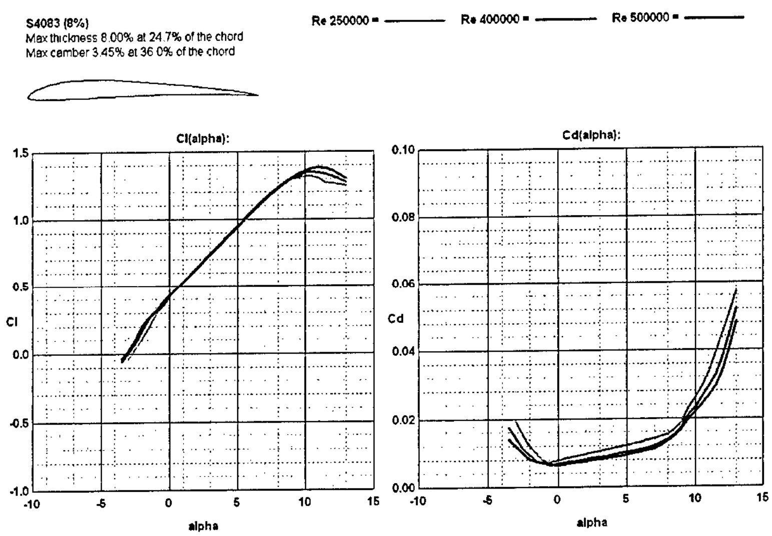


Figure 4.2 Drag Polar, CL v alpha, and Cd v alpha Curves for S4083 Airfoil

4.1.4 Tail Design

Once the wing dimensions were calculated and the airfoil was selected, calculations were performed to obtain the desired dimensions for the tail. This factor limited the height of the vertical stabilizer to 10 inches and the horizontal stabilizer to 22 inches. This coincides with the competition rules, which stipulates that the horizontal stabilizer cannot exceed a quarter of the wingspan.

In order to optimize the tail effectiveness, a boom 42 inches long was used. Numerous symmetric airfoil designs were made available. However, the NACA 0009 airfoil was immediately selected for the horizontal stabilizer because of its reduced drag and it produces zero lift at cruise (Figure 4.3). This selection improved the theoretical stability characteristics.

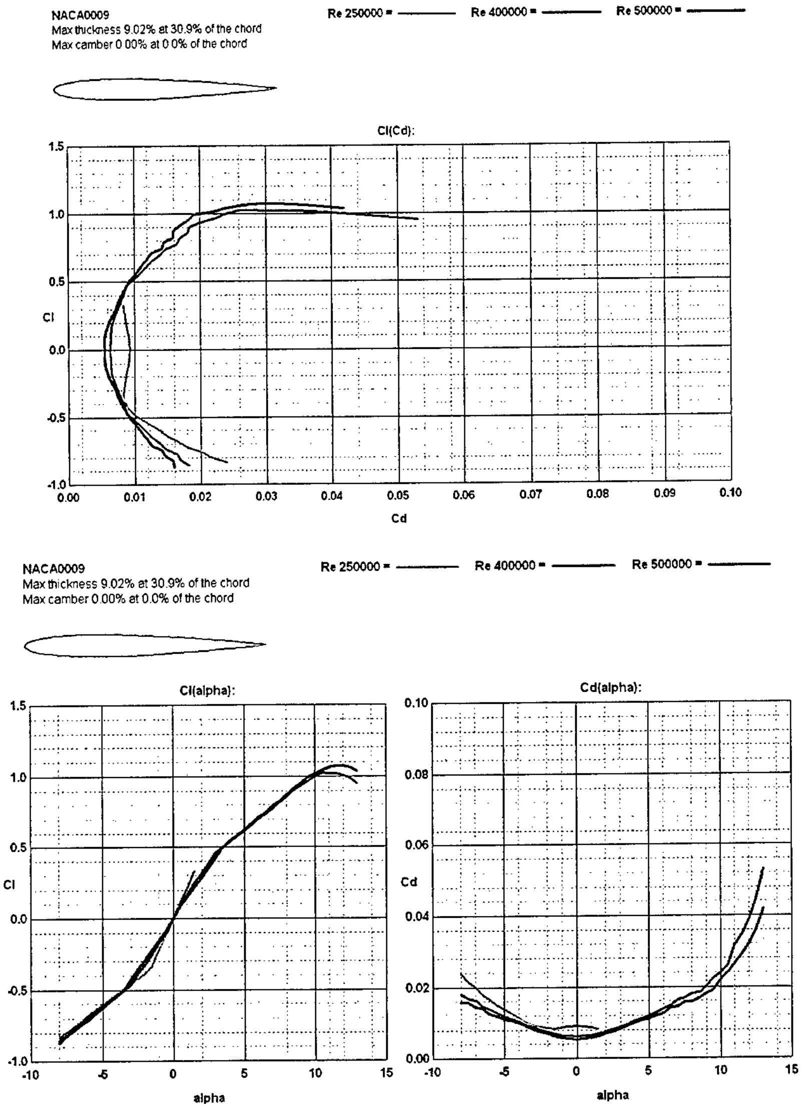


Figure 4.3 Drag Polar, CL v alpha, and Cd v alpha Curves for NACA0009 Airfoil

The stability of the aircraft was further enhanced, by optimizing the tail section area. The area was calculated with the following formulas. The volume coefficients of 0.55 and 0.031 were used for the horizontal and vertical stabilizers, respectively.

$$Area_{HorizontalStabilizer} = (Coefficient_{HorizontalVolume})^*(MAC_{MainWing})^*(Area_{MainWing})/(Arm_{HorizontalStabilizer})$$

$$Area_{VerticalStabilizer} = (Coefficient_{VerticalVolume})^*(Span_{MainWing})^*(Area_{MainWing})/(Arm_{VerticalStabilizer})$$

Using these empirical equations, the horizontal stabilizer and vertical stabilizer areas were calculated to be 219 inches squared and 99.5 inches squared, respectively.

The mean aerodynamic chord (MAC) for each tail section was determined using the vertical stabilizer's height, horizontal stabilizer's span, and the respective surface areas. Using a taper ratio of 0.85 (Raymer 85), an 8 inch tip chord and 11.9 inch root chord were calculated for both the horizontal and vertical stabilizers.

4.1.5 Fuselage Shape

Various shapes were considered for the fuselage. It needed to be aerodynamically efficient while being able to hold the payload. Therefore, the fuselage is shaped after a NACA0035 airfoil and constructed in three parts/modules. This airfoil is symmetric and has a thickness to chord ratio of 35 percent of the fuselage, which provides adequate aerodynamic streamlines that reduce drag and gives ample room for the payload. The Drag Polar, CL v alpha, and Cd v alpha are shown in Figure 4.4.

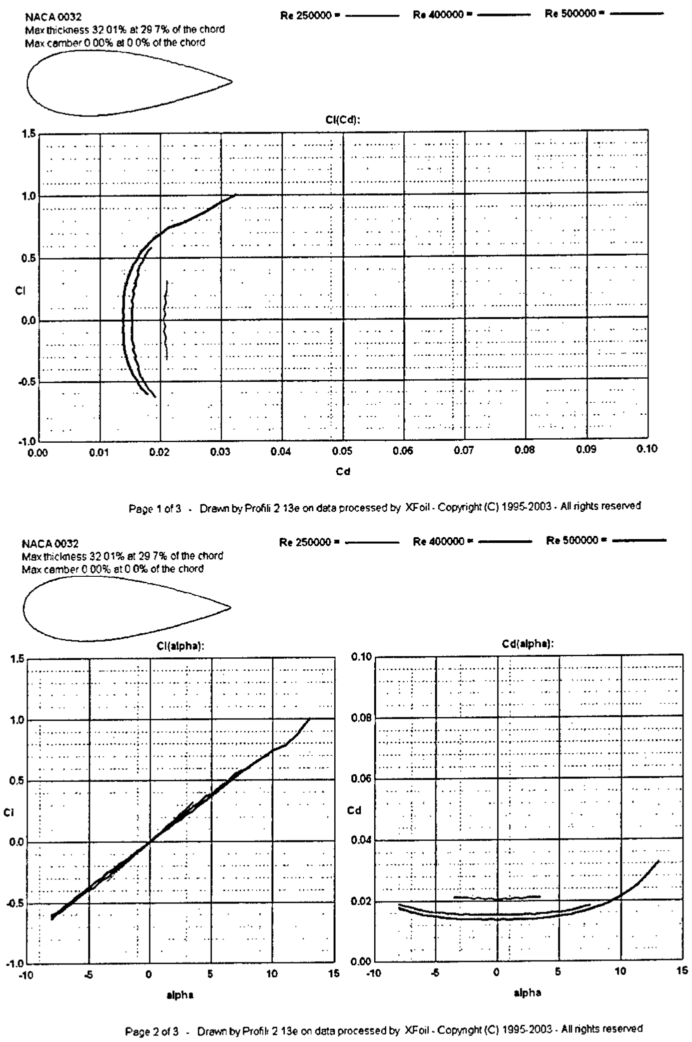


Figure 4.4 Drag Polar, CL v alpha, and Cd v alpha Curves for NACA0035 Airfoil

4.2 Structures

During the conceptual design phase the mission parameters and FOM were used to create a basic structure of the aircraft. In the preliminary design more focus was placed on the aircraft's structural requirements:

- Feasibility of primary structure locations.
- Load paths of primary structural components.
- Materials selection for said components.

- Sizing of components with a safety factor of 1.25 for yield and 1.5 for ultimate.
- Practicality of structural weight to lift available.
- Survivability of structure in respect to impact loads from hard landings.
- Manufacturability, maintenance and life cycle cost were estimated.
- Fit into the shipping container described in the mission profile.

Trade studies were conducted on these design requirements. These studies provided the necessary tools to investigate structural alternatives in location, applied loads, material selection, sizing, and weight of each component. They also played a part in deciding on the overall assembly of the aircraft. Estimates for the aircraft weight and center of gravity were determined. The information gained from these studies produced an aircraft structure that was sound and efficient.

4.2.1 Factor of Safety

The preliminary analysis used factors of safety to produce initial sizing. The analysis conducted to determine the aircraft's structural components optimal thicknesses were subject to factors of safety of 1.25 for yield and 1.5 for ultimate. If sizes were deemed to thin for practical purpose a margin of safety was added in the detailed design.

4.2.2 Wing Structure

The wing structure consists of two main portions, the spar and skin. Since the spar mainly resists bending and skin torsion, these two structural elements were individually analyzed.

The two loading cases determined the tapered shape of the spar: distributed aerodynamic loading and a point load applied at the tip during the wing test. The distributed load was analyzed using Shrenks approximation. These two cases generate different bending moments, as seen in Figure 4.5.

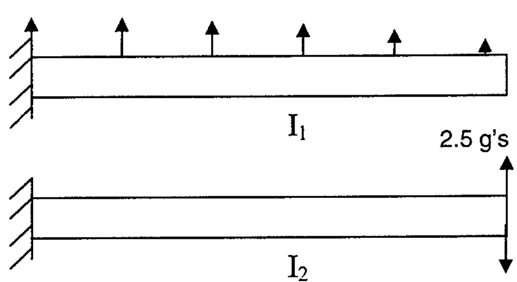


Figure 4.5 Distributed and point loads.

The incorporation of these resulted in the tapered pattern shown in Figure 4.6. The spanwise taper of the spar gives a better strength to weight ratio for the given loading cases. During the preliminary analysis of the spar structure, the bending moment due to the point load of the landing gear was considered to be insignificant compared to the bending moment of the distributed load; therefore, it is not considered separately.

The preliminary thickness needed for the composite spar was calculated by modeling the minimum thickness needed for the spar while retaining a factor of safety of 1.5 is 0.0045 inches for the top and bottom layer.

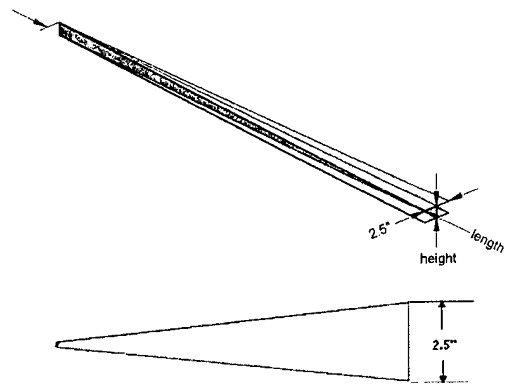


Figure 4.6 Illustration of spanwise profile of the spar.

The primary requirement of the skin is to provide the smooth surface designed by the aerodynamics group and also serve as a platform for the control surfaces. The skin also needs to be as light as possible for overall aircraft performance, while still transferring the torsional loads to the wing joiner. A fiberglass resin matrix was chosen to build the skin. This would give a hard and smooth aerodynamic surface, has an acceptable strength to weight ratio, and is easy to manufacture to the shape of the airfoil.

The wing skin was analyzed using a MATLAB script. The worst case aerodynamic loading was assumed in sizing the thickness of the skin. The wing skin was considered a closed single cell web in torsion. The shape of the actual airfoil was used in the analysis. The preliminary thickness required with a factor of safety of 2.0 is 0.025 inches. The preliminary thickness being very small requires that a larger thickness be used so that the skin can be manufactured accurately. This will increase the factor of safety and should not have a great effect on the final weight of the wing.

4.2.3 Wing Joiner/Central Platform

The central platform is the main load bearing component of the structure, acting as the wing joiner, supports the payload, and attaches the motor mount to the structure. The platform is found at the root of the wing where it experiences both torsion and bending loads; therefore, both types of loading were analyzed.

Once the wing span and configuration was determined, it became necessary that the central platform be used as the method by which the wings attach to the fuselage. The central platform is shaped (Figure 4.7) so that the wings can simply be attached to a hard point in the wing. Its length reaches into the span and four bolts fasten each wing to the fuselage.

Since the central platform is the main structural link between the wings, payload bay, and motor, strength is a major issue. The platform was first thought to be constructed of carbon-foam-carbon sandwich structure. However if the payload hold began to leak in flight, the foam may become wet and the structure fail. Therefore, a single carbon laminate was used for precaution. The critical thickness of the platform was determined by modeling the platform as a flat plate. By applying point loads when considering the wings, landing gear, and tail boom a laminate thickness of 0.096 inches was found. Loads that were not considered in the calculation were the torque caused by the motor and empennage. These will be considered in a Finite Element Model (FEA) done during the Detailed Design.

The spar which is made of carbon fiber will be adhered to a hard-point to ensure complete transfer of loads to the central platform.

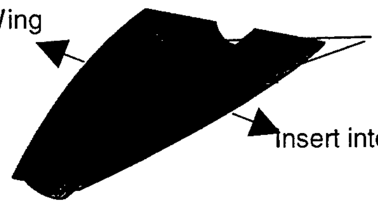


Figure 4.7 Central Platform

4.2.4 Tail Configuration

The primary purpose of the tail is to act as a platform for control surfaces and enhance stability. Three load paths were assumed critical in the design: boom, vertical and horizontal stabilizer. As shown in Figure 4.8, loads were transferred from the stabilizers causing bending and torsion, which were the main forces acting on the boom. The loads applied to these stabilizers were similar to the main wing loads and, therefore, were modeled in a similar fashion.

Two designs were considered for the boom, a truss and a beam. Due to manufacturing limitations, the truss was deemed too complicated and unreliable. Therefore, a beam design was chosen. Several beam cross-sections were considered: square and solid, square and hollow, circular and solid, and circular and hollow. These cross sections were analyzed using the figures of merit: ease of manufacture, plane bending to weight ratio, out of plane bending to weight ratio, torsion to weight ratio, and durability (Table 4.1). The FOMs were compared using a comparative system consisting of +,0, and – (good, satisfactory, and unsatisfactory, respectively). The ease of manufacture and durability were given a weight of one, while the "plane bending to weight ratio", "out of plane bending to weight ratio", and "torsion to weight ratio" were given a weight of two because of their overall effect on the aircrafts performance.

	Figure of Merit (x weighting)						
(R) – Rejected (S) – Selected	Ease of Manufacture (x1)	Plane Bending to Weight Ratio (x2)	Out of Plane Bending to Weight Ratio (x2)	Torsion to Weight ratio (x2)	Durability (x1)	Sum of Ratings	Decision
Square Solid	+	+	-	-	+	0	R
Square Hollow	+	+	-	0	0	+1	R
Circular Solid	-	-	+	+	+	+2	R
Circular Hollow	-	0	+	+	0	+3	S

Table 4.1 FOM of Cross Section Selection for Tail Beam.

After the analysis was complete, the circular hollow tube design was considered the most efficient because of its weight and ability to handle the predicted loads. With forces acting in different directions, a circular tube is desirable because it has uniform behavior. The hollow configuration was chosen instead of the solid configuration in order to lighten the aircraft structure and move the inertial mass away from the neutral axis. The radius arm of the cross section is responsible for the ability of the hollow tube to take torsion loads without weight penalty associated with the solid tube. This produced greater bending and torsion stiffness in reference to a minimized cross sectional area.

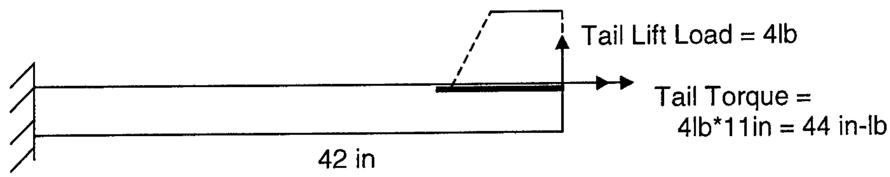


Figure 4.8 Tail Boom Load Model.

Due to availability of material, a boom with an outside diameter of 0.5 inches was selected, while the inner diameter was found through calculations. Using Figure 4.8, two trade studies were performed in order to determine the inner diameter of the boom. The first trade study investigated bending. The boom was modeled as a cantilever beam with a 4 pound point load 42 inches from the quarter chord. By analyzing the stresses through the cross section of the loaded member, the minimum inner cross sectional diameter was found to be 0.4966 inches. The second trade study focused on torsion loads. The torsion of the boom was modeled as a 4 pound side load located at the top of the rudder 11 inches away from the tail boom. By studying the torque applied at the tail, the resulting minimum required inner diameter was found to be 0.4985 inches. Since the inner diameter found in the first case study is smaller than that found in the second, it is deemed the critical inner diameter.

Trade studies conducted on model aircraft tail structures led to the selection of a single vertical and bottom mounted horizontal stabilizer. The studies showed that most tail structures do not contain a spar and are usually comprised of a foam core with fiberglass skin. However, performance aircraft increased bending stiffness by adding a small amount of carbon fiber tow to the top and bottom of the quarter chord. Due to the nature of the competition, the performance aircraft tail structure was selected. The skin thicknesses of the stabilizers were determined by the same modeling as the main wing. The required thickness was calculated to be 0.015 inches of +/- 45 degree fiber glass.

4.2.5 Fuselage Structure

The fuselage is considered the primary load bearing structure, housing the payload, batteries, central platform, tail boom, and motor mount. The main component that is loaded is the central platform, which is used as a wing joiner, payload support, and motor mount attachment. The internal structure of the fuselage is shown in Figure 4.9.

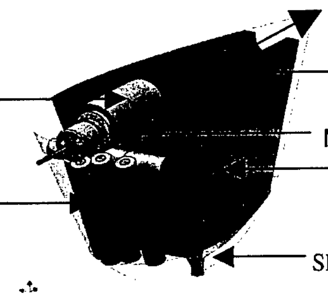


Figure 4.9 Internal Structure of Fuselage

All of the loads from the motor, boom, wings, landing gear servo, and payload will be transferred through the single platform.

The skin of the fuselage was initially to be made as a sandwich structure to save weight, however such thin sheets of carbon can be porous so a single carbon laminate was decided upon. The only section of the payload volume that may deform critically under the weight of water is the face that is perpendicular to the wingspan. Therefore, this face will be modeled as a flat plat and analyzed during the Detailed Design.

4.2.6 Aircraft Weight and Center of Gravity

After the fuselage platform components were initially sized, Autodesk Inventor was used to create a solid model. By inputting the material properties of each element into this computer-generated model, the mass and its specific center of gravity for each component were determined.

The estimated empty weight of the aircraft was 9.5 pounds. This value does not include the 8.8 pounds of water payload and 1.5 pounds of battery. The fully loaded aircraft weight was estimated to be 19.8 pounds and the center of gravity was found to be approximately 0.10 inches in front of the quarter cord.

Much attention was focused on the change in center of gravity during payload release. For this reason the center of gravity for the payload was placed under the center of gravity for the empty weighted aircraft. If the payload's center of gravity were to shift before or aft the empty weighted aircraft, the pilot would need to trim the control surfaces which take energy from the system reducing its efficiency.

4.2.7 Water Deployment System (WDS)

During the conceptual design phase it was decided that a circular valve would be used. However, during the preliminary design phase testing was done in order to verify this decision. Several valve designs (gate valve, ball valve, and cylindrical valve) were considered for payload delivery. Some were tested to evaluate their effectiveness and reliability. Because our final score depends partially on the time that it takes to deliver the payload, the chosen valve must open quickly. Flight time can also be penalized heavily if any water is released after a certain point in the mission; therefore, this valve must also close quickly and effectively.

Through testing (See Research and Testing Plan), a cylindrical valve was determined to be the most effective. It was concluded that the valve would have a tapered cylinder with a matching tapered receptacle (Figure 4.10). A spring was added to draw the two tapered pieces closer together, providing a good seal.

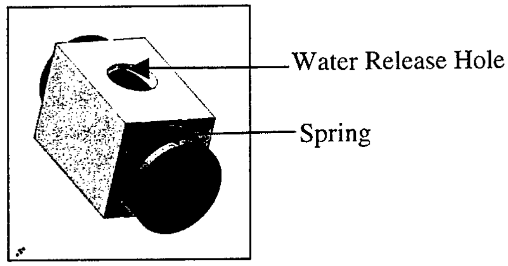


Figure 4.10 Cylindrical Valve

When water begins to flow out of the payload chamber a vacuum will be created unless the volume lost is replaced with another fluid. To offset this affect, a hatch will be opened synchronous with the tapered cylinder valve to allow air to fill the void. This hatch should be aligned with the onset of air to assist the outflow of water by creating a small amount of dynamic pressure.

The mission profile requires that the aircraft's payload be refilled during the mission. The time that it takes to fill the water tank is part of the flight score. To speedily refill the water tank two holes atop the fuselage will accept a fitting made from a two liter soda bottle cap. The same dilemma of a vacuum being created exists for emptying the soda bottles as does the water tank. While rules specify that the two liter soda bottle can not be modified, the bottle cap may. A tube with a single one way valve should reach the height of the bottle in its refill position (Figure 4.11). This should relieve the negative pressure caused by the outflow of water. Water pumps are being considered to assist in the refill process.

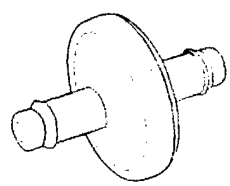


Figure 4.11 Bottle Cap Vacuum Release Valve

4.2.8 Motor Mount

A trade study was preformed taking into account the thrust and landing loads that were transferred to the primary structure. It was found that these loads should be transferred through carbon mounting plate in order to assure the durability of the aircraft. The motor is attached to a pre-impregnated Kevlar motor mount and the motor mount is then attached to the central platform as seen in Figure 4.12. The material was chosen for its thermodynamic properties and resistance to shear from load bearing holes in its structure. It is also believed that the mount is an aircraft saving device due to its ability to fail at high impact loads. This quality would theoretically save the motor in the event of a low speed nose impact.

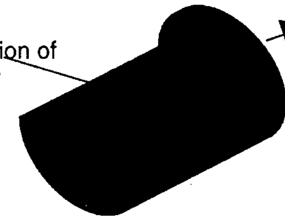


Figure 4.12 Motor Mount

4.2.9 Landing Gear

During conceptual design a tail dragging landing assembly was selected. Selecting this type of landing gear configuration is critical for refilling of the water tank. While the payload is being refilled, air in the payload bay must escape. Using a tail dragger allows for the nose of the aircraft to be pitched upwards, allowing of the air to escape. If there is any air trapped in the payload bay, the total desired volume of water will not fit.

The main landing gear was designed to mount between the wing and the central platform (Figure 4.13). This design was chosen for its simplicity and for structural integrity. Mounting the landing gear on the wings was also considered. However, aircraft with wing mounted landing gear often have problems with structural integrity during landing. When experiencing a hard landing which exceeds the designed load factor, the wings may break due to impact and the aircraft may be unable to complete the mission.

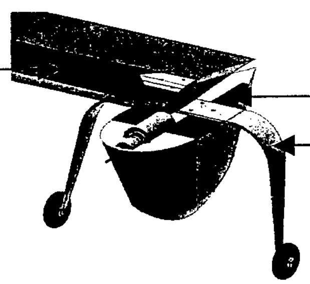


Figure 4.13 Main Landing Gear Placements

When analyzing the landing gear, a five g-loading profile was assumed because the wind gusts at the location of competition are known to rise to 30 miles per hour. The landing gear was modeled as a flat plate and showed that the gear be made of 10 plies of carbon fiber at zero degrees. In the Detailed design this number is verified with a finite element model.

4.3 Propulsion

During the preliminary design, the minimum required thrust to complete the missions was calculated and the optimum propulsion system configuration was found. Since, takeoff requires the most power throughout the mission; it was used to establish the minimum required thrust. The required thrust of 5.7 pounds was calculated using the aircraft weight and wing geometry combined with the maximum takeoff distance. It was imperative to find a propulsion system configuration that would provide a minimum of 5.7 pounds of thrust and draw less than 40 amps of current, which was stipulated by competition guidelines. A gear box is used to lower current draw and allow for a larger propeller to achieve the required thrust.

In the conceptual design, the propulsion system was determined to have a single motor with an approximate 1.5 pound battery pack. According to competition rules a Graupner or Astro motor must be used. From past experience and success, a motor from the Graupner 3300 series was chosen. These motors are known to be reliable and demonstrate high performance.

The Graupner 3300 series offer motors with different numbers of windings, which allow peak efficiencies at different voltages. MotorCalc software was used to determine the best combination of propeller geometry, battery cells, and engine windings. Each configuration's pitch speed, flight time, and current were calculated. Some of the chosen configurations are found in Table 4.2.

Motor and Battery Configuration							
# of Wind-ing	Gear Ratio	# of cells	Pitch Speed (mph)	Flight Time	Propell er Geo-metry	Thrust (oz)	Current (Amps)
5	2.5:1	16	71.9	1:53	16x16	92	41.2
5	2.5:1	16	52.1	1:43	18x12	127	45.6
6	2.5:1	16	57.9	2:05	18x16	95	37.4
6	2.5:1	18	65.8	2:02	17x16	98	38.2
6	2.5:1	18	62.0	1:52	18x16	109	42.2
7	2:1	16	57.9	2:37	17x16	75.6	29.9
7	2:1	18	57.3	2:10	18x16	93	36.1

Table 4.2 Graupner Motor 3300 series

The optimum propulsion system for this competition has high static thrust, high pitch speed, long flight time, and minimal battery weight while abiding by competition regulations. Gear ratio, propeller diameter and pitch and number of cells are optimized to get the desired values of thrust, flight time and pitch speed. Large diameter propeller provides large static thrust yet low pitch speed. Gearing is used to achieve highest efficiency for the propeller thus increasing flight time. By analyzing the MotorCalc results (Table 4.2), the Graupner 3300-7 motor with a two to one gearbox coupled with 18 Sanyo 1300 mAh cells and 18x16 inch propeller was chosen to achieve the desired thrust of 5.8 pounds resulting in a pitch speed of 57.3 mph. This configuration offered a reasonably high static thrust, high pitch speed, and sufficient flight time while drawing less than 40 Amps; therefore, offering the optimum performance.

4.4 Final Aircraft Configuration

At the conclusion of the preliminary design, the design parameters and sizing trades had all been investigated and the final aircraft component selection and systems architecture was ready to be selected. The preliminary design determined the sizing of the wing and tail geometry, primary structure dimensions, and propulsion configuration. The wingspan and chord length were calculated to be 8 feet and 12 inches, respectively. The S4083 airfoil was selected for the wing, while the NACA0009 was chosen for the tail. The most efficient shape for the spar was found to have a spanwise taper. The aircraft's takeoff weight was found to be approximately 19.8 pounds and the center of gravity was determined to be 1.5 inches behind the leading edge of the wing. An 18 inch diameter propeller with a 16 inch pitch, Graupner 3300-7 motor, and 18 Sanyo 1300 cells compose the propulsion system.

5.0 Detailed Design

With the information gained from the preliminary design phase, a foundation was laid for finalizing component and system architecture selection. The aerodynamics group calculated final wing and tail geometry while the structures group finalized the design of the spar, central platform, and payload bay. The structures group also conducted a detailed analysis of the internal architecture of the fuselage and the propulsion group determined the propeller, motor, and battery cell optimal configuration.

5.1 Aerodynamics

Based on the design parameters and sizing trades investigated during the preliminary design phase, performance characteristics were determined and modified to achieve the mission specifications.

5.1.1 Mission Profile and Flight Performance

The aerodynamic detailed design focused on the mission profiles, which were outlined in the conceptual design phase. In order to optimize the final time and flight score, and hence, the final score, each mission was analyzed for both conventional and performance flight modes:

- Conventional mode: The conventional mode focuses on flight efficiency, minimizing the number of batteries used.
- Performance mode: The performance mode was dominated by the concept of maximizing airspeed in order to minimize flight time.

In order to calculate flight time, each mission was profiled by the aircraft's cruise speed, maximum velocity, turn rate, and clime rate, which were determined from the wing geometry, weight, and thrust. Performing numerous calculations (Raymer 27-110), the cruise and maximum velocities were found to be 57.1 and 84.0 feet per second, respectively. Similarly the turn and climb rates were calculated to be 76.8 degrees per second and 14.2 feet per second, respectively. Each mission flight time was then calculated from these numbers.

The estimated flight times and resulting scores are found in Table 5.1. All of the missions were evaluated with no wind conditions at Standard Temperature and Pressure (STP) and all flight scores included an estimated one-minute assembly time.

Mission Profiling Matrix					
	Conventional		Performance		
Mission (x difficulty factor)	Time (min : sec)	Flight Score	Time	Flight Score	Escore
Fire Fight (2x)	2:16	7.820	2:03	8.786	.966
Ferry (1x)	2:09	0.467	1:41	0.597	.13

Table 5.1 Evaluation of the mission times and flight modes

The two mission flight scores were compared in both conventional and performance modes. In each case, water drop flight scores received the highest flight score. Therefore, this proves the initial assumption that the mission with the most difficulty offered higher scoring potential was correct. This is true whether the aircraft flies in conventional or performance mode.

As shown in Table 5.1, each mission's flight score increased while flying in performance mode. The Fire Fight mission's flight score increased over 10% with the implementation of high performance maneuvering. While not to the same degree as the Fire Fight mission, the Ferry mission flight score also increase while flying in performance mode. Therefore, it was deemed that the performance mode increases the flight score. However, this does not automatically mean that the performance mode will optimize the final score.

The RACs of the two different flight modes were calculated for both missions. While the performance mode reduced flight time, it also required high battery capacitance and weight, thereby penalizing the RAC. The conventional flight and performance modes yielded RACs of 8.02 and 8.332, respectively. However, although the RAC yielded by the conventional flight mode is better than that yielded by the performance mode, the final scores received by these modes with RAC effects were determined to be 96.8 and 105.3, respectively. Therefore, it was determined that high performance mode would increase the overall final score.

5.1.2 Takeoff Distance

Calculations were preformed to determine the aircrafts takeoff distance. Using an estimated stall speed of 44.1 feet per second, the takeoff speed was found to be 48.5 feet per second. For low Reynolds numbers, (250,000 based on mean chord), induced drag is neglected. Therefore, by integrating the ground acceleration (Raymer 565) into Newton's second law, the takeoff distance was found. In order to account for the ground roll, the takeoff distance was increased by 10 percent. A ground-roll distance of 134.7 feet was achieved with an estimated 6.0 pounds of thrust calculated by the propulsion group.

5.1.3 Stability and Control

An aircraft's stability is defined by its ability to return to equilibrium after experiencing a small perturbation. Static stability is a prerequisite for dynamic stability. Pitch stability corrects pitch deflections through a relationship between the main wing and the horizontal stabilizer. For an aircraft to be considered stable the calculated static margin is required to be between 0.25-0.35 of the chord. Yet this will provide an extremely stable aircraft which will require a lot of control effort to turn. Since this competition is performance oriented, a relaxed stability margin of 0.10-0.15 was selected. This relationship determined aerodynamic center to be 1.5 inches behind the center of gravity. Therefore, the airplane proves to be stable, yet maneuverable, as required for the competition. Roll stability was achieved by implementing the high wing design.

Dynamic stability necessitated the placement of the payload cg at the cg of the aircraft, thus restricting the cg movement to a minimum. Furthermore, the restriction of placement of the total cg of the aircraft on the quarter cord provided that the pitching moment of the aircraft would not change during water release portion of the flight.

5.1.4 Control Surface Sizing

To achieve good control characteristics of the aircraft proper control surface sizing is imperative. Using historical guidelines, aileron chord was selected to be 0.2 of the cord or 2 inches and the aileron span to be 0.54 of the cord or 26 inches (Raymer 126-128). Rudder was found to be 9in high and 5in wide fulfilling the guidelines of 90% vertical tail span and 50% chord. Lastly, the guidelines for the elevator are 90% of the span and up to 30% of the chord. The actual size of the elevator was found to be 22in in span and 3 inches in chord. The extension of the elevator span was done to simplify the manufacturability of the tail section.

Pitch stability corrects pitch deflections through a relationship between the main wing and the horizontal stabilizer. For an aircraft to be considered stable the calculated static margin is required to be between 0.15-0.20 of the chord. This relationship, determined by the positions of the center of gravity and the aerodynamic center, was found to be 1.5 inches. Therefore, the airplane proves to be stable, yet maneuverable, as required for the competition. Roll stability was achieved by implementing the high wing design.

5.2 Structures

The structural detailed design focused on the component and systems architecture selection. Integrating and sizing dimensions in order to obtain a proper aircraft load distribution dominated this phase. The main parameter was meeting the competition requirement that the aircraft fit within a 4x2x1 foot container.

5.2.1 Wing Structure

The two load carrying components of the wing are the spar and skin. For analysis, the spar was modeled as a cantilever beam, while the skin was modeled as a shear web. The foam, not being directly in the load path, was not separately modeled.

The spar is designed to transfer the bending forces applied to the wing onto the central platform. The loading of the spar was broken into two cases, a distributed load approximating the maximum aerodynamic loading during flight and a point load representing the load seen during the judge's wing test. This load is taken into separate consideration since the wing will deflect differently than it will during natural distributed aerodynamic loading. MSC-Patran was used to create a finite element model (Figure 5.1). After much iteration an optimal laminate was chosen. Four layers of uni-directional carbon were used in the vertical section of the I-beam, and three layers for each top and bottom horizontal section.

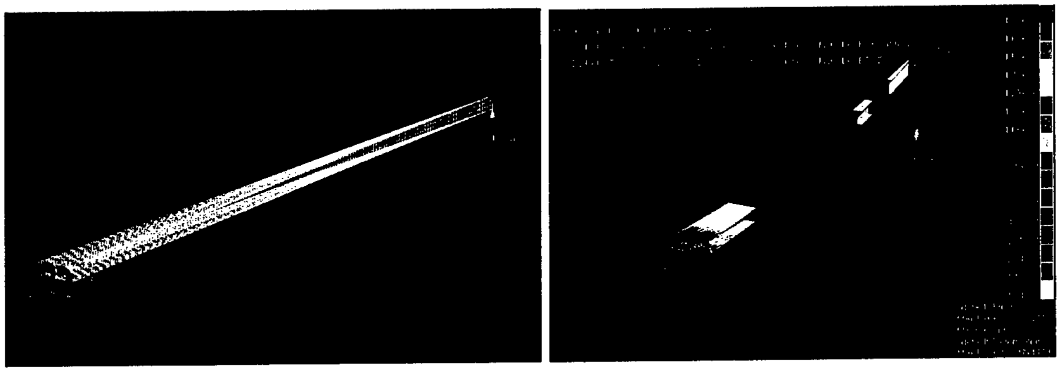


Figure 5.1 von Mesis Stress and Displacement of Spar

The wing skin thickness required for a factor of safety of 1.5 was too small to be manufactured properly. Using MATLAB script, the actual manufactured thickness of 0.01 inches was found, which is two plies of fiber glass.

5.2.2 Tail Configuration

The structural detailed design of the tail section was focused on the component integration of the fuselage, boom, tail and sizing dimensions (Figure 5.2). The boom is designed to attach to the central platform which is the key load carrying component in the fuselage.

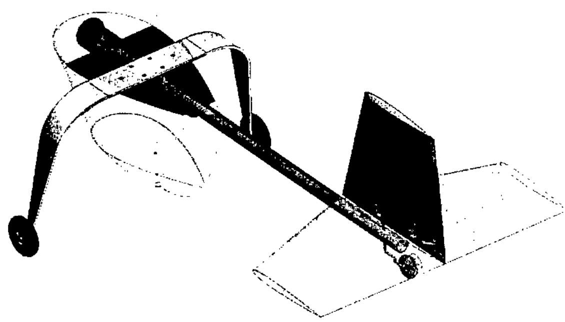


Figure 5.2 Integrated Tail Assembly.

A boom was designed with a 0.5 inch outer diameter and a 0.4966 inch inner diameter. However, a pre-made shaft was selected because of the complexity of manufacturing one from carbon fiber. Unfortunately, a shaft with these dimensions could not be obtained. Therefore, a 0.9 inch shaft with an inner diameter of 0.8 was chosen. The geometry of the boom was verified by applying the estimated loads to the model discussed in the preliminary design. The resulting factor of safety is order of magnitudes greater than that required.

In order to attach the empennage to the boom, analysis was conducted to see how many screws it would take to ensure that the tail stay attached. It was deemed that three bolts were necessary. The horizontal was sandwiched between the vertical and boom. In order to ensure a secure fit, the boom was attached to a hardpoint found in the vertical tail (Figure 5.2). The hardpoint was adhered to the foam and skin of the vertical using cavicil.

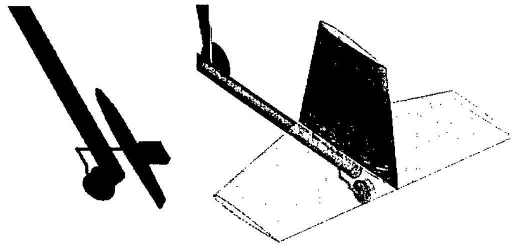


Figure 5.2 Tail Configuration.

While deciding on how the tail should attach to the boom, concerns were addressed about the possibility of a crash damaging the structure. Therefore, nylon screws that shear in the event of a crash were selected in order to absorb damaging kinetic energy.

Two servos were used to power the control surfaces; one was used for the rudder and the other for the elevator. They are connected to the batteries via a wire harnesses also designed to shear.

5.2.3 Fuselage Structure

The detail design of the fuselage focuses on the placement of the CG and the dimensioning of its internal components. The placement of the aircrafts internal components must be placed with precision and accuracy. However, the internal components of the fuselage must also be able to withstand the applied loads. Within the fuselage, the main internal load bearing structure is the central platform. The integration of all aircraft components are found in Figure 5.3.

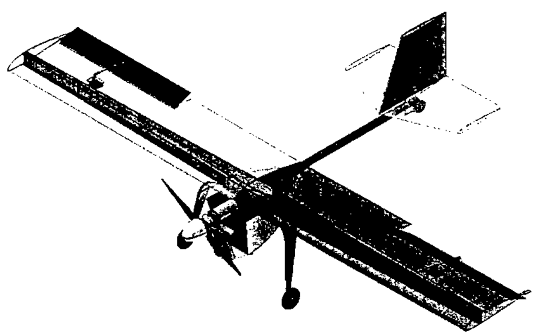


Figure 5.3 Structural Component Integration

There is ample room for the placement of the batteries, receiver and WDS within the fuselage because of the shape which selected in the conceptual design phase. With the receiver and WDS locations fixed, the CG was able to be placed on the quarter chord by locating the 1.35 lb battery pack in approximately 7.5 inches in front of the quarter chord. However, if the actual CG location is inconsistent with the theoretical location due to liberties taken in manufacturing, the batteries can be adjusted shifting the CG closer to its proper location. Due to the placement of the WDS, the CG remains at the same location before and after the payload drop.

As described in the preliminary structural design, the central platform is to absorb all dynamic loads. Because the platform will be exposed to water, this component must not allow water to leak. Therefore a sandwich structure made of carbon-foam-carbon will not be used; instead a single thicker laminate will be used. In order to verify the thicknesses found during preliminary design, a model of the platform was generated with MSC-Patran software (Figure 5.4). The model, however conservative, encompasses the loads generated during a high speed jerk pull up maneuver (approximately 5g's). By applying the proper loads to this model and a factor of safety of 1.5, an iterative process determined the desired material thickness to be six layers of carbon fiber, approximately 0.078 inches thick. The model experiences a maximum stress of 1.3×10^4 psi at the location of the bolt holes (Figure 5.4).

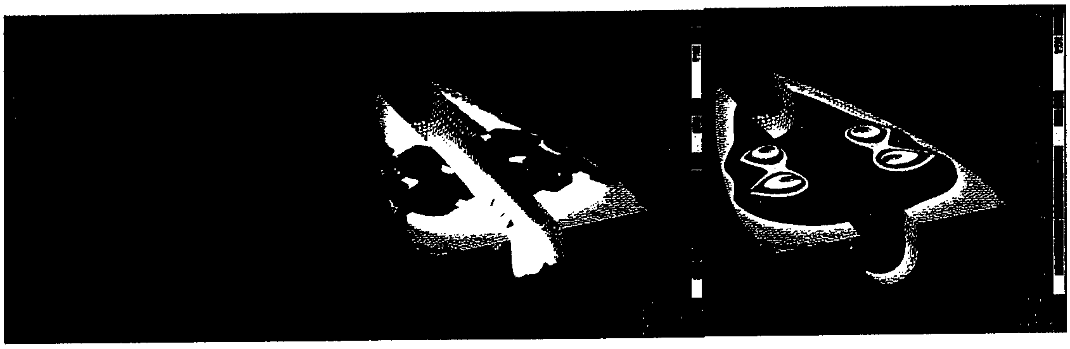


Figure 5.4 von Mesis Stress and Displacement of Platform

As discussed in the preliminary design, the fuselage/water tank is also made of carbon fiber. During modeling, a wall thickness of four plies of carbon laminate was determined. In order to reduce the number of plies, tension bars were integrated to the fuselage at points of highest stress. This lowered the number of plies necessary for the water tank walls to three plies of carbon laminate.

5.2.4 Water Deployment System (WDS)

The water deployment system consists of three major components, the servo controller, cowling and tapered cylindrical valve (Figure 5.5). From the closed position the valve must turn 90 degrees to fully open. From wind tunnel testing, the cowling must rotate approximately 15 degrees to allow the pressure to be minimized. Therefore, the rods', located on the servo arm, orientations must differ in order to achieve the different rotations necessary for each mechanism.

The servo controller rotates the servo-arm which activates the cowling and valve simultaneously via small diameter threaded rods. An additional arm is attached to each mechanism and accepts the threaded rod via threaded receptacles which are allowed to rotate within the mechanism arm. This configuration will allow each mechanism to be "fine tuned".

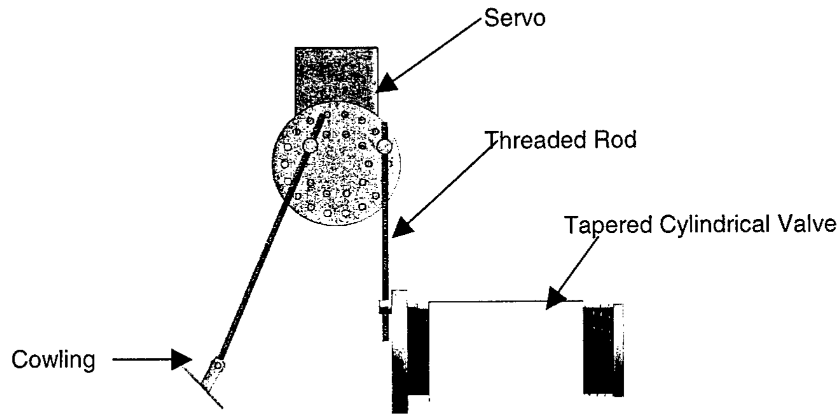


Figure 5.5 WDS Valve Mechanisms

5.3 Propulsion

The final propulsion system was selected from comparing both actual and analytical results. There were discrepancies in the dynamometer results and those calculated using MotorCalc software. The MotorCalc efficiency of the Graupner 3300-7 motor was 67% for the configuration chosen, while the dynamometer efficiency results were over 85%. With the actual efficiency being higher, the thrust output is expected to increase. The mission profiling provided the overall usage of power during mission. For the firefighting mission the Sanyo 1300SCR will provide more then sufficient power for the mission. The energy stored in the batteries is barely sufficient to complete the ferry mission.

5.4 Final Aircraft Configuration

The final configuration was an aircraft with a high-wing, and a moderate aspect ratio of 9.60. Additionally the wings contained no sweep or taper because of the increased cost with negligible increase in performance. The aircraft had good gliding capabilities because of high 15.32 lift to drag ratio. The fuselage was shaped like a tapered and swept wing giving a minimal drag penalty. The tail section was a conventional T-tail with span of 22 inches and height of 11 inches. The landing gear was a tail dragger configuration with the main wheels 2.48 inches behind the CG with an aircraft ground clearance of approximately 1.88 inches. These dimensions were considered necessary so the fuselage does not scrape should a hard impact landing occur. The propulsion system is made up of a Graupner 3300-7 motor, 18x16 propeller, and 18 Sanyo cells. The aircraft breaks down into four individual modules when placed in shipping mode: two wings, fuselage and tail boom.

5.5 Final Aircraft Configuration Calculations (Table 5.2)

Geometry

Aircraft Length	47.6 in
Aircraft Span	96 in
Aircraft Height	19.5 in
Main Wing Configuration	High Mono-Wing
Main Wing Airfoil	S4083
Main Wing Area	960 in^2
Taper Ratio	1
Aspect Ratio	9.60
Main Wing Control Area	90 in^2 per wing
Horizontal Stabilizer Area	218.9 in^2
Vertical Stabilizer Area	99.5 in^2
Tail Airfoil	NACA 0009
Horizontal Stabilizer Control Area	66 in^2
Vertical Stabilizer Control Area	45 in^2

Performance

CL max (actual)	1.151
L/D max (actual)	15.32
Maximum Rate of Climb	14.43 ft/sec
Stall Speed	44.1 ft/sec
Takeoff Speed	48.5 ft/sec
Cruise Speed	57.1 ft/sec
Maximum Speed	84.0 ft/sec
Takeoff Distance Loaded (Ground Roll)	149 ft
Takeoff Distance Unloaded (Ground Roll)	74 ft

Weight Statement

Airframe Weight	9.04 lbs
Manufacturer's Empty Weight	10.39 lbs
Payload (4 Liters of Water)	8.80 lbs
Battery Weight	1.35 lbs
Motor Weight	1.90 lbs
Control System	0.75 lbs
Takeoff Gross Weight	19.19 lbs

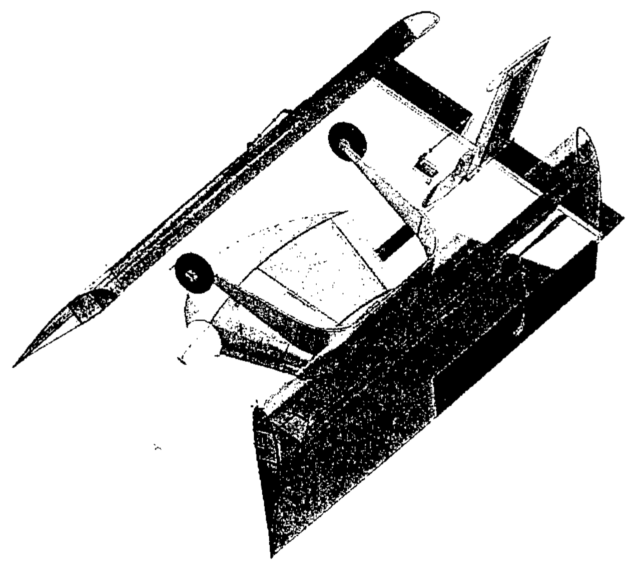
Systems

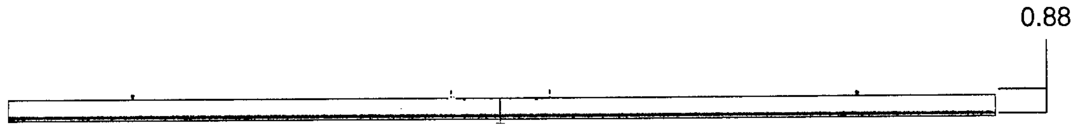
Motor	1 x Graupner 3300-7
Propeller	18x16 thin carbon prop
Gear-ratio	(2:1)
Batteries	Sanyo 1300 (18 cells)
Servos	HS-225MG (2), HS-545BB (2)
Reciever	HPD-07RB (PCM)
Handset	Prism 7X (PCM)

5.6 Rated Aircraft Cost

Coefficient	Item	Cost Per Unit	Value		Total MFHR
	Empty Weight		10.39	lbs	
A	**A * MEW**		**3116**	$	
	# of Engines		1.00	#	
	Battery Weight		1.50	lbs	
B	**B * REP**		**2247**		
WBS 1.0	Wing Span		7.50	ft	
	Max Chord		1.00	ft	
	Wing Span * Max Chord	10	7.50	ft^2	75
	Ailerons	5	1.00		5
	Flaperons	7.5			0
	Ailerons + Flaps	10			0
	Ailerons + Spoilers	10			0
	Ailerons + Flaps + Spoilers	15			0
	WBS 1.0 Total				**80**
WBS 2.0	Fuselage Length		5.00	ft	
	Fuselage Width		0.40	ft	
	Fuselage Height		0.83	ft	
	L * W * H	20	1.67	ft^3	33.33333
	WBS 2.0 Total				**33.33333**
WBS 3.0	Vertical Surface w/o control	5	0.00	#	0
	Vertical Surface w control	10	1.00	#	10
	Horizontal Surface <25%	10	1.00	#	10
	WBS 3.0 Total				**20**
WBS 4.0	# Servos / Motor Controller	5	7.00	#	35
	WBS 4.0 Total				**35**
	MFHR Total				**168.3333**
C	**C * MFHR**		**3366.667**		
	TOTAL RAC		**8.729**		

5.7 Detailed Drawings

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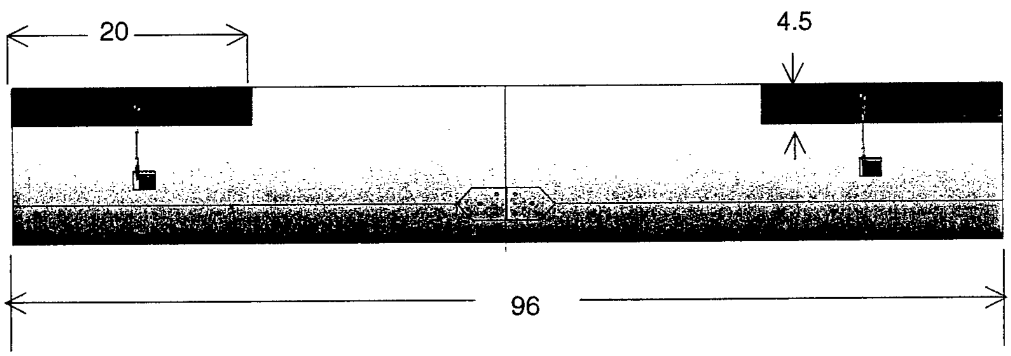
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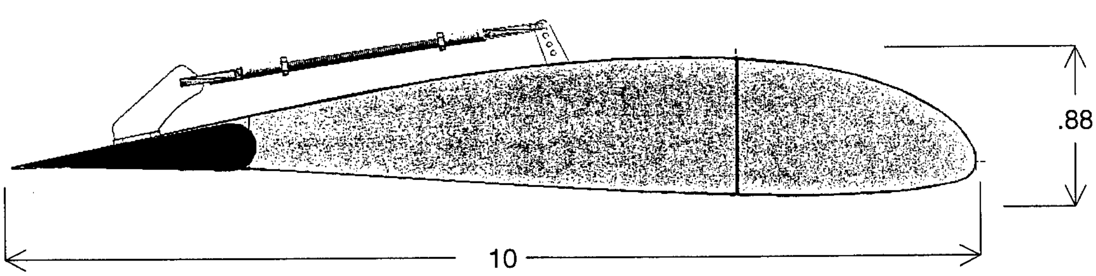
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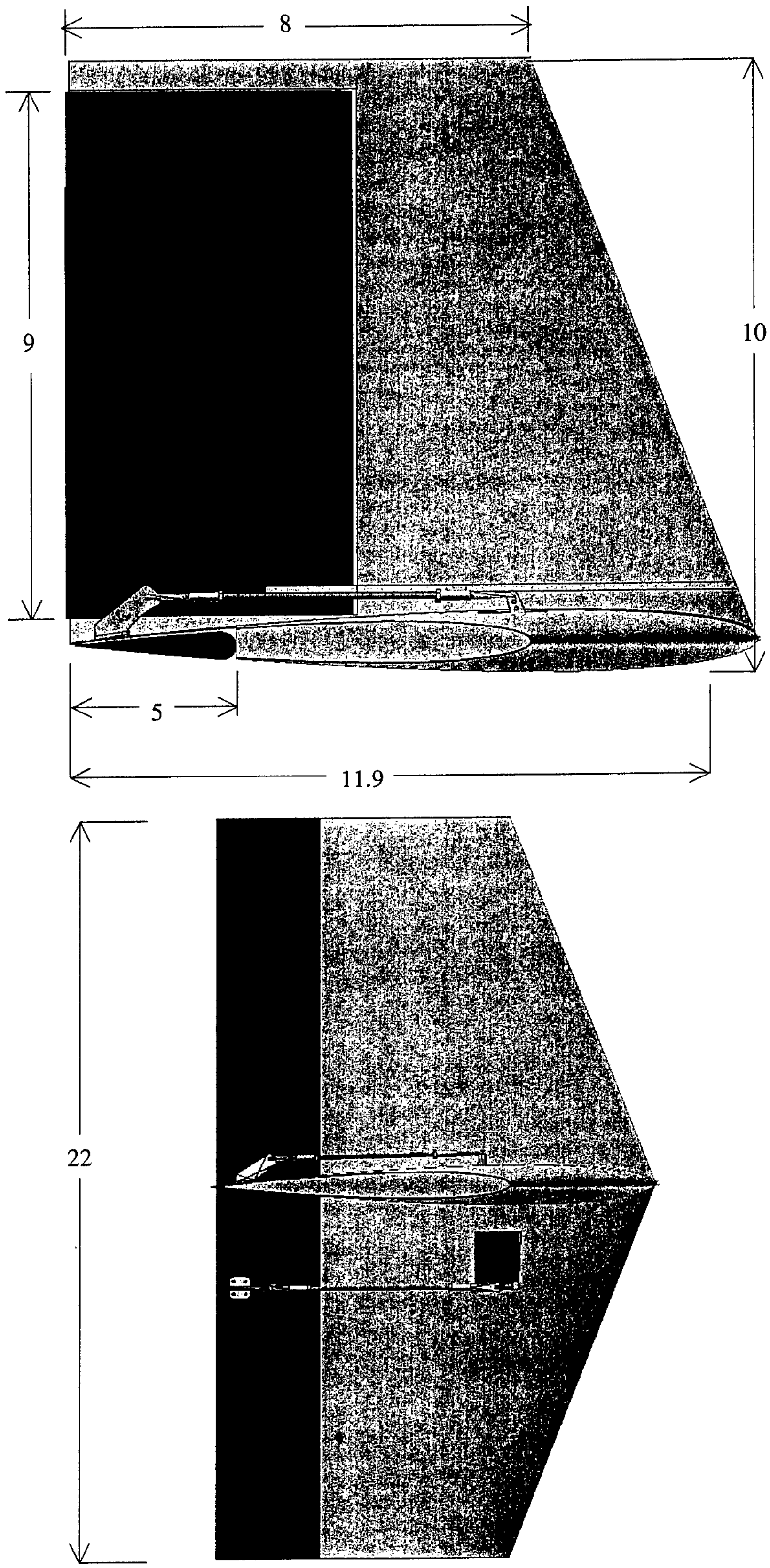
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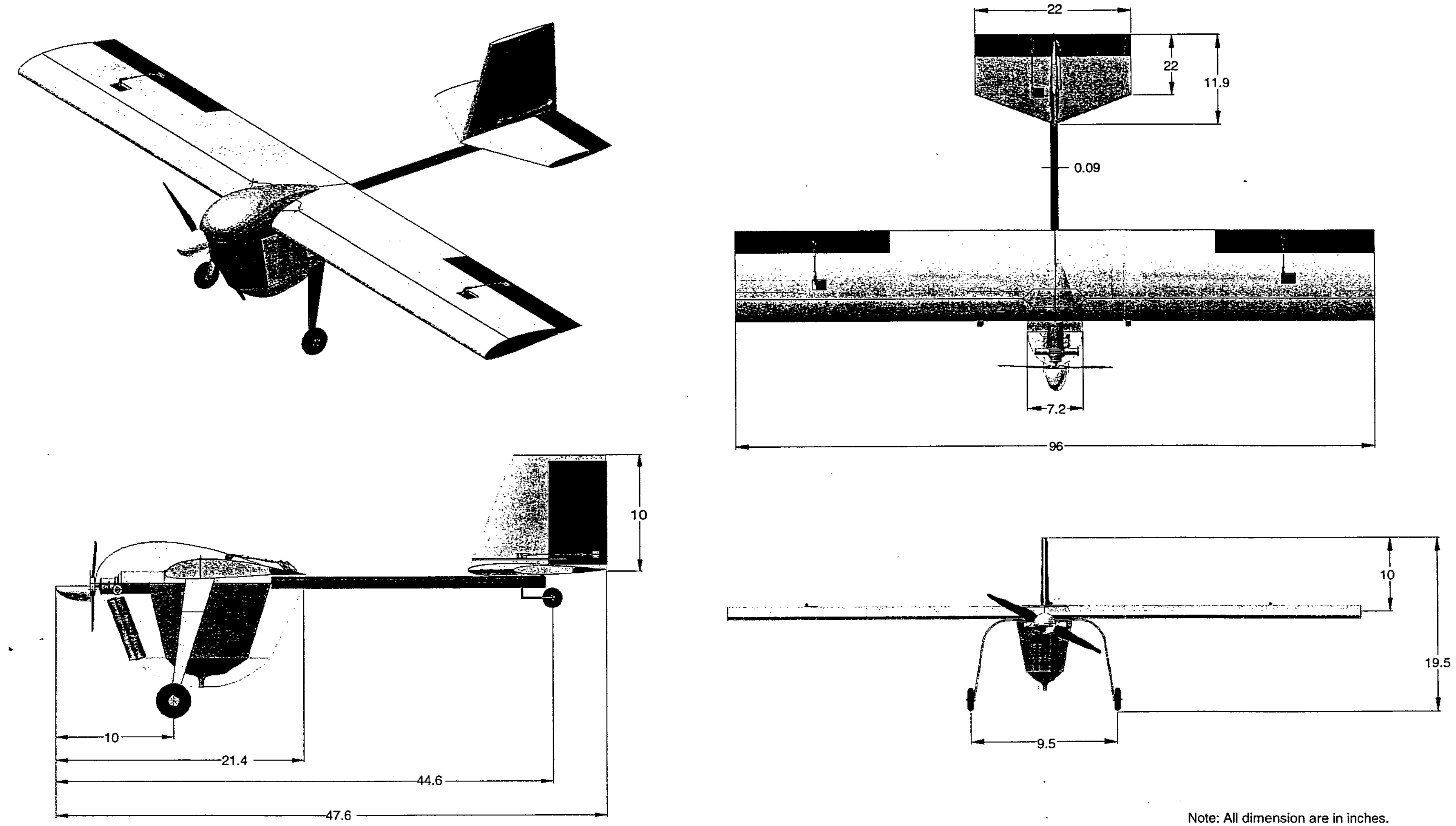


Note: All dimension are in inches.

6.0 Manufacturing Plan

Most of the aircraft is made of carbon fiber or fiberglass and epoxy resin. The techniques to manufacture each individual part vary in order to create the shapes with the level of precision required.

Depending on the requirements for each part, the most efficient fabrication method is selected to give the level of precision needed and minimize the cost, time, materials, and tooling. An estimation of the relative costs of each manufacturing technique is shown in Table (6.1) below.

	Material Use	Training	Difficulty	Tools/ Equip-ment	Tooling	Time	Total
Hot Wire	0	1	2	1	1	1	6
Vacuum Bag	0	1	0	1	0	2	4
Mold	2	0	0	2	2	2	8
LaserCamm	1	1	0	2	0	1	4
Autoclave	0	1	1	2	0	2	6

Table 6.1 Fabrication processes and their estimated relative costs

Some parts require more precise dimensions than others, therefore the technique with the lowest cost that can still deliver the accuracy required for the part to be useful is selected.

6.1 Wings

The most critical property of the wings is the airfoil designed by the aerodynamics group. The wings must accurately portray the shape of the chosen airfoil or the aircraft would not perform as expected. Templates of the airfoil were cut using a LaserCamm and a hot wire was used to cut the wings. This method was inexpensive and readily available. Space for the spar and hardpoint were cut out and the foam sanded. Small pockets cut were cut lengthwise to hold servos and brake wires. Bi-directional fiberglass was bonded to the top and bottom of the wings to provide the torsional stiffness. Painted mylar enclosed the wing and then the assembly cured in a vacuum bag.

6.2 Spar

The spar must fit inside the wing and connect to the wing sleeve. The critical dimension of the spar is the thickness and is ultimately determined by the number of layers of unidirectional carbon used. A section from the wing cores was cut for the spar.

First the vertical piece of spar will be adhered to the hardpoint. After the carbon fiber vertical spar was cured was sanded to shape, and then adhered to the wing foam cores that have a section cut for these pieces. The horizontal section of the spar was applied to the wing foam cores then vacuum packed and allowed to cure. Some cavicil was used where the horizontal spar piece should be attached to the vertical section to ensure continuity between the two. After curing, this component was sanded to fit the wing profile and prepped for fiberglass skin.

6.3 Hardpoint

The hardpoint is a crucial piece that transfers wing loads to the fuselage via the spar. To ensure that this piece fits the exact profile of the wing cross-section it will be milled using a 4-axis CNC mill. To ensure that all the wing loads are transferred to the hardpoint, the spar has been adhered to the hardpoint during its construction. Four countersunk holes are drilled into the hardpoint so that the bolts that will hold the wings to the fuselage do not protrude into the oncoming airstream.

6.4 Platform

The platform will experience many types of loading and must be constructed carefully. A mold is made from a flat plate with a round bar running down the center. The mold is covered in release wax so that the piece can be removed after construction. Carbon fiber is lied over the mold, on top of the carbon fiber is an absorbent which will draw excess epoxy out of the part. All this is vacuum bagged and left to cure.

6.5 Skin & Fuselage

The purpose of the skin is to provide additional structural stiffness, house the payload and streamline the fuselage. To produce the designed fuselage accurately plugs will be made with pieces of foam. The foam will be cut using a 3-axis CNC mill. The foam will be treated with a gel coat to obtain a smooth finish, then covered with release wax. After the plug is prepped for usage fiberglass or carbon fiber will cover the plug and all is set in a vacuum bag allowing curing. Once cured, the excess areas were trimmed and sanded. The fuselage is then adhered to the platform using epoxy mixed with cavacil and micro balloons.

6.6 Motor-mount

The fabrication method for the motor-mount was determined to be the vacuum bag because the tight dimensions obtained through creating a mold was not needed for the motor-mount to be functional. Therefore the vacuum back was the method with the smallest cost. A piece of circuit board was cut to the required size. A thin layer of foam was made using a hot wire. Unidirectional carbon fiber was laid on the foam, followed by bi-directional carbon. The carbon foam sandwich structure was vacuum packed and cured. A small mold, a tube just larger than the motor was used to create the motor mount which will cradle the motor. The mold is covered with release wax then covered in a carbon-foam-carbon structure. After its construction, a flat panel is adhered perpendicular to the cradle where the motor is bolted.

6.7 Landing Gear

The landing gear must withstand impact loads and be symmetrical so that the aircraft rolls down the runway straight. These requirements mean that precise and durable parts are needed, negating the importance of low cost fabrication techniques. Therefore, it was decided to make the landing gear with a mold and autoclave. A mold was created by rolling a piece of aluminum to size. Preimpregnated unidirectional carbon fiber was placed in the mold and placed in an autoclave for curing. Then the part was trimmed to size and holes were later drilled for mounting to the aircraft and wheel attachment.

7.0 Research and Testing Plan

Most of the aircraft is made of carbon fiber or fiberglass and epoxy resin. The techniques to manufacture each individual part vary in order to create the shapes with the level of precision required.
Depending on the requirements for each part, the most efficient fabrication method is selected to give the level of precision needed and minimize the cost, time, materials

7.1 Objectives

The purpose of testing confirms or denies a function of a design configuration or design component in the aircraft. The research and testing plan objectives were to gain insight into the drag and lift values of the airfoil types, determine the optimum motor and battery configuration, and the effect of changing the release mechanisms of the payload. Water release tests, wind tunnel testing, and valve testing confirmed or denied if the design used would be sufficient and efficient in the overall final design. These tests allowed the design to achieve maximum development of that particular component or design configuration in turn achieving the best performing aircraft.

7.2 Schedules

A research and testing schedule was organized. Due to the abundant data from Michael Selig's research (Selig) airfoil wind tunnel testing was deemed unnecessary for airfoil selection; however, wind tunnel tests were performed in order to optimize the WDS.

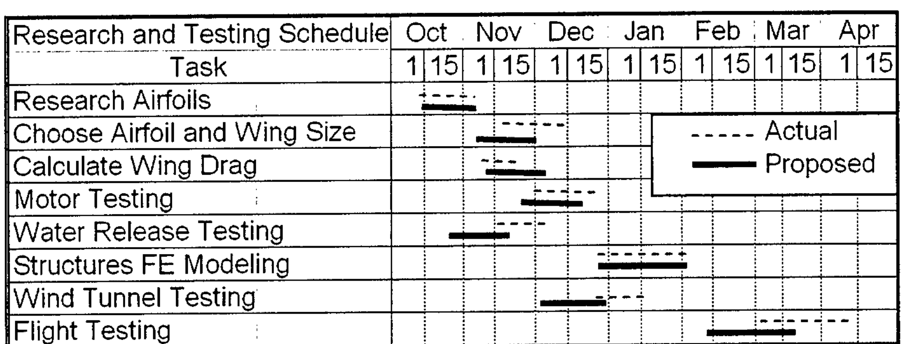


Figure 7.1 Research and Testing Schedule Task

7.3 Testing Procedures

The landing gear is tested for static load bearing ability, as well as for its dynamic properties. The rear landing gear is secured in a cantilever position and 20 pounds of weight is attached to the end of the landing gear. This test is to ensure that the part will be able to hold the entire weight of the airplane. Next, the impact performance of the landing gear is tested by tying the test weight to the end of the landing gear with a string, and the load dropped from a small distance. This will simulate the landing impact of the aircraft on the landing gear. If the landing gear bounces too strongly, damping will be needed to keep the airplane from bouncing back into the air while trying to land.

The structural proof test of the spar is conducted in a similar fashion. This test is to learn whether the spar will withstand the loading being placed on it, as well as to find the deflection of the spar under point loading at the tip of the wing.

The first step of flight testing is to recreate the judges' wing test. The aircraft is lifted by the wing tips for a short period of time. This step is important for safety. Next, the motor is throttled up to full speed while holding the aircraft tail. A series of very simple and gentle maneuvers are performed to check for the general stability of the aircraft. After the flight worthiness and stability have been established, the airplane is then put through some more advanced maneuvers. It is important to know how the aircraft will handle.

7.4 Check-lists

Testing, assembly, pre-flight and post-flight checklists were developed to track testing milestones and ensure safety of flight and personnel.

Testing Checklist

X	Wind tunnel testing
X	Water release testing
	Flight testing
X	Static motor testing
	Dynamic motor testing
X	Structures testing

Figure 7.2 Testing Checklist

Pre-Flight Checklist

	Ensure all batteries are fully charged: motor, receiver and controller
	Check electrical connections and wires for integrity and abrasion
	Inspect fuselage, wings and tail for obvious damage
	Ensure boom and is installed properly
	Inspect landing gear for attachment rigidity
	Ensure wheels spin freely
	Check all flight controls respond to given commands
	Check propeller for cracks, dents and dings
	Check forward fuselage area near motor for FOD
	Ensure payload valve secure
	Install fuse and check for correct rating
	Start and rev motor; listen for unusual vibrations

Figure 7.3 Pre-Flight Checklist

Post-Flight Checklist

	Check wing attachment points for damage
	Inspect wings for structural integrity and skin for ripples
	Inspect motor mount and firewall for cracks or delaminations
	Inspect motor connecting bolts for tightness
	Check motor, motor speed control and battery pack for overheating damage
	Ensure vertical and horizontal tail sections are firmly attached and no loosening has occurred
	Check electrical connections for abrasion or overheating
	Check fuse is not blown
	Inspect propeller for cracks, dents or dings

Figure 7.4 Post-Flight Checklist

7.5 Results

7.5.1 Airfoil Results

The results from the airfoil data are shown in Figure 7.5.

Airfoil Data

Airfoil	Cl	Cdo	Wing L/Dmax	Aircraft Drag [lb]
S4083	1.3	0.008	14.7	1.82
E214	1.3	0.01	13.6	2.00
E423	1.95	0.017	13.1	2.64
S1210	1.8	0.015	12.9	2.45
S1223	2.1	0.02	11.7	2.91

Reynolds number 300,000

Figure 7.5 Airfoil Data

The difference in drag values between the S4083, the chosen airfoil for the Rain of Terror, and the S1223 is 1.09 pounds. This 60% increase in drag is substantial because it would require additional thrust. In attempts to keep the battery weight low, the choice of the S4083 airfoil was prudent because less thrust is required, which corresponds to less battery weight.

7.5.2 Motor Testing Results

Static testing was performed on a dynometer and data collected using Labview software. A relevant portion of the Graupner Ultra 3300-5 motor data is shown below.

Graupner Ultra 3300-5 (5-winding) Motor Data

	# Cells	Volts	Motor RPM	Motor Eff [%]
MotorCalc Data	18	18	7955	52.5
Test Data	18	18	5700	87.0

Figure 7.8 Graupner Motor Data

It was clear that actual static testing must be performed to validate MotorCalc software results.

7.6 Water Release Testing

The water release testing consisted of a bucket with a hole, a scale, and a timer. The bucket was filled with water, weighed, and the water was released with the timer running. The bucket was reweighed to determine the amount of water that was released. The water release testing was done without a tube and then with a four foot two inch long tube. The time it took to release the water was plotted against the amount of water released, Figure 7.9.

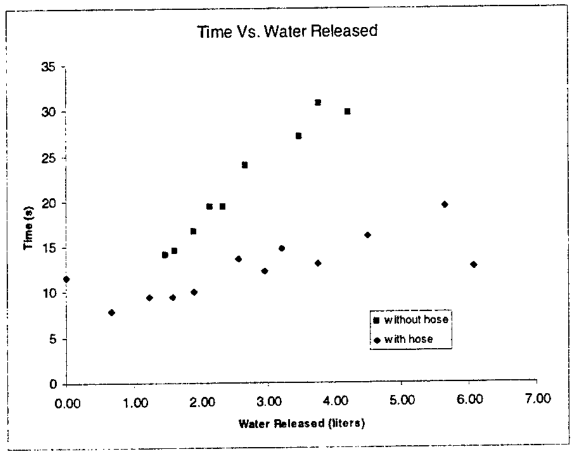
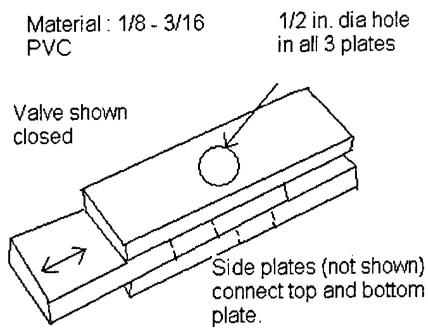


Figure 7.7 Time Vs. Water Released Without the Hose

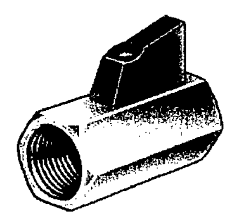
The results confirmed that the design component of a tube attached to the fuselage would have more effective time to release the water. The benefit of having a long hose dangling from the fuselage was increased water flow rate due to increased height difference from the water source to exit. This device has been deemed illegal for safety reasons; the hose must not drag on ground after landing. To raise the hose after payload release. This would cause the need for more batteries, motors and servos, resulting in an increase in the RAC by 0.35. The aircraft design was configured without a tube attached as a result.

7.7 Valve Testing

During the conceptual design phase various methods for releasing the payload were considered: sliding gate valve and "off the shelve" ball valve, and a cylindrical valve. Figure 7.10 show the gate valve and the ball valve. Determining how the water is released and how it is integrated into the aircraft system were important factors in the decision process. Each valve had its advantages and disadvantages for the aircraft performance.



Gate Valve



"Off The Shelve" Ball Valve

Figure 7.8 The Design Valves Tested

During the preliminary design, each gate valve was acquired and tested to determine the most effective and sufficient design. The release knobs were tested to see how smooth it turned and how much force would be required to turn the knobs. The gate valve design included a flat plate with a one half inch hole, which could slide in and out of a sleeve that housed an o-ring in order to assure no water leaked when it was in its closed position. However, the gate valve needed complicated mechanisms to open the valve causing the valve to be of larger dimensions. These necessary dimensions failed to fit in the designated valve space on the aircraft. The "off the shelve" ball valve design was bulky in size, and therefore, weighed heavier than the gate valve. A problem that came up when testing the "off the shelve" valve was that large amounts of torque was required to open it. Both suggested valves had problems either with weight or size.

A modified cylindrical valve was then tested. This modified valve was press fit into a receptacle, both which were machined from a Teflon. Teflon was used for its low coefficient of friction. Overall this design mechanically worked with the efficient material of Teflon. This "modified" designed valve and its device to operate it fit well in the designated space. However, this "modified" design also required more torque to open the valve than the servos provided. After more modification and design thought to reduce the amount of torque needed to operate this design while keeping its effectiveness and ease of manufacture, a tapered cylinder was created with a matched tapered receptacle. A spring was used in order to draw the two tapered pieces closer together, providing a good seal. Yet, vacuum grease added to aid the seal of the two tapers. The "modified" valve is in Figure 7.11 below.

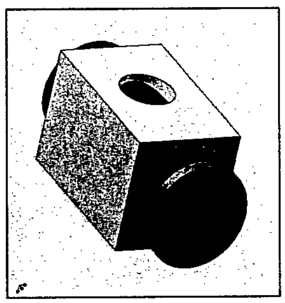


Figure 7.9 "Modified" Valve Used in Final Design

In conclusion for the release mechanism, the "modified" valve proved to be the most effective and a sufficient design for the Reign of Terror's water release mechanism.

7.8 Wind Tunnel Testing

The wind tunnel located at the UCSD Engineering Building II, room 107 (Figure 7.12) was used to determine if the cowling was beneficial to help release the water from the fuselage and determine if the fuselage shape provides extra lift. The speed of the water release is crucial for the mission's success and was the main inspiration for wind tunnel testing.

A .3 scaled model of the aircraft was created and tested. The model was mounted in the small wind tunnel testing space as shown in the Figure 7.12. The model was rotated at the mount left and right to change the angle of attack during testing. The pressure readings were taken from holes drilled along the top and bottom of the model, and the tubes were attached to the wind tunnel pressure reader.

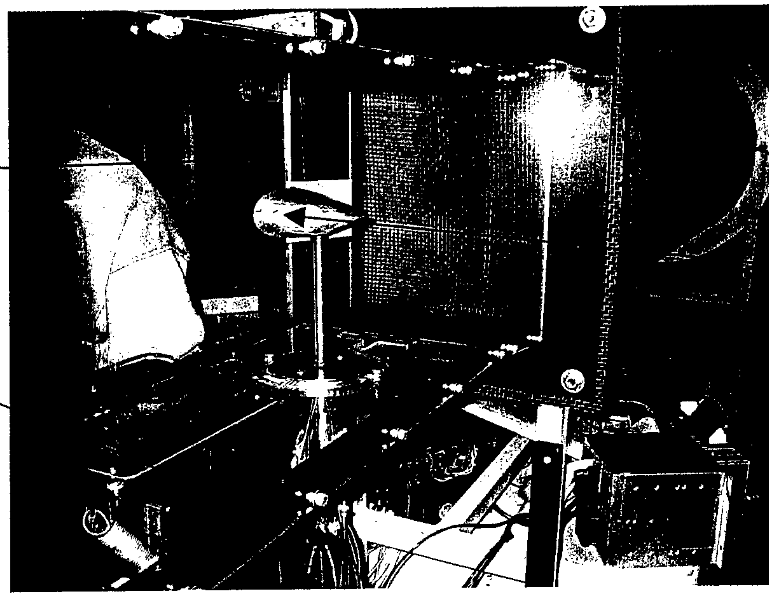


Figure 7.10 The Wind Tunnel Setup With Model

There were two tests: without the cowling and with the cowling. The pressure readings were taken at angles of attack of –10, -5, 0, 5, and 10 degrees at a range of velocities of zero to 150 mph with 10 degree increments.

The first experiment, analyzed the airflow over the model without cowling as seen in Figure 7.13.

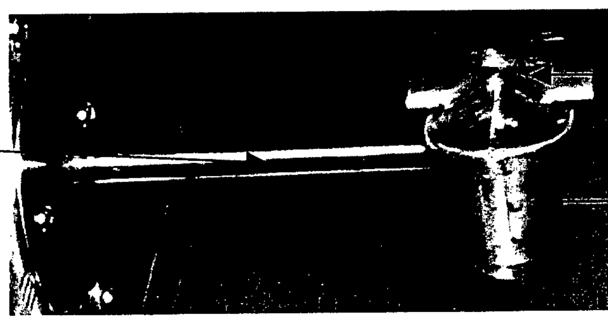


Figure 7.11 30% Scaled Fuselage and Partial Wing Model of Final Design

The second test had the same .3 scaled model, but included a cowling located 2.5 inches from the leading edge of the fuselage, which is on the very bottom of the structure, Figure 7.14.

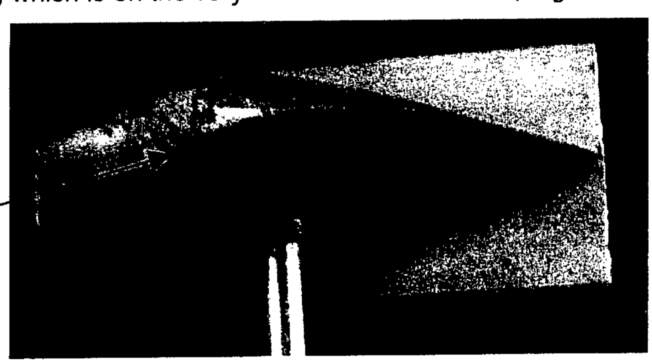


Figure 7.12 30% Scaled Model with Cowling

The pressure readings for the two experiments were converted to coefficients of pressure for comparison and plotted in Figure 7.15.

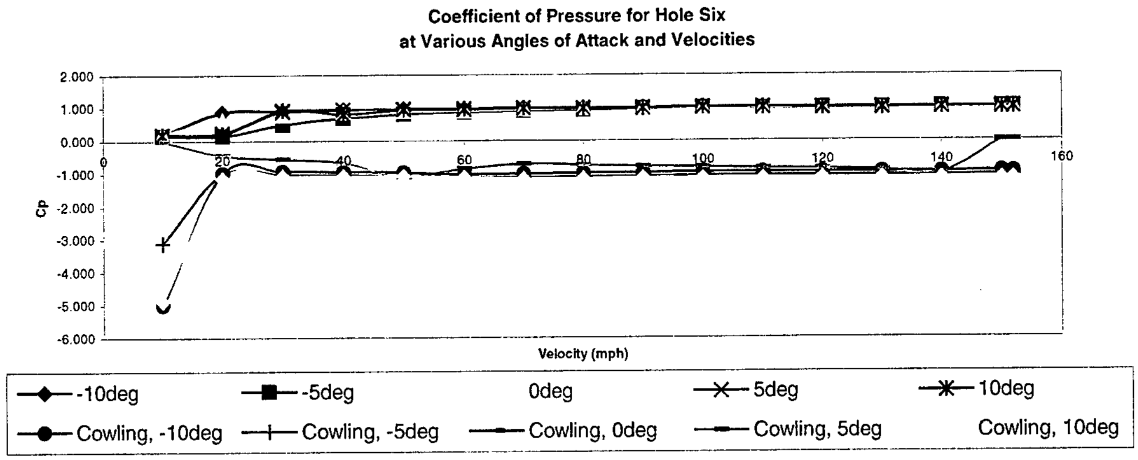


Figure 7.13 Wind Tunnel Results at Pressure Hole 6

The data showed that there was a pressure differences when using the cowling. The coefficients of pressure became negative with the cowling, causing a vacuum like reaction at the cowling's location. This result is beneficial in releasing the water from the payload bay. From the results, it was concluded that the addition of a cowling to the fuselage, in front of the hole where the water is released, will increase the water flow through the hole and reduce the time to completely release the water, thus, improving the mission flight time.

From the coefficients of pressure, the normal force coefficient was determined using the trapezoidal rule to integrate across the fuselage body. From the normal force coefficient the coefficient of lift was determine from by the relationship:

$$L > N\cos(\beta) .\ A\sin(\beta) \text{ where } \beta = \text{angle of attack (radians)}$$

The data for the axial force was undetermined because the wind tunnel was not equipped to take data along the axial length of the fuselage. The lift calculated was plotted against each angle of attack tested, Figure 7.16.

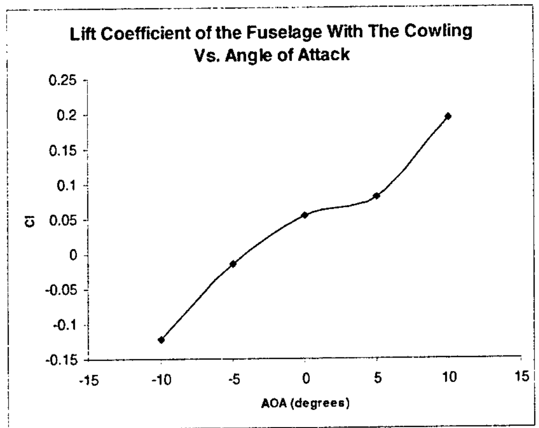


Figure 7.14 Lift Coefficient of the Fuselage with the Cowling Versus Angle of Attack

The results showed that as the angle of attack was increased, the fuselage produces more lift. The fuselage shape has been confirmed to be a sufficient design in wind tunnel testing, and thus, accepted as the final design of the fuselage.

7.9 Lessons Learned

The airfoil data and resulting calculations for the total drag on the aircraft varied. Choosing an airfoil because of its high coefficient of lift (Cl) value is not always the best choice. Comparing the S4083 and S1223 airfoils, it is clear that the latter Cl is much higher. However, the additional drag associated with the S1223 airfoil creates more than a pound of additional drag.

The main lesson learned from the motor testing was the actual test data varied greatly from MotorCalc software calculations. In the case of the Graupner Ultra 3300-5 motor, the motor speed and efficiency with 18 volts is grossly in error. MotorCalc predicts a motor speed of 7955-rpm and 52.5% efficiency while actual static test data gave 5700-rpm and 87% efficiency. Results from the actual test data were used in determining the best motor-battery configuration.

The valve testing proved that a cylindrical valve works better than gate and ball valves; and a taper cylindrical valve works better than a cylindrical valve. These tests also demonstrated that the payload is released faster with a cowling than without a cowling. This shows that the speed of the aircraft has a negative effect on the release of the fluid. Therefore, by blocking the air flow, a vacuum is created and the fluid is pulled out.

8.0 References

1. Abbot I.H., Theory of Wing Sections, Dover Publications Inc, 1949.
2. Nelson, R., Flight Stability and Automatic Control, McGraw Hill, 1989.
3. Nicolia, L..M., Fundamentals of Aircraft Design, METS Inc, 1984.
4. Peery, D.P., Aircraft Structures, McGraw-Hill Inc., 1982.
5. Raymer, D.P., Aircraft Design: A Conceptual Approach, AIAA Education Series, 1989.
6. McMaster-Carr Supply Company. Copyrighted 2004. http://www.mcmaster.com/ Accessed: February 2, 2004.