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By Gregory Page, Chris Bovias, and the student participants of D/B/F 2004.
Compiled by Stephen Brock, AIAA Student Programs

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American Institute of Aeronautics and Astronautics
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1801 Alexander Bell Dr., Ste 500
Reston, VA 20191-4344

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This report is made up of the combined reports of 37 separate teams of students who entered the 2004 Design, Build & Fly Competition. The objectives of the Design, Build & Fly Competition were to have students teams design, build and fly unmanned remote control electric aircraft designed for two specific missions: a ferry mission and a fire bomber mission. A "fly-off" took place on the Cessna Air Field in Wichita, KS, in April 2004. Winners of the contest: 1st place, Oklahoma State University; 2nd, Oklahoma State University; 3rd, University of Illinois at Urbana-Champaign. Cessna, the Office of Naval Research and the AIAA Foundation supported the Design, Build & Fly Competition.

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2003-2004 Design, Build, Fly Competition

Place: Wichita, KS—Cessna Airfield

Ranking by Total Score
Order Within this Document

Position	University	Entry Name	Paper Score	RAC	Flight Score	Total Score
1	Oklahoma State University	OSU Black	94.9	6.74	12.1	170.46
2	Oklahoma State University	OSU Orange	80.53	6.71	11.95	143.49
3	University of Southern California	Scquirt	84.85	8.62	10.07	99.13
4	University of Illinois Urbana/Champaign	Fiberglass Overcast	93.65	9.51	9.73	95.82
5	Cal Poly San Luis Obispo	Moist	91.5	7.84	7.98	93.08
6	La Sapienza Rome	Galileo V	90.8	9.04	4.18	42.02
7	Queen's University	Squirt	86.05	6.57	3.11	40.65
8	Virginia Tech	Turbulence Syndrome	88.25	15.4	4.76	27.3
9	University of Florida	Swamp Stuka	62.23	12.56	5.41	26.83
10	Western Michigan University	Yeager - Bomber	70	20.27	5.42	18.72
11	Mississippi State University	Phyxius	88.1	9.5	1.88	17.41
12	University of California Los Angeles	ENGINuity-IGI	64	15.45	1.73	7.17
13	Clarkson University	Kamikazes	90	17.21	1.22	6.38
14	Georgia Institute of Technology	Buzzweiser	92.5	7.21	0.36	4.64
15	Cal Poly Pomona	Paani Hawaii Jahaz	76.05	16.26	0.66	3.1
16	University of California San Diego	Rain of Terror	91	9.74	0.24	2.21
17	Washington State University	Spirit of Procrastination	85.5	8.72	0.17	1.71
18	Case Western Reserve University	Flying Nemo	80.7	12.3	0.22	1.45
19	West Virginia University	Right Flyer	54.5	13.9	0.27	1.05
20	Columbia University	The Leaking Lion	67	15.8	0.24	1.03
21	University of Central Florida	Poseidon's Fury	87.25	12.55	0.11	0.8
22	*La Sapienza Rome	Michelangelo	84.45	15.04	0	0.01
26	*West Virginia University	Sting Ray	33.5	15.82	0	0
37	†West Point	Knight Wings	43.85	100	0	0
36	°University at Buffalo	MAX DRAG	59	100	0	0
24	*University of Texas Arlington	Volantis	63.87	17.5	0	0
35	°University of Colorado	Water Buffalo	64.75	100	0	0
34	†United States Military Academy	Boombatz Grandioso	65.15	100	0	0
25	*University of Arizona	Aircat 2004	68	18.99	0	0
33	†Miami University	Swoop-a-Loop	68.45	100	0	0
32	°University at Buffalo	Buffalo Bomber	76	100	0	0
23	*Istanbul Technical University	ATA-5	77.68	17.87	0	0
31	°Cal Poly Pomona	Death from Above	78.8	100	0	0
30	°University of California Los Angeles	Aquanaut	79.5	100	0	0
29	†Middle East Technical University	Anatolian-Craft	83	100	0	0
28	°City College of New York/CUNY	Rhino	87	100	0	0
27	†University of Texas Austin	121.5	88.5	100	0	0

2003-2004 Design, Build, Fly Competition

Place: Wichita, KS—Cessna Airfield

Ranking by Paper Score

Position	University	Entry Name	Paper Score	RAC	Flight Score	Total Score
1	Oklahoma State University	OSU Black	94.9	6.74	12.1	170.46
4	University of Illinois Urbana/Champaign	Fiberglass Overcast	93.65	9.51	9.73	95.82
14	Georgia Institute of Technology	Buzzweiser	92.5	7.21	0.36	4.64
5	Cal Poly San Luis Obispo	Moist	91.5	7.84	7.98	93.08
16	University of California San Diego	Rain of Terror	91	9.74	0.24	2.21
6	La Sapienza Rome	Galileo V	90.8	9.04	4.18	42.02
13	Clarkson University	Kamikazes	90	17.21	1.22	6.38
27	†University of Texas Austin	121.5	88.5	100	0	0
8	Virginia Tech	Turbulence Syndrome	88.25	15.4	4.76	27.3
11	Mississippi State University	Phyxius	88.1	9.5	1.88	17.41
21	University of Central Florida	Poseidon's Fury	87.25	12.55	0.11	0.8
28	°City College of New York/CUNY	Rhino	87	100	0	0
7	Queen's University	Squirt	86.05	6.57	3.11	40.65
17	Washington State University	Spirit of Procrastination	85.5	8.72	0.17	1.71
3	University of Southern California	Scquirt	84.85	8.62	10.07	99.13
22	*La Sapienza Rome	Michelangelo	84.45	15.04	0	0.01
29	†Middle East Technical University	Anatolian-Craft	83	100	0	0
18	Case Western Reserve University	Flying Nemo	80.7	12.3	0.22	1.45
2	Oklahoma State University	OSU Orange	80.53	6.71	11.95	143.49
30	°University of California Los Angeles	Aquanaut	79.5	100	0	0
31	°Cal Poly Pomona	Death from Above	78.8	100	0	0
23	*Istanbul Technical University	ATA-5	77.68	17.87	0	0
15	Cal Poly Pomona	Paani Hawaii Jahaz	76.05	16.26	0.66	3.1
32	°University at Buffalo	Buffalo Bomber	76	100	0	0
10	Western Michigan University	Yeager - Bomber	70	20.27	5.42	18.72
33	†Miami University	Swoop-a-Loop	68.45	100	0	0
25	*University of Arizona	Aircat 2004	68	18.99	0	0
20	Columbia University	The Leaking Lion	67	15.8	0.24	1.03
34	†United States Military Academy	Boombatz Grandioso	65.15	100	0	0
35	°University of Colorado	Water Buffalo	64.75	100	0	0
12	University of California Los Angeles	ENGINuity-IGI	64	15.45	1.73	7.17
24	*University of Texas Arlington	Volantis	63.87	17.5	0	0
9	University of Florida	Swamp Stuka	62.23	12.56	5.41	26.83
36	°University at Buffalo	MAX DRAG	59	100	0	0
19	West Virginia University	Right Flyer	54.5	13.9	0.27	1.05
37	†West Point	Knight Wings	43.85	100	0	0
26	*West Virginia University	Sting Ray	33.5	15.82	0	0

CESSNA ONR / STUDENT DESIGN/BUILD/FLY COMPETITION

An AIAA Student Activity

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

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

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

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

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

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

Position	University	Entry Name	Paper Score	RAC	Bomber Score	Ferry Score	Flight Score	Total Score
1	Oklahoma State University	OSU Black	94.90	6.74	11.73	0.37	12.10	170.46
2	Oklahoma State University	OSU Orange	80.53	6.71	11.66	0.29	11.95	143.49
3	University of Southern California	Scquirt	84.85	8.62	10.07	0.00	10.07	99.13
4	University of Illinois Urbana/Champaign	Fiberglass Overcast	93.65	9.51	9.39	0.34	9.73	95.82
4	University of Southern California	Scquirt	84.85	8.62	10.07	0.00	10.07	99.13
5	Cal Poly San Luis Obispo	Moist	91.50	7.84	7.53	0.44	7.98	93.08
6	La Sapienza Rome	Galileo V	90.80	9.04	3.87	0.31	4.18	42.02
7	Queen's University	Squirt	86.05	6.57	2.74	0.37	3.11	40.65
8	Virginia Tech	Turbulence Syndrome	88.25	15.40	4.58	0.18	4.76	27.30
9	University of Florida	Swamp Stuka	62.23	12.56	5.20	0.21	5.41	26.83
10	Western Michigan University	Yeager - Bomber	70.00	20.27	5.21	0.22	5.42	18.72
11	Mississippi State University	Phyxius	88.10	9.50	1.50	0.38	1.88	17.41
12	University of California Los Angeles	ENGINuity-IGI	64.00	15.45	1.43	0.30	1.73	7.17

13	Clarkson University	Kamikazes	90.00	17.21	0.98	0.24	1.22	6.38
14	Georgia Institute of Technology	Buzzweiser	92.50	7.21	0.00	0.36	0.36	4.64
15	Cal Poly Pomona	Paani Hawaii Jahaz	76.05	16.26	0.47	0.20	0.66	3.10
16	University of California San Diego	Rain of Terror	91.00	9.74	0.00	0.24	0.24	2.21
17	Washington State University	Spirit of Procrastination	85.50	8.72	0.00	0.17	0.17	1.71
18	Case Western Reserve University	Flying Nemo	80.70	12.30	0.00	0.22	0.22	1.45
19	West Virginia University	Right Flyer	54.50	13.90	0.00	0.27	0.27	1.05
20	Columbia University	The Leaking Lion	67.00	15.80	0.00	0.24	0.24	1.03
21	University of Central Florida	Poseidon's Fury	87.25	12.55	0.00	0.11	0.11	0.80
22	*La Sapienza Rome	Michelangelo	84.45	15.04	0.00	0.00	0.00	0.01
23	*Istanbul Technical University	ATA-5	77.68	17.87	0.00	0.00	0.00	0.00
24	*University of Texas Arlington	Volantis	63.87	17.50	0.00	0.00	0.00	0.00
25	*University of Arizona	Aircat 2004	68.00	18.99	0.00	0.00	0.00	0.00
26	*West Virginia University	Sting Ray	33.50	15.82	0.00	0.00	0.00	0.00
27	†University of Texas Austin	121.5	88.50	100.00	0.00	0.00	0.00	0.00
28	°City College of New York/CUNY	Rhino	87.00	100.00	0.00	0.00	0.00	0.00
29	†Middle East Technical University	Anatolian-Craft	83.00	100.00	0.00	0.00	0.00	0.00
30	°University of California Los Angeles	Aquanaut	79.50	100.00	0.00	0.00	0.00	0.00
31	°Cal Poly Pomona	Death from Above	78.80	100.00	0.00	0.00	0.00	0.00
32	°University at Buffalo	Buffalo Bomber	76.00	100.00	0.00	0.00	0.00	0.00
33	†Miami University	Swoop-a-Loop	68.45	100.00	0.00	0.00	0.00	0.00
34	†United States Military Academy	Boombatz Grandioso	65.15	100.00	0.00	0.00	0.00	0.00
35	°University of Colorado	Water Buffalo	64.75	100.00	0.00	0.00	0.00	0.00
36	°University at Buffalo	MAX DRAG	59.00	100.00	0.00	0.00	0.00	0.00
37	†West Point	Knight Wings	43.85	100.00	0.00	0.00	0.00	0.00

†Not present at the Fly-off
*Present at Fly-off, but did not complete a successful flight
°Present at Fly-off, but did not complete Tech Inspection

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

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

See you next year - Greg Page: Contest Administrator

Right Flyer

2004 DBF Competition
West Virginia University



Authors

Anthony N. Favorito
Lucas S. Roselius
Sandeep Vennam
Bruce Bennett
Reece Mcgregor
Brian C. Abe
Danielle Corbin
Shanti D. Hamburg
Frank Wineland

Advisors:

John Loth and Richard Guiler

1 **Executive Summary** 1
1.1 Overview of the Design Development 1
1.2 Design Alternatives Investigated 2
1.3 Highlights of the Development process 2
2 **Management Summary** 2
2.1 Structure of the design team 2
2.2 Configuration and Schedule 3
3 **Conceptual Design** 4
3.1 Mission Requirements 4
3.1.1 Aircraft Storage, Timed Assembly, Takeoff and Landing 4
3.1.2 Fire Fight Mission (FFM) 4
3.1.3 Mission Penalties 5
3.1.4 Ferry Mission 5
3.1.5 Aircraft cost-effectiveness 5
3.2 Aircraft Configurations Studied 5
3.3 Numerical Figured of Merit (FOM) 6
3.3.1 Takeoff Distance 7
3.3.2 Water Deployment 7
3.3.3 Rated Aircraft Cost (RAC) 7
3.4 Mission Modeling and Analysis 8
3.4.1 Total Aircraft Weight 8
3.4.2 Drag 8
4 **Design** 9
4.1. Aircraft Comparison 10
4.1.1 Comparison of Weights, Performance and Geometry 11
4.1.2 Discussion on Other Aircraft 11
4.2 Critical Design Parameters 13
4.2.1 Engineering Requirements 13
4.3 Component Selection and System Breakdown 13
4.3.1 Stability and Control System 14
4.3.2 Propulsion System 15
4.3.2.1 Motor Size and Number 16
4.3.2.2 Propeller Pitch and Diameter 17
4.3.2.3 Battery Weight and Number and Cells 17
4.3.3 Load Factor Limit 17
4.4 Landing Gear 17
4.5 Payload Support and Deployment System 18

4.6 Aircraft Characteristic Input 18
4.7 Final Aircraft Performance Analysis 19
4.8 Predicted Performance 19
5 **Manufacturing Plan** 20
5.1 Manufacturing Process Investigated 20
5.1.1 Center Section 21
5.1.2 Mid-wing 21
5.1.3 Outer-wing 21
5.1.4 Wing Characteristics 21
5.2 Analytical Methods Used 23
5.3 Selected Processes and materials 23
5.4 Manufacturing Milestones 25
5.4.1 Critical Path 25
6 **Testing Plan** 26
6.1 Test objectives and Schedules 27
7 **Process of Creating the Mold** 27
8 **Engineering Drawings** 29
8.1 DBF100-01 30
8.2 DBF100-02 31
8.3 DBF600-01 32
8.4 DBF200-01 33
9 **References** 34

List of Symbols:

FFM = Fire Fight Mission

FOM = figure Of Merit

RAC = Rated Aircraft Cost

UAV = Un-manned Aerial Vehicle

' = Feet

" = Inches

ft = Feet

in = Inches

lb = Pounds

oz = Ounces

CL = 3-D Lift Coefficient

P_1 = Pressure in Water Bladder (lbf)

V_1 = Entrance Velocity (ft/s)

V_2 = Exit Velocity (ft/s)

ρ = Density (lb/in^3)

y = Height Differential in Y-direction (in)

z_1 = original height of water (in)

z_2 = exit height of water (in)

MEW = Manufacturers Empty Weight (lb)

REP = Rated Engine Power (Watts)

MFHR = Manufacturing Man Hours

A, B, C = Constants set forth in rules

V_{Stall} = Stall Velocity (ft/s)

V_{Max} = Maximum Velocity (ft/s)

Cd_{Para} = Parasitic Drag

Cd_i = Induced Drag

cg = Center Of Gravity

cp = center of pressure

L/D = Lift Over Drag

W_{TO} = Take Off Weight

W_E = Empty Weight

AR = Aspect Ratio

Lbf = Pound Force

min = minutes

L = Liters

G = Gravity

Ni-Cad = Nickel Cadmium

mA = Micro Amps

RPM = Revolutions per Minute

Amps Exp = Experimental Amps

Watts Exp. Experimental Amps

RPM Exp = Experimental RPM

Thrust Exp = Experimental Thrust

% Diff. Amps = Percent Difference of Amps

% Diff. Watts = Percent Difference Of Watts

% Diff. RPM = Percent Difference of RPM's

% Diff. Thrust = Percent Difference of Thrust

Prop = Propeller

S_W = Wing Area

Sf = Fuselage Frontal Area

S_{Tires} = Frontal Area of Three Tires

S_{Strut} = Frontal Area Of Landing Gear Strut

W = Gross Weight

L = Lift in Level Flight

Q_{stall} = Dynamic Pressure at Stall

$V_{Rotation}$ = Velocity at Takeoff (ft/s)

C_{Avg} = Average Chord

Cd_v = Drag Coefficient of Vertical Tail

Cd_o = Drag Coefficient of Wing Airfoil

Cd_f = Drag Coefficient of Fuselage

Cd_h = Drag Coefficient of Horizontal Tail

Cd_t = Drag Coefficient of Tires

Cd_s = Drag Coefficient of Strut

1 Executive Summary

This report has been compiled to explain the overall design and the reasons for the design of the Right Flyer. When designing this aircraft, the goal was to develop a UAV that would be able to meet all of the requirements of the AIAA Design, Build, Fly competition, and an aircraft that would be looked at as innovative and unique.

1.1 Overview of the Design Development

In the beginning of the design phase, it had already been decided that a flying wing would be developed to compete in this year's competition. The main aspects that had to be decided were the materials, the control of the aircraft, and the testing plane.

The materials were very important because a main goal was to keep the plane as light as possible. To achieve this, it was decided that composite materials such as carbon, fiber glass, and honey comb would be used for the main shell. The payload deployment system was also looked at very carefully. To cut down on weight, surgical tubing was used in place of heavy material such as PVC or regular plastic tubing. It was also decided that a light weight bladder had to be used. To accomplish this, a helium balloon was chosen as the tank. Every opportunity to cut any extra weight from the valves and other non-tubing aspects of the deployment system was also taken advantage of. Over half of the original plastic on the shut-off valve was removed and a large bulky section of the PVC that connects the bladder to the tubing was also cut off. The batteries were also made especially for the competition. After discussions with the manufacturer, they agreed to make these battery packs extra light by eliminating any unnecessary weight.

Control was important because historically, flying wings have been difficult to design and control. It was decided that wing morphing would be used as the means of controlling the aircraft. This eliminates verticals which in turn eliminates weight and also feeds directly into the second of the two main goals of this aircraft which was to design an innovative and unique aircraft.

Because of the uncertainty and difficulty involved with wing morphing, it was important to have a very strict testing plane. First a small half scale glider was made from foam. The glider was used to make sure that the washout angle was correct to make the aircraft fly stable. After the success of the small glider, a larger glider was cut out of foam so that it could be used as the mold for the main shell. Before the composites were laid over the mold however, several more test flights were done with the full scale model. It was determined that the aircraft would be stable during flight and that the lay-up could begin.

1.2 Design Alternatives Investigated

The main alternative to the design was that originally a Horton II airfoil was going to be used. The original plane for the aircraft actually called for this airfoil. After some research however, it was discovered that at low Reynolds numbers the Horton II was found to suffer from boundary layer separation due to its excessive thickness and its moment coefficient is not positive enough for an un-swept flying wing. Another design alternative that was looked at was the addition of verticals. This had to be examined in the event that the wing morphing fell through. To get accurate data on the wing with the addition of verticals, a wind tunnel model was made and tested.

1.3 Highlights of the Development process

To ensure that everything could fit into the final composite shell, it was decided that a mockup of the center section of the wing would be created. This allowed the propulsion, landing gear and electronics team to layout the main body of the aircraft. The board allowed for the team to determine what elements of the aircraft would fit in what section and it also allowed for different deployment ideas. By being able to set up the deployment system the team was able to try different distances and different exit points for the payload. This mockup also made the installation of the parts into the actual plane a quick and accurate process.

2 Management Summary

The West Virginia University design team for this year's competition consisted of nine Mechanical and Aerospace Engineering students. The team was divided into smaller sub-groups that worked on certain assignment areas of the entire aircraft design. This approach was utilized to create a quality aircraft in the allotted time.

2.1 Structure of the design team

In order to appropriately divide up the work that needed to be done to compete, the ability level and skills of the team members were factors used to create four sub-groups. The aerodynamics controls and flight testing team was responsible for the stability of the aircraft and for studying wing morphing as a control method. The propulsion, landing gear, and electronics team was responsible for determining the best battery/motor/propeller combination, the design of the landing gear, and the wiring and layout of the electronics. They also studied the amount of charging cycles the batteries need to go through to reach their peak performance level. The CAD, procurement, and manufacturing team were responsible for creating the CAD drawings of the aircraft which served as a design and layout aid. They also were responsible for solving the problems of how to implement each design idea and creating the manufacturing plan and process. The fund-raising team was responsible for acquiring the funds necessary to build an aircraft and attend the competition.

One team member was designated as team leader who was in charge of all of the sub-groups. The leader ensured that each sub-groups completed their tasks timely. The sub-groups were created with at least one upperclassmen or more knowledgeable person per team. Table 2.1.1 shows the design team members and their main assignment areas. Other minor design problems were present and needed solving but this table shows the distribution of the major tasks that needed completed.

Table 2.1 – Team structure and assignment areas

Team Leader: Lucas Roselius			
Aerodynamics Controls and Flight Testing	Propulsion, Landing Gear and Electronics	CAD, Procurement and Manufacturing	Fund Raising
Danielle Corbin Lucas Roselius Frank Wineland Shanti Hamburg	Brian Abe Bruce Bennett Reece McGregor	Tony Favorito Sandeep Vennam Ryan Church Ryan Boring Lucas Roselius	Danielle Corbin Reece McGregor

2.2 Configuration and Schedule

In the beginning of the design phase, the team determined the major design accomplishments and dates by which they should be achieved. The team held weekly meeting times to discuss progress with the faculty members Richard Guiler and Dr. John Loth. The sub-groups met to work individually on their own assignment areas outside of this weekly team effort.

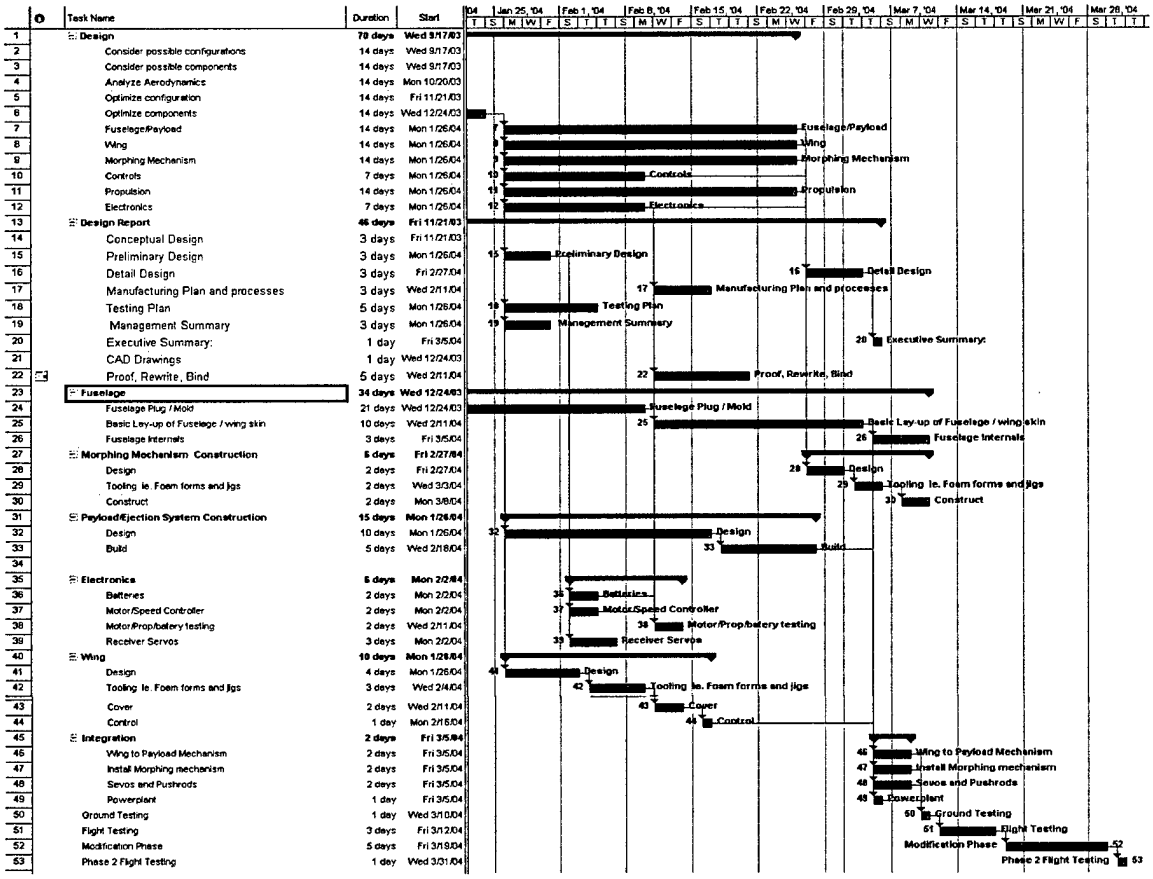


Figure 2.2 – Schedule Breakdown

3 Conceptual Design

3.1 Mission Requirements

There are two missions that the teams may choose from. The first is the "Fire Fight" mission. The second is the "Ferry Mission".

3.1.1 Aircraft Storage, Timed Assembly, Takeoff and Landing

All aircrafts must fit in a 2'x1'x4' box. This is to keep all aircraft about the same size and also for ease in transportation to the competition. The maximum takeoff distance is 150'. The aircraft is required to land on the runway to obtain a score but may roll off afterwards. Airplanes that roll off may be retrieved and carried across the finish line. Aircrafts which are damaged upon landing must take an "incomplete" for that lap.

3.1.2 Fire Fight Mission (FFM)

The firefight mission requires that the plane begin the mission empty. The plane will then be loaded with up to four liters of water from a maximum of four 2-liter soda bottles. After takeoff, the plane will dump the water on the downwind leg of the track and land. The plane will then be refilled and the mission will be flown again. On all laps flown, the plane must complete a 360-turn in the direction opposite of the base and the final turn on the downwind

leg. The water may only be dumped between the upwind and downwind turn marker flags on the downwind leg of the track. The maximum exit hole diameter is ½". The pilot must call "Dump on" and "Dump off" to show control over the dumping on the downwind leg. The score will be obtained by multiplying the weight in pounds of water carried by two, and dividing by the mission flight time.

3.1.3 Mission Penalties

Any early, late, or inadvertent dumping will incur a 3 minute penalty. Excessive spilling during filling will incur a one minute penalty. Gross and/or deliberate water spilling will result in a disqualifying run which counts as a flight attempt. A 3 minute penalty will result if any water is left in the plane upon landing. If the second lap is not attempted or completed, a 3 minute penalty will result.

3.1.4 Ferry Mission

The ferry mission consists of the plane flying four laps and landing. On each lap the plane must complete a 360° –turn in the direction opposite of the base and the final turn on the downwind leg. The mission will be scored by taking the inverse of the mission time.

3.1.5 Aircraft cost-effectiveness

Any early, late, or inadvertent dumping will incur a 3 minute penalty. Excessive spilling during filling will incur a one minute penalty. Gross and/or deliberate water spilling will result in a disqualifying run which counts as a flight attempt. A 3 minute penalty will result if any water is left in the plane upon landing. If the second lap is not attempted or completed, a 3 minute penalty will result.

3.2 Aircraft Configurations Studied

Conventional Tailed

Conventional aircrafts have a trimmed lift to drag ratio, trimmed maximum lift coefficient, trimmed maximum lift coefficient in landing and/or takeoff (with the flaps down).

Canard

Issues related to the use of canards include:

- Reduces subsonic-supersonic aerodynamic center shift.
- Needs to be balanced unstable to take full advantage of performance. (W.H. Mason Aerodynamic Design Options and Requirements) If balanced unstable, the control system design is expensive.
- Drawback: the downwash from the canard unloads wing (for forward swept wing concepts this is good).
- If balanced stable, CL on the canard is much higher than the wing.

• Acceptable high angle of attack lateral/directional characteristics are very hard to obtain.

Tailless

Tailless is also known as the flying wing or blended body configuration. The advantage is that eliminating the surface area of the fuselage and control surfaces reduces the parasite drag, improving the aerodynamic efficiency. The weight of the tail is eliminated. This concept can work well if extreme maneuverability is not needed. The volume distribution can be an issue.

3.3 Numerical Figured of Merit (FOM)

Figures of merit are ways to contrast and compare different design options. After exploring each option, the figures of merit show the best combination for a successful design. In this mission the most important design elements are overall weight which influences rated aircraft cost, mission time and payload deployment. The rated aircraft cost will also be discussed in this section.

FOM 1

Because there were several options when it came to materials, weight and strength were determined to be the most important design factors. Since the team had access to composites, it was decided that carbon, fiber glass, honeycomb, and Kevlar would be used. This would greatly reduce the weight of the aircraft without losing any strength or rigidity. This reduction in weight directly reduces the rated aircraft cost. This also makes the aircraft more streamline. This will help the aerodynamics witch will directly effect mission time. The lower weight also makes it easier to achieve lift off in the allotted 150 ft. This is important because a penalty is accessed to any team not taking off in that distance.

FOM 2

Mission time is important because it is figured into the flight score. It is also important because the rules state that the mission must be completed within 10 minutes. As stated above, the selection of composites will aid in aerodynamics which will reduce drag allowing faster flight time. The reduced weight will also allow for a better lift to drag ratio which will increase flight time. To save on the weight, retractable landing gear was not considered. To reduce the drag created by leaving the landing gear exposed during flight, wheel pants will be installed over the wheels.

FOM 3

The final design parameter is payload deployment. The goal in the design of this system was to keep it as light as possible, while insuring that the entire payload is deployed. This is important because the rules state that any payload that remains in the aircraft will result in a penalty of time added to the flight score. Besides emptying the entire payload from the

aircraft, it is also important to try and hold the maximum payload of four liters. This is because there is also a penalty for dumping less than the maximum payload.

3.3.1 Takeoff Distance

The aircraft must take off within 150 ft or the sortie will be disqualified, there is no advantage for shorter takeoff distances. The takeoff distance is defined as:
Takeoff distance = (0.0 if distance is greater than 150ft) or (1.0 of distance is less than 150ft)

3.3.2 Water Deployment

This year's mission requires the team to design a mechanism to store up to four liters of water and control the dumping of it over a specified length of the course. Bernoulli's Equation, which assumes steady and incompressible flow, was applied to two points along the streamline of water flow to analyze the situation:

$$P_1 + .5\rho V_1^2 + \gamma z_1 = P_2 + .5\rho V_2^2 + \gamma z_2 \qquad (3.1)$$

Point one would be the most elevated water in the water compartment and point two would be the free jet of water exiting the plane. This equation states that the total pressures must remain constant along a streamline. One way to increase exit velocity, V_2, is to increase P_1, the pressure in the water bladder. Another is to increase the height differential (increase the relation of $z_1 - z_2$) between where the water is stored and where it will exit. Pressurizing the water compartment is illegal, so it was decided that the place of exit would be mounted as far below the water in the water compartment as was possible. This strategy to maximize water flow could not be taken to an extreme, though, because by lengthening the pipe the water exits vertically downward, drag would be added to the aircraft. It was decided to let the hose run down behind the landing gear because this will allow the water to flow out slightly faster and would not create much extra drag than what is already produced by the landing gear.

3.3.3 Rated Aircraft Cost (RAC)

Rated aircraft cost is important because the final team score is based on the equation

$$\text{SCORE} = \text{Written Report Score} * \frac{\text{Total Flight Score}}{\text{Rated Aircraft Cost}} \qquad (3.2)$$

Because the rated aircraft cost is used as a denominator it is very important to keep it as low as possible. The actual rated aircraft score is based on the equation

$$RAC = \frac{(A * MEW + B * REP + C * MFHR)}{1000} \qquad (3.3)$$

where A,B and C are constants set fourth in the rules and MEW is manufacturers empty weight, REP is rated engine power and MFHR is manufacturing man hours.

3.4 Mission Modeling and Analysis

Comparisons between different wing lengths and amount of water storage and score all played a large role in determining the best aircraft configuration.

3.4.1 Total Aircraft Weight

Each component of the aircraft was given an approximate weight if the actual value was not known. Since the configuration of this aircraft has no empennage or tail it reduces the weight significantly.

Table 3.4.1 – Weight Breakdown

Component	Weight
Motor	1lb 7.2oz
Propeller	0.6oz
Servos	1.9oz
Landing gear	
Propulsion battery	1lb 7oz
Receiver	1.1oz
Receiver battery	2.9oz
Wing	2.5lbs

The final weight of the aircraft with 1K x 3K carbon is 10lb empty equipped and 18lb with 4 L of water.

3.4.2 Drag

Parasitic - Performance calculations ranging from Vstall to Vmax=200 ft/s

Vmax at 65% Cdpara=2.76552 and Vmax at 100% Cdpara=3.79653

Induced drag

Vmax at 65% Cdi=.001165 Vmax at 100% Cdi=.000617

Table 3.4.2 – Percent Power

	V (ft/s)	Dtotal (lbf)	Cd = Cdi+Cdpara
65% Power	122	2.967	0.0172
100% Power	143	3.943	0.0166

4 Design

Since the beginning of manned flight, engineers, scientists and aviation enthusiasts have tried to improve the efficiency, control and operating range of the aircraft they design. When people look at the efficiency and elegance of bird flight in nature, our aviation achievements seem to pale in comparison. Inspired by a gliding tree seed called Zanonia (Macrocarpa) and the flight of Seagulls, work done by Alexander Lippisch, Reimar and Walter Horten, and John Northrop offered a chance at achieving some of the efficiency of bird flight with swept wing tailless aircraft.

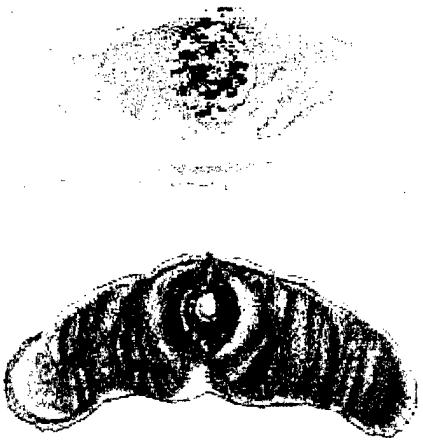


Figure 4.1 – The Zanonian seed that inspired the flying wing

Swept wing tailless aircraft exhibit a highly efficient configuration with low parasitic drag, and up to 30% less total drag. A minimum of two control surfaces is needed for pitch, roll and yaw. Propulsion, power plant, fuel and payload are all built into the wing structure, which allows the design of a very lightweight rigid structure. (Northrop, 1947) Finally the smooth lines with very few protrusions allow the design of an aircraft with a minimum electromagnetic signature, which has become very important to today's military operations.

Unfortunately, there are a number of problems and design compromises inherent in tailless aircraft, which currently do not allow them to exploit many of their potential advantages. These problems and Compromises are as follows:

1) Tailless aircraft have a narrow Center of Gravity (cg) range. (variations from design cg must be trimmed using elevons)
2) Many designs experience adverse yaw problems during maneuvering. Current designs mitigate the adverse problems using a relatively large built in wash out at the wing tips, which causes a drag increase during all flight regimes, but is only needed during maneuvering flight.
3) Susceptible to dangerous tip stalls. (Risk minimized using built in wash out)
4) Extreme elevon throws can initiate separated flow at the wing tips and cause a tip stall. This is made worse with increased trim for cg compensation.

In the late 1950's the Astrophysics Department of Mississippi State University and Dezo George Falvy conducted a thorough aerodynamic study of the Horten IV tailless sailplane. This study concluded that the aircraft could be redesigned to achieve a theoretical Glide ratio of 50:1 and many of the poor handling characteristics could be remedied if there was a way to adjust the washout in flight. (George-Falvy, 1997) Dr. Reimar Horten and Dr. Karl Nickel also came to this same conclusion. (Horten, 1993), (Nickel, 1994) The Hortens and Nickel also modeled elevon control inputs as effective washout. Together with Siegfried Panek, Dr. Horten's last design, the PUL-10 in the late 1980's was a very efficient well handling two place tailless general aviation aircraft, which represents the state of the art, but still needed the design compromises found in earlier aircraft such as a fixed 10 degrees of washout. (Bullard, 2003)

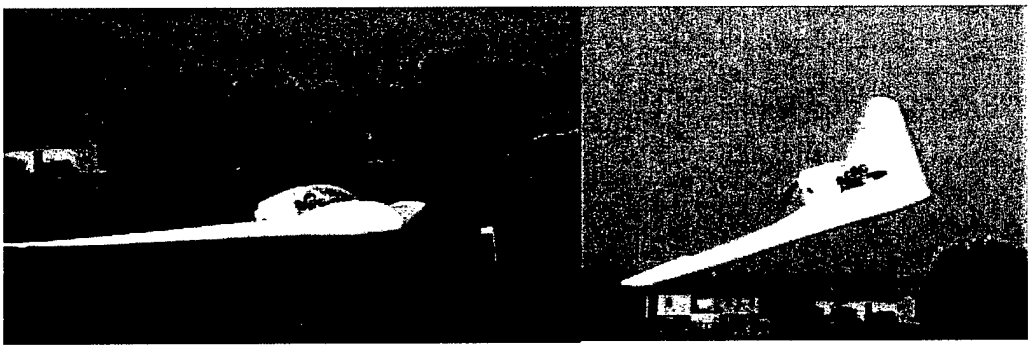


Figure 4.2 – Horten PUL-10 during flight-testing in Germany 1997. (Bullard, 2003)

Advances in composite materials, computational and experimental aerodynamics and actuator technology, allow us today, to design a tailless aircraft with wing morphing. One aspect the wing morphing technology can immediately address is the compromise of fixed washout. Active morphing of the wing, variable washout, allows washout to be used only when needed. This eliminates the corresponding drag for all non-maneuvering flight regimes. Therefore, swept wing tailless aircraft can finally come close to reaching their full potential.

4.1. Aircraft Comparison

This section was used to document features of similar aircraft that may be relevant to the overall configuration selection of the design. This report focuses on the main wing, but information collected in this section will be used in follow-on reports for other configuration items.

To determine the most important factors of the wing design, the AIAA rules for this year's competition were examined. The size restriction was a major factor in determining wing dimensions. The wing must fit in a 4' X 2' X 1' box. This limited the total length of the wing to eight feet. Knowing this, the first objective of the competition is to take off within 150 feet. A high lift design was necessary for this aircraft. Straight and level flight performance was also a desirable characteristic. High wind could possibly cause time to be lost due to perturbed

flight. Poor stability could also cause the aircraft to crash. The aircraft must also be designed to have enough maneuverability to complete a 360° degree turn quickly. Payload release was also a major factor in the wing design. Payload is approximately 1/3 of the entire weight. A rapid payload release will cause the wing to experience a sudden high increase in load. The wing will undergo a major L/D change in a short period of time.

4.1.1 Comparison of Weights, Performance and Geometry

The following table is a collection of weights and geometric parameters of similar aircrafts. No performance data was listed for these particular aircraft. The data found here was used to aid in the design aspects of the proposed aircraft.

Table 4.1.1 – Comparisons of weights and geometrical parameters

Name	Type	W_{TO}	W_E	Span (b)	AR	Airfoil	Source
Syracuse University	High-wing, tail dragger	19 lbs	12.5 lbs	7 ft	7	Clark Y	Page, 2004
UCLA	High-wing, pusher, twin boom	16 lbs	8.5 lbs	12 ft	13	NACA-4415	Page, 2004
WVU	The Right Flyer	18 lbs	10 lbs	8.2 ft	5.83	MH 78	WVU, 2004
Pepe	High- wing, tail dragger	14.2 lbs	7 lbs	8.2 ft	13	SD7003	Page, 2004
Queen's University	High-wing, tail dragger	16.3 lbs	8.8 lbs	10 ft	10	Selig-1210	Page, 2004
Sub-Zero	High-wing, V-tail, tricycle gear	17.3 lbs	9.8 lbs	11.25 ft	10.5	Selig-73014	Page, 2004

4.1.2 Discussion on Other Aircraft

The mission of the 2001 Design, Build Fly competition was to carry as many tennis balls as possible around a course as fast as possible. The Utah State "A" was one of the more radical designs for this competition. The configuration of the Utah State "A" was an elliptic wing and elliptic tail monoplane with retractable landing gear and a horizontal NACA 0015 fuselage. The main wing was designed as an elliptic wing placed in a mid-wing position. The mid-wing location allows the aircraft to maneuver very well. An elliptic plan-form provides the best overall lift distribution. Un-swept, untwisted, elliptic shaped wings produce the least amount of induced drag. This aircraft was designed to takeoff and land in short distances.

The ability to land in a short distance decreased flight time, which increased overall team score.

The Oklahoma State "Orange" (OSU) was a low-wing, v-tail monoplane. Payload capacity was very important during this competition. The need for a large fuselage limited the OSU team to either a low or mid-wing design. The low-wing offers a fair amount of maneuverability compared to the high-wing. Low-wing aircraft also benefit from short takeoff and it also makes for an easy landing. These two aspects were very important for this competition. Landing was a major concern of the OSU team. The v-tail design avoids tail damage during rough landings. A single tractor style propeller decreases flow separation over the fuselage, this increases stall performance. The sandwich fuselage allowed the OSU team to maximize the amount of space available under the DBF weight limit. This type of fuselage also optimized the structural stability of the aircraft.

San Diego State (SDSU) designed an aircraft to have a low-wing, traditional tail, with a lifting body fuselage. The lifting body fuselage allows the entire wing/body section of the aircraft to be used as a lifting surface. SDSU found that 24 softballs was the optimum payload for this competition. A fairly thick fuselage airfoil was needed to carry the heavy payload. The main wing had a rectangular plan-form with tapered wing tips. SDSU attempted to take advantage of the short takeoff and self-landing afforded by a low-wing design. This team was very concerned about landing ability of the aircraft. The landing gear was designed with only two struts and a strut to support the centerline of the aircraft. The tractor style propeller gives this aircraft an advantage during climb. The lifting body design was excellent for stall performance with this propeller location.

The Utah State "Zephyr" (USU) was a low-wing conventional tail aircraft designed to complete two of three mission proposals for the 2003 DBF competition. The two missions were to deploy sensor equipment and missile decoying. The Utah State team attempted to optimize the aerodynamic forces on the aircraft. The strict takeoff and landing requirements most likely affected wing location for this aircraft. Takeoff and landing distance was limited to 120 ft. for this competition. The low-wing location allows decreased takeoff and landing distances. A low wing is a self landing configuration, which was also a major concern for the competition. To decrease the drag effects it was necessary to efficiently retract the landing gear.

The Texas A&M "Flash Flood" (TAMU) was built under very similar constraints as the 2004 aircrafts. The Flash Flood team felt that stability and control were the most important factors for their aircraft. This aircraft was designed to have a low mono-wing, conventional tail, single fuselage, three engine tractor style aircraft. The low wing was used because it provided the easiest, fastest access to the payload. Engines were placed on both wings and nose of the Flash Flood. This team also decided to have one horizontal and one vertical tail

behind the wing. This traditional tail provided the necessary stability and control the aircraft needed. A tricycle landing gear was used to provide steering control without risking ground loops or trip-overs. During this particular competition course selection was not chosen until the day of the competition. An asymmetric design would have made it difficult for the team to perform stability and control calculations with out knowing the course layout.

4.2 Critical Design Parameters

4.2.1 Engineering Requirements

The aircraft design has to conform to basic engineering requirements that ensure safe and controllable flight. Most design requirements have values that must be met for the aircraft to compete. These requirements are shown in Table 4.2.1. The aircraft was designed to meet or exceed any design requirements.

Table 4.2.1 – Engineering Requirements for final aircraft

Engineering Requirements	Required	Goal
Gross Weight (lbf)	< 55	< 20lbs
Takeoff Distance (ft)	< 150	< 150
Mission Time (min)	< 10	< 10
Radio Fail Safe Mode	YES	YES
Maximum Aircraft Water Capacity (L)	4	4
Maximum Motor and Batteries Current Draw (Amps)	40	< 40
Maximum Battery Weight (lbf)	5	< 5lbs
Disassembled Dimensions (ft)	4 X 2 X 1	4 X 2 X 1
Rated Aircraft Cost		< 10
Upright and Inverted Wing Lift Test (g)	2.5	2.5

4.3 Component Selection and System Breakdown

The Right Flyer design can be broken down into several basic systems; propulsion, skin/structure, landing gear, water fill/dump, and control mechanisms. Each will be constructed as an independent module and integrated in final assembly.

Propulsion will be handled by and Astroflight Cobalt 60 direct drive motor spinning a 13x8 two blade propeller. Power is supplied from two 15 cell 1700mA Ni-Cad packs assembled by Diversity Model aircraft. Packs will be wired in series to provide optimal voltage with sufficient capacity. Should the Cobalt 60 prove insufficient it will be possible to install a Cobalt 90; though this would require re-balancing the aircraft due to the sensitivity of blended-wings to cg location.

Rolling hardware is a tricycle arrangement of Robart struts. Main gear will be equipped with breaks to allow shorter taxi rolls reducing time spent on ground. Nose gear will incorporate the release nozzle of the water mechanism. Steering will be controlled with a mechanical mixer running off the wing-tip morphing servos. All gear assemblies have a carbon fiber casing on the strut as well as wheel pants to minimize drag. Retracts were proposed to attain absolute minimum drag but it was felt that they would add too much weight and would simply be another point of possible failure.

Control is through a Futaba seven channel transmitter and receiver combination running two Futaba 5051 servos and two Futaba S3305 servos and an Astro Flight Model 204D electronic speed control. Wing tip and release servos are all high-torque variety due to the large airfoil areas that need to be moved and the stiffness of the release valve.
Water release is the most mechanically complex student-assembled component of the aircraft. Storage is a reinforced Mylar party balloon with a quick fill top opening providing air venting and water loading. Outlet piping exits from the bottom of the balloon through surgical Penrose drain tubing and light weight plastic elbows at all right angel corners. Flow control is a light weight plastic garden hose valve which will be directly operated by a high torque servo. Tubing exits the aircraft through the nose wheel faring and drops the full length. This design creates the maximum fluid head to reduce the total emptying time.

Airframe structure and skin are entirely inseparable in this design. Our carbon/honeycomb skin will provide nearly all structure. Six will be used, one at either side of the center-body and one at either side of the mid-wing segment. The center body ribs will mate with the inner mid-wing ribs to provide solid mounting when assembled. Both outer-most ribs are at the transition between rigid and morphing sections acting a spar guides and servo mounting points. Several other points will have uni-directional carbon reinforcements, and the top of the aircraft from center to outermost rib will have large areas of honeycomb material. It is felt that this configuration will provide more than sufficient strength at minimum weight.

4.3.1 Stability and Control System

Because of the flying wing design, there is no empennage on this aircraft. Therefore no calculations need to be made. The only control surfaces will be on the wing itself.
Geometry: For the flying wing, we followed the design of the Horten Brother's PUL-10. Our aircraft has a wing sweep of 35 degrees, and a wing span of 8 feet. Three different airfoils are used in this design. A 19% MH78 is used at the center of the flying wing. Then at 1/3 of the way down the wing a 14% MH78 is used. Finally, the tips are fashioned after the NACA-0010

Control Style: On a normal flying wing, elevators and ailerons are replaced with elevons. The elevons both turn and pitch the aircraft. However, our design utilizes the concept of wing

morphing to eliminate the usage of elevons. The term morphing consists of changing the washout (variable washout) at the tips of the aircraft. By morphing the latter 33% of the wing, we are able to control the aircraft with better aerodynamic performance. The drag on the aircraft will theoretically be lower due to the absence of deflecting elevons. This allows the air to flow over the wing without any obstruction. A half scale glider was constructed to test the preliminary design of a trimmed 10 degrees washout. This showed to be a tad to much. Once the full scale glider was constructed, a washout angle of 5 degrees was chosen to be the trim factor. After the glider was balanced, the 5 degrees of trim proved to be sufficient. The glider also showed that after finding the correct cg and cp, once the aircraft stalled the nose drops down and allows the aircraft to regain control rather than going into a spin. This is optimal for any aircraft.

Control Hinging: Because of the variable washout, only two servos will be needed to control the aircraft. They will be placed near the leading edge and will turn a control rod that ultimately pivots the tips of the wing on its leading edge. This will morph the wing and allow for control.

Stability: a flying wing relies on it instability in order to maneuver. This was observed by Lippish, Northrop and the Horten Brothers. If vertical stabilizers were added to the flying wing, the maneuverability would decrease making it harder to fly the objectives. Therefore no verticals have been put into the design of this aircraft.

Turning: Our design was taken mainly from the PUL 10 except for the wing morphing. The PUL 10 used control surfaces to turn. Where as with our wing morphing we will use our entire wing tips to control the aircraft. We will use more relative area while also reducing our RAC for the competition.

Descent and landing: To land and stop we will be using a combination of flaps and brakes. Since a flying will gets high ground affect lift while landing it will be a challenge to actually land, so a flap system will create enough drag to land and wheel brakes will stop the aircraft.

4.3.2 Propulsion System

The program P-Calc available at http://brantuas.com/ezcalc/dma.asp was validated using experimental data from the Astro Flight web site www.astroflight.com. On this site, experimental data was given for several different propulsion combinations. After that data was obtained, the same combinations were run through P-Calc to see how accurate the program was. After running the different combinations, it was found that P-Calc was accurate within 25%. This seams high however all of the numbers predicted by P-Calc were over predicted. With this information, it was determined that as long as the numbers given by P-Calc were under the maximum 40 amps, the combination would be safe to use.

Table 4.3.2.1 – Experimental Data vs. P-Calc

Prop	Amps Exp.	Amps P-Calc	Watts Exp.	Watts P-calc	RPM Exp.	RPM P-calc	Thrust Exp.	Thrust P-Calc	% Diff. Amps	% Diff. Watts	% Diff. RPM	% Di Thr
12x6	20	23.876	660	642.9	10300	10535.3	13.75	12.5	19.38	2.6678	2.2844	1
12x8	25	28.477	800	745.6	10000	10057.9	16.67	14.498	13.91	7.2933	0.5793	14.
14x7	30	37.3	950	907.2	9700	9142.93	19.79	17.64	24.33	4.7155	5.743	12

The goal in designing the propulsion system was to create the lightest system that would provide enough power to complete all of the mission requirements. To aid in the motor, battery, propeller combination, the internet program P-calc was used. After running several combinations, it was determined that an Astro Flight 660 series motor with 30 battery cells and a 13 X 8 propeller was the most efficient combination. This setup will provide enough thrust for lift off, draws just under 40 amps and is relatively light. Once this combination was determined from P-calc, actual motor testing was done. To determine the RPM's an RPM gauge was used. While doing the flight testing and amperage meter was used to check the amperage that was being drawn during flight. After verifying the accuracy of the Propulsion numbers through P-calc and actual testing, it was determined that the above combination is sufficient.

Table 4.3.2.2 – P-Calc results for Actual Propulsion System

motor	Batteries Nicads	Prop	Amps P-Calc	Watts P-calc	RPM P-calc	Thrust P-Calc
Cobalt 60 direct	30	13x8	34.188	855.6	9465	16.64

4.3.2.1 Motor Size and Number

The main criteria used in selecting the motor for the preliminary design, was take off distance, weight, amps drawn, and RAC. Because of weight considerations, there was never

an option to use more than one motor. Since the number of motors was set at one, it was necessary to find a single motor that could produce enough thrust to allow take off within the designated 150 ft of runway, but would also draw 40 or less amps which is the maximum amperage allowed within the rules.

4.3.2.2 Propeller Pitch and Diameter

The main goal in selecting the propeller was efficiency. Since larger propellers are more efficient, a large propeller was considered. However, because of the low speeds at which this aircraft will travel, it was determined that a smaller propeller should be used due to the fact that they have smaller pitch to diameter ratios which allow for better efficiency at flight speed. Besides that, ground clearance also had to be considered when selecting the propeller for the aircraft. A reduction in propeller diameter saves weight in the propeller as well as with landing gear because the gear can be shorter.

4.3.2.3 Battery Weight and Number and Cells

The number of batteries for this aircraft was a major design factor. Because of the low weight that will be accomplished by using composite materials, the batteries were one of the heaviest components that could be varied. To make a final determination on the number of cells, the total battery power for the mission was looked at. The goal was to have just enough power to finish the mission. Of course, a factor of safety has to be introduced. Because of natural occurrences such as wind and rain, it is not good enough to assume laboratory conditions.

4.3.3 Load Factor Limit

Because the course for the competition requires the aircraft to make turns from both the front to back leg of the course and vise versa, as well as complete a 360 degree turn on the downward end of the course, the load factor had to be determined. The load factor for this aircraft is above the required 2.5 g's listed in the competition rules.

4.4 Landing Gear

There were many factors involved in considering the landing gear for the aircraft; such as weight, drag, stability, and position. A retractable system was considered, but decided against it due to the weight. The landing gear we chose is manufactured by Robart. The landing gear being used is 8 inches long and is rated for 18 pounds. The landing gear is going to have a 3in. wheel on the front and a 2.5in wheel on both of the rear gear assemblies. The three arms are of the landing gear have a shock on them; this is going to help in controlling the aircraft on the runway, before takeoff and after landing. The arms for the landing gear are going to be connected to the aircraft via an aluminum connection.

4.5 Payload Support and Deployment System

This year's mission requires the team to design a mechanism to store up to four liters of water and control the dumping of it over a specified length of the course. The water is stored in a foil bladder made from a balloon which holds approximately 3 liters. A PVC pipe runs from the bladder to the outside of the plane. Penrose tubing connects this pipe to a valve. A servo is mounted directly to this valve which will control the flow of water from the bladder. The water then continues from the valve through another piece of Penrose tubing and exits behind the landing gear. By mounting the end of the tube well below the bladder, a greater pressure difference is developed and the water will drain faster. The bladder will be mounted directly above the center of gravity of the plane. This will insure no change in moments about the center of gravity during dumping. The bladder storage system has three main advantages. It has a very small height and can conform to the inside of the plane. This is essential for the flying wing since the shape of the plane cannot be changed to accommodate the payload. Since the bladder is flexible it will conform to the water which will help decrease the sloshing of the water. The bladder will also be mounted on top of the battery pack which will help cool them. A cowl will be placed over the entire outside assembly in order to improve the aerodynamic properties of the plane.

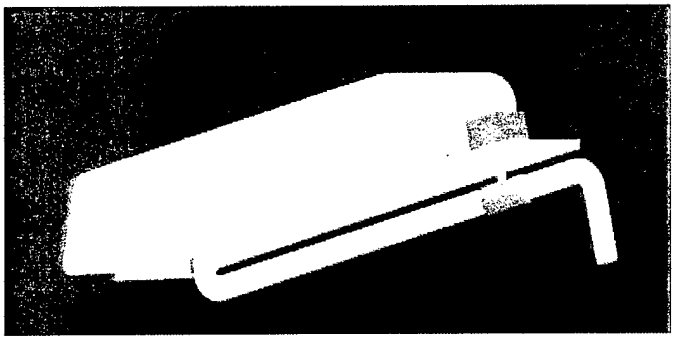


Figure 4.5 – Payload Deployment System

4.6 Aircraft Characteristic Input

Table 4.6 – Areas of Main Components

Wing Area Sw=	Sw=	9.752	ft^2
Fuselage frontal area Sf=	Sf=	0.01	ft^2
Frontal area of three tires or Stires=	Stires=	0.03125	ft^2
Frontal area of landing gear structure	Sstrut=	0.038194444	ft^2
Gross weight W=lift L in level flight= (Loaded)	L=W=	20	lbf
WING LOADING		32.81378179	oz/ft^2

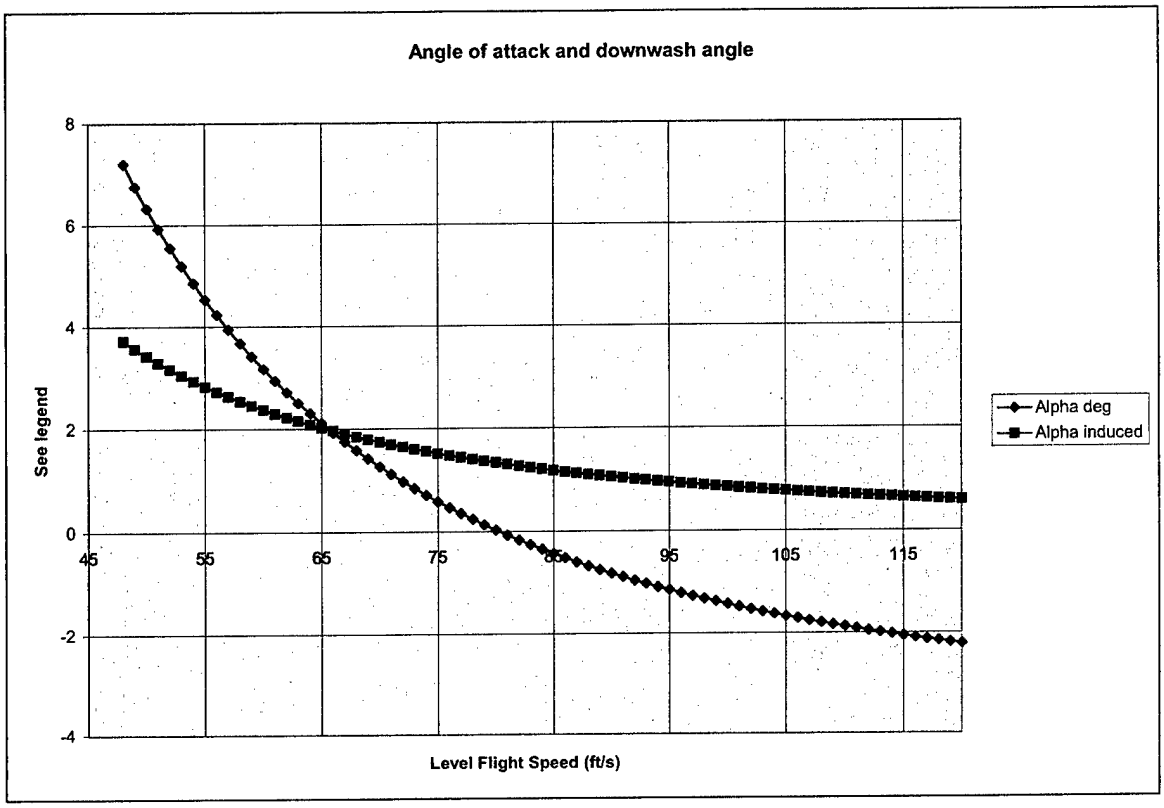


Figure 4.6 – Angle of Attack and Downwash Angle

4.7 Final Aircraft Performance Analysis

Table 4.7.1 – Stall Characteristics

qstall =W/(CLstall*Sw)=	qstall(ft^3/s)=	2.838562
Vstall=SQRT(2*qstall/rho).	Vstall(ft/s)=	48.87082

Table 4.7.2 – Calculations for Takeoff Distance

Vrotation=Vstall*1.20	Vrotation=Vstall*1.20 (ft/s)	58.6449849	m/s	17.87499
Rolling Resistance Force measured (Uk=.2) (lbf)	Ffr=Fn*Uk (lbf)	2.77	N	12.32096
Ground Acceleration	(Tstatic-Fdrag)/m=a (ft/s2)	14.4578	m/s2	4.406737
Time to Vrotation	t=(Vrotation-Vo)/a (sec)	4.056286911	sec	4.056287
Takeoff Distance	x=1/2*a*t^2 (ft)	118.9404423	m	36.25305

4.8 Predicted Performance

The aircrafts performance numbers are listed in the chart below. This chart shows the current year's aircraft compared to past years aircraft of 2002; which placed third, and also Utah States "Zephyr" Aircraft. Looking at the RAC, this design has a good head start on both

of the other planes that were competitive. If everything goes according to plan the RAC is going to give this aircraft a great advantage in the competition.

Table 4.8 – Rated Aircraft Cost

Aircraft Performance		Right Flyer	Zephyr
Takeoff	Distance (ft)	118.94	115.8
	Time (s)	4.1	4.7
	Acceleration (ft/s^2)	14.5	N/A
Flight speed	Stall (ft/s)	48.9	36.3
	Cruise (ft/s)	122	108.5
	Max (ft/s)	143	N/A
L/D	Cruise	6.7	4.4
	Max	5.1	18.1
Rated Aircraft Cost		9.428	13.338

5 Manufacturing Plan

When designing an aircraft from scratch within a specified time frame nothing is more important than the manufacturing plan and process. The manufacturing plan outlines all of the possible ways of construction and chooses the best one depending on some analytical characteristics. Then, a manufacturing plan sets a schedule by which everything needs to be completed, in order to keep the construction moving. Without a schedule certain parts of the aircraft might be waiting on other parts that were completed late and thus holding up construction.

To construct the aircraft following the design requirements the best combination of manufacturing processes and materials needed to be used. The following processes were investigated, and the best one was chosen.

5.1 Manufacturing Process Investigated

First it was decided that the aircraft needed to be developed in multiple pieces. Different from a conventional aircraft where the tail wing and fuselage could be created separately, the wing needed to be completed in segments. The construction segments were decided by first how it could be cut out of foam and then what would fit into the box and finally what material would be covering the segment. The following discusses options for constructing the center section, mid-wing, and outer-wing.

5.1.1 Center Section

The center section of the aircraft was designed to hold most of the internal components, such as, the motor, batteries, water dumping system and controls. Since it is so thick it needs to be very strong. Balsa was not even considered for his section, rather it was between composites. The possible choices were fiberglass, carbon fiber, and kevlar. Kevlar being the lightest but most expensive was used over areas that did not need extreme strength, the carbon fiber was used over the structural components and on the leading edge since it was more available than Kevlar and somewhat stronger. Since the concern was on strength for the center section it was laid up with a honeycomb center to make it rigid. Figures below show the lay-up for the skin components of each section.

5.1.2 Mid-wing

The construction of the mid wing is very similar to the center section, in that they both need to be very structurally solid and strong. The mid-wing is still a shell with carbon fiber at the leading edge and on the weight bearing sections and Kevlar everywhere else. The wing servos are mounted in the mid-wing section and will attach to the outer wing through a common wing.

5.1.3 Outer-wing

The outer wing is responsible for the control of the aircraft. The skin of the outer wing will be some what flexible but will need to hold shape while warping. It will be 2 ply 3k carbon with a center rib to help keep the shape. The carbon will flex while still keeping the shape of the airfoil on the outer wing section.

5.1.4 Wing Characteristics

When determining the wing characteristics, the major goal was to design the lightest wing possible that would allow the aircraft to complete the flight missions. After several iterations it was determined that the following geometry provided the best performance. This iteration can be seen on the following page.

Table 5.1.4 – Geometric Wing Characteristics

span b=	b=	7.54	ft
Aspect ratio AR=b^2/Sw=	AR=	5.8297375	
Average chord c=Sw/b=	Cavg=	1.2933687	ft
2D lift curve slope Clalpha per degree=	Clalpha/degree=	0.1	per degree
Alpha0 at zero lift in degrees	alpha0=	-4	degree
3D slope CLalpha=Clalpha/(1+57.3*Clalpha/(pi*e*AR))	CLalpha3D=	0.0668177	
CLstall	Clstall=	0.85	
Drag coefficient of wing airfoil CDo=0.008+0.002*Cl(.6)^2	Cdo=	0.015	
Drag coefficient of vertical tail Cdv=	Cdv=	0.001	
Drag coefficient of fuselage Cdf=	Cdf=	1	
drag coefficient of horizontal tail Cdh=	Cdh=	0.001	
Drag coefficient of tires Cdt=	Cdt=	1.2	
Drag coefficient of strut Cds	Cds=	0.8	

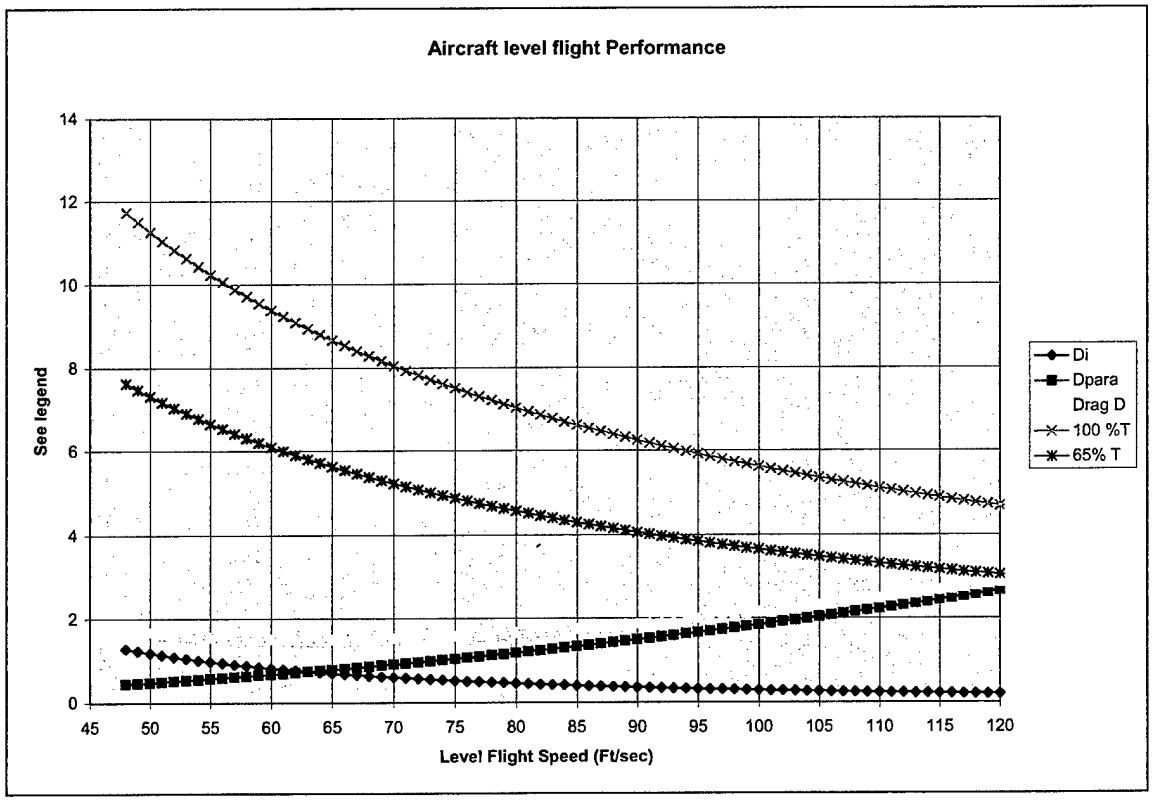


Figure 5.1.4 – Aircraft Level Flight Performance

5.2 Analytical Methods Used

All possible avenues for construction must be compared using analytical methods. To decide on what would be the best combination of process and material would be each FOM was given a rating depending on importance then with those ratings each material and process is also rated. A score for team required skill level was determined in which shows all possible skills needed and how many people on the team can do them.

5.3 Selected Processes and materials

After following the above procedure optimum processes and materials were chosen for each section of the aircraft.

Shell: It was decided to lay up the entire shell of our aircraft in one step. To do this there is a lot of preparation needed to get the lay up correct, on the first try since mistakes would be devastating in this step. The skin over the center section and mid-wing will be two layers of 3K carbon composite sandwiching a layer of quarter inch honeycomb. The skin covering the outer-wing will be two layers of 3K carbon without the honeycomb to 2/3 span. From that point the material changes to 1K carbon to allow flexibility for the wing morphing mechanism. There will be a carbon rod that extends the entire front edge and will provide a large portion of the strength as well as help keep the joints together.

Creating the Mold: The first step to creating a composite shell is to make a mold with male and female parts. We started out with a 9'x 2'x 5' block of foam. First we sketched out a rough 2-D version of the aircraft at center. Then we trimmed off the excess foam from the block so it was easier to work with. Next we cut the block into sections where the taper ratio changed. While the sections were being prepped the airfoil templates were being cut out of thin aluminum. Airfoil templates needed for foam cutting are an upper edge and a lower edge with a lip on both sides to guide the hot wire. The templates were then attached to the foam block sections and using a hot wire we cut out our mold. Finally the engine cowl could be cut out of the top female center section. Using a template cut out of aluminum and a hot wire the engine cowl was removed from the female mold and will be attached to the male mold.

Prepping the Mold: Since foam is very porous it needed addition prepping before any composite could be laid up in it, otherwise the final product wouldn't be smooth and would stick to the mold. To prep the mold the first step is to make each female section a smooth surface; this was accomplished by sanding it and covering it with packing tape. Then all of the female sections are joined together. Next the male mold is prepped with a sketch of the lay up and indented placement of the hatches and honeycomb areas. A detailed sketch of the lay up and a side view of the skin can be found below. Immediately prior the lay up all molds will be coated with PPT, a mold-release compound. Finally a piece of Mylar long

enough to extrude out of the mold is inserted into the trailing edge of the male mold. This will allow the male mold to be removed from the skin after it has cured.

Laying up the Shell: When the mold is completely prepped the lay up can begin. First all pieces of composite fabric needed are laid out on a flat surface on top of plastic. Resin is then poured on top and worked into the fabric with brushes or spackle knives. Once the fabric is completely covered another sheet of plastic is laid on top. Now all of the air and excess resin needs to be forced out of the saturated fabric. This can be done with rollers or spackle knives. In industry the ideal lay up is 80% fabric, 20% resin; the best hand lay up possible is about 64% fabric, 36% resin. After the fabric is compressed between two sheets of plastic, it can be cut out in to the desired shapes needed for the design. The cut pieces are then placed into the mold and smoothed out. The mold is then placed inside a vacuum bag and all of the air is pumped out to allow atmospheric pressure to compress the mold. The mold is then placed inside a heated room an allowed to cure for at least 12 hours.

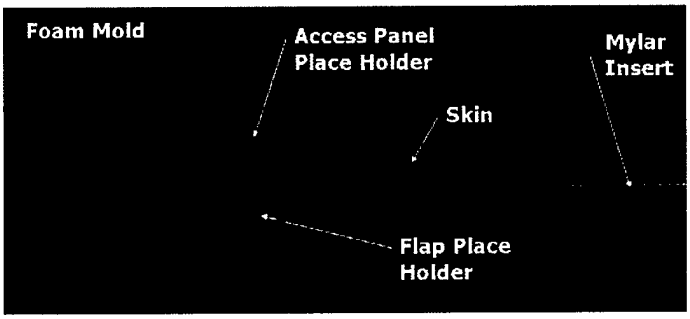


Figure 5.3 – Shell Lay up Cross Section

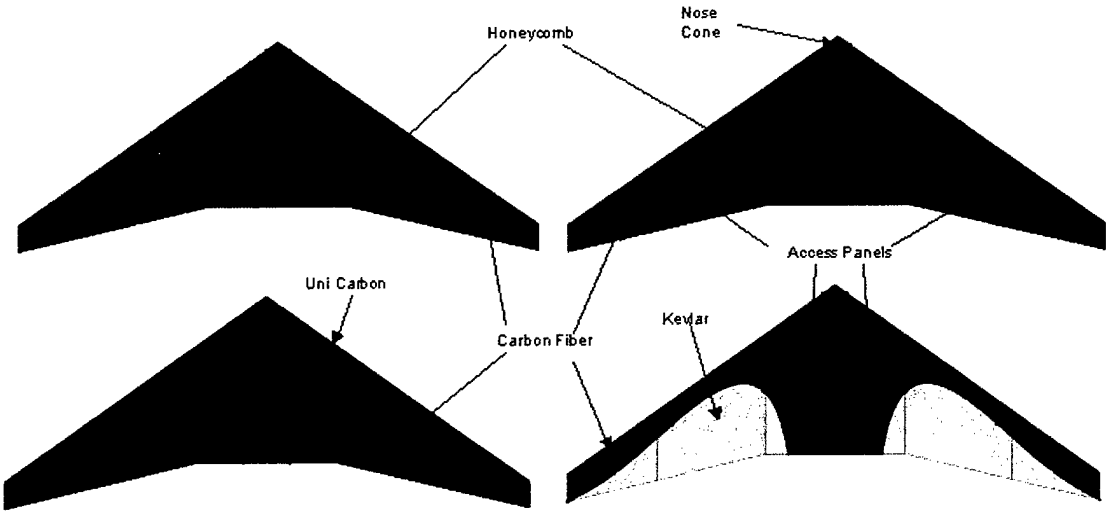


Figure 5.4 – Shell Lay up

Laying up the Ribs: The only place ribs will be used is at the joints where the wings come apart and at the start of the wing morphing section. They will be laid up with two layers of carbon sandwiching honeycomb and plywood inlays. The carbon/honeycomb will give them strength and the plywood inlay will give us a place to run connectors, wires, and to attach servos. The ribs will be laid out similarly to the skin except on a flat surface. The ply wood will take the place of the honeycomb where needed and the whole sheet will be vacuum bagged. Once it has cured the ribs will be cut out.

Internal Construction: Once all of the components are ready to be put into the aircraft the interior of the aircraft must be prepped with a coating of ----- that will allow adhesives to be used on the composite. When the inside of the shell is ready the ribs are glued into place and all the internal components are attached.

5.4 Manufacturing Milestones

5.4.1 Critical Path

Once a process and materials are chosen a schedule was made to keep the entire task on track and ensure there are no delays. The critical path is made from dependencies of a process on another process. With this schedule the team will be able to overlap processes that don't need one another therefore completing the project much faster and more efficient.

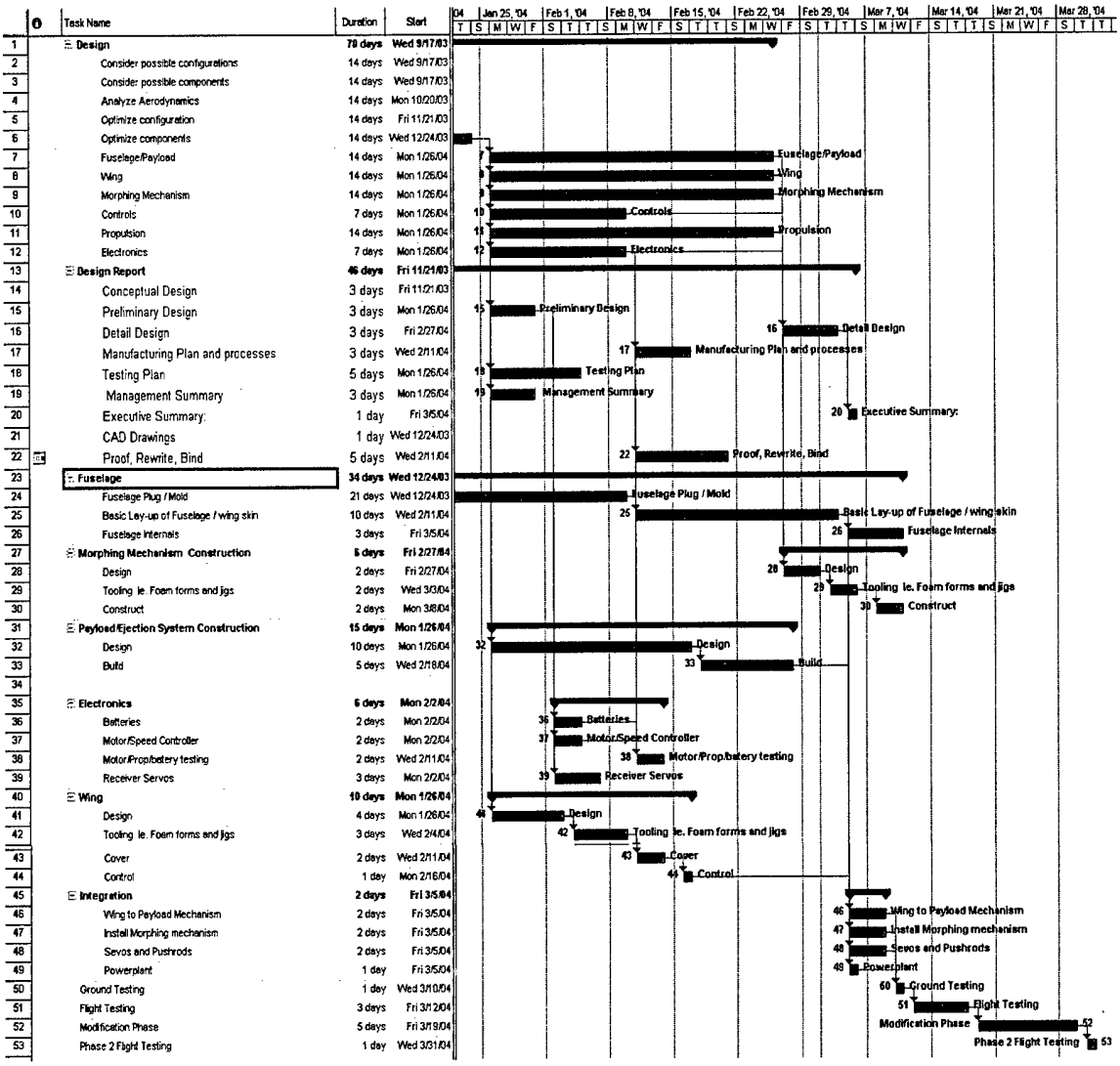


Figure 5.5 – Breakdown of Manufacturing Schedule

6 Testing Plan

Testing began with ½ scale foam core glider models to run wind-tunnel testing and drag coefficients as well as basic glide and stall characteristics. It may be possible to experiment with a micro-servo radio system mounted in one of these gliders so that the pilot can get some feeling for control response prior to the first flight of the actual UAV. Modifications for this step would be relatively easy if elevons were used rather than the full wing-morphing control proposed for the final aircraft.

Flight testing on the actual craft will be conducted in several stages. First, a completely stripped flight: without breaks, water delivery system, water tanks, and flaps. This first flight will be using a ½ scale flight demonstrator as a trainer for the pilot. As the pilot becomes more comfortable with the handling of the craft, and control problems diminish, the remaining equipment will be installed in the aircraft.

Once flights have been made with breaks, flaps, and the water delivery system loaded flights will begin. In a slightly counter-intuitive manner, we will begin with the tanks full so that there are no concerns with loss of stability due to movement of water inside the aircraft. Again, the pilot will be allowed time to become comfortable with the new handling characteristics. To prepare for in-flight dumping several flights will also be made with tanks only partially full so that the pilot experiences the most un-stable conditions the aircraft will encounter in competition.

Finally the vertical stabilizers will be progressively scaled down with continual trial flight to be sure that they are not necessary for reliable flight. At this point flight testing will be complete and competition practice can begin in earnest, along with any mechanical optimizations which may occur to the team.

6.1 Test objectives and Schedules

The objectives of testing were to find the correct placement of the CG, the stability and maneuverability of the aircraft and drag coefficients as well as basic glide and stall characteristics.

Table 6.1 – Testing Schedule for Critical Components

Test #	Test Subject	Results	Date
2004-101	Motor/Battery	Found the life of the battery necessary for each mission	11/(15-20)/2
2004-201	Warping Servo	Tested the torque of the servo	1/(5-7)/2004
2004-202	Valve servo	testing the torque of the servo	1/(5-7)/2004
2004-301	Nose Cone Impact	testing the strength and durability of the nose cone	2/(20-21)/2004
2004-401	Payload Release	Finding the total time needed to release the water	2/(10-20)/2004
2004-501	1/2 Scale Model	Initial flight tests	11/(20-23)/200
2004-502	Full Scale Model	Flight test to find the final CG placement	2/(1-5)/2004

7 Process of Creating the Mold

Before manufacturing the large scale aircraft, A ½ scale version was created to check stability and cg location.

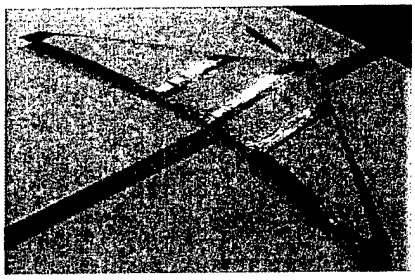


Figure 7.1 – ½ Scale Prototype

After the small model proved that the design was stable, large scale templates were created from aluminum.

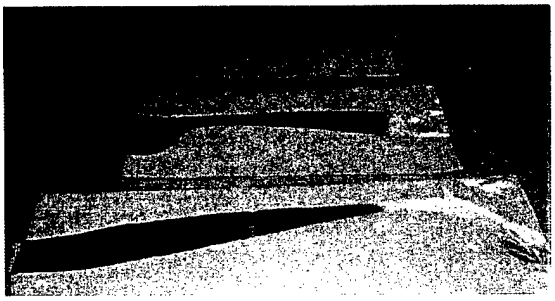


Figure 7.2 – Aluminum Templates

Once the templates were cut out, they were attached to the foam as seen below.

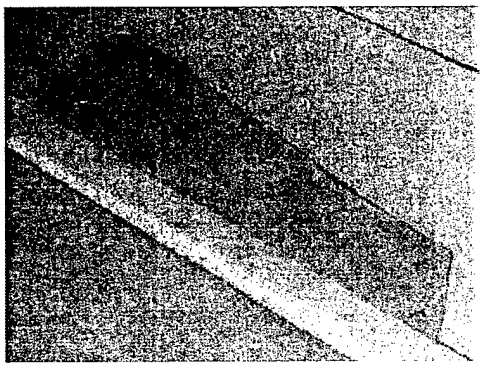


Figure 7.3 – Template Attachment

Once the templates were attached to the foam, a hot wire was used to cut out the mold.

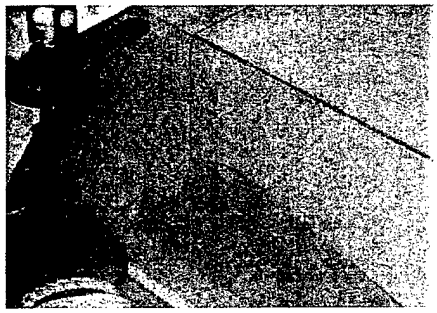


Figure 7.4 – Hotwire Cutting Mold

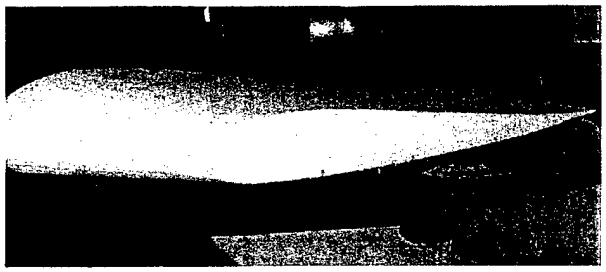


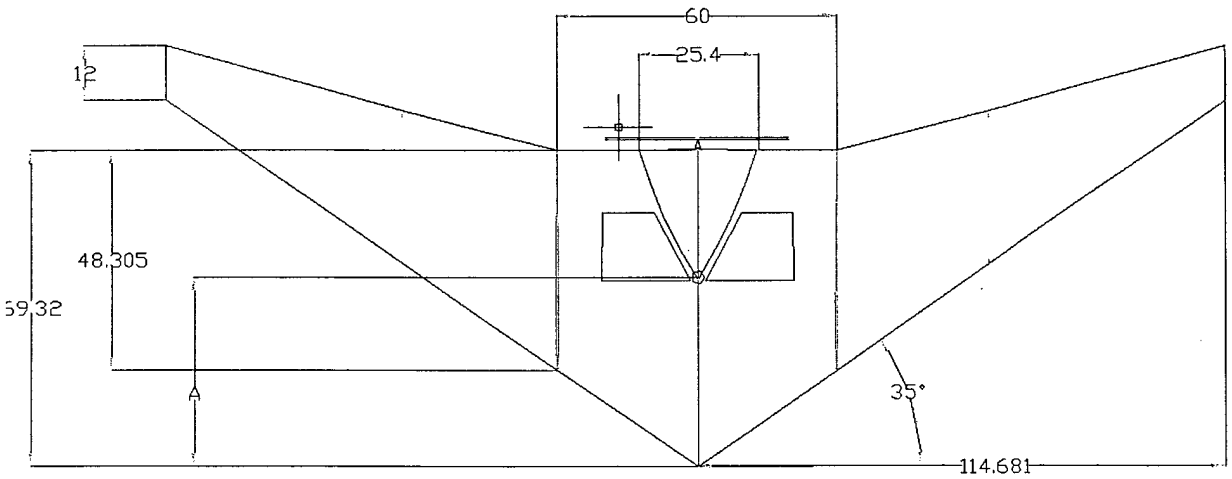
Figure 7.5 – Finished Mold After Cutting is Completed.

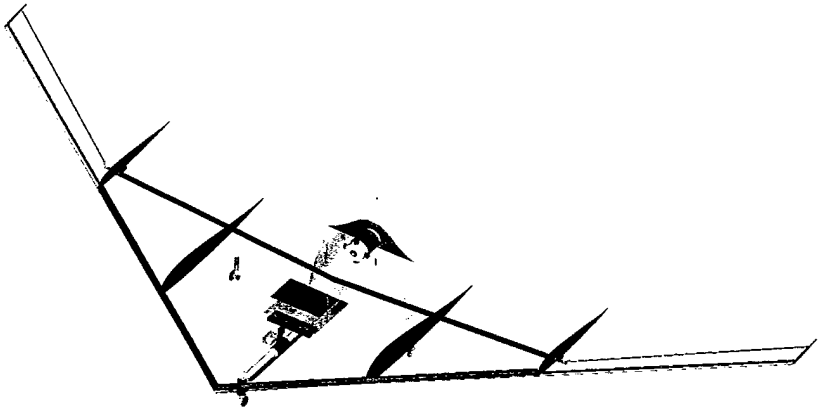


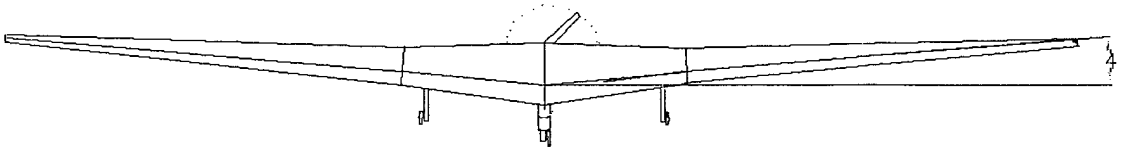
Figure 7.6 – Complete Mold

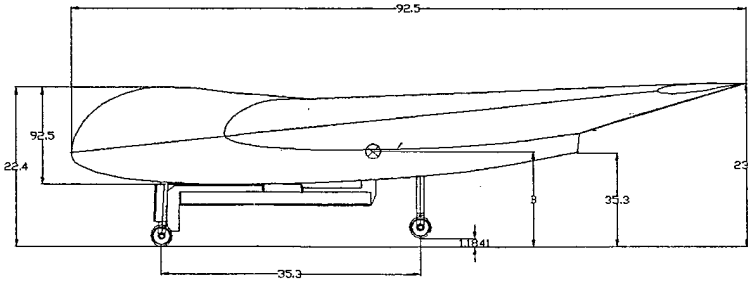
8 Engineering Drawings

	DBF100-01	1	





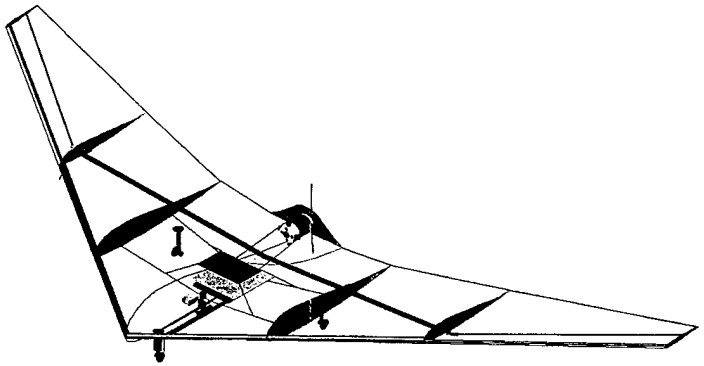


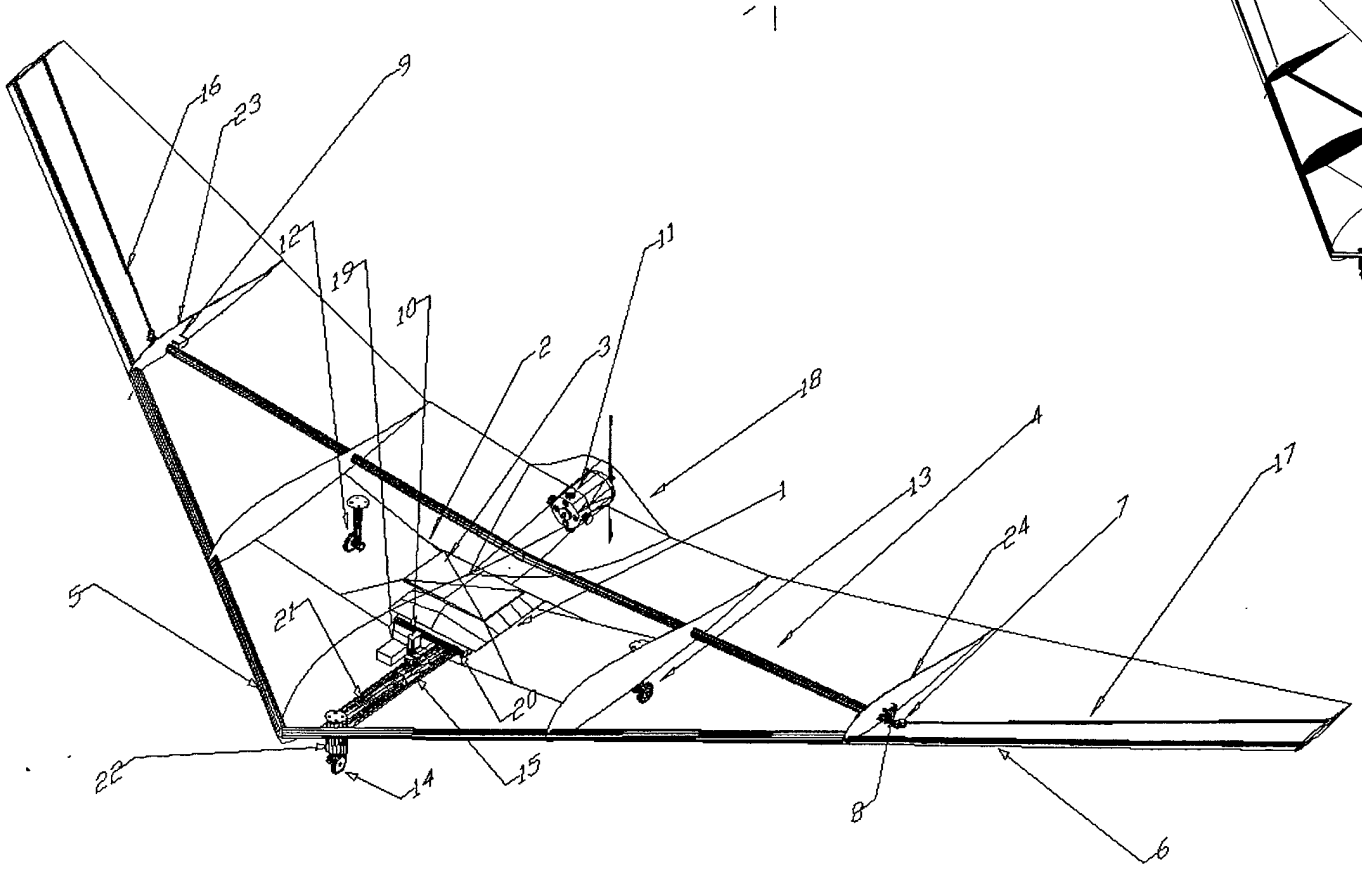


Notes:
"A" and "B" is the location of the center of gravity
A=41.36cm
B=13.4cm

	1	1	DBF100-02	Internal Assembly	
			Parts List		
			Not To Scale	2004 DBF West Virginia University	
Wright Flyer			Measurements		
Project			Cm	3-View	
				DBF100-01	

--- DBF100-02	1	--

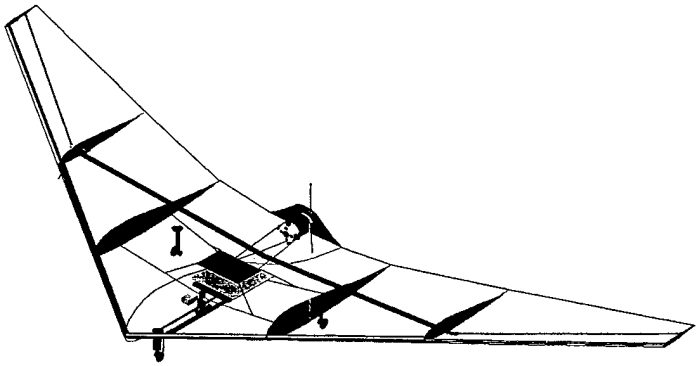


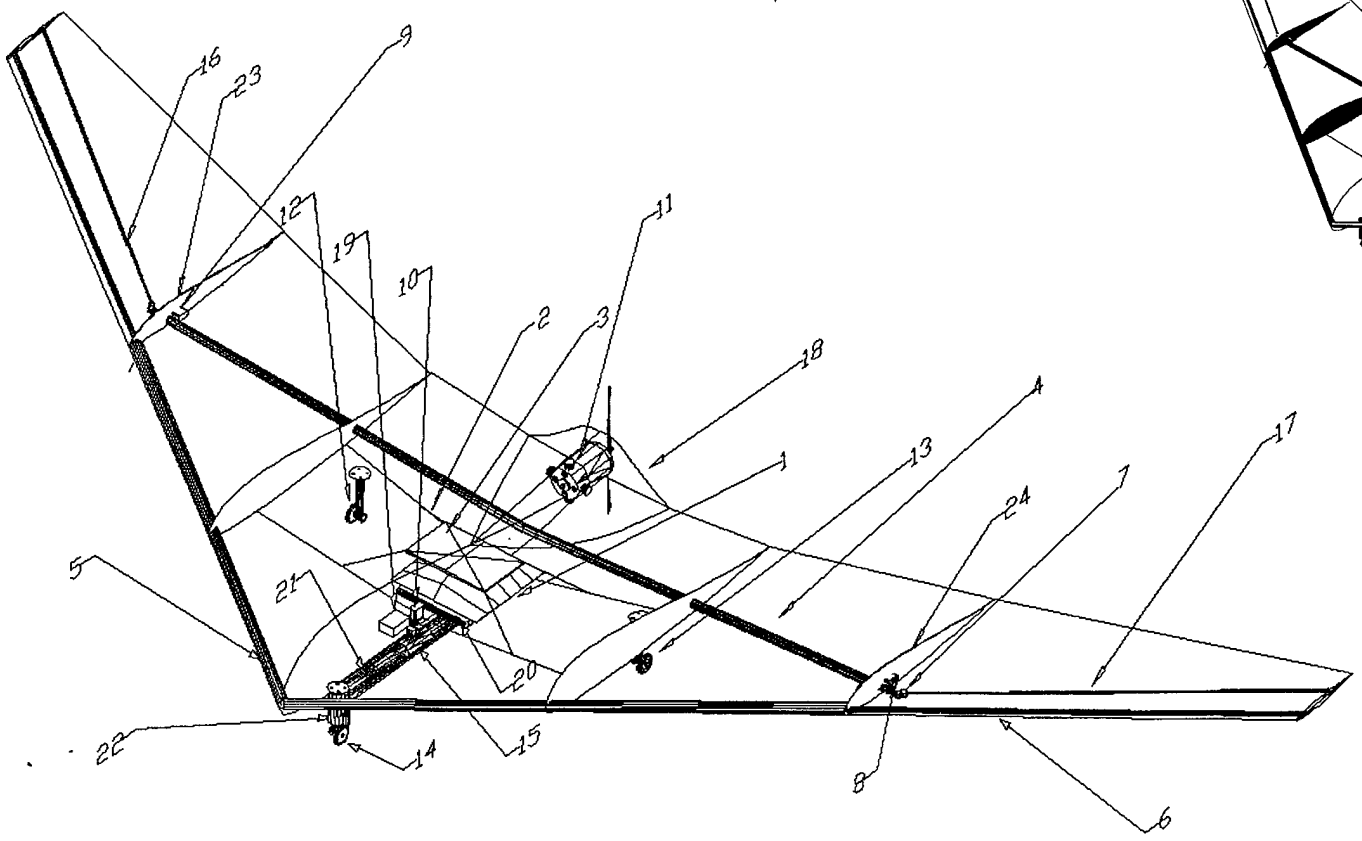


	26	1	DBF300-03	Left Joiner Rib	
	25	1	DBF300-03	Right Joiner Rib	
	24	1	DBF300-02	Left Transition Rib	
	23	1	DBF300-02	Right Transition Rib	
	22	3	DBF500-04	Landing Gear Furring	
	21	1	DBF200-03	Piping (D .5in)	
	20	2	DBF400-02	Battery Pack	Sanyo
	19	1	DBF700-01	Receiver	
	18	1	DBF400-02	Motor Mounting Plate	
	17	1		Left Control Arm	
	16	1		Right Control Arm	
	15	1	DBF200-02	Ball Valve	
	14	1	DBF500-01	Nose Landing Gear	Robart
	13	1	DBF500-02	Left Landing Gear	Robart
	12	1	DBF500-03	Right Landing Gear	Robart
	11	1	DBF400-01	Motor	Astroflight
	10	1	DBF200-04	Valve Servo	Futaba
	9	1	DBF300-01	Right Warping Servo	Futaba
	8	1	DBF300-01	Left Warping Servo	Futaba
	7	2	DBF600-01	Warping Mechanism	
	6	2		Outer Leading Edge Spar	
	5	2		Center Leading Edge Spar	
	4	1		Lateral Spar	
	3	1	DBF200-05	Attachment Plate	
	2	1		Payload Restraints	
	1	1	DBF200-01	Payload Assembly	
				Parts List	

			Not To Scale	2004 DBF West Virginia University
Wright Flyer				
Project				Assembly

--- DBF100-02	1	--



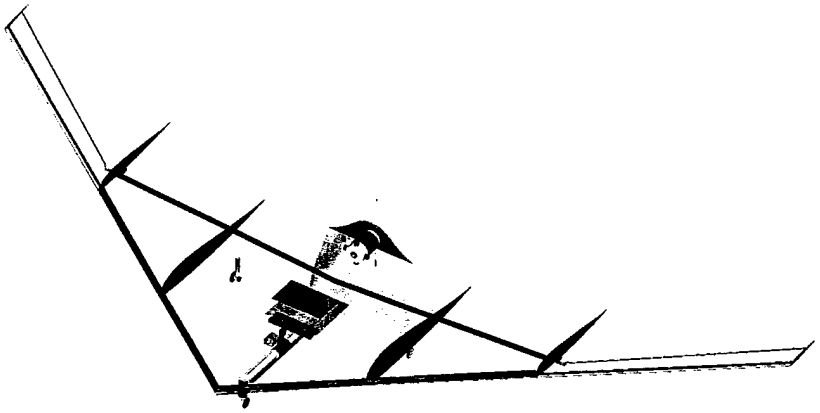


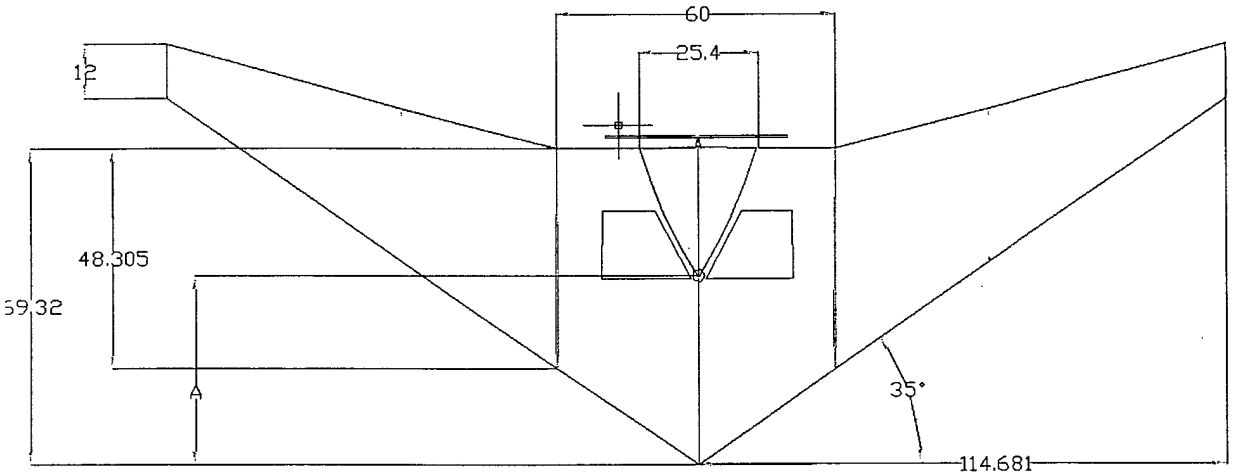
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24	1	DBF300-02	Left Transition Rib	
23	1	DBF300-02	Right Transition Rib	
22	3	DBF500-04	Landing Gear Furring	
21	1	DBF200-03	Piping (D .5in)	
20	2	DBF400-02	Battery Pack	Sanyo
19	1	DBF700-01	Receiver	
18	1	DBF400-02	Motor Mounting Plate	
17	1		Left Control Arm	
16	1		Right Control Arm	
15	1	DBF200-02	Ball Valve	
14	1	DBF500-01	Nose Landing Gear	Robart
13	1	DBF500-02	Left Landing Gear	Robart
12	1	DBF500-03	Right Landing Gear	Robart
11	1	DBF400-01	Motor	Astroflight
10	1	DBF200-04	Valve Servo	Futaba
9	1	DBF300-01	Right Warping Servo	Futaba
8	1	DBF300-01	Left Warping Servo	Futaba
7	2	DBF600-01	Warping Mechanism	
6	2		Outer Leading Edge Spar	
5	2		Center Leading Edge Spar	
4	1		Lateral Spar	
3	1	DBF200-05	Attachment Plate	
2	1		Payload Restraints	
1	1	DBF200-01	Payload Assembly	

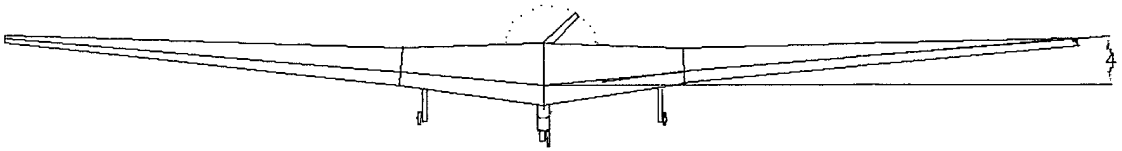
Parts List

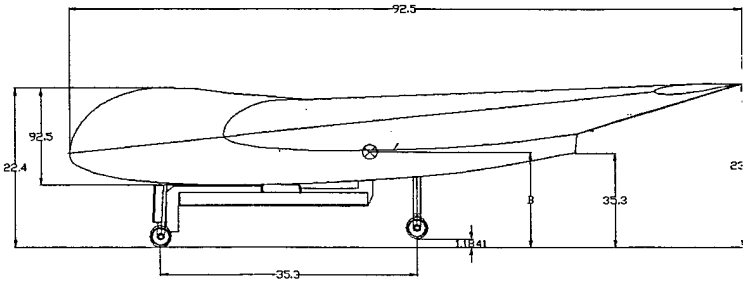
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Wright Flyer				
Project				Assembly
				DBF100-02

DBF100-01	1	





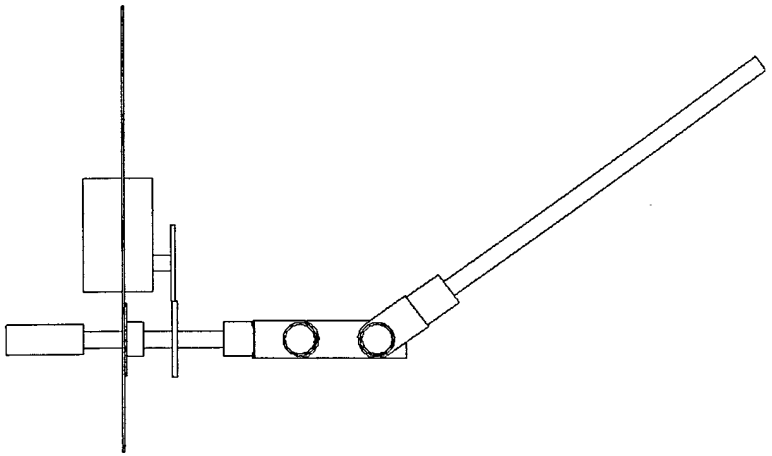


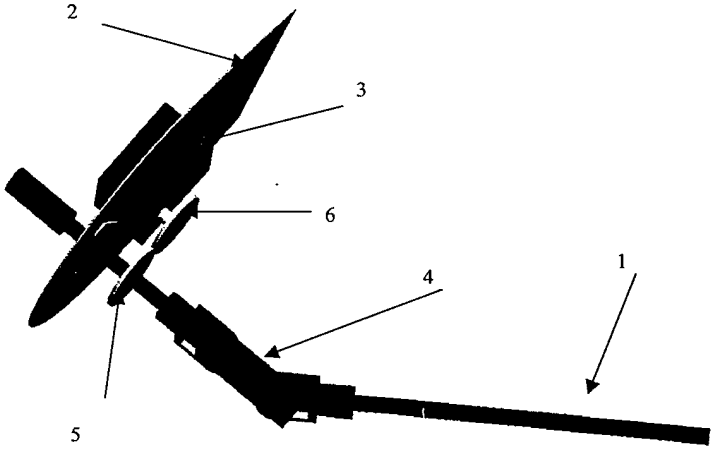


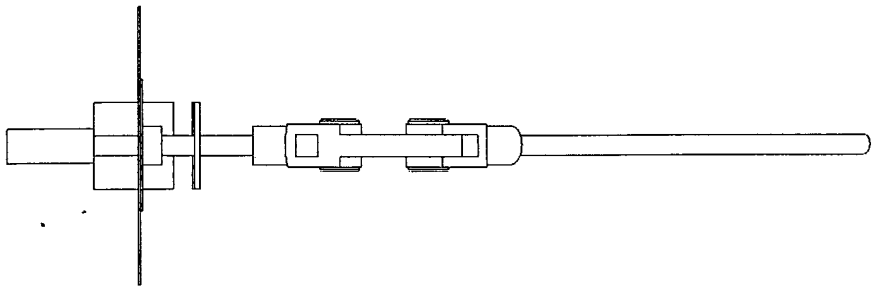
Notes:
"A" and "B" is the location of the center of gravity
A=41.36cm
B=13.4cm

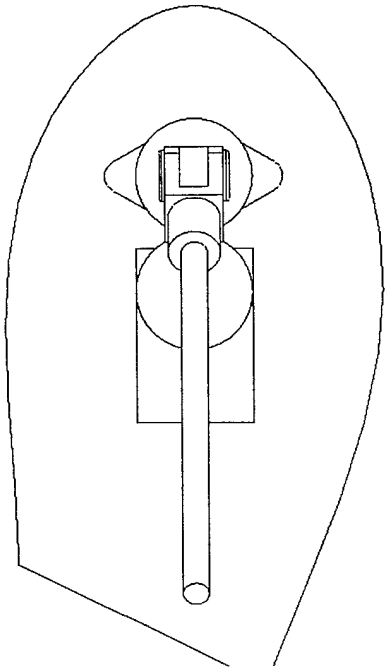
	1	1	DBF100-02	Internal Assembly	
			Parts List		
			Not To Scale	2004 DBF West Virginia University	
Wright Flyer			Measurements		
Project			Cm	3-View	
				DBF100-01	

DBF600-01	1	



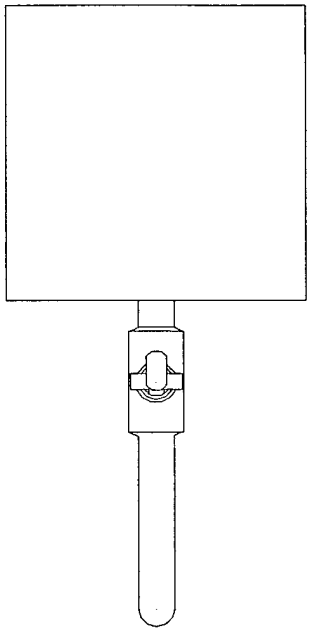


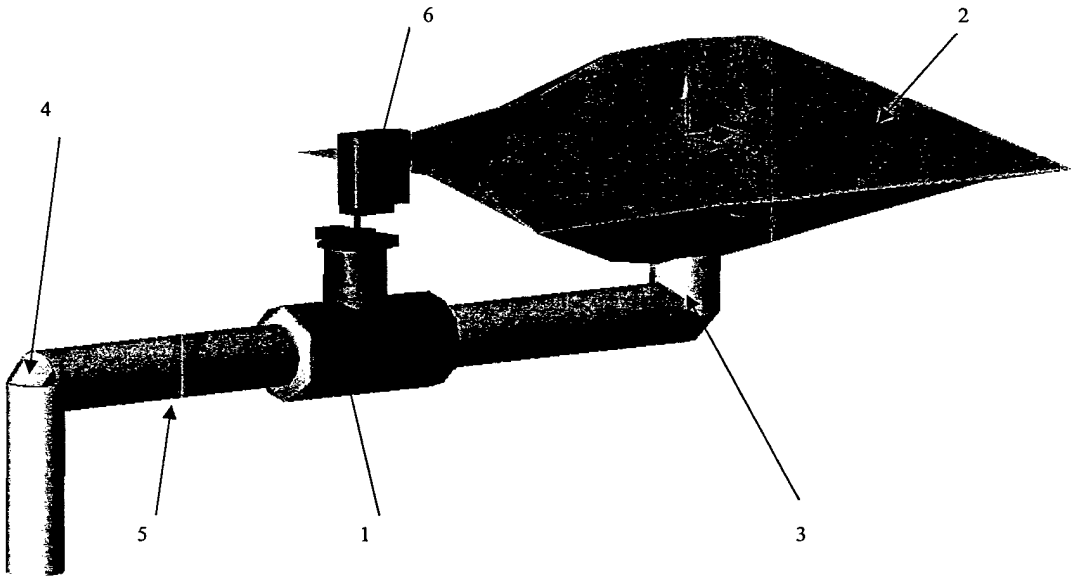


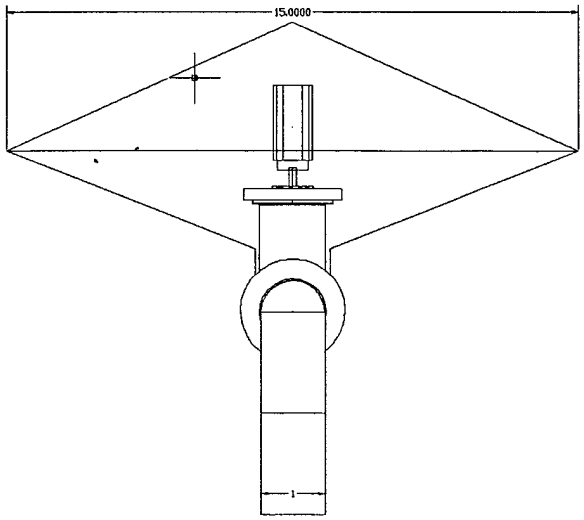


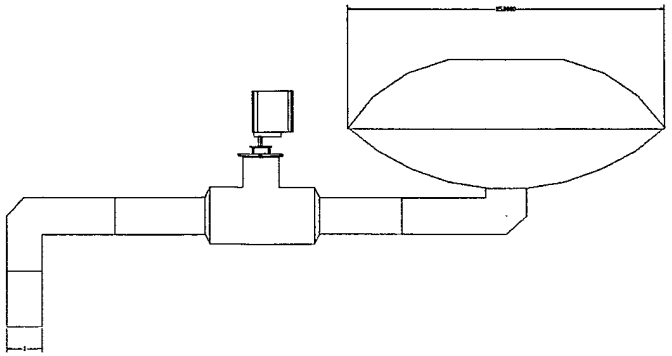
	6	1	DBF200-04	Transmission Gear	
	5	1	DBF200-03	Output Gear	
	4	1	DBF200-05	Linkage	
	3	1	DBF200-05	Warping Servo	Futaba
	2	1	DBF200-01	Transition Rib	
	1	1	DBF100-02	Control Arm	
	Parts List				
			Not To Scale	2004 DBF West Virginia University	
Wright Flyer					
Project				Warping Mechanism	
				DBF600-01	

---	DBF200-01	- 1 -









	6	1	DBF200-04	Valve Servo	Futaba
	5	1	DBF200-03	Piping (D .5in)	
	4	1	DBF200-05	Front Elbow	
	3	1	DBF200-05	Rear Elbow	
	2	1	DBF200-01	Water Bladder	
	1	1	DBF100-02	Valve	
			Parts List		
			Not To Scale	2004 DBF West Virginia University	
Wright Flyer					
Project				Payload Assm.	
				DBF200-01	

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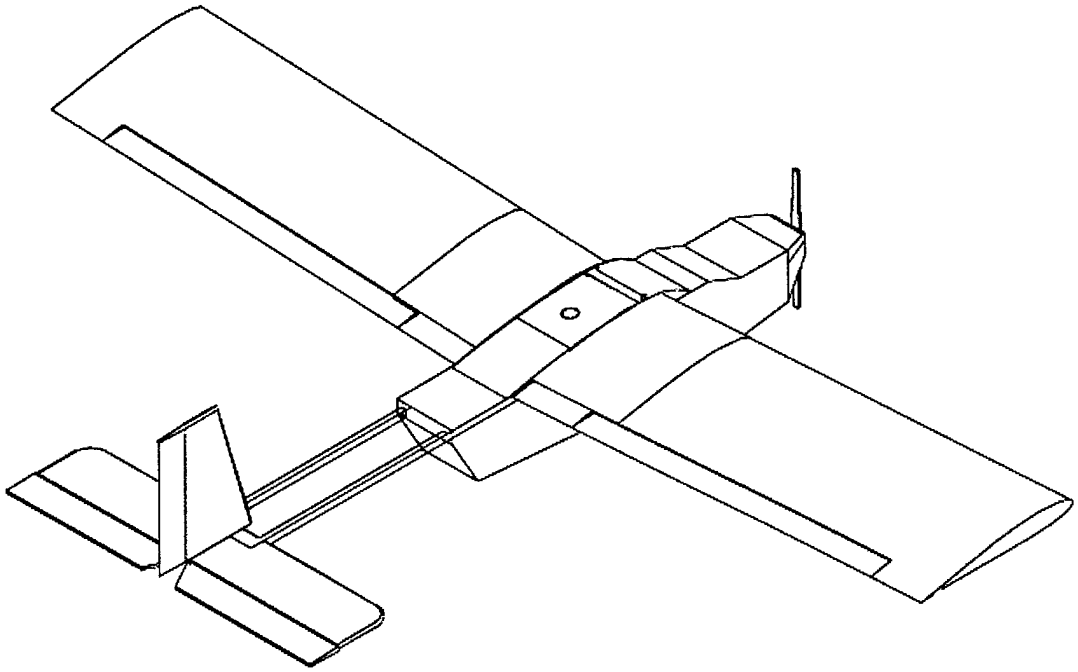
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AIAA Sponsored,
Cessna Design/Build/Fly Competition



Columbia University Design Report
"Leaking Lion"
3/9/04

Table of Contents

1.Executive Summary 3
1.1 Overview and Highlights of the design Development 3
1.2 Design Alternatives Investigated 4
2. Management Summary 5
2.1 Management Plan 5
Conceptual Design 7
Preliminary Design 7
Detail Design 7
Manufacturing 7
Documentation 7
3. Conceptual Design 8
3.1 Mission Statement 8
3.1.1The Fire Fight 8
3.1.2 The Ferry 8
3.2 Design Decisions and FOM's: 8
3.2.1 Wing Dimensions (WD): 9
3.2.2 Distance from CG to Horizontal Stabilizer (CG-HS): 9
3.2.3 Distance from CG to Propeller Plane (CG-PP): 9
3.2.4 Cross-Sectional Area of Fuselage (CSA): 9
3.2.5 Rated Aircraft Cost (RAC): 10
4. Preliminary Design 10
4.1 Wing Design 10
4.2 Fuselage 12
4.3 Payload 13
4.4 Propulsion 14
5.Detailed Design 17
5.1 Schematic Drawings 17
5.2 Dimension and Component Tables 22
Total Aircraft 22
5.4 Final Rated Aircraft Cost Spreadsheet 26
6. Manufacturing Plan and Processes 27
6.1 Wings Manufacturing 27
6.2 Fuselage Manufacturing 28
6.3 Payload Manufacturing 31
6.4 Manufacturing milestones 31
7. Testing Plan 32
Testing Date / Period 32

1.Executive Summary

This year Columbia University is introducing a rather conventional concept to the student design/build/fly competition. Everyone involved this year is new to the project and in fact this team is new to the University. We have managed however, to designed, and build a remote controlled aircraft. Although we have come this far, it is worth mentioning that the aircraft's performance may be hampered a bit by lack of experience in the field. Nonetheless, with the aid of colleagues, hobbyist and our passion for engineering design, our first airplane design ever looks to be a successful venture to date as this first generation aircraft takes off for the sky.

While we entered the competition last year, we were unable to get the funding, experience, and executive board support needed to make this possible. Rather than scrapping the dreams of building a flying machine, this year's team has made it an obsession to learn from past mistakes and achieve success with this new breed of aircraft design.

We used the I-DEAS CAD package during the careful design phase of this first generation craft. The design process took over six months to complete due to the insistence of total system integration. Since this was our first attempt at it, we wanted every last detail pertaining to mounts, battery and servo placement, among others to be included into the design before the first tool was picked up. Much has been learned so far from this experience, and as result many things will change for next year. Some snapshots and functional views of the drafting are included in the relevant parts of this report.

The following report provides an overview of the design, manufacturing and testing procedures developed for this year's competition. The plane has been designed to complete both the firefight and the ferry missions.

1.1 Overview and Highlights of the design Development

Since this was our first attempt as a team to engineer a plane we were strict in keeping the overall design simple. No complicated mechanisms or designs involving expensive manufacturing processes were considered. Due to budget constraints we limited our design such that every part could to be manufactured using conventional tools from our mechanical engineering department shop.

With that said, during the conceptual design phase of the development process we performed a Tagucchi-like analysis on the most cost effective configurations by varying the plane parameters that met some basic requirements. Rather than implementing a complicated computer algorithm that could iterate through thousands of different parameters, we narrowed down the number of possible combinations that would work using Tagucchi's method of

designing. The parameters that were varied were the wing area, wingspan, number of battery cells, and motor/propeller combinations. With the aid of the MotoCalc © and FoilSim © programs we were able to input the results into the L8 array for the Tagucchi analysis for the different configurations. Some of the trends found during this preliminary design will be talked about in later chapters.

The detail design phase was rather an investigative work. Given the lack of availability of an aeronautical engineering department, we were forced to use outside resources such as MotoCalc © and FoilSim © software to provide us results about the computational aerodynamics. While the trends found in the results made sense, we were unable to pin-point the actual theory behind it and accepted the results as given. Given scale of the design, we believed that the results obtained from the software are valid and that taking their results for granted would not compromise the overall soundness of our design. For the case of structural analysis, we looked at various model airplanes drawings with a similar scale to get an idea of the materials used, and construction methods utilized. Based on this research we reversed engineered plans that would fit our requirements and gave our design a factor of safety of two. The structural analysis was verified using a finite element modeling method from the CAD package utilized. While this robust design may compromise the overall performance of the design, we want to emphasize that as our first attempt we are not intending to win the competition but rather to learn from this experience. Overall, the detailed design provided us with a stable, robust, medium performance plane design that should fly.

1.2 Design Alternatives Investigated

Many different aircraft configurations and components were considered and evaluated. Early in the design process we examined two different tail configurations: the conventional aft tail and the V-tail. However, during the preliminary design we opted for the conventional design given the results obtained from the Tagucchi analysis. Different battery configurations were considered using the Motocalc software for the Sanyo SRC 2440mAh single cells. In particular we looked at flying times for the battery packs ranging between 25 and 30 cells for the Astroflight Cobalt motors 660 and 661. We found during that a configuration of 28 cells with a Astroflight Cobalt 660 motor worked the best based on the power versus total energy tradeoffs. Similarly, different tank dimensions were studied for carrying the full payload. Fluid dynamic equations were utilized to minimize the water drop time. Along with that, different Venturi nozzles were designed and tested as part of this design.

2. Management Summary

2.1 Management Plan

We learned from the past entry that a plan was necessary early in this project to succeed. Thus for this year's competition, the team identified the major design milestones and the approximate dates by which they should be achieved as to design/build/fly the airplane.

After having done the conceptual design as a team, our small team divided into two sub-groups: The fuselage and tank team, the wings other components group. The fuselage group was responsible to generating the design of the fuselage, tank payload and static and dynamic analysis on the plane. The wing group was responsible for generating the wing structure. This group worked in conjunction with the fuselage group to work out the details on the junction of the two pieces. The wings group also worked in the propulsion design thereby investigating possible battery/motor/propeller combinations. Calder Hughes and Gilberto Mendez were chosen as group managers to make sure the team stayed on schedule. They also performed the role of group leaders and were responsible for delegating work to all the team members. A treasurer and secretary were selected to keep track of the budged, ordering materials and handling the paper work. Both teams met on a weekly basis to talk about the progress and to set the goals for the following week. While we had Dr. Vaicaitis as our advisor, we found ourselves meeting regularly with the mechanical engineering lab technician Bob Stark. Mr. Stark gave us valuable information and guided us throughout the whole designing and manufacturing process.

The following table lists the design personnel and their assignment areas.

Project managers: Calder Hughes and Gilberto Mendez	
Treasurer: Allan Fong	Procurement and logistics: Tucker McClure
Fuselage Leader: Calder Hughes	**Wings and other Components** Leader: Gilberto Mendez
Greg Rucks Megan Mackenzie	Ameresh Kollipara Tucker McClure Gidon Ofek

Table 1 Team Architecture, personnel and assigned areas

The following is a milestone chart of the plan organized followed by a table with a breakdown of the active participants and his/her involvement within the project.

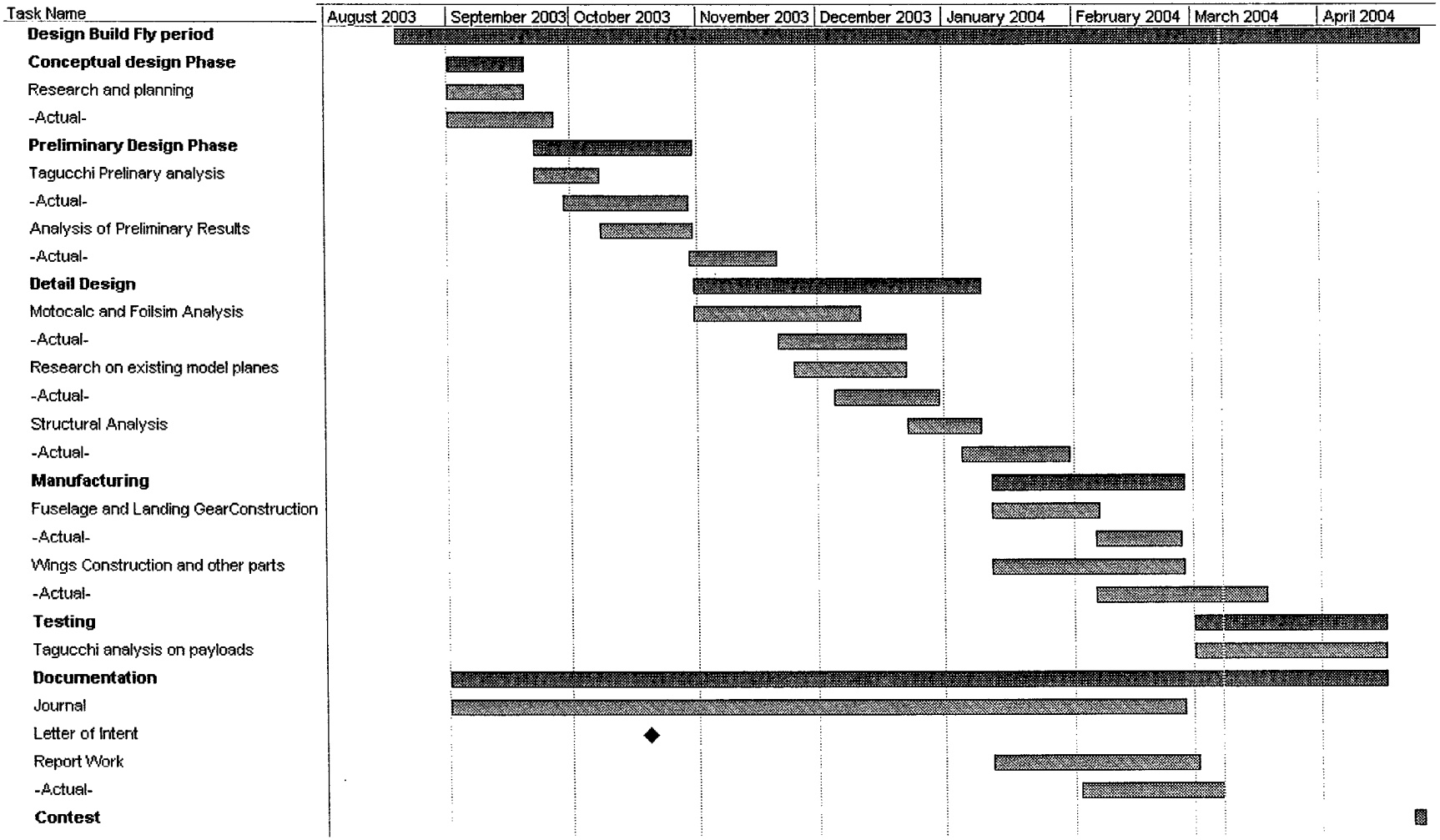


Figure 1 Milestone graph shows the planned and actual schedule.

	Ameresh Kollipara	Calder Hughes	Megan Mackenzie	Tucker McClure	Gilberto Mendez	Greg Rucks	Gidon Ofek
Conceptual Design							
Brainstorming of ideas	5	5	4	3	5	4	3
Research and planning	5	5	3	3	5	4	2
Preliminary Design							
Tagucchi parameters definition	1	2	2	1	5	2	1
Alternative Designs	5	3	3	3	3	4	1
Analysis and results interpretation of Tagucchi	5	3	2	4	5	3	1
Detail Design							
Drafting Package	1	5	3	1	2	2	1
Manufacturing							
Manufacturing method selection	3	2	3	4	5	3	1
Fuselage manufacturing	1	5	5	1	1	5	1
Wings Manufacturing	5	1	1	4	4	1	1
Landing Gear manufacturing[1]	-	-	-	-	-	-	-
Electronic design manufacturing	4	3	1	1	5	3	1
Documentation							
Letter of intent	-	5	-	-	5	-	-
Final Report	1	5	3	3	5	3	-

Table 2 Breakdown of the active participants and his/her involvement within the project

[1] Not done at the time this report was written

3. Conceptual Design

3.1 Mission Statement

The flight score will be dependent on the best time trial for two different mission types. As there are only two mission types to choose from, the Fire Fight and the Ferry mission, there were no design options based on alternate mission configurations. For both missions we were limited in regards to the size of the airplane by the fact that the entire plane must fit within a 2-1-4 ft box. The plane must also be capable of taking off and landing on a 150 ft runway, and must complete the total mission in less than ten minutes. Both missions take place on a course consisting of two poles at 1000ft apart.

3.1.1The Fire Fight

For this mission the objective is to take off with, and carry, 4 liters (maximum) of water to a designated drop zone, where the payload is to be emptied through a 1/2" diameter hole. The drop zone is defined as the downwind leg of flight about the specified course. A member of the flight crew must announce the start and finish of the dumping process by calling out, "Dump On", and "Dump Off". During this downwind leg, a single 360 degree turn must be initiated. The aircrafts tank must be filled from 2 2-Liter bottles, and the time taken to fill the water tank will be included in the overall mission time. During the timed period, the aircraft must be loaded, take off, drop its initial 4-liter payload, land, and repeat the task a second time. All of this must occur in under ten minutes. If the second lap is not completed, a time penalty of three minutes will be added to the team's time. If the second lap is not attempted, the second lap will be timed as "incomplete".

3.1.2 The Ferry

The ferry mission is the high-speed component of the competition and requires only that the aircraft to complete 4 laps and land. There is not water payload for this mission. As with the Fire Fight, the aircraft must complete a 360 degree turn during the downwind leg of each loop. The direction of the turn must be opposite that of the base and final turns.

3.2 Design Decisions and FOM's:

As most of our team has limited experience in aircraft design we focused our efforts on a conventional design even in the conceptual stage. Our initial analysis centered on manipulating the basic dimensions and ratios of a flat winged, single engine, tractor configuration. Alternate tail configurations *were* considered to reduce drag, and increase our flight controls. The split tail

configuration was abandoned because it increased drag without adding any real stability or turning capability. The V-tail configuration was also considered, but was abandoned due to the complexity of construction and ultimately control. We also considered multi-pitch wings that would add stability, and decrease the chord necessary to achieve the necessary wing area. These were abandoned because they would have required several more points of attachment due to the packing limitations stipulated by the mission parameters.

Consequently, we based our FOM's on the dimensions of the fuselage, wings, and tail, as well as their positions relative to each other.

3.2.1 Wing Dimensions (WD):

Restraining the Length to be less than 4 ft., and maintaining enough wing area to support the aircraft we manipulated the wing chord, length, and thickness. A scale of 1-5 was used to assess desirability based on aspect ratio and L/D.

3.2.2 Distance from CG to Horizontal Stabilizer (CG-HS):

As this distance changed, the tail area required to counteract the pitching moment of the wings varied. A scale of 1-5 was used to evaluate desirability of this relationship. Construction, shipping, and flight control were all considered.

3.2.3 Distance from CG to Propeller Plane (CG-PP):

The distance from the CG to the nose is an important element to balancing the aircraft. The further the motor is from the CG, the more moment it exerts about the CG. This moment acts to offset the moment generated by the other components and the tail. A shorter length, however, is desirable for drag reasons as well as scoring. As the distance from the nose of the CG decreases more components must be moved forward to balance the aircraft. A limit is reached when the volume needed for the components necessary for balance exceeds the amount of space available between the water tank (centered on the CG) and the back of the motor mount. A scoring system from 1-5 is used to evaluate the desirability of these trade-offs.

3.2.4 Cross-Sectional Area of Fuselage (CSA):

A small cross-sectional area is clearly desirable from a drag perspective. The presence of the water tank, and the time dependency on draining the tank introduces a contradictory incentive. The greater the tank height, the more pressure will be present at the exit hole and the faster the water will drain. Thus a compromise must be struck between tank dimensions and

overall cross-sectional area of the fuselage. A scale of 1-5 is again used to evaluate the different potential cross-sectional areas.

3.2.5 Rated Aircraft Cost (RAC):

RAC is defined by the competition rules.

$$RAC = \$(thousands) = (A * MEW + B * REP + C * MFHR)/1000 \quad (1)$$

The following tables provide examples of the design modifications considered, and their FOM ratings in the aforementioned categories.

Trial Sample	WD	CG-HS	CG-PP	CSA	RAC
1	Ch=16", L=4', Thick=1.25'	32"	16"	60 in^2	9.82
2	Ch=17", L=4', Thick=1.25'	36"	18"	65 in^2	10.20
3	Ch=18", L=4', Thick=1.25'	40"	20"	70 in^2	10.43
4	Ch=19", L=3.75', Thick=1.25'	44"	22"	75 in^2	10.52
5	Ch=20", L=3.5', Thick=1.25'	48"	24"	80 in^2	11.46

Table 3: Trial Configurations for FOM calculations

Trial	WD	CG-HS	CG-PP	CSA	RAC
1	0	1	3	4.5	5
2	3.5	2.5	5	5	4
3	5	4	4.5	4	3
4	4	5	2.5	3	2
5	2	3	2	1	1

Table 4:FOM ratings for the trials defined in Table 1. Note these are only a sample of the rated designs.

4. Preliminary Design

4.1 Wing Design

We knew we wanted to keep the plane simple in terms of calculations and construction and to be easy to fly – we're beginners. Our trade off, we accepted, was speed. From previous observations and research, we knew that the wings, in order to create good lift on a slow plane, would need to be thick with low camber. Furthermore, we knew that the easiest design to build was straight across, with without tapering or significant changes throughout. This was also the easiest to calculate.

We used the program MotoCalc to optimize our plans for the wing (in tandem with the engine and battery configuration). Among other design characteristics, we started with some estimates of our plane's overall weight (4 pounds of water plus a lot of guessing – about 13 pounds before motor or batteries), necessary power, battery needs, and wing dimensions, using about 13% thickness, 2% camber, and an angle of attack of about 5°. We tried high thickness and lower camber for ease of handling, but this proved too slow, so lower thickness and slightly higher camber followed. Our plane would not have the speed to justify a significantly higher camber. The angle of attack, as the third major variable in the decision for the airfoil, was tried in many positions for each likely configuration of thickness and camber until we found the numbers we liked the best. We reduced some of the weight due to batteries, optimized the propeller/pitch, and followed the suggestions of MotoCalc's "Opinion" feature, which responds to result calculations.

The design parameters decided upon are as follows: 8' span, 18" cord, 9% thickness, 1.7% camber, 2.8° angle of attack. A dihedral would be small, probably 1-2°, and we will decide on it later. This, with the information from the motor, battery, and propeller decisions should result in a 43 mph level flight speed at 66% throttle (lift of about 19 pounds), and a 29 mph stall, which should give us about 9:00 minutes of flight time – enough to complete the course at our estimated speeds and taking into consideration flight with then without water.

Once the airfoil shape was satisfactory, we looked to fleshing out the wing. We knew it would be a standard rib-and-spar design, so to get a feel for the spacing of ribs in other models, we observed some existent large kits for guidance. We decided on balsa ribs placed 3" apart, pieced together with 0.25" x 0.5" pine spars – 4 on top and 2 on bottom (this would also require a flat bottom, as discussed later). The entire wing would then be connected to the fuselage with <THINGS> penetrating the ribs. The wing would be covered with <WING MATERIAL>. Plans were drawn for the final construction with a few more minor adjustments and the necessary parts were ordered.

The tail was even more straight forward. Again for ease, the horizontal stabilizer was given no aerodynamic shape. We decided flat was best, as long as it operated as a decent

control surface. The rudder decision was similar. There will be no dihedral angle for the horizontal stabilizer.

4.2 Fuselage

It was decided that the most important function of the fuselage was to adequately house all the components necessary for the operation of the plane. The water tank turned out to be the determining factor along with the chord of the wing which we wanted to entirely contain with the body for obvious reasons. Since it is generally desirable, from a drag standpoint, to minimize frontal area, the frontal dimensions of the fuselage were made large enough to provide a volume sufficient to transport the payload and internal components, and no larger. The drag coefficient was thus minimized with respect to frontal area

$$C_D = \frac{drag}{.5\rho V^2 A} \qquad (2)$$

The length was similarly limited. Since long, planar surfaces tend to induce instability, the longitudinal length of the fuselage was made only so long as to ensure its carrying capacity, and no longer. The length was also constrained by the competition rules, as it must fit in a 4x2x1 ft box. Similar to length, a trade was made for the height, which was limited according to the height of the payload tank. Streamlining was considered a desirable yet difficult implementation to the design. We opted against streamlining along the sides and viewed it as a trade between making the fuselage longer and wider and extending a streamlined geometry rearward. Such a design would have been heavier and would have required a larger frontal area, creating undesirable drag characteristics. The bottom of the fuselage was streamlined, however, such that ¼ of the its length was angled upward at an angle of approximately 20° at the rear. The shorter length of the fuselage introduced the issue of attaching the tail. A support system was adopted in which two rods of aluminum, anchored along ¼ of their length to the inside of the top wall of the fuselage, would extend rearward and support the tail.

Another consideration was strength. High-density foam was readily available to us and seemed to have a desirable strength to weight ratio. A standard epoxy could be applied to the foam to increase its strength and resistance to damage, without adding significant weight. Since the introduction of asymmetry would produce unpredictability with respect to flight behavior, the bottom was settled upon as the ideal plane from which to access the inside of the plane, but this introduced the issue of providing sufficient strength to support the payload. A method of balsa wood railing was thus used to provide extra strength to the removable panels on the bottom. (See Manufacturing Process) Also, it was decided that the walls should provide the bulk of support to the payload tank, so a balsa wood caging system was adopted whereby the weight of

the payload would be supported through shear along the walls rather than from below. This left the bottom free to be removable allowing for ease of construction and access to electronics.

The balsa wood between the tank and the fuselage side also provides necessary space to run wires between the back and front ends of the aircraft. We will need to run wiring between the batteries (which are located both in front of and behind the payload), as well as to the various servos locations: the engine, the tail, the landing gear.

4.3 Payload

The payload tank was designed to hold 4 Liters of water, with minimal excess room. Since the fuselage dimensions, and consequently the overall drag of the airplane, are dependent on the payload size, we kept the volume at the 4 Liter range. The payload dimensions are 11.25"x4.75"x6.625" (length x width x height) along the exterior surfaces, and interior surfaces are curve and therefore difficult to dimension. Suffice it to say that the tank does indeed hold 4 Liters of water. Based on pressure alone, the ideal tank would be a very tall, skinny column of water, since pressure is proportional to the height of water in the tank.

$$P = \gamma h \tag{3}$$

Bernoulli's equation states that increased pressure will increase the rate of flow of water leaving the tank. Since the cross-sectional area affecting the drag on the overall aircraft is a function of the height times the width, we increased the cross-section in the height direction while minimizing the width of the water tank.

A second reason we modeled our tank as a long, tall, rectangle was to decrease side to side imbalance of the plane during flight. Obviously a cylinder would minimize the maximum distance the water must flow to reach the exit hole, however the cylindrical shape would allow more space for the water to move from side to side, and proved difficult to consturct. Since the payload is a liquid we are concerned about the effects of sloshing which will shift the balance of the aircraft. During take-off and landing, the tank is respectively full and empty, and so water shifting and imbalance should not occur (by design the tank will hold just enough water, and will contain no empty spaces when full). We expect to be in steady flight when the payload is dropped, and in fact, part of the emptying will occur while the plane is turning. At this point, it is important to have control of our plane laterally. With the smaller width, any splashing and deviations will be relatively close to the center of mass compared to a wider tank. We are also considering the addition of baffles to the tank to further minimize this effect. With even just a barrier down the middle, the effect would seem to be minimized.

The bottom of the tank is angled toward the exit hole, which is also angled to create a funnel. The exit hole goes from 1 inch at the top to ½ inch at 45 degrees. Like the fuselage, the

tank is constructed of high density foam, covered with an epoxy for strength, and water-proofing. The tank has been tested and confirmed to hold 4 Liters of water without leaking.

4.4 Propulsion

Based on the lift curve generated by Motocalc, and the overall drag of the plane, we calculated the amount of thrust needed to power our aircraft.

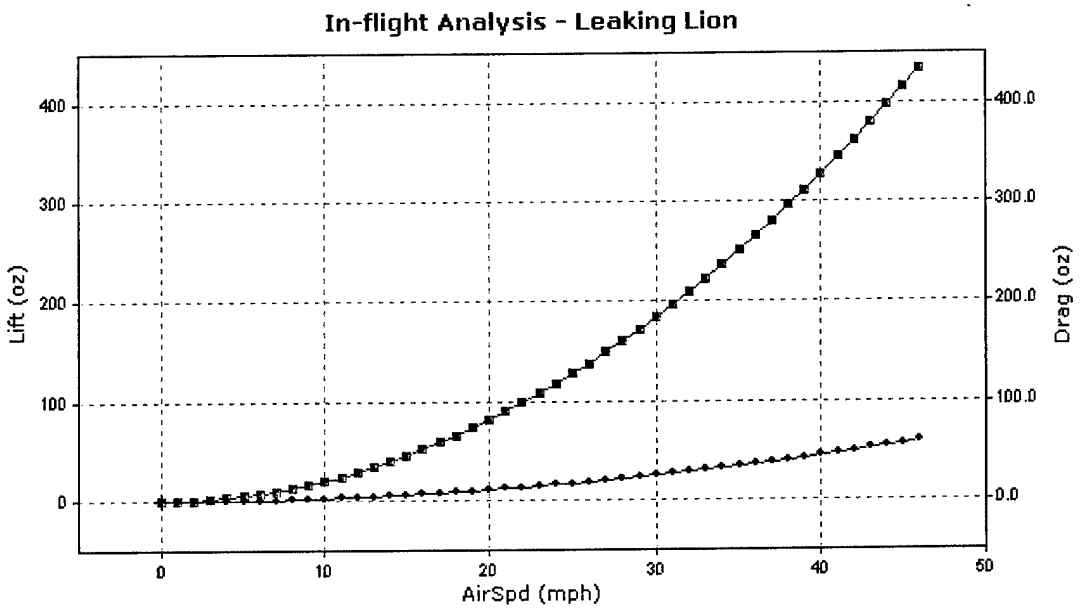


Figure 2 lift drag without payload

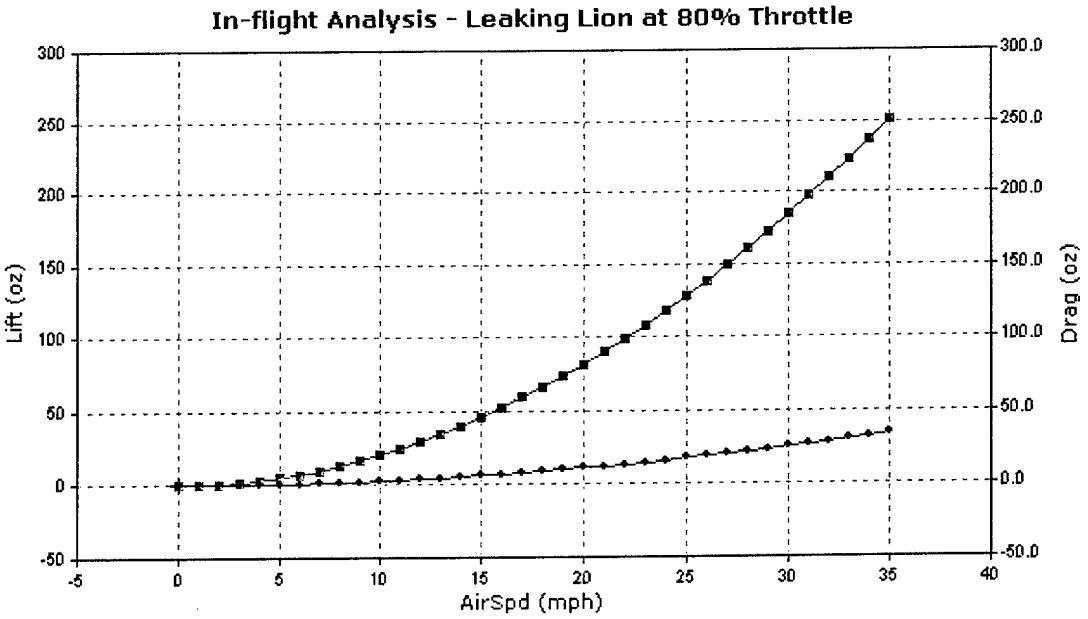


Figure 3 lift and drag per air speed with payload

These values were obtained for the following configurations:

Motor: Astro Cobalt 60 11T#23 #661; 347rpm/V; 2.5A no-load; 0.103 Ohms.
Battery: Sanyo 2000SCE; 26 cells; 2400mAh @ 1.2V; 0.007 Ohms/cell.
Speed Control: Astro 210; 0.003 Ohms; High rate.
Drive System: My gear box; 14x10 (Pconst=1.31; Tconst=0.95) direct drive.
Airframe: Leaking Lion ; 1728sq.in; 204.8oz; 17.1oz/sq.ft; Cd=0.062; Cl=0.45; Clopt=0.62; Clmax=1.07.

Assuming an optimal speed range of 25-40 mph, we selected a 1.3 horsepower motor with a 14x10 Master Airscrew Propeller for fire-fight mission and a 14x8 Master Airscrew Propeller for the high speed mission. The following are the characteristics of the propeller for both missions.

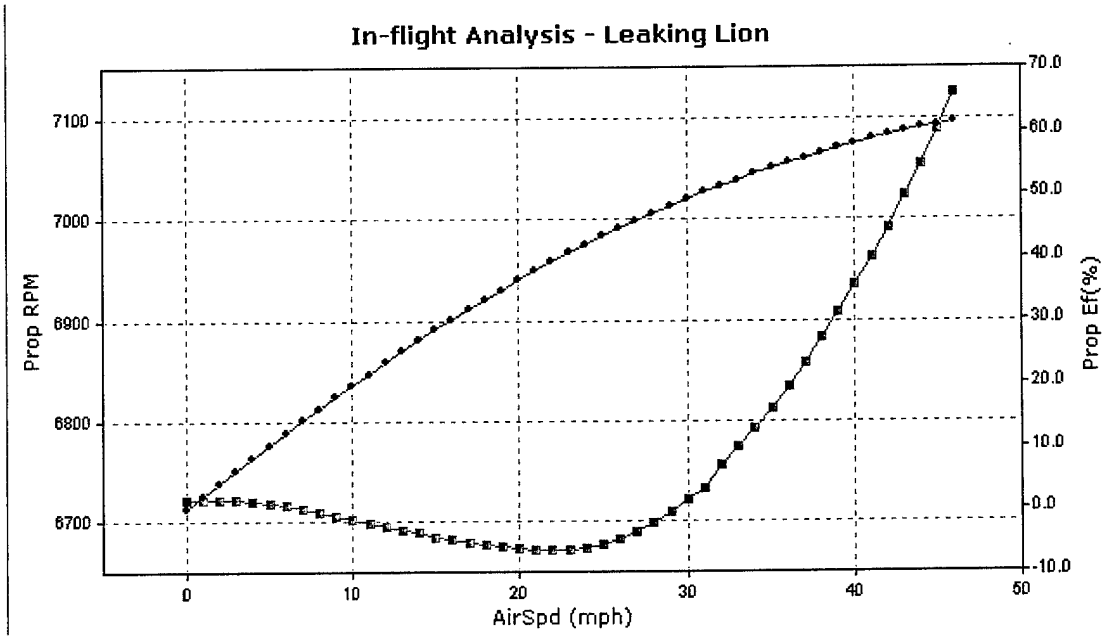


Figure 4 propeller RPM and efficiency versus air speed no payload

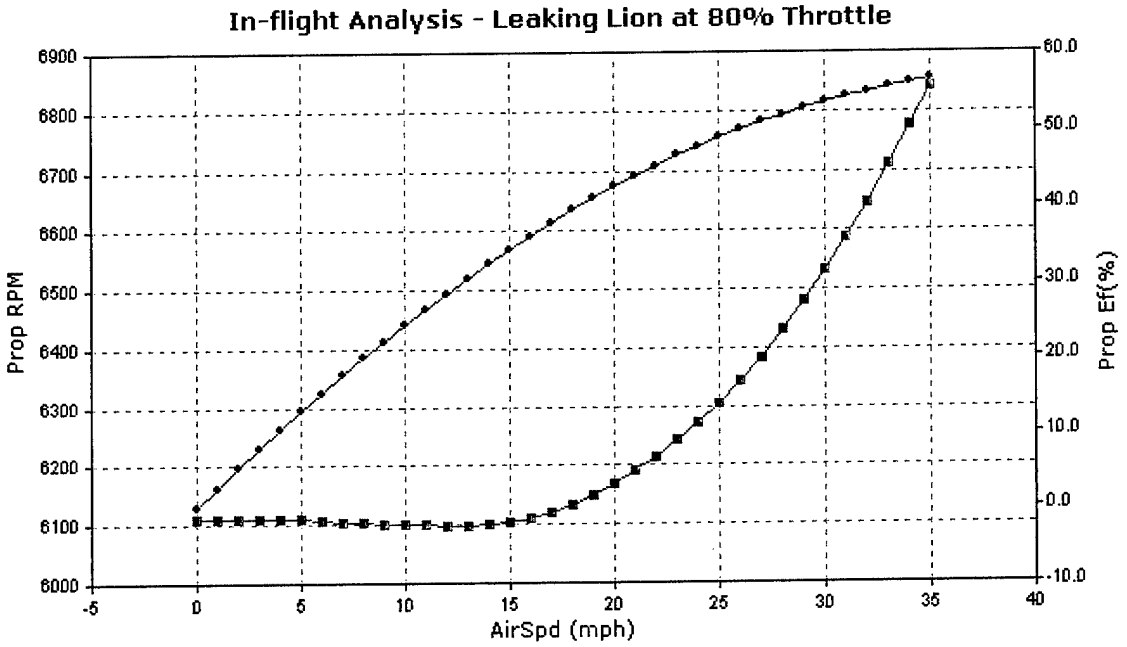
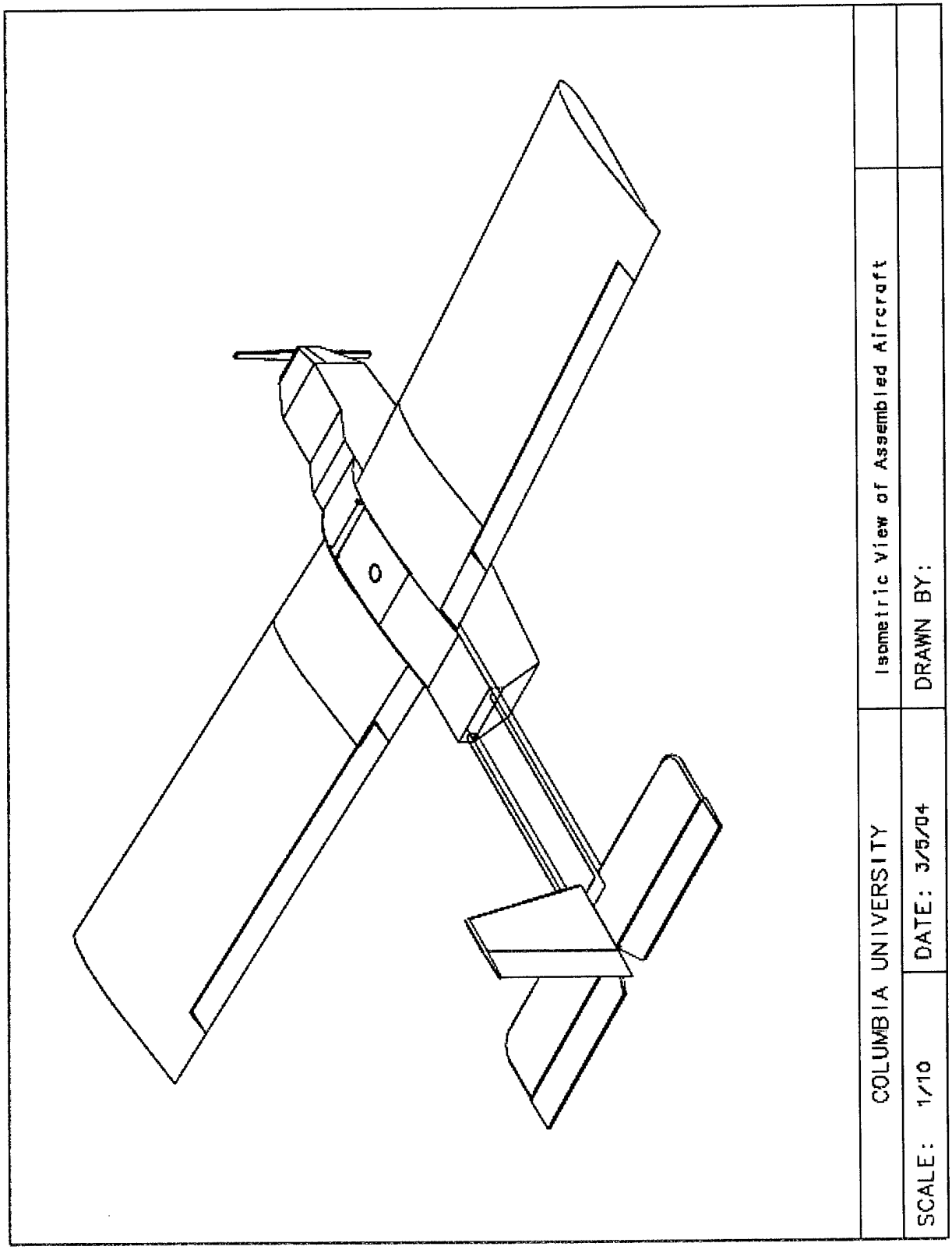
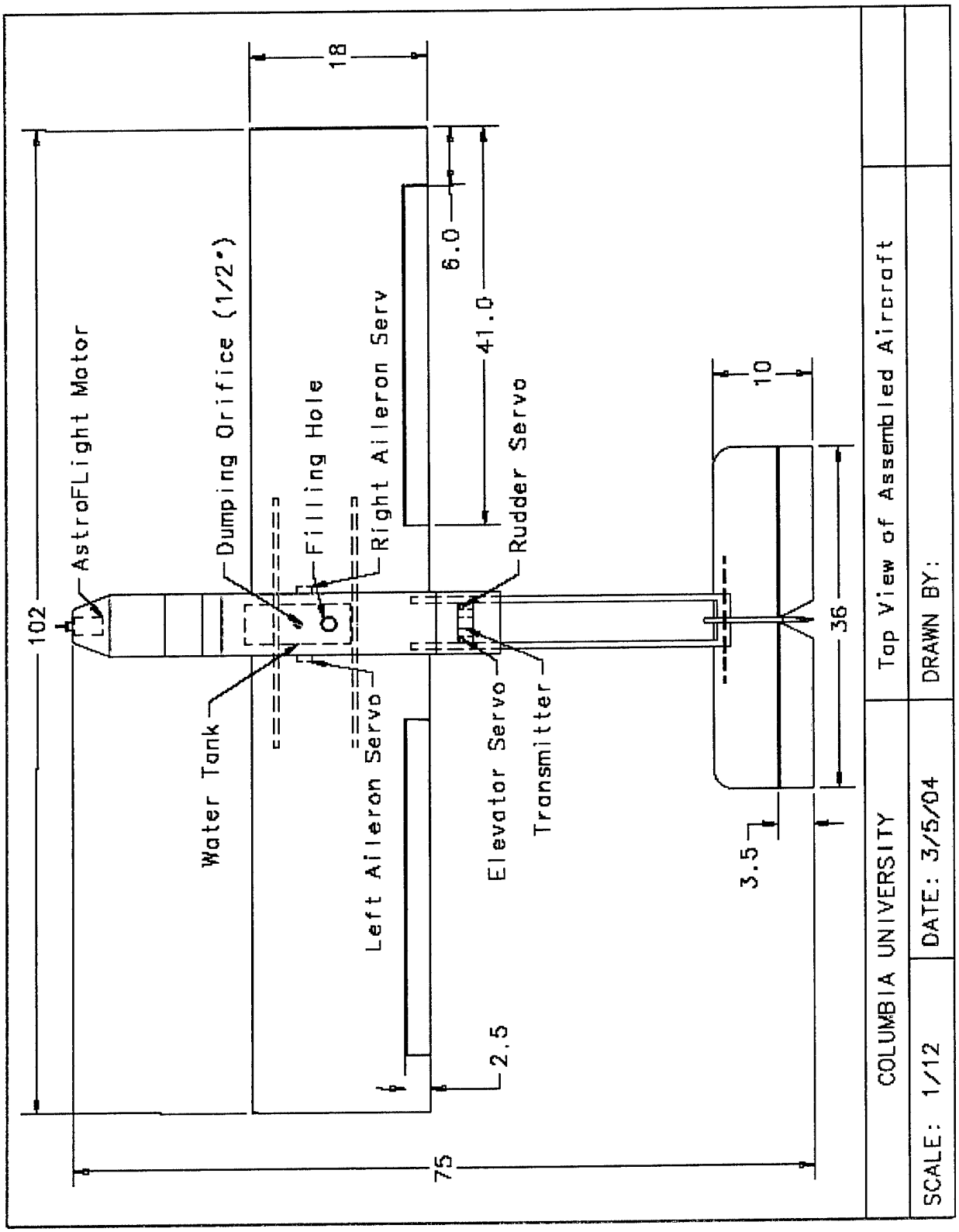


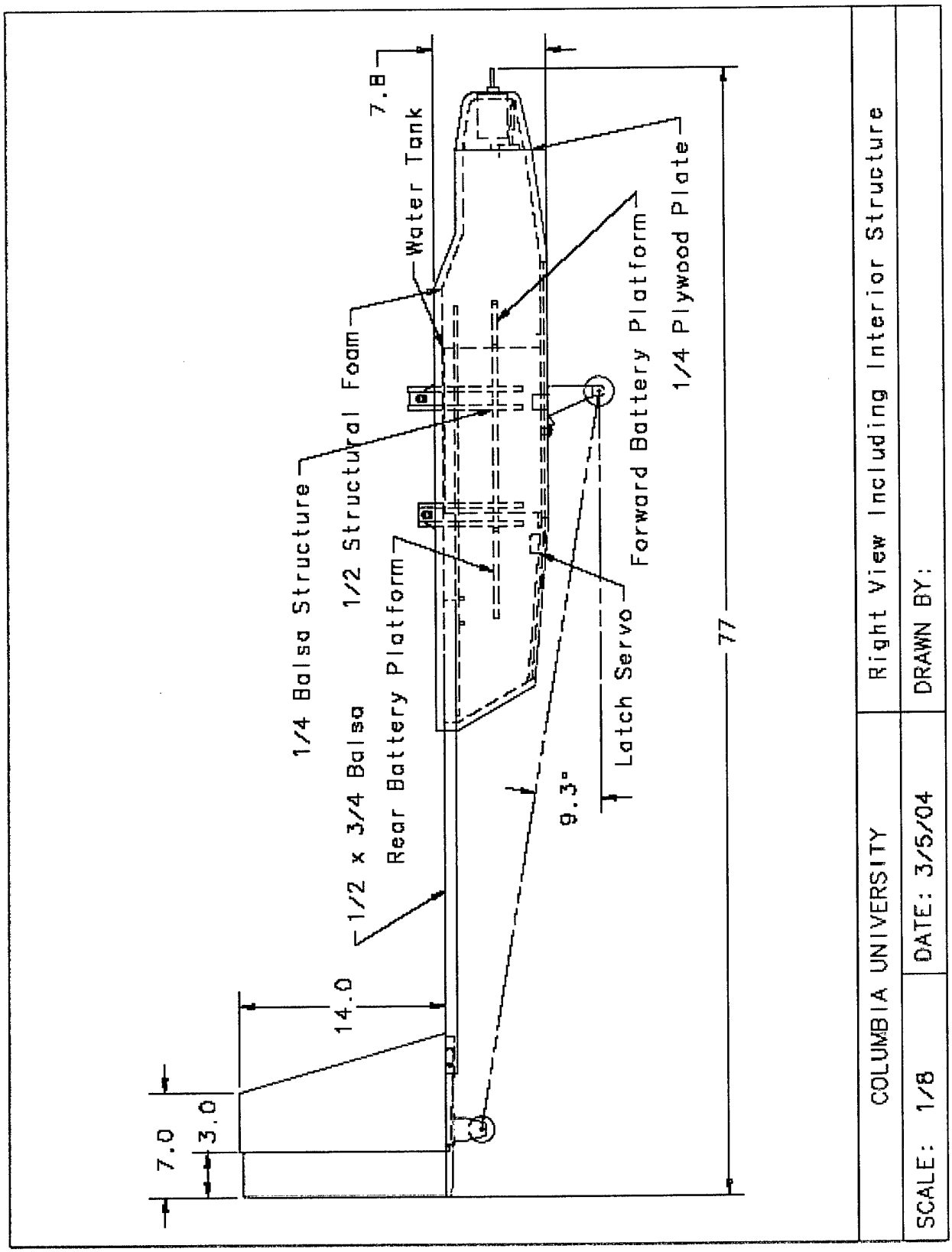
Figure 5 propeller rpm and efficiency versus airspeed

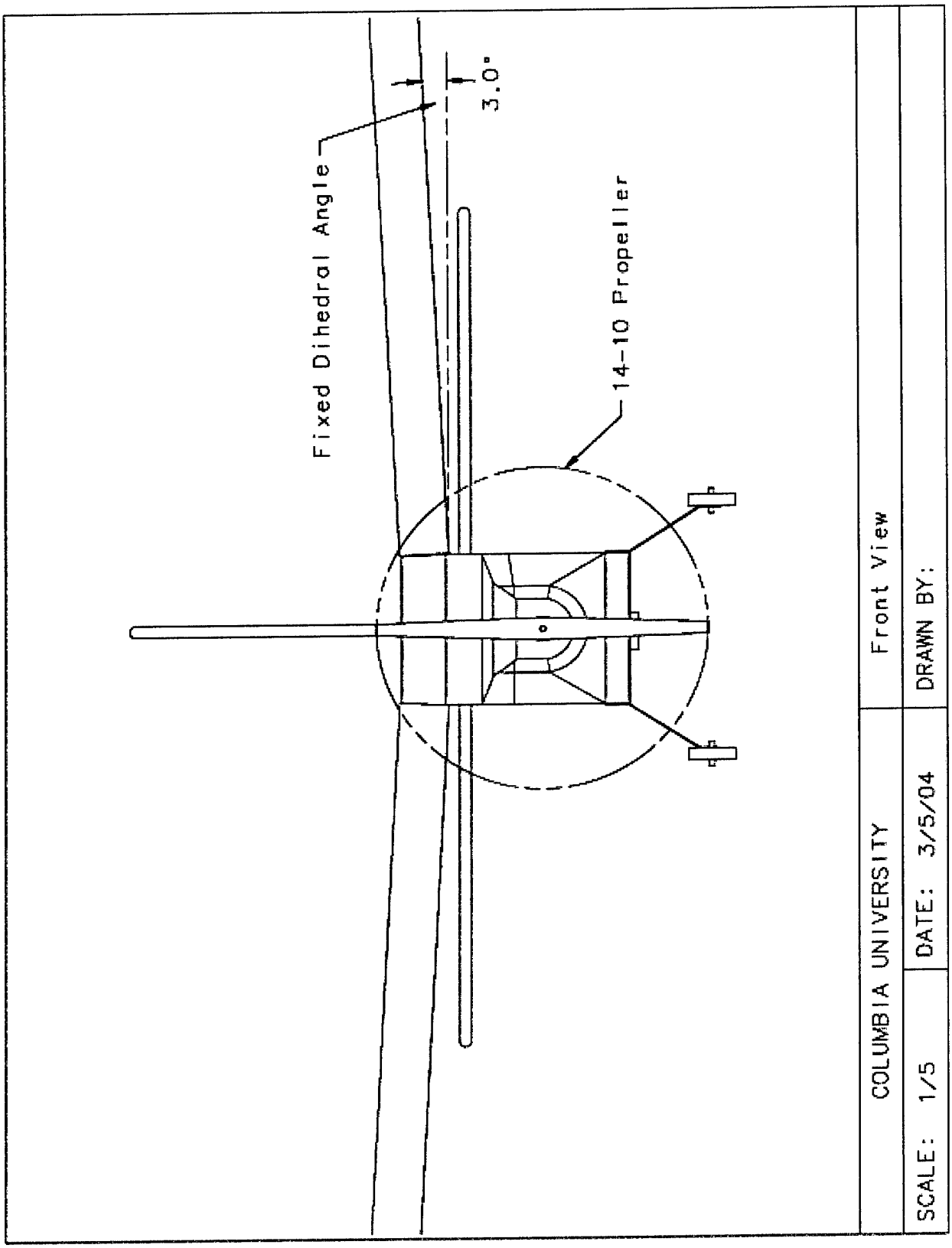
5.Detailed Design

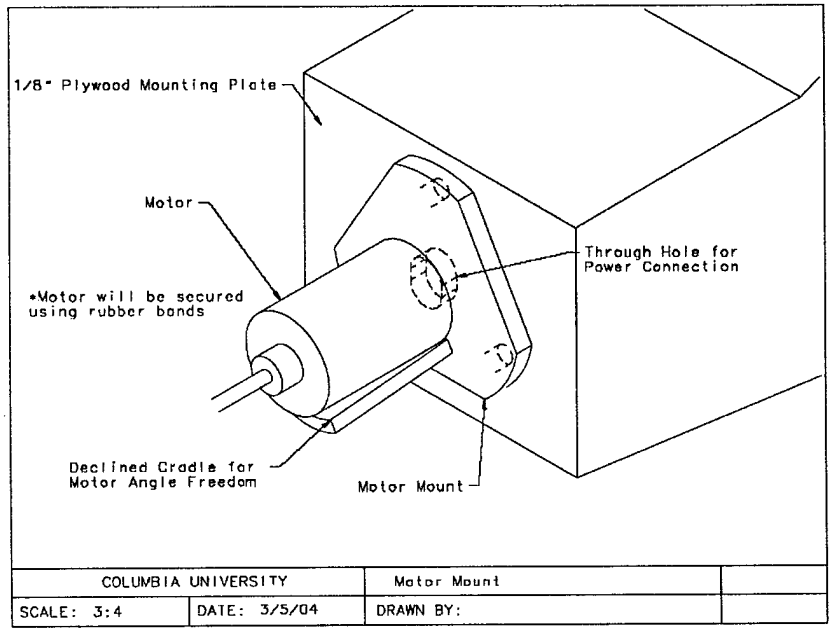
5.1 Schematic Drawings



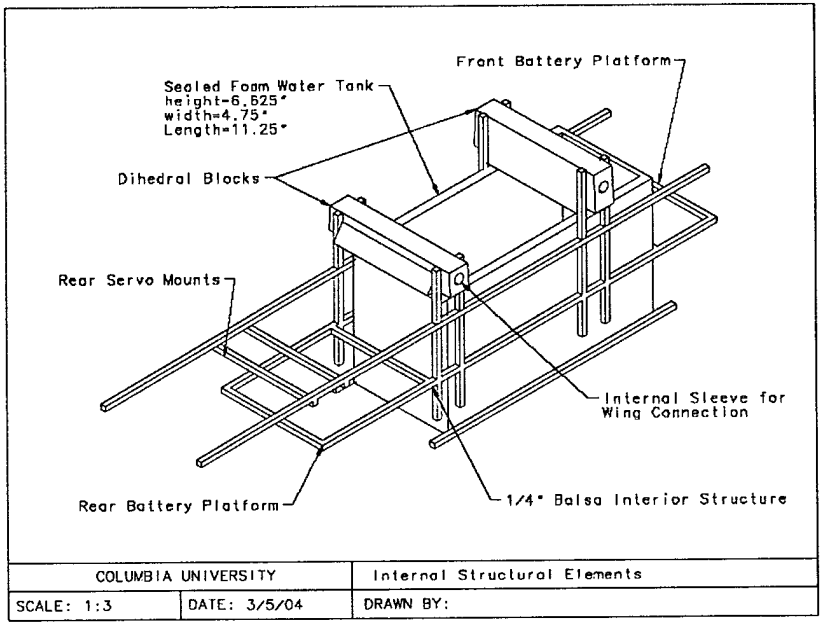

AstroFLight Motor
Dumping Orifice (1/2")
Filling Hole
Right Aileron Serv
Rudder Servo
Water Tank
Left Aileron Servo
Elevator Servo
Transmitter
102
18
6.0
41.0
10
36
3.5
2.5
75
COLUMBIA UNIVERSITY
Top View of Assembled Aircraft
SCALE: 1/12
DATE: 3/5/04
DRAWN BY:


7.8
Water Tank
1/4 Balsa Structure
1/2 Structural Foam
1/4 Plywood Plate
1/2 x 3/4 Balsa
Rear Battery Platform
Forward Battery Platform
Latch Servo
9.3°
77
14.0
7.0
3.0
COLUMBIA UNIVERSITY
Right View Including Interior Structure
SCALE: 1/8
DATE: 3/5/04
DRAWN BY:


Fixed Dihedral Angle
3.0°
14-10 Propeller
Front View
COLUMBIA UNIVERSITY
SCALE: 1/5
DATE: 3/5/04
DRAWN BY:



COLUMBIA UNIVERSITY		Motor Mount	
SCALE: 3:4	DATE: 3/5/04	DRAWN BY:	



COLUMBIA UNIVERSITY		Internal Structural Elements
SCALE: 1:3	DATE: 3/5/04	DRAWN BY:

5.2 Dimension and Component Tables

Length	75 in
Span	98 in
Height	7.8 in
Chord	18 in
Aspect Ratio	5.67
Wing Area	1734 in^2

Table 5: Aircraft Geometry

Airtronics Servo 94102 (4X)
Airtronics RD6000 Sport Radio
Airtronics Narrow Band Reciever 92777
Battery Configuration: 4 linear pods of 7 Batteries
Master Airscrew Propeller
Astro Challenger Cobalt 60 Motor

Table 6: Aircraft Systems; Components and Electronics

Component	**Weight (oz)**
Astro Flight Cobalt 60 Motor	24
Engine Mount	4
Foam payload tank	2.25
Weight 4 L of water	140.8
1 Battery (Total of 28 batteries)	58.8
5 Servos and connector	8.5
2 Wings	25.2
Fuselage (minus balsa wood)	10
Balsa wood cage	?
Propeller	7.8
Tail	?
AIRFRAME	
fuselage	3.25 lbs
PROPULSION SYSTEM	1.5 lbs
Engine + Engine Mount	
CONTROL SYSTEM	3.8 lbs
Batteries + Servos	
PAYLOAD SYSTEM	.25 lbs
payload tank + balsa wood structure	
PAYLOAD	8.8 lbs
Total Aircraft	
EMPTY WEIGHT	~8.8lbs
GROSS WEIGHT	~17.6lbs

Table 7: Weight Measurements

5.3 Dynamic analysis of plane Design

Motor: Astro Cobalt 60 11T#23 #661; 347rpm/V; 2.5A no-load; 0.103 Ohms.
Battery: Sanyo 2000SCE; 26 cells; 2400mAh @ 1.2V; 0.007 Ohms/cell.
Speed Control: Astro 210; 0.003 Ohms; High rate.
Drive System: Astro 035-15 2.38:1 Gearbox; 14x10 (Pconst=1.31; Tconst=0.95) direct drive.
Airframe: Leaking Lion ; 1728sq.in; 204.8oz; 17.1oz/sq.ft; Cd=0.062; Cl=0.45; Clopt=0.62; Clmax=1.07.
Filter: 40A max; 45A max (ESC).
Stats: 41 W/lb in; 30 W/lb out; 20mph stall; 27mph opt @ 55% (17:50, 106°F); 32mph level @ 60% (15:01, 113°F); 369ft/min @ 8.9°; -294ft/min @ -7.1°.

Sea Level, 29.92inHg, 59°F
Throttle 80%

AirSpd (mph)	Drag (oz)	Lift (oz)	Batt Amps	Motor Amps	Motor Volts	Input (W)	Loss (W)	MGbOut (W)	MotGb Ef(%)	Shaft Ef(%)	Prop RPM	Thrust (oz)	PSpd (mph)	Prop Ef(%)	Total Ef(%)	Time (m:s)
17.0	8.1	59.3	21.3	26.6	20.0	532.8	153.0	379.8	71.3	57.2	5760	64.8	37.5	36.0	20.6	6:46
18.0	9.1	66.4	21.3	26.6	20.0	532.9	153.1	379.8	71.3	57.2	5759	64.1	36.5	37.7	21.6	6:46
19.0	10.2	74.0	21.3	26.6	20.0	533.0	153.1	379.9	71.3	57.2	5758	63.4	35.5	39.3	22.5	6:46
20.0	11.3	82.0	21.3	26.6	20.0	532.9	153.1	379.9	71.3	57.2	5759	62.7	34.5	40.9	23.4	6:46
21.0	12.4	90.4	21.3	26.6	20.0	532.7	153.0	379.7	71.3	57.2	5760	62.0	33.6	42.5	24.3	6:46
22.0	13.6	99.2	21.2	26.5	20.0	532.3	152.8	379.5	71.3	57.3	5764	61.2	32.6	44.0	25.2	6:47
23.0	14.9	108.5	21.2	26.5	20.1	531.6	152.4	379.2	71.3	57.3	5769	60.4	31.6	45.4	26.0	6:47
24.0	16.2	118.1	21.2	26.4	20.1	530.7	152.0	378.7	71.4	57.4	5777	59.6	30.7	46.8	26.9	6:48
25.0	17.6	128.2	21.1	26.4	20.1	529.7	151.5	378.2	71.4	57.4	5785	58.7	29.8	48.1	27.7	6:49
26.0	19.0	138.6	21.0	26.3	20.1	528.6	150.9	377.6	71.4	57.5	5795	57.9	28.9	49.4	28.4	6:51
27.0	20.5	149.5	21.0	26.2	20.1	527.2	150.3	376.9	71.5	57.6	5805	57.0	28.0	50.6	29.2	6:52
28.0	22.1	160.8	20.9	26.1	20.1	525.5	149.4	376.1	71.6	57.7	5819	56.1	27.1	51.8	29.9	6:54
29.0	23.7	172.4	20.8	26.0	20.2	523.6	148.5	375.1	71.6	57.8	5834	55.2	26.3	52.9	30.6	6:56
30.0	25.3	184.5	20.7	25.8	20.2	521.5	147.5	374.0	71.7	58.0	5851	54.3	25.4	54.0	31.3	6:58
31.0	27.1	197.1	20.6	25.7	20.2	519.2	146.5	372.7	71.8	58.1	5869	53.3	24.6	55.0	32.0	7:00
32.0	28.8	210.0	20.4	25.5	20.2	516.7	145.3	371.4	71.9	58.3	5889	52.3	23.8	55.9	32.6	7:03
33.0	30.7	223.3	20.3	25.4	20.3	514.1	144.1	369.9	72.0	58.4	5910	51.4	23.0	56.8	33.2	7:06
34.0	32.6	237.0	20.1	25.2	20.3	511.2	142.8	368.3	72.1	58.6	5933	50.4	22.2	57.7	33.8	7:09
35.0	34.5	251.2	20.0	25.0	20.3	508.1	141.5	366.6	72.2	58.8	5957	49.4	21.4	58.5	34.4	7:12
36.0	36.5	265.7	19.8	24.8	20.4	504.8	140.1	364.8	72.3	59.0	5981	48.4	20.6	59.3	35.0	7:16
37.0	38.6	280.7	19.6	24.6	20.4	501.4	138.6	362.8	72.4	59.2	6008	47.4	19.9	60.0	35.5	7:20
38.0	40.7	296.1	19.5	24.3	20.5	497.6	137.0	360.6	72.5	59.4	6036	46.4	19.2	60.7	36.0	7:24

Motor performance calculations take ambient temperature and heating effects into account.

Color Key: Propeller Stalled | Stall Speed @ Clmax=1.07 | Level Flight @ Clopt=0.62 | Level Flight @ Cl=0.45

Figure 6. In flight Analysis without payload 80% throttle

Motor: Astro Cobalt 60 11T#23 #661; 347rpm/V; 2.5A no-load; 0.103 Ohms.
Battery: Sanyo 2000SCE; 26 cells; 2400mAh @ 1.2V; 0.007 Ohms/cell.
Speed Control: Astro 210; 0.003 Ohms; High rate.
Drive System: Astro 035-15 2.38:1 Gearbox; 14x10 (Pconst=1.31; Tconst=0.95) direct drive.
Airframe: Leaking Lion ; 1728sq.in; 296.8oz; 24.7oz/sq.ft; Cd=0.062; Cl=0.45; Clopt=0.62; Clmax=1.07.
Filter: 40A max; 45A max (ESC).
Stats: 28 W/lb in; 20 W/lb out; 25mph stall; 33mph opt @ 68% (10:45, 129°F); 38mph level @ 76% (8:30, 146°F); 144ft/min @ 2.9°; -354ft/min @ -7.1°.

Sea Level, 29.92inHg, 59°F
Throttle 80%

AirSpd (mph)	Drag (oz)	Lift (oz)	Batt Amps	Motor Amps	Motor Volts	Input (W)	Loss (W)	MGbOut (W)	MotGb Ef(%)	Shaft Ef(%)	Prop RPM	Thrust (oz)	PSpd (mph)	Prop Ef(%)	Total Ef(%)	Time (m:s)
17.0	8.1	59.3	21.3	26.6	20.0	532.8	153.0	379.8	71.3	57.2	5760	64.8	37.5	36.0	20.6	6:46
18.0	9.1	66.4	21.3	26.6	20.0	532.9	153.1	379.8	71.3	57.2	5759	64.1	36.5	37.7	21.6	6:46
19.0	10.2	74.0	21.3	26.6	20.0	533.0	153.1	379.9	71.3	57.2	5758	63.4	35.5	39.3	22.5	6:46
20.0	11.3	82.0	21.3	26.6	20.0	532.9	153.1	379.9	71.3	57.2	5759	62.7	34.5	40.9	23.4	6:46
21.0	12.4	90.4	21.3	26.6	20.0	532.7	153.0	379.7	71.3	57.2	5760	62.0	33.6	42.5	24.3	6:46
22.0	13.6	99.2	21.2	26.5	20.0	532.3	152.8	379.5	71.3	57.3	5764	61.2	32.6	44.0	25.2	6:47
23.0	14.9	108.5	21.2	26.5	20.1	531.6	152.4	379.2	71.3	57.3	5769	60.4	31.6	45.4	26.0	6:47
24.0	16.2	118.1	21.2	26.4	20.1	530.7	152.0	378.7	71.4	57.4	5777	59.6	30.7	46.8	26.9	6:48
25.0	17.6	128.2	21.1	26.4	20.1	529.7	151.5	378.2	71.4	57.4	5785	58.7	29.8	48.1	27.7	6:49
26.0	19.0	138.6	21.0	26.3	20.1	528.6	150.9	377.6	71.4	57.5	5795	57.9	28.9	49.4	28.4	6:51
27.0	20.5	149.5	21.0	26.2	20.1	527.2	150.3	376.9	71.5	57.6	5805	57.0	28.0	50.6	29.2	6:52
28.0	22.1	160.8	20.9	26.1	20.1	525.5	149.4	376.1	71.6	57.7	5819	56.1	27.1	51.8	29.9	6:54
29.0	23.7	172.4	20.8	26.0	20.2	523.6	148.5	375.1	71.6	57.8	5834	55.2	26.3	52.9	30.6	6:56
30.0	25.3	184.5	20.7	25.8	20.2	521.5	147.5	374.0	71.7	58.0	5851	54.3	25.4	54.0	31.3	6:58
31.0	27.1	197.1	20.6	25.7	20.2	519.2	146.5	372.7	71.8	58.1	5869	53.3	24.6	55.0	32.0	7:00
32.0	28.8	210.0	20.4	25.5	20.2	516.7	145.3	371.4	71.9	58.3	5889	52.3	23.8	55.9	32.6	7:03
33.0	30.7	223.3	20.3	25.4	20.3	514.1	144.1	369.9	72.0	58.4	5910	51.4	23.0	56.8	33.2	7:06
34.0	32.6	237.0	20.1	25.2	20.3	511.2	142.8	368.3	72.1	58.6	5933	50.4	22.2	57.7	33.8	7:09
35.0	34.5	251.2	20.0	25.0	20.3	508.1	141.5	366.6	72.2	58.8	5957	49.4	21.4	58.5	34.4	7:12
36.0	36.5	265.7	19.8	24.8	20.4	504.8	140.1	364.8	72.3	59.0	5981	48.4	20.6	59.3	35.0	7:16
37.0	38.6	280.7	19.6	24.6	20.4	501.4	138.6	362.8	72.4	59.2	6008	47.4	19.9	60.0	35.5	7:20
38.0	40.7	296.1	19.5	24.3	20.5	497.6	137.0	360.6	72.5	59.4	6036	46.4	19.2	60.7	36.0	7:24
39.0	42.8	311.9	19.3	24.1	20.5	493.6	135.4	358.2	72.6	59.6	6067	45.4	18.5	61.3	36.6	7:29

Motor performance calculations take ambient temperature and heating effects into account.

Color Key: Propeller Stalled Stall Speed @ Clmax=1.07 Level Flight @ Clopt=0.62 Level Flight @ Cl=0.45

Figure 7 in flight analysis with payload 80 % throttle

5.4 Final Rated Aircraft Cost Spreadsheet

The following is the spreadsheet analysis performed for the rated aircraft cost based on the parameters studied during the Taguchi analysis. The values included in the figure below show the final parameters, yielding a rated aircraft cost of $11.92 per thousand as shown below.

variables
set values

Description	value	variable	units
Manufactures Empty weight multiplier	300	A	$
Rated Engine Power multiplier	1500	B	$
Manufacturing Cost Multiplier	20	C	$/hour
Manufactures empty weight, Actual Airframe Weight With all flight and propulsion Batteries but without payload	9.55	mew	lb
Rated Engine Power	3.33125	rep	
Manufacturing Man Hours	202.695	MFHR	

MFHR break down

WBS 1 wings	125
WBS 2 Fuselage	52.6953
WBS 3. Empennage	20
WBS 4 flight systems	5

Calculation of Weight

Total Battery Weight	3.33125
Payload Weight	8.7824
Weight motor	1.21875
Weight rest of plane	5
Total Weight lbs	18.3324
Total Weight oz	293.3184
Total Empty Weight w/ payload lbs	13.7824
Empty weight in oz	220.5184

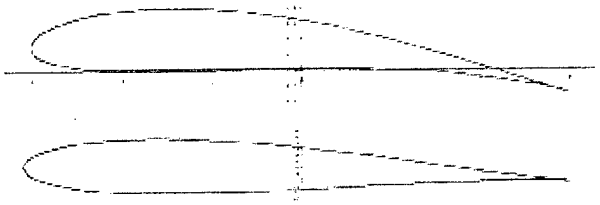
Description of Design

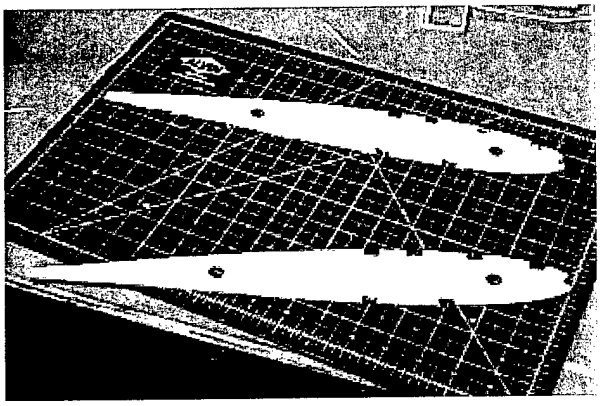
# of Engines	1
wing Span [ft]	8
Chord [ft]	1.5
# wings	1
water payload liters	4
Control Function multiplier.	1
fuselage body length [ft]	5.91667
fuselage width [ft]	0.59375
fuselage height [ft]	0.75
# Vertical Surfaces with no active control	0
# Vertical Surfaces with an active control	1
# horizontal surface	1
number of cells	26
Type of Battery Cell	subC
Current per battery cell mAH	2400
weight per battery cell	0.12813

Rated Aircraft Cost ($x1000) 11.92

6. Manufacturing Plan and Processes

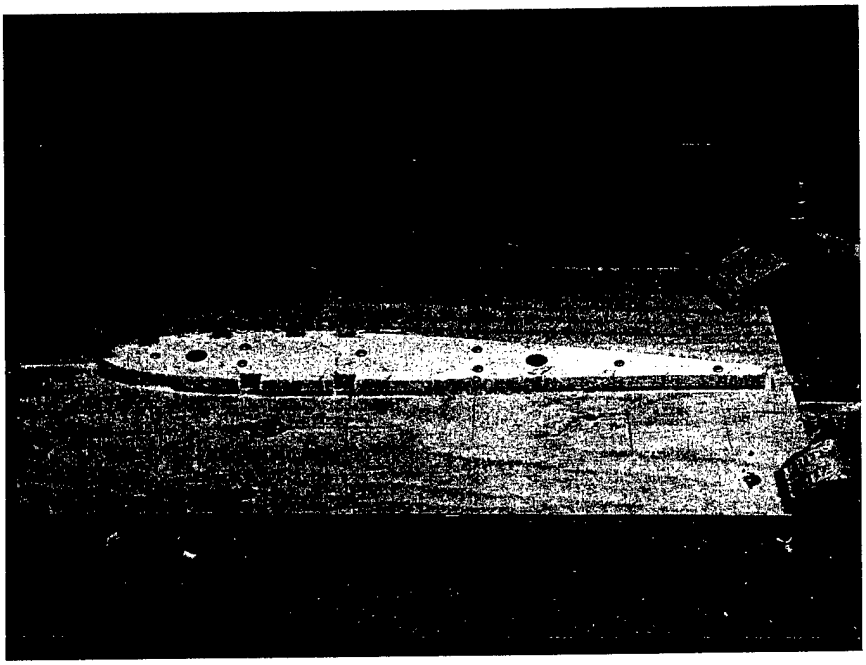
6.1 Wings Manufacturing



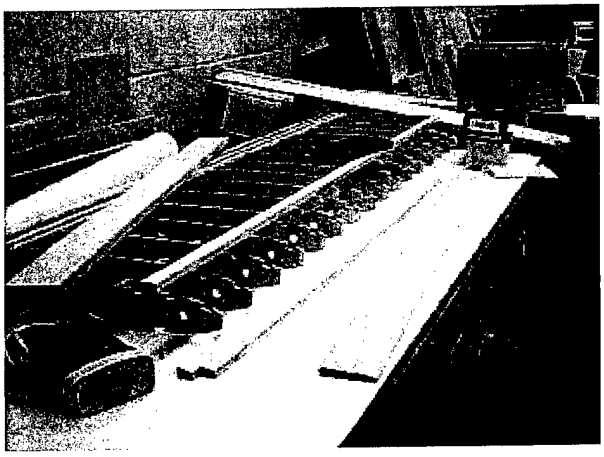


With the wings designed, we began to figure out the best way to manufacture the pieces and assemble everything. We used NASA's FoilSim II with our wing numbers from MotoCalc to verify that things looked correct and to derive coordinates. We graphed these coordinates in I-DEAS and, for the sake of construction, adjusted the bottom to be more nearly flat. This gave us more stability and the ability to flight at higher speeds. The lift loss was calculated to be the same as that gained by ailerons. Based on the research done and the reverse engineering processed we needed 42 pieces cut with precision, so cutting the airfoils manually was not an option. Instead, we fed the coordinates of the airfoil into the CNC milling machine to cut the ribs out of 1/16" thick pieces of balsa wood.



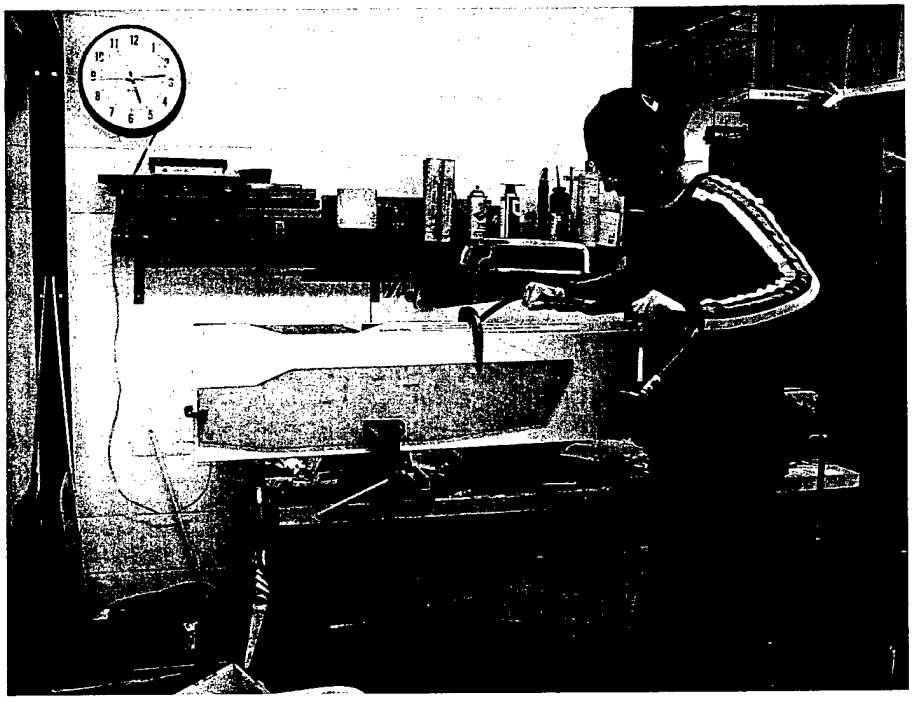


Once cut, these pieces were sanded down finely so as not to snag and split. With the ribs made and the plans drawn, we assembled the pieces into the wing shape. Near the fuselage, the ribs were doubled for strength, since they are be pierced by the connecting block connecting the wings to the fuselage. The leading edge (not pictured) is a pre-cut rounded nose shape of balsa. The ailerons are also balsa blocks, manually sanded to the correct shape. These pieces were relatively expensive and time consuming, but reliable and easy to work with later. The horizontal and vertical stabilizers are simply ribbed with leading and trailing spars for simplicity of the design. The rudder and elevator will be sanded like the ailerons. The wings and tail will all be covered with monokote.



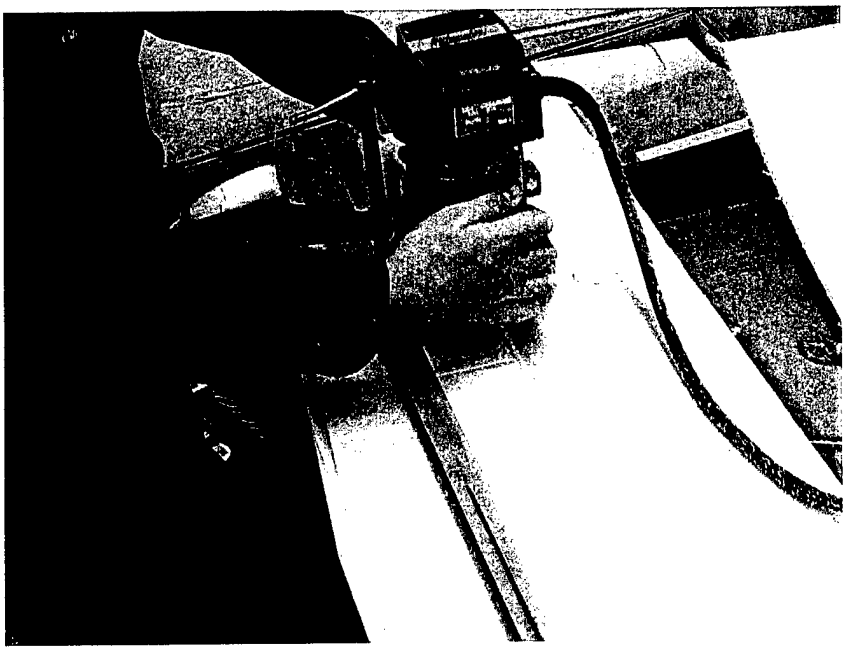
6.2 Fuselage Manufacturing

High-density foam proved highly available, sufficiently strong, and satisfactorily lightweight, especially considering its strength, to satisfy our design needs. The cutting method involved using a bow assembly with a resistance-heated wire connected to two 12-volt batteries. Large sheets were first cut to 1/2" thickness; then clamped between two wooden templates to be cut into the desired cross sections.



One important consideration in the manufacturing process of the fuselage was to ensure that the two sides of the fuselage were exactly identical. Also, the shapes should be repeatable in case a second fuselage was needed. Generally, a template-style manufacturing method was adopted whereby a model would be made of the shape of the sides of the fuselage. Specifically, two templates were made using a jigsaw and two ¼ inch sheets of plywood. Two fuselage walls were cut at one time to ensure their likeness. Once the two templates were aligned and attached using clamps and a vice, the foam profiles of the main fuselage were cut with the hot wire. The other, less complex shapes, were cut by an exacto knife and straightedge.

One innovation of our design was a removable bottom to the aircraft. This feature allowed easy access to the internal components and allowed for easy balancing as well as repairs. Balsa wood rails were used to guide and secure the bottom face into place. The balsa rails were inserted and glued into grooves along the sides of the fuselage walls created using a router and guide.



The corner connections between foam walls were notched using the same technique in order to increase the surface contact between sides. Standard Elmer's Glue was used for the foam-to-foam connections as regular modeling glues turned out to melt the foam. The shear strength Elmer's Glue turned out to be significant, and was found to withstand significant forces.



Finally, in order to ensure strength and durability of the fuselage, a coating of epoxy was given to the entire body of the aircraft. This provided a 'Hard Candy Shell', that served to both strengthen and protect the foam construction.

6.3 Payload Manufacturing

Rather than using a store bought tank, which would likely have possessed less than ideal dimensions, we chose to construct our own tank. Sticking with our foam construction methods we constructed a tank using ¼" foam panels that was 6.626" tall, 4.74" wide, and 11.25" long along the outside edges. The interior was just large enough to hold 4-liters of water. In order to achieve a water tight vessel, the inside of the container was coated with a very thin layer of epoxy. The end result was excellent. The container was perfectly sealed and extremely light.

6.4 Manufacturing milestones

An integral part of the manufacturing plan is the schedule. The following figure lists the major components of the aircraft and the time periods over which there were built.

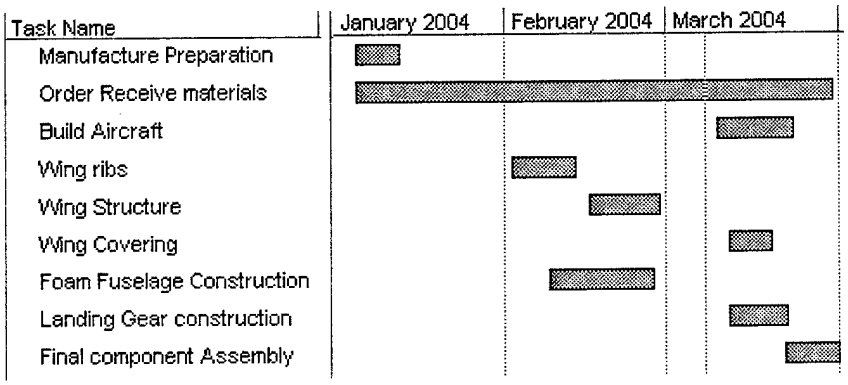


Figure 8: Manufacturing plan schedule

7. Testing Plan

Fall Semester:

	Testing Date / Period
Tank construction techniques:	November
Timed release of payload:	November
Foam fuselage construction tests	December

January:
'No test schedule due to School Holiday'

February:

	Testing Date / Period
Computer Model of Wing	Feb. 6th, finite element analysis (IDEAS)
Computer Model of Fuselage	Feb. 13th, finite element analysis (IDEAS)
Computer Model of Boom and Tail	Feb. 20th, finite element analysis (IDEAS)

March:

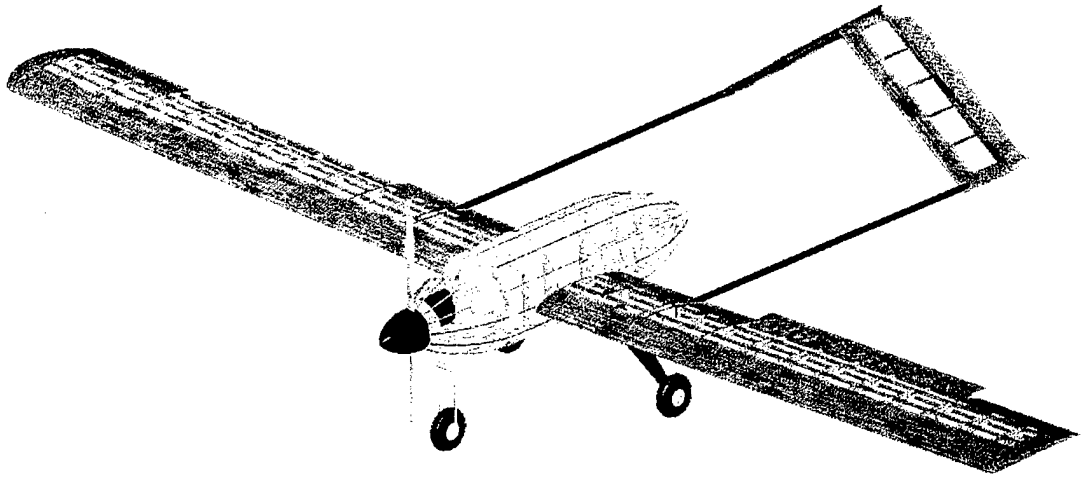
	Testing Date / Period
REPORT DUE!	N.A.
Aircraft Balancing	March 13th, establish center of gravity, fine adjustment of flexible components (ie. batteries)
Wing Tip Test	March 20th, simulation of competition test, assure structural integrity of aircraft
Component Testing	March 20th, testing of flaps, ailerons, rudder, drop mechanism, channel mixing, ect.
First Flight Test (w/o payload)	March 26th-27th, Flight testing at Brooklyn Airfield, basic take-off and landing exercises

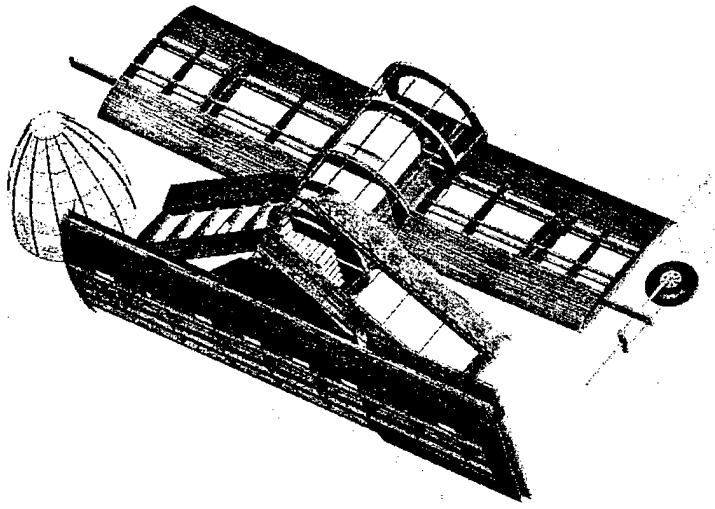
April:

	Testing Date / Period
Second Flight Test (w/ payload)	April 2nd, basic take-off and landing exercises with payload. Incremental increase in payload volume.
Third Flight Test (w/ and w/o payload)	April 9th, simulate missions, time trials, PEDAL TO THE METAL!
Final Adjustments (and Repairs?)	April 16th through 18th, final adjustments and repairs before competition.
COMPETITION!!!	SEE YOU THERE.

2003/2004 AIAA
CESSNA/ONR DESIGN BUILD FLY COMPETITION
DESIGN REPORT

POSEIDON'S FURY





THE UNIVERSITY OF CENTRAL FLORIDA
DEPARTMENT OF
MECHANICAL, MATERIALS, AND AEROSPACE
ENGINEERING

Table of Contents

1.0	**EXECUTIVE SUMMARY**		**1**
	1.1	**Conceptual Design Summary**	**1**
		1.1.1 Conceptual Design Alternatives	1
		1.1.2 Conceptual Design Tools	1
		1.1.3 Conceptual Results	1
	1.2	**Preliminary Design Summary**	**1**
		1.2.1 Preliminary Design Alternatives	1
		1.2.2 Preliminary Design Tools	2
		1.2.3 Preliminary Results	2
	1.3	**Detail Design Summary**	**2**
		1.3.1 Detail Design Tools	2
		1.3.2 Detail Design Results	2
2.0	**MANAGEMENT SUMMARY**		**2**
	2.1	**Division Responsibilities**	**3**
		2.1.1 Propulsion	3
		2.1.2 Aerodynamics	3
		2.1.3 Structures	3
	2.2	**Scheduling and Information Management**	**5**
3.0	**CONCEPTUAL DESIGN**		**5**
	3.1	**Mission Requirements**	**5**
	3.2	**Structural Analysis**	**5**
		3.2.1 Possible Structural Solutions	6
		3.2.2 Structural Comparison Analysis	8
	3.3	**Aerodynamic Analysis**	**11**
		3.3.1 Possible Aerodynamic Solutions	11
		3.3.2 Aerodynamic Comparison Analysis	12
	3.4	**Propulsion Analysis**	**16**
		3.4.1 Possible Propulsion Solutions	17
		3.4.2 Propulsion Comparison Analysis	18
	3.5	**Conceptual Summary**	**20**

4.0 **PRELIMINARY DESIGN** 22

4.1 Mission Model 22

4.2 Aerodynamic Analysis 22

4.2.1 Airfoil Parameters and Sizing Trades 22

4.2.2 Airfoil Analysis and Optimization 23

4.2.3 Wing Parameters and Sizing Trades 23

4.2.4 Wing Analysis and Optimization 24

4.2.5 Empennage Parameters and Sizing Trades 25

4.2.6 Empennage Analysis and Optimization 26

4.2.7 Flight Mechanics 26

4.3 Structural Analysis 26

4.3.1 Fuselage and Drain System Parameters and Sizing Trades 27

4.3.2 Fuselage and Drain Systems Analysis and Optimization 27

4.3.3 Wing Attachment Analysis and Optimization 28

4.3.4 Main Landing Gear Analysis and Optimization 34

4.4 Propulsion Analysis 37

4.4.1 Battery Parameters and Sizing Trades 37

4.4.2 Motor Parameters and Sizing Trades 38

4.4.3 Propeller Parameters and Sizing Trades 39

4.4.4 Propulsion Analysis and Optimization 40

4.5 Predicted Mission Performance 42

5.0 **DETAIL DESIGN** 42

5.1 Component Selection/System Architecture 42

5.2 Sized Aircraft Data Tables 43

5.2.1 Geometry Data Table 43

5.2.2 Performance Data Table 44

5.2.3 Weight Statement Table 45

5.2.4 Systems Data Table 45

5.3 Final Aircraft Specifications/Drawing Package 45

Top Assembly 46

Exploded View 47

Stowed Configuration 48

Custom Valve Assembly 49

Propulsion Schematic 50

5.4 Final Aircraft RAC Calculation 51

6.0 MANUFACTURING PLAN 51
6.1 Major Component Manufacturing 51
6.2 Skill Matrix 52
6.3 Manufacturing Milestones – Planned and Actual 52
6.4 Cost Analysis 53

7.0 TESTING PLAN 54
7.1 Test Objectives and Schedule 55
7.2 Flight Testing Checklists 55
7.3 Summary of Test Results 56
REFERENCES 57

List of Figures

Figure 2.1: Management Tree for *Poseidon's Fury* 3
Figure 2.2: UCF Project Timeline 4
Figure 3.1: Valve Considerations 7
Figure 3.2: Wing Attachment Considerations 8
Figure 3.3: C_L Comparison Plot 15
Figure 3.4: Conceptual Aircraft Solid Model 21
Figure 4.1: Flight Mechanics Data 26
Figure 4.2: Custom Stereolithography Ball Valve 28
Figure 4.3: 3-D Model of Wing Attachment 31
Figure 4.4: Distributed Load Longitudinal Stresses 31
Figure 4.5: Distributed Load Displacement 32
Figure 4.6: Wing Tip Load Longitudinal Stresses 32
Figure 4.7: Wing Tip Load Displacement 33
Figure 4.8: Main Landing Gear Load Application 34
Figure 4.9: Aluminum Main Gear Maximum Von Mises Stress 35
Figure 4.10: Maximum Tensile Stress for Carbon Fiber Gear 36
Figure 4.11: NiCad Battery Comparison 38
Figure 4.12: Propulsion Test Setup 41
Figure 5.1: Turn Radius as a Function of Velocity and Load Factor 44
Figure 5.2: Turn Rate as a Function of Velocity and Load Factor 44
Figure 6.1: UCF Manufacturing Timeline 52
Figure 6.2: Various views of completed aircraft 54

List of Tables

Table 3.1: Structural Decision Matrix 9
Table 3.2: Aerodynamics Decision Matrix 14
Table 3.3: Propulsion Decision Matrix 19
Table 3.4: RAC Worksheet 20
Table 4.1: Modified Eppler 216 Parameter Values 21
Table 4.2: Wing Parameters 24
Table 4.3: Optimized Parameters 24
Table 4.4: Stability and Empennage Parameters 25
Table 4.5: Empennage Optimized Parameters 26
Table 4.6: Fuselage and Tank Parameters 27
Table 4.7: Summary of Maximum Normal Stresses 33
Table 4.8: Sanyo Battery Specifications 37

Table 4.9: Sanyo Actual Useful Capacity 38
Table 4.10: Motor Parameters 39
Table 4.11: Current vs. Power Available w/ 75% Eff. Propeller 39
Table 4.12: Propulsion Test Summary 41
Table 4.13: Predicted Mission Performance Parameters 42
Table 5.1: Geometry Data Table 43
Table 5.2: Performance Data Table 45
Table 5.3: Weight Statement Table 45
Table 5.4: System Data Table 45
Table 5.5: Final Rated Aircraft Cost 51
Table 6.1: Skill Level Matrix 52
Table 6.2: Manufacturing Cost Breakdown 53
Table 7.1: Test Objectives and Schedule 55
Table 7.2: Pre-Flight Checklist 56
Table 7.3: Flight Testing Checklist and Schedule 56
Table 7.4: Summary of Test Results 56

1.0 EXECUTIVE SUMMARY

This report discusses the design and development of *Poseidon's Fury*, the 2004 University of Central Florida entry to the AIAA Design/Build/Fly competition. The following outline introduces the alternatives, tools, and results used or found in each of the three design phases.

1.1 Conceptual Design Summary

The conceptual summary outlines the first part of the design process. It highlights special considerations and final decisions that transferred to the preliminary design phase.

1.1.1 Conceptual Design Alternatives

The conceptual design phase highlights major aircraft components and selects several design alternatives for comparison. It also introduces the Rated Aircraft Cost (RAC) as an important scoring factor in the competition. At this stage, the structures team focused on the comparison analysis of five aircraft components: the tank, drain orifice, fuselage cross-section, and wing attachment. The aerodynamics team investigated the number of wings, wing location, planform area, wing tips, empennage, and control surfaces. Finally, the propulsion team compared three different motor parameters and two different propeller parameters.

1.1.2 Conceptual Design Tools

Decision matrices were the prime conceptual design tool although simple calculations and SolidWorks sketches gave perspective on each of the parameters. Figures of merit were chosen for each component and weighted according to importance. These matrices helped the team organize ideas and come to a reasonable final result.

1.1.3 Conceptual Results

At the end of the conceptual design phase, *Poseidon's Fury* contained a molded fiberglass tank, an ball valve as the drain orifice, a donut shaped wing attachment, and a circular cross-section for the fuselage. It also had a mid-mounted, single wing configuration with shaped tips, a rectangular planform, and flaperons for control surfaces. The propulsion system was narrowed down to a single motor/single propeller system with fixed vectoring and direct drive. The propeller was chosen to be composite and two bladed. At this point, more analysis was required for the empennage design but alternatives were reduced to a conventional or an inverted v-tail configuration.

1.2 Preliminary Design Summary

The preliminary design phase took the conceptual design alternatives and further investigated their strengths and weaknesses. It also established good estimates for component sizes.

1.2.1 Preliminary Design Alternatives

The preliminary design started with a mission model of the desired aircraft performance. With that in mind, the structures, aerodynamic, and propulsion teams began making sizing trades for each component. After recognizing the approximate size and shape of the aircraft, the structures team looked

for weak points in the structure; while the aerodynamics team analyzed stability characteristics, and the propulsion team continued analysis of the motor, propeller, and batteries.

1.2.2 Preliminary Design Tools

For the preliminary design, the aerodynamics team developed a computer program to combine the possible aerodynamic parameters. The team used this information to rank the combinations in terms of RAC and then decided the best configuration to meet the desired performance specified in the mission model. The structures team used basic stress calculations and CosmosWorks finite element analysis software to define the weak points in the structure and assure that the materials would last in flight.

1.2.3 Preliminary Results

At the conclusion of the preliminary stage, *Poseidon's Fury* had a 9.33 ft wingspan with an Eppler 216 modified airfoil and an inverted V-tail. Additionally, basic aerodynamic properties were calculated. Further investigation into the tank/valve configuration proved that a sculpted, non-rectangular tank and custom valve design were necessary to achieve the desired drain time of the mission model. Failure points of the aircraft structure were found at the wing joint and landing gear, and the design was modified to accommodate the anticipated stresses. Finally, the results of the propulsion analysis showed that an AstroFlight Cobalt 60 geared motor with a narrow range of propeller dimensions and approximately 2.5 lbs of Sanyo 1700 batteries could provide enough thrust for the duration of the flight specified in the mission model.

1.3 Detail Design Summary

The detail design aimed to provide the final configuration for all aircraft components. A drawing package at the end of the detail section displays the final configuration.

1.3.1 Detail Design Tools

Tools for the detail design included SolidWorks and AutoCAD 2000. The team used SolidWorks to produce the 3-D models and AutoCAD to produce the 2-D assembly drawings

1.3.2 Detail Design Result

At the conclusion of the detail design, the RAC was calculated to be 12.45. Major components selected were similar to the preliminary results. The flight performance characteristics were summarized in Table 4.14.

2.0 MANAGEMENT SUMMARY

In order to increase group efficiency, Team *Poseidon's Fury* was divided into the 3 subgroups (Figure 2.1). The divisions consisted of: Structures, Aerodynamics, and Propulsion. Each division included a lead design engineer, additional design engineers, and underclassmen. The head engineers were to, within their division, oversee all progress, make final design decisions, and report statuses periodically to the Captain. The Captain managed the team through the head engineers to ensure clear communications and steady progress. Even though this structure was put in place, each member was

involved at some point in each of the areas. This was partly to ensure that all members would be familiar with each part of the design for future projects.

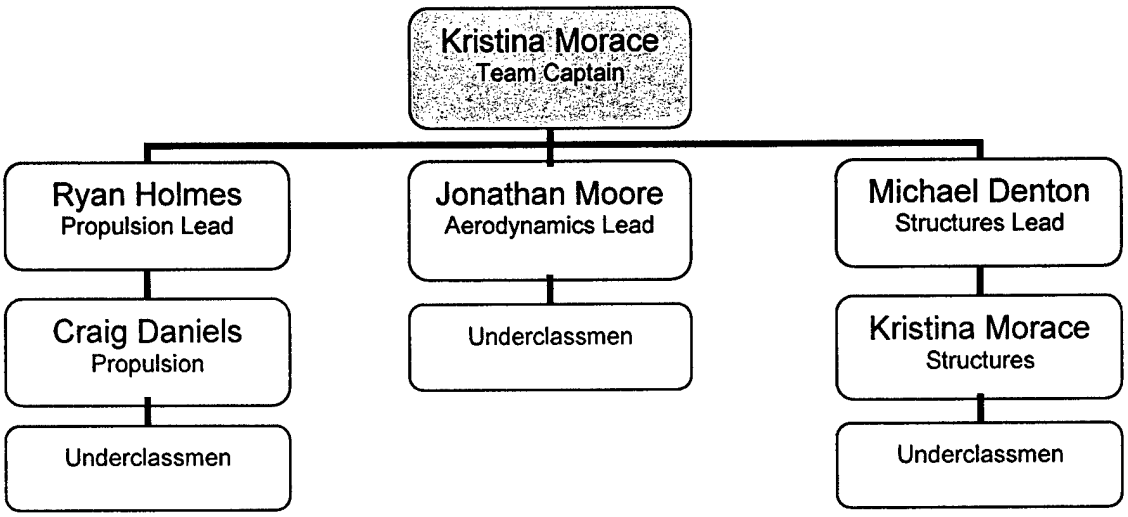


Figure 2.1: Management Tree for *Poseidon's Fury*

2.1 Group Responsibilities

Each group was given certain aspects of the airplane to study, design, and construct. While most of the work was conducted individually, some interdivision cooperation was needed to settle disputes over large design aspects of the airplane. Each individual group will also be responsible for the repair of their individual components during test flights and competition.

2.1.1 Propulsion

This group focused on all of the project's electronics. This included choosing which model motor to use for the aircraft and which, if any, gearbox with which to supplement it. Once a motor was selected the division's next job was to determine the most efficient battery configuration considering power output, weight, and cost. Then the division had to decide the optimal propeller pitch and diameter. Propulsion was also responsible for the wiring and testing of these components.

2.1.2 Aerodynamics

The main responsibility of this group was studying and designing all aspects of the airplane that were affected by any aerodynamic forces. This included choosing the airfoil type, wing and control surface dimensions, and empennage configurations. The group studied and tested different possibilities and chose the best design based on stability, drag, and fuselage integration. It also had to cooperate with the structures team in analyzing the aerodynamics of the airplane's main body and its overall effects.

2.1.3 Structures

The structures group was responsible for making the aircraft strong, durable, and lightweight. It focused on designing the fuselage, tank, drain orifice, and all attachment points necessary to the modular design.

ID	Task Name
1	**DESIGN**
2	Projected Conceptual Design
3	Actual Conceptual Design
4	Projected Preliminary Design
5	Actual Parametric
6	Projected Detail Design
7	Actual Detail Design
8	Design Complete
9	**TESTING**
10	Projected Drain Tests
11	Actual Drain Tests
12	Projected Spar Testing
13	Actual Spar Testing
14	Projected Motor/Prop Tests
15	Actual Motor/Prop Tests
16	Projected Flight Testing
17	Actual Flight Testing
18	Testing Complete
19	**CONSTRUCTION**
20	Projected Tank Construct.
21	ActualTank Construct.
22	Projected Wing Construct.
23	Actual Wing Construct.
24	Projected Fuselage Construct.
25	Actual Fuselage Construct.
26	Projected Tail Construct.
27	Actual Tail Construct.
28	Projected Comp. Integration
29	Actual Comp. Integration
30	Redesign
31	**2004 DBF REPORT DUE**
32	**2004 DBF COMPETITION 4/23-4/25**

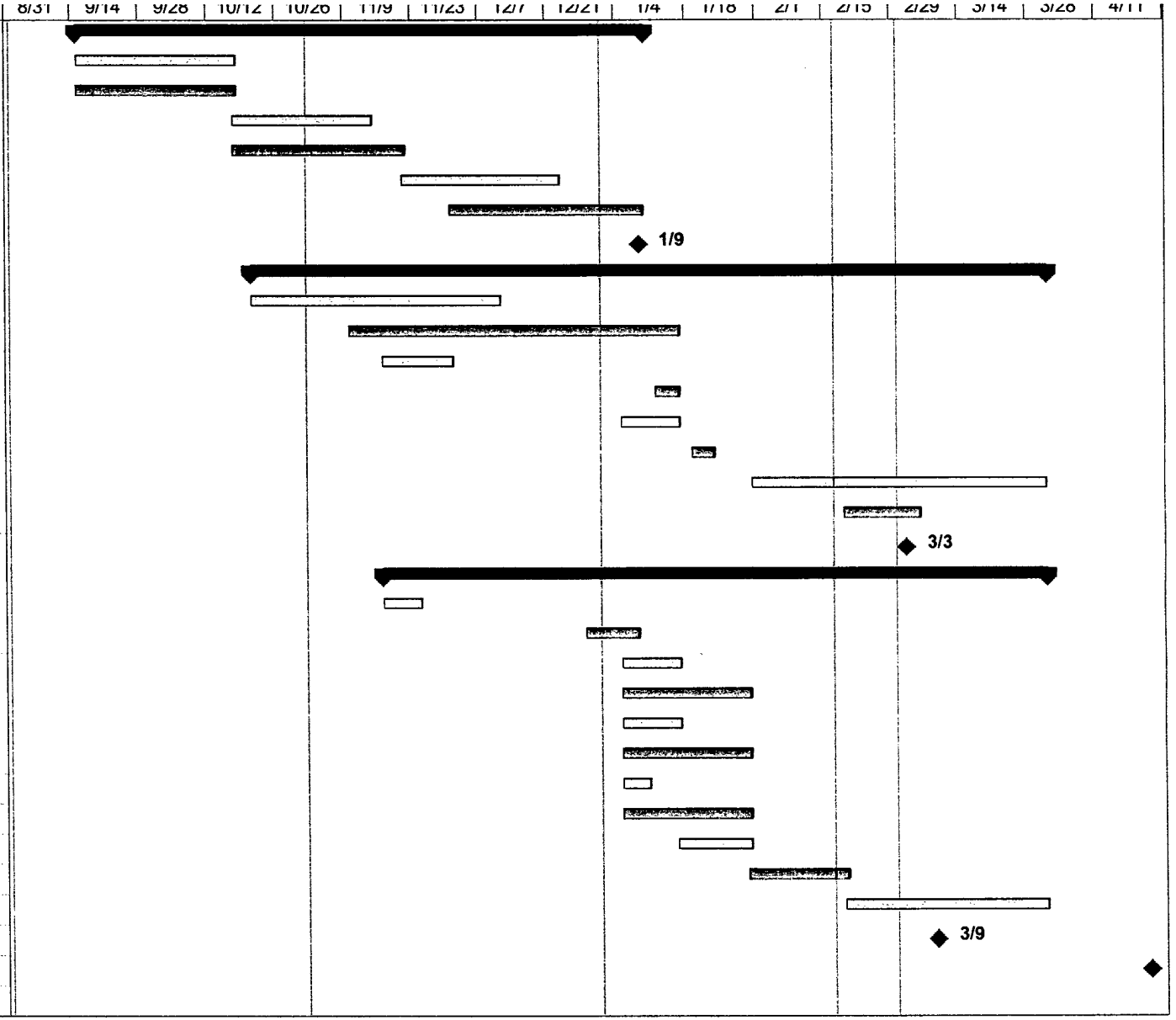


Figure 2.2 UCF Project Timeline. Green bars represent proposed timeframes. Red bars represent actual timeframes. Black Diamonds represent milestone markers.

It also chose the materials for each aircraft component and tested the weakest points of the design. Finally, the structures group produced the drawing package and assembly drawings for manufacturing.

2.2 Scheduling and Information Management

To ensure project progress a timeline was made for individual tasks and was broken up into three phases: design, construction, and testing. Each task had a planned timeframe and was given a deadline for completion. These deadlines and the AIAA Design/Build/Fly competition were added to the end of Figure 2.2 as milestones. More detailed schedules of manufacturing and testing can be found in sections 6.0 and 7.0.

3.0 CONCEPTUAL DESIGN

The conceptual design stage focused on determining the general aircraft components according to mission requirements. All three design groups, structures, aerodynamics, and propulsion, assigned important parameters to the aircraft components and compared the components through simple calculations and general aerodynamic and structural knowledge. The parameters were assigned weighting factors numbered 1-5 corresponding with "unacceptable" to "excellent" respectively, allowing components to be judged in a final conceptual decision matrix.

3.1 Mission Requirements

The AIAA Student Design/Build/Fly Competition requires each aircraft to complete two missions. Mission 1 is a heavy lift mission requiring the aircraft to fly two laps around a prescribed course while carrying up to four liters of water and dumping that water on the downwind leg. During this mission, water must not spill or leak from the aircraft or the team will suffer a time penalty of three minutes. Mission 2 is simply a time trial mission requiring the aircraft to complete four laps around the course unloaded. In both missions the aircraft must be equipped with an AstroFlight or Graupner style electric motor, contain less than 5 lbs of batteries, draw less than 40 amps, and weigh less than 55 lbs. Finally, the aircraft is required to fit inside a 4'x2'x1' box.

3.2 Structural Analysis

The strength and durability of the aircraft structure become more important when carrying a heavy load and flying in adverse weather conditions. If the weather conditions were unfavorable, the aircraft must be secure enough to stay together during flight and strong enough to withstand a hard landing. Therefore, desired outputs focused on strength constrained by only the essential aerodynamic and propulsive outputs and by the constraints of the box. Undesirable outputs resulting in an unsafe or under-supported design were discarded in the decision matrices.

The five general components chosen for structural analysis at this stage were the aircraft's water tank and drain orifice, fuselage cross-section, wing attachment, and landing gear. Analysis began by researching both off-the-shelf and easy-to-manufacture water tanks and drain orifices. Since 4 liters and 0.5 inches are the maximum allowable water volume and drain orifice diameter, respectively, only tanks large enough to hold four liters and valves meeting the diameter requirement were considered. The tank

directly or indirectly affected analysis of the four other components; in general, the aircraft's structural concept was designed around the aircraft's payload. The fuselage cross-section refers to the frontal shape of the fuselage, and wing attachment refers to the structural member that connects the wing and fuselage together. Landing gear analysis concentrated solely on the main gear.

3.2.1 Possible Structural Solutions

Tank: Due to the decision that the aircraft would be designed around the tank, a good tank design choice was integral to the overall design of the aircraft. Sloshing was considered first to reduce the dramatic effect it could have on the aircraft flight characteristics. The team determined that incorporating baffles into the design would result in the most ideal conditions, reducing or eliminating sloshing all together. Secondly, the size of the tank was considered. Since the size of the fuselage depended on the size of the tank, the tank must be compact to keep the fuselage cross-section to a minimum and in turn increase the team's overall score. Research revealed that a good base for the tank cross-section started at a size of 6 inches in both width and height. This cross-section allows the fuselage's frontal profile to be held to approximately 8 inches, which will give the aircraft an aesthetically pleasing look without having a large effect on the RAC. The competition limits the amount of water to be carried in the tank to 4 liters or approximately 1 gallon. It was decided early in the conceptual phase that the design would incorporate the maximum load allowed since this allows the team to modify the amount of water carried up to the maximum. With that in mind, if the team finds that carrying less water results in a higher overall score then it will have the option of adjusting the amount of water carried without having to modify the design of the aircraft.

Starting with a one-gallon payload, several different choices for tanks were found. The first tank consideration was a polyethylene terephthalate (PET) plastic tank. This type of tank is durable, inexpensive, and easily obtained from any number of container supply centers. A one-gallon PET tank was found with dimensions that fit the team's goal of a fuselage cross-section no larger than 8 inches. The next tank design considered was a bladder. Essentially, this is nothing more than a heavy-duty balloon carried inside the fuselage. A balloon-like tank adds an elastic force assisting in deploying the payload but lacks rigidity and dependability against leaks during flight. A third tank alternative was an off-the-shelf juice bottle. A juice bottle was found with dimensions that would fit within the aircraft's fuselage design. This provides a tank that is easily obtained and inexpensive. The final tank considered was a self-manufactured fiberglass tank that could be built to the dimensions and shape that the team decided was best.

Drain Orifice: The next area of consideration for the structural team was the drain orifice. As mentioned previously, the competition limits the size of the drain orifice to 0.5 inches and does not allow assisted draining, i.e. no pumps are allowed. The first of the five different types of valves considered was the ball and string type valve. As can be seen in Figure 3.1, the servo pulls on the twine which is connected to a rubber ball that forms a seal with the bottom of the fuselage. The next valve considered was a trap door type design. The door would be mounted flush with the fuselage and hinged to allow rotation. A pin

would secure the door in the up and sealed position. This pin would be connected to a servo which when activated would release the pin allowing the door to drop open thus releasing the water payload. Another consideration was a ball valve. This would be mounted to the tank with a servo connected to the arm of the ball valve. Rotation of the arm opens and closes the valve as needed. After considering the ball valve, a solenoid valve seemed a valuable option to explore. Although similar to the ball valve, the solenoid valve uses its own motor and power source for valve activation instead of a servo. The final consideration was an opening at the top of the fuselage that acted as a fill point and dump point. The aircraft would fly inverted and dump the payload while upside down.

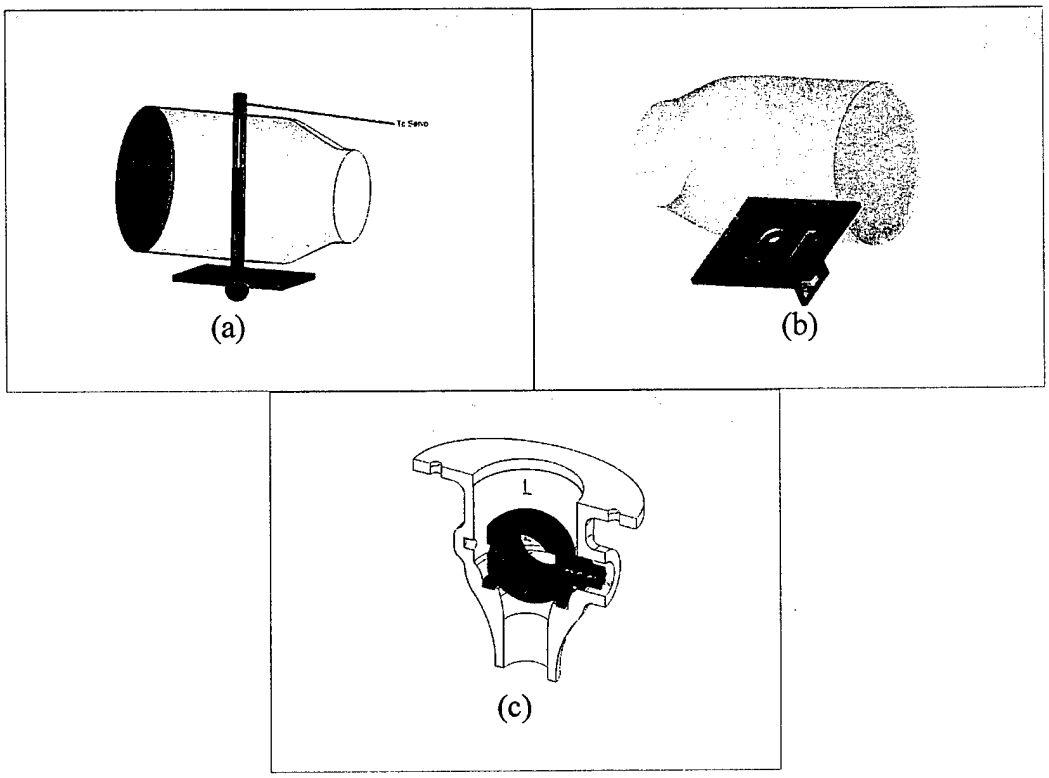


Figure 3.1: Valve Considerations: (a) Ball and String, (b) Trap Door, (c)Ball Valve

Fuselage Cross-Section: The primary design considerations for the fuselage were a rectangular profile, blended-wing profile, and a circular profile. The rectangular fuselage is a typical R/C type flying "box". The blended wing as its name states blends the wing and the fuselage together. There is little to no separation with respect to the fuselage and wing. This type provides a smooth transition from wing to fuselage. The final type of fuselage considered was the circular type. This is the typical fuselage used in many commercial airliners.

Wing Attachment: The wing attachment was the most crucial component in the structural analysis of this airplane. This was primarily due to the concentration of loads and stresses encountered in this area. Furthermore, the wing attachment is also dependent on what type of wing configuration is used. Two types of wing attachments were considered. The first attachment considered was the pass-through design. This design incorporates a spar which passes through the fuselage to the opposing wing. The

other option would be the "donut". As can be seen in Figure 3.2, this design allows the tank to sit within the fuselage between the wings. The wings are attached to the protruding stems.

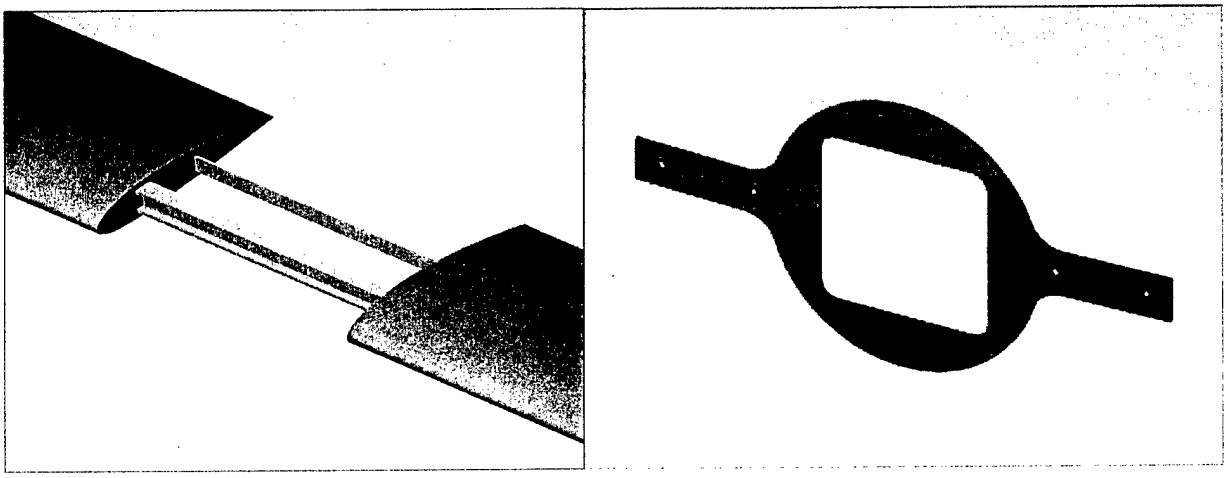


Figure 3.2: Wing Attachment Considerations: Left View: Pass-Through, Right View: Donut

Landing Gear: Considerations for different types of landing gear were constrained to off-the-shelf R/C type landing gear. The three landing gear configurations considered were the fixed tricycle, retractable tricycle, and pneumatic fixed tricycle landing gear.

3.2.2 Structural Comparison Analysis

The most common parameters chosen for structural analysis included manufacturability, durability, cost, and weight. Manufacturability refers to the ease of manufacturing according to the skills available to the team; durability refers to the components' ability to withstand the competition elements; and cost and weight refer to the actual manufacturing cost and the RAC of the aircraft, respectively. Other parameters chosen were stability, drag, reliability, and aesthetics. These parameters are component specific and are explained further in each analysis section. A decision matrix of each structural comparison can be found in Table 3.1.

Table 3.1: Structural Decision Matrix

		Manufacturability	Durability	Stability	Reliability	Weight (RAC)	Size (RAC)	Drag	Cost	Aesthetics	TOTAL SCORE	FINAL OUTCOME
Tank	Juice Bottle	2	3	5	3	4	2	-	5	-	24	Fiberglass
	PET Bottle	4	3	5	3	5	3	-	5	-	28	
	Bladder	5	1	1	2	5	5	-	5	-	24	
	Fiberglass	4	4	5	5	4	5	-	4	-	31	
Drain Orifice	Ball & String	2	2	4	2	3	-	-	4	-	17	Ball Valve
	Trap Door	3	3	3	2	3	-	-	3	-	17	
	Solenoid	4	4	5	4	2	-	-	3	-	22	
	Ball Valve	5	5	5	4	5	-	-	5	-	29	
	Upside Down Dump	5	2	1	1	5	-	-	5	-	19	
Fuselage	Rectangular	5	-	-	-	-	-	3	4	3	15	Circular
	Circular	4	-	-	-	-	-	5	4	4	17	
	Blended	2	-	-	-	-	-	4	3	5	14	
Wing Attachment	Donut	4	4	-	-	4	-	-	-	-	12	Pass-Through
	Pass-Through	5	4	-	-	4	-	-	-	-	13	
Landing Gear	Fixed	5	4	5	5	5	-	3	5	2	34	Fixed
	Retracts	3	3	3	3	4	-	5	2	5	28	
	RoboStruts	4	5	5	5	3	-	2	3	4	31	

5-Excellent, 4-Good, 3-Fair, 2-Poor, 1-Unacceptable

Tank: The first discarded tank concept was the bladder. It received an excellent ranking in both manufacturability and weight but held an unacceptable mark in durability and stability. On a rough landing or turbulent flight, the bladder could rupture and leak within the fuselage. However, this was unacceptable in terms of the overall design and in terms of the competition rules. Also, while the bladder payload empties, the bladder itself changes size and could move within the fuselage causing instability in flight. The juice bottle was the next discarded concept. Although rigid and durable, unlike the bladder, it received poor scores in manufacturability and size. Since the juice bottle is "off-the-shelf", it is difficult to incorporate baffles into its pre-made design. It is also impossible to change the size and shape of the bottle for future modifications.

The last two possible designs, the PET bottle and the fiberglass tank, scored no poor or unacceptable marks and fell closely together in total score. The fiberglass design excels in reliability and size. It can be molded and is seamless allowing flexibility in the preliminary design and preventing leaks from connecting parts. Overall, the fiberglass design is the best design option for the tank.

Drain Orifice: The first abandoned drain concept was the "upside-down dump". As stated previously, this concept requires the aircraft to fly inverted while deploying the payload. It received unacceptable scores in both stability and reliability because the pilot would be required to roll an aircraft carrying a heavy load then keep it steady while dumping. The trap door concept was abandoned quickly thereafter. It scored only fair and poor marks in each parameter, including a poor mark in reliability. Reliability for this design

and for the next three valve concepts refers to the likelihood of the valve to leak. Note that leaks are penalized according to competition requirements. Similarly, the ball-and-string concept received a poor mark in reliability. The solenoid valve and ball valve both had good reliability ratings and excellent stability ratings. The main difference between the two is weight. The solenoid valve is much heavier as it contains a motor and electrical package, whereas, the ball valve would only require a small servo and is powered from the servo battery pack. Due to the ball valve's excellent ratings, it was further investigated in the preliminary design phase.

Fuselage: Only three options were considered for the fuselage cross-section. A rectangular cross-section is much easier to manufacture than a circular or blended section giving it an excellent manufacturability rating. But, its sharp corners give it only fair drag and aesthetic qualities. In terms of aesthetics, a blended body has excellent appeal, but in terms of manufacturability it has difficult building qualities. From looking at the total scores, the best cross-section design appeared to be the circular cross-section. It had good and excellent scores in all applicable parameters.

Wing Attachment: As mentioned previously, there are two possible options for the wing attachment design: the "donut" and pass-through. The pass-through is an excellent choice for low and high wing attachments. It is a proven design and simple to manufacture. However, if the wing were placed in the center of the fuselage (mid-wing), the water tank must be compromised to accept the pass-through. This increases the level of difficulty in manufacturing the tank to accept the pass-through without leaking. Therefore, the "donut" is a better design for the mid-wing setup. Durability of both designs is equal; they both allow the bending loads created by the wings to be distributed evenly throughout the aircraft in a similar manner. Also, the weight of either setup was similar and was therefore given a score of good. As expected, the overall scores for both wing attachment designs were similar.

Landing Gear: Early in the design comparison a tricycle landing gear design was chosen to best support the plane. None of the main gear options compared were unacceptable, but the fixed gear received a poor rating in aesthetics. Weighed against this option, the retractable landing gear had the best aesthetic appeal, for it hides the gear internally during flight. However, aesthetics was its only excellent rating. It received fair marks in manufacturability, durability, stability and reliability. Durability, stability, and reliability in this section refer to the performance of the gear on the runway. It is much more difficult to install a retract configuration than a fixed configuration, and retracts are more likely to collapse on landing. Fixed gear scored good and excellent ratings in all sections except for drag and aesthetics. Therefore, the fixed gear was the best concept for this aircraft.

Empennage Attachment: Three options were considered for the attachment of the empennage to the fuselage. The single boom attachment scored the lowest of the three options. It was given a fair rating with respect to aesthetics and durability. The twin-boom attachment scored slightly higher since it was decided that two tail booms are aesthetically more pleasing then a single boom. Also, the twin boom is more desirable than a single boom because it prevents tail flutter which is a problem commonly encountered in single boom designs from past UCF entries. The extended fuselage design only scored

fair ratings in manufacturing and weight due to the added difficulty in designing and building as well as the added weight encountered with this design.

3.3 Aerodynamic Analysis

Aerodynamic considerations of an aircraft were at best difficult and require experience and detailed analysis. Every component of the design affects the overall aerodynamic performance of the aircraft [2]. Variables kept in mind throughout the design included air density, aircraft speed, and aircraft weight [3]. These variables later helped produce important outputs of lift and drag and, ultimately, the overall aircraft performance. Limitations to the overall aerodynamic design occurred when overlapped with propulsion and structural constraints. For instance, the size of the fuselage is governed by the amount of payload and flight hardware that must be carried. Increasing the size of the fuselage increases the amount of form drag of the aircraft, degrading performance [3]. Nevertheless, a conceptual analysis dictated that an overall study of the aerodynamics take place. To accomplish this, only the major aerodynamic surfaces—the wing parameters and the tail configuration—were analyzed. Additionally, the possible aerodynamic routes were judged upon performance in straight and level flight and turning flight. The mission criteria were also studied to provide a conceptual overview of the required performance envelopes. These criteria highlighted ideas that were further investigated and eliminated ideas that vastly degraded either airplane performance or cost.

3.3.1 Possible Aerodynamic Solutions

Number of Wings: The first solution considered for the aerodynamic team was how many wings to use. There were two possible solutions. The first was a single wing design and is typical of what is seen in most modern aircraft. Another possible solution was using multiple wings such as a tandem or bi-wing design.

Wing Vertical Location: The vertical location of the wing on the fuselage is more a choice of preference than of aerodynamics [5]. Structural factors vastly affect this parameter as opposed to aerodynamic effects. However, some effect does still exist with respect to drag and stability [10]. Three options were chosen for the vertical location of the wing on the fuselage. The low-wing configuration places the fuselage on top of the wing. The high-wing configuration is the inverse of the low-wing design and is typical of what is seen in many radio controlled and civilian trainer aircraft. The mid-wing configuration is characterized as attachment at any location between the high-wing and low-wing.

Wing Planform: The next area of consideration was the wing horizontal planform. This parameter was analyzed due to its effect on the overall lift and the drag of the aircraft. Additionally, the lift distribution created by the horizontal planform has a large effect on the structure and weight of the wing [2]. There were three possible solutions with respect to the wing planform. The first was what is known as the tapered configuration. This type of wing planform creates a wing that tapers from the root near the fuselage to the wing tip. This planform is typically seen on many sailplanes. The next option was the reverse tapered planform. The reverse tapered planform is similar to the tapered planform. The final

consideration for wing planform was the non-tapered or "straight wing". As its name implies, this is a wing which has no change from root chord to tip chord.

Wing Tip Configurations: When considering the design of the wing tip, there were three typical options. First, the clipped wing tip design does not incorporate any modifications to the wing tip. In other words, the wing tip chord is the same as the root chord. Another option was placing winglets at the tip of the wing. This design places a vertical surface at the tip of the wing. The third and final option was the shaped tip. This method gives an elliptical shape to the planform of the wing at the tip [2].

Wing Control Surfaces: The wing control surfaces are the primary means of aircraft roll control and lift-drag polar manipulation [2]. Several options were specified by the competition rules. The first option is ailerons. Ailerons provide a means of banking or rolling the aircraft by reducing the lift on one wing and increasing lift on the other causing a force imbalance. This in turn causes the airplane to roll [2]. The next option considered was the ailerons and flaps combination. Incorporating the flaps allows the manipulation of lift curve, increasing lift [3]. Another option utilized ailerons and spoilers. The spoilers are placed on top of the wing and act to destroy the lift created by the wing [3]. This is useful when landing high lift aircraft by offsetting the ground effect phenomenon [3]. Additionally, the option of utilizing ailerons, flaps, and spoilers is provided. The final option was the use of flaperons. Flaperons incorporate the ailerons and flaps into one moving surface [2].

Tail Configuration: There were five areas to consider for possible solutions to the design of the tail. The conventional tail is typical of what is seen in a majority of most R/C aircraft today [8]. It utilizes a vertical stabilizer as well as a horizontal stabilizer to provide pitch and yaw control [2]. Another option was the T-Tail. As its name implies, the conventional horizontal surface is placed at the top of the vertical surface giving the appearance of a T [2]. Another option was the cruciform design. This is similar to the T-Tail except that the horizontal surface is placed in the middle of the vertical surface rather then at the top [2]. The V-Tail is a "radical" design which essentially meshes the vertical and horizontal surfaces of the conventional tail into one surface placed at an angle forming a "V". The inverted V-tail is identical to the standard V-tail except that it is placed upside-down [2].

3.3.2 Aerodynamic Comparison Analysis

As stated earlier, the abstract nature of the conceptual design phase dictated that the hundreds of aerodynamic and mission factors for each component be simplified into a more universal set of criteria.

Aerodynamics: To fully account for the aerodynamic considerations of the aircraft, an appraisal of the flight conditions was made. From this appraisal it was found that the basic forces on an aircraft—lift, weight, thrust, and drag, were the most universal parameters of any aircraft. However, due to the aerodynamic equations, these could be further simplified to just lift, drag, and weight.

Mission Parameters: Originally, the RAC was the sole consideration for mission score; however, such a broad range of other characteristics existed that a separate consideration labeled mission compatibility was created.

Stability and Control: As with any aircraft, a significant amount of thought must go into the design of the control and stability of the aircraft. As such, the effects on stability and control of the aircraft were added to the evaluation criteria.

Manufacture and Design Criteria: Ignoring the importance of manufacturing and design criteria can be a fatal mistake. Composite construction might be the best option, but without the proper tools and knowledge the component will fail. The team decided that with any part considered, the difficulty of designing and manufacturing the parameter would be taken into account. Additionally, due to the limited amount of time to construct the final design, the effect on construction time was also analyzed.

As with any aerodynamic surface, the geometry, location, and number of surfaces determine its effectiveness in the above criteria. As such, several characteristics were compared to observe the effect they had on the above criteria. These effects where then placed into a decision matrix found in Table 3.2, weighting the importance of the criteria.

Number of Wings: This parameter was simplified in the beginning to the advantages and disadvantages of a single wing versus multiple. From low-speed aerodynamic theory, it is known that the larger the lifting area, the more lift is produced, thus a multiple wing approach is definitely favored. However, due to both the increased lift, and increased surface area, drag was negatively affected twofold by the multiple wing approach. Additionally, the RAC and manufacturing time for a multiple wing aircraft are significantly increased. Therefore, from this analysis, the single wing is the best approach.

Wing Vertical Locations: The first wing location to be abandoned was the low-wing design. This design produces roll instability characteristics and poor interference drag qualities [5]. The high-wing was also abandoned due to the considerations needed for the landing gear as well as the added drag this design produces [5]. The mid-wing was chosen since it has been shown to have lower drag qualities and neutral roll stability [5].

Wing Planform: Of the three types of wing planforms available, the reversed tapered wing scored low due to the depreciated effects produced with respect to drag and required structural weight. It is also a poor design with respect to the stability and control of the aircraft. The tapered and non-tapered wing planform have similar qualities, however, the non-tapered wing is easier to manufacture and has no depreciable effects on the RAC [15].

Wing Tip Configurations: The wing tip structure has been found to have a drastic effect on the overall drag of the aircraft [2]. The first solution to be discarded was the winglet design because of the difficulty in manufacturing and the degraded effect caused to the RAC. The shaped and clipped tips provided similar scores. Both are easy to manufacture and have little effect on the RAC. However, the shaped-tip configuration was chosen due to its excellent drag qualities as compared to the clipped tip [10].

Table 3.2: Aerodynamic Decision Matrix

		Lift	Drag	Weight	Size	Stability	Control	RAC	Mission Comp	Manf. Skill Required	Manf. Time Required	TOTAL SCORE	FINAL OUTCOME
Number of Wings	Mono	5	5	5	5	3	3	5	3	4	4	42	Mono
	Multiple	5	2	1	1	3	3	2	3	3	2	25	
Verticle Location	High	3	2	3	3	5	4	3	2	4	4	33	Mid
	Mid	3	5	4	5	4	5	3	4	3	4	40	
	Low	3	2	3	3	2	3	3	2	4	4	29	
Wing Planform	Tapered	3	5	4	3	4	4	3	3	2	3	34	No-Taper
	No-Taper	5	3	3	3	4	4	5	4	5	5	41	
	Rev. Taper	3	1	1	3	1	1	3	3	2	3	21	
Wing Tip	Winglet	3	5	3	3	4	4	2	2	2	3	31	Shaped
	Shaped	3	4	4	5	4	3	4	3	4	3	37	
	Clipped	3	2	2	5	3	3	5	3	5	5	36	
Control Surfaces	Ailerons	3	3	5	5	3	3	5	3	5	5	40	Flaperons
	A+F	5	4	3	4	4	5	3	4	4	4	40	
	A+S	3	5	3	4	3	3	3	3	4	4	35	
	A+F+S	5	5	2	3	4	5	2	5	3	3	37	
	Flaperons	4	4	4	4	4	4	4	4	5	4	41	
Empennage	Conventional	3	3	5	3	3	5	3	4	4	4	37	Inverted V or Conventional
	T-Tail	3	3	3	5	3	2	3	3	3	4	32	
	Cruciform	3	3	4	4	3	2	3	3	3	4	32	
	V-Tail	3	5	4	3	3	2	5	4	3	3	35	
	Inverted V-Tail	3	5	3	3	3	4	5	5	3	3	37	

5-Excellent, 4-Good, 3-Fair, 2-Poor, 1-Unacceptable

Wing Control Surfaces: Several options were researched for their advantages and disadvantages in control and weight, along with RAC considerations. To compare the effects of the various options, data were generated from a low-speed aerodynamic analyzer and graphed for a NACA 2412 airfoil. The results can be seen in Figure 3.3.

The ailerons-only option is both lighter in weight and offers a lower overall RAC score than all other control surface options. However, this roll-control-only system does not allow the aircraft to change mid-flight to suit the current flight regime. Thus the results are increased stall, takeoff, and landing speeds. To reduce these characteristics airfoils may be modified with flaps and spoilers [2]. Flaps increase lift by shifting the lift curve up from the standard position [3]. The result is greater lift at slower speeds. This option offers shorter landing and take-off distances; however, the risk of floating above the runway due to ground effect during landing is increased [2]. The effect of spoilers was also analyzed. The lift slope is shifted down from the standard position demonstrating the destruction of lift [3]. Spoilers help reduce ground effect and landing roll out distance, but at the cost of lift and drag [2]. Spoilers offer the greatest

benefit in combination with flaps; protecting against the dangers of floating while still generating large amounts of lift. However, flaps and spoilers increase the competition RAC score and provide benefit at greater cost. The final solution was flaperons. Flaperons have a reduced RAC score, but still provide many of the same landing and take-off characteristics of the flaps and spoilers option [2].

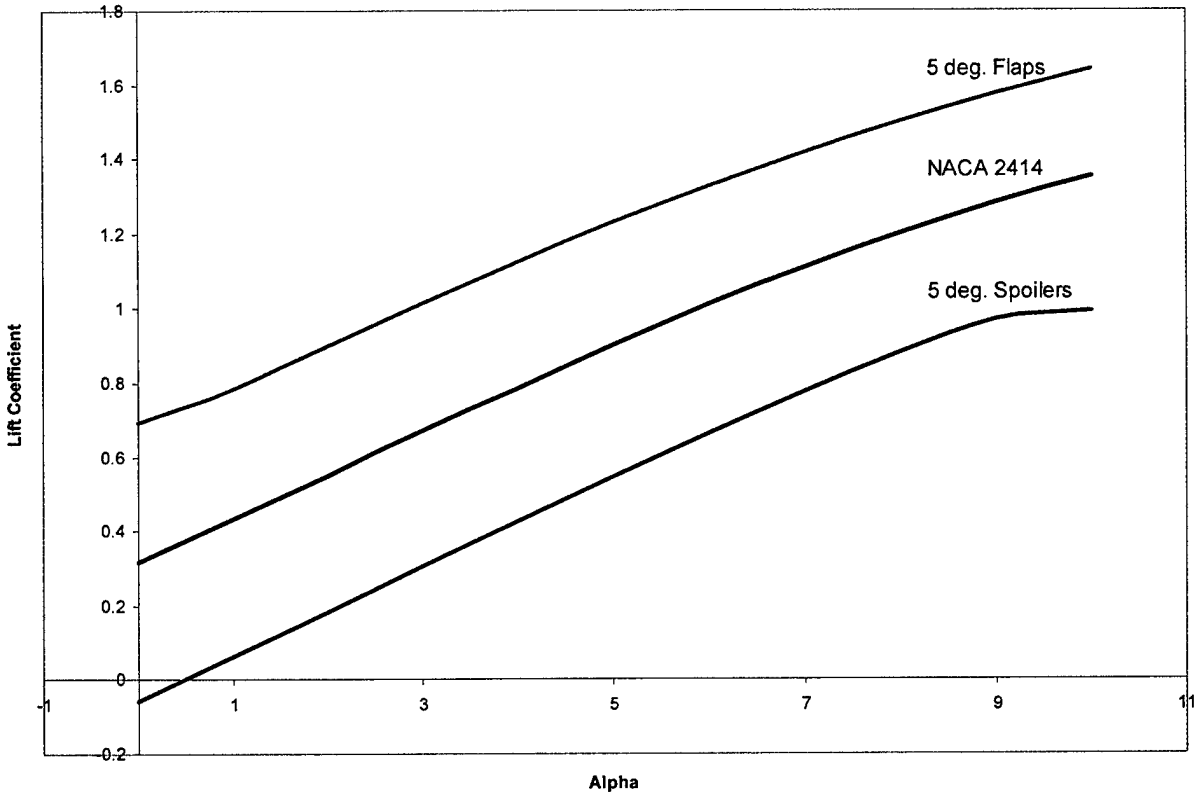


Figure 3.3: C_L Comparison Plot

Additional Wing Parameters: Several other wing design concepts were researched as possible solutions to ensure stability and low drag.

Dihedral: Dihedral is the angle from the horizontal plane that the wing is angled. This simple increase of the wing's angle increases the roll stability of the aircraft. This change comes with no penalty to the overall RAC score of the aircraft. However, the difficulty of manufacturing a wing is significantly increased with dihedral [3].

Washout: Washout is the angle of twist along the span of the wing. By washing out a wing, the aircraft becomes less prone to tip stall [15]. This wing modification also comes with no RAC penalty. A general reduction in lift however is noted. Also, washing out a wing only works over a very small range of attack angle [15].

Blended Airfoil: A blended airfoil is a wing that contains two different airfoil geometries—one at the tips of the wing and the other at the root. This is most often done to replicate the ideal lift distribution over the wing while maintaining several other aspects of the wing [2]. However, this is a

very difficult feature to design. Additionally, construction with the tools available would be nearly impossible.

Leading Edge Flaps: Leading edge flaps, or slats, are another lift increasing device. However, interviews with modelers and other aerospace engineers have revealed that these devices are often not effective for the flight regimes in which model aircraft fly, and they prove very difficult to manufacture [15].

NASA "Safe" Wing Concept: This concept, which was heavily researched in the 1970's, was NASA's attempt to create a nearly spin-proof wing [8]. The result was highly effective. By dropping the outer portions of the leading edge of the wing, that portion of wing is flying at a different flight regime. The result is that while the inner portions of the wing are stalling, the outer portions are still producing lift. This results in holding off wing stall. This concept has virtually no impact on the overall RAC. It has been shown to be easily manufactured, and has been proven to work several times in the model aircraft flight regimes [8].

Tail Configuration: The first option considered for the tail configuration was the conventional tail. This tail configuration is the easiest to design and construct; however, it has a negative effect on the competition RAC score. The T-tail increases the efficiency of the tail by moving the horizontal surface out of the turbulent air generated by the propeller. However, at high angles of attack, as might be encountered during take-off, the backwash of the wing can block the tail, reducing the elevator's ability to control the pitch of the aircraft. If the aircraft stalls in this condition, the pilot may not have enough pitch control to escape. Additionally, to accommodate the stresses at the top of the vertical stabilizer, the structure must be strengthened, increasing weight [5].

The cruciform configuration was considered next. This configuration was used to help alleviate some of the weight gain of the T-tail. By moving the horizontal component of the tail to the middle of the vertical stabilizer, the structure can be much weaker. However, the cruciform tail is not as efficient as the T-tail, and has many of the same stall characteristics due again to high alpha backwash [5]. The last two options considered were the V-tail and inverted V-tail. Both tails provide the most amount of controllability for the smallest RAC penalty. The standard V-tail produces a slight control coupling [15]. This coupling, known as adverse roll-yaw coupling, results in the nose of the aircraft dropping significantly during a turn. This is difficult to null out of the aircraft and its severity is difficult to predict. The inverted V-tail solves this problem and essentially allows the nose to maintain its position in a turn more predictably then the standard V-tail [15]. Few drawbacks exist with the V-Tail. Both the standard and Inverted V-tails are more difficult to design and configure than a conventional setup. Another drawback to the inverted configuration is its low ground clearance in the case of a single tail boom, or its weight in the case of a twin-tail-boom system [15].

3.4 Propulsion Analysis

The main components of the propulsion system are the batteries, the motor controller, the motor, the drive mechanism and the propeller or ducted fan. The inputs to the system are the weight of the aircraft,

the time and total energy needed to complete the mission, and the power required to sustain flight. The output is the power available. The propulsion analysis began with research and brainstorming to discover as many potential solutions as possible. Then, the analysis criteria was developed, and the RAC impact of each solution was evaluated. Finally, this information was used to perform a complete analysis and find the best possible solution for each component. The final design should incorporate a configuration of these components that produces as much speed as possible for at least the minimum amount of time needed to complete the longest mission, which is presumed to be the ferry mission.

3.4.1 Possible Propulsion Solutions

Thrust Vectoring: Thrust vectoring is a technique that can be used to reduce the take-off distance and increase the in-flight maneuverability of an aircraft. At a minimum, it involves rotating the propeller about the pitch axis to produce a vertical lift component at slow airspeeds, at the expense of the horizontal thrust component of a conventional fixed propeller. In extreme cases, thrust vectoring can incorporate many small, ducted air streams that can be manipulated semi-independently to facilitate delicate maneuvering at low airspeed.

Multi-motor configurations: The rules permit the use of multiple motors to power the aircraft. Customarily, multiple motors or engines are used to provide a safety factor in the advent of motor failure or to provide additional power to support heavy loads.

Single Motor: The single motor is the most common power plant for small aircraft [2]. It is generally lighter, more efficient and easier to employ than any other configuration.

Ducted Fan: One or more commercially available ducted fan units may be used individually or in conjunction with the conventional motor and propeller.

Drive Mechanism: There are two fundamental methods of transferring power from a source to an output. The first is the direct drive, which generally uses the main shaft from the motor, or some direct extension, to transfer the power to the output. The second option is to use a gearbox to manipulate the speed and torque of the shaft to maximize the efficiency of the system for a given condition.

Propellers: The rules permit the use of any commercially available propeller. They are manufactured from a variety of materials; the most common types are carbon fiber, nylon, and wood. The actual dimensions of the propeller were determined in later analyses, but propeller type was selected in this phase based on cost and durability.

Batteries: The batteries constitute a major portion of the aircraft gross weight, so their configuration must be considered carefully to minimize the weight and maximize the useful power. The rules specify that only Nickel Cadmium batteries can be used in competition, and like the propeller, the exact specifications of the batteries were determined in a later phase. The cells are available with capacities as high as 2400 mA-hr and as low as a few hundred mA-hr. In general, the weight of the cell increases as the capacity increases. Groups of the battery cells can be connected in series or in parallel to increase the voltage or amperage, respectively.

Motor Controller: The motor controller is responsible for controlling the voltage that the motor sees when it is running. The specifications of the motor controller depend on the specifications of the battery packs and the motor, so the analysis of the motor controller was performed in a later design phase.

3.4.2 Propulsion Comparison Analysis

For the propulsion comparison analysis, there were ten criteria used at various times in the analysis of the components of the propulsion system. These ten criteria can be grouped into three major categories: Manufacturing, RAC and Aircraft Performance. These criteria were then placed into the decision matrix in Table 3.3, weighting the importance of each. RAC equations were taken from the competition rules.

Thrust Vectoring: Thrust Vectoring is a relatively delicate operation, and incorporating this capability into the design would have required more time and skills than were available. It added to the complexity of the design and would require the employment of at least one extra servo and additional structural support, which would increase the RAC. If the concept were successfully incorporated it would reduce the take-off distance, but this is probably the only benefit. Table 3.3 shows that it is clearly inferior to the fixed propeller configuration. The relative Manufacturers Empty Weight (MEW), Rated Engine Power (REP), and Manufacturing Man Hours (MFHR) variables were determined by assuming the simplest application of thrust vectoring which employed one extra servo and 0.15 lbs of additional structural weight.

Thrust Vectoring RAC increase = (300* .2 + 1500*0 + 20*5)/1000 = +.160 minimum (Eqn 3.1)

Motor/Propeller Configuration: Table 3.3 summarizes the analysis of the three most promising motor/propeller configurations: Single Motor/Single Propeller, Multi Motor/Multiple Prop and Multi Ducted Fan. These configurations represent the most efficient versions of their particular genre. Simplification was needed to avoid the analysis of an endless array of configurations produced by adding or removing motors and propellers. The Multi-Motor/Multi-Prop and Ducted Fan options both represent a significant increase in the overall complexity of the design. They would require more manufacturing resources. With the addition of each motor or fan, the RAC increases dramatically, and the reliability and efficiency of the system declines. Assuming that each additional motor or ducted fan unit weighs 1.5lbs and using a conservative battery weight of 3lbs, it was found that the RAC increased as follows:

RAC increase = (300* 1.5 + 1500*1*3 + 20*0)/1000 = +4.950 (Eqn 3.2)

Thus, a configuration of multiple motors results in an adverse increase in RAC. For this reason, a multiple motor with multiple propellers and a multiple ducted fan scored poorly in Table 3.3. This led to the elimination of these two configurations as possible solutions for the aircraft.

Table 3.3: Propulsion Decision Matrix

		Manf. Skill Required	Manf. Time Required	Manf. Cost Required	Rated Engine Power (RAC)	Manf. Empy Weight (RAC)	Manf. Man Hours (RAC)	Take Off Distance	Flight Speed	Flight Time	Efficiency	TOTAL SCORE	FINAL OUTCOME
Motor	Thrust Vectoring	2	2	2	2	2	2	4	4	3	4	27	Fixed
	Fixed Propeller	3	3	3	5	5	5	3	4	3	3	37	
	Single Motor/Single Prop	3	3	3	5	5	3	-	3	-	4	29	Sing. Motor / Sing. Prop
	Single Motor/Multiple Prop	1	1	1	5	2	-	-	3	-	2	15	
	Multi Motor/Multiple Prop	2	2	2	2	2	3	-	3	-	2	18	
	Multi Ducted Fan	1	1	1	1	1	1	-	3	-	2	11	
	Gearing	-	-	-	-	2	-	4	3	-	-	9	Direct Drive
	Direct Drive	-	-	-	-	4	-	3	4	-	-	11	
Propeller	Two Bladed	-	-	-	-	3	-	-	-	4	3	10	Two Bladed
	Multiple Bladed	-	-	-	-	2	-	-	-	2	2	6	
	Composite	-	-	-	-	5	-	-	-	3	3	11	Composite
	Wood	-	-	-	-	4	-	-	-	3	3	10	
	Nylon	-	-	-	-	3	-	-	-	3	3	9	

5-Excellent, 4-Good, 3-Fair, 2-Poor, 1-Unacceptable

Drive Mechanism: The analysis of the drive mechanism was not particularly conclusive. Table 3.3 shows that the direct drive mechanism scored the highest, but they only differ in three categories: MEW, take-off distance and flight speed. The MEW was slightly higher for the geared drive because of the weight of the gearbox and the increased weight of the propeller. The propeller diameter and overall size and weight were usually twice as large for geared motors. The large propellers were more efficient at low speed, which was why the geared drive scored higher in the take-off distance, but the efficiency at higher speeds is lower, thus the low flight speed. Still the team must stay committed to reducing the MEW. Assuming at the time of this analysis that the weight of the gearbox was 0.125lbs and the large propeller weighed 0.2 lbs more than the small prop for the direct drive:

$$\text{Geared RAC increase} = (300 * 0.325 + 1500 * 0 + 20 * 0)/1000 = +.0975 \quad \text{(Eqn 3.3)}$$

Propeller: At this point there were still too many unknowns to determine the actual dimensions of the propeller, but the propulsion team can evaluated the benefits of using a particular type of propeller. The two main physical characteristics of propellers are the number of blades and the propeller material. Table 3.3 shows that a two-bladed composite propeller is probably the best choice for the final design because the composite material is much lighter than its multiple-bladed wood or nylon counterparts.

3.5 RAC Conceptual Summary

After concluding the conceptual analysis, an estimated RAC was calculated using the worksheet in Table 3.4 on the following page. Ranges remain for areas in the wing, fuselage, battery weight, and empty weight and will be investigated further in future design analyses. The estimated RAC range is between about 10 and 17 at this stage. Although the lower value is desired, due to its inverse effect on the total score, it cannot be assumed best until optimal flight performance is finalized. From the conceptual analysis, the structures team decided on using a fiberglass tank since it can be shaped to fit the aircraft. The ball valve was chosen for the drain valve primarily because of the ease of acquiring it and the cost. It was decided to use a circular fuselage due to the good drag and weight characteristics found. Since a circular fuselage was decided upon, the donut was chosen as the wing attachment even though the pass-through structure won. This was done to prevent increasing the difficulty of manufacturing a tank that would allow the pass through. The choice of landing gear was the fixed type. This type is typical of what is seen on most aircraft and is therefore considered the most reliable and easy to integrate.

Table 3.4: RAC Worksheet

MFHR	Quantity	WBS Multiplier	WBS Total
Wing			
Span =	8' ~ 10'	10	54 ~ 83
Chord =	.67' ~ .83'		
# of Wings =	1		
Flaperons =	1.5	5	7.5
Fuselage			
Body Length =	<4'	20	36 ~ 55
Width =	.67' ~ .83'		
Height =	.67' ~ .83'		
Empennage			
V-Tail =	1	15	15
Flight Systems			
Servos =	6	5	30
		MFHR Range	142.5 ~ 190.5
		# of Engines =	1
		Battery Weight (lbs) =	2 ~ 5
		REP Range	2 ~ 5
		Empty Weight (lb) =	15 ~ 20
		MEW Range	15 ~ 20
		A =	300
		B =	1500
		C =	20
		RAC Range	10.35 ~ 17.31

The aerodynamics team decided on using a single-wing configuration which will be mounted on the mid-section of the fuselage. The wing will have a non-tapered (rectangular) planform as this is the most efficient design with respect to the overall RAC. Unfortunately, this design incorporates poor drag qualities. To overcome this issue, the aerodynamics team decided on a shaped tip which, as stated previously has excellent drag-reduction qualities. Further design consideration is required before a control system can be decided upon. From the initial research, it was determined that the advantages gained from a spoiler system did not outweigh the associated cost in rated aircraft score. From the decision matrix, it was decided that flaperons offered the best control option. However, if more control is needed, the system can be quickly converted to either ailerons only, or an ailerons and flaps option. Finally, the wing will employ some combination of dihedral, "Safe" wing technology, and other drag-reducing and stability-increasing geometries to further improve the overall performance of the wing. From the tail configuration decision matrix it can be seen that the conventional and V-tail configurations are nearly equivalent in control and weight. However, due to the advantages in control and RAC of the inverted V-tail, it was selected as the best design for the aircraft.

After careful consideration, the propulsion team decided on using a single motor/single propeller configuration. This choice was primarily dictated by the advantageous effects to the RAC. The type of propeller to be used was decided to be a standard two-bladed propeller due to the excellent efficiency results found. The propeller material will be a composite material since this also has an advantageous effect on the propeller efficiency.

Conceptual considerations from each group were integrated into the solid model seen in Figure 3.4. This model will be updated throughout the design process so that the team may have a continual look at the overall aircraft configuration.

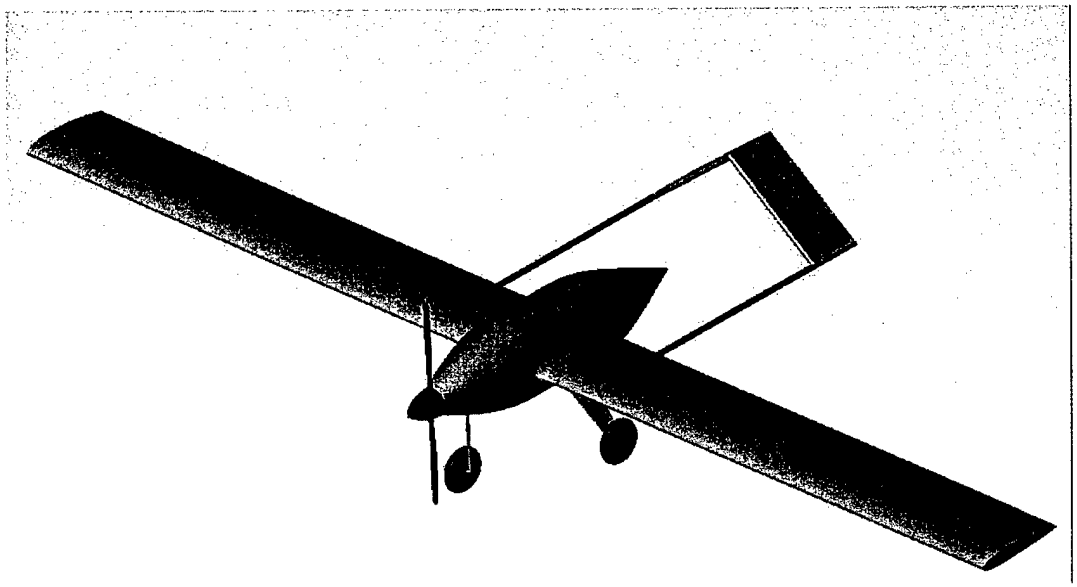


Figure 3.4 Conceptual Aircraft Solid Model

4.0 PRELIMINARY DESIGN

The preliminary design stage further investigated the aircraft components selected in the conceptual phase. Reasonable assumptions were made and discussed to continue analysis in each of the three design areas. The main goal of the parametric design was to narrow aircraft component sizes using theoretical formulas and computer simulations.

4.1 Mission Model

The aircraft is expected to complete two missions as described in the Mission Requirements section. The required course layout was divided into several sections to more accurately model the flight profile of the aircraft throughout each mission.

- Aircraft Loading—This phase occurs only during Mission 1 and is characterized by the pit crew disabling the aircraft, filling the payload section, and transiting times to and from the pit box.
- Takeoff Roll—This section encompasses the distance the aircraft rolls along the runway during takeoff. It is important to note that the aircraft will be loaded during Mission 1 and unloaded during Mission 2. This will vastly affect take-off distance and velocity.
- Upwind leg—The upwind leg is defined as the distance the aircraft travels in the air from takeoff to the 1st turn.
- Turn 1—Turn 1 aligns the aircraft with the downwind leg of the course. The radius was calculated as a result of both the load factor and velocity
- Downwind leg—The downwind leg of the flight contains two 500 foot straight-aways with a 360 degree turn in the middle. The aircraft's weight will be decreasing, due to the payload release, during Mission 1.
- Turn 2—Turn 2 aligns the aircraft along the runway.
- Final Approach—The aircraft will be descending and slowing to landing velocity.
- Landing Roll—The aircraft will touchdown well before its initial position for the mission. It will roll to a complete stop just past the start position. It is important to recognize the aircraft will be empty for both missions during this phase.
- Go-Around leg—This leg connects turn 2 to turn 1 by way of a 1000 foot straightaway along the direction of the runway.

4.2 Aerodynamic Analysis

The aerodynamic analysis parameterizes the major aerodynamic areas to find the most efficient, mission-capable aircraft design. Aerodynamics has been divided into three major sections: Airfoil Geometry, Wing Geometry, and Stabilizing Surfaces.

4.2.1 Airfoil Parameters and Sizing Trades

To best judge the suitability of each airfoil to the mission requirements of *Poseidon's Fury*, a set of ranges were established for the typical airfoil performance characteristics from typical values. These ranges are shown in Table 4.1.

Minimum Coefficient of Lift – The coefficient of lift range was established to ensure any airfoil chosen would be able to adequately lift the aircraft weight. Rated aircraft cost considerations increased the lower limit of this parameter in an effort to reduce the wingspan of the aircraft. This also has a reducing effect on the overall aircraft weight by reducing the necessary wingspan.

Zero-Lift Coefficient of Drag – This parameter was established as an average of typical airfoil parasitic drag values.

Lift to Drag – High airfoil efficiency is paramount to the *Poseidon's Fury*. As such, an average was taken of a selected group of airfoils and then set as the lower bound of the range. The average lift-to-drag ratio of any airfoil tested for the *Poseidon's Fury* then had to exceed this bound.

Thickness – Space is a major consideration for the wing of the *Poseidon's Fury*. Enough thickness must be provided to accommodate the wing structure and the outboard control servos. From these requirements, a lower bound was established. The upper bound was determined from boundary layer effects. As the thickness increases the boundary layer tends to separate reducing the efficiency of the wing.

Constructability – A constructability requirement was also added to evaluate the shape of the airfoils. The more curved the shape, the more difficult it becomes to accurately construct the airfoil.

4.2.2 Airfoil Analysis and Optimization

The final airfoil design is based on the Eppler 216 airfoil. Of the 30 airfoils tested, it provided the best results. However, the thickness-to-chord ratio fell outside of the bounds of the airfoil. As a result the airfoil was modified to increase this ratio to 0.125. The result was an increase both in efficiency and lift. Table 4.1 shows the final results for this airfoil.

Table 4.1: Modified Eppler 216 Parameter Values

Parameter	Range	Modified Eppler 216
Min. Coefficient of Lift	> 1.5	1.643
Zero Lift Coefficient of Drag	< 0.06	0.037
Avg. Lift to Drag	> 11	13.9
Thickness to chord	0.11 - 0.13	0.125
Constructability	Relative Ease of Construction	This goal is achieved

4.2.3 Wing Parameters and Sizing Trades

As seen in the conceptual design, a series of studies was generated to analyze the effectiveness of a wing's design. From these studies it was determined that a more in-depth study was needed to best generate the performance required by the competition rules. The parameters generated were for steady, level flight at a reasonable flight speed.

Lift – As with any wing, the single most important performance-defining aspect is lift. From the conceptual design, the maximum aircraft gross weight shall not exceed 25 pounds. As such, for steady level flight, the wing must produce enough lift to support this load at the stall speed of 30mph.

Lift to Drag Ratio – The lift to drag ratio was established as a parameter to ensure the efficiency of the wing. This parameter represents the average efficiency of the wing over a typical range of flight velocities for remote controlled aircraft.

RAC – The rated aircraft cost parameter was chosen to limit the overall rated aircraft cost of the aircraft. Though the wing has little effect on the overall score, this parameter was seen as a way to ensure the score would remain as low as possible. This parameter was used for all aspects of the aircraft it affected.

Manufacturability – Manufacturability was determined through experience from building remote control aircraft. It is a highly subjective term, contrasting the projected time to build the wing with the tools required to build it.

Wing Geometry Parameters – The wing geometry parameters define the characteristics of the wing. The chord, span, taper ratio, and aspect ratio were manipulated to find the most mission compatible design. This combination of wing dimensions, when added to the modified Eppler 216 airfoil, must provide the performance dictated by the above parameters. Various ranges were selected based on experience for each dimension. Taper ratio and aspect ratio were limited based on mission requirements, RAC efficiency, and three-dimensional aerodynamic affects. The parameters discussed in this section may be found in Table 4.2.

Table 4.2: Wing Parameters

Performance Parameter	Range
Lift at 30 miles per hour	> 25 lbs
Nominal Lift to Drag Ratio	> 11
Rated Aircraft Cost	< 80
Manufacturability	Determined from Experience
Wing Geometry Parameters	
Taper Ratio	1
Aspect Ratio	> 10
Chord	8 – 13 inches
Span	72 – 120 inches

4.2.4 Wing Analysis and Optimization

Over 100 wing configurations were tested varying the above parameters. Table 4.3 shows the final wing geometry parameters.

Table 4.3: Optimized Parameters

Wing Parameter	Value
Chord	10 in
Span	108 in
Wing Area	1080 in^2
Taper Ratio	1
Aspect Ratio	10.8
Max Lift Coefficient	1.643
Max Drag Coefficient	0.2
Max Lift to Drag Ratio	13
Rated Aircraft Cost	75

4.2.5 Empennage Parameters and Sizing Trades

- Performance Parameters – The empennage determines the stability of the aircraft [9]. As such, only a few performance and size parameters were established for the empennage. It was established that the empennage be designed to provide static stability and have enough control surface area to change the aircraft's attitude. The volume tail coefficient was utilized for this purpose [8]. The range of the volume tail coefficient was determined from an analysis of other statically stable aircraft and basic rules of thumb.

- Empennage Sizing Parameters – The sizing of the empennage did have three major restrictions. The first required that the tail panels be thick enough to accommodate a servo in each panel. Running servo control arms from another location is both highly stressful on the servo and would be difficult to place in an inverted V-tail configuration. The second restriction was implemented to accommodate rated aircraft rules that mandate the tail span be no more than twenty five percent of the main wing span of the aircraft. The final restriction required that the total area of the tail be the same regardless of the final configuration.

- Other Design Parameters – Several other parameters were included as basic rules of thumb. The vertical tail should be no smaller than eight percent of the wing area [5]. The Horizontal area should be near fifteen percent of the wing area [5]. Finally, the tail airfoil should be symmetric and thick enough to allow for the placement of servos inside. Table 4.4 shows the final parameters and their ranges. It should be noted that for the V-tail design no limit was placed on decalage or any other parameter, these would be indirectly constrained by the other design parameters.

Table 4.4: Stability and Empennage Parameters

Parameter	Range
Volume Tail Coefficient	0.35 - 0.55
Total Tail Area	Equivalent for all tail configurations
Tail Span	< 27 inches
Vertical Tail Area	around 93 square inches
Horizontal Tail Area	around 173 square inches
Airfoil Section	Symmetrical

4.2.6 Empennage Analysis and Optimization

The final tail design parameters can be found in Table 4.5. The tail was designed by first configuring the aircraft with a conventional tail and using the resulting tail areas to form the V-tail. The tail moment arm was designed as a function of the horizontal tail volume coefficient. The horizontal tail volume coefficient was set at a nominal value, and the moment arm length was determined. This process was iterated until a final value was obtained that fit all the tail parameters. Additionally, the control surface sizes were determined from experience and suggested rules of thumb [5].

Table 4.5: Empennage Optimized Parameters

Parameter	V-Tail
Total Area	228.5 square inches
Chord	7 inches
Horizontal Span	26.875 inches
Decalage	110.6 deg
Tail Moment Arm Length	37 inches
Control Surface Total Area	57 square inches
Volume Tail Coefficient	0.405
Tail Airfoil	NACA 0012

4.2.7 Flight Mechanics

Given the above design criteria, a basic aerodynamic performance model can be constructed of the aircraft. Basic lift, drag, and stability calculations using low speed aerodynamic principles may be made to better understand the performance of the aircraft. Figure 4.1 shows plots of the lift and drag with respect to velocity and attack angle. It was also found that the aircraft static margin was 0.25 [9].

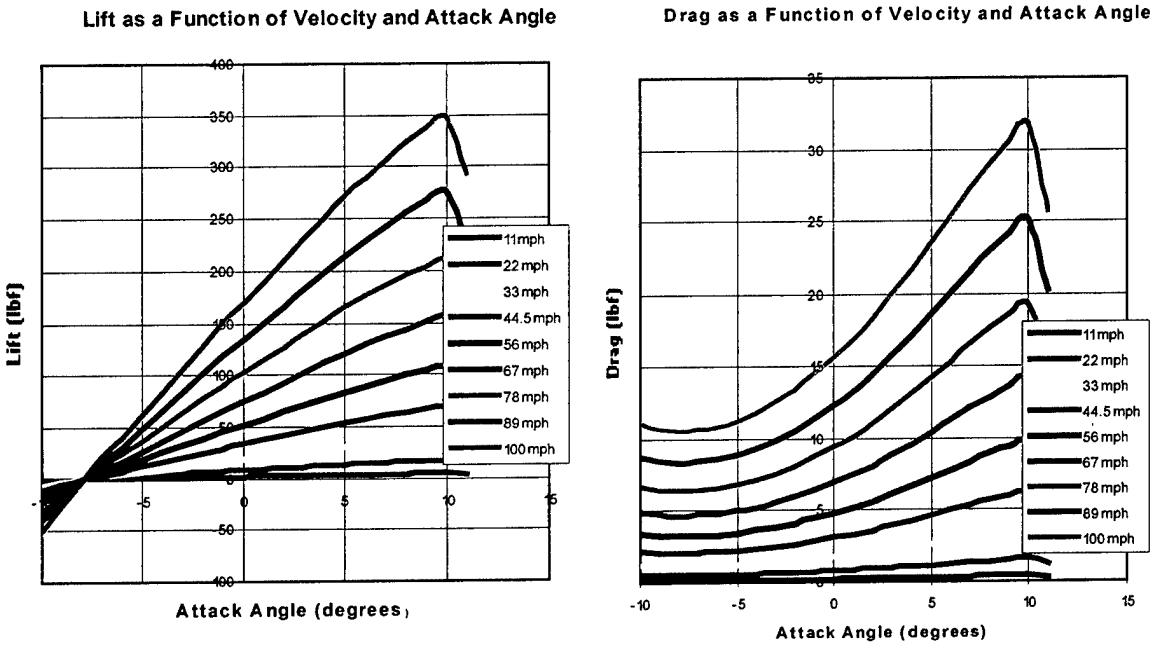


Figure 4.1: Flight Mechanics Data

4.3 Structural Analysis

Considerations for structural analysis of the aircraft were divided into several categories: the fuselage and drain system, wing attachment, and main landing gear. Analysis of the tank and drain system required investigation into the orifice discharge coefficient as well as sizing trades with respect to the fuselage. Analysis of the wing attachment focused on material, strength, and size. Finally, landing gear analysis was limited to two of the most common materials, aluminum and carbon fiber, to determine which would be best for the competition.

4.3.1 Fuselage and Drain System Parameters and Sizing Trades

Storage limitations posed a constraint on the overall fuselage length. The fuselage needed to be as long as possible to create enough space for internal components such as the tank, servos, and batteries. Increased surface area also yielded more attachment points for any outside components such as the front and main landing gear. Furthermore, considerations for how the fuselage would fit within the designated 2' x 4' x 1' box had to be evaluated along with the negative effects of the RAC with increases in length. The tank was then considered since it must fit within the fuselage while still allowing room for internal components. The parameters for the fuselage and rectangular tank can be seen in Table 4.6.

Table 4.6: Fuselage and Tank Parameters

	Fuselage	Tank
Length	24 in	9 in
Width	8 in	5.25 in
Height	8 in	5.25 in
Shape	Circular	Rectangular

4.3.2 Fuselage and Drain System Analysis and Optimization

To inspect aspects of tank drainage, a math model was created using theories in basic fluid mechanics with particular interest being in conservation of mass and Bernoulli's principles. The equation for estimating the drain-time, T, can be shown to be:

$$T = \sqrt{\frac{2 \cdot y_o}{g}} \cdot \left(\frac{A_{tank}}{C_D \cdot A_{orifice}} \right) \qquad \text{(Eqn 4.1)}$$

where

- y_o is the initial water height
- A_{tank} is the top area of the tank
- $A_{orifice}$ is the area of the drain orifice
- g is the gravity constant
- C_D is the discharge coefficient

During calculations and empirical testing, y_o and C_D were set as dependant variables with respect to water volume and orifice design, respectively. The A_{tank} variable was considered independently and subsequently selected for trade analysis. The final variable, $A_{orifice}$, became a constant after applying the 0.5-inch diameter requirement specified in the competition rules. The value for $A_{orifice}$ was therefore 0.196 in^2.

Before deciding on limits for the tank size, assumptions were made regarding the maximum tank height and the drain coefficient. Knowing that the size of the fuselage depends upon the size of the tank, the tank height needed to be kept at a minimum to reduce the negative effects of drag and to generate the lowest achievable RAC value. Of all variables, the value for C_D becomes the most intriguing. It is constrained solely by the design of the orifice and can be improved without sacrificing other factors like RAC. From Fox and McDonald, typical values for the drain coefficient range from 0.6 to 1.0 with 1.0 corresponding to the ideal orifice design, a design which produces little to no flow losses [6]. Notice that

for the same tank an orifice with a 0.6, C_D increases drain-time by 60% compared to an orifice with a 1.0 C_D. For the sake of being conservative, the orifice was assumed to have a C_D equal to 0.6.

Using Equation 4.1 the anticipated drain time for 4 liters of water in a rectangular tank is approximately 70 seconds. This was undesirable according to the mission model. To optimize the drain system design the rectangular tank was remodeled into a tapered shape giving it a higher frontal profile and a sloped bottom towards the drain. Most importantly, a custom valve was designed to prevent losses and in turn increase the discharge coefficient. To optimize the valve, a 3-D model was created using SolidWorks in which the interior was designed smooth to prevent disturbances in the water flow, and the drain orifice was fixed at the 0.5 inch maximum requirement. To insure that the creation of the valve was precise, it was created using a method called stereolithography. Stereolithography is a rapid prototyping method which builds the design out of resin using the 3-D solid model (Figure 4.2). Testing revealed that the custom tank and valve design empties the 4 liters in approximately 27 seconds.

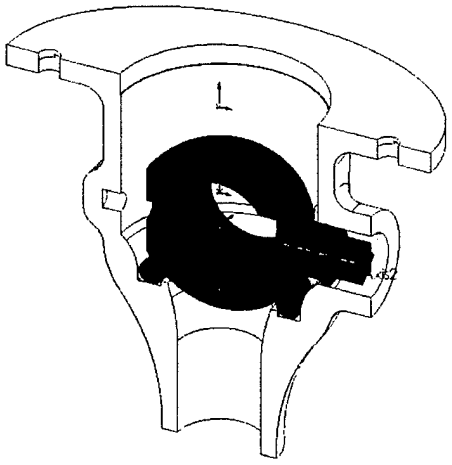


Figure 4.2: Custom Stereolithography Ball Valve

4.3.3 Wing Attachment Analysis and Optimization

The wing attachment is considered one of the most crucial areas of the aircraft. It was decided early in the design phase that the wing spar would be a balsa plywood laminate core sandwiched between epoxy-impregnated carbon fiber to ensure a lightweight assembly. Restrictions on the height of the spar were constrained to the maximum thickness of the airfoil, which is 1.25 inches. However, a smaller height would be beneficial in weight reduction. The spar was analyzed using both a tip load and uniformly distributed load as found in steady level flight with no gusts. During technical inspection at the competition, the aircraft must be lifted at the tips of each wing while the plane is fully loaded. This is done to simulate a 2.5g load on the aircraft. The estimated weight of the aircraft was assumed to be approximately 25 lbf, which results in a 12.5 lbf load at each tip. The team determined that the tip load test would stress the spar more then any other loads that may be encountered during a regular flight regime. Therefore, the tip load would be the load used to size the height and thickness of the spar. An

initial determination for the height of the spar was 0.875 inches. This gives adequate room to cover the wing with balsa sheeting without interfering with the spar during assembly. Since the drag loads are relatively small when compared to the bending loads from lift, they were ignored to simplify the analysis.

The first method to analyze the stresses in the spar involved theoretical calculations. The uniformly distributed load was the first load type to be analyzed. Since the spar would be a composite beam, the approximate theory for bending of sandwich beams used by Gere [7] was used for theoretical calculations. This theory gives the maximum normal stresses in each material. Equation 4.2 and 4.3 pertain to the carbon fiber and balsa laminate core, respectively.

$$\sigma_{x1} = \frac{-MyE_1}{E_1 \cdot I_1 + E_2 \cdot I_2} \qquad \text{(Eqn. 4.2)}$$

$$\sigma_{x2} = \frac{-MyE_2}{E_1 \cdot I_1 + E_2 \cdot I_2} \qquad \text{(Eqn. 4.3)}$$

where

- M is the bending moment produced by the load
- y is the distance from the neutral axis
- E_1 is the elastic modulus of the carbon fiber skin
- E_2 is the elastic modulus of the balsa laminate core
- I_1 is the polar moment of inertia of the carbon fiber skin
- I_2 is the polar moment of inertia of the balsa laminate core

For the distributed load, M is found by replacing the 10 lbf distributed load with an equivalent load. This results in a bending moment, M, of 240 lbf-in. Similarly for the point load, the resulting bending moment was found to be 540 lbf-in. Mechanical properties of carbon fiber impregnated with epoxy resin and 1/8 inch aircraft plywood were found at www.efunda.com [13] and www.cstsales.com [14], respectively. The elasticity modulus for the carbon fiber layer is $10.11 * 10^6$ psi. The balsa laminate core has an elasticity modulus of $2.02 * 10^6$ psi.

Solutions of equations 4.2 and 4.2 for the distributed load showed the maximum normal stresses within the carbon fiber skin and balsa laminate core to be 16.03 ksi and 3.13 ksi, respectively. These results are reasonable since the carbon fiber elasticity modulus and balsa modulus differ by a factor of 5. The above maximum stresses result in a factor of safety for the carbon fiber and balsa core laminate of 13.5 and 3.2 respectively.

Replacing the bending moment value of the distributed load with a tip load value of 540 lbf-in results in higher normal stresses as would be expected due to the larger moment. The maximum normal stresses for the carbon fiber skin and balsa core laminate due to the tip load were found to be 32.1 ksi and 6.3 ksi, respectively. These stress values can be assumed to be much larger then those that could be found in steady level flight with no gust loads. Moreover, the factor of safety for the carbon fiber skin and balsa

core laminate is 6.8 and 1.6, respectively. It should be noted that the above values for stress assume that the material is ductile. To classify carbon fiber as ductile is not very accurate. However, due to the relatively thin layer of carbon fiber to be laid on the spar, it was assumed to be ductile for this analysis to reduce the complexity of the solutions.

The above solutions were then verified using CosmosWORKS FEA which operates within Solidworks 3-D modeling software. CosmosWORKS software utilizes the same method of FEA as does NASTRAN and is therefore considered an adequate tool for the wing spar analysis. The wing spar was modeled with Solidworks 3-D modeling software as can be seen in Figure 4.3.

Note that due to the symmetry of the wing spar attachment, only half of the structure was modeled to reduce the amount of time needed to run the analysis. The wing attachment is an assembly consisting of a balsa laminate core sandwiched by the epoxy-impregnated carbon fiber. Boundary conditions were applied to the spar to simulate the actual conditions encountered during flight. This included the ends of the center cut "donut" being fixed and the sides of the donut being held normal to their surface to prevent translation of the part in that direction. Hence, the spar was only allowed to translate and rotate in the transverse direction. Material properties identical to those used for the theoretical calculations were then assigned to the various components. Meshing of the model was done with a very fine mesh to ensure accuracy of the solution. The part was meshed with a relatively small element size of 0.1946 inches. This resulted in a total of 30448 elements.

The first load analyzed with CosmosWORKS was the distributed load. A uniformly distributed load of 0.1852 lbf/in (10 lbf / 54 inches) was applied to the spar. To assign boundary conditions, the donut was fixed at the cut edges to prevent translation and rotation, while the faces of the donut were fixed so that they could only rotate in the transverse direction to prevent the donut from "twisting" during analysis. A solution of the distributed load resulted in a maximum normal stress of 11.2 ksi within the carbon fiber skin and 4.36 ksi within the balsa laminate core as can be seen in Figure 4.4. The maximum displacement of 1.66 inches was found at the outer tip as can be seen in Figure 4.5.

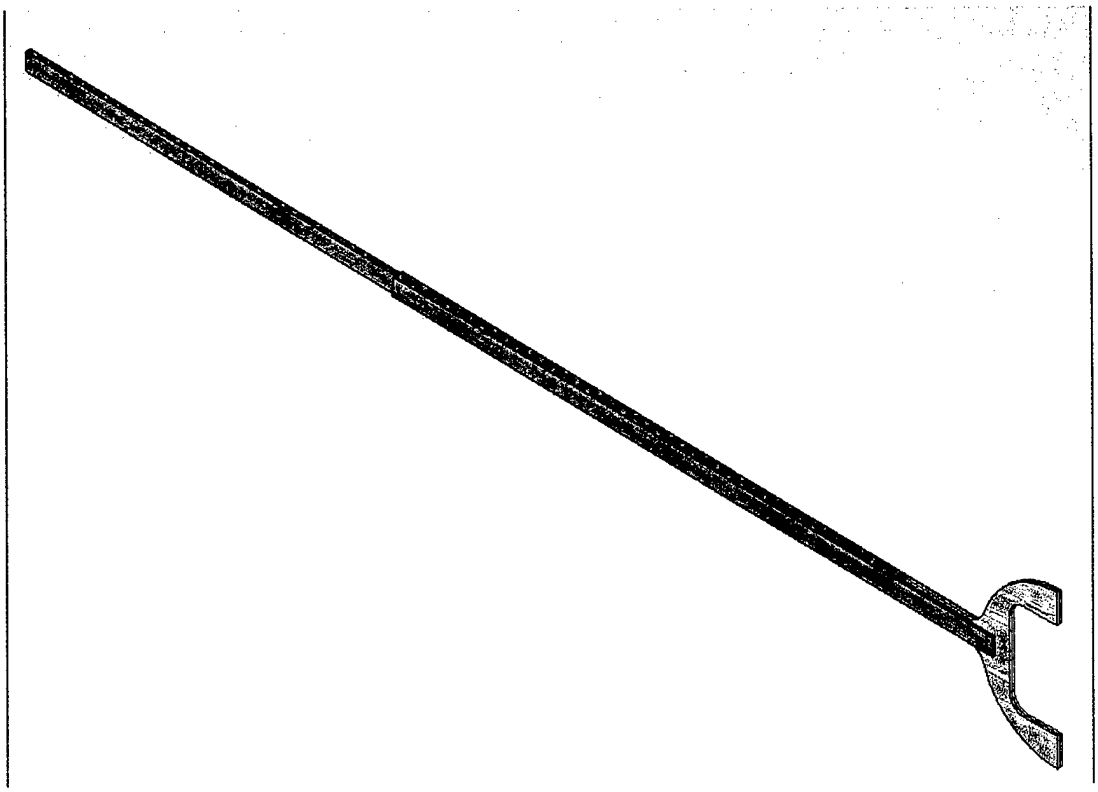


Figure 4.3: 3-D Model of Wing Attachment

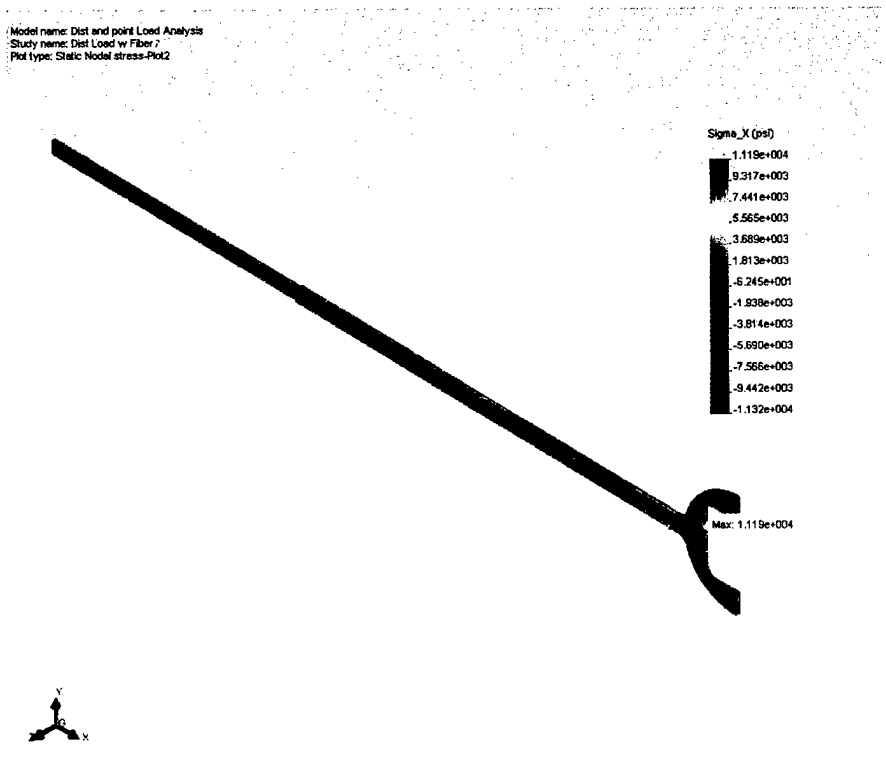


Figure 4.4: Distributed Load Longitudinal Stresses

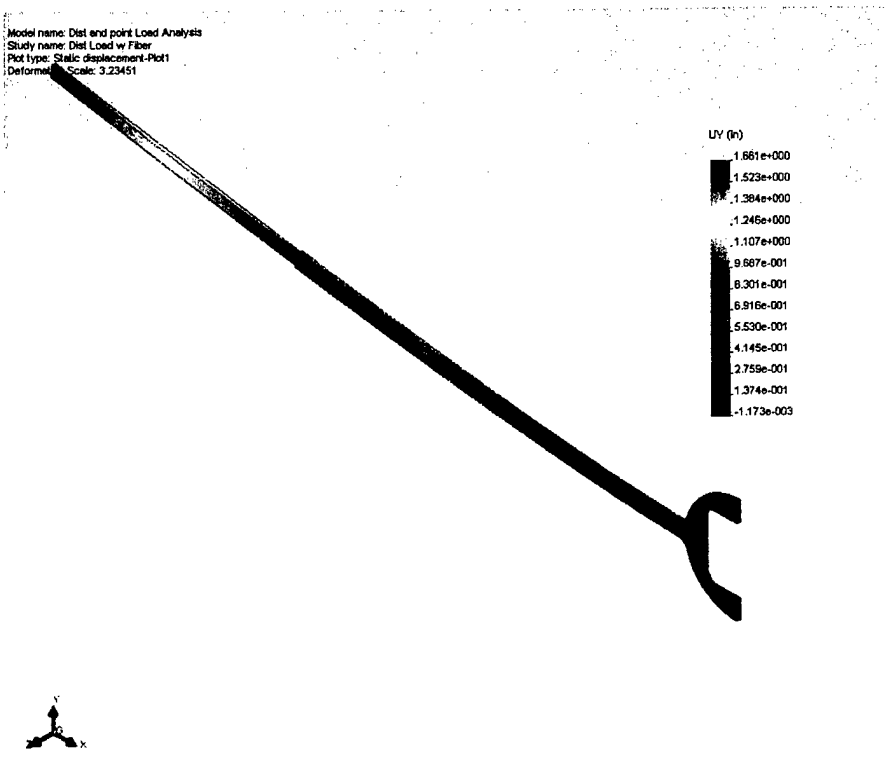


Figure 4.5: Distributed Load Displacement

Analysis of the tip load was the next type of load to be analyzed within CosmosWORKS. A tip load of 10 lbf was applied at the outside edge. After the analysis was finished, the stresses and deformation for the point load were plotted as shown in Figures 4.6 and 4.7.

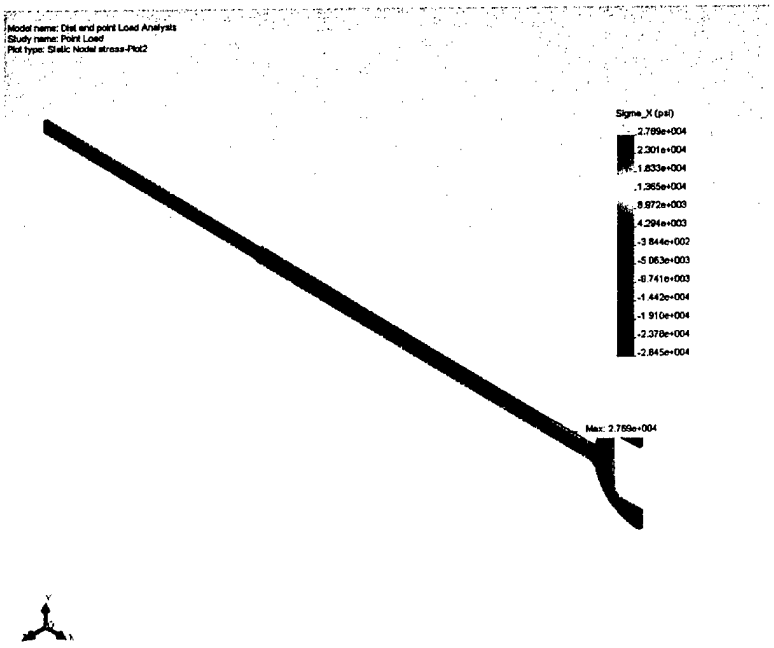


Figure 4.6: Wing Tip Load Longitudinal Stresses

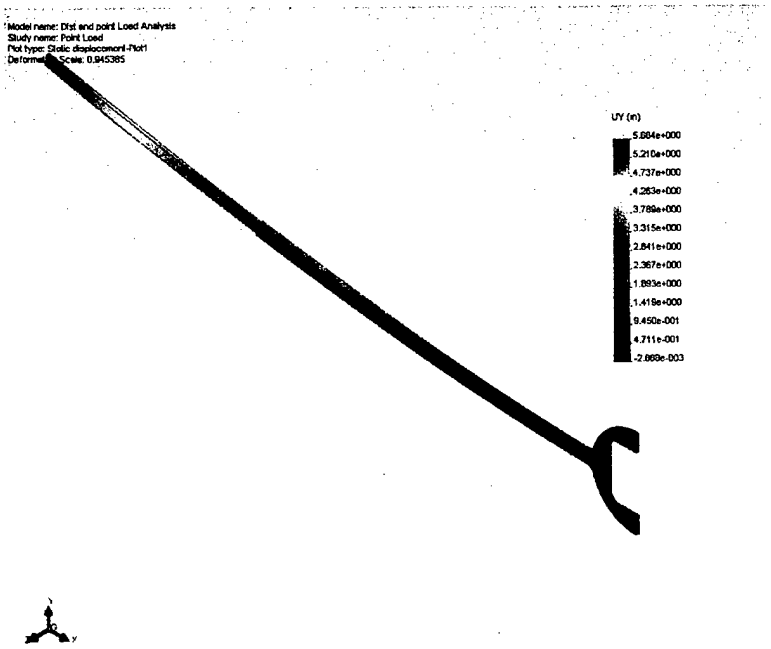


Figure 4.7: Wing Tip Load Displacement

The maximum stress within the carbon fiber measured by CosmosWORKS for the wing tip load was 27.7 ksi and occurred at the inner root of the spar as expected. The maximum deflection encountered was 5.68 inches at the outer tip. The maximum stress within the balsa core was found to be 5.01 ksi. Both values are within 20% of those found by the theoretical calculations. Moreover, these values are more realistic of what would be found within the actual spar since the spar is identical to what will be built. A summary and comparison of the theoretical solutions and FEA solutions is shown in Table 4.7.

Table 4.7: Summary of Maximum Normal Stresses

	Summary of Maximum Normal Stresses			
	Point Load		Distributed Load	
	Carbon Fiber	Balsa Laminate	Carbon Fiber	Balsa Laminate
Theoretical Solution	*32.1 ksi*	*6.26 ksi*	*16.03 ksi*	*3.13 ksi*
CosmosWORKS Solution	*27.8 ksi*	*5.01 ksi*	*11.2 ksi*	*4.36 ksi*
% Difference	*13.40%*	*20.00%*	*30.13%*	*28.20%*
Factor of Safety	*6.8*	*1.6*	*13.6*	*2.3*

As can be seen by the above table, the results from the theoretical calculations solution are similar to the stresses found by CosmosWORKS FEA. The differences between the two methods can be attributed to the geometry of the solid model being more complicated then the simple cantilever beam and the donut picking up some of the stresses at the root of the spar. Considerations of the factor of safety were taken from the values giving the lowest factor of safety, i.e. the theoretical calculations, to be conservative. The smallest factor of safety found for the carbon fiber was 6.8, which is considered excellent. Although the factor of safety for the balsa core was found to be only 1.6, it can safely be assumed that any localized areas of failure within the balsa core will not cause failure of the spar since the carbon fiber skin is able to

carry any increased loads due to any localized failure in the balsa core. This is due to the carbon fiber skin's relatively large factor of safety.

In conclusion, the spar should be built from balsa plywood laminated to a thickness of 0.5 inches. This core will then be sandwiched with .018 inch of carbon fiber impregnated with epoxy resin, which results in a total spar height of .875 inches. This method results in a factor of safety of 1.6 for the balsa core and 6.8 for the carbon fiber skin.

4.3.4 Main Landing Gear Analysis and Optimization

The analysis of the main landing gear was done by modeling the gear in Solidworks and analyzing the gear with CosmosWORKS. A typical R/C main gear was measured and modeled as can be seen in Figure 4.8.

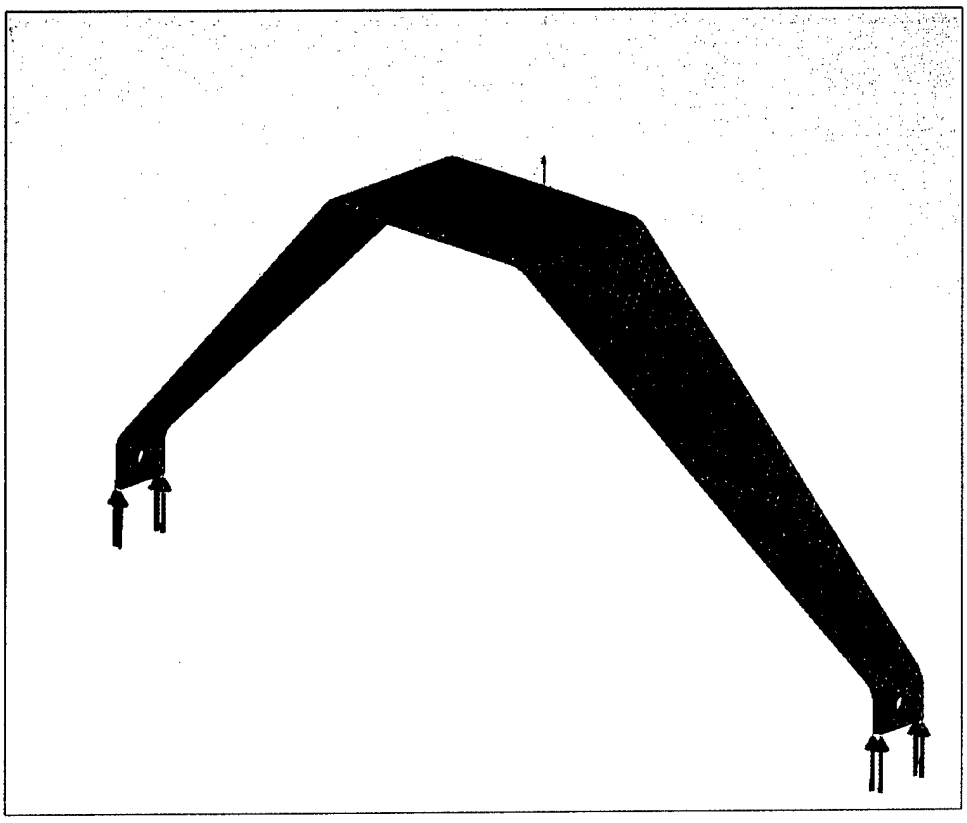


Figure 4.8: Main Landing Gear Load Application

The gear was analyzed in CosmosWORKS using the mechanical properties of both 6061-aluminum and epoxy impregnated carbon fiber. An aluminum landing gear tall enough to clear the propeller of the aircraft would weigh approximately 0.7200 lbf. Therefore, a carbon fiber landing gear was considered for weight reduction. The approximate weight of the carbon fiber gear is 0.431 lbf. This results in a total weight savings of 0.289 lbf, which is approximately 1.5% of the total weight of the aircraft.

Determination of the load to be applied was the initial weight of the unloaded aircraft, 10 lbs, multiplied by a factor of 3 to account for landing loads. This resulted in a total of 15 lbs applied at each tip as also shown in Figure 4.8.

The gear was then fixed at the top flat surface to restrain it. The 6061-aluminum landing gear was analyzed first. Determination of the stresses within the aluminum gear was found using the Maximum Von Mises Stress Criterion. Young and Budynas [12] state that this theory predicts the yielding in ductile materials when the distortion energy per unit volume of the material equals or exceeds the distortion energy per unit volume of the same material when it is subjected to yielding in a tensile test. As can be seen from Figure 4.9, the maximum Von Mises stress in the aluminum landing gear was found to be 20.6 ksi.

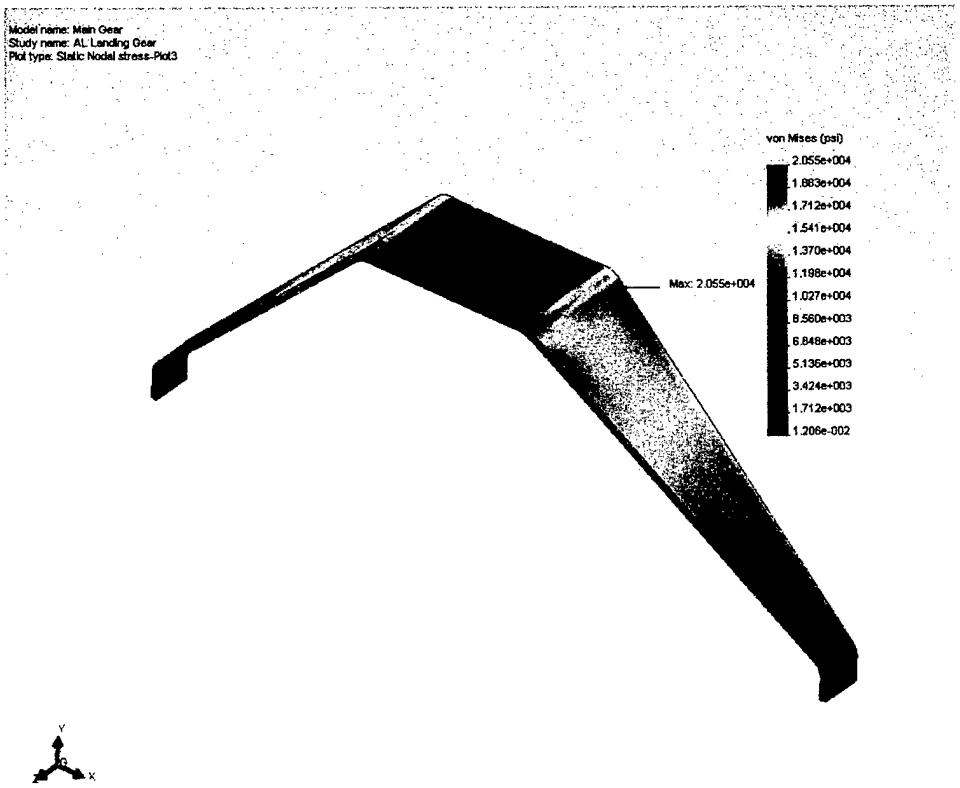


Figure 4.9: Aluminum Main Gear Maximum Von Mises Stress

This stress results in a factor of safety of 0.39 for the landing gear under a heavy landing condition. This condition is unacceptable because it indicates that the gear has yielded. Since CosmosWORKS is a linear analysis tool, the results after yielding become inaccurate. It can safely be said that the gear is well beyond its yield point and would be unusable for any further flights.

Analysis of the carbon fiber gear realized results that were far better than the aluminum gear. However, determining an accurate factor of safety was difficult since assigning the carbon fiber as either a ductile material or brittle material is difficult. Unlike the spar, which has only a small layer of carbon fiber, the

main gear is entirely composed of carbon fiber and determination of the factor of safety becomes more complex. Carbon fiber is a "filament-reinforced material which consists of a base material in which high-strength filaments are embedded" [7]. Unlike ductile materials, the ultimate stresses in compression are much higher than those in tension. For the sake of this analysis, the carbon fiber gear was classified as a brittle material. Therefore, determination of the factor of safety could be found using the Maximum Normal Stress Criterion.

This theory states that the factor of safety is:

$$FOS = \frac{\sigma_{Limit}}{\sigma_1} \qquad \text{(Eqn 4.4)}$$

Requirements for this criterion are that the material must be brittle. A brittle material is loosely defined as a material which has different compressive and tensile properties. This theory also assumes that the ultimate strength of the material is the same in both tension and compression. However, cracks and irregularities which can weaken the strength of the material in tension tend to close when subjected to compression and hence do not affect the compressive strength of the material. Therefore, the maximum tensile stress encountered will be allocated as σ_{Limit} since it can be assumed that the gear will fail in tension rather then compression. Figure 4.10 shows the maximum tensile stress found from the analysis.

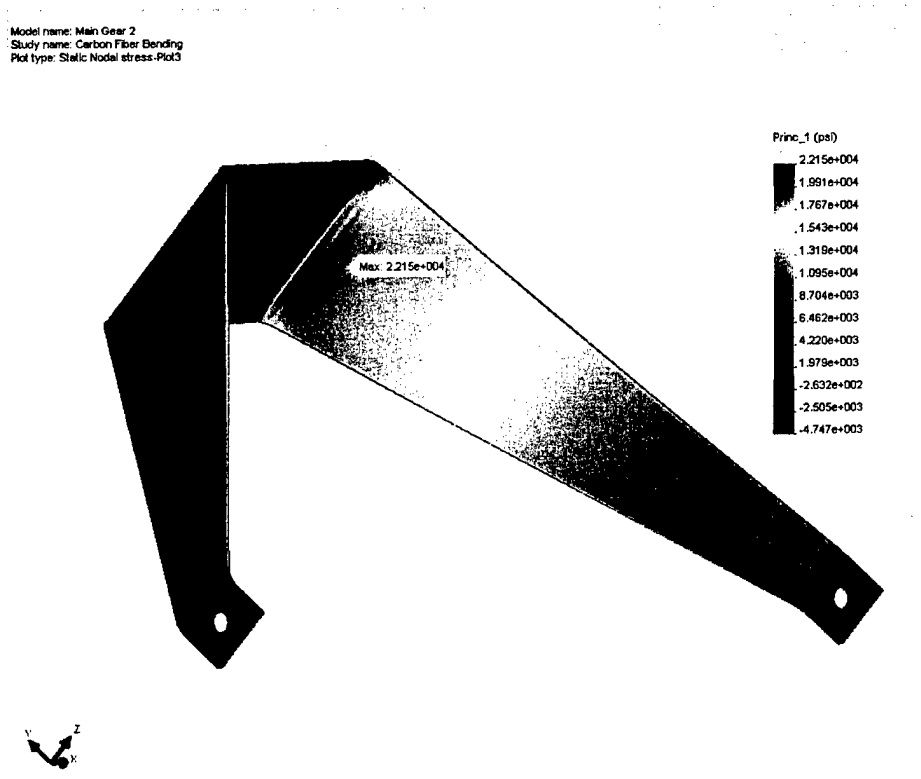


Figure 4.10: Maximum Tensile Stress for Carbon Fiber Gear

As can be seen in Figure 4.11, the maximum tensile stress is 22.2 ksi. This gives a factor of safety of approximately 2.0.

In conclusion, analyses of both carbon fiber gear and aluminum landing gear were done to determine which had the best attributes. The aluminum landing gear was heavier and had poor stress results when compared to the carbon fiber gear. Although the carbon fiber gear is more expensive, the savings from weight reduction and reliability of the gear under heavy loading far outweighs the actual cost of the gear.

4.4 Propulsion Analysis

The goal of the propulsion analysis was to select a combination of motor, batteries, and propeller that provided the best performance for the lowest RAC. The performance of the propulsion system was primarily a function of thrust and the RAC was dominated by the battery weight as shown in Equation 4.5.

$$Propulsion\ System\ RAC = \$1500 * Battery\ Weight + \$300 * Total\ Weight \quad \text{(Eqn 4.5)}$$

The aerodynamics team estimated that the aircraft would need to produce 4 lbf of thrust, at 50 mph, for approximately 3 minutes to complete the longest sortie, and an average of 5 lbf of thrust at takeoff, to achieve liftoff within 150 ft during the loaded mission. These estimates indicated that the minimum continuous power required was approximately 400W. The following section describes the experiments and trade study evaluations that were used to find a configuration that satisfied these requirements at a minimal RAC penalty.

4.4.1 Battery Parameters and Sizing Trades

The project rules dictate that only NiCAd type batteries can be used in the design. Sanyo offers a wide variety of battery sizes with varying capacities, weight and internal resistances; these are the primary battery parameters. Table 4.8 shows a sample of battery sizes that may fulfill the requirements. As mentioned in the previous section, the battery weight contributes to the RAC twice, once in the Total Weight, and again in the Rated Engine Cost. It was imperative that the final selection provided adequate power for the minimum required period of time, while minimizing the total weight.

Table 4.8: Sanyo Battery Specifications [17]

Sanyo Battery Specifications							
Model	**Capacity**	**Weight**	**Dimensions**		**Resist.**	**Size**	**Voltage**
			Diam.	**Length**			
	[mAh]	**[g]**	**[mm]**	**[mm]**	**[mOhm]**		**[V]**
CP-2400SCR	2400	60	22.9	43.5	4.5		1.2
KR-2300SCE	2300	58	23.0	50	5.5	5/4SC	1.2
RC-2400	2300	60	23.0	43.5	3.5	SC	1.2
N-1900SCR	2000	54	23.0	43	4	SC	1.2
KR-1700AU	1700	36	17.0	49.5	17	A	1.2
KR-1500AUL	1500	31	17.0	43	16	4/5A	1.2

Table 4.8 shows that the weight of the battery cells increase with capacity, and the internal resistance decreases with capacity. In addition, there is a drastic decrease in weight between the 2000 mA-hr and 1700 mA-hr cells.

Figure 4.11 is a plot by Fly diversity aircraft [16], of the output of two NiCad batteries offered by another manufacturer. The plots illustrate that the actual output, in amp-hours, is significantly lower than the advertised specification; this is a typical trend for all batteries in high-output applications. In these instances, as large currents pass through the battery, the internal resistance causes the cell to produce heat and thus reduce the useful capacity and voltage.

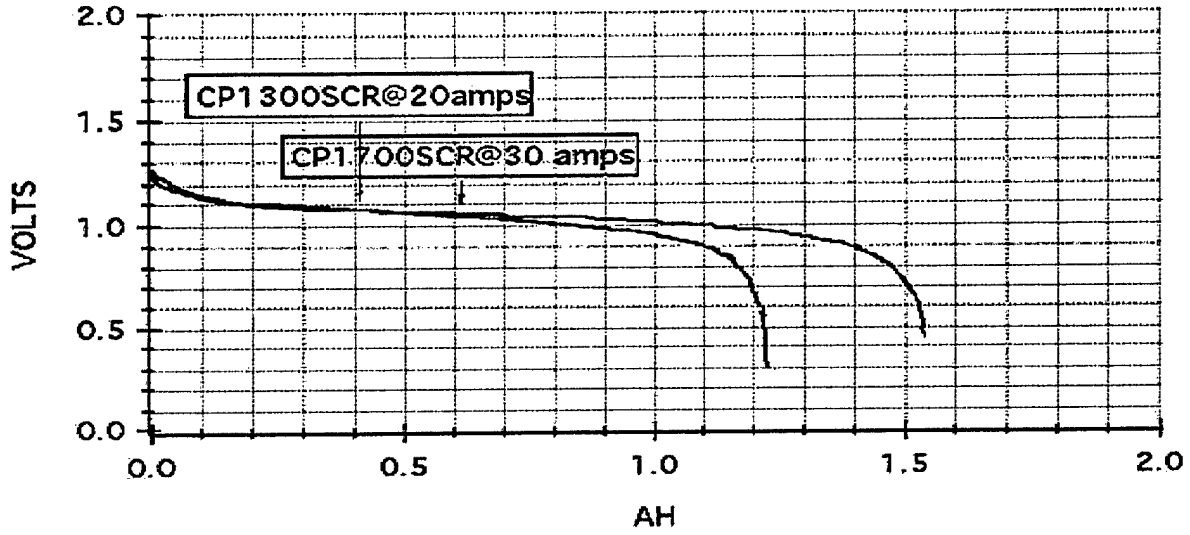


Figure 4.11: NiCad Battery Comparison

Table 4.9 predicts the actual useful capacity of several Sanyo cells based on an anticipated 20% reduction in capacity due to internal resistance and then lists the continuous available current each cell can provide for a 3-minute flight.

Table 4.9: Sanyo Actual Useful Capacity

Model	Advertised Capacity (mAh)	Actual Useful Capacity (mAh)	Maximum Continuous Current for 3 min (A)
CP-2400SCR	2400	1920	38.4
KR-2300SCE	2300	1840	36.8
RC-2400	2300	1840	36.8
N-1900SCR	2000	1600	32.0
KR-1700AU	1700	1360	27.2
KR-1500AUL	1500	1200	24.0
KR-1400AE	1400	1120	22.4

4.4.2 Motor Parameters and Sizing Trades

The competition rules limit the motor manufacturer to Graupner of AstroFlight. The analysis was restricted to AstroFlight Cobalt motor models because they have an excellent reputation and offer a variety of high power motors. Table 4.10 lists the main parameters investigated, maximum shaft power,

maximum voltage, maximum current, shaft speed, motor efficiency, motor weight, anticipated battery weight, and anticipated RAC of the motor with light-weight batteries. Of these, the motor weight, battery weight, maximum shaft power, and RAC were most important. The table shows that all of the parameters increase in magnitude as the motor size increases, except for the maximum current and the shaft speed. The maximum shaft power-to-weight ratio also increases with motor size.

Table 4.10: Motor Parameters

	Max Shaft Power (W)	Max Voltage (V)	Max Current (A)	Shaft Speed (RPM)	Motor Efficiency (%)	Motor Weight (lbf)	Battery Weight (lbf)	RAC
Cobalt 40	750	24	35	6000	76	1.13	1.90	3.76
Cobalt 60	1200	36	35	4200	83	1.50	3.90	5.58
Cobalt 90	1400	48	35	3800	84	2.13	3.80	7.18

Table 4.11 shows the expected shaft power provided by the manufacturer [16], and the anticipated power available to the aircraft with a 75% efficient propeller versus the current consumed on the Cobalt 40 @20V, Cobalt 60 @32V, and the Cobalt 90 @40V. The data shows that the maximum power available to the aircraft by the Cobalt 40 was 375W, the maximum power produced by the Cobalt 60 was 675W and the maximum produced by the Cobalt 90 was 840W. Clearly the Cobalt 40 was incapable of producing the required 400W on 20V. It may be possible to increase the output of the Cobalt 40 slightly by maximizing the voltage rating at 24V, but the margin for error is small and the high current consumption was undesirable. The Cobalt 90 produced a maximum of 840W, which is nearly double the minimum required, but the motor is over 0.5 lbf heavier than the Cobalt 60 and the additional weight of the battery pack would cause another substantial increase in the RAC and reduce or final score by as much as 25%.

Table 4.11: Current vs. Power Available w/ 75% Eff. Prop.

	Power Available w/ 75% Eff. Propeller		
Current	Cobalt 40	Cobalt 60	Cobalt 90
(A)	(W)	(W)	(W)
10	110	190	210
15	170	270	360
20	230	390	480
25	285	480	600
30	330	585	720
35	375	675	840

The Cobalt 60 was selected because it produced a maximum of 675W with a 75% efficient propeller, which was well above the minimum power required of 400W. It also produced 480W of power at 32V and 25A. In addition, the current consumption was low enough to use the lightweight Sanyo KR1700 battery cell, which will help to reduce the RAC and increase the final score.

4.4.3 Propeller Parameters and Sizing Trades

The propeller has two dimensions, pitch and diameter. The thrust from any given propeller is a function of the rotational speed and the diameter, while the pitch determines the efficiency of the propeller at a

given flight speed. The efficiency increases as the pitch to diameter ratio approaches 1 and as the free stream velocity approaches the pitch speed [4]. There are multiple, complete theories that predict the static and dynamic thrust of a propeller as a function of rotational speed and free stream velocity, but the computations involved with these theories are long and elaborate. In addition, the accuracy of the results is low for small applications such as this. But empirical data shows that for this application, the thrust produced by a propeller is nearly a linear function of the free-stream velocity. This rule coupled with Equations 4.4.2 and 4.4.3 make it possible to streamline the propeller selection process [4].

$$PowerAbsorbed = 1.31 * Diameter^4 * Pitch * \left(\frac{RPM}{1000}\right)^3 \quad \text{(Eqn. 4.6)}$$

(Diameter and Pitch are measured in feet)

$$PitchSpeed = Pitch * RPM \quad \text{(Eqn. 4.7)}$$

Using equations 4.6 and 4.7 with a rotational speed of 4000 RPM, pitch speed of 50 mph, and 750W of shaft power absorbed, the following range of propellers was selected. Static thrust tests were conducted to verify that the initial static thrust approaches the anticipated required static thrust needed to produce an average of 5 lbf at takeoff. The results of these tests are in the following section.

$$19in \leq Diameter \leq 21in$$
$$14in \leq Pitch \leq 16in$$

4.4.4 Propulsion Analysis and Optimization

To validate the component selections listed in the preliminary trade studies, a test stand was developed to measure and record the static thrust, motor rpm, current consumption, voltage drop across the motor, and the motor run time. The team acquired a new AstroFlight Cobalt 60, 36 Sanyo KR1700 cells and a 20x15 APC propeller. Three tests were conducted using the new motor, batteries and propeller and an old 18x12 propeller and Sanyo 2400 battery pack. The results of the tests are summarized in Table 4.12

Table 4.12 shows that the maximum thrust was achieved in Test 2, with the Sanyo 2400 mA-hr cells and 20x15 propeller, but the projected RAC was extremely high due to the weight of the batteries. Test 3 resulted in the lowest RAC, 5.58, but the average static thrust was also much lower. In all three tests, the run time easily satisfied the required 3 minutes and provided a potentially priceless margin of error. The runtime for Test 3 was substantially longer than the results predicted in Section 4.4.1. The 1700 mA-hr cells were only expected to provide 24A for about 3.5 minutes. The issue was never completely resolved, but the current consumption was not a linear function of time and the average current consumption listed in Table 4.12 was calculated from a discrete set of data that was collected by hand, so the percent error may be high. The average static thrust value for Test 3 is lower than expected, but the initial static thrust values observed in the test (not shown here) were in excess of 7.0 lbf and remained in that range for the first 1.5 minutes before dropping off later in the test.

Table 4.12: Propulsion Test Summary

	Run Time (min)	Avg. Static Thrust (lbf)	Avg. Voltage Drop (Volts)	Avg. Current Consumption (Amps)	RAC
Test 1 18x12 Prop Sanyo 2400	8	6.0	34.2	18.4	8.01
Test 2 20x15 Prop Sanyo 2400	5	7.3	31.5	22.6	8.01
Test 3 20x15 Prop Sanyo 1700	5	5.75	25.5	24.1	5.58

The initial propulsion system requirements provided by the aerodynamics team stipulate that the propulsion system should be capable of producing 4 lbf of thrust at 50 mph for 3 minutes, and an average of 5 lbf thrust at takeoff. Based on the results of the trade studies and functional test results it was determined that the optimal propulsion system configuration was an AstroFlight Cobalt 60, 36 Sanyo KR1700 cells, and a 20x15 propeller. This result is a compromise between the specified requirements of the system and a need to minimize the final RAC of the aircraft. It should produce the thrust that was requested by the aerodynamics team for the majority of the flight envelope, but may result in lower than expected top speed. Correspondence with the competition organizers indicated that teams will be allowed to change propellers at any time. The propulsion team recommends flight-testing of this configuration to validate the propeller selection.

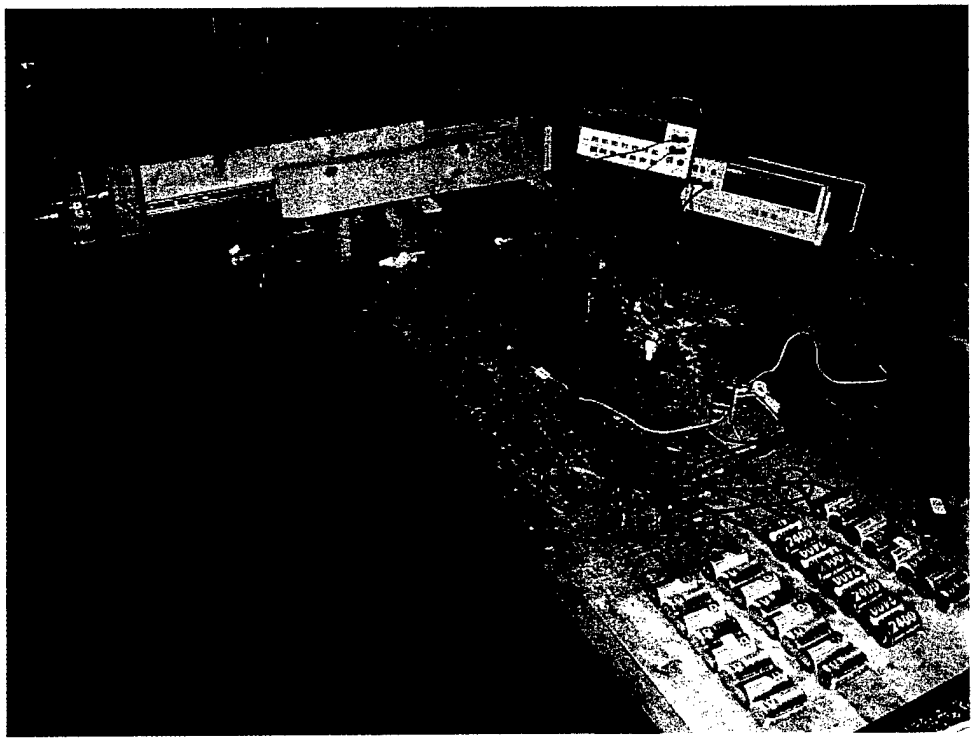


Figure 4.12: Propulsion Test Stand Setup

4.5 Predicted Mission Performance

The predicted mission performance of the aircraft is given in Table 4.13. The table includes the performance parameters as well as the estimated Total Mission Times for the various phases of each mission as described in the mission modeling. Several total time breakdowns are also provided.

Table 4.13: Predicted Mission Performance Parameters

Mission 1: Fire Bomber			
Flight Phase	**Velocity (mph)**	**Distance (ft)**	**Time (s)**
Loading time	--	--	60
Takeoff	31	125	5
Upwind	37	375	5.75
Turn 1 (1.22g's)	45	596.6	5.8
Downwind	45	1758	27
Turn 2 (1.16g's)	28	596.6	4
Final Approach	26	248	6.34
Landing Rollout	0	502	16.68
	Total Engine Run Time (minutes)		1.48
	Total Flight Time (minutes)		2.35
	Total Mission Time (minutes)		4.35
	Total Mission Score		7.660
Mission 2: Ferry Flight			
Flight Phase	**Velocity (mph)**	**Distance (ft)**	**Time (s)**
Takeoff	25	42	2.3
Upwind	64	458	4.9
Turn 1 (2g's)	64	318	3.4
Downwind	64	1637	16.4
Turn 2 (2g's)	64	318	3.4
Go Around	64	1000	10.6
Final Approach	26	249	6.3
Landing Roll	0	502	16.68
	Total Engine Run Time (minutes)		2.43
	Total Flight Time (minutes)		2.53
	Total Mission Time (minutes)		2.82
	Total Mission Score		0.36

5.0 DETAIL DESIGN

The detail design summarizes the components and systems from the preliminary design and displays the aircraft in a detailed drawing package. The final aircraft data are tabulated and the final RAC is computed.

5.1 Component Selection/System Architecture

The components selected for the final design of *Poseidon's Fury* can be divided into the following five sections: fuselage, drain system, wing, tail, and propulsion system. The fuselage consists of a main section and a removable cone section containing nine and five bulkheads, respectively. The main section of the fuselage cradles the drain system through holes cut through the bulkheads leaving room for the

valve to protrude from the fuselage underside. As mentioned in the preliminary design, the drain system consists of a 4-liter molded fiberglass tank and a custom ball valve. The custom ball valve can be opened and closed by a servo on the pilot's command.

The mid-wing connects to the fuselage through a donut assembly creating a support for the spar but avoiding the difficulties of a pass-through. It is designed with one spar connecting to the donut, a stabilizing rod counteracting twist, and a series of ribs to keep the modified Eppler 216 airfoil shape. The root of the wing permanently attaches to the fuselage but detaches in the center for disassembly into the box. Finally, the double-boom empennage attaches to the wing and supports the inverted V-tail. The entire empennage assembly breaks down for an easy fit into the box. Note, that the inverted V-tail section is made from a NACA 0012 airfoil and contains the same specifications as indicated in the preliminary design.

The propulsion system includes the motor, batteries, and propeller of *Poseidon's Fury*. The motor selected is an AstroFlight Cobalt 60 with superbox and is mounted at the nose of the aircraft. It connects to 36 battery cells, also located in the nose of the aircraft. The batteries are Sanyo 1700 mA-hr and weigh 3 lbs. Finally, the propeller selected is an APC 20x15 as discussed in the preliminary design section.

5.2 Sized Aircraft Data Tables

The data tables in this section display the geometry data, performance data, weight, and systems data of *Poseidon's Fury*.

5.2.1 Geometry Data Table

The geometry data table, Table 5.1, contains all primary physical dimensions of the aircraft.

Table 5.1: Geometry Data Table

Geometry	Value
Length (in)	58.5
Fuselage Diameter (in)	8.3
Main Wing Span (in)	112.0
Flaperon Volume (in^3)	12.3
Main Wing Area (in^2)	1176.0
Aspect Ratio	10.7
Dihedral Angle (Degrees)	1.5
Main Wing Airfoil	Eppler 216M
Tail Span (in)	27.5
Ruddavator Volume (in^3)	7.0
Tail Area (in^2)	250.9
Tail Airfoil	NACA 0012
Distance Between Tail Booms	26.5
Payload Capacity (Liters)	4.0
Orifice Diameter (in)	0.5

5.2.2 Performance Data Table

The performance data table, Table 5.2, shows various flight and handling characteristics through all flight regimes. Figures 5.1 and 5.2 depict the aircraft's turn radius and turn rate, respectively. All values are in meters and seconds.

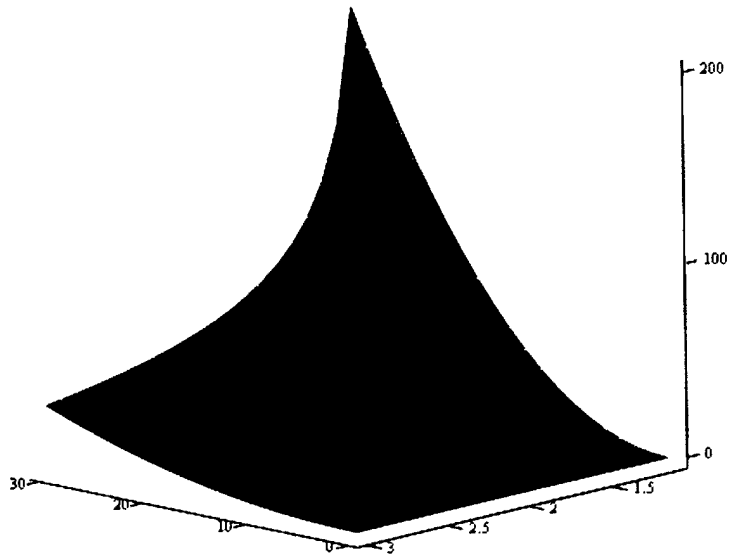


Figure 5.1: Turn Radius (Vertical Axis) as a Function of Velocity (Left Horizontal Axis) and Load Factor (Right Horizontal Axis)

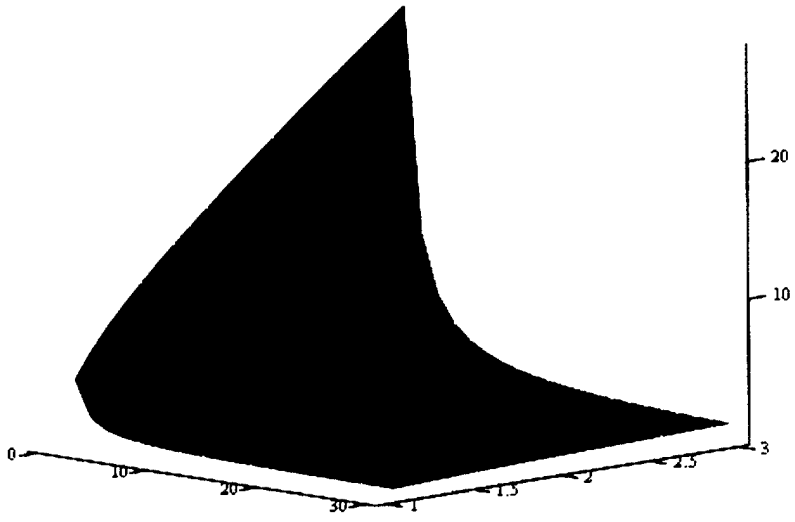


Figure 5.2: Turn Rate (Vertical Axis), as a Function of Velocity (Left Horizontal Axis) and Load Factor (Right Horizontal Axis)

Table 5.2: Performance Data Table

Performance Data	Gross Weight	Empty
Stall Velocity (mph)	26	21
Takeoff Velocity (mph)	31	25
Cruise Velocity (mph)	45	65
Max Velocity (mph)	71	71
Max Rate of Climb (ft/min)	1228	2101
Maximum Lift to Drag Ratio	11.577	11.577
C_L Max	1.8	1.8
Turn Radius at Cruise (2g) (ft)	78	152
Turn Rate at Cruise (2g) (deg/s)	79	55
Takeoff Distance (ft)	129	42

5.2.3 Weight Statement Table

Table 5.3 provides a list of the primary systems individual weight as well as the manufacturer's empty weight and gross weight:

Table 5.3: Weight Statement Table

Description	Total Weight (lbf)
Airframe	8.38
Propulsion System	4.99
Control System	0.61
Payload System	0.67
Payload	8.46
Manufacturers Empty Weight	14.65
Gross Weight	23.11

5.2.4 Systems Data Table

Table 5.4 includes all contributors to the control system and their names and types.

Table 5.4: System Data Table

Radio	Futaba 9 Channel
Receiver	Futaba 9 Channel PCM
Tail & Nose Gear Servos	Hitec Mini
Wing Servos	Futaba Low Torque
Valve Servo	Hitec High Torque
Batteries	Sanyo 1700 mAhr
Motor	AstroFlight Cobalt 60 Superbox
Propeller	APC 20x15
Gear Ratio	2.75 to 1

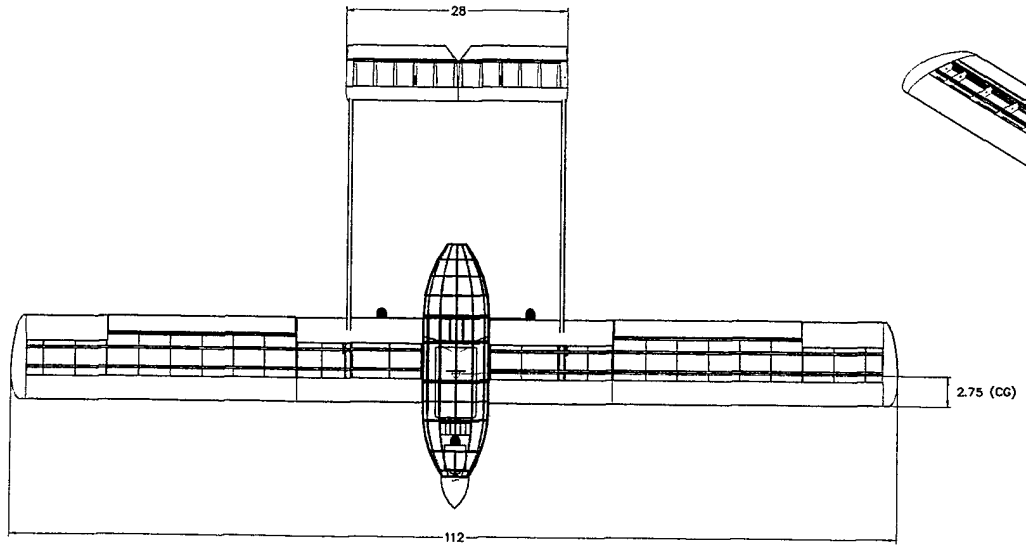
5.3 Final Aircraft Specifications/Drawing Package

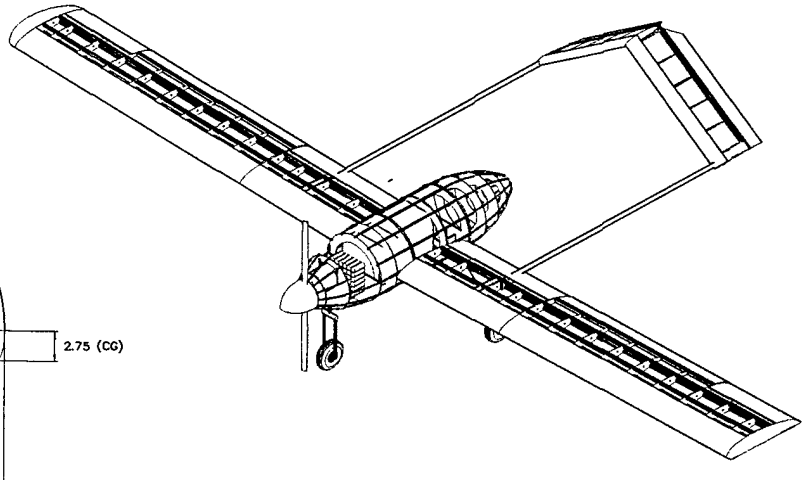
The drawing package displays the aircraft in five separate manners. The 3-view drawing displays the top, side, front, and isometric view of *Poseidon's Fury* and contains the major dimensions of the aircraft. The exploded view shows how the aircraft disassembles and the stowed view show how it fits into the box. The two final drawings take a closer look at the custom valve configuration and the propulsion schematic.

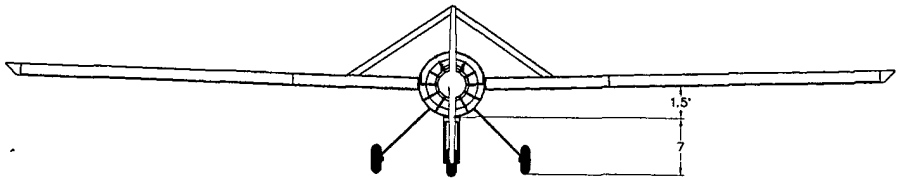
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DBF04 – 1010	1	–

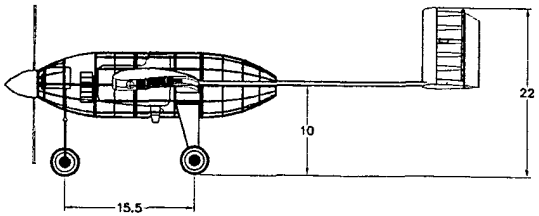
REVISIONS

ZONE	AUTHORITY	REV	DESCRIPTION	DATE	APPROVED









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.XX = ±.010
.XXX = ±.005
ANGLES MAY VARY ± 2°

PERFECT COAXIALITY IS REQUIRED FOR ALL RELATED FEATURES AT MMC.

SURFACE ROUGHNESS √ NOT TO EXCEED 63 RMS ON FINISHED SURFACES

BREAK SHARP EDGES .005 TO .015

MATERIAL

APPROVALS		DATE
DRAWN	M. DENTON	2/01/04
CHECKED	J. MOORE	2/06/04
ENGINEER	K. MORACE	2/10/04
PROJECT ENGINEER	R. HOLMES	2/21/04
APPROVED	C. DANIELS	2/21/04

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TITLE: POSEIDON TOP ASSY

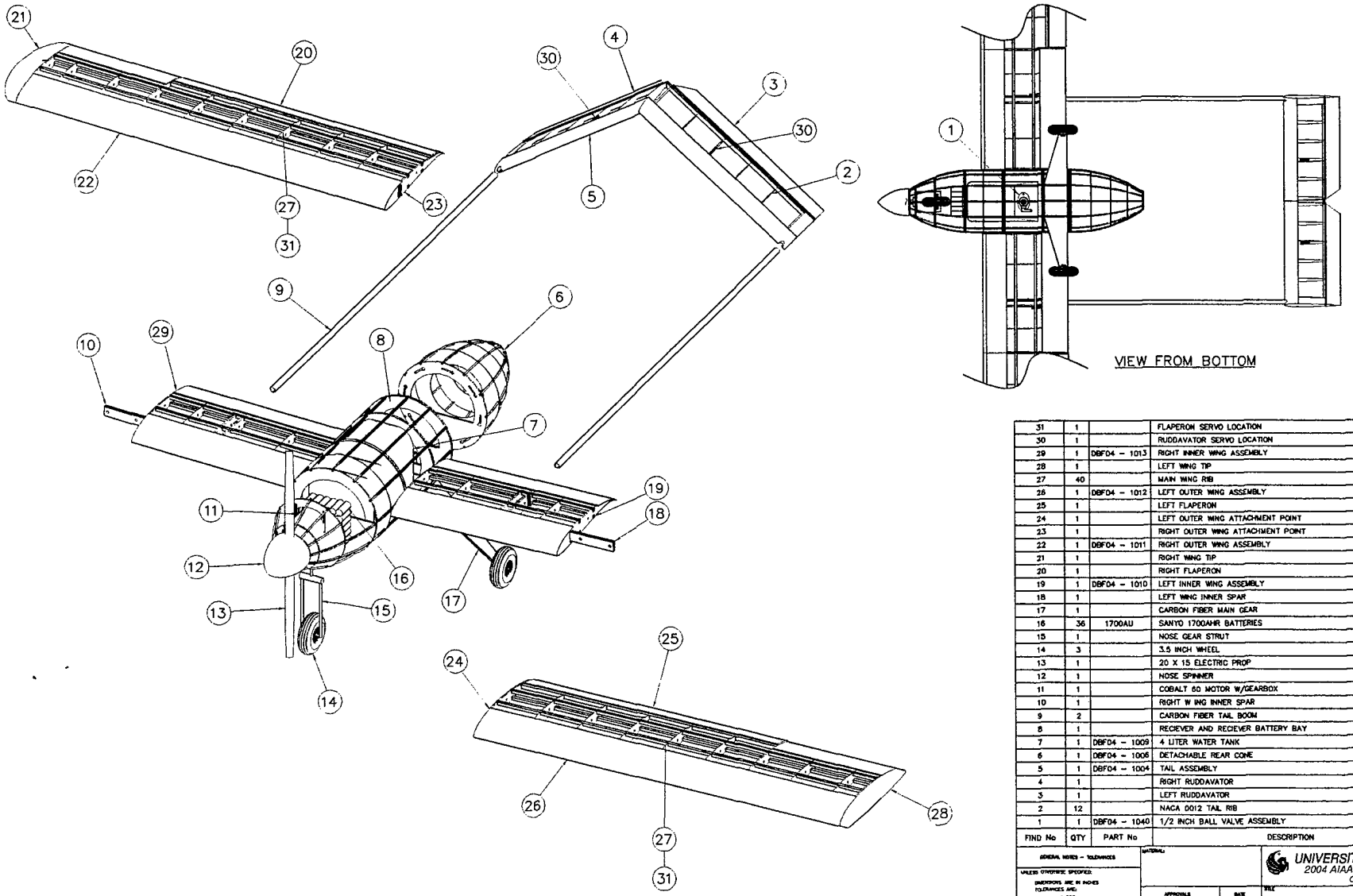
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D	DBF04 – 1010	–

SCALE: 1/16 | SHEET 1 OF 1

DWG. NO.	REV.	SHT.
DBF04 – 1020	1	–

REVISIONS

ZONE	AUTHORITY	REV	DESCRIPTION	DATE	APPROVED



VIEW FROM BOTTOM

FIND No	QTY	PART No	DESCRIPTION	MATERIAL
31	1		FLAPERON SERVO LOCATION	
30	1		RUDDAVATOR SERVO LOCATION	
29	1	DBF04 – 1013	RIGHT INNER WING ASSEMBLY	
28	1		LEFT WING TIP	BALSA
27	40		MAIN WING RIB	LITE PLY
26	1	DBF04 – 1012	LEFT OUTER WING ASSEMBLY	
25	1		LEFT FLAPERON	BALSA
24	1		LEFT OUTER WING ATTACHMENT POINT	
23	1		RIGHT OUTER WING ATTACHMENT POINT	
22	1	DBF04 – 1011	RIGHT OUTER WING ASSEMBLY	
21	1		RIGHT WING TIP	BALSA
20	1		RIGHT FLAPERON	BALSA
19	1	DBF04 – 1010	LEFT INNER WING ASSEMBLY	
18	1		LEFT WING INNER SPAR	LITE PLY/CARBON FIBER
17	1		CARBON FIBER MAIN GEAR	CARBON FIBER
16	36	1700AU	SANYO 1700AHR BATTERIES	
15	1		NOSE GEAR STRUT	
14	3		3.5 INCH WHEEL	
13	1		20 X 15 ELECTRIC PROP	NYLON COMPOSITE
12	1		NOSE SPINNER	
11	1		COBALT 60 MOTOR W/GEARBOX	
10	1		RIGHT W ING INNER SPAR	LITE PLY/CARBON FIBER
9	2		CARBON FIBER TAIL BOOM	CARBON FIBER
8	1		RECIEVER AND RECIEVER BATTERY BAY	
7	1	DBF04 – 1009	4 LITER WATER TANK	FIBERGLASS
6	1	DBF04 – 1008	DETACHABLE REAR CONE	
5	1	DBF04 – 1004	TAIL ASSEMBLY	
4	1		RIGHT RUDDAVATOR	BALSA
3	1		LEFT RUDDAVATOR	BALSA
2	12		NACA 0012 TAIL RIB	LITE PLY
1	1	DBF04 – 1040	1/2 INCH BALL VALVE ASSEMBLY	RESIN

NOTES:

1. FIBERGLASS TANK FIXED BETWEEN BULKHEADS 5, 6, AND 7.

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.X = ± .020
.XX = ± .010
ANGLES MAY VARY ± 2°

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BREAK SHARP EDGES .005 TO .015

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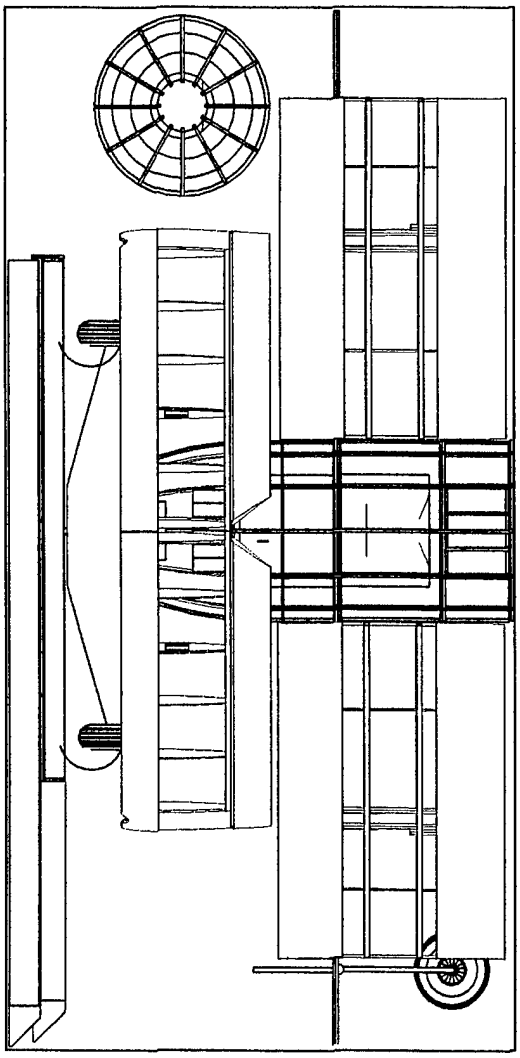
APPROVALS		DATE
DRAWN	M. DENTON	2/01/04
CHECKED	J. MOORE	2/06/04
ENGINEER	K. MORACE	2/10/04
PROJECT ENGINEER	R. HOLMES	2/21/04
APPROVED	C. DANIELS	2/21/04

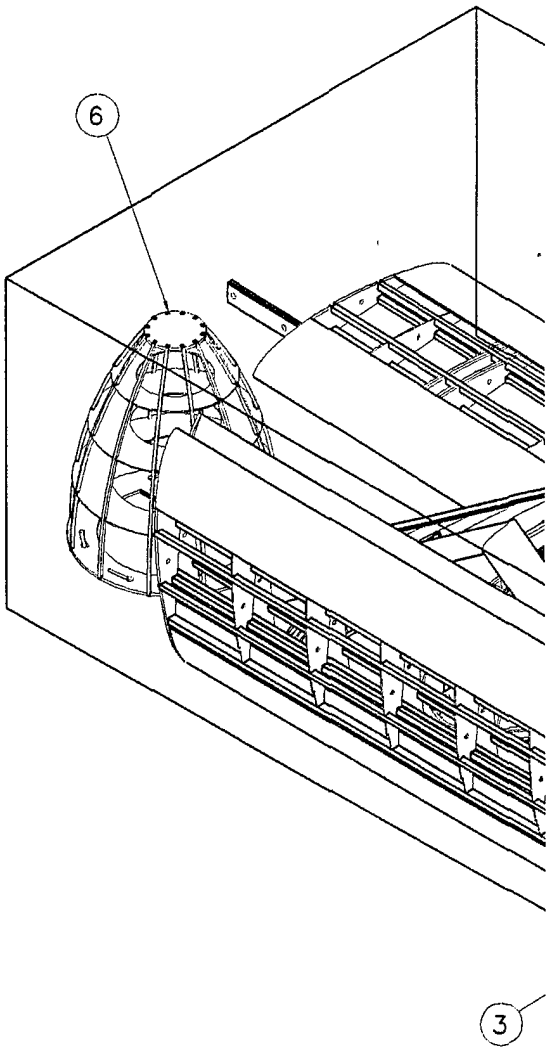
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ORLANDO, FLORIDA

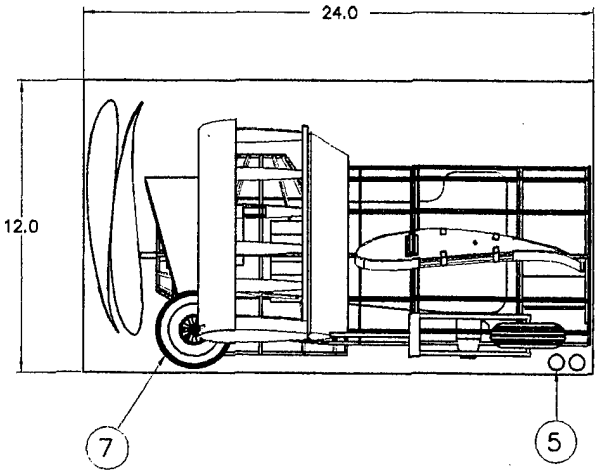
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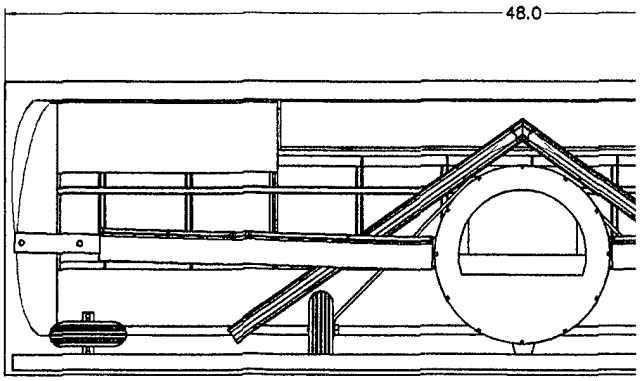
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D	DBF04 – 1020	–

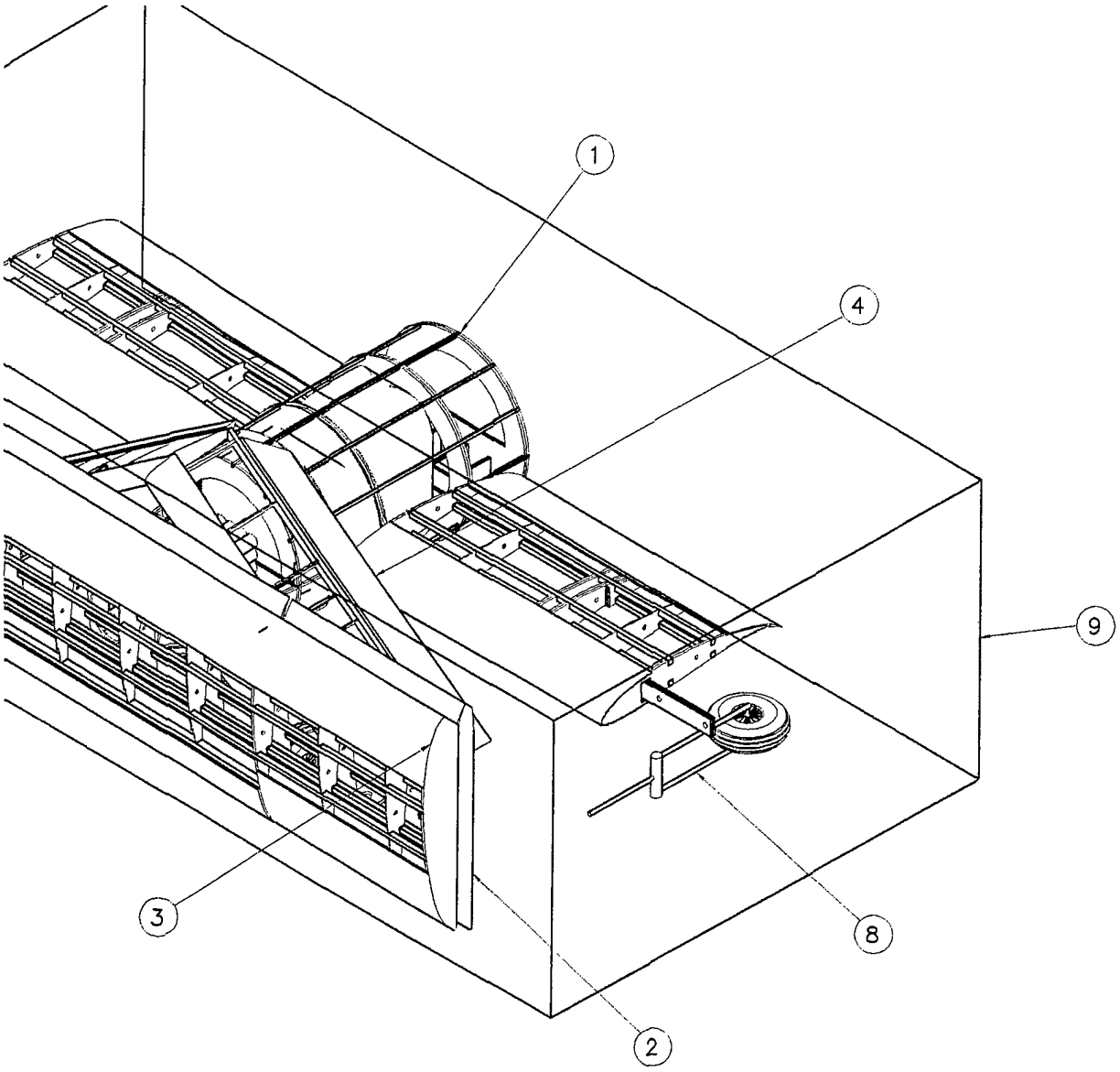
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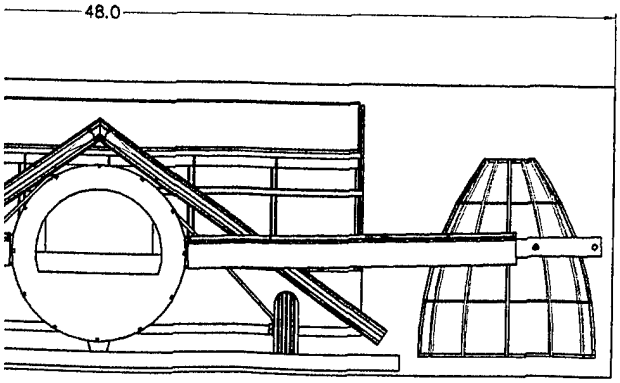



6
3


24.0
12.0
7
5


48.0





FIND No	QTY	PART No	DESCRIPTION	MAT
9	1		1' x 2' x 4' BOX	
8	1	DBF04 – 1008	NOSE GEAR ASSEMBLY	
7	1	DBF04 – 1007	MAIN GEAR ASSEMBLY	
6	1	DBF04 – 1006	DETACHABLE REAR CONE	
5	2	DBF04 – 1005	CARBON FIBER TAIL RODS	
4	1	DBF04 – 1004	TAIL ASSEMBLY	
3	1	DBF04 – 1003	RIGHT OUTER WING ASSEMBLY	
2	1	DBF04 – 1002	LEFT OUTER WING ASSEMBLY	
1	1	DBF04 – 1001	FUSELAGE ASSEMBLY	

GENERAL NOTES – TOLERANCES

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.XX = ±.010
.XXX = ±.005
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SURFACE ROUGHNESS √ NOT TO EXCEED 63 RMS ON FINISHED SURFACES

BREAK SHARP EDGES .005 TO .010

MATERIAL:

APPROVALS		DATE
DRAWN	M. DENTON	2/01/04
CHECKED	J. MOORE	2/06/04
ENGINEER	K. MORACE	2/10/04
PROJECT ENGINEER	R. HOLMES	2/21/04
APPROVED	C. DANIELS	2/21/04

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ORLANDO, FLORIDA

TITLE: STOWED CONFIGURA

SIZE: D | DWG. NO.: DBF04 – 1030

SCALE: | SHEET 1 (

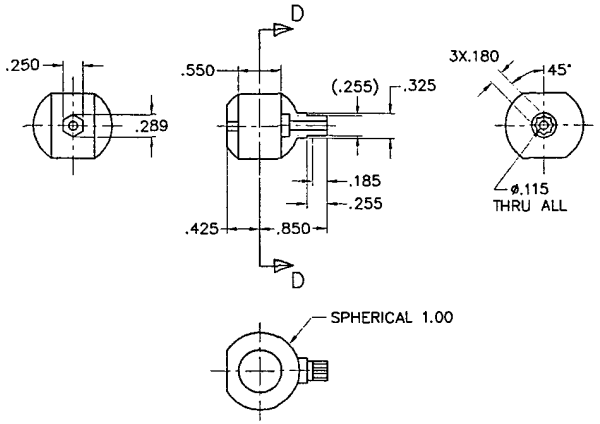
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5 | 4 | 3 | 2 | 1

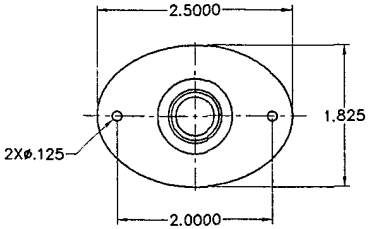
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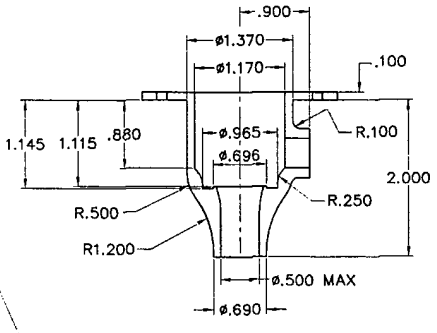
REVISIONS

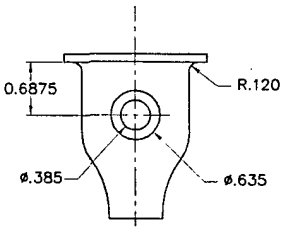
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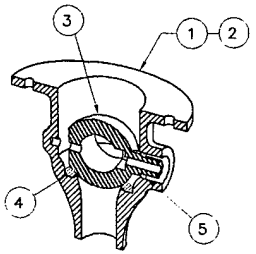
BALL VALVE DETAIL







HOUSING DETAIL



ASSEMBLY

FIND No	QTY	PART No	DESCRIPTION	MATERIAL
5	1	2–010	O–RING	VITON
4	1	2–209	O–RING	VITON
3	1		BALL VALVE	STEREOLITH
2	1		VALVE HOUSING	STEREOLITH
1	1		VALVE ASSEMBLY	STEREOLITH

GENERAL NOTES – TOLERANCES

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TOLERANCES ARE:

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.XX = ±.010
.XXX = ±.005
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BREAK SHARP EDGES .005 TO .010

MATERIAL: SEE CHART

APPROVALS	DATE
DRAWN M. DENTON	2/01/04
CHECKED J. MOORE	2/06/04
ENGINEER K. MORACE	2/10/04
PROJECT ENGINEER R. HOLMES	2/21/04
APPROVED C. DANIELS	2/21/04

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ORLANDO, FLORIDA

TITLE: VALVE ASSEMBLY

SIZE	DWG. NO.	REV.
C	DBF04 – 1040	–

SCALE 1/1 | DO NOT SCALE | SHEET 1 OF 1

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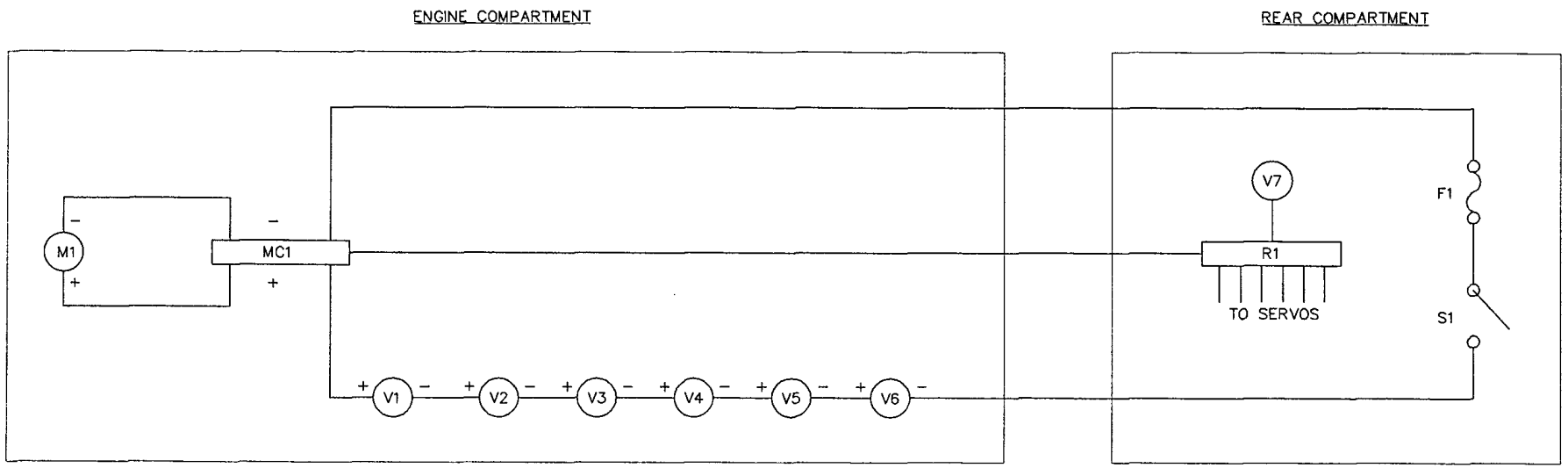
DWG. NO. DBF04 – 1050 | REV. 1 | REV. –

REVISIONS

ZONE	AUTHORITY	REV	DESCRIPTION	DATE	APPROVED

NOTES:

1. ALL WIRE TO BE 14 GAUGE SILVER PLATED TEFLON COATED CONNECTING WIRE.
2. ALL WIRE CONNECTORS TO BE RATED FOR 40VDC AT 40AMPS.



SYMBOL	QTY	PART No	DESCRIPTION	SUPPLIER
F1	1		SAFETY FUSE	MAXI-BLOW
M1	1	6615	COBALT 60 MOTOR W/GEARBOX	ASTRO FLIGHT
MC1	1	204D	MOTOR CONTROLLER	ASTRO FLIGHT
R1	1	R138DP	RECIEVER	FUTABA
S1	1		KILL SWITCH	
V1	1	KR-1700AU	BATTERY PACK (6 INDIVIDUAL CELLS)	SANYO
V2	1	KR-1700AU	BATTERY PACK (6 INDIVIDUAL CELLS)	SANYO
V3	1	KR-1700AU	BATTERY PACK (6 INDIVIDUAL CELLS)	SANYO
V4	1	KR-1700AU	BATTERY PACK (6 INDIVIDUAL CELLS)	SANYO
V5	1	KR-1700AU	BATTERY PACK (6 INDIVIDUAL CELLS)	SANYO
V6	1	KR-1700AU	BATTERY PACK (6 INDIVIDUAL CELLS)	SANYO
V7	1	2001	RECIEVER BATTERY PACK	DURATRAX

GENERAL NOTES – TOLERANCES

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.XX = ± .010
.XXX = ± .005
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BREAK SHARP EDGES .005 TO .010

MATERIAL

APPROVALS		DATE
DRAWN	C. DANIELS	2/22/04
CHECKED	M. DENTON	2/22/04
ENGINEER	K. MORACE	2/22/04
PROJECT ENGINEER	J. MOORE	2/22/04
APPROVED	R. HOLMES	2/22/04

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ORLANDO, FLORIDA

TITLE: WIRING SCHEMATIC

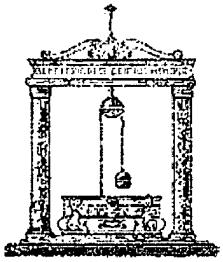
SIZE: D | DWG. NO. DBF04 – 1050 | REV. –

SCALE: | SHEET 1 OF 1

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ROMA
"La Sapienza"
FACOLTÄ DI INGEGNERIA



Michelangelo Team



CESSNA / STUDENT
ONR / DESIGN/BUILD/FLY
COMPETITION
An AIAA Student Activity

Table of Contents

1.0 EXECUTIVE SUMMARY

1.1 CONCEPTUAL DESIGN

1.1.1 Valuable Conceptual Design

1.1.2 IPT Evaluation

1.1.3 Concept al Results

1.2 PRELIMINARY DESIGN

1.2.1 Preliminary Design Solutions

1.2.2 Preliminary Design Tools

1.2.3 Preliminary Results

1.3 DETAIL DESIGN

1.3.1 Detail Design Scolutions

1.3.2 Detail Design Tools

1.3.3 Detail Design Result

1.4 MANUFACTURING AND PROTOTYPE TESTING

2.0 MANAGEMENT SUMMARY

2.1 TEAM ORGANIZATION AND PERSONNEL ASSIGNMENT WORKING AREAS

2.2 RESPONSIBILITIES OF EACH TECHNICAL GROUP

2.2.1 Aerodynamics and Flight Mechanics Groups

2.2.2 Propulsion Group

2.2.3 Structures and Systems Groups

2.3 SCHEDULING, DOCUMENT, AND CONFIGURATION CONTROL

3.0 CONCEPTUAL DESIGN

3.1 DESIGN PARAMETERS

3.1.1 Primary Design Parameters

3.1.2 Design Parameter Sensitivity Analysis

3.1.3 Rated Aircraft Cost Analysis

3.2 AIRCRAFT CONFIGURATION

3.2.1 Alternative Fuselage and Tail Configuration

3.2.2 Alternative Wing Configuration

3.3 STRUCTURAL CONFIGURATION

3.3.1 Figures of Merit

3.3.2 Assumptions Made and Design Parameters Investigated

3.3.3 Alternative Wing Structures

3.3.4 Alternative T-tail Structures
3.3.5 Alternative Fuselage Structures
3.3.6 Alternative Landing Gear Configurations
3.4 ALTERNATIVE POWER PLANT CONFIGURATIONS
3.5 ANALYTICAL TOOLS
3.5.1 Optimization Program Architecture
3.5.2 Other Methods Used
3.6 FINAL AIRCRAFT CONFIGURATIONS

4.0 PRELIMINARY DESIGN
4.1 INITIAL TRADE STUDY RESULTS
4.1.1 Study of Model and Number of Batteries
4.1.2 Study of Take Off Battery Performances
4.1.3 Study of Cruise Battery Power Requirements
4.1.4 Study of Wing Planform Area Requirements
4.1.5 Study of Wingspan Requirements
4.1.6 Study of Aspect Ratio Requirements
4.1.7 Study of Plane Structural Weight Effects
4.1.8 Study of Rated Aircraft Cost Effects
4.2 AERODYNAMIC CONSIDERATIONS
4.2.1 Wing Figures of Merit, Assumptions, and Analysis
4.2.2 Fuselage Figures of Merit, Assumptions, and Analysis
4.2.3 Tail Figures of Merit, Assumptions, and Analysis
4.3 STRUCTURAL ANALYSIS
4.3.1 Figures of Merit
4.3.2 Design Parameters and Trade Studies Investigated
4.3.3 Wing Assumptions and Structural Analysis
4.3.4 Tail Assumptions and Structural Analysis
4.3.5 Structural Analysis
4.3.6 Main Gear Assumptions and Structural Analysis
4.4 PROPULSION ANALYSIS
4.4.1 Figures of Merit
4.4.2 Design Parameters and Trade Studies Investigated
4.4.3 Weighted Decision of Motors
4.4.4 Investigation of Graupner Motor
4.5 ANALYTICAL TOOLS
4.5.1 Optimization Program Architecture

4.5.2 Other Methods Used

4.6 FINAL AIRCRAFT CONFIGURATIONS

4.6.1 Wing and Power Loading

5.0 DETAIL DESIGN

5.1 AERODYNAMIC PERFORMANCE ANALYSIS

5.1.1 Final Configuration Features

5.1.2 Estimated Mission Performance

5.1.3 Takeoff and Climb

5.1.4 Flight Conditions

5.2 PROPULSIVE PERFORMANCE ANALYSIS

5.3 STRUCTURAL CONSIDERATIONS

5.3.1 Fuselage Structural Details

5.3.2 Wing Structural Details

5.4 DETAIL DESIGN ASSUMPTIONS AND COMPARISONS

6.0 MANUFACTURING PLAN

6.1 MANUFACTURING PROCESS INVESTIGATED

6.2 PROCESSES SELECTED FOR COMPONENT MANUFACTURING

6.2.1 Fuselage Manufacturing Process and Tooling

6.2.2 Wing and Tail Manufacturing Process and Tooling

6.2.3 Landing Gear Manufacturing Process and Tooling

6.3 ANALYTIC METHODS INCLUDING COST, SCHEDULING AND SKILLS MATRIX

6.3.1 Manufacturing Cost

6.3.2 Skills Matrix

6.3.3 Manufacturing Scheduling

***7.0* TESTING**

7.1 STATIC TESTS

7.2 FLIGHT TESTS

8.0 REFERENCES

1.0 Executive Summary

This report has been redacted by Michelangelo Team (University La Sapienza of Rome , Italy) in the sphere of the works performed for attending at the AIAA 2003/2004 Design/Build/Fly (DBF) Competition .Specifically in this paper we describe the mainlines followed by the group during several tasks of planning, design, construction and testing of an electric and propeller driven unmanned aircraft according to competition rules. At first glance, we want outline the main goal of the competition is to realize an optimized aircraft in terms of aerodynamic, propulsive, flight and structural characteristics, subjected to a continuum control of costs in order to minimize the Rated Aircraft Cost.

1.1 Conceptual Design

In first of all the group planned the whole operation defining two macrophases in their turn divided in a series of steps concerning several areas of work. First macrophase involves, in sequence:

- Planning
- Production
- Flight tests and model tuning ;

Second macrophase regards:

- Definition and organization of spare parts (and relative tools of installation);
- Managing of travel logistic;
- Planning of tasks for each team member during the competition.

As regards first macrophase, an IPT (Integrated Product Team) was formed to reach following goals:

- CPD (Concurrent Product Definition);
- DFMA (Design for Manufactoring and Assembly);
- DAT (Determinant Assembly Techniques).

IPT performed several studies in the areas of Aerodynamic, flight mechanics, propulsion, systems and structures during which the relative groups were in constant contact each other. CPD was obtained integrating the best design concept in each area of work where relatives groups performed sensitivity studies scrutinizing each rule of the competition to focus design specifics with the aim to maximize overall scoring. Each group proposed some design concepts in their respective technical areas to meet defined design requirements;then IPT performed a design optimization analysis which results were used to create figures of merit to compare various concepts and evaluate different approaches of aircraft definition to achieve the highest possible score in terms of RAC as regards design/building competition and best flight performances concerning flight competition. After several evaluations for each iteration of optimization process, an optimum region was determined according to whom each group tailored their respective ideas into their final concepts.

1.1.1 Valuable Conceptual Designs

The aerodynamic group conceived three degrees of analysis: analysis of wing airfoil according to low Reynolds number foreseen, definition of fuselage geometry subjected to propeller position and, at least, wing and horizontal tail position with respect to the fuselage keeping in count the location of tank vane. Different combinations of fuselage shapes, tank shape and dumping water hole, wing configurations and tail configurations were evaluated for the components that made up the different configurations. Concurrently structural evaluations was performed in order to select only viable alternatives in terms of feasibility of construction, RAC minimization, control of weight and piloting . Three propulsion system alternatives were defined evaluating several schemes generated combining number of engines, propeller characteristics and position, estimated weight and drags, length of flight and battery characteristics .

1.1.2 IPT Evaluation

A decision making process was performed after each group of IPT defined all possible schemes of solution concerning respective work ambits. Our process was split in two steps: during first one score sensitivities were determined using an optimization code where all aspects of the mission were keep in count; the code was developed employing models of weight and drag, aerodynamic and propulsive effects of various components and RAC. Time of construction and RAC was effected by each model which, moreover, in turn adjusted the overall score of a specific configuration. Subsequently, six parameters were selected. The performance program adjusted six parameters using random variables and found local maximums in the performance of the plane. Weighted decision matrices were constructed using the sensitivities studies produced from these models. Finally configurations were evaluated in the weighted decision matrices for scoring potential.

1.1.3 Conceptual Results

The result of the conceptual design work, with the highest scoring potential was a mid-high wing monoplane with a Double Mobile Independent Tail (DMIT) in which, moreover, is inserted a three-blade propeller (a single engine configuration with a pusher propeller resulted to be the optimum propulsive scheme).

1.2 Preliminary Design

After having frozen aircraft configuration the groups focused their attention to aircraft sizing and control surface sizing, structural design and propulsion components definition and configuration In detail, aerodynamics and flight mechanics groups defined number, location and size of control surfaces, valuating ,concurrently, static stability of the aircraft versus manoeuvring necessities. The structures group, assumed use of wood and polystyrene for structure, performed static and dynamic stress analysis of the aircraft enclosed therein simulations of drop test and aircraft lifting by tips. Propulsion group determined the best combination of motor, reducer, propeller and batteries to be tested.

1.2.1 Preliminary Design Solutions

During preliminary design work, aerodynamicists and flights mechanics evaluated wing, fuselage and tail sizing combinations , concurrently defining dimensions of surface controls. Then structurists began to work for a preliminary sizing of primary structure and definition of construction procedures; tank vane geometry, place for engine and radio racks in addiction to space for control cables was determinant design ties. Meantime propulsion group evaluated different engine, reducer, batteries and propeller configurations in order to scale down number of system components and overall weight.

1.2.2 Preliminary Design Tools

Preliminary modelling of the aircraft was initiated by developing bi-dimensional and then three-dimensional models employing Studio 3-D program in a single computer work-station: in this way all members of the IPT developed a feel for initial component sizing and placement early in the design. Then this was exported in a F.E. program in order to allow aerodynamicists to perform their fluidodynamic studies whose results influenced both structurists work, in terms of geometries of each part of the airplane, and propulsion group activity finalized to evaluate the complete number of batteries to employ with the aid of a dedicated computer code. During the whole work, all group work stations were connected each other by a LAN in order to reduce transfer data times.

1.2.3 Preliminary Results

Once preliminary design phase was ended, type, sizes and construction procedures of aircraft components were finalized. The flight-mechanics group determined the static stability derivatives and provided geometries, dimensions and tolerances on wing, tail, fuselage and control surfaces to the structures group, while gave a propulsion mission profile to the propulsion group which so tailored own studies towards well-defined targets in terms of electricity consumption.

The structures group integrated aerodynamics data with forecasted mission loads in order to define primary structure. At the end of this step final dimensions and makeup of each primary structural component were available for the detail design phase.

1.3. Detail Design

The detail design was generated from components defined during preliminary design phase. This time aerodynamicists, with the help of CFD (Computational Fluid-dynamics), performed dynamic stability analysis valuating handling characteristics and control systems requirements. Meantime, the propulsion group finalized propulsion system characteristics in terms of engine model, batteries number (and model), propeller model. Instead the Structures group was able to complete a three dimensional model and detailed drawings of surfaces, structures and subsystems only after having received a report from aerodynamicist about the feasibility (luckily confirmed) of employing rotating (in a conjugated way) wings

as large ailerons in order to eliminate apposite mobile surfaces for their purpose maximizing, in such way, the RAC. During whole work also pilot's requirements were considered.

1.3.1 Detail Design Solutions

Detail design alternatives in each group focused on secondary components and process used in the construction and operation of the aircraft. Once again figures of merit system was implemented to analyze the specific impact of each decision to score. Components system, like cargo handling, gears, electric wire and control cable routing, transmission power (between engine and propeller) mechanism, wings and elevator insertion mechanisms were finalized. Different methods and hardware alternatives for realizing the aircraft components were considered. Moreover manufacturing processes were examined for each component. The propulsion group tested the performance of different propellers in combination with different models and numbers of batteries and motor types during real test and computers simulations via dedicated software; this group also evaluated some different types of fuse holders and methods for charging batteries developing specific procedures to be followed during the competition. Flight mechanics group determined control system alternatives in the form of servo requirements and control characteristics.

1.3.2 Detail Design Tools

The aerodynamics group used Ansys (in which was imported 3-D Studio model) code for evaluation of the aerodynamic field around the fuselage and its modification due to propeller, wings and simulated communications relay device presence; airfoils were analyzed employing "Profili" code while stability derivatives was determined using "Fluent" code. The structure group also employed Ansys program for Finite Elements analysis while final design and details of each component was obtained under a CAD program. A scheduling program was implemented to develop a manufacturing process schedule. The Propulsion group used the "Motocalc" software to evaluate engine model, batteries and propeller possible combinations and a dynamometer to compare actual system performance to analytical predictions.

1.3.3 Detail Design Result

At the end of the detail design phase, all aircraft geometries and dimensions and system components (and their respective lay-out) had been selected while aircraft performance capabilities and control characteristics had been determined. It was, therefore, possible to develop a manufacturing process. Moreover, finally detailed production drawings had been prepared together with emanation of some recommendations for construction process, to be verified by apposite checks audits.

1.4.Manufacturing and Prototype Testing

As in all constructive process, the project was subjected to some refinements during works; audits were useful to control critical points of realization like lower side of wings near the trailing edge, high degree of smoothness of surfaces of wing roots and CWB (Center Wing Box) which had to be side by side without interference (pernicious for wing rotation) during flight, alignment of ribs holes in the tail for the exact insertion of the pin above the elevator and alignment of the tail with the fuselage axe. As regards flight test, it was determined the utility of providing wing tips with slim soft steel appendices working as skates to prevent wings from damages due to uncontrolled rolls during take-off/landing manoeuvres due to relative short distance between wheels of main gear. On the other hand this solution is aerodynamically (and so in terms of energy consumption) better than the adoption of a larger distance between main gear wheels.

2.0 Management summary

2.1 Team Organization and Personnel Assignment Working Areas

The Michelangelo team was divided in five technical groups: aerodynamics, flight mechanics, structures, propulsion and systems. Each technical group was comprised of a lead engineer; structures group was comprised of two component specialists. Figure 1 outlines the architectural structure of the team. The chief engineer was responsible for overall team direction and performance as well as ensuring that each group had the tools and information necessary to complete assigned tasks. The lead engineer of a group was responsible for coordinating the efforts of the underclassmen and, in the case of structures group, also the efforts of the component specialists to meet the overall goals of that technical group, as set forth by the chief engineer. Together, the chief and lead engineers assured that the design process followed a logical progression from day to day and ensured that every aspect was properly considered.

2.2 Responsabilities of Each Technical Group

According the aim of an IPT, we can affirm the aircraft conceived by Leonardo team was the result of interoperability of all groups each other. The effectiveness of such an integration is demonstrated by the fact that each decision made by one group affected the constraints placed on the conceptual design of the other groups. Each group agreed upon each single concept before further progress was made. Moreover, resposabilities of each group were shared among its members. Managers individual contributions are displayed in figure 1.

Project manager: Massimo Florenzi		
Treasures: Giovanna Orlando	Procurement: Simone Berardi	Logistics: Clelio Ruscitti
Aerodynamics Leader: Clelio Ruscitti	**Propulsion** Leader: Simone Berardi	**Structures** Leader: Giovanna Orlando
Giovanna Orlando Massimo Florenzi Simone Berardi	Clelio Ruscitti Giovanna Orlando Massimo Florenzi	Clelio Ruscitti Massimo Florenzi Simone Berardi

2.2.1 Aerodynamics and Flight Mechanics groups

Aerodynamic group, after incorporated competition regulations, oversaw the mathematical modelling process as regards fluiddynamics field due to fuselage and propeller. Meantime propulsion group made some rough estimates of performance parameters involved during the flight. Then both groups combined their first evaluations and generated new and more refined estimates of performance parameters for themselves and the other groups as a base for conceptual studies. Subsequently aerodynamics group, in conjunction with flight mechanics group, progressed to analyzing different aircraft configurations by scoring potential. Figures of merit were defined to further rate the scoring potential of each concept. The final result of the aerodynamic conceptual study was the final aircraft configuration. However developing of mathematical model followed up through refinements in order to improve lift, reduce drag and interference between fuselage and wing, wing and equilibrator, this one and tail also evaluating spin prevention by an analysis concerning the TDPF (Tail Damping Power Factor) which, automatically conducted to a correct tail dimensioning . In this way was possible to estimate a first degree of stability and controllability of the aircraft .

2.2.2 Propulsion Group

The propulsion group was responsible for the design and evaluation of different motor, propeller, reducer and battery (in terms of number and models) combinations. Of course the concepts were judged on the effective use of available power, possible performances of each component and effect on Rated Aircraft Cost. During the conceptual phase, the propulsion group decided how many engine and propellers would be required to produce the thrusts (specially during take-off run) calculated by classic mathematical models. Moreover it was decided whether a pusher or a tractor configuration would have been implemented. At least the propulsion group developed analytical and experimental means by which the wide selection of motors and propellers was narrowed to a few combinations to be prototype tested. Also this group continued in during following steps an analysis more and more refined of the components

checking validity of preliminary and historical data. Prototype analysis allowed the group to further refine the combination of propulsion components (especially in terms of type and number of batteries and selection of reducer once the motor was chosen) to best met the mission requirements and help minimize the Rated Aircraft Cost. Propulsion group was the only affected , in terms of scheduling, by typical delivery delays of parts.

2.2.3 Structures and Systems Groups

This group was responsible for the structural design, the integration of components and at least, but not last, c.g. position and weight controlling. It worked together with system group in order to keep in count storage and handling needs for cargo and defining spaces for wire and cables passage. In this way , during the conceptual phase, the structures group examined several different methods of building the general structures required. As kinds of materials was fixed, the structures group evaluated only few possible type of woods employable. An other task was the definition of a construction algorithm to be followed as regards definition of attaches of pieces each other. An other head-ache was the definition and integration of the water tank and it's shape to minimize the moment effects, automatic complete dumping of water minimizing servos quantities in order to affect as less as possible RAC. Finally it was clear from the beginning that it was necessary to reduce weight of the rear portion of fuselage without weaken resistance but also without adopting sharp boom (aesthetically not accepted from members of Michelangelo team) threatening to preserve fluidynamics field originated in the fore portion of fuselage. During the preliminary design phase, the structures group used loading parameters given by the aerodynamics and propulsion groups to decide on the quantity and size of primary structural components in the wing, fuselage and tail. Due to decision of mounting the propeller on the trailing edge of the tail, it was possible to adopt a short undercarriage, bypassing typical problems of form resistance. However this idea obliged to simulate drop test in order to evaluate the capability of the structures to absorb violent shocks (for example due to improvise gusts) during landing with low damping (due to rigid connection of wheels to fuselage). This problem was, besides, exacerbated by the fact that internal structures of the airplane presented a large discontinuity due to the presence of the cargo compartment . So It was necessary to conceive a box structure in order to absorb torsion and flexion. During the detail design phase, the landing gear and primary structures were tested for compliance to preliminary parameters. Further analysis resulted in the sizing of the secondary structures. Finally the structures group also considered all tooling and construction methods to be used during the manufacturing phase. Final drawings and implementation of the manufacturing processing were under the responsibility of structures group.

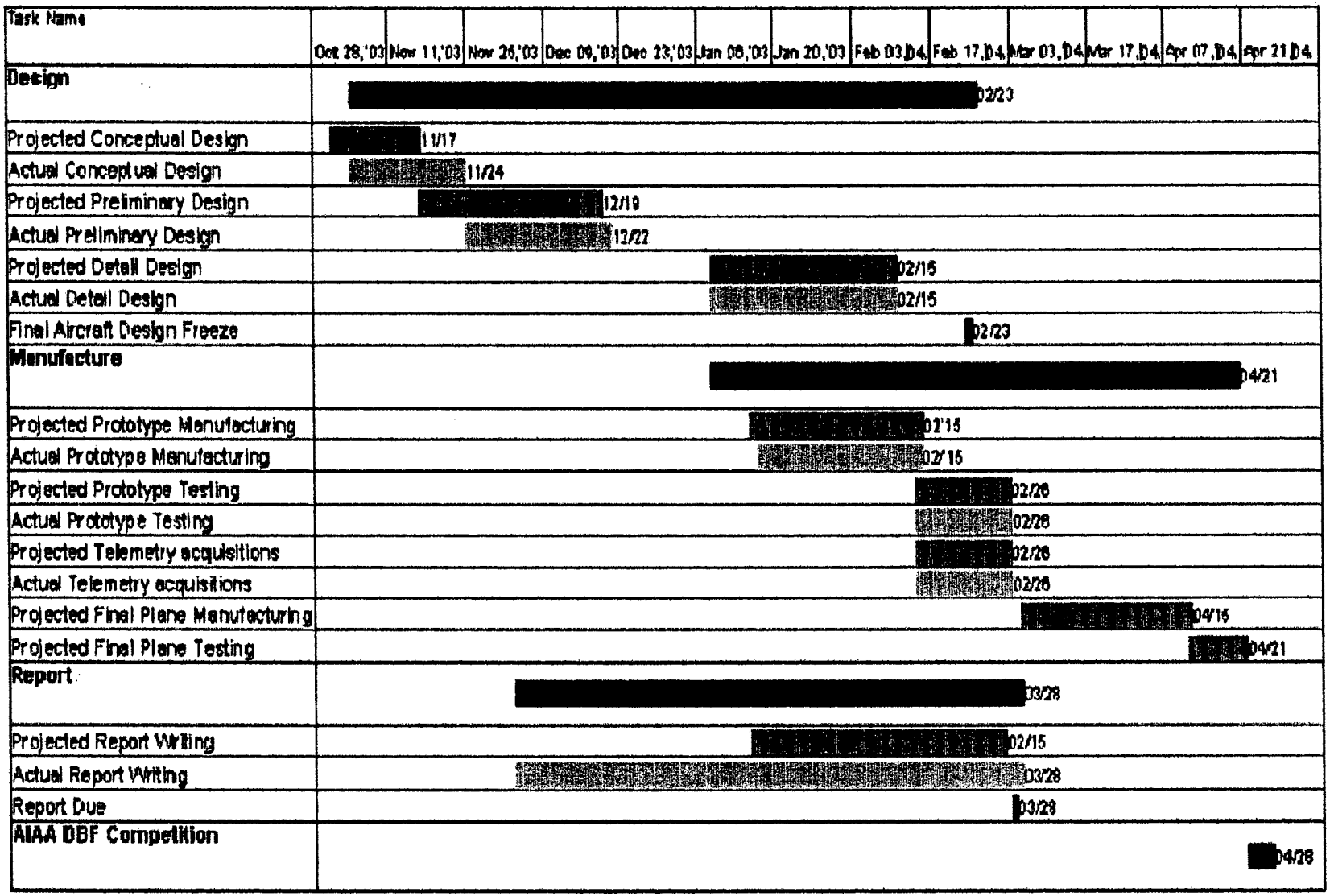


Figure 2: Milestone Chart

2.3 Scheduling, Document, and Configuration Control

In order to meet the competition deadline, the design process was performed through strict milestones, just identified, threatening to maintain correction margins for the structures and the systems eventually to overwork during tuning phase in the second macro phase. The design process was divided into the conceptual design, preliminary design, detail design and manufacturing phases. Dates were set for the completion of the ingredients for each of these phases. During design and manufacturing process several minutes were redacted at all steps: they were determinant for the realization of this report. Figure 2 is a milestone chart developed for the design process

3.0 Conceptual design

Above all during concept selection process there were performed sensitivity studies on the competition in the areas of aerodynamics, flight mechanics, aircraft systems, propulsion and structures. Then ideas for accomplishing all aspects of the competition were generated. Subsequently each concept was discussed to determine the benefits and penalties of implementation. Ideas were combined and evolved until a final concept was reached. The final configuration chosen was best suited for the contest rules and mission, allowing for the most scoring potential: moreover this one resulted compatible with our construction skills and general costs. Mission requirements were: the aircraft must fit in a 2 foot wide by 1 foot high by 4 foot long box; maximum take-off distance must be 150 ft. Maximum mission time is 10 minutes.

Mission requirements:

The 2 possible competition missions were named "Fire Fight" and " Ferry". Each mission flown will be scored using a degree of difficulty factor divided by the mission flight time. The required mission tasks for each of these 2 missions are described below.

- Aircraft storage

  It was required that the aircraft be disassembled and fit into a 4 ft x 2 ft x 1 ft box. All electrical connections should be keyed so they cannot be misassembled.

- Takeoff:

  For all missions there is a maximum take off distance requirement of 150 ft. The wheels must be off the runway within this distance. If the aircraft is unable to meet this constraint, a 0 score will result for that sortie .

- Fire Fight:

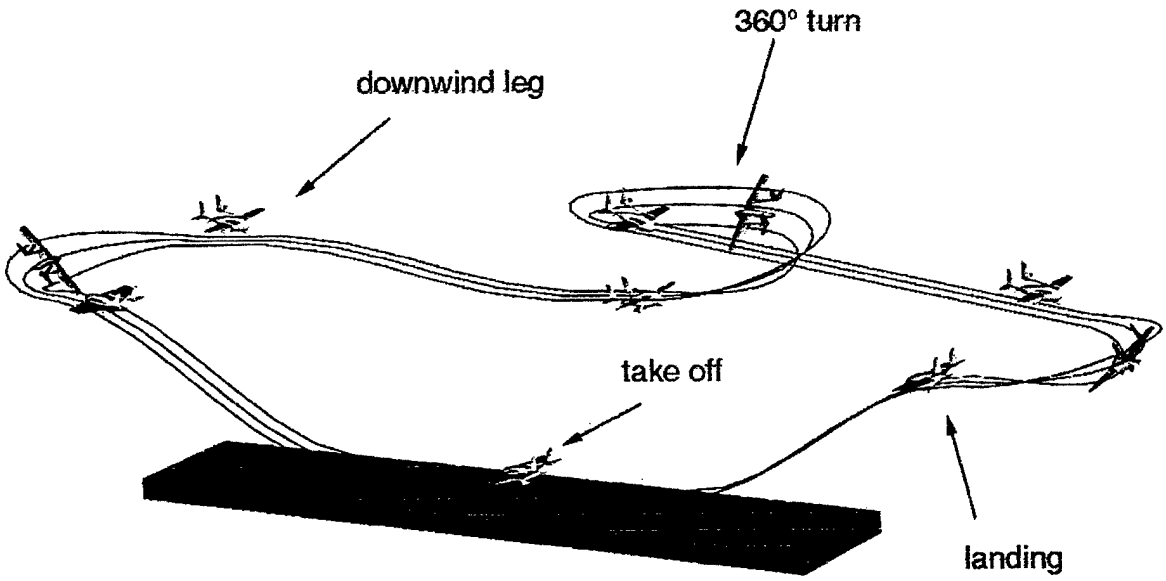
  For this mission the aircraft must be loaded, take off, dump its water load during the downwind leg and 360 degree turn, and return to land. This circuit it has to be repeated twice. Maximum allowed aircraft water capacity is 4 litres and the loading must be effected through four 2 liter plastic "soda" bottles that cannot be pressurized. This mission was given a difficulty factor of 2.0.

- Ferry:

  For this mission the aircraft must complete four laps with 360 degree turn, without loading the water.

  Aircraft cost-effectiveness:

  The final competition score will depend on a Rated Aircraft Cost (RAC), which is a function of the complexity and technology of the design. It is essential that the RAC be as low as possible, since the competition score is inversely proportional to RAC.



3.1 Design Parameters

A performance program (excel sheet) was written to investigate the design parameters of the plane over the entire mission. The program was used to perform sensitivity to each design parameter was known, an overall strategy was developed and implemented during the three design phases.

3.1.1 Primary Design Parameters

The performance program considered all phases of competition flight and RAC. Six primary parameters were chosen as major contributing factors in the development of the plane: weight of batteries, weight of water tank, power used in cruise, power used in takeoff, plan form area of the wing and span of the wing; these ones were weighted by a global performance factor obtained combining single performances of each component.

The six factors were used to maximize the scoring potential of the design. The program found that the highest scoring plane configurations carried the limit of 4 litres of weight for the water tank. Takeoff distance was limited to 137.6 ft. The full 150 ft feet was also used by every high scoring configuration . Another limiting factor was the number of batteries. Each high scoring configuration used 96% to 100% of the energy capacity of the batteries. These tendencies confirmed the plane needed to fly efficiently in cruise, while still taking off under the limit with a payload of the full water tank. Figure 3 summarizes the sensitivity analysis results. Takeoff power was found to be between 400 and 530 watts, while cruise power was between 270 and 310 watts. The wing plan form ranged from 8.5 and 9 square feet with a span of 85.2 to 89.6 in. The above parameters were given to the propulsion and structures group so that refined analysis could be made during the conceptual design phase.

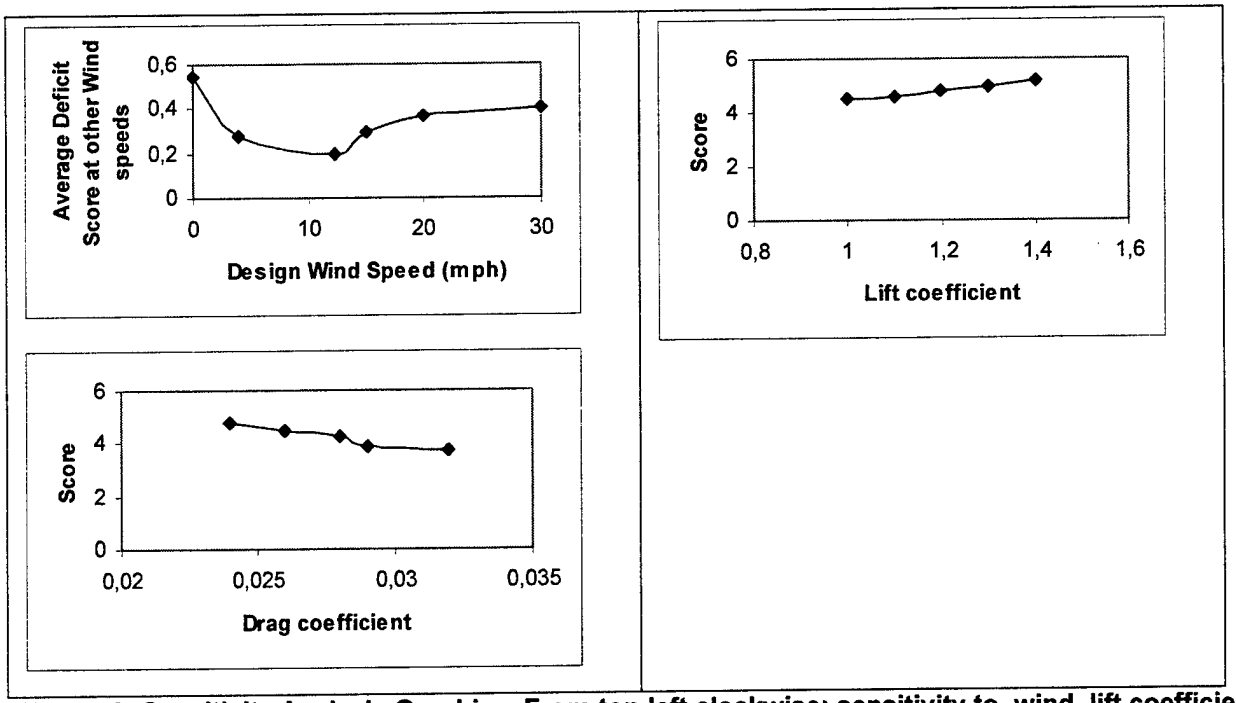
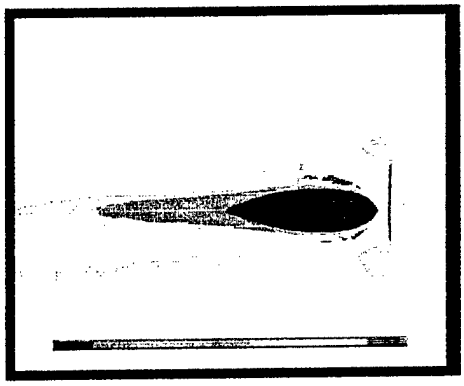


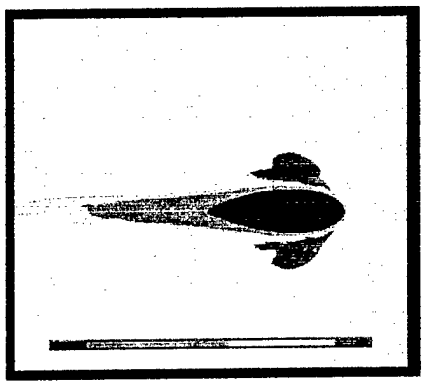
Figure 3: Sensitivity Analysis Graphics. From top left clockwise: sensitivity to wind, lift coefficient and drag coefficient

3.1.2 Design Parameter Sensitivity Analysis

Sensitivity checks were made regarding other major parameters. The parameters consisted of the following: weight model of plane, lift and drag coefficients, aerodynamic effects of propeller and fuselage length. The parameters were fixed within the program and therefore had to be used with care to avoid to affect the outcome with macroscopic errors. The sensitive of the major parameters helped to optimise the plane and determine the desired values of the parameters within the program.

- Weight model of the plane: weight models for the plane were developed from historical data. This models included different combinations of kind of woods and amounts of glue and rivets. Also polystyrene were considered for its quality of resistance and weight and the combination of wood and polystyrene use had been decided. About materials, besides the various more expensive alternatives surely in terms of price and time of realization, this is based on the bright idea of the mole maker French C.P.Leroux. The idea foresees the realization of a strong body thanks to the polystyrene covered again with vinyl glue and paper from packages. Not to underestimate the facility with which shape is possible the initial block. With the same procedure wings and plans of tail have been realized while for the boom you has been realized in balsa. The weight models, when analyzed through the optimization, showed that the performance of the plane was very sensitive to the overall structural weight. Definitive selection couldn't be done due to realization definition problems, but, at least, the weight model sensitivity did show that construction material selection would be very important to the final design.





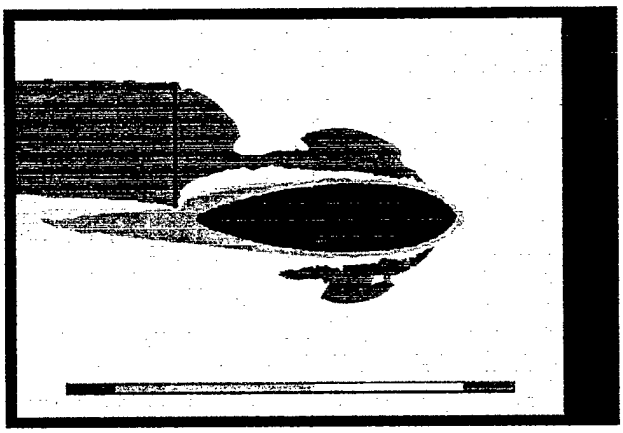


Figure 4: in clockwise direction: Velocity Field around Fuselage with fore propeller, no propeller and aft propeller.

- Lift and drag coefficients: analysis was accomplished by comparing the three optimal scoring capabilities of several lift and drag coefficients. Fewer batteries were required for higher coefficients of lift and lower coefficients of drag. The better performance prompted an airfoil study for maximum lift coefficients ranging from 1.1 to 1.5, and from 1.5 to 1.8 with flap and slat full extended.
- Aerodynamic effect of propeller: propeller position influenced heavily drag as you can see in CFD studies of figure 4. We studied different propeller position and we decided to put it in the rear side of the fuselage for the aerodynamic interference with the fuselage and to avoid the propeller damage too.
- Fuselage length: it was investigated for sensitivity by dividing the length into spaces needed to carry the payload and additional length for the nose and tail structures. A constraint to this analysis was imposed by transportation tank limited length. The results concluded that the fuselage needed to be slightly long than transportation tank. The total length was analysed using a combined aerodynamic and structural model.
- Wind sensitivity: it was investigated so that the performance could be optimised in preliminary design. To accomplish this goal, the six major parameters were optimised at various wind speeds and compared to other winds according to our historical data. Figure 3 shows an average score deficit for wind speeds from 0 to 30 mph. The plane optimised at 15 mph resulted to have the smallest average deficit, and 9mph with flap and slat full extended.
- Time on ground: this was the final sensitivity analysis. Resulted showed that time was a major factor and that decreasing ground time would drastically boost the score in terms of time of flight.

3.1.3 Rated Aircraft Cost Analysis

Another concern was the impact that Rated Aircraft Cost (RAC) had on the design. To evaluate the RAC, a computer program was written to calculate the effects of the RAC parameters from a few configuration

inputs. One of the major factor limiting configurations through the RAC was the total battery weight: this one was counted in both the Rated Engine Power and in the Manufactures Empty Weight. Additionally, the Rated Engine Power Multiplier was the largest of the multiplier values. All three factors combined to give the weight of the batteries the greatest influenced over the design; in practice it was necessary to develop a high performing aircraft from an aerodynamic point of view in order to minimize drag that is energy dissipation translating in necessary number of batteries .Other important factors were the wingspan, empty weight of the aircraft and body length. To give an idea of the relative importance of these three components, the battery weight had a greater impact on RAC than all three of the other factors combined. Other factors, while not directly having a large effect on the RAC, greatly increased the importance of certain components. For example, the number of engines did not significantly increase the Manufacturing Man-Hours; however, when the power for the extra motors and the Rated Engine Power were considered, the impact of adding an additional motor became large. Number of servos and mobile surfaces was at least determining for the final configuration of the plane (for this reason rotating wings had been adopted). Using the RAC program, the rating of each configuration could be immediately evaluated.

3.2 Aircraft Configuration

Once the primary design parameters were identified and investigated, the various aircraft component alternatives were evaluated. For each major component, the Figures of Merit (FOM) were outlined and used to judge the various alternatives through a weighted decision matrix. The matrices determined the alternative that best meet the Figure of Merit.

3.2.1 Alternative Fuselage and Tail Configuration

Figures of Merit (FOM) in the analysis of the fuselage and tail configurations included:

- Rated Aircraft Cost: since RAC strongly affects the overall score of the aircraft, RAC was heavily weighted.
- Take off and landing: from the design parameter studies, the take off lift coefficient and take off roll were key to producing a high scoring airplane.
- Handling qualities: handling qualities in terms of smoothness of water dumping.
- Drag performance: with battery weight counting twice in the RAC, a successful design must make efficient use of the available power and complete the mission quickly.

Once the design parameters and figures of merit for the fuselage and tail were defined, numerous configurations and combinations of configurations were considered. After the concept generation phase, the most promising alternatives were evaluated further. For further analysis, several assumptions were made including: the fuselage could be sized to hold the water tank; the propulsive efficiencies were approximately equivalent while the difference in structural weight was negligible. The parameters could be

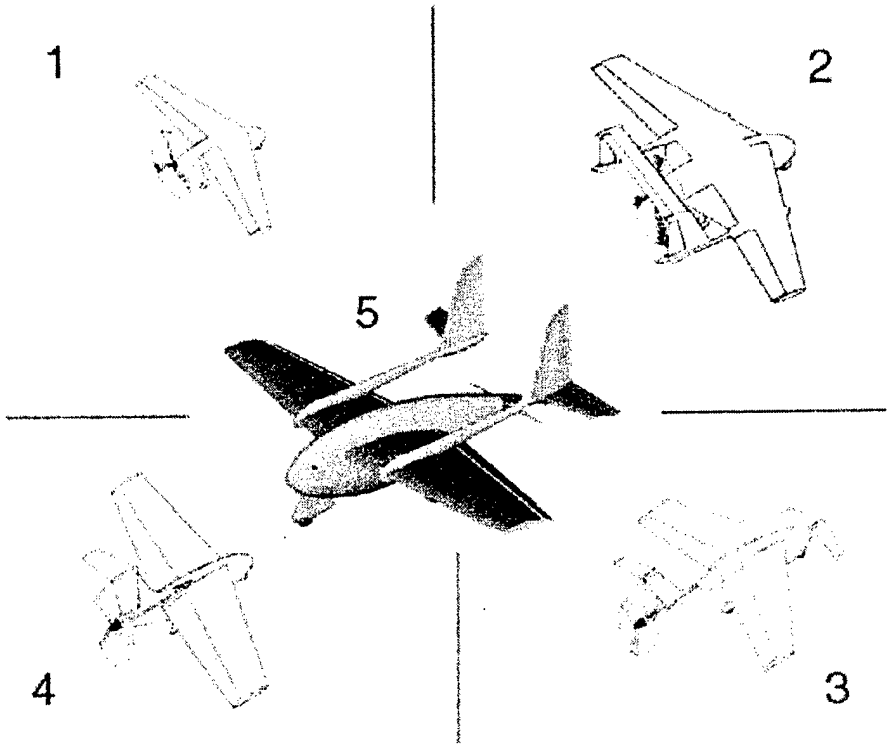
considered constant during conceptual configuration. Given the assumption and the figures of merit above, the alternatives were evaluated as follows:

1. Flying wing: the flying wing had an obvious advantage in the RAC by removing the tail and relative control surfaces. The flying wing had some problems, namely extended take off roll and poor handling qualities in gusty winds due to a lower wing loading.
2. Conventional A: the conventional fuselage with a double T-tail was set as standard for comparing the various configurations. For A version we intend a plane with low wing and cross tail.
3. Canard: the canard configuration had excellent stall characteristics; however, by preventing the main wing from stalling, this design limits the maximum lift at take off. The canard configuration had the added benefit of flexible motor set-ups, both tractor and pusher.
4. Conventional, T tail: a conventional fuselage with a T-tail had the same performance as the conventional tail alternative, but had the RAC advantage previously mentioned
5. Conventional B: for B version we intend a plane with high wing and a double cross-tail: this design was relatively straightforward with well-documented performance characteristics.

In order to evaluate the Rated Aircraft Cost figure of merit for the various configurations, the RAC of each alternative was calculated. The results of the analysis are listed below with configuration, RAC value and then the RAC benefits of that configuration:

1. Flying wing (RAC of 8.3) does not have a horizontal tail surface or elevator servos
2. Conventional fuselages (RAC= 8.2) doesn't present any difference between A and B version.
3. Canard (RAC of 7.8) is slightly penalizing in terms of RAC
4. Conventional fuselage with T-tail (RAC 8) has passive vertical surface instead of active

A weighted decision matrix was used to evaluate the alternatives according to the figures of merit to determine the configuration with the highest scoring potential. Weighting factors were assigned with different magnitudes according to the defined mission sensitivities. While RAC is the dominating factor, it is not the only consideration. For instance, while the flying wing has the highest RAC, the poor performance of the design with regard to the other figures of merit limited its overall scoring potential. The results of the decision matrix can be seen in figure 5. This arrangement best met the figures of merit because of an improved RAC advantage and average performance qualities. Due to short distance between wing and tail, a Double Mobile Independent Tail (DMIT) configuration was preferred.



Figures of merit	Weigthing factor	Model 1	Model 2or5	Model 3	Model 4
RAC	0.4	8.3	8.2	7.8	8
Drag	0.15	2	1	2	4
Construction	0.25	6	8	7	6
Handling qualities	0.2	1	5	4	2
Score	-	5.32	6.43	5.97	5.7

Figure 5: Fuselage and Tail Configuration Weighted Decision Matrix

Wing fuselage connection: as regards wing-fuselage connection we considered the following idea. We decided to adopt a mid wing, so we had to decide how realize this one avoiding to improve drag and to obtain an advantage in the RAC score. Generally mid wing aircraft are realized with a wing, with moving parts(as ailerons) which is also a part fuselage: this way is the better to preserve stability but it isn't the most useful in terms of RAC score. We thought to proceed with a all moving wing "Rotating Wing", in this way we obtain the number of servos reduction.

The rotation of the wing is permitted by the pitch control that being independent for every wing and makes the two wing independent.

We have installed flap with gap of 0.1 c and slat both to manual driving. Of course we were more concentrated to drag reduction than stability and to obtain this goal we have done so that the fuselage

remains with zero angle of incidence and this guarantees a best efficiency of the propeller and to preserve it from the bumps in phase of takeoff and landing.

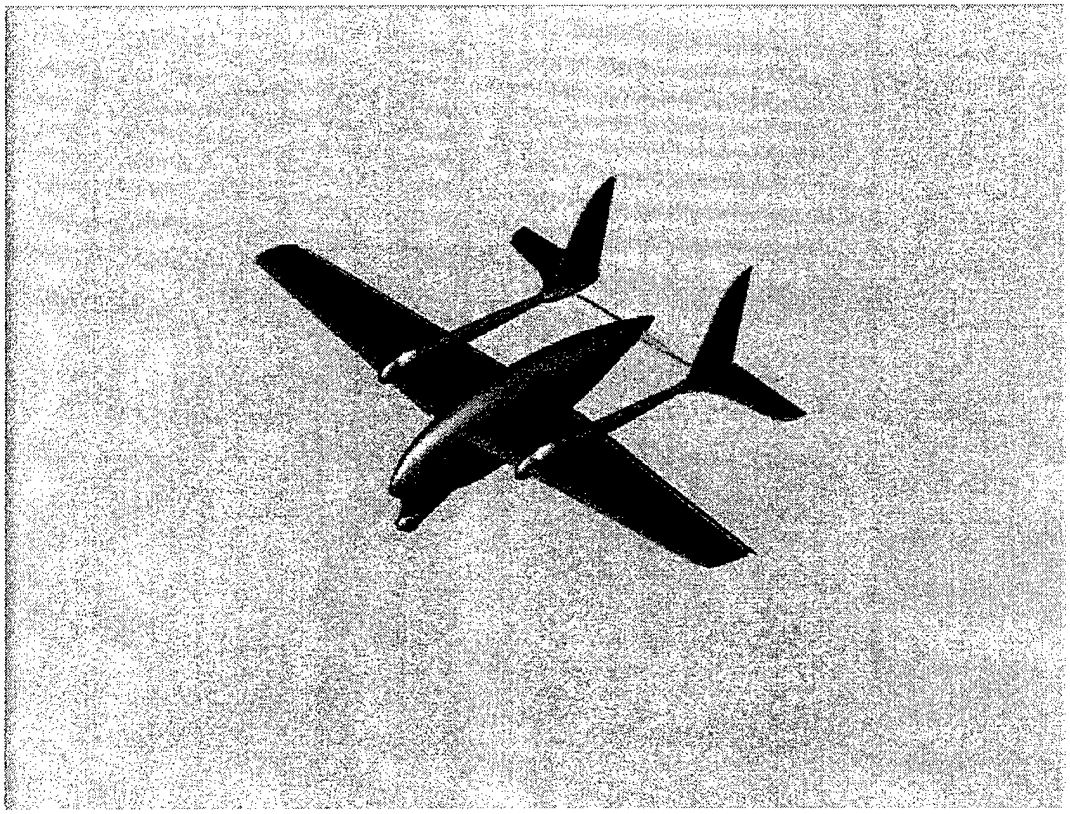


Figure 6: A first rendering of the final project M9xx

3.2.2 Alternative Wing Configuration

Figures of merit in the analysis of the wing configurations included

- Construction: the alternative selected must be as light as possible and, meantime, as resistant as possible. Moreover it must be within the team's construction abilities in order to be manufactured correctly and without spending too time.
- Stability: the stability contributions of the wing location were estimated from historical data. Mid-high (our solution) mounted wings typically provide roll stability, while low mounted wings can produce instability.

The three basic alternatives for wing location were considered: low, high and sunshade wing attachment. For analysis the following assumptions were made: the taper ratio was one for better stall characteristics and ease of construction; the wing has not sweep angle.

- High wing: the high wing did not significantly interfere with water tank expelling, produced roll stability without adding dihedral into the wings. This configuration has not been applied because of the troubles of the airflow with the fuselage.
- Low wing: the low wing alternative has less roll stability, which might require dihedral. Moreover it presented the same problems of mid wing, in terms of water tank handling. This configuration has not been applied for the same motives written up.

Figure of merit	Weighting factor	High	MID	LOW
Stability	0.075	1	0	-1
Payload expulsion	0.35	1	0	0
Construction	0.15	0	-1	0
Payload interference	0.35	1	-1	1
Gear Interface	0.075	0	0	1
Score		0.775	-0.5	0.35

Table 1 : Wing Configuration Weighted Decision Matrix.

The weighting factors of the wing decision matrix in table 1 shows the relative importance of each figure of merit in the wing selection. While the payload and construction played a large part in the decision matrix, the stability and landing gear issues were secondary effects. Since the high mounted wing better addressed more of the important figures of merit than the other alternatives, the high wing rated the highest and was selected as the wing configuration for the aircraft.

3.3 Structural Configuration

3.3.1 Figures of Merit

A topic area of the structural conceptual design was the development of figures of merit to represent mission objectives for each component to perform. The mission objectives chosen were designed to represent crucial areas within the competition that were affected by different structural components.

The figures of merit which were developed for the conceptual structural design were:

- Scoring Potential: each aircraft component had a direct effect on the ability the design to achieve a high score. Since the size, shape, and capability of each component can affect both aircraft performance and Rated Aircraft Cost, the design of each component is important to perform the mission in the most efficient way. In order to maximize scoring potential this figure of merit is the overall result of all figures of merit analysed.
- Strength and Weight: the weight of each component influences both Rated Aircraft Cost and aircraft performance in every phase of flight. Therefore the proper balance between the strength of a component and the weight of the component was necessary to determine.
- Formability: the ability to produce the desired component shape with the least amount of man-hours and components has an affect on the quality of the component, the overall weight of the structure, and the Rated Aircraft Cost
- Ease of construction: certain construction techniques required more time and skill to produce complex shapes such as high camber airfoils and streamlined bodies or smooth nose. The ability to produce such shapes directly affected the capabilities of the aircraft in all phases of flight
- Reparability: during the flight-testing phase and the competition, minor to moderate damage was expected. On site repair of the damage was desired to reduce amount of time.
- Cost: while a score efficient structure was desired, the cost of certain construction methods and materials have a drastic affect on the validity of some concepts. However this figure of merit had been attenuated by sponsors.
- Durability: during flight operations, the aircraft was expected to experience hard landings. In order to be competitive, the aircraft had to withstand the conditions and operations of the competition through many cycles.

While the figures of merit were valid for all structures, several specialized figures of merit were developed for individual components. These figures included:

- Taxing: the landing gear must be capable of precise aircraft manoeuvring while the aircraft is on the ground. The mission advantage was to provide quick turnaround times between different phases of the mission sortie.
- Landing Performance: also contributing to the aircraft score was time spent in the landing phase of the sortie.
- The ability for the landing gear to both endure hard landings and damp out the landing force with out springing the aircraft back into the air was important to achieving a time efficient landing manoeuvre.

3.3.2 Assumptions Made and Design Parameters Investigated

During the conceptual design phase, assumptions were made to allow an unbiased look at different alternatives. For the four primary components analysed, loading conditions were assumed based on simple models produced from available material samples, such as were those of the weight.

Parameters that had the most influence on score were identified for each primary component. These influenced not only the structural design but also the conceptual studies of the aerodynamics and propulsion groups. The parameters investigated for each structural component included scoring potential, efficient use of structural weight, strength, dimensional alternatives of the component, different material and construction alternatives, and effects on Rated Aircraft Cost.

3.3.3 Alternative Wing Structures

The figures of merit used to investigate wing alternatives were strength to weight ratio, formability, ease of construction, durability, reparability, and cost. Additional constraints were imposed on the wing design as a result of the functions of the wing: to generate lift and transfer that lifting force to the rest of the aircraft. To create lift, the wing had to be capable of holding an airfoil shape. For the high aspect ratio indicated by the mathematical model, the wing was required to be rigid enough to prevent a decrease in the angle of attack at the wing tips due to the aerodynamic loads. Due to initial estimates, the wing design was expected have an area of approximately 1462.43 in^2 and to experience a 3 g-load during a turn at maximum gross weight. Both the functions of the wing must be met on a weight budget for the wing to be efficient. Material alternatives and construction methods were considered. The three types were conventional build-up of lightweight wood skeleton, conventional build-up of carbon fibre skeleton and skin (there were the possibility of employing stitched composites at our university), and composite skin with polystyrene core. However even if we were conscious of the advantages related to the use of composite materials we remained focused on the original idea.

In addition to the structural build-up of the wing, conceptual alternatives for internal system components of the wing were generated. These included servos, wire and control cables routing, and how the wing will mate and transfer loads to the rest of the fuselage. A final conceptual sketch of the wing structure can be seen below.

Figures of Merit	Weighting Factor	Conventional Buildup with Wood	Conventional Buildup with Carbon Fiber	Composite Buildup with Foam Core
Strength to Weight Ratio	0.3	-1	1	0
Formability	0.25	-1	0	1
Ease of Construction	0.15	0	0	1
Durability	0.15	0	1	1
Repairability	0.1	-1	0	0
Cost	0.05	1	-1	-1
Score	I-	-0.6	0.4	0.5

Table 2: Wing and Tail Structural Decision Matrix

3.3.4 Alternative T-tail Structures

Similar considerations as those used on the wing were made for the tail. The figures of merit used to investigate wing alternatives were strength to weight ratio, ease of construction, durability, reparability and cost. The three types of tail construction alternatives were conventional build up of lightweight wood skeleton, conventional build up of carbon fibre skeleton and skin and composite skin with polystyrene core. The function of horizontal tail was to apply longitudinal stabilizing moment to provide control in aircraft pith; the function of vertical tail was to apply laterally stabilizing moments to provide control in aircraft yaw. The tail unit needed to be lightweight, durable and able to adjust the incidence angle.

3.3.5 Alternative Fuselage Structures

The fuselage concept was developed as a combination of requirements. Figures of merit involved were: easy of construction, strength, formability, durability, reparability, and cost. The fuselage must be capable of handling the optimum amount of payload with the lowest drag possible while carrying all radio equipment, batteries, and the propulsion system. The fuselage also serves as the convergence point of all other aircraft components supporting the weight of the entire plane and transferring moments and loads between components. Several different construction techniques were evaluated, including conventional

build-up with wood, conventional build up with carbon fibre, solid polystyrene construction, and monocoque. Solid polystyrene construction; it consisted of a solid fuselage of polystyrene covered again with vinyl glue and paper from packages.with epoxy hollowed out for payload and equipment.

Figures of Merit	Factor	Conventional Buildup with Wood	Conventional Buildup with Carbon Fiber	Solid Foam Buildup	Monocoque
Strength to Weight Ratio	0.3	1	0	-1	0
Formability	0.25	0	1	-1	0
Ease of Construction	0.25	1	-1	1	-1
Durability	0.05	0	-1	1	1
Repairability	0.1	1	0	-1	1
Cost	0.05	1	-1	-1	0
Score	-	0.75	-0.15	-0.4	-0.1

Table 3: Fuselage Structural Concept Decision Matrix

Other techniques being evaluated were:

- Conventional build-up with carbon fibre; it was built up of the same frame work as conventional build up with wood however the use of carbon fibre skin allows loads to be transferred throughout the skin
- Conventional build-up with wood had been evaluated first: it consisted of frames and stringers to carry loads and maintain shapes. The framework was covered with a cover that had no load bearing capability. The method featured very lightweight construction but had the potential to lose the advantage if the structure was required to be very rigid.

- Monocoque fuselage; it consisted of a fully load bearing outer shell made of composite material built up on foam molds. Monocoque provided a very lightweight rigid structure that can be formed into almost any shape.

3.3.6 Alternative Landing Gear Configurations

The gear concept also was developed to satisfy mission requirements. Figures of merit involved: dimensions, weight, aerodynamic drag, ease of construction, rotation attitude and landing performance.

Various configurations were available but we finally chose the low-tricycle configuration since this was the most robust and permitted the transport of the fuselage in the fixed-dimension box without disassembling, with all the advantages in terms of time. Once the landing gear configuration was determined, configurations of the main gear were considered. The primary function of the main gear was to absorb energy during landing; it had to be lightweight, durable, and have a low drag. Figures of merit involved: dimensions, weight, aerodynamic drag, ease of construction, landing performance, and damping effects.

Figures of Merit	Weighting Factor	Tricycle	Quad	Bicycle	Tail Wheel
Ground Tracking	0.20	1	-1	-1	-1
Landing Performance	0.15	0	1	0	-1
Chaulked Fuselage Attitude	0.15	1	1	-1	-1
Braking	0.15	0	1	1	-1
Weight	0.10	0	-1	1	0
Ease of Construction	0.07	0	-1	-1	0
Drag	0.06	0	-1	1	0
Component Mating	0.06	1	-1	-1	1
Cost	0.06	0	0	0	0
Score	-	0.41	-0.04	-0.17	-0.59

Table 4: Landing Gear Concept Decision Matrix

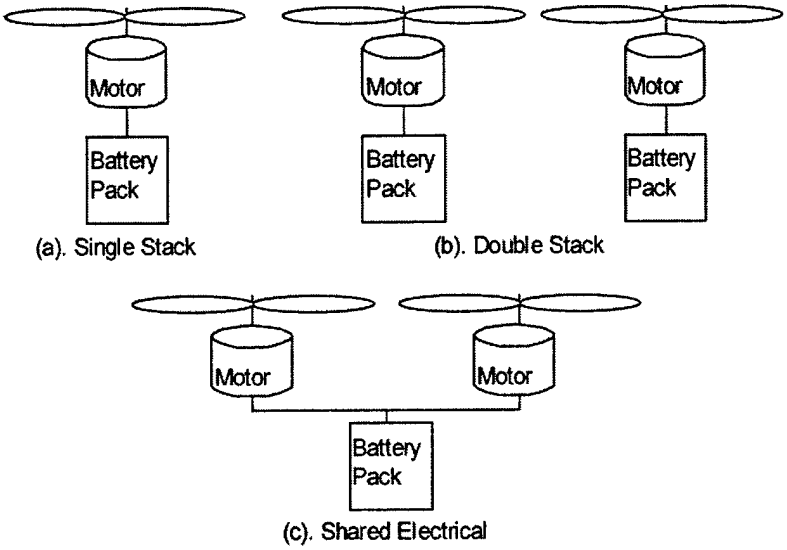
Figures of Merit	Weighting Factor	Bow	Aluminium legs	Leaf Spring
Landing Performance	0.25	0	-1	0
Weight	0.25	0	-1	0
Realization	0.25	0	-1	0
Damping	0.15	0	0	1
Ease of Construction	0.07	1	-1	0
Drag	0.03	0	-1	0
Score	-	0.07	-0.35	0.15

Table 5: Main Landing Gear Conceptual Decision Matrix

3.4 Alternative Power Plant Configurations

The propulsion system had three main components: batteries, motor, and propeller. Three major configuration ideas were considered: single stack, double stack, and shared electrical. Figures of merit (FOM) were developed to evaluate the three configurations. These included:

- Rated Aircraft Cost (RAC): the propulsion system is very influential in the RAC function.
- Score Effects: the propulsion system had influence in every variable of the score function.
- Power Produced: the power produced by the motor has direct effect on aircraft performance. Power also requires battery performance, which affects the weight of the aircraft and RAC.
- Weight: the battery weight was determined to consume the majority of the overall RAC.
- Efficiency: the ability to produce the required performance with the lowest number of batteries was important in both aircraft flight performance and Rated Aircraft Cost.
- Price: although the highest performance possible was important, the cost of components limited the level of components used.



Figures of Merit	Weighting Factor	Single Stack	Double Stack	Shared Electric
RAC	0.35		-1	-1
Power Produced	0.25		1	1
Weight	0.2		-1	0
Efficiency	0.15		-1	0
Total Number of Fuses	0.05		-1	1
Score	-		-0.5	-0.05

Figure 7: Battery-Motor-Propeller Alternatives and Propulsion Weighted Decision Matrix

The figure above shows the propulsion alternatives and decision matrix used to evaluate the different propulsion alternatives. From the decision matrix, the single stack was found to be the best option. The single stack had the lowest RAC and while meeting the required thrusts and speeds to achieve takeoff and climb. The single stack could provided enough power, thus eliminating the need for two motors. Anything that can be done to make the system more efficient was desirable. Another draw back to using two motors was the loss in efficiency. In the RAC equation, the battery weight was the most detrimental. In order to increase the efficiency of the engine, several cooling concepts were considered. The concepts investigated included air-cooled, water-cooled, and ice-cooled systems. An air-cooled system was chosen as the best concept based on cost, weight, and complexity.

3.5 Analytical Tools

3.5.1 Optimisation Program Architecture

The performance program was written to help analyse each phase of the mission individually as a contribution to the overall score. The program contained mathematical models for the structural weight, propulsive efficiency, and aerodynamic characteristics. The models were written into the program so that refinement and expansion could be done quickly and simply. The structural weight and propulsive models were constructed from our historical data, as calculated from component build-up. The aerodynamic model was constructed using a simple drag build-up method suggested by Jensen (1990) and the coefficient of lift corrected for aspect ratio of the wing.

After the flight characteristics of the plane were calculated from the four models, the performance characteristics for each phase were calculated:

- The time and energy consumption required to takeoff under the 150 feet limit.
- Time, energy consumption, and power required to climb at a vertical speed of 2.5 fps.
- Time, distance and g-loading during turning were calculated from weight and stall characteristics.
- Time and distance to slow down from cruise velocity to stopping. Time in cruise given the selected power settings.

Finally, the times, distances, and energy consumption were summed and compared. The overall values of the three parameters were used to calculate total flight time, flight path distance, and total battery usage. Rated Aircraft Cost was further calculated, and the Flight Score was calculated using both flight time and RAC.

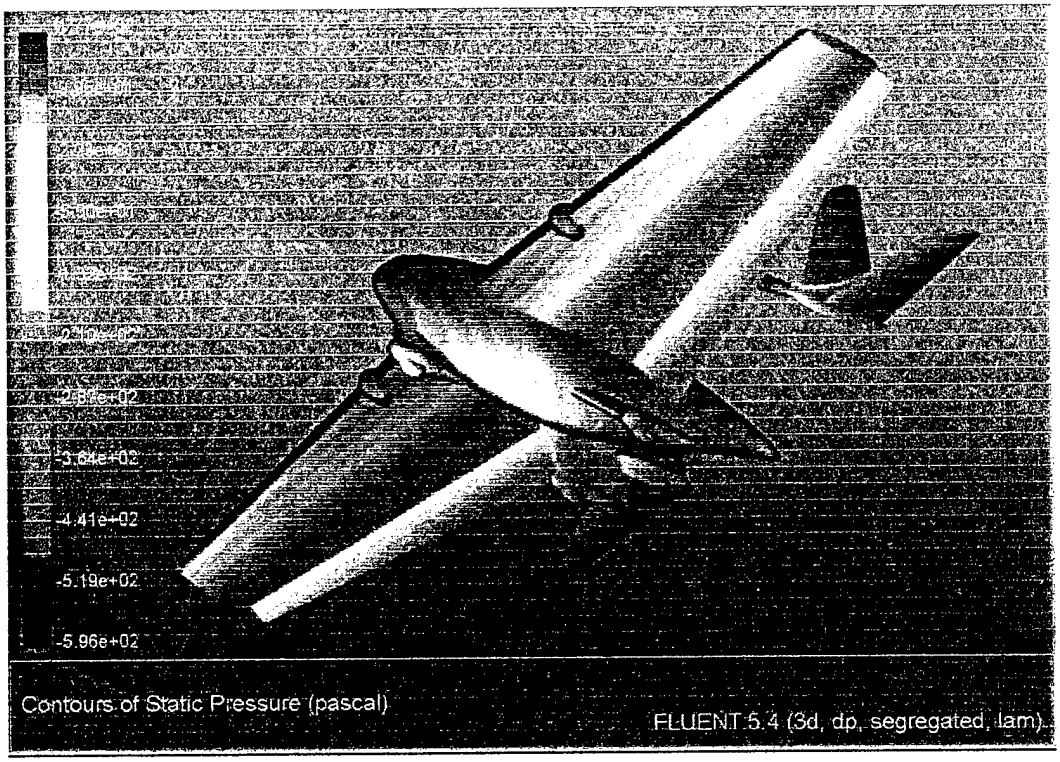
3.5.2 Other Methods Used

During the conceptual design phase, the various groups used weighted decision matrices to decide upon important decisions. Decision matrices were used so that the opinions of group members did not bias the decision, but rather the best configuration was chosen in a more numerical form. The graph of weight and volume effects along with RAC calculator were used to access parts of the factors input into the decision matrices.

3.6 Final Aircraft Configuration

When the final ranking phase was completed the design that stood out on top was the "drop-type" fuselage with "sunshade" wings and pushing propeller shown below. The configuration had good handling qualities and a reasonable take-off distance, one that fell within the prescribed 150 feet. A high mounted wing is a typical configuration of a cargo aircraft, the tail wing span fell within the 25% of the span of the greatest span horizontal surface while still providing adequate control of the plane. The propulsion system was composed of a single propeller tractor set-up. The landing gear (included the

nose) was constructed of wood. The final configuration was the best compromise between all of the ranking factors.



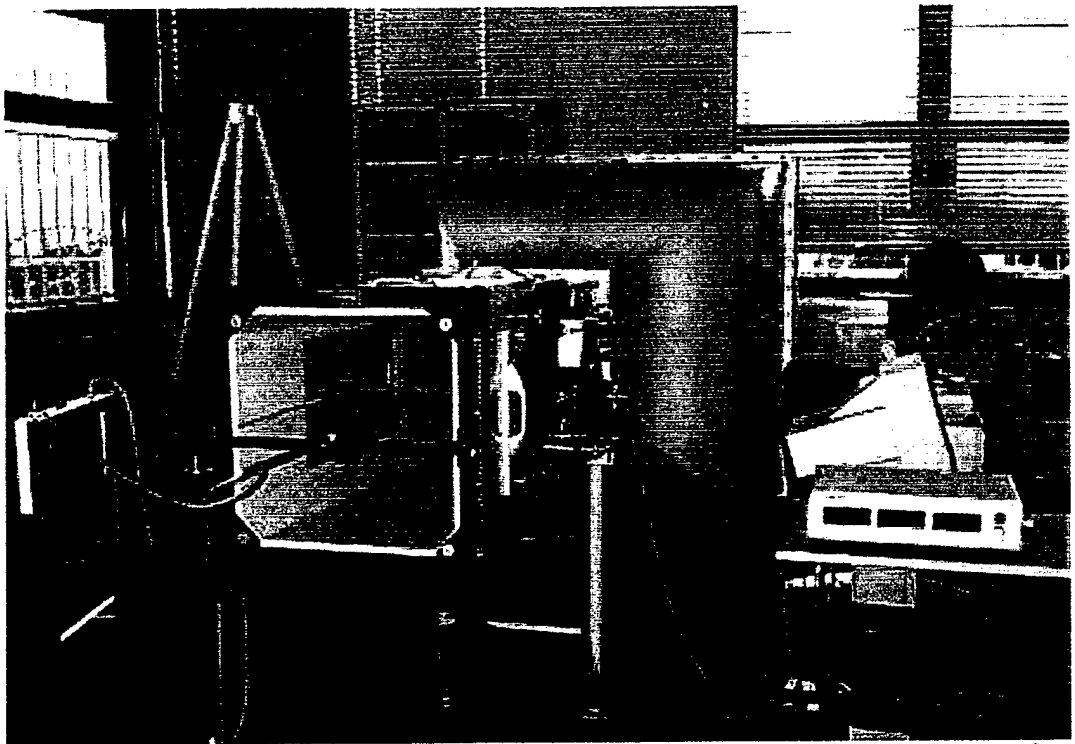


Figure 8,9: Fluent analysis & wind tunnel (Plint & Partners TE44) with scale balsa wood model

For the aerodynamics coefficients evaluation two tests have been executed; one by computer numerical analysis using a numerical fluid dynamics program (Fluent) and a second one by the realization of a scale model (1:4.44) to use in the wind tunnel. Half model installed in the test section of the Plint & Partners TE44 wind tunnel at I.T.Aer. "F. De Pinedo" in Rome. The results of this two tests have been different: the wind tunnel outputs are 30% less in terms of resistance in opposition to the computer analysis. This is probably due to an excessive mesh dimension. This parameter is actually limited by the calculus power of our calculators and cannot be reduced (increase of computational load). The model construction is based on balsa wood lacquered to obtain a near-ideal surface. To test the capability to avoid low Reynolds airfoil-stall using a turbulence inductor, a sand paper stripe has been used (on the leading edge) to obtain a simple and low cost material.

4.0 Preliminary Design

Once conceptual design phase was over, each group began preliminary design phase. The optimisation program was refined with preliminary models and used to find parameters to begin this phase: the last one were employed as in wing and fuselage aerodynamic and propulsive plant sizing.
Then dimensions and tolerances for components were passed on the structures group so that trade studies and structural analysis could be performed on primary structures. At this point propulsion group was able to identify some possible combinations for testing in the detail design phase.

4.1 Initial Trade Study Results

After the conceptual phase was ended, the weight, drag, propulsive and RAC models in the optimisation process were updated. The previous models were considered only as conceptual simulations because the previous versions were developed from the La Sapienza Aerospace Department, NACA and Sponsor historical database. The new program optimised features of the plane and helped to develop trade studies for each parameter in relationship to final score. The optimisation program was only able to output the local maximums for plane performance; the global maximum was solved by graphical means.

4.1.1 Study of Model and Number of Batteries

As regards model, batteries were selected evaluating energetic characteristics versus weight: by the way it was redacted a classification involving batteries with not more then 60mOhm internal resistance (that is able to furnish high currents during discharge phase without overheating). So it was possible to obtain reliable batteries with the best energetic performances affecting as less as possible RAC in terms of batteries-pack weight. As it results from the reading of the following matrix, the best option is represented from Sanyo RC2400.

CAPACITY WEIGHT	vs. CELL NAME	CAPACITY	INERTANCE	WEIGHT	VOLTAGE
40.7	Sanyo RC2400	2400	0.0032	59	1.2
39.6	Sanyo 2300SCE	2300	0.0055	58	1.2
39.6	SanyoCP-2400SCR	2300	0.0053	58	1.2
38.3	Sanyo 1800SCE	1800	0.0065	47	1.2
38.0	SR 2000 MAX	2000	0.0035	53	1.2
35.8	Sanyo 3000CR	3000	0.0032	84	1.2

Table 6: Batteries Rank

As regards the number of batteries, this one was assumed, in terms of energy requested, as a start point for the definition process of propeller and plan form. A desirable number of batteries was twelve but it was almost impossible to go under 18.

4.1.2 Study of Take Off Battery Performances

Batteries system resulted good with respect to competition constraint of 40 A while enables the aircraft to take off in 137.6 ft granted by 7.05 lb of fixed point thrust and 4.188 lb at 39.37 ft/sec.

4.1.3 Study of Cruise Battery Power Requirements

Similar to take off power, the battery power required during cruise was also examined. The score comparison has maximums ranging from 400 to 530 watts and an optimal value of 443 watts. Again, the broad range gives a margin of safety for the propulsion group to work within. moreover it is possible to maintain a 1.54 lb of thrust at a cruise speed of 60.14 ft/s during 7.56 minutes (time equivalent to a flight of 29002.62 ft , i.e. a distance double longer than that of the competition).

4.1.4 Study of Wing Plan form Area Requirements

The wing plan form area was shown to have a direct relationship to score. The relationship existed on one side of the optimal value and dropped off on the other side; also in our opinion this is due to the added propulsive requirements to get the plane to take off under the limit of 150 feet (i.e. 137.6 ft): in

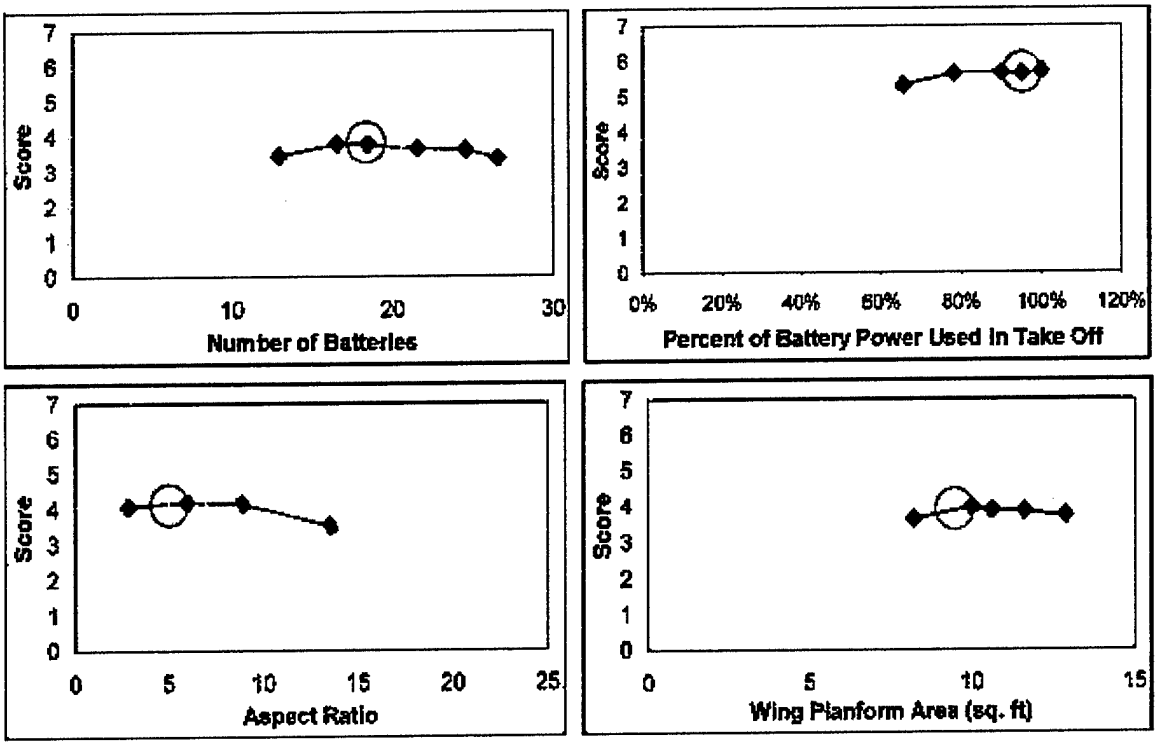


Figure 9: Trade-off Analysis Results. From top left clockwise: number of batteries required, take off power required as a percent of available power, wing plan form area requirement, aspect ratio requirement. Circles represent the optimal value found by the program

such a way we had to trade-off number of battery (to be minimized in terms of RAC and airplane weight) versus maximum take-off length. As regard the relationship above mentioned, up to the optimal, this one appeared to well represent the aircraft because of the cruise efficiency decreases as plan form increases. All of the above features of the linear trend can be seen in figure 9.

4.1.5 Study of Wing Span Requirements

Wingspan was shown to have a rectangular relationship with optimal score. Also in our opinion, rectangular scatter indicates that many different spans can be considered optimal. Really also in this step we traded off wingspan with chord length in order to complain with wing plan form area previously fixed. The maximum length of airplane box transport was determinant in the decision process as it was preliminary decided to minimize assembling time avoiding to realize multi-pieces wings. The final value determined was 85.8 in.

4.1.6 Study of Aspect Ratio Requirements

Due to the bent of the optimisation process to an efficient propulsive system, the optimal scores were graphed versus corresponding aspect ratios in order to illustrate the aerodynamic trend toward efficient planes In this way the optimal range of aspect ratios was from 5 to 6.

The value chosen was 5.22 (see fig 9).

4.1.7 Study of Plane Structural Weight Effects

Plane structural weight was heavily take in account in the overall performance of the plane during the conceptual phase. Also weight revealed a linear relationship with score. Optimal weight was determined 17.63 lbs .So, a decrease in the overall structural weight of the plane increased the overall performance and score of the plane.

4.1.8 Study of RAC effects

In the idea of minimizing RAC, a final study of the effects of RAC on performance and score was completed during the preliminary phase; so it was possible to trade-off minimization RAC with keeping the plane within good performance parameters. The optimal plane had a RAC value of 11.25 within a range of 11 to 12 for the top 5 configurations.

4.2 Aerodynamic Considerations

Using the design parameters determined during the preliminary phase, it was possible to perform a further analysis in order to refine the design. The areas examined included: wing area, tail sizing, fuselage , wing/fuselage connection, stability and control.

4.2.1 Wing Figures of Merit: Assumptions and Analysis

Preliminary considerations have to be done are the following.

Airfoil selection is related to the following parameters: projected air-speed, Reynolds numbers corresponding to maximum and minimum airspeeds, construction technique and relative effective feasibility, airfoil behaviour at angle of attack in proximity of stall, and drag developed in the range of possible angle of flight. It is clear, at this point the airfoil to be selected must develop a minimum drag if it is desired to construct an airplane adapt to rapid flight without enhancing energy consumption. In fact, if an airfoil was characterized by an high C_d, it would have been requested an installation of a motor with necessary and adequate power in order to obtain aimed performances with consequent weight and consumption problems . From an analytical point of view, there are other considerations to be done in order to perform the optimisation process.

Specifically it was necessary to define figures of merit that, for the wing analysis, included:

- Total Flight Score: since the score is used to rank the aircraft for the competition, total flight score was set as the primary figure of merit. The wing sizes that produced the greatest scoring potential ranked highest. The flight score included the RAC effects of the configuration.
- Performance: this one quantifies the effects not considered in the optimisation; namely, tip strike, strength, and deflection. The performance merits tended to shorten the wing to ensure that the wing could realistically achieve the characteristic calculated in the optimisation.

The wing was modelled in the optimisation program to provide estimates of the flight score based on the wing area and span required for the mission profile. On the highest scoring configurations, the chord varied from 15.63 to 19.3 inches while the wingspan ranged between 85.2 and 89.6 in. The optimisation also included other factors involved in the analysis of the wing dimensions, such as the takeoff power and cruise power. The power necessary for takeoff and cruise was related to the wing area through calculations of lift and thrust. Therefore, a study of the motor performance was necessary in the analysis of the wing dimensions. Another aspect of sizing the wing was deciding whether a greater chord length or a longer wingspan best achieved an increase in plan form area.

The wing span length was penalized heavily in the RAC calculations, while the chord was only a minor factor. Increasing the wing chord produced more wing area, but decreased the aspect ratio. Higher aspect ratios produced drag efficient lift production as predicted by the conceptual assumptions. The optimisation program altered the constraints to find the best scoring configuration.

The optimised wing was then slightly oversized to allow for both construction and aerodynamic margins of safety. The wingspan and chord were 87.4 in and 17.43 in, giving a wing plan form of 10.156 sq. ft. The coefficient of lift necessary for takeoff predicted by the optimisation program was 2.5. Using the assumption of 80% of $Cl_{\max}$ at takeoff, the wing airfoil required a $Cl_{\max}$ around 2. Employing preliminary consideration, taking in count the estimated low Reynolds numbers (nearly 200000) on the go, aerodynamicists identified , among optimisation results, a family of airfoils (projected by Dr Richard Eppler) which is very efficient in the range of modelling Reynolds numbers (from 60000 to 200000); respective diagrams were determined during wing tunnel test at Stoccarda University. By the way, keeping in mind estimated structure weight and wing joint kind (realized inserting an aluminium pipe previously passed through upper fuselage), a 0.15 thickness was adopted. At least, as structurists wanted a ribbed and binded wing, the attention was focused on EPPLER 793 airfoil (fig.10).

If characteristic curves trend was observed in the preceding diagram, it would be clear the chosen airfoil is characterized by a $Cl_{\max}$ value of 1.55 while $C_{M_{med}}$ is -0.1. So, according to the competition rule of maximum take off run length (150 ft) it is clear how the obtaining of more and more high $Cl_{\max}$ values is determinant. Proceeding this way we'll need to provide the airplane with secondary control surfaces (as flaps and slats).

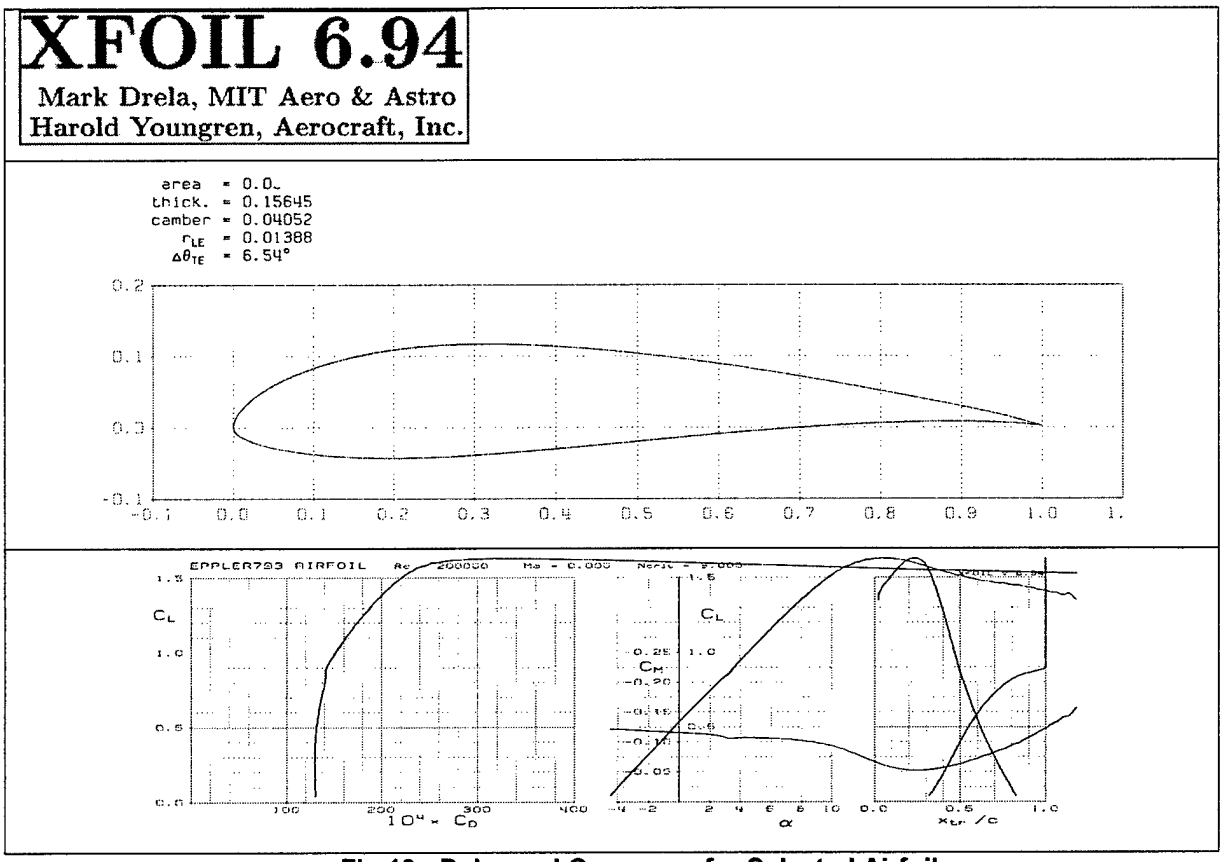


Fig.10 : Polar and Cm curve for Selected Airfoil

4.2.2 Fuselage Figures of Merit, Assumption and Analysis

The Figures of Merit for the fuselage analysis included:

RAC: the main RAC affect of the fuselage was the body length. So, in order to reduce the RAC, the body length was minimized but still contained all components and , above all, was less length than maximum dimension of transportation box.

- Drag: this one, on the fuselage was minimized in order to reduce the thrust required ultimately decreasing the battery weight. This is realized by a "new" concept: double mobile independent tail (DMIT). The wings are free to rotate for climbing the aircraft while the fuselage can remain parallel to the flight direction. This can be used during takeoff and water dumping phases in order to obtain the required high C_l value but also a reasonable C_d. This factor contribute to a higher score.

Further studies into effects of the fuselage were performed to obtain better drag models and stability characteristics. Water tank dimensions governed the optimisations of fuselage, in terms of both section and length. The fuselage was broken into three sections: the propulsion compartment, cargo bay and tail section. The necessary area for each portion was evaluated independently. Propulsion equipment, such

as the motor and speed controller, needed to fit the confines of the water tank compartment that's centred right in the C.G..The shape of the water tank was studied to achieve the best performance in terms of fuselage aerodynamic (compactness), flight stability and easy construction. The governing factor in the tail section dimensioning were propeller position (with relative mechanical transmission) and the T.P.D.F. (Tail damping Power Factor).

The three parts combine to form the total fuselage length of 46.6 in. By designing fuselage around the components, rather than vice versa, the length of the fuselage was minimized.

4.2.3 Tail Figures of Merit, Assumption and Analysis

As in this work propeller position was fixed ab initio, tail positioning and dimensioning must be defined in a way such to not perturb air flux directed towards propeller.

With this particular tail the TPDF has been calculated considering only one (of two) vertical tail. The other vertical empennage was not considered in order to take into account the horizontal empennage shadow effect during a corkscrew dive.

$$TPDF = \frac{S_e l_e^2}{S(b/2)^2} \frac{S_1 l_1 + S_2 l_2}{Sb/2}$$

where

S_1, S_2 = tail surfaces not put in the shade by elevator at specifics angles of attack

l_1, l_2 = distances from c.g. and, respectively, tail areas S_1, S_2.

4.3 Structural analysis

4.3.1 Figures of merit

During the preliminary design phase several figures of merit were used to represent different mission features for structural design.

1. Scoring potential: every aspect of each component had the potential to influence score through the ability to efficiently perform a task.
2. Weight: the weight of each component played an important role in both aircraft performance and RAC.
3. Ease of construction: more than anything construction time was determinant each time it was necessary make a choice.
4. Aerodynamic considerations: the design of the fuselage, wing, tail, wing/tail coupling, power transmission and landing gear influenced lift, drag and aircraft stability.
5. Cost: as Leonardo construction was sponsorised, only pieces reconsidered too expensive weren't adopted.

4.3.2 Design Parameters and Trade Studies Investigated

During the preliminary design phase parameters for primary components were identified for structural analysis. The design parameters investigated included

- Weights for primary components
- The number of composites reinforcements for foam fuselage, tails and wing
- Landing gear dimensions

During the structural analysis, different approaches to the design of each parameter were investigated to identify tradeoffs. The trade studies compared structural alternatives in the areas of material selection for each component, strength vs. weight tradeoffs, component deformation and aerodynamics. Using information from the studies assisted in the design of each parameter to achieve each components highest scoring potential.

4.3.3 Wing Assumptions and Structural Analysis

The wing structure was constituted of two half wings in which a rod had to be inserted: this one had to pass though the holes in the water tank. Wing-rod insertion was modelled as a two DOF, degree of freedom system ,while wing/ fuselage section (realized by the rod) was modelled as a SDOF, single degree of freedom system (see fig.17): each half wing (included the portion with the rod inserted) was considered as a cantilever . For simplicity of the analysis, the lift and drag loads were assumed evenly distributed along the lengths of the wing portion. Since the bending stresses in the wing caused by the lift will be the dominant factors contributing to failure, an evenly distributed load assumption will provide a more conservative analysis. When the distributed load was resolved into a resultant point load, the load was applied further out on the wing compared to a resolved elliptically distribution. Therefore the calculated bending moment will be larger. The same reasoning was used for justification for the drag load analysis. As a reinforced panels for wings was adopted, buckling analysis was easily conducted without surprises.

A trade study on three or two spar cross-sections, was analysed in order to provide the needed structural integrity with the least amount of material and weight. The investigation included calculating the bending, shear and axially tensile and compressive stresses that spars would encounter under conditions of a fully loaded aircraft at 3g load. At gross weight, the conditions resulted in a vertical wing loading of 2.34 lbs per square inch. The results of spars analysis showed that two main spar located, respectively, at 10% and 30% of the airfoil chord would be sufficient to support the expected loads.The shear stress on the rib was calculated from the aerodynamic moment on the wing. The shear stress showed to be extremely low compared to the material properties of the wood.

As regards bending, statically safety factor adopted was 2.2. The wing structural assembly weight was estimated at 2.204 lbs.

4.3.4 Tail assumptions and Structural analysis.

The moments created by the tail for longitudinal stability were a function of both airfoil used for the cross-section and the control surface deflection angle. The loading cases assumed for the structural design of the tail was loads of a 10-degree elevator deflection at a flight speed of 63.79 fps. The

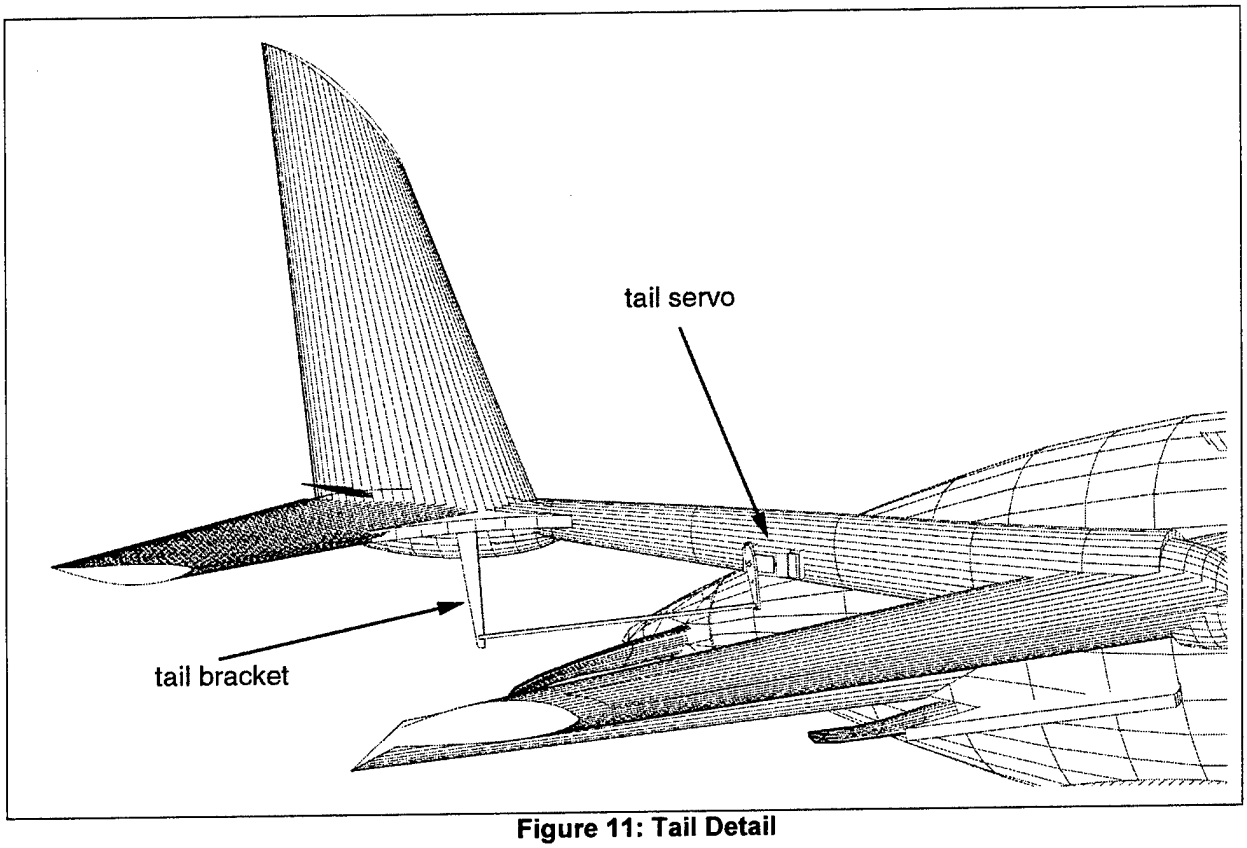


Figure 11: Tail Detail

loading case corresponded to an emergency control reaction at high airspeeds. Loads were also modelled from the aerodynamic moments and forces created by the airfoil. The stress analysis was performed in a similar manner . The horizontal tail is all mobile and has a single spar. Fin is high 17.4 in while span-elevator is 28.2 in and generates a maximum aerodynamic moment of 29.76 dyne per in.

4.3.5 Structural Analysis

In order to simplify structural analysis, our calculus's were tailored by torsion considerations. Airplane structure is based on torque box configuration whose basement is able to support flexion thanks to under reinforcements.

When applying loads to the fuselage, the following assumptions were made. All loads where modelled as resultant point loads with a safety factor of 1.5. The loads were developed as if the aircraft was loaded with the water tank and 18 batteries and experiencing a 3g loading. Engine torque was also analysed at maximum power settings. These assumptions were based on information from the preliminary design aerodynamic and propulsive studios and historical data.

4.3.6 Main Gear Assumptions and Structural Analysis

Only main gear effect was modelled as concentrated forces applied under the basement edges of the fuselage . Tyres were considered as solid springs :in this way, reaction soil transmitted to fuselage can be considered not concentrated forces of infinite magnitude .

Good results was obtained, by strong aluminium gear-legs, during tests performed to verify fuselage strength in the cases of a two-wheel touchdown and a one-wheel touchdown (drop test) in MLW (Maximum Landing Weight). A gear load factor of 3 was assumed for ordinary two-wheel landings. The gear load factor caused the landing weight to increase by a multiple of 1.5 per wheel for two-wheel landing conditions. As regards drop test, the multiple was increases to 5.

4.4 Propulsion Analysis

For the preliminary design phase for the propulsion system we availed ourselves of Motocalc program for optimising the major components: batteries, motor, and propeller. The program simulated the complete propulsion system from the power used by the batteries to the power produced by the propeller in flight. Components were allowed to vary in such a way as to match and/or exceed the output values estimated for thrust, current, power, and efficiency for the overall propulsion system. Estimated values were provided from the optimised aerodynamic aspects of the plane. Four propulsive components were varied in the optimisation program: battery cell count, propeller dimensions, motor-size, and gearbox ratio. Six motor series were examined for the program: the Astro Cobalt 25, 40, and 60 and Astro FAI 25, 40 p/n 642, and 40 p/n 643 series. The gearbox is entirely self-built. See below for component's choice criteria.

4.4.1 Figures of Merit

Figures of merit were developed to evaluate the design parameters. The figures of merit represented the relation between efficient propulsion system and the competition goals and regulations. The figures of merit used were:

- Score: in all aspects of the design, score was the ultimate figure of merit. All aspects of every design decision had the potential to influence the overall scoring potential of the aircraft
- Efficiency: the propulsion system affected score in both flight performance and Rated Aircraft Cost. Efficiency of each part was optimised to offer good performance in both areas
- Weight: the weight of the propulsion system was determined to be a large player in the Rated Aircraft Cost because battery weight was heavily penalized. Therefore reducing the weight of the batteries as well as other propulsion system components can provide a dramatic score improvement
- Current: the current draw on the batteries was a limiting factor set forth by the competition regulations

- KV Values: the KV Value of a motor was a ratio that relates RPM to voltage. Selecting a motor with the proper KV Value allowed thrust to be produced at acceptable current levels
- Energy Density: the energy density of the batteries was crucial to determining which batteries provided the most energy with the lowest increase in battery weight.
- Historical Data: historical data was invaluable in predicting efficiencies and performance during the trade studies performed on the different propulsion components

4.4.2 Design Parameters and Trade Studies Investigated

Following are component's choice criteria of propulsive group. As said, we used Motocalc program.

- P/D ratios: by fine-tuning the ratio of propeller pitch to propeller diameter, it was possible to increase the efficiency of the motor and thereby reduce the number of batteries required.
- Propeller Diameters: the propeller diameter has a direct affect on the amount of current required to maintain an RPM setting. Since the propulsion system is limited to 40 amps of current the propeller diameter was a limiting design parameter.
- Number and type of batteries: overall aircraft efficiency was a crucial parameter on the competition. An efficient propulsion system would find the optimum balance between power produced and batteries required. Balance was crucial to reach the aircrafts maximum scoring potential.
- Size and type of motor: Different motors were optimised for operation in specific ranges. Motor selection was important with the most potential to operate efficiently under the predicted flight constraints

4.4.3 Weighted Decision of Motors

The results of a weighted decision matrix were analysed for the FAI motors and Graupner motors. From the decision matrix resulted FAI motor series score were all fairly close in relationship with each other. Apparently Graupner motor efficiency is generally lower than the FAI motor series, but to determine which motor was best for the design, other parameters must be considered.

4.4.4 Investigation of Graupner Motor

Following there is a numerical analysis of performance behaviour based on Motocalc program. From the first inset, left to right, top to bottom, we have total efficiency vs. airspeed (mph); power (W) vs. airspeed; thrust vs. airspeed; current vs. airspeed.

Figure of Merit (FOM)	Weight	FAI 25	FAI 40 / 642	Graupner Ultra 920
Number of Batteries	0.3	0.9	1	
Current	0.25	0.7	1	
Takeoff Thrust	0.25	1	1	
Cruise Power	0.1	1	1	
Efficiency	0.1	0.9	0.8	
Score		0.90	0.98	

Table 7 : Motor Weighted Decision Matrix

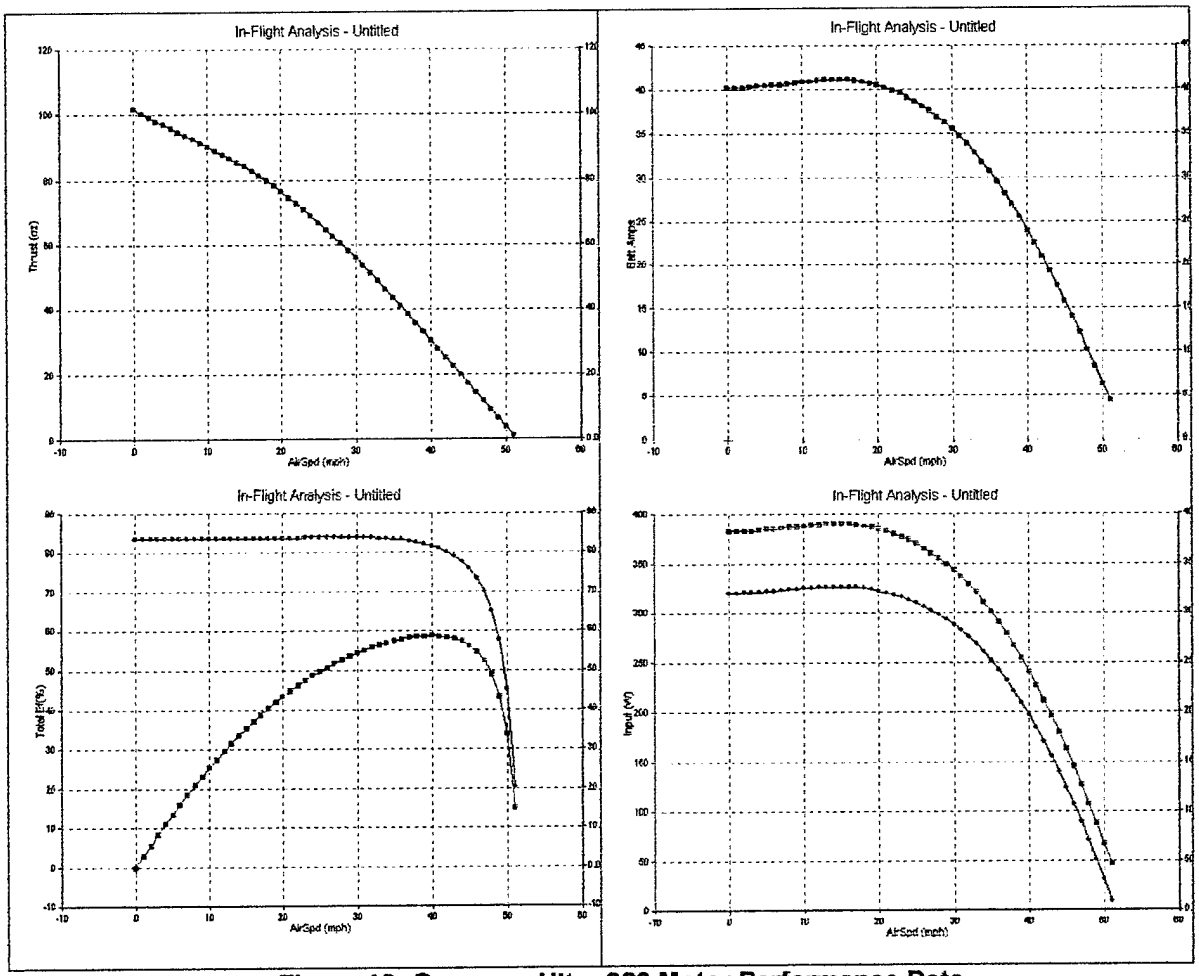


Figure 12: Graupner Ultra 920 Motor Performance Data.

4.5 Analytical tools

4.5.1 Optimization Program Architetture

After the conceptual phase was completed, the drag, weight, propulsive and RAC models were updated. The following models describe the updates to the program:

- Drag Model: the weight model was reconstructed according to the drag build-up method found in Raymer (1999). The method gave a closer estimate of drag without requiring the parameters developed in the preliminary and detail design phases. The drag model also included the drag polar and lift curve for the chosen airfoil. Different values were used for both cruise and take-off Reynolds numbers. Conservative margins of safety were also adopted.
- Weight Model: the weight model was reconstructed according to the structural and propulsive assessments presented earlier in the conceptual design phase.
- Propulsive Model: the propulsive model was compared to the analytical model used by the propulsion group in its analysis. The model showed that the conceptual model was a little bit too conservative (it was probably due to the aerodynamics refinements).
- RAC Model: the RAC was also updated to meet all the RAC requirements, and better represented the features chosen in the conceptual design phase.

Drag, weight propulsive and RAC models were placed in the program along with the parameters that best support an optimal design. The program was adapted so that the scoring potential was maximized and trade studies occurred using the optimisation process

4.5.2 Other Methods Used

Other than optimisation, several other programs were used to calculate the various parameters of the design. First we started with 3D Studio MAX, then aerodynamics group used ANSYS for Computational Fluid Dynamics (CFD) and structures group for a structural analysis, while the flight mechanics group used Fluent to verify stability, then the layout was transferred to the CAD program for three-dimensional modelling.

4.6 Final Aircraft Configuration

At the end of preliminary, the primary components of the configuration were all sized and placed according to the figures of merit. The fuselage length was minimized using a "drop-type" fuselage to accommodate the water tank and keeping the overall drag low. As said the wing configuration was a D.M.I.T.. The full control of the airplane is achieved by only 2 servos moving the horizontal tail.

4.6.1 Wing and Power Loading

Wing and power loading for the preliminary configuration was 2.34 lb/ft2 and 26.3 lb/hp. The two loading values were a compromise to achieve the correct mission balance and achieve the best scoring potential. Higher wing loading allowed the plane to penetrate through the wind better, achieve higher cruise

velocities, and makes the configuration less susceptible to gusting conditions because greater changes in pressure differential are required to disturb the plane. Drawbacks to high wing loading occurred at takeoff and climb. Higher wing loading was not helpful during the critical phases.

To overcome the wing loading disadvantage, higher power loading was required to help the plane overcome takeoff and climb requirements. Therefore, a compromise was met between wing and power loading to result in the preliminary configuration.

5.0 Detail Design

After the primary components were sized in the preliminary phase, secondary components were sized and added. The aerodynamics group had the task of verify the preliminary design. As a model of analysis we preferred to use the Fluent program, so a 3D model within the simulator was created. This analysis garanted a more global view of pressures acting on the aircraft with respect to experimental methods (i.e. wind tunnel). Moreover, it was able to verify that the downwash angles used for tail keying were correct; it has been used to determine the dynamic stability and verifying the initial hypothesis: the fact that besides not having a dihedral angle the fluxes interaction between fuselage and wing permitted a good behaviour on roll.

5.1 Aerodynamics Performance Analysis

The optimization code was modified to provide data on the flight performance of the final configurations. The use of an airfoil analysis program allowed the characteristics of the airfoils of the wing and tail to be included in the analysis. The following sections contain results from the two computer codes, representing how the plane would perform.

5.1.1 Final Configuration Features

The final aircraft configuration is a rather unconventional one. A "drop-type" fuselage with a trapezoidal wing, with a Double Mobile Independent Tail (DMIT) slightly less than a quarter of the wing span to avoid RAC penalty. The wings can rotate 12 degrees around fuselage pitch axis. In this way we obtain a better propeller configuration cause of the its steady position and pushing trust; in fact the fuselage remains parallel to the ground in every conditions. A more complete description of the aircraft configuration features is listed in the following table.

Fuselage		**Wing**	
Length (overall)	46.6 in	Airfoil	Eppler
Maximum width	6.1 in	Span	85.8 in
		Chord	22.4 in

Horizontal Tail Properties		Area	1462.43
Airfoil	Eppler	Incidence angle	2.6 °
Span	14.1 ft	Incidence angle (zero lift)	6.85 °
Chord	7.4 in		
Vertical Tail Properties		**Total aircraft**	
Airfoil	NACA 0012	Rated Aircraft Cost	11.25
Height	17.4 in		
Chord	7.4 in		

Table 8 : Final Aircraft Configuration Features

5.1.2 Estimated Mission Performance

The estimated mission performance of the configuration was calculated from the performance code used in the optimization. Each stage of the mission was individually evaluated. The time consumed for each step was of particular interest. Table 9 shows the time spent in each portion of the mission profile. The time spent in cruise, both loaded and unloaded, was the total time at cruise velocity, including all turns and straight legs. Also included in Table 9 were the expected flight velocities and distance of each phase. The performance information was calculated assuming a wind speed of 5 mph. Accounting for all phases of the mission as well as the RAC, the aircraft was predicted to score 1.75 times the report score.

Mission Components	Time Spent	Distance	Velocity
Takeoff unloaded	5.8 sec	87.81 ft	15.12 fps
Climb unloaded	9 sec	---	29.25 fps
Cruise unloaded	240 sec	---	70.13 fps
Takeoff loaded	6.6 sec	137.6 ft	20.82 fps
Climb loaded	15.1 sec	---	48.15 fps
Cruise loaded	280 sec	---	81.72 fps

Table 9: Time Spent in Mission Phases. Value predicted by performance program.

5.1.3 Takeoff and Climb

The takeoff distance for loaded conditions was 137.6 ft from the optimization code. Takeoff velocity was approximately 20.82 fps. The plane needed to accelerate after takeoff to a velocity of 60 fps to maintain the proper climb speed. For unloaded conditions, the takeoff decreased significantly to 87.81 feet, and takeoff velocity also decreased to a speed of 15.12 fps. When the plane accelerated to climb velocity at unloaded conditions, the plane moved at an airspeed of 29.25 fps. The time for takeoff and climb scenarios can be seen above in Table 9.

5.1.4 Flight Conditions

The aircraft was designed to perform best in a 5 mph wind. At this wind speed, the empty cruise speed was projected to be 85.1 fps while the loaded speed was projected at 79.6 fps.

In order to close aerodynamic considerations, we produced the following picture by fluent in which it is possible to appreciate the uniform pression field behind the propeller and wings. It is evident the low degree of perturbation .

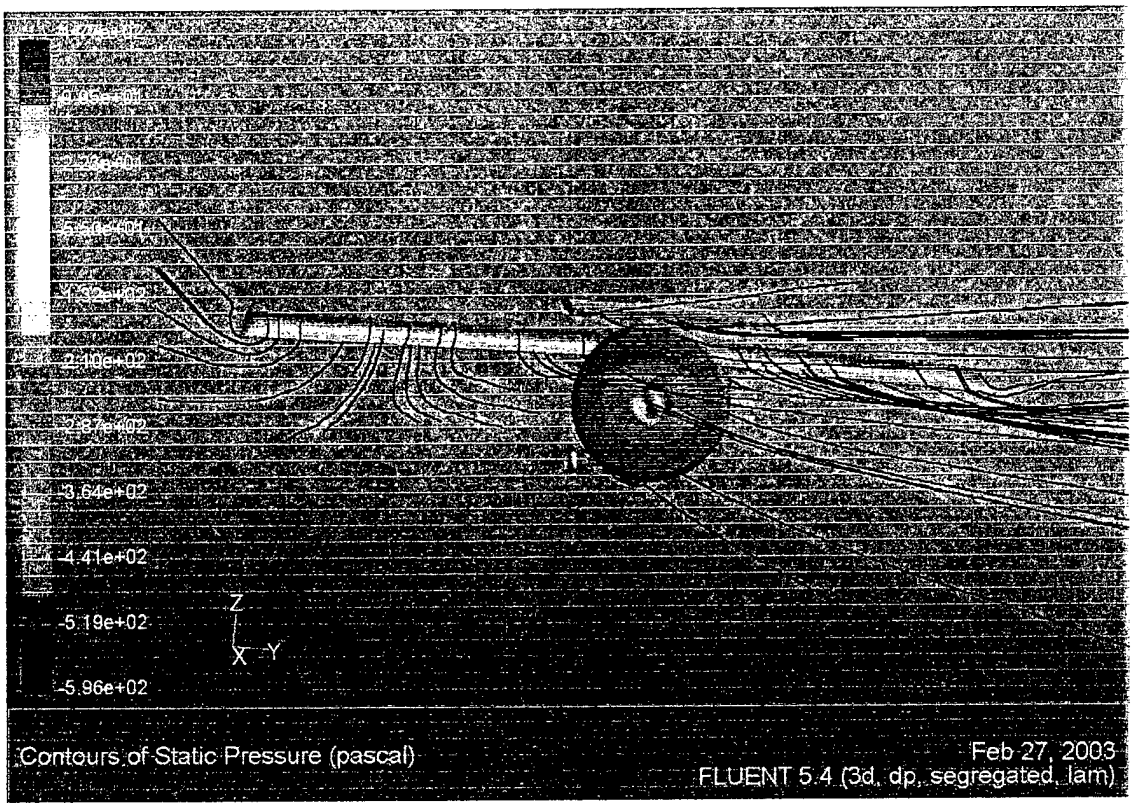


Figure 13: Pressure Field behind the Plane (project n°3)

5.2 Propulsive Performance Analysis

The final propulsion system was designed in the preliminary phase using analytical methods. Using a dynamometer, the propulsion group tested several of the top motors that remained at the end of the

preliminary phase. The final motor and propeller combination was decided to be the Graupner with a 20 inch diameter propeller, 18 batteries were needed to produce the thrust (they were of course NiCad batteries). The final output was 6.6 pounds at fixed point, using approximately 700 watts from the batteries; 4.4 pounds (with approximately 600 watts) at rotation; cruise thrust 1.7 pounds and drew 400 watts from the batteries. The values above were used for the construction of the prototype and refined the optimal results.

5.3 Structural Considerations

The focus of the detail design phase was the integration of the primary components with each other and the various systems comprising the aircraft as a whole. The details of the four major aircraft components, the fuselage, wing, tail, and gear, were designed to insure form, fit, and function of each component.

5.3.1 Fuselage Structural Details

The "drop-type" form of the fuselage was designed to carry the payload represented by inner tank which is loaded with water. Maximum allowed water capacity is four liters. Fuselage design was accomplished employing Studio 3D program and then importing the data in CAD to shape the fuselage around internal components while minimizing wetted surface area and optimizer materials.

The basement gear-carrier and load carrier has been dimensioned to support shear and axial stresses. Shear stresses are caused by the contribution of the reaction of the front landing gear and of that given by torsion: those due to torsion would exist in case of an asymmetrical landing. Axial stresses are due to the inflexion caused by the reaction of the wing at the moment of touchdown, following the reaction of the front landing gear. Vertical bulkheads realize the connection bridge-structure –basement allowing the vertical load transfer (i.e. lift); moreover they realize the transfer of torsional moments between bridge-structure /basement. The two rear connections "seized" to the bridge-structure and basement transfer axial stresses due to flexion induced by drag generated by the horizontal empennage during pitching manoeuvres. Front gear support divides between the bridge-structure and the basement stresses of the front landing gear. Another function is to prevent the bridge-structure from sliding in the direction of the basement; such sliding would take place in case of a landing on the front gear.

The aft section was designed for purely aerodynamic reasons which made it preferred to a boom.

5.3.2 Wing Structural Details

We have taken in consideration that the wings, in order to be transported inside a fixed-dimension box, had to be necessarily detached from the fuselage, implicating, of course, a series of troubles. Those troubles have been resolved by adopting a systems made up of an aluminium tube to insert in proper holes in the fuselage (water tank) and the wings. On this tube wing -roots plug themselves. Since we have chosen an "all-mover" wing instead of a more classical wings + ailerons, we designed a system made up of a hinged pivot on a proper wing rotation system. Before we proceed in this study, we want to

examine another critical load situation that is corresponding to the wing tip lift test, whose successful overcoming is necessary to be admitted to the fly-off. The load is a concentrated-type one and is 4.165Mdyne(41.65N). We verified that in order to have a structure that is able to absorb, in case of a lift load, axial stresses of 342.289 psi (2.36 MPa) and shear stresses of 72.51 psi (0.5 MPa; stresses given by wing tip loads are smaller), it was sufficient to adopt spars with slabs having transversal section of 0.039 x 0.039 square inch, stringers having sections of 0.236 x 0.236 square inch; wing cover 0.059 in thick except for the sectors between the two spars where it has to be thicker (0.118 in). The reason for this solution appears clearly if we examine the following figure where it is shown with red arrows shear stress trend, almost entirely absorbed in the area comprised between the two spars. We determined at last that the value of the force acting on the wing rotation pivot is 12 Mdyne (12N), so it was necessary to adopt a pretty massive mechanism since a more slim junction would not be able to resist to such a load.

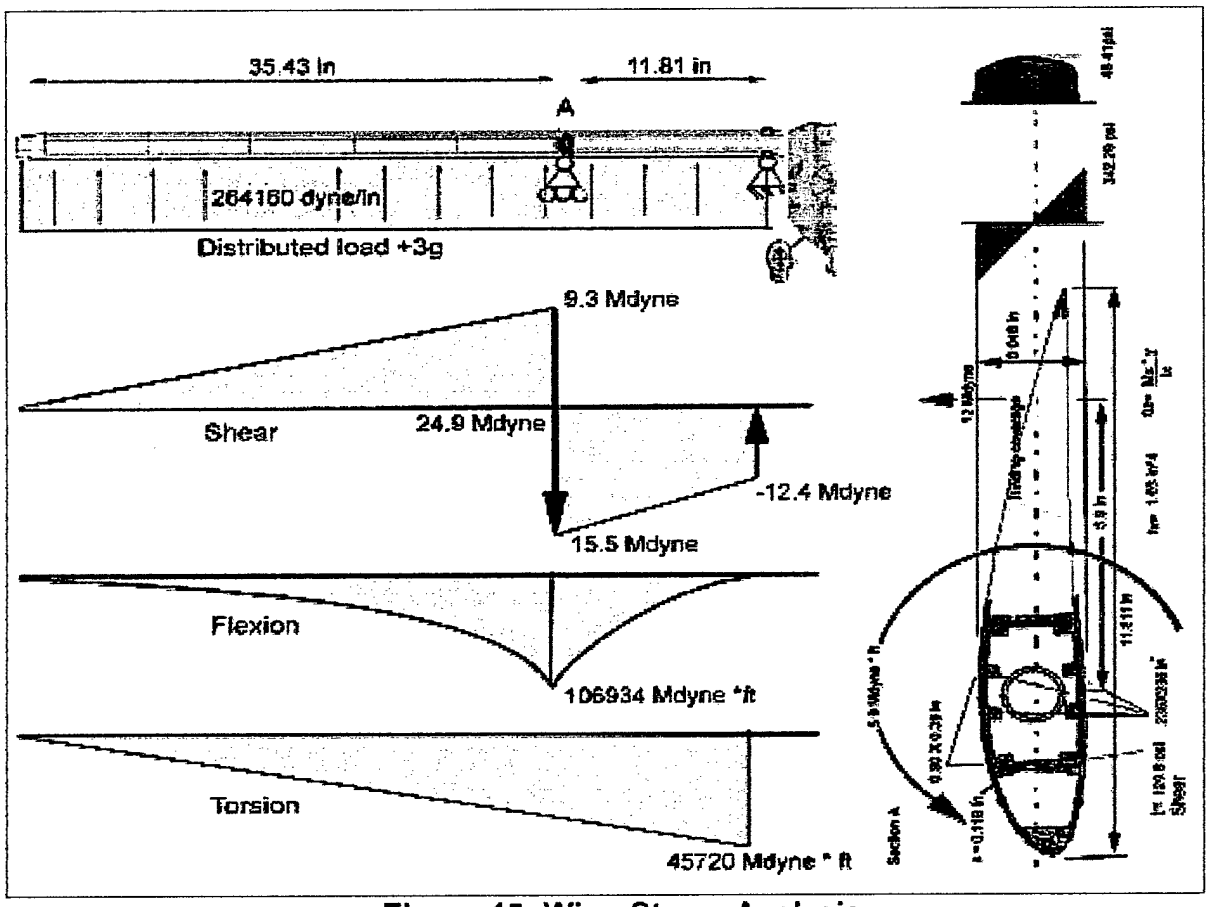


Figure 15: Wing Stress Analysis

To determine the stresses to which is subject the aircraft during possible manoeuvres, once fixed maximum and minimum load factors, we made a preliminary study to define diagram of manoeuvring

loads: in this way we have rapidly visualised aircraft structural limits. Then we proceeded to the stress analysis within the structure. Following are the results of shear, flexion and torsion trends.

As we can see, the sections to be studied are those in correspondence of the 'A' point in figure 15 and the wing root: in the 'A' point we have a shear discontinuity, while the maximum value of torsion is in proximity of wing root.

5.4 Detail Design Assumptions and Comparisons

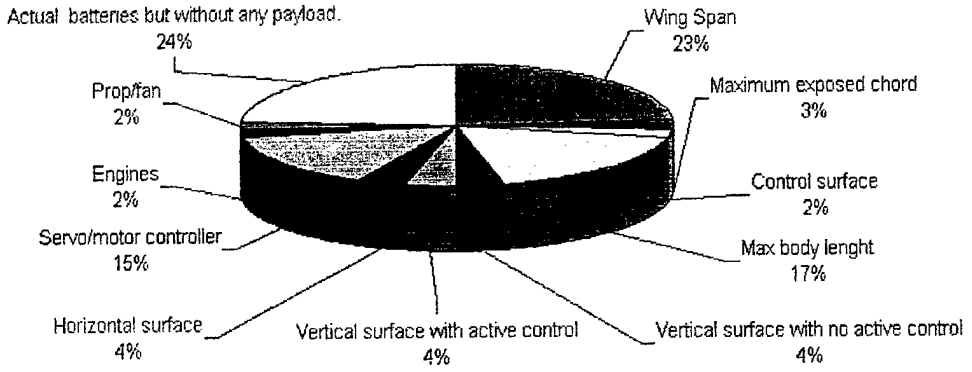
During detail design analytical assumptions made in preliminary design were reinforced by static structural testing, propulsion dynamometer testing and finally prototype flight-testing. The prototype aircraft, less payload and batteries weighted about 11.24 lbs during flight test number one. Luckily estimated total weight of 18.74 lbs in preliminary design became17.63 lbs.

Center of gravity resulted to be in the estimated position (30% of equivalent swept wing chord). In

Aircraft Component	Station (in)	Weight (pounds)	Moment (pound*in)
Batteries			
Propulsion battery	13	2.64	34.32
Servo battery	12	.55	6.6
Propulsion system			
Engine	5	.52	2.6
Propeller	27	.26	7.02
Watt-meter	0	0	0
Aircraft structures			
Fuselage (with hatch)	9	2.87	25.83
Main gear	1	.15	.15
Nose gear	12	.11	1.32
Tail	26	.33	8.58
Wing	1	2.2	2.2
Payload			
Water	0	8.8184	0

Geometry	Datas	Measures
	Length	46.6 in
	Wing Span	85.8 in
	Height	34 in
	Wing Area	1462.43 sq in
	Aspect Ratio	5.22
	Control Volumes	2250 cubic ft
Performance		
	CL max	2.5
	L/D max	11
	Max Rate of Climb (ft/sec)	48.15
	Stall Speed (ft/sec)	36.45
	Max Speed (ft/sec)	61.05
	Take-off Field Length (empty)	87.81
	Take-off Field Length (gross weight)	137.6
Weight Statement		
	Whole Airframe (lbs.)	5.66
	Propulsion System (lbs.),batteries included	4.4
	Control System (lbs.)	1.41
	Payload System (lbs.)	0.2
	Empty Weight (lbs.)	11.67
	Payload (lbs.)	8.8184
	Gross Weight (lbs.)	18.67
Systems		
	Radio Used	Graupner X-3810
	Servos Used	3
	Battery Configuration Used	Series
	Motor Used	Graupner Ultra 920
	Propeller (Nominal)	18/14 (3-bladed)
	Gear Ratio (Reducer)	1:9.8

Table 10: Aircraft Data



Manufacturing Man Hours: Work Breakdown Structures				
Aircraft Component	Man Hours/Unit	Aircraft Parameter	Individual Hours	Individual Contributions
Wing WBS				
Wing Span	8 hr/ft	7.94	63.5	1270
Maximum exposed chord	8 hr/ft	0.98	7.8	156
Control surface	3 hr/surface	2 surfaces	6	120
Fuselage WBS				
Max body length	10 hr/ft	4.59	45.9	918
Empenage WBS				
Vertical surface with no active control	5 hr/surface	2 surfaces	10	200
Vertical surface with active control	10 hr/surface	1 surface	10	200
Horizontal surface	10 hr/surface	1 surface	10	200
Flight Systems WBS				
Servo/motor controller	5 hr/servo	8 servos	40	800
Propulsion System WBS				
Engines	5 hr/engine	1 engine	5	100
Prop/fan	5 hr/prop	1 propeller	5	100
Total Manufacturing Man Hours				**4064**

	Actual airframe weight, lb., with all flight and propulsion batteries but without any payload.	$100	12.63 lb.	NA	**1263**
	"Total Battery Weight" defined as weight of propulsion batteries.	$1,500	1 Engine 3.3 lb. Batteries	NA	**4950**
	Sum of assembly hours defined by Work Breakdown Structure (WBS).	$20/hour	-	190.25	**3964**
					10177

Predicted Rated Aircraft Cost = 10.177

Table 11: Rated Aircraft Cost Worksheet and, Pie chart. Predicted RAC of 10.177

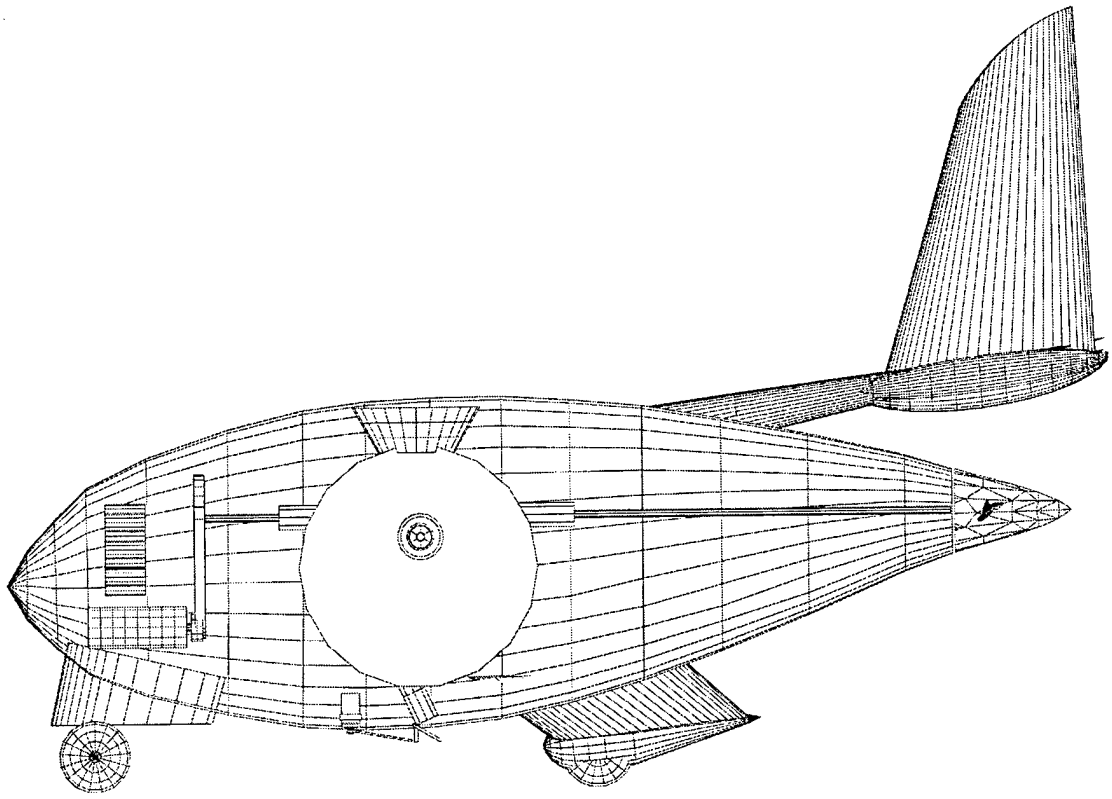


Fig16: internal view

6.0 Manufacturing plan

6.1 Manufacturing Process Investigated

In order to produce the aircraft, several different manufacturing techniques would have to be employed for the different components. Several different methods for obtaining the desired shapes (specially the wing airfoil) , the surface finishes (specially in terms of interfaces between mobile surfaces and fixed) and smoothness of mechanisms were investigated for each component. The figures of merit used to evaluate different manufacturing techniques included the ability to produce the desired shapes and finish, cost of tooling and manufacturing, build time, required materials (in terms of different woods and metal pieces), skill levels and process repeatability.

6.2 Processes Selected for Component Manufacturing

In order to produce the aircraft, some different manufacturing techniques were evaluated. The figures of merit used to evaluate different techniques included ability of wood cutting, cost of tooling and manufacturing, build time, skill levels, and process repeatability. Drawings in scale 1:1 were employed for cutting of pieces from woods. This technique resulted very useful as allowed to minimize errors during cutting, sizing and shaping of singular pieces. Mainly cyan acrylic glue and contact glue were used.

6.2.1 Fuselage Manufacturing Process and Tooling

In first of all, frame spaces for spars and reinforces alignment was determinant for the correct alignment of the transmission shaft with respect to the nose. Then it was necessary to reinforce each clutch of pieces (not jointed) with angular reinforces glued at respective edges. Also the alignment of the holes through which rods (for wings) had to pass was realized carefully in order to avoid interferences during mounting and flight. Obviously, during structure mounting, control cables, shaft, then motor and wires and at least servos were positioned. Fuselage was covered after test (with the airplane fixed on the terrain, of correct working either of mobile surface either of engine and propeller.

6.2.2 Wing and Double Mobile Independent Tail Manufacturing Process and Tooling

The wing and the double mobile independent tail had an architecture of spars and polystyrene foam core. During construction also dimensions (as wing-span and double tail span) was strictly controlled : as regards double tail, it resulted slightly wider than projected, so it was necessary to shrink it by reducing tips thickness. Continuing with fuselage analogy, also alignment of holes for the passage of the rod for wing attachment was strictly controlled. We paid attention also to low side of wing in correspondence to trailing edge due to particular shape of airfoil and its relative difficulty of conforming. Double tail structure was constructed with analogue technique used for wing. They were positioned at the end of the two booms jointed on the wings.

6.2.3 Landing Gear Manufacturing Process and Tooling

Each wheel of main landing gear was connected to fuselage by an aluminium leg screwed to fuselage basement. They where also streamlined. Bow gear can steer .

6.3 Analytic Methods Including Cost, Scheduling and Skills Matrix

6.3.1 Manufacturing Cost

With the preliminary design study complete, manufacturing and tooling costs were estimated. Luckily our work was greatly sponsored by a modelling shop.

6.3.2 Skills Matrix

In order to assign tasks for the manufacturing process it was necessary to develop a matrix of the skills required for each task. Table 12 contains the skills matrix. In the matrix a component that requires a lot of skill in a certain area was rated two, a component that required average skill in an area was score with one and a component that required no skill in an area received a zero. The columns of the skill matrix represent required skills in the manufacturing process; the rows represent the major assemblies and system of the aircraft. Members were assigned to components matching their expertise. An external laboratory consultant observed our work.

Primary Aircraft	Wood & polystyrene	Wood & polystyrene	Resin modelling	Radio Equipment	Electrical work	CAD modelling

Assemblies and Systems	cutting	gluing		Installation		
Wing	2	2	0	2	1	2
Fuselage	2	2	1	2	2	2
Landing gear	0	0	0	0	0	1
tail	2	2	0	2	2	2
Propulsion system	0	0	0	2	2	2

Table 12: Skills Matrix For Leonardo Team

6.3.3 Manufacturing Scheduling

The aircraft was constructed in four assemblies: the fuselage (included landing gear), wing, and the two booms included horizontal and vertical tails. The assemblies were constructed to allow components to be constructed simultaneously. Considerations such as material availability were of critical importance to maintaining a smooth manufacturing process. Figure 20 is the milestone chart developed for the manufacturing process.

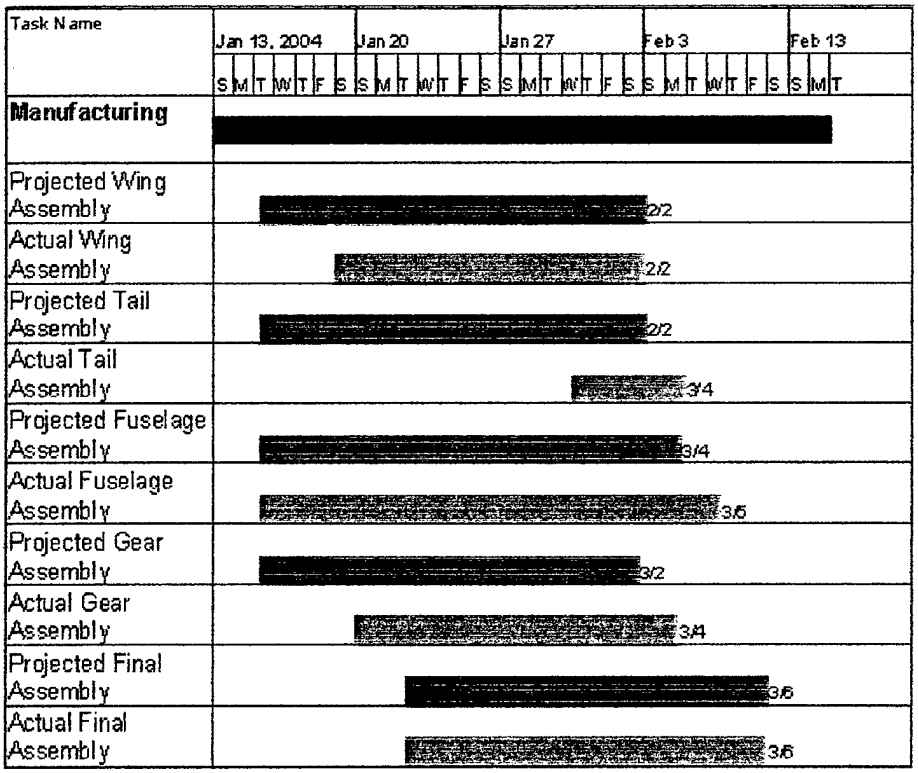


Fig.20 Manufacturing Schedule

7.0 Testing

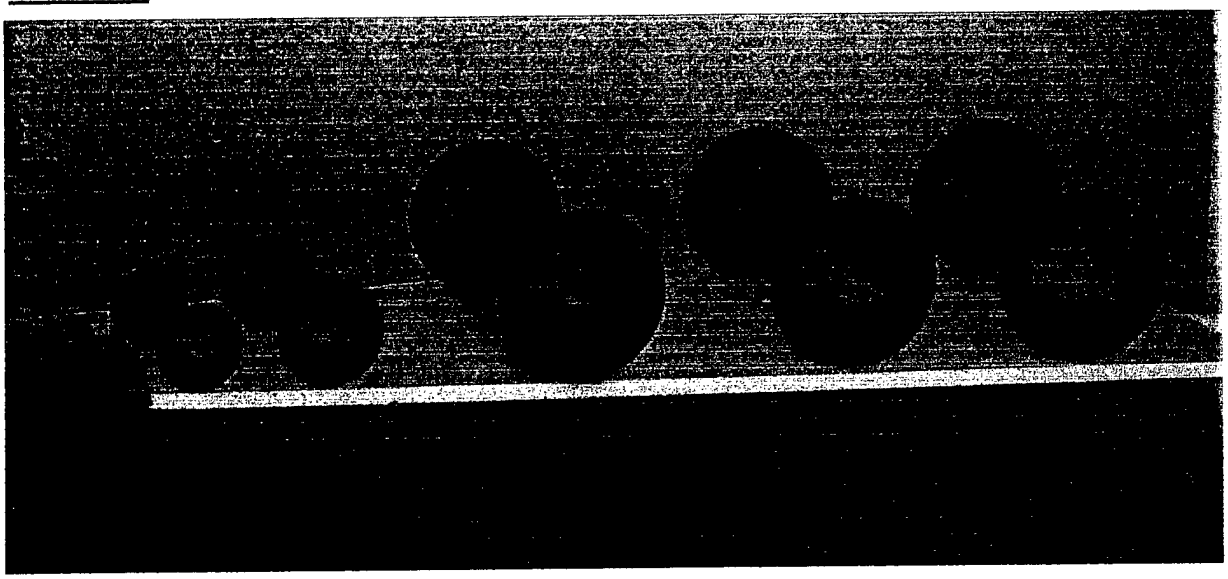
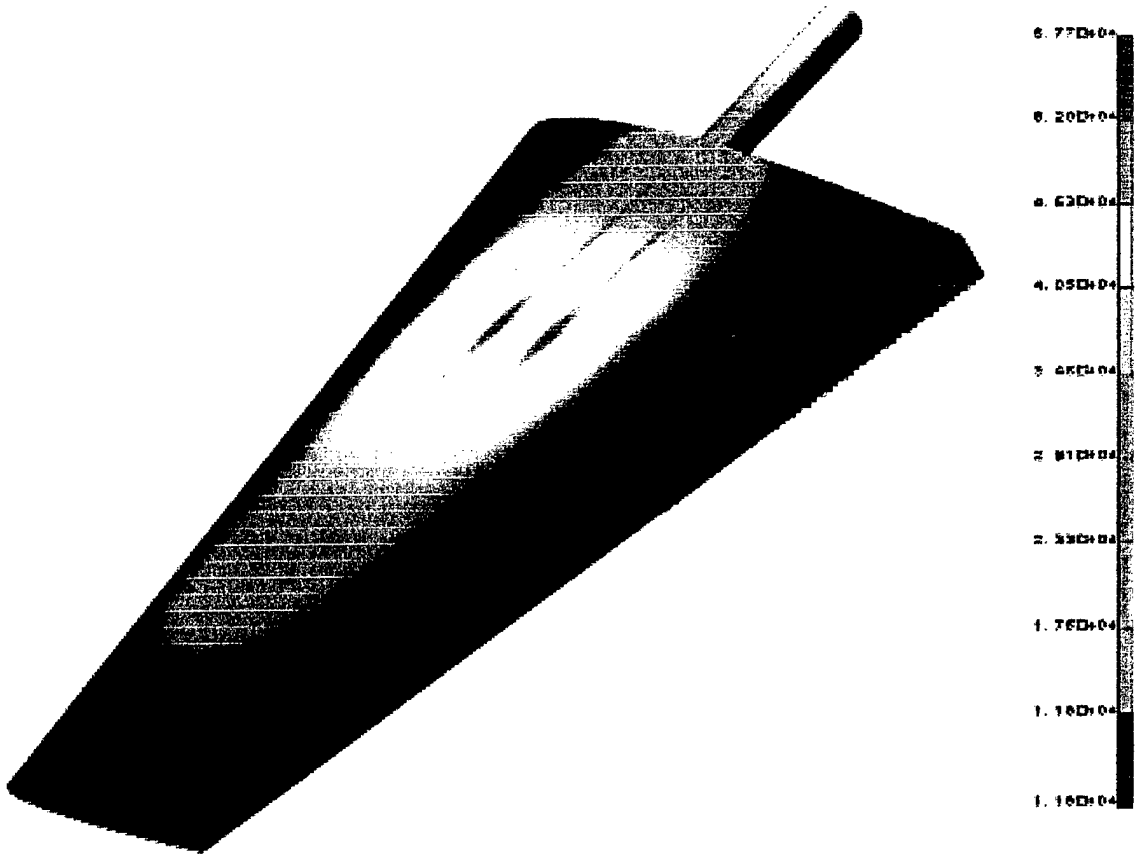


Fig.21 Simulation of wing trapezoidal load during Michelangelo construction. Wings were loaded employing some gym weight. In this picture a 3g load (37.56 lb)

Fig.22 Finite Element Method Analysis (wing load)



7.1 Static Tests

To be sincere, our tests (static) began during airplane assembly. Of course wings were the main character of our tests. Mainly we verified their strength when subjected to lift (that is to say airplane weight) supposed to be elliptic. In the following image (figure 22) we can see our static test.

An another test was to verify smoothness of wing connection to the fuselage in order to ensure a rapid and sharp assemblage of them during competition. This last operation resulted very useful as we had to refine several times all our plug-connections and in one case it was necessary to replace a spring of plug system. Moreover we tested also smoothness degree as regards shaft rotation. We observed it was necessary to insert two axle box (*in carbon, we apologize for this historical licence!*), one in the fin an the other in the trailing edge of the "sunshine". We didn't considered fatigue tests as low values of static loads in a wood contest. Motor and propeller vibrations didn't manifested a serious risk for structural integrity. Another test repeated continuously was the auctioning of control cables correct working in terms of degree of rotation of control surfaces. Last tests involved the discarge of the water throught the orifice of ½ inch. The test was successful the tank is emptied of its four liters of water in about 45 seconds.

7.2 Flight Tests

All the tests of flight have been conducted exclusively thinking about the two missions of the DBF of this year. The two missions are been simulated, fire-fight and ferry, verifying in the first one the suit emptying of the reservoir during the downwind leg including 360° in about 50 seconds, and in the second the complete result of the 4 anticipated turns from the mission even if to the limit of the duration of the batteries. The behavior of the Michelangelo has also satisfied particularly broadly our expectations the delicate part regarding the choice of the most proper helix. The lengths of take-off also result them in the limits imposed by the rule of competition.

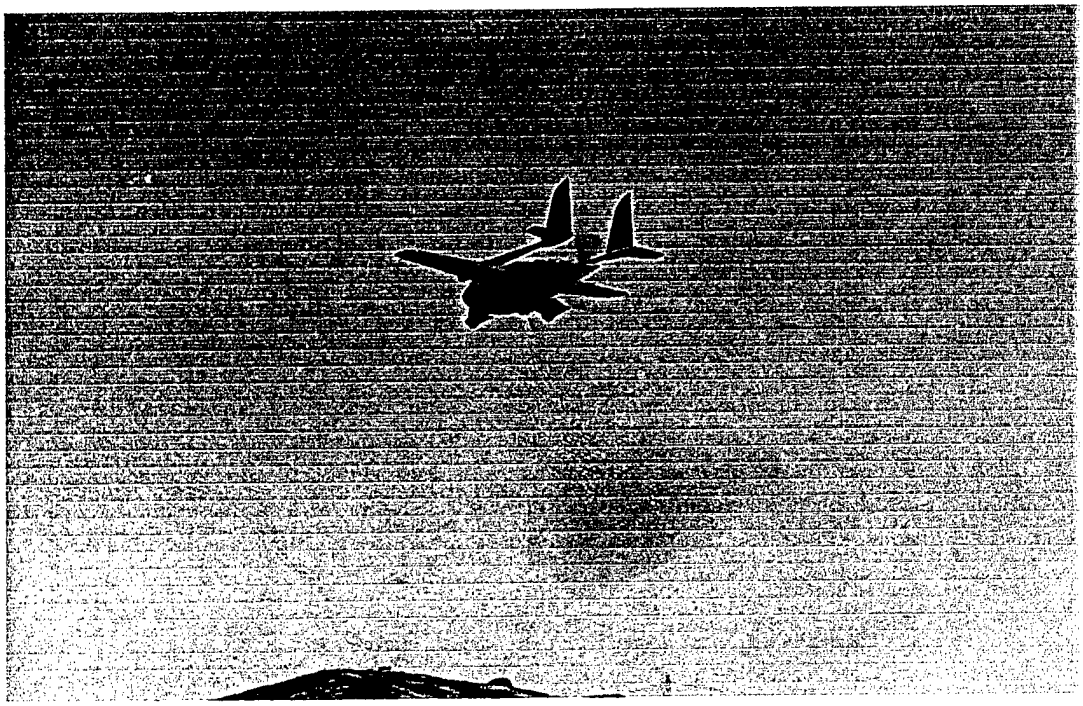


Fig.22 The first water dumping flight!

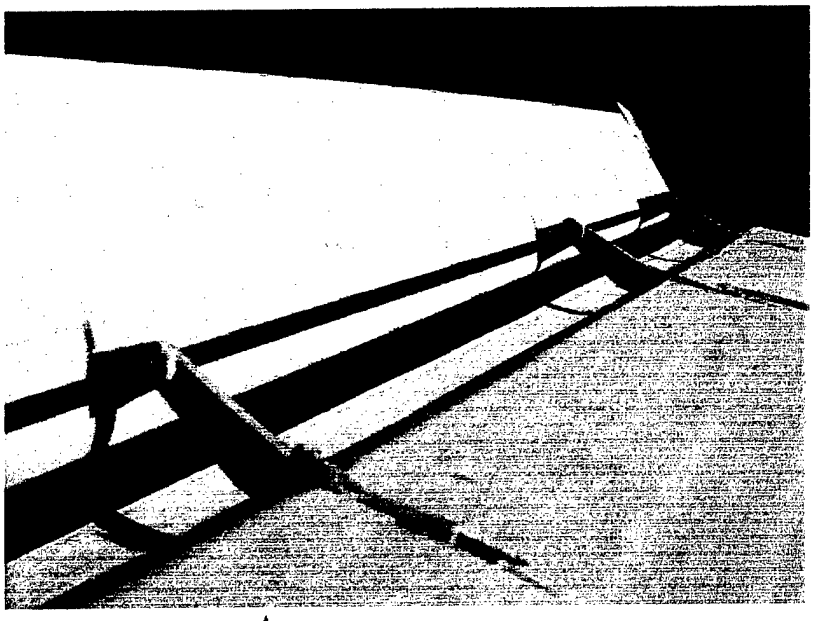


Fig.23 Particular of the manual-armed flap

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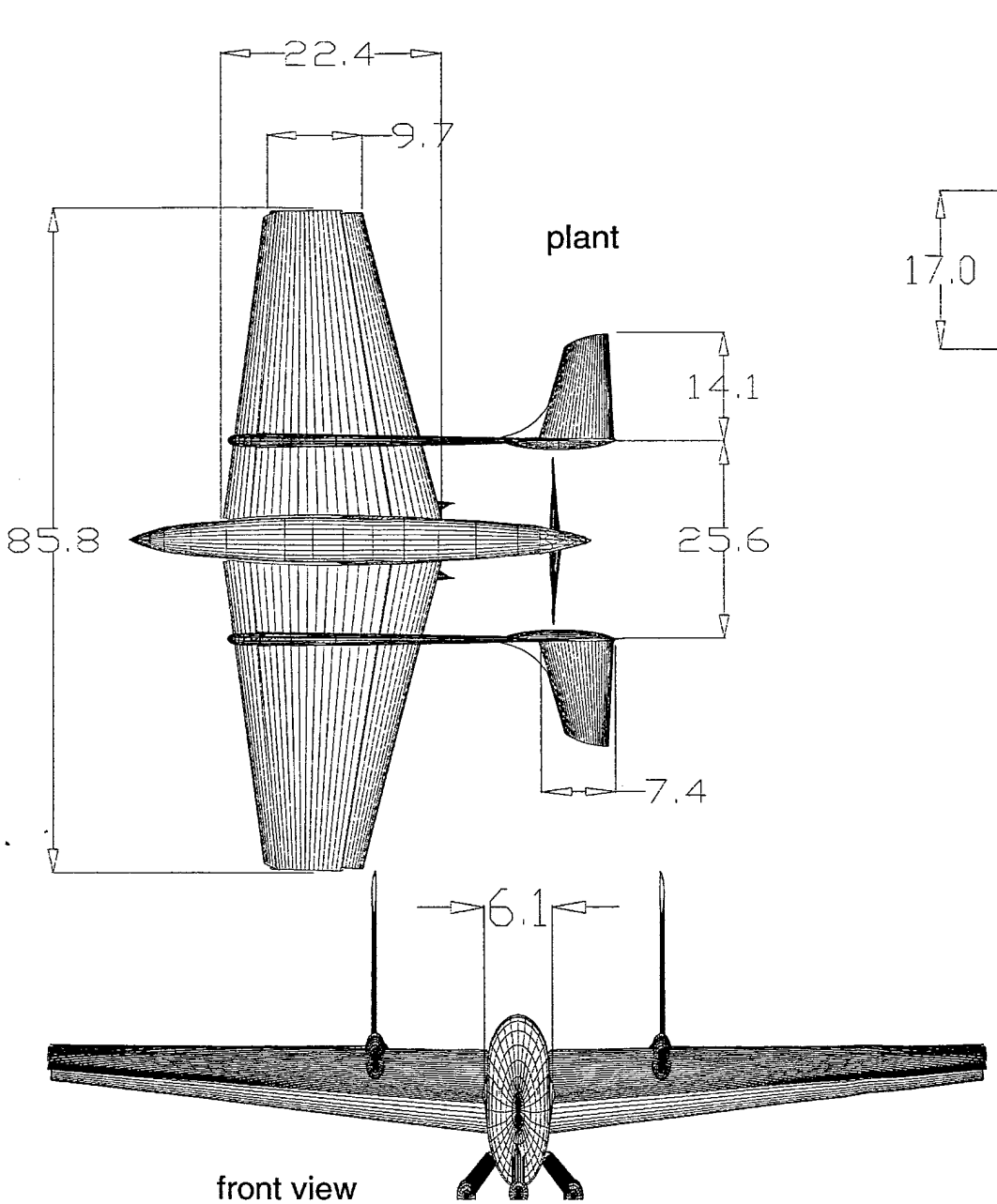
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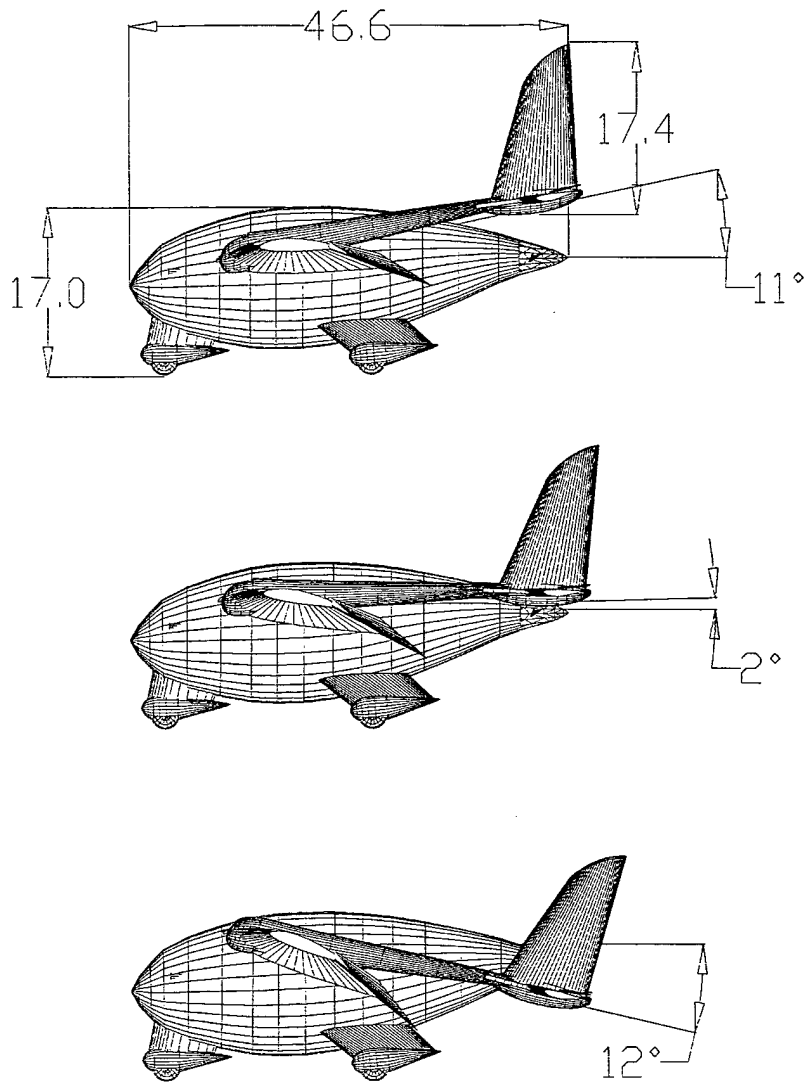
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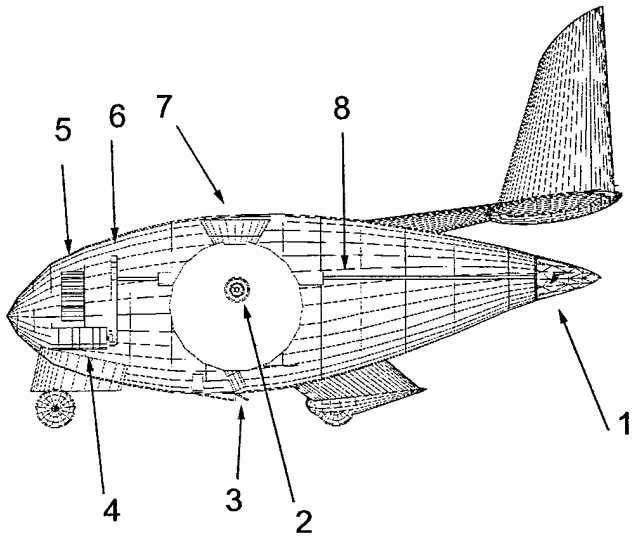


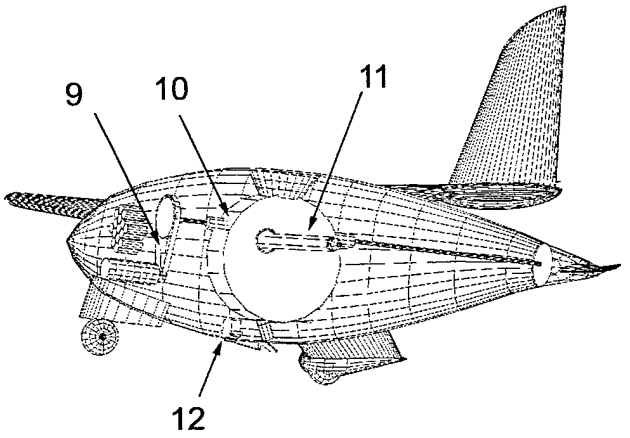
plant

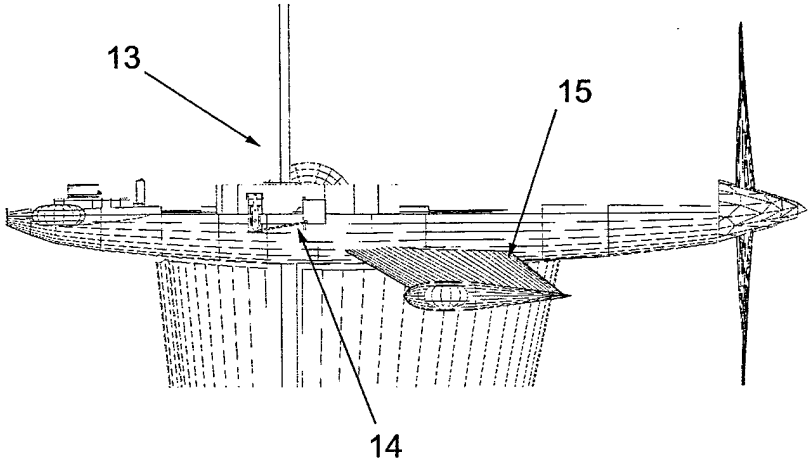
front view



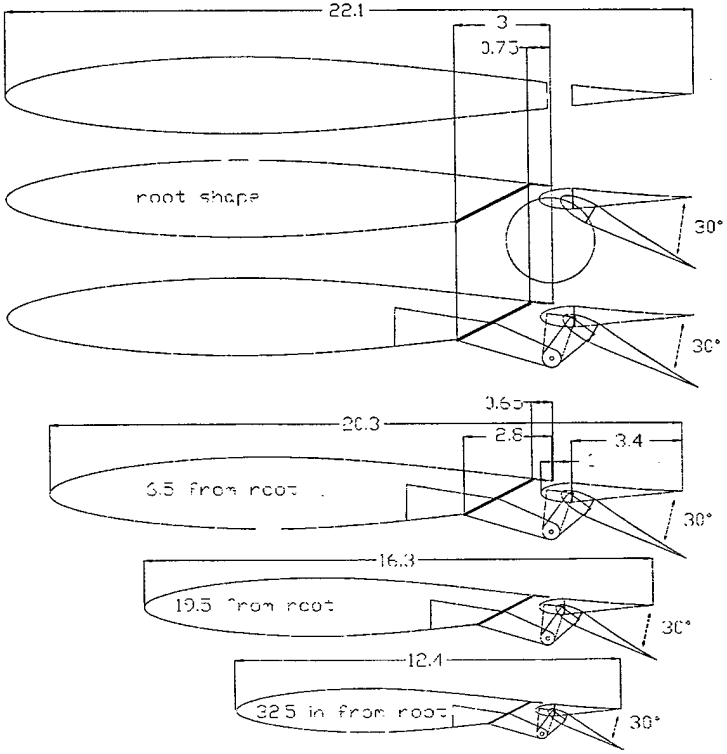
tail excursion range

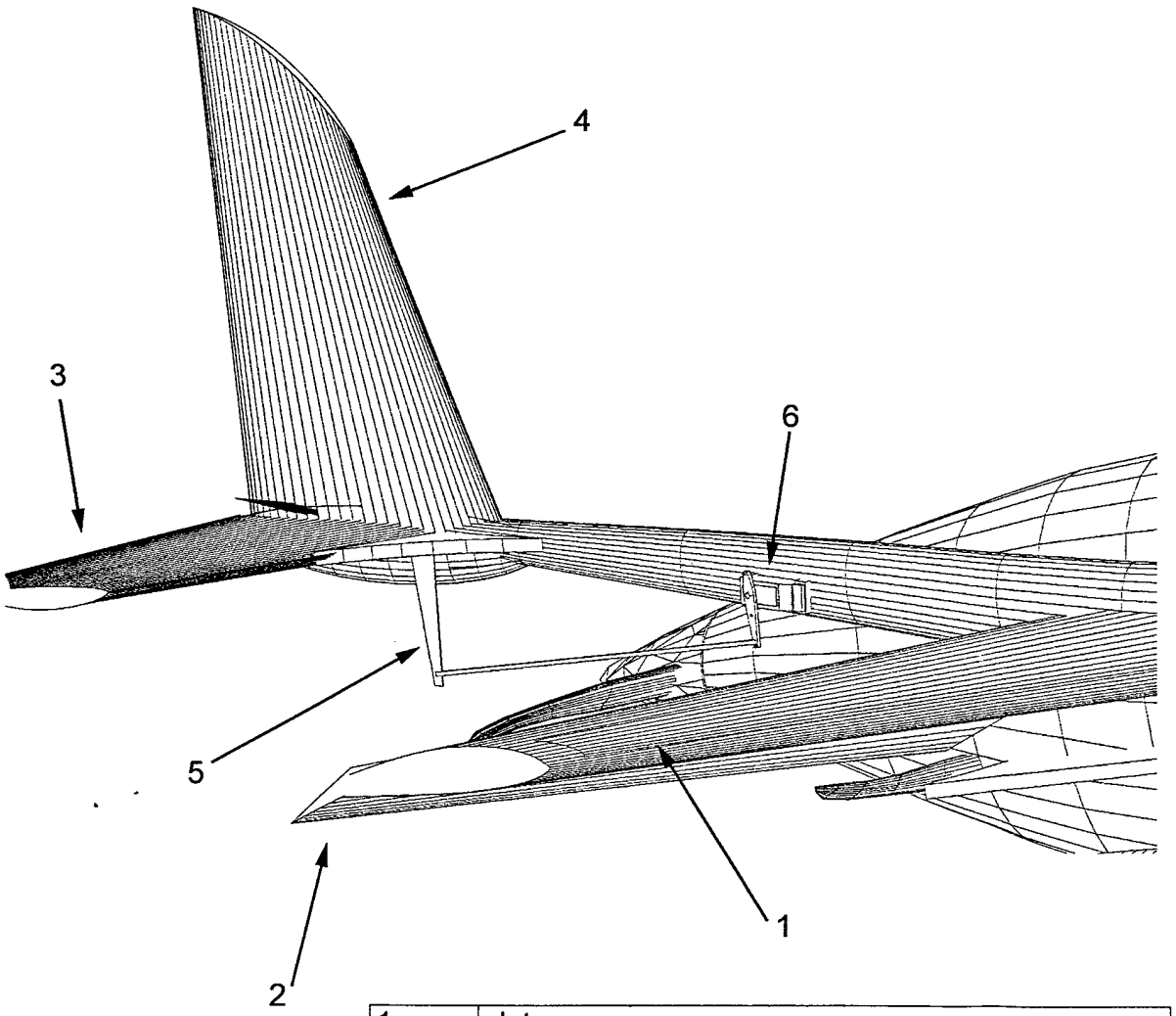




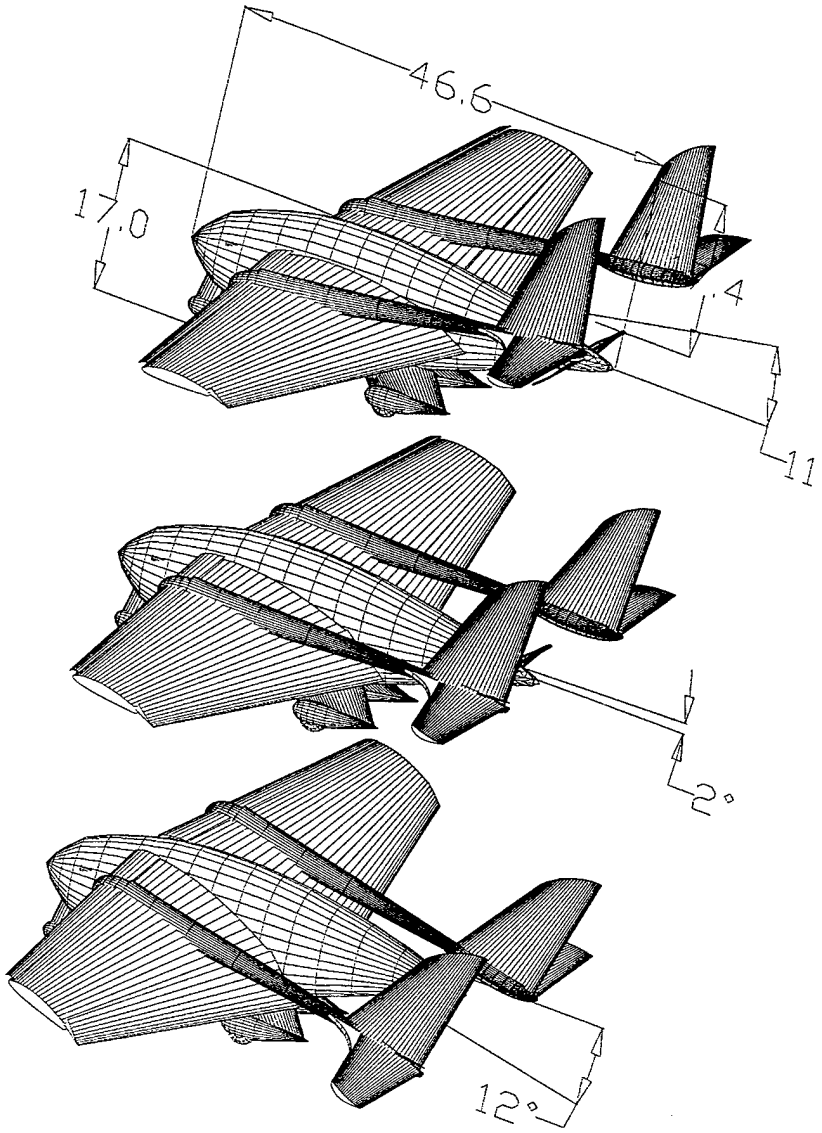


1	propeller & spinner
2	water tank with the pass-through rod hole
3	water damping orifice
4	motor
5	cells
6	gear
7	water intake
8	propeller shaft
9	transmission belt
10	cylindrical water tank
11	wing rod
12	water dump servo
13	wing rod





1	slat
2	flap
3	all mobile elevator
4	rudder
5	elevator brace



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

DESIGN REPORT

ATA-5

ISTANBUL TECHNICAL UNIVERSITY

Table Of Contents

1.0 Executive Summary 1
1.1 Conceptual Design 1
1.1.1 Overview 1
1.1.2 Development Process 1
1.1.3 Conceptual Design Tools 1
1.1.4 Results 2
1.2 Preliminary Design 2
1.2.1 Overview 2
1.2.2 Development Process 2
1.2.3 Preliminary Design Tools 2
1.2.4 Results 3
1.3 Detail Design 3
1.3.1 Overview 3
1.3.2 Development Process 3
1.3.3 Results 3
2.0 Management Summary 4
2.1 Situation Analysis and Work Flow Diagram 4
2.1.1 Situation Analysis 4
2.1.2 Work Flow Diagram 4
2.2 Architecture of the design team 4
2.2.1 Chart of Personal and Assignment Areas 5
2.3 Milestone Chart, Planned and Actual 8
3.0 Conceptual Design 9
3.1 Competition Requirements 9
3.1.1 Aircraft Requirements 9
3.1.2 Mission Requirements 9
3.1.3 Rated Aircraft Cost 10
3.2 Design Parameters 10
3.2.1 Weight 10
3.2.2 Aerodynamic Performance 11
3.2.3 Handling qualities 11
3.2.4 Power 11
3.2.5 Manufacture 11
3.2.6 Strength 11
3.2.7 Payload 11
3.3 Aircraft Configurations 12
3.3.1 Alternative Fuselage and Tail Configurations 12

3.3.2 Alternative Wing Configurations 13
3.3.3 Alternative Structure Configuration 14
3.3.4 Alternative Propulsion System Configurations 15
3.3.5 Alternative Landing Gear Configurations 16
3.4 Design Tools 17
3.5 Final Aircraft Configuration 17
4.0 Preliminary Design 17
4.1 Initial Sizing 18
4.1.1 Weight Calculation 18
4.1.2 Wing Airfoil Selection 19
4.1.3 Wing Geometry 19
4.1.4 Fuselage Shape Assumptions 19
4.1.5 Empennage Assumptions 21
4.2 Propulsion Analysis 21
4.2.1 Power to Weight Ratio 22
4.2.2 Motor-Battery-Propeller Alternatives 22
4.3 Structural Analysis 22
4.3.1 Figures of Merit 22
4.3.2 Design Parameters and Trade Studies 23
4.3.3 Wing Structure 23
4.3.4 Fuselage Structure 24
4.3.5 Tail Structure 25
4.4 Numerical Tools 26
4.4.1 Optimization Program Architecture 26
4.5 Final Aircraft Configurations 26
5.0 Detail Design 26
5.1 Performance Anaylsis 26
5.1.1 Final Configuration Features 26
5.1.2 Estimated Mission Performance 27
5.1.3 Handling Qualities 28
5.2 Propulsive Performance Analysis 28
5.3 Structural Considirations and Analysis 28
5.3.1 Fuselage Structural Consideritions 28
5.3.2 Wing Structural Considirations 29
5.3.3 Tail Structural Considirations 29
5.3.4 Landing Gear 29
5.4 Drawing Package 29
6.0 Manufacturing Plan 30
6.1 Figures of Merit 30
6.1.1 Availability of Materials 30
6.1.2 Skill Level 30

6.1.3 Time 30
6.1.4 Strength to Weight Ratio 30
6.2 Manufacturing Processes 31
6.2.1 Wings 31
6.2.2 Empennage 31
6.2.3 Fuselage 31
7.0 Testing Plan 32
7.1 Test Objectives and Schedules 32
7.2 Summary of Test Results and Lessons Learned 32
References 33

1.0 Executive Summary

ITU undergraduate students have organized to design an unmanned air vehicle that will compete in 2004 Cessna ONR Design Build Fly Competition, a tradition of the leading organization of the sector, AIAA. Every year the mission requirements are updated. By the year 2004 the design must overcome both fire and ferry missions. In order to complete the two missions successfully and to get the highest score, an airplane was designed by ITU ATA-5 Team, as reported below.

1.1 Conceptual Design

1.1.1 Overview

Aim of the conceptual design is to learn the other designers' projects and to be inspired from their ideas. For this purpose a competitor study was made by ITU ATA-5 Team. All team members worked for conceptual design. First of all, mission needs were considered and similar home-built models, fire fighting aircrafts were investigated. The collected database was helpful in having an idea about the dimensions of our aircraft and more significantly it made the alternative concepts become obvious. Then the Figure of Merit (FOM) analyses of the alternative configurations were performed. Some of the promising results began to be more significant. In FOM analysis, Rated Aircraft Cost (RAC), weight, ease of manufacture, strength and performance criteria were considered. During the FOM analysis phase, wing-tail, fuselage, propulsion, landing gears, and the structural alternatives were evaluated in order to carry out Fire Fighting Mission (FFM) and Ferry Mission (FM) accurately. At the end, the most promising concept has been selected.

1.1.2 Development Process

Alternative fuselage-tail arrangements were conventional, bi-wing, canard, flying-wing or twin-boom with conventional or V-type tailed. The desired fuselage shape should have the form and dimensions that make drag force minimum. Wing location options were high, mid or low. Data sheets of producer firms were examined to qualify the best arrangement of motor-battery-propeller. Location, shape, and structure of the landing gears were selected carefully. Tricycle or tail-dragger type landing gears that are made of composite or aluminum were thought. Appropriate tank and dump off mechanisms were studied into make our plane capable of carrying the maximum amount of water during the FFM, dumping it off rapidly in 1000 ft distance and giving effect the ferry mission minimum for increasing the velocity. Throughout specifying the configuration, professional RC aircraft pilots assisted the team to finalize configuration design. Material alternatives like composite, aluminum and wood were considered for the manufacture of components.

1.1.3 Conceptual Design Tools

Design alternatives were optimized for the best flight performance and the lowest Rated Aircraft Cost (RAC) to get highest overall score. The drag was the most important problem to solve and the solution was made by using various software. The dropped payload was another parameter that affects the performance and aerodynamics. FORTRAN codes were written to specify the performance,

aerodynamics, power and control parameters. These parameters were eligible both ferry and fire fight missions so optimization was done.

1.1.4 Results

After the FOM analysis that was carried out for the defined components, the concept of our plane was formed. It was decided to select a high wing configuration fixed to the center of the fuselage. The tail was conventional type. A tractor type motor that can provide the necessary power was selected. Tricycle landing gears were chosen as the most appropriate solution.

Although retractable landing gears have good drag characteristics, they were not considered because of their weight, production difficulties, and RAC. Fuselage, wings, tail, and structural elements were decided to be made of composite. Thus, it was possible to manufacture the fuselage in the desired form and to minimize the form drag.

1.2 Preliminary Design

1.2.1 Overview

In order to make the conceptual design more advanced, aerodynamics, structure and propulsion subgroups that worked in coordination were organized. At this stage of the design, weight estimation, profile selection, wing and tail sizing, selection of engine-battery-propeller combinations, performance calculations, landing gears and tank sizing were carried out.

1.2.2 Development Process

Preliminary design started with the weight estimation calculations that were carried out with the help of the statistical data and reference books like REF. [1] and REF. [2]. Alternate profiles for the wings that are going to carry the weight were selected. By considering the calculated wing area, several combinations of span, chord and wing plan form were determined for design and off design conditions. Motor power that can get over the estimated drag and make the aircraft reach the velocity, which gives the lift required, was examined. Together with the performance calculations, average mission time, in other words the necessary energy amount was determined. Therefore, motor-battery-propeller combination matrix got smaller. As a result of the dimension and weight calculations, the landing gears were formed. The structure group combined the fuselage, wing and tail precisely. Tail and control surface sizing was made by the help of stability calculations. After the formation of the fuselage, the form of the aircraft appeared. Preliminary design sizes became the first step of iteration to find the detailed design sizes.

1.2.3 Preliminary Design Tools

Sketching the three dimensional technical drawing of the model helped the groups guessing possible problems and solving them immediately. All groups advanced their ideas on the CAD program, Catia. The results of preliminary sizing rearranged the limits of design and iteration began with this step. The iterative process was coded in FORTRAN. The data for iteration flow from performance to aerodynamics, aerodynamics to structures, structures to propulsion and

propulsion to performance. During this iteration some other possible values have appeared and at last, the final model was designed to get the highest total score.

1.2.4 Results

The aircraft was designed considering efficient aerodynamic performance, power, control characteristics, and strength criteria to obtain the best score. The exact decisions of sizes, places of propulsion and control systems and construction methods were finalized. The performance group determined the lower values of velocity, required thrust, and radius of turn and time of dropping payload. The aerodynamics group decided the control surfaces dimensions according to static stability and the interval values of wing, tail and fuselage dimensions. The initially considered motor types were Astro Flight brand, Cobalt 60 and Cobalt 90 models. A few alternatives for the batteries and the propeller were determined by Motocalc program [9]. Fuselage (manufactured from composite and supported with bulkheads and longerons), spars, landing gears and water tank formed a strong structure.

1.3 Detail Design

1.3.1 Overview

At this stage of design, it was planned how to manufacture the fuselage, wings, landing gears and structural elements, whose final dimensions were known, and how to combine these components. Final motor-battery-propeller combination was determined. The location and the installation of these components, receivers, velocity controllers, servos, control sticks, and connection cables were planned.

1.3.2 Development Process

As a result of iterations, some values changed .Fuselage, wings, landing gears and water tank dimensions were finalized. Astro Flight Cobalt 90-model motor was chosen. As a result of tests, the most appropriate motor-propeller combination and the battery package with the highest ampere/weight ratio were selected. Performance characteristics of the motor and the batteries were determined. Cooling paths for these components were added to design. Servos needed to move the control surfaces and the water dumping mechanism were selected. The control system that consists of remote control, receiver, and velocity controller were chosen. At the end of this stage, the design and combination of all components and mechanisms were finished. Afterwards, the stage of manufacture and installation began.

1.3.3 Results

Dimensions of all the elements were calculated and the components that are going to be procured from the market were chosen. Before the last stage; manufacturing; technical drawings of all the components were made and the installation techniques were determined. Manufacture planning for the aircraft and for the ones who will work in this stage, was made.

2.0 Management Summary

The Istanbul Technical University ATA-5 project team for this year's competition is composed of twenty five Aeronautical and Aerospace Engineering students, uniformly selected from all years. The organization and design planning was made so that to finish all design, manufacturing and testing steps in most efficient way.

2.1 Situation Analysis and Work Flow Diagram

In order to organize team and make a schedule in most productive way first thing is to determine the workload and team abilities. This requires a work flow diagram on which all preconditions are easy to see. After identifying all works in order, next step is to match the students' abilities to the works. This approach requires to use past years' statistics and to comprehend students' abilities. All must be made in very systematic way.

2.1.1 Situation Analysis

This stage includes listing of both owned and needed design and manufacturing tools and instruments, human power and knowledge sources. The aim is to collect enough information on present condition and needs. All information gathered from this stage will be used at next steps of the design and manufacturing. This procedure is not time limited, it continues till project finishes, however most serious work must be made at the beginning of the project.

2.1.2 Work Flow Diagram

Work Flow Diagram is the ideal chart on which all steps of the project is plotted including iterations. While drawing this chart, current state results gathered from situation analysis is included. Also past years' results, design and manufacturing methods to be used are other important factors affecting the flow.

2.2 Architecture of the design team

With the help of the work flow diagram team members are charged to subtasks according to their abilities and experience. The team was divided into seven sub-groups at design phase and three subgroups at manufacturing phase. For documentation, budget and logistic jobs organizing groups are not needed. Instead, three team members are charged to keep the related tracks. Aerodynamics subgroup also serviced in stability and control calculations. Similarly structural subgroup was also responsible of weight and balance analysis, and water depot design. Beside of the aerodynamic and structural subgroups the power plant subgroup and landing gear subgroup are also formed because of the topics' importance.

Even trough subgroups' tasks are different from each other they are not sharply divided. A subgroup's results affect other groups' work, so the relation among subgroups must be supplied. A group consisting of subgroups' leaders is formed called chief group. This group discusses and evaluates

subgroups results. Design subgroups work through the chief group's decisions. This type of multidisciplinary work helps to reach more optimized results.

Like design subgroups, manufacturing subgroups are organized so that to divide the workload uniformly. These groups are general manufacturing subgroup, electrical components subgroup and landing gear manufacturing subgroup. General manufacturing is responsible of producing wings, tail and body including all structural main parts and water depot. Electrical components subgroup is responsible of settling wires, servo actuators, push roads, batteries, engine and other radio controlling and safety units. According to past years' statistics landing gear is the most problematic part over the craft, so a landing gear manufacturing subgroup is formed, different from last years. Picture 1 represents ATA-5's architecture.

2.2.1 Chart of Personal and Assignment Areas

Chart 1 represents organization of the team meanwhile Table 1 lists all members in order and their assignment areas. As explained before personal list and members' assignment are made after situation analysis and work flow identification. Also table 2 shows how each member of the design team contributed to the completion of the aircraft

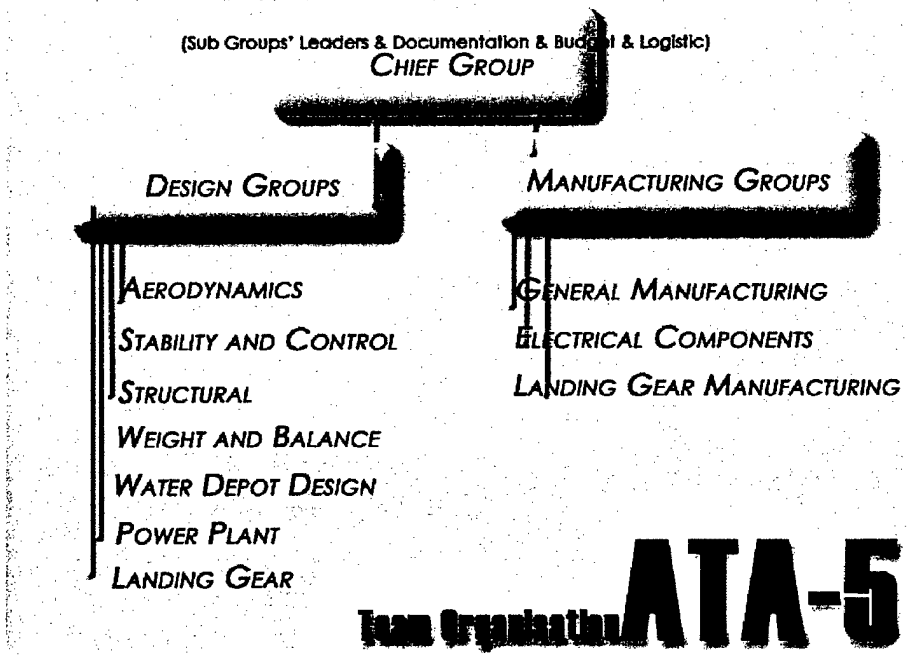


Chart 1: Organization of the ATA-5 Team

Chief Group:

Leader:Gunay KAHYAOGLU
Treasurer: A. Levent SEN
Documentation: Banu YUCEL
Logistic: Gurdal TUGAY
&
Sub-Groups Leader

ATA5

Design Sub-Groups

Manufacturing Sub-Groups

AERODYNAMICS

LEADER: GUNAY KAHYAOLGU

BILIM ATLI

EMRAH OZGUMUS

UGUR SEVILMIS

STABILITY AND CONTROL

LEADER: BILIM ATLI

UGUR SEVILMIS

GURDAL TUGAY

METIN ACAR

TOLGA KURTULUS

LANDING GEAR

LEADER: ERHAN EREN

TEVFIK UYAR

HILAL CENDEK

STRUCTURAL

LEADER: ZAFER OZNALBANT

TAYLAN AKSONGUR

TEVFIK UYAR

MUSTAFA SERDAR TEKCE

MUSTAFA COBAN

MEHMET KARAGOZ

ALPER SERT

WEIGHT AND BALANCE

SAME MEMBERS OF SRUCTURAL GROUP

WATER DEPOT DESIGN

SAME MEMBERS OF SRUCTURAL GROUP

POWER PLANT

LEADER: EMRAH OZGUMUS

UGUR SEVILMIS

MIRAC AKSUGUR

GENERAL MANUFACTURING

ZAFER OZNALBANT

TAYLAN AKSONGUR

MUSTAFA SERDAR TEKCE

MUSTAFA COBAN

MEHMET KARAGOZ

UGUR SEVILMIS

MIRAC AKSUGUR

GURDAL TUGAY

METIN ACAR

LANDING GEAR MANUFACTURING

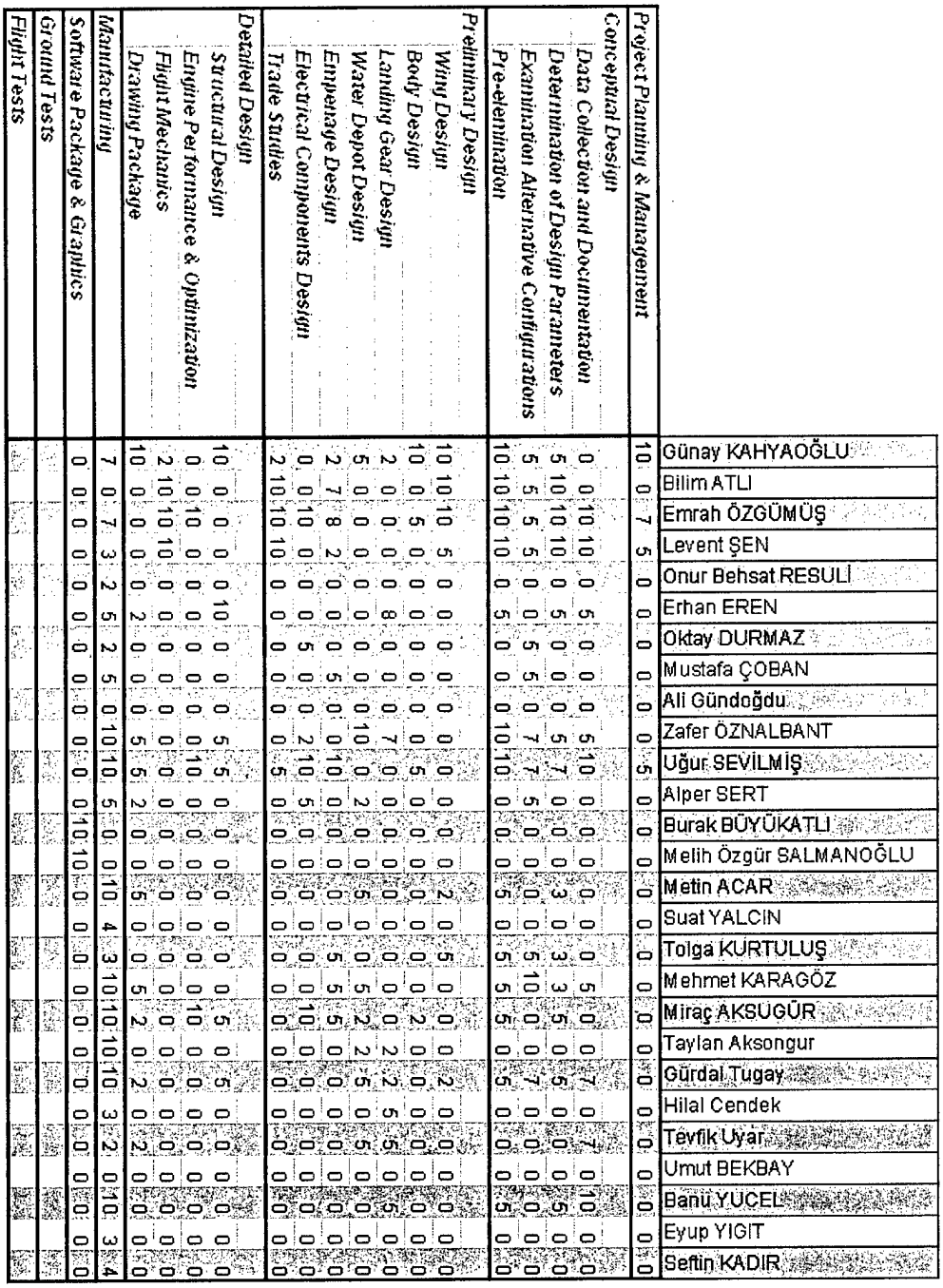
SAME MEMBERS OF SRUCTURAL GROUP

ELECTRICAL COMPONENTS

SAME MEMBERS OF POWER PLANT GROUP

Academic Advisors:
Prof. Dr. Yurdanur K. Tulunay
Prof. Dr. Mehmet Ş. Kavsaoğlu

Table 1: ATA-5 Team members



Project Planning & Management																																
Conceptual Design																																
Data Collection and Documentation	0	0	10	10	0	5	0	0	0	5	10	0	0	0	0	0	0	0	5	0	0	7	0	10	0	0						
Determination of Design Parameters	5	10	10	10	0	5	0	0	0	5	7	0	0	0	0	0	3	0	3	3	5	0	0	0	5	0	0					
Examination Alternative Configurations	5	5	5	5	0	5	0	0	0	7	7	5	0	0	0	0	0	5	10	0	0	7	0	0	0	0	0	0				
Pre-elemination	10	10	10	10	0	5	0	0	0	10	10	0	0	0	0	0	5	0	5	5	5	5	0	0	0	5	0	0	0			
Preliminary Design																																
Wing Design	10	10	10	5	0	0	0	0	0	0	0	0	0	0	0	2	0	5	0	0	0	2	0	0	0	0	0	0	0			
Body Design	10	0	5	0	0	0	0	0	0	5	0	0	0	0	0	0	0	0	0	0	0	2	0	0	0	0	0	0	0	0		
Landing Gear Design	2	0	0	0	0	8	0	0	0	7	0	0	0	0	0	0	0	0	0	0	0	0	2	2	5	5	0	0	0	0		
Water Depot Design	5	0	0	0	0	0	0	0	0	10	0	2	0	0	0	5	0	0	0	0	0	5	2	2	5	0	5	0	0	0	0	
Empenaye Design	2	7	8	2	0	0	0	5	0	0	10	0	0	0	0	0	0	5	5	5	5	0	0	0	0	0	0	0	0	0	0	
Electrical Components Design	0	0	10	0	0	0	5	0	0	2	10	5	0	0	0	0	0	0	0	0	0	10	0	0	0	0	0	0	0	0	0	0
Trade Studies	2	10	10	10	0	0	0	0	0	5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Detailed Design																																
Structural Design	10	0	0	0	0	10	0	0	0	5	5	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Engine Performance & Optimization	0	0	10	0	0	0	0	0	0	10	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Flight Mechanics	2	10	10	10	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0	0
Drawing Package	10	0	0	0	0	2	0	0	0	5	5	2	0	0	0	5	0	0	0	0	0	0	2	0	0	2	0	0	0	0	0	0
Manufacturing	7	0	7	3	2	5	2	5	0	10	10	5	0	0	10	4	3	10	10	10	10	10	3	2	0	10	3	4				
Software Package & Graphics	0	0	0	0	0	0	0	0	0	0	0	0	0	10	10	0	0	0	0	0	0	0	0	0	0	0	0	0				
Ground Tests																																
Flight Tests																																

10	Günay KAHYAOĞLU
0	Bilim ATLI
7	Emrah ÖZGÜMÜŞ
5	Levent ŞEN
0	Onur Behsat RESULİ
0	Erhan EREN
0	Oktay DURMAZ
0	Mustafa ÇOBAN
0	Ali Gündoğdu
0	Zafer ÖZNALBANT
-5	Uğur SEVİLMİŞ
0	Alper SERT
0	Burak BÜYÜKATLI
0	Melih Özgür SALMANOĞLU
0	Metin ACAR
0	Suat YALCIN
0	Tolga KURTULUŞ
0	Mehmet KARAGÖZ
0	Miraç AKSUGÜR
0	Taylan Aksongur
0	Gürdal Tugay
0	Hilal Cendek
0	Tevfik Uyar
0	Umut BEKBAY
0	Banu YUCEL
0	Eyup YIGIT
0	Seftin KADIR

Table 2: Management Summary

2.3 Milestone Chart, Planned and Actual

Chart 2 shows the milestone chart of ITU ATA-5 Team.

Task Name	Start	Finish
Conceptual Design	**Fri 01.08.03**	**Fri 24.10.03**
Data Collection and Documentation	Fri 01.08.03	Fri 12.09.03
Determination of Design Parameters	Mon 08.09.03	Fri 03.10.03
Examination Alternative Configurations	Fri 03.10.03	Fri 24.10.03
Pre-elemination	Sun 19.10.03	Fri 24.10.03
(Actual)Conceptual Design	***Fri 01.08.03***	***Mon 10.11.03***
(A) Data Collection and Documentation	*Fri 01.08.03*	*Fri 26.09.03*
(A)Determination of Design Parameters	*Fri 19.09.03*	*Thu 09.10.03*
(A)Examination Alternative Configurations	*Thu 09.10.03*	*Thu 06.11.03*
(A) Pre-elemination	*Thu 06.11.03*	*Mon 10.11.03*
Preliminary Design	**Fri 24.10.03**	**Thu 11.12.03**
Wing Design	Fri 24.10.03	Fri 14.11.03
Body Design	Fri 31.10.03	Thu 27.11.03
Landing Gear Design	Thu 27.11.03	Thu 11.12.03
Water Depot Design	Fri 31.10.03	Thu 04.12.03
Empenage Design	Thu 27.11.03	Thu 11.12.03
Electrical Components Design	Thu 27.11.03	Thu 04.12.03
Trade Studies	Fri 31.10.03	Thu 04.12.03
(Actual)Preliminary Design	***Mon 10.11.03***	***Mon 12.01.04***
Wing Design	*Mon 10.11.03*	*Mon 24.11.03*
Body Design	*Mon 10.11.03*	*Mon 15.12.03*
Landing Gear Design	*Mon 10.11.03*	*Mon 05.01.04*
Water Depot Design	*Mon 10.11.03*	*Mon 22.12.03*
Empenage Design	*Mon 15.12.03*	*Mon 12.01.04*
Electrical Components Design	*Mon 24.11.03*	*Mon 22.12.03*
Trade Studies	*Mon 17.11.03*	*Fri 26.12.03*
Detailed Design	**Thu 11.12.03**	**Wed 18.02.04**
Structural Design	Thu 11.12.03	Thu 22.01.04
Engine Performance & Optimization	Thu 11.12.03	Thu 12.02.04
Flight Mechanics	Thu 11.12.03	Thu 08.01.04
Drawing Package	Thu 18.12.03	Wed 18.02.04
(Actual)Detailed Design	***Fri 26.12.03***	***Fri 19.03.04***
Structural Design	*Fri 26.12.03*	*Fri 23.01.04*
Engine Performance & Optimization	*Fri 26.12.03*	*Fri 06.02.04*
Flight Mechanics	*Fri 26.12.03*	*Fri 06.02.04*
Drawing Package	*Mon 26.01.04*	*Fri 19.03.04*
Manufacturing	**Thu 18.12.03**	**Mon 08.03.04**
(A) Manufacturing	***Mon 26.01.04***	***Mon 15.03.04***
Ground Tests	Wed 11.02.04	Tue 17.02.04
Ground Tests	*Tue 16.03.04*	*Mon 22.03.04*
Flight Tests	Wed 18.02.04	Tue 16.03.04
Flight Tests	*Tue 23.03.04*	*Mon 29.03.04*
Documentation	Mon 01.12.03	Wed 03.03.04

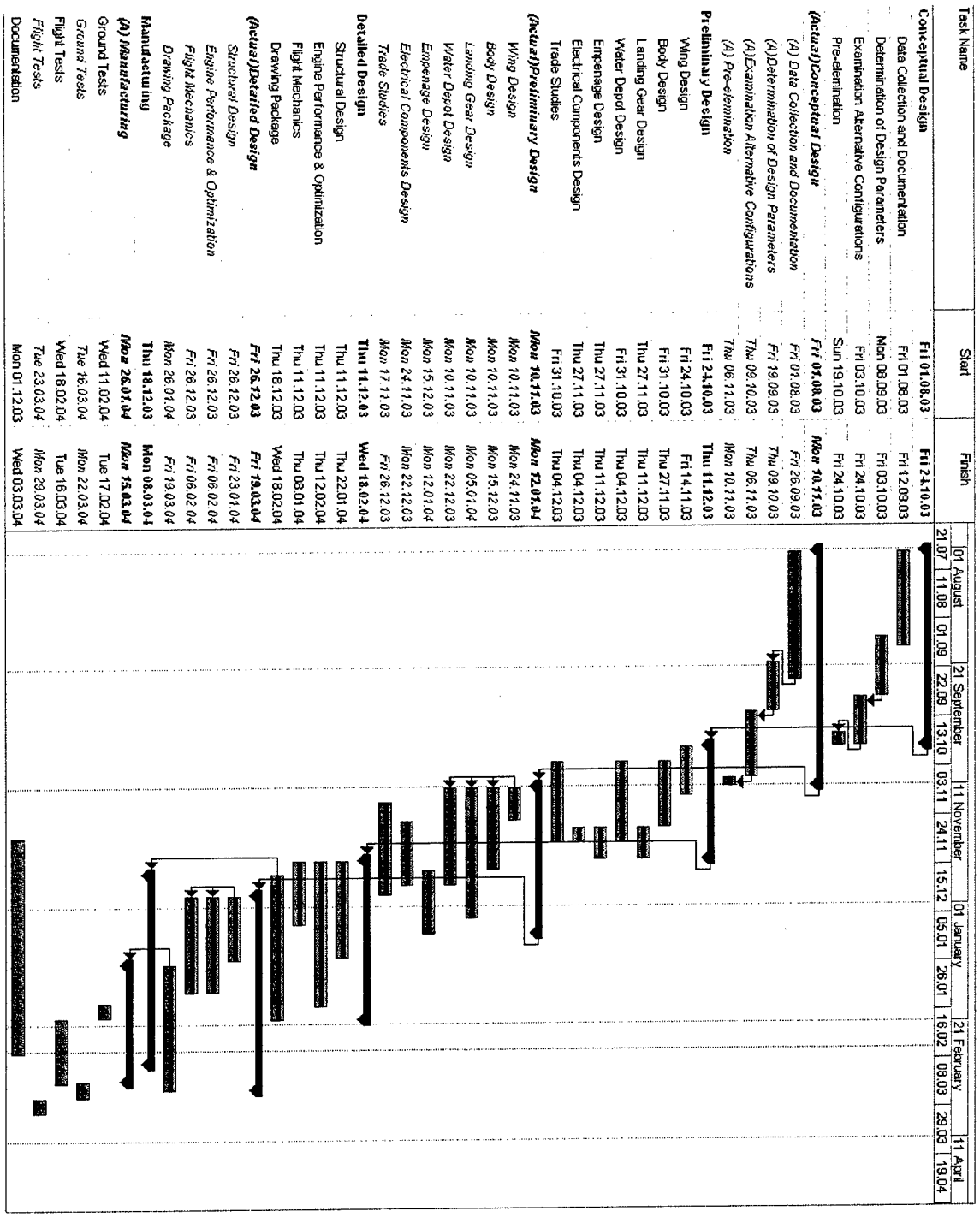


Chart 2: Milestone Chart

3.0 Conceptual Design

The first step of the conceptual design was to analyze the competition requirements. Similar home-built radio controlled aircrafts and the former competitors were investigated, considering the aircraft requirements. This database helped to be produced the alternative configurations. Synchronously with researches, mission requirements and affecting factors of the score were evaluated. At this point, the importance of RAC is cleared. Thus, design parameters of the aircraft, which will complete the missions successfully and get the high score, were determined. After making final ranking chart of each design for each FOM, features that produced the final configuration selected.

3.1 Competition Requirements

3.1.1 Aircraft Requirements

Some parameters, which restrict the design, have been indicated with competition rules. Aircraft requirements point the general configuration (radio controlled, unmanned air veichle except rotary wing or lighter than air), propulsion system (electric powered, propeller driven) and limit battery weight, and total weight of the aircraft. Thus, the matrix of alternatives were narrowed down. Especially the limits about propulsion system were needed to choose the motor-batteries-propeller correctly and they had to work in a good harmony. Another way to overcome limits about propulsion was to design and manufacture an aircraft which has good drag characteristics.

3.1.2 Mission Requirements

This year there are two missions in the competition, the Fire Fighting Mission (FFM) and the Ferry Mission (FM). Since they had opposite characteristics, a perfect optimization was required to get a high total score.

- General Mission Information

Aircraft must fit in a 2-ft wide by 1-ft high by 4-ft long box. This rule limited some sizes of the aircraft such as wing-span and fuselage length. Another rule was about performance: The aircraft wheels must be off the runway within the distance of 150-ft. This was needed to have a sufficient thrust-to-weight ratio and wing-loading. Also all missions must be completed in 10 minutes.

- Fire Fighting Mission (FFM)

At this mission aircraft will carry maximum 4 liters water and dump it off during the downwind leg. The flight mission will consist of two sorties, which have loading, taking-off, dumping and landing phases. Aircraft must complete a single 360-degree turn in the direction opposite of the base and final turns on the downwind leg of each lap. Since water must be emptied in 1000-ft distance, aircraft must fly at enough low velocity in order to dumped the water in sufficient time. Briefly, two important criteria about the FFM are "heavy lift" and "slow flight". Single flight score of FFM will calculated with:

SFS: 2*lbs water/mission time

- Ferry Mission (FM)

A high speed flight will be performed during FM which consist of taking-off, completing four laps, and landing. Aircraft must complete a single 360-degree turn in the direction opposite of the base and final turns on the downwind leg of each lap. Single flight score of FM will calculated with:

SFS: 1/ mission time

It was considered that the higher score got in the FFM, the higher rank got in the competition, due to the multiplier of the FFM is two times the FM. In the past years misssions there was a water carrying mission which we had agreed that the most difficult mission. It is really difficult to carry a liquid payload. Dumpig the water off during flight makes the mission more difficult. After investigating the mission and aircraft requirements, we primarily focused on the question: "how can we design an aircraft that carries maximum water capacity in minimum time".

3.1.3 Rated Aircraft Cost

Overall score will be computed from Written Report Score (WRS), Rated Aircraft Cost (RAC) and Total Flight Score (TFS), which would be summation of the best Single Flight Score from each mission. RAC was more effective on overall score than TFS according to a roughly made calculation. RAC, which was used to estimate the aircraft cost, was a function of empty weight, number of engine, total battery weight, number and sizes of wing, control surfaces type, fuselage sizes and number of servo or motor controller. The most important factor of the RAC is battery weight. Battery weight is both counted in manifacturers empty weight and the rated engine power. An aircraft cannot be designed without the RAC analysis,so a computer program written. Several configurations given as input and the RAC of the configurations calculated. The RAC rating of each component had a dominant effect on the decision.

3.2 Design Parameters

Design parameters were very important over the development of conceptual design. After the competition study, design parameters determined to form an aircraft have a good aerodynamic performance, low empty weight, high power, strong structure, low RAC and carry maximum payload. These parameters, which are connected with each other, were explained separately below. Design parameters would be used to determine the "weight factors" in the FOM analysis in the following stages of conceptual design.

3.2.1 Weight

Results of the historical investigations, known weight range of some components and payload weight gave an opinion about the aircraft total weight. A linear interpolation was made to predict the weight of the airplane according to the historical data. It was predicted that an aircraft carrying 8-9 lbs of payload has 20-25lbs empty weight. Power limitations, taking-off in 150 ft, dumping the water in the required time with low stall speed, reaching high speeds in FM, low RAC and actually making the empty weight less to carry more water in FFM were the reasons to reduce the total weight. Therefore, making the total weight less is especially important design parameter. Using light materials and optimum number of batteries were considered as solution.

3.2.2 Aerodynamic Performance

Wings with high aerodynamic efficiency are necessary to be successful in FFM, which required heavy lift and in FM, which required high speed. A fuselage form that minimized the drag was desired and rough surface that generate turbulence in laminar flow were especially avoided because thrust, which was very valuable, should not be wasted by the drag force. Actually, low drag force would bring low mission time and higher score. Also higher lift and lower drag coeficients *obtains* fewer batteries. An optimization on scoring potential of wing area, span and lift coeficients made. The predicdet wing area was about 8-10ft^2, the wing span was about 7-8 ft and the maximum lift coefficient was estimated to be between 1.2 and 1.4. Manufacturing and external components would be effective on drag.

3.2.3 Handling qualities

For a radio-controlled aircraft, which was not an acrobatic type, stability was desired due to the difficulties about control. Handling of this type of aircrafts was harder especially in bad weather conditions that would happen during competition.

3.2.4 Power

As a result of competitor study and motor manufacturer investigation with considering the conjectural weight of aircraft, a motor would be needed between the power ranges of 1000-1500Watt. There were many motor-battery-propeller combinations. This propulsion system affected the overall score twice. It should produce high flight performance and also have low RAC. Motor and especially batteries were the most effective parameters on RAC.

3.2.5 Manufacture

Being able to manufacture the designed components was another important parameter when limited opportunities were considered. The difficulties in manufacturing prevents us building the aircraft on time. There would be less time to test the manufactured aircraft,which is a big problem.

3.2.6 Strength

Since completing the competition, aircraft had to endure to expected and unexpected loads. High g-loads were exposed during turning and landing, torsion over the wings was some critical design parameters. As a historical experience, crashes during landing frequently happened. Thus, strength of landing gears was especially important.

3.2.7 Payload

The weight of the payload was an important problem. An optimization code was written for the weight of the water. Maximum weight of the water cuold be 8.8 lb. But the assumption of the heavier payload carried the higher score got was not proofed according to our optimization code. According to the program the payload must be 8-8.5 lb for the highest score. The orientation of the payload was another problem. The height of the tank affects the speed of dumping. More faster dumping gives faster flight in the down-wind leg. But the shape of the fuselage and of course the total drag will be

changed according to the orientation of the payload. According to the optimization program written. The height of the tank found to be 5-6 inches for the best score.

3.3 Aircraft Configurations

Aircraft's fuselage, wing, tail, structure, propulsion, landing gear configurations were evaluated according to design parameters and their weight factor. After these FOM analyses, final configuration was decided.

3.3.1 Alternative Fuselage and Tail Configurations

Parameters considered in the Figures of Merit (FOM) analysis of the fuselage and tail configurations are as follows:

RAC: Fuselage dimensions, weight, horizontal and vertical surfaces have effects on RAC. However, only horizontal and vertical surfaces were evaluated because it was very difficult to consider others at this phase. After a roughly made RAC calculation, the weighting factor of RAC was decided.

Drag Characteristics: Drag characteristics have high weight since it was increased the thrust requirement and the mission time. In other words, it would affect the overall score twice.

Handling: Considering probable bad weather conditions, stability and high control efficiency were important for completing the mission maneuvers, take-off and landing successfully.

Ease of Manufacture: RAC contained the ease of manufacture for tail configurations. Also for fuselage, configurations had small advantages or disadvantages. Thus, weight factor of manufacture was considered as low.

After determining the design parameters and the figures of merit for the fuselage-tail, primary configurations selected:

A) Conventional: It had been preferred during the aviation history, so it had well known characteristics. It had relatively good drag and handling qualities.

B) Conventional – V Tail: Complexity of the tail control surfaces could be difficult to handle for a human-controlled aircraft.

C) Bi-wing: Interference of two wings is biggest problem in addition, it has drag, RAC and manufacture disadvantages.

D) Flying Wing: Since there was no horizontal tail and no one more servo, this configuration had a RAC advantage. However, the flying wing is not so good at handling qualities.

E) Canard: Separate two horizontal surfaces causes RAC penalty. The canard configuration have good stall charecteristics, but this configuration reduces the lift due to the downwash at the main wing.

F) Twin boom: It produces easiness for fuselage manufacturing. On the other hand, this fuselage generates more drag due to the discontinuity. Also, the vibrations of the booms make the aircraft bad at handling qualities.

The result of the RAC analysis of configurations can be seen in table 3, decision matrix can be seen in the table 4. The conventional type fuselage got the highest rank.

RAC Analysis of Configurations:

	A	B	C	D	E	F
WBS-3	20	15	20	10	20	15
WBS-4	30	30	30	25	30	30
Total	50	45	50	35	50	45
Addition to RAC (Thousands $)	1	0, 9	1	0, 7	1	0, 9

Table 3: RAC Analysis Decision Matrix.
(WBS Stands for Work Brake Down Structure which is explained in the rules)

RAC was changing about %0, 3 between the best and worst configurations. Thus, a low weight factor was given to RAC.

		Configurations					
Figures of Merit	Weighting Factor	A	B	C	D	E	F
Drag	0,40	1	1	-1	1	0	-1
Handling	0,30	1	0	0	-1	0	-1
RAC	0,25	-1	0	-1	1	-1	0
Manufacture	0,05	0	0	-1	0	0	1
Score		0,45	0,40	-0,70	0,35	-0,25	-0,65

Table 4: Fuselage Decision Matrix

3.3.2 Alternative Wing Configurations

Parameters considered in the Figures of Merit (FOM) analysis of the wing configurations:

Stability: One of the most important contributors to lateral stability is wing location.

Tank Interaction: Spars of the wings which would pass through the fuselage had not to prevent the tank location or water loading-dumping.

Gear Interaction: In an uncontrolled landing, there was a possibility hitting the wing to ground. Keeping the wing high could reduce this risk.

Ease of Manufacture & Assembly: It is also important to have a blended fuselage-wing connection and assemble wing easily.

The selected primary configurations were:

High Wing: It has stability and landing safety advantages.

Mid Wing: Its qualities are between high and low wing configurations. Also, spars might be preventive for water tank.

Low Wing: It is an unstable configuration and most dangerous one for landing. Also spars could be obstacle for the dumping-off mechanism.

The result of the decision matrix can be seen in the Table 5. The high wing got the highest rank.

		Configurations		
Figures of Merit	Weighting Factor	High	Mid	Low
Stability	0,40	1	0	-1
Tank Interaction	0,25	0	-1	-1
Gear Interaction	0,20	1	0	-1
Manufacture	0,15	0	-1	0
Score		0,60	-0,40	-0,85

Table 5: Wing Configuration Decision Matrix

3.3.3 Alternative Structure Configuration

Parameters considered in the Figures of Merit (FOM) analysis of the structure configurations:

Strength-Weight Ratio: It was not important to be strength but also low weight was important for an aircraft structure. High strength to weight ratio was desired, it was heavily weighted.

Formability: Since smooth surfaces were desired for low drag, formability was very important.

Maintenance: There were no opportunity to manufacture an aircraft from the beginning due to damage, aircraft should be repaired easily. Mishaps could happen during flight tests.

Ease of Manufacture: This was the least important parameter because of the abundance of manpower.

RAC: It did not directly affect the structure configuration.

Wing and Tail Structure: Alternatives for wing and tail structures were wood and composites.

Table 6 shows the decision matrix of the materials.

		Configurations		
Figures of Merit	Weighting Factor	Balsa-Wood	Carbon Fiber	Glass-Fiber with Foam
Strength-Weight Ratio	0,4	-1	1	1
Formability	0,3	-1	1	1
Maintenance	0,2	-1	0	0
Ease of Manufacture	0,1	1	-1	0
Score		-0,9	0,6	0,7

Table 6: Structure Decision Matrix

Landing Gear Structure:

The most critical stage of the flight for a radio-controlled aircraft was landing. The biggest structural problem of past years competitors were landing gear damages. Thus, landing gear design was

particularly important. It was desired to withstand about 3g-loads, be flexible to absorb shocks and have a form that produces minimum drag.

Alternatives for landing gear structure were aluminum and composites. There were no difference between their shock endurance but composites were naturally flexible. On the other hand, aluminum required a spring or damper to absorb shocks. These extra components increased the weight and made production harder. Thus it was decided to produce the main landing gear from composite.

3.3.4 Alternative Propulsion System Configurations

Parameters considered in the Figures of Merit (FOM) analysis of the propulsion system configurations:

Power to Weight Ratio: Required power range had been estimated as 1000-1500Watts. One and two motor alternatives at this power range were studied. Their power to weight ratios was calculated. Also the effects of batteries were considered to the weight. Although there were good twin motor alternatives, as a statistical result, single motors had higher ratios than twin motors.

RAC: According to the manufacturer's catalogs, required battery number for single or twin motor, which would produce equal total power, were almost same. However, twin motor had 1.25 times bigger Rated Engine Power Cost.

Safety: In an adversity during landing, it was probable to hit the propeller to ground.

Performance: Location of the motor that affected fuselage sizes was effective on stability and surface drag. Also, torque effect and flow disturbance had a direct effect on aircraft performance.

Ease of Manufacture & Assemble: Extra surfaces would be necessary to mount the motor at the exterior of fuselage and also it might have some problems at the interior.

Alternative motor location, which was at nose, tail or wings, was studied. At the same time, single or twin motor alternatives were evaluated.

Tractor: It was a historical trend that to mount a single engine to the nose. It required less tail surface and increased the stability. Also, cooling the motor was very easy. Another advantage of this configuration was being the propeller in undisturbed air. On the other hand, aircraft flied in a turbulence that was caused by the propeller and this increased the surface drag.

Pusher: Since it allowed the aircraft flying in undisturbed air, the surface drag decreased. Also, getting shorter of the fuselage, wetted area was reduced. The biggest problem was the probability of the propeller to hit ground during take-off and landing and to make longer the landing gear to prevent this. Tail boom should be larger than other configurations. Servo and battery connection could have some difficulties.

On Wings: It was an application of twin motors. Since fuselage and tail did not affect from turbulent flow, it produced low fuselage surface drag and easy control. Also, torque effect was removed by the propellers which rotating opposite sides. In addition to RAC, another disadvantage was being required a cover that mount the motors to the wings.

The result of the decision matrix can be seen in the table 7. The tractor type got the highest rank.

		Configurations		
Figures of Merit	Weighting Factor	Tractor	Pusher	On Wings
Power to Weight Ratio	0,30	1	1	0
RAC	0,25	1	1	-1
Safety	0,20	0	-1	1
Performance	0,15	0	0	1
Manufacture & Assemble	0,10	1	0	-1
Score		0,65	0,35	0

Table 7: Purpulsion Decision Matrix

3.3.5 Alternative Landing Gear Configurations

Parameters considered in the Figures of Merit (FOM) analysis of the landing gear configurations:

Landing performance: Some safety problems made landing more difficult and after touching down, aircrafts' behaviors were different for each configuration. Since safety landing and ground stability were desired, this parameter was heavily weighted.

Take-off performance: Slope of the aircraft when it was preparing to take-off, reduced the ground roll distance and time. Sizes and location of landing gears could allow this slope. It was an important parameter to overcome the rule which was about taking-off at 150-ft.

Drag: The landing gears were main external components generating drag. Thus, a configuration desired that minimize drag.

RAC: Retractable types of landing gears required extra servos that increased the RAC about %1.

Ease of Manufacture & Assemble: Mounting of landing gears to the fuselage and connect them with servos could have some difficulties.

For radio-controlled aircrafts, trends were fixed tricycle or tail-dragger types. Also, retractable tricycle was considered as third alternative.

Tricycle: Since main landing gear was in backside, landing was relatively more easy and safety. The front gear prevented hitting the propeller to the ground. Also, ground stability was high. During cruise, this configuration generated more drag than tail-dragger type due to the gears sizes.

Tail-dragger: Length difference of front and rear gears obtained slope which increased the angle of attack while aircraft was on the ground. Thus, take-off distance and time decreased. In this configuration, rear gear was used to turn the aircraft on ground. Since rear gear and rudder were controlled form the same servo, it was easier to connect them.

Retractable: Although it had a drag advantage, weight, complexity, RAC, difficult manufacture and assembling were the negatives.

Table 8 shows that tricycle combination was better than the other combinations according to our weighted figures of merit.

		Configurations		
Figures of Merit	Weighting Factor	Tricycle	Tail-dragger	Retractable
Landing performance	0,30	1	0	1
Take-off performance	0,15	0	1	0
Drag	0,20	0	0	1
RAC	0,15	0	0	-1
Manufacture & Assemble	0,20	1	1	-1
Score		0.50	0.35	0.10

Table 8: Landing Gear Decision Matrix

3.4 Design Tools

The conceptual design started with a historical investigation form world wide web to have an idea about different concepts and their performance. Also reference books produced some initial values. A computer program written to evaluate the RAC. Several configurations given as input and the RAC of each configurations calculated. By the help of an optimization code, the performance and the RAC was optimized for the probabilitiy of the highest score. Throughout specifying the configuration, professional RC aircraft pilots assisted the team to finalize conceptual design.

3.5 Final Aircraft Configuration

At the end of the conceptual design phase, form of the aircraft, some initial sizes and performance parameters were estimated. These values were shown in table 9. A conventional, high-wing, powered by a tractor motor, composite made, tail-dragger geared aircraft was created.

Empty Weight: 20-25lb Payload Weight: 8-8,8b	Wing Area: 8-10ft^2 Wing Span: 7-8ft
Power Requirement: 1000-1500Watts	C_{Lmax}: 1,2-1,4
V_{stall}: 40-50ft/s	V_{max}:70-85 ft/s

Table 9: Initial Sizes and Values

4.0 Preliminary Design

For the standard engineering design phase preliminary design began as soon as the conceptual configuration was completed. The historical data and the model, formed in the conceptual design, were the beginning criteria. The parameters are the first values for iterations. After finding the aerodynamic and propulsion calculations, the dimensions of wing and fuselage have appeared. As a result the dimension of tail was found, and also the tolerance values. The aim was to narrow down the intervals of dimensions. For this purpose all group leaders come together and narrowed it down.

4.1 Initial Sizing

Once the conceptual configuration and the first estimates of the aircraft made, the performance group made an initial performance calculations to determine weight, wing geometry, airfoil, propulsion, general dimensions and stability of the aircraft.

4.1.1 Weight Calculation

The database created in the conceptual design phase was used for the first estimations of the weight of the aircraft. Evaluated aircrafts that has similar characteristics with our design were investigated for the weight estimations. Every dot in the figure 3 represents an aircraft investigated.Two methods used for calculating the weight. One of the methods was REF. [1] approach and the other was REF. [2] approach. According to the results REF. [2] method was thought to be more feasible. The calculation with REF. [2] approach was shown in figure 3.

1. A linear relation between the logarithm of the empty and total weight was found by the help of similar aircrafts data.
2. The relation was:

   $$\log(W_0) = 1.0439 \times \log(W_e) + 0.0703$$

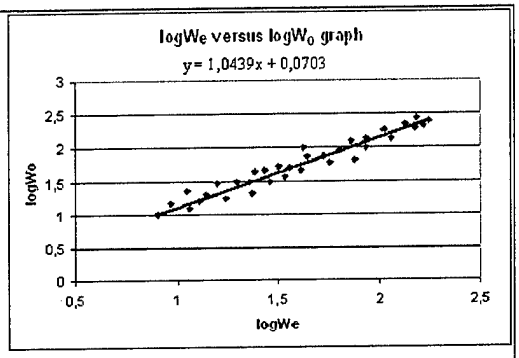
3. W_e values were calculated with the estimated W_0 values for $W_{payload}$=8lb and first log (W_0) column was formed.
4. Second log (W_0) column was calculated with $\log(W_0) = 1.0439 \times \log(W_e) + 0.0703$ relation.
5. Errors between these two log (W_0) columns was calculated and compared.
6. Minimum error occurred for the W_0=31lb
7. Initial weight values:

   W_0 = 31lb

   W_p = 8lb

   W_e = 23lb

Total weight would be tried to reduce and payload would be tried to maximize during the remaining phases of design and manufacture.



$$\log(W_0) = 1.0439 \times \log(W_e) + 0.0703$$

estimated

$\log(W_0)$

Wo(lb)	We(lb)	logWo	logWo	error(%)
25	17	1,3548	1,3979	3,1869
26	18	1,3807	1,4150	2,4839
27	19	1,4052	1,4314	1,8626
28	20	1,4284	1,4472	1,3100
29	21	1,4506	1,4624	0,8158
30	22	1,4717	1,4771	0,3714
31	23	1,4918	1,4914	0,0299
32	24	1,5111	1,5051	0,3939
33	25	1,5296	1,5185	0,7254
34	26	1,5474	1,5315	1,0283
35	27	1,5645	1,5441	1,3060
36	28	1,5810	1,5563	1,5614
37	29	1,5969	1,5682	1,7970
38	30	1,6123	1,5798	2,0146
39	31	1,6271	1,5911	2,2167
40	32	1,6415	1,6021	2,4042

Figure 3: Weight Estimations

4.1.2 Wing Airfoil Selection

From the airfoil database of software lots of laminar airfoils investigated. The first parameter for the airfoil selection was the C_l value. In the conceptual design phase the first estimations about the C_l value was over 1.4. The maximum C_l / C_d ratio was the second parameter for the selection. Investigating the profiles, three options were evaluated. These three promising airfoils were DAE21, SG6043 and Eppler 211 [6,8]. The DAE-21 airfoil showed the best characteristic between the airfoils. Figure 4 shows the polar characteristics of the airfoil DAE-21.

4.1.3 Wing Geometry

Estimating the weight and the airfoil of the aircraft, the wing area and the speed were optimized. To specify the wing area stall velocity had to be calculated. Due to time required for dumping off water 20 seconds, stall speed was selected 45.28 ft/s as a design parameter. Optimized wing area was 9.7 ft^2. The empirical data from the Ref. [1] shows that the aspect ratio of homebuilt was approximately 6. The value 6 was used for further calculations. The span of the wing was calculated as 7.6 ft. In order to arrange the distribution of lift well, a taper ratio of 0.5 was selected by the aerodynamics group. The taper ratio was chosen by the help of the experiences of the model aircraft pilots. It is known that, a wing with a taper ratio of 0.5 has similar characteristics with an ideal elliptic wing. Determining the taper ratio, the root, the tip and the mean chords of the aircraft were calculated. The mean chord was 1.3 ft the design speed was too slow so the sweep angle was designed to be 0^0 at the quarter chord. The historical data provided from REF. [1] guided us to choose the twist angle. Wings were twisted - 3^0 to prevent tip stall and produce approximately an elliptical lift distribution. Best cruise conditions were calculated at an incidence angle of 3^0. 1^0 of dihedral angle were designed to improve the roll stability of the aircraft. Initial Flaperons dimensions were selected from REF. [1]. In order to reduce the tip vortexes wingtip is thought to be used. Due to the windy condition and the RAC, winglets were not considered.

4.1.4 Fuselage Shape Assumptions

Drag is the most important problem for sizing the fuselage. Minimizing the drag provided to carry much water in tank and decreasing the power required for motor. The optimum weight of water was calculated as 8 lbs, so with the shape of fuselage is optimized for connection with wings and tails, shape of tank and most importantly minimum drag. Sections of fuselage are included in drawing part.

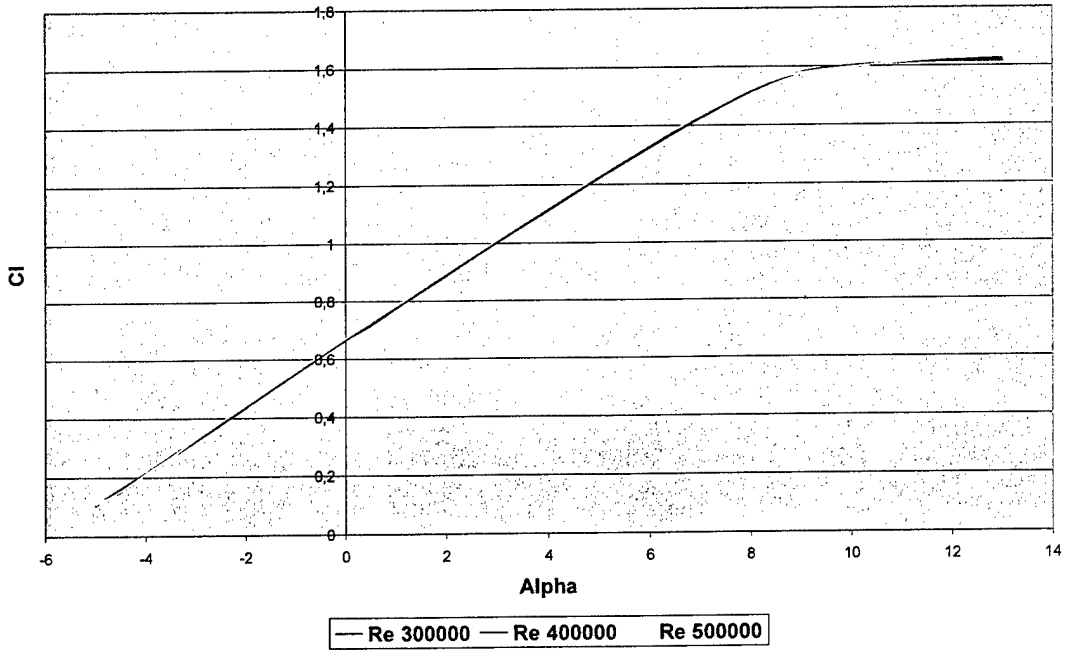


Figure 4a: Cl-Alpha Graph of DAE-21

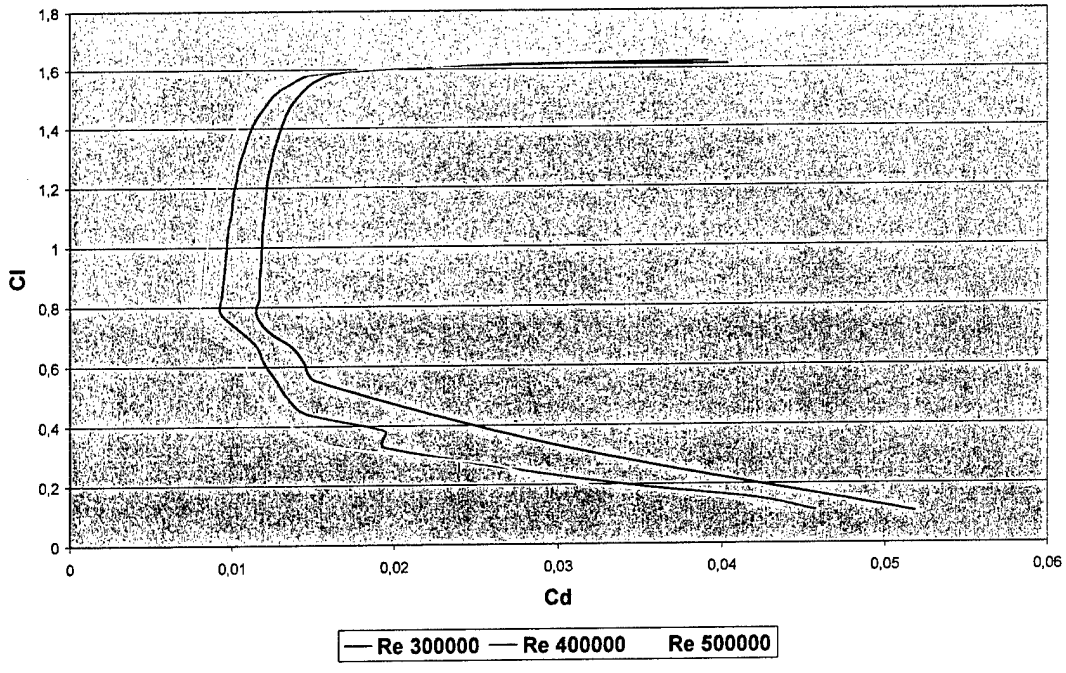


Figure 4b:Cl-Cd Graph of DAE-21

4.1.5 Empennage Assumptions

The first dimensional tail estimations were made according to REF. [1]. The area of horizontal tail was 0.85 ft^2 and the area of the vertical tail was 0.67 ft^2. The static margin was considered to be between 10% and 15%. To satisfy the static margin the horizontal tail area was increased to 1.26 ft^2. The span of the horizontal tail was maximized without exposing RAC penalty. The span of the horizontal tail was considered to be 1.77 ft. The aspect ratio of the horizontal tail was 2.5. The taper ratio of the horizontal tail was 0.5 which was estimated by the help of the REF. [1]. The sweep angle of the horizontal tail was considered to be 7^0 at the quarter chord. The sweep angle of the horizontal tail is bigger than the wing sweep angle in order to control the horizontal tail if the wing stalls. Also the sweep angle of the horizontal tail was selected 30^0 by the help of the REF. [1]. NACA0009 and NACA0009 were the alternatives for the horizontal and vertical tails. NACA0009 was chosen for both because of its good moment characteristics. Incidence angle of the horizontal tail was 0^0 where it generates minimum drag. Location of the horizontal tail was a critical point which affects control of the aircraft. Vertical position of the horizontal tail produces to be able to recover from stall unless elevator enters the wing wake. Thus, horizontal tail was considered to be in line with wing by the help of the REF. [1]. Another point is that, the rudder should not to be in a turbulent wake of horizontal tail when aircraft is spinning. For spin recovery vertical tail was located sufficiently forward with respect to the horizontal tail.

4.2 Propulsion Analysis

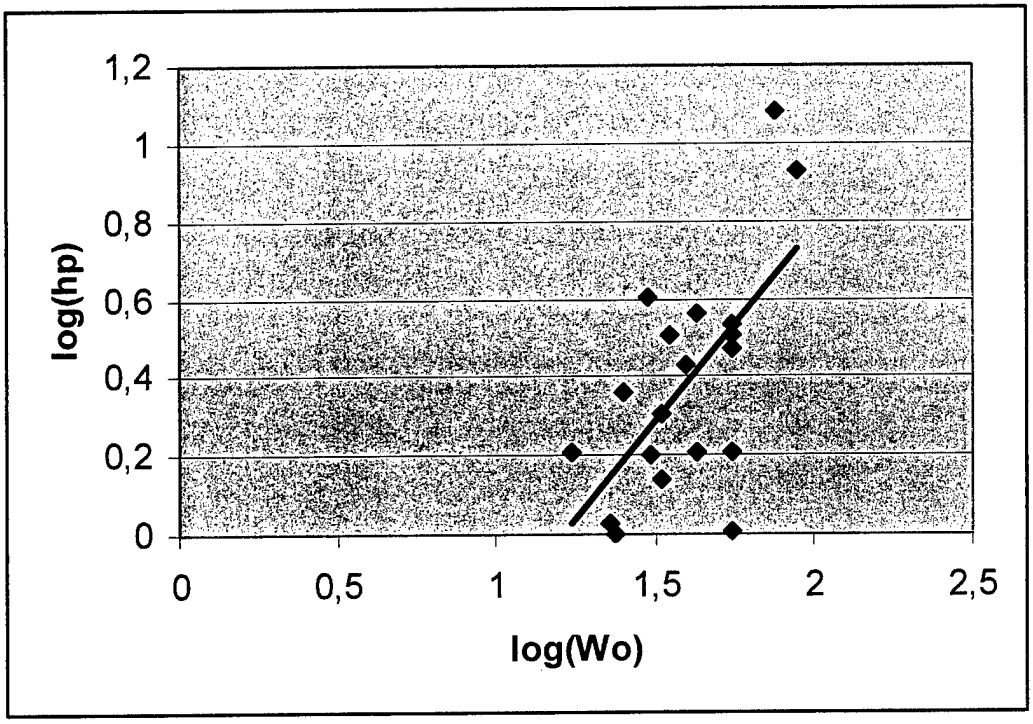


Figure 5: Power to Weight Ratio Estimation

4.2.1 Power to Weight Ratio

Required power was calculated by 3 different ways. Firstly, historical trends of the competitors were evaluated. Results were showed in Fig.5. For estimated weight value of 31lb power to weight ratio was found 0,062 according to this graphic.

REF. [1] gave a statistical data as second approach. Similar aircraft's power to weight ratio changes between 0,04-0,08. According to the third method, which was empirical formulas in REF. [1], power to weight ratio was calculated 0,048. As a result, required power estimated over the 1200Watt.

4.2.2 Motor-Battery-Propeller Alternatives

During conceptual design phase, using a single motor had been decided. Motor alternatives, which would generate 1200 Watt, were direct drive or geared type, 660 or 690 models of Astro Flight. However, geared motors require bigger propeller which increases the drag and landing gear lenghts. Also their gear boxes have mechanical losses of about 8%-12% and they increase the weight. Thus, geared type motors were eliminated.

Remaining motors could be used with propellers which has 14, 15 or 16inch diameter. Alternatives for propeller material were wood, nylon and composite. A composite made propeller would be used due to its strength and high performance.

Also number of batteries was another parameter in order to produce more power. Only battery weight had an effect on Rated Engine Power part of RAC. Thus, small amount of light batteries were considered to use. These motors required at least 20 batteries to work. 660 motors produce maximum power with 30 batteries and 690 motors produce maximum power with 40 batteries. It means that, one battery's weight must be at the range of 2-2,6oz because of the 5lb total battery weight limitation. PANASONIC or SANYO batteries are general trends for competitors and professional RC pilots. We took the data sheets of the batteries from their producer and the values were graphed using excel to help us predicting the duration of the batteries. Their all products were investigated and SANYO's 2400RC, 2300SCE, and 2400SCR batteries were considered as alternatives due to their high Capacity to Weight ratio.

4.3 Structural Analysis

After determinig the dimensions of the wing, fuselage and the tail the structural analysis has began. The landing gear dimensions was determined in the structural analysis phase.

4.3.1 Figures of Merit

In structural design several figures of merit were used for improving the high score potential of the aircraft. The figures of merit in the structural analysis included:

- Weight: The weight of each component was very important for the aircraft performance and the RAC.
- Ease of construction: The ease of construction was another important point because of the limited materials and manufacturing skills of the construction members.

- Aerodynamic considerations: The aerodynamic forces and moments acting on the components and the drag of the landing gear were another important point for the structural analysis.
- Cost: The costs of every material were also important because of the economic problems having the desired material.

4.3.2 Design Parameters and Trade Studies

The design parameters of the structural analysis were:

- Material selection
- Weight
- Number of spars and their shapes of the tail an the wing
- Landing gear dimensions
- The shell thickness of the wing, tail and the fuselage

In the structural analysis these design parameters were used to optimize the structural components of the aircraft. Feasibility of the materials for the structural components, the weight vs. strength trades, and the trade studies made in the other alternatives of the structural components were investigated. For the ease of the investigation, aerodynamic forces and moments acting on the wing were considered as the wing geometry was elliptical.

4.3.3 Wing Structure

The structural analyze began with the wing analyze. Knowing that the lift would act on to the wing, it was very important to decide the connection of the wing with the body. Also the wings strength must be enough in order to prevent its deforming. The wings were thought to produce from foam core and composite cover. Software was used for determining the type of the composite and the number of the layers. Analyzing the structure of the wings, the distribution of the forces was assumed as the wings were elliptical. This assumption made analyze easily.

Three alternatives were considered for the spars. These three promising alternatives were an I beam, two C beams and a tube at the quarter chord.

The I beam was considered to built up with composite and the location of the I beam were thought to be at the quarter chord of the wing. Some other joints must be added because one spar would not be enough to carry all the loads and moments. If the material would be fiber glass, it might be too heavy to satisfy the desired strength. These disadvantages let us not to built the I beam.

The second alternative was two C beams. The advantage of two C beams was there would not be any other joint to carry the wings. One of the C beams would be at the behind of the quarter chord and the other would be at the ahead of the quarter chord. But the analyze pointed us that the weight of the C beams would be more than the I beam, and the beams could not be oriented at the inner points of the wing due to the thickness of the wings.

The cylindrical tubes were considered to be built up with composite or aluminum. To satisfy the desired strength, the composite must be carbon fiber. The difficulties of forming a cylindrical tube with carbon fiber were the biggest disadvantage of the composite material. The aluminum was considered for the material of the tube. The calculations show that 0.8 inch inner diameter and 0.9 inch out diameter tube was enough for carrying a 4 g load. The diameter of the spar is small enough to be

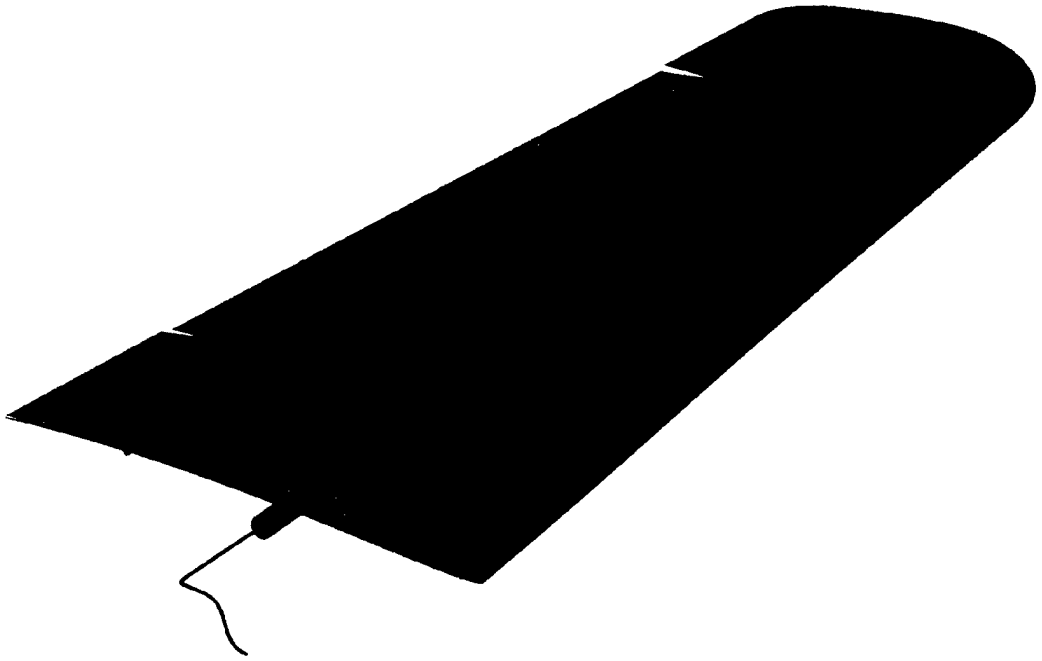


Figure 7: Considered Spar of the Wing

oriented at the inner parts of the wings. The weight of the spar was in the acceptable limits. Another advantage of the tube is the wires of the control cables would be put in the tube as seen in figure 7.

4.3.4 Fuselage Structure

In analyze of the fuselage, aerodynamic considerations were very important. Knowing that the drag force must be reduced, the planned shape of the fuselage has not changed. To satisfy the required strength two material options were compared. One of them was carbon fiber sheet, and the other was glass fiber sheet.

The strength of the carbon fiber was enough to carry all the loads acting on the fuselage. However manufacturing carbon fiber structure was very difficult. The connection of wing and tail requires some other structural elements. With the structural elements the carbon fibers become heavier.

Based on experience, the strength of the glass fiber was not enough to carry the loads acting on the fuselage. On the other hand, it was easier to manufacture. Bulkheads and longerons were planned to satisfy the required strength. The bulkheads were oriented in order to make the connection of the primary components easier.

Because of the dimensional limitations of the rules, the fuselage must be produced in two pieces. The connections of the pieces were considered to achieve with bots on the longerons. This combination was less heavy than the carbon fiber structure.

4.3.5 Tail Structure

Due to the forces acting on the tail is less than the forces acting on the wing the structure analysis were easier. The calculations pointed that the foam core with one sheet of fiber glass cover had enough strength to beat the forces and the moments. The connection of the horizontal tail to the fuselage was achieved with two joints. These were two bolts as seen in figure 8. The elevator was thought to be cut from the produced tail. To satisfy the desired strength, balsa woods were fixed to the surfaces of the elevator and coupling tail surfaces.

Likewise the horizontal tail the vertical tail were planned to be built up with foam core and fiber glass composite structure. The connections of the vertical tail were thought to be achieved with fixing the vertical tail to the bulkheads in the fuselage.

A cage structure were planned to connect the horizontal and vertical tail. The cage structure was considered to be built up with wood, which satisfies the required stress. The cage structure would let the connection easier.

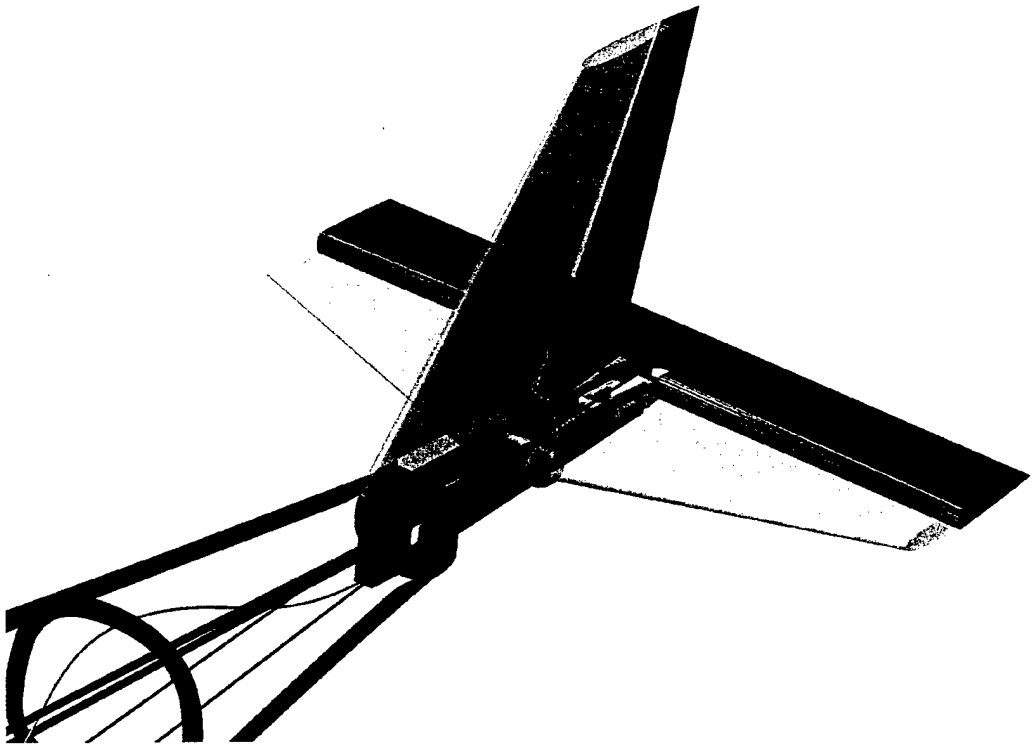


Figure 8: The Connection of the Empennage to the Fuselage

4.4 Numerical Tools

4.4.1 Optimization Program Architecture

After the conceptual design the data for the aircraft was optimized by the performance group. Every group gave the values and every group gets the required data from performance group. To solve optimization problems FORTRAN and Excel was used. The optimization was used at every steps of design process. The carpet graphs are used to determine intervals and the promising solutions is calculated in FORTRAN. The optimization began between the ferry and fire fighting missions and had an end by sizing the tail. During RAC and performance calculations optimization was done. After every optimization process the calculations were done not to make mistake.

4.5 Final Aircraft Configurations

Final configuration results are included in table 11 as:

Wing Span	7.12 ft
Wing Tip Chord	0.85 ft
Wing Root Chord	1,7 ft
Tail Area	8.6 ft^2
Propulsion Alternatives	Cobalt-60, Cobalt-90
Material Aircraft	Composite
Wing Loading	3.6 lb/ft^2

Table 11: Final Aircraft Consideration

5.0 Detail Design

After sizing the componenets in the pleriminary design phase, the performance group determined stability, performance and the handling qualities of the arcraft. Manufacturing process and detailied drawings for the manufacturing of the components developed by the stuctures group. The propulsion grup determined the final motor, baterry and the propoller combination that were going to installed to the aircraft.

5.1 Performance Anaylsis

After determining the final configuration features, performance group used a code to analyse the estimated mission performance, handling qualities and the purpulsive performance.

5.1.1 Final Configuration Features

The final aircraft was a conventional, blended high wing with a medium aspec ratio about 6. The fuselage was designed to cover all the components reduce low drag. There was a convantional tail, the area of the horizantal tail was 1.26 ft^2 . which is out of the RAC penalty. The landing gear was 1 ft tall. More detail is shown in table 12. Altought the aircraft have not builted yet, the RAC was estimated as 19,300.

Fuselage

Lenght	5.39 ft
Maximum widht	1.21 ft

Tail properties

	Horizontal	Vertical
Airfoil	NACA 0009	NACA 009
Span	1.77 ft	0.98 ft
Chord at root	0.95 ft	0.95 ft
Chort at tip	0.52 ft	0.39 ft
Area	1.26 ft^2	0.67 ft^2
Taper ratio	0.5	0.4

Wing

Airfoil	DAE-21
Span	7.12 ft
Chord (mean)	1.3 ft
Area	8.6 ft^2
İcidence angle	3^0
Taper Ratio	0.5
Flaperon area	0.5 ft^2

Total Aircraft

Maximum Weight	31 lb
Payload Weight	8 lb
RAC	19,300

Table 12 : Final Aircraft Configuration Features

5.1.2 Estimated Mission Performance

A performance code were used for optimizing the estimated mission performance of the configuration. The time needed for each step were calculated. Table 13 shows the estimated time needed for each step. Total time spent for Ferry Mission and Fire Fighting Mission wa s estimated as 169 second and 256 second.

Mission Steps	Time Spent	Distance	Velocity
Take off unloaded	3.4 sec	42 ft	62 fps
Climb unloaded	11.3 sec	-	81 fps
Cruise unloaded (all laps)	154 sec	-	93 fps
Ground time for loadind	23 sec	-	-
Take off loaded	8.6 sec	98 ft	70 fps
Climb loaded	24 sec	-	76 fps
Dumping off (8 lb of water)	18 sec	-	56 fps
Cruise loaded (both laps)	182 sec	-	76 fps

Table 13 : Estimated time spent in mission steps.

5.1.3 Handling Qualities

Dumping off payload made the handling qualities very important. After determining the final dimensions of the aircraft the handling qualities were calculated and trimmed. Any unwanted behaviour of the aircraft would cause the aircraft crash. Due to the importance of the handling qualities static margin of the aircraft were calculated 13%. The neutral point was calculated and the center of the gravity selected to statisfy the stability. Arranging the center of gravity point were tought to success by replacing the batterries. The final location of the tank was choosen after the prediction of center of gravity, for fitting the CG of the aircraft and the CG of the full tank. The control power was calculated for the rudder and elevator.

5.2 Propulsive Performance Analysis

The thrust needed for the aircraft were calculated by the propulsion group. A software was used for estimating the propulsive performance of the aircraft. The power needed were calculated as 1.55 hp. Determined final motor was Astro Flight Cobalt 90 which is the only motor that satisties the needed thrust. Gear box was determined as unneccessary for producing the needed thrust. The power-loss handicap of gear box were prevented in this way. To satisty the best performance a 16x10 propeller was determined. The batteries were choosen as SANYO 2400SCR.

5.3 Structural Considirations and Analysis

In the deail design phase, the integration and the structural analysis of the components were determined. A software were used in the structural analysis. Mainly four major aircraft structures were considered, the fuselage, the wing, the tail and the landing gear.

5.3.1 Fuselage Structural Consideritions

The fuselage were formed to cover the inner components (payload, batteries, motor...ect.), as possible as to reduce the drag. In order to reduce the drag force, the fuselage was considered to built as a fish shaped shell. Due to the shell structure of the fuselage, bulkheads and longerons were added as structural elements. About all of the stresses acts on the fuselage were considered to be carried by the bulkheads and longerons. The bulkheads were very well oriented in order to provide the connections of the compenents easier. The orientation of the bulkheads also provide carriying forces acting on the other major components of the aircraft. All the fuselage, bulkheads and longerons were considered to built up with compisite. Due to the dimensional limitations of the rules, the fuselage mus be in two parts. Theese parts were considered to be connected by th longerons. The water tank is tought to be fixed in the fuselage. The tank also tought to built up with composite. In order to load the water tank and removing the batteries a hatch was considered to built on top of the airplane. There is a mechanism on the hatch allows to load tha tank easily with a sliding motion. Another hatch was considered to built for removing the empenage. The fuse is placed on to the airplane in order to reach and remove easily.

5.3.2 Wing Structural Considirations

The wing structure was cosisted of foam core and composite covering. The high density foam core holds the airfoil shape and the composite cover There was a cylindrical spar to cary the aerodynamic forces and moments acting on the wing. İn order to fix the wing there were two joints. The joints were planned to slide in a deck and it could be fixed at any desired position. Cylindrical spar and sliding joints allow us to change the incidence angle of the wings. Wooden ribs were used for the connection of body and thewing. The servos acting to flaperons were digged in the wing in order to reduce the drag. The connection cables of the servos were conected to the body in the cylindrical tube spars. Composite rods used for the hinge mechanism for the flaperon and the wing.The hinge mechanism between the flaperon and the wing is designed to minimise the gap. The structural problems between the flaperon and the wing lead us to use wooden plates for the sides.

5.3.3 Tail Structural Considirations

The tail structure is same as the wing structure. Both vertical and horizontal tails were made up with foam core with composite cover. The vertical tail was fixed in the tail bulkheads. The horizantal tail was connected with a spar to the fuselage. All the push rods and the servos were mounted in the body to minimize the drag force. There was a hatch at the bottom of the fuselage that provides to remove the tail and reach the servos and the push rods easily.

5.3.4 Landing Gear

The orientation of the landing gear were determined before. The main landing gear was carriying 85 percent of the aircraft weight, and it must be 1 ft high because of the safe of the propoller. Difficulties for producing a landing gear led us to buy a langing gear. The landing gear was fixed to one of the bulkheads. For the transmission of the stresses a wooden plate was used.

5.4 Drawing Package

Drawing package is in the Last five pages of this document.

6.0 Manufacturing Plan

In manufacturing phase a plan is required to be able to produce the craft on time in sufficient strength and cost. As we done before, while organizing the team, in this step we again analyzed the situation and draw a manufacture flow diagram. This provided as required design parameters under general name "Figures of Merit".

6.1 Figures of Merit

6.1.1 Availability of Materials

All materials for construction are available except very low density glass-fiber. Because we will not use so much low-density, material availability is not a figure of merit (FOM). Not the material but the machine availability is a FOM. Every time it is not possible to find the needed machine to produce some specific parts, such as body and landing gears.

6.1.2 Skill Level

From past years' DBF competitions sufficient experience on composites was obtained. Hand up composite material process increases the error constants. Limiting errors in a sufficient bound is mostly dependent to the workers.

Metal and wood forming experience is still poor.

6.1.3 Time

All materials chemical properties and their process types related to time are different. While processing the materials these differences are took in consideration.

6.1.4 Strength to Weight Ratio

Over the plane different locations require different strengths. Wood, composites, metals and others have different strength values.

The balsa wood is very light but the balsa is not strength enough.

Similarly metallic parts can be very strength but it is heavy.

Composite materials have a special condition. Controlling the layer size and using different materials in layers can serve very good strength results. Besides, investigating the strength to weight ratios, it is again possible to say that composite solutions are best. Processing the composites' fiber directions in proper way decreases the weight to sufficient levels. Mainly two type of material is used in the layers. One is carbon, and other is glass hardened in a chemical like epoxy. Carbon-fiber has better strength to weight ratio but also has bigger density value. In other side glass-fiber can meet more deflection.

Foam and other non-structural materials are only preferred to fill a thin-wall shell parts. They are light and only used to form a part, like wing, empennage.

6.2 Manufacturing Processes

6.2.1 Wings

Glass-fiber composite laminated over wing-formed foam is the approach. Controlling the layer number allow to control the strength. Wing formed foam is produced by hot-wire process. Another approach is to use structural elements and apply fewer layers over foam. To find which approach is better, finite elements methods and experiments can be used. We used the second approach because pipe type structural spar is also good assembly element.

6.2.2 Empennage

Same approaches for wings are discussed for tails. Because forces are not so much high, non structural form is enough for empennage. Also this type of tail will allow easy montage.

6.2.3 Fuselage

Glass-fiber shell, supported with structural units like bulkheads and ribs is our selection. Carbon-fiber is not used because of its process difficulties. This selection has high strength to weight ratio. The body will be produced using a mold. Using mold increases the time and costs but helps to produce lighter and more aerodynamic models.

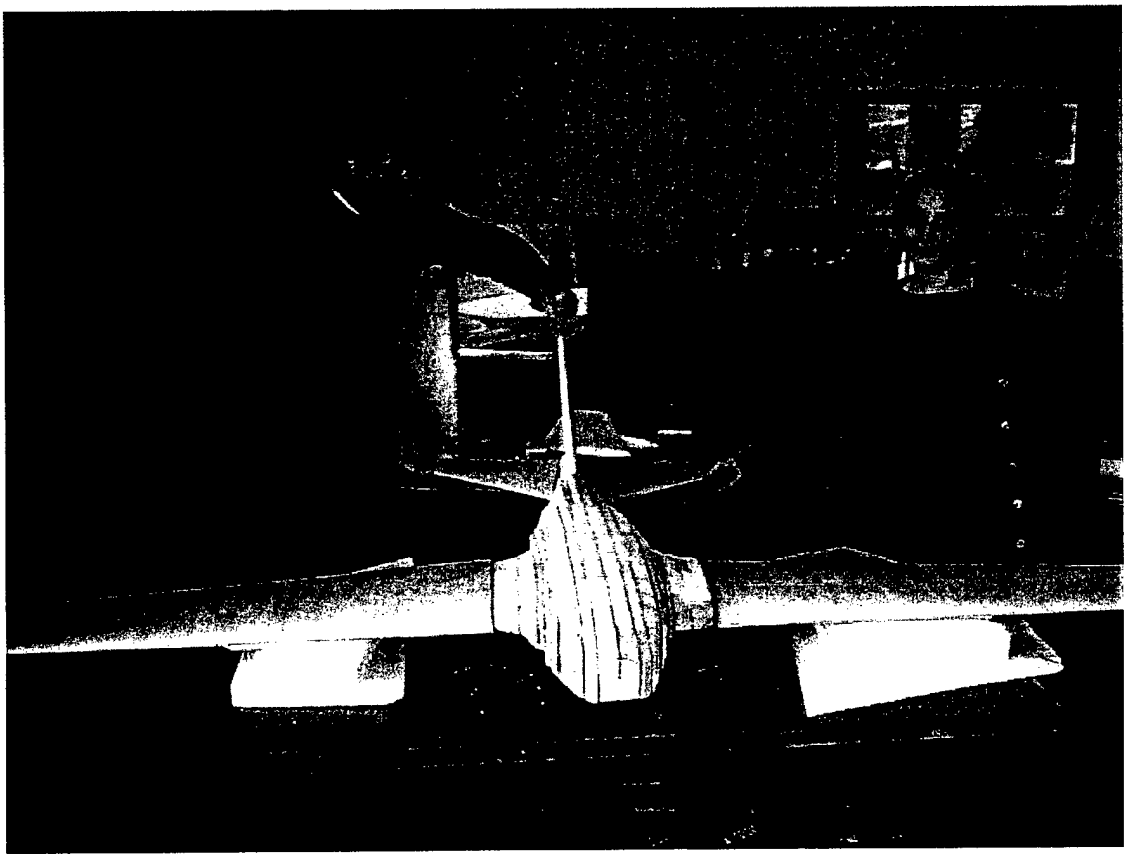


Figure 9: The Mold of the Fuselage

7.0 Testing Plan

7.1 Test Objectives and Schedules

The main purpose of the both ground and flight tests are obtaining the real values and characteristic of the craft. Gathered feedbacks will allow optimizing the plane.

In the case of flight mechanics, locations of the plane's center of gravity, horizontal and vertical stabilizers, and water depot are thought to be control for optimization. Meanwhile the engine's and other electrical components' dynamic and static characteristics will be investigated. Also structural samples will be made for both prediction of the weight and integration to numerical application.

The following chart shows schedule planned for tests.

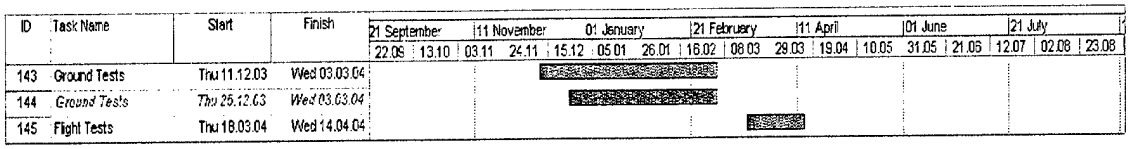


Chart 3: Test Schedule Chart

7.2 Summary of Test Results and Lessons Learned

All tests are not finished yet. However till the point we reached it is possible to say that, predictions and acceptations used in design process are still weak. It is also true that numerical and analytic methods and tools give better results comparing to the past years' calculations, but not enough. Experimental data must be integrated to the above-mentioned methods for more realistic consequences.

Also manufacturing quality is increasing comparing the past years records. However, improvements on processing and material science are necessary for lighter solutions.

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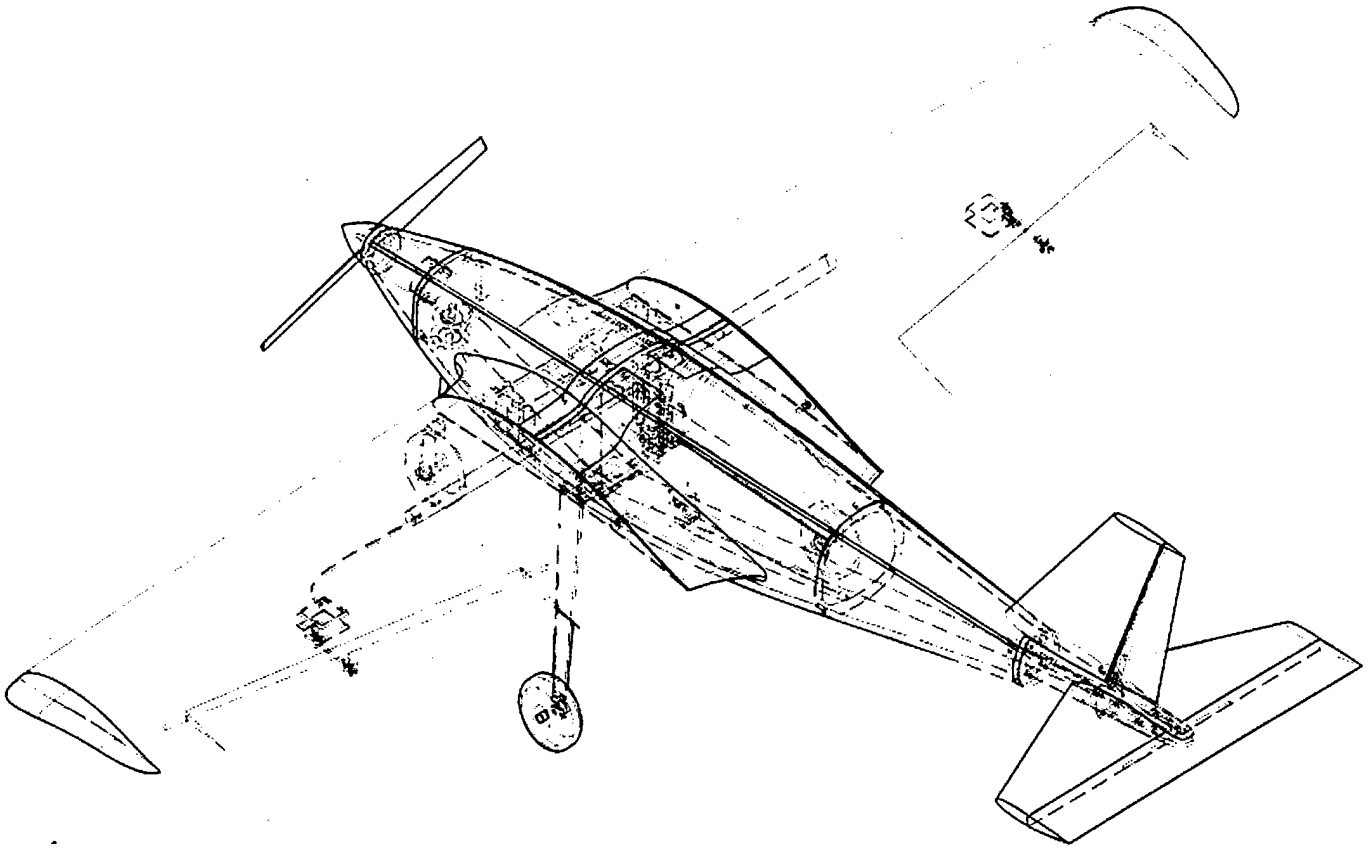
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Technical Advisor: Prof. Dr. Mehmet S KAYSAOGLU

Team Leader: Cuneyt KAHYACOLU

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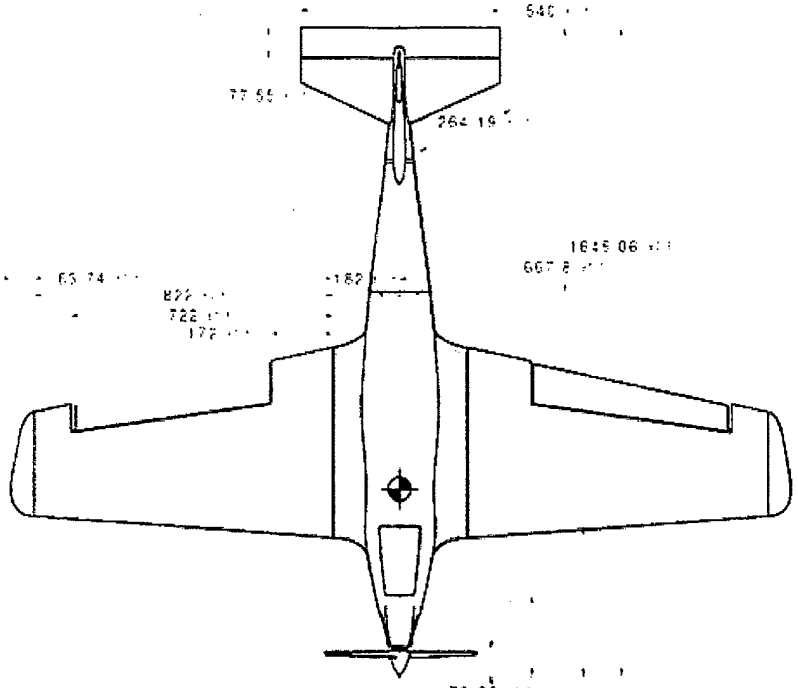
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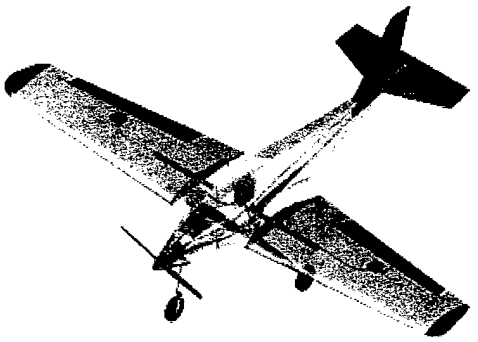
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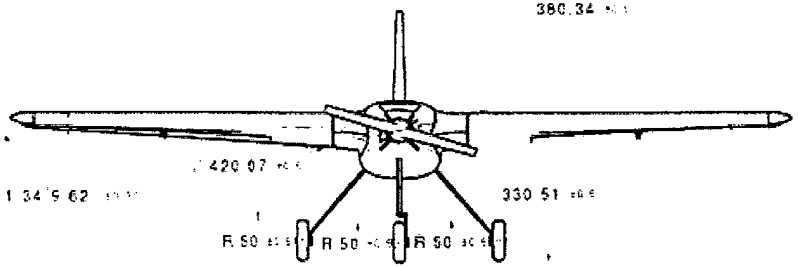
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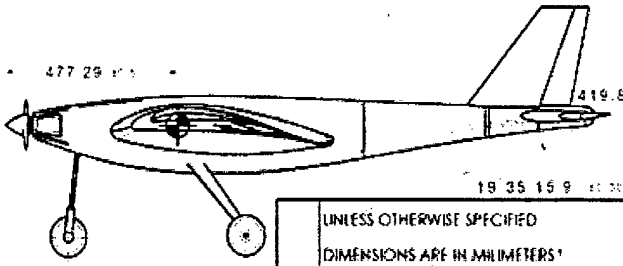
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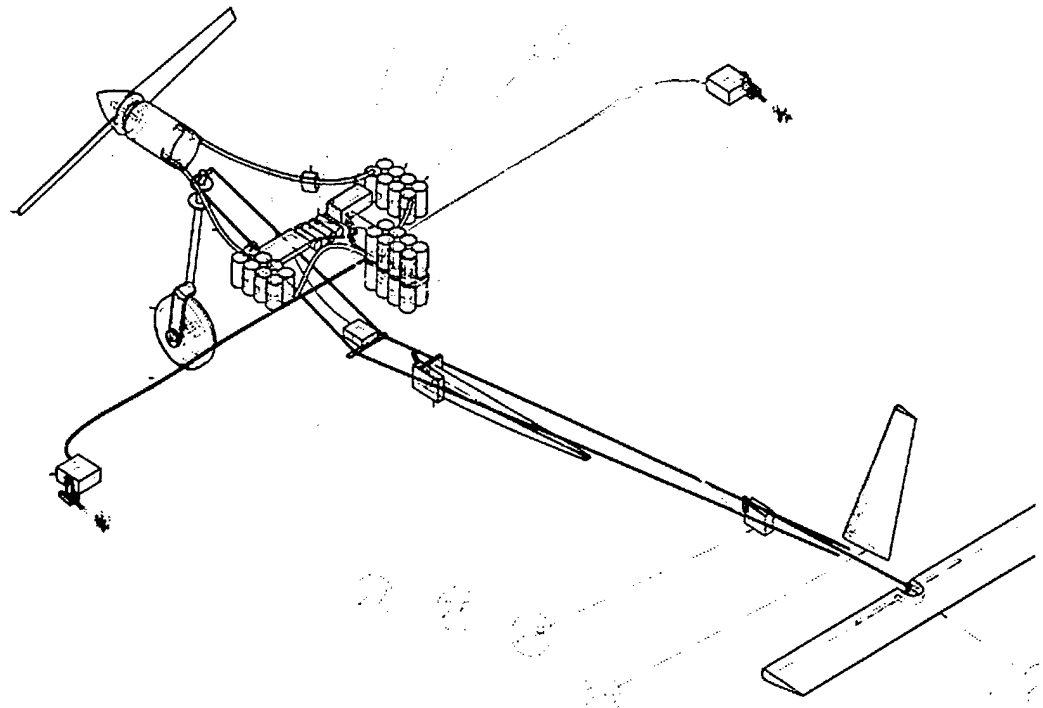
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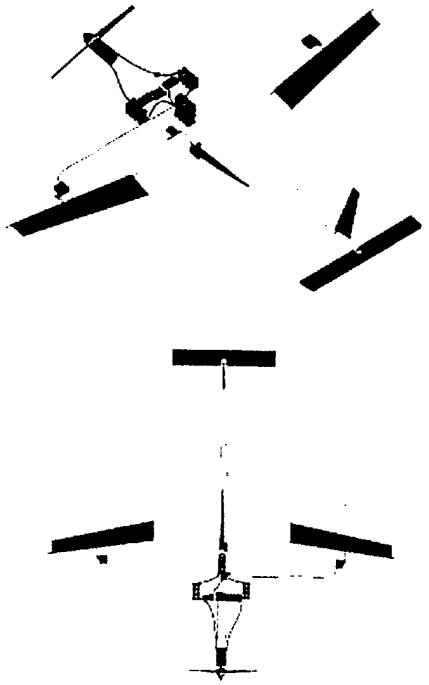
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3 VIEW & OVERALL DIMNSIONS

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DESCRIPTION	REVISIONS	SHEET 2 OF 5
	CHECK BY	DATE





PARTS SPECIFICATIONS				
	QUANTITY	DESCRIPTION	PART NAME	MATERIAL
17	2		flapeleron	foam glass fiber composite
18	1	works together with rudder	nose wheel	
19	1	engine-batteries cable		
20	1	between engine and power supply	fuse	
21	36 cells	SANYO SCR 2400 NiCd	BATTERY PACK	
22	1		receiver	
23	1		PROPELLER	carbon or wood
24	1		speed-controller	
25	1		rudder	
26	1	open/close/actuation of depot orifice	servo-actuator	
27	1	between rudder and nose landing gear	servo-actuator	
28	1	actuates the elevator	servo-actuator	
30	1	astro cobalt 90 direct drive	engine	
31	2	flapeleron-receiver cable		
33	1	power suppy of receiver	battery pack	
34	2	actuates flapeleron	servo actuator	

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Project Owner: Prof. Dr. Yurdanur TULUNAY

Technical Advisor: Prof. Dr. Mehmet S KAVSAOGLU

Team Leader: Gunay KAHYAOGLU

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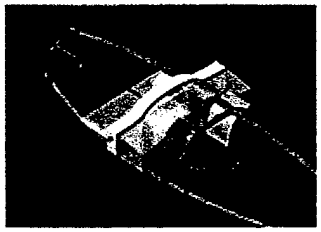
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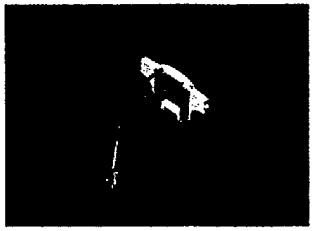
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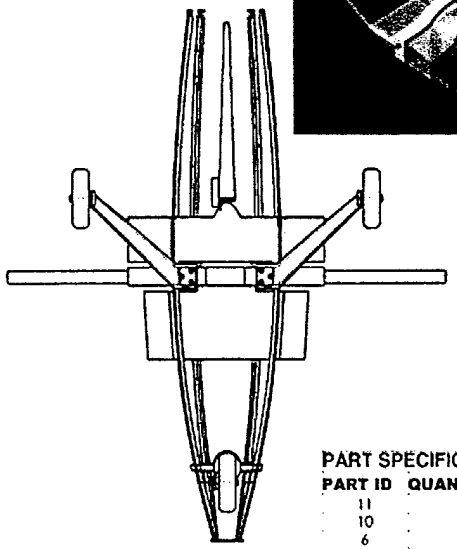
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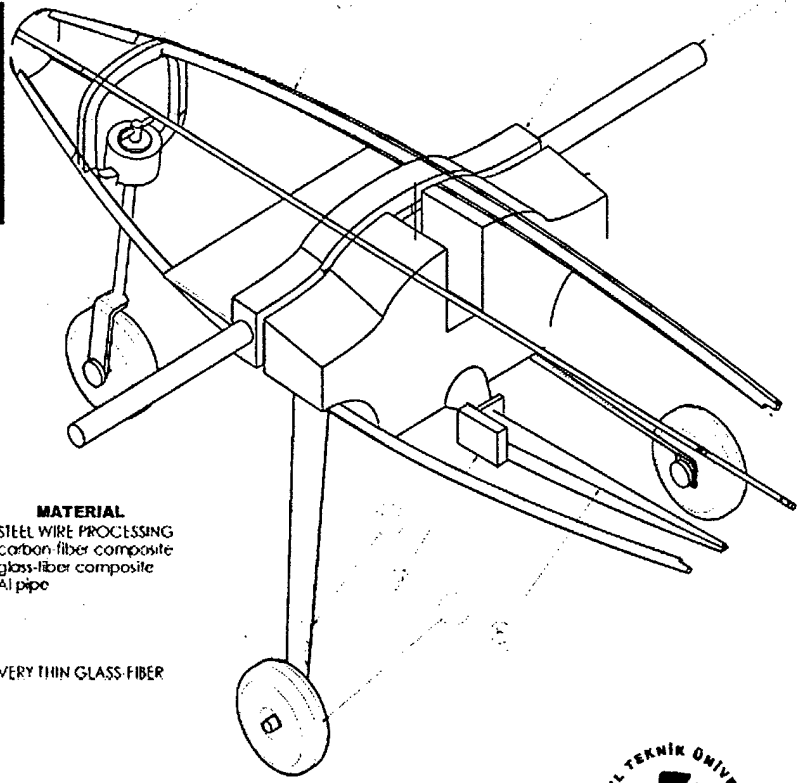
POWER SYSTEM & ELECTRICAL COMPONENTS

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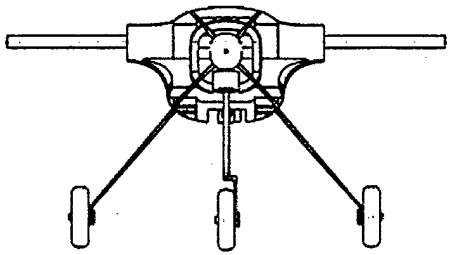


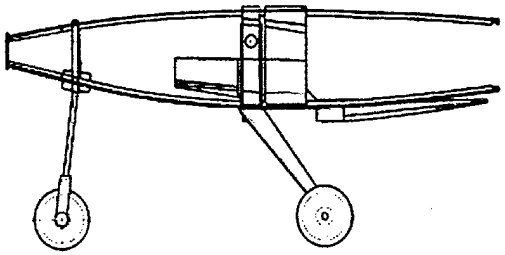




PART SPECIFICATIONS

PART ID	QUANTITY	PART NAME	DESCRIPTION	MATERIAL
11	1	NOSE LANDING GEAR		STEEL WIRE PROCESSING
10	2	MAIN LANDING GEAR		carbon-fiber composite
6	4	longerons		glass-fiber composite
31	2	SPAR	CARBON-FIBER PIPE	Al pipe
9	1	SERVO ACTUATOR	FOR WATER DEPOT MECHANISM	
8	1	PIPE	FOR WATER DEPOT MECHANISM	
7	1	BULKHEAD NO1	MAIN STRUCTURAL UNIT FOR NOSE LANDING GEAR	
15	1	WATER DEPOT		VERY THIN GLASS-FIBER





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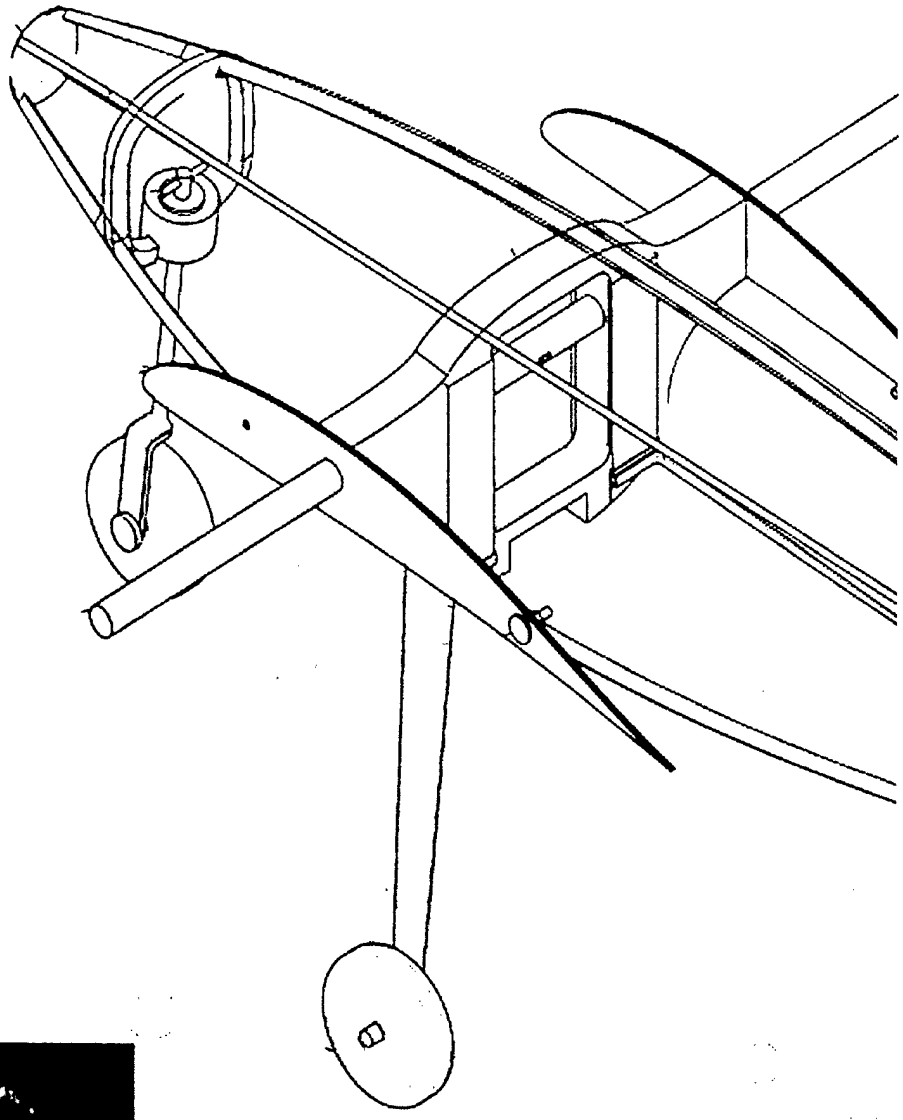
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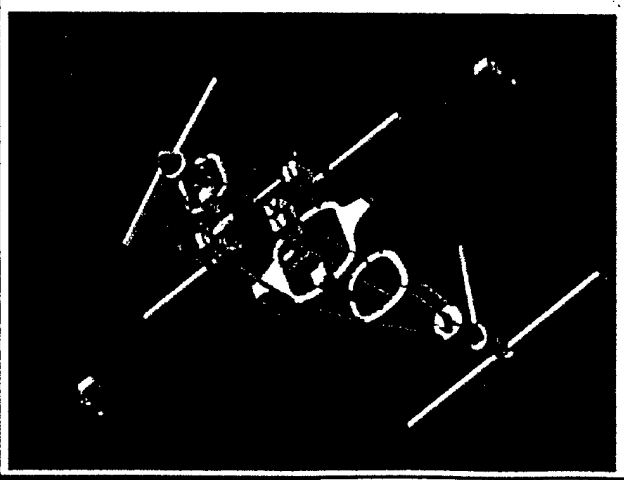
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LANDING GEAR & WATER DEPOT



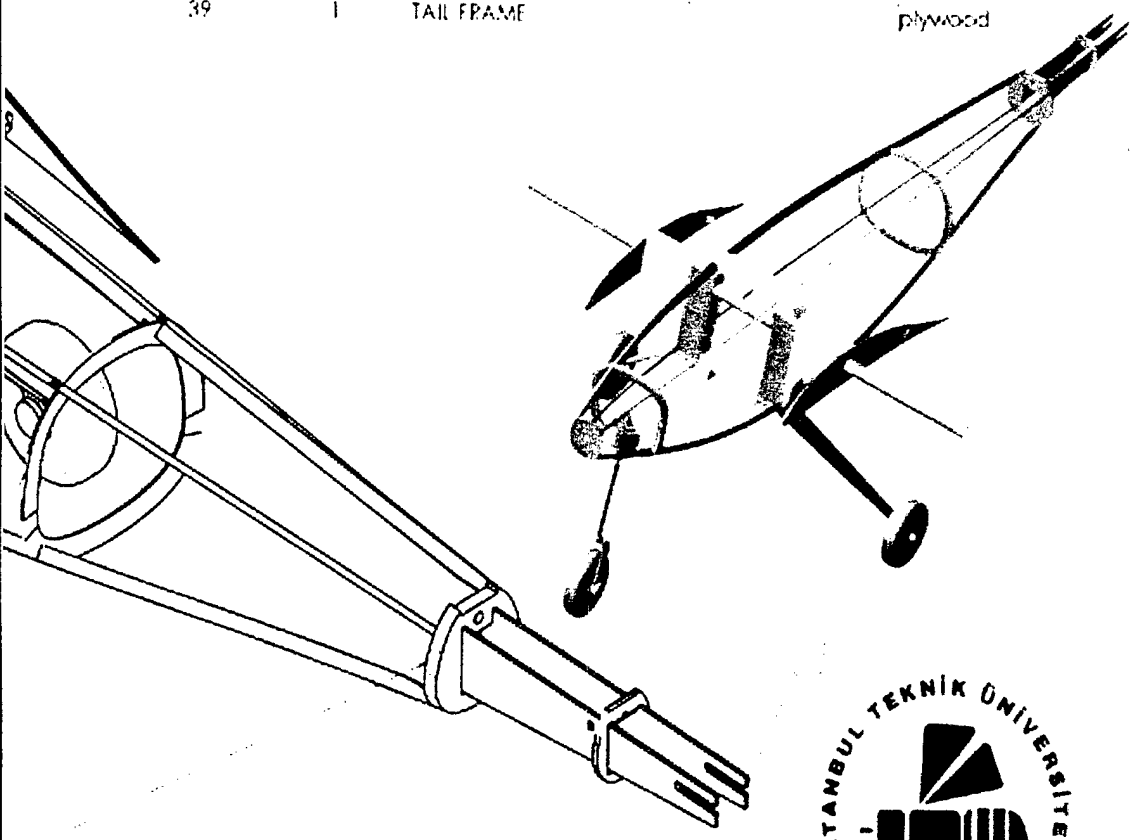


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PART SPECIFICATIONS

PART ID	QUANTITY	PART NAME	DESCRIPTION	MATERIAL
21	1	NOSE BULKHEAD		glass-fiber composite
5	1	LONGERON		" "
6	4	BULKHEAD NO1		" "
18	1	NOSE LANDING GEAR		steel
12	2	WING ROOT RIB		plywood
13	2	BODY ROOT RIB		plywood
31	2	SPAR	CARBON-FIBER PIPE	
17	3	WHEEL	ALL WHEELS ⌀10cm	
29	10	FASTENER	SCREW	steel
30	10	BOLT		
9	2	CONJUCTION BULKHEADS		glass-fiber composite & wood
10	1	BULKHEAD NO4		
39	1	TAIL FRAME		plywood



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MAIN STRUCTURAL COMPONENTS

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

-Design Report-

The University of Texas at Arlington

"Volantis"

March 2004

Table of Contents

1. Executive Summary 3
2. Management Summary 3
2.1. Team Organization 3
2.1.1. Fuselage Group 4
2.1.2. Propulsion Group 4
2.1.3. Wing and Tail Group 4
2.1.4. Performance Analysis Group 4
2.1.5. Manufacturing Group 4
2.2. Planned and Actual Timing of the Design Process 5
3. Conceptual Design 6
3.1. Introduction 6
3.1.1. Aircraft Configuration 6
3.1.2. Payload Configuration 7
4. Preliminary Design 8
4.1 Introduction 8
4.2 Main Wing 8
4.3 Tail 9
4.4 Fuselage 10
4.5 Water Tank 11
4.6 Motor 11
4.6.1 Motor Sizing 11
4.6.2 Propeller Pitch and Diameter 11
4.6.3 Battery Weight and Number of Cells 12
4.7 Finite element analysis of the wing spar 12
4.7.1 Dimensions 13
4.7.2 Loads 13
4.7.3 Material Properties 13
4.7.4 1g loading 14
4.7.5 4g loading 16
5. Detail Design 20
6. Manufacturing Plan and Process 23
6.1. Introduction 23
6.2. Fuselage 23
6.2.1. Processes Investigated 23
6.2.2. Fuselage Construction Process 23
6.3. Main Wing 24
6.3.1. Processes Investigated 24
6.4 Tail Boom 24
6.5 Manufacturing Milestone Chart for Scheduled Events and Timing 25
7. Testing Plan 26
7.1 Testing Date and Objectives 26
7.2 Check-lists 26
7.2.1 Battery Runtime 26
7.2.2 Thrust Measurement 26
7.2.3 Test Flight 27
7.3 Test Results and Lessons Learned 27
References 28

1. Executive Summary

This document details the methods used by the student team at The University of Texas at Arlington to design, build, and test a radio-controlled aircraft for the AIAA 2003/04 Design Build Fly Contest. The contest offers two possible missions: one where the aircraft makes four flights around a preset course and a second which the aircraft flies two flights with a water payload which is dumped in flight. The course flown will have 1000 ft. between turning points. One 360° turn will be made in the opposite direction of the course turns.

Each element of the project had to be planned carefully before an attempt was made to build the component. The team had to draw on the expertise of underclassmen with experience in making composite model aircraft to construct the airplane and seniors made design decisions.

Weight was the primary concern for each component. Test parts were made to compare the weight of one building material against another. For example, to compare plywood and fiberglass as wing building materials, the team made three test wing sections: one with only fiberglass, one with plywood, and one with plywood and fiberglass. The conclusion was that using fiberglass on the outer third of the wing and 1/64" plywood in the inboard portion of the wing would offer the greatest strength and weight.

The entire design was focused on performance. This is why the team decided to devote an entire design group to aircraft performance. Each group was required to bring their design models to the performance analysis group for review. For example, when the wing group arrived at desired wing design, they had to consider the weight requirements given by the performance group. When they were unable to reach the performance group's expectations, other methods of reducing weight were investigated. This process continued iteratively throughout the design of the whole aircraft until a working solution was reached.

The methods used for design and construction were selected with the purpose of increasing simplicity and improving performance. To this end, research in methods of construction and design features was made. The result of this research was to direct the design process in the appropriate direction.

2. Management Summary

2.1 Team Organization

The team is comprised of eleven students in Aerospace and Mechanical Engineering. In order to optimize the design process, The University of Texas at Arlington "Volantis" team decided to divide tasks into five parts;

1. Fuselage
2. Wing and tail
3. Propulsion, and
4. Performance analysis
5. Construction

For each component, senior students led and had responsibilities for the following areas.

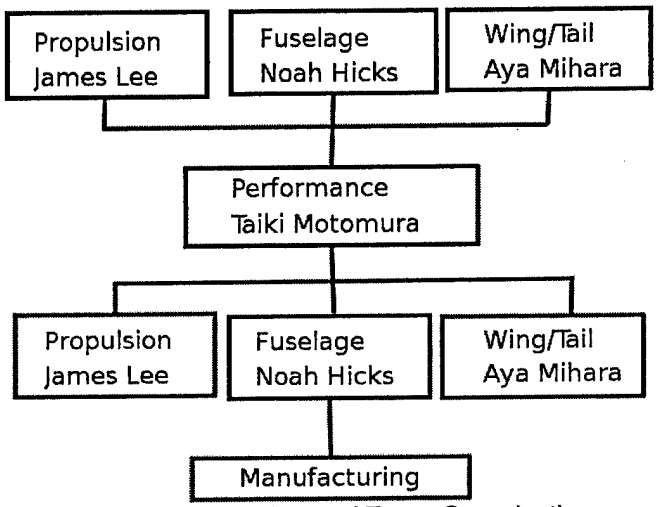


Figure 2.1: Flow Chart of Team Organization

First, fuselage, propulsion, and wing/tail groups decided the ideal dimensions for each component as shown in Figure 2.1. Next, the data was given to the performance group to check whether it would meet the aircraft performance requirements. Third, the performance analysis would be sent back to the components group for modification. After iterative calculations and the dimensions had been set, the manufacturing group started constructing the airplane parts by parts.

2.1.1 Fuselage Group

Fuselage group was responsible for designing shape, motor, battery, water tank position, main wing and tail attachments, and landing gear design. The main idea of the design was to reduce parasite drag and the Rated Aircraft Cost as much as possible. The fuselage width was minimized to reduce drag, and also helps water tank to dump the water out by gravity loading. The center of gravity (CG) was carefully studied to decide where to place components in the fuselage. The water tank must be placed at the CG to prevent shifts that would cause an unstable condition during flight. The primary concern regarding the structure was at the wing attachment and landing gear attachment where the strongest forces, lift and weight act.

2.1.2 Propulsion Group

The purpose of the propulsion system was to have sufficient thrust to take off in 150 ft and runtime to complete the mission. Thrust depends on propeller size and blade count, however, higher thrust increased current draw, shortening battery life. It was essential to maximize the thrust while maintaining the operation time enough to complete the flight.

2.1.3 Wing and Tail Group

Since the aircraft was expected to be a heavy lift slow flight model, the research of the airfoil was aimed for high lift and low drag in relatively low speed. Once the airfoil has been selected, dimensions

have been calculated based on how much lift was necessary to achieve the required mission. Wing and tail sizing is critical for the flight performance.

2.1.4 Performance Analysis Group

Performance analysis group gathered the information given from fuselage, wing and tail, and propulsion group to study take-off distance, cruise speed, and thrust required for best performance. If the designs from components do not meet the expected performance, the data had to send back to the components group for modification.

2.1.5 Manufacturing Group

Once the dimensions were decided, materials had been chosen. This process involved weight, strength, and availability of the materials in commerce.

2.2 Planned and Actual Timing of the Design Process

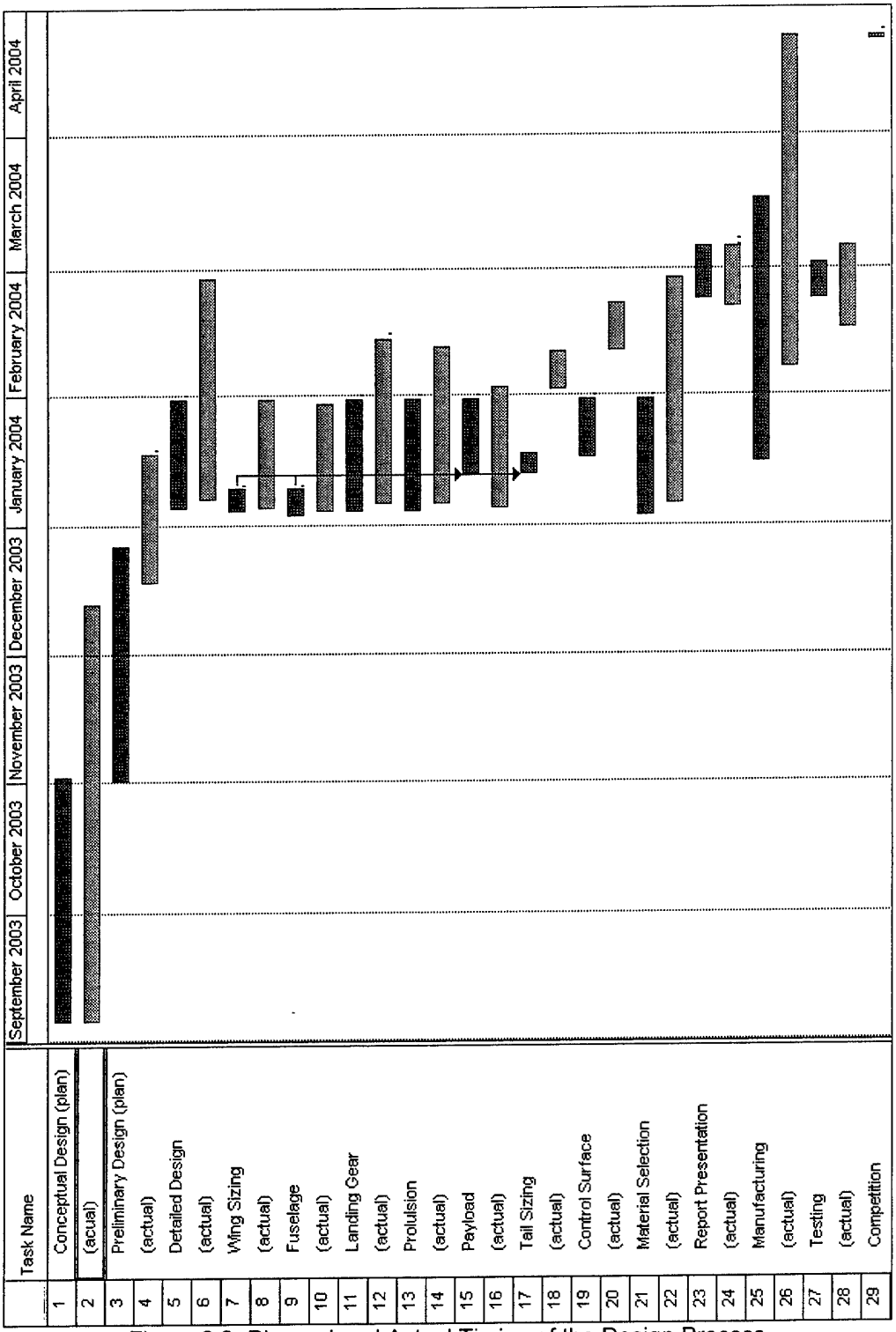


Figure 2.2: Planned and Actual Timing of the Design Process

3. Conceptual Design

3.1 Introduction

In the process of selecting a suitable design, the critical features must be defined. Since this aircraft will be judged on a numerical scoring system, the factors that most directly contribute to the final score are of primary importance.

The foremost contributors in scoring are the mission time and rated aircraft cost. Therefore all conceptual design has been evaluated based on these considerations or Figures of Merit (FOM).

Conceptual analysis was broken down into three areas: aircraft configuration, materials selection, and payload systems. Each area was analyzed based on how it affected the mission time and rated aircraft cost. Additionally, each area had to be considered for compliance with the others. Conceptual analysis of each design configuration provided a initial guess by which to verify our qualitative analysis.

3.1.1 Aircraft configuration

The team decided that there were a vast number of aircraft design concepts and that only a few would apply to this aircraft. Initial evaluations resulted in three feasible concepts suitable as aircraft designs:

1. Flying wing
2. Canard
3. Conventional layout

Each conceptual design was tested to evaluate rated aircraft cost and flight time. Each option presented merits and shortcomings and the team conceptually and qualitatively evaluated each to arrive at the conclusion that a conventional layout would be most functional.

Flying Wing

The flying wing is aerodynamically efficient and challenging at the same time. Total parasite drag on a flying wing is reduced since there are no tail surfaces protruding. However, a flying wing design is very sensitive to center of gravity location. Since this aircraft is carrying a fluid payload tank sloshing became a concern very early in the analysis. A lack of suitable ideas to prevent it was a significant factor in eliminating this as a design option.

Rated aircraft cost is very low. This is a derivative of the fact that little or no fuselage elements exist and by implementing a sweep angle there is no need of vertical tail surfaces.

One of the unique advantages of the flying wing configuration, span loading, has already been eliminated as an option per the rules requirement that no payload be carried in the wing.

Canard Conceptual Design

Investigation of advantages in the canard design found it to be a strong candidate. Chief among the advantages found was that the canard's horizontal surfaces are all lifting. This means that pitch control is accomplished on a lifting surface rather than the negative lifting surface of a conventional design. Since all horizontal surfaces are creating lift, the main wing would operate at a lower lift coefficient thus lowering induced drag. By reducing induced drag the aircraft can fly faster on a given propulsion system. This reduces the mission time and therefore contributes directly to the flight score.

Factors that detracted from the canard designs suitability were primarily related to stability. Creating a design that exhibits positive static stability requires balancing the center of gravity between the main wing the canard. Previous experience with canards has indicated that incorrectly locating the center of gravity caused problems that were difficult to rectify.

Conventional Design

The conventional design was considered to be the most suitable because it offered proven design methods. Familiarity with the required design parameters for stable flight characteristics was the primary reason for its selection. The main feature of this design is a forward wing and aft-tail surfaces. Stability is achieved through a down lifting tail surface and a center of gravity forward of the center of lift. This is a simple concept to design the airframe with and was chosen primarily for that reason.

During the conceptual design process of the conventional configuration the team realized that there were opportunities to optimized the conventional design. Investigating efficiency concepts that could be applied to the conventional platform resulted in consideration of the pusher configuration. The advantage presented by a pusher configuration is that turbulent air in the rotating slipstream of the propeller does not pass over the entire fuselage. This should reduce skin friction drag and thus increase the overall aerodynamic efficiency. We

There were several tail configuration options in the conventional design. Among them were v-tail, conventional and cruciform. We choose the v-tail to reduce rated aircraft cost and simplify structural problems.

3.1.2 Payload Configuration

Among the many options available to carry the water payload, the team identified three objectives that needed to be achieved:

1. The payload needed to be removed in the minimum time with out the use of pumping to assist flow-rate.
2. The water tank should be designed to contribute a minimum to the fuselage RAC volume.
3. Plumbing apparatus should be kept to a minimum to reduce the opportunities for leaking to develop and viscous friction.

Initially the team thought that the time required to remove the water payload would be a severely controlling factor in the flight speed, possibly requiring an aircraft with slow flight capability. However, testing preformed in November 2003 indicated that this would not be the case. Performing water dumps from a gas tank filled to 2 liters through a 0.5-inch orifice provided us with dump times of 11 seconds. The expected cruise speed of 47 ft./s flow across a 1314 ft flight path (1000 ft plus estimated 50 ft turning radius) allows 26 seconds of dump time, which is more than double the required time.

However, though the dump time is not a factor in required flight speed it is a factor in energy efficiency. Each second flown with the water on board is consuming valuable electrical energy from the batteries. With the motor producing the manufacturer's maximum power at 1200 Watts it is drawing close to 33amps from the batteries. The projected runtime at this current draw is barely 6min. Furthermore, the voltage drop that occurs between the first and second flights in the Fire Bomber mission reduces thrust available, which adversely affects the take-off distance on the second flight. All of these reasons contributed to the need to extract payload as quickly as possible.

To satisfy Objective 1, the team concluded that the tank should have a rectangular shape and sharp corners in the entire design except at the bottom near the exit. At this location in the tank the bottom and sides must be shaped to allow the water to smoothly enter the exit orifice. Having a sudden contraction in the flow area would produce pressure losses that would be undesirable.

Objective 2 is introduced constraints to the fuselage design. It was concluded that by using the fuselage as the part of the tank structure, the tank would inflict minimal increase on the rated aircraft cost. In concrete terms, this means that the team determined that the left and right side walls of the tank should be the left and right sides of the fuselage.

The essence of Objective 3 is to reduce the complexity of payload release valve. To do this we first surveyed the available pre-manufactured options. Lack of availability of these valves was a difficulty that weighed heavily on design choice

To satisfy these design objectives the team concluded that the final tank configuration should have the following characteristics.

1. Height of the tank should be greater than width to maximize hydrostatic pressure at the tank exit.
2. The tank surface should have a curved bottom to create laminar flow at the exit.

Implementing these features should provide a functional tank design.

4. Preliminary Design

4.1 Introduction

In this chapter, the results of preliminary estimation of each component, main wing, tail, fuselage, water tank, motor, and propeller, based on the conceptual design are shown. Also, the importance of the shape and size of each component is mentioned.

The methods to estimate Preliminary design are base on the book, "Design of Aircraft", written by Thomas C. Corke.

4.2 Main wing

Starting from the design used in last year's aircraft, the wing design group found several areas that could be improved. Since this year's aircraft requires both heavy lifting capability and efficient cruise, a wing planform that could achieve an elliptical lift distribution was sought. An elliptical lift distribution can be nearly perfectly attained by using a taper ratio, λ, of 0.4. The team found it difficult to make a wing with a 0.4 taper ratio due to uneven hotwire heating. Thus the design was compromised for manufacturing requirements to have a taper ratio of λ=0.45.

From an airfoil database, provided by Nihon Aero Student Group, NASG, the Selig 2048 has been chosen. Comparison of lift, moment, and drag curves on the NASG airfoil database provided the basis for airfoil selection. Shown below is the airfoil data of S2048.

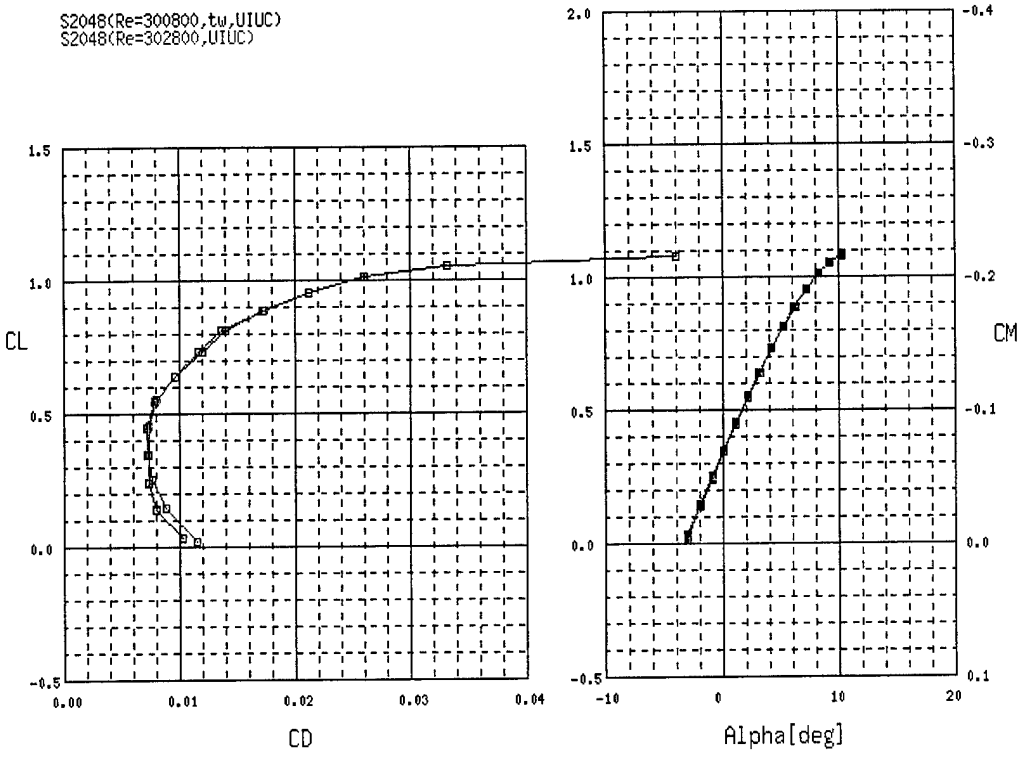


Figure 4.1: Polar Curve - S2048 airfoil

Table 4.1: Characteristics of S2048 airfoil

Thickness	0.0863	Leading edge radius	0.0042
Camber	0.0193	Trailing edge angle[deg]	4.1655

Below are the design parameters of main wing and drag estimation.

Table 4.2: Design Parameters of main wing

Design Parameters		Unit
S	11	ft^2
A	9.1	-
Λ_{LE}	5.73	deg
λ	0.40	-
b	10.0	ft
c_r	1.6	ft
c_t	0.6	ft
m.a.c.	1.2	ft

Total Drag	2.116027	lbf

4.3 Tail

The total surface area of conventional tail and the total surface area of V-tail should be the same. From this theory, size of V-tail is determined. The airfoil for the V-tail is the flat plat to reduce the total weight. Below are the optimized data and estimated shape of V-tail.

Table 4.3: Tail sizing

V-tail property		Unit
λ	0.6	-
Cr	1.4	ft
r	1.316569	ft
Λ_{LE}	20	degree

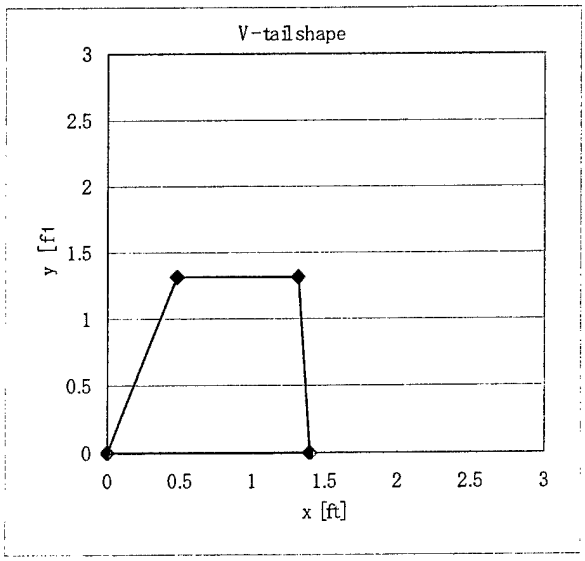


Figure 4.2: V-tail Planform Shape

4.4 Fuselage

Fuselage shape is primarily a function of the water tank. The water tank is 3 inches wide and the honeycomb material used to make the fuselage side is 0.25 inches thick. Thus the outside width of the fuselage is 3.5 inches. This shape increases the height of the fuselage to raise hydrostatic water pressure at the tank.

The estimation of fuselage drag is below.

Table 4.4: Fuselage Analysis

Flight Regime Data:		**Unit**
Cruise Alt.	1,000	ft
V	51.79	ft/s
ρ	0.0744742	lbm/ft^3
q	3.1021383	lbf/ft^2
μ	0.000012	lbm/(f-s)
ν (cruise)	0.0001611	ft^2/s
Dimension Data:		
D-max	1.0833333	ft
L/D	4.81	-
L	5.21	ft
S	11	ft^2

Total Drag:	**0.145**	**lbf**

The below is the figure of estimated fuselage size.

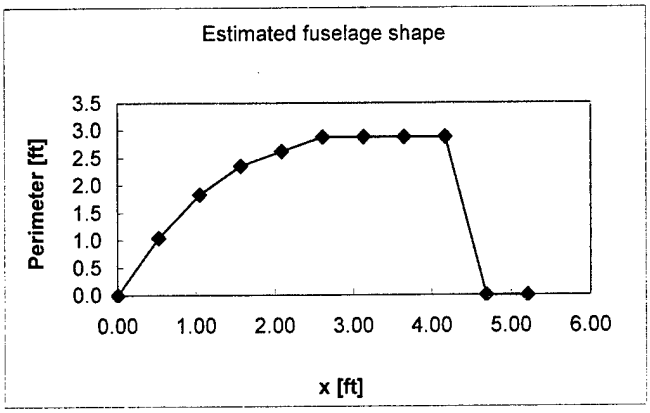


Figure4.3: Fuselage Shape

Table 4.5: Fuselage Size

Estimated size of fuselage		**Unit**
Width	0.291666667	ft
Length	2.421	ft
Height (Max)	1.083	ft

4.5 Water tank

Since the water tank is an integral part of the fuselage, its size is determined by where the fuselage bulkheads are placed. The fuselage was designed to have flat sides in the area of the water tank. As following the shape of the fuselage, the estimation of size of water tank become below. By increasing the height, hydrostatic water pressure increases. The volume of the tank is almost 4 liters. The valve system is operated by a microservo driving a plunger into the hole at the bottom of the tank.

Table 4.6: Water Tank Sizing

Estimated size of water tank		Unit
Width	3	In.
Height	10	In.
Length	8.15	In.

4.6 Motor Sizing.

Motor selection was based the several requirements including power output, current draw, weight, and its effect on the RAC. An increase in the size or number of motors will result in an increase in power witch translates into shorter takeoff distances and higher flight speeds. However, increasing the number of motors will increase the Rated Engine Power. Also, increasing the size of a single motor will increase the Manufacturers Empty Weight. An increase in the REP or MEW will result in an undesirable increase in the Rated Aircraft Cost. For this reason it is important to select a single motor that balances weight and output power.

Through the performance analysis, it was determined that a minimum thrust of 200 oz is required in order to meet the take-off distance requirement of 150 ft. This required thrust along with takeoff velocity was used to determine the minimum power requirements of the motor. The effect of propeller and gearbox efficiencies must also be considered when selecting a motor. It was determined that the motor must be capable of producing a minimum of 800 W. This requirement, along with the manufacturer restriction, reduced the available options to the AstroFlight 60, or the AstroFlight 90.

Analysis of these two options was done using MotoCalc. Each motor was evaluated by calculating its performance over the manufacturers recommended input voltage and for a verity of different propeller sizes. I was determined that the AstroFlight 60 would not be capable of providing the required thrust without exceeding the maximum allowable current draw. However, it was determined that the AstroFlight 90 would be able to provide the required thrust without exceeding the maximum allowable draw.

4.6.2 Propeller Pitch and Diameter.

The propeller selection affects the RAC only through the MEW. Propeller selection should be based on aircraft requirements. These requirements include required thrust and cruse velocity. Due to relatively small power margins, it is important to make efficient use of the available power. Large propellers with a low pitch are efficient at low speed while higher pitch propellers are more efficient at higher speeds. With takeoff performance a primary concern, it is necessary to determine a propeller that is capable of producing enough thrust to accomplish take-off while also being capable of providing enough thrust to reach cruise speed. Low speed and high-speed conditions demand different propeller configurations. Due to a lack of available variable pitch propellers in the appropriate diameter it is important to strike a balance between takeoff performance and maximum airspeed. Propeller sizing is also affected by physical limitations such as ground clearance as well as the fuselage configuration. Within the selected airframe configuration, the maximum propeller diameter is between twenty and twenty two inches. To obtain the required thrust with a propeller of this diameter it is necessary to employ the use of a three of four blades with a pitch of 14 to 16 inches.

4.6.3 Battery Weight and Number of Cells.

The battery affects the RAC through MEW and REP. As the battery weight increases the RAC increases. The battery selection also affects the maximum airspeed and the endurance. As the total weight of the battery increases the RAC will be increased due an increase both the MEW and REP. Also important to consider is that as the number of cells increases, the maximum flight speed will potentially increase. This increase in flight speed will contribute an overall decrease in the score. The capacity of the individual cells directly effect the endurance of the aircraft as well at the maximum number of cells. Analysis completed with MotoCalc suggests that a battery constructed out of thirty-six Sanyo CP-2400SCR cells will provide adequate potential and capacity while minimizing the battery weight and satisfying the five pound maximum battery weight rule.

4.7 Finite element analysis of the wing spar

The wing is reinforced with a one-inch diameter carbon fiber rod 36 inches long. The rod extends 16.25 inches into the wing in each direction. Three and a half inches in the center of the rod is contained in the fuselage and used to connect the wings to the sidewalls of the fuselage. A finite element model was created for one half of the wing spar. The wing loading was applied to the bottom of the spar. That pressure load is equivalent to the force created by the wing in the 16.25-inch area contained within the wing. Furthermore, the degrees of freedom were constrained to zero within the section that was inserted inside the fuselage.

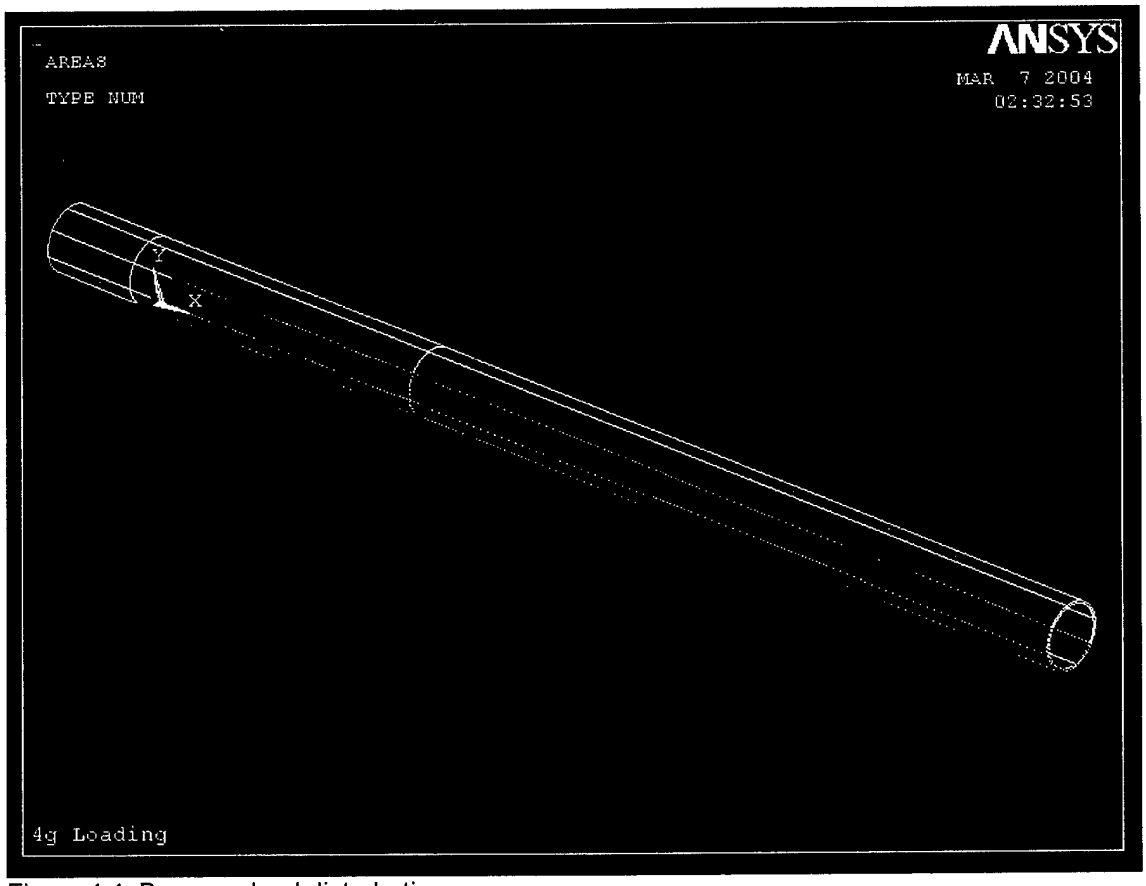


Figure 4.4: Pressure load distrubution

4.7.1 Dimensions

The pipe is 18 inches in length and extends from the center of the fuselage through 16.25 inches into the wing. The wall thickness of the pipe is 0.35 inches, which corresponds to roughly seven layers of carbon fiber.

4.7.2 Loads

Within the Figure 4.4, the red section on the bottom of the pipe indicates the pressure loading which corresponds to the lift force. In this case for a maximum of 4g-flight load, the corresponding pressure loading is 3.2 Psi. The degrees of freedom are constrained to zero in all directions in the area behind the center of the coordinate axis. There is no load shown in this short area. The spar was tested in Ansys for two different loadings. First test concerned the loading corresponding to a maximum flight load of 1g. Second test identifies with a loading corresponding to a maximum flight load of 4g's.

4.7.3 Material properties

Since the carbon fiber rod is an orthotropic material, the properties for all three directions have to be defined. The elastic modulus, Poissons ratio, and the modulus of rigidity for the x, y, and z directions are required to calculate the stresses and the deflection.

Table 4.7: Material Properties of carbon fiber wing tube

Direction	Modulus if elasticity (lbf/in^2)	Poisons ratio	Modulus of rigidity (lbf/in^2)
X	24.8 E 6	0.329	0.9 E 6
Y	1.41 E 6	0.329	0.9 E 6
Z	1.41 E 6	0.329	0.9 E 6

As the table shows, the modulus of elasticity is higher in the x direction. This is due to the fact that carbon fiber is strongest in the direction they are oriented. The majority of the carbon fibers are oriented in the x direction along the length of the pipe; therefore the properties differ in the x direction. Due to symmetry, the y and z direction have the same material properties.

4.7.4 1g Loading

The deflection, the stress in the y-direction, and the Von Mises stresses were calculated when applying the pressures for the 1g-flight load.

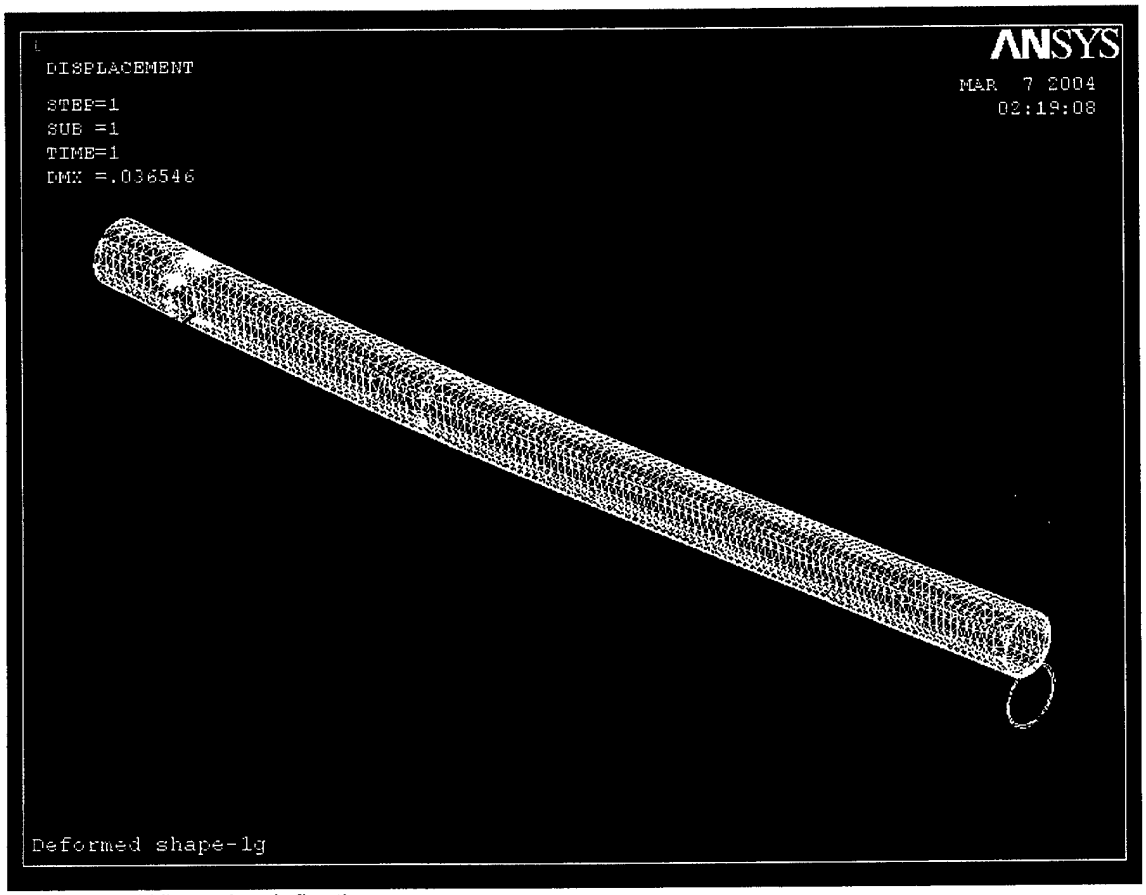


Figure 4.5: Wing tube deflection

Figure 4.5 shows the maximum deflection. The value for the maximum deflection shown in the top left corner (DMX) is 0.0366 inches. This is a relatively small value for a 16.25-inch pipe, and should be well within the flexure range of the pipe.

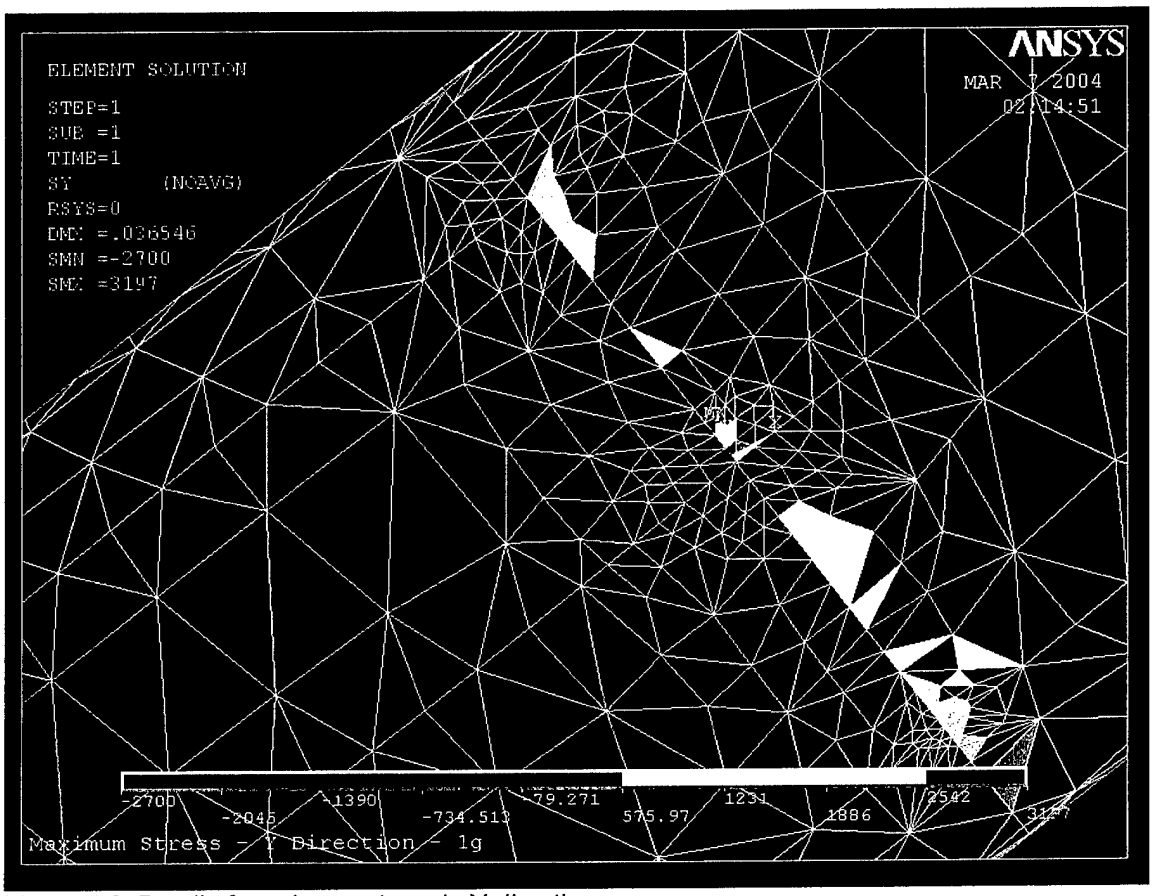


Figure 4.6: Detail of maximum stress in Y-direction

Figure 4.6 and 4.7 and below show the maximum stress in the y direction and the overall maximum stress of the pipes. The maximum stresses occur at the connection between the spar and the fuselage. While the wing is being pushed upward, simultaneously the spar is bent upward in the y direction and the high stress areas are created on the top and bottom of the connection to the fuselage. The maximum stress in this case occurs at the bottom.

The maximum stress in the y direction is 3.2 ksi and the overall maximum stress in the pipe is 5.6 ksi. The maximum tensile strength that was given for the pipe by the manufacturer is 200 ksi. Therefore for this type of loading, the wing spar will not fracture, delaminate, or fail in any other manner.

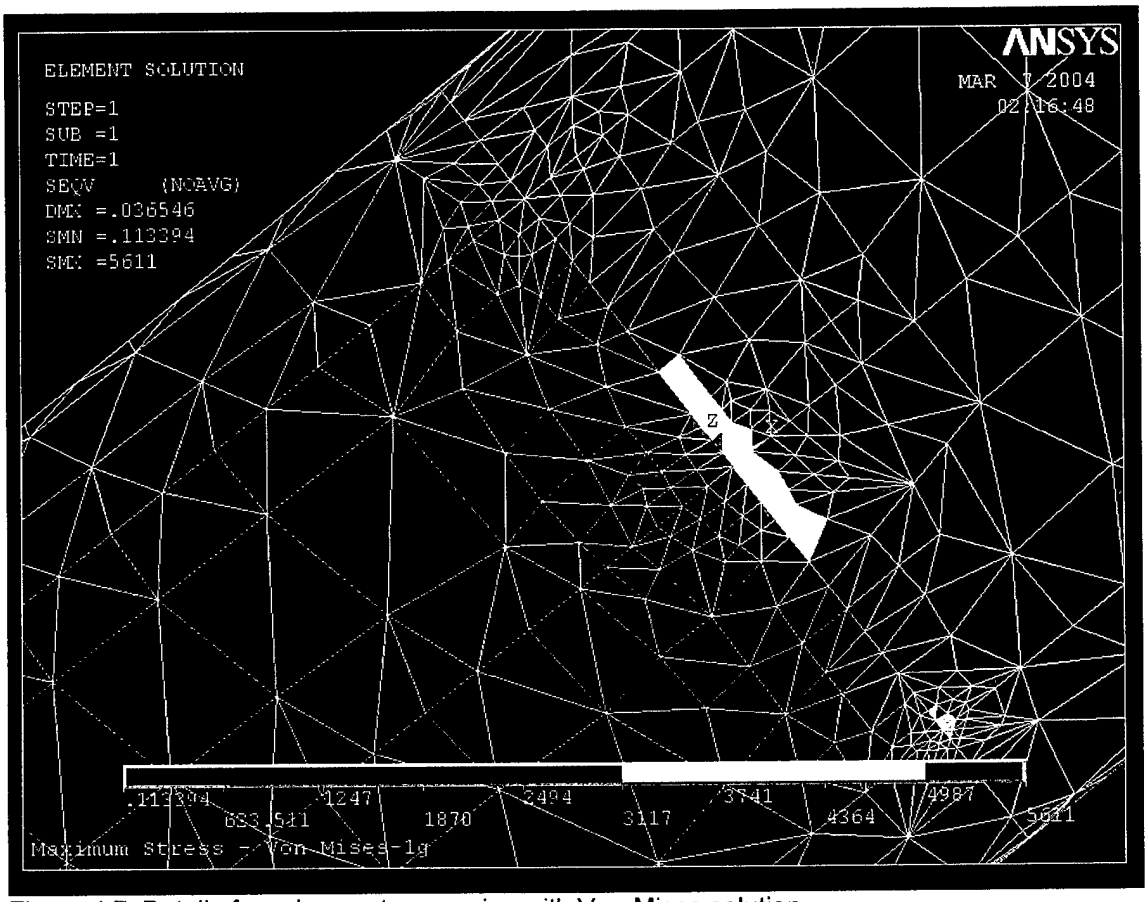


Figure 4.7: Detail of maximum stress region with Von-Mises solution

4.7.5 4g Loading

As compared to the maximum loading that will be applied on the spar during flight, the 4g loading was chosen to ensure a 50% safety factor. By doing this, we guarantee that the wings will withstand incidental increased loadings such as sudden gushes of wind. As before, the pressure value corresponding to a 4g maximum flight loading is applied. As shown in a previous Figure 4.7 the corresponding pressure is 3.2 psi. The deflections and stresses were calculated using this loading along with the identical degrees of freedom constraints as the previous model.

Figure 4.5 shows the maximum deflection of the spar. The value for the maximum deflection shown in the top left corner (DMX) is 0.1568 inches. The value is relatively small within the flexure range of the pipe and the wing.

As with the preceding model, the high stress areas were at the connection between the spar and the fuselage. The maximum value occurs at the bottom. Figure 4.9 shows the maximum stress is 15.9 ksi in the y direction. For the 4g loading state, Figure. 10 and 11 shows the overall maximum stress. The maximum stress value is 27.3 ksi. Both these values are well below the maximum tensile strength of 200 ksi.

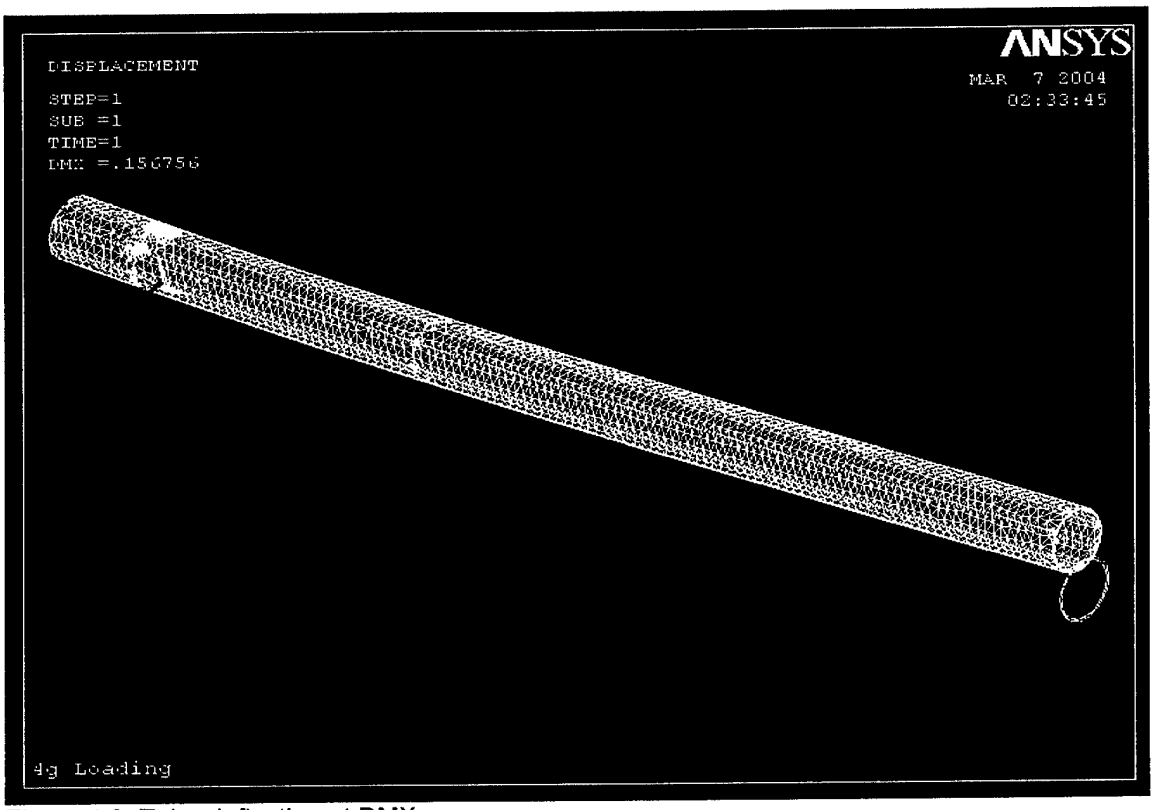


Figure 4.8: Tube deflection at DMX

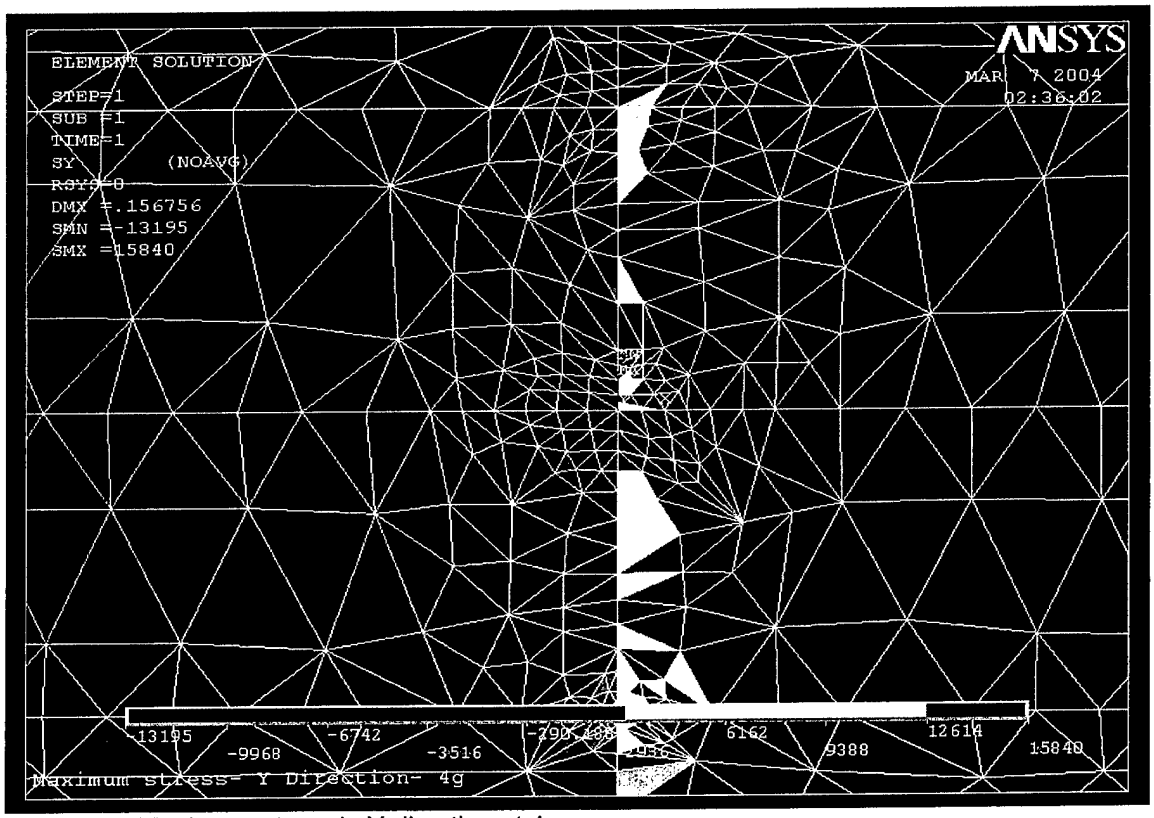


Figure 4.9: Maximum stress in Y-direction at 4g

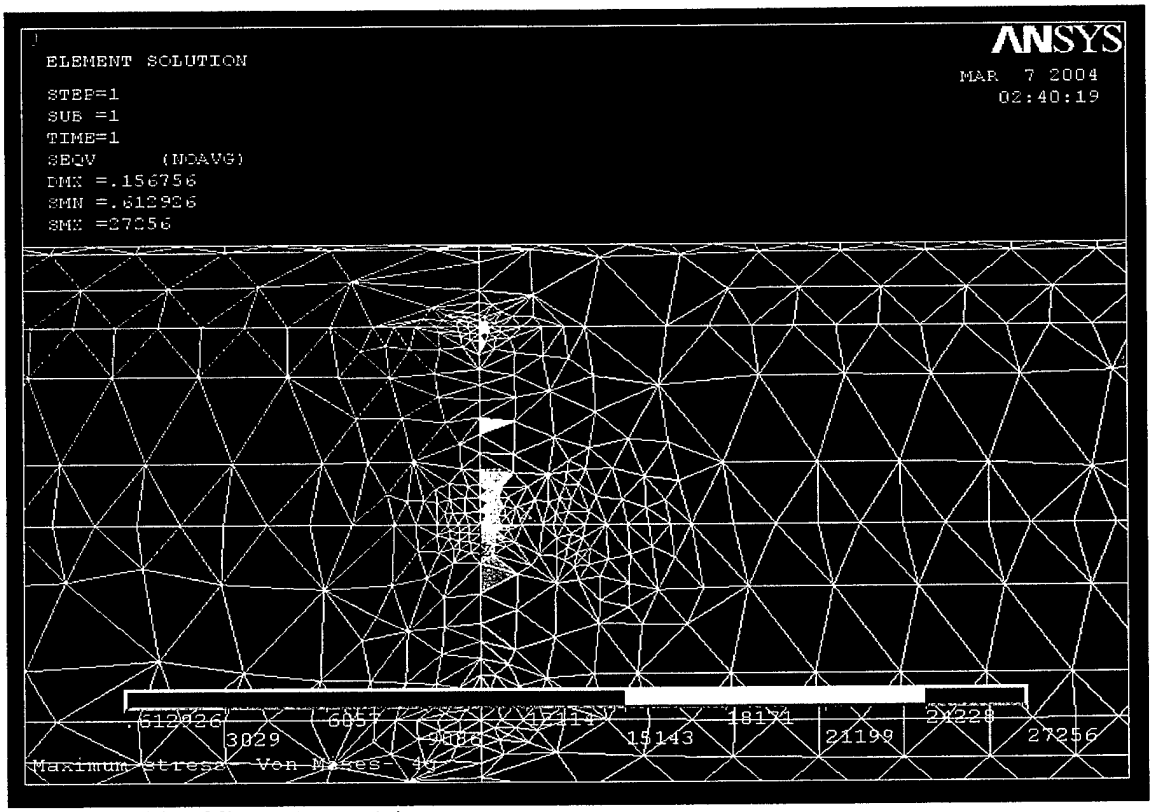


Figure 4.10 Von-Mises stress at 4g

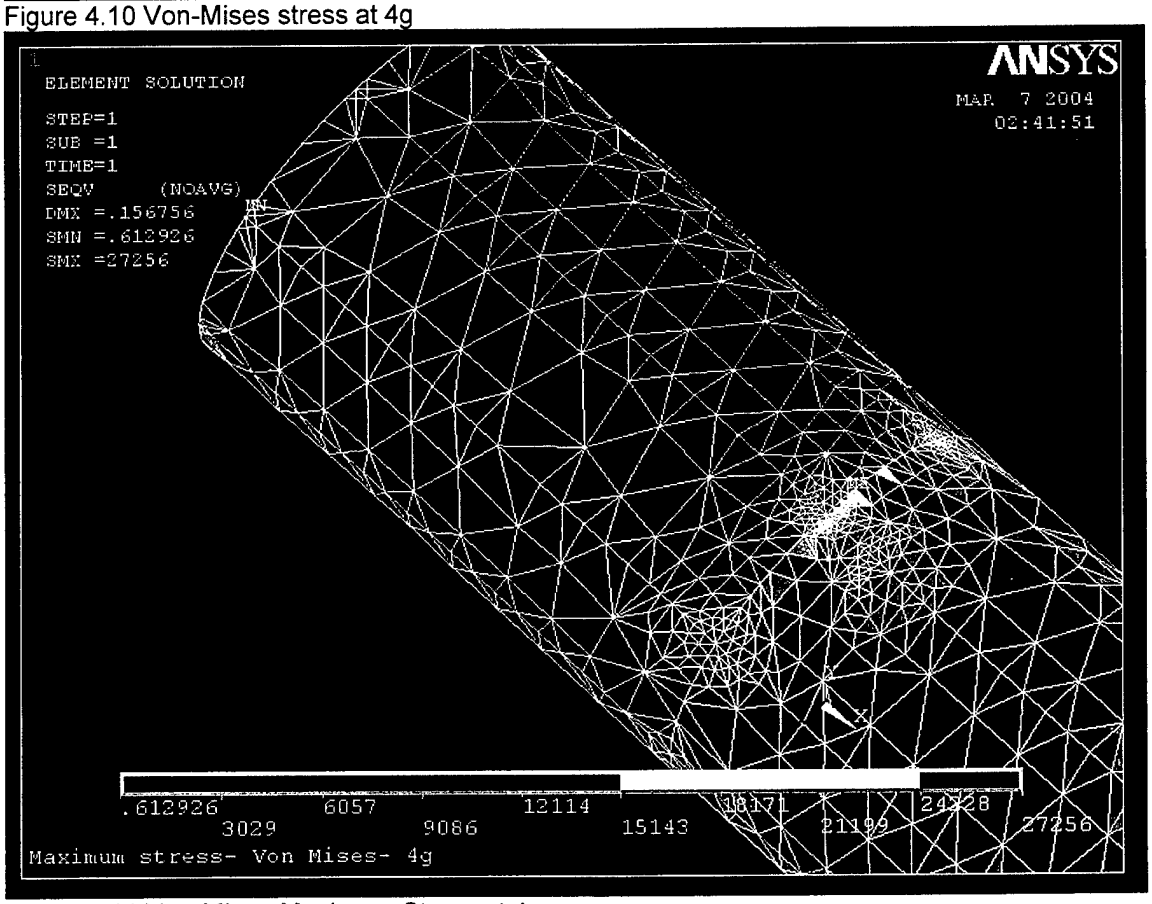


Figure 4.11 Von-Mises Maximum Stress at 4g

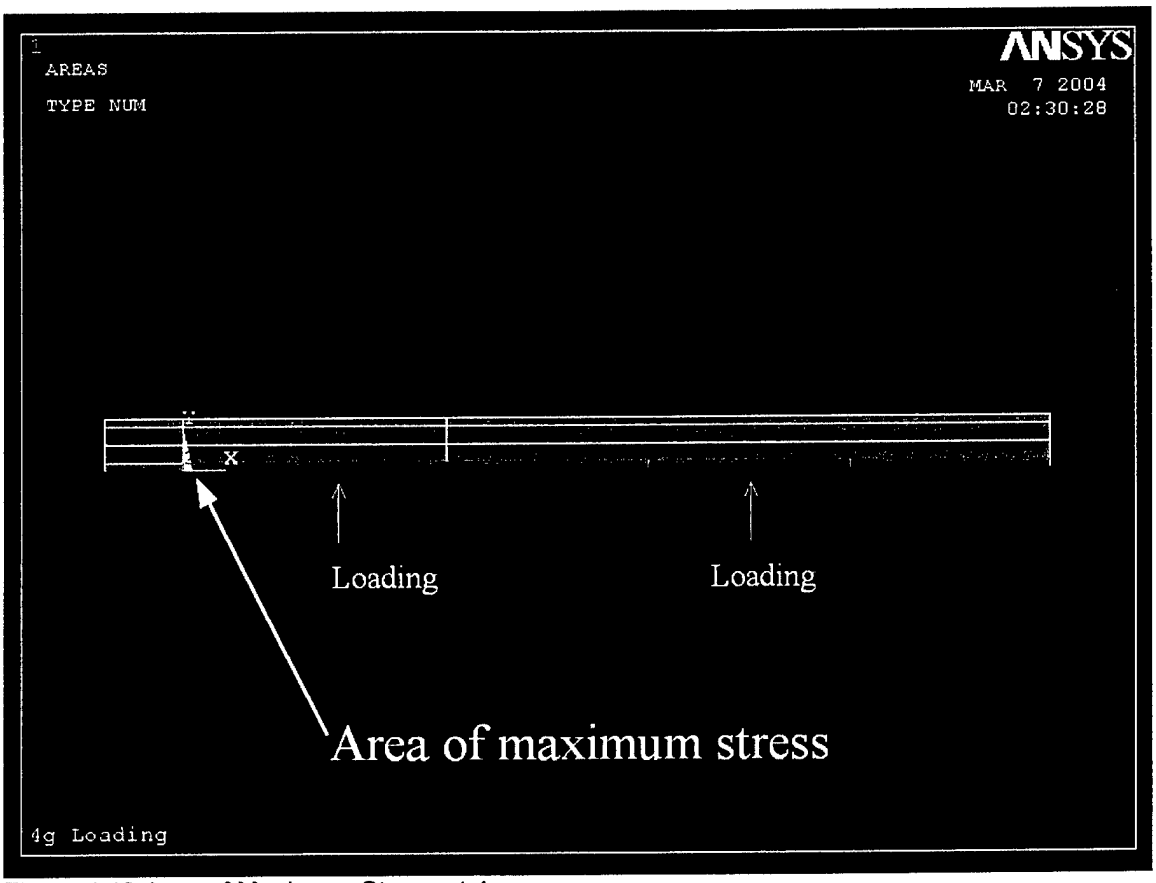


Figure 4.12 Area of Maximum Stress at 4g

Figure 4.12 shows the area of the maximum stresses and the loading areas.

In the Ansys model, there are not any significant deformations that will compromise the structural properties of the wing while applying a 1g-flight load and 4g-flight load. In addition, the stress values are far below within the range that the wing spar can be handled without significant deformation. Due to the spars structural rigidity, it is able to withstand delaminate or fracture with the 4g loading which includes a 50% safety factor.

5. Design Detail

In the water tank design, the plunger is one of the important systems. The size of tank is 3"× 8.15"×10". For the plunger system, a micro-servo has been set on the top of water tank. Also, two push rod guides will be fixed inside the tank. To make the water stable, four baffles will be fixed inside the tank. The 0.5" exit hole is going to be set behind the main landing gear to reduce the static pressure local static pressure at the exit.

Two different sizes of carbon tubes, 1" diameter and 0.393" diameter, have been used as main wing spar. The larger size has been used for the root side of main wing. The smaller size has been used for the tip side of main wing. To fix and prevent rotating the main wing to the fuselage, two short carbon tubes are fixed into each side of main wing root. Also, ribs are fixed on each end section to prevent the bending force on the end of carbon tubes. The surface of the wing is sheeted by 1/64" plywood sheet. The tip section of the main wing is held to the fuselage by a single 0.285" diameter carbon tube and rotation is prevented by aluminum studs placed forward and aft of the carbon tube.

A special consideration was paid to the heat sink design because higher efficiency of the batteries results in more power to the motor.

The batteries are set vertically inside the fuselage. Each of the batteries are parallel and perpendicular to each other in order to minimize the contacting surfaces, hence reducing the conduction of heat between the individual cells. Included in this consideration is the fact of trying to get the heat generated by the batteries away from them and into the convection area. Therefore lithium grease is used between the batteries to drawn the heat away from the source and then redirected towards an overlaying aluminum plate. Adding a vent to the front of the model to achieve forced convection then dissipates the resulting energy. The overall result is a combined heat transfer system that will allow more heat to flow through the system and cool the battery pack.

Table 5.1: Rated Aircraft Cost

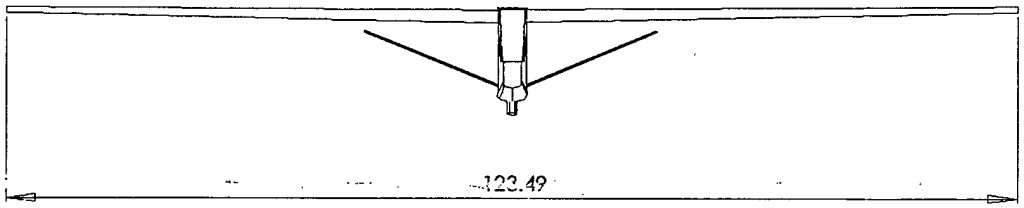
Coef.	Description	Value	Unit
A	**Manufactures Empty Weight Multiplier**	**300**	**$**
B	**Rated Engine Power Multiplier**	**1500**	**$**
C	**Manufacturing Cost Multiplier**	**20**	**$/hour**
MEW	**Manufacturers Empty Weight**	**16.47**	**lb**
	Battery (Sanyo 2400SCR - 2.05 [oz]) X 37	4.841	lb
	solder & wire	0.1	lb
	Wire and solder	0.1	lb
	Motor (Astroflight cobalt 90)	2.125	lb
	Landing Gear	0.2	lb
	Main Wing	4.02	lb
	Tail	1	lb
	Fuselage w/ water tank	3.5	lb
	1.3 [oz] Servo S3003 (Standard Servo) X 4	0.325	lb
	Servo S3101(Micro servo) w/ wire	0.0375	lb
	Astroflight 204 (speed controller) w/ wire	0.11	lb
	7 channel Receiver w/ wire	0.109	lb
REP	**Rated Engine Power**	**4.841**	**lbs**
MFHR	**Manufacturing Man Hours**	**235.923**	**hr**
	1.0 Wing		
	Wing Span	10.292	ft
	Chord	1.517	ft
	10 hr/ft^2 * wing span *chord*#wings	156.1	hr
	Ailerons	1	
	5hr*control function multiplier	5	hr
	Total hour	**161.090**	**hr**
	2.0 Fuselage		
	Body length	4.721	ft
	Width	0.292	ft
	Height	1.083	ft
	Volume	1.492	ft^3
	20 hr/ft^3 * Body length * Width * Height	**29.833**	**hr**
	3.0 Empennage		
	V-tail	**15**	**hr**
	4.0 Flight Systems		
	5 hr/servo or motor controller		
	4 Standard Servo	20	hr
	1 Micro Servo	5	hr
	1 Motor controller	5	hr
	Total hour	**30**	**hr**
RAC	**Rated Aircraft Cost**	**16.92**	**$(Thousands)**

Table 5.2: Data of sized airplane

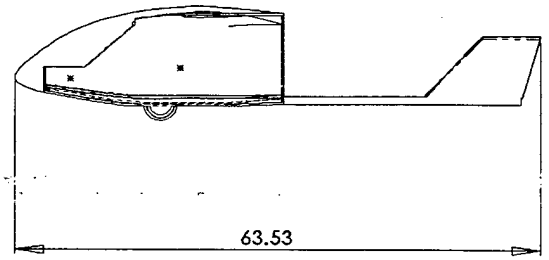
Geometry		Unit
Length	5.21	ft
Span	10	ft
Height	1	ft
Wing area	11	ft^2
Aspect Ratio	9.090909091	-
Control volumes	52.1	ft^3
Performance		
Cl max	1	-
L/D max	20.4	-
Max R.C.	535.92	ft/min
Stall speed	35	ft/s
Max speed	52	ft/s
Take-off length (Empty)	57.5	ft
Take-off length (Gross)	143.8	ft
Weight Statement		
airframe	9.12	lb
propulsion system	7.1	lb
control system	0.725	lb
payload system	0.377	lb
payload	8.8	lb
empty weight	16.47	lb
gross weight	25.27	lb
Systems		
Radio	Futaba T6XA PCM	
Servos	Futaba S3101	
Battery configuration	36 Sanyo SCR Nicad Cells	
Motor	Astroflight Cobalt 90	
Propeller	20x14 pusher 4 blade	
Gear ratio	2.75	

UT Alington "Volantis" Three view
All dimensions in inches

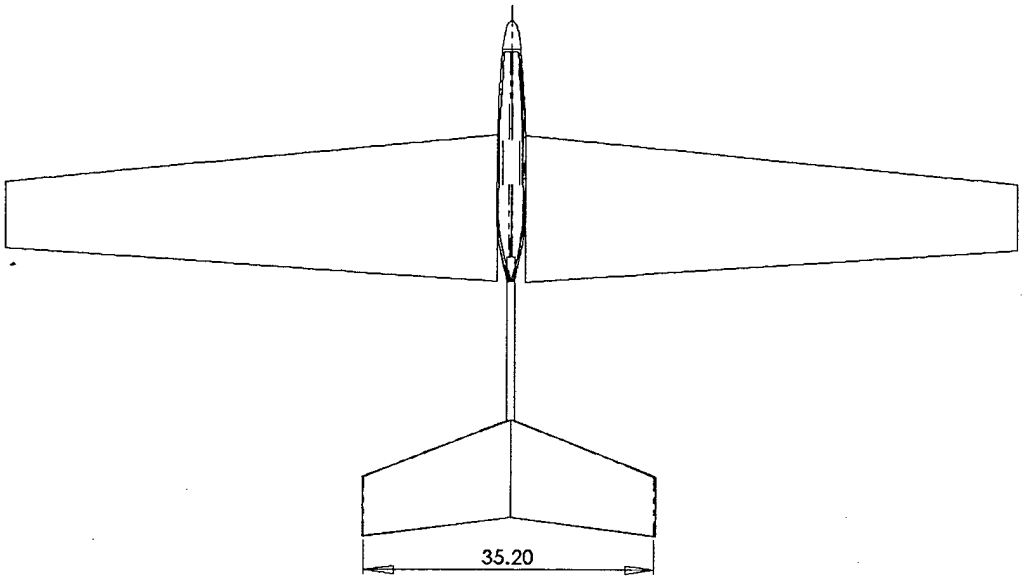
Front View



Left Side View



Top View



6. Manufacturing Plan and Processes

6.1 Introduction

The main Figures of Merit used in determining construction methods of this aircraft were weight and availability. Skill was not as large a restriction as expected since the team was able to overcome several construction challenges early in the manufacturing process. Creating a fuselage with complex curves from honeycomb and fiberglass without using a negative mold was initially thought to be an impossible task. However, sufficient planning and testing resulted in a completed fuselage product that showed satisfactory results. Thus manufacturing skill, while an important consideration in determining the manufacturing process to use, was found to not be a restrictive factor.

One of the most important parts of this project is the manufacturing plan and processes. The manufacturing plan and process for this aircraft was categorized into three parts:

1. Wing
2. Fuselage
3. Tail

Since the weight is a critical part of an aircraft, consideration was given to composite materials such as honeycomb, fiber glass cloth, carbon fiber cloth, carbon fiber tunings, extruded blue Styrofoam, balsawood, four litter plastic bottle, epoxy and polyester resin.

6.2 Fuselage

6.2.1 Processes Investigated

Initially the manufacturing plan considered for the fuselage was shaping extruded blue Styrofoam and inserting a four litter plastic bottle and covered with fiberglass cloth. The plan changed when a donation of honeycomb was made to the project. Cost is a significant factor in all aspects of this design and the price of this honeycomb made a very reasonable choice. The method used to construct the fuselage from honeycomb involved first shaping the extruded blue foam and then covering it with honeycomb sheet sandwiched between fiberglass cloth. Honeycomb gives a higher structural strength than a fiberglass-foam matrix and offers greater internal volume in the fuselage.

6.2.2 Fuselage Construction Process

In chronological order the process used to make the fuselage was:

1. Cut the extruded blue Styrofoam in to the dimension using hotwire
2. Shape the foam to the exact shape using sand paper
3. Cover the shaped foam with fiber glass cut to size and wetted with epoxy resin
4. Cover the foam with honeycomb sheet cut to size
5. Cover the honeycomb with two layers of fiberglass cloth cut to size wetted with epoxy resin.
6. Cover the outer layer of fiberglass with Mylar plastic sheet
7. Cover the fuselage set-up with waxed paper
8. After covering with waxed paper set the fuselage lay-up into a vacuum bag and let the epoxy resin coated fiberglass cure for the next forty-eight hours under vacuum pressure.
9. After forty-eight hours take the set up and peal of the wax paper and Mylar. Remove the foam from the part using acetone.
10. Install bulkheads and water tank in the fuselage
11. Shape and lay-up fiberglass canopy
12. Insert Carbon fiber tail boom to complete fuselage structure

6.3 Main Wing

6.3.1 Processes Investigated

Manufacturing processes available for the wing are based on the processes used to manufacture model airplanes. Consideration was given to several manufacturing methods.

1. Wooden built up wing with ribs and spars
2. Foam core wing with fiberglass covering
3. Foam core wing with plywood covering

Each method was evaluated with consideration for the following features of merit:

1. Weight
2. Building skill required
3. Strength

It was quickly realized that a hybrid solution would provide advantages that would not be realized by adopting a single solution. Foam core construction offers a quick build up time, where the shape of the entire wing can be quickly obtained. A built up wing is very light and does not require special construction equipment beyond a flat building surface and wood cutting tools but a spar offers

The manufacturing plan consider for the main wing cut to the selected airfoil shape covered with two layers of fiberglass, one layer of birch light plywood or one layer o birch light plywood and a one layer of fiberglass. After testing all three options we found that the two layer of fiberglass give as the best weight to strength ratio. The process of manufacturing the main wing entailed the following steps.

1. Cutting out the extruded blue Styrofoam with hotwire.
2. Cutting the foam wing in to cross-sections in required length.
3. Cutting holes through the wing to for inserting wing tubes.
4. Inserting the wing tubes and socks in to the wing and gluing the to the wing.
5. Inserting ribs made of plywood between the cross-sections of the wing where the wing tubes end.
6. Ones all the wing tubes and ribs are inserted the wing pieces are glued together the wing is ready for to be cover with glass.
7. Cut the fiberglass cloth to the required length.
8. Wet the fiberglass cloth with epoxy resin.
9. Lay the fiberglass cloth over the wing.
10. Cover the foam and fiberglass cloth combination with Mylar.
11. Cover the wing set-up with waxed paper.
12. Vacuum bag the set-up until the fiberglass and epoxy cured.
13. After the fiberglass harden over the foam peal off the Mylar plastic cover and the waxed paper
14. and the wing should be ready for use.

6.4 Tail Boom

The design of the aircraft calls for a pusher propeller aircraft. A carbon fiber tail boom inserted into the end of the aircraft comprises the tail boom. At the aft end of the carbon tubing a V-tail is attached to the carbon tail boom with steel inserts. Initially the team thought that the tail airfoil should be made similarly to the wing describe above. Computing the center of gravity of each component, however, revealed that the design was very sensitive to tail weight and that it must be minimized.

To reduce the tail weight the team redesigned it using spruce and balsa wood. The new process includes making the tail surface main frame out of balsawood and spruce and covering it with heat shrink plastic. This gives us sufficient weight reduction to meet our weight requirement. The manufacturing process is as follows:

1, Cut the wing frame out of balsawood and spruce to the size
2, Glue the frame together using superglue
3, Layout the heat shrinks plastic
4, Attach the control surfaces to the tail wing
5, Attach the servos to the wing
6, Attach the tail to the fuselage

6.5 Manufacturing Milestone Chart for Scheduled Events and Timing

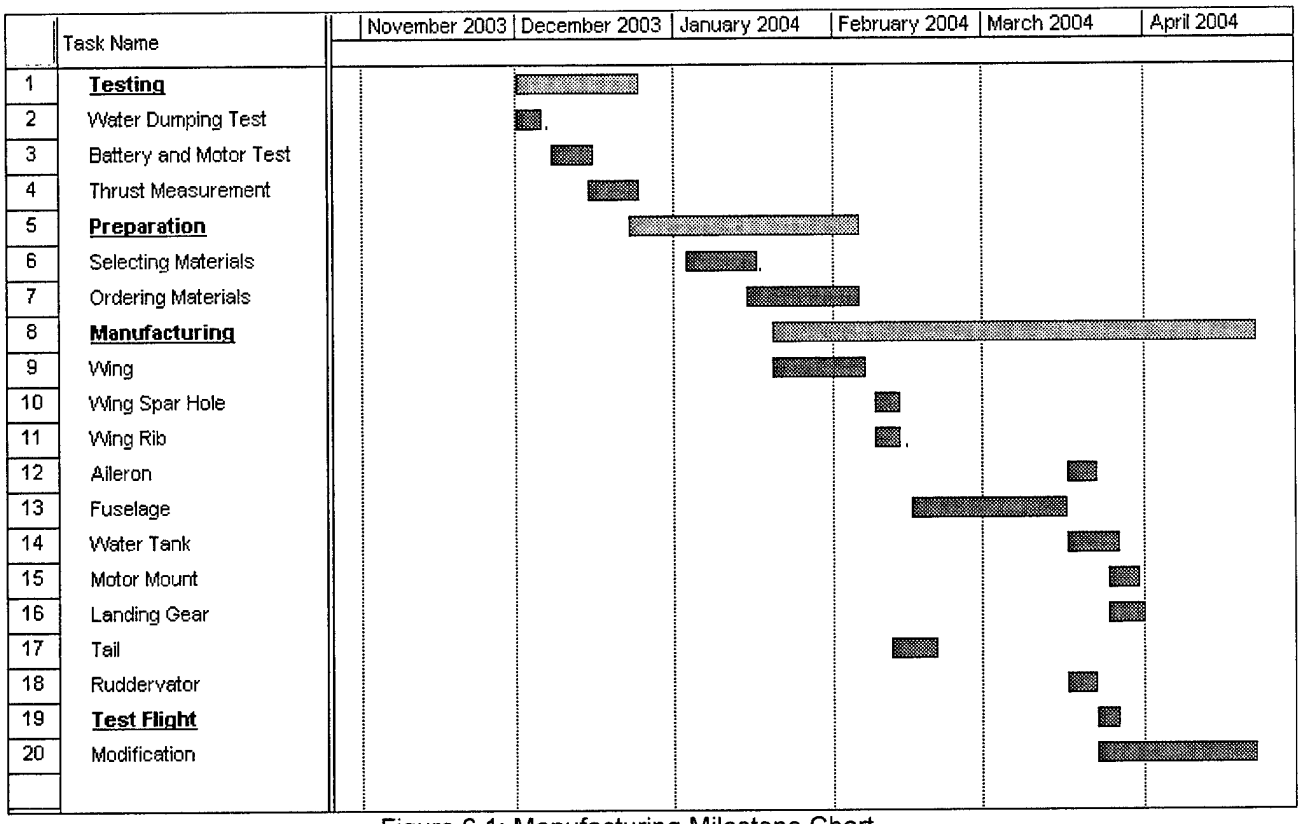


Figure 6.1: Manufacturing Milestone Chart

7. Testing Plan

7.1 Testing Date and Objectives

Testing is the part that gives ideas of designing and manufacturing of the aircraft their validity. The results were collected and analyzed for modifications necessary to optimize the aircraft performance. The tests have been divided into several sections to study the components of the aircraft.

Table 7.1: Testing Date and Objectives

Test	Date	Objectives
Water Dumping	Dec. 1 – Dec. 5, 2003	Measure discharge time to empty the water tank
Battery Runtime	Dec. 5 – Dec. 7, 2003	Measure battery runtime with throttle full position
Thrust Measurement	Dec. 15 – Dec. 24, 2003	Measure thrust with different propeller
Weight	Jan. 5 – Jan. 15, 2004	Measure weight for composite finish form, honeycomb sandwich fiberglass, and plywood sheet finish plate
Structural Strength	Jan 26 – Feb. 22, 2004	Make sure components made can resist in bending, torsion, point load
Control Surface	Feb. 3 – Feb. 25, 2004	Check servos, control surface deflections
Final Check	Mar. 15 – Mar. 23, 2004	Final aircraft checkout
Test Flight	Mar. 23 – Mar. 31, 2004	Measure take-off distance, flight time

7.2 Check-lists

7.2.1 Battery Runtime

Battery runtime has been estimated based on the flight time required to finish the mission.

Table 7.2: Battery Runtime

Test Part	Status	Time
Battery	Full Throttle	> 5 minutes, (estimation)

7.2.2 Thrust Measurement

Thrust has to satisfy the maximum thrust needed to take-off and climb.

Table 7.3: Thrust Measurement

Test part	Status	Thrust
Motor &Propeller	Full Throttle	> 13 lbs

7.2.3 Test Flight

Table 7.4: Test Flight Checklist

Phase	Parts	Status
Pre-flight	Battery	Fully charged, connected, secure
	Fuselage	Main wing connection, secure
	Tires	Check grids, inflation
	Control Surfaces	Free & correct
	Empennage	Tail connection, secure
	Propeller & spinner	Check
Motor Start	Fuse	Secure
	Propeller area	Clear
Taxi	Ruddervator	Check for directional control
Take-off	Throttle	Full
	Take-off distance	< 150 feet
Fire Flight	Water tank valve	Open & discharge water

7.3 Test Results and Lessons Learned

Test	Results and Lessons Learned
Water Dumping	Average of 13 seconds to discharge 4 liters of water with free drain through 0.5 inch exit diameter
Battery Runtime	More than 5 minutes using 2 blades, 22" x 18 propeller
Thrust Measurement	7 lb of thrust produced using 2 blade, 22" x 18 propeller, increase pitch / number of blades necessary to establish 13 lbs or more
Weight	Since honeycomb sandwich fiberglass is strong yet flexible and lightweight, it has been applied to build fuselage. Comparing weight and strength of wings coated with fiberglass and plywood, those were assigned to have plywood sheet for the root part and fiberglass for the tip part.
Structural Strength	Wing spars carried enough strength to prevent wing from bending or material failure during flight.

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AIAA 2004 DESIGN BUILD & FLY

University of Arizona

Project Advisor: Sergey Shkarayev

Team Leader:

Mike Woodard

Team Members:

Roger Aure
Belinda Holmes
Lasse Mæland
Dustin Wright

7. March 2004

Table of Contents

1 EXECUTIVE SUMMARY 1

1.1 OVERVIEW OF THE DESIGN DEVELOPMENT 1

2 MANAGEMENT SUMMARY 2

3 CONCEPTUAL DESIGN 2

3.1.1 Design Requirements 2

3.2 MISSION OUTLINES 3

3.2.1 Mission 1: Fire Bomber: Heavy Lift, Slow Flight 3

3.2.2 Mission 2: Ferry: High Speed Flight 4

3.3 MISSION SCORING 4

3.4 DESIGN CONSTRAINTS 4

3.5 APPROACH USED IN CONCEPTUAL DESIGN 5

3.6 RESULT OF CONCEPTUAL DESIGN 6

4 AIRPLANE PRELIMINARY DESIGN 7

4.1 CRITICAL DESIGN PARAMETERS 7

4.1.1 Aircraft Length and Fuselage Size 7

4.1.2 Wing Area 7

4.1.3 Wingspan 7

4.1.4 Motor Size 7

4.1.5 Propeller 8

4.1.6 Battery Weight and Number of Cells 8

4.2 RESULTS OF PRELIMINARY DESIGN 8

5 AIRPLANE DETAIL DESIGN 9

5.1 AIRPLANE DESIGN SUMMARY 9

5.2 WEIGHT CALCULATIONS 10

5.3 WING DESIGN 10

5.3.1 General Considerations 10

5.3.2 Airfoil Selection 11

5.3.3 Wing Area 12

5.3.4 Taper Ratio 13

5.3.5 Aspect Ratios and Chord 15

5.3.6 Twist 16

5.3.7 Modular Design 18

5.3.8 Final Specifications of Wing 18

5.4 TAIL DESIGN 19

5.4.1 Center of Gravity and Radius of Gyration 19

5.4.2 Longitudinal Aerodynamic Coefficients 20

5.5 FUSELAGE DESIGN 20

5.5.1 Configuration: 20

5.5.2 Approach 20

5.5.3 Solution 20

5.6 PROPULSION 21

5.7 3-D MODELING 21

5.8 AIRCRAFT FINAL SPECIFICATIONS 22

5.9 DRAWINGS 23

6 AIRPLANE DESIGN ANALYSIS 27

6.1 LINEAR STABILITY ANALYSIS 27

6.2 NONLINEAR STABILITY ANALYSIS 28

6.3 PERFORMANCE 30
6.3.1 Construction of Drag Polar *30*
6.3.2 Available Thrust *31*
6.3.3 Maximum Rate of Climb *32*
6.3.4 Take-off Length *32*
6.4 MATERIAL SELECTION 33
6.4.1 Wing martial justifications and lay out *33*
6.4.2 Nose material justification and layout *34*
6.4.3 Fuselage (center section) Material type and layout *34*
6.4.4 Tail material types and layout *35*
6.4.5 Horizontal and Vertical stabilizers' materials and layouts *36*
6.5 OVER VIEW OF MATERIALS USED FOR EACH MAJOR COMPONENT 36
6.6 STRUCTURAL ANALYSIS 38
7 MANUFACTURING 41
8 TESTING PLAN 42
8.1 FLIGHT TEST CHECKLIST 42
9 CONCLUSIONS 43
10 REFERENCES 44
11 APPENDIX A: RATED AIRCRAFT COST 45
12 APPENDIX B: DENSITY CONSTANTS 45
13 APPENDIX C: MATLAB LINEAR STABILITY PROGRAM 46
14 APPENDIX D: MATLAB NONLINEAR STABILITY PROGRAM 48
15 APPENDIX E: ROOT LOCUS 50
16 APPENDIX F: WING CALCULATIONS 51
17 APPENDIX G: MASS PROPERTIES 52
18 APPENDIX H: PUGH ANALYSIS 54
19 APPENDIX I: MECHANICAL PROPERTIES OF BALSAWOOD 57
20 APPENDIX J: STRUCTURAL ANALYSIS 58

1 Executive Summary

This report provides an overview of the design, manufacturing, and testing procedures used in the development of the University of Arizona entry in the 2003 Design, Build, and Fly competition. The aircraft was designed to complete to designated missions: A fire fighter mission (heavy lift and slow flight) and a ferry mission (fast flight).

1.1 Overview of the Design Development

Throughout the design process, heavy emphasis was placed on simplicity of design. Rather than developing the aircraft from scratch, it was decided to make use of prior experiences and make use of these to save time and achieve a better design. Some of the group members have long experience in RC aircraft building and some have worked with similar designs before. It would therefore be natural to make use of the knowledge and experiences these persons have. There is also a wide range of internet pages that covers model aircraft building, and especially there are some pages dedicated to electric driven RC aircrafts. Early on it was decided that previous mistakes and experiences from AIAA design, build and fly competitions should be used to the fullest extent. Therefore old reports were recovered and members from prior competitions were contacted. The members of the local chapter of the AIAA also proved to be a valuable resource in the design process.

At an early stage the important factors of the design were identified, some of these were: high lift requirement, slow speed requirement, minimizing of drag, difficult cargo to carry (sloshing of water/change of center of gravity), motor requirements and structures requirement. The team senior members were then assigned their own responsibilities. The senior member led a team of lower class men in their design groups. The design groups developed their designs, and these were discussed at weekly meetings with the rest of the members. Some of this design required practical testing, while other parts were only done on paper. After some weeks the preliminary design were done, and a preliminary design was decided. This design was carried on to detailed design. Important factors in the preliminary design were: large wing area to allow slow flight, large aspect ratio to minimize drag, placing water tank at center of gravity to minimize the movement of the center of gravity when deploying water, ease of maintenance and part exchangeability hence a modular design and use of cheap lightweight materials instead of expensive and exotic polymers and composites. The last criterion was set to avoid unnecessary expensed and added complexity of manufacturing. During this process configurations ranging from flying wings to traditional high wing aircrafts were investigated, but it was early on concluded that the traditional configuration was the most favorable.

The detailed design took up most of the time from end September to mid December. Important factors in this process were selection of materials, configuration, analyzing strength and performance of component and drawing of components in SolidWorks. To help in this process theory and routines were programmed up in Microsoft Excel and Matlab. These programs were used to analyze different configurations. Detailed analyzes of wings, structures and stability and control were central elements of this process.

The design was concluded in mid December, and since that point the senior members and the AIAA members have been involved in manufacturing the aircraft parts.

2 Management Summary

As described in the last section the senior group was given different responsibilities and led design groups consisting of AIAA members at the university. The responsibilities were:

Task	*Roger Aure*	*Lasse Mæland*	*Belinda Holmes*	*Mike Woodard*	*Dustin Wright*
Responsible person	Aerodynamics and wings	Stability and control, water container	Structures	Tail, overall design matching, team leader, materials.	Propulsion, landing gear, electronics and SolidWorks modeling

Table 2.1: Task break-down

The team had design meetings with the AIAA-members every Monday throughout the fall and the winter. Every Wednesday the team met with the faculty advisor Dr. Sergey Shkarayev to discuss the progress and problems/questions that might show up.

The tasks were assigned after interest, previous knowledge and need. The team leader was chosen to keep track of the overall development and coordinate the individual development. This was to insure proper interaction between the different designs and ensure an overall effective design. Table 2.2 shows how the design process was performed

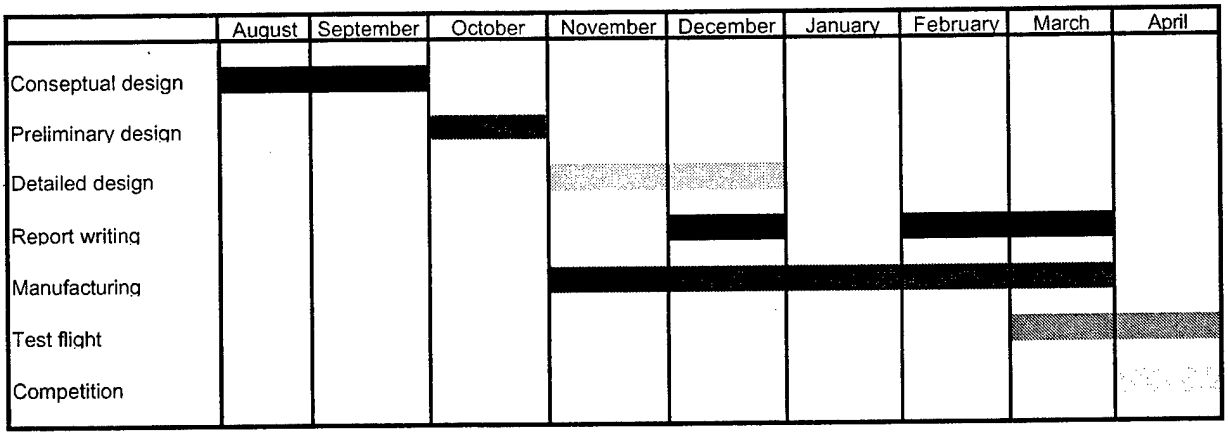


Table 2.2: Gant Diagram

Each team leader had the responsibility to develop an optimum sub-design for the overall design. This could only be done by regular contact with the other design groups. The main instrument to keep this contact was the regular Monday meetings.

3 Conceptual Design

3.1.1 Design Requirements

The AME 420 AIAA design team will design, build, and fly an aircraft tailored to the 2004 Cessna/ONR Student Design/Build/Fly Competition. Teams taking part in the competition must utilize their aircraft to complete in a maximum of five flight attempts, two different missions: a heavy lift, slow flight "fire bomber" mission and a high speed flight "ferry" mission. The aircraft, as specified by the competition rules, must fit into a 4' by 2' by 1' box employing innovative techniques for rapid assembly and disassembly. The runway distance for aircraft take off and landing is 150 feet. In order to obtain a score for an individual flight, a team's aircraft's wheels must clear the runway within that distance and the aircraft must land on the same runway; however, the competition allows for the aircraft to roll off the runway on landing. Competing teams will have a maximum of 10 minutes to complete each mission. Finally, teams must secure all payloads to the aircraft by mechanical means, which excludes such forms of restraint as tape and Velcro.

At the beginning of the competition and prior to any flights, teams must assemble their aircraft for safety inspections. The competition does not take into consideration the length of time required to assemble the aircraft and does not prohibit outside advice or assistance during the assembly process. Aircraft are to remain assembled while waiting in the queue, and teams are to install propulsion batteries once they reach the 3rd "On Deck" position. The team may provide a flight support crew of a maximum of five people: a pilot, an observer, and 3 ground crew members, who will have sole permission to reload the aircraft payload. The aircraft must complete its mission receiving only minor damages, which the competition defines as those damages that the team can repair within 30 minutes of the completion of the aircraft's flight. Finally, the aircraft must fly at a sufficient altitude for safe terrain clearance while maintaining visual contact with the operator and judges.

3.2 Mission Outlines

3.2.1 Mission 1: Fire Bomber: Heavy Lift, Slow Flight

This mission centers on the release of a liquid water payload in mid-flight over a designated region of the competition flight course shown in red in figure 1. Each team will load their aircraft with liquid water while on the runway, take-off within the designated 150 feet of the runway, fly along the designated competition course, perform a 360° retrograde maneuver while releasing its liquid water payload, and safely land the aircraft back on the runway. Teams may load, take-off, fly, dump, and land up to two times in the designated maximum of ten minutes. Teams must load their aircraft utilizing four 2-liter plastic soda bottles and may not implement any pressurization system to assist the loading process. The maximum allowed aircraft payload capacity is 4 liters. The aircraft can release its payload from an orifice no larger than 0.5" in diameter and must fly slow enough to allow sufficient time for the water to empty between the competition upwind and downwind turn markers or flags. Premature or delayed payload release as well as inadvertent payload release during landing will result in a 3-minute time penalty per occurrence. Excessive water spillage during the loading process will result in a 1-minute time penalty per infraction. In addition, incomplete payload release will result in a 3-minute time penalty upon completion of the final lap.

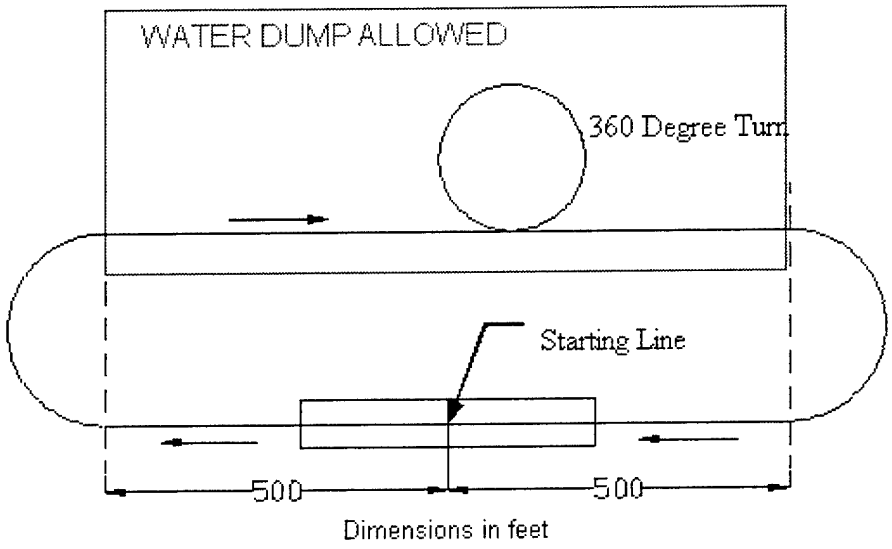


Figure 3.1: AIAA 2004 course map

3.2.2 Mission 2: Ferry: High Speed Flight

This mission awards the most points to aircraft that can complete the prescribed course in the shortest amount of time. Aircraft must take-off within the designated 150 feet of the runway, complete four laps around the competition course, which includes a 360° retrograde maneuver, and safely land the aircraft back on the runway. Aircraft will not carry a payload for this mission.

3.3 Mission Scoring

The competition assigns a difficulty factor to both the "Fire Bomber" and "Ferry" mission. Due to the additional difficulties of housing, flying, and releasing a liquid payload, the competition doubles the score of every team's "Fire Bomber" mission performance. The competition computes single flight scores differently for each genre of mission.

One calculates the "Fire Bomber" mission single flight score by dividing twice the weight of the payload in pounds, i.e. payload weight multiplied by the difficulty factor, by the total mission time in minutes, which includes all additional penalties incurred.

One calculates the "Ferry" mission single flight score by taking the inverse of the mission time, i.e. the difficulty factor divided by the mission time. The total flight score is the combination of the best single flight scores from both mission types. In addition, the competition takes into consideration the rated aircraft costs as described by the following equation:

Rated Aircraft Cost ($ Thousands) = (A*MEW + B*REP + C*MFHR)/1000

Where: A = Manufacturers Empty Weight Multiplier = $300
B = Rated Engine Power Multiplier = $1500
C = Manufacturing Cost Multiplier = $20/hour
MEW = Manufacturers Empty Weight (Aircraft weight [lb] with all flight and propulsion batteries, but no payload)
REP = Rated Engine Power = (1 + .25(#engines)) * Total Battery Weight

One computes the total competition score by utilizing the following equation:

Score = (Written Report Score * Total Flight Score)/ Rated Aircraft Cost
Equation 2

3.4 Design Constraints

There are no limitations to the aircraft configuration other than that it cannot have a rotary wing or be lighter than air. Next, the payload cannot be carried internal to the wing proper; however, for blended wing body configurations the inner 9 inches of semi span is considered fuselage and can hold payload. The propulsion system is limited to one or more electric motor/propellers combinations, from the Graupner of Astro Flight families of brushed electric motors. The propeller must be commercially available, and can be modified by clipping the tips or by painting for balance. It is acceptable to use commercial ducted fan units as well. Direct drive or gear or belt reduction is allowed. The maximum current allowed is 40 amps. This will be enforced by a 40-amp fuse in-line from the positive terminal of the current source. The current must be supplied from NiCad batteries. The batteries must have shrink-wrap or other comparable protection over the contact points. All disconnects must be "fully insulated" style connectors. No more than 5 lbs of batteries can be carried on the mission. This battery pack can only power the propulsion and payload systems. The Radio Rx and servos must be connected to a separate battery pack. The batteries cannot be replaced or charged between sorties during a flight period. Aircraft and pilot must be AMA legal, implying a TOGW less than 55 lbs. It is also a requirement that the plane has flown prior to the contest date.

The aircraft will be submitted to a safety inspection performed by a contest safety inspector prior to any test flights or competition flights. The inspection will encompass a physical inspection of the craft to ensure structural integrity. All components must be sufficiently secured to the aircraft by verifying that all fasteners

are tight and have either safety wire, locktite (fluid) or nylock nuts. All electronic wiring will be inspected to insure adequate wire gauges and connectors. Team must notify inspector of estimated maximum current draw. The radio range will be checked with motor on and motor off. All control surfaces must move in the correct sense. Also, the payload system will be submitted to general integrity check. The structural verification of the aircraft will be performed by lifting the aircraft by the wingtips. This will also be performed with the aircraft inverted. This test will require the complete payload of the plane. The center of gravity locations (both loaded and unloaded) must be marked on the fuselage. The radio fail-safe mode will be checked. The motor must have a mechanical arming system independently of the onboard Rx switch. The device must have a contest type "blade" fuse and be located in a manner allowing it to be accessed standing behind the propeller(s) for tractor aircraft. Fuses must be accessible from outside the aircraft. The aircraft Rx must be on and the throttle verified to be "closed" before the motor arming switch is activated.

3.5 Approach Used in Conceptual Design

Lack of testing time and design complexity were the primary shortcomings of the 2003 DBF teams plan. The results of the 2003 competition supported the idea of reducing complexity with relatively conventional aircraft placing highest. While some of the new ideas served to reduce drag, weight, or cost somewhat, the overall performance of the design usually suffered. For these reasons it was determined to focus on keeping the design as simple as possible. Not only will this produce a more reliable system of time-tested systems and configurations, but it will also allow more flying time to ensure that the design is optimized before the competition.

The main tools in the conceptual design process were brain storming and Pugh analysis. The design team started off with some meetings where all members came up with all possible solutions they could think of. These designs were organized after configurations. The main groups were based on engine configurations, these were: single-engine, twin-engine, pushers and tractors. Each of these main groups was divided into subgroups based on wing configuration, tail configuration and landing gear configuration. When everything was organized the Pugh analysis was utilized. Pugh analysis allows the qualitative comparison of several configurations by comparing a reference datum to the others for important design characteristics. A different datum is chosen as reference each time. After a few relative comparisons are completed trends usually develop indicating that one of the options meets the design criteria better than the other configurations. The completed Pugh analysis grids may be found in appendix H.

Besides the general configuration of the aircraft, the important physical factors in further development were identified. These factors were decided on the basis of the missions and the requirements from the competition rules. Simple math and physics were used to do calculations. Important factors that were considered were: take-off length, cruise speed, speed when deploying cargo, power available and cargo weight.

3.6 Result of Conceptual Design

The tables below show the result of the conceptual design.

Configuration	Description	Reason
Wing	High wing	Increased stability, ease of manufacture, easy to assemble/disassemble, cost.
Tail	Conventional tail	Ease of manufacture, easy to assemble/disassemble, prop wash over tail
Landing gear	Tricycle	Safety in take-off and landing, ground maneuvering.
Engine	Tractor	Easy to mount and maintain. Easier construction of tail

Table 3.1: Aircraft configuration

Driving parameters in further development		
Performance	Stall speed	~30 feet/sec
	Cruise speed	~45 feet/sec
	Take-off length	~100 feet
	Rate of climb	~6 feet/sec
	Total weight	~25 lbs
Weights	Payload	Prevent water sloshing
Stability	Power consumption	Change of CG. during water release.
		minimize
Endurance	Drag	minimize

Table 3.2: Driving parameters in further development

4 Airplane Preliminary design

The focus of preliminary design was to determine approximate dimensions, weight, power, and other physical characteristics for an optimum aircraft, which would best meet objectives for the two missions specified. The main areas of concern were wing area and span, empennage size, battery weight, motor/propeller combination, wing strength and the movement of CG when the water was drained.

It was decided to do this process as quantitative as possible. Realizing that using a quantitative method instead of crunching too many numbers would make the process more effective and faster was the major reason for this. The members in the design group and the AIAA members have enough experience with RC aircrafts to do this process quantitative. A qualitative approach would be a time consuming process. Arriving faster at a preliminary design and spend more time on detailed design was considered to be important in this phase to optimize the aircraft as much as possible.

4.1 Critical Design Parameters

Based on results obtained in the conceptual design phase, the critical design parameters were revised to reflect the performance parameters considered most influential to the overall score. These included main wing dimensions, power plant, empennage dimensions and battery weight.

4.1.1 Aircraft Length and Fuselage Size

The main purpose of the fuselage is to carry the payload, the batteries and the electronic equipment. Therefore the fuselage has to be sized to carry at least 4 liters of water and the batteries. This gave a needed volume of the center section were it was natural to have both these payloads placed. The overall length of the aircraft is dependent of the wings and which moment they create. Due to this the overall length has to be decided taking the wings into consideration. Another important factor is the storage of the aircraft. The overall length can not be greater than the storage box, or it has to be divided into modules that fit in the box.

4.1.2 Wing Area

For a fixed gross weight, decreasing wing area increases wing loading and raises the minimum drag airspeed. This results in higher flight speeds and reduced mission flight times. However, increasing wing loading also increases the stall speed, which increases takeoff distance. Since mission requirements call for short take-off, high flight speed and slow flight speed when draining water, wing loading must be a compromise. The conclusion from the design group was to go for a relative high wing area to reduce stall speed and hence increase safety factors in flight.

4.1.3 Wingspan

For a given wing area, a higher wingspan produces more efficient, faster wing while increasing the bending moment enforce at the root of the wing. A higher wings span also reduces the induced drag. Since available power is a major issue in this design, drag reduction became an important issue. It was therefore decided to go for a large as possible wing span. Other limitations on wingspan were manufacturability, storage of the wing in the aircraft storage box, and speed of assembly.

4.1.4 Motor Size

Motor selection was based upon two factors: costs and needs. As the design team had a fairly small budget and wanted to keep the costs down, this was an important factor. However, now compromises that would lead inadequate performance were accepted. The motor choice was made on the basis of this. Another important factor was that there were several motors available from previous years design build and fly competitions. The final choice was to use one of these motors.

4.1.5 Propeller

The propeller choice was made after the motor choice. Due to the mission specifications a lot of power is required. The power is a function of both the motor and propeller. When the size of the motor was done propellers that would give as much power as possible were researched. Other considerations that were made were need for acceleration and maximum speed. The propeller diameter was also a major factor since this affects the undercarriage and ground clearance.

4.1.6 Battery Weight and Number of Cells

The maximum allowed battery weight is 5 lbs and the maximum allowed current is 40 A. This formed the basis for the preliminary choice of number of batteries and battery types, hence the battery weight. Several batteries were investigated and compared, before one type of batteries was selected.

4.2 Results of Preliminary Design

The preliminary design process resulted in a more refined and detail aircraft specification than after the conceptual design process. The results are presented in the table below. These results were utilized in the detailed design process which occupied a major part of the design process.

Design Parameters After Preliminary Design		
Wing	Wing Area Wing Span	10~13 ft^2 9~12 ft
	Wing Profile	High lift/low drag/good stall properties
	Taper Wing Layout	Yes 3 sections
Fuselage	Fuselage length Fuselage width	5~7 ft 5~7 inches
	Fuselage height Fuselage layout	8~10 inches 3 sections
Tail Motor/prop. Undercarriage	Tail Motor Propeller Undercarriage	Conventional Astro Flight Cobalt 60 14~16 in. diameter, 4~6 in. pitch Tricycle

Table 4.1: Results of preliminary design

5 Airplane Detail Design

5.1 Airplane Design Summary

Table 5.1 summarizes all calculated or assumed design parameters for the aircraft including payload. Subsequent sections summarize the design process utilized in obtaining these numbers and provide the reasoning behind all assumptions.

m = mass	$?_8$ = density of air	c = wing chord
h_{cg} = distance to center of gravity	distance between neutral point and center of gravity of wing	h_{acw} = distance to aerodyn. center of wing
b_t = span of tail	b_e = span of elevator	S_w = wing surface area
AR_t = tail aspect ratio	AR_e = elevator aspect ratio	k_y = radius of gyration
a = aircraft lift slope curve	α = aircraft angle of attack	α_w = wing angle of attack
$K_?$ = wing-taper-ratio factor	K_H = horizontal-tail-location factor	(de/da) = downwash gradient at horizontal tail
$U_{8\,trim}$ = trim velocity	M = Mach number	Re = Reynold number

Table 5.1a: Parameters used

c_t = tail chord	c_e = elevator chord	h_N = location of neutral point
h_{act} = distance to aerodyn. center of tail	distance between aerodynamic centers of tail and wing	b_w = span of wing
S_t = tail surface area	S_e = elevator area	AR_w = wing aspect ratio
a_w = wing lift slope curve	a_t = tail lift slope curve	a_e = elevator lift slope curve
Θ_T = thrust angle	? = taper ratio	K_A = wing-aspect-ratio factor
$(a_w)_0$ = wing angle of attack at zero lift	$(C_m)_0$ = coefficient of moment at zero lift	i_t = tail set angle

Table 5.1b: Parameters used

5.2 Weight Calculations

In the initial airplane sizing, the team summed the chosen components to establish the preliminary airplane weight. Table 5.2 summarizes mass contributions from all components considered in the team's calculations and models:

Component	Mass (kg)
Motor	0.84
Payload	4
Wings	2.5
Nose	1.27
Batteries servos & receiver	2.3
Tail	0.6
S	11.51

Table 5.2: Mass Distribution

5.3 Wing Design

5.3.1 General Considerations

There are several considerations that must be made when designing the wing. The factors that are critical in our design process must be identified, and that eventually will be the deciding factors in the design. For the aircraft and the mission it is design for, the most important limiting factors are:

1. The aircrafts is propelled by an electrical motor.
2. The aircraft must be able to take off in a distance of 150 feet.
3. For the part of the mission the water is dropped, the aircraft has to be able to fly slowly enough for 4 liters to drain from the tanks.
4. The aircraft must fit in a 4'x2'x1' box, assembled or not assembled.

By analyzing these four factors, the impact on the design can be identified.

The fact that the aircraft is propelled by an electrical motor means there is a limited amount of power available. This limits the maximum thrust available and how long the thrust will be available. A large weight of batteries must also be carried to give power to this motor. All these factors result in the conclusion that the aircraft should have as little drag as possible to best make use of this power.

The take-off requirement is crucial to the mission. Failure to meet this requirement will result in a score of 0 in this part of the mission. The natural conclusion from this factor is that the wing has to be able to produce high lift at low airspeeds, or the aircraft must accelerate quickly. The last alternative excludes itself since we have limited power available. This leaves the high lift option. To meet this requirement three reasonable alternatives are available: a high lift wing, high lift devices for take-off/landing and a large wing area, or a combination of these.

Part of the mission is to dump 4 liters of water during a downwind leg with a 360 degree turn. To be able to do this we have to assure that the aircraft can fly slowly enough to allow 4 liters of water to drain during this sequence. This result in almost the same alternatives as above: a high lift wing, high lift devices and a large wing area. Since in this case the aircraft does not need to accelerate, it is important to be able to maintain good control over the aircraft. This basically means that if the aircraft stalls, it should be stable in stall and be easy to recover fom stall. These properties are also important in take-off and especially landing. To meet this requirement there were two alternatives: use an airfoil that stalls gradually or twist the wing to ensure that is stalls from root to tip.

To fit in the transport box the wing cannot be more than 4' in span or it have to be made up of sections of no more than 4 feet each. It is easy to see that a for a aircraft of 25 lbs W_{TOmax} a 4 foot wing would require a

enormous chord to be able to get into the air without having a huge engine. This means that the wing have to be made up of sections.

Since the aircraft is going to operate at low airspeeds it is important that the airfoil have good properties at low Reynolds numbers. This will greatly impact the choice of airfoil.

To sum up from these general considerations the following conclusion can be drawn:

1. Airfoil with high $C_{Lmax.}$
2. Airfoil with high C_L/C_D at angles of attack for cruise
3. Airfoil with good low Reynolds number properties.
4. Airfoil with gradual transition to stall.
5. The wing has to have a pretty large area.
6. Possibilities to incorporate high lift devices such as flaps.
7. Drag reduction measures → tapering of wing, aspect ration, winglets or laminar flow airfoil.
8. Twist of wing to improve stall properties.
9. Sectional wing

5.3.2 Airfoil Selection

Selection of airfoils is a science in itself. There are a huge number of airfoils available to choose from, or you could design your own airfoil. To design your own airfoil would mean enormous amount of work and testing. There is no need to do all that work as long as there are a lot of good airfoils available. Besides, designing a new airfoil only pays of when you are trying to achieve some special optimization such as a 1% reduction of drag reduction on a transport jet.

RC-aircraft have been built for almost a hundred years. There is plenty of expertise available to consult and most airfoils have been tried at model aircrafts. By using own experience, books on model aircraft design, contacting experienced RC-aircraft builders and utilizing the program X-FOIL, the airfoil choice was done. From all the airfoils evaluated, an airfoil that best met the following criteria was chosen:

1. High $C_{Lmax.}$
2. High C_L/C_D at angles of attack for cruise (0-5°)
3. Good low Reynolds number properties.
4. Gradual transition to stall.
5. Thickness > 12% (to allow structural strength).
6. C_m > -0.15

The result from the selection was the NACA6412. This airfoil meets all these criteria and it is also been used successfully on low airspeed RC-aircrafts before.

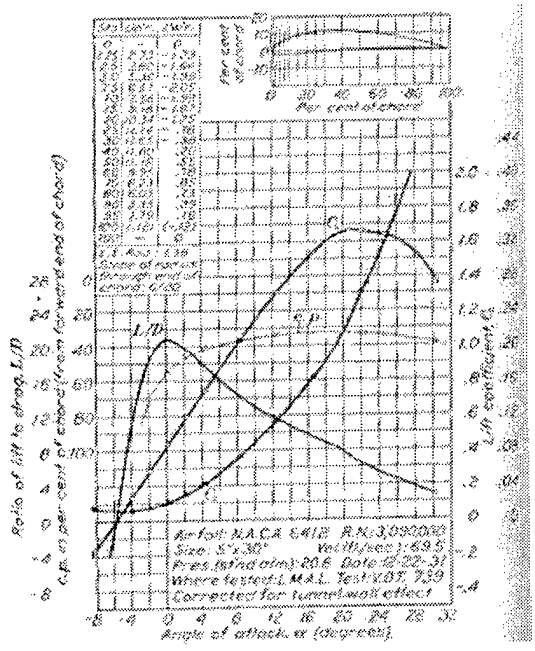
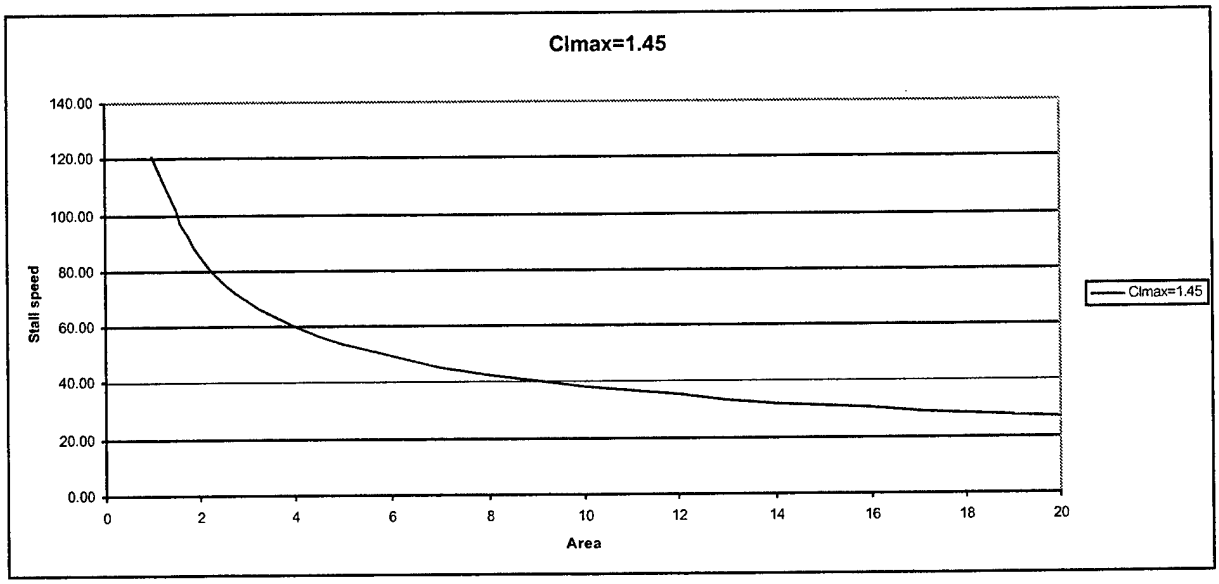


Figure 5.1, NACA 6412 Polar [1]

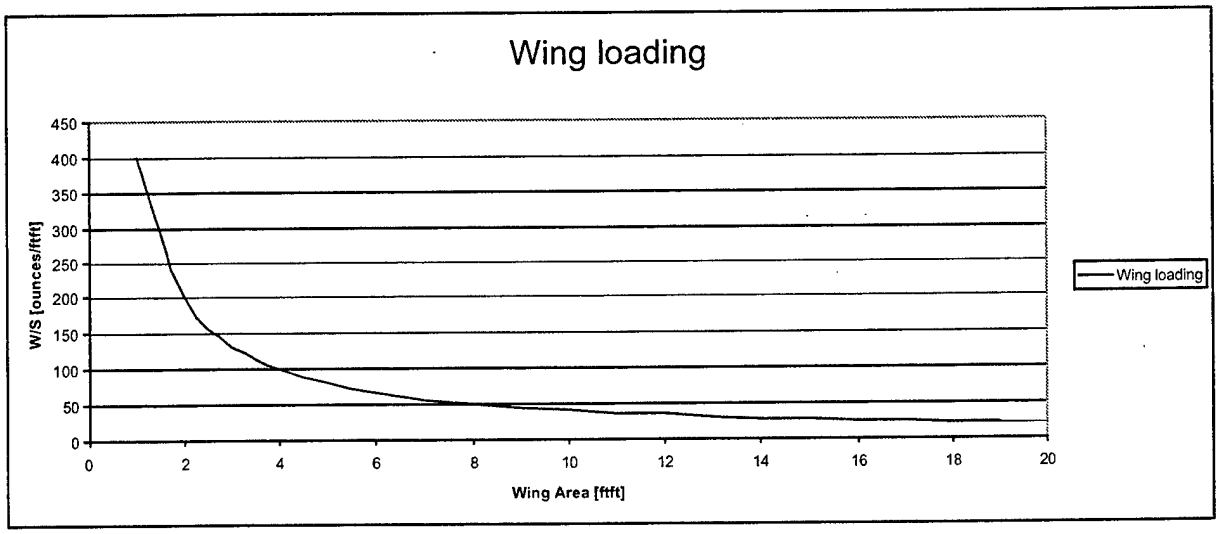
5.3.3 Wing Area

After choosing an airfoil the next step in the design process was to choose a wing area. The wing area was chosen on the basis of our maximum desired stall speed (~35 feet/sec) and the 2-D C_{Lmax} of the airfoil for a Reynolds number of 500 000. When performing these calculations the wing area was chosen at about 12 ft^2.

The wing load was also an important factor in this decision, as recommended by Andy Lennon [Ref.3] we desired to keep the wing loading below 45 ounces/ft^2



Graph 5.1 , Stall Speed vs. Wing Area



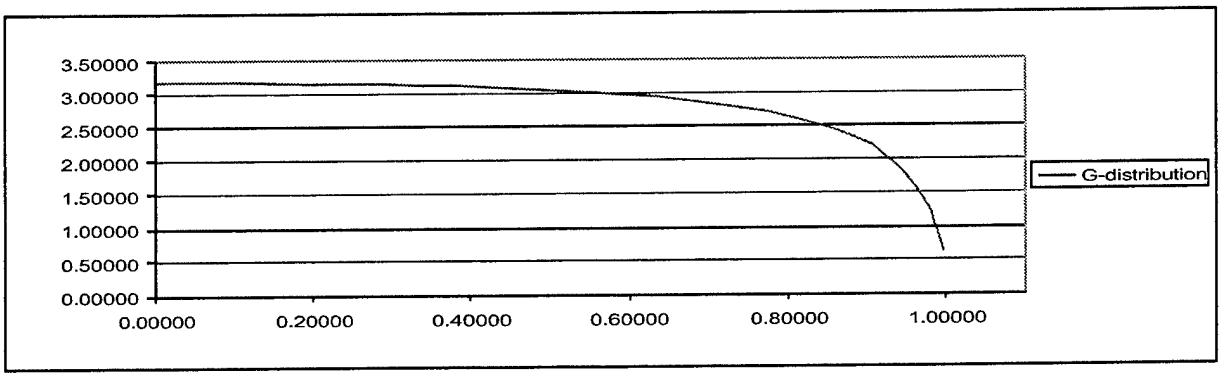
Graph 5.2 , Wing Loading vs. Wing Area

5.3.4 Taper Ratio

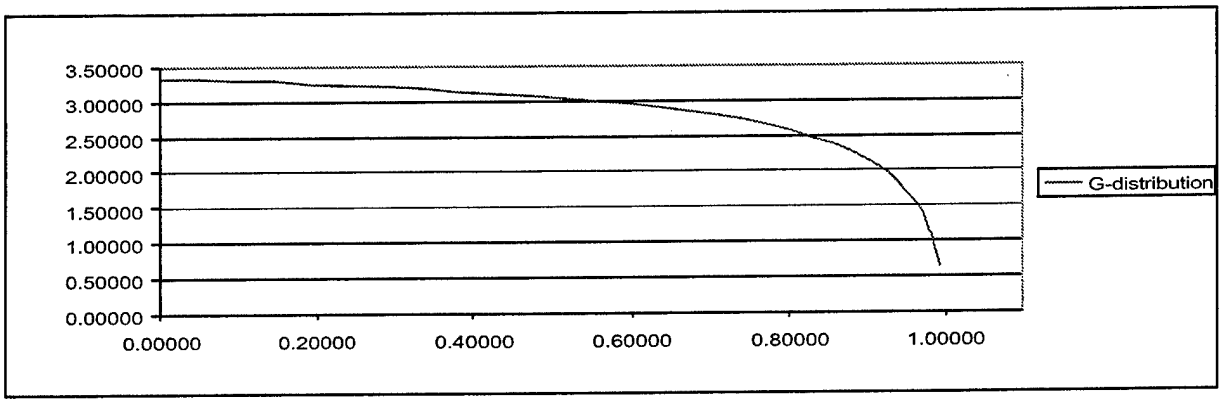
A wing that has an elliptical load-distribution is proven to have the least lift induced drag. To have a wing with an elliptical load-distribution it is necessary to build a wing of elliptical plan form. This is very hard to do structurally and would involve a substantial amount of time. However, by incorporating a taper of the wing, almost the same lift distribution as by having an elliptical wing would be achieved. A close to constant downwash velocity along the span would be the result; hence reduce the lift-induced drag to a minimum.

To investigate the effect of incorporating a taper in the wing among other things, Prandtl Lifting-Line theory was utilized. Microsoft Excel was utilized to program up these theoretical methods. In this program, airfoil properties, atmospheric properties, aircraft weight, taper ratio, twist of the wing and the angle of attack of the wing can be entered. These variables can all be changed to analyze the effect of the changes.

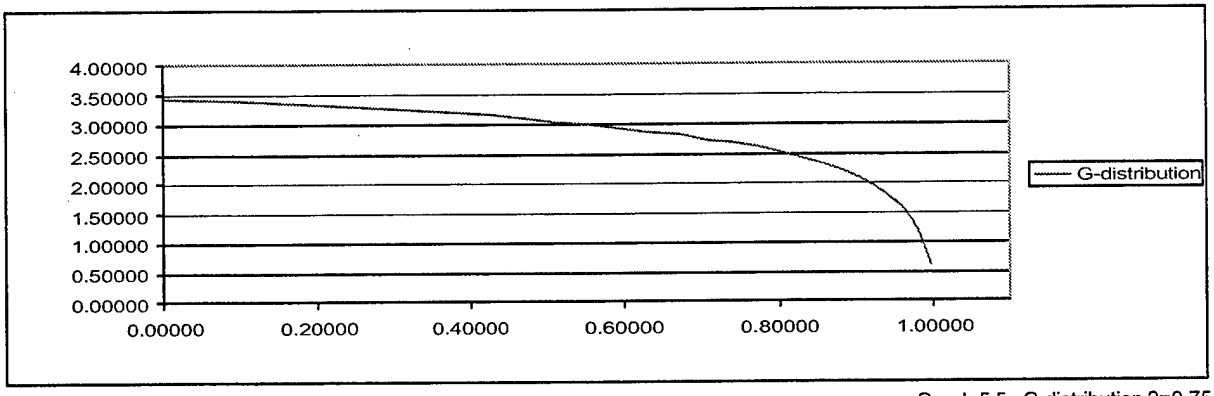
Several taper ratios were analyzed to see how the drag was reduced. The final choice was to go for a taper of 0.75. It is proven that a taper of 0.65 [3] gives the load distribution that is closest to an elliptical load, hence giving the least lift-induced drag. However the more taper we incorporate, the larger root chord or wingspan we have to use to maintain the same lift of the wing. A larger span is beneficial to reduce drag, but makes the wing harder to build structural without using expensive or heavy materials. A larger span also means problems with transportation (ref. box requirements). A larger root chord means that the Reynolds number will increase. To reduce drag it is desired to keep the Reynolds number below Re_{crit} to ensure a laminar flow as possible. Considering all these criteria and that the change in taper ratio from 0.75 too 0.65 would not give much gain in reduction of drag.



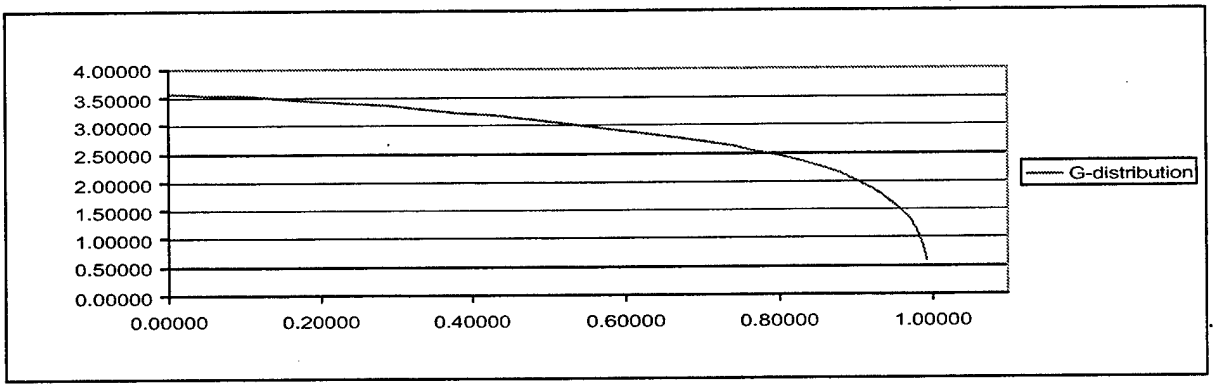
Graph 5.3 , G-distribution ?=1.0



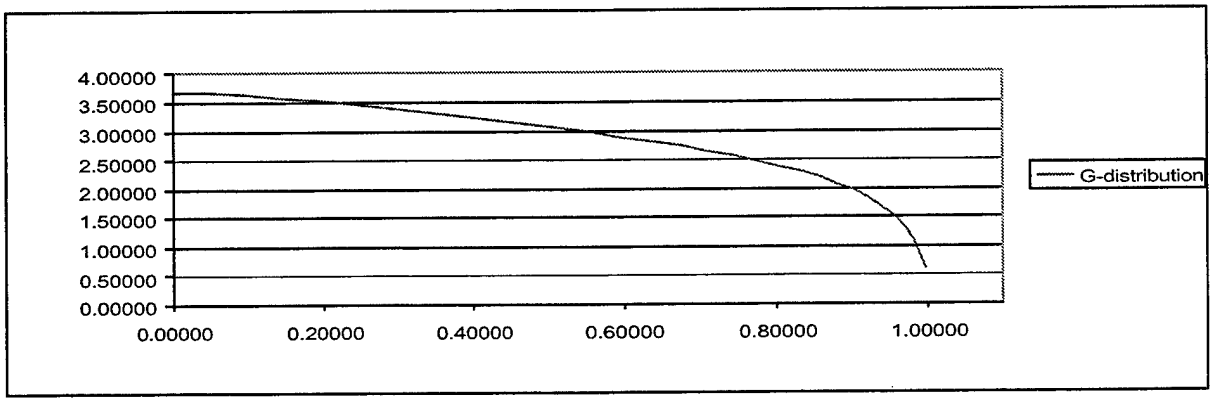
Graph 5.4 , G-distribution ?=0.85



Graph 5.5 , G-distribution ?=0.75



Graph 5.6, G-distribution ?=0.65



Graph 5.7, G-distribution ?=0.55

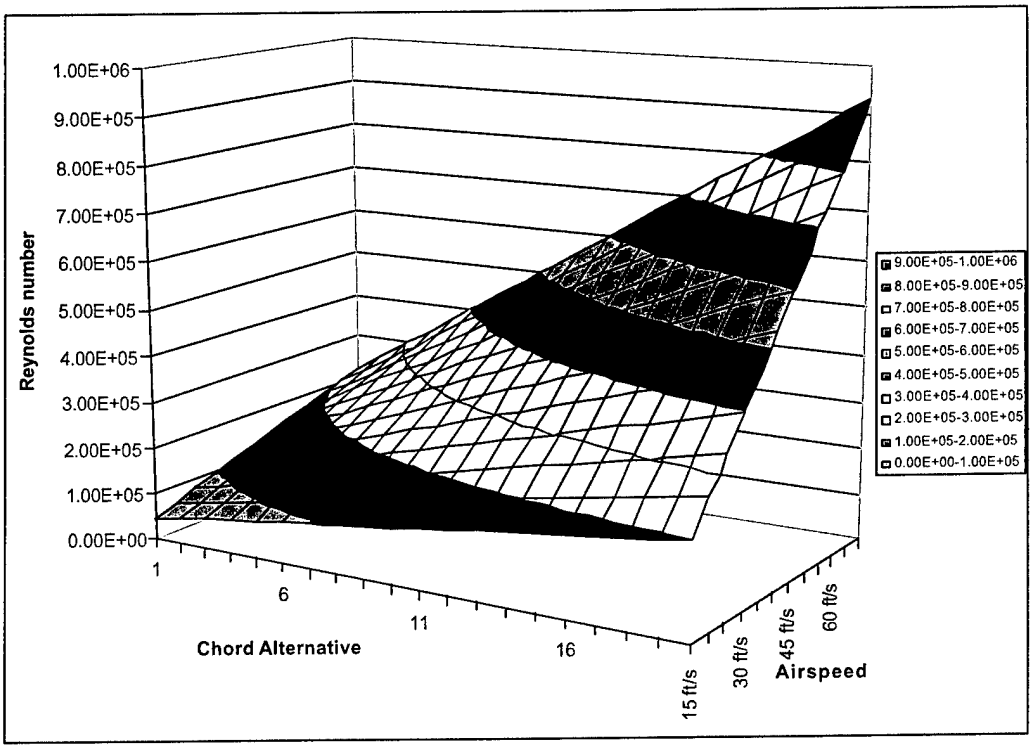
5.3.5 Aspect Ratios and Chord

The aspect ratio considerations have briefly been mentioned before. There are three major considerations to make:

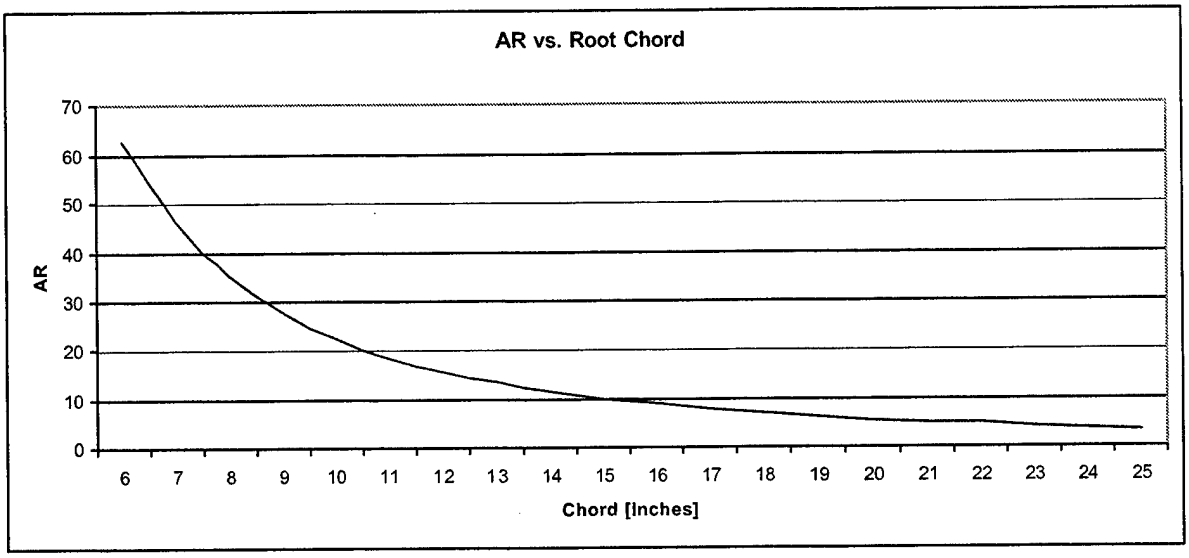
1. The larger aspect ratio, the less lift induced drag.
2. The lesser aspect ratio, the larger chord for the same wing area, which means higher Reynolds number.
3. A larger aspect ration means that the wing is more difficult to manufacture ensuring structural strength without using expensive or heavy materials.

As the goal is to keep the drag at a minimum we decided to go for an aspect ratio as large as possible. Investigating other successful designs from the AIAA Design, Build & Fly contest, revealed that most of them had an aspect ratio of about 10. When looking at successful new commercial jets, that owes much of its success to increased fuel efficiency due to reduced drag; it was revealed that most of them had increased the aspect ratio from previous transport jets. Most of them now had an aspect ratio of about 9. Based on own experience an aspect ratio in that area is typical for RC-aircrafts of this configuration

The selection of aspect ratio was also done by considering the chord of the wing. From selection of taper ratio it was known that the tip chord was going to be 75% of the root chord. The aspect ratio then had to meet both the requirement of wing area and taper ratio (in other words chord). It was desirable to keep the flow over the wing as laminar as possible. The chord was therefore selected such that the Reynolds number would stay beyond Re_{crit} for the desired cruise speed at 45 ft/sec, and that an aspect ratio of 9-10 was possible to obtain.



Graph 5.8, Reynolds number as a function of airspeed and chord length



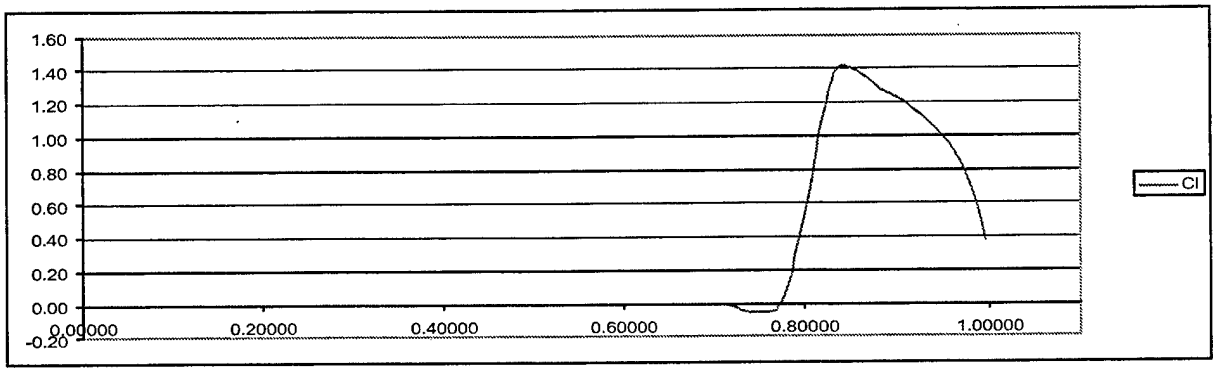
Graph 5.9, Aspect ratio as a function of chord length

5.3.6 Twist

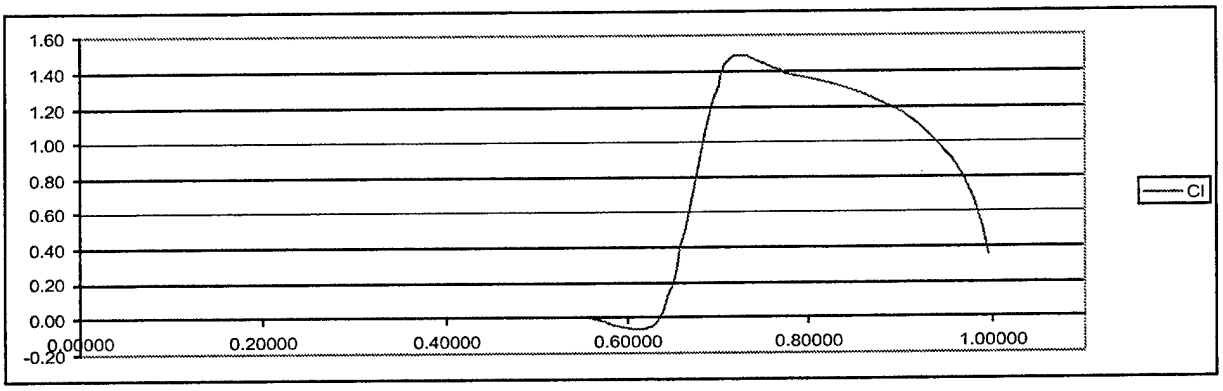
The aircraft will in some segments of the flight operate close to the stall speed. It is therefore crucial that the aircraft behaves stable in stall. There are several ways of achieving this: the wing can be twisted, use a computer to prevent stall/initiate stall recovery and an airfoil that stalls very nicely can be used. Usage of a computer (autopilot) is too expensive, so that was quickly ruled out as an alternative. An airfoil that behaves well in stall has already been chosen. To further improve the stall properties the twist option was incorporated as well.

By twisting the wing in such a way that the tip of the wing has a less angle of attack than the root of the wing, the wing stalls from root to tip. This lead to a stall pattern that is gradual, and control over the ailerons is maintained as long as possible. There will not be any surprises when the aircraft stalls and the pilot will be able to detect the stall before it get so bad that he/she will not be able to recover from it without substantially deviating from normal flight procedures. The risk of having the tip of the wing stall and send the aircraft into a potentially destroying spin, will also be reduced.

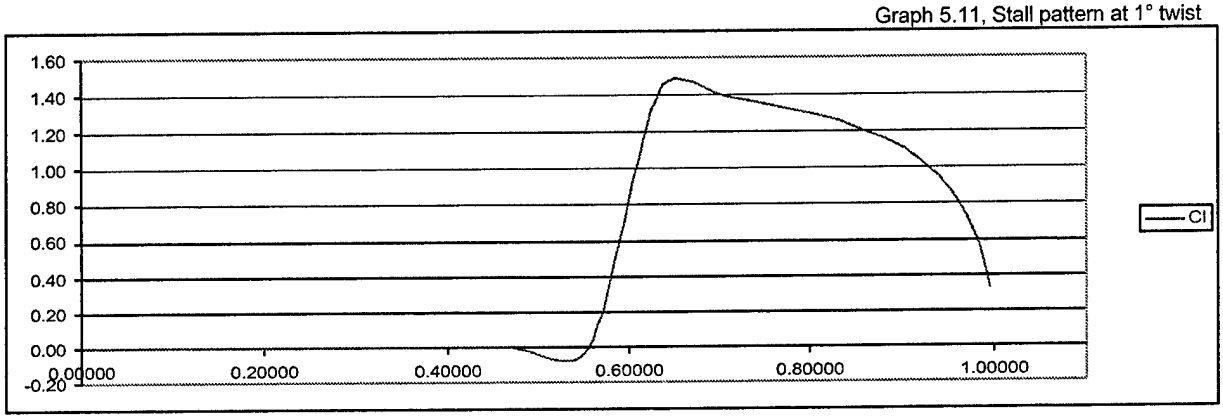
To investigate what the ideal twist angle was, the Excel program was utilized. By adjusting the wing to stall angle without any twist, the twist angle was gradually changed until the wing did not stalled in the area the aileron would be situated. The result of this analysis was to use a twist of 3°.



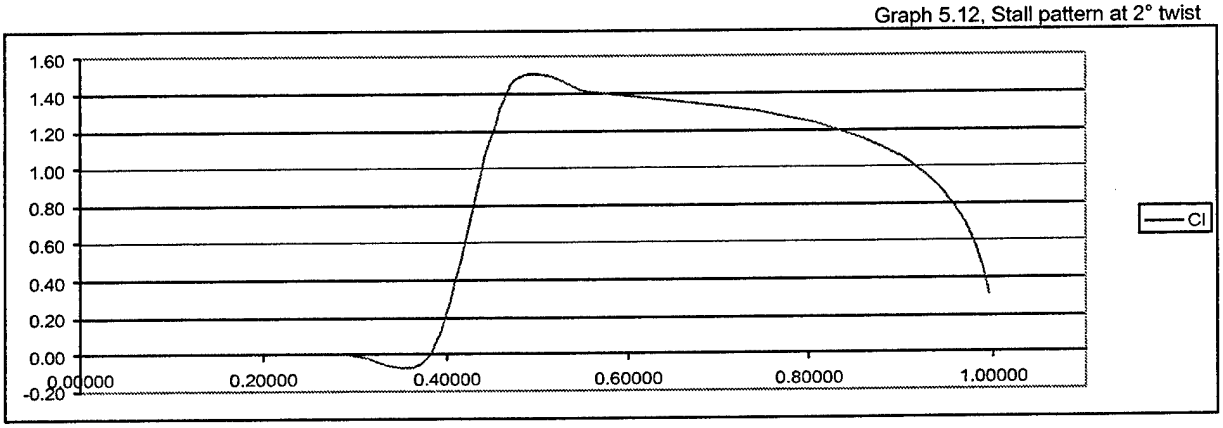
Graph 5.10, Stall pattern at 0° twist



Graph 5.11, Stall pattern at 1° twist



Graph 5.12, Stall pattern at 2° twist



Graph 5.13, Stall pattern at 3° twist

5.3.7 Modular Design

To be able to keep the wingspan chosen and transport the wing in the required box, a modular design was incorporated. The wing will be divided into 3 sections: a right and left outer section and a center section. These sections will all be less than 4 feet in length. By doing it like this the outer and center sections can be kept as separate designs. These sections can be altered separat ely from each other, allowing to experiment with different designs if needed. This is especially relevant to the center section. There are no plans to use flaps initially, but if these are needed to be able to fulfill the take-off requirements, they can easily be incorporated in the center section without having to build the whole wing from scratch again. The center section is also intended to seal of the water tank, and will blend with the fuselage of the aircraft.

The section design also gives us greater flexibility and operability. Different wing designs and wing features can be tested and if for some reason break a part of the wing, it can easily be replaced without having to make an entire new wing. This will save time and money.

5.3.8 Final Specifications of Wing

Specification		Value
Airfoil		NACA6412
	C_{Lmax}	1.45 (Re = 500 000)
	C_{La0}	0.6947
	C_{m0}	-0.145
	a_{0l}	-6.5°
Wingspan (b)		11 feet
Taper ratio (?)		0.75
Root chord (c_{root})		15 inches
Tip chord (c_{tip})		11.25 inches
Wing Area (S_w)		12.5 Sqr. feet
Twist		3°

Table 5.3: AIAA 2004 course map

5.4 Tail Design

In determining the area of the airplane's tail, Equation 1 was utilized, it is derived from empirical data for RC aircraft and found in Reference 3:

$$S_t = \frac{0.2 \times S_w \times 2.5}{(h_{act} - h_{cg})}$$

Equation 1.

(where S_t and S_w are measured in in^2)

Given the area of the tail, S_t, the tail aspect ratio, AR_t, was set equal to 5 based on the value suggested in Reference 1, and calculated both the span and chord of the tail. Furthermore, the AR_t of the tail was used to calculate the lift-curve-slope of the tail, a_w.

5.4.1 Center of Gravity and Radius of Gyration

Initially, a value of 0.0175 was utilized for the location of the center of gravity, h_{cg}, based on a suggested value of .05c to .075c forward of h_{acw} from Reference 1. Using this value to calculate the neutral point, h_N, the tail down set angle, i_T, was calculated to be approximately 9°. It was decided that this high of a value for the tail down set angle would produce too much drag on the plane, therefore the center of gravity was moved to reduce the static margin, $(h_N - h_{cg})$ thus reducing the tail down set angle to a more optimal value that fell between 3° and 4°.

In order to establish the mass distributions of the airplane components thus defining the radius of gyration for the airplane, k_y, the necessary value for the radius of gyration was calculated, so that the airplane flight characteristics would provide acceptable frequency and damping of the short period. The team utilized Equations 2-5 to establish this relationship.

$$\zeta_{S.P.} = \sqrt{\frac{a_t}{\mu(h_N - h_{cg})}} \frac{k_y}{\bar{c}}$$

Equation 2.

$$\zeta_{Ph} = \frac{1}{\sqrt{2}} \frac{C_{d0}}{C_{l0}}$$

Equation 3.

$$\omega_{sp} = \left(\frac{\bar{c}}{k_y}\right) \sqrt{a_t (h_N - h_{cg}) \frac{g}{\bar{c} C_{L0}}}$$

Equation 4

$$\omega_{ph} = \sqrt{2} \frac{g}{u_0}$$

Equation 5

Utilizing the developed Solid Works model the actual radius of gyration for the aircraft was established This value was then implemented, 0.24 m, into Equations 25 and compared it to the desired values previously calculated. It was found that the frequency of the short period is 1.76 Hz, the frequency of the phugoid 0.094 Hz, the damping of the short period is 0.88, and the damping of the phugoid is 0.11.

5.4.2 Longitudinal Aerodynamic Coefficients

Under the assumption that the moment equals zero in steady level trimmed flight and utilizing known values for S_t, S_w, a_w, a_t, h_{cg}, h_{acw}, h_{act}, de/da_w, and a, a Matlab program was constructed to calculate the coefficient of moment at zero lift,$(Cm)_0$, the neutral point, h_N, and the tail down set angle, i_T. This program was then used to verify that all design conditions were met for the given wing, tail, and fuselage design as well as component mass distributions. Appendix B contains this Matlab program.

5.5 Fuselage Design

5.5.1 Configuration:

Several different configurations were considered in order to optimize the aircraft design. The requirement that the aircraft must fit into a box of prescribed dimensions drove the decision to implement a sectioned wing. This design allowed for a large wing span that produced desirable flight characteristics. Furthermore, in order to complete the mission, to carry and drop 4 kg of water, the aircraft design must include a watertight compartment with the ability to drain during flight on command. In addition, the design must compensate for the large, heavy battery pack needed to propel the aircraft. Furthermore, it was decided to implement a conventional tail into the design for reason of simplicity, i.e. a conventional tail requires no servo mixing and provides for easy construction.

5.5.2 Approach

It was decided to approach the problem with a modular design. This would lend itself to easy assembly and disassembly, thus alleviating the hassle of selecting an acceptable configuration to fit in the box. Several other features of this approach were also found very attractive. Modules lend itself towards easy repairs during test flight phase and competition. Also, one can quickly refit the plane with a different tail, or wing sections if the team decides to do so during test flight phases.

5.5.3 Solution

The stability of the aircraft is very sensitive to cg location so it was realized that it would be beneficial to put the batteries and water close to the desired cg location. In order to achieve this, a water tank with a tubular battery compartment inside was designed. This thus formed the central module that would transfer all loads between the other modules. Next a nose module with energy absorbing struts was conceived. The tail section was built as a standalone module. And the wing section connects as a module. This structure lends itself toward adjustments during test flight phase, which was one of the objectives of the design.

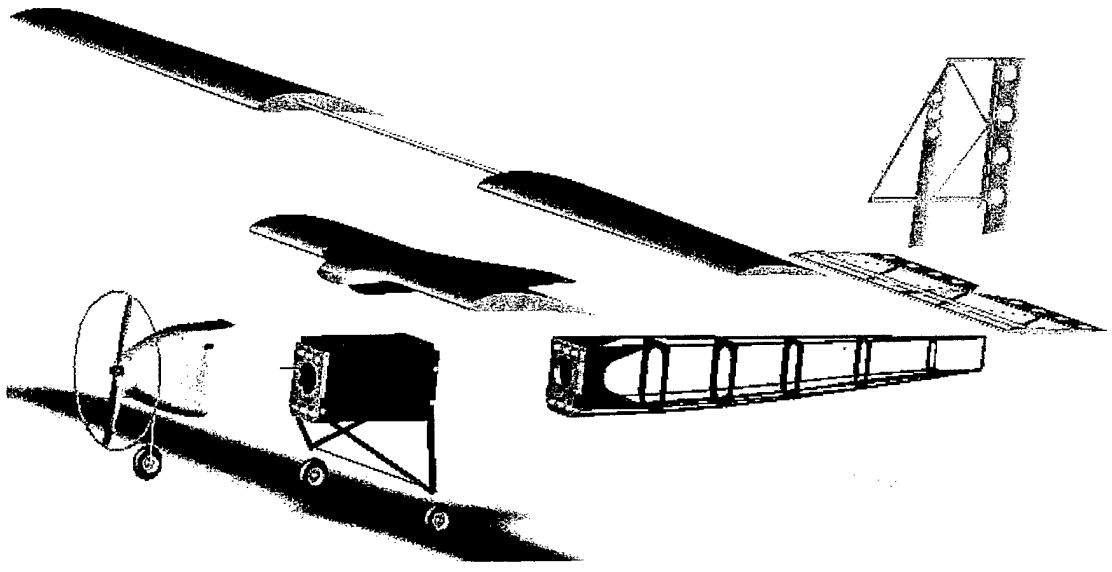


Figure 5.2 Modular airplane configuration

5.6 Propulsion

The propulsion of the aircraft is limited in two respects, the competition rules and cost.

The requirements for propulsion generated by the competition rules are as fallows:

Electric brushed motor
Maximum of 5lb. Ni-Cad batteries
Maximum current draw of 40amps controlled by a fuse

To minimize the cost of the propulsion system the team will reuse the electric motor used by previous DBF teams. The motor is an Astro Flight Cobalt 60 capable of spinning a 15 or 16 inch diameter prop easily. The pitch of the prop will be chosen by flight testing, however the high RPM generated by the electric motor will require a small pitched prop, probably a 2-3, however this may limit take off length, implying the necessity of a larger pitch 4-6. Flight-testing over a wide range of pitches will ensure the best compromise for our competition.

Energy will be stored for propulsion by the 5lb of available Ni-cad batteries. Sub C cells will grouped in sticks six batteries long, as seen in figure 17 below.

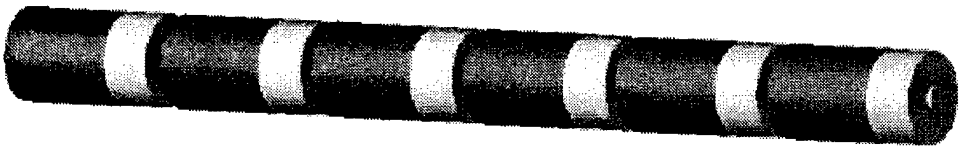


Figure 5.3: Battery pack

The sticks will then be arranged in a pack of seven, with a total battery count of 42. Each cell weighs 0.112 lb, which would allow us to carry a total of 45 cells, however we will only carry 42 because after assembling the pack extra undo weight is added.

Each cell is rated 1.2 volts for a total battery pack voltage of 50.4 volts. Assuming an electric motor efficiency of 80% and a propeller efficiency of 75%, there will be 1.65 Hp available for trust at take off.

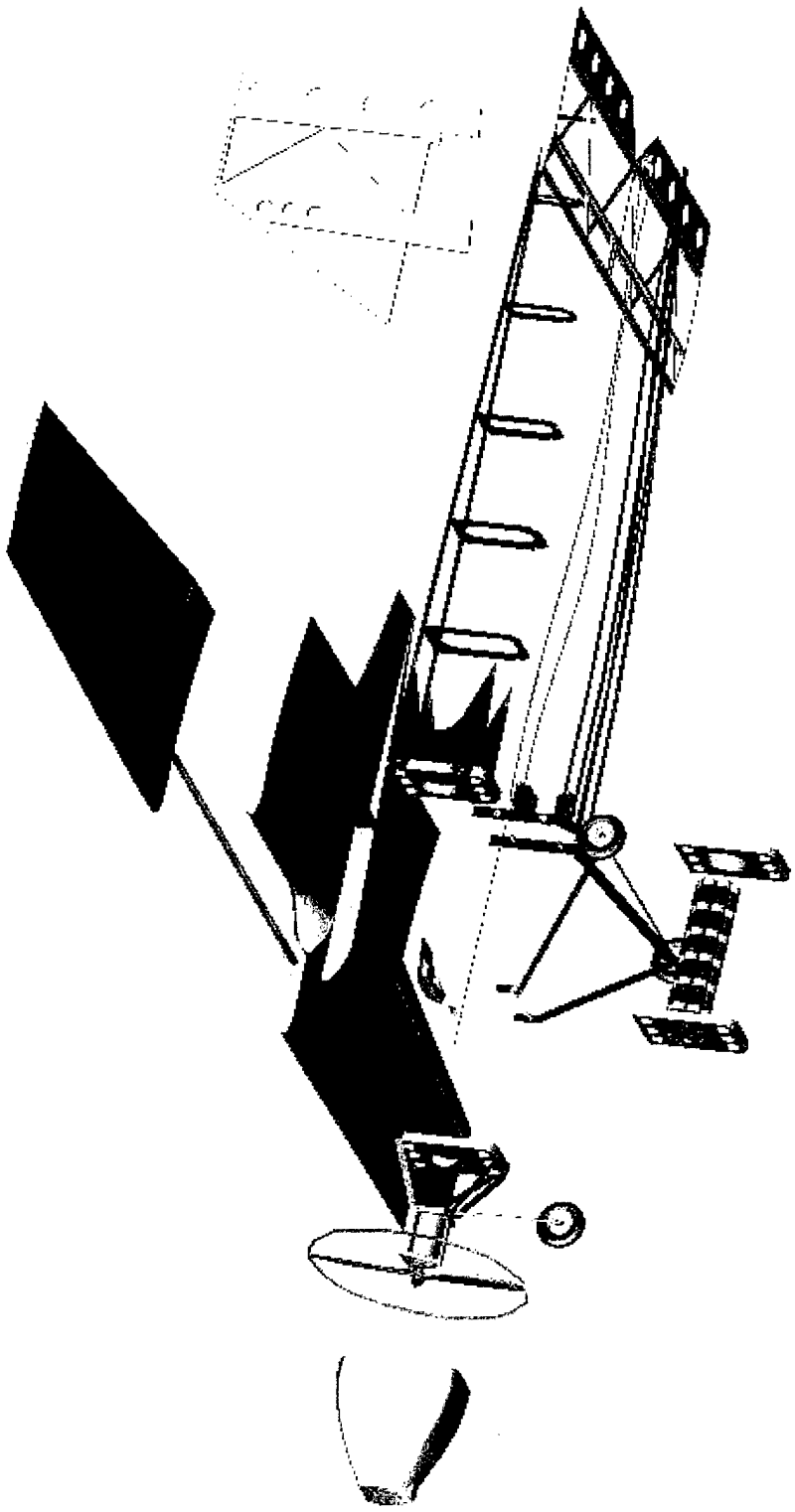
5.7 3-D Modeling

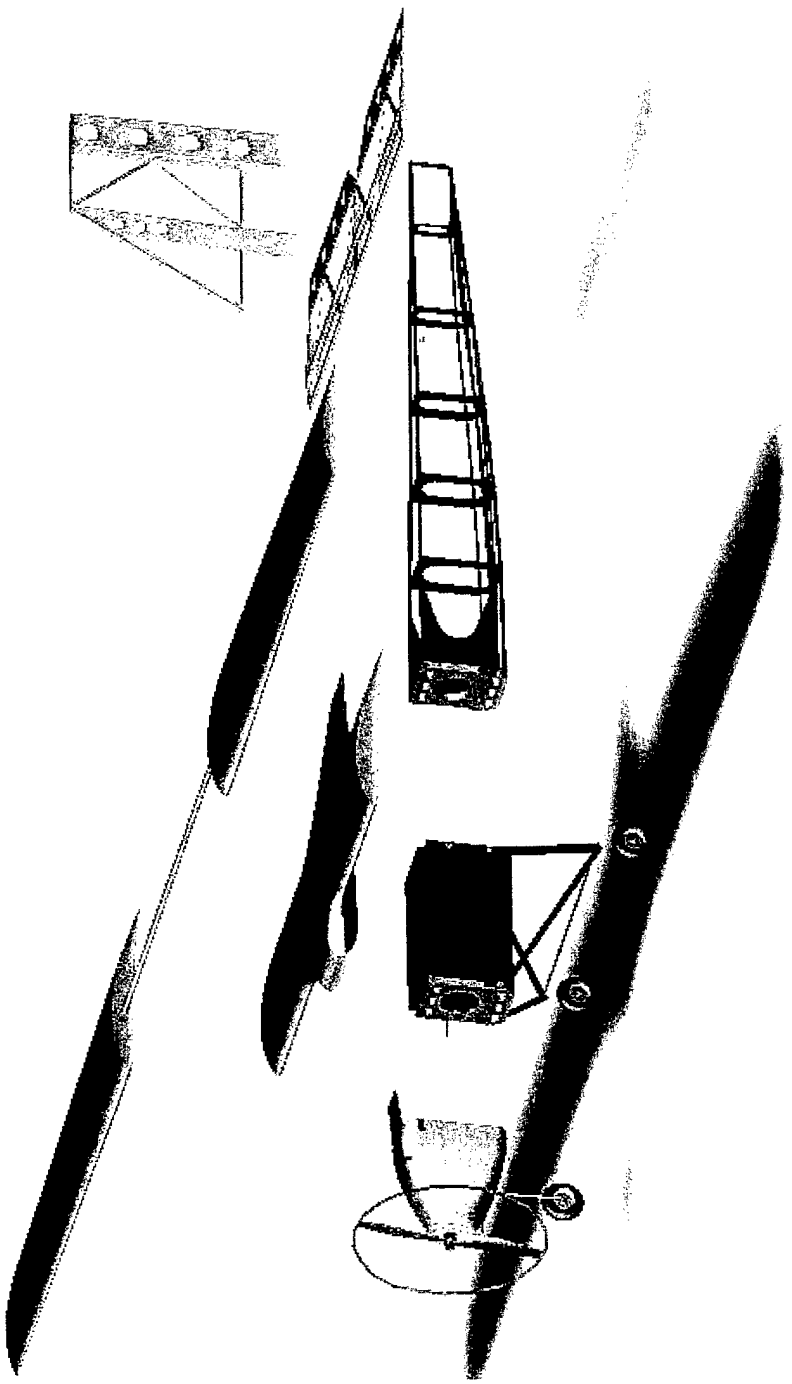
Once the fuselage, wing, and tail initial design had been completed, the entire aircraft was modeled using SolidWorks 3D CAD software. The model was made as accurate as possible down to the bulkhead cut-outs, individual NiCad cells, and even fasteners. Using density calculations the weight of the fuselage, tail, batteries, motor with mount, and landing gear were determined. A list of the densities used may be found in appendix A. The wing's weight was determined using a similar method, but its weight was calculated based on the amount of each material that would compose each cross section and the total wingspan. With the weight distribution of all critical aircraft components determined, SolidWorks is able to quickly calculate the necessary mass properties to begin the stability analysis and refine the subsystem designs in both filled and empty payload conditions. Remarkably, the total takeoff weight found using this much more detailed weight analysis method was 25.5 lbs, less than a pound different than the initial estimate.

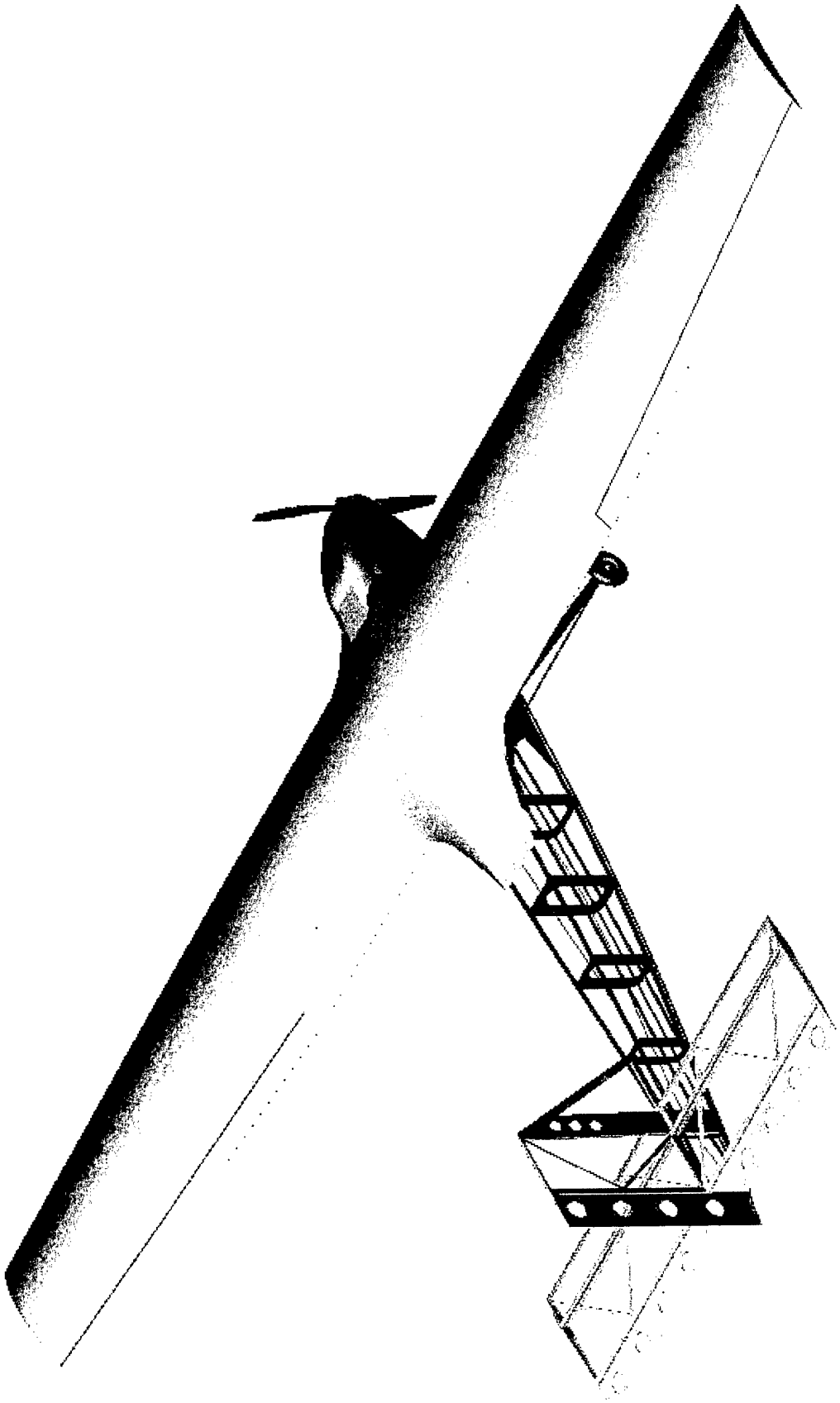
5.8 Aircraft Final Specifications

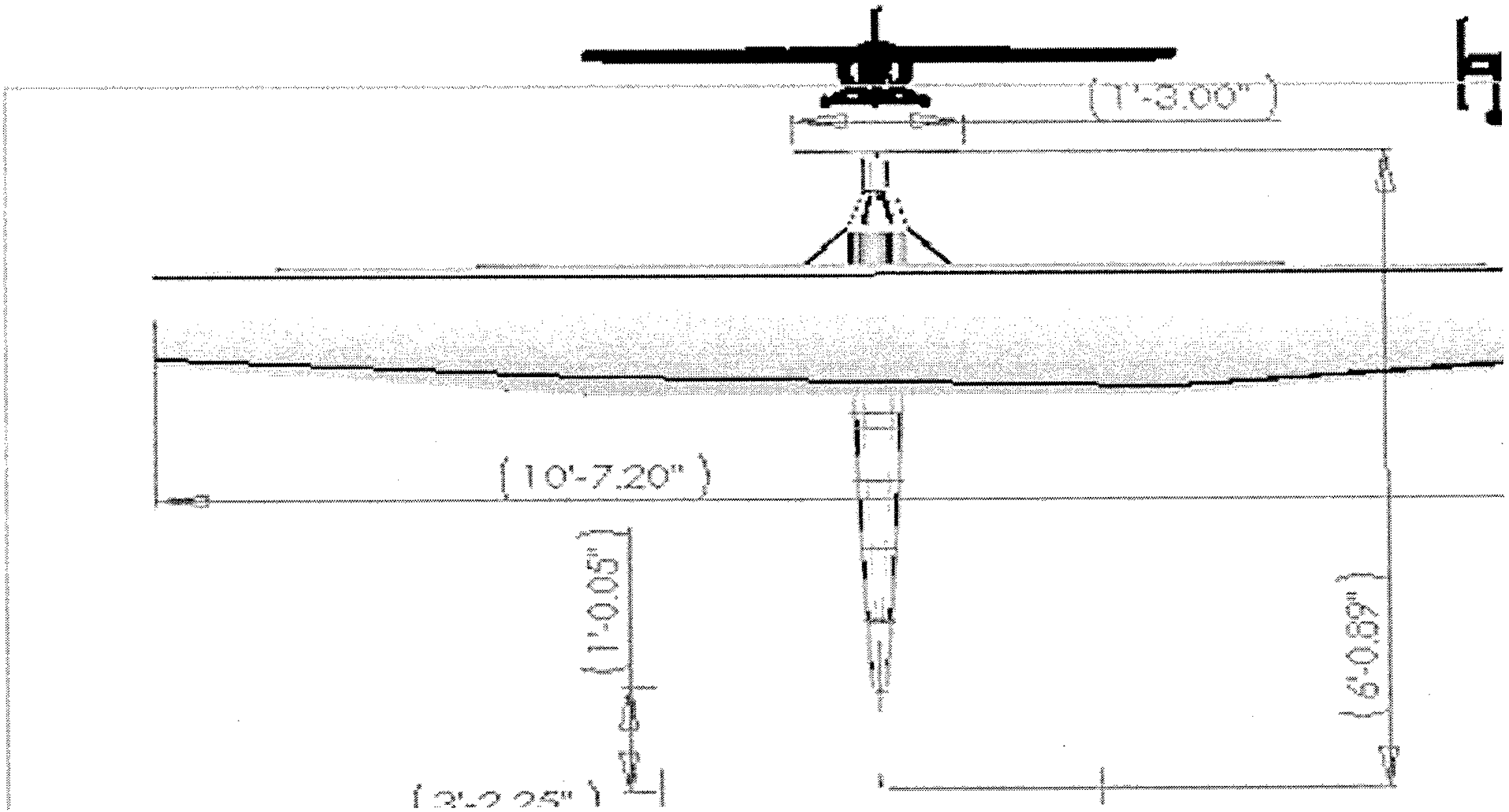
Geometry	**Value**
Length (ft)	6.17
Span (ft)	10.67
Height (ft)	2.17
Wing Area (ft^2)	12.5
Aspect Ratio	10.5
Main Wing Airfoil (ft)	NACA6412
Horizontal/Vertical Stabilizer	Symmetric plate
Weight Statement	**Value**
Airframe (lb_m)	13.2
Propulsion System (lb_m)	7.4
Control System (lb_m)	0.6
Payload (lb_m)	8.8
Manufactures Empty Weight (MEW) (lb_m)	21.2
Gross Weight (lb_m)	30.0
Systems	**Details**
Servos	6 x Futaba
Battery Configuration	42 x sub C cells
Motor	Cobalt 60
Gear Ratio	none
Propeller (nominal)	14x4
Brakes	none
Landing Gear	Tricycle aluminum
Performance	**Value**
Cl_{max}	1.45
Stall speed (ft/sec)	36
Take-off field length (ft)	96
Mac Rate of Climb Empty (ft/sec)	11.41
Mac Rate of Climb Full (ft/sec)	8.1
Minimum drag speed (ft/sec)	50

5.9 Drawings








(1'-3.00")
(10'-7.20")
(1'-0.05")
(3'-2.25")
(4'-0.69")

6 Airplane Design Analysis

6.1 Linear Stability Analysis

A program in Matlab was developed to solve the linear system shown as equations 6.1, a program was also written to solve the non-dimensional, longitudinal equations of motion to analyze the longitudinal stability of the airplane.

Appendix B contains the code of the program in its entirety.

$$\frac{\partial}{\partial t}\begin{bmatrix} u \\ w \\ q \\ \theta \end{bmatrix} = \begin{bmatrix} -2C_{d0}\frac{u_\infty^2}{\mu c g} & C_{l0}(1-2a(\frac{1}{\pi AR}))\frac{u_\infty^2}{\mu c g} & 0 & \cos(\theta_0) \\ -2C_{l0}\frac{u_\infty^2}{\mu \bar{c} g} & -(C_{d0}+a)\frac{u_\infty^2}{\mu \bar{c} g} & 1-\frac{g}{u_\infty}\frac{u_\infty^2}{\mu \bar{c} g}\frac{\partial C_l}{\partial q} & -\sin(\theta_0) \\ 0 & -aS_{man}\frac{\frac{u_{\infty_0}^2}{\mu \bar{c} g}u_\infty^2}{\bar{c} g \frac{k_y}{\bar{c}}\frac{k_y}{\bar{c}}} & \frac{g}{u_\infty}\frac{\frac{u_{\infty_0}^2}{\mu \bar{c} g}u_\infty^2}{\bar{c} g \frac{k_y}{\bar{c}}\frac{k_y}{\bar{c}}}\frac{\partial C_m}{\partial q} & 0 \\ 0 & 0 & 1 & 0 \end{bmatrix}\begin{bmatrix} u \\ w \\ q \\ \theta \end{bmatrix}$$

Equations 6.1

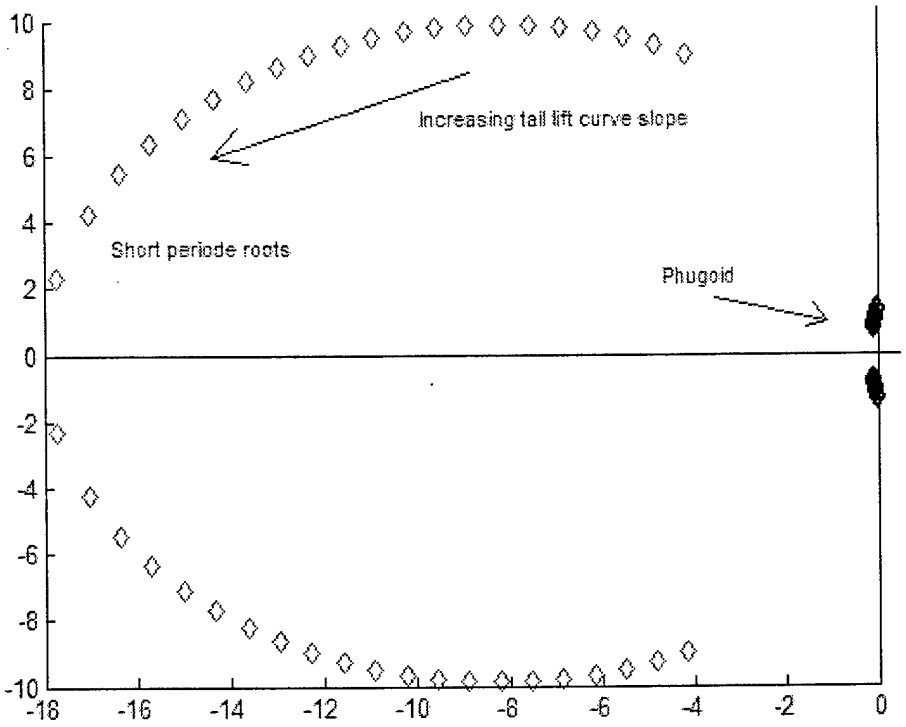
Inputting the longitudinal aerodynamic coefficients and equilibrium conditions for the airplane, this program outputs the eigenvectors and eigenvalues of that particular airplane state. Table 6.1 summarizes the eigenvectors, eigenvalues, short period frequency and damping, as well as phugoid frequency and damping for the steady level flight equilibrium.

				Normalized Eigenvectors				
	EigenValues	Frequency	Damping Ratio	u/uo	w/uo	q/ω	θ	
SP_1	-14.0881 + 7.269i	1.7586	-0.88	0.0015	0.0988	1	0.0622	Mag
				-19.686	50.112	0	27.29	Phase
SP_2	-14.0881 - 7.269i	1.7586	-0.88	0.0015	0.0988	1	0.0622	Mag
				19.686	-50.112	0	-27.29	Phase
Phugoid_1	-0.096 + 0.8375i	0.0943	-0.11	1	0.1526	0.7278	0.8572	Mag
				0	-2.9904	-5.3655	78.0056	Phase
Phugoid_2	-0.096 - 0.8375i	0.0943	-0.11	1	0.1526	0.7278	0.8572	Mag
				0	2.9904	5.3655	-0.8572	Phase
			Frequency Ratio	18.64899				
			Damping Ratio	8				

Table 6.1 Linear system characteristics

Clearly, the values for both the frequency and damping ratios satisfy th discrepancy between the value of the ratios as predicted by Equations 2-5, approximately 5, and the values as calculated by the computer program. However, considering that Equations 2-5 are simply approximations used to determine the order of magnitude of the frequency and damping of both the phugoid and short period, it is assured that the design fits within all design constraints.

This program could also be utilized to investigate the motion of the aircrafts root-locus upon varying the stiffness of the short period. This was implemented by plotting the roots for a tail lift curve slope that varied from 0, to 2*pi. The result can be seen in graph 7.1. This provided insights into the stability of the aircraft. And as one can infer the aircraft is very stable and one expects the aircraft to handle satisfactorily since the frequency ratios are higher than 5.

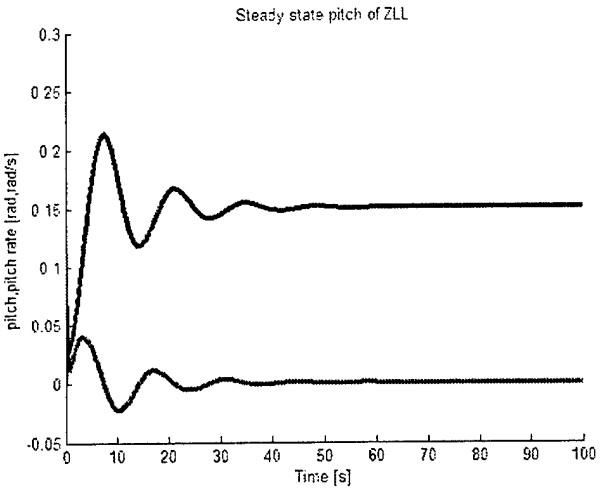


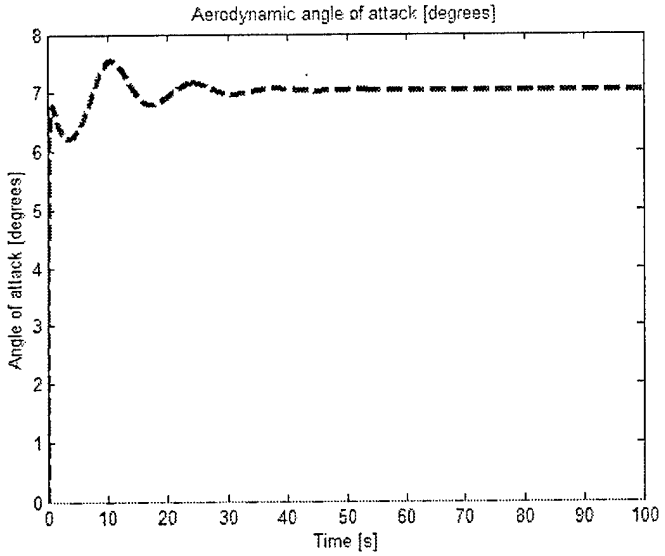
Graph 6.1

6.2 Nonlinear Stability Analysis

The governing equations of motion of the airplane were derived by balancing all forces on the airplane. Next the nonlinear dimensional differential equations that govern the longitudinal motion of the airplane was derived, i.e. the global X and Z positions, the pitch, pitch rate, and the components of velocity with respect to the body fixed coordinate system. Appendix C contains the code for a Matlab program that utilizes an ordinary differential equation solving subroutine to solve the nonlinear dimensional differential equations for the airplane. Note that the program contains a function that utilized all of the previously calculated variables needed to obtain the desired results. Given initial conditions and all airplane parameters, the program generates a series of plots describing the airplane position in the X and Z axes, the airplane velocity, as well as the airplane pitch as a function of time.

Examples of these are presented in graph 6.2.





Graph 6.2

6.2.1.1 Nonlinear Equilibrium State for T/mg = 0.0, 0.1, 0.2, 0.3

Table 6.2 summarizes the equilibrium conditions for the respective thrust-to-weight ratios. From this table and the team's calculation, one can observe that the airplane climbs for all thrust-to-weight ratios above and descends for all ratios below 0.08, the design ratio. Additionally, one observes that the trim conditions of the airplane do not change with change in the thrust-to-weight ratio, only the attitude of the aircraft.

State				
T/mg	u	q	?zll (rad)	?eq
0	14.1	0	0.0571	-0.089
0.1	14.1	0	0.1649	0.0169
0.2	14.1	0	0.266	0.118
0.3	14.1	0	0.375	0.227

Table 6.2: Attitude at equilibrium

The case of thrust-to-weight ratio equal to 0.2 displays some interesting results. The linear system predicts a stable but lightly damped phugoid in this equilibrium state. However, the nonlinear system displays a very stable oscillation about the equilibrium point. This oscillation shown in the nonlinear system somewhat contradicts the prediction made by the linear system about the same equilibrium point, thus exemplifying a deficiency of the linear system. Furthermore, for the case of thrust-to-weight ratio equal to 0.3, the linear system predicts instability while the nonlinear system reveals a stable airplane. This discrepancy epitomizes the inherent limitation of solely relying on linear models.

6.2.1.2 Root Locus for T/mg = 0.0

Appendix D contains the plot of the root locus for the specified condition. Since the short period locus has a negative real component nearly two orders of magnitude greater than the negative real component of the phugoid, one can conclude that the phugoid dictates the stability of the airplane. In addition, the locus corresponding to the condition where thrust-to-weight ratio equals 0.3, the locus has a positive real component, thus confirming the before mentioned instability predicted by the linear system.

6.3 Performance

6.3.1 Construction of Drag Polar

Construction of drag polar was made by utilizing methods described in Airplane Aerodynamics and Performance {Ref. 4}. The methods describe a procedure to calculate C_{d0} based upon wings and fuselage. Major terms that are influence these calculations are area of wing and fuselage (S_w and S_f) and the wetted areas of the same structures (and $S_{w.w}$ and $S_{w.f}$). The result of the C_{d0} calculations was a C_{d0}=0.035. We find this reasonable since most aircrafts has a C_{d0} in the area 0.015 too 0.04.

$$C_{d0} = C_{d0w} + C_{d0f}$$

$$C_{d0w} = R_w \cdot R_w \cdot R_w \{1 + L'(\frac{t}{c}) + 100(\frac{t}{c})^4\} \cdot \frac{S_{w.w}}{S_w} = 0.014$$

$$C_{d0f} = R_{wf} \cdot C_f \left\{ 1 + \frac{60}{(\frac{l_f}{d_f})^3} + 0.0025\left(\frac{l_f}{d_f}\right) \right\} \cdot \frac{S_{w.w}}{S_w} = 0.016$$

$$C_{d0} = 0.014 + 0.00627 = 0.035$$

In the further calculations we chose to consider only induced drag due to the wing lift. Moving into induced drag due to lift from the fuselage would be to make it too advanced for our needs. The design of a model aircraft with all its uncertainties and imperfections in manufacture will make our calculations deviate from reality anyway, so the added complexity of these calculations was not prioritized.

The C_{d0} was utilized in the Excel work book designed by us to construct the drag polar. The workbook calculates the lift induced of the wings utilizing Prandtl Lifting Line Theory. The same workbook also calculat es the lift.

The drag polar shows us that the minimum drag speed is at about 50 mph. This is very beneficial, since this is the area the aircraft is designed to fly at.

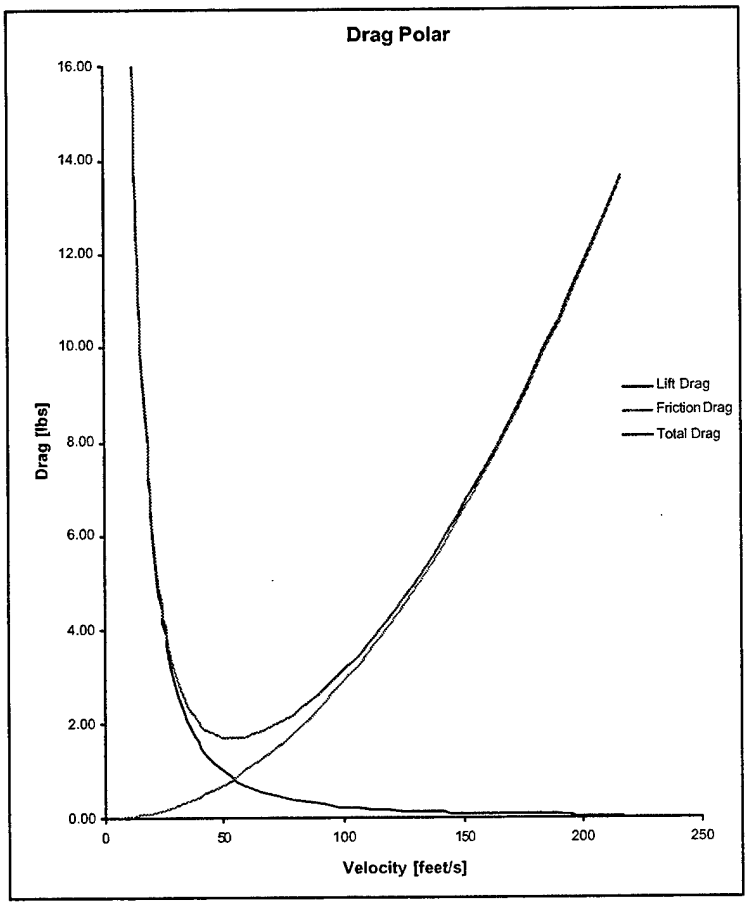


Figure 6.1: Drag polar

6.3.2 Available Thrust

To get an idea of what the maximum speed of the aircraft would be and what the maximum rate of climb would be, it was decided that a curves for available thrust should be constructed. Construction of several curves was done to represent the thrust at different efficiencies of the propeller. The chosen efficiencies were ?=1.00, ?=0.95, ?=0.90, ?=0.85, ?=0.80 and ?=0.75. The construction was done by doing utilizing a program (PropSelector) for choosing propellers for RC aircrafts often used by experienced builders. The program let you manipulate with different propellers, airspeed and RPM. The program was used because previous good experience with it and the acknowledgement from the experienced RC builders.

As shown on the plot, the maximum speed of the aircraft will be at about a 100 ft/s at all efficiencies. The spread in the curves lies further down in the speed range. It will make significant difference if the propeller have ?=1.00 or ?=0.75 in cruise. It is therefore best to assume the latter case. A propeller efficiency of 0.75 is not unreasonable to assume in real life, and we expect it to be in this area.

It should be noted that these calculations have some uncertainties associated with them. But, the calculations are accurate enough for this use.

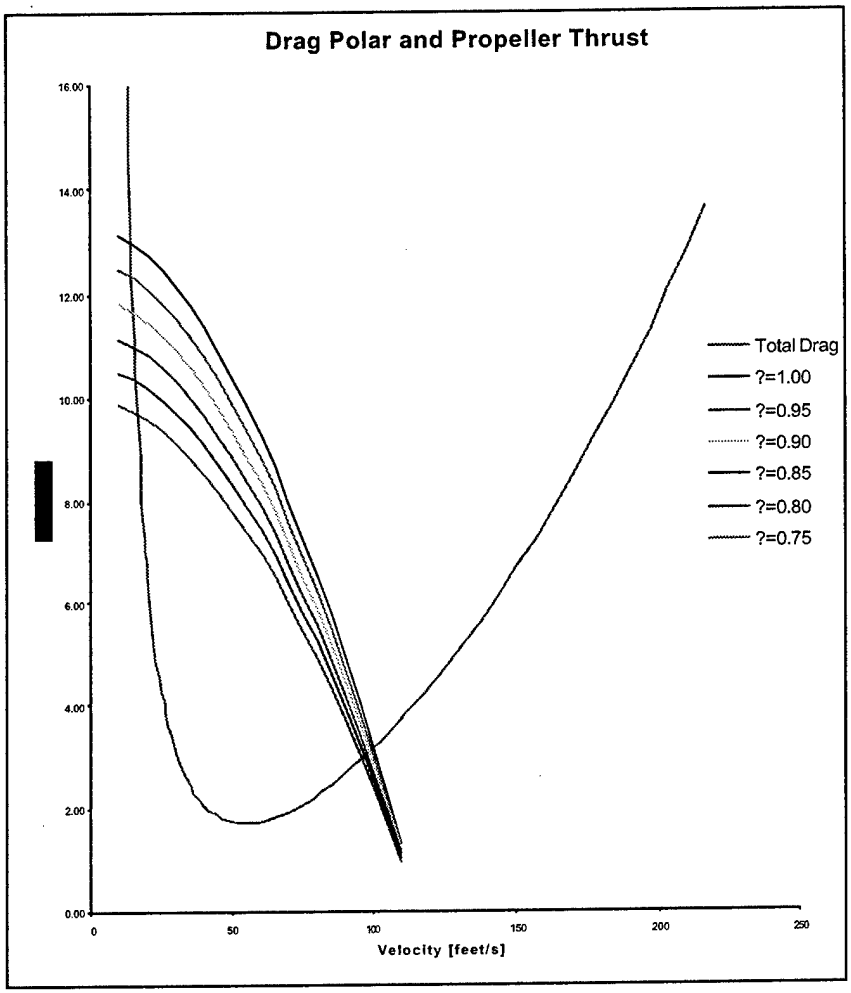


Figure 6.2: Thrust and drag polar

6.3.3 Maximum Rate of Climb

The maximum rate of climb occurs under the flight conditions that the difference between available power and required power is the largest. In other words this is when the difference between available thrust and drag is the largest. Utilizing the Excel workbook, this point is identified to be at about a velocity of 40 feet/sec. This is about $1.15xV_{stall}$, which is the design climb out speed. This is very favorable, since this is the most critical phase of the flight and hence the phase the maximum rate of climb is needed. The maximum rate of climb at the worst case scenario with an aircraft weight of about 30 lbs is (m/s):

$$R.C. = \frac{P_{av} - P_{req}}{W} = 8.1\,ft/s$$

The maximum rate of climb with no payload (aircraft weight of 21.2 lbs) is (m/s):

$$R.C. = \frac{P_{av} - P_{req}}{W} = 11.42\,ft/s$$

The results are satisfactory for the design and seem reasonable for an aircraft of this size. The rate of climb is also comparable to other designs of similar size and power.

6.3.4 Take-off Length

The take-off length was calculated by using Newton's laws. Terms taken into consideration were thrust, drag, aircraft weight, take-off speed and friction coefficient to the ground. The friction coefficient was set conservatively to 0.1 which is equivalent to soft field coefficient. Even though the aircraft will be flown off pawed runway during competition, this friction coefficient was chosen to correct for the high friction in the undercarriage itself. The take-off speed was calculated using $1.15V_{stall}$. The rotation itself will be initiated before this, but this is the velocity the take-off should be done at and the first part of the climb out. After the initial part of the climb out the aircraft should be accelerated to about $1.3V_{stall}$, this will ensure a safe climb out velocity with satisfactory clearance to stall speed.

The result of the calculations was a take-off distance of about 96 feet. This is well below the required 150 feet, and gives adequate room for the uncertainties these calculations contain due to the size of the aircraft. The thrust available and the ground friction coefficient are the largest uncertainties.

6.4 Material selection

6.4.1 Wing martial justifications and lay out

The three main materials to be used in the wing are:

1. Foam, the same kind used in packing of electronics (polystyrene)
2. Balsa Wood, wing skin $1/20^{th}$ of an inch thick
3. Hex-Cell, forming an "I-beam" main spar

These material will be used because of their lightweight, and when combined, high strength. When the wing is loaded under bending the upper and lower extreme most surfaces experience the heights compression and tension loading, while the middle of the wing is only under shear stress. The bulk of the wing will be formed with a foam core, which is extremely strong, relative to weight, under shear and compression loading but week under tension. The balsa skin is very strong under tension and compression loading. When the two are combined they form a lightweight composite of high strength for a much lower cost than that of a textile bases system.

The center of the wing is under both a bending moment and a focused point load, due to the attaching hardware. In the case of highly concentrated load such as a bolt, therefore stated foam core and balsa skin would not be an adequate choice. This is because a bolt could easily be pulled threw the balsa and foam, even with a large washer. To solve this problem and to make it possible to bolt the wings to the fuselage a hex cell spar is introduced to the center section of the wing. The hex-cell spar will take the concentrated loads of the wing mounting bolts and distribute the along the length of the center section of the wing. The hex-cell spar will also increase the over all bending strength of the center section, as well as absorb the concentrated loads generated by the attaching points of the wing tips. Figure 6.3 below is a typical cross section of the center portion of the wing.

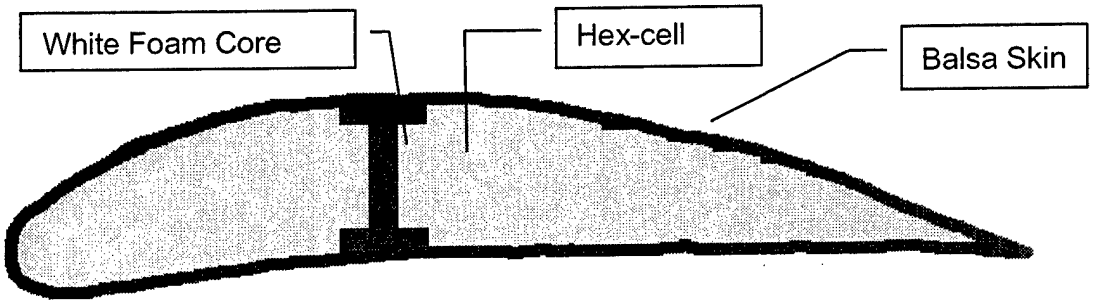


Figure 6.3 Wing material cross section

6.4.2 Nose material justification and layout

It was desired of the nose section of the aircraft to be able to have an energy absorbing characteristics during a crash that should permit in the reuse of the electric motor, instead of bending the shaft as in years past. To make this possible a system had to be designed to fail under loading conditions that would occur in a crash, but not fail under normal operating conditions. To ensure that the motor mounts would indeed fail in a predictable manner a material with know failure characteristics had to be used. Due to the nature with which wooden and composite structures fail, a splintering and cracking in a brittle mode, it is nearly impossible to predict how the failing structure is dissipating energy. For this reason it was decided to use a much more predictable material under failure. Aluminum fails with plastic deformation, which is a much easier failure mode to predict energy dissipation under. The use of aluminum also enables the designer to accurately model failure of the motor mounts using sophisticated FEA analysis tools. Figure 6.4 shows the deformed motor mount assembly after a front-end impact. The impact force represents a fully loaded aircraft flying straight into the ground at a speed of 25 MPH. Note that the aluminum struts have deformed in buckling. Buckling is a form of failure that can easily be calculated both by hand and by FEA analysis tools. It should also be noted that the motor shaft has not deformed due to the impact. The shaft was loaded at an angle of 45 deg with respect to its axis; this would represent a more realistic crash scenario where the prop could exert a moment on the shaft.

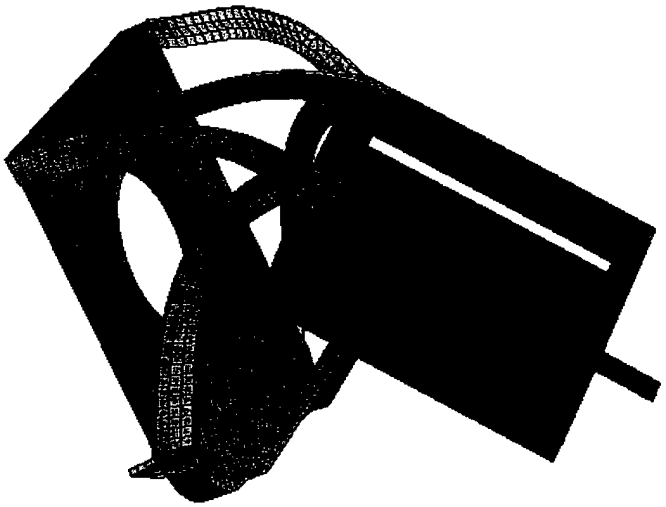


Figure 6.4: Deflected motor mount

6.4.3 Fuselage (center section) Material type and layout

The Center section of the fuselage has to hold both the water and the batteries, as well as transfer the weight of the rest of the aircraft to the wing and to the main gear. This requires both a strong and water tight structure. The materials chosen to make the center section strong will also be able to make it watertight. The use of the a fiberglass inner liner will ensure a water tight "wet" center section, and the fiber glass will also and a significant amount of strength the highest stressed region of the fuselage. The Center section of the Fuse will be constructed form two main Hex-Cell bulkheads at each end. A very thin wall tube will join the center of the bulkheads, and a birch ply wood skin will be wrapped around the side and bottom in a continuous fashion. The whole assembly will be bonded with epoxy together, and then a fiber glass and epoxy resin inner liner will be wet laid inside the tank, ensuring a 100% sealed tank, and further reinforcing all joined parts. Figure 6.5 shows the center section and its water payload.

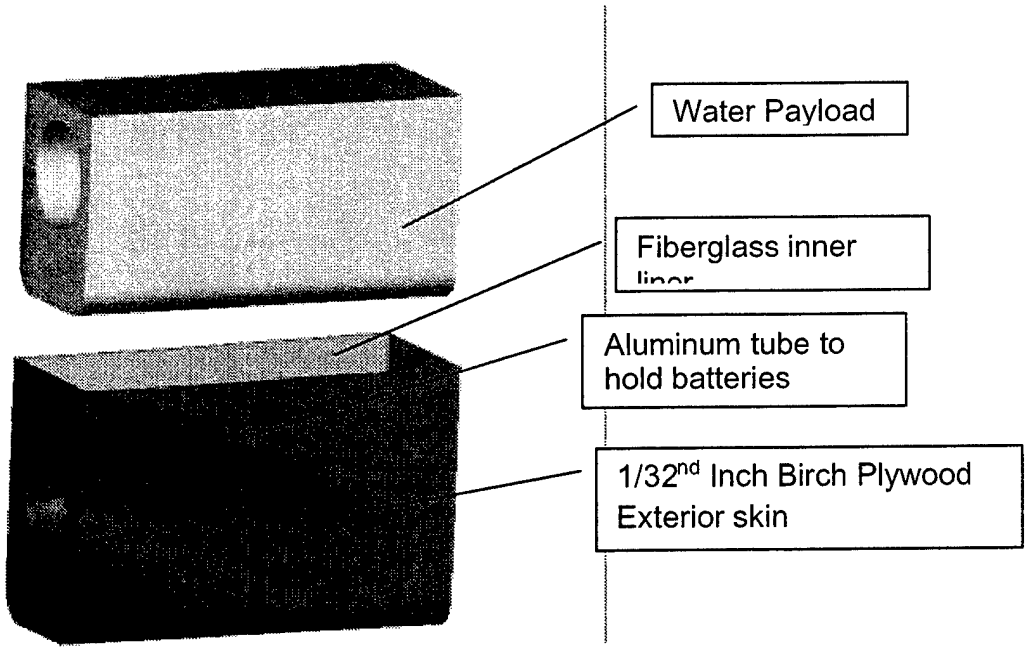


Figure 6.5: Center "WET" section of fuselage

The central tube to hold the batteries is to be made form aluminum because of its superior heat dissipating characteristics. The thin walls of the aluminum tube should allow the heat generated by the batteries to be dissipated to the water. The water will act as a large thermal reservoir. This will also be supplemented by the forcing air threw the batteries. The front nose fearing has a scoop just behind the high-pressure region of the prop. The scoop will supply a slight pressurized airflow to the tank to aid in water dumping. The majority of the air, however, will be forced threw the batteries and out the tail of the aircraft.

6.4.4 Tail material types and layout

The rear portion of the fuselage will be constructed using a well-proven, time-tested technique. The Rear portion of the fuselage will be constructed using bulkheads to form the cross section of the airframe at evenly spaced intervals, and long stingers running the length of the assembly. Each of the bulkheads will have a series of notched for the stringers, ensuring a straight and strong fuselage. This technique for construction has no "X" bracing or triangulation in the wooden structure; instead the iron on plastic covering known as Mono-Coat acts as a stressed skin. This arrangement of bulkheads, stringers, and mono-coat is also a vary coast effective and easy to manufacture solution. Complex fiberglass and or other composites structures can easily be as much as 100 times more expensive with little to no weight saving and little to no strength increase.

The bulkheads will be made of 1/32-inch thick birch plywood with 1/8-inch square balsa stiffeners around the perimeter. All of the bulkheads can be seen in the measured drawing section of the appendix I. The long stringers will be made of 3/16-inch square spruce. The tail will also contain all the electronics need for fight, except those for propulsion. To carry this equipment and to further strengthen the root of the tail, a series of 1/32-inch birch ply wood gussets and 1/8-ich birch plywood trays were added to between the first and second bulkheads. Figure 6.6 show how the whole tail system comes together.

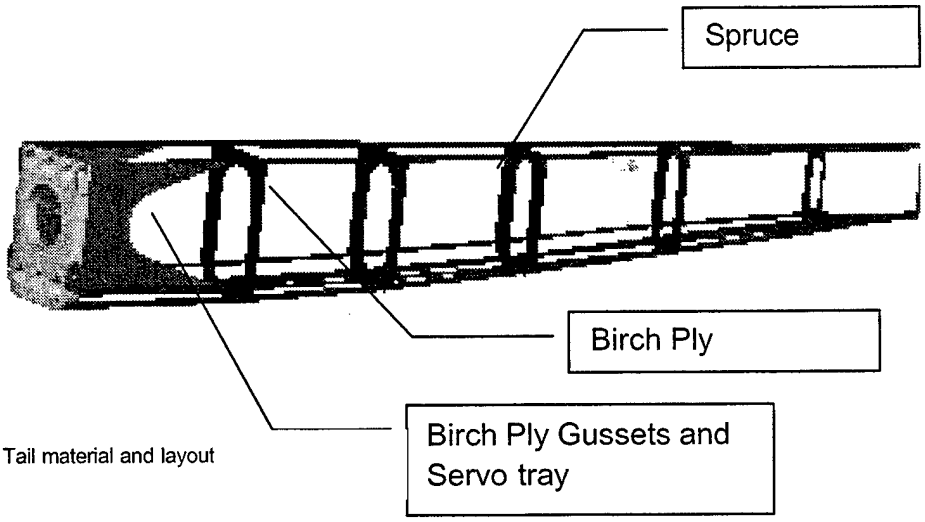


Figure 6.6: Tail material and layout

6.4.5 Horizontal and Vertical stabilizers' materials and layouts

The Horizontal and vertical stabilizers will be made form balsa wood. The horizontal stabilizer will be constructed to form a "truss" with a high amount of triangulation. A cross section that more resembles a symmetric airfoil will be generated by running two strips down the length of the horizontal stabilizer, also helping to increase the overall strength of the structure. The vertical tail will be constructed around a central mast, that will insert threw the horizontal stabilizer and into the fuselage, where one nylon bolt will secure both the vertical and horizontal stabilizers.

6.5 Over view of materials used for each major component.

Wing:

Foam core
Balsa skin
Hex-Cell (honey comb) "I beam" spar
0.125" central and joint ribs
Monokote Iron-on plastic covering

Nose:

Aluminum motor mount struts
Front gear attaches to back of old motor mounts
Fiberglass fearing (painted)

Fuse:

Hex-cell bulkheads
Birch ply side
Aluminum tube
Fiberglass tank sealer (inside of tank)
Monokote Iron-on plastic covering

Tail:

Birch ply bulkheads with balsa stiffeners
Spruce stringers
Birch ply gussets and servo tray
Monokote Iron-on plastic covering

Horizontal and vertical stabs:

Balsa trusses
Fiberglass and CA hinges
Monokote iron-on covering (sealed control surfaces)

Landing gear:

Aluminum struts
Stream lined connecting bar
Steam lined wheel pants.
Monokote Iron on plastic covering

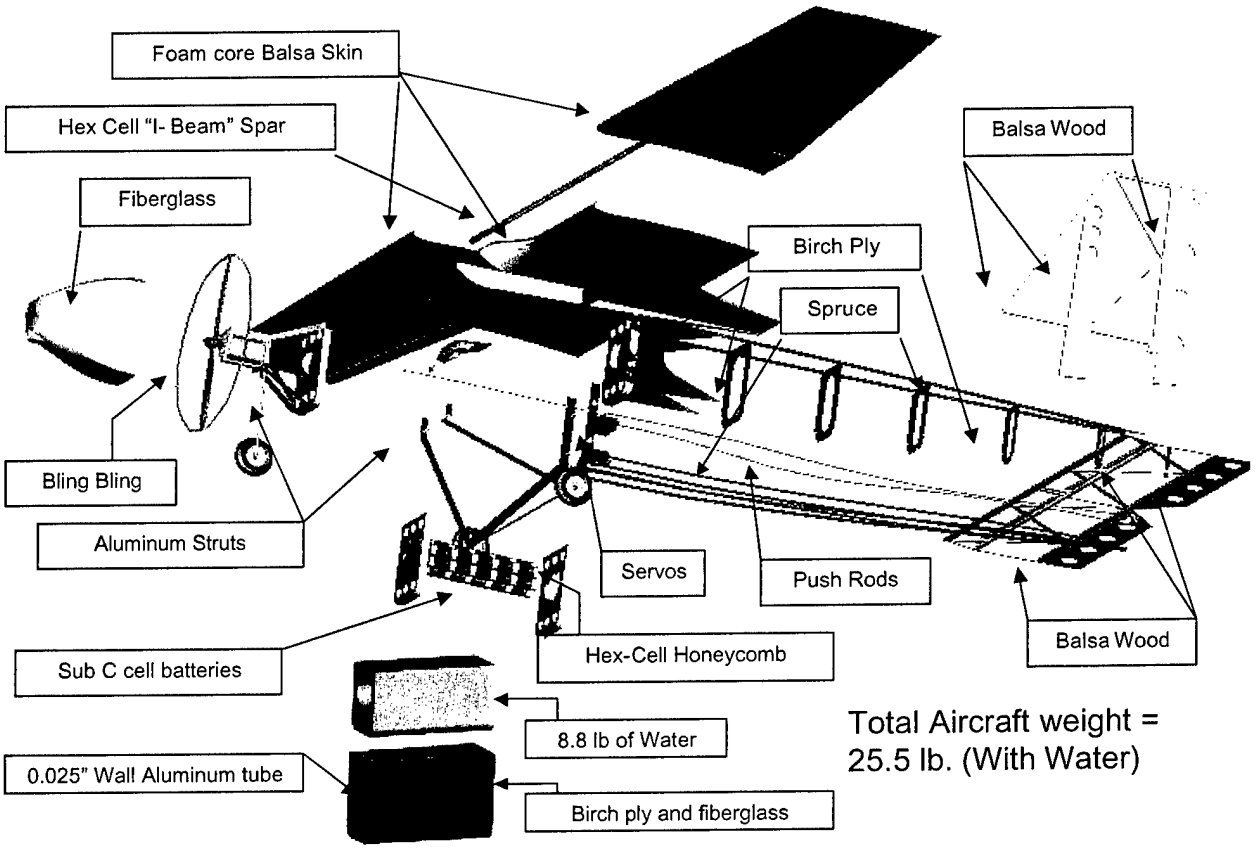


Figure 6.7: Final aircraft layout and materials used.

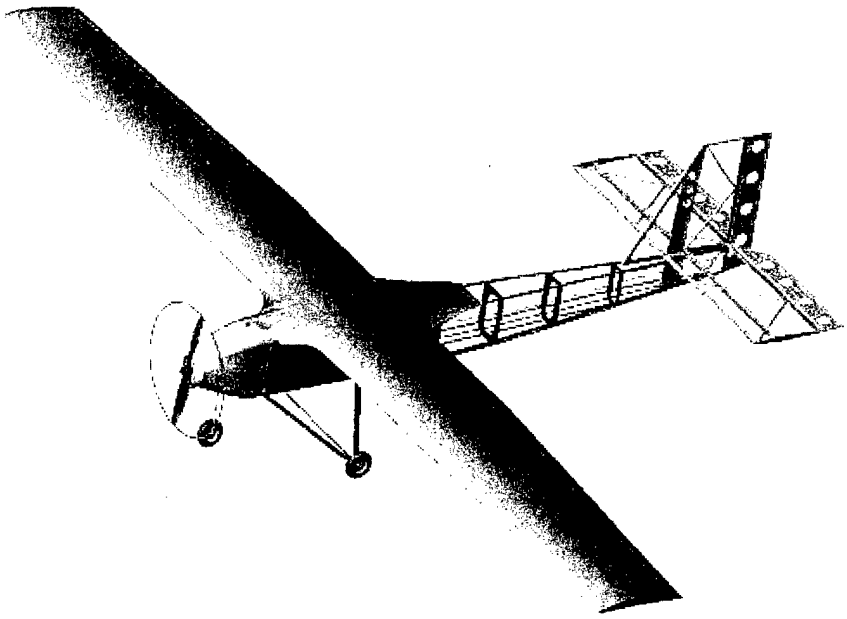


Figure 6.8: Final aircraft assembly.

6.6 Structural Analysis

A structural analysis was conducted using the finite elements method with the assistance of the ANSYS software program. The air frame was modeled using beam and shear elements and then loaded to simulate various conditions. To simulate the most extreme conditions an aircraft mass of 30 pounds was used. Figure 6.9 shows the configuration of the model.

The conditions simulated are in-flight and landing. The model was loaded under load factors n = 1, 2, 3, 4 and 5 times the aircraft weight, and then loaded until stress in the frame exceeded the maximum compressive strength. The mechanical properties of balsa wood used for the analysis can be found in appendix I.

The results of this analysis can be found in Table 6.3 below:

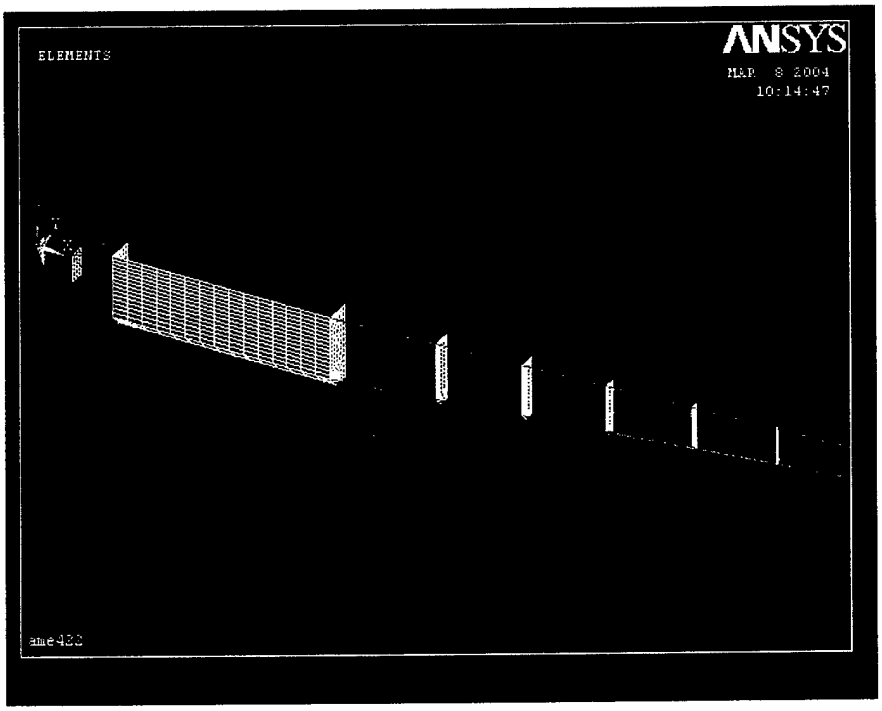


Figure 6.9 Configuration of ANSYS model.

Load Factor, n	Maximum Stress (psi)			
	Case 1: In-flight		Case 2: Landing	
	Compressive	Tensile	Compressive	Tensile
1	249.19	408.96	329.32	266.90
2	231.97	811.05	344.75	239.74
3	345.63	1213.1	360.30	212.36
4	459.28	1615.2	375.84	256.03
5	-	-	391.39	321.05

Table 6.3: Results of analysis.

Appendix J contains plots of the stress distribution throughout the model for each load case and highest load factor before failure. It can be seen that for the load case simulating the in-flight condition the point of maximum stress occurs at the rearward point where the wing attaches to the fuselage. For the case of landing the point of maximum stress occurs at the rearward point where the landing gear attaches to the fuselage.

Figures 6.10 and 6.11 below are contour plots of the stress distribution for the cases simulating the failure of the structure.

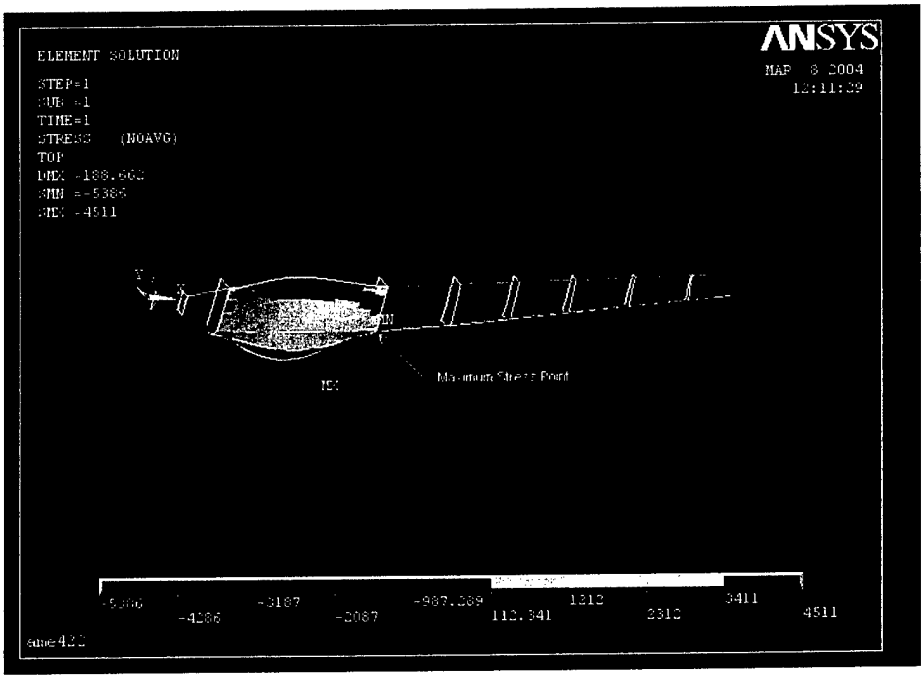


Figure 6.10: Stress distribution during landing.

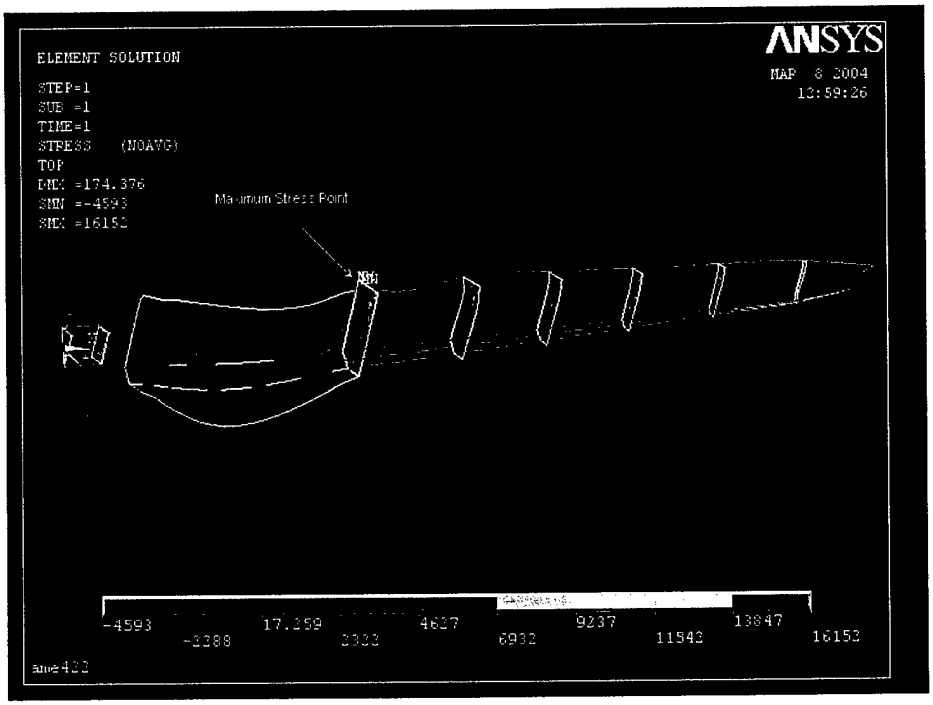


Figure 6.11: Stress distribution during flight

It was found that failure occurred for a load factor of n = 7 when simulating landing. The failure is compressive failure and occurs in element 655. This element is highlighted in figure 6.10. For loading of n = 6, the maximum compressive stress was 462.35 psi. For loading of n = 7, the maximum compressive stress was 538.58 psi. Since the maximum compressive strength of balsa wood is approximately 500 psi failure occurs at n = 7.

For the case of in-flight the structures fails when loaded with four times the aircraft weight. The failure is tensile failure in this case, and the element in which this occurs is shown on figure 6.11. Table 6.3 above shows the maximum tensile stress is 1615.2 psi, which exceeds the ultimate tensile strength of 1375 psi.

It can be seen that the risk of failure of the structure is greater in the case of in-flight. Considering the mission requirements of the aircraft design, failure of the structure is not considered to be an issue, as the aircraft should not be placed under such load cases during it's missions.

7 Manufacturing

A designated person will be identified to oversee the entire manufacturing process in order to ensure that the manufacturing goes as planned. This also provides an opportunity to get experience in identifying and delegating tasks to members of the construction team. The person will make sure that all components and tooling are available when needed. Also, he will function as a director of the manufacturing process and be responsible for schedules of progress in the work.

The first parts to be manufactured are the vertical and horizontal stabilizers and all other subsequent "2 D" parts. "2 D" parts are parts which can be built directly off a flat print out of the part. The construction of the vertical stabilizer will be out lined here, and represents the techniques used for all "2 D" parts.

Firsts a full scale copy of the measured drawing is taped securely to a flat work surface, the drawing will act as a guide in placing the balsa wood sticks in position. Next the balsa wood can be cut to the correct length assembled and glued all on top of the measured drawing, ensuring a correctly built part.

The wing sections are formed using a foam bow and wing profiles made of wood. The wing profiles are glued on the ends of the closed cell foam using temporary adhesive, and the hot wire is drawn threw the foam using the wing profiles. The balsa skin is then glued on the surface and "Clamped" in place using sand bags and the cut off portions of foam.

The water tank is formed by cutting two bulk head form Hex cell and then wrapping the birch plywood around the hex cell. A wet lay up of fiberglass is then used to seal the interior of the tank. This is accomplished by

To start construction of the main fuselage a full-scale top view measured drawing is printed out. The drawing should then be tapped to a flat work surface. The top of the fuselage was designed to be flat for the ease of manufacture. The fuselage will be able to be built right off of the full scale drawing ensuring a straight member. Next all of the bulk heads must be manufactured. Again a full scale measured drawing is printed out of each bulk head, and is pasted to birch ply using a temporary adhesive. The parts can then be cut using the lines on the paper. The fuselage can then be built top side down right on the paper drawing.

Parts are and components will be acquired both from previous designs and new procurements. The parts will be ordered online and well in advance in order to reduce the cost and to ensure good logistics. Furthermore, parts that are of a more trivial nature will be bought at local store as needed, such as glue and wood.

As modules are completed they will be subjected to a final inspection to ensure that they comply with the requirements imposed in their design such as confirming the mass distribution and dimensional constraints. Next, they will undergo simple tests to confirm the integrity when exposed to nominal load cases. When all modules are complete they will be assembled to form the complete aircraft and design integrity will be tested and confirmed and test flight program will ensue.

8 Testing Plan

The aircraft testing will begin as soon as the constructions are done. The constructions will hopefully be done by the last week of March, and the flight tests will begin immediately. The purpose of the flight tests is to confirm/disconfirm the calculations made in the design. If the results deviate by a large amount in a way that is unfavorable modifications should be undertaken. This is one of the reasons for building the aircraft in modules: it makes it easy to modify if needed. The tests will be done continuously until the date of the competition. This is not only for the reasons mentioned above, but also to familiarize the pilot with the airplane and prepare for the missions flown during the competition.

8.1 Flight Test Checklist

The following checklist is prepared to ensure a safe conduction of the first test flights. If the procedures are followed and the flights conducted as described, it will reduce the risk of loosing the aircraft. It also gives valuable information on how the aircraft performs and what has to be adjusted/modified.

Flight Test	
Pre Start Checklist:	
	Check wing connection points fastened and locked.
	Check all modules fastened and locked
	Check landing gear fastened and locked
	Check all electrical points connected
	Check propeller for wear and notches
	Turn on receivers
	Do preflight check of all control surfaces
After Start Checklist	
	Check motor for proper operation
Taxi test	
	Low speed taxing, check for nose gear operation
	High speed taxing, check for rudder, elevator and aileron operation
Flight test 1.	
	Take-off and climb straight ahead at maximum power to safe altitude (dependent on the experience of the pilot)
	One complete circuit with a full stop landing
Flight test 2.	
	Take-off and climb straight ahead at maximum power to safe altitude (dependent on the experience of the pilot)
	Shallow bank turns to right and left
	Stalls straight forward and in shallow turns
	Full stop landing
	Check structures and connections
Flight test 2.	
	Take-off and climb straight ahead at maximum power to safe altitude (dependent on the experience of the pilot)
	Step turns
	Test of full deflection of aerodynamic surfaces.
	Full stop landing
	Check structures and connections

Flight test 3.	
	Load aircraft to maximum take-off weight
	Take-off and climb straight ahead at maximum power to safe altitude (dependent on the experience of the pilot)
	Shallow turns to right and left
	Steep turns to left and right
	Stall straight ahead
	Stalls in turns
	Full control surfaces deflections
	Release of cargo
	Full stop landing
	Check structures and connections
Flight test 4, 5, etc.etc	
	Load aircraft to maximum take-off weight
	Take-off and climb straight ahead at maximum power to safe altitude (dependent on the experience of the pilot)
	Check structures and connections after each landing.

9 Conclusions

From what we can see, the design of the aircraft has been successful. The modular design allows easy transportation and eases the maintenance. The modules are also easy to interchange if the design needs modification after the test flight stage. When it comes to performance the aircraft will cruise at minimum drag speed which reduces the power consumption, and it will take-off at the maximum rate of climb speed. The calculated take-off distance is well below the requirements. How accurate these calculations are will be very interesting to see. This will reveal itself in the next couple of weeks when the test flights start.

However, the team is confident that this is a successful design and that the aircraft will perform well in the competition.

10 References

[1] Comprehensive Reference Guide to Airfoil Sections for Light Aircrafts, Aviation Publications, Wisconsin, 1982.

[2] Aerodynamics for Engineers, John J. Bertin, Prentice Hall , Upper Saddle River, 2002.

[3] Basics of R/C Aircraft Design, Andy Lennon, Appletree Publications, Philadelphia, 1999

[4] Airplane Aerodynamics and Performance, Dr. Jan Roskam and Dr. Chuan-Tau Edward Lan, DARcorporation, Lawrence, 1997.

11 Appendix A: Rated Aircraft Cost

MEW	22	lbs
Rated Aircraft cost	17.355	k$
Man hours:		
Wings	12	hours
Fuselage:	23	hours
Horizontal+vertical stabilizer	6	hours
Control surfaces	5	hours
Servos + controls	15	hours
Engine mounting	8	hours
Sum:	69	hours

The aircraft rated cost is found to be 17 355 USD. This estimate is based on the techniques supplied in the rules of the competition.

12 Appendix B: Density Constants

Material	Density (lbm/in^3)
Steel	0.282
Aluminum	0.098
NiCad Battery	0.092
Birch	0.0257
Spruce	0.0126
Hex-cell	0.005
Balsa	0.003
Foam	0.0003

13 Appendix C: Matlab Linear Stability Program

```
clear all;
close all;

%constants
g = 9.81;
uequil = 14;  %equilibrium axes
wequil = 0;     %15.8 m/s when no elevator
elevdeg = 0;
thrustmg = 0.00
thrustdirdeg = 0; %Desing has propeller alligned with fuselage
thetaequil = 0; %calculated for T/mg=0;
cl0 = .678;
cd0 = 0.0646;
cbar =0.339;

speedequil = sqrt((uequil*uequil) + (wequil*wequil));
%thetaequil = (thetadegequil * pi)/180;

elevdeg = 0;
thrustmg = 0;
thrustdirdeg = 0;

theta0 = thetaequil - atan(wequil/uequil);

ct0 = cos(theta0);
st0 = sin(theta0);

theta0deg = (theta0 * 180)/pi;

uinf = speedequil;
omega = g/uinf;

cbar =0.339;
mu = 41.62;
a = 5.7;
at=4.488;
marm = 3.08;
ARw = 10;
alfw0deg = 1;
tzLLdeg = 8.48;

ARw = 10;
alfw0deg = 1;
%tzLLdeg = 8.48;

cdstar = 0.05;
Cm0 = 0.08;
dcldelev = 0;
dcmdelev = 0;
sm = 0.1;

kycbar = 0.7;
St=0.1914;
Sw=1.1526;
alfw0 = (alfw0deg * pi)/180;
%tzLL = (tzLLdeg * pi)/180;
thrustdir = (thrustdirdeg * pi)/180;
```

```
elev = (elevdeg * pi)/180;
lt=cbar*3.08;
par1 = (uinf * uinf)/(mu*cbar*g);
par2 = (par1 * uinf * uinf)/(cbar * g * kycbar * kycbar);
dcldq = (St/Sw)*(cbar*marm/uinf)*at;
dcmdq=-dcldq*marm;
K = 1/(pi*ARw);


A = [-2*cd0*par1, cl0*(1-2*a*K)*par1, 0, -ct0;
    -2*cl0*par1, -(cd0+ a)*par1, 1-omega*par1*dcldq, -st0;
    0, -a*sm*par2, omega*par2*dcmdq, 0;
    0, 0, 1, 0;];

 %x0=[0 ;0.1;0;0];
 %for i=1:1:5000
 %    x11(:,i)=x0;
 %    x=A*x0;
 %    x0=x0+x*0.01;
 %    Time(i)=i*0.01;
 %end
 %plot(Time,x11(4,:))
 %hold on
 % plot(Time,x11(3,:),'r')
[Eigvec,Eigval]=eig(A)

c=max(Eigvec);
c=1./c;
n=diag(c);
scalEig=Eigvec*n;
fase=180/pi*atan(imag(scalEig)./real(scalEig));
Mag=abs(scalEig);

Lambda_1=Eigval(1,1)
Magnitude_eigen_vector=Mag(:,1)'
Phases=fase(:,1)'

Lambda_2=Eigval(2,2)
Magnitude_eigen_vector=Mag(:,2)'
Phases=fase(:,2)'

Lambda_3=Eigval(3,3)
Magnitude_eigen_vector=Mag(:,3)'
Phases=fase(:,3)'

Lambda_4=Eigval(4,4)
Magnitude_eigen_vector=Mag(:,4)'
Phases=fase(:,4)'
%break
%break
figure
hold on

ReEig=real(Eigval);
ImgEig=imag(Eigval);
plot(ReEig,ImgEig,'+')
break
```

14 Appendix D: Matlab Nonlinear Stability Program

```
%Solution of nonlinear equations of motion for ame427 final design case


%Initialization of global variables
clear all; close all;
global m g rhoair cbar cbare cba rt b StaMarg MoArm Sw St Se ARw ARt ARe a aw at ae awo it Cmo dedalf
        Cdo de T Thetat Ky


%dimensions
m=11.5;
g=9.81;
rhoair=1.225;


cbar=0.339;
cbare=0;
cbart=0.196;
b=3.23;
Ky=0.24;

StaMarg=0.1*cbar;
MoArm=1.05;

xh=0.8;

Sw=1.1526;
St=0.1914;
Se=St*(1-xh);

ARw=10;
ARt=5;
ARe=25/((1-xh)*5)

% lift curve slopes

a=5.70;
at=4.488;
ae=2*pi-2*acos(1-2*xh) + 2*sin(acos(1-2*xh));
aw=5.21;

% angels

awo=0.019; %rad
it=0.0766;
Thetat=-(8.5*pi/180); %Thrust

% Moment coeff.

Cmo=0.08;

% downwash

dedalf=0.33;
% Parasite drag coeff
```

```
Cdo=0.05;

%Contol inputs
de=0*pi/180;
T=0.1*m*g%0.088*m*g;

%State Variables
u=13.76;
w=0;%1.46;
q=0;
X=0;
Z=-250;
Theta=0 ;%.122;

%State vector
x=[u;w;q;X;Z;Theta];

Tspan=[0 100];

[T,y]= ode23(@nonlinear,Tspan,x);

b=atan(y(:,2)/y(:,1));
plot(T,(b*180/pi),'--')
%plot(T,(b),'--')
figure
V=((y(:,1).^2)+(y(:,2)).^2).^(0.5);

plot(T,V)
xlabel('Time [s]')
ylabel('Velocity [m/s]')
title('Velocity of cg of the aircraft')

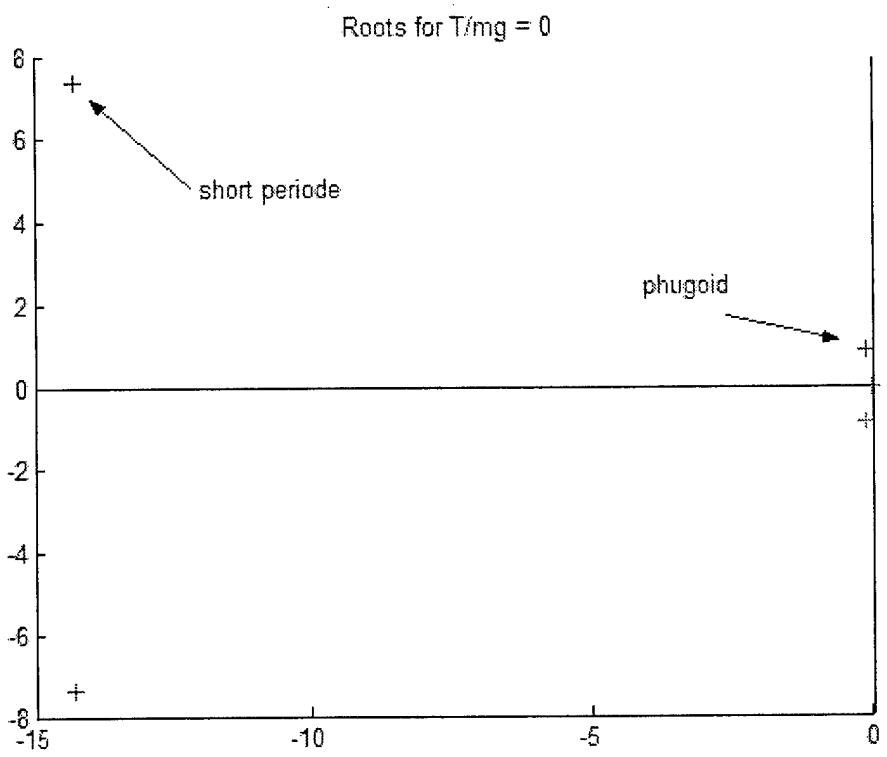
figure

hold on
plot(T,y(:,3),'r')
plot(T,y(:,6),'b')
xlabel('Time [s]')
ylabel('pitch,pitch rate [rad,rad/s]')
title('Steady state pitch of ZLL')
hold off

figure


plot(y(:,4),-y(:,5))
xlabel('Distance (m)')
ylabel('Altitude (m)')
title('Position of cg of aircraft')
```

15 Appendix E: Root Locus



16 Appendix F: Wing Calculations

```
clear all;
close all;

M = 11.5;
g = 9.81;

MeanChord = 33.9; %mean wing chord in cm
WingArea = 11526;  %wing area in cm^2
MomentArm = 105;   %moment arm in cm to calculate tail area
WingSpan = 340;   %wing span in cm

MeanChordUS = MeanChord/2.54; %mean wing chord in in
WingAreaUS = WingArea/(2.54*2.54); %wing area in in^2
MomentArmUS = MomentArm/(2.54); %moment arm in in

TailAreaUS = (.2*2.5*MeanChordUS*WingAreaUS)/MomentArmUS; %Tail area in in^2
TailArea = TailAreaUS*(2.54*2.54) %tail area in cm^2

AspectRatiotail = 5;
AspectRatio = (WingSpan/MeanChord);

DepsilonDalpha = 0.33;

hacw = 0.25;
hact =  3.54;
hcg = .40;

Atail = 2*pi*(AspectRatiotail/(AspectRatiotail+2));
Awing = 2*pi*(AspectRatio/(AspectRatio+2));

StoverSw = (TailArea/WingArea);
A = Awing + (StoverSw*Atail*(1-DepsilonDalpha));

AtoverA = (Atail/A);

hN = hacw + (StoverSw*AtoverA*(1-DepsilonDalpha)*(hact-hacw));
NecessaryAlpha = (M*g)/(.5*1.225*WingArea*.0001*A*16*16);
NecessaryAlphaDeg = NecessaryAlpha * 180/pi
Cm0 = -A*(hcg-hN)*(NecessaryAlpha);
iT = Cm0/(Atail*StoverSw*(hact-hacw)*(1-(StoverSw*AtoverA*(1-DepsilonDalpha))));
Alphawing0 = StoverSw*AtoverA*iT;

TrimSpeed = sqrt(M*g*((hN-hcg)/(.5*1.225*WingArea*.0001*Cm0)))
```

17 Appendix G: Mass Properties

Mass properties of master assembly with water

Output coordinate System : -- default --

* Includes the mass properties of one or more hidden components.

Mass = 11.4898 kilograms

Volume = 0.0622 cubic meters

Surface area = 5.6789 square meters

Center of mass: (meters)
X = -0.2977
Y = -0.0666
Z = 0.2957

Principal axes of inertia and principal moments of inertia: (kilograms * square meters)
Taken at the center of mass.

Ix = (1.0000, -0.0000, -0.0014)	Px = 0.6599
Iy = (-0.0014, -0.0972, -0.9953)	Py = 1.5192
Iz = (-0.0001, 0.9953, -0.0972)	Pz = 2.0566

Moments of inertia: (kilograms * square meters)
Taken at the center of mass and aligned with the output coordinate system.

Lxx = 0.6599	Lxy = 0.0001	Lxz = -0.0012
Lyx = 0.0001	Lyy = 2.0515	Lyz = 0.0520
Lzx = -0.0012	Lzy = 0.0520	Lzz = 1.5243

Moments of inertia: (kilograms * square meters)
Taken at the output coordinate system.

Ixx = 1.7156	Ixy = 0.2279	Ixz = -1.0126
Iyx = 0.2279	Iyy = 4.0744	Iyz = -0.1744
Izx = -1.0126	Izy = -0.1744	Izz = 2.5934

Mass properties of master assembly with out water

Output coordinate System : -- default --

* Includes the mass properties of one or more hidden components.

Mass = 7.4569 kilograms

Volume = 0.0582 cubic meters

Surface area = 5.4155 square meters

Center of mass: (meters)
X = -0.2983
Y = -0.0470
Z = 0.3045

Principal axes of inertia and principal moments of inertia: (kilograms * square meters)
Taken at the center of mass.

Ix = (1.0000, 0.0001, -0.0012)	Px = 0.6105
Iy = (-0.0012, -0.0929, -0.9957)	Py = 1.4961
Iz = (-0.0002, 0.9957, -0.0929)	Pz = 2.0181

Moments of inertia: (kilograms * square meters)
Taken at the center of mass and aligned with the output coordinate system.

Lxx = 0.6105	Lxy = 0.0002	Lxz = -0.0011
Lyx = 0.0002	Lyy = 2.0136	Lyz = 0.0483
Lzx = -0.0011	Lzy = 0.0483	Lzz = 1.5006

Moments of inertia: (kilograms * square meters)
Taken at the output coordinate system.

Ixx = 1.3185	Ixy = 0.1047	Ixz = -0.6785
Iyx = 0.1047	Iyy = 3.3688	Iyz = -0.0584
Izx = -0.6785	Izy = -0.0584	Izz = 2.1807

18 Appendix H: Pugh Analysis

T Tail	Conventional Tail	T Tail	V Tail	Canard Tail	Twin Boom
	C1	C2	C3	C4	C5
Manufacturability	1	0	-1	1	-1
Cost	1	0	-1	0	0
Weight	1	0	1	1	0
Durability/Reliability	1	0	1	1	0
Complexity	1	0	-1	0	-1
Worse Than Datum	0	0	3	0	2
Better Than Datum	5	0	2	3	0
Equal to Datum	0	5	0	2	3
Raw Score (Better-Worse)	5	0	-1	3	-2
Normalized Score	1	0.2857	0.1429	0.7143	0

Conventional	Conventional Tail	T Tail	V Tail	Canard Tail	Twin Boom
	C1	C2	C3	C4	C5
Manufacturability	0	-1	-1	-1	-1
Cost	0	-1	-1	-1	-1
Weight	0	-1	1	-1	-1
Durability/Reliability	0	-1	-1	0	-1
Complexity	0	-1	-1	-1	-1
Worse Than Datum	0	5	4	4	5
Better Than Datum	0	0	1	0	0

Equal to Datum	5	0	0	1	0
Raw Score (Better-Worse)	0	-5	-3	-4	-5
Normalized Score	1	0	0.4	0.2	0
Canard	Conventional Tail	T Tail	V Tail	Canard Tail	Twin Boom
	C1	C2	C3	C4	C5
Manufacturability	1	-1	-1	0	-1
Cost	1	0	-1	0	0
Weight	0	-1	1	0	-1
Durability/Reliability	0	-1	-1	0	-1
Complexity	1	0	-1	0	-1
Worse Than Datum	0	3	4	0	4
Better Than Datum	3	0	1	0	0
Equal to Datum	2	2	0	5	1
Raw Score (Better-Worse)	3	-3	-3	0	-4
Normalized Score	1	0.1429	0.1429	0.5714	0

V Tail	Conventional Tail	T Tail	V Tail	Canard Tail	Twin Boom
	C1	C2	C3	C4	C5
Manufacturability	1	1	0	1	1
Cost	1	1	0	1	1
Weight	-1	-1	0	-1	-1

Durability/Reliability	1	-1	0	1	-1
Complexity	1	1	0	1	1
Worse Than Datum	1	2	0	1	2
Better Than Datum	4	3	0	4	3
Equal to Datum	0	0	5	0	0
Raw Score (Better-Worse)	3	1	0	3	1
Normalized Score	1	0.3333	0	1	0.3333

19 Appendix I: Mechanical Properties of Balsawood

Density, ? = 0.00324 lb/in^3

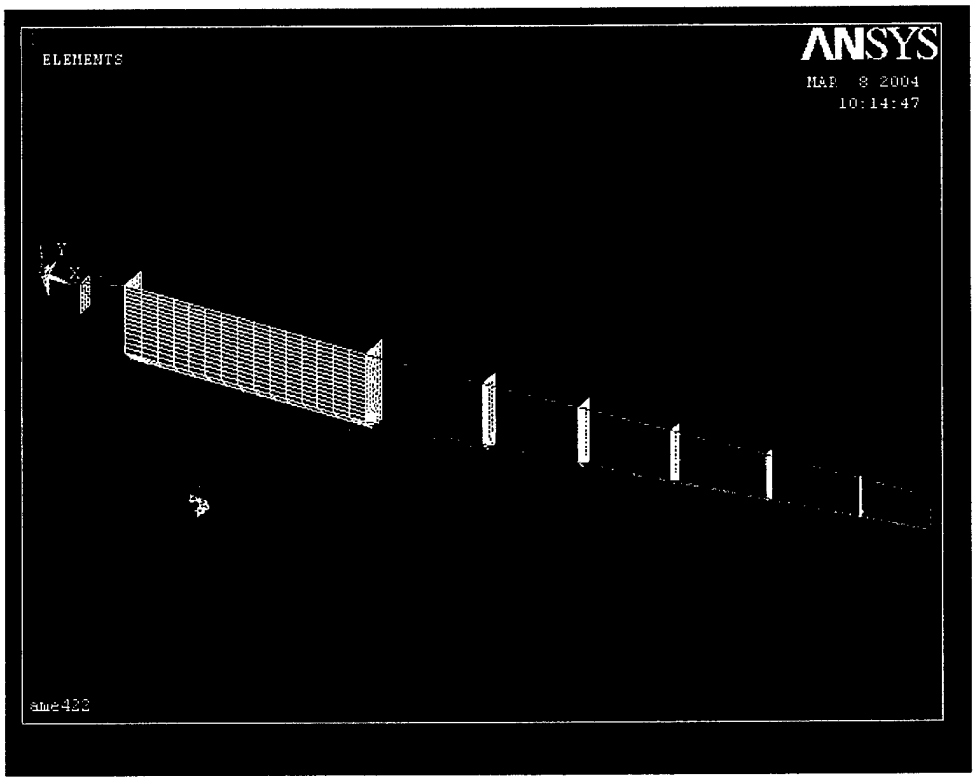
Shear Modulus = 13.92 ksi

Compressive Strength = 500 psi

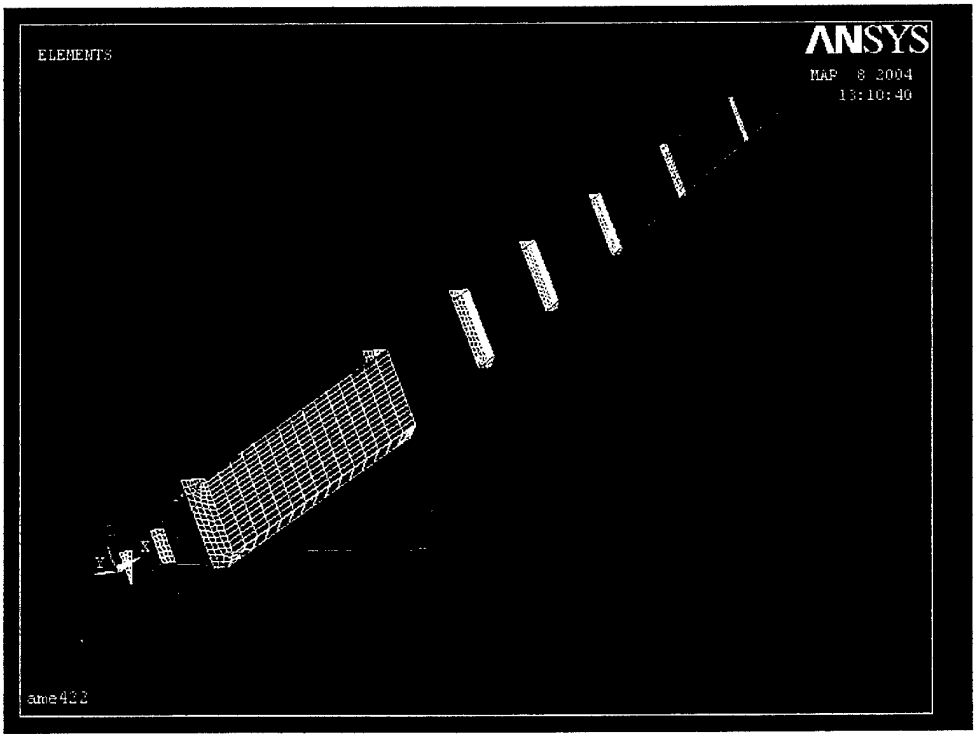
	Weight in Pounds per cubic foot		
Compression Strength (PSI)	**6**	**11**	**15.5**
Compression Strength (PSI)			
Parallel to the Grain (End-Grain)			
Maximum crush strength	750	1910	2950
Modulus of elasticity	330000	768000	1164000
Perpendicular to grain (Flat Grain)			
Modulus of elasticity - High Value	16000	37000	55000
Modulus of elasticity - Low Value	5100	13000	19900
Bending Strength (PSI) 3 point load static bending			
Stress at proportional limit	825	1725	2535
Modulus of elasticity	280000	625000	925000
Tensile Strength (PSI)			
Parallel to grain (End-Grain)	1375	3050	4525
Shear Strength (PSI)			
Parallel to grain - High Value	180	360	522
Parallel to grain - Low Value	158	298	425

20 Appendix J: Structural Analysis

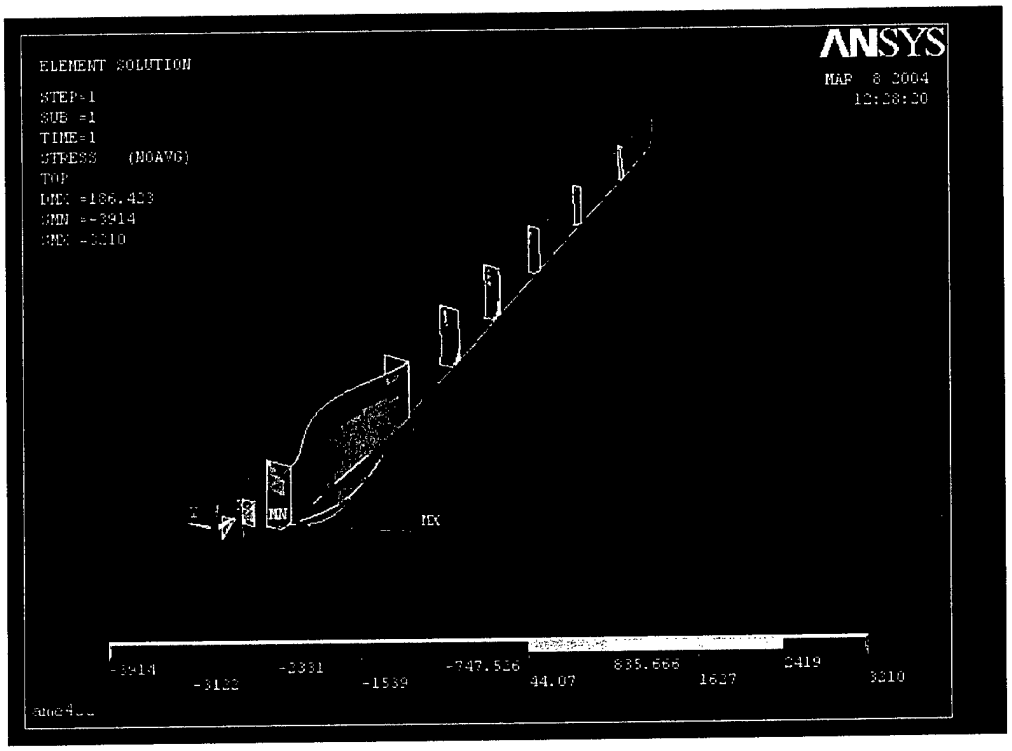
J.1: Configuration of model for landing simulation.



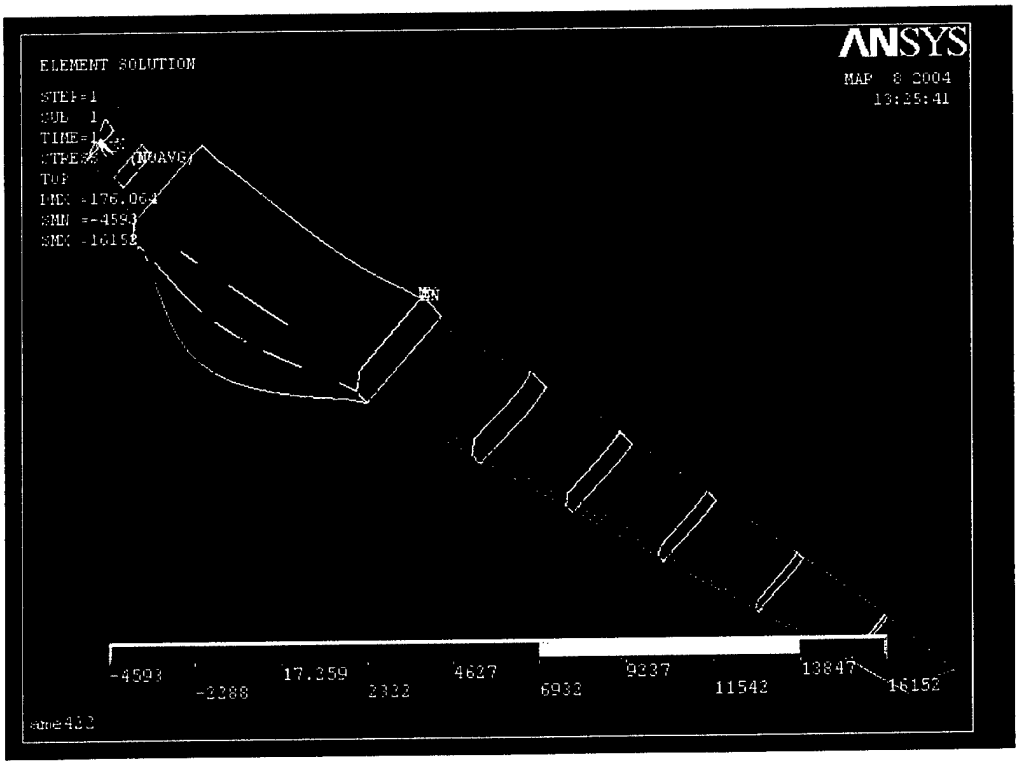
J.2: Configuration of model for in-flight simulation.



J.3: Stress distribution simulating landing case for load factor of 5.



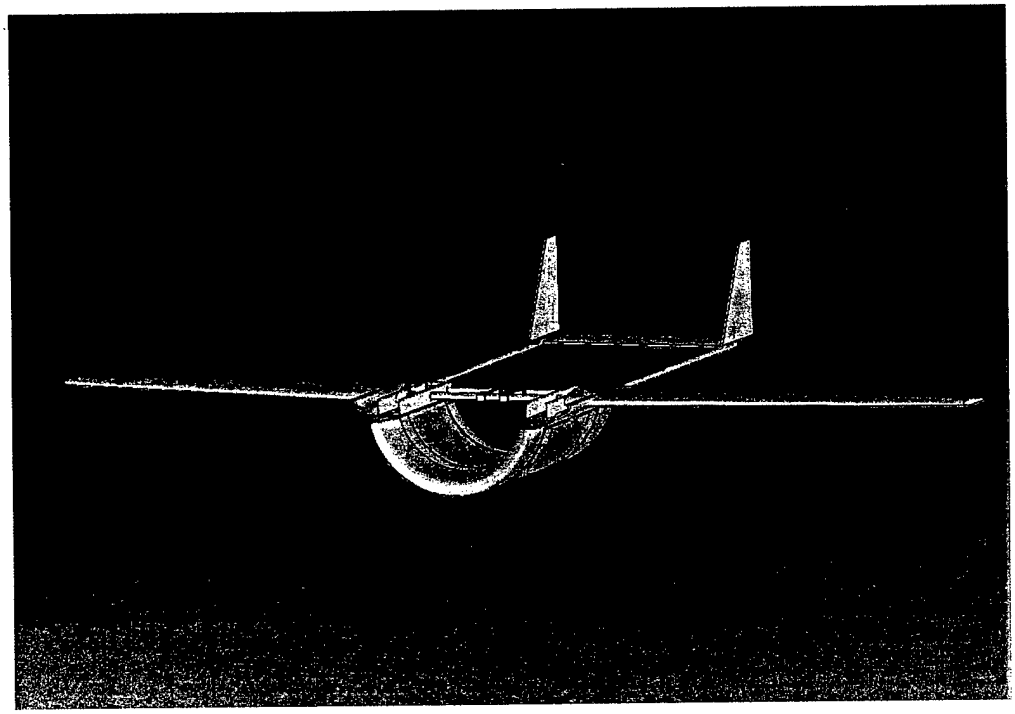
J.4: Stress distribution simulating in-flight case for load factor of 3.



West Virginia University®
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AIAA Student Design/Build/Fly

Team StingRay



Group Members:

Raphael W. Alwin Dodrill

Scott Boas

Brian Launer

R. J. Claus

Micah Arter

CAD Assistants:

Anthony Favorito

Sandeep Vennam

Advisors:

John Loth

Richard Guiler

Table of contents

1 Introduction 1
2 Rules and Requirements 1
2.1 General Requirements 1
2.2 Mission Requirements 2
2.2.1 Fire Bomb 2
2.2.2 Ferry mission 3
2.3 Aircraft Configuration 3
3 General Design Considerations 5
3.1 Wing 5
3.2 Fuselage 5
3.3 Tail 6
3.4 Internal Components 7
3.4.1 Water Storage System 7
3.4.2 Water Dump Mechanism 7
3.5 Landing Gear 7
3.6 Numerical Figures of Merit (FOM) 8
3.6.1 Takeoff Distance Rating (TDR) 8
3.6.2 Fire Bomb Rating (FBR) 9
3.6.3 Ferry Rating (FR) 9
3.6.4 Rated Aircraft Cost (RAC) 9
3.6.5 Total Figures of Merit (TFM) 10
3.6.6 Relation between Figures of Merit and Mission Requirements 10
4 Aircraft Comparison 11
4.1 List of items with major impact on design 11
4.2 Comparison of Weights, Performance and Geometry 12
4.3 Discussion on other aircraft 12
5 Aircraft Design 14
5.1 Wing 14
5.1.1 Wing design considerations 14
5.1.2 Finalized wing design 14
5.2 Fuselage 16
5.2.1 Conventional (First Generation) Design 16
5.2.1.1 3-D Drawings of the Conventional Design 17
5.2.2 Ducted Fan (Second Generation) Design 20
5.2.2.1 Drawings of the Ducted Fan Design 21
5.2.3 Chosen Design 23
5.3 Tail 23
5.3.1 Tail Design Considerations 23
5.3.2 Finalized Tail Design 24
5.4 Internal Components 25
5.4.1 Considerations about the Internal Components 25
5.4.1.1 Final Decisions on the Internal Components 25
5.4.2 Water Storage System 25
5.4.2.1 Considerations on Water Storage 25

5.4.2.2 Final Choices on the Water Storage System 25
5.4.3 Water Dump Mechanism 26
5.4.3.1 Thoughts about the Water Dump Mechanism 26
5.4.3.2 Decisions about the Water Dump Mechanism 26
5.4.4 Control System 27
5.4.5 Propulsion system 27
5.4.5.1 Motor size and number 28
5.4.5.2 Propeller pitch and diameter 29
5.4.5.3 Battery weight and number and cells 29
5.4.5.4 Component Placement 29
5.5 Landing Gear 30
6 Manufacturing Plan 31
6.1 Critical Manufacturing Path 31
6.2 Wing 32
6.3 Fuselage 33
6.3.1 Conventional Design 33
6.3.2 Ducted Fan Design 34
6.4 Tail 35
6.5 Internal Components 35
6.5.1 Water Storage System 35
6.5.2 Water Dump System 35
7 Rated Aircraft Cost 36
8 Testing Plan 36
8.1 Test Glider 37
8.2 Aircraft 37
8.2.1 Ground Test 37
8.2.2 Flight Testing 38
9 Test Results 39
9.1 Test Glider 39
10 Conclusion 39
11 References 41

1 Introduction

This report documents the design of an aircraft, which has been entered into the 2004 AIAA student Design Build Fly competition; for the West Virginia University StingRay team. The design was a team effort to evaluate each component of an aircraft and make optimizations to allow the best performance possible to be achieved from the resulting flight vehicle; within skill, budget, and time constraints.

2 Rules and Requirements

The AIAA rules and requirements had to be taken into consideration arrive at an aircraft design which is most appropriate. The information listed in the following sections was provided by the AIAA Student Design/Build/Fly Competition (AIAA Rules, 2004).

2.1 General Requirements

Since this aircraft will be entered into the AIAA Design Build Fly competition, certain requirements must be accounted for in the design. The most important ones are summarized below.

1) Aircraft must fit in a 2-ft wide by 1-ft high by 4-ft long (interior dimensions) box. Note: The aircraft does not need to "fold" to fit in the box, but may employ "plug-in" joints for rapid assembly/disassembly. All electrical connections should be keyed so they cannot be misassembled. Tape may be used as a non-structural "latch" to hold components in place, such as taping a wing joint when using a plug-in spar arrangement to keep the wing from sliding loose.
2) Take-off on a 150 ft runway
3) On landing the aircraft must land on the runway (but may roll off) to obtain a score for that flight.
4) All payloads must be adequately secured using mechanical means. Tape and Velcro are not acceptable forms of restraint.

5) Maximum mission time is 10 minutes.

2.2 Mission Requirements

Two possible mission profiles were provided. Since the Ferry mission is easier than the Fire bomb, they have been assigned different difficulty factors. The difficulty factors for the Fire Bomb and the Ferry mission are 2.0 and 1.0 respectively. The course layout is shown in Figure 1.

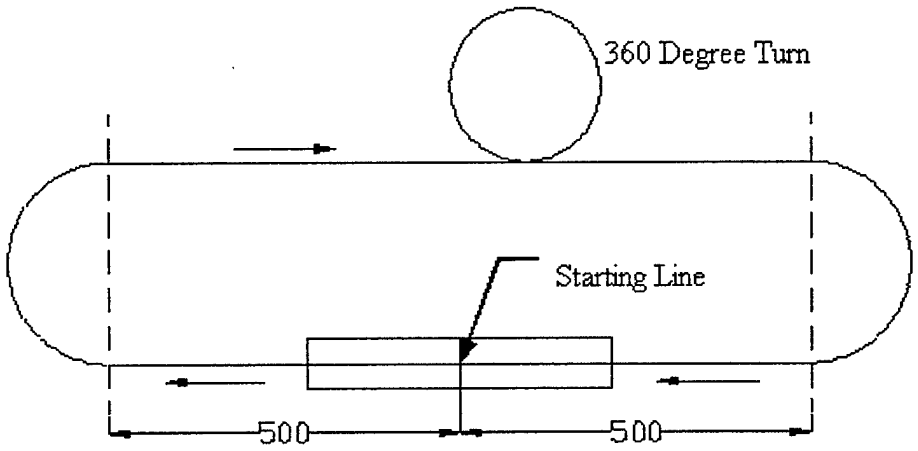


Figure 1 – Course Layout

2.2.1 Fire Bomb

The Requirements for the Fire bomb mission are as follows.

1) Aircraft will begin the mission time empty.
2) The flight mission will consist of two sorties.
   a) Aircraft will be loaded, take-off, dump its water load during the downwind leg, and return to land.
   b) Aircraft will be reloaded, take-off, dump its water load during the downwind leg, and return to land.

3) Maximum allowed aircraft water capacity is 4 liters. *(Note: Aircraft will be emptied then filled to overflow during the tech inspection. Total volume must be 4 liters or less.)*
4) Teams will load the aircraft during the mission from four 2-liter plastic "soda" bottles. Teams may use gravity loading or "pumped" loading. Soda bottles may NOT be pressurized to assist loading.
5) On all laps flown the aircraft must complete a single 360° turn in the direction opposite of the base and final turns on the downwind leg of each lap.
6) Water may only be dumped during the downwind leg between the upwind and downwind turn markers/flags. *Aircraft must fly slow enough to allow sufficient time for the water to be emptied.*
7) Maximum dump orifice diameter is 0.5 inch. *(Measured at the exit so it may be easily verified during the tech inspection).*
8) The pilot must call "Dump On" and "Dump Off" so the observer judge can easily verify that the water dump is restricted to the downwind leg.

2.2.2 Ferry mission

The Ferry mission follows the outline below.

1) Aircraft must take-off, complete four laps, and land. The aircraft will have no water payload for this flight
2) On all laps flown the aircraft must complete a 360° turn in the direction opposite of the base and final turns on the downwind leg of each lap.

2.3 Aircraft Configuration

The requirements on the aircraft configuration and components are outlined below.

1) The aircraft may be of any configuration except rotary wing or lighter-than-air.
2) No payload may be carried internal to the wing proper. Payload in the "fuselage" may be carried in the area where the wing carry-over structure passes. For blended-wing configurations, the "fuselage" is defined to be the

inner most 9 inches of semi-span; the remainder is "wing" for payload considerations.

3) Must be propeller driven and electric powered with an unmodified over-the-counter model electric motor. May use multiple motors and/or propellers. May be direct drive or with gear or belt reduction.
4) All motors must be from the *Graupner* or *Astro Flight* families of brushed electric motors, only (motor should have manufacturers label visible to the judges during the technical inspection).
5) For safety, each aircraft will use a commercially produced propeller. Teams may modify the propeller diameter by clipping the tip, and may paint the blades to balance the propeller. No other modifications to the propeller are allowed. Commercial ducted fan units are allowed.
6) Motors and batteries will be limited to a maximum of 40 Amp current draw by means of a 40 Amp fuse (per motor or pack) in the line from the positive battery terminal to the motor controller. Only ATO or blade style plastic fuses may be used. ("Maxi" size Slow Blow, 1.15"x0.85". Available online www.Mcmaster.com part #7460K51 $1.66 each)
7) Must use over the counter NiCad batteries. For safety, battery packs must have shrink-wrap or other protection over all electrical contact points. The individual cells must be commercially available, and the manufacturers label must be readable (i.e. clear shrink wrap preferred). All battery disconnects must be "fully insulated" style connectors.
8) Maximum battery pack weight is 5 lb. Battery pack must power propulsion and payload systems only. Radio Rx and servos MUST be on a separate battery pack. Batteries may not be changed or charged between sorties during a flight period.
9) Aircraft and pilot must be AMA legal. This means that the aircraft TOGW (take-off gross weight with payload) must be less than 55 lb, and the pilot must be a member of the AMA.
10) Since this is an AMA sanctioned event, the team must submit proof that the aircraft has been flown prior to the contest date (in flight photo) to the

technical inspection team. Contest supplied qualified pilots will be available to teams who require them.

3 General Design Considerations

In an attempt to optimize the performance of the aircraft, different configurations of the various components were considered. These considerations were then evaluated to achieve at the most appropriate aircraft configuration.

3.1 Wing

Since the wing is the major lift producing device, as well as an important control component, on the aircraft, it is critical to find the most appropriate configuration. The placement options are: low-wing, mid-wing, and high-wing. The high-wing configuration offers more stability; while the low-wing makes use of the ground effects to provide more lift during takeoff, which could have adverse effects during landing. The mid-wing configuration offers a balance between the other two options; however its supporting structure occupies space within the fuselage which could be used for other purposes. As a result of these considerations; the high wing was chosen because it allows the best landing characteristics and offers better stability in flight.

3.2 Fuselage

Fuselage aerodynamic considerations were developed based on the sum of the volumes of the internal components. Length, diameter and overall fuselage shape are functions of the arrangement of the internal components. Fuselage components are generally arranged in a manor producing circular conic leading and trailing sections, and a central cylindrical section. Fineness ratio does not need to be optimized for the design of an R/C aircraft. However, fineness ratio should be kept within an acceptable range. Based upon these considerations, the team came up with two designs which best fulfilled the considerations above; a Ducted Fan, and a more conventional fuselage based on the Von Karman ogive.

3.3 Tail

Since the tail provides the horizontal and vertical stability of the aircraft, an analysis of several configurations was conducted. Common tail arrangements are: conventional tail, T-tail, H-tail, and V-tail. Less common tail configurations are: cruciform tail, inverted V-tail, Y-tail, and the twin-tail. The conventional tail design places the vertical stabilizer at or near the fuselage centerline and provides adequate stability and control while reducing the tail weight. The T-tail places the horizontal stabilizer over the vertical stabilizer. This configuration utilizes the flow effects, caused by the joint between the horizontal and vertical tails, to allow the size of the vertical tail to be decreased; it also places the horizontal stabilizer out of the wake of the wing, which permits the reduction of the horizontal tail size. Conversely, this tail arrangement requires the increase of the structural weight to be able to support the horizontal tail. The V-tail reduces the wetted area of the tail by combining the vertical and horizontal tail surfaces. However, this configuration is subjected to the "adverse yaw-roll" effect (a coupling between yaw and roll), which tends to counteract the yawing by rolling in the opposite direction. The H-tail reduces the required horizontal tail size and the vertical tail height. It also places the vertical stabilizers outside of the disturbed airflow of the fuselage. This tail configuration, however, requires an increased structural weight.

The boom tail, which was chosen for this aircraft, places an H-like tail on two booms mounted on the wing. This configuration does not require the extra structural weight of the horizontal stabilizer (which was necessary for the H-tail) because it is supported at both ends, and must not carry the load of the vertical tail sections; it also reduces the wetted area leading to the tail. This configuration incorporates the advantages of the H-tail; and when combined with a pusher-propeller aids in keeping the tail from stalling, since the tail is inline with the airflow from the propeller.

3.4 Internal Components

The internal components consist of the water tank, battery packs to run the motor, the motor mount, a battery pack for the receiver, and the receiver itself. Some of the servos may need to be mounted inside the fuselage. Some of these items can/must be purchased off the shelf; however others may have to be built. This section addresses possible construction methods for these components.

3.4.1 Water Storage System

Only two options were available on the structure of the water tank; either to have a physical tank that would be placed inside the fuselage, or to integrate the "tank" into the skin of the fuselage itself. Since the fuselage construction utilizes composites; it was decided that the water storage tank would be integrated into the fuselage itself. This could be achieved by lining the inside shell of the aircraft with a layer of fiberglass/epoxy composite. Baffles could then be installed into this "tank" to keep the water from sloshing during flight.

3.4.2 Water Dump Mechanism

The water dumping mechanism allowed us to become creative. One Idea was to control water flow with an exit valve that would be mounted on the bottom of the aircraft. A servo would work the valve and control the water flow. Another idea would be to control the water flow by opening and closing the water vent. With this idea a ball bearing would be used to plug up the ventilation tube which allows us to control the water flow. However, when utilizing an exit draining out of the bottom of the tank, plugging the vent would not keep the water from draining; since an air column would form in the exit, allowing water to exit. Thus it was deemed vital to regulate the exit rather than the vent hole.

3.5 Landing Gear

We had to use a tricycle landing gear since we had the propeller mounted in the rear. The landing gear could either be constructed or bought. The Robart

struts would be a very good choice. A home-built landing gear (seen in Figure 2) could be constructed using carbon rods, an aluminum hinge, and an R/C car shock. Another alternative would be to design a shock absorbing mechanism that would be integrated into the fuselage design.

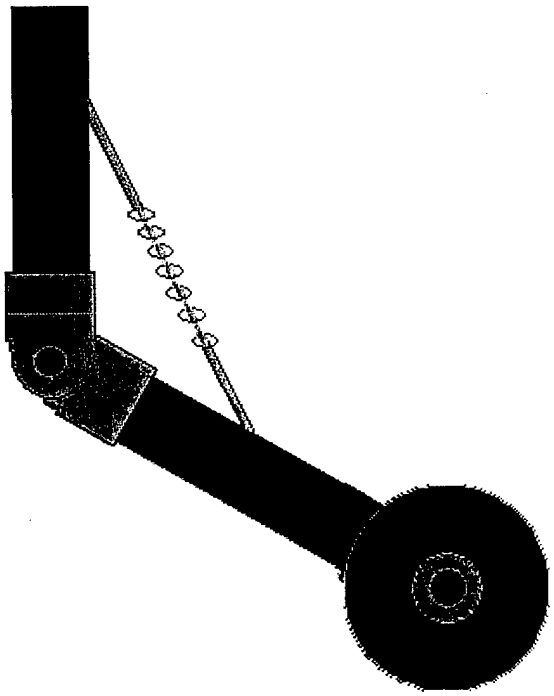


Figure 2 – Home-Built Landing Gear Design

3.6 Numerical Figures of Merit (FOM)

A system using FOM, shown below, will be used to evaluate the performance of the aircraft. The evaluation will reflect the mission requirements for the specified mission being flown and provide direct feedback for the performance of the aircraft.

3.6.1 Takeoff Distance Rating (TDR)

The aircraft must takeoff the runway (meaning wheels off the ground) within 150 feet of the starting point. If the aircraft does not takeoff within 150

feet, then it will be disqualified. There is no advantage for a short takeoff; therefore, TDR is defined as

TDR= (0.0 if takeoff is >150ft) (1.0 if takeoff is <150ft)

3.6.2 Fire Bomb Rating (FBR)

The FOM reflects the fire bomber mission. The mission has a difficulty rating of 2.0 and is defined as

FBR= 2.0/Estimated Flight Time*10min

3.6.3 Ferry Rating (FR)

This FOM was used for the ferry mission who has a difficulty of 1.0 and is defined as

FR= 1.0/Estimated Flight Time*10min

3.6.4 Rated Aircraft Cost (RAC)

The RAC was required by the contest rules and was used as a way to rate the cost of the aircraft on a scale which was made by the contest officials. The RAC includes the structure of the aircraft as well as the components involved in manufacturing the aircraft. It is defined as

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

Where MEW is manufacturer's empty weight, defined as the Actual airframe weight with all flight and propulsion batteries but without any payload: REP is rated engine power and is defined as

REP= (1+.25*(# engines-1)) * Total Battery Weight (lbs)

And the MFRH is the manufacturing man hours, defined as

MFRH= 10hr/ft^2 Wing Span * Chord * # wings

+ 5 hr * control_function_multiplier

+ 20 hr/ft^3 Body Length x Width x Height

+ 5 hr/servo or motor controller

+ 5 hr/Vertical Surface without an active control, 5hr/Vertical Surface with an active control, and 10 hr/Horizontal Surface. A horizontal surface is a "wing" if it is more than 25% of the span of the greatest span horizontal surface

3.6.5 Total Figures of Merit (TFM)

The TFM was calculated from the best two mission scores, the TDR, and the RAC on the following equation

TFM= TDR(FBR+FR)/RAC

Where input is a function that returns the sum of the larger of the two arguments.

3.6.6 Relation between Figures of Merit and Mission Requirements

Table 1 lists the relation between the FOM and the Mission Requirements. The storage of the aircraft does not depend on the score received as long as the aircraft fits into the proper size box: therefore, it was not included in the table.

Table 1 – Relationship between FOM and Mission Requirements

Mission Requirements	FOM
Takeoff	TDR
Fire Bomber DF	FBR
Ferry DF	FR
Aircraft Costs-Effectiveness	RAC

Table 2 – Numerical Figures of Merit

Mission Requirements	FOM
Takeoff	1
Estimated Time for Fire Bomb	9
Fire Bomber DF	2.222222222
Estimated Time for Ferry	7
Ferry DF	0.142857143
Aircraft Costs-Effectiveness	9.08
Overall FOM	0.260471296

4 Aircraft Comparison

This section was used to document features of similar aircraft that may be relevant to the overall configuration selection of the design. This report focuses on the main wing, but information collected in this section will be used in follow-on reports for other configuration items.

4.1 List of items with major impact on design

To determine the most important factors of the wing design, the AIAA rules for this year's competition were examined. The size restriction was a major factor in determining wing dimensions. The wing must fit in a 4' X 2' X 1' box. This limited the total length of the wing to eight feet. Knowing this, the first objective of the competition is to take off within 150 feet. A high lift design was necessary for this aircraft. Straight and level flight performance was also a desirable characteristic. High wind weather conditions could possibly cause time to be lost due to perturbed flight. Poor stability could also cause the aircraft to crash. The aircraft must also be designed to have enough maneuverability to complete a 360 degree turn quickly. Payload release was also a major factor in the wing design. Payload is approximately 1/3 of the entire weight. A rapid payload release will cause the wing to experience a sudden high increase in load. The wing will undergo a major L/D change in a short period of time.

4.2 Comparison of Weights, Performance and Geometry

The following table (Table 3) is a collection of weights and geometric parameters of similar aircrafts. No performance data was listed for these particular aircraft. The data found here was used to aid in the design aspects of the proposed aircraft.

Table 3 – Comparisons of weights and geometrical parameters

Name	Type	W_{TO}	W_E	Span (b)	AR	Airfoil	Source
Syracuse University	High-wing, tail dragger	19 lbs	12.5 lbs	7 ft	7	Clark Y	AIAA webpage
UCLA	High-wing, pusher, twin boom	16 lbs	8.5 lbs	12 ft	13	NACA-4415	AIAA webpage
WVU	High-wing, twin boom	17 lbs	9.5 lbs	10 ft	11	NACA-4409	AIAA webpage
Pepe	High- wing, tail dragger	14.2 lbs	7 lbs	8.2 ft	13	SD7003	AIAA webpage
Queen's University	High-wing, tail dragger	16.3 lbs	8.8 lbs	10 ft	10	Selig-1210	AIAA webpage
Sub-Zero	High-wing, V-tail, tricycle gear	17.3 lbs	9.8 lbs	11.25 ft	10.5	Selig-73014	AIAA webpage

4.3 Discussion on other aircraft

The mission of the 2001 Design, Build Fly competition was to carry as many tennis balls as possible around a course as fast as possible. The Utah State "A" was one of the more radical designs for this competition. The configuration of the Utah State "A" was an elliptic wing and elliptic tail monoplane with retractable landing gear and a horizontal NACA 0015 fuselage. The main wing was designed as an elliptic wing placed in a mid-wing position. The mid-wing location allows the aircraft to maneuver very well. An elliptic plan-form provides the best overall lift distribution. Un-swept, untwisted, elliptic shaped wings produce the least amount of induced drag. This aircraft was designed to takeoff and land in short distances. The ability to land in a short distance decreased flight time, which increased overall team score.

The Oklahoma State "Orange" (OSU) was a low-wing, v-tail monoplane. Payload capacity was very important during this competition. The need for a large fuselage limited the OSU team to either a low or mid-wing design. The low-wing offers a fair amount of maneuverability compared to the high-wing. Low-wing aircraft also benefit from short takeoff and it also makes for an easy landing. These two aspects were very important for this competition. Landing was a major

concern of the OSU team. The v-tail design avoids tail damage during rough landings. A single tractor style propeller decreases flow separation over the fuselage, this increases stall performance. The sandwich fuselage allowed the OSU team to maximize the amount of space available under the DBF weight limit. This type of fuselage also optimized the structural stability of the aircraft.

San Diego State (SDSU) designed an aircraft to have a low-wing, traditional tail, with a lifting body fuselage. The lifting body fuselage allows the entire wing/body section of the aircraft to be used as a lifting surface. SDSU found that 24 softballs was the optimum payload for this competition. A fairly thick fuselage airfoil was needed to carry the heavy payload. The main wing had a rectangular plan-form with tapered wing tips. SDSU attempted to take advantage of the short takeoff and self-landing afforded by a low-wing design. This team was very concerned about landing ability of the aircraft. The landing gear was designed with only two struts and a strut to support the centerline of the aircraft. The tractor style propeller gives this aircraft an advantage during climb. The lifting body design was excellent for stall performance with this propeller location.

The Utah State "Zephyr" (USU) was a low-wing conventional tail aircraft designed to complete two of three mission proposals for the 2003 DBF competition. The two missions were to deploy sensor equipment and missile decoying. The Utah State team attempted to optimize the aerodynamic forces on the aircraft. The strict takeoff and landing requirements most likely affected wing location for this aircraft. Takeoff and landing distance was limited to 120 ft. for this competition. The low-wing location allows decreased takeoff and landing distances. A low wing is a self landing configuration, which was also a major concern for the competition. To decrease the drag effects it was necessary to efficiently retract the landing gear.

The Texas A&M "Flash Flood" (TAMU) was built under very similar constraints as the 2004 aircrafts. The Flash Flood team felt that stability and control were the most important factors for their aircraft. This aircraft was designed to have a low mono-wing, conventional tail, single fuselage, three

engine tractor style aircraft. The low wing was used because it provided the easiest, fastest access to the payload. Engines were placed on both wings and nose of the Flash Flood. This team also decided to have one horizontal and one vertical tail behind the wing. This traditional tail provided the necessary stability and control the aircraft needed. A tricycle landing gear was used to provide steering control without risking ground loops or trip-overs. During this particular competition course selection was not chosen until the day of the competition. An asymmetric design would have made it difficult for the team to perform stability and control calculations with out knowing the course layout.

5 Aircraft Design

This finalized design incorporates the best aspects of each individual component configuration considered. The following sections discuss the design considerations and resulting configuration; they also list the important parameters for each.

5.1 Wing

5.1.1 Wing design considerations

In the original design, the wing featured an un-swept trailing edge and had a swept leading edge. In trying to construct this design, it was noted that this design would make it hard to keep our quarter chord spar straight. To ease building the design was changed so that both the trailing edge and leading edge should be swept to allow the spar, running along the ¼ chord, to remain a straight line.

5.1.2 Finalized wing design

The wing contribution to the Rated Aircraft Cost (RAC) is proportional to the sum of the wingspan and maximum chord length. A rectangular plan-form, with an aspect ratio equal to 1, provides the minimum RAC per unit area. A penalty is placed on high aspect ratio wings. An attempt to design the most efficient airplane possible while maintaining acceptable handling characteristics

was a primary goal. Wing plan-form has a significant effect on aircraft performance. Since the trapezoid wing plan-form ace be made to have close to en elliptical lift distribution, yet are easier to construct than al elliptical wing; they prove to be more favorable choice than the rectangular wings evaluated. Wing configuration greatly affects the performance of an aircraft. Depending on the given wing configuration, greater lift slope may be obtained, efficiency increased, drag reduced, and structural requirements simplified. Winglets were not considered for any case because of the small performance advantage compared to the large rated aircraft cost penalty.

These considerations led to the decision on utilizing the trapezoidal wing plan-form. The next step was to choose the airfoil to be used. The driving factors in the airfoil selection were the implications on the RAC, and the aerodynamic efficiency (L/D). Since the NACA 2412 airfoil provides a high Lift/Drag ratio, is easy to cut, and is widely used in R/C and General Aviation aircraft; it was deemed to be the best option for the desired purpose.

In an attempt to achieve a more elliptical wing loading it was decided to divide the wing into a central section with a rectangular profile, and the trapezoidal wing tips. The size and magnitude of the vital parameters are listed in Table 4.

Table 4 – Main wing Parameters

Parameter	Value	Units
Overall Span (b)	97.5	in
Span of stubs on fuselage	3	in
Wingtip Span (b_W)	37	in
Wingtip Tip Chord (C_t)	6	in
Wingtip Root Chord (C_R)	14	in
Wing Area (S_W)	824	in^2
Wing Aspect Ratio (AR_W)	11.53671117	
Airfoil	NACA 2412	
Wingtip Sweep Angle	3.094058059	Degrees
Wingtip Sweep Angle	-9.211026541	Degrees
Dihedral Angle (G)	0	Degrees
Angle of Attack (a)	6	Degrees
Wingtip Washout Angle	5	Degrees

5.2 Fuselage

Since the fuselage is the primary section of the aircraft; two different designs were developed to allow the choice of the optimum profile. These include a more conventional shape and a ducted fan system. They are discussed below in more detail.

5.2.1 Conventional (First Generation) Design

It was desired to design a fuselage with smooth dimension changes and blended curves without significantly increasing surface area. The main concern was to prevent flow separation over the fuselage. The Von-Karman Ogive is a symmetric body of revolution. Due to the low drag coefficient this fuselage shape produces it was expected to perform well in adverse weather conditions. The vital data concerning the fuselage is listed in Table 5.

The standard design underwent several changes to achieve optimal performance. It was decided that the best way to create the fuselage was to make the entire fuselage a water tank. In order to do this we needed to move most of the components in the fuselage to other parts of the aircraft. To do this,

servos and the receiver pack were moved to the center section of the main wing. Upon calculating the center of gravity, it was discovered that some of the components would need to be placed further forward than the main wing. To solve this problem a baffle was sketched into the design allowing for some payload space forward of the water tank. This would allow us to correct the center of gravity and created more room to mount components.

Table 5 – Conventional Fuselage Parameters

Parameter	Value	Units
Length	22	in
Max Diameter	6	in
Profile	Von-Karman Ogive	

5.2.1.1 3-D Drawings of the Conventional Design

The illustrations are shown below in Figure 3 through Figure 7 depict a CAD model of an aircraft utilizing the conventional fuselage design.

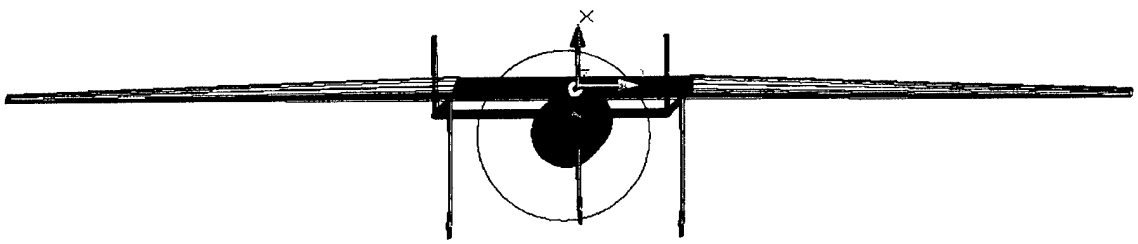


Figure 3 – Front view of aircraft

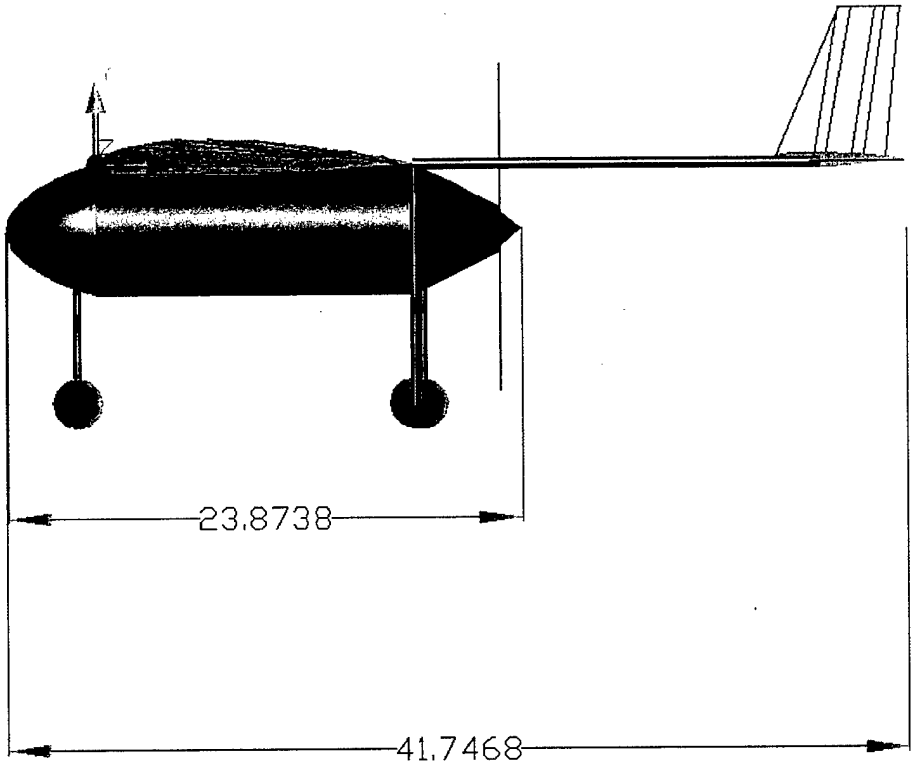


Figure 4 – Side View of Aircraft

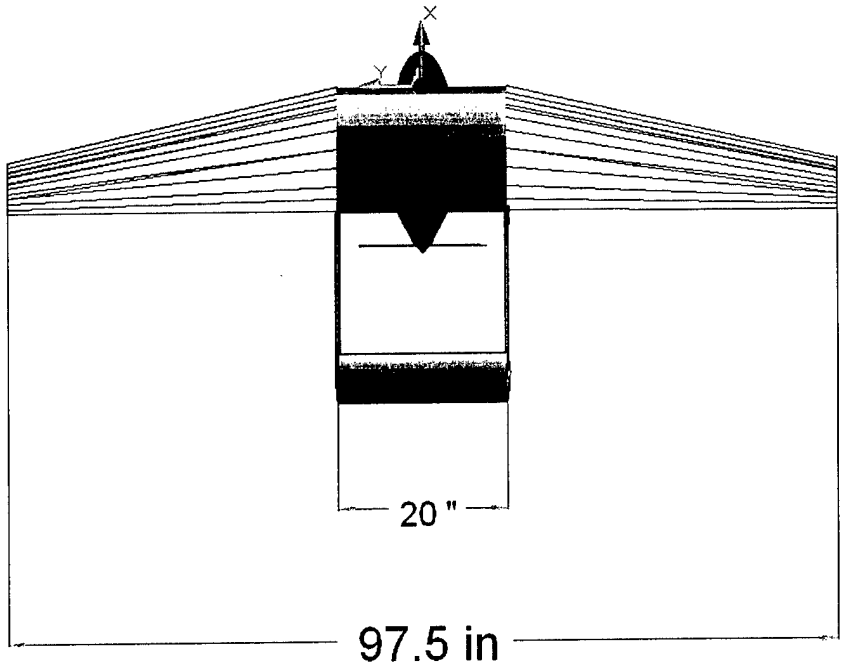


Figure 5 – Top View of Aircraft

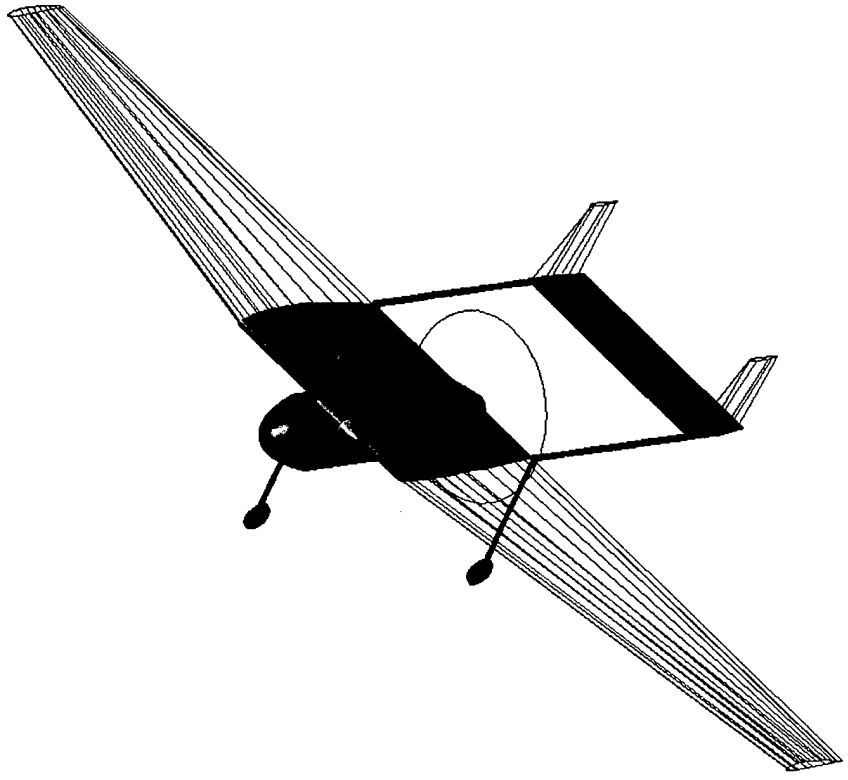


Figure 6 – Upper Isometric View of Aircraft

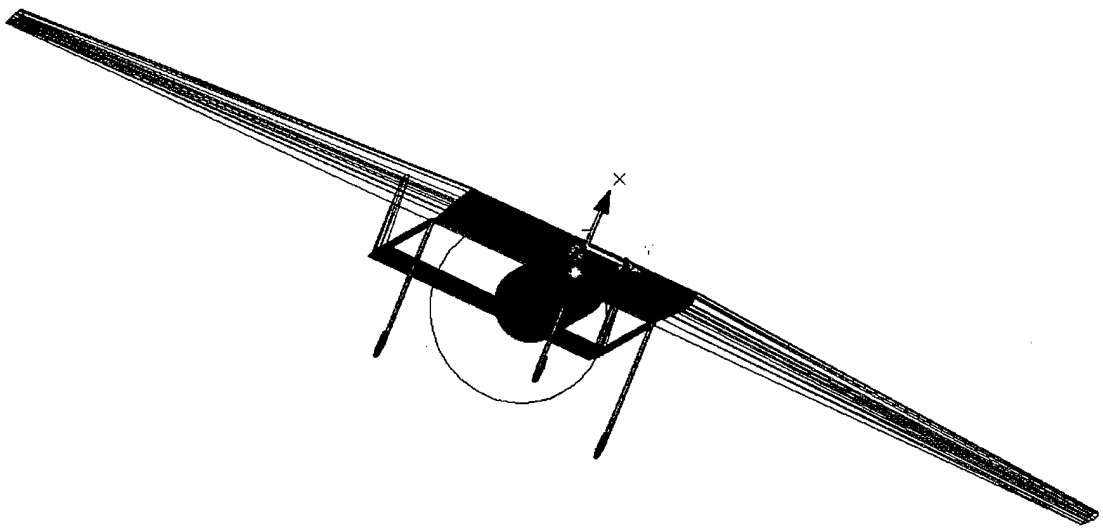


Figure 7 – Lower Isometric View of Aircraft

5.2.2 Ducted Fan (Second Generation) Design

This year the difficulty was a quick draining tank on board. The competition model aircraft will need to travel around a designated course while at the same time carrying and unloading up to four liters of water. To accomplish this goal, several factors must be considered. Overall weight is of primary concern, along with an acceptable thrust-to-power ratio for the chosen electric motor. One major step towards increasing the overall thrust is to reduce the aircraft drag as much as possible. Another improvement to thrust can be accomplished by keeping the propeller out of the fuselage wake. This is an especially critical aspect for an aircraft using a pusher type prop, since the propeller itself operates inside the turbulent wake of the fuselage.

Replacing the conventional tank/fuselage with a semicircular shrouded tank/fuselage can eliminate the problem of propeller inefficiency operating in wake drag. Centering the motor in the shroud removes any obstruction directly in front of the prop and dramatically increases power plant efficiency. By creating a shroud exit which is larger in diameter than the shroud inlet (known as wake expanding), the thrust-to-power ratio can be increased. By running the propeller

blade tips in a groove inside the shroud, a trapped vortex is formed which deflects energizes the outflow to a larger prop wake diameter. This prop wake is then used to fill shroud/tank wake drag. Although the semicircular shroud will increase the total surface area of the aircraft, the increase in skin friction will be offset by the addition of increased lift from a semi shroud; in addition, a 15% increase in thrust is achieved. An added benefit of this shroud is an increased water depth compared to a conventional fuselage depth. This larger difference in the total height of the tank will aid in faster discharge of the water. The vital sizing parameters are listed in Table 6.

Table 6 – Ducted Fan Fuselage Parameters

Parameter	Value	Units
Length	18	in
Width	17	in
Height	12	in

5.2.2.1 Drawings of the Ducted Fan Design

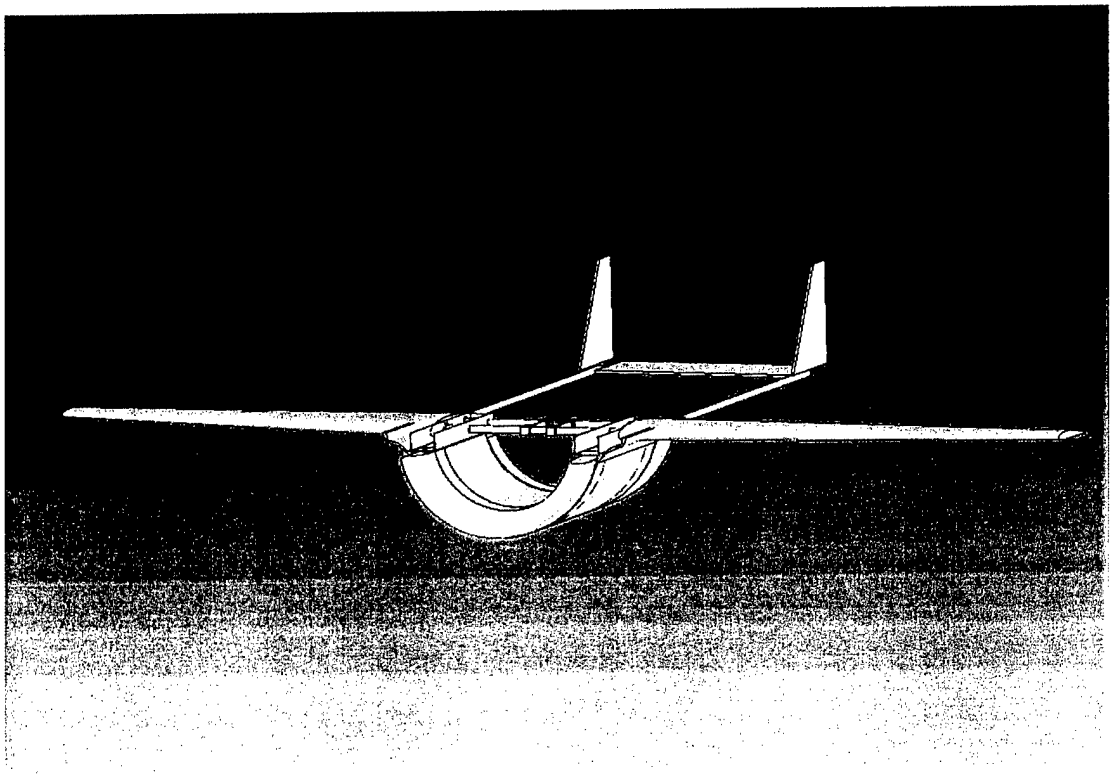


Figure 8 – 3-DIsometruc View of the Ducted Fan Fuselage

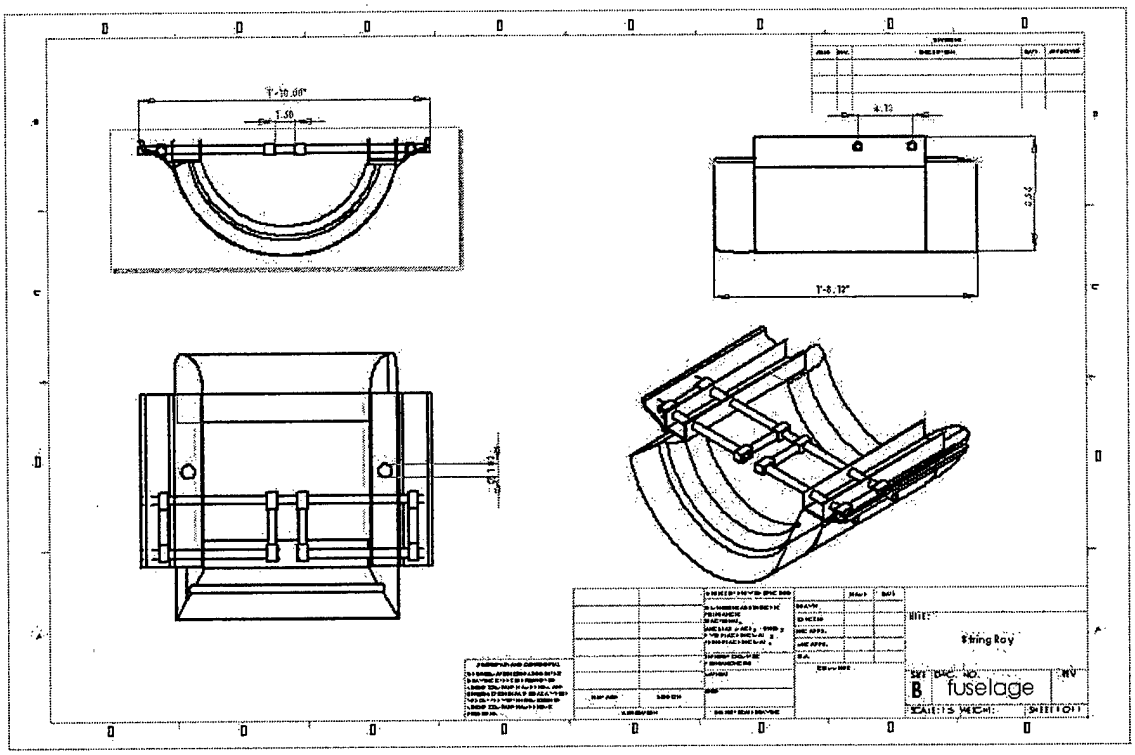


Figure 9 – 3-View Drawing of the Fuselage

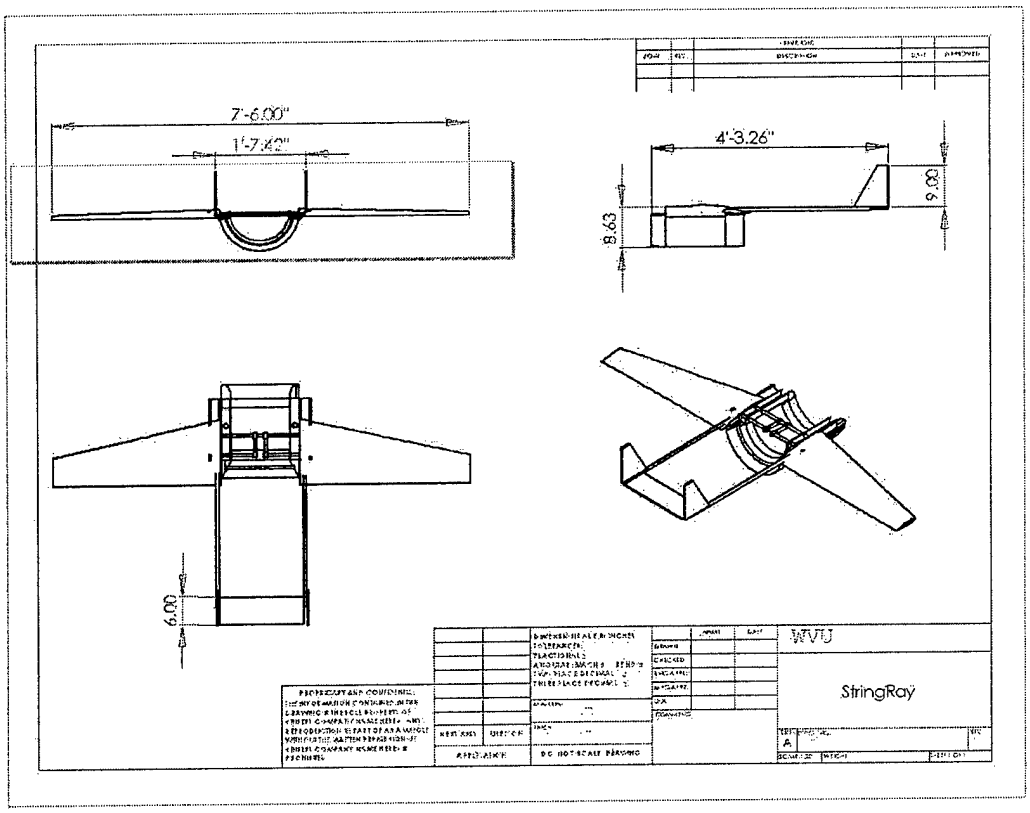


Figure 10 – 3-View Diagram of the Aircraft

5.2.3 Chosen Design

The Ducted fan design was chosen because the majority of the vital components; i.e. shock absorbing gear supports, water tank, etc.; were integrated into the construction of the fuselage.

5.3 Tail

5.3.1 Tail Design Considerations

Two designs were considered for the stabilizers; a balsa frame covered with monocote, and a carbon/honeycomb composite lay-up with rounded leading edge (made from balsa stock) with a solid balsa control surface. The composite structure proves to be the better of the two since it is stronger, lighter, and more aerodynamic than the sizing required for the balsa frame design.

5.3.2 Finalized Tail Design

Since this aircraft makes use of a pusher-prop, it was decided to make use of the boom tail. Another factor that prompted this choice was that the stall characteristics of the horizontal stabilizer were improved by placing it almost right inline with the prop-wash. The physical profile of the stabilizers is a flat plate with a rounded leading edge and sharp trailing edge. This design allows ease of production and lower drag than using a "fat" airfoil shape. The sizing parameters are listed in Table 7. Figure 11 below shows the approximate location of the tail on the fuselage.

Table 7 – Vital Tail Parameters

Horizontal Tail			Vertical Tail		
Parameter	Value	Units	Parameter	Value	Units
Span	19.375	in	Span	9	in
S_{HT}	116.25	in^2	S_{VT}	72	in^2
l_{HT}	27	in	l_{VT}	29	in
Aspect Ratio	3.229166667		Aspect Ratio	1.125	
Taper Ratio	1		Taper Ratio	1.66667	
Chord	6	in	Root Chord	10	in
			Tip Chord	6	in
			Sweep Angle (Λ_{LE})	23.962489	Degrees

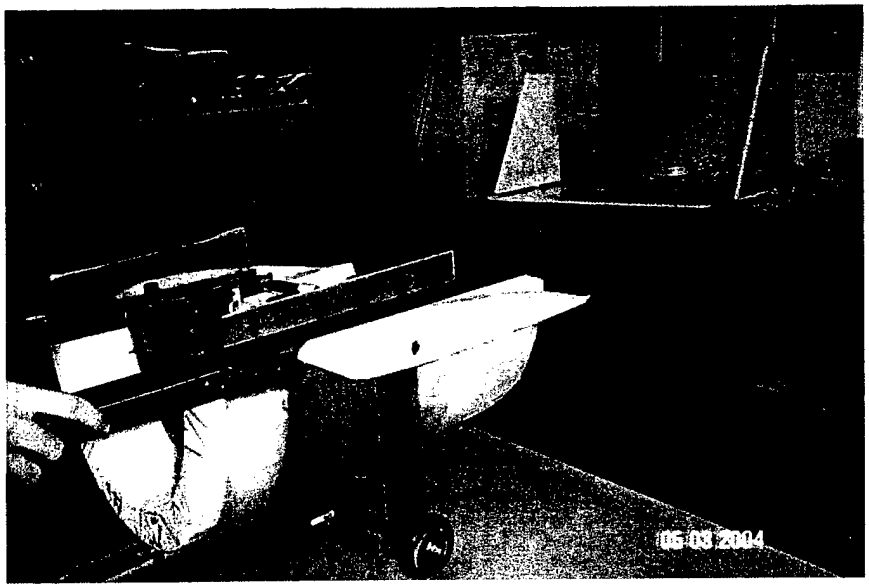


Figure 11 – Approximate Location of the Tail

5.4 Internal Components

5.4.1 Considerations about the Internal Components

Internal components didn't change very much. Initially, the servos were going to be the Hitech hs-81 metal gear servos. These are micro servos that would have fit in our main wing section. They were replaced with some standard servos as worries about torque and stripped gears began to arise. The receiver battery pack was also changed, from a 4.8 volt pack to a 6.0 volt pack, to boost the servo power.

5.4.1.1 Final Decisions on the Internal Components

The internal components will be kept to a minimum. The internal components will consist of the water tank, battery packs to run the motor, the motor mount, a battery pack for the receiver, and the receiver itself. These components will be attached to the structure of the fuselage; i.e. servos can be placed atop the motor mount.

5.4.2 Water Storage System

5.4.2.1 Considerations on Water Storage

The water was initially going to be stored in a tank that would sit inside the fuselage. For simplicities sake, it was decided that the fuselage should be turned into the water tank. This could be done by constructing the fuselage as a waterproof tank during the initial composite lay-up.

5.4.2.2 Final Choices on the Water Storage System

Since the Ducted Fan fuselage design was chosen; the water storage tank will be integrated into the skin of the fuselage itself. The walls of the tank will be lined with a fiberglass-epoxy layer; utilized to provide a water-proof seal. Baffles, also made up of the glass composite, will be installed to keep the water from sloshing.

5.4.3 Water Dump Mechanism

5.4.3.1 Thoughts about the Water Dump Mechanism

The water dumping mechanism was going to be designed to siphon the water out of the tank; seen in Figure 12. After tests were run, it was discovered that utilizing just a vent at the top of the tank, and an exit hole (controlled by a valve) at the bottom would allow a faster release of payload. Being able to start and stop water flow was incorporated into our design from the beginning. It was later realized that it would be easier to have a way to open the tank and then let it all flow out.

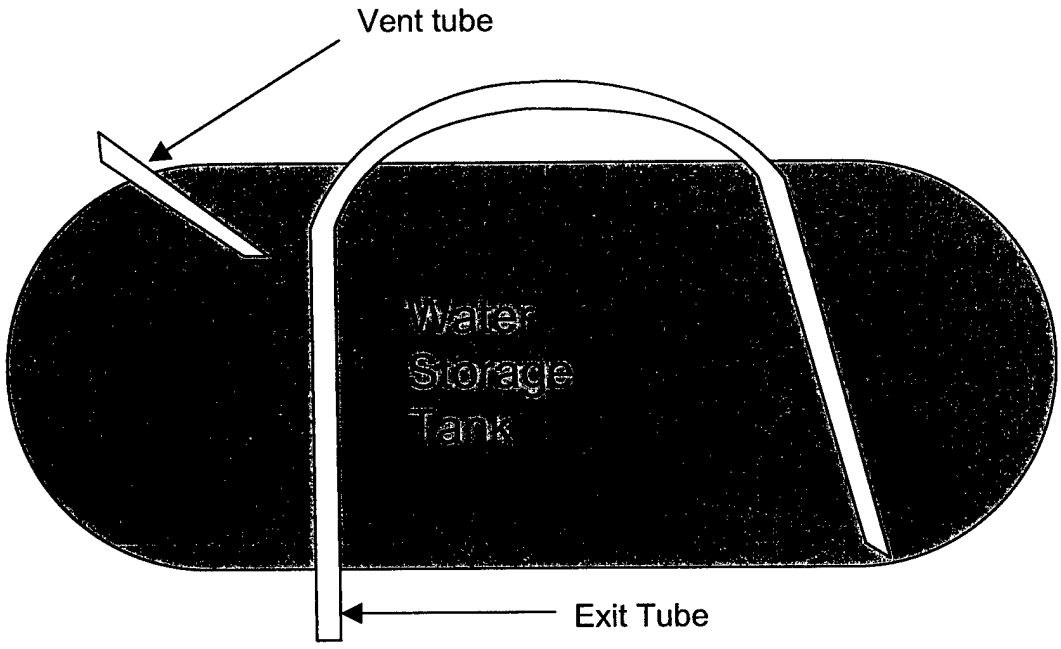


Figure 12 – Siphon Dump Mechanism

5.4.3.2 Decisions about the Water Dump Mechanism

Since draining the water right out of the bottom of the water tank proved to be the best method for dumping the water; it was decided to have vent holes at the top of the fuselage. This vent could also be fitted with a quick-release hose fitting for use during the filling procedure. A ½" ID fitting would be placed at the

bottom at the tank to serve as the exit hole. This exit would then be plugged with a stopper, which would be pulled out by means of a servo. This stopper would have to be re-inserted prior to re-filling the fuselage.

5.4.4 Control System

The aircraft will be controlled using 6 servos and an electronic speed control. The elevator will be controlled with one servo. The elevator servo will be mounted in the fuselage and will have push rods traveling down one of the tail booms. The Rudder and the front nose wheel steering will share a servo. This servo will be mounted in the fuselage and have pushrods running down one of the tail booms. It will also have another pushrod going to the front wheel. The two rudder sections will be connected with a rod so they will move in unison. Using bell cranks we can use one servo for the ailerons, and one for the flaps. The aileron and flap servos will be mounted on the motor. The water dump mechanism will also use a servo. It will be mounted near the exit hole. An electronic speed control will control the throttle on our aircraft. Another servo will be added to control the airbrakes that will be used to stop the aircraft.

5.4.5 Propulsion system

The program P-Calc available at http://brantuas.com/ezcalc/dma.asp was validated using experimental data. Comparisons were made between three specific sets of motor, amperage and thrust data collected for the Astro-flight Cobalt 60 Direct drive motor. The same motor, battery and propellers were then entered into the P-calc program to see how accurate the program was. After running the different combinations, it was found that P-Calc was accurate within 7.3% in power, 2.3% in motor RPM and 15% in thrust. With this information, it was determined that the P-calc program could be used as an effective design tool that fairly accurately predicts power usage, motor RPM and with a correction factor the total thrust.

Table 8 – Experimental Data vs. P-Calc (Bouchard, 2004)

Prop	Amps Exp.	Amps P-Calc	Watts Exp.	Watts P-calc	RPM Exp.	RPM P-calc	Thrust Exp.	Thrust P-Calc	↔	% Diff. Watts	% Diff. RPM	% Diff. Thrust
12x6	20	23.876	660	642.9	10300	10535.3	13.75	12.5	↔	2.6678	2.2844	10
12x8	25	28.477	800	745.6	10000	10057.9	16.67	14.498	↔	7.2933	0.5793	14.96
14x7	30	37.3	950	907.2	9700	9142.93	19.79	17.64	↔	4.7155	5.743	12.2

The goal in designing the propulsion system was to create the lightest system that would provide enough power to complete all of the mission requirements. The validated P-calc program was then used to try all the possible motor, battery, propeller combinations for the Astroflight 60 and 90 motors. The results of these many iterations were that an Astro Flight 660 series motor with 30 battery sub-C NiCd cells and a 13 X 10 propeller was the most efficient combination. This setup will provide enough thrust for lift off, draws just under 40 amps and was the lightest propulsion package which provided the needed power and endurance. Once this combination was determined from P-calc, actual motor optimizing was done. To determine the RPM's an RPM gauge was used. While doing the flight testing and amperage meter was used to check the amperage that was being drawn during flight. After verifying the accuracy of the Propulsion numbers through P-calc and actual testing, it was determined that the above combination is sufficient.

Table 9 – P-Calc results for Actual Propulsion System

Motor	Batteries Nicads	Prop	Full load Amps	Watts	RPM	Thrust (lbs)
Cobalt 60 direct	o CP-SCR-17	13x10 APC	39.188	855.6	9465	16.64

5.4.5.1 Motor size and number

The main criteria used in selecting the motor for the preliminary design, was take off distance, weight, amps drawn, and RAC. Because of weight and RAC considerations, there was never an option to use more than one motor. Since the number of motors was set at one, it was necessary to find a single motor that could produce enough thrust to allow take off within the designated 150 ft of runway, but would also draw 40 or less amps which is the maximum amperage allowed within the rules.

5.4.5.2 Propeller pitch and diameter

The main goal in selecting the propeller was efficiency. Since larger propellers are more efficient, a large propeller was considered. However, because of the low speeds at which this aircraft will travel, it was determined that a smaller propeller should be used due to the fact that they have smaller pitch to diameter ratios which allow for better efficiency at flight speed. Besides that, ground clearance also had to be considered when selecting the propeller for the aircraft. A reduction in propeller diameter saves weight in the propeller as well as with landing gear because the gear can be shorter.

5.4.5.3 Battery weight and number and cells

The number of batteries for this aircraft was a major design factor. Because of the low weight that will be accomplished by using composite materials, the batteries were one of the heaviest components that could be varied. To make a final determination on the number of cells, the total battery power for the mission was looked at. The goal was to have just enough power to finish the mission. Of course, a factor of safety had to be introduced. Because of natural occurrences such as wind and rain, it is not good enough to assume laboratory conditions.

5.4.5.4 Component Placement

The motor was placed just aft of the ¼ chord of the wing. This placement is such that the prop will spin in a half "shroud" such that a 15% increase in thrust results. The batteries were placed in channels on either side of the fuselage. These channels allow the batteries to be moved fore/aft to achieve the desired CG location. The locations of these components can be seen in Figure 13.

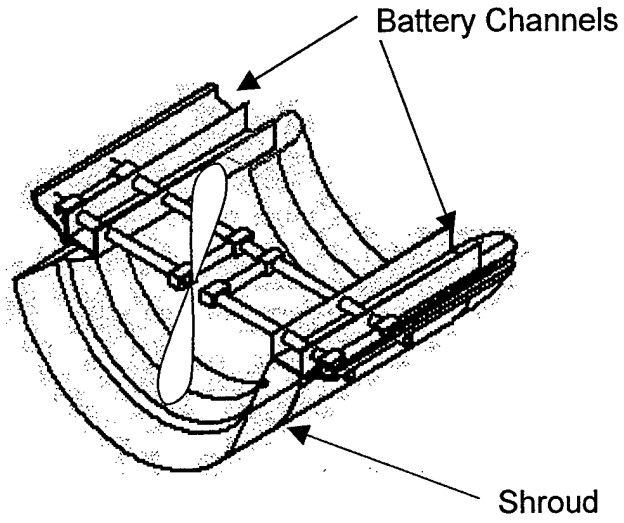


Figure 13 – Internal Component Location

5.5 Landing Gear

The landing gear will be in a tricycle pattern under the fuselage. A carbon tube runs across the bottom of the fuselage serving as an axle. A piece of fiberglass sheeting runs down either side of the fuselage connecting to the axle to act as shocks. A metal channel runs along the bottom of the fuselage with a standard tail wheel is attached to the front of the channel to provide steering capability (i.e. functioning as a nose wheel). This can be seen in Figure 14 below.

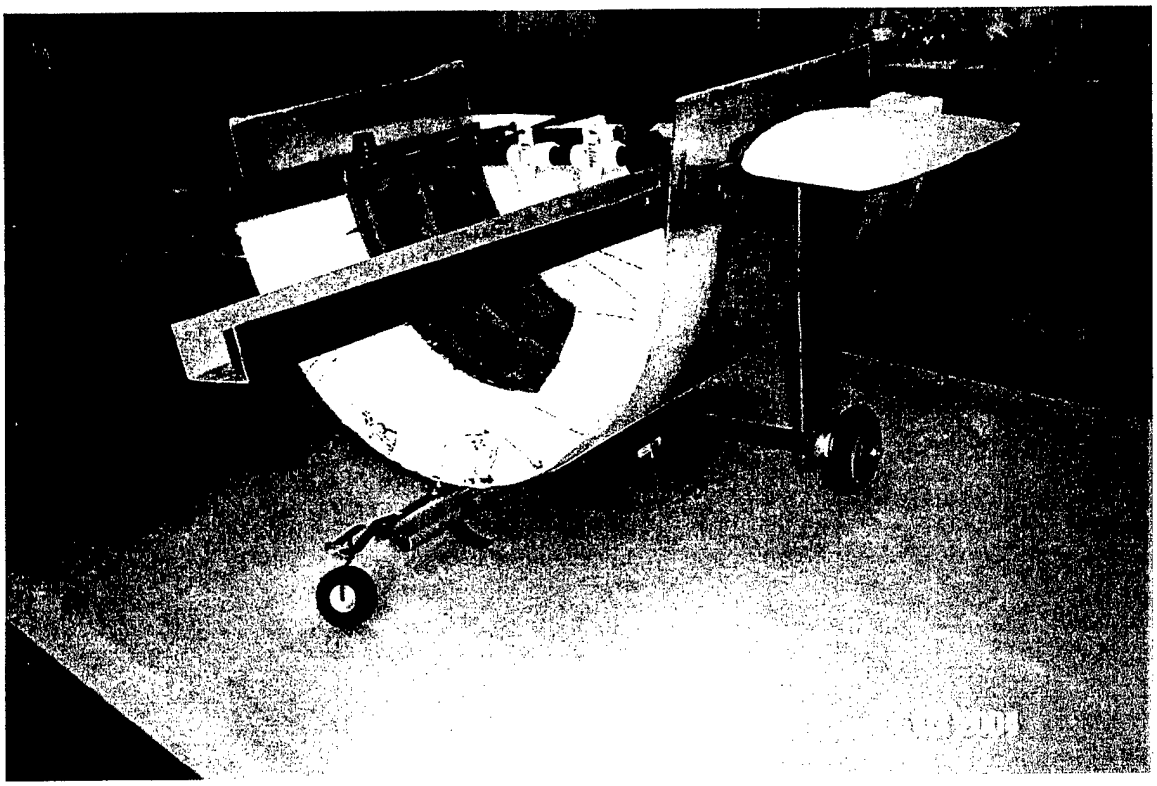


Figure 14 – Actual Fuselage with Gear Mounted seen from Front Right

6 Manufacturing Plan

Once the Design was finalized, a procedure for constructing the aircraft had to be decided upon. The construction process was divided into four main sections, which could be manufactured simultaneously. The fabrication of these components is outlined below.

6.1 Critical Manufacturing Path

Figure 15 below shows a Gant Chart illustrating the Critical path for the manufacturing of the aircraft.

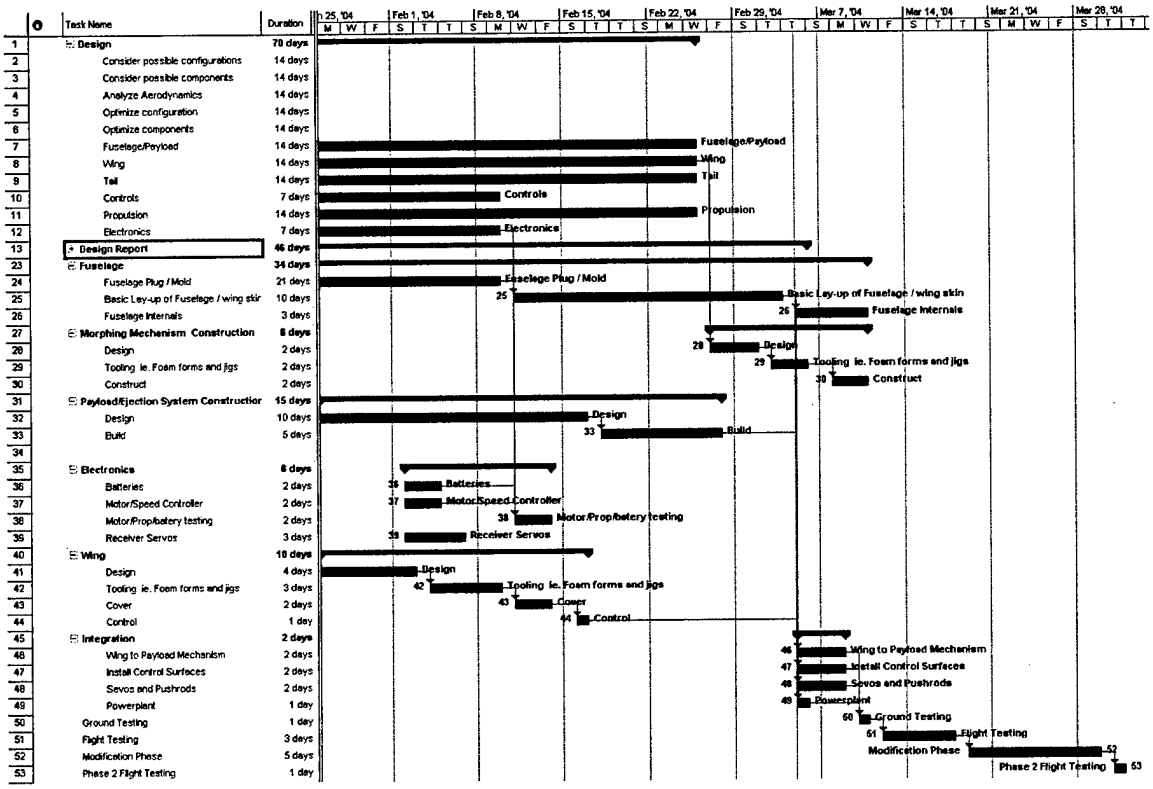


Figure 15 – StingRay Gant Chart

6.2 Wing

The construction of the wing has been divided into three components: the central wing section, and the tapered wingtips. The wing tips will consist of a foam core, cut out with by means of a hotwire. A carbon tube (spar; 1/2" OD, 3/8" ID) will run the length of the wing, with a section sticking out the root end (for attachment to the fuselage); a second tube of the same size was installed aft of the ¼ chord line to provide a sturdy, non-rotational, joint for the removable sections. The leading edge of the wingtips will be sheeted with 1 ply of 3K, 5.7 oz, carbon fabric; then covered with 1ply of 2.4 oz fiberglass fabric; and finally unidirectional carbon tape will be applied to the ends. This assembly will be covered with monocote to provide a smooth surface. Figure 16 depicts the curing of the composite lay-up on the wingtips.

The fuselage was constructed with wing "stubs" on either side; consisting of the same structure as the wingtips. These stubs encase the receiving mechanism for the wingtips. Two tubes (5/8" OD, ½" ID) run across the fuselage; these will act as the receiving ends for the tubes in the wingtips.

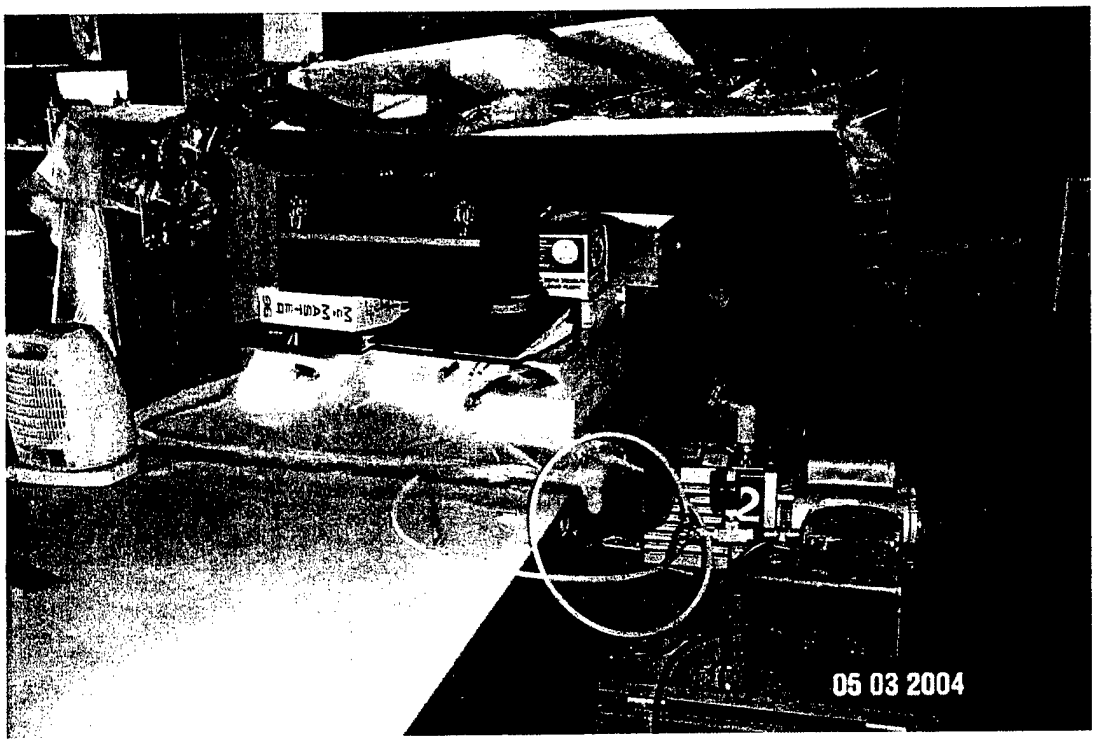


Figure 16 – Curing the Wingtips

6.3 Fuselage

Since two separate fuselage designs were discussed; it was necessary to address the issue of how both could be manufactured.

6.3.1 Conventional Design

It was decided that the fuselage would be constructed in upper and lower halves to allow easy access to the internal components/structure. Since the fuselage profile is a symmetric body of revolution, an internal wooden mold will be turned on a lathe. Once this has been achieved, a band saw will be used to create a recess, where the central wing will be placed. This mold will then be covered with wax, to act as a mold release.

A carbon/honeycomb (3K, 5.7 oz Carbon fabric/Eurocomposites Nomex 3/16" honeycomb) lay-up will be placed on this mold. A seam, made up of several layers of carbon, will run around the aircraft. This will allow the shell to be separated to come off the mold, and allow an access point to the internal components; see Figure 17 for a general cross-sectional diagram of the fuselage

construction. Finally, the central wing section will be attached to the fuselage. This joint will then be filleted by means of silicon glue.

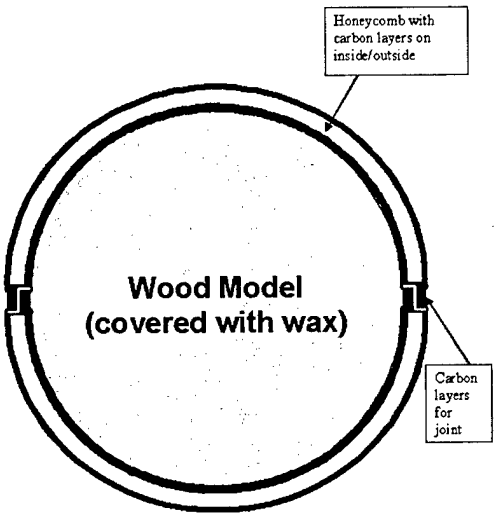


Figure 17 – Cross-sectional view of Fuselage Constructional Lay-up

6.3.2 Ducted Fan Design

The 4 liter tank was fabricated out of commercially available 0.040" thick flat fiberglass panels It was in the form of a semi circular trough of 17" OD by 13" ID or 2 " thick gap for water storage 3 ribs two external and one in the center controlled its shape, on top of each end were riveted 0.015' thick aluminum channels of 2" width and 1" legs. These also help to maintain the shape and support the 0.5 " diameter carbon rods supporting the wings and the tail assembly. The trough is also useful for battery placement and easy center of gravity adjustment. The tank was fitted with a standard drinking bottle cap on each side to serve as vents. The water drained quickly in less than 40 seconds though a 0.5 inch orifice glued inside a 0.5" CPVC elbow.

The tank forms the shroud for the propeller and also serves as center wing section similar to the Gull STOL wing, where the suction of the propeller inflow creates the lift. A special trapped vortex aft shroud is used to induce spreading of the prop wake and filling the wake of the shroud so as to reduces it's drag

6.4 Tail

The boom tail chosen is centered about the two carbon tubes acting as the booms. Since the servos controlling the rudder and elevator are placed inside the fuselage; pushrods were run along the booms to provide control of the rudders and the elevator. Since sturdy stabilizers are required, they were cut out of a lay-up of carbon/honeycomb (3K, 5.7 oz Carbon fabric/Eurocomposites Nomex 3/16" honeycomb) composite; however, since the control surfaces are quite small and require a sharp trailing edge, they will be cut out of balsa wood. To provide a smooth finish, these components will then de covered with monocote; this also serves as a hinge between the control surface and its respective stabilizer.

6.5 Internal Components

Most of the internal components; such as servos, receivers, batteries, etc.; will be purchased. However, the water tank, fuselage, and dump mechanism will have to be custom built.

6.5.1 Water Storage System

The Water storage tank will be integrated into the Fuselage itself. This is achieved by installing a layer of fiberglass/epoxy composite on the inner side of the fuselage; with baffles throughout to prevent sloshing. However, in the case of the ducted fan design, the majority of the fuselage shell itself will serve as the water tank.

6.5.2 Water Dump System

The release mechanism was kept very simple. It consists of a ½ in. ID elbow, placed such that it is located at the lowest point in the tank, and a plug, designed to keep the water from the exit of the elbow. A servo was used to pull the plug out of the elbow; allowing the water to flow freely out of the tank.

7 Rated Aircraft Cost

The rated aircraft costs which is required by the competition rules is Shown in Table 10.

Table 10 – Rated Aircraft Cost Calculation

Coef.	Description	Value	StingRay
A	Manufacturers Empty Weight Multiplier	$300 (NEW)	300
B	Rated Engine Power Multiplier	$1,500	1500
C	Manufacturing Cost Multiplier	$20 / hour	20
MEW	Manufacturers Empty Weight	Actual airframe weight [lb] with all flight and propulsion batteries but without any payload.	10.125
REP	Rated Engine Power	(1+.25*(# engines-1)) * ***Total Battery Weight [lbs]***	2.875
		"Total Battery Weight" will be the weight of the propulsion battery pack(s) as determined by the judges scale during technical inspection. Total propulsion battery pack weight may not exceed 5 lb, but it may be lighter.	2.875
MFHR	Manufacturing Man Hours	Prescribed assembly hours by WBS (Work Breakdown Structure).	
		MFHR = Σ WBS hours	
		WBS 1.0 Wing(s): (NEW)	
		10 hr/ft^2	
		Wing Span * Chord * # wings	9.479166667
		Wing Span is longest distance perpendicular to fuselage axis on any wing. Chord is maximum exposed wing chord on any wing. For a blended wing-body, exposed wing starts 9 inch out from the body centerline.	8.1
		5 hr * control_function_multiplier	2
		ailerons = 1	
		flaperons = 1.5	
		ailerons + flaps = 2	
		ailerons + spoilers = 2	
		ailerons + flaps + spoilers = 3	
		WBS 2.0 Fuselage (NEW)	
		20 hr/ft^3	
		Body Length x Width x Height	2.125
		Length is maximum body length. Width is maximum body width. Height is maximum body height (does not include landing gear height). Maximum body width and maximum body height may occur at different fuselage locations.	
		For a blended wing-body, body width is fixed at 18 inch.	
		Note: Maximum length of the body is defined to be the longest longitudinal length possible to measure on the aircraft, and may include spinner and part of vertical or horizontal surfaces.	
		WBS 3.0 Empennage	
		5 hr/Vertical Surface (Any vertical surface, including winglets, struts, end plates, ventral etc) with no active control	0
		10 hr/Vertical Surface (Any vertical surface) with an active control	20
		10 hr/Horizontal Surface. A horizontal surface is a "wing" if it is more than 25% of the span of the greatest span horizontal surface.)	10
		A "V" tail is considered to be a Vertical surface without control (5 hr) plus a horizontal surface with controls (10 hr), for a total of 15 hr.	0
		WBS 4.0 Flight Systems	
		5 hr/servo or motor controller (2 servos aileron/flap, 1 for payload, 1 for brakes, 1 for rudder/steering, 1 for elevator and a motor controller	35
		MFHR TOTAL:	86.70416667
	Rated Aircraft Cost, $ (Thousands) = (A*MEW + B*REP + C*MFHR)/10		9.084083333

8 Testing Plan

The purpose of the test plan is to minimize the risk of having any problems prior, during, or after each flight.

8.1 Test Glider

A one-half scale glider was created to do simple tests on. For the test glider we just wanted to make sure that the aerodynamics of the real aircraft would work. We considered a flight of 50 feet, from a 5 foot height, to be successful.

8.2 Aircraft

This section addresses the testing procedure to be performed on the actual aircraft, which will be entered into competition.

8.2.1 Ground Test

The ground test will be administered prior to every day's flight. The ground test will make sure that the aircraft is structurally ready to fly. It will also determine if all the controls are working properly. Table 11 below shows the steps taken in the ground test.

Table 11 – Ground Test Procedure

Test	Description	Criteria to move forward
Structural Test		
Payload	Secured and hooked up properly	Pass or Fail
Control Arms	Connections are good between servo and control surface	Pass or Fail
Bolts/screws	Make sure screws/nuts/bolts and fasteners are tight	All missing fasteners must be replaced
Wing	Wing is attached and secure to aircraft	Pass or Fail
Tail	Tail is attached and secure to aircraft	Pass or Fail
Engine	Engine mount secured to engine/aircraft	Pass or Fail
Gears	Gear attached and secured to plane	Pass or Fail
Control Surfaces		
Ailerons	Functioning properly (left and right turning)	Pass or Fail
Elevator	Functioning properly (up and down)	Pass or Fail
Rudder	Functioning properly (left and right)	Pass or Fail
Flaps	Functioning properly (extended and up)	Pass or Fail
Throttle	Functioning properly (idle and full)	Pass or Fail
Payload	Functioning properly (closed and release)	Pass or Fail
Steering	Functioning properly (left and right)	Pass or Fail
Brakes	Functioning properly (applying and off)	Pass or Fail

8.2.2 Flight Testing

The flight test plan will be administered during the first few flights. It will give us an idea of how the aircraft will perform with respect to takeoff, landing, cruise flight, maneuvering, and payload mechanism operation. A copy of the flight test plan can be seen in Table 12.

Table 12 – Flight Test Procedure

Test Flight		Criteria To move Forward
Empty	Empty payload (takeoff, laps, landing)	Successful flight, Pilot must be Satisfied with his control over the plane
	*trim airplane as necessary for flight	
1/4 Full	1/4 Full payload (takeoff, laps, landing)	See Above
	*trim airplane as necessary for flight	
1/2 Full	1/2 Full payload (takeoff, laps, landing)	See Above
	*trim airplane as necessary for flight	
Full	Full payload (takeoff, laps, landing)	See Above
	*trim airplane as necessary for flight	
Course Circuit	Full payload, fly circuit and test dumping water	See Above
	*trim airplane as necessary for flight	
	Note: If any major changes made in test flight, go back through to the beginning of the checklist.	

9 Test Results

This section addresses the results acquired from the various testing phases conducted on the performance of the aircraft design.

9.1 Test Glider

The test glider was constructed to evaluate the aircraft's stability and aerodynamic performance during flight. It flew quite stably over distances exceeding 50 ft. This test model illustrated that it was quite feasible to construct an R/C aircraft based on this design.

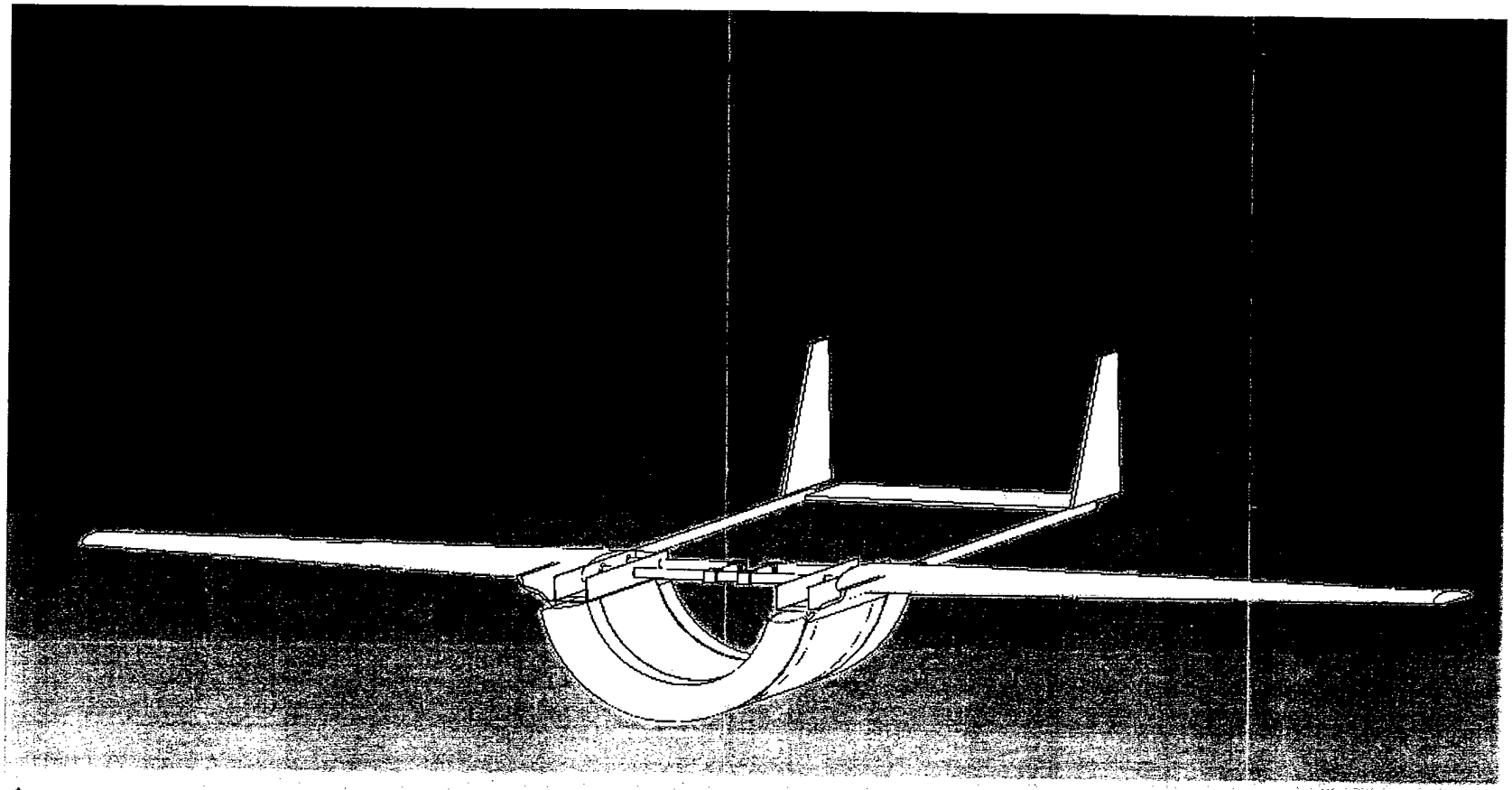
10 Conclusion

This attempt in creating an aircraft best suited to achieve the requirements implemented by the AIAA was the result of a lot of teamwork and optimizations. Several options for the configurations for each component were considered, and the ones best suited to the desired performance were chosen. Each chosen component design was then evaluated in detail to determine the best, most feasible, method for construction; and how these sections would be assembled to

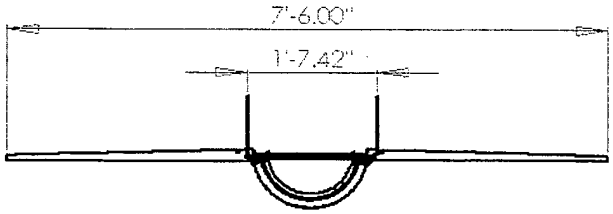
produce the completed aircraft. Thus the entire aircraft was designed around the rules and requirements to produce the best all-around flight vehicle for the competition.

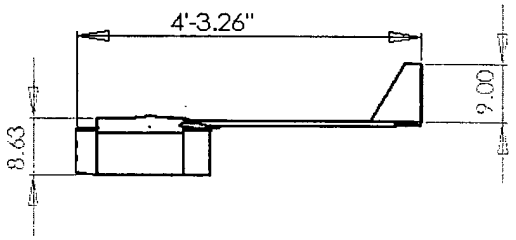
11 References

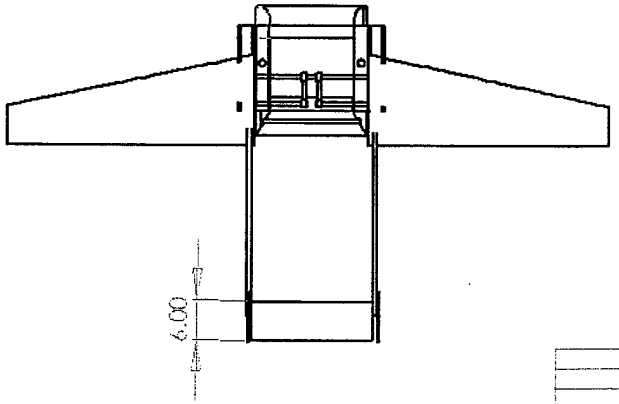
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2) Boucher B., http:www.astroflight.com, 2004
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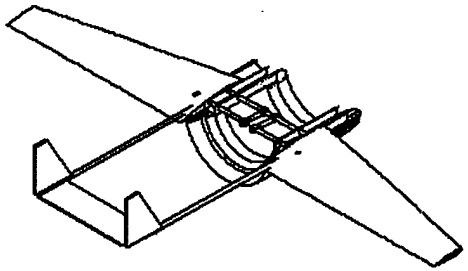


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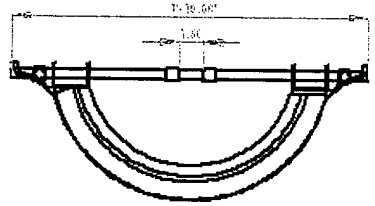
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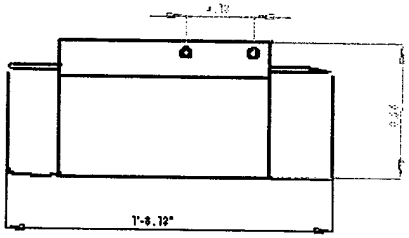
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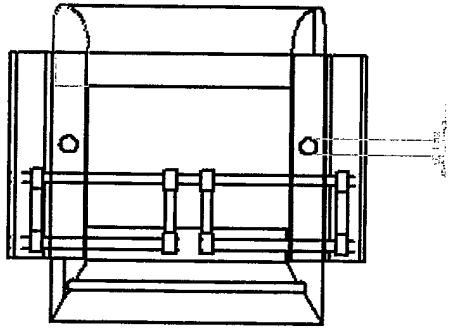
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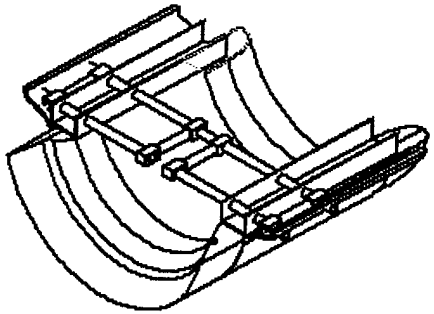
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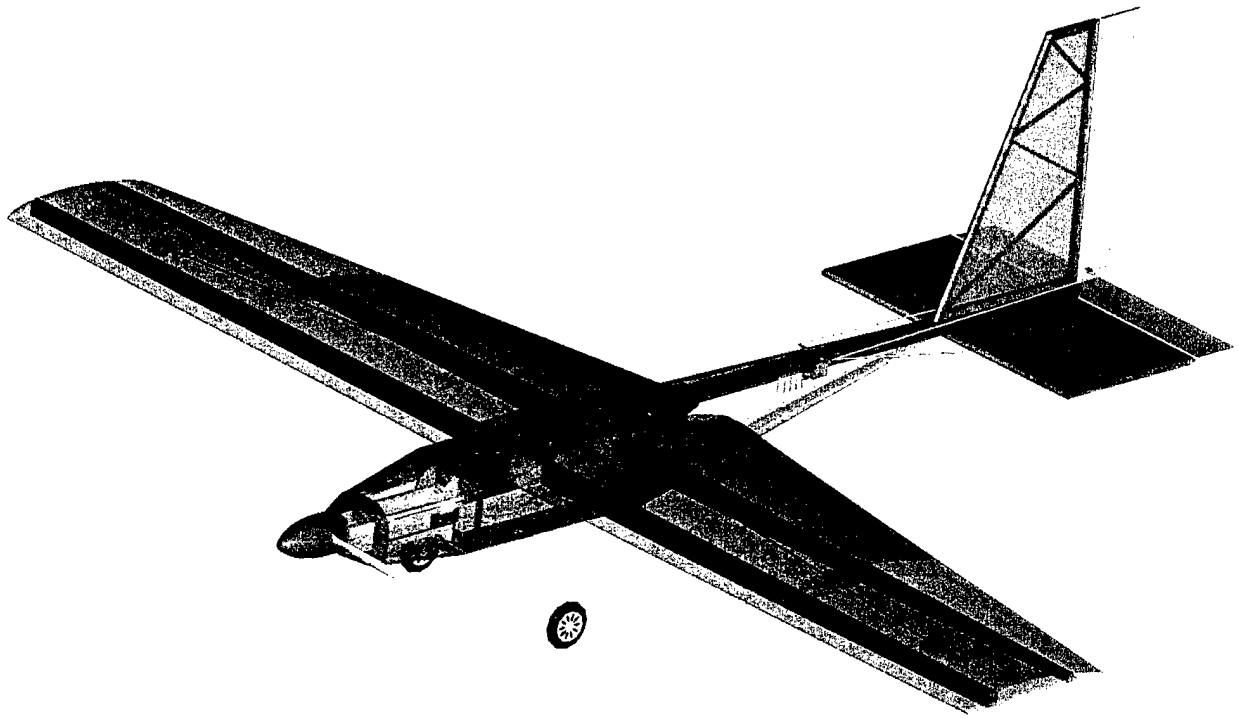
2003 / 2004 Design / Build / Fly Team

TABLE OF CONTENTS

1.0 EXECUTIVE SUMMARY 1
1.1 Conceptual Design 1
1.2 Preliminary Design 2
1.3 Detail Design 2
2.0 MANAGEMENT SUMMARY 3
Aerodynamics 3
Propulsion 3
Structures 3
3.0 CONCEPTUAL DESIGN 5
3.1 Mission Requirements 5
3.2 Figures of Merit 6
3.2.1 Required Time 6
3.2.2 Required Skill 6
3.2.3 Reliability 6
3.2.4 Stability and Control 7
3.2.5 Effect on Rated Aircraft Cost 7
3.2.6 Effect on Average Flight Speed 7
3.3 Assumptions Made 7
3.4 General Configurations 8
3.4.1 Monoplane 9
3.4.2 Biplane 9
3.4.3 Tandem 10
3.4.4 Flying Wing 10
3.5 Design Parameters 11
3.5.1 Wing Planform 11
3.5.2 Landing Gear Configuration 12
3.5.3 Empennage Configuration 12
3.5.4 Engine Configuration 12
3.5.5 Payload/Aircraft Size 13
3.6 Features that Produced Final Configuration 14
4.0 PRELIMINARY DESIGN 15
4.1 Figures of Merit 16
4.1.1 Required Time 16
4.1.2 Required Skill 16
4.1.3 Speed 17
4.1.4 Component Weight 17
4.2 Design Parameters Investigated 17
4.2.1 Fuselage Shape and Assembly 17
4.2.2 Fuselage Materials 18
4.2.3 Spar Cross Section and Material 18
Al – 2014-T6 20
Balsa Wood 20
Fiberglass/Balsa 20
Carbon-Fiber 20
Lite-Ply/Balsa 20

4.2.4 Wing Structure 21
4.2.5 Landing Gear Cross-Sections 22
4.2.6 Tail Assembly 22
4.3 Tail Sizing 23
4.4 Aircraft Performance Estimates 24
4.4.1 Weight Breakdown 24
4.4.2 Mission Model and Estimated Performance 25
4.4.2 Stability Characteristics 29
5.0 DETAIL DESIGN 32
5.1 Component Selection 32
5.1.1 Propulsion Configuration 32
5.1.2 Airfoil selection 34
5.1.3 Wing Placement 36
5.1.4 Control Surface Sizing 37
5.2 Systems Architecture 37
5.2.1 Connection of Wing / Fuselage / Tail Assembly 37
5.2.2 Wiring / Servos / Receiver Placement 38
5.2.3 Water Dispensing Mechanism 38
5.3 Aircraft Configuration 38
5.3.1 Final Configuration 38
5.3.2 Weight and Balance Sheet 45
5.3.3 RAC Calculation 46
5.4 Ongoing Aircraft Optimization 47
5.4.1 Second Generation Aircraft 47
5.4.2 Motor Optimization 48
5.4.3 Number of Batteries 49
6.0 MANUFACTURING PLAN 50
6.1 Figure of Merits 50
6.1.1 Performance 50
6.1.2 Weight Estimation 50
6.1.3 Construction Skill Required 50
6.1.4 Construction Time Required 50
6.2 Manufacturing Processes Investigated 51
6.2.1 Fuselage Construction 51
6.2.2 Water Tank Valve 52
6.2.3 Wing Materials 52
6.2.4 Wing Sweep / Spar 52
6.2.5 Spar Material 53
6.2.6 Wing Attachment 53
6.2.7 Tail Design 53
6.2.8 Empennage Design 53
6.3 Final Manufacturing Process 54
6.3.1 Front Fuselage 54
6.3.2 Mid-Fuselage / Water Tank 54
6.3.3 Rear Fuselage 55
6.3.4 Empennage 55

6.3.5 Wings 55
6.3.6 Landing Gear 56
7.0 TESTING PLAN 57
7.1 Static Tests 57
7.2 Flight Test Check List 57
7.3 Flight Tests 58
7.3.1 February 1, 2004 58
8.0 REFERENCES 59



1.0 EXECUTIVE SUMMARY

Students at The University of Texas at Austin's Department of Aerospace Engineering and Engineering Mechanics designed and built an electric RC airplane in order to compete in this year's Cessna/ONR Student Design Build Fly Competition. This year's mission profile involves flying two different mission sorties: a "fire bomber" mission in which the aircraft is twice loaded with water which is then dumped during flight, and a "ferry" mission in which the aircraft simply has to take off and complete 4 laps. The aircraft must takeoff within 150 feet and withstand being lifted by the wingtips fully loaded. The aircraft was also required to fit inside a four foot by two foot by one foot box; however, this year the aircraft assembly was not timed. The aircraft was designed in three main phases: conceptual design, preliminary design and detail design.

The team began the design process with a conceptual design phase in which the team eliminated general configurations from consideration. Then in the preliminary design, the aircraft components were sized and optimized, stability analysis was performed, and performance analysis was done. In the detail design process the aircraft's specific components were selected, the control surfaces were sized, and as the plane components were built and tested, improvements were made to ensure the performance goals of the aircraft were met and/or exceeded. Note that the team decided that it was better to design a non-optimal aircraft which we could build and fly at the competition than to design, but not have time to build or fly, the optimal aircraft on paper. In conjunction with the detailed design phase, a manufacturing plan was created to construct the aircraft and flight tests were performed to verify and further improve performance. In each phase of the design specific figures of merit were used to more quantitatively judge which specific feature was best.

1.1 Conceptual Design

The figures of merit selected for the conceptual design phase were the required skill, required time, reliability/construction time, stability and control, rated aircraft cost effects, and speed. Next, design concepts such as, monoplane, biplane, tandem wing, and flying wing were scored based on the figures of merit. For the best concept, the design parameters investigated were the motor configuration, wing planform, landing gear configuration, empennage configuration and finally whether we should build a large (carry nearly all 4L of water) or small (carry ~1L) aircraft. The design selected was a monoplane with tapered wing planform, a tail dragger landing gear, and a conventional empennage; however, we were unable to decide definitively between the small and large aircraft concepts. Since the team had been told that money was not as much of a factor this year (unlike in previous years) and the team was large compared to previous years, it was decided that both aircraft would be designed and a fly-off between the two planes would occur before the competition to decide which plane would compete. Two separate teams were not entered into the contest itself because many of the (electrical) components were still shared between the two teams which saved money, the department did not want to send everyone

from both teams to the competition (as in previous years, we are allowed to send only 10 members), and finally once the fly-off was complete we were able to focus everyone on making the winning aircraft better. The paper focuses solely on the design and performance estimation of the large aircraft so that the paper could be written as the design progressed. There was very little, if any, difference between the design process of the small aircraft and the large aircraft.

1.2 Preliminary Design

The preliminary design figures of merit selected were the component weight, required time, required skill, and expected effect on aircraft speed. The design parameters investigated were spar cross-section and material, wing/empennage structure, fuselage shape, fuselage materials, landing gear cross-sections, and the empennage structure. Once the aircraft's weight was more accurately determined, the static thrust and propeller pitch speed needed for takeoff were determined. Finally, the stability of the aircraft was checked and an estimate of the aircraft's performance was done to obtain an expected score.

1.3 Detail Design

The detail design phase began with selection of the propulsion configuration from the propulsion requirements determined in the preliminary phase. Then the airfoil was selected, the wing was located on the fuselage, and the control surfaces were sized. The details of the fuselage/wing connection, the water release mechanism, and the servo wiring placement were also determined. Also, the overall aircraft performance was calculated again with the most current aircraft parameters and weight. The first generation large aircraft (4.0 L) outscored the first generation small aircraft in actual flight tests; however, both groups are making second generation aircraft. Thus, the overall aircraft continues to be optimized as test flights continue up until the week before the competition.

2.0 MANAGEMENT SUMMARY

In the summer of 2003, the 2004 UT DBF team began to prepare for this year's competition by deciding on the individual group leaders (from the previous year's team) and beginning the conceptual design of the aircraft. By the beginning of the Fall Semester, the team had recruited the new freshmen for this year's competition. At the start, the team consisted of 28 people; however, the team quickly lost members due to schoolwork, and in the end consisted of 1 faculty advisor, 1 graduate student, 3 seniors, and 8 underclassmen who were all aerospace engineering majors. The faculty advisor's role was to provide guidance when requested and act as an intermediary between the department and the team on matters of funding.

To keep the team motivated a Project Manager was chosen to oversee the project and the team was divided into three distinct groups, each with members from the other two groups to ensure that each group knew what the others were doing. The groups were Aerodynamics, Propulsion, and Structures. Table 2.1 shows the organizational structure of the team.

Table 2.1: Organizational Structure of the Team.

Project Manager: Clint Kam		
Aerodynamics ***Leader***: Chris Moore	**Propulsion** ***Leader:*** Clint Kam	**Structures** ***Leader:*** Bill Tandy
Stephanie Sellers Clint Kam	Rodrigo Serrano Chris Hein Chris Moore	Kyle Riggen Ubong Etukudo Conor Bingaman Vikram Garg Cinnamon Wright Kelly Walsh Stephanie Sellers

The team leaders and project manager developed a production schedule for the upcoming semesters during the summer with major and minor tasks. Table 2.2 is the original timeline with both the planned start and finish dates and the actual finish dates for the tasks. Whenever a deadline could not be met, the timeline was slightly altered and the team attempted to compensate by finishing other tasks ahead of schedule.

Under the direction of the team leaders, each group was given assigned tasks to complete over a certain amount of time. The workload was split among the group members according to each member's abilities

as evenly as possible. Each building task was assigned to at least two people so that safety requirements were met and to ensure that the task would be completed.

Table 2.2: Project Timeline.

TASK	START DATE	FINISH DATE	ACTUAL DATE COMPLETE
Start-Up Tasks	**7/15/2003**	**8/30/2003**	**9/6/2003**
Assemble Design Team	8/28/2003	8/30/2003	9/6/2003
Get Budget Approval	7/9/2003	7/13/2003	7/13/2003
Letter of Intent	7/6/2003	7/6/2003	7/9/2003
Conceptual Design	**8/28/2003**	**10/14/2003**	**10/14/2003**
Read and Discuss Contest	8/28/2003	9/6/2003	9/6/2003
Configuration Analysis	9/6/2003	9/30/2003	9/30/2003
Design Parameter Analysis	9/30/2003	10/14/2003	10/14/2003
Conceptual Design Written	9/30/2003	10/14/2003	10/14/2003
Preliminary Design	**10/14/2003**	**12/27/2003**	**12/31/2003**
Design Parameter Analysis	10/14/2003	11/8/2003	11/8/2003
Critical Dimension Sizing	11/8/2003	12/2/2003	12/2/2003
Stability Analysis	12/2/2003	12/20/2003	1/1/2004
Takeoff Velocity Code Written	12/10/2003	1/1/2004	1/10/2004
Preliminary Design Written	12/20/2003	1/14/2004	1/30/2004
Detail Design	**12/31/2003**	**2/1/2004**	**2/4/2004**
Propulsion Systems Selected	1/3/2004	1/10/2004	1/22/2004
Aerodynamic Analysis	1/3/2004	1/18/2004	1/21/2004
First Flight Analysis	1/21/2004	1/24/2004	2/2/2004
Design Iteration/Improvement	1/24/2004	4/19/2004	-
Detail Design Written	1/20/2004	2/1/2004	2/26/2004
Manufacturing Plan	**1/13/2004**	**1/21/2004**	**1/21/2004**
Final Airplane Construction	1/18/2004	1/21/2004	1/22/2004
Manufacturing Plan Written	2/14/2004	2/18/2004	2/20/2004
Initial Test Flight	**1/21/2004**	**1/21/2004**	**2/1/2004**
Flight Tests	**2/1/2004**	**4/19/2004**	-
Performance Modifications	**2/2/2004**	**4/19/2004**	-
Compile Paper	**3/1/2004**	**3/8/2004**	**3/8/2004**
Contest	**4/23/2004**	**4/25/2004**	**4/25/2004**

3.0 CONCEPTUAL DESIGN

The conceptual design process involved determining the general aircraft configuration and several design parameters. Each general design was scored by figures of merit and the winning design proceeded to preliminary design. In order to determine the overall shape of the aircraft for preliminary design, several design parameters were investigated.

3.1 Mission Requirements

This year's competition had two required missions: a "fire bomber" mission and a "ferry" mission; also, the aircraft had takeoff within 150 feet and fit within a 4 foot by 2 foot by 1 foot box. In addition, the aircraft had to withstand a "wingtip test" in which the aircraft was lifted fully loaded, both right-side up and inverted, by the wingtips. Each flight mission required the aircraft to fly laps around the course shown in Figure 3.1; each number (such as number 4) between the tick marks denotes a specific section (the required 360° turn) of the course used to model the aircraft performance. The flight score is calculated by dividing a difficulty factor by the mission flight time. The fire bomber mission consisted of loading the plane with up to 4 liters of water, taking off, and then dumping the water in-flight in sections 3 through 5 in Figure 3.1. The aircraft then landed and flew another lap. The difficulty factor for the fire bomber mission was equal to 2 times the number of pounds of water carried. The ferry mission consisted of flying the 4 laps without any payload and had a difficulty factor of 1.0.

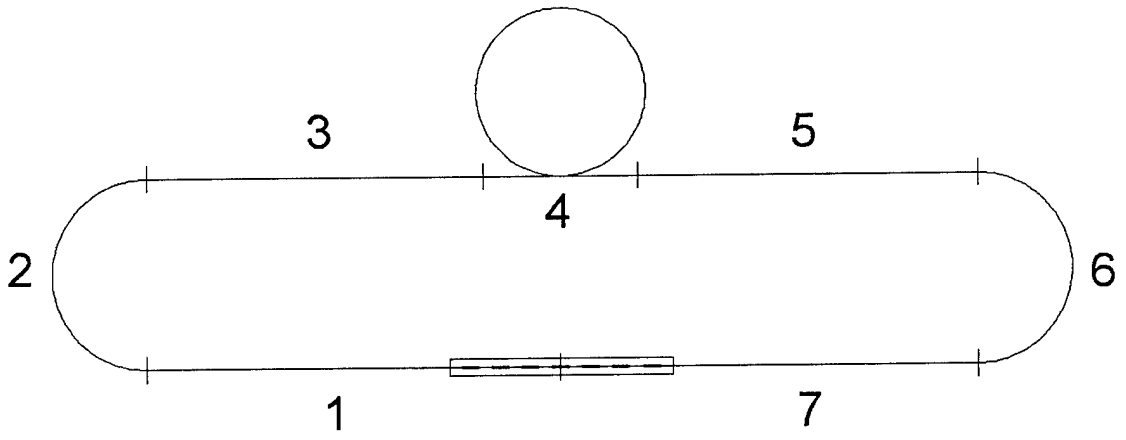


Figure 3.1: Mission Course Layout.

3.2 Figures of Merit

The figures of merit were selected this year to minimize the design risks and maximize the overall score of the aircraft. In other words, the team decided that it was more important to have a plane that would be able to compete at the competition instead of a plane that would win on paper, but was never successfully flown, either because of design difficulty, excessive construction time or skill required. In order to maximize the score while accounting for potential design setbacks or shortcomings, the figures of merit used were time and skill required to build, the design reliability, the inherent stability of the design, the design/parameter's expected Rated Aircraft Cost (RAC) shown in Table 3.1, and the overall effect of the design/parameter on the average flight speed of the aircraft. The complete conceptual design figure of merit table is shown in Table 3.2. The figure of merits and their relative weights are discussed below.

3.2.1 Required Time

Most of the team members have busy schedules with just their courses, let alone with DBF as an extracurricular activity. Therefore, since the contest dictates a tight schedule, an important factor in the feasibility of each component's configuration was the time required to design and build the configuration. It would be unwise to choose a configuration that could potentially require more time to test and develop than was available before the contest date. The basic assumption was that complicated/new configuration designs would increase the time required to build the component because many members were new and inexperienced. This figure of merit was given a weight of 1.0.

3.2.2 Required Skill

An important factor in the feasibility of each component's configuration was the ability of the team to build the configuration correctly. Complicated configuration designs would increase the skill, and material cost (due to mistakes) required to build the component. Also, configurations which required more skill to build would inherently be more probable to contain slight errors which might reduce the flight reliability of the aircraft. This figure of merit was given a weight of 2.0.

3.2.3 Reliability

Reliability of the configuration was a concern because in previous years, our aircraft have crashed before or during the competition. The reliability of a configuration or a parameter is a measure of its complexity and difficulty/strain on the pilot. This figure of merit was given a weight of 2.0.

3.2.4 Stability and Control

The inherent stability of a design concept/parameter was important because certain designs are more easily made stable than others, lending to a larger margin of error in the design, and thus safer flight. Because of the history of high gusts in Kansas, a high degree of control was desirable for the aircraft. This figure of merit was given a weight of 3.0.

3.2.5 Effect on Rated Aircraft Cost

Each design parameter was evaluated by the team to have a certain effect on the overall Rated Aircraft Cost of the aircraft for a given payload. The lower the figure of merit score for a given parameter, the more that parameter would raise the Rated Aircraft Cost of a given design; that parameter being the only difference between aircraft concepts. This figure of merit was given a weight of 4.0 since it directly influences the final score for the aircraft.

3.2.6 Effect on Average Flight Speed

The flight score that any given design could achieve was dependent on the design's total flight time. For a given thrust in level flight, the only ways to increase the cruise velocity (and thus reduce the total flight time of the mission) would be to reduce the aircraft's drag or decrease the aircraft's overall weight. Also, this factor was important because in the past the team has had problems with the flight time exceeding the batteries' lifetime, so shortening the flight time or reducing the thrust required was a priority. This figure of merit was given a weight of 4.0.

3.3 Assumptions Made

Several assumptions were also made in the calculation of the Rated Aircraft Cost for each configuration. Ideally, a given aircraft design would maximize the amount of payload relative to the total aircraft weight. Thus it was assumed that the payload weight fractions from previous competitions would be similar to this year's aircraft, and when applicable, equal to the fraction achieved by our previous teams. For the configuration analysis, it is also assumed that 4.0 Liters of water will be carried.

Also, based on an RC rule of thumb, it was further assumed that 50 W of battery power would be required per pound of total aircraft weight. It was assumed that the brand of battery used last year, which performed very well, would be used. They provide 1 V per cell at high amp loads and weigh 2.25 oz/cell. The motor configuration would be assumed to run at 35 A so that the current would not be too close to 40 A, and yet still be large enough to allow for a low total battery weight. This allowed the team to calculate the needed battery weight for the aircraft based on the loaded weight from the payload fraction calculation. Finally, it was assumed that each configuration would have one propeller and one motor, since the contest strongly penalizes more than one motor in terms of the rated aircraft cost.

The maximum wing loading was estimated using steady flight assumptions (L = W, T = D), resulting in the equation:

$$\frac{W}{S} = \frac{1}{2}\rho\left(\frac{V_S}{1.2}\right)^2 C_{L,max} \qquad (3.1)$$

where W/S is the wing loading, ρ is the density at sea level, V_s is the stall velocity of the aircraft, and $C_{L,max}$ is the maximum lift coefficient. The factor of 1.2 is a safety factor included to prevent stall on landing. The pilot requested that the stall speed be no greater than 30 mph so that his approach speed would not be too great; therefore a stall speed of 30 mph was assumed for each configuration. Also for the analysis we assumed that no flaps would be used, since they would increase the RAC. The last two years aircraft had airfoils with a $C_{L,max}$ of 1.3. From equation 3.1, with these assumptions, a wing loading of 2.1 lbf/ft^2 is needed for each configuration. With the aircraft weight from the payload weight fraction and the wing loading, it was then possible to find the necessary wing area for a given configuration. It was further assumed, based on previous year's experience, that an aspect ratio of ~8 would be a good balance between increasing structural weight and aerodynamic efficiency for single wing aircraft. However, it should be noted that this aspect ratio gave the single wing designs a wingspan greater than 8 feet, but this was not a problem because the fuselage could have the inner wing section permanently attached such that the two outer wing sections were still only 4 feet each. For aircraft with two lifting surfaces, an aspect ratio of 9 would be viable because the lower wing loading each wing experiences would relieve some of the structural difficulties associated with a larger aspect ratio.

Since the competition does not require timed assembly, the fuselage could be easily made into two sections which would attach during assembly. This allowed the fuselage to be made longer than 4 feet in order to reduce the parasite drag on the fuselage and provide a sufficient moment arm for the tail to act on. The fuselage was assumed to be 5.25 feet long with a fineness ratio of 8. This gave each configuration a fuselage with a maximum width and height of 0.66 feet.

3.4 General Configurations

The first step in designing an airplane is determining the general shape of the airplane. Four different airplanes configurations were chosen for analysis. These configurations are described below. The RAC was estimated for each configuration and the breakdown is shown in Table 3.1.

3.4.1 Monoplane

The monoplane configuration has one wing and a horizontal stabilizer. Its versatility in design as well as relative ease in manufacturability makes it very attractive. It has been successfully designed and flown by current members in previous years, meaning the team had prior experience in designing and building this type of aircraft. The reliability of this configuration was decided to be slightly less than a biplane because of the monoplane's generally higher wing loading, which requires faster landing velocities. It also had a longer construction time than the flying wing because the fuselage would have to be in two sections to fit in the box. The monoplane was considered by the team to have the most inherent stability and control; this style of aircraft is well documented in both design and performance which gave the DBF team a large database from which to draw. The monoplane was considered a relatively fast configuration owing to its lower parasite drag and weight relative to the biplane or the tandem configurations.

Based on aircraft data from previous contests, the historical payload weight fraction for monoplanes was estimated to be 0.37. This led to an empty weight of 15.0 pounds and thus a wing area of 11.3 ft^2. For a wing with an aspect ratio of 8, the wingspan would be 9.4 ft and the chord would be 1.2 ft. The battery weight for a 23.8 pound loaded monoplane would be 4.8 pounds. For a monoplane, it was assumed that there would be 1 rudder, 1 elevator, 2 ailerons, 5 servos, and 1 motor controller.

3.4.2 Biplane

The biplane configuration is similar to that of the monoplane, except with the addition of a second wing above the main wing. In previous Design, Build, Fly competitions this design has done very well owing in part to both its high reliability and its good controllability; however, there are certain design issues such as the relative location of the top wing to the bottom wing that must be accounted for appropriately. The skill and time required to build the aircraft would be the greatest of the four configurations because of the need to build two wings and two wing pylons. Because of the large amount of surface area, the parasite drag of the biplane was considered large and therefore the configuration was considered slow.

Based on previous year's aircraft data, the historical payload weight fraction for biplanes was estimated to be ~0.36. This leads to an empty weight of 16.3 pounds. Each wing for the biplane had a wing area of 6.0 ft^2. For a wing with an aspect ratio of 9, the wingspan would be 7.5 ft and the chord would be 0.8 ft. The battery weight for a 25.1 pound fully loaded biplane would be 5.0 pounds. For a biplane, it was assumed that there would be 1 rudder, 1 elevator, 2 ailerons, 5 servos, 1 motor controller, and two wing pylons (vertical surfaces) without control surfaces on them.

3.4.3 Tandem

The tandem wing airplane consists of two similar sized wings, one in front and one in rear of the airplane. The tandem wing's horizontal stabilizer creates lift and controls the aircraft's pitch. The forward wing's downwash can make control difficult for the pilot and reduce the effective lift coefficient of the rear wing if not properly designed. Previous DBF teams have had direct experience with the design and implementation of the tandem wing configuration without much success, therefore leading this year's team to give the tandem design a low reliability score. This configuration, like the biplane, required the construction of two wings and therefore needed more time and skill in building it. Similar to the biplane, the tandem configuration has a large surface area, and therefore a large parasite drag resulting in slower flight speeds for a set amount of thrust.

Based on a previous year's design, the assumed payload weight fraction for the tandem was 0.36. This leads to an empty weight of 16.3 pounds. Each wing on the tandem therefore had a wing area of 6 ft^2. For a wing with an aspect ratio of 9, the wingspan would be 7.5 ft and the chord approximately 0.8 ft. The battery weight for the tandem would need to be 5 pounds. For the tandem, it was assumed that there would be 1 rudder, 1 elevator, 2 ailerons, 5 servos, and 1 motor controller.

3.4.4 Flying Wing

The flying wing is one of the most efficient designs for an airplane; however, stability is a strong limitation of the design. The flying wing can have substantially less drag because the lower aircraft weight translates into less induced drag and the elimination of the horizontal tail slightly reduces the parasite drag. Therefore the plane is able to fly at greater speeds given the same amount of thrust as other configurations. Difficulties arise, however, in the manufacturing and design of the plane since the airfoil requires special consideration, such as wash-out. Also, the flying wing is less suited for payloads that have large volumes compared to the other types of aircraft. However, previous UT DBF teams have had some success with the flying wing configuration.

The flying wing aircraft built several years ago at UT had a payload weight fraction of 0.39, corresponding to an empty weight of 13.8 pounds for this year's competition. The flying wing therefore needed a wing area of 10.8 ft^2, a wingspan of 9.0 ft and a chord of 1.2 ft. The battery weight for a 15.1 pound aircraft would be 4.5 pounds. For the flying wing, it was assumed that there would be 2 rudders, 0 elevators, 2 ailerons, 5 servos, and 1 motor controller and that the fuselage would only be 3 feet long, slightly less than 18 inches wide (the limit imposed by the contest rules), and 5 inches tall.

Table 3.1: Configuration Rated Aircraft Cost Comparision.

Description	Monoplane (Large)	Biplane	Tandem	Flying Wing	Monoplane (Small)
MEW	**15.0**	**16.3**	**16.3**	**13.8**	**5.9**
REP	**4.8**	**5.0**	**5.0**	**4.5**	**0.8**
# of Engines	1.0	1.0	1.0	1.0	1.0
Total Battery Weight (lb)	4.8	5.0	5.0	4.5	0.8
MFHR	**213.5**	**241.5**	**231.5**	**196.6**	**81.5**
Wings	**117.8**	**130.0**	**130.0**	**113.0**	**32.7**
Wing Span (ft)	9.4	7.5	7.5	9.0	4.7
Max Chord (ft)	1.2	0.8	0.8	1.2	0.6
Control Function Multiplier	1.0	1.0	1.0	1.0	1.0
# of Wings	1.0	2.0	2.0	1.0	1.0
Fuselage	**45.7**	**51.5**	**51.5**	**33.6**	**3.8**
Fuselage Length (ft)	5.3	5.3	5.3	3.0	2.5
Fuselage Width (ft)	0.7	0.7	0.7	1.4	0.2
Fuselage Height (ft)	0.7	0.7	0.7	0.4	0.4
Empenage	**20.0**	**30.0**	**20.0**	**20.0**	**20.0**
Vertical Surfaces w/no control	0.0	2.0	0.0	0.0	0.0
Vertical Surfaces w/ control	1.0	1.0	1.0	2.0	1.0
Horizontal Surfaces	1.0	1.0	1.0	0.0	1.0
Flight Systems	**30.0**	**30.0**	**30.0**	**30.0**	**25.0**
# of Servos	5.0	5.0	5.0	5.0	4.0
# of Motor Controllers	1.0	1.0	1.0	1.0	1.0
TOTAL RATED AIRCRAFT COST	**16.0**	**17.2**	**17.0**	**14.8**	**4.5**

3.5 Design Parameters

For the initial stage of the conceptual design, several design parameters were investigated for their performance advantages and disadvantages for the chosen monoplane configuration. The design parameters that were chosen for further analysis were then considered in the preliminary design phase.

3.5.1 Wing Planform

The team considered three distinct planform shapes for the wing(s). The wing planforms considered were rectangular, tapered, and elliptical. The rectangular platform offers easy construction that the team was comfortable with from previous year's experience. However, the rectangular planform is the least efficient of the three planforms due to large wingtip-losses. The tapered wing planform has the advantage of more efficient production of lift while remaining relatively easy to construct; however, tapered wings can suffer tip stall due to the low Reynolds number and the thin airfoil at the wing tips. Furthermore, tapered wings, while producing more lift per unit area than the rectangular wing, do not have a significant Rated Aircraft Cost advantage over the rectangular wings because the manufacturing man-hours cost is calculated by adding eight dollars per foot of wing span and eight dollars per foot of

maximum exposed chord, instead of charging per square-foot of wing area as in some previous years. A taper ratio of 0.5 would give a good balance between the increase in Oswald's efficiency and the increase in the RAC. This taper ratio also reduced the chance of tip-stall and would only require slight wash-out to prevent this from occurring. The elliptical planform is the most efficient wing shape in that it minimizes the induced drag of the wing. Unfortunately the elliptical wing is extremely difficult to construct and has a high likelihood of tip-stall. For these reasons, a tapered planform was chosen for this year's aircraft.

3.5.2 Landing Gear Configuration

The team examined three landing gear configurations: tail dragger, tricycle, and retractable tricycle. The tail dragger configuration is inherently unstable, having the tendency to want to flip around as it moves, such that the plane faces backwards. Neither tricycle configuration contains this instability; they were therefore both given scores of 1. However, a tail dragger would be a short, simple, lighter option (meaning less RAC) with less drag than a tricycle. The tricycle configuration had been used by the design team in previous years, but would have relatively large amounts of drag even with fairings and would weigh more than the tail dragger. The retractable landing gear would have little to no drag during flight; however, it would be the heaviest option, the most complicated and difficult to construct, and it would raise the RAC because of the extra servo. It would also require adequate space in the wing root to fold up into.

3.5.3 Empennage Configuration

The empennage configurations considered consisted of conventional, T-tail, canard, and V-tail. The conventional and T-tail are somewhat versatile, easily constructed, and well documented designs that have good stability and control characteristics. The T-Tail, with the extra structure required to place the horizontal tail on top of the vertical tail was more difficult and required more time to build; however, the horizontal tail would act as a winglet for the vertical tail, adding yaw stability. The V-tail, on the other hand, is much more difficult to build and not quite as controllable, but a little less expensive in rated aircraft cost as the conventional configuration. The canard presents aerodynamic and control problems, due to the downwash on the wings, and has met with little success from previous UT DBF teams. Also, it required a little more skill than to build, and still has the RAC disadvantage over the V-Tail.

3.5.4 Engine Configuration

The engine configuration was a decision between a pusher, puller, or multiengine pusher/puller. The pusher configuration is better to resist aircraft crashes. The puller is more conventional and is sometimes required by the geometry of the aircraft. The multiengine configuration can allow propeller selection that results in a higher propeller pitch speed and thus a higher overall aircraft speed and has the advantage that if one engine fails, the plane could still possibly land safely. However, the use of two engines would

greatly increase the RAC. The main disadvantage of the pusher configuration this year was the question if the water would clear the propeller as it was draining. If it did not, the impact of the water on the propeller could have serious consequences and at the very least would require more structure holding the motor in place. Therefore, the puller configuration was chosen.

3.5.5 Payload/Aircraft Size

Two payload sizes (and thus aircraft sizes) were compared: a large payload which would carry all or nearly all of the water possible, and a small payload which would carry about 1 Liter of water. Whichever design was chosen to be the best would then have the payload amount optimized for the highest score. The analysis to determine the RAC was done as before, with several different assumptions. We assume that the small aircraft carries 1 L of water, and uses smaller capacitance batteries (which weigh less) than the larger aircraft because the flight time was expected to be shorter and thus a lower total battery time was needed. Also, the smaller motor was only rated up to 30 amps, so this was assumed to be the current drawn from the batteries. Finally since the control surfaces were small, only one servo was needed for the ailerons. The RAC of the smaller aircraft is shown in Table 3.1. It was necessary to account for the difference in the water carried when applying the figures of merit, as shown in Table 3.2, to decide which aircraft configuration would be best. This was done by accounting for the difference in the RAC effect column: the original ratio of the large aircraft to the small aircraft was 3.5; however, the score of the large aircraft if everything else is the same would be 4 times larger than the small aircraft because of the difference in water carried. Therefore 3.5/4.0 = 0.875 is the effective RAC effect of the small aircraft and the large aircraft is given an effective RAC of 1.0. The small aircraft was expected to complete the course slightly faster than the large aircraft because of the reduced time to fill the aircraft's water tank and the decrease in turn time (because of the increased g-loading).

Table 3.2: Conceptual Design Figures of Merit.

	Component	Time Required (x1)	Skill Required (x2)	Reliability (x3)	Stability & Control (x3)	RAC effect (x4)	Speed (x4)	Final Score
General Configuration	**Monoplane**	1.00	1.00	0.95	1.00	0.93	0.90	**14.25**
	Biplane	0.80	0.85	1.00	0.95	0.86	0.80	12.99
	Tandem	0.85	0.85	0.80	0.85	0.87	0.85	12.78
	Flying Wing	0.90	0.80	0.80	0.80	1.00	1.00	13.70
Wing Planform	Rectangular	1.00	1.00	1.00	1.00	1.00	0.80	14.20
	Tapered	0.95	0.95	0.95	0.95	0.95	0.95	**14.25**
	Elliptical	0.75	0.50	0.90	0.90	0.80	1.00	12.55
Landing Gear Configuration	**Tail Dragger**	1.00	0.95	0.95	0.95	1.00	0.80	**13.90**
	Tricycle	0.95	1.00	1.00	1.00	0.95	0.75	13.75
	Retractable	0.50	0.60	0.75	1.00	0.90	1.00	13.05
Empennage Configuration	**Conventional**	1.00	1.00	1.00	1.00	0.90	0.90	**14.20**
	T-Tail	0.95	0.90	0.90	0.90	0.90	0.90	13.55
	V-Tail	0.90	0.75	0.80	0.95	1.00	0.95	13.85
	Canard	1.00	0.95	0.80	0.80	0.90	1.00	13.70
Engine Configuration	Pusher	1.00	1.00	0.80	1.00	1.00	0.95	14.60
	Puller	1.00	1.00	0.90	1.00	1.00	0.95	**14.70**
	Multi-Engine	0.95	1.00	1.00	1.00	0.85	1.00	14.35
Payload Amount	**Small**	1.00	1.00	1.00	0.95	0.88	1.00	**14.35**
	Large	0.90	0.95	1.00	1.00	1.00	0.90	**14.40**

3.6 Features that Produced Final Configuration

The winning general configuration was the monoplane; however, the analysis was unable to determine whether the small or large aircraft concept should be built. The monoplane was mainly chosen due to the ease of construction and the stability and controllability of the design relative to the flying wing which came in second. While the monoplane did not have the best RAC or the best speed, it was competitive enough in those categories to warrant its selection based on its qualities in the others. The wing planform selected for the monoplane was the tapered wing, because it had more efficient lift generation than the rectangular planform and was still simple and quick to build compared to the elliptic wing. The tail dragger gear was selected mainly due to the pilot's preference. Finally, the conventional tail was chosen because, while it wasn't the fastest or the cheapest (RAC) solution, it did have the overall best features. As mentioned in the executive summary, the team was split into two groups, one which built the small aircraft and another which built the large aircraft. The preliminary and detail design which follows relates specifically to the large aircraft, though the procedures followed for the small plane were essentially equivalent with minor differences due to scaling related problems.

4.0 PRELIMINARY DESIGN

Once the general configuration was selected in the conceptual design phase, the expected aircraft weight (from previous year's payload weight fractions) was refined and optimized by sizing and determining the materials for the various structural components such that the resulting component gave just enough strength to carry the design loads with a 1.25 factor of safety. The tail surfaces were then sized with tail volume coefficients, stability analysis was performed on the design, and an estimate of the design's mission performance (i.e. flight time) was obtained by assuming steady level flight for the entire course and coordinated turns. The end goal of the preliminary design was therefore to analyze the effect of several design parameters so that the aircraft weight could be both optimized and determined with some accuracy, and to analyze the aircraft's general stability and expected mission performance.

The overall aircraft dimensions used for the analysis were those that were generated in the conceptual design phase. Two years ago the DBF team wrote and used an optimization code which iterated through various values for the aircraft's dimensions that were important to the performance of the aircraft (C_L, wingspan, chord, etc.); however, that year's aircraft illuminated a flaw with such optimization: all of the aircraft components are optimized to be near failure. Despite individual component safety factors, the system as a whole has little to no safety factor built in; if one component fails (say a wing stalls due to wind gusts), then the whole system fails. This was illustrated (and reviewed on video by this year's team) several times during the testing of that year's aircraft in which the plane went out of control several times and crashed. It was also noted that the final optimized dimensions were very similar to those designed during the conceptual phase. In fact, each component was no more than 1 or 2 percent different from their optimal counterpart and the overall conceptual design configuration scored within ~5% of the optimal design. The team therefore decided to not pursue optimization in this way; instead we opted to optimize the aircraft by designing the aircraft, test flying the plane, and then iterating the design to incorporate improvements. In this way we were guaranteed an aircraft that could fly at competition and, depending on the number of iterations completed, one that was close to optimal.

Therefore, instead of spending more time optimizing the aircraft dimensions to get the best mission time possible (since small dimension changes had a very negligible effect on the RAC), it was decided to build the aircraft designed in the conceptual phase. This approach had the advantage that the team was able to empirically test the aircraft components and reduce their weight since past experience had shown that actual components almost never performed as well as predicted. Whether this was due to poor assumptions (wood is not isotropic) or just the natural variability of the construction materials (balsa wood's density can vary by a factor of 2 or more), it does not change the risk that optimizing the aircraft dimensions based on the expected theoretical performance of the components presents. Building a conservative plane and then improving its weight also had the advantage that the aircraft would be much

more likely to successfully fly at competition under possibly poor wind conditions since the dimensions were conservative and the aircraft would be flight tested much more than if the dimensions were extensively optimized before construction began. The team concluded that the overall score could be more easily improved by optimizing the actual aircraft instead of the initial dimensions, given the assumption that the initial dimensions were 'close' to optimal.

It was also noted that, while a reduction in a proven aircraft's weight would increase the steady level flight velocity by a little, its main impact on the aircraft's score would be the reduction in the RAC. For example, if the team were to cut 10% from the aircraft's weight (non-battery) then the RAC of the aircraft from the conceptual phase would be reduced by ~2 percent and if the team cut 10% from the battery weight, then the RAC would decrease by ~6 percent because the battery weight is counted twice. The goal of design iterations was therefore to reduce the weight of the aircraft sufficiently so that fewer battery cells would be needed to allow for takeoff. The reduction in thrust (and thus velocity) would be mostly offset by the weight reduction.

4.1 Figures of Merit

Figures of merit were selected for the preliminary design phase in order to eliminate certain design parameters and to allow for the best scoring airplane with the conceptual design dimensions to be built. The complete figure of merit table is shown in Table 4.3.

4.1.1 Required Time

Since the team is composed entirely of volunteers and the contest dictates a tight schedule, an important factor in the feasibility of each component's configuration was the time required to build the configuration. Complicated configuration designs would increase the time required to build the component because many of the members were new and inexperienced. This figure of merit was given a weight of 1.0.

4.1.2 Required Skill

Another important factor in the feasibility of each component's configuration was the ability of the team to build the configuration. Complicated configuration designs would increase the skill, and material cost (due to mistakes) required to build the component. This figure of merit was given a weight of 1.0.

4.1.3 Speed

The flight score any given design could achieve was dependent on the design's total flight time. For a given thrust, one way to reduce the total flight time of the mission would be to reduce the aircraft's drag (the weight is accounted for below). This figure of merit was given a weight of 2.0.

4.1.4 Component Weight

In order to compare the weight of each component's various possible configurations, the worst-case loading for each component was considered. The minimum weight for a given configuration that could withstand the worst-case load was then compared against other configurations of the same component. This figure of merit was given a weight of 3.0.

4.2 Design Parameters Investigated

4.2.1 Fuselage Shape and Assembly

The fuselage design, both its shape and assembly, was vital to success in this year's competition because the score is calculated by the difficulty factor divided by the total time of the run. The fuselage design affects the ability of a given aircraft configuration to carry the desired payload and the total mission time for that aircraft due to drag advantages or disadvantages. The fuselage shapes considered were a horizontal extended symmetric airfoil, a vertical airfoil, and a conventional cylindrical tube. In order for the symmetric airfoils to have a reasonable thickness (<15%) the airfoil would need to be greater than 4 feet long and therefore would require two sections which was considered difficult to make with an airfoil fuselage. However, airfoil shapes are extremely quick and easy for the team to manufacture. The main disadvantage to the vertical airfoil was that in crosswinds the airfoil would be at an angle of attack and, being a low Aspect Ratio wing, it would generate a lot of drag making the vertical airfoil relatively slow. From experience and research on Wichita's wind history the importance of considering crosswinds was critical. The horizontal airfoil fuselage was used for a plane two year's ago and the stability issues created by the fuselage's quarter chord being in front of the CG were difficult to overcome since the CG could not be moved all the way up to the fuselage's quarter chord. Similar problems were expected if an airfoil were to be used for this year's aircraft and therefore a low skill score was given to the horizontal airfoil fuselage. Both airfoil shaped fuselages were potentially heavier than the cylindrical fuselage due to the need for shear web structures inside the airfoil.

In previous years it was found that it was difficult and time consuming to manufacture light cylindrical shapes. However, through several year's worth of improvements it was determined that, in fact, cylindrical shapes could be easily manufactured if styrofoam was a major component of the structure.

This manufacturability is reflected in the realtively higher construction time and skill required scores given in the figure of merit matrix to the cylindrical fuselage. However, since the box that the plane must fit in was only 4 feet long, this required the cylindrical fuselage to be made in two pieces that could then be put together. An ingenious method of assembling the two pieces was decided upon and minimized the extra complexity of the configuration. As can be seen in the design drawings, the rear fuselage had one longeron that protruded from the rear section and would fit tightly into a slot in the front section. The two sections were then secured together by putting several large rubber bands around two rods that protruded slightly from each section. These rubber bands also went over the wing spars and held the wing in place. While there were initial doubts if this method would be stiff enough, the rubber bands were found to meet the stiffness requirements easily through actual wingtip testing in which the aircraft was loaded in a cyclic manner. Table 4.3 shows the final ranking for the fuselage shapes and the cylindrical shape was selected for the aircraft despite the difficulties involved in construction.

4.2.2 Fuselage Materials

Once the fuselage shape was determined to be cylindrical, it was necessary to select the material to be used internally to support the payload and connect the tail section. The materials considered were balsa wood, a fiberglass/balsa composite, and monokoted styrofoam. Carbon fiber was not considered due to radio troubles experienced in a past plane which was traced back to interference caused by carbon fiber structural members around the receiver. The fiberglass/balsa composite allowed the team to add strength only where it was needed such as to the wing carry-through. The fiberglass/balsa option did take both more time and skill than the other options, but the team was relatively experienced with fiberglass and so this difficulty was limited.

The monokoted styrofoam was a new, novel idea for the team this year. From experience the team knew that styrofoam performed well enough in compression, and monokote worked well in tension. Through experimentation the team concluded that for large portions of the aircraft the structure could be built entirely of the styrofoam and monokote 'composite' with a balsa wood framework. Where structurally necessary, balsa spars were inserted into portions of the styrofoam. Because of the ease of construction both the time required and skill required were high. Due to the low weight of both styrofoam and monokote the weight figure of merit was assigned the highest value. The material chosen was the monokote/styrofoam combination.

4.2.3 Spar Cross Section and Material

In order to predict the weight of the spar, both the spar material and the spar geometry had to be chosen. The worst-case load on the spar's structural design was determined to be the "Wingtip test" stated in the competition rules. In this test point loads at each wingtip support the fully loaded plane, which was

assumed to weigh 23.8 pounds. All spar designs considered must be at least strong enough to withstand this test. Also, the estimated maximum torque with ailerons fully deployed was considered as a constraint when necessary; however, it was in general less limiting than the bending case. The torque from the ailerons on the wing spar configuration was estimated to be 17 ft-lbs.

The spar materials investigated included 2014-T6 aluminum, balsa wood, a fiberglass/balsa composite used last year, carbon-fiber, and, in another new concept for this year, a lite-ply/balsa laminate. The aluminum spar cross-sections considered were a hollow circular geometry and box beam geometry. The balsa wood and fiberglass-balsa composite spar cross-sections considered were a rectangular box beam and a solid rectangular beam. The only carbon-fiber cross-section considered was a hollow circular cross-section. In order to properly design the spars, tapered cross-sections were needed because of the wing taper. It was assumed that the airfoil's maximum height at the root was 1.75 inches while at the tip it was only 1 inch. If the spar height was only 1 inch high over the entire wingspan, the spar would need to be excessively thick to support the bending load. Also, two spars were always assumed so that the rubber band attachment method could be used to secure the wing assembly to the fuselage without the wing twisting about the spar. The two spar assembly also had the advantage that the spars could be made straight since the quarter chord would always be between the two spars. The single spar would have to be angled along the quarter chord sweep so that the foam would not need to carry much of a load.

In order to find the minimum weight for a given material/cross-section combination, the following flexure formula was used

$$\sigma_u = \frac{M_{max}\, y}{I} \qquad 4.1$$

where σ_u is the ultimate stress, I is the moment of inertia and is determined by the spar geometry, y is the distance from the neutral axis, and M_{max} is the maximum moment the spar experiences from the load. Since the limiting load was a point load at the wing tip (equal to one-half of the plane's mass) and the maximum stress will occur at a height, y, equal to one-half of the spar's maximum height, (4.1) can be simplified into three different cases based on the spar's moment of inertia.

The following torsion formulas for (a) rectangular and (b) thin walled members were used to find the minimum weight:

$$\tau_{max} = \frac{T}{\alpha h w^2} \qquad 4.2a$$

$$\tau_{max} = \frac{T}{2tA_m} \qquad 4.2b$$

where, in (4.2a), τ_{max} is the maximum shear strength of the material, T is the applied torque from the deflection of the flaperons, h is the height of the spar, w is the width of the spar, and α is a dimensionless constant that depends on $^{h}/_{w}$ obtained from Table 4.3 in [1]. In (4.2b), A_m is the area enclosed by the median line, and t is the thickness of the spar.

For each of the materials, σ_u was obtained from previous year's test data in which three-point loads were applied until failure and, in aluminum's case, looking up the published ultimate stress value for 2014-T6 aluminum. Table 4.1 shows densities and ultimate stresses of the various materials

Table 4.1: Experimental Values for Material Properties.

	Al – 2014-T6	Balsa Wood	Fiberglass/Balsa	Carbon-Fiber	Lite-Ply/Balsa
σ_u (ksi)	60	1.7	3.1	7.5	4.3
τ_{max} (ksi)	4	0.3	~ 0.6	~ 2.5	~ 0.9
ρ (lb/ft^3)	175	7.2	10.6	75	12

Two approaches were used to determine the spar dimensions for each configuration that gave the minimum weight and supported the wingtip load: one for rectangular cross sections and the other for the circular spar geometry. Both geometries were constrained by a maximum height/diameter that was based on the quarter chord thickness of the wing. For preliminary design purposes, this thickness was assumed to be 1.75 inches. The hollow spars were also constrained by an estimated minimum thickness to avoid thin wall buckling. For the metal spars this was assumed to be 1/32 -inch, for the carbon-fiber and the balsa wood it was assumed to be 1/16 -inch. The last constraint on the spars was material availability. For the metal spars, another constraint is that the aluminum is only readily available in discrete diameters leading to certain spar geometries being unattainable.

For the rectangular geometries, the torsion formula (4.2a) was solved for the minimum width assuming the height of the spar was equal to the maximum allowed height. The maximum stress the spar would experience was then calculated from the flexure formula for the wingtip load. If this exceeded the ultimate stress value shown in Table 4.1, the width was increased so that the maximum stress in the spar was less than the ultimate stress. The weight of the 4-foot long spar was then calculated from the dimensions found and is shown in Table 4.2.

For the hollow circular cross-section, the flexure formula was solved first. Assuming the wall thickness to be equal to the minimum thickness to avoid buckling, the minimum diameter required to satisfy the wingtip load was calculated. If this diameter exceeded the maximum allowed, the thickness was increased such that the diameter was equal to the maximum allowed. The maximum shear experienced

in the spar was then found from formula (4.2b) based on the estimated pitching moments produced with full flaps down. If this shear exceeded known, representative values, then the wall thickness was further increased until the spar could withstand the required loading. The weight of a 4 ft long spar was then calculated.

Table 4.2: Spar Material/Cross-Section Minimum Weights.

Material	Cross-Section	Weight (oz)
Al	Hollow-Circular	3.82
Al	Hollow-Rectangular	4.5
Lite-Ply/Balsa	Solid	3.12
Balsa	Hollow-Rectangular	3.89
Balsa	Solid	2.95
Carbon	Hollow-Circular	4.22
Fiberglass/Balsa	Solid	3.3
Fiberglass/Balsa	Hollow-Rectangular	3.2

The estimated time and skill required for each spar design was discussed in a group meeting. The aluminum spars would require machine shop access and skills that the team did not have, so they were scored correspondingly low. The composite laminate spars, such as the Lite-Ply/Balsa and the fiberglass/balsa designs, required skills with composites that several, but not all, of the team had from previous years. The hollow rectangular balsa wood spar designs required more time and slightly more skill than the solid balsa spar configurations, and were therefore scored slightly lower in both categories. The spar materials and cross-sections were assumed not to affect the speed of the flight. A solid balsa spar was selected initially based on its figure of merit score (see Table 4.3). However, after several years of wing failures due to flight conditions just beyond the predicted flight envelope it was decided that a margin of additional strength would be worth the additional weight. After building and testing several spars the Lite-Ply/Balsa laminate was found to add sufficient resistance to twisting and, as an added benefit, prevented the balsa wood from warping.

4.2.4 Wing Structure

Four wing structures were considered; a solid foam core with a monokote surface, foam ribs covered by monokote, balsa ribs with monokote, and foam ribs with birch sheeting. The solid foam core could be easily and quickly produced, but would weigh the most. The foam ribs would weigh the least, but take the longest to make and have higher drag than the solid foam wing because the monokote tends to bow slightly in-between the ribs. In addition, the team encounters warping problems when working with thin sheets of styrofoam. The balsa ribs, since they are denser than foam, would need to be thinner in order to make the wing lighter; however, after building and testing a sample wing the monokote was found to bow too much in-between the ribs as the thickness of the ribs decreased, and so the balsa rib wing would be heavier and have more drag than the foam rib wing. Foam ribs with thin birch sheeting instead of a

monokote surface eliminated the bowing of the monokote, but added weight and significant time to the wing construction. The foam core wing with monokote was chosen.

4.2.5 Landing Gear Cross-Sections

The team considered three different landing gear extension cross sections: a hollow circular beam, a solid circular beam, and a standard solid tapered rectangular beam. Both of the circular beams would require extra structure in the wing since the wheel base needed to be further apart for ground stability than the fuselage allowed. The standard solid tapered main gear commercially sold was bent such that the wheelbase was increased and yet the gear was still mounted on the fuselage. The aircraft drawings in the detail design show the standard gear more clearly. The hollow circular beam would be the lighter, but wider than the comparable solid circular beam and therefore, even after adding fairings, the hollow beam would have slightly more drag. Also, previous teams have already machined a connector between the landing gear strut and the extension for a solid cross section; however, this connector would not work for a hollow cross section and so new connectors would need to be machined, a skill the team did not have. Since properly sized tapered rectangular main gear were sold commercially, the time and skill required for them were minimal. The final landing gear's pieces are labeled in the detail design drawing package and show that the team chose the solid tapered rectangular beam.

4.2.6 Tail Assembly

A conventional design warranted a tail structure that would provide both strength against vibration effects and adaptability for the construction portion of the competition, while remaining as light and easy to construct as possible. The dimensions of the control surfaces were determined from the preliminary design. However, the supporting structure was decided from several options. The first was to use two vertically spaced beams, which would be inserted into the back of the main fuselage and then slant towards the vertical tail structure. The beam design would provide strength against vibration due to the distance between the beams and the relatively solid structure of the beams. Additionally, this would make it relatively easy to construct in terms of the beam. The main drawback was that the angle precision required to perfectly align the tail section with the fuselage caused concern. The next design feature was the use of balsa, or Styrofoam ribs as a main structural component. This would satisfy the light weight requirement. By using longerons along the inner diameter of the ribs the vibration could be damped as well as providing a method of attachment to the main structure. To test the manufacturability of the concept a full sized tail section was built using this method. Although the strength was impressive the construction skill and time required would prove to be its undoing. The third and chosen method was to use a single basswood beam. The torque characteristics from simple tests on sample pieces were deemed acceptable, while construction time proved quick and efficient when inserted into a solid section of styrofoam.

Table 4.3: Preliminary Design Figures of Merit.

	Component	Time Required (x1)	Skill Required (x1)	Speed (x2)	Weight (x3)	Final Score
Fuselage Shape	Horizontal Symmetric Airfoil	1.00	0.80	1.00	0.90	6.50
	Vertical Symmetric Airfoil	1.00	1.00	0.90	0.90	6.50
	Cylindrical	0.90	0.95	0.95	1.00	**6.75**
Fuselage Shear Web Materials	Balsa Wood	1.00	1.00	0.00	0.88	4.64
	Fiberglass/Balsa	0.90	0.95	0.00	0.95	4.70
	Monokoted Styrofoam	1.00	1.00	0.00	1.00	**5.00**
Spar Configuration	Al (H,C)	0.70	0.60	0.00	0.77	3.62
	Al (H,R)	0.50	0.40	0.00	0.66	2.87
	Lite-Ply/Balsa	0.90	0.85	0.00	0.95	**4.60**
	Balsa (H,R)	0.85	0.95	0.00	0.76	4.08
	Balsa (S,R)	1.00	1.00	0.00	1.00	5.00
	Carbon (H,C)	0.50	0.45	0.00	0.70	3.05
	Fiberglass/Balsa (H,R)	0.70	0.85	0.00	0.89	4.23
	Fiberglass/Balsa (S,R)	0.90	0.90	0.00	0.92	4.57
Wing Structure	**Foam core w/monokote**	1.00	1.00	1.00	0.75	**6.25**
	Foam ribs w/monokote	0.70	0.70	0.90	1.00	6.20
	Balsa ribs w/monokote	0.90	0.90	0.85	0.89	6.17
	Foam ribs w/birch sheeting	0.65	0.80	1.00	0.80	5.85
Landing Gear Cross-Sections	Hollow Circular Beam	0.90	0.90	0.98	0.95	6.61
	Solid Circular Beam	0.95	0.95	1.00	0.90	6.60
	Solid Rectangular Beam	1.00	1.00	0.90	1.00	**6.80**
Empennage Structure	**Single Element Beam**	1.00	1.00	0.00	0.90	**4.70**
	Balsa/Foam Ribs	0.65	0.80	0.00	1.00	4.45
	Slanted Beams	0.90	0.90	0.00	0.85	4.35

4.3 Tail Sizing

Tail volume coefficients computed from previous year's aircraft were used to initially size the vertical and horizontal tails. The horizontal tail volume horizontal tail coefficient used was 1.3 and the vertical tail volume coefficient used was 0.08. The horizontal tail volume coefficient is defined as

$$V_H \equiv \frac{S_H}{S_W}\frac{L_H}{\bar{c}} \qquad 4.3$$

where V_H is the horizontal tail coefficient, S_H is the horizontal tail planform area, S_w is the wing planform area, L_H is the distance from the wing ac to the horizontal tail ac, and $\bar{c}$ is the wing mean aerodynamic chord (MAC). The vertical tail volume coefficient is defined as

$$V_V \equiv \frac{S_V}{S_W}\frac{L_V}{b} \qquad 4.4$$

where V_V is the vertical tail coefficient, S_V is the vertical tail planform area, L_V is the distance from the wing ac to the vertical tail ac, and b is the wing span.

The distance from the tail's ac to the wing ac was set by the fuselage length of 5.25 feet. The horizontal tail's span was further constrained to be 2 feet so that the horizontal tail may be permanently attached to the back fuselage section and still fit in the box. The vertical tail was tapered by 50 percent so that the ac of the vertical tail would be closer to the aircraft's vertical CG and so that when the rudder was used, the aircraft would roll less than if the ac was higher. If it was assumed that the wing ac was a quarter of the way along the fuselage and that the trailing edge of the tails were at the end of the fuselage, it was possible to size the horizontal and vertical tails from (4.3) and (4.4). The calculated dimensions are shown in Table 4.4.

Both tails were made as flat plates instead of airfoils because this allowed the tail to be built with only a balsa frame with a monokote membrane instead of either a foam or ribbed airfoil shape with a central spar. The balsa frame method was extremely quick and easy to construct and was ~25 percent lighter. The horizontal tail's vertical placement was such that the fuselage could taper down equally on each side and therefore produce no lift at zero angle of attack. So that the less efficient tail was not producing lift (and therefore induced drag), the tail incidence was set to counter the downwash angle resulting from the wing. The detail design phase more precisely sets the incidence of the tail such that almost no lift is needed from the tail at cruise conditions.

Table 4.4: Empennage Parameters (inches).

Horizontal Span	24
Horizontal Chord	14.5
Horizontal Incidence (rad)	0.035
Vertical Height	19
Vertical Root Chord	16
Vertical Tip Chord	8

4.4 Aircraft Performance Estimates

4.4.1 Weight Breakdown

To get an estimate for the aircraft performance, it was necessary to determine the aircraft's weight more precisely. It was assumed that each aircraft configuration would use 5 servos: 1 for the rudder and the front gear, 2 for the ailerons, 1 for the elevator, and 1 for the water release valve. For the Futaba servos that the team used, the weight per servo was 1.5 oz. The receiver, battery, controller, and fuse that the team used weighed 1.1, 3.4, 2.2, and 1.6 ounces respectively. The motor used in previous years was too small for this year's larger aircraft, so it was decided to use the next larger motor offered by Astroflight, the Cobalt 640S which has a 3.1:1 gear ratio and a weight of 18 ounces. It was assumed that a propeller of ~17 inches would be needed which weighs ~4 ounces (if it's an electric propeller) while the corresponding spinner weight of 2.1 ounces. The commercial main landing gear available weighed 6.5 ounces and the 3-inch diameter rubber wheels that the team selected weighed 1.6 ounces apiece, while

the tail gear weighed 0.7 ounces. The first estimate used for the battery weight was the conceptual design phase weight of 76.8 ounces; however, when all the weights were summed, the final loaded aircraft weighed 22.3 pounds instead of the 23.8 pounds predicted in the conceptual phase. The number of needed batteries and the aircraft dimensions were then recalculated as in the conceptual design phase and the new expected weight of the components found until the weight difference between iterations became small. The other weights listed in Table 4.5 were calculated while investigating the design parameters for the aircraft.

Table 4.5: Preliminary Weight Estimate

Component	Weight [oz]
Wing and Spar	43.8
Vertical Tail	2.1
Horizontal Tail	2.6
Tail Gear	0.7
Main Gear	6.5
Wheels	3.4
Fuselage	36.5
Motor	18.0
Propeller	4.0
Spinner	2.1
Motor Controller	2.2
Radio Reciever	1.1
Battery, Fuse, and Wiring	6.5
Propulsion Batteries	65.3
Servos	9.8
Payload	140.8
EMPTY AIRCRAFT WEIGHT (lb)	**12.8**
LOADED AIRCRAFT WEIGHT (lb)	**21.6**

4.4.2 Mission Model and Estimated Performance

In order to quantify the mission performance, several assumptions were necessary. The first assumption dealt with the flight plan for each mission which consisted of flying laps around the course shown in Figure 3.1. Since the flight score was dependent on the total flight time, the flight plan was designed to minimize the time while limiting the pilot difficulties. This was accomplished by limiting the flight plan such that the aircraft flew at full throttle for the entire course except for after the landing approach turn when the plane's airspeed was reduced to ~1.2 times the stall speed of the aircraft.

For both missions, it was assumed that the aircraft was in steady level flight for the entire distance of the linear course sections (sections 1, 3, 5, and 7 when not landing). Also, it was further assumed that quasi-steady, coordinated level turns were performed. This allowed the mission flight time to be approximated as a water loading segment (for the fire bomber mission), a takeoff segment, a landing segment, a cruise segment, and a turning segment. The cruise flight time was determined by dividing the minimum linear

course distance (3000 feet for the fire bomber mission and 7500 feet for the ferry mission) by the cruise velocity and multiplying by 1.1 to account for the fact that the aircraft is not always in steady level flight. The turn velocity was assumed to be equal to the cruise velocity and the load factor was assumed to be 2.5 for the first turn during the fire bomber mission and 4.4 for when the aircraft was empty (as in the ferry mission). The ferry mission had a higher load factor because there was no payload for the mission, while a load factor of 2.5 was used for the ferry mission because the wingtip test simulates a 2.5g wing loading with the full payload and the wing design, with a safety factor, was just strong enough to pass the wingtip test.

The steady level cruise velocity was found by setting the thrust equal to drag and the lift equal to weight. This required the drag of the aircraft to be estimated, which was done with an empirical model; details of this model can be found in [2]. Unfortunately, this model was made for large jet aircraft at (typically) very large Reynolds numbers; however, since the team was unable to find anything better, it was decided to use the model as a crude drag estimate. We particularly expected the predicted parasite drag to be wrong because of the propeller's effect on the flow about the fuselage. First, the model assumes a parabolic drag polar with constant coefficients of the form

$$C_D = C_{Do} + KC_L^2 \qquad 4.5$$

where C_{Do} and K are assumed constant since the low speed of aircraft removes the mach number dependence. To calculate C_{Do}, the equivalent parasite area method was used. That is, the contribution of drag from friction was summed according to

$$C_{Do} = C_{Df} = \sum_k \frac{1.1 f_k}{S_W} \qquad 4.6$$

where f_k is the equivalent parasite for each component of the airplane, S_W is the exposed surface area of the wing, and the factor of 1.1 is a "safety factor" to hopefully ensure that the drag is over estimated. The team desired to over estimate the drag because this would be easy to fix by merely reducing the aircraft thrust with a decrease in the number of battery cells. The equivalent parasite, f_k, is determined using the average skin friction coefficient, an interference factor accounting for imperfect connections between components (e.g. the wing connection to the fuselage), and a form factor that accounts for thickness effects. The induced drag coefficient, K, was calculated using the standard form

$$K = \frac{1}{\pi A_W e} \qquad 4.7$$

where A_W is the aspect ratio of the wing and e is Oswald's efficiency estimated from the wing aspect ratio and taper ratio [3].

The propeller thrust as a function of velocity was also estimated in order to have a better performance estimate. According to Electricalc, a commercial propeller program, the pitch velocity corresponds to the

velocity at which the static thrust of the aircraft is reduced by a factor of two. Electricalc also predicted a linear relationship between thrust and velocity until well after the pitch velocity, so the thrust was modeled as

$$T(V) = T_0\left(1 - \frac{V}{2V_p}\right) \qquad 4.8$$

where T is the thrust for a given velocity, T_o is the static thrust, V is the velocity, and V_p is the pitch velocity of the propeller-motor combination. To ensure that the plane did not loose thrust and stall due to strong gusts, it was necessary to ensure that the stall velocity of the aircraft not approach the non-linear region of the thrust versus velocity curve. Therefore, the pitch velocity was assumed to be at least 40 mph; which allowed for gusts greater than 20 mph before the aircraft would lose thrust.

The takeoff with the full water payload was the most constraining requirement and was therefore used to determine the needed static thrust. The takeoff time (and distance) was found by integrating the 1-D equations of motion numerically, assuming a $C_{L,TO}$ of 1.2 (~90 percent of the assumed $C_{L,max}$), a wheel friction of 0.03, an aircraft drag polar as computed above, and airplane dimensions as designed in the conceptual phase. Figure 4.1 shows the velocity of the plane at the end of the runway on the left axis as a function of static thrust for several pitch speeds. If a pitch speed of 55 mph is assumed, a static thrust of ~7 pounds would be needed in order to takeoff in 150 feet. For this propulsion configuration, the loaded takeoff time was 7.5 seconds and the unloaded time was 5.6 seconds. The landing phase was estimated to take 20 seconds, and the time to fill up the water containers was estimated to take 15 seconds after some simple experiments were performed with a 2 L bottle.

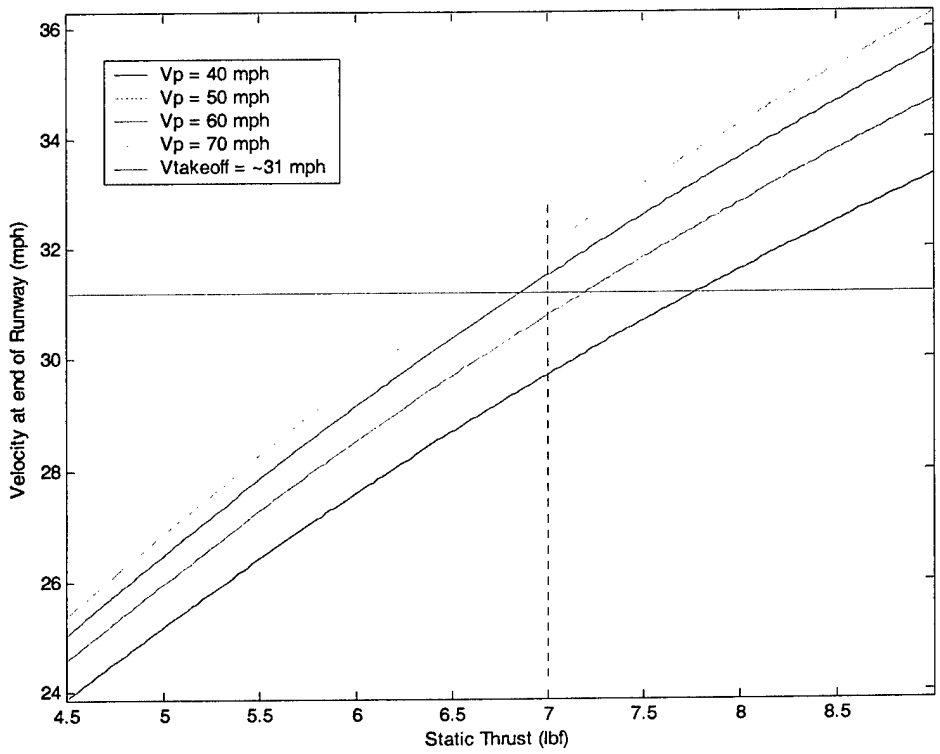


Figure 4.1: Velocity vs. Static Thrust at end of runway.

By setting the thrust equal to the drag and lift equal to weight at cruise, the following equation can be found

$$0 = \left(\frac{1}{2}\rho S_W C_{Do}\right)V^2 + \left(\frac{T_o}{2V_p}\right)V + \left(\frac{2kW^2}{\rho S_W}\right)\frac{1}{V^2} - T_o \qquad 4.9$$

where the variables are as defined earlier. Equation 4.9 can be solved with a Newton-Raphson technique for the cruise velocity. The values used for the aircraft are shown in Table 4.6; as are the cruise velocities determined from (4.9).

Finally, in order to estimate the flight time of each mission, the time spent in the turns must be accounted for. This was done by assuming quasi-steady, level turns as mentioned earlier. This allowed for the calculation of the turn time as

$$t_{turn} = \frac{\theta * V_c}{g\sqrt{n^2 - 1}} \qquad 4.10$$

where t_{turn} is the turn time, θ is the number of radians in the turn, V_c is the velocity at which the turn is taken, g is gravity, and n is the load factor of the turn. Table 4.6 shows the time breakdown for the missions and the total estimated flight time and overall flight score. The flight time estimate is most

sensitive to error in the estimated value for the friction drag coefficient, C_{Do} and the landing time. A change of 10% in C_{Do} would change the overall time by 2.4%, and if the aircraft rolls off the runway, 20 or more seconds might be spent retrieving it and positioning the plane at the starting line.

Table 4.6: Aircraft Performance Parameters.

Parameter	Value
Wing Area (ft^2)	10.6
Static Thrust (lbf)	7
Pitch Velocity (mph)	55
C_{Do}	0.0278
k	0.0541
RAC	14.4
Ferry Mission	**Fire Bomber Mission**
Weight (lbf)	
12.8	21.6
Cruise Velocity (mph)	
51.9	50.4
Takeoff & Landing (sec)	
26	85
Turns	
29	17
Cruise	
108	45
Total Time (sec)	
163	147
Overall Score (without the paper score)	
0.52	

4.4.2 Stability Characteristics

A MATLAB stability code was written to calculate the longitudinal stability of the aircraft. This code took the aircraft parameters from the conceptual and preliminary design phases and calculated various stability parameters. The most important parameter in predicting the longitudinal aerodynamic response is the location of the aircraft aerodynamic center (X_{ac}) relative to the aircraft center of gravity (X_{cg}) which was estimated by using approximate positions for the various components. The difference $X_{ac} - X_{cg}$ normalized by the mean aerodynamic chord is termed the static margin which must be positive (if the distances are measured from the aircraft nose) in order for the airplane to be statically stable.

In order for the longitudinal stability of the aircraft to be calculated, the geometry of the airplane must be considered; that is, the location of the horizontal tail with respect to the wing. The longitudinal distance, L_h, between the horizontal tail aerodynamic center and the wing aerodynamic center and the vertical displacement, H_h, between the aerodynamic centers are used to calculate the downwash from the wings

and its effect on the horizontal tail. The downwash model used was empirical, and is given in [2]. Also, $C_{L_{\alpha,WB}}$ and $C_{L_{\alpha,HT}}$ were calculated from the wing aspect ratio (assuming the half-chord sweep angle is approximately zero and the fuselage effect is negligible) and the horizontal tail aspect ratio. Finally $X_{ac,WB}$ was estimated by placing the wing a quarter of the way along the fuselage. Throughout the longitudinal stability analysis, the fuselage's effect was assumed to be negligible as was assumed by [2] for a symmetric, cylindrical fuselage. With these quantities the X_{ac} of the whole aircraft could be calculated according to

$$X_{ac} = \frac{C_{L_{\alpha WB}} X_{acWB} + C_{L_{\alpha H}} (1-\varepsilon_\alpha) \eta_H \frac{S_H}{S_W} X_{acH}}{C_{L_{\alpha WB}} + C_{L_{\alpha H}} (1-\varepsilon_\alpha) \eta_H \frac{S_H}{S_W}} \qquad 4.11$$

where $C_{L_{\alpha WB}}$ is the lift curve slope for the wing body (as mentioned, assumed to be equal to that of the wing), X_{acWB} is the location of the wing-body aerodynamic center, $C_{L_{\alpha H}}$ is the horizontal tail lift curve slope, ε_α is the downwash angle slope, η_H is a tail efficiency factor, S_H is the horizontal tail area, S_W is the wing area, and X_{acH} is the position of the horizontal tail aerodynamic center. Using this method to calculate X_{ac}, the static margin was calculated as 0.31. Hence, the airplane was statically stable. While this value seemed high, the desired static margin was ~0.2, the team decided a slightly sluggish response would be acceptable for the initial test flights and could be optimized during the test flights. The team flew the first test flight in this configuration and changed the static margin gradually through multiple flights by moving the batteries until the pilot was satisfied with the stability and control qualities of the aircraft.

As with longitudinal static stability, the airplane must also be stable laterally; that is, the plane must return to its original operating condition when perturbed by a gust of wind from the side or any sideslip angle. To ensure lateral static stability, the total yawing moment stability derivative, $C_{n\beta}$, must be positive. The two major contributions to yawing stability stem from the fuselage and the vertical tail. The fuselage tends to be destabilizing while the vertical tail is stabilizing. Thus, the vertical tail size must be such that lateral stability is maintained. Using empirical relations for the contribution of the fuselage and vertical tail to $C_{n\beta}$ from [4], the value of $C_{n\beta}$ was determined to be 0.14. Thus, the airplane is laterally statically stable.

Before proceeding with the design, the longitudinal and lateral dynamic performance of the airplane was calculated. Using steady, level flight assumptions, the stability derivatives were approximated using relations from [2]. Table 4.7 shows the calculated nondimensional derivatives for the aircraft. Since the aircraft mach number is small, many of the derivatives are equal to zero. Correspondingly, the eigenvalues for the longitudinal response of the airplane in steady, level flight are shown in Table 4.8. Both modes display convergence. The short period mode is highly damped and should be unobservable

in flight, while the phugoid mode is lightly damped, but has a long oscillatory cycle and should be controllable even if it turns out to be slightly unstable.

Table 4.7: Nondimensional Derivatives.

Derivative	Value	Derivative	Value	Derivative	Value
C_{X_α}	0.15	C_{m_α}	-1.61	C_{Z_α}	-4.57
$C_{X_{\dot\alpha}}$	0.0	$C_{m_{\dot\alpha}}$	-5.25	$C_{Z_{\dot\alpha}}$	-2.0
C_{X_q}	0.0	C_{m_q}	-13.7	C_{Z_q}	-5.21
C_{X_u}	0.0	C_{m_u}	0.0	C_{Z_U}	0.0
$C_{X_{\delta_E}}$	0.0	$C_{m_{\delta_E}}$	-1.3	$C_{Z_{\delta_E}}$	-0.5

Table 4.8: Eigenvalues to Step Input at Cruise Velocity.

	Eigenvalues	**Natural Frequency (rad/sec)**	**Damping Ratio**
Short Period Mode	-2.8 ± 4.76i	5.52	0.51
Phugoid Mode	0.01 ± 0.21i	0.21	-0.05

The dynamic lateral stability was not calculated because the two references known to the team disagreed on some of the approximations used to find the lateral stability derivatives. Thus the team decided that the aircraft should at least be controllable laterally, if not dynamically stable, since the aircraft had static lateral stability and was a standard monoplane aircraft. Flight tests later showed that this assumption was correct as no instabilities manifested themselves in crosswind gusts of ~10 mph.

One of the major difficulties in determining the stability derivatives was the calculation of the moments of inertia for the airplane. The facilities to conduct a dynamic measurement to estimate the inertia characteristics of the aircraft were insufficient at the time. The team resorted to AUTOCAD. AUTOCAD provides inertia calculations based on a unit mass. The moments were found for the balsa, Styrofoam, and payload using AUTOCAD and then later scaled by the appropriate density and translated to the proper coordinate system. The materials are assumed homogeneous in this calculation. While this is not physically true, it was determined that the assumption should provide reasonable results in estimating the dynamic performance of the airplane. The contribution of the motor, batteries, and landing gear to the moments of inertia were approximated by using point masses at the specified locations of these components.

5.0 DETAIL DESIGN

Preliminary design chose several design parameters, made certain assumptions, and then designed several aircraft components. The overall aircraft information from the preliminary design is shown in Table 5.1. The purpose of detail design was to match components and systems that would allow the aircraft to achieve the desired characteristics given in the preliminary design.

Table 5.1: Preliminary Aircraft Parameters.

GENERAL PARAMETERS	
Empty Aircraft Weight	12.8 lb
Loaded Aircraft Weight	21.6 lb
Expected Total Flight Score	0.52
WING PARAMETERS	
Wing Span	8 ft
Mean Aerodynamin Chord	1.32 ft
Taper Ratio	0.62
Main Spar Height (root)	1.75 in
Main Spar Height (tip)	1 in
Main Spar Thickness	.75 in
Reynolds Number	~650000
PROPULSION PARAMETERS	
# Motors	1
Static Thrust	7 lbf
Picth Velocity	55 mph
AERODYNAMIC PARAMETERS	
$C_{L,MAX}$	1.3
$C_{L,TO}$	1.2
$C_{L,CRUISE}$	0.38
C_{do}	0.0258
k	0.0566

5.1 Component Selection

5.1.1 Propulsion Configuration

It was decided in the conceptual design phase that the airplane would have 1 motor. As mentioned earlier, last year's motor was barely powerful enough for the contest's ~6 pound payload, therefore the team decided to use the next larger motor, the Astroflight Cobalt 640S with a 3.1:1 gear ratio. Further analysis into which motor would perform the best was done after the aircraft was shown to fly, and is presented in the ongoing improvements section. The batteries used in the previous year were chosen by examining 177 different Ni-Cad batteries and ranking them based on their weight per battery cell, internal

resistance, and total milliamp-hours. The SR 2400 Max batteries won the comparison due to their very low internal resistance, low weight, and decent capacitance. Further examination this year yielded the same conclusion for the large aircraft; however, it was found that the SR 1200 Max batteries were the best for the small aircraft because of scaling issues.

The static thrust this motor needed to produce in order to takeoff in 150 feet was found in preliminary design to be 7 lbf and the desired pitch speed of the propeller was 55 mph. Static propeller tests were performed in order to determine the number of batteries, the diameter of the propeller, and the pitch of the propeller needed to meet the static thrust requirement. Only electric propellers were used because they are more efficient at the expected rotation rates of an electric motor and they are ~60% lighter than the standard airfoils. No experimental method for determining the pitch speed was available to the team, so the team relied on the commercial programs Electricalc and Motorcalc to estimate the pitch speed and considered all the configurations with at least 90% of the desired pitch speed. Since the programs were fairly accurate (±10%) in their static thrust predictions, it was believed that the pitch speed estimates would be of similar error. The programs predicted that the pitch to diameter ratio had to exceed ~0.65 in order to meet the desired pitch speed. Initial tests revealed that any propeller with a diameter smaller than 16 inches could not meet the 7 lbf static thrust and the 55 mph pitch speed requirements. Table 5.2 shows an abbreviated list of the propeller data collected. The propulsion configuration chosen was the 18x12 propeller with 21 batteries because the other configurations had either too little static thrust or too small a pitch speed. While this did not have the desired pitch speed, it did produce more static thrust than needed, it had a plenty of battery lifetime at full throttle to complete the missions, and it reduced the RAC of the aircraft by 2 points (~14%). This reduction more than offset the slightly longer mission time that resulted from the slower cruise velocity. Finally, the takeoff distance remained the same due to the extra static thrust.

Table 5.2: Propeller Test Data.

Diameter (in)	Pitch	# Cells	Static Thrust (lbf)	Pitch Speed (mph)	Max Power Lifetime (min)
17	10	20	7.01	42	5.4
17	10	21	7.48	43	5.1
17	12	20	5.94	49	4.8
17	12	21	6.29	51	4.6
18	10	20	7.84	40	4.7
18	10	21	8.31	42	4.5
18	12	20	6.18	47	4.4
18	12	21	7.13	49	4.2
19	12	20	7.36	45	4.1
19	12	21	7.60	47	3.8

5.1.2 Airfoil selection

With the sizing parameters complete, the airfoil for the wing had to be selected. In order to select the airfoil, a search was done through the NASG online catalog of airfoils [5]. The airfoils were then narrowed down with several constraints. The main constraint on the airfoil was that it must generate the desired lift coefficient for the airplane. This corresponded to a $C_{L,max}$ of 1.3 and a $C_{L,cruise}$ of ~0.4. Furthermore, the previous year's team experienced a great deal of trouble cutting the airfoil trailing edge, so the trailing edge had to be sufficiently thick so the shape could be cut with some uniformity. Finally, the airfoil was not considered if it was excessively thick/thin (>13% or <9%) or if it was highly cambered. Thick airfoils will generally have more drag, thin airfoils will present spar weight problems and generally stall at lower angles of attack; and highly cambered airfoils generally have a large moment, which the team was uneasy with since no empirical data was available on the NASG website.

Searching through the database and comparing to airfoil shapes, the list was narrowed to eight airfoils for detailed comparison; for example the Clark Y was discarded because its $C_{L,max}$ was only 1.15. The airfoil that the team decided to use for the aircraft was selected from this group by examining the drag coefficient properties of the airfoil around $C_{L,cruise}$, and it's stall angle. The Eppler E214 airfoil and the SG 6042 airfoils had the best overall properties of the eight airfoils. The measured airfoil polar curves at the closest Reynolds number available on the NASG website are shown below and the data summed in Table 5.3. The SG 6042 has a larger stall angle and a higher $C_{L,max}$ at the takeoff Reynolds number of ~200000, and the cruise characteristics $C_{D,cruise}$ is approximately the same as the E214. The E214 was also thicker, though not by much, which would allow for taller spars which would save weight; however, since the wing was designed with two main spars on each side of the quarter chord, the weight savings would be minimal. In the end, the SG 6042 airfoil was selected based on its higher $C_{L,max}$ at takeoff and slightly lower C_{Do} at cruise.

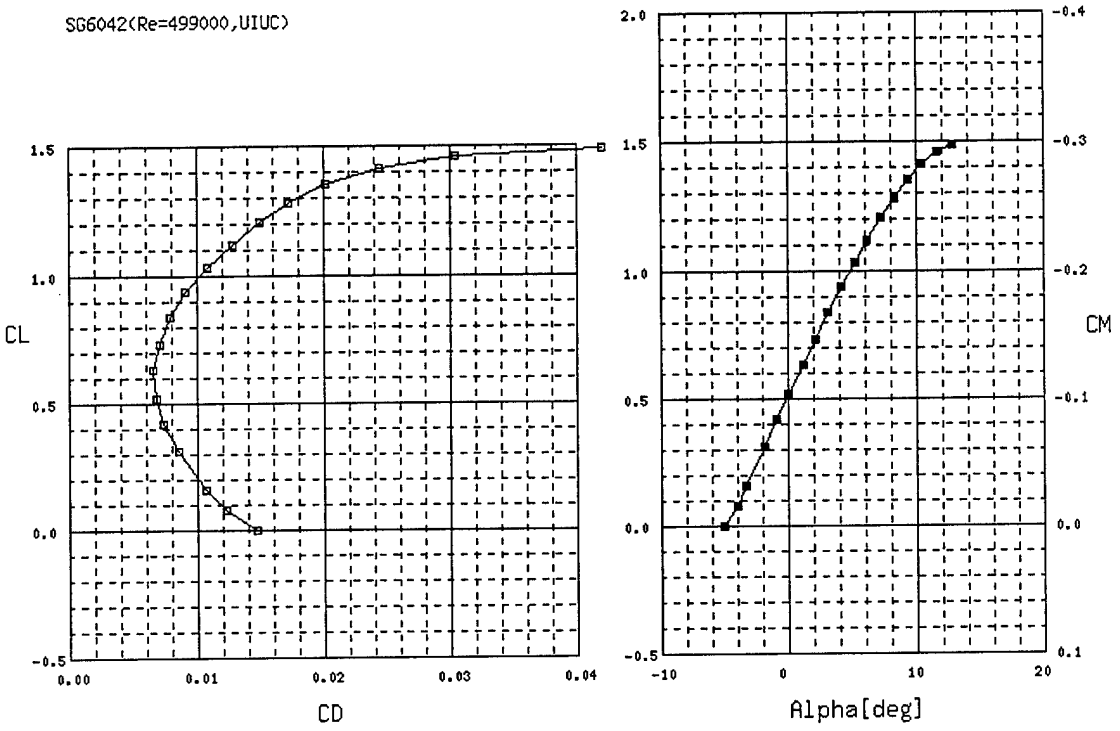


Figure 5.1a: SG6042 [5]

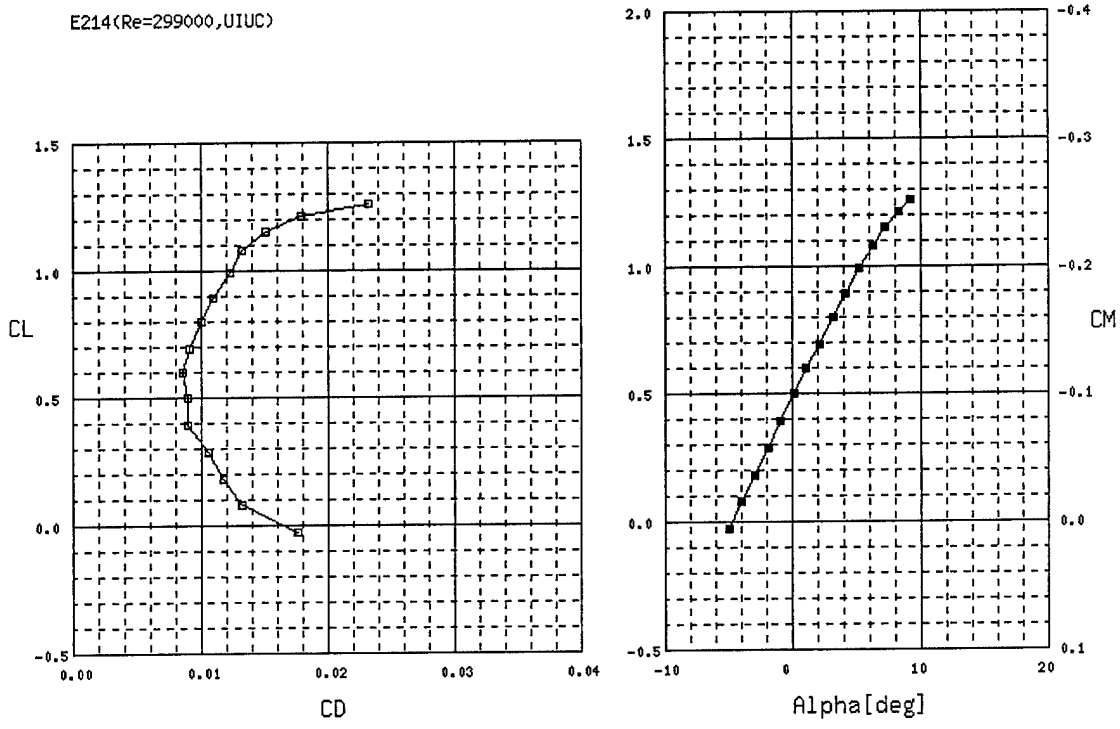


Figure 5.1b: E214 [5]

Table 5.3: Airfoil Property Comparison.

Property	SG 60423	E214
Takeoff $C_{L,max}$	1.39	1.31
$C_{D,cruise}$	0.008	0.009
Thickness	0.10	0.11
Takeoff Stall Angle (°)	12.5	11.4

5.1.3 Wing Placement

To place the wing, it was desirable to minimize the trim drag and ease of control for the pilot. Hence, when in steady level flight, no elevator deflection should be required to maintain level flight. This meant that the aircraft's aerodynamic center should be placed relative to the aircraft's CG such that the moment from the aircraft's lift would exactly cancel the wing's nose down moment, provided that the tail was producing no lift. This can be achieved by examining the equations of motion. Dividing the steady, level flight equations of motion by the dynamic pressure and substituting in the stability derivatives for the aerodynamic coefficients yields

$$\begin{aligned} C_T - C_{D0} - KC_L^2 - (W/\bar{q}S_W)\gamma = 0 \\ C_{Lo} + C_{L\alpha}\alpha + C_{L\delta_E}\delta_E - W/\bar{q}S_W = 0 \\ C_m = C_{mo} + C_{m\alpha}\alpha + C_{m\delta_E}\delta_E = 0 \end{aligned} \qquad 5.1$$

For level flight, $\gamma = 0$ and the equations can be solved for α and δ_E:

$$\begin{aligned} \alpha &= \frac{C_{m\delta_E}\delta_E(C_L - C_{Lo}) + C_{L\delta_E}C_{mo}}{C_{L\alpha}C_{m\delta_E} - C_{m\alpha}C_{L\delta_E}} \\ \delta_E &= \frac{-C_{L\alpha}C_{mo} - C_{m\alpha}(C_L - C_{Lo})}{C_{L\alpha}C_{m\delta_E} - C_{m\alpha}C_{L\delta_E}} \end{aligned} \qquad 5.2$$

where α equals the angle of attack in radians and δ_E equals the elevator deflection angle in radians. From these equations, the angle of attack and the elevator deflection for steady level flight can be calculated for each mission. Setting the $C_{m\alpha}$ equal to $-C_{L\alpha}$ times the static margin, the deflection angle of the elevator can be written as

$$\delta_E = -\frac{C_{mo} + (\bar{X}_{cg} - \bar{X}_{ac})(C_L - C_{Lo})}{C_{m\delta_E}} \qquad 5.3$$

This allows the placement of the aircraft ac relative to the CG. Since the position of the tail is known, the wing can therefore be placed on the fuselage such that $\delta_E = 0$ for the expected cruise condition. This requires a fairly precise determination of the moment coefficient of the airfoil; which the NASG or UIUC did not have. Without a better alternative, the team used the software code Javafoil to predict the moment coefficient. To place the wing, the position was first estimated (as in preliminary design) and put into the weight and balance CG sheet. Equation 5.3 was then solved for the position of the wing so that

the elevator deflection needed at cruise was zero. The new position of the wing was then put back into the CG spreadsheet to get the new CG. This process was iterated until convergence and then the aircraft's dynamic longitudinal stability was checked again.

5.1.4 Control Surface Sizing

A large elevator are was needed in order to allow the plane to rotate on takeoff because the fuselage was relatively short and the horizontal tail was smaller than ideal because the contest restricted the tail span to one quarter of the wingspan. Thus, the elevator was made to span the entire horizontal tail in order to provide the necessary nose-up moment on takeoff. Even with the entire tail span elevator, the aircraft should not be overly sensitive to small elevator deflections due to the large static margin of the aircraft. This was verified during flight testing.

The rudder for the aircraft was designed to be along the entire span of the vertical tail since our pilot preferred to fly with the rudder. The bottom of the rudder was cut at an angle so that it would not interfere with the elevator, even at maximum elevator deflection.

The ailerons were sized using aircraft from previous competitions for guidance so that the aircraft would have controllable roll response. If the ailerons were too big, then the aircraft would be overly sensitive to small deflections at cruise velocity, but if the ailerons were too small the aircraft would be in danger during takeoff because of the slow velocity. Therefore using history as a guide, the ailerons were made to span the outer half of the wing.

5.2 Systems Architecture

Most component analysis is discussed in the Manufacturing Plan, Section 6.0. Aspects of the aircraft warranting further detail in its operation and assembly of the aircraft are discussed below.

5.2.1 Connection of Wing / Fuselage / Tail Assembly

The wing and tail assemblies are connected to the fuselage via a series of rubber-bands. The rubber bands wrap around two brass rods that extend 0.5" from either side of the fuselage and are permanently fixed to the structure of the aircraft. The front brass rod is through the balsa support for the motor on the fuselage assembly slightly in front of the wing carry throughs. The rear brass rod is through the support pieces for the tail assembly and is very near to the water container. The wing rests on top of the water container between the two rods. The wing carry throughs were designed to be the major structural component attaching the wings to the fuslage. By pulling a rubber band from the front rod over the wing carry-through structure and finally back to the rear rod, all three assemblies are held together at once.

Three heavy-duty rubber bands on each side of the payload box were determined to be the minimum number to be able to properly hold the structure together.

5.2.2 Wiring / Servos / Receiver Placement

To minimize the lengths of the servo wires and to keep the center of gravity directly over the center of the water container, the receiver was placed in the tail assembly as far forward as possible. The rudder and elevator servos were also placed at this location and were directly connected to the receiver. Only the wing servos and motor controllers required wire extensions to reach the receiver. The motor controller wire extension and wing wire extensions fit easily between the water box and overhead hatch to the receiver at the tail assembly of the aircraft. Nylon push rods were connected between the rudder and elevator servos and their corresponding control surfaces.

5.2.3 Water Dispensing Mechanism

The issue of dropping the water was deceptively difficult. Although the concept of pulling a plug sounded easy, in practice the team encountered several difficulties. The first task was to decide on a method. One method discussed was to slide a piece of wood away from the hold. Another method was to plug the hole and pull the stopper with a rod connected to a servo. A third method was to use a flexible tube which would initially be turned so that the exit was above the waterline, but then to drop the water the tube exit would be rotated.

To determine a final solution each method was built to a rough approximation of the concept. It was determined that the sliding method leaked excessively, while the flexible tube created problems with the amount of free space required to rotate a six inch section. It was thus determined that a plug and servo combination provided the best method for success.

The final step involved in the design of the water container was the problem of dealing with the sloshing water during critical periods such as take-off and turning. To reduce the impact of the sloshing it was determined that it was necessary to reduce the amount of distance the water could travel before reaching a surface. The solution was to install baffles inside the water container in evenly spaced intervals. Testing before and after the insertion of the baffles showed marked improvement in the reduction of sloshing.

5.3 Aircraft Configuration

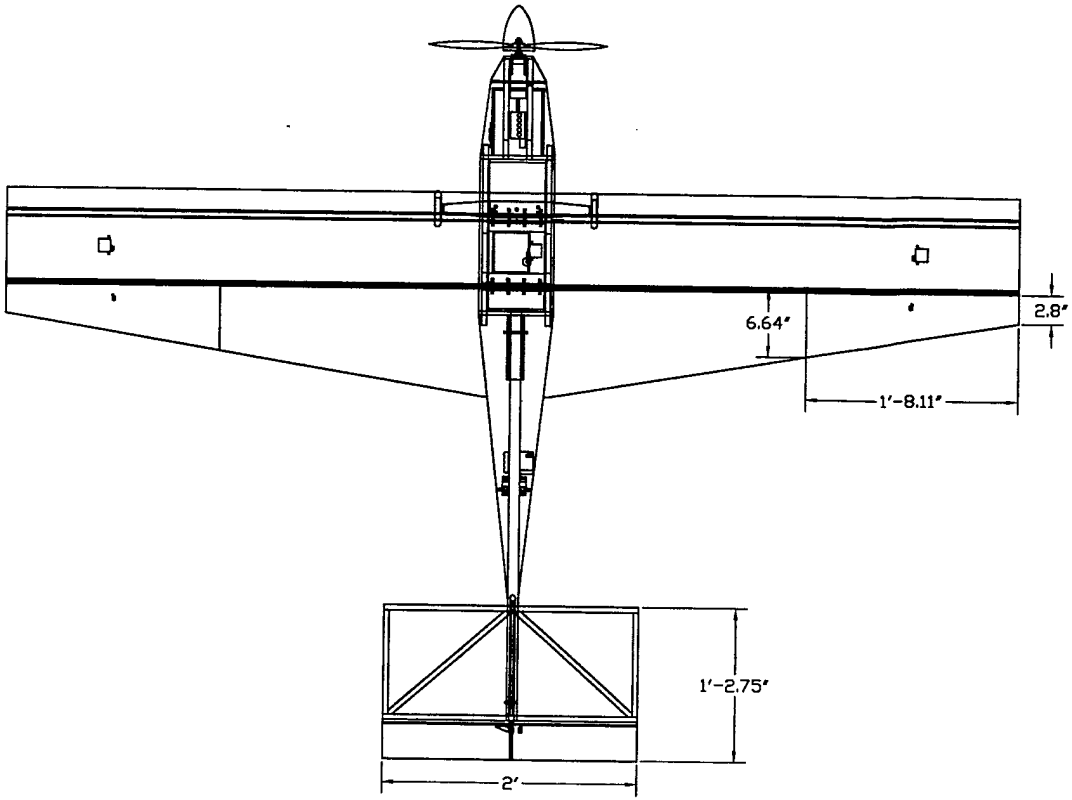
5.3.1 Final Configuration

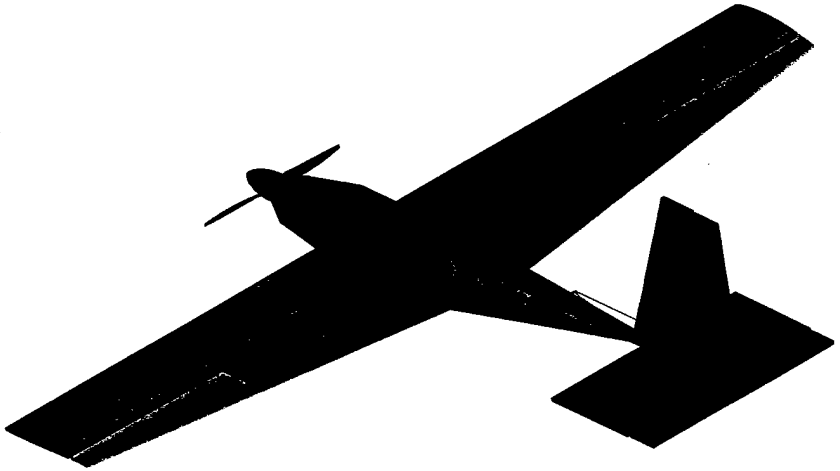
The aircraft configuration as of writing the paper is shown in Table 5.4 and in the following drawing package. Updated performance estimates with the most recent weight and propulsion characteristics

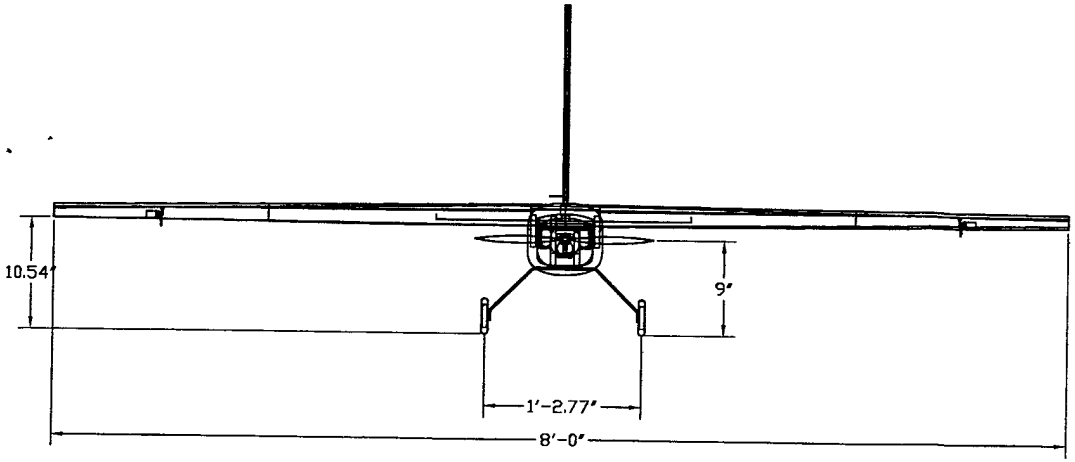
were calculated in the same manner as was done in the Preliminary stage. Several changes are being made as a result of further analysis and flight tests which will lighten the aircraft. These are discussed in the future improvements section.

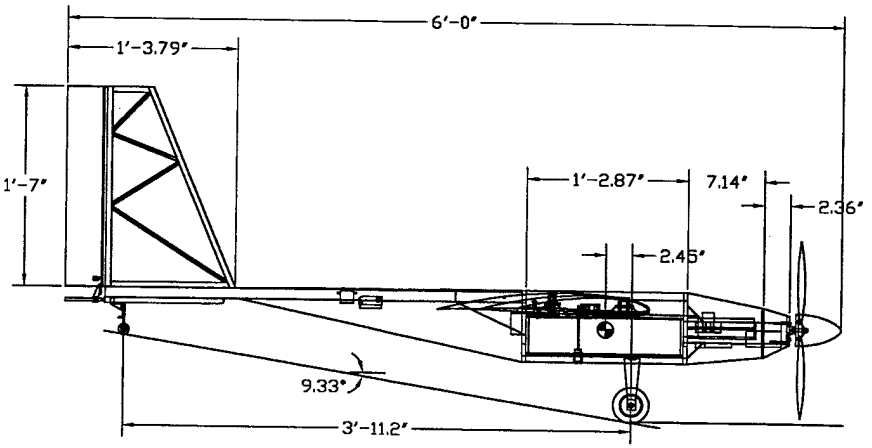
Table 5.4: Final Aircraft Data.

GEOMETRIC DIMENSIONS	
Fuselage Length	5.9 ft
Fuselage Height	0.59 ft
Fuselage Width	0.56 ft
Wing Span	8.0 ft
Mean Aerodynamic Chord	1.33 ft
Taper Ratio	0.62
Wing Area	10.64 ft
Aspect Ratio	6
Aileron Area (total)	1.32 ft^2
Elevator Area (total)	0.54 ft^2
Rudder Area (total)	0.44 ft^2
PERFORMANCE	
Wing Airfoil	SG 6042
$C_{L,max}$	1.4
L/D_{max}	13.1
Maximum Rate of Climb (loaded)	9.4 ft/sec
Stall Speed (loaded)	45.8 ft/sec
Cruise Speed (empty)	76.3 ft/sec
Cruise Speed (loaded)	74.3 ft/sec
Takeoff Field Length (empty)	107 ft
Takeoff Field Length (loaded)	142 ft
Total Predicted Flight Score	**0.52**
WEIGHT STATEMENT (lb)	
Airframe	6.9
Propulsion System	1.4
Propulsion Batteries	2.9
Control System	0.3
Payload System	0.4
Payload	8.8
Empty Weight	**11.9**
Loaded Weight	**20.7**
MISCELLANEOUS SYSTEMS	
Motor	AstroFlight Cobalt 640S
Gear Ratio	3.1 : 1
Batteries	21 SR 2400 Max in series
Propeller	18 x 12 APC Electric
Radio Controller	Futaba PCM 1024
Servos	Hobbico CS-35MG



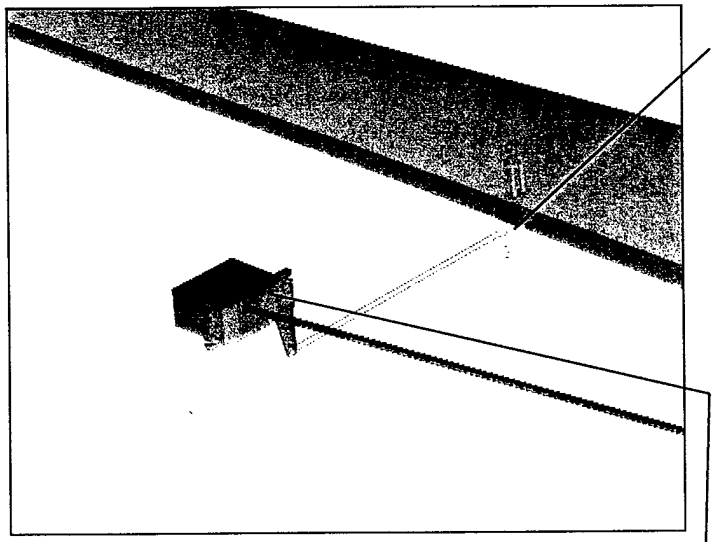




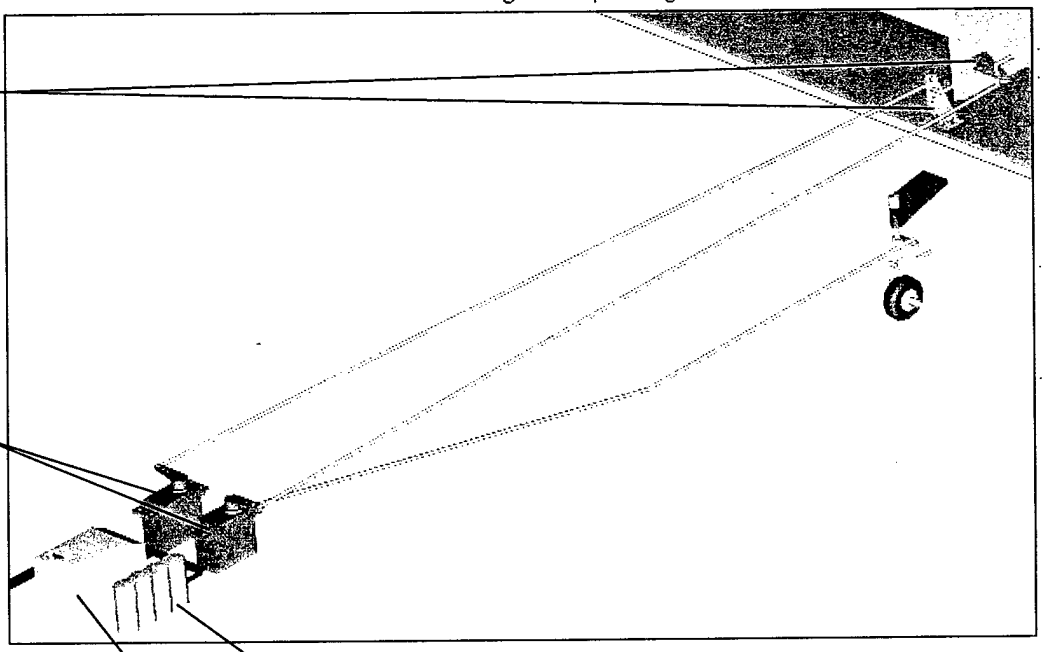


UT DBF 2004
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Scale: 1" = 1'-0"

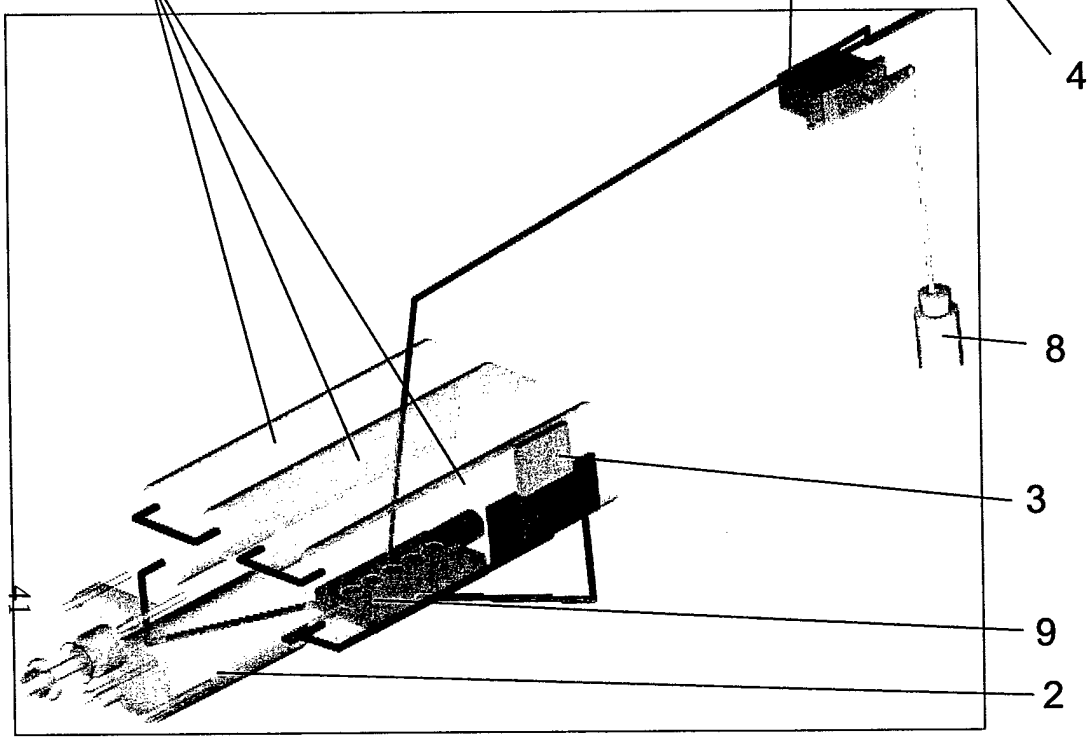
Right Wing



Aft Fuselage / Empennage

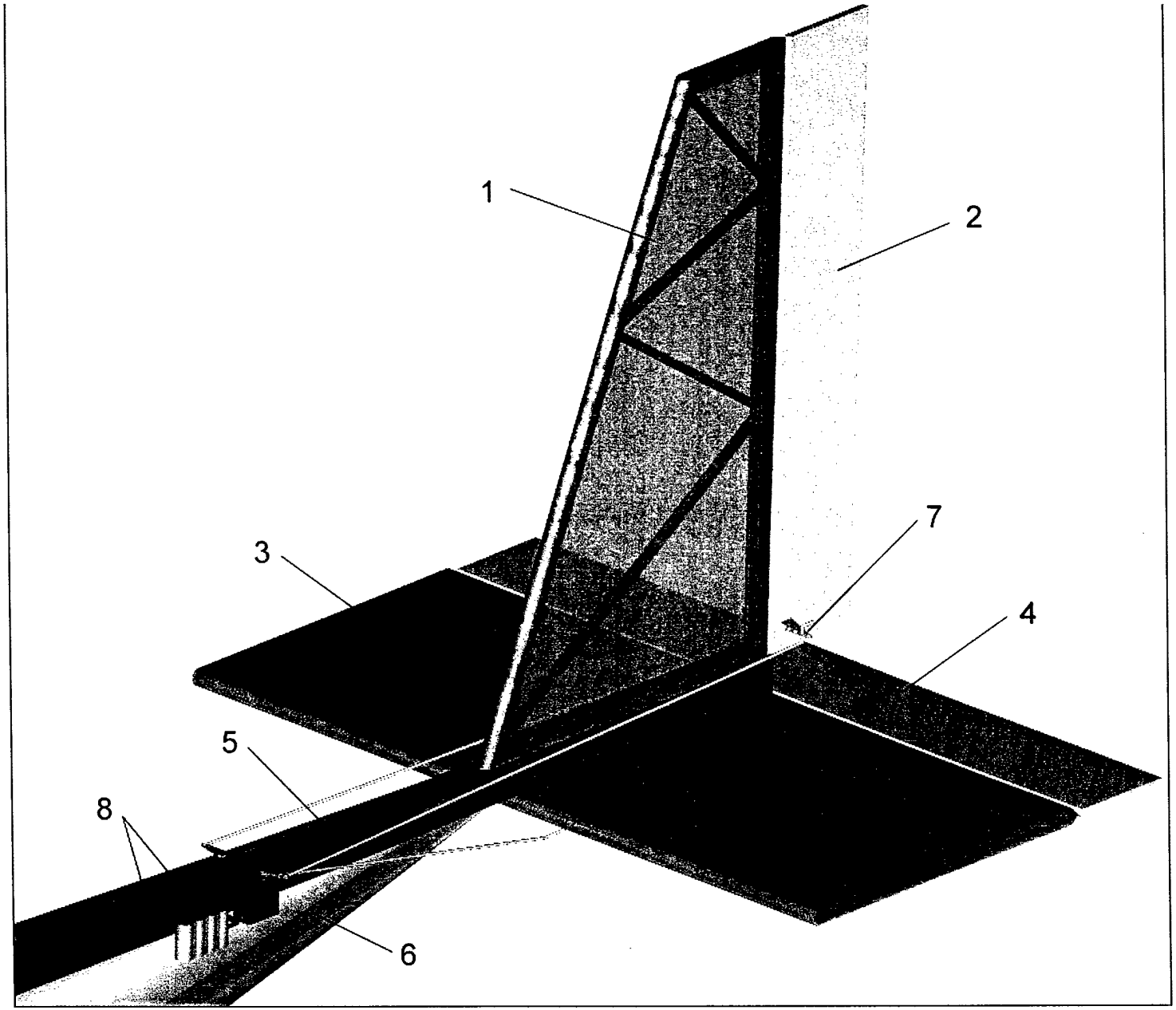


Nose / Mid-Fuselage



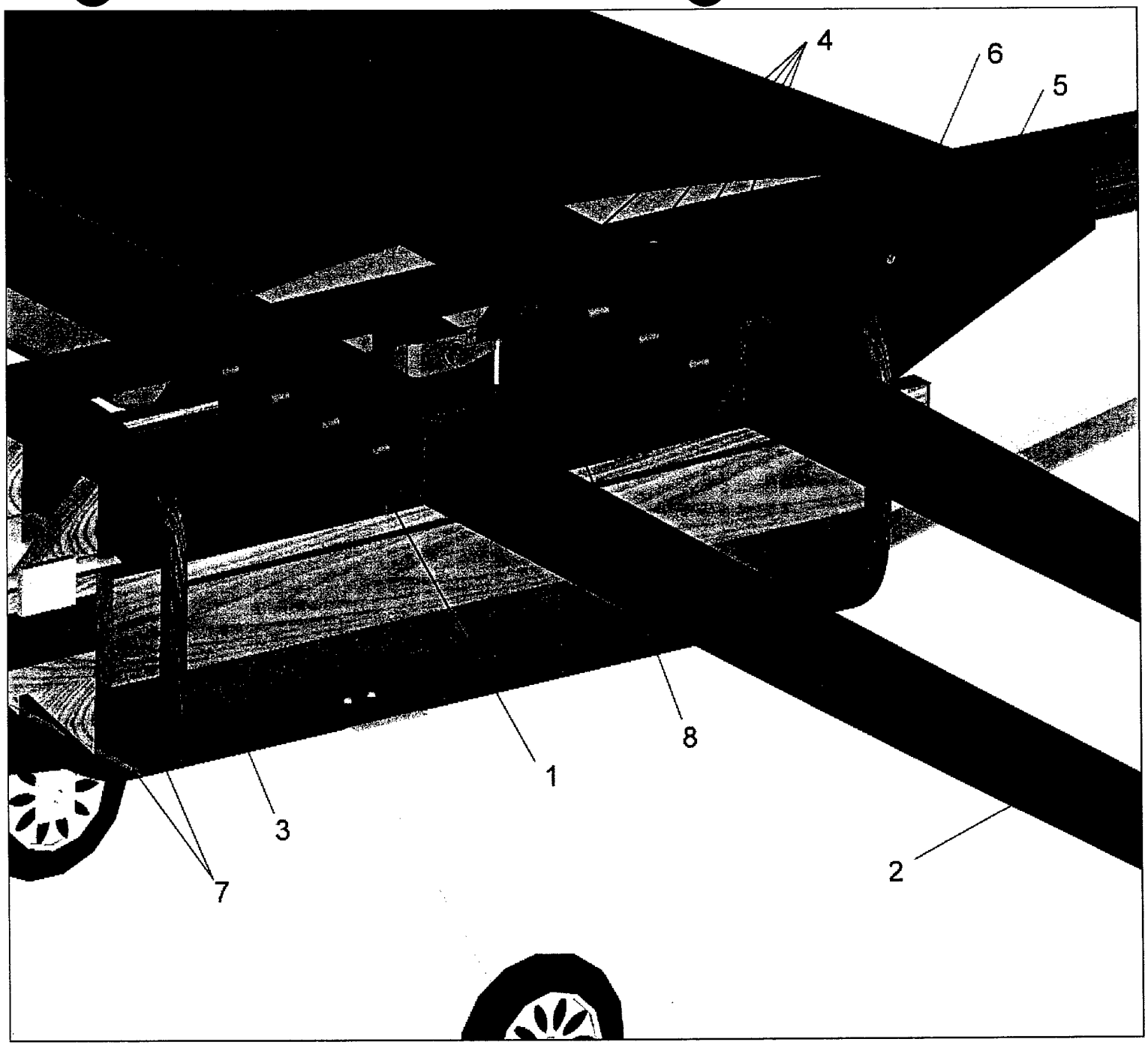
Systems Architecture	
1	Battery
2	Motor
3	Fuse
4	Receiver
5	Receiver Battery
6	Servo
7	Control Horn
8	Water Plug
9	Motor Controller

UT DBF 2004
Dwg #: 2 of 5
N.T.S.



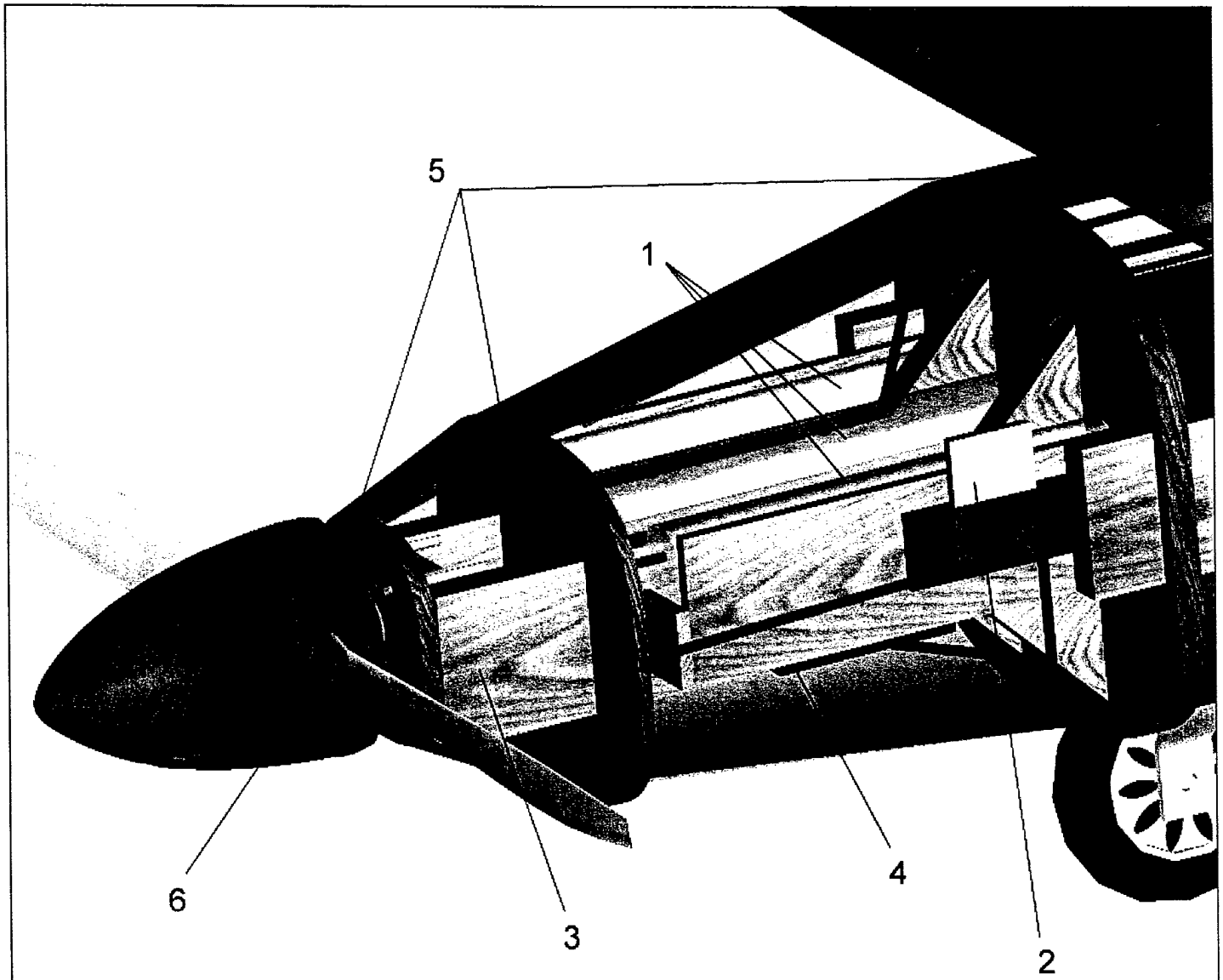
Empennage Design	
1	Vertical Stabilizer
2	Rudder
3	Horizontal Stabilizer
4	Elevator
5	Longeron
6	Styrofoam Shaping
7	Control Horn
8	Electronic Equipment

UT DBF 2004
Dwg #: 3 of 5
N.T.S.



Mid-Fuselage Design	
1	Wingbox
2	Laminated Spars
3	Water Tank
4	Securing Bolts for Wings
5	Tail Connection Box
6	Securing Pin for Tail
7	Support Beams
8	Lid for Water Access

UT DBF 2004
Dwg #: 4 of 5
N.T.S.



Nose Design	
1	Battery
2	Fuse
3	Motor
4	Motor Controller
5	Bulkheads
6	Spinner / Propeller

UT DBF 2004
Dwg #: 5 of 5

5.3.2 Weight and Balance Sheet

In order to estimate the CG position of the plane, a weight and balance spreadsheet was created shown in Table 5.5. The total empty weight of the plane calculated from the individual parts was 12.1 pounds, in good agreement with the scale measurement for the entire plane, which was 11.9 pounds. The extra weight is most likely due to uncertainties in the weights measured. The CG position of the fire bomber configuration and the ferry configuration was the same because the placement of the water container was iterated until the water's CG was on the CG of the whole aircraft. To more accurately determine the aircraft's CG, the load on both the nose gear and main gear were measured with two scales so that the weight fraction on each could be measured. The CG found in this way contained less error than determining the CG with the individual components as the balance spreadsheet does, and so it is reported in Table 5.5 as the actual CG position. To determine where the most possible weight could be saved, the weight breakdown of the current aircraft in terms of percentages is shown in Figure 5.2. Notice that the payload weight fraction is higher than estimated in the conceptual design phase.

Table 5.5: Weight and Balance Spreadsheet.

	Weight (oz)	**Position (in)**
Empennage Assembly	25.1	49.5
Front Fuselage Assembly	37	17
Wings	46	13
Propulsion Batteries	47	6
Motor Controller	1.9	6
Motor	17.5	1.5
Propeller/Spinner	4.3	-1.75
Main Landing Gear	8	15
Tail Gear	3	64
Reciever & Reciever Battery	3.4	45
Payload	140.8	17
EMPTY WEIGHT (lbf)	**12.1**	
EMPTY DECOY CG (in)	**16.8**	
LOADED WEIGHT (lbf)	**20.9**	
LOADED CG (in)	**16.9**	
ACTUAL EMPTY CG (in)	**16.95**	
ACTUAL LOADED CG (in)	**17.00**	

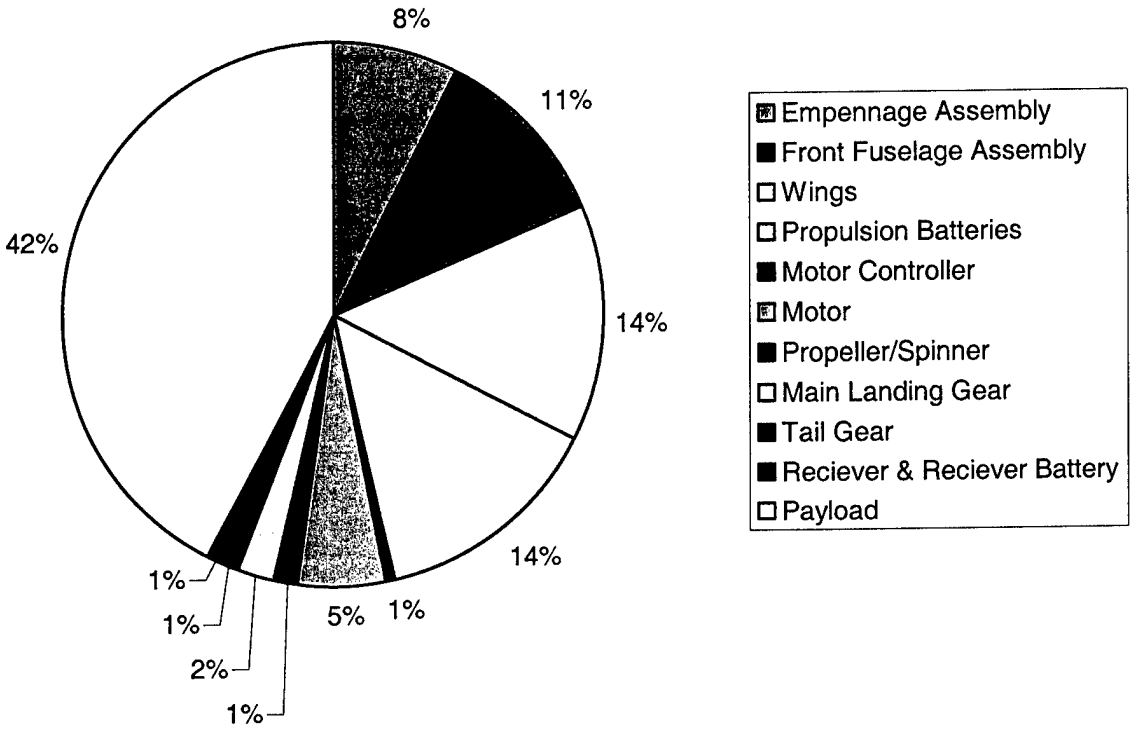


Figure 5.2: Component Weight Percentages.

5.3.3 RAC Calculation

An aircraft cost model, or Rated Aircraft Cost, is given in the contest rules and is calculated in the following manner:

$$\mathrm{RAC} = \frac{(300 * \mathrm{MEW} + 1500 * \mathrm{REP} + 20 * \mathrm{MFHR})}{1000} \qquad 5.4$$

where MEW is the Manufacturers Empty Weight, REP is the Rated Engine Power, and MFHR is the Manufacturing Man Hours; all of which are defined in the contest rules. Table 5.6 shows the calculated RAC for the final aircraft design.

Table 5.6: RAC breakdown.

Description	Monoplane
MEW	**11.9**
REP	**2.9**
# of Engines	1.0
Total Battery Weight (lb)	2.9
MFHR	**224.0**
Wings	**135.0**
Wing Span (ft)	8.0
Max Chord (ft)	1.6
Control Function Multiplier	1.0
# of Wings	1.0
Fuselage	**39.0**
Fuselage Length (ft)	5.9
Fuselage Width (ft)	0.6
Fuselage Height (ft)	0.6
Empenage	**20.0**
Vertical Surfaces w/no control	0.0
Vertical Surfaces w/ control	1.0
Horizontal Surfaces	1.0
Flight Systems	**30.0**
# of Servos	5.0
# of Motor Controllers	1.0
TOTAL RATED AIRCRAFT COST	**12.5**

5.4 Ongoing Aircraft Optimization

Following the successful flight by the aircraft, the team began investigating ways to improve the aircraft's performance. The team began construction of the second generation aircraft the week this paper is due. Several improvements to the plane were incorporated and are discussed below. Also, further study was done to determine if another motor configuration would perform better than the current configuration.

5.4.1 Second Generation Aircraft

The second generation aircraft had several improvements and time saving construction methods incorporated. Examples include changing the empennage design from built-up balsa ribs to a solid Styrofoam section with a single basswood spar longeron, and a simplification of the motor mount structure. The wing attachment received a major overhaul, from removable wing boxes held down by

rubber bands to a fixed, tight-fitting wing box. The center of gravity was also more precisely placed over the center of the water container to improve the aerodynamic properties while the water emptied from the aircraft. The current wings on the plane are solid Styrofoam; however, they were originally supposed to be ribbed to save weight. Unfortunately, the ribs failed the initial strength tests and took excessive time to build so that the solid core wings were built instead. The lite-ply laminated balsa spars worked so well in initial testing that they were incorporated into all future wing designs. To give a little more room for wing insertion in the wing boxes an additional half inch was added to the spars. Finally, the tail was adjusted to maintain the zero elevator deflection in cruise to compensate for the new wing location. The overall weight savings between the two aircraft was just over a pound, therefore justifying the additional design work involved in making the optimizations.

The tail assembly was originally balsa ribs with two vertically spaced longerons connecting the ribs together. Although the concept created a lightweight structure there were a couple of issues which warranted a redesign. First, due to the coating in-between the ribs there were noticeable ridges in the tail, leading to concerns about unnecessary drag. Second, the time required for construction (on the order of several weeks) was deemed unacceptable. The solution was to create a solid Styrofoam section with a single stiffening beam inserted into the center. The beam attaches the tail to the fuselage via a built-up slot. The overall construction time was reduced to less than a day with virtually no loss in performance.

As test flights continue, the wing placement relative to the aircraft cg and the horizontal tail incidence will be adjusted so that almost no lift and no elevator deflection is needed from the tail for steady level flight at cruise conditions. This should significantly reduce the drag of the aircraft at cruise conditions because the horizontal tail, with its low aspect ratio, inefficiently produces lift. This adjustment is needed because the airfoil moment coefficient obtained through Javafoil is inaccurate and to a lesser extent because the aircraft's expected cruise velocity may be wrong due to the inaccuracies of the drag model used. Changing the wing position relative to the horizontal tail will change the static margin; however, the aircraft's current static margin is actually a bit too high, so decreasing it is not expected to be a problem.

5.4.2 Motor Optimization

The first area of improvement investigated was the purchase of a new motor. The rules of the competition require the motors be brushed Astroflight or Graupner motors; however, since the Graupner motors are difficult to obtain, they were not considered. The team investigated 7 additional Astroflight motor/gearbox combinations: the Astroflight 625S, 625G, and 625; the 640G and 640; the 660 and 661. These motors with their respective gearboxes were examined with the aid of the commercial programs Electricalc and Motorcalc. When the two programs were not in consensus, Electricalc's values were used because previous static tests had shown that Electricalc's estimate was generally more accurate.

The analysis for each motor/gearbox combination was performed by iterating over many different combinations of propeller geometry and number of batteries. Propellers with diameters between 10 and 20 inches and pitches between 5 and 20 were examined, as well as between 17 and 23 batteries. An individual configuration was discarded if it could not provide at least 80% of ideal static thrust (5.6 lbf) and 85% of pitch velocity (47 mph). The thrust cutoff was 80% because the combination might make the aircraft lighter and thus require less thrust for takeoff, and when compared with propeller tests, the predictions were typically 10% or more below the actual value. It was found that the 625 direct drive motor configurations were unable to produce the required thrust under 40 amps and that the 660G motor configurations needed more than 25 cells to deliver the required thrust. The table below shows properties of the best configurations.

The analysis showed that the Astroflight 625S, 625G, and 625 was not powerful enough to meet the aircraft requirements, as expected. However, the analysis did show significant promise in the Astroflight Cobalt 660 motor with 1:1 gearing even though the motor weighs 4 oz more than the current motor. Several different configurations for this motor could, in theory, provide the necessary thrust and pitch speed, and reduce the number of battery cells. Table 5.7 shows the various possible configurations. The two most attractive propeller configurations are shown in bold and are the 16x10 propeller with 18 cells and the 16x12 propeller with 18 cells. Both combinations will be tested when the motor arrives. The 17x12 propeller would be a good choice as well, but the configuration lifetime gives less room for error on the flight time estimation.

Table 5.7: Astroflight Cobalt 660 Configurations Examined.

Number of Cells	Diam (in)	Pitch (in)	Thrust (lbf)	Pitch Speed (mph)	Lifetime (min)
17	16	10	7.0	49.0	4.7
17	16	12	6.9	57.0	4.2
17	17	12	7.8	53.0	3.8
17	18	12	8.4	50.0	3.4
18	**16**	**10**	**7.6**	**51.0**	**4.3**
18	**16**	**12**	**7.6**	**59.0**	**3.9**
18	17	12	8.4	56.0	3.5

5.4.3 Number of Batteries

The RAC of the aircraft is strongly influenced by the number (weight) of batteries that the aircraft requires. In light of this, the majority of the team's optimization up to this point was focused on the reduction of the number of battery cells; however, it is possible that the addition of another cell or two for the current motor (640S) would increase the thrust sufficiently (and therefore decrease the flight time) to justify the additional batteries. Therefore, propeller tests are still being run and flight tests performed with different amounts of cells to determine the best number of cells.

6.0 MANUFACTURING PLAN

The design and construction of the UT-DBF aircraft was highly influenced by previous year's experience but also included several significant innovations. For the initial aircraft, aspects such as simple construction and design were judged more important than other factors. After flight testing however, these pieces were redesigned to reduce weight and increase efficiency.

6.1 Figure of Merits

A figure of merits system was developed to determine how the aircraft's components would be designed and built. These merits included performance advantages, weight estimation, construction skill, and construction time. Although the cost of the materials is a concern, the variance of price for the designs was not significant enough to warrant a figure of merit. The figure of merits and the designs discussed are in Table 6.1.

6.1.1 Performance

The performance of the component was considered the most important figure of merit. How the component may affect the aircraft's drag and controllability is taken into account for performance. Also, performance is considered in how the design technique adversely affects and connects to other components of the aircraft. This figure of merit was given a weight of 3.

6.1.2 Weight Estimation

Due to the restrictive takeoff requirement, saving weight became a critical concern for the aircraft. Weight estimations were considered for the component designs and were given a figure of merit weight of 3.

6.1.3 Construction Skill Required

Most members of the team were previous DBF veterans and were thoroughly experienced in the typical construction techniques. New designs however were presented this year that warranted the need for a construction skill figure of merit. If the new design is too complicated, it may actually be less effective than the older proven techniques. This figure of merit was given a weight of 2.

6.1.4 Construction Time Required

As with any project, extensive amounts of time could not be devoted to minor components. However, the more time spent on an individual part typically corresponds with higher quality. A balance had to be found between the two when choosing component designs. For these reasons, a figure of merit with a weight of 1 was given to the construction time required.

6.2 Manufacturing Processes Investigated

Although all of the aircraft's components received design attention, some parts needed extra consideration and discussion. Those particular components are listed below and had the figure of merits system applied to them.

Table 6.1: Manufacturing Plan Figure of Merit.

	Option	Time Required (x1)	Skill Required (x2)	Performance (x3)	Weight Estimation (x3)	Final Score
Fuselage	**Solid Foam**	1	1	0.7	0.8	**7.5**
	Built-Up	0.4	0.5	1	1	7.4
	Foam Ribbed	0.7	0.7	0.6	0.9	6.6
Water Tank Valve	**Plastic Plug-Drain**	1	1	0.7	0.8	**7.5**
	Time-Release Valve	0.7	0.3	0.7	1	6.4
	Turn-Release Valve	0.9	0.6	1	0.4	6.3
Wing Materials	**Full Foam**	1	1	1	0.8	**8.4**
	Foam Ribbed	0.7	0.9	0.5	1	7
	Built-Up	0.6	0.8	0.9	0.9	7.6
Wing Sweep / Spar	**Swept Forward - 2 Spars**	1	1	1	1	**9**
	Unswept	0.8	0.8	0.7	0.8	6.9
	Swept Back	0.6	0.7	0.4	0.4	4.4
Spar Materials	Simple Balsa Sticks	1	1	0.7	0.9	7.8
	Laminated Balsa Sticks	0.9	0.9	1	1	**8.7**
	Fiberglassed Balsa Sticks	0.7	0.8	0.9	0.8	7.4
	Aluminum Rods	0.7	0.7	0.7	0.5	5.7
	Carbon Rods	0.4	0.4	0.8	0.9	6.3
Wing Attachment	Wingbox	0.6	0.9	1	0.8	7.8
	Rubber-Bands	1	1	0.9	1	**8.7**
Tail Design	Airfoil Shaped	0.8	0.7	1	0.9	7.9
	Balsa Frame	1	1	0.8	1	**8.4**
Empennage Design	Aluminum Rod	0.9	0.8	0.7	0.7	6.7
	Torsion Box	0.7	0.7	1	0.7	6.9
	Two-Piece Angled Balsa	0.8	0.7	0.9	0.9	7.6
	Single Balsa Stick	1	1	0.8	1	**8.4**

6.2.1 Fuselage Construction

The manufacturing of the fuselage for the desired shape and strength had to be analyzed. Shapes for the fuselage included an airfoil shape, a cylindrical shape, and also a general streamlined shape. The airfoil shape was dismissed because of aerodynamic reasons. The cylindrical shape could be constructed by two methods. One consisted of cutting styrofoam into the round shape and then sanding down the front and rear sections for a smooth transition. The second method involved creating many circular balsa ribs that would be wrapped with monokote for a smooth shape. Since the cross-sectional area for the motor mount, payload box, and control surface attachment were all fixed, a general streamlined shape matching these dimensions would fit better than a set cylindrical shape. Styrofoam could easily be shaped to fit the

transition between these components. The general streamlined shape was thought to be the easiest to construct.

6.2.2 Water Tank Valve

The design of the water tank itself seemed very straightforward to all members of the team: a simple styrofoam structure with monokote waterproofing. The actual release mechanism, on the other hand, was not as simple of a choice. Three options were investigated that included a plastic plug-drain method, a time-release valve, and a turn-release valve. The plastic plug-drain would consist of a commercial bought stopped plug and its corresponding open piece. A servo mounted above would then pull the plug to release the water. Very easy to do but had poor performance because you could not 're-plug' the hole in flight. The time release valve method was based off the idea that the water would soak a thin membrane during takeoff just enough that it could be ruptured by a high G maneuver later in flight. Although this is by far the lightest method, it would require extensive testing and uncertainties are rampant. Turn release valves offer the best performance because they can be closed, but all were found to be far too heavy for our purposes.

6.2.3 Wing Materials

Several wing construction designs were investigated including a full foam wing, a foam ribbed wing, a balsa ribbed wing, and a built-up combination wing. The full foam wing was by far the easiest and quickest to construct but also potentially the heaviest. A foam-ribbed wing is very light but has a greater workload. A new idea for this year was the built-up combination wing that consisted of a foam leading edge and balsa sheets for the remaining area of the wing. The combination wing was believed to be the most complicated and also most time extensive design. Its promise of weight savings was debatable but very possible.

6.2.4 Wing Sweep / Spar

The spar through the wings was also a major factor in the design of the wing. The aircraft's wing could be swept forward, back, or unswept without a significant change in aerodynamics, but with major structural considerations. In the swept-forward arrangement of the wing, a primary heavy-duty spar could be located near or slightly forward of the average quarter-chord and a second smaller spar could be located closer to the rear. Since the aircraft's wings taper down, the smaller rear spar have to taper accordingly through the entire length of the wing. The swept back arrangement of the wing would be the most difficult to handle structurally because the quarter-chord would be angled back and its cross-sectional area taper down substantially. Multiple spars, including possibly angled spars, would be needed to be able to support this structure, and the swept back arrangement would result in problems with the wing spar to

fuselage connection. The unswept wing would share some problems of the swept back wing because the leading edge of the wing still sweeps back, resulting in a shifting quarter-chord.

6.2.5 Spar Material

The considered materials for the spars included simple balsa sticks, fiberglassed balsa sticks, aluminum rods, carbon rods, and lite-ply/balsa laminated spars. The simple balsa sticks were the easiest to buy and taper if necessary for the spar. Fiberglassing the spar greatly increases its strength but also the weight and construction time. Aluminum rods are the most restrictive in terms of design. Metal work cannot be machined easily so the wing would practically have to be designed around the spar. Carbon rods have the potential for great strength with low weight, but for the high quality of construction needed, the carbon is a questionable material to use successfully. The lite-ply/balsa laminate provided the greatest torque protection and added to the ultimate strength.

6.2.6 Wing Attachment

The method of attachment of the wings involved unique difficulties. To not affect the center of gravity of the aircraft, the payload box had to be placed underneath the location of the wings. A high wing was decided for stability, but a simple way to access the payload box was not an easy challenge. One idea investigated was a wingbox that the wings' spars slid into which had attachment tabs to hold the wingbox onto the fuselage. A second idea involved a technique in which part of the wing's spar would slide into a reinforced connection in the fuselage above and on either side of the payload. To stiffen the structure sufficiently, a fixed carry-through box would be located over the payload that attached to both wings. The last idea was actually a method used by many radio-controlled modelers. High-strength rubber bands would be strapped over the wing spars and wrapped around pins forward and aft of the wings. A fixed carry-through box connected to both wings would be necessary for the rubber bands to function properly.

6.2.7 Tail Design

A couple techniques were studied for the vertical and horizontal surfaces. The traditional foam symmetrical airfoil over balsa spar technique was a proven UT-DBF method. A rectangular balsa frame with monokote covering resulted in a simple flat plate design. Although not aerodynamically the best solution, this idea was also considered.

6.2.8 Empennage Design

The only major manufacturing difficulty remaining was the design of the aircraft's empennage. The tail had to be removable to fit within the contest's required packaging box. Due to the long distance required for the light-weight support from the payload box to empennage, flutter became a concern in the design.

A single aluminum rod extending back, a torsion box, vertically spaced balsa pieces, and a single basswood spar were all considered. The aluminum rod and torsion box would not need to taper down but would be more difficult in attaching to the central fuselage. The angled balsa pieces would be able to taper from the required height of the payload box down to a reasonable area for the control surfaces to attach to, but creating a precise angle with the needed tolerances was considered very challenging. The single basswood spar was stiffer than the balsa and was easiest to manufacture. At the wide payload box the structure would be easy to connect because it would be able to fit into a fitted slot.

6.3 Final Manufacturing Process

Items highlighted in Figure 6.1 show the techniques that were attempted. The test aircraft generally used the more experimental processes. New techniques included a built-up fuselage for the best performance and weight, built-up wing for the best weight, and the one piece Styrofoam-balsa structure for the empennage for its high performance and weight. Rubber bands, proven last year, were chosen as the attachment method of the wings. Thus, the final design incorporated a solid foam fuselage and wing, a wing box attachment with rubber bands, and a single stick for the empennage design. The following is detailed manufacturing instruction used for the final version of the aircraft. Table 6.2 displays the schedule for the final aircraft construction.

6.3.1 Front Fuselage

The fuselage was one of the more complex portions of the aircraft and the first to be built. The entire design was done within AutoCAD 2004 first to draft templates for the styrofoam cutouts. The templates took into account supporting balsa sticks and plates that would carry the load. The motor was mounted onto a 1/4" poplar bulkhead because of vibration concerns. Two balsa wood sticks .5" x 2" extended from the motor bulkhead back through two additional bulkheads. All attachments were made using the West System's Resin and Hardener epoxy. The second bulkhead, constructed with a 1/4" sheet of balsa wood acted as a shaping member and support for the batteries. The third bulkhead acts as the front support of the water tank. Balsa sheets were cut to form battery compartments between the second and third bulkheads. The third bulkhead was a 1/4" balsa sheet cut to shape and laminated with 1/8" liteply. Triangular pieces were made to further support the two balsa sticks to the third bulkhead. Styrofoam pieces were cut to finish the shape of the nose section of the fuselage. Top Flite's Econokote plastic covering was hot-ironed onto the styrofoam for a smooth finish.

6.3.2 Mid-Fuselage / Water Tank

The water tank was made out of a solid block of styrofoam that was hot-wired into an aerodynamic outer shape with a open funnel shape on the inside for the water. Econokote was applied over the styrofoam to act as waterproofing. The water tank was housed between the third and fourth fuselage bulkhead. Two

1/2" x 2" balsa support beams and a 1/8" x 5.5" lite-ply plate connected the two bulkheads. The balsa beams were mounted high on the water tank and were used for mounting the wings. A lite-ply plate on the underside of the tank acted as a support for the water and was epoxied to both bulkheads and the water tank. The release mechanism for the water consisted of a plug-drain technique. A commercially purchased PVC cylindrical piece with a hole was mounted through the lite-ply plate and a small 1/2" diameter brass piece was used to restrict the water flow. To stop the water, a second plastic plug was purchased that would fit within the open plastic piece. The lite-ply plate also acted as a mounting point for the landing gear discussed later. The cover for the water tank consisted of a second 1/8" lite-ply plate with a hole to allow access to the water. Balsa sticks measuring 1/2" x 3/4" were epoxied around the opening to help mount a lid and the release servo. The release servo was attached to a piano-wire rod that lowered to the plastic plug below. By rotating the servo the plug would be pulled and the water released. The top hatch was also made with styrofoam by the hot-wire technique. The entire exterior of the tank and the top hatch were then monokoted for smooth airflow.

6.3.3 Rear Fuselage

To attach the removable tail onto the rest of the aircraft, a triangular shaped mount was constructed from balsa wood and lite-ply. The mount was then epoxied onto the fourth fuselage bulkhead. The shape of the tail was cut with a single block of styrofoam that tapered from the shape of the fourth fuselage to the 1" x 1" supporting longeron spar. Two servos and the receiver equipment were placed within the styrofoam of the rear fuselage. A single bolt, which also doubles as a rubber band anchor is used to secure the 1" x 1" supporting longeron of the tail to the triangular mount attached the rest of the aircraft. The entire structure was then covered in monokote.

6.3.4 Empennage

Both the vertical and horizontal surfaces were built similarly. Balsa sticks were epoxied together to form the frame of each surface, and smaller crossways balsa sticks were glued to stiffen the frame. A half-cylindrical shaped balsa stick was added to the front of the frame as the leading edge. The actual control surfaces were thin balsa sheets cut to shape. To cover the vertical and horizontal surfaces, monokote was ironed onto the frame. A heat-gun was used to tighten the covering. The control surfaces were hinged onto the vertical and horizontal stabilizers with small strips of monokote. Nylon servo control horns were then mounted to the rudder and elevator. Push rods then connected the control horns to the servos on the rear fuselage.

6.3.5 Wings

A double spar technique with a swept forward wing was determined to be the best solution. The spars were laminated balsa sticks with the front spar slightly larger than the rear spar and both tapering down

through the entire length of the wing. Both spars also extended an additional 3.5" at the root to be able to slide into the carry-through box.

A simple solid foam wing technique was utilized to construct the wing. Templates were first designed using AutoCAD 2002 software to create an airfoil shape with slots for the inner two spars. The templates were placed onto the appropriate width sheet of styrofoam and cut out using a hot-wire. With the wing shape created, the spars were then epoxied into the wing. After a 24 hour setting time, slits were cut into the foam for servo wiring and a hole was also cut to mount the servo inside the wing. Ailerons were created by simply cutting off the appropriate portion of the wing with the hot-wire. Monokote plastic covering was hot-ironed onto the wing to smooth its surface. The wingbox was constructed out of balsa sheets and lite-ply. Two box structures were created, one for the front spar, the other for the rear spar. These box structures were then epoxied onto the support longerons of the fuselage. Holes were drilled within the wing spars and wingbox to allow a bolt to pass through and lock the wing in place.

6.3.6 Landing Gear

The main landing gear was store-bought stock landing gear. Mounting holes were drilled at the top of the landing gear so that it may connect to the fuselage. The tail gear was also a stock gear bought commercially. Connecting push rods extended from the tail wheel to the same servo that the rudder utilized on the fuselage so they could be steered in unison.

Table 6.2: Manufacturing Timeline – Second Generation Aircraft Design

TASK	START DATE	FINISH DATE	ACTUAL DATE COMPLETE
Fuselage	**2/1/04**	**2/13/04**	**2/15/04**
Make Templates	2/1/04	2/1/04	2/1/04
Build Water Tank	2/2/04	2/5/04	2/5/04
Build Motor Mount	2/2/04	2/2/04	2/2/04
Make Front Fuselage Support	2/2/04	2/4/04	2/4/04
Cut / Attach Styrofoam Shaping	2/204	2/12/04	2/10/04
Build Wing Box	2/7/04	2/10/04	2/12/04
Attach Servo Water Release	2/8/04	2/8/04	2/13/04
Attach Landing Gear	2/8/04	2/8/04	2/10/04
Build Tail Connection	2/8/04	2/10/04	2/13/04
Econokote Front Fuselage	2/11/04	2/13/04	2/15/04
Empennage	**2/14/04**	**2/20/04**	**2/22/04**
Build Frame for Stabilizers	2/14/04	2/15/04	2/16/04
Build and Attach Control Surfaces	2/14/04	2/18/04	2/17/04
Econokote Stabilizers	2/15/05	2/18/04	2/20/04
Mount Servos / Push Rods	2/19/04	2/20/04	2/22/04
Wings	**2/21/04**	**2/29/04**	**3/2/04**
Make Templates	2/21/04	2/21/04	2/23/04
Make Wing Spar	2/21/04	2/21/04	2/23/04
Cut Stryofoam	2/22/04	2/22/04	2/25/04
Cut Control Surfaces	2/22/04	2/22/04	2/25/04
Add Servos / Wiring	2/23/04	2/25/04	2/28/04
Attach Wing Spar	2/24/04	2/27/04	2/29/04
Monokote and Finish Wings	2/28/04	2/29/04	3/2/04

7.0 TESTING PLAN

7.1 Static Tests

Static testing was of fundamental performance to the team this year. Enhanced focus and labor was dedicated to building samples of nearly every concept discussed to compare real world results. For instance, the wing structure was built using both balsa ribs and solid stryrofoam. Although the balsa ribbed wings were aesthetically pleasing, the time required to construct them was magnitudes greater than that of the solid styrofoam wing; on the order of a week compared to one hour. In addition, the ultimate strength and torque testing showed negligible difference between the two concepts. Based on this experience the figures of merit were modified to reflect the results of the tests.

The propeller tests are covered in the detail design phase and verify that the static thrust produced by the motor configuration will be sufficient. Other tests conducted include the different concepts of emptying the water (which was discussed briefly in section 5.2.3), spar construction, motor mounts, and tail construction. The tail construction in particular was given special attention. From speaking with professional modelers it was suggested that instead of using styrofoam the team should consider using balsa ribs and then apply a covering to the built-up structure. To conduct the experiment detailed templates were created in Auto-CAD and carefully cut and set aside for use. Then, over the span of several weeks a tail structure was meticulously built and covered using a combination of birch wood sheeting and monokote. The end result was a lightweight and strong tail section. However, the team then built a tail section of styrofoam and a single, thicker spar of balsa within one hour. Again, the ultimate strength and torque tests were used to compare the two sections. The results showed little difference between the two tails, with any improvements in the ribbed section paling in comparison to the amount of time required to construct it. In the event of an unforeseen accident or crash (which required a rebuild) the result would have been a catastrophic loss of labor and time.

Before the first flight the aircraft was dropped from 2 feet off the ground to verify that the landing gear could withstand a hard landing. Also, the wingtip test was performed, where the fully-loaded aircraft was lifted by the wingtips. Finally, a range check with the controller and aircraft separated by 1,000 feet was completed. All tests were successful and so the aircraft was cleared for its first flight.

7.2 Flight Test Check List

Prior to each flight, the following check list was and will be completed to ensure the airworthiness of the aircraft.

1. Aircraft surfaces are undamaged and clean.
2. Check rubber bands tightness. Wing and tail components are securely fastened to the fuselage.

3. Check that the landing gear is securely fastened. Vertical tail is properly secured. Propeller / spinner has been installed properly.
4. Check control surfaces (rudder, elevator, and ailerons) for deformations and excess play in their movement. Check for solid connection between servo arms, horns, and extension rods.
5. Servo and battery wires are snuggly fit. Receiver and transmitter battery power sufficient.
6. Hatches are completely closed and secured.
7. Switch on the receiver. The rudder, elevator, nose gear, and ailerons all rotate in the proper direction. Motor successfully engages / disengages. Controls continue to operate correctly with the motor on full power. Range check.
8. Perform a wing loading test by lifting the aircraft at the wingtips.

7.3 Flight Tests

7.3.1 February 1, 2004

The first flight test of the aircraft occurred on February First, in gusting winds of about fifteen knots. The aircraft had actually been completed two weeks earlier; however, weather and pilot availability prevented an earlier flight. On the first flight the pilot tested the handling qualities and the glide capabilities of the aircraft without any payload. The main goal of the flight was to verify stability and prove the aircraft was flight worthy. The flight was a success; however, the motor did quit halfway through the flight, about five hundred feet from the runway. Fortunately, the pilot was able to glide in for a smooth landing with no concerns about aircraft stability or glide performance. Upon inspection it was discovered that the motor controller had burned out in flight. The department's electrician inspected the controller and found that the problem was due to a manufacturing defect. Despite the electric failure the team was exceptionally pleased with the overall result. It was seen that the placement of the center of gravity and the center of lift were correct and that the aircraft was stable. In addition, the aircraft was controllable in gusting winds such as what is expected in the competition. The only concern the pilot had, outside of the motor controller, was a request for a little less stability.

Unfortunately, additional flights were curtailed by the lack of a functioning motor controller. However, the team has recently acquired multiple controllers and anticipates testing the aircraft extensively in the months leading up to the competition.

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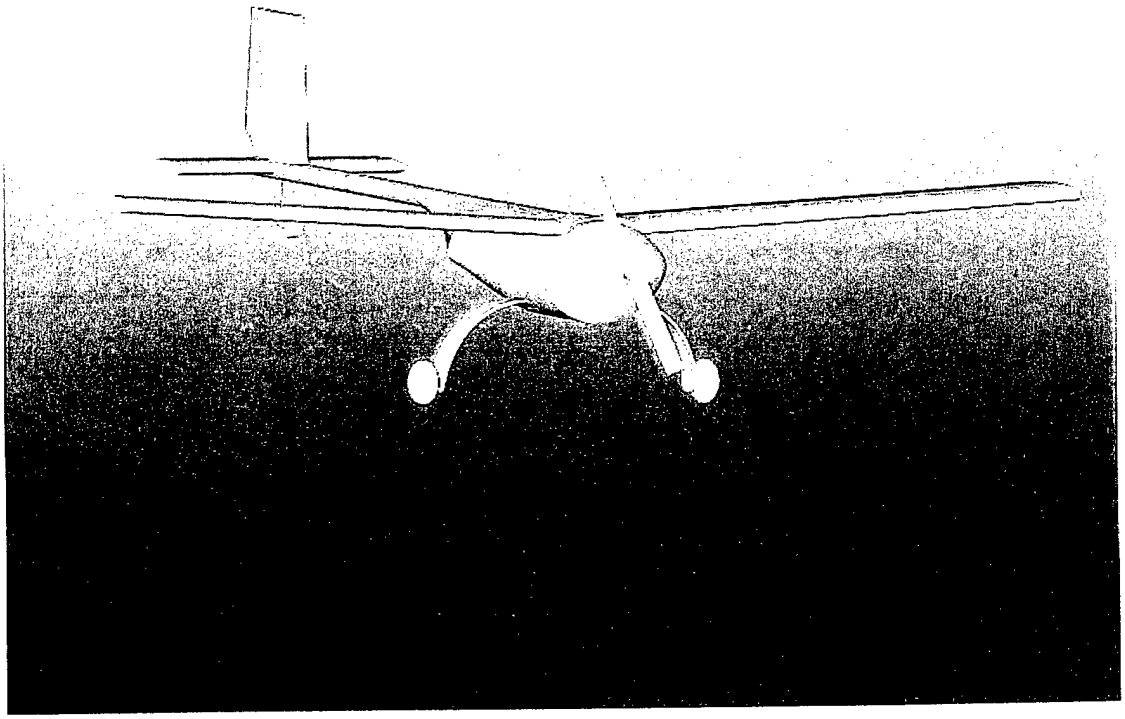
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City College of New York

2003/2004 CESSNA/ONR AIRPLANE DESIGN/ BUILD/ FLY COMPETITION



THE RHINO

-THE FIREMAN OF THE FOREST-

Table of Contents

1.0 Executive Summary 1
1.1 Highlights of Development Process 1
1.2 Design Alternatives 1
2.0 Mangment Summary 2
2.1 Team Architecture 2
2.2 Technical Groups Responsibilities 3
2.3 Scheduling & Documenting 3
3.0 Conceptual Design 5
3.1 Problem Statement 5
3.2 Mission Selection 7
3.3 Alternative Configuration Concept 8
3.4 Summary 17
4.0 Preliminary Design 18
4.1 Mission Model 18
4.2 Wing Design 18
4.3 Tank Design & Sizing 22
4.4 Fuselage Design 24
4.5 Static Stability Estimation 24
4.6 Dynamic Stability Estimation 26
4.7 Drag Calculations 27
4.8 Motor & Battery Selection 29
4.9 Performance 32
4.10 Lift & Drag Calculations 32
4.11 Optimization 34
5.0 Detail Design 35
5.1 Component Selection and System Architecture 35
5.2 Final Rated Aircraft Cost 40
5.3 Sized Aircraft Table 41
5.4 Drawing Package 42
6.0 Manufacturing Plan and Processes 46
6.1 Material Selection 46
6.2 Manufaturing Processes 47
6.3 Manufacturing Milestone chart 52
7.0 Testing Plan 53
7.1 Testing Objectives 53
7.2 Lessons Learned and Testing Results 55
7.3 Conclusion 57
Selected Reference 58

1.0 Executive Summary: the executive summary comprises a narrative description highlighting the major areas in the development process for the final configuration and a broad description of the range of design alternatives investigated.

1.1. Highlights of Development Process: The first major highlight in the development process was the specification of the payload. Initially it was assumed that the maximum payload of 4 liters was required to obtain the maximum score. However, an extensive analysis based on initial estimates of weight and relative performance, determined that the best payload was 3 liters.

The second highlight in the development of the design developed out of a desire to minimize the weight of the aircraft. In general composite materials are the materials of choice for high strength to weight ratios. The team lacked previous experience and knowledge of composite materials and their use, so early in the design a small fiberglass test-piece with curves was fabricated as well as a sheet of balsa wood reinforced with fiberglass. The results of the strength to weight ratios were impressive, that the decision was made to employ composite materials extensively through out the design. The ability to use composites also translates into the ability to have surfaces of high curvature that tend to provide for low drag as well as aesthetics.

The requirements for the specification of the propulsion system initially proved to be elusive. However, after the analysis methods featured in an article in the "Quiet Flyer" magazine was applied to the design based on the Astroflight family of motors, parameters such as thrust, number of cells, type of cells, and type of motor were easily specified. The final specifications came from using the computer based motor-analysis program called Motocalc.

Almost anyone can make anything fly with just enough thrust. The problems of sustainable and reliable flight is in the stability and control characteristics of the airplane. To ensure the reliability and survivability of the design, the stability and control methods featured in the textbook called Performance, Stability, Dynamics and Control of Airplanes, by B.N. Pamadi were understood and applied to the airplane. An example of the influence of this book on the design is that initially the tail-boom had been sized to fit the assembly box, upon extensive spreadsheet analysis it was increase by ten inches so as to satisfy the directional stability requirement. As a result the of the book, the survivability of the design has improved.

1.2 Design Alternatives: All known design alternatives from the payload to the type of cells were investigated. The nature of the mission as well as the competition site, enable many alternative to be eliminated without extensive analysis. An example is that V-tails tend not to be suitable for the windy condition experienced at Wichita. Also, historically entries with twin motors do not perform as excellently as single motored entries. Analyses and evaluations proved that such design alternative are not suitable for the desired performance of the design.

2. Management Summary:

The City College of New York team is composed of twelve mechanical engineers and three from other engineering majors. In order to accomplish the tasks efficiently and successfully compete; a project timeline had to be created.

2.1. Team Architecture:

The CCNY RHINO team consists of three groups, Aerodynamics, Structures and Systems. Each group had its own leader as shown in Figure 1. There has been a weekly meeting to review the assigned tasks and to evaluate the performance of the groups. The chief engineer and group leaders have worked together to make sure that each team had the tools and information to complete the tasks efficiently. The project manager wrote a journal of every team meeting and worked with chief engineer to keep track of the assigned tasks & draft an agenda for the following week.

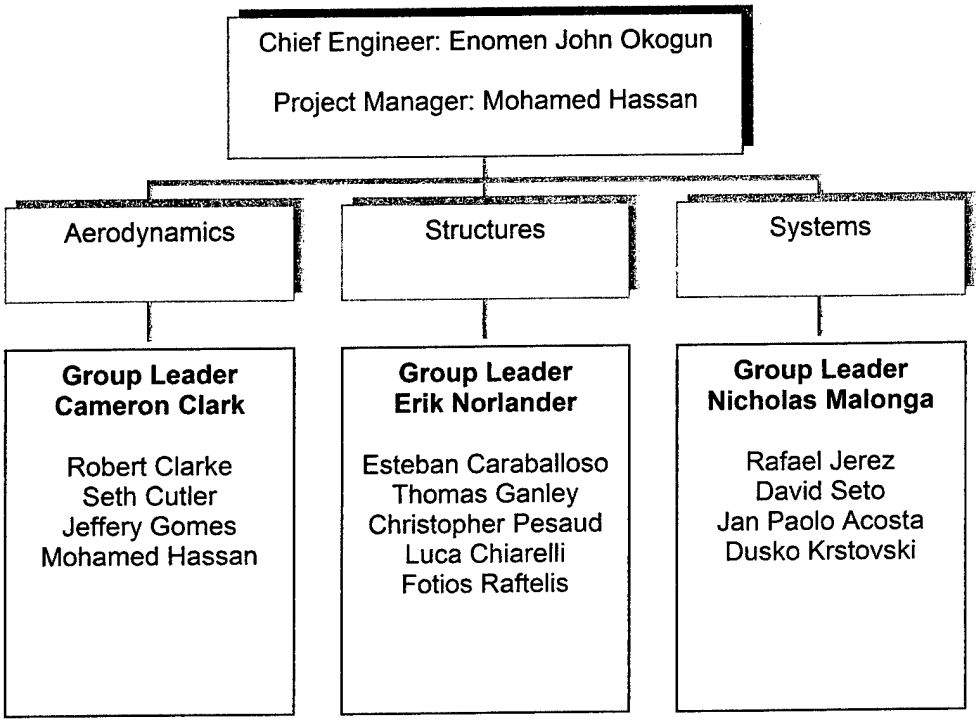


Figure 2.1: Organizational structure of the team and their positions

2.2. Technical Groups Responsibilities:

During the meetings, group members were discussing different aircraft configuration concepts. The decision made was based on mission requirements and the design constraints. Each group had responsibilities and they were divided among its members as presented in Figure 2.1.

2.2.1. Aerodynamics Group:

During conceptual design the team developed analytical and experimental ways to choose the amount of payload as it would lead to a competitive design thus a high score. In the preliminary phase the group investigated different concepts for the wing, tail and fuselage also stability and control of the aircraft were considered.

2.2.2. Structures Group:

The structures group was responsible for designing the payload systems, material selection and construction of the aircraft. The aerodynamics group provided the specification of the components of the aircraft. CAD drawings were created using Pro/ENGINEER as it would be the first step in the manufacturing process. Tank, Batteries and motor sizes were examined and placed in a certain way in order to maintain the stability of the aircraft while airborne.

2.2.3. Systems Group:

The systems group was responsible for the motor selection, batteries combination, propeller size, wiring of the electrical components, and programming the control pad. The group used different tools to optimize the amount of batteries to be used that can be enough to produce the desired thrust and meet the aircraft configuration.

2.3. Scheduling and Documenting:

Time was a crucial component for the project therefore a project timeline schedule was created. The design process was broken up into three components: conceptual, preliminary and detail designs, then Fabrication and testing of the established design. Due date was set for each phase. The project timeline chart is illustrated in Figure 2.2 and shows the estimated and the actual timing of the important design processes.

Figure 2.2: Project Timeline, the blue strips represent the projected time for the task while the red strips represent the actual time.

3. Conceptual design:

In the conceptual design, mission requirements and design specifications were considered. One of the two missions was chosen, in order to allow a more focused method for the design process; design alternatives have been narrowed down. The alternatives included basic aircraft specifications, landing gear type, manufacturing and assembly of the aircraft. Rated aircraft cost and other figures of merit were considered in order to achieve a competitive design.

3.1 Problem Statement

3.1.1. Mission Requirements: The aircraft to be designed and fabricated must fulfill the following:

- **Aircraft Storage and Timed Assembly:** The aircraft has to fit into a 4 ft by 2 ft by 1 ft box, assembled or disassembled. If the aircraft is disassembled to fit in the box, electrical connections have to be permanently installed, tapes can only be used to hold components in place but cannot be used as a structural latch.
- **Mission Time:** The maximum time for the mission is 10 minutes.
- **Weight:** The maximum takeoff-gross-weight of the aircraft must be less than 55 lb.
- **Takeoff Distance:** The aircraft must takeoff in 150 ft with all wheels off the ground.
- **Mission Tasks:** The aircraft has to perform one of two possible missions, as follows:

(I) Fire bomber Mission: This mission was designated to be a heavy lift/slow flight mission. The aircraft can carry and have a maximum capacity of 4 liters of water. The mission consists of a sortie that will be done twice which is

Aircraft will be loaded, take-off, dump the water during the downwind leg, and return to land.

The tank will be loaded with 2-liter "soda" bottles; methods used to assist loading water are allowed except pressurizing the bottles. The aircraft must complete a single 360^0-turn in the opposite direction of the base and final turn on the downwind leg of each lap. The water can only be dumped during the downwind turn markers/flag (see figure 1). The maximum orifice diameter allowed for dumping water is 0.5 inches.

$$\text{Mission Score} = \frac{\text{DF x Weight of Water}}{\text{Time}}$$

Were DF is the difficulty factor = 2.0

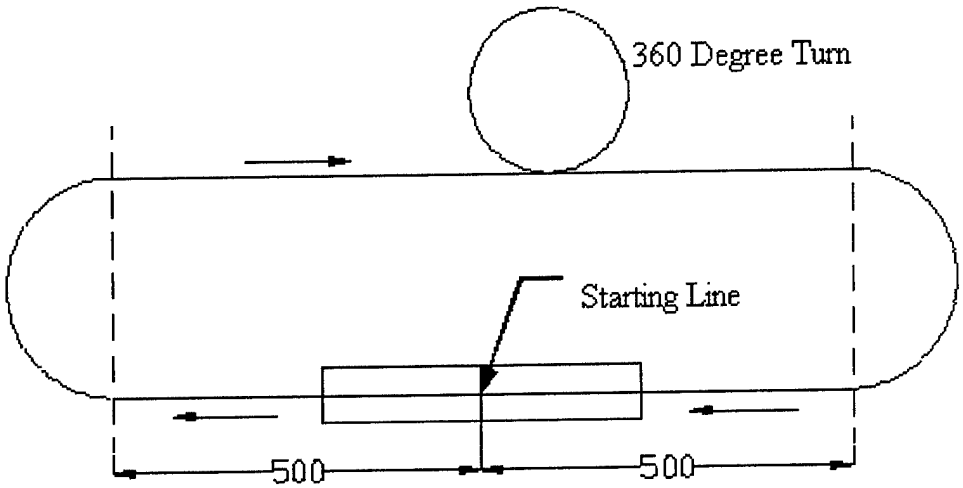
The starting and ending weight of the team's ground based storage tanks, taken when entering or leaving the flight box, will determine the weight of water.

(2) Ferry Flight Mission: This mission is designed for an aircraft with high-speed flight capability. The mission will consist of 4 laps. The aircraft must complete a single 360^0-turn in the opposite direction of the base and final turn on the downwind leg of each lap as shown in figure 1.

$$\text{Mission Score} = \frac{\text{DF}}{\text{Time}}$$

Were DF is the difficulty factor = 1.0

Figure 1: Course Layout



Course Layout
Shown to Scale

-Propulsion: The aircraft must be propeller driven and electric powered with an unmodified over-the-counter model electric motor & Nickel-Cadmium batteries. Motors and batteries will be limited to a maximum of 40 Amp current draw by means of a 40 Amp fuse (per motor or pack) in the line from the positive battery terminal to the motor controller. Only ATO or blade style plastic fuses may be used. Multiple motors and/or propellers can be used. All motors must be from the Graupner or Astroflight families of brushed electric motors. For safety, each aircraft will use a commercially produced propeller. The maximum battery pack weight is 5 lb and it must power propulsion and payload systems only.

3.1.2. Design Objectives: It was determined that the aircraft designed to fulfill the requirements above in order to be competitive also has to complete the following design objectives.

- Score: The final score of the competition is calculated by:

$$\text{Score} = \frac{\text{Total Report Score x Total Flight Score}}{\text{Rated Aircraft Cost}}$$

Therefore to maximize the score achievable, the Rated Aircraft Cost should be minimized, the Total Flight Score must be maximized and the Report Score should also be maximized.

- Survivability: Wichita, Kansas is the location of the competition, and the winds in Wichita can exceed 30 ft/s. Since the competition organizers stop all flights at 30 ft/s, the aircraft must be able

to fly at a maximum wind speed of 30 ft/s in the direction of flight. Also, in order to design against wind gusts the aircraft must possess adequate stability and control characteristics. Designing for survivability increases the chances of the aircraft to complete the mission tasks.

- **Weight:** From a historical perspective the aircraft must be less than 18 lbs to be competitive. Also, to improve the aircraft's competitiveness, the airframe of the aircraft including the batteries, motor and accessories must be able to carry it's own weight. That is, the payload weight must equal the weight of the aircraft minus the payload.

3.2. Mission Selection: Based on the scoring methods of the two mission choices, the ferry flight mission was determined to be less competitive than the fire bomber mission. This is based on the fact that an aircraft designed to meet the ferry flight mission must have the inverse of "R.A.C multiplied by mission" that equals or is greater than "2 multiplied by the weight of the water carried, divided by R.A.C divided by mission time" to be competitive. Therefore the ferry flight mission was ignored and the Fire Bomber mission became the focus of the design.

The initial assumption was to design an airplane that would carry 4 liters of water, because it was assumed that this would give the best score. To prove this an analysis was performed to compare each mission capability for 1 through 4 liters. For each mission the total flight score or TFS, the TFS divided by the TOGW of the aircraft, and the TFS divided by the square root of the TOGW of the aircraft were estimated and compared to the 4-liter mission. To do this, an analysis was done to determine the best speed to fly in which adequate distance and time will be allowed for the water to be completely dumped, for each mission. The weight of the aircraft for each mission was also estimated. The dumping time was estimated through empirical tests. An important parameter in this analysis was the radius of the 360°-turn, because it was used to eliminate velocities that would require large radii. After evaluating the average speeds, 65.5 ft/s was found to be the best speed for all missions, over 49.2 ft/s and 82.0 ft/s. 49.2 ft/s was too slow and 83.0 ft/s gave too much radii. Table 1 below shows the summary of the results.

Table 1: Mission-Task-Selection Analysis Summary. The F.O.M are in square brackets.

Payload (L)	TOGW (lbs)	TFS	TFS/TOGW	TFS/(sqrt TOGW)	Total F.O.M
4	17.6	0.123198 [4]	0.006985 [1]	0.029335352 [3]	8
3	15.6	0.116159 [3]	0.007527 [2]	0.029569 [4]	9
2	13.2	0.106203 [2]	0.008029 [4]	0.0292 [2]	8
1	11.0	0.083239 [1]	0.007551 [3]	0.025071 [1]	5

The estimation of the TFS, the TFS/TOGW, as well as the TFW/(sqrt TOGW) was done in order to also determine what mission capability would provide the best performance and stall speed.

Stall speed became a factor due to the slow flight capability of the aircraft. Each mission had figures of merit, based on the estimated values. The figures of merit are: 4 – first, 3 – second, 2 – third, 1 – worst. The figures of merit for each quantity are listed in square brackets. Therefore, the estimated TFS, etc. in table 1 served as figures of merit in order to maximize the TFS as well as the overall performance of the aircraft. Based on the analysis, a mission capability of 3 liters proves to be more beneficial in meeting our design objectives and the mission requirements. Therefore, a fire bomber mission with a mission capability of 3 liters was selected as the mission's task.

3.3. Alternative Configuration Concepts: Before determining the best aircraft configuration that would best meet the design requirements and objectives, it was important to determine the shape of the tank first. Then, the aircraft configuration that is best able to carry the payload, i.e. the tank, can be chosen.

3.3.1 Tank Concepts: the factors that affect the dimensions of the tank are described below:

(1) Length: The smaller the length or chord wise dimension, the better the stability of the airplane and the slow flight capability of the airplane. This is because, since water is not a rigid body it will move with every disturbance and this will tend to shift the center of gravity. Therefore in order to maintain the center of gravity range within the static margin this dimension has to be as small as possible. Furthermore, the larger the dimension, the longer the chord will have to be as well as larger tail volume will be required and thereby increasing the trim drag.

(2) Width: The smaller the width or span-wise dimension of the tank, the less drag the airplane will have. This is because a small width will give a smaller frontal area and reduce the fineness ratio of the fuselage, both of which tends to lead to lower drag.

(3) Height: The greater the height of the tank, the greater the volumetric flow rate through the orifice. This is because the pressure of the water above the base of the tank where the orifice will be is in direct proportion to the height of the water. This higher pressure will enable the water to flow out of the tank faster, thereby reducing the time of flight and increasing the flight score.

The tank dimensions for a specific volume do not necessarily affect R.A.C., because R.A.C. is based on volume and not the individual dimensions. The types of tanks considered are, a rectangular polygon, a sphere, and an ellipsoid. Each factor affecting the dimensions of the tank was used as a figure of merit to screen each concept. The dimensions for a tank with a maximum capacity of 4 liters are used to determine the figures of merit. The figures of merit were determined by normalizing the calculated dimensions in each column to the best value within that column. The results are shown below in table 2.

Table 2: Figures of Merit table for types of tanks based on a 4 L maximum capacity.

Tank Type	F.O.M: Length	F.O.M: Width	F.O.M: Height	Total F.O.M
Rect. Polygon	1	1	0.759615	2.759615

Sphere	0.498734	1	0.759615	2.25835
Ellipsoid	0.55493	1	1	2.55493

Based on the analysis, a rectangular polygon was deemed the best types of tank.

3.3.2. Airplane Concepts: Having maximized TFS without compromising performance in the selection of the mission task and in the selection of the tank type, the selection of the airplane concept will be based on the following figures of merit:

I. High Lift Capability: The mission task chosen, that is fire bomber with a 3-liter payload capability. This is more than 6.5 lbs; therefore the aircraft concept must have achievable high-lift capability or HLC, without compromising the ability of the airplane to fit in the assembly box. For evaluation, maximum F.O.M. is 1 and the lowest is 0.

II. Rated Aircraft Cost: Minimizing the RAC of the airplane concept serves to maximize the final score. Certain design concepts due to their configurations have inherently low RAC. Some of factors that affect RAC are the empty weight of the aircraft, the volume of the fuselage and wings, the propulsion and flight system components. The rated aircraft cost for each airplane concept will be determined by calculating the worst deemed RAC it can possess. For evaluation, the lowest RAC will be divided by the RAC of each concept and the result will be the F.O.M. assigned that concept. Therefore, the maximum F.O.M. is 1 for the aircraft with the least RAC, and the aircraft with the highest RAC will have the least F.OM.

III. Static Stability Capability: Static Stability Capability or SSC provides for the design objective of survivability and completion of mission. The design concept chosen will have to be inherently stable and have ease of control so the pilot can easily counter any errant gusts or winds. For evaluation, maximum F.O.M. is 1 and the lowest is 0.

IV. Reliability and Familiarity: Reliability and familiarity or RF of the design concept chosen enables completion of the mission, and increases its survivability. This figure of merit serves the design objective for survivability. For evaluation, maximum F.O.M. is 1 and the lowest is 0.

V. Low drag: Survivability involves more than stability and control it also involves power consumption. To meet the power requirement necessary for minimal power consumption it is important for the airplane concept to have low drag or LD characteristics and be able to meet the range and endurance requirement. For evaluation, maximum F.O.M. is 1 and the lowest is 0.

The design concepts are outlined below.

- 1- Biplane: The biplane concept can enable the aircraft to achieve high-lift capability, but the wings may also be too close to allow proper airflow thereby reducing its lift capability as shown in figure 2A. Also, Biplanes usually require external bracing or struts. Bracings such as wires would reduce the ability for LD of the airplane concept, while struts would increase the R.A.C. This concept may be high on HLC and SSC, but will score poorly on RAC and LD. For RF it will perform moderately. The RAC calculated for a biplane concept is 18.1.

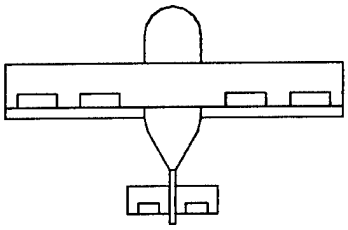
-2- Canard: The horizontal stabilizer in front of the wing, or canard, used to provide pitching stability is a lifting surface; thereby it may increase the high lift capability moderately as shown in figure 2B. A major advantage of using a canard is its ability for good center of gravity travel. This capability helps the survivability of the airplane concept, because the payload is water and it will shift the c.g. position anytime the airplane accelerates in any of its 6 degrees of freedom. However, the location of the vertical stabilizer may reduce its SSC. Also, canards may increase R.A.C if canard flaps are used, because apparently each flap requires one servo. The RAC calculated for a canard concept is 15.15.

-3- Flying Wing: As shown in figure 2C has two main advantages its capability for LD, and low RAC. Flying wings may not be appropriate for High Lift capability without compromising the ability of the aircraft to fit in the box. This is mainly due to the airfoils necessary for high lift tends to have significant moments about the aerodynamic center, and the airfoils used in flying wings should have very low moments about the A.C. Furthermore, flying wings possess very low survivability or SSC, especially in at a site like Wichita. Flying wings will also rate very low in RF. The RAC calculated for a flying wing concept is 12.25.

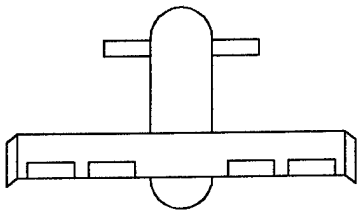
-4- Monoplane: The monoplane is the best compromise of all the aircrafts considered As shown in figure 2D. It is a design concept that rates high on RF, and it can be designed for survivability or SSC, and high lift capability without jeopardizing the ability of the aircraft to fit in the assembly box. However, its ability for LD is moderate. The RAC calculated for a monoplane concept is 12.79.

Figure 2: Different Plane Concepts

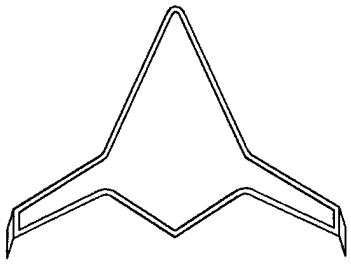
(A) Biplane



(B) Canard



(C) Flying Wing



(D) Monoplane

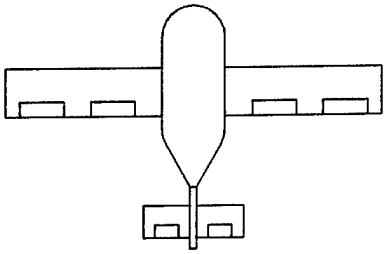


Table 3 below shows how the aircraft concepts rank against each other.

Table 3: F.O.M. ranking of the airplane concepts considered:

Airplane Concepts	HL	RAC	SSC	RF	LD	Total
Biplane	1	0.68	1	0.5	0	3.18
Canard	0.75	0.81	0.75	.25	0.5	2.81
Flying Wing	0	1	0	0	1	2
Monoplane	0.5	0.83	1	1	0.5	3.83

Therefore, based on the figures of merit considered and in the manner in which they were evaluated, the best airplane concept for the desired design objectives is the monoplane.

The next concept configurations are outlined below.

3.3.3. Tail Concepts: a major consideration for the tail surface is that the servo that controls the rudder(s) should be able to be linked to the nose wheel in a tricycle landing gear configuration or the tail wheel in a tail-dragger configuration. With a maximum of 1, the best tail concept will be evaluated on the following figures of merit:

I. Static Stability Capability: The primary function of the tail is to provide static stability, and this is very important for survivability of the airplane in completing the mission.

II. Low Drag: Certain tail concepts lower the total drag of the airplane, because of increases in effective aspect ratios, which then reduces the required wetted area. Other tail concepts may require more wetted area because of their location within the wakes of the fuselage or tail.

III. Weight: Certain tail concepts require more weight due to their configuration, and this may have to limit the ability of the airplane especially in completing the desired mission task.

IV. Rated Aircraft Cost: Tail concepts also influence the rated aircraft cost. The maximum contribution to RAC deemed attainable by a tail concept shall be calculated, and the lowest RAC shall be divided by the RAC of each tail concept to determine the FOM of the corresponding tail concept. The RAC is computed by determining how many surfaces with active controls the tail concept will have, and the number of servos required in controlling the active surfaces. Also, if the servo required for the directional control or the rudder, cannot be connected to the nose wheel of a tricycle gear or the tail wheel of a tail dragger, a servo is added in computing the tail concept's contribution to RAC.

V. Assembly Box Considerations: This figure of merit or ABC accounts for the ease and lack of complexity with which a tail concept can be fitted into the assembly box.

Below, the tail concepts considered are detailed and evaluated.

-1- Conventional Tail: Conventional tails are advantageous because they provide sufficient stability and control, and tend to have lower weight than most other tail concepts. Lower tail weight can help to reduce trim drag, because of c.g. position with respect to the aerodynamic center. However, the horizontal stabilizer may be in the wake of the fuselage and wing. Although this can be accounted for by increasing the size of the horizontal stabilizer, this will also increase the overall drag of the airplane.

(2) T-Tail: The vertical tail can be smaller than that of the conventional tail because of the winglet effect created by the horizontal stabilizer due to the increase in effective aspect ratio. The horizontal tail can also be lifted out of the wakes of the fuselage and wing, and this can also mean it will be of less size than that of the conventional tail. These reductions of size of the T-tail configuration could be helpful in enabling the aircraft to fit in the assembly box. However, the fin of the T-tail is heavier than the conventional tail because of the loads from the horizontal stabilizer. Also, the T-tail has to be carefully placed in order to avoid the deep stall, whereby the horizontal stabilizer is caught between the wakes of the fuselage and wing. A T-tail may also have to disassembled in order to fit within the assembly box.

(3) Cruciform Tail: The cruciform tail is a compromise between the conventional tail and T-tail. The horizontal stabilizer can be placed out of the wing wake, and there would be a less weight penalty due to the vertical stabilizer. However, there is no aspect ratio reduction. Also, there will be a discontinuity in the movable section of either the horizontal stabilizer or the fin that would mean that an extra servo might be needed for complete control, thereby causing an increase in RAC.

(4) H-Tail: The H-tail concept is good for the assembly box considerations, since the fins may not require any dismantling. There is an effective increase in aspect ratio of the horizontal stabilizer due to the winglet effects of the fins. However, added strength is needed for the horizontal tail, the horizontal tail is also in the wake of the wing, and the need to control two moveable rudders may increase the RAC.

(5) V-Tail: V-tails in general tend to reduce the wetted area of the tail, as well as the rated aircraft cost of the airplane. However they tend to produce adverse yaw, and they also lack survivability at the competition site due to a limited capability in static stability. Also the ruddervators are quite complex. V-tails are the simplest to build but the most complex to design.

(6) Inverted V: The solution to the adverse yaw experienced in V-tails is the inverted V-tail. It also, produces a complimentary roll, which enhances coordinated turns. The inverted V-tail also has the other advantages of the V-tail, i.e. lowers RAC and wetted area. However, as well as the complexity of the ruddervators, its major disadvantage is that it requires extra ground clearance. This may not be acceptable for a tail dragger landing gear configuration.

(7) Y-Tail/Inverted Y-Tail: This configuration does not require the complexity of the ruddervators of V-tails and inverted V-tails, because it uses a rudder on the fin. Furthermore, the horizontal stabilizer is kept out of the wakes of the wing, and uses a lower surface area with respect to the conventional tail. However, it may require some complexity for assemblage into the assembly box.

(8) Boom Tail: The option of the tail boom is included in the tail concepts, because it provides flexibility in attaining the required tail volumes without unnecessarily increasing the wetted area of the fuselage, or the fuselage weight, and in enabling the aircraft to fit in the assembly box.

Table 3 below shows how the aircraft concepts rank against each other.

Table 4: F.O.M. Ranking of the Tail Concepts considered:

Airplane Concepts	ABC	RAC	SSC	Weight	LD	Total
Conventional	1	1	1	0.75	0	3.75
T	0.5	1	0.75	0	1	3.25
Cruciform	0.5	0.87	1	0.5	0	2.87
H	1	0.87	1	0	0.5	3.37
V	0.5	1	0	0.75	0.75	3.25
Inverted V	0.5	1	0.25	0.75	1	3.5
Y/Inverted Y	0.25	1	0.5	0.5	0.5	2.75
Boom	1	0	1	1	1	4

Based on the figures of merit considered and in the manner in which they were evaluated, the best tail configuration concept is the horizontal tail with a tail boom.

3.3.4. Landing Gear Concepts: Two landing gear concepts are considered, the tail dragger and the tricycle landing gear. Although the tricycle landing gear is of greater familiarity than the tail dragger, it appears that due to the size of the tank selected earlier, the center of gravity of the airplane may be very close to nose of the airplane. Therefore the stability and the controllability the tricycle-gear will provide the airplane on the ground may not be sufficient. An approximation of this distance is 7 inches. Tail draggers also pose control problems on the ground.

Furthermore, both concepts can help to save a servo for control of the airplane on the ground, by connecting it to the servo that controls the rudder. However, only certain tail concepts will provide this reduction in RAC. Also, usually the main gear of the tricycle landing gear is placed almost at the center of gravity of the airplane, but this will interfere with the tank orifice. For the tail dragger configuration the mina gear is mounted below the leading edge of the wing. Therefore, the tail dragger configuration will not interfere with the mission task. The figures of merit are contribution to RAC, controllability of the airplane the ground (CAG), familiarity (F), and mission task consideration (MSC). Table 5 below summarizes the results.

Table 5: Landing Gear Concept and Figures of Merit Summary

Landing Gear	RAC	CAG	F	MSC	Total
Tricycle	1	0	1	-1	1
Tail Dragger	1	0	0	1	2

The FOM for MSC came out negative for the tricycle landing gear because it will be a hindrance to the mission task, i.e. it will be detrimental to the mission task. Therefore, the tail dragger best meets the requirement for the airplane concept.

3.3.5. Wing Concepts: Wing concepts include Planform and position on fuselage. The choice for wing planforms are, elliptical, rectangular, swept, tapered, and delta planforms. Since the airplane shall be flying at low subsonic speeds the delta planform will not be considered. Another planform

concept that will not be considered is the swept wing, because this planform better serves airplanes in the high subsonic region or canard aircrafts. Therefore the planform concept will be chosen from three remaining planforms, i.e. rectangular, elliptical and tapered.

-1- Figures of Merit for Wing Planform:

I. Stall progression: The rectangular planform offers the best stall progression, followed by the elliptical planform. The rectangular planform stalls from the root and progresses towards the wing tip. This is good because during a stall the ailerons remain effective. The reverse is the case for tapered wings, unless washout is used

II. Assembly Box Considerations: If a tapered planform is used, the wingspan will be larger than that of a rectangular wing and this can cause complications in fitting the aircraft in to the assembly box.

III. Complexity: The complexity of the elliptical design in designing and fabrication is unfavorable. While the complexity due to including washout in the tapered planform also increases complexity in design and fabrication.

The final ranking chart below summarizes the results for wing planform selection. The rectangular planform is selected based on the evaluation of the figures of merits assigned each planform concept.

The wing can be positioned in three different locations on the fuselage, in a high wing, mid –wing, or low wing configuration. The Figures of Merit considered in order to make a selection are outlined below.

-2- Figures of Merit for Position on Fuselage:

I. Obstructions: This figure of merit is important because, in order to reduce the weight and stress on the sides of the fuselage, it is better to have a wing structure that carries-through the fuselage. By having the wing structure pass through the fuselage rather than attach at the side of the fuselage, reinforcements that add weight can be eliminated. Therefore the figure of merit assesses the best compromise for a carry through structure for the wing and obstructions that can be tolerated.

Each position concept creates an obstruction. The high-wing concept obstructs accessibility to the tank for refilling of water, the mid-wing obstructs the tank itself, and the low-wing concept obstructs the tank orifice for water dumping. This figure of merit eliminates the mid-wing. The flow of water through the dump orifice may be slowed down if the water had to pass through a tube. For the high wing, tubes can be created to have the water pass through when filling the tank, and since there is no restrictions, this tube will create a scenario with more flexibility.

II. Static Stability: For increased survivability, high-wing concepts have more stability than low wing concepts. Although this can be corrected in low-wings by adding dihedral, but dihedral of the wing has a destabilizing effect on directional stability. A high-wing concept will require less

dihedral than a low wing concept for lateral stability. Furthermore, high wing concepts generate more lift than low-wing concepts since the flow over the top surface of the wing is not obstructed, unlike the low-wing concept.

The final ranking chart below illustrates the selection of the high wing concept. The negative value used against the mid-wing concept shoed that this concept is detrimental.

3.3.6. Motor Concepts: At this stage of the design process selecting the number of motors to use, and a choice between a tractor or a pusher configuration narrows the propulsion system down. The maximum amount of motors considered was two, because more than two motors may not be efficient in power, weight, and thrust considerations. The number of engines considered will be evaluated first, and based on the following figures of merit.

I. Weight: Using two motors positioned on the wings will increase the weight of the wing due to reinforcements. Also if the motors are placed in tandem, one on the nose and one on the tail, the fuselage weight will likely be increased as well. Also, using the weight of the Astrolight 40 geared and 60 direct motors, it was concluded that two motors are heavier than one.

II. Complexity: The complexity of wiring two motors increases significantly, over one. Two motors also require two speed controllers.

III. Rated Aircraft Cost: The contribution of the number of engines to the RAC is evaluated using the Rated Engine Power or REP. The REP for one motor is 1 multiplied by total battery weight, and the REP for two motors is 1.25 multiplied by the total battery weight. At this time it is assumed that the battery weight for two motors and one motor is the same. The FOM for RAC for one motor will be higher. This is because to determine the FOM each configuration's REP divides the lesser REP.

The final ranking chart below shows that one-motor configuration is superior based on the figures of merit considered.

A one-motor monoplane with a T-tail concept may not fit well with a pusher configuration. Other than the unusualness of the configuration, the rear of the fuselage may need reinforcing and this may lead to additional weight. Therefore, it is best to use a tractor configuration for the motor.

3.3.7. High lift Configuration: the estimated weight for the concept is between 15 and 18 pounds. Therefore the airplane must have high lift capability. The possible options for high lift are flaperons, flaps, and a high lift airfoil. The two figures of merit used are L/D, RAC and complexity. L/D was used because flaps and flaperons generate significantly lower L/D at high angles of attack. This is detrimental because flaps will also be used during level flight and not just during take-off and landing. The RAC contribution for flaps is 2, flaperons is 1.5, and a high-lift airfoil with ailerons is 1. Furthermore, using flaps and flaperons will increase the number of servos used. In terms of complexity, flaperons are the worst. They are not a good idea from a control standpoint

and require and experienced pilot as well as complex TX mixing. The final ranking chart below summarizes the results and shows that it's best to design the wing with a high-lift airfoil.

3.3.8 Final Ranking Chart:

Mission-Task-Selection Analysis Summary

Payload (L)	TOGW (lbs)	TFS	TFS/TOGW	TFS/(sqrt TOGW)	Total F.O.M
4	17.6	4	1	3	8
3	15.6	3	2	4	9
2	13.2	2	4	2	8
1	11	1	3	1	5

Figures of Merit for types of tanks based on a 4 L maximum capacity.

Tank Type	F.O.M: Length	F.O.M: Width	F.O.M: Height	Total F.O.M
Rect. Polygon	1	1	0.759615	2.759615
Sphere	0.498734	1	0.759615	2.25835
Ellipsoid	0.55493	1	1	2.55493

F.O.M. ranking of the airplane concepts considered:

Airplane Concepts	HL	RAC	SSC	RF	LD	Total
Biplane	1	0.68	1	0.5	0	3.18
Canard	0.75	0.81	0.75	0.25	0.5	2.81
Flying Wing	0	1	0	0	1	2
Monoplane	0.5	0.83	1	1	0.5	3.83

F.O.M. Ranking of the Tail Concepts considered:

Concepts	ABC	RAC	SSC	Weight	LD	Total
Conventional	0.75	1	1	0.75	0	3.5
T	0.75	1	1	0	1	3.75
Cruciform	0.5	0.87	1	0.5	0	2.87
H	1	0.87	1	0	0.5	3.37
V	0.5	1	0	0.75	0.75	3.25
Inverted V	0.5	1	0.25	0.75	1	3.5
Y/Inverted Y	0.25	1	0.5	0.5	0.5	2.75
Boom	1	0	1	1	1	4

Landing Gear Concept and Figures of Merit Summary

L. Gear	RAC	CAG	F	MSC	Total
Tricycle	1	0	1	-1	1
Tail Dragger	1	0	0	1	2

Wing Planform Summary

Planform	Stall	Box	Complex	Total
Elliptical	0.5	1	0	1.5
Rectangular	1	1	1	3
Tapered	0	0	0	0

Wing Position Summary

Position Concepts	Obstrct'n	Static Stability	Total Points
High-Wing	1	1	2
Low-Wing	0.5	0	0.5
Mid-Wing	-1	0.5	-0.5

Summary of Number of Motors Considered

# Of Motors	Weight	complxty	RAC	Total
One	1	1	1	1
Two	0	0	0.8	0.8

Summary of High-lift Concept

Concept	complxty	L/D	RAC	Total
Flaps	0.5	0	0.5	1
Flaperons	0	0	0.7	0.7
High-Lift Airfoil	1	1	1	3

3.4 Summary: The final ranking chart indicates that the airplane concept will have a single engine, high-wing monoplane with a tail-dragger landing gear, and a T-tail. The wing shall have a rectangular planform, a 2-D high-lift airfoil, and have only ailerons. Last but not the least, the payload shall be a tank in the shape of a rectangular polygon with a capacity of three liters.

4. Preliminary Design

4.1. Mission Model

With the key elements of the mission requirements as discussed in the conceptual design section in mind, the following factors were taken into consideration:

- Aircraft speed: 66 ft/s (20 m/s), which was determined from a combination of V_{stall} (42.7 ft/s) and wind speed, which was taken to be the average wind speed (23.3t/s) in Wichita. If the wind direction is the same as the flight direction, the aircraft has to fly above 65.3ft/s (42.7 ft/s + 23.3t/s = 65.3 ft/s).
- Maximum payload of 3 liters, with a flying time of 80 seconds per lap, and a tank-filling time of 30 seconds. Therefore the total time of a mission was estimated to be 3.4 minutes.
- Takeoff Gross Weight: 17.6 lbs.
- Altitude: 1500 ft.
- Flight speed range between 49.2 ft/s and 82 ft/s
- Maximum turn load factor of 2 at 82 ft/s
- Minimum flight speed of 49.2 ft/s with a maximum turn load factor of 1.25. As shall be pointed out shortly, this condition gives the minimum turn radius and enables the wing to fit in the assembly box.

4.2 Wing Design

4.2.1. Wing Area Sizing. It is desired that the aircraft fly slower/faster than 66 ft/s. Therefore; it was necessary to calculate the wing area (S) required to sustain flight at multiple speeds. Table 4-1 illustrates the design parameters taken into account while determining the wing area. It shows that a wing area of 7 ft^2 will sustain flight at a minimum flight condition with a flight speed at 49 ft/s and a turn load factor (n) of 1.25, and at a maximum flight condition with a flight speed at 82 ft/s and a turn load factor of 2.00. Also, wing areas greater than 8 ft^2 reduce the ability of the disassembled airplane to fit in the assembly box. Therefore, the critical design parameters are: capability of flight at 49 ft/s w/o head wind, capability of a flight at 65.6 ft/s with head wind and a maximum load factor of 1.25, and capability of flight at 82 ft/s with a maximum load factor of 2; all of which must result in a reasonable turn radius.

V (ft/s)	N	C_L	S (ft^2)	R (ft)
49.2	1.25	1.00	7.0	310.75
65.6	1.50	1.00	4.7	467.4
65.6	2.00	1.00	6.3	467.4
82.0	2.00	1.00	4.0	624.0

Table 4-1 Summary of speed and load factor versus wing area

4.2.2. Airfoil Selection, Phase I. Since this mission calls for slow flight while dumping water, only high lift airfoils were considered. Therefore, the design parameter in this phase is high-lift capability, defined by $C_{l,\alpha=0^\circ} \geq 1.0$ and $C_{l,max} \geq 2.0$. To determine the proper airfoil for this mission, a list of high-lift airfoils were compiled from the University of Illinois, Urbana Champagne database of airfoils. Several airfoils were considered but, since many of them didn't meet our minimum requirements of $C_{l,\alpha=0^\circ} \geq 1.0$ and $C_{l,max} \geq 2.0$, the search was narrowed down to 4 airfoils: 2 Epplers, 420 and 423, as well as 2 Seligs, 1220 and 1223.

4.2.3. Airfoil Selection, Phase II. To further analyze these airfoils, C_d vs. α (Figure 4-1) and L/D vs. α (Figure 2) were plotted. Therefore the design parameters in this phase are low drag, as well as high lift to drag over the AOA range. As shown in the figures, S1210 has significant increase in drag at an angle of attack (AOA) of 8°. The figures also show that E420 has the highest drag among the airfoils for just about any angle of attack. Based on these observations, S1210 and E420 were eliminated from consideration.

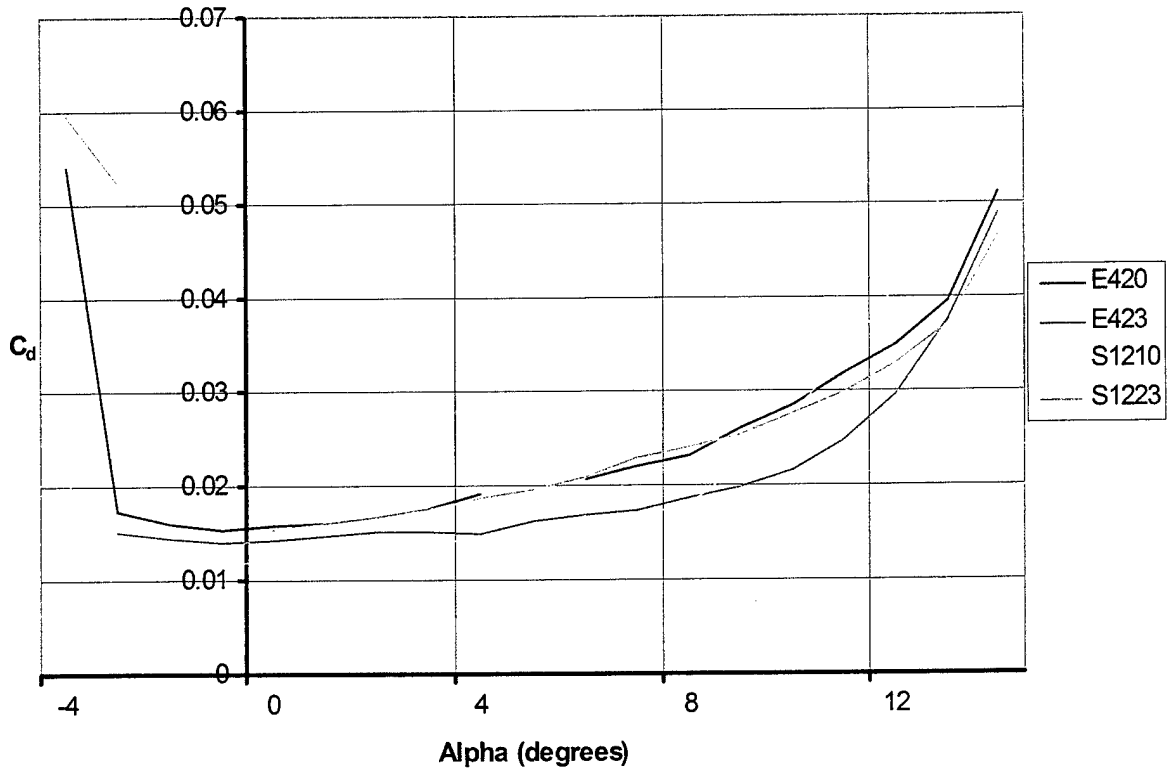


Figure 4-1 Drag versus angle of attack for various airfoils

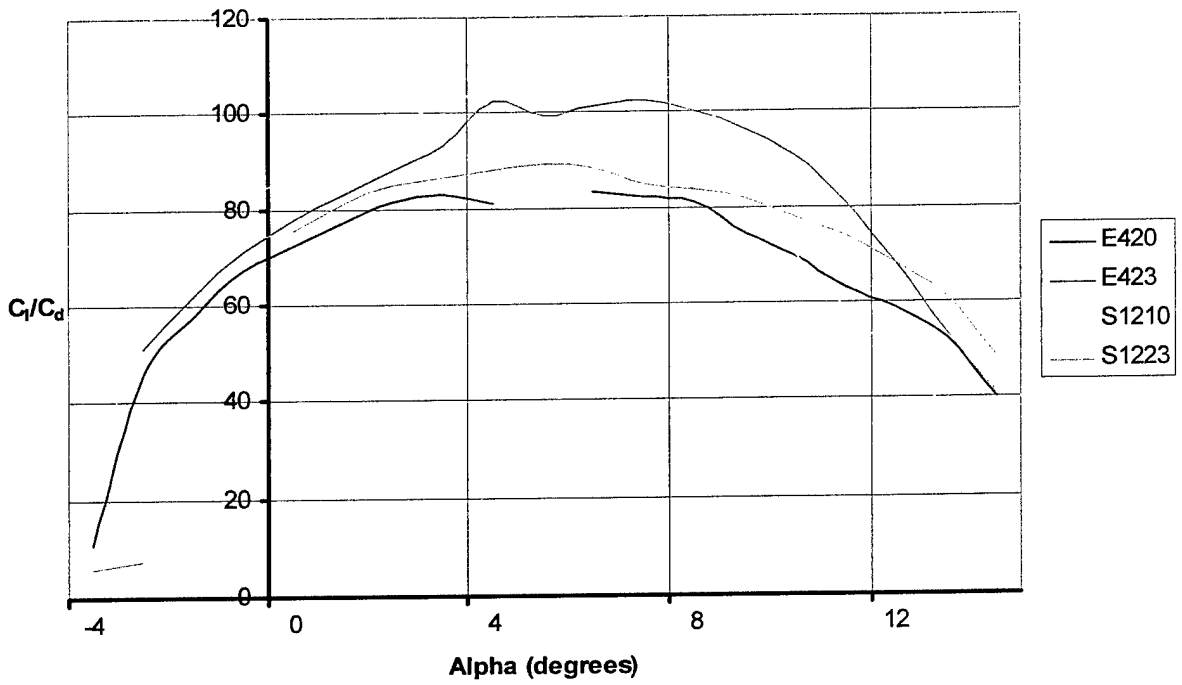


Figure 4-2 Ratio of lift to drag versus angle of attack

4.2.4 Airfoil Selection, Phase III. To facilitate the choice between the remaining two airfoils, their lift curves were extrapolated to finite wing values (see Figure 4-3). As the figure shows, S1223 has the higher C_{Lmax}; but it also has an inverted stall for $-3° \leq \alpha \leq 0°$. Although the aircraft is not intended to fly in this region, the region of inverted stall made S1223 undesirable given the possibility that the aircraft could slip into that region. S1223 also proved to be less desirable because it has a lower moment coefficient curve than E423 (see Figure 4-4). Therefore based on the design parameters of moment coefficient and finite wing characteristics, E423 is the designated airfoil for the wing.

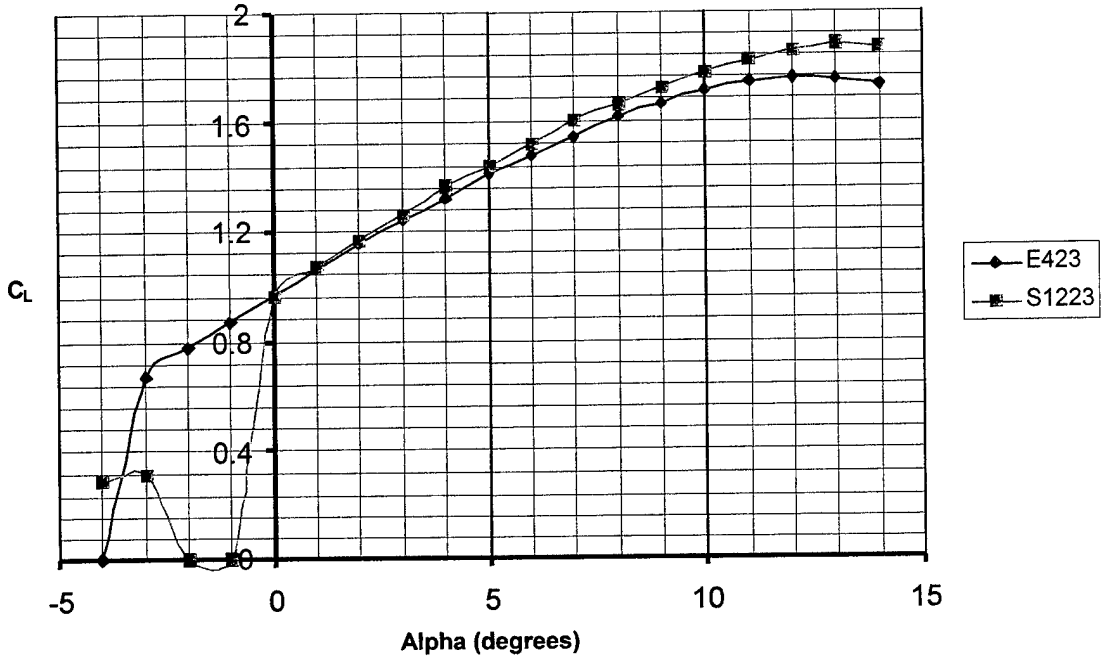


Figure 4-3 Lift versus angle of attack for multiple finite wings

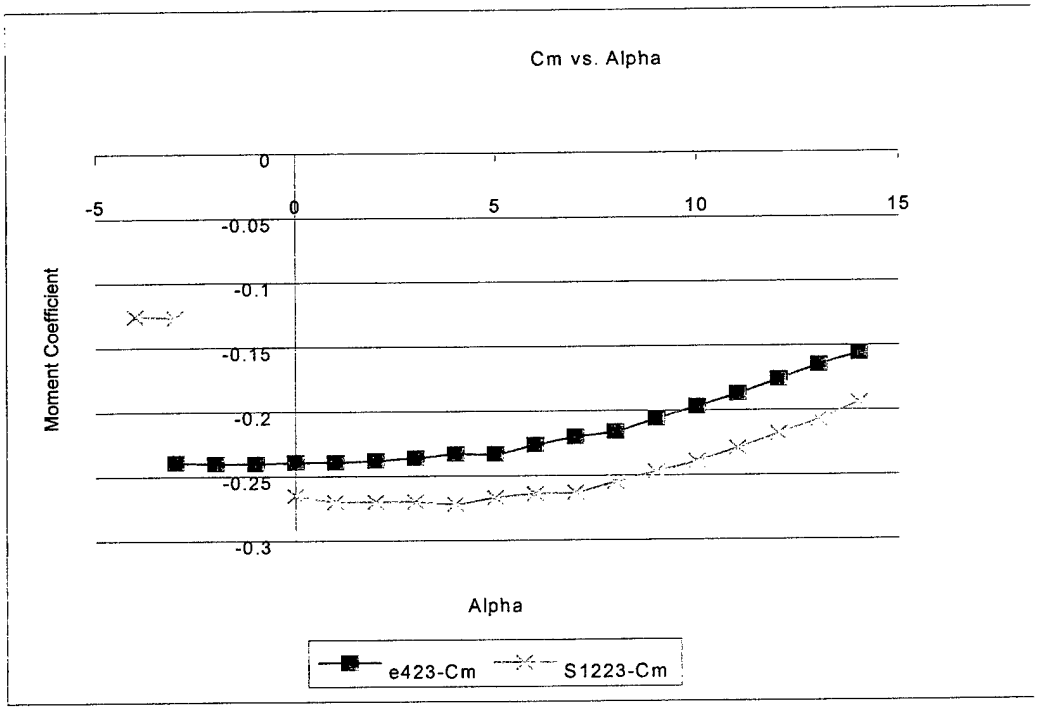


Figure 4-4: Moment Coefficient versus angle of attack

4.2.5 Wing Sizing. As discussed in section 4.1, the wing area was constrained by the cruise speed and load factor to 7ft^2. Based on this, the following design parameters place constraints on the dimensions/sizing of the wing:

- Static margin (based on the length of the tank)
- Assembly box dimensions
- Flow separation due to slow flight
- Effect of aspect ratio on induced drag

S (ft^2)	AR	b (ft)	c (ft)
7	7	7	1
7	8.3	7.6	0.92
7	10.1	8.4	0.83

Table 4-2 Summary of wing dimensions considered

Table 4-2 displays the possible wing dimensions for an area of 7 ft^2. Since induced drag decreases as the aspect ratio increases and also because the longer the chord the more likely flow separation will occur over the wings, a chord length of 1 ft was eliminated from consideration. Due to the static margin, a chord length of 0.83 ft was eliminated from consideration, because the smaller the chord the less the static margin and this will limit the location of the tank. Since for survivability, the aircraft must be flyable for a c.g. range over the length of the tank, because a worst-case scenario of the sloshing of water from the front of the tank to the rear of the tank was assumed. Therefore, 0.92 ft is the designated chord length; this sets the remaining parameters of the wing as highlighted in the table.

4.3 Tank Design and Sizing

From the conceptual design the tank capacity was determined to be 3 liters. Also, the best type of a tank was determined to be a rectangular polygon. The design parameters in the design of the tank are:

- Static margin: this affects the length of the tank. It is assumed that due to the possibility of water sloshing from in the tank the c.g. range will vary from front of the tank to the back of the tank. Therefore the greater the length of the tank the greater the static margin that is necessary. The static margin affects the size of the horizontal tail; the larger it is, the larger the tail volume, which is directly proportional to the area of the tail. Length of the tank was fixed at 4 inches, because a static margin of 2 inches forward and aft of the c.g. was estimated to be a good value. This value is later verified in the Static Stability section.

- Hydrostatic Pressure: This parameter affects the height of the tank. The pressure of the water above the 0.5-inch diameter orifice affects the speed of the flow of water out of it. The hydrostatic pressure of a column of water above a reference point is directly proportional to the height of the column of water. Therefore it is desirable to maximize the height of the tank. From initial empirical tests with a Tropicana orange juice carton, any height from seven inches and above is desirable.
- Drag in turns: The width of the tank determines the width of the airplane, and the same applies to the height. Therefore the tank dimensions determines the shape of the fuselage In order to minimize the drag from the cross-flow over the fuselage in turns, a cross-section shape of an ellipsoid with the short axis vertical was desired.

After all the design parameters above were considered a tank with composite shape was chosen; a pyramid attached to a rectangular polygon (see figure 4-5).

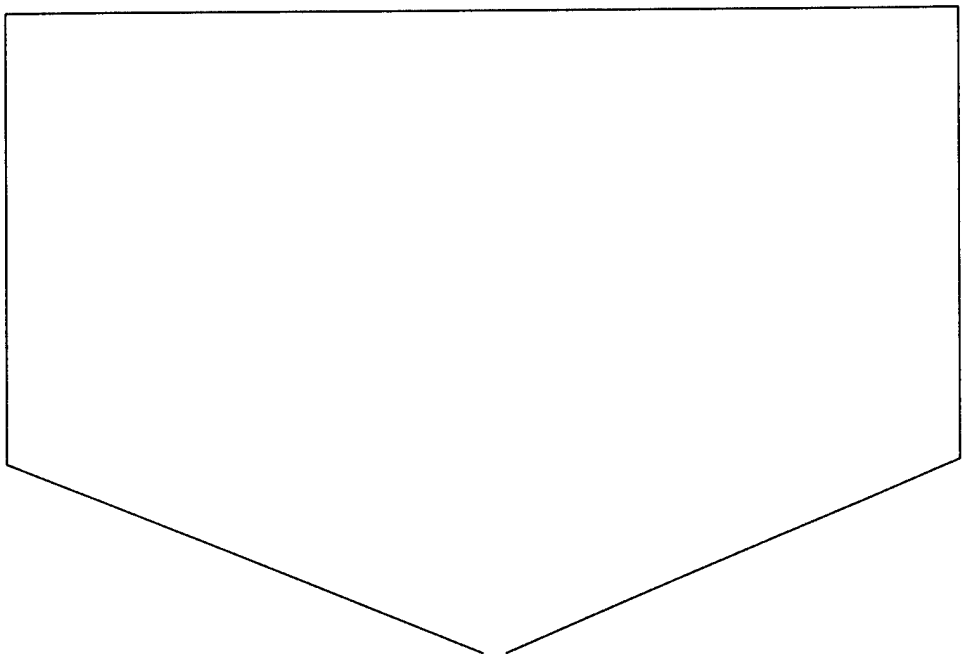


Figure 4-5: Cross-sectional Shape of Tank

Two tank sizes were considered. The first involved having the pyramid section to contain 0.25 liters of water and the second involved 0.5 liters of water. The dimensions for both tank sizes are presented in table 4-3. Tank 2 was chosen because it met the design parameters.

Tank Type	Rec. Poly. Vol.	W (in)	L (in)	H (in)	Pyramid H(in)	Total H (in)
Tank 1	2.50	7.63	4	5	4.08	9.08
Tank 2	1.275	3.8394	2.4	5	1.91	6.91

Table 4-3 Tank sizing summary and indication of selection

4.4. Fuselage Design

The fuselage was designed and sized with the following parameters in mind.

a. Conservation of Space: by conserving space within the fuselage and ensuring that all the space within it are just sufficient to fit the water tank, the batteries and other accessories without any waste of space. If space is not conserved the volume of the fuselage will be higher and so will the RAC. Also conservation of space helps to minimize the drag due to skin friction and possibly the pressure drag.

b. Wing-Fuselage Junction: the fuselage should have a constant width at its junction with the wing. This is because if the fuselage starts to taper in to the tail, a loss of lift will occur due to the pressure effects, in that vicinity. This is parameter is especially valid, because of the potentially large difference between the width of the tank and the root chord of the wing.

c. Elliptical Cross-section: An elliptical cross section is desirable with the long axis horizontal, because of the potential reduction in the drag during sideslip or due to cross flow of air over the fuselage.

Using the Pro/E and taking the consideration of the tank dimensions and the potential chord of the wing, the fuselage was dimensioned as follows: length of long axis is 12 inches, length of short axis is 8.4 inches, and the length of the fuselage from the end of the nose to the portion just before the taper to the tail is 14 inches, on the upper surface.

4.5 Static Stability Estimation

4.5.1 Static Directional Stability. An aircraft is said to be directionally stable if it has the capability to restore itself to the same direction as the oncoming wind when disturbed from steady level flight. This criterion is met by having a positive non-zero value for the directional stability derivative (C_{n_β}), which is composed of contributions from the wing, fuselage and tail.

The wing contribution to directional stability is results from dihedral and sweep, which is ignored because the wing is unswept. There are two methods for estimating the effect of dihedral on directional stability: the simple strip theory, which ignores induced drag effects, and an empirical formula, which is reserved for low subsonic speeds and includes induced drag effects. The application of the strip theory resulted in abnormally high values for the directional derivative and was therefore disregarded. Application of the empirical formula resulted in more reasonable values. Coupling these results along with the empirical formula's inclusion of induced drag effects led to the decision to use the empirical method to analyze the wing contribution to directional stability.

Analyses of the fuselage and tail contribution to directional stability are confined to a single empirical formula for each, which were taken from Reference 1.

Summing each contribution to directional stability showed that the Rhino has $C_{n\beta} > 0$ for all angles of attack for which it was designed (see Figure 4-6). In particular, $C_{n\beta} = 0.00141$ when the aircraft is in level flight ($\alpha = 0°$).

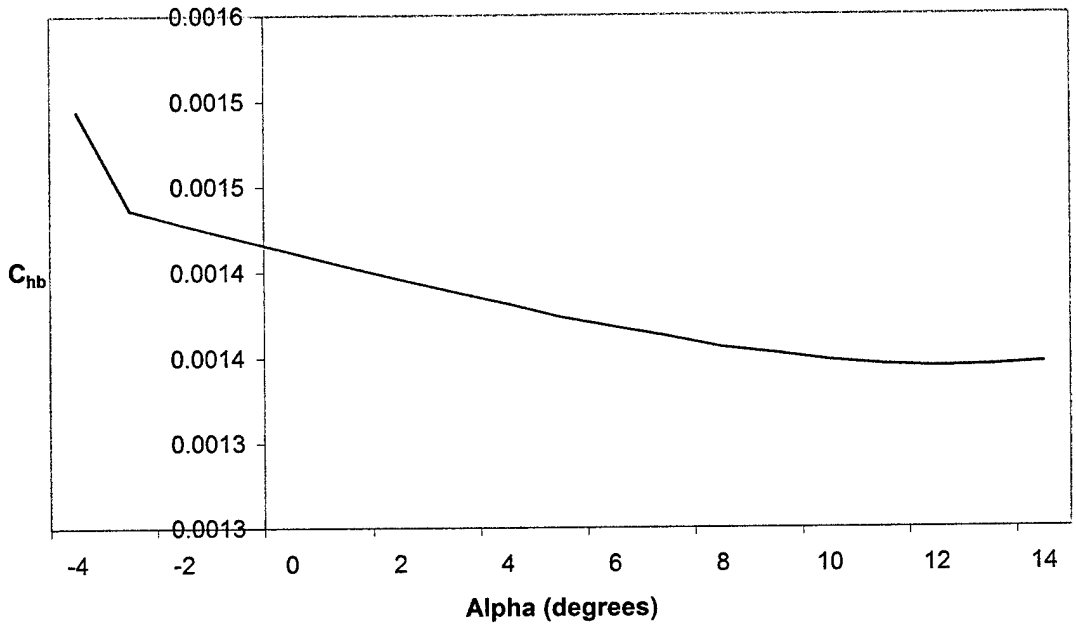


Figure 4-6 Directional stability coefficient vs. angle of attack

4.5.2 Static Longitudinal Stability. Like directional stability, the longitudinal stability derivative (C_m) is also composed of contributions from the wing, fuselage and tail. Each component was analyzed using empirical methods taken from Reference 1 and consequently, the individual analyses shall not be discussed. The main issue for longitudinal stability is having positive values for the stick-fixed neutral point (N_o) and the static margin (H_n). N_o represents the center of gravity location when the aircraft is neutrally stable. H_n is just another term used to describe the level of static longitudinal stability of the aircraft. Obtaining positive values for these parameters was achieved by varying the tail length, which changes the location of the center of gravity. For an aircraft to be statically stable, the center of gravity must remain ahead of N_o in pitch, which the design tail length of 3.46 ft. does. It also allows all components of the fuselage to fit into the box. Table 4-4 summarizes the longitudinal stability parameters.

Forward c.g.			Aft c.g.		
$d(C_m)/d(C_L)$	N_o	H_n	$d(C_m)/d(C_L)$	N_o	H_n
-0.586	0.654	0.586	-0.222	0.654	0.222

Table 4-4 Summary of longitudinal stability parameters

4.5.3 Static Lateral Stability. The lateral stability coefficient ($C_{l\beta}$) is equal to the sum of the individual contributions from the fuselage, wing and tail surfaces. The direct contribution of the fuselage is negligible, thus it can be ignored. However, the fuselage significantly interferes with the performance of the wing, thereby making an indirect contribution to lateral stability. The wing's contribution depends upon the wing-fuselage interference, wing dihedral and wing leading edge sweep.

The wing-fuselage contribution depends upon the placement of the wing. In a high-wing configuration, the inboard section of the right wing experiences an upwash in positive sideslip and a resultant increase in angle of attack. Conversely, the inboard section of the left wing experiences a downwash and a resultant decrease in angle of attack. This imbalance in lift results in a stabilizing/restoring rolling moment for high-wing configurations. Similarly, the induced rolling moment in a low-wing configuration is destabilizing. Understanding this concept made choosing a high-wing configuration all the more practical.

Using the strip theory analysis found in Reference 1, it was determined that wing dihedral and sweep have stabilizing effects in lateral motion. However, as discussed in section 4.5.1, the strip theory ignores the effects of induced drag. Given this, it was decided to use a more accurate empirical formula found in Reference 1 to estimate the wing contribution to lateral stability. This formula is valid for untwisted, straight tapered wings; the wing for the Rhino meets both criteria.

The contribution of the vertical tail was also estimated using an empirical formula taken from Reference 1. It takes into account the vertical tail side force, the horizontal and vertical distances between the center of gravity and the vertical tail aerodynamic center. The rolling moment for the vertical tail side force depends upon the angle of attack.

The empirical formulas referenced above assume that the aircraft is in positive sideslip. As previously discussed, an aircraft in positive sideslip needs an induced rolling moment to restore stability. This rolling moment must be negative according to the standard sign convention. Therefore, $C_{l\beta}$ must be negative at all angles of attack. Table 4-5 summarizes $C_{l\beta}$ at extreme and level angles of attack for which the Rhino was designed to fly.

α	$C_{l\beta}$
-3	-0.0834
0	-0.0833
14	-0.0812

Table 4-5 Summary of lateral stability coefficients at extreme and level angles of attack

4.6 Dynamic Stability Estimation

The following table displays the dynamic stability derivatives for the Rhino, which were calculated using empirical relations contained in Reference 1.

Longitudinal		Lateral-Directional	
$C_{L\alpha}$	0.104	C_{yp}	-0.227
$C_{m\alpha}$	-0.019	C_{lp}	-0.231
$C_{m\alpha}$	0.019	C_{np}	-0.343
$(C_{Lq})_t$	0.075	C_{yr}	1.05E-04
$(C_{mq})_t$	-0.283	C_{lr}	0.322
$C_{L\alpha_dot}$	-49.39	C_{nr}	-0.124
$(C_{m\alpha_dot})t$	-0.079	$C_{y\text{-}\beta\ dot}$	0.003
		$C_{l\text{-}\beta\ dot}$	0.000
		$C_{n\text{-}\beta\ dot}$	-0.002

Table 4-6 Summary of dynamic stability derivatives

4.7 Drag Calculations

Drag calculations for the aircraft involved solving for the following critical variables:

Reynolds Number, Re,

Drag Coefficient independent of C_l, C_{D0},

Form Factor, FF,

Coefficient of Friction, C_f,

Wetted Surface Area, S_{wet}, is the surface area of the entire object.

Planform Surface Area, S_{ref}, this is the area of a single view of the object, such as the top or side view

The S_{wet} of the fuselage was obtained by summing the surface area of its three segments, two cones and a cylinder. At any circular connecting surface the areas taken of any circle were subtracted. The S_{ref} was taken by summing the areas of two corresponding triangles and one rectangle.

To obtain the Reynolds Number, the following formula was used:

Re = (Velocity * Length)/ (Kinematic viscosity)

Velocity = 65.6 ft /s,

Kinematic viscosity = 1.6499 ft^2/s, at an altitude of 2000ft

Either, laminar or turbulent boundary layers determines Cf. If the size of the Reynolds number is 300,000 or below, it's Laminar, if it's greater it is Turbulent.

In laminar boundary layers, according to Blasius' friction law,

$C_f = 1.328 / (Re)^{1/2}$

In Turbulent Boundary Layers, according to the Schlicting empirical formula,

$C_f = 0.455 / (\log10\ (Re))^{1/2}$

Ideally, all results should be laminar.

To calculate FF, for the fuselage, $FF = 1 + 60/(FR^3) + 0.0025FR$, where FR is the fuselage fineness ratio, or fuselage length/diameter.

For the wing, horizontal and Vertical tails,

$FF = [1 + L\ (t/c) + 100((t/c)^4)]\ R$

R is a constant equal to 1.05. Ideally for the wing, the flow of air should be laminar, and we adjusted the dimensions of the wings to ensure this.

For the wing, tails fuselage and landing gear,

$C_{D0} = FF * C_f * S_{wet}/S_{ref}$.

The landing gear includes a single strut and wheel that is 4 inches in diameter and 0.5 inches wide.

As for the tail boom, we needed to calculate the reference length, the sum of the fuselage length and the boom length and by doing so use that information to calculate the C_f.

For the tail boom,

$C_{D0} = 1.05 * C_f * S_{wet}/S_{ref}$

Item	C_{Do}
Horizontal tail	0.00632
Vertical Tail	0.006276
Tail Boom	0.013587
Fuselage	0.0576
Wing	0.01467
Landing gear	0.00797
Total	0.106

Table 4-7 Summary of component drag values

This number was then used to calculate the effective drag acting upon our plane. Using the following equation, it was possible to calculate drag acting upon the system;

$$D = AV^2+B/V^2$$

$$A = \tfrac{1}{2}C_{Dp}*\rho*S$$

$$B = KW/(\tfrac{1}{2}\rho*S)$$

$$K = 1/\pi*e*AR$$

Item	Value
Air density (ρ)	$2.2409*10^{-3}$ (slugs/ft^3)
K	0.03995
A	8.35E-5 (slugs/ft^2)
B	1577.741 (slugs/ft^2)

Table 4-8 Summary of drag parameters

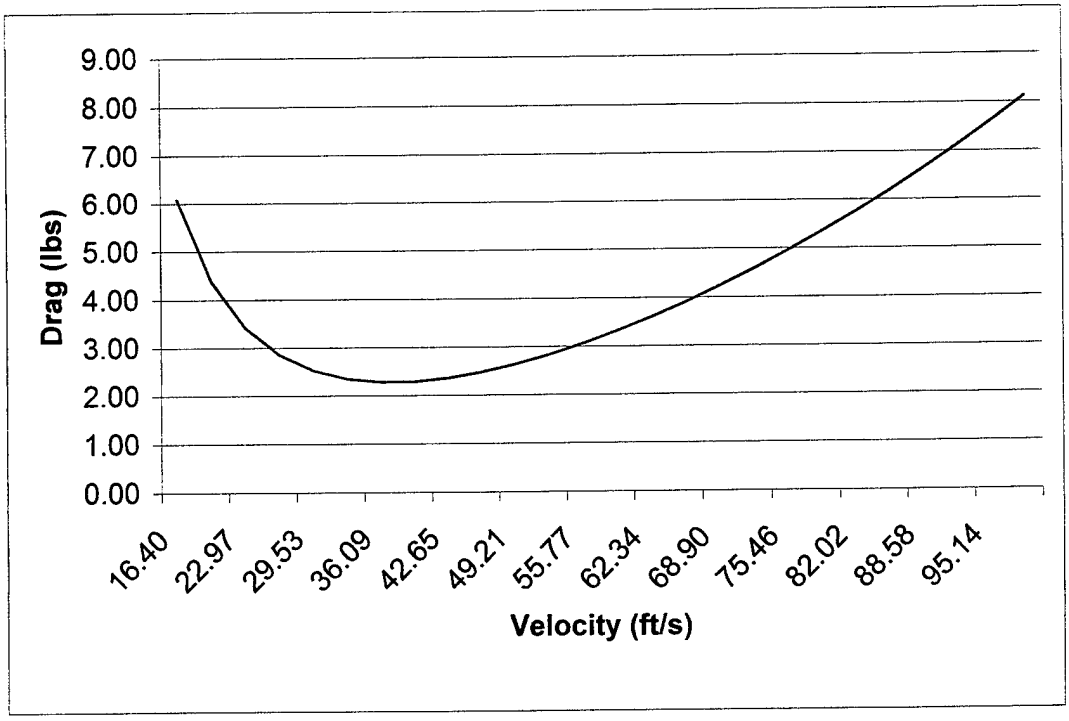


Figure 4.6 Relationship between velocity and drag

4.8 Motor and battery selection

4.8.1 Initial calculations

The team's major requirements in the power system included a run time of at least 3.4 minutes, a minimum airspeed of 80.7 feet per second, the ability to sustain strong wind as well as a good climb and a minimum thrust of 9 pounds based on take off requirement. Also, the rules of the competition restricted the battery weight to 5 pounds.

Initial calculations based on formulas found in an airplane magazine, showed that the airplane would be able to perform well if the propulsion system used an Astro 60 direct drive. To power the motor, a battery pack of 30 cells was suggested from the manufacturer's website. From this initial analysis, four different types of cell matched our requirements. Those cells were: RC-2000, CP-2400 SCR, and RC-2400. The following table summarizes our results:

Cells	*Number of Cells*	*Run Time (min)*	*Battery pack Weight (lb)*	*Thrust (lb)*
RC-2000	**30**	**3.428571429**	**3.75**	**12.99375**
CP-2400 SCR	**30**	**4.114285714**	**3.91875**	**12.99375**
RC-2400	**30**	**4.114285714**	**3.91875**	**12.99375**

Table 4-9: Initial propulsion system analysis

Run Time: We used the battery pack mah rating to determine how long the needed current could be delivered in minutes. This is summarized by the following formula:

Duration= 60 x (capacity (mah) / 1000) / current (amp)

Motors	Drive	Type of Cell	Number of Cells	Propeller Sizes	Battery weight (oz)	Max Airspeed (ft/s)	Max Run Time (m:s)
Astro Cobalt 60 FAI 6T#20 #660	Geared	RC-2400	30	20x10	2.08	74.8	5:10
Astro Cobalt 60 11T#23 #661	Direct drive	RC-2400	30	14x10	2.08	86.5	3:44
Graupner Ultra 3500-8 30V #6346	Direct drive	RC-2400	27	14x10	2.08	82.1	4:27

Table 4-10: MotoCalc Analysis Summary

Thrust (lb): Thrust was determined by dividing the output power by airspeed.

Those preliminary calculations, even though not a hundred percent accurate, allowed the team to get a rough idea of the kind of system combinations able to sustain the plane and therefore narrow the large range of power systems possible with the five-pound battery limit.

4.8.2. Motocalc analysis

Once the initial calculations were done, Motocalc was used to evaluate the results we obtained from the initial calculations and also consider other alternatives that the program might be able to provide us with. The power systems providing the most promising results were selected for further analysis. The parameters investigated included, motors, the gearing ratio, and the propeller dimension, the type of cell and the quantity. Based on the team's requirement and the competition's restrictions, the best power systems using motors from the Astroflight and Graupner family were selected for further analysis. These motors selected are: an Astro Cobalt 60 11T#23 #661 and a Graupner Ultra 3500-8 30V #6346 motor. An Astro Cobalt 60 FAI 6T#20 #660 motor with a gear ratio of 2.7:1 from previous Cessna/ONR DBF entries was also analyzed. The summary of analysis is presented in Table 4-10 above.

The power system using an Astro Cobalt 60 11T#23 #660 geared gave the best-run time of 5:10. However, the average maximum airspeed of 74.8 feet per second is a little less than the requirement maximum airspeed of 80.7 ft/s.

The power system using the Astro Cobalt 60 11T#23 #661 direct drive assured us the most excellent airspeed of 86.5 feet per second; however its maximum run time of 3:44, leaves no room for error based on the run time requirement of 3.4 minutes.

Finally, the power system using the Graupner Ultra 3500-8 30V #6346 predicted an acceptable airspeed of 82.1 feet per second along with a run time of 4:27. Most importantly, it would be able to provide such a performance with only 27 cell of type RC-2400, and this will reduce the battery pack weight by 6.27 ounces.

The Astro Cobalt 60 11T#23 #661 direct drive and the Graupner Ultra 3500-8 30V #6346 systems give the best overall results, based on run time and speed requirements. However, the Graupner Ultra 3500-8 30V #6346 was eliminated for further analysis, because it is less accessible than both the Astro Cobalt 60 11T#23 #661 direct drive and the Astro Cobalt 60 11T#23 #660. Furthermore, analysis of the thrust provided by each motor as a function of speed, shows that the Graupner Ultra 3500-8 30V #6346 provides the lowest values (see figure below). The following graph contrast the thrusts provided by each motors:

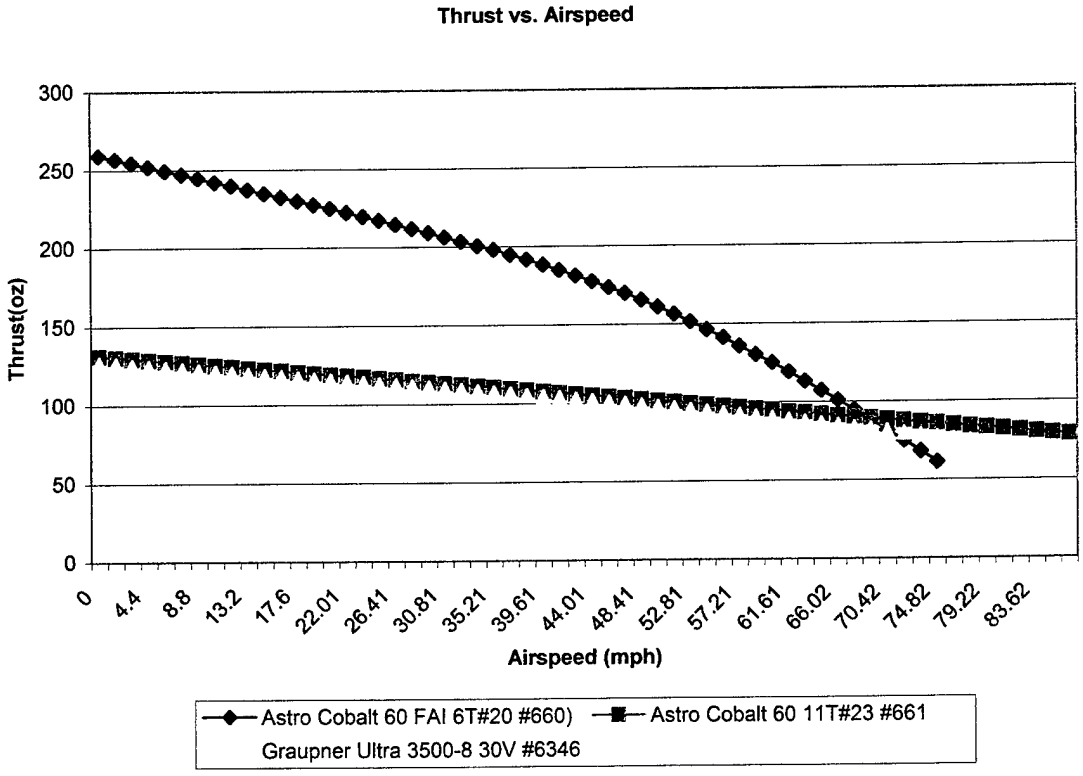


Figure 4-7: Thrust versus airspeed curves of the three motors.

The analysis of the remaining motor systems, Astro Cobalt 60 11T#23 #661 direct drive and the Astro Cobalt 60 11T#23 #660 geared will be reserved for the test section, where both systems will be tested statically and dynamically for verification of these results.

4.9 Performance

The performance of an aircraft is essentially a statement of its capabilities and a different selection of these will be specified dependent on the needs that need to be placed on the airplane. For the team the different performance features served as design criteria or at the very least desirable objectives, around which the plane would be designed. Analyses of the characteristics, which would maximize the airplane, were all performed.

Simple assumptions were made, which allowed the team to perform basic performance estimates and develop methods of analysis and calculations. The team performed the following performance calculations in our attempt to define our airplane as accurately as possible:

- Take-off performance
- Lift and Drag Calculations
- Power requirements

4.9.1 Take-off performance. For take-off to be successful, there is a minimum required take-off velocity which much first be achieved. This minimum take-off velocity, V_{stall} was assumed and considered to be proportional to the stall velocity, V_s.

Alternatively, V_{stall} was calculated directly, as a function of:

The weight of the plane, W

Wing area, S

The maximum lift coefficient, C_{Lmax}

Air density at an altitude of 2000 ft, ρ

$$V_{stall} = 1.2(2W/ \rho * S * C_{Lmax})^{1/2}$$

Using this approach, the team received a result of 51.27 ft/s for our minimum take-off velocity. This number was then used in the calculations of multiple parameters, which were all apart of our design specifications.

4.10 Lift and Drag Calculations.

In our performance analyzes, the team utilized the following equations to achieve our lift values, which were then used for further calculations

$$L = \tfrac{1}{2} C_l * \rho * V_{stall} * S$$

The values used for C_l were extrapolated into 3D from the ones obtained from our analyses of the airfoil in XLR5. These values were dependent on alpha and as such our lift is also dependent upon the angle of attack, alpha (α).

The drag calculations, which were then performed, were also dependent upon the achieved V_{stall}, as well as the previously acquired the values for C_d; both of which were achieved.

$$D = \tfrac{1}{2} C_{Dp} * \rho * V^2 * S$$

Where V here is taken as our maximum velocity. This is done to take into consideration, the largest possible drag, which we would encounter, as can seen from the relationship between drag and velocity as shown in the previous section.

These analyses were also performed for a range of different values of V, so an understanding of the relationship between the various design parameters could be established and understood.

4.10.1 Power Requirements. Power requirements refer to the required thrust, T_{req}, which is the minimum thrust that is needed to ensure the plane takes off. This is a major concern for our design, because it is essential there be enough power for our plane to take-off. Our team analyzed the thrust requirements for the plane as a function of both aircraft weight and drag, and in addition to that, an analysis of thrust as we varied the take-off angle was also conducted. It was essential that we compare the thrust needs of the plane at varying angle for take-off so as to ensure the plane never reached its stall angle. The relationship between thrust and aircraft weight and drag is given by the following equation:

$$T_{req} = D + W*\sin\theta$$

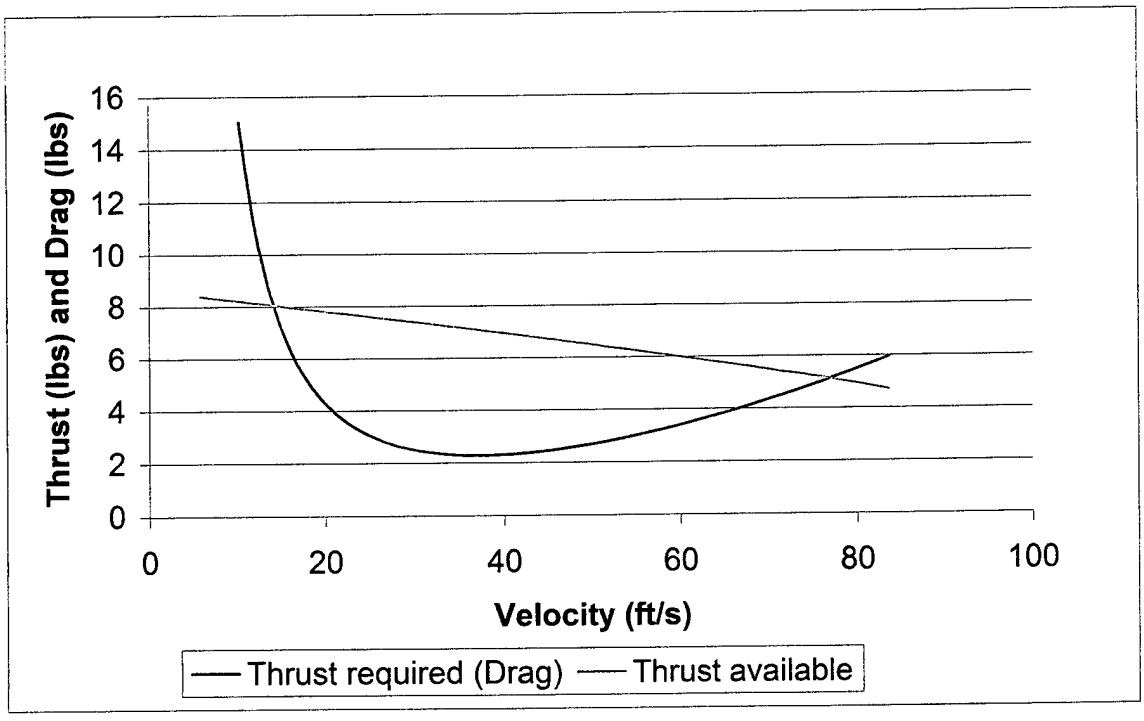


Figure 4.8 Plot of thrust required and thrust available (Astro 60 direct drive) versus velocity

Figure 4.7 shows the relationship between the required drag available for flight and the parabolic drag curve, the two were super-imposed upon each other to illustrate the fact that over the aircraft's velocity interval, there is sufficient thrust to maintain flight. This fact is crucial to accomplishing the design goals of the plane.

4.10.2 Excess Power. Another important parameter to understand is the excess power the engine produces. The excess power is given as:

Excess Power = $(T_{req} - D)*V_{stall}$

Since T_{req} already includes a drag component, which is subtracted, it is possible to rewrite the equation in the following manner.

Excess Power = $V_{stall}*\sin\theta$

Just as was done with thrust, the team also calculated the excess power for the aircraft for different values of θ. A single value for the excess power at was obtained, however varying θ, does produce a different number for the excess power. This is because the plane would be acting at different drag and lift numbers.

4.10.3 Rate of Climb. The rate of climb of the aircraft, as its name suggests is the rate at which the plane increases altitude. The rate of climb was obtained using the following equation:

R/C = Excess Power/W

which is further reduced to:

R/C = $V_{stall}*\sin\theta$

Since there is a fixed value for V_{stall}, the R/C is only dependent upon θ.

4.11 Optimization

It is essential to optimize the performance of the aircraft, so as to achieve maximum performance. For this purpose the team decided it was best to optimize the total maximum range of the aircraft as well as the its endurance.

4.11.1 Range

The total range of an airplane consists of its climb, cruise and descent segments, where the cruise segments are usually the longest. To maximize the aircraft range, the team noted that a maximization of the L/D ratio must be accomplished, this is obtained when drag is minimized. But more importantly focus was placed on finding the C_l, which would yield maximum range.

$C_{l\ max\ range} = 3^{1/2}/3\ C_{l(L/D)max}$

$C_{l(L/D)max} = (C_{D0}*\pi*AR*e)^{1/2}$

This is the C_l, where drag is minimized.

$C_{l\ max\ range} = 0.934516$

Flying at this optimized C_l would give the aircraft its maximum range.

4.11.2 Endurance

Another performance criterion, which was maximized is the endurance, it was seen that this must be performed if our aircraft was to accomplish the mission at optimal performance.

For the endurance the team looked at the Velocity required to maintain maximum the Lift to Drag ration, $V_{(L/D)max}$.

$V_{(L/D)max} = (2W/\rho*S)^{1/2}*(k/C_{D0})^{1/4}$

$= 37.233808$ ft/s

5.0. Detail Design Report

5.1. Component Selection and Systems Architecture:

At this stage of the design, the wings, the fuselage, the horizontal stabilizer, the vertical stabilizer and the propulsion system have been specified. The detail design of the airplane now requires the design and selection of the internal structure and components of the aircraft. The internal structures are designed to withstand the loads that occur at different phases of flight. For the internal structure to be able to withstand the varying loads, it is designed based on a load limit.

5.1.1. Load Limit: The load limit is the maximum load factor that the airplane will endure in its flight envelope. To determine it, the load factors at different flight phases were compared. The flight phases considered are sustained turn rate where the speed and altitude are maintained so that thrust equals the drag, climb, and high angle of attack whereby an instantaneous change of angle of attack can result in a high load factor. For the sustained turn rate with a thrust of about 5.7 pounds and at a flight speed of 65.6 ft/s, the load factor is 2. A climb speed and angle of 50 ft/s and 12 degrees, at a thrust setting of 6.5 pounds the load factor comes to 1.3. The third flight phase considered is the highest angle of attack at a flight speed of 65.6 ft/s. The load factor is highest here, and it has a value of 3.5. The table below summarizes these results.

Flight Phase	Load Factor
Sustained Turn Rate	2
Climb	1.3
High Angle of Attack	3.5

Table 5.1: Load factors at different flight phases

The load limit is determined from these values by selecting the maximum and multiplying it by a factor of safety of 1.5. Therefore, the load limit is 5.25.

5.1.2. Wing Structure: The loads on the wings are made up of the aerodynamic lift and drag forces and structural elements. The structures of the wings usually have an internal component and skin that resists torsion and a spar that resists the bending moments. Ribs or foam cores are used to resist torsion. From the manufacturing processes investigated foam core was chosen over ribs. Spars are the primary structure in wings that resist the bending moments. The materials that can be used for spars are spruce wood or carbon fiber reinforced-balsa wood. The later was chosen based on the anticipated strength to weight ratio. An analysis was done in Pro/ Mechanica to visualize the stresses the spar will have to resist. Figure 6.1 shows the static stresses in a spar undergoing the wing tip test. Fixing the left end of the spar simulated the tip test and a load of 2.5 times the loaded weight of the aircraft was placed at the other tip. The image on the left shows that most of the bending stresses are at the top of the spar, and the image on the right indicates there are also shearing stresses within the spar where it is fixed.

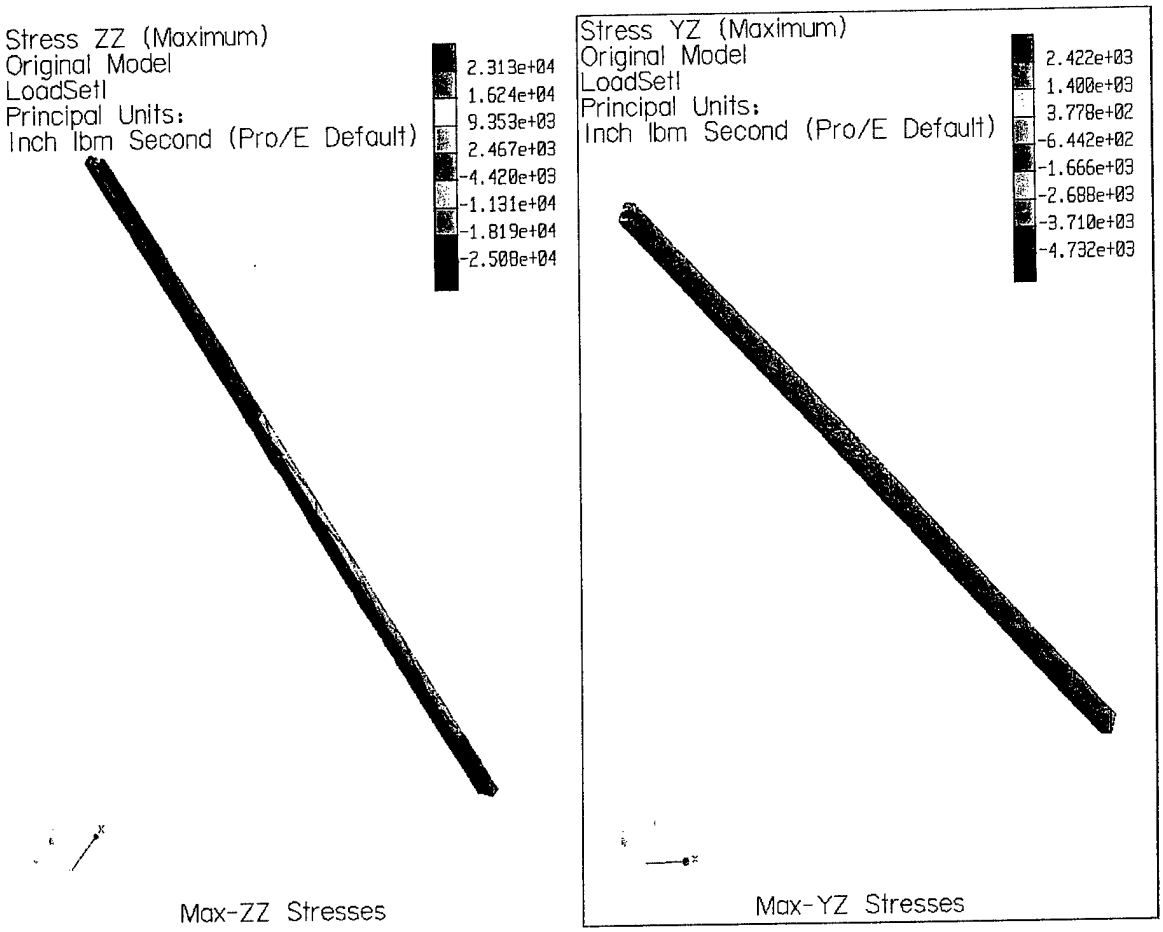


Figure 5.1: Static stress results for wing tip test in Pro/ Mechanica.

The figure above on the spar indicates the rational of curing carbon fiber reinforcement around a balsa wood cure. The top layer of carbon fiber will take the bending stresses, while the shearing stress at the connection point can be accounted for in the wing-fuselage connection.

5.1.3. The Wing-Fuselage Connection: The requirement that the aircraft must fit in the assembly box warranted the use of a specialized wing-fuselage connection. The requirements for this connection is that it must provide the dihedral of the wing, fit in the upper chamber of the fuselage above the tank, and be able to fix the spar to the fuselage. The materials used where bi-directional carbon fiber, because of its strength to weight ratio, layered three times within a balsa wood form. The three layers gave this component superior strength.

5.1.4. The Landing Gear: The selection of the landing gear type took place during the conceptual design phase. A tail-dragger configuration was selected. In the detailed design of the landing gear, a choice of a two-piece main gear rather than a one piece was made because of the ease of disassembly. Also, the belly of the fuselage, which has an elliptical cross-section, would have made a one-piece main gear harder to attach. The length of the landing gears was

determined by ensuring that the tip of the propeller will always be two inches from the ground and the leading edge of the wing will not be have more than 7 degrees incidence when all three wheels are sitting on the ground. The carbon fiber was the material of choice for the main gear due to its high strength to weight ratio over Aluminum. The wheel track or the distance between the two main gear tires was determined by scaling it to that of a Cessna 182 with respect to wingspan.

5.1.5. Fuselage: The fuselage can be considered to be supported at the location of the center of lift of the main wing. Therefore, the loads on the fuselage structure are then due to shear force and bending moments about that point. The various components that add to the load are the weights of the payload, the batteries, the wing structure, motor, fuselage structure, and tail control lift force. Table 5.2 gives the load break down and the resulting moments about the location of the wing aerodynamic center.

Load Type	Magnitude(lbs)	(x/L)start	(x/L)end	(x/L)resultant	M@xCL (ft-lbs)
Batt-1	2.625	0.10156	0.198	0.14978	-0.058
Batt-2	1.3125	0.2135	0.255	0.23425	0.082
Fuse Struct.	2.3	0	1	0.5	0.755
Motor	2	0	0.0833	0.04165	-0.26
Wing Struct.	2	0.1146	0.344	0.2293	0.115
Payload	6.7	0.12	0.203	0.1615	-0.07
Tail Struct.	1	0.8146	1	0.9073	0.735
Tail Lift	-1.884806367	0.861	0.861	0.861	-0.058

Table 5.2: Load Summary for fuselage

To withstand these loads, the type of structure used for the fuselage is a semi-monocoque. The shell withstands tension and is made out of two layers of bi-directional carbon fiber. The semi-monocoque structure uses bulkheads and longerons to withstand the compressive buckling. Four bulkheads are used in total; each bulkhead is 5/16 inches thick and has approximately 84 square inches of surface area. Balsa wood was reinforced with fiberglass and carbon fiber to create the bulkheads. The first bulkhead is connected to the motor its reinforcement is carbon fiber. The next two bulkhead serves to hold the motors, one pack on each, and the tank in-between the two. They are connected by two longerons located on each side of the internal fuselage. Each longeron is wrapped with carbon fiber thread for reinforcement. The bulkheads are mounted onto the longeron through drill holes on each bulkhead. The figure below illustrates the architecture.

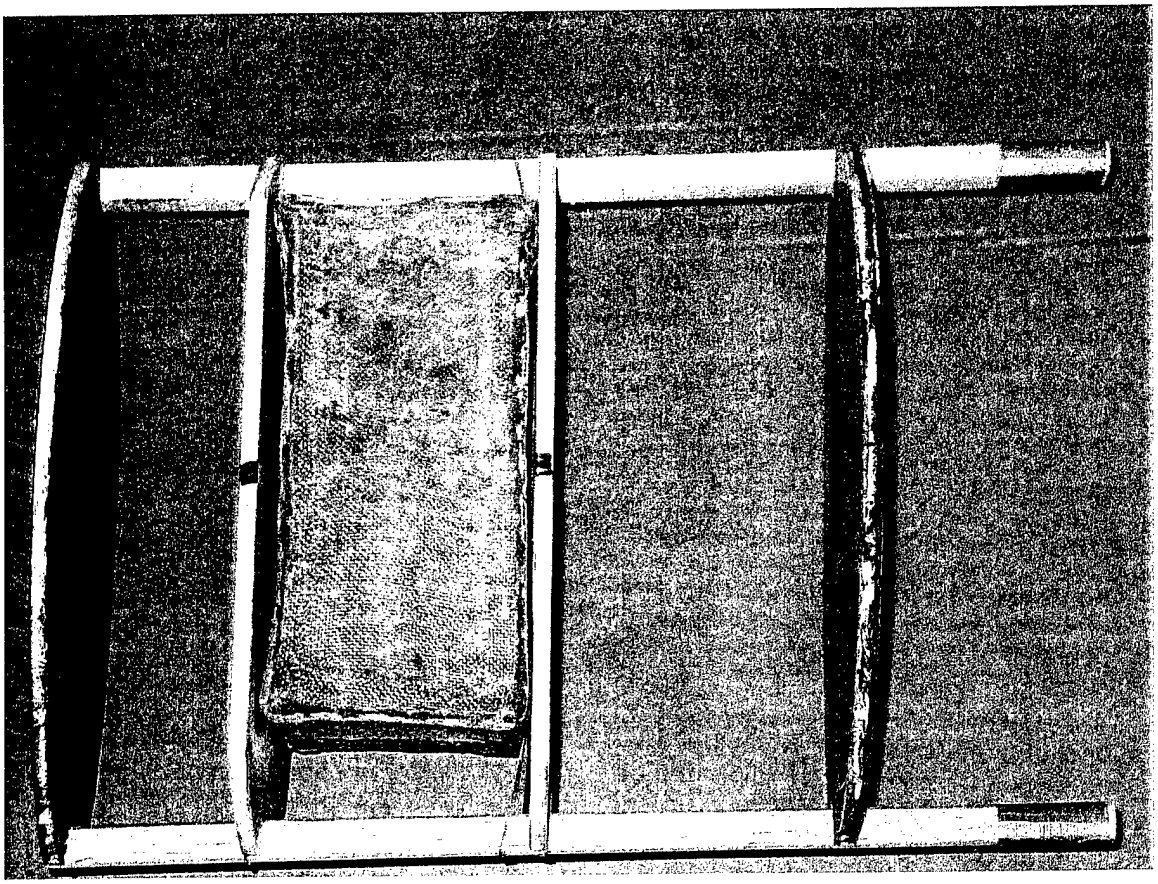


Figure 6.2: Architecture and placement of water tank, bulkheads, and longerons. The longerons are cut at the intersection with the last bulkhead.

5.1.6. Servos Selection

The servos were chosen through a process that will eliminate the servos that does not meet the needs of the group. The most obvious need is the compatibility of the controller to the receiver and the receiver to the servos. The criteria that were used in the decision of the compatible servos started off with the dimensions of the servos and after the dimensions were chosen, the servos was eradicated by weight. The judgment of the servos proceeded on to the amount of ounces of torque per inch and finally the speed of the servo.

Picking the size of the servos was crucial to the design. The chosen sizes were based on where the servos are going to be placed (of course, those positions was known not to be permanent, there may be some change but hopefully not too drastic). The heights of the servos were the key concern of this decision-making. If the height was too large then it was not one of the considerations because of the discreetness. The size of the servo will meet all the needs of the different placements thus the concern for size. The more inconspicuous the servo is, the less it will affect the aerodynamics of the model. The servo for the wing will be attached to the last bulkhead along with the servo for the tank. The servo for the ruder and the landing gear will be

attached to the tail boom along with the servo that is attached to the elevator. After searching for servos that will fit the roughly drafted design, the weight was then considered. If the weight was great enough to present a significant disposition the current center of gravity, then it was discarded from servos made available to use. The previous servo weights from earlier competition models were also taken into consideration.

The amounts of ounces of torque per inch were also another critical factor in the result of the servos. If the servos were not powerful enough to put the any part of the tail, wings, or landing gear in motion then it was believed not to be a practical servo to use. The servos also needed to be able to react to a command in a good amount of time, meaning it has to be able to execute a command in fashion that will fulfill the duties of the function in the timeframe needed.

5.1.7. Receiver Selection

On deciding which receiver to use for the airplane, many compatible receivers were found and one by one where then eliminated, until one was left, which was best suited for the job at hand. All the receivers that were found to be compatible with the transmitter (T9C) were in the tower hobbies website (It was a group decision to order most of the equipment from tower hobbies).

All Futaba receivers that are 8 or 9-channel were compatible with the transmitter to be used. All the 9-channel receivers were eliminated because they cost much more than the 8-channel receivers. After that, other receivers were eliminated because they did not have the frequency that was assigned to the group. Finally, the rest of the receivers were eliminated concerning specific dimensions (height, width, length, weight), until the last receiver left was the best one for the job at hand. The servo also would be able to attach to the last bulkhead so it would be close to half the servos and be close enough so the other servos on the tail boom would not need such long extension cords to connect it to the receiver.

After choosing the receiver, a crystal had to be bought separately. The website showed a crystal that is compatible with the receiver, but the decision was made to call up tower hobbies and ask them if it was possible to acquire a different crystal which was still compatible with the transmitter to be used. It turned out it was possible, and in the end, a better more proficient crystal was obtained.

Choosing a module was not much of a challenge. Tower hobbies only had one module compatible with the frequency to be used therefore that module was chosen.

5.1.8 Systems Architecture:

Figure 6.2 shows the internal architecture of the fuselage. The rear bulkhead, that is the one on the right above, serves as a mount for the servo that controls the water tank valve, the servo that controls both ailerons, the receiver, and the battery for the receiver. The motor mount is mounted in front of the first bulkhead, and the speed controller is mounted behind the same

bulkhead. The battery packs are separated into two for center of gravity purposes. The first battery pack is attached to the second bulkhead, and the second battery pack is attached to the third bulkhead. As shown in the figure 6.2, the tank is sandwiched between the second bulkhead and the third bulkhead.

The servo that controls the rudder and the tail-dragger is mounted in the tail-boom and the wires run through the tail boom. The servo that controls the elevator is mounted beneath the horizontal stabilizer.

5.2. Final Rated Aircraft Cost:

A Manufactures Empty Weight Multiplier	300 $
B Rated Engine Power Multiplier	1500 $
C Manufacturing Empty Weight	20 $/hour
# Engines	1
Battery Weight	3.91875 lb
WBS 1.0 Wings	
# Wings	1
Wing Area	7.058333 ft^2
Control function multiplier (Ailerons)	1
WBS 2.0 Fuselage	
Fuselage Volume	3.254563 ft^3
WBS 3.0 Empennage	
Vertical surface (Struts) with no active control	0
Vertical surface with active control	1
Horizontal surface with active control	1
WBS 4.0 Flight Systems	
# Servos	4
MFHR=∑WBS (Work Breakdown Structure)	
WBS 1.0 Wings	75.58333
WBS 2.0 Fuselage	65.09125

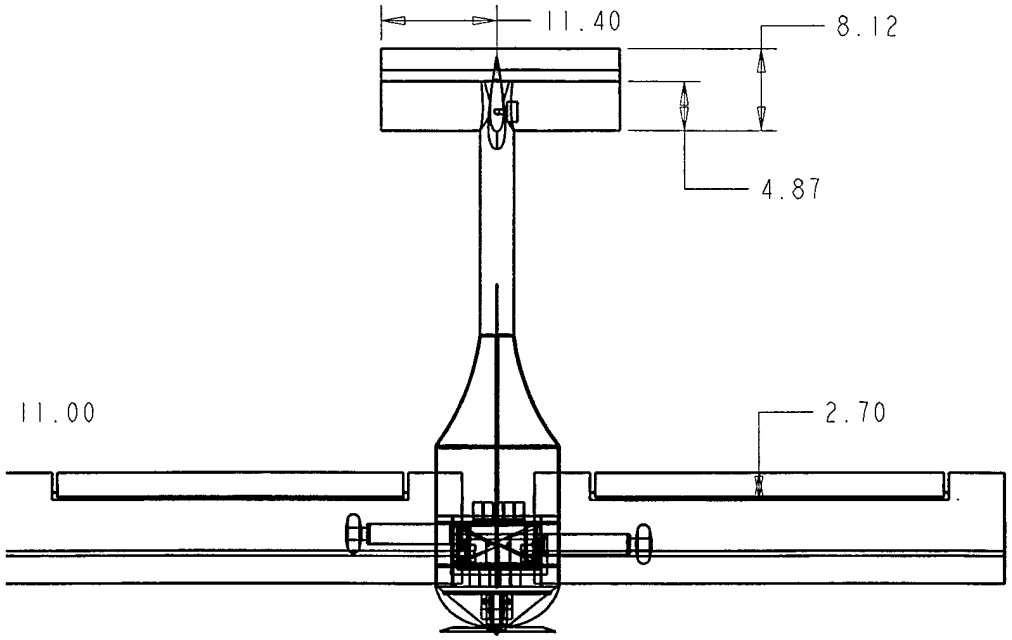
WBS 3.0 Empennage	20
WBS 4.0 Flight Systems	20
MEW (Manufacturing Empty Weight)	11 lb
REP (Rated Engine Power)	3.91875
MFHR (Manufacturing Man Hours)	180.6746
FINAL COST	12.79162 $(Thousands)

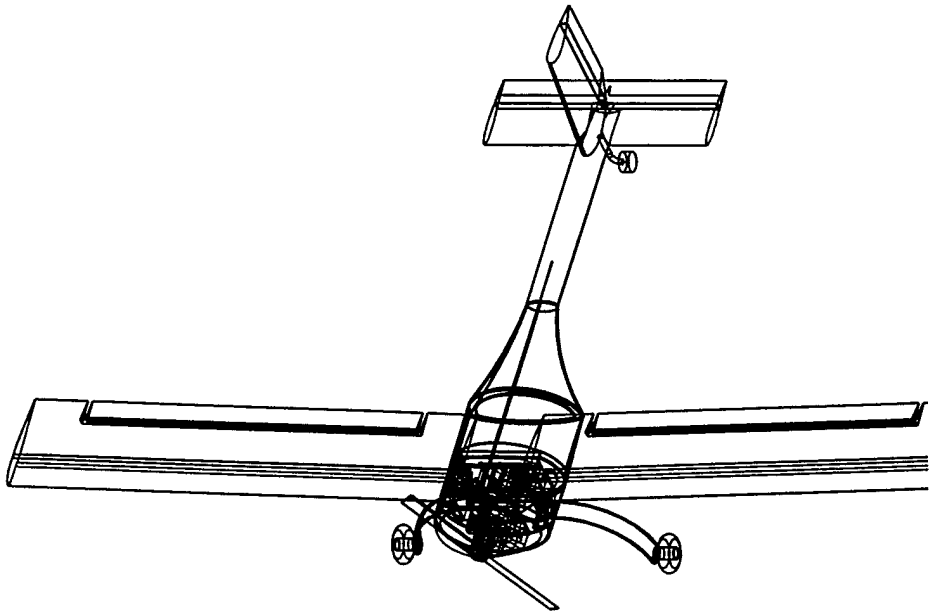
5.3. Sized Aircraft Table:

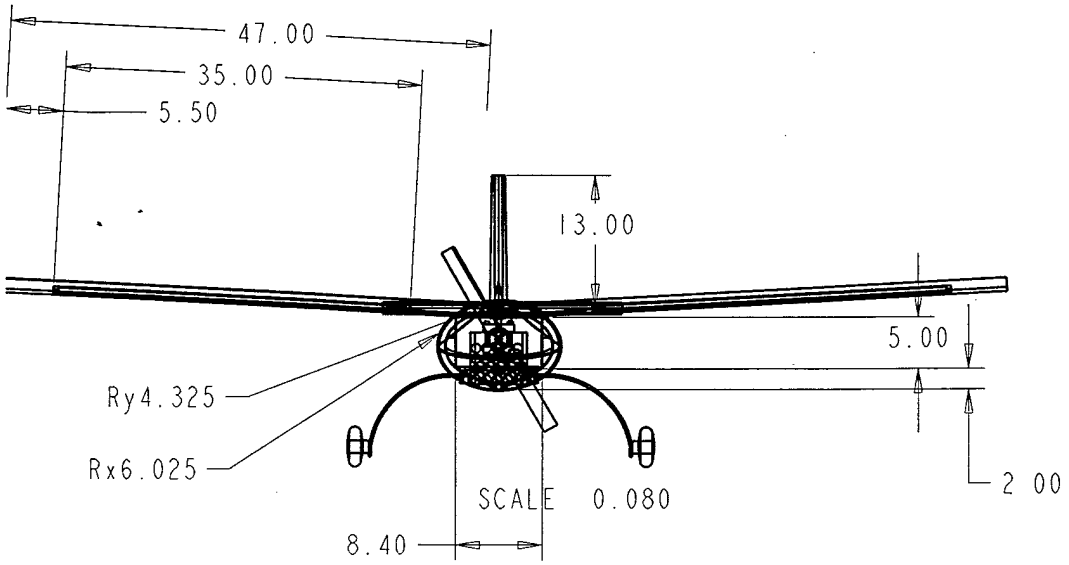
Geometry	
Length	
Height	
Span	7.6 ft
Wing Area	7.0 ft^2
Aspect Ratio	8.3
Horizontal tail volume	0.504
Vertical tail volume	0.044
Performance	
$(C_L)_{max}$	1.75
$(L/D)_{max}$	27.3
$(R/C)_{max}$	37 ft/s
V_{stall}	42.7 ft/s
V_{max}	76 ft/s
Take-off length (empty)	14 ft
Take-off length (gross)	34 ft

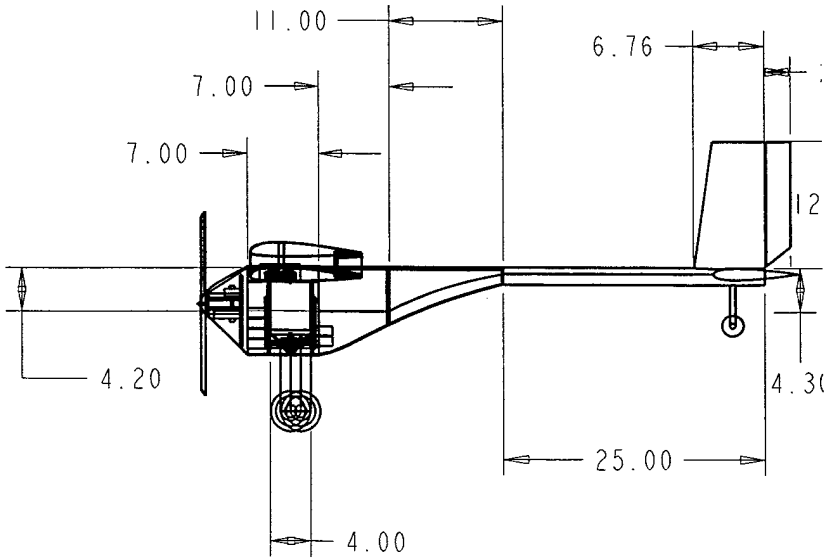
Systems	
Radio	Futaba 9CAP
Servos for:	
Ailerons	Futaba S9151
Elevator	Futaba S3002
Rudder	Futaba S9650
Tank hatch	Futaba S9151
Battery Config.	30 cell, 2400 mAHr.
Motor	Astro Flight 60 #661
Propeller	14 by 10
Gear ratio	None
Weight Statement (lb)	
Airframe	4
Propulsion system	6
Control system	0.5
Payload system	0.5
Payload	6.6
Empty weight	11.0
Gross weight	17.6

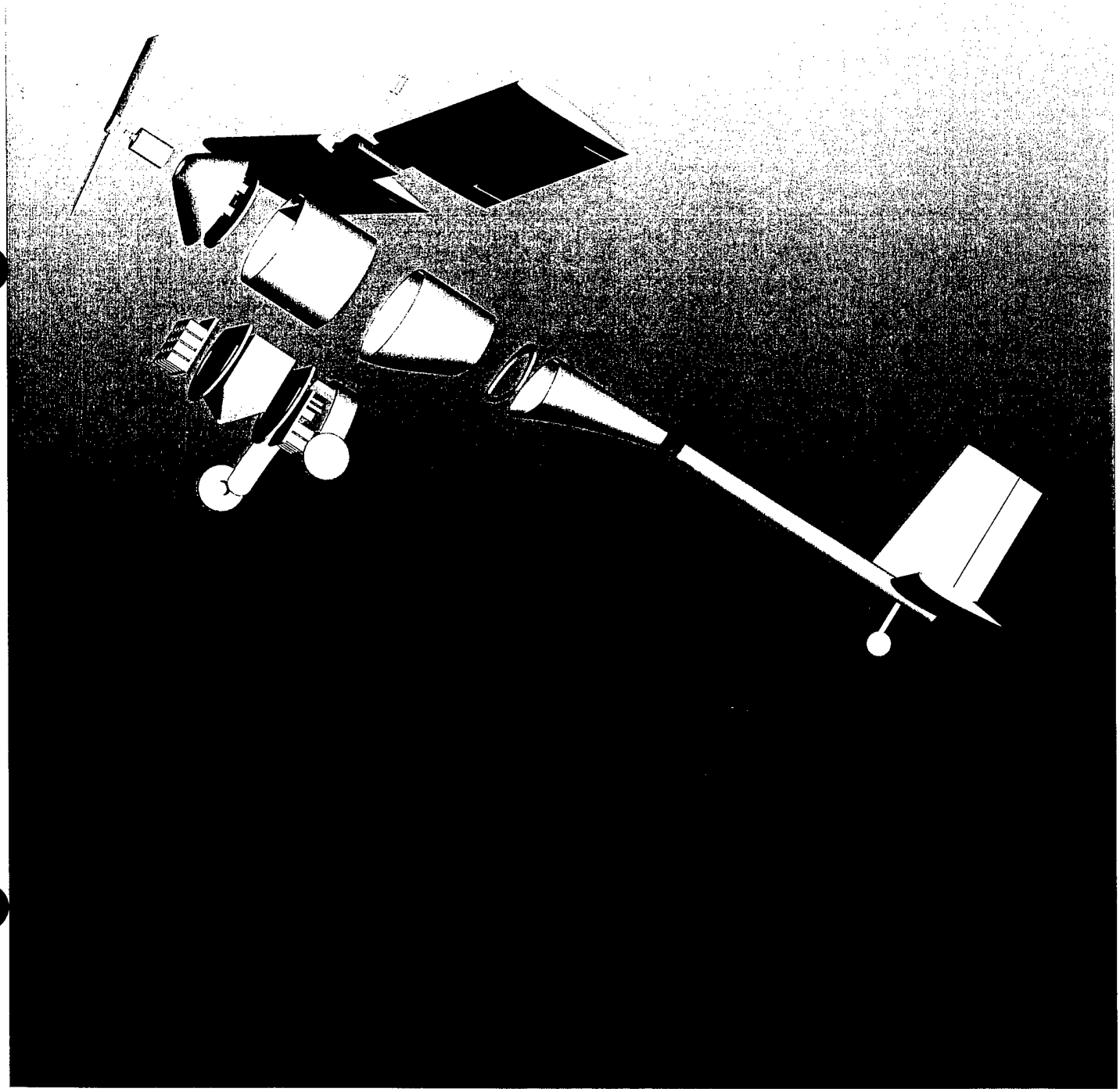
5.4. Drawing Package: The drawing package includes a 3-view drawing of the of the airplane, and exploded 3-D view of the airplane.

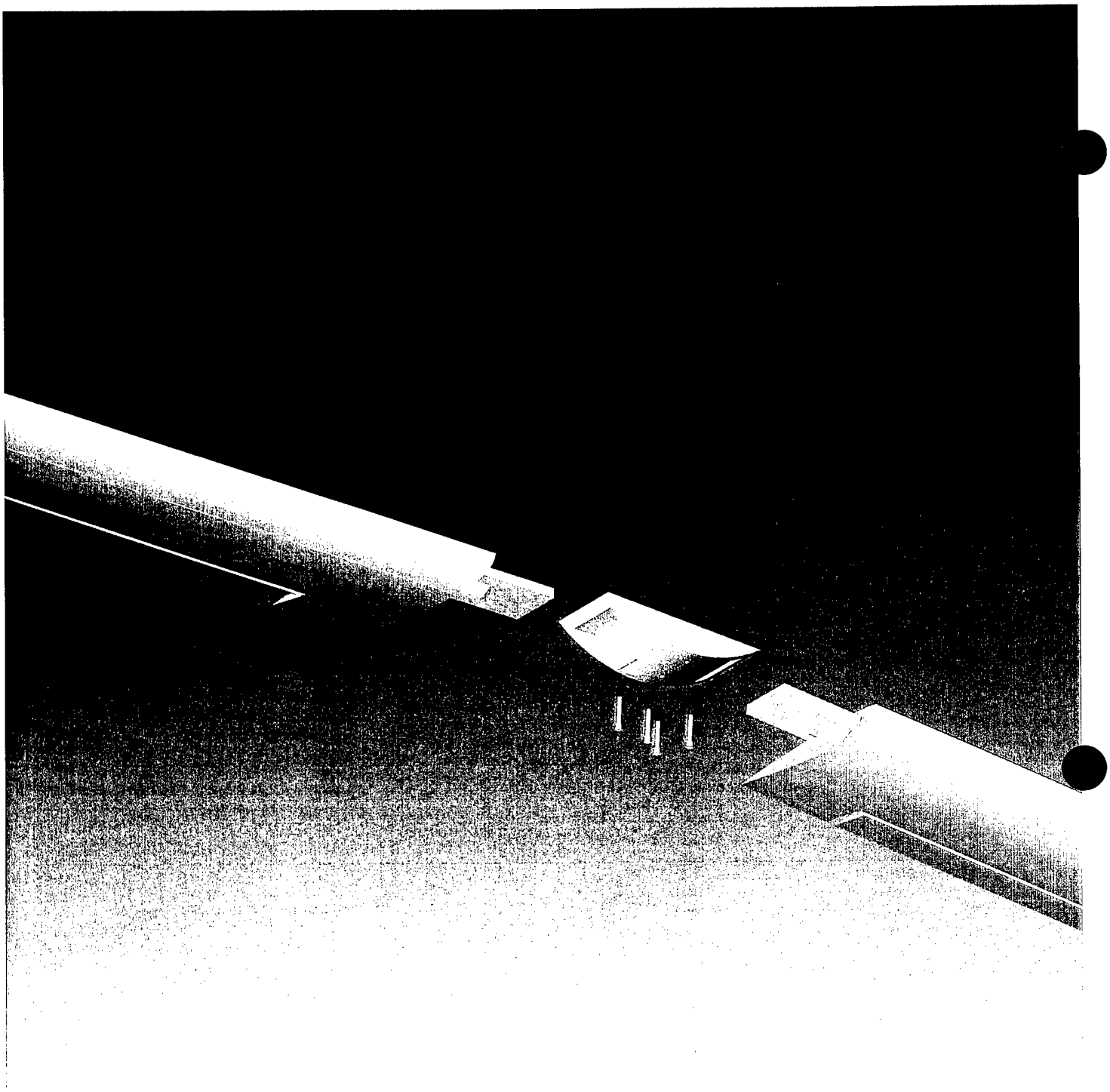

11.40
8.12
4.87
11.00
2.70


SCALE 0.080


47.00
35.00
5.50
13.00
5.00
Ry4.325
Rx6.025
SCALE 0.080
2 00
8.40


11.00
6.76
7.00
7.00
4.20
25.00
4.00





6. 0. Manufacturing Plan and Processes:

This report documents the manufacturing process selected for the manufacture of major components and assemblies of the final design. It includes the figures of merit used to screen competing manufacturing concepts as well as a manufacturing milestones chart.

6.1. Figures of Merit for Material Selection

A figure of Merit table was used for the selection of materials used in the manufacturing of the aircraft. A weighted average is used in order to give more points to the categories that were considered more important. The scoring of the FOM categories was done as follows.

Availability – Scored on a scale of 1 to 2. 2. 2.2 is given to an item readily at hand. 1 was given to an item that had to be ordered.

Required skill level – This category was scaled from 1 to 4, 1 being highest skill level required.

Cost – This category was scaled from 1 to 4, 1 being the most expensive.

Strength to Weight ratio – This category was deemed the most important and given a scale of 1 to 5. The tank, in which strength was not a significant factor, was given the same grade for all materials considered negating this category.

Assembly difficulty – This category was scaled from 1 to 4, 1 being the most difficult.

Time to complete – This category was scaled from 1 to 4, 1 being the longest to complete.

When all the scores were calculated they were averaged, and the manufacturing process with the highest score in its category was chosen.

		Availability	Required Skill level	Cost	Strength weight ratio	Assembly difficulty	Time to complete	Average
Spars	balsa core carbon fiber	1	3	3	5	4	3	3.1667
	aluminum (commercial)	1	4	1	3	4	4	2.8333
	spruce	1	3	4	2	4	3	2.8333
Bulkheads	balsa	2	4	4	1	4	4	3.1667
	balsa core carbon fiber	1	4	3	5	3	4	3.3333
	balsa core fiberglass	1	4	4	3	3	4	3.1667
Landing Gear	aluminum (commercial)	1	4	3	3	4	4	3.1667
	carbon fiber	1	3	4	5	3	4	3.3333
Fuselage skin	fiberglass	1	3	3	3	3	4	2.8333
	carbon fiber	1	3	2	5	3	4	3
	monokote (balsa core)	2	2	4	4	3	2	2.8333

Wings and tail	balsa frame	2	3	4	3	4	3	3.1667
	foam core fiberglass	1	4	4	3	3	4	3.1667
	foam core carbon fiber	1	4	3	5	3	4	3.3333
Tank	plastic (commercial)	1	4	2	5	4	4	3.3333
	fiberglass	1	3	4	5	4	4	3.5
	carbon fiber	1	3	3	5	2	3	2.8333
Wing Box	balsa core carbon fiber	1	4	3	5	3	3	3.1667
	spruce core carbon fiber	1	4	3	4	3	3	3
Tail Boom	foam core fiberglass	1	4	4	3	4	4	3.3333
	foam core carbon fiber	1	4	3	5	4	4	3.5
	balsa frame	2	3	4	2	3	3	2.8333

6.2. Manufacturing Processes: The processes used to manufacture major components are described below.

6.2.1. The tank: The material chosen for fabrication of the tank was fiberglass. Initially we considered a commercial plastic container and carbon fiber also but fiberglass was the best material to fit our needs. The reason we chose fiberglass was because of its ability to be made in any shape or size we desired, its lightweight and its low cost. The fact that the tank would not be subjected to any significant forces ruled out carbon fiber or any other unnecessarily strong, heavy, or expensive material. The tank was to be placed in the center of gravity of the aircraft so as not to create a CG shift when the water is emptied during flight.

Once the design was chosen, fabrication began. We decided to use a mold less composite method due to the simple geometry and small size of the tank. We started by creating a plug of the bottom of the tank out of Styrofoam. When the desired size and shape was attained the foam was coated in mold release wax and 2 layers of liquid mold release agent. 1 layer of fiberglass was laid over the plug and allowed to dry. When drying was complete we pulled the foam out of the fiberglass shell. The plug was ruined but we made it expecting not to use it again. We constructed the top of the tank in the same fashion and finally wrapped the top and bottom of the tanks with one sheet of fiberglass to seal it.

6.2.2. Water exit Tube: The water exit tube is to be constructed of a simple rubber tube. The tube will stick through a hole in the bottom of the fuselage. A spring hinge with a crank arm will be used to open the mechanism to release the water. The rubber tubing will act as a gasket to keep the water from spilling.

6.2.3. Air Vent Tube: The vent tube will be a small diameter rubber tube exiting through the top of the airplane. At the top of the tube a plastic elbow connector will be used to align the tube into the oncoming air stream. This will give a slight pressure assist to help empty the tank more quickly.

6.2.4. Landing Gear: Carbon fiber was chosen as the material for the landing gear according to our material selection method. Both one-piece and two-piece landing gear were considered. Due to the compound curvature of our fuselage we decided that mounting the landing gear on the surface of it would be unnecessarily complicated, therefore the two-piece design was chosen. We then had to decide between using a u-shaped and a flat cross section for the gear. A flat cross section was chosen because of the difficulty in mounting a curved cross section to the interior structure of the plane.

The landing gear was constructed using a mold less composite method. As with the tank, this method was chosen due to the simple geometry and small size of the parts. A foam block was formed on the shape and size of the gear. Since both would be identical we decided to use the same plug to make both and then cut them in half. The foam was covered in plaster and painted. It was then coated with mold release wax and two layers of liquid mold release agent. The part was made using four layers of carbon fiber fabric and epoxy.

The landing gear had to be removable to be able to fit the plane into the two-foot by four foot box. The connection method we used was as follows. Two pieces of balsa covered with carbon fiber were attached between two of the bulkheads. A slot was cut in each of them to allow the landing gear to slide in 1½ inch. Two holes were then drilled through the wood and landing gear in which two screws would be used to hold the gear in place. The fuselage was slotted and screw holes were drilled in it to allow the screws to be removed when taking off the landing gear.

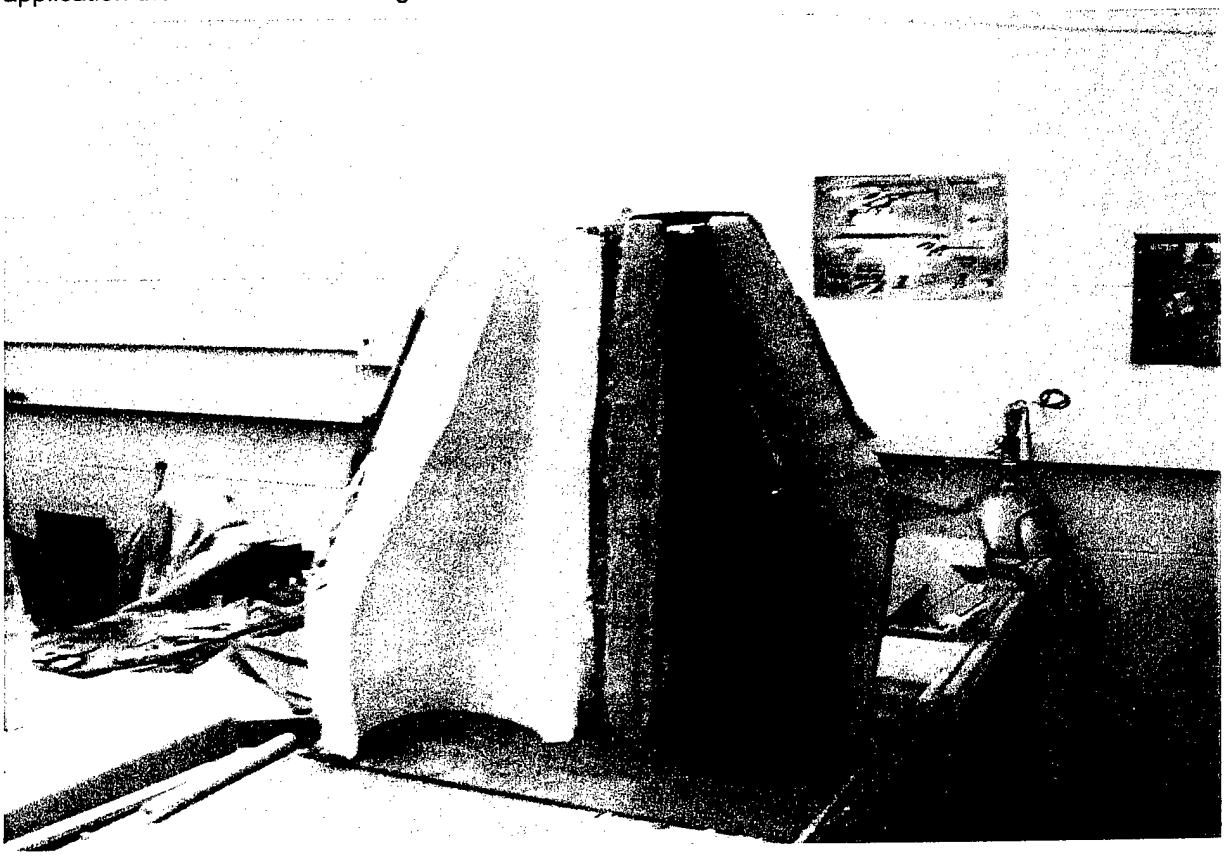
6.2.5. Fuselage: In order to make a fuselage of the strong lightweight carbon fiber and fiberglass materials, we had to create a mold that matched the aircraft's design specifications. To create the mold we first needed to create a plug, which matched the fuselage's geometry to actual scale. Once the plug was completed we could then create the mold and subsequently lay-up the reinforced fiber shell.

To begin the process of creating the plug we first printed full size cross sections of the fuselage. The cross sections were taken every two inches beginning at the nose-fuselage connection and ending at the rear where the tail boom would be connected. We decided to make the nose cross-sections every one-inch, as the nose was shorter and had a steeper slope. Once we attained the printouts we then began to create disks of polystyrene foam two inches thick to match the printed cross sections. Paper cross-sections were replaced by cardboard, as this would give us a clearer guide to where sanding should stop. Adhering the disks to one another, in sequence, we attained the general shape the aircraft would take. The foam was then sanded to

the appropriate geometry. Initially we had reservations about using cardboard as a sanding guide. We found it to add an additional 1.65 inched to the total length of the fuselage. We found this additional length to be negligible, as it would be spread out through the length of the aircraft.

Once the foam plug was finished, a thin layer of plaster was added to coat the surface and make it smooth. The plug was finished with ten coats of spray sealant and was prepared to be molded with three layers of parting wax and two coats of PVA releasing film. We first attempted to cast the mold using a mixture of plaster, plaster of Paris, and shredded fibers. We found that this process took too long and did not yield good results. The plaster compounds took too long to cure and set up. We also found that the moisture in the plaster softened the plug itself and made it weak. The first plaster mold that was pulled failed. This method was abandoned at once and we took to a new strategy.

Polyurethane expanding foam proved to be the best remedy to our problem. Creating two cradle-like boxes we caste the left and right halves of the fuselage using the expanding foam. This method proved to be successful in that it took only several minutes for the foam to harden and was completely sand-able to a desired shape. To complete the molds we added plaster to fill in the gaps and air pockets the expanding foam did not fill. The same process was used for the nose. After sealing with spray sealant we were ready to cast our parts. Before casting the parts we checked our molds against our plug and found that through repetitious sanding and plaster application the molds were no longer true to our desired shape..



The picture above shows the polyurethane expanding foam molds. The plug is in the mold on the right.

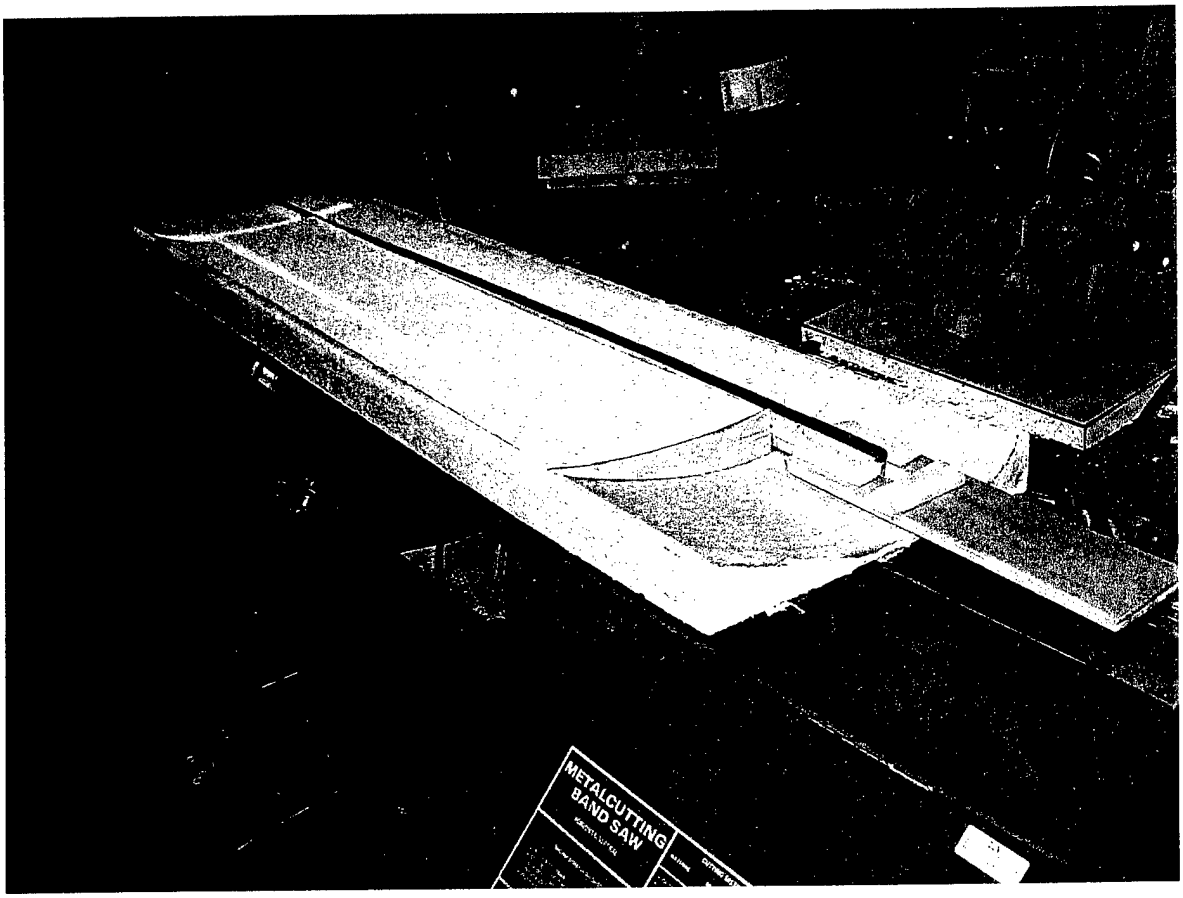
In order to meet the test flight schedule, time was now short and it was decided that we would use our molds as cradles and cast our parts directly on to the plug itself. The plug was prepared with three layers of parting wax and two layers of PVA releasing film. After the parts had cured, they were released from their molds. The results were two mirror image halves of our desired fuselage.

Since the polyurethane nose mold was not sanded repeatedly, we found that its shape was still true. We were then able to cast the part using the original casting technique.

6.2.6. Fuselage-Wing Connection: To control Cg the wing connection is located vertically above the tank; the size of the tank limited the size of the connection to 1.5 inches as a maximum vertical distance from the top off the fuselage arc. In the fabrication of the connection the angle was achieved with basic geometry, the size of the drawing was done largely oversized to minimize error. This drawing was then taken and fitted on a cross-sectional view of the fuselage at the connection point to be able to fit it onto the mounting surface. This shape (the arc of the top of the fuselage above the tank and a flat bottom) was used to fabricate master forms out of 16-gauge sheet aluminum. A connection box was constructed using ¼-4-36 inch balsa sheeting and ½ inch square balsa stock, these were attached into a x-sectional rectangle with internal dimensions of 3-1/2 inch, and external dimensions of 1-4 inches the length of the box was 1 foot, which is oversized allowing for ripping, shaping and trimming. This 'box' was lined with fiberglass in epoxy resin in its internal surfaces. The box was then ripped in half on a band saw and had the 2 mating ends ground to the proper angle using a vertical belt sander with an angularly adjustable slide plate. The mounting ends were then bonded using 2-ton 2-part epoxy using a foam jig cut into the angle to maintain position. The top and bottom of the box was then coated with a single layer of carbon fiber once again being hand-laid with epoxy resin. To fill the upper and lower volume block balsa (4-3inch) was shaped to the angle of the box on its mating surfaces while being left over sized to allow shaping and trimming using the master forms. When the desired shape was achieved the entire external surface was skinned with 1 layer of carbon fiber weave. The box openings were cut and ground out carefully using a Dremel. 4 holes were drilled all the way through the connection using a drill press, 2 holes per side to affix the male connections from the wings when they are slid into the box from either side. Washers and nuts were affixed with epoxy on the underside of the connection, set in flush. The connection was affixed to the fuselage skin using 2-part 2-ton epoxy and ¾ inch pop-rivets.

6.2.7: The Wing and Spars: The wings were made from fiber-reinforced foam core. The foam core was purchased from home depot and cut with a hotwire foam cutter into the shape of the

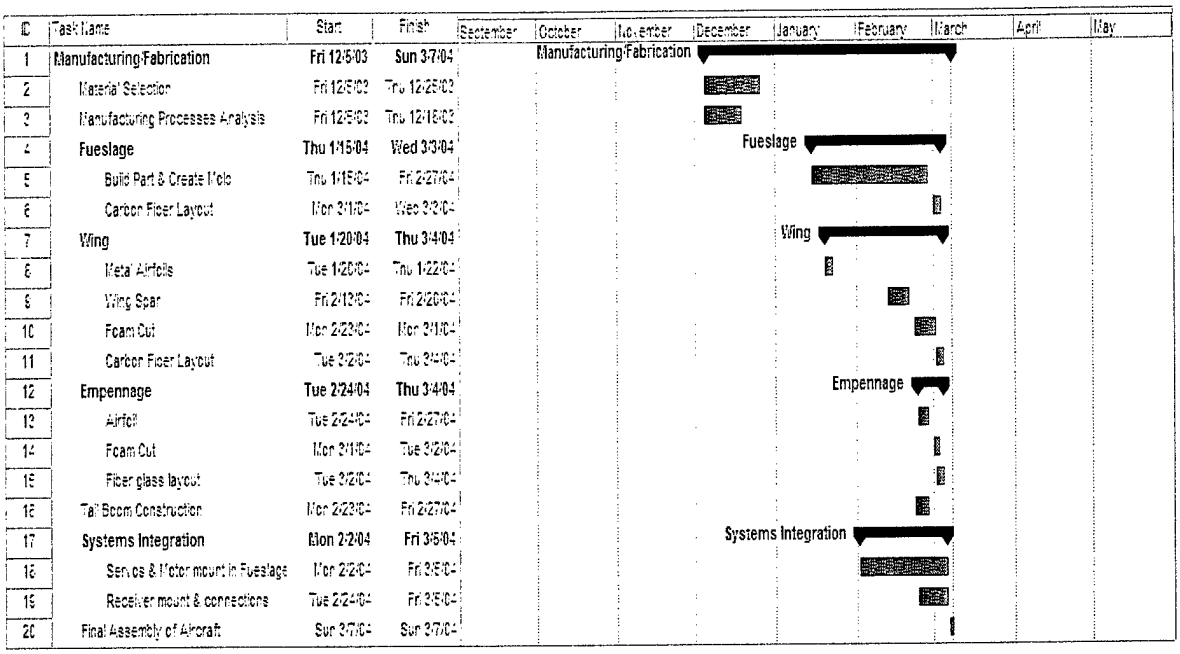
wing. The fiber was then cured over the foam with the spar and place. The picture below illustrates the manufacturing process used, and shows the wing prior to curing with the fiber.



The spars were made by attaching several blocks of balsa wood with epoxy, curing a strip of carbon fiber over the top and bottom, and then stripping into two to the desired width. A final layer of carbon fiber was cured along the length of the spars.

6.2.8. The Tail Boom and Tail: the tail boom was fabricated out of foam core covered with carbon fiber. The foam core was first cut with a foam cutter to the specifications of length and shape. Then it was cured over with two layers of carbon fiber. The horizontal tail was also constructed in a similar manner, except that sand-able lightweight filler was used over it, since the foam served as a male mold. The vertical tail was constructed out of a foam core as well except that a layer of fiberglass was cured over it and covered with a plastic monokote to smoothen the surfaces.

6.3. Manufacturing Milestones Chart:



ID	Task Name	Start	Finish
1	Manufacturing/Fabrication	Fri 12/5/03	Sun 3/7/04
2	Material Selection	Fri 12/5/03	Thu 12/25/03
3	Manufacturing Processes Analysis	Fri 12/5/03	Thu 12/18/03
4	Fueslage	Thu 1/15/04	Wed 3/3/04
5	Build Part & Create Mold	Thu 1/15/04	Fri 2/27/04
6	Carbon Fiber Layout	Mon 3/1/04	Wed 3/3/04
7	Wing	Tue 1/20/04	Thu 3/4/04
8	Metal Airfoils	Tue 1/20/04	Thu 1/22/04
9	Wing Spar	Fri 2/13/04	Fri 2/20/04
10	Foam Cut	Mon 2/23/04	Mon 3/1/04
11	Carbon Fiber Layout	Tue 3/2/04	Thu 3/4/04
12	Empennage	Tue 2/24/04	Thu 3/4/04
13	Airfoil	Tue 2/24/04	Fri 2/27/04
14	Foam Cut	Mon 3/1/04	Tue 3/2/04
15	Fiber glass layout	Tue 3/2/04	Thu 3/4/04
16	Tail Boom Construction	Mon 2/23/04	Fri 2/27/04
17	Systems Integration	Mon 2/2/04	Fri 3/5/04
18	Servos & Motor mount in Fueslage	Mon 2/2/04	Fri 3/5/04
19	Receiver mount & connections	Tue 2/24/04	Fri 3/5/04
20	Final Assembly of Aircraft	Sun 3/7/04	Sun 3/7/04

7.0 Testing Plan

7.1 Testing Objectives

The team conducted a series of tests on the aircraft. The different tests allowed the team to evaluate the performance of the aircraft. A projected timeline of the tests is as shown in Figure 7.1. A checklist for static and dynamic testing can be viewed on Figure 7.2. The Objectives of each test are stated in table 7.1. A successful testing will be possible if all tests are properly performed according to the security checklist which can be seen on Table 7.2 and thus would insure the safety of the aircraft. The current status is that the aircraft is still in the manufacturing phase and some of the tests will begin after the first test fly.

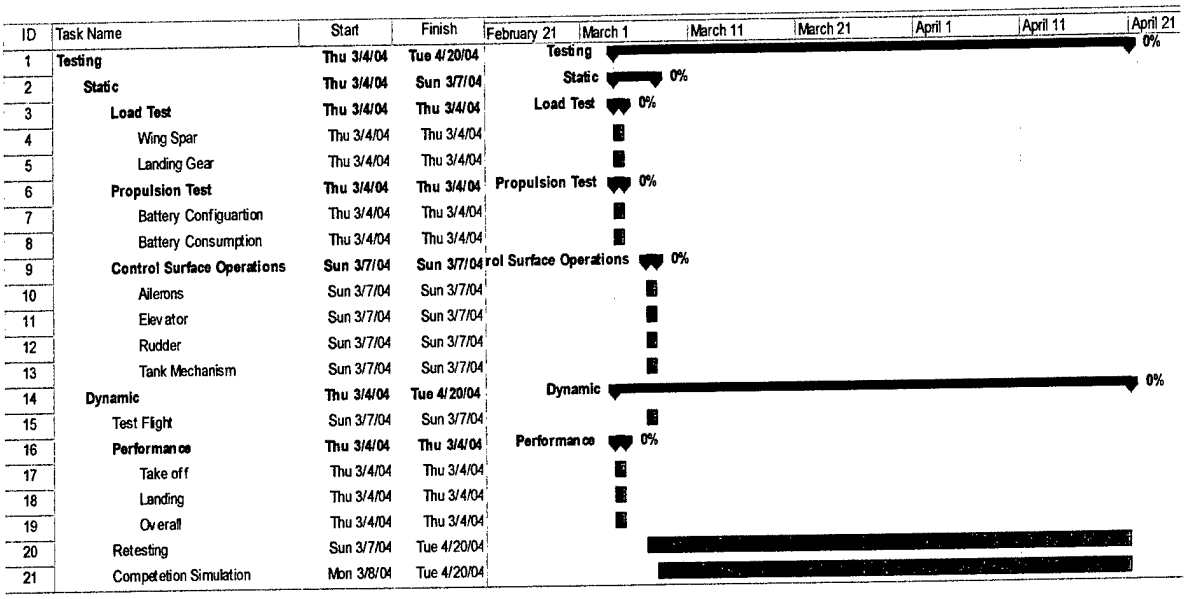


Figure 7.1 Testing Timeline

Test	Test objective
Propulsion Test	Collect Motor and battery Information data.
Wing Spar Test	To Check the strength and weight of wing and fuselage spars.
Landing Gear Test	To Check the strength of the landing gear.
Control Surface Operations	To Check the performance of the servos in controlling the different aircraft parts.
Performance	To Check the performance of different parts of the aircraft while in motion.
Competition Simulation	Simulating the actual competition.

Table 7.1 Test Objectives

Static Testing Checklist				
Test Type	Measured Values and Checklists			
Load test	Wing Spars	Landing Gear		
Propulsion Test	Battery Configurations	Battery Consumption & Motor		
Control Surface Operation	Ailerons	Elevator	Rudders	Tank Mechanism

Dynamic Testing Checklist				
Test Type	Measured Values and Checklists			
Performance of the Aircraft				
Take Off	Distance	Time	Climb angle	
Landing	Distance	Time	Flare	Nature of Touchdown
Overall	Dumping time	Max Radius	Max Altitude	Speed

Competetion Simulation	Test Run 1	Test Run 2	Test Run 3	Test Run 4

Figure 7.2 Static & Dynamic tests Checklists

Security Checklist	
Tests	Check
1. Aircraft inner components are in their proper place	
2. Securing the Aircraft's external connection (Propeller, Wings, tail)	
3. Structural Integrity (upright and inverted wing lift tests will be performed)	
4. Verify all controls (servos) move in the proper sense.	
5. Integrity of the Payload System	
6. Radio Range Check	
7. Radio Fail/Safe Test	

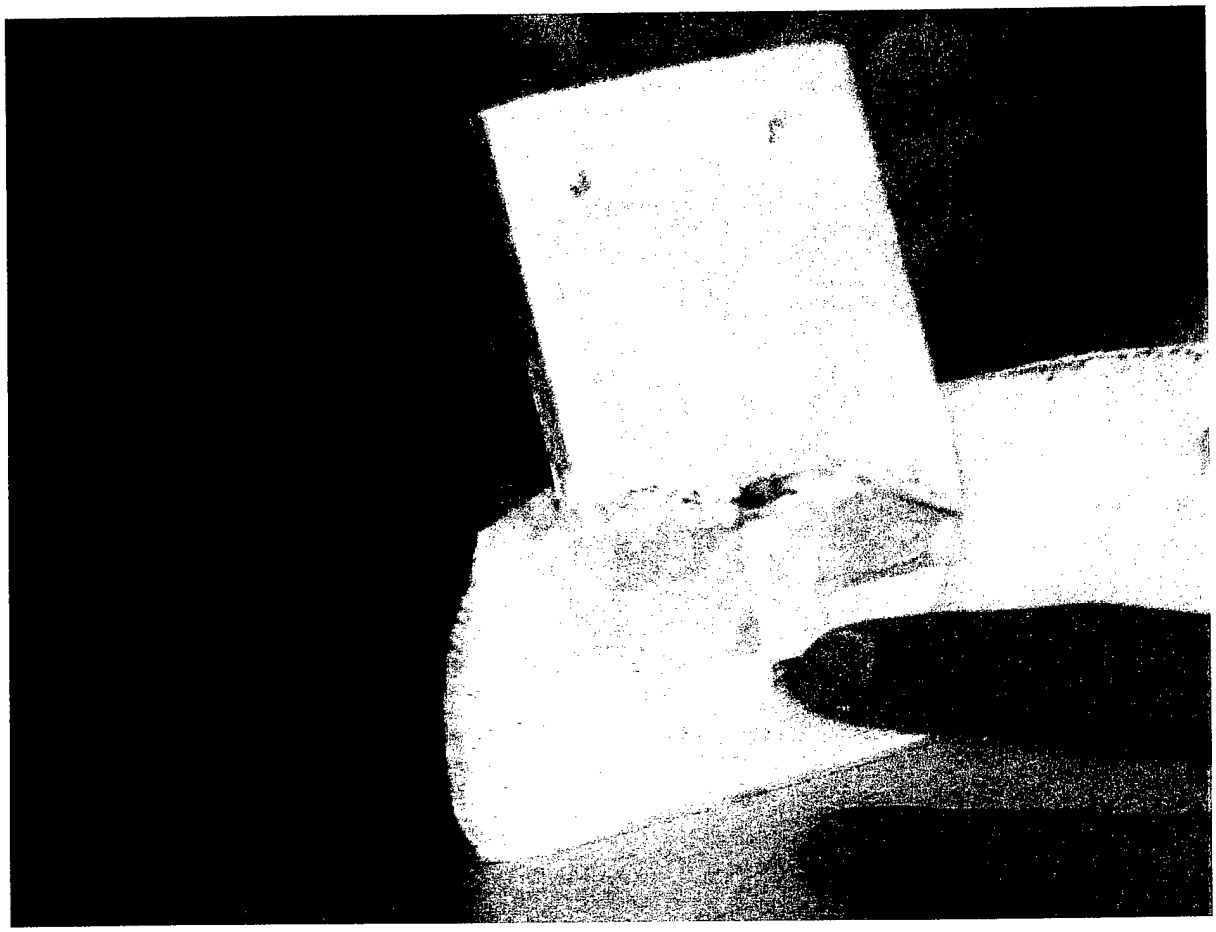
Table 7.2. The Security Checklist must be checked before every flight.

7.2 Lessons Learned & Test Results:

7.2.1. Motor Run time test: From the preliminary design two propulsion systems were selected for testing. The Astro Cobalt 60 11T#23 #661 direct drive using a 14 by 10 propeller, and the Astro Cobalt 60 11T#23 #660 with a 2.7:1 gear ratio using a 20 by 10 propeller. Both systems use 30 cells of Sanyo RC-2400. The objective of this test was to verify the run time. This is important because the propeller is the load on the motor and maximum run time is desired. Initially the objective included comparing the run times of both systems, but the Astro Cobalt 60 11T#23 #661 was delayed in transit from the vendor. The tests is analyzed by ear of when the motor starts fading.

Astro Cobalt 60 11T#23 #660 with a 2.7:1 gear ratio using a 20 by 10 propeller, powered by 30 Sanyo RC-2400 cells, began to fade after 7.2 minutes, after 8.20 minutes there was sharp fade, and it stopped completely at 9.17 minutes. When the Astro Cobalt 60 11T#23 #661 direct drive becomes available this static test will also be repeated, and both systems will be compared. Further verification of the run-time will occur in flight and compared to the in-flight run-time using Moto-Calc of 5.25 minutes for the geared system and 3.7 minutes for the direct system. The superior system will be chosen based on the performance of the airplane during the detailed test flights.

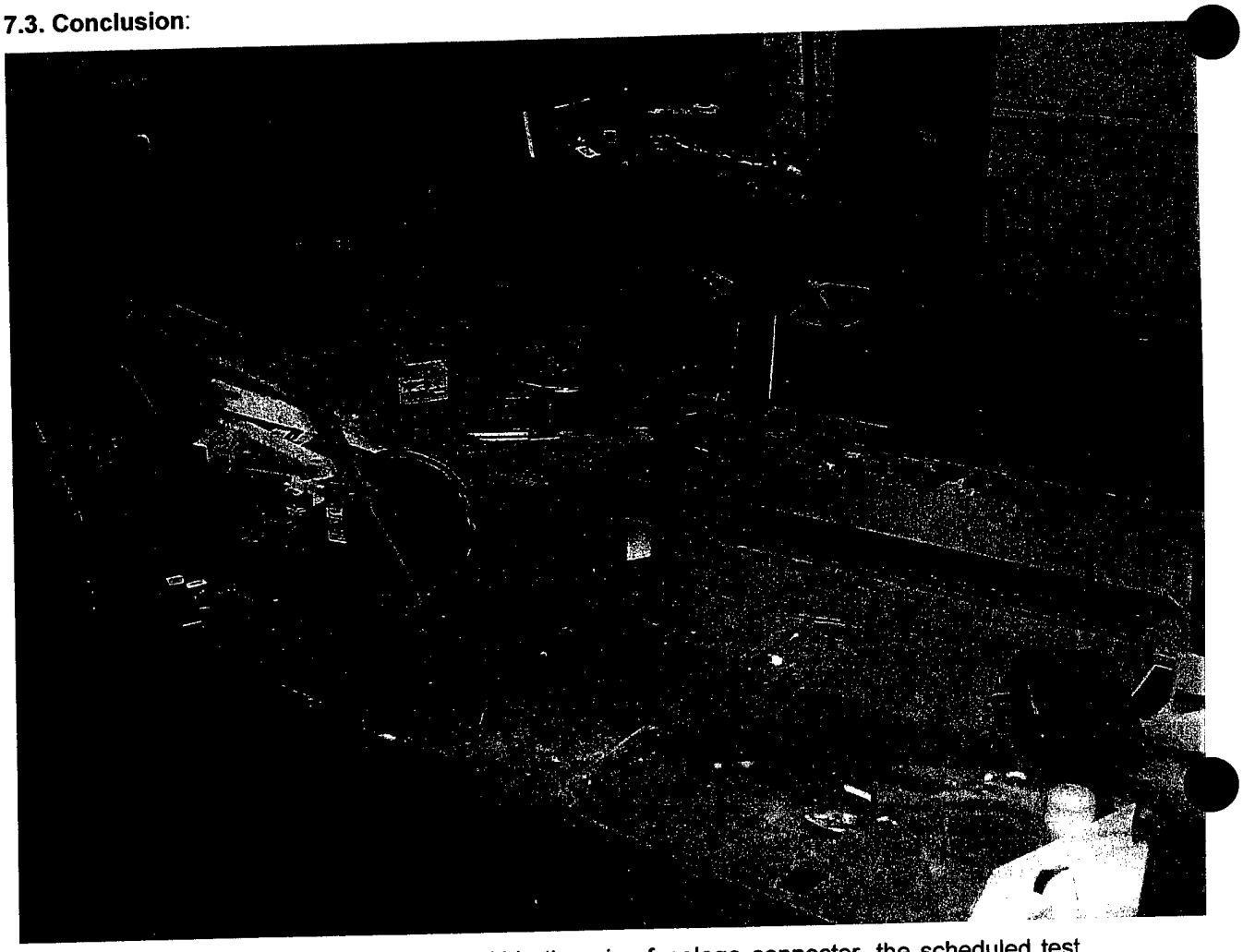
7.2.2. Wing Tip Test: Prior to the test, the aircraft was loaded except for the water payload and a wing tip test was performed in order to satisfy this technical requirement performed on competition day. The test resulted in the failure of the connection between the spar and the wing-fuselage connection. The picture below illustrates the failure. The failure seems to have occurred as a result of stress concentration in the junction of the spar and the spar-connector. The failure appears to have originated within the connector and propagated to the fiber-reinforced sides. Apparently this is the same shear stress that was predicted in the finite element analysis simulation of the wing-tip test, featured in the detailed design (see figure 5.1, the image on the right).



The lessons learned from this test are:

1. The junction between the spar and the spar-connector needs internal reinforcing or a complete re-design.
2. The aircraft is too heavy. The weight of the aircraft without the water payload comes to more than 20 pounds. Strict weight management will be adhered to, as the manufacturing processes of most of the structural components shall have to be re-visited. The structural design will also need to be revisited, in order to meet the target weight of 17.6 pounds loaded.

7.3. Conclusion:



Due to the failure of the spar connector within the wing-fuselage connector, the scheduled test flight has been postponed for a month, so that all structural and manufacturing processes can be revisited and revised. The picture above shows the nearly completed aircraft.

Selected Reference:

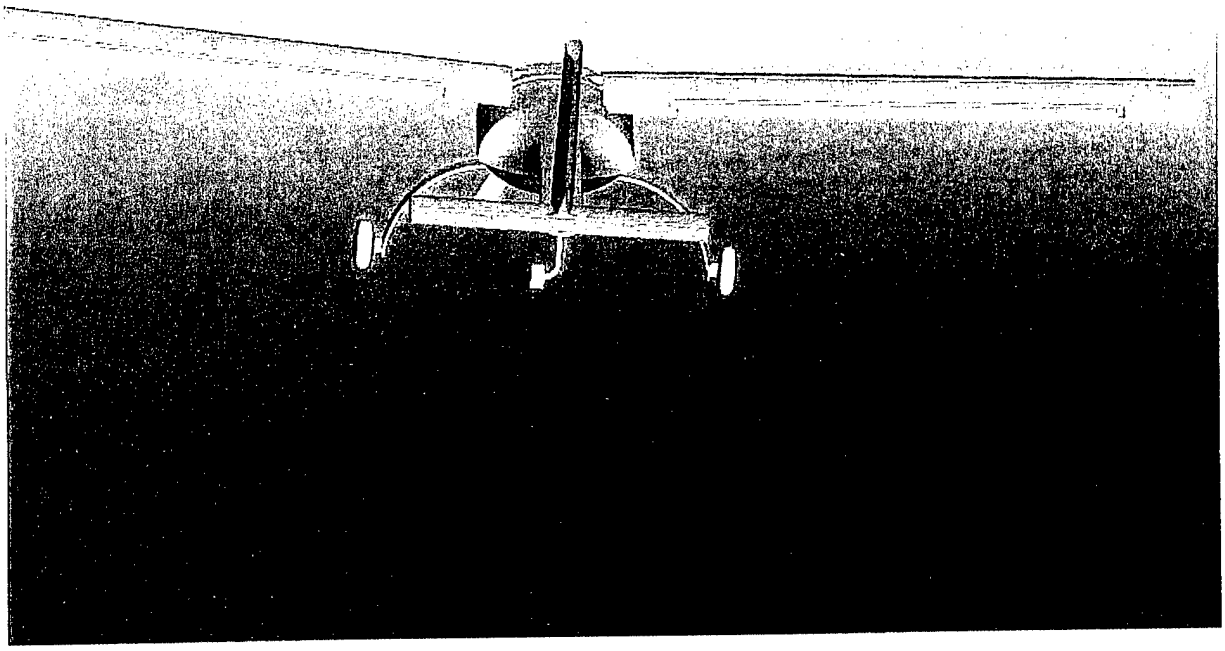
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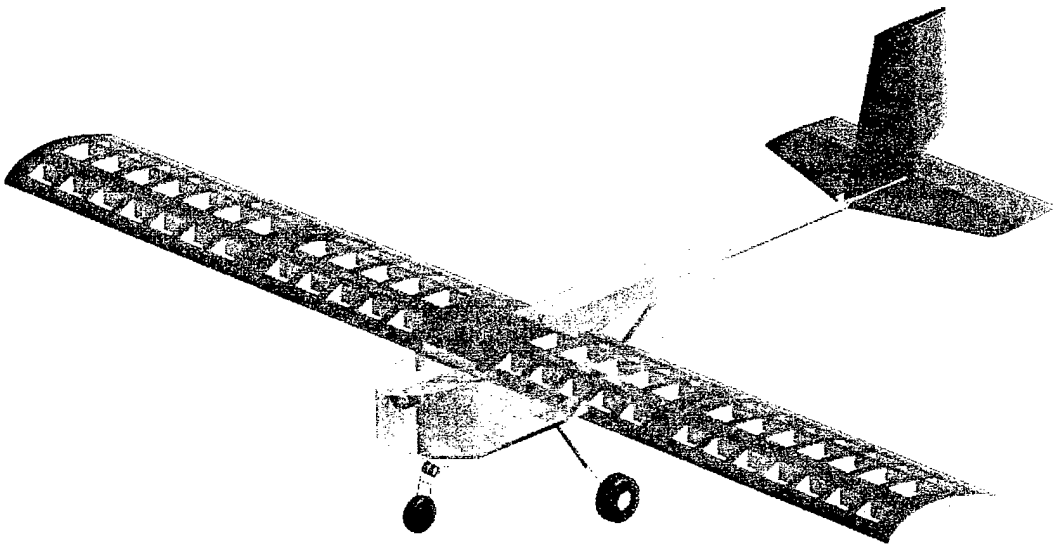


The long day wanes: the slow moon climbs: the deep; Moans round with many voices. Come, my friends, 'T is not too late to seek a newer world. Push off, and sitting well in order smite; The sounding furrows; for my purpose holds; To sail beyond the sunset, and the baths Of all the western stars, until I die. It may be that the gulfs will wash us down: It may be we shall touch the Happy Isles, And see the great Achilles, whom we knew. Tho' much is taken, much abides; and tho' We are not now that strength which in old days Moved earth and heaven, that which we are, we are; One equal temper of heroic hearts, Made weak by time and fate, but strong in will To strive, to seek, to find, and not to yield.

Excerpt from Ulysses - Alfred Lord Tennyson

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

Design Report



"Daha Basit"
Middle East Technical University
March 2004

TABLE OF CONTENTS

CHAPTER 1- EXECUTIVE SUMMARY 3
1.1 DESIGN HIGHLIGHTS 3
1.2 DESIGN DEVELOPMENT 3
1.2.1 Conceptual Design 4
1.2.2 Preliminary Design 4
1.2.3 Detailed Design 5
1.2.4 Manufacturing 6
1.2.5 Testing 6
CHAPTER 2 - MANAGEMENT SUMMARY 7
2.1 ORGANIZATION OF THE DESIGN TEAM: ORGANIZATION AND ASSIGNMENT AREAS 7
2.2 WORK AREAS OF TECHNICAL GROUPS 8
2.2.1 Aerodynamics Group 8
2.2.2 Propulsion Group 8
2.2.3 Stability and Control Group 9
2.2.4 Structure Group 9
2.3 THE MILESTONES CHART 9
CHAPTER 3 - CONCEPTUAL DESIGN 11
3.1 MISSION REQUIREMENTS 11
3.2 COMPETITOR STUDY 12
3.2.1 Wing Geometric Parameters and Placement 15
3.2.2 Alternative Tail Configurations 16
3.2.3 The Fuselage 17
3.2.4 The Landing Gears 18
3.2.5 Propulsion System 19
3.3 THE SCORING SYSTEM 19
3.3.1 Figures of Merit (FOM) 20
3.3.2 Results of the Trade-Off Analysis 20
3.4 WATER DUMPING SYSTEM CONCEPTS 21
CHAPTER 4 - PRELIMINARY DESIGN 23
4.1 SELECTION OF STALL SPEED AND ESTIMATION OF MAXIMUM LIFT COEFFICIENT 24
4.2 INITIAL WEIGHT ESTIMATION 25
4.3 AERODYNAMIC SIZING AND GEOMETRY SELECTION 26
4.3.1 The Wing 26
4.3.2 Empennage Group and Stability 32
4.4 PROPULSION SYSTEM 33
4.5 STRUCTURAL DESIGN ANALYSIS 35
4.5.1 Figures of Merit 36
4.5.2 Fuselage 36
4.5.3 Wing 36
4.5.4 Landing Gear 37
4.5.5 Empennage 37
CHAPTER 5 - DETAIL DESIGN 38
5.1 STRUCTURAL DESIGN ANALYSIS 38
5.1.1 Fuselage 38
5.1.2 Wing 38
5.1.3 Landing Gears 39
5.1.4 Empennage 39

5.1.5 Water Dumping Mechanism 40
5.2 PROPULSION 40
5.3 RATED AIRCRAFT COST ANALYSIS 40
5.4 SIZED AIRCRAFT DATA 42
CHAPTER 6 – MANUFACTURING PLAN AND PROCESSES 47
6.1 FIGURES OF MERIT (FOM) 48
6.2 MANUFACTURING OF WINGS AND EMPENNAGE 48
6.3 MANUFACTURING OF TAIL BOOM 49
6.4 MANUFACTURING OF FUSELAGE 49
6.5 MANUFACTURING OF LANDING GEARS 50
6.6 MANUFACTURING OF WATER TANK 50
CHAPTER7-TESTING PLAN 51
7.1 STATIC TESTS 51
7.1.1 Spar Test 51
7.1.2 Landing Gear Test 52
7.2 FLIGHT TESTS 53

CHAPTER 1- EXECUTIVE SUMMARY

This project intends to design, manufacture and fly an unmanned, electric powered, radio controlled aircraft that can best meet the mission requirements specified by the rules of the 2004 Cessna/ONR Student Design/Build/Fly Competition. Two missions are identified, first of which is to evacuate maximum 4 liters of water. The second mission is to fly the aircraft without payload. The instructions of the competition are open to various design options. Hence, some major highlights are defined.

1.1 DESIGN HIGHLIGHTS

The highlights for the design followed in this project are; Simplicity, Minimum Weight, Efficiency, Availability and Minimum Cost.

First of all, the design should be as simple as possible. This reduces the risks which might be encountered during production and operation process. Moreover, the unnecessary complications will have negative effect on the cost of the aircraft. The requirements should be adhered; and only necessary concepts should be applied.

Next the weight of the aircraft, which is one of the most significant concerns of every aerospace vehicle design. All 'over safe' structural elements, excess aerodynamic surface areas and excess thrust power bring 'unnecessary and dead weights' to the aircraft, and should be avoided.

Another concern is the efficiency of the design, which refers to the general performance of the aircraft. The air vehicle should complete its mission with the highest score at minimum cost.

Commercial of the shelf (COTS), availability of the components of the aircraft is the fourth concern. In case of any breakdown, the materials used in the production of the aircraft should be available and be easily replaceable.

In addition to these the cost of the aircraft should be taken into account. An air vehicle, which is designed and built in its simplest forms from the most available materials of the shelf, will be an efficient design as well as the cheapest one.

It is clear that all these five highlights are related to each other and for an optimum design these conditions must be met as much as possible.

1.2 DESIGN DEVELOPMENT

The design team initialized a deep competitor study on July 15th, 2003 before the rules of the competition were announced. The parameters like the wing loading, power loading, and tail volumes were computed for future statistical analysis. Those configurations which achieved good performance were identified and the research and statistical studies conducted by the previous design teams are analyzed in detail.

The corresponding codes, spreadsheet programs and a wide database of material weights and properties were developed at the same time.

Following the announcement of the rules of the DBF-2004 Competition on July 28th, 2003, the conceptual design is initiated, which was followed by the preliminary and the detail design phases.

1.2.1 Conceptual Design

During this phase, the different aircraft configurations were studied by varying critical design parameters so that the optimum configuration based on the mission requirements can be obtained. A computer program was developed to calculate the parameters and the Figures of Merit (FOM) of the design. The configuration with the highest score was chosen according to a scoring system developed by the design team. Also different production techniques and materials are investigated. The configuration yielding the highest value score were chosen according to an evaluation system devised by the design team.

The scoring system consisted of attributing scores to alternate designs from different perspectives such as; aerodynamics, structure, weight, ease of manufacturing, cost, flight times and mission effectiveness. Each one of these aspects had a different weight, and the scores given to each component by the team members were weighed by these factors and a final total score for each design were obtained for final evaluation.

Results of the Conceptual Design:

At the end of these analyses; a high wing monoplane with a conventional tail and single tractor engine were chosen as a final configuration. High wing configuration was selected because it is more stable and easy to manufacture when it is compared with low-wing and mid-wing configurations. Also conventional tail configuration was chosen instead of T-tail, V-tail and canard configurations due to the fact that it has a fairly simple and strong structure, shorter take off distance and a better control effectiveness. The wing is made by foam/balsa rib. It is produced by covering the foam with balsa sheets and locally removing the foam/balsa to approach the classical rib construction. The wing is reinforced with a single U profile aluminum spar. The fuselage is rectangular with thin walled aluminum construction and fiber-glass/film skin (not stress-carrying). The tail is also balsa/foam construction and attached to the fuselage by a tail boom instead of a lofted fuselage. Also since it is desirable to have an aircraft center of gravity (cg) position essentially independent of the water tank configuration, the planform centroid of the payload was designed to be close to the aircraft center of gravity (cg) position.

1.2.2 Preliminary Design

In preliminary design a more detailed analyses was made and most competitive and effective aircraft that meets the mission requirements was selected from the possible configurations. Since in the first mission, fire bomber, the aircraft is required to discharge 4 liters of water in 1000 feet and a 360o loop at the loop a test was performed to determine the approximate time of discharging the water in the required distance. This time interval reveals the approximate 'minimum flight speed', i.e. the stall speed. This shows that stall speed is the most critical performance parameters of our design. Thus preliminary design was initiated by determining the stall speed. Then initial weight estimation was

made and maximum lift coefficient was determined. Then, the iterative sizing procedure is initiated with the collaborative work of design groups such as; formed as aerodynamics, stability and control, structure and propulsion groups. During this study, the design tools shifted from statistical ones towards more analytical ones.

In this phase, two software programs, which were revised and improved version of the programs called "PreSize" and "WeiSt" were developed. They were used for aerodynamic calculations and structural analysis. The iterative process has been initiated using the data generated at the conceptual design phase. First, the wing section was selected followed by wing planform area, aspect ratio, taper ratio, sweep angles, incidence angle of the wing, dihedral angle, wing twist and location of the wings on the fuselage. Following the calculation of required empennage areas and moment arms, the fuselage has been designed. Simultaneously, the weight estimations are updated regularly. After the aircraft geometry and weight estimation is completed, the propulsion system has been selected in the preliminary design phase. At the end of the iterative process the results were validated with 2-D viscous and 3D potential flow models in order to obtain the aerodynamic parameters such as lift, drag and pitching moment characteristics.

Results of the Preliminary Design:

A number of competitor designs were made to determine which one yields the superior design compared to others. This analysis is performed by the use of computer program "PreSize". The results have been showed that Eppler 423 airfoil section yields a lower value for wing area. This brings a decrease in both take off gross weight and Rated Aircraft Cost (RAC). Therefore Eppler 423 airfoil is selected for our aircraft because of its high lift characteristics at low Reynolds numbers which bring advantages from the point of view of weight and rated aircraft cost. Also the analysis showed that the aspect ratio, which minimizes the weight of the wing, is 8. On the other side flaperon configuration is chosen taking the second mission requirements into account and considering the advantageous in the aspects of weight and RAC. Also FOM analysis for propulsion has showed that Astro Cobalt 60 11T# 23 motor gives a better performance for our design. Analysis showed that 2400 mAh batteries are suitable for the mission. And the number of batteries is estimated as 30. Also the team decided to use different propellers for the two missions and they were decided to be chosen during the flight tests.

1.2.3 Detailed Design

The aerodynamic, propulsion and stability characteristics are determined in the preliminary design. Detail design was focused mainly on the structural aspects including joints, assembly points etc. to be used further in the manufacturing phase. Control surface sizes and travel amounts are refined and the final configuration of the aircraft is fixed. Corresponding drawings and tables including final aircraft data are done accordingly. In case of inconsistency, the team returned back to the previous analysis and modified the design.

1.2.4 Manufacturing

In this phase, the team members are organized in small groups each tasked to a particular job to increase the efficiency and the aircraft was manufactured. In fact at the beginning blue prints for the parts are prepared for manufacturing the aircraft.

1.2.5 Testing

In the testing phase, the observed performance of the aircraft was compared to those that were expected. Major modifications are done in order to assure a "safe" operation of the aircraft and optimize the mission performance.

CHAPTER 2 - MANAGEMENT SUMMARY

The success of the projects requires engineering skills depending on the teamwork rather than the individual work. It is the output of the team which constitutes the integrity of multi-disciplinary projects. Since a team is composed of groups of individual people, each person contributes a work in his/her discipline to the project. It is much like forming VOLTRON, a famous cartoon; each member of the team fulfils a certain task. The design process should be organized/planned in such a way that there should be a great cooperation between team members. The design process requires multi-disciplinary contributions, so the cooperation is very important in order to increase the efficiency of the groups and the final output.

The design team of METU has set up an organized process by planning/scheduling the milestones of the design as described in the following sections of this chapter.

2.1 ORGANIZATION OF THE DESIGN TEAM: ORGANIZATION AND ASSIGNMENT AREAS

The design team of METU is named as 'Anatolian-Craft Team'; and it is composed of 10 members. Under the leadership of the team advisor, the team was divided into four major technical groups as aerodynamics, stability & control, structure and propulsion. However, people in these categorized groups also participated to other groups according to their interests, since they were all Aerospace Engineering Students; everyone had some word to say about all subjects. Therefore, the group formation was a bit flexible.

The groups do not follow a hierarchical order. On the contrary, all of the group members are under the supervision of the team advisor. The role of the team leader is to maintain the coordination of the team and track the processes to be completed within the planned time frame. Figure 2.1 illustrates the team's organizational structure.

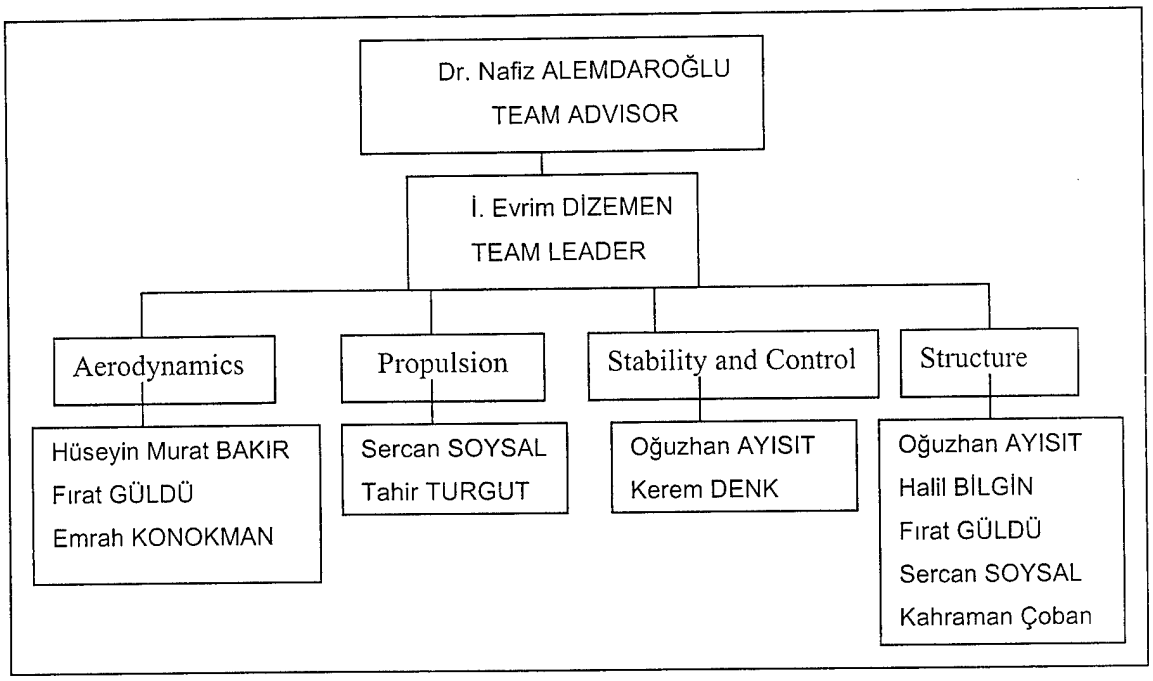


Figure 2.1 METU DBF Team architecture including design personnel and their assignment areas

2.2 Work Areas of Technical Groups

As mentioned above the group formation was flexible, however each group was assigned a very well defined task to be accomplished. Members in their assigned groups were responsible for that part of the design project. Since the design process is composed of multi-disciplinary projects, at previous stages of the project weekly meetings have been conducted. However, as the project preceded these meetings were conducted very often. The work areas of each technical group are described under the corresponding subtitles given below.

2.2.1 Aerodynamics Group

The aerodynamics group was responsible for the aerodynamic properties of the design project, which aimed first of all to provide maximum lift and minimum drag for this design. Since the every component of the aircraft which is subjected to the airflow is responsible for the drag, the importance of the work of this group was great, especially at the beginning of the design process. One of the most important tasks of the group was to determine the airfoil shape and sizing of the wing. All calculations and measurements were performed according to these data. Owing to the collaboration with the other groups, other tasks can be declared as shaping the fuselage, drag reduction of structural members immersed in the flow, and cooling of the engine and batteries were also accomplished

2.2.2 Propulsion Group

Analysis, optimization and verification of the propulsion system performance were among the responsibilities of the propulsion group. Since electric motor rather than piston engines had to be

used, the first task of this group was to study main characteristics of the electric motors, such as weight, performance and energy (battery) need. The major task of the group was to perform a trade-off analysis for the determination of the best motor/propeller/battery system configuration.

2.2.3 Stability and Control Group

In order to qualify an air vehicle as a flying machine, it should not only be lifted-off by air but also it has to be stable and controllable. Therefore, while the aerodynamics group tried to lift-off the aircraft, the stability and control group tried to "fly" it. The main responsibility of the group is to size the empennage group with the control surfaces so as to require adequate stability and controllability. Moreover, this group is responsible for the performance analysis of the aircraft.

2.2.4 Structure Group

Since the most important part of the aircraft is thought to be its structure, this group involves more people than the other groups. The main task of this group was to construct the aircraft as light as possible according to the other parameters concerning the strength of the structural members. In order to construct the structure as light as possible while keeping the strength as required, the material selection, design and analysis of the structure is very important and critical. These tasks were all accomplished by the structure group. Besides, the group determined the manufacturing processes and plans for the structural members.

2.3 The Milestones Chart

Since the time for design and manufacturing the aircraft is limited, the team had to use the time effectively. In order to achieve this goal, a complete time scheduling of the whole project was made at the very beginning of the project. The project plan includes schedules of design, manufacturing and testing activities as a whole since the competition is required to design, build and fly the aircraft. Time chart for the entire project is given in Figure 2.2. Design and manufacturing processes were sequenced in the light of experiences of the team from the previous years' competitions. It is not usually allowed to overlap the stages of the project and the milestones chart has been followed as closely as possible under the control of the team leader.

Description	Start Date
Start	
Competitor Study	15.07.2003
Rules Declerated	28.07.2003
Conceptual Design (Planned)	04.08.2003
Conceptual Design (Actual)	04.08.2003
Preliminary Design(Planned)	01.09.2003
Preliminary Design (Actual)	04.09.2003
Detailed Design(Planned)	13.10.2003
Detailed Design (Actual)	29.10.2003
Prototype Manufacturing(Planned)	01.12.2003
Prototype Manufacturing (Actual)	26.01.2004
Prototype Testing(Planned)	15.12.2003
Prototype Testing (Actual)	16.02.2004
Prototype Flight Testing(Planned)	19.01.2004
Prototype Flight Testing (Revised Plan)	11.03.2004
Actual UAV Manufacturing (Planned)	16.02.2004
Actual UAV Manufacturing (Revised Plan)	19.03.2004
Actual UAV Flight Test (Planned)	15.03.2004
Actual UAV Flight Testing	05.04.2004
Design Journal	15.07.2003
Design Report (Planned)	01.01.2004
Design Report (Actual)	05.01.2004
Contest	23.04.2004

Figure 2.2 METU DBF Project Schedule Chart

CHAPTER 3 - CONCEPTUAL DESIGN

At the first step of the conceptual design the mission requirements were analyzed and by considering the design principles of the team a guideline was determined. Then at the next stage, the capabilities, which the aircraft was supposed to have, was determined. After that, for each component of the aircraft (wing position, fuselage configuration, tail arrangement, landing gear, propulsion system etc.), a quantitative trade-off analysis between different alternatives was performed in the light of the figures of merit which are determined by analyzing the mission requirements. Following the trade off analysis, the team started to analyze the performance and score of the alternative concepts in order to identify the aircraft that best suits to the mission requirements.

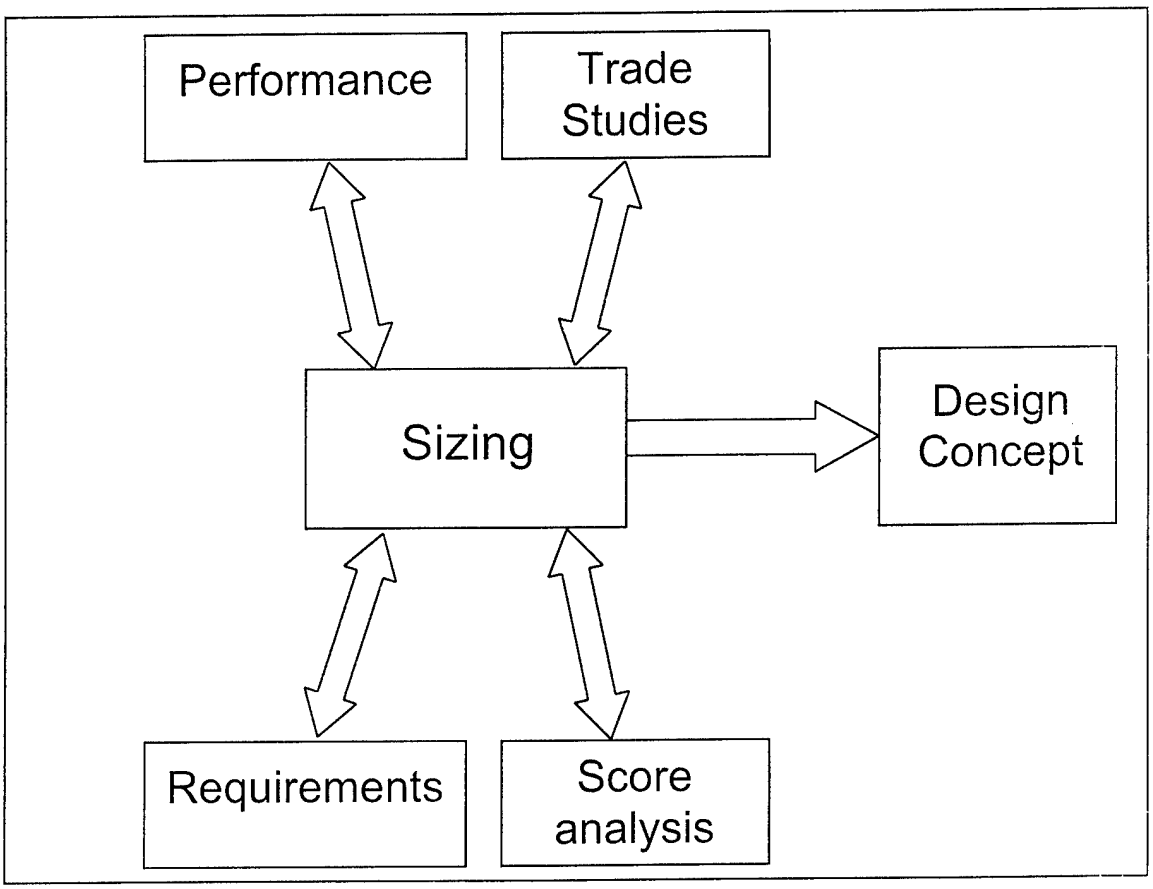


Figure 3.1

3.1 MISSION REQUIREMENTS

The key elements of mission requirements can be roughly stated as:

- Rotary wing and any component lighter-than-air are not allowed.
- The aircraft must be propeller driven electrically powered with a maximum total battery weight of 5 lbs.

- The aircraft must fit into a box having dimensions of 2 feet wide by 1 foot high by 4 feet long (interior dimensions).
- Payload can only be carried in the fuselage. For flying wings, the "fuselage" must be the inner most 9 inch of semi-span; the remainder is "wing" for payload considerations.
- In any mission type, the aircraft should be able to take-off from a maximum ground roll distance of 150 ft.

Water deploying system requirements:

- Maximum allowed aircraft water capacity is 4 liters.
- The aircraft will be loaded four 2-liter plastic "soda" bottles. Gravity loading or "pumped" loading can be used. Soda bottles may not be pressurized to assist loading.
- Water may only be dumped during the downwind leg which is 1000 ft long. Aircraft must fly slow enough to allow sufficient time for the water to be emptied.
- Maximum dump orifice diameter must be 0.5 inch.
- There is ferry mission consisting of high speed flight.

Guide Lines:

Aircraft should be capable of flying with heavy payload (4 lt.) and at high speed in separate missions. So, aircraft should have both high lift with low drag. To achieve this criterion, a wing with flaperon can be used. The inverse flaperon can also be used in the high speed mission to get rid of excess lift. Also the structure should be considered. It will carry 4 lt. of water, hence it should be strong enough, but should not be too heavy. The water tank and dumping system should be light, safe and easy to produce. In addition to that the configuration should satisfy the following principles which were given in the part 1.1 Design Highlights.

- As simple as possible
- As light as possible
- As efficient as possible
- As available as possible
- As cheap as possible

3.2 Competitor Study

When the task is to design something; a chair, a car or an aircraft, the hardest part is to know where and how to start. To be able to start the conceptual part of design probably requires the most 'activation energy' and a significant amount of knowledge. Therefore, at this point, the necessary action to be taken is gathering information on previous designs and then comparing them in terms of the given requirements.

Taking a look at the properties of previous examples would be the least time consuming and the most efficient way to start a design process because it assures that the design team has a number of ideas to evolve thus the canvas is ready for a rough preview of the conceptual design.

In the oncoming parts of this section, a group of graphs which demonstrates the general trends of the teams of 2002-2003 DBF competition in the selection of some design alternatives.

Together with the past years' experiences, competitor study gave some important results:

- In terms of stability aspects high wing configuration is better.
- For simplicity conventional tail configuration is more successful.
- A highly tapered wing is unnecessary.
- Thin and highly cambered airfoils are widely used among the top scored designs.
- Try to achieve a mono-engine configuration.
- Try to keep the structural weight as low as possible.

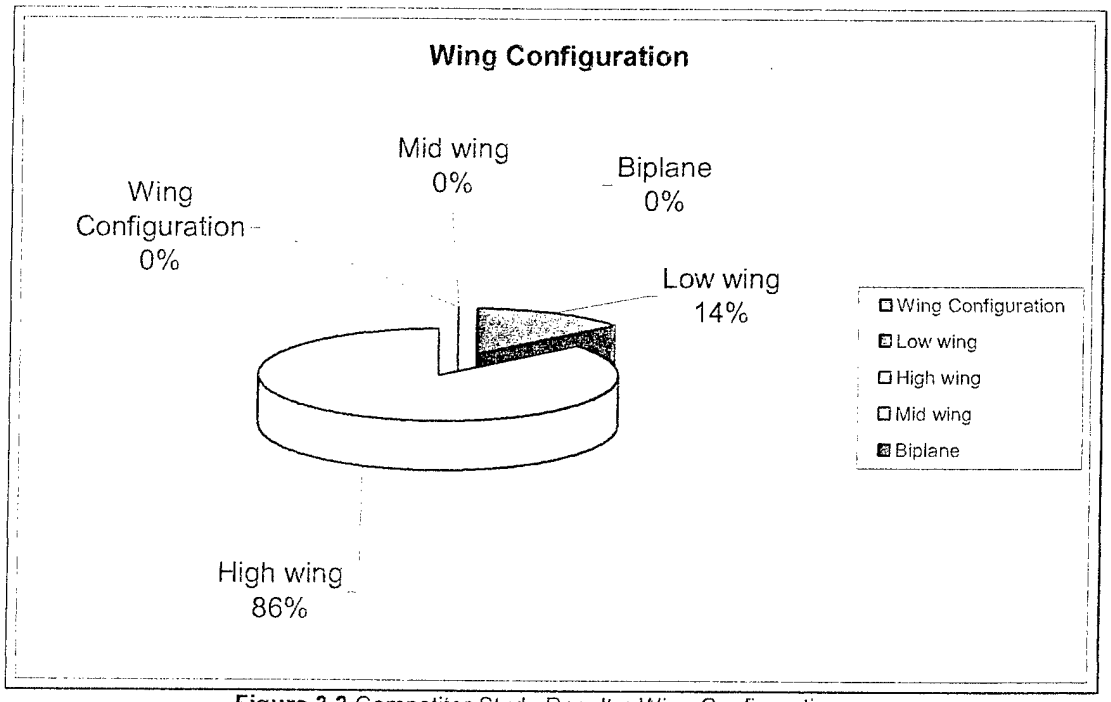


Figure 3.2 Competitor Study Results; Wing Configurations

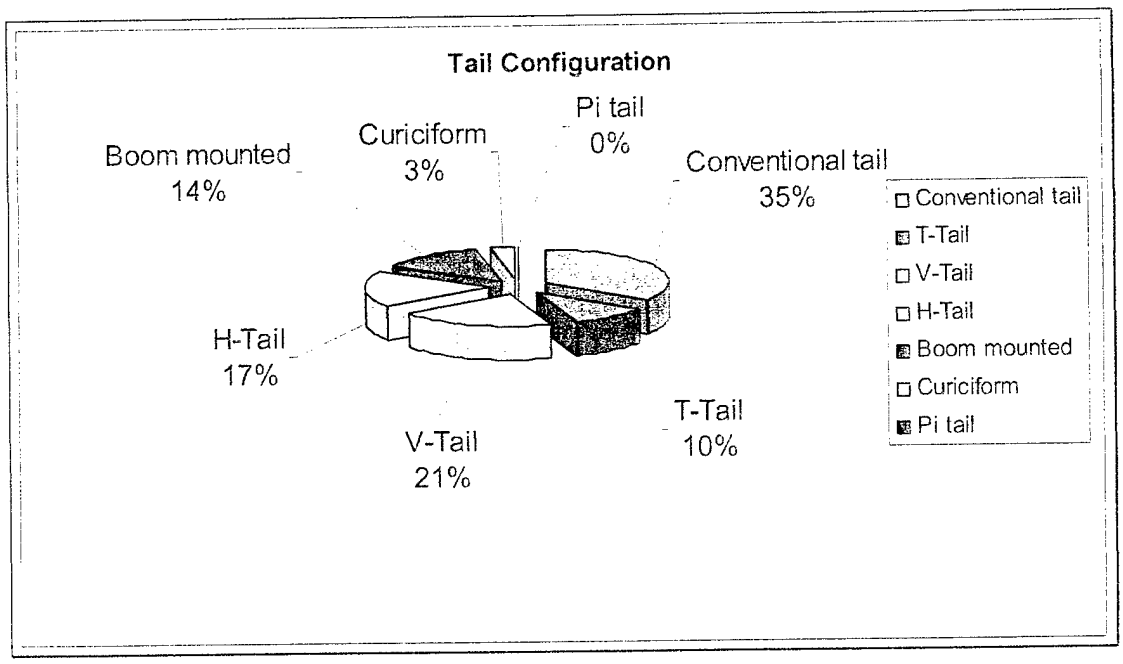


Figure 3.3 Competitor Study Results; Tail Configurations

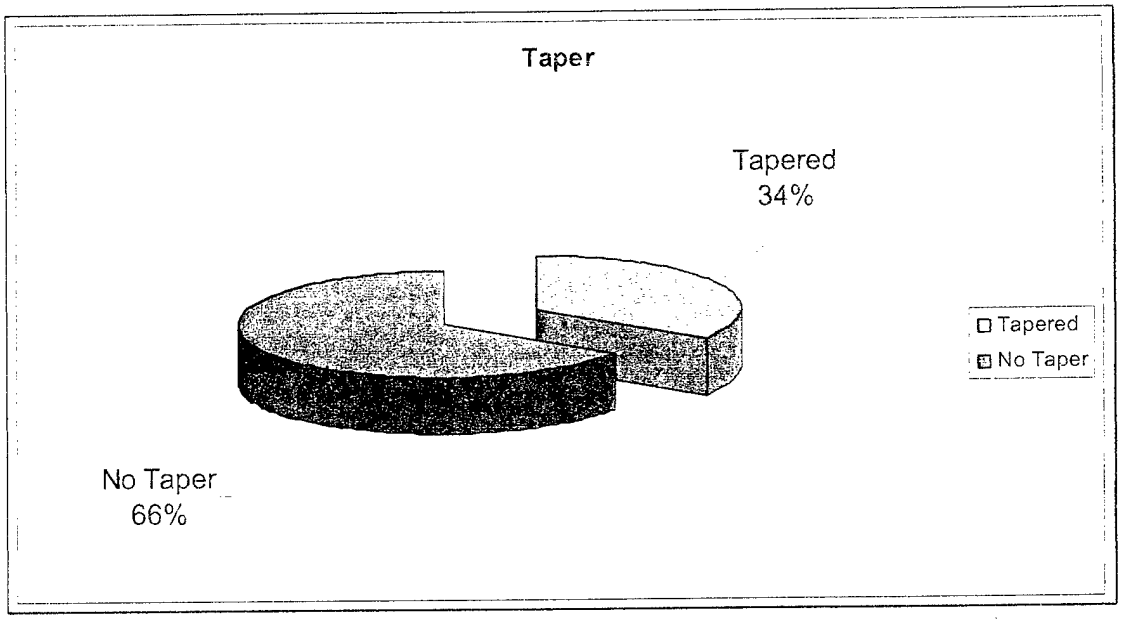


Figure 3.4 Competitor Study Results; Use of Taper

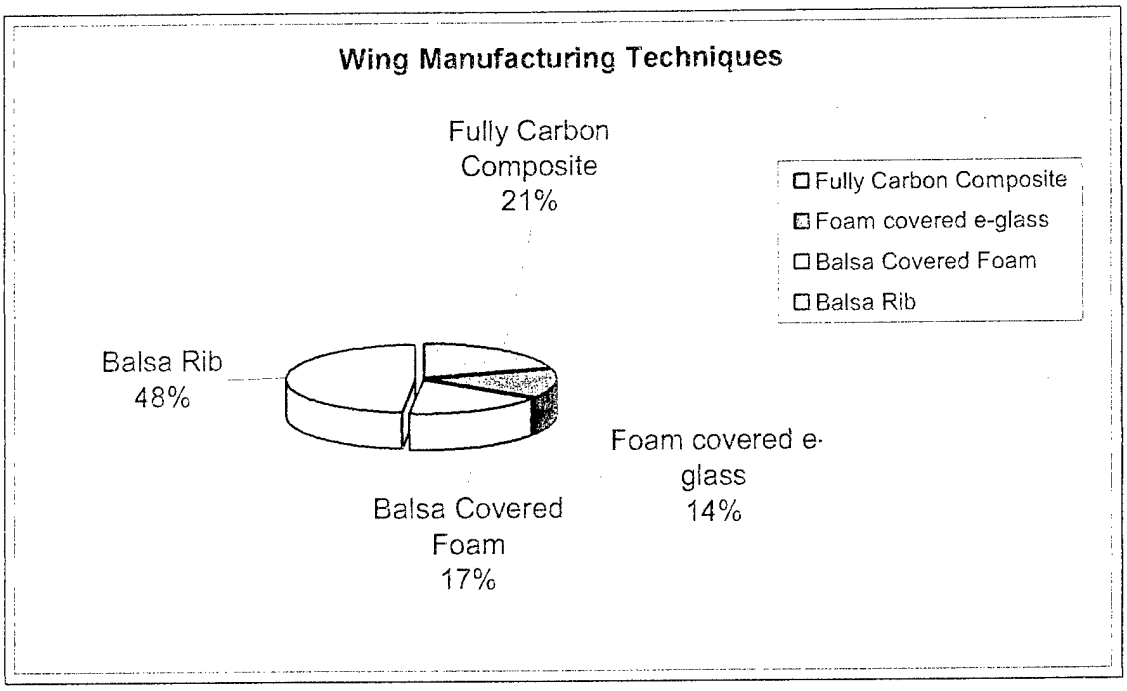


Figure 3.5 Competitor Study Results; Manufacturing Techniques

3.2 Alternative Configuration Concepts

In this section each component of the aircraft was investigated separately and trade-off analysis were done between the alternatives related with them. Competitor study and mission requirements are the main driving factors to reduce the number of these alternative concepts,

Competitor study stage is where the properties of a number of aircrafts that have the similar mission characteristics with the current design are evaluated. In the determination of possible configuration concepts, aircrafts that had been involved in the past years DBF Competitions' were taken as the basis.

In the oncoming subsections, conceptual design considerations of each component of the aircraft are summarized.

3.2.1 Wing Geometric Parameters and Placement

There are a number of parameters that determines the geometrical design of a wing such as sweep, twist, dihedral, aspect ratio, taper ratio, wing orientation with respect to the fuselage.

Aspect ratio, taper ratio, dihedral and wing incidence and wing vertical location are the key parameters that describe the performance of an aircraft's wing flying at low Reynolds's Number.

- Aspect ratio: A finite aspect ratio creates 3-D effects, namely induced drag due to tip vortices. The higher the aspect ratio, the lower the induced drag is. From this point of view, a high aspect ratio wing is seemed to be aerodynamically preferable to a low aspect ratio wing. Also, (L/D)max increases with increasing aspect ratio. However, aspect ratio is restrained by structural and manufacturing aspects.

- Taper Ratio: It is known that the elliptic wing has the most effective lift distribution that can be obtained from a given planform area from the point of view of induced drag. Because of structural and manufacturability considerations, almost no wing has elliptic planform. For a rectangular wing, aspect ratio defines how close the lift distribution on the wing is to that of elliptic planform.
- Dihedral: Wing dihedral is the angle of the wing with respect to the horizontal when seen from the front. Positive dihedral tends to roll the aircraft level whenever it is blanket. Generally high-wing configuration does not need a dihedral because of the inherent dihedral effect of this wing configuration.
- Wing incidence: It is the pitch angle of the wing with respect to the fuselage. Wing incidence angle is chosen minimize the drag at some operating condition, usually cruise. The incidence angle is chosen such that when the wing is at the correct angle of attack for the selected design condition, the fuselage is at the angle of attack for minimum total drag.

3.2.2 Alternative Tail Configurations

Main function of a tail in an aircraft is to stabilize and trim the aircraft. Tail performs these tasks with its surfaces that stabilize the aircraft, called stabilizer, and that supply the sufficient control force for pitch and yaw motions, called elevator and rudder. Different tail configurations were illustrated in Figure 3.6 and discussed in this section all of which has different aerodynamic, stability, control and structural characteristics.

Because of its simplicity and good structural attributes, conventional tail is the most suitable configuration. However the horizontal tail should be sufficiently far back that at stall the wake of the horizontal tail does not mask the rudder on the vertical tail.

T tail is the second most preferred configuration. For the T tail, the structure is heavier; the vertical tail must be strength to support the aerodynamic load and weight of the horizontal tail. On the other hand, the horizontal tail acts as an endplate on the vertical tail, allowing the vertical tail to experience a smaller induced drag and higher lift curve slope; hence the aspect ratio of the vertical tail can be made smaller when the T tail configuration is used. However in T-tail configuration the horizontal tail is not in the slipstream of the propeller, so take off distance of an aircraft with T-tail configuration is longer that that with a conventional tail. Moreover, horizontal of a T-tail configuration may be ineffective of controlling the aircraft in the case of stall because it lies in the wake of wing in such cases.

V tail is advantageous in terms of weight and wetted area considerations. However the handling qualities of an aircraft with V Tail are not good when compared with other configurations.

The canard configuration carries a positive lift that yields an efficient configuration. Also stall characteristics of an aircraft with a canard configuration is good. However, the adjustment of the center of gravity is difficult in this configuration.

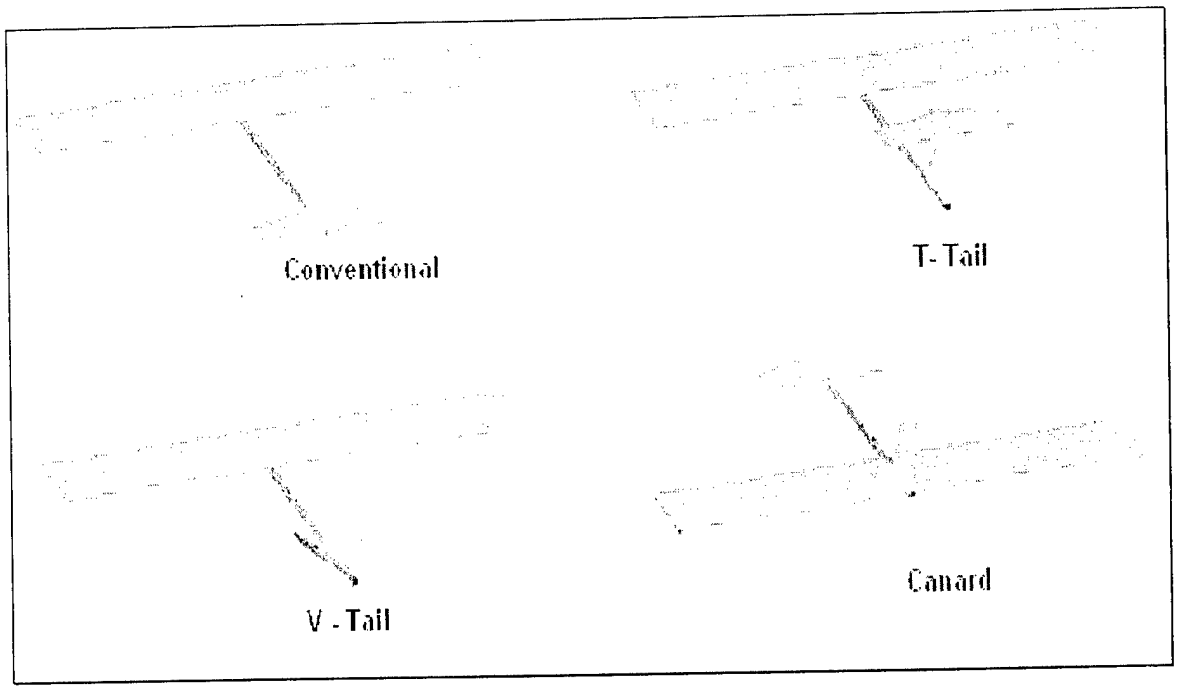


Figure 3.6 Basic Tail Configurations

3.2.3 The Fuselage

Fuselage configuration is determined by the cross-sectional and lateral view of the fuselage. The cross-sectional view is mainly driven by the required interior dimensions and manufacturability of the structural members that build up the fuselage. Selection of the lateral view, however, is an aerodynamic and artistic consideration. Different fuselage configurations that are obtained by changing the fairing of the fuselage towards the tail is shown in figure 3.7. Tail boom, tail boom with aft fairing and lofted fuselage configurations with the rectangular or circular/elliptical fuselage cross sections are the ones that are considered during conceptual design of the fuselage.

Analysis and engineering judgments reveal that the fully lofted fuselage with circular/elliptic cross-section is the most preferable one in terms of aerodynamics. However it is the heaviest configuration among the others. Tail boom configuration, on the other hand, prefers the lightest weight but with a penalty of drag. Tail boom with an aft fairing is a compromise between these two extremes. It offers acceptable weight and drag characteristics.

It is obvious that it would be better to have a circular cross-section instead of rectangular one from the point of view of aerodynamics. However if the manufacturability of the alternatives is considered, rectangular cross-section seems to be more suitable. It also provides flat surfaces for assembly of other components such as wing, tail and landing gears.

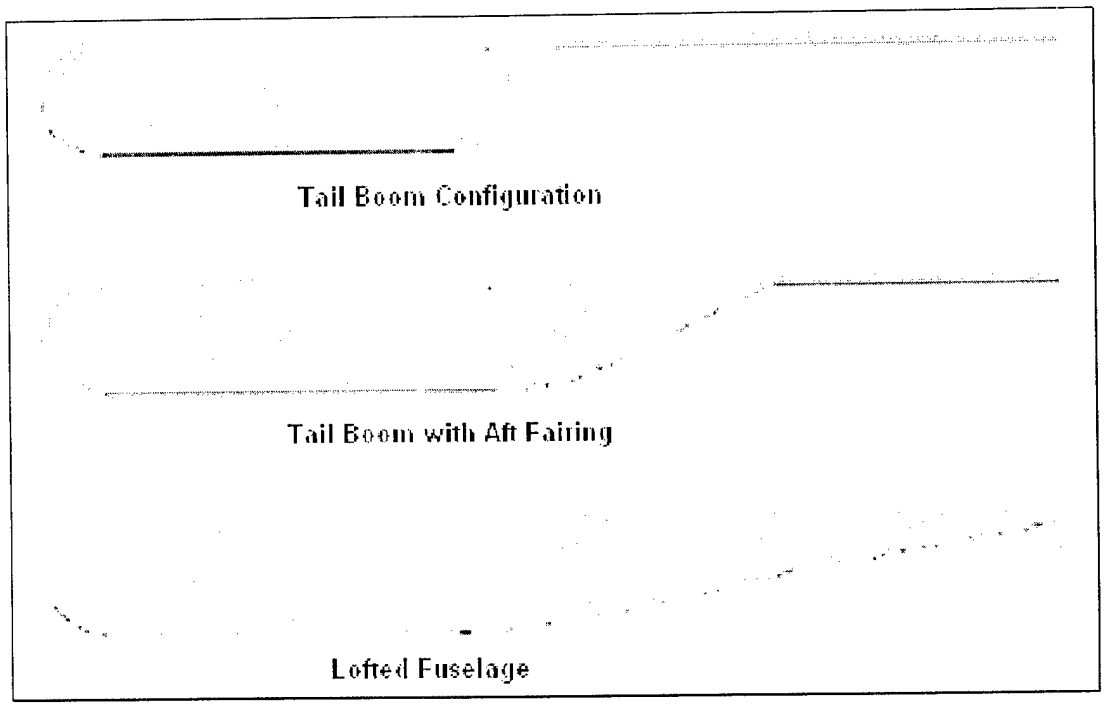


Figure 3.7 Fuselage Lofting Alternatives

3.2.4 The Landing Gears

Landing gears are the components that determine the handling quality characteristics of the aircraft on the ground and take-off and landing performances. There are two configurations that can be thought as alternatives for our case which are the tail dragger and tri-cycle configurations as shown in figure 3.8.

Another aspect to be decided in the design of the landing gears is the mount point of the main landing gear. Although the connections and joints are the subjects of detailed design stage, at the conceptual design stage mount points of landing gears should be determined without going into the details of joints. This analysis is itself a trade off besides the selection of landing gear configuration. One alternative is to mount the landing gear to the wing; another is to mount it to the fuselage. The former is used generally for low wing configurations.

In an aircraft, the landing gears are the strongest structural elements because they must withstand high impact loadings in landing. Being structurally strong means being heavy; therefore the landing gears should be 'critically' designed, that is their margin of safety should be smaller than that of other structural members in order to save weight. At this stage, determinations of possible materials that can be used to manufacture landing gears have to be made. There are three possible material alternatives as aluminum, steel and carbon-composite. Aluminum is relatively light compared to steel but it has low strength than steel. High-strength aluminum alloys offer good strength characteristics but they do not permit some manufacturing operations such as welding. Composite materials have relatively high strength and low weight; but in the case of proper manufacturing. In order to have desired properties, carbon-composite manufacturing has to be almost perfect. Moreover, composites

also do not allow many manufacturing operations. Also, elastic characteristics of aluminum and composites are not as good as that of steel.

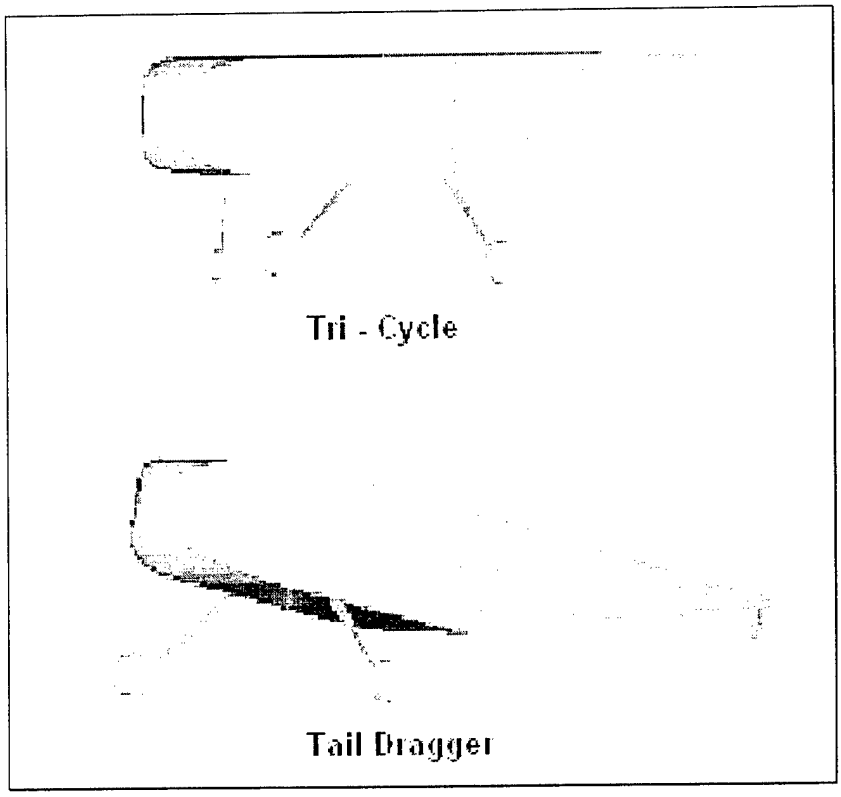


Figure 3.8 Landing Gear Alternatives

3.2.5 Propulsion System

In the conceptual design of the propulsion system, main concerns were to decide on the location and number of engines. Also, propeller configuration alternatives were traded off.

Multi-engine option provides a safe landing in case of an engine failure during flight. However there is no need for such safety feature because of the high aspect ratio of the wing which provides enough gliding characteristic to the aircraft. Also the multi engine concept has some problems with propeller synchronization. Another problem of having multiple engines is that in such a case the engines must be mounted on the wing. This has two possible penalties; firstly the wings may need to be strengthened; secondly the engines may disturb the flow over the wings. A more detailed numerical evaluation of multi-engine considerations is taken into account more detailed in the Table 3.1.

Other decision that has to be made in the conceptual design phase is whether the aircraft would be a pusher or a tractor one. Although a pusher-prop configuration was also taken into consideration, it was found to be unnecessarily complicated. Therefore, pusher-prop configuration was eliminated.

3.3 The Scoring System

During the conceptual design phase, a number of possible configurations for each component of the aircraft are determined. However it is not enough to list all of these configurations; in fact

determining the superior concept among all is more important. In order to find out the best concepts for the components, the design team developed a scoring system and performed a Figure of Merit analysis (FOM). This analysis revealed the advantages and disadvantages of all the concepts by weighting them according the scoring system. The scoring system is a combination of analytical and statistical evaluations. For analytical evaluations a critical parameter is selected and an estimation of the parameter for different choices was made; as done in the case of determining the effect of aspect ratio on the structural stiffness of the wing.

On the other hand, the competitor study is used as the main resource for developing statistical evaluations.

In the FOM analysis, the key design alternatives was scored over 12 for each of the figure of merit together with a multiplier determined by the design team according the importance of the task. A score of 1 is given for a very low contribution to design performance whereas a score of 12 is given for an apparent superiority over the other alternatives/

3.3.1 Figures of Merit (FOM)

Aerodynamics: The major scoring criterion for aerodynamic performance of a lifting surface is its lift to drag ratio, whereas for non-lifting components it is directly the drag caused by that component, i.e. the skin friction drag.

Stability & Control: In this part of the FOM analysis, individual contributions of each component to the stability and controllability of the design.

Structural Stiffness: Load carrying capacity of each member is evaluated by this FOM. A member with sufficient stiffness together with high elasticity gets higher score from this FOM.

Weight: Each component is scored according to its contribution to total aircraft weight. Obviously, a design alternative weighing less gets the highest score for this FOM.

Cost: Each design alternative is scored for its cost which is calculated by the given rated aircraft cost model.

Manufacturability: A better configuration does not mean that it must be selected among the others because sometimes manufacturing capabilities are not enough to produce it. Especially for complex configurations, manufacturability is the major reason for refusing such alternative configurations. In this FOM, each alternative is scored according to its ease of manufacturing by assigning high values to `easy to manufacture` alternatives.

3.3.2 Results of the Trade-Off Analysis

Table 3.1 illustrates the scores of each design alternatives corresponding to the prescribed figures of merit. The conceptual form of the aircraft is shaped by selecting the alternatives that gets the highest score. The selections are:

- Wing Configuration: A non-tapered wing with aspect ratio between 6 to 9 and located at the top of the fuselage (high wing).
- Tail Configuration: A conventional tail configuration.

- Landing Gear Configuration: Tri-cycle type non-retractable landing gears without a suspension system. Main landing gear is mounted on the fuselage.
- Propulsion system: A single tractor engine.

CONCEPTUAL DESIGN ALTERNATIVES			Aerodynamics	Stability & Control	Structural Stiffness	Weight	R.A.C.	Manufacturability	Maintenance	Weighted Average
Wing Geometry		Multiplier	5	4	4	5	3	3	-	
Aspect Ratio		High (9-12)	12	8	4	8	4	-	-	6.7
		Mid (6-9)	**10**	**10**	**8**	**12**	**7**	**-**	**-**	**8.5**
		Low (3-6)	3	12	12	9	12	-	-	8
Taper Ratio		**None(1)**	**7**	**-**	**10**	**12**	**12**	**12**	**-**	**8.6**
		High(0.7-0.9)	8	-	10	10	11	6	-	**7.5**
		Mid (0.35-0.7)	12	-	12	9	9	6	-	**8.3**
		Low(0-0.35)	8	-	10	8	7	4	-	**6.4**
Wing Position w.r.t. fuselage		**High**	**8**	**12**	**12**	**12**	**12**	**11**	**-**	**11**
		Mid	10	9	8	10	11	6	-	**10**
		Low	12	6	10	11	11	12	-	**10**
Tail Configuration		Multiplier	5	5	3	4	2	3	-	
		Conventional	**10**	**12**	**12**	**11**	**12**	**12**	**-**	**11**
		T-Tail	11	6	11	9	12	10	-	**9.5**
		V-Tail	10	4	11	12	11	8	-	**9**
		Canard	12	2	12	12	12	12	-	**9.7**
Landing Gears		Multiplier	2	-	5	4	2	3	1	
Configuration		**Tricycle**	**12**	**-**	**12**	**10**	**-**	**11**	**-**	**9.2**
		Tail Dragger	8	-	10	12	-	12	-	**8.8**
Main LG Position		**Under Fuselage**	**12**	**-**	**12**	**11**	**-**	**12**	**-**	**9.6**
		Under Wing	10	-	7	12	-	10	-	**7.8**
Retractable		Yes	12	-	6	5	4	6	8	**6.4**
		No	**2**	**-**	**12**	**12**	**12**	**12**	**12**	**11**
Suspension		Oleo	12	-	12	3	-	4	6	**6.7**
		Spring	4	-	10	10	-	8	11	**7.2**
		None	**10**	**-**	**10**	**12**	**-**	**12**	**12**	**9.8**
Number of Engines		Multiplier	2	3	5	4	2	3	1	
		Single	**12**	**10**	**12**	**12**	**12**	**12**	**12**	**12**
		Double	6	12	8	9	9	5	6	**8.2**

Table 3.1 FOM analysis for the conceptual design

3.4 Water Dumping System Concepts

Any means of pressurizing the water tank is not allowed, the tank must be emptied by gravity alone.

It is desirable to have an aircraft center of gravity (cg) position essentially independent of the water tank configuration. That implies planform centroid of the payload to be close to the aircraft cg. A

rectangular shape of water tank was considered in order to adjust water tank cg easily. However, according to fuselage shape and battery placement water tank shape may be changed after detailed design.

Dumping system of the water tank is a very important subject to be determined carefully. Mechanism of the dumping is considered consisting of stopper and servo. Some gasket must be used at the stopper in order to prevent water leakage. According to the requirements of the competition, dumping hole must be half of inch; since this hole is small some accelerating mechanisms might be designed. For this since any pressurizing mechanism is not allowed, some aerodynamic characteristics may be used. For example, Bernoulli's principle can be used, that is, air may be accelerated locally at the dumping hole.

Also holes must be opened at the top of the tank, in order water to be dumped fast.

These holes must not cause any water leakage, hence placement of these top holes will be determined after fuselage and tank shape is certain.

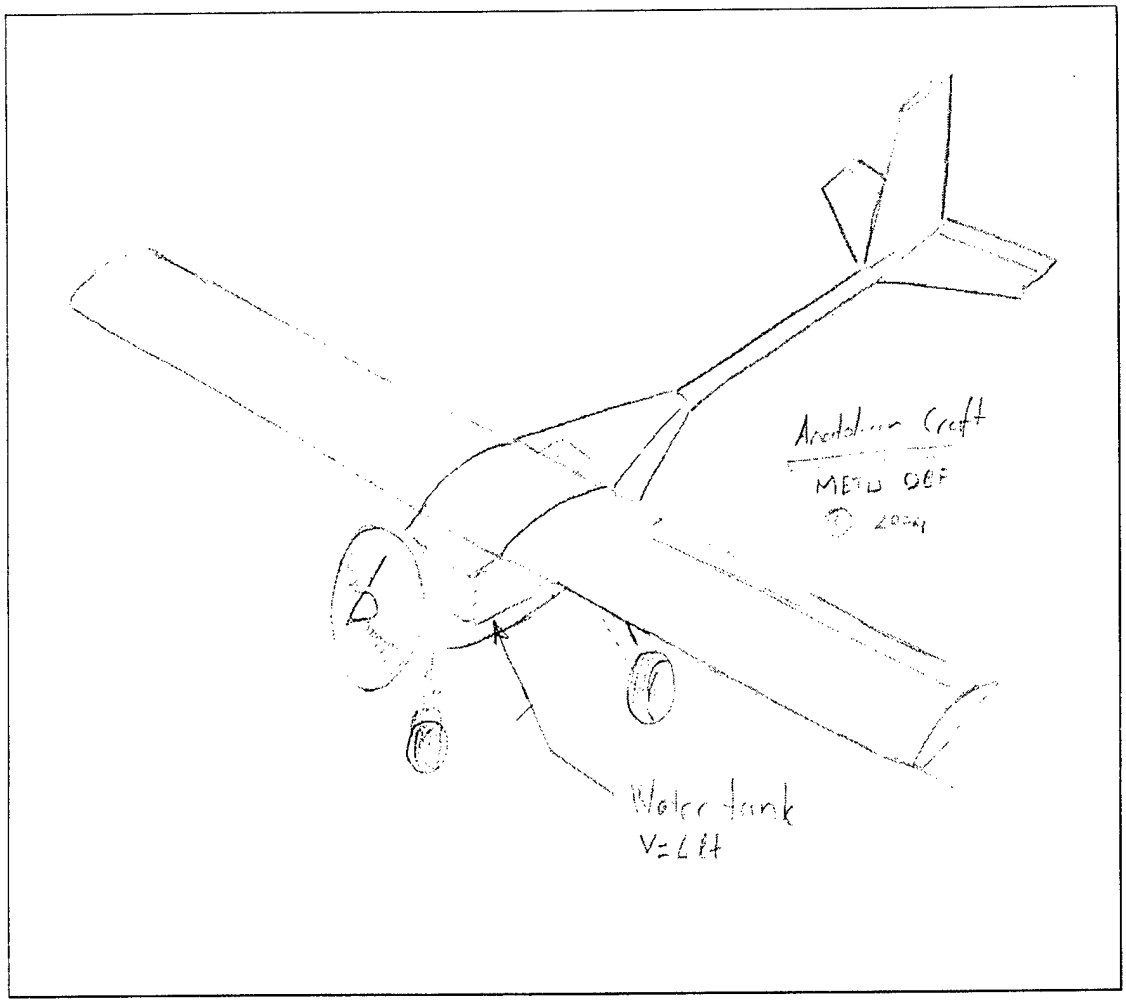


Figure 3.10 Conceptual Sketch of the Aircraft

CHAPTER 4 - PRELIMINARY DESIGN

Preliminary Design is the stage at which the most competitive and effective aircraft that meets the mission requirements is selected from the possible configurations. These possible configurations are developed in the Conceptual Design phase. Approximate geometry, weight, power and other characteristics of the final configuration is determined and almost fixed in the preliminary design stage. Airfoil and planform selection, number of batteries, motor-propeller combination, size horizontal and vertical tail surfaces, size of control surfaces and are the major sizing parameters to be determined. Also strength of the main structural elements is to be determined in preliminary design stage.

There are many different approaches in sizing of an airplane. Usually a parameter, which is critical to the mission requirements, is selected as the basis for the sizing. In our case, the stall speed seemed to be the most critical performance parameters after studying the mission requirements. Therefore the design team agreed to employ the sizing study in basis of stall speed. This sizing sequence starts with a selection of the stall speed and immediate estimates of the take-off gross weight and maximum lift coefficient. This is followed by sequential sizing of aerodynamic surfaces, propulsion systems and structural elements. With the use of these initial sizing values, the weight estimation is refined and a revision of concept layout is done when necessary. This is in fact an iterative procedure, which is expected to converge after a sufficient number of cycles.

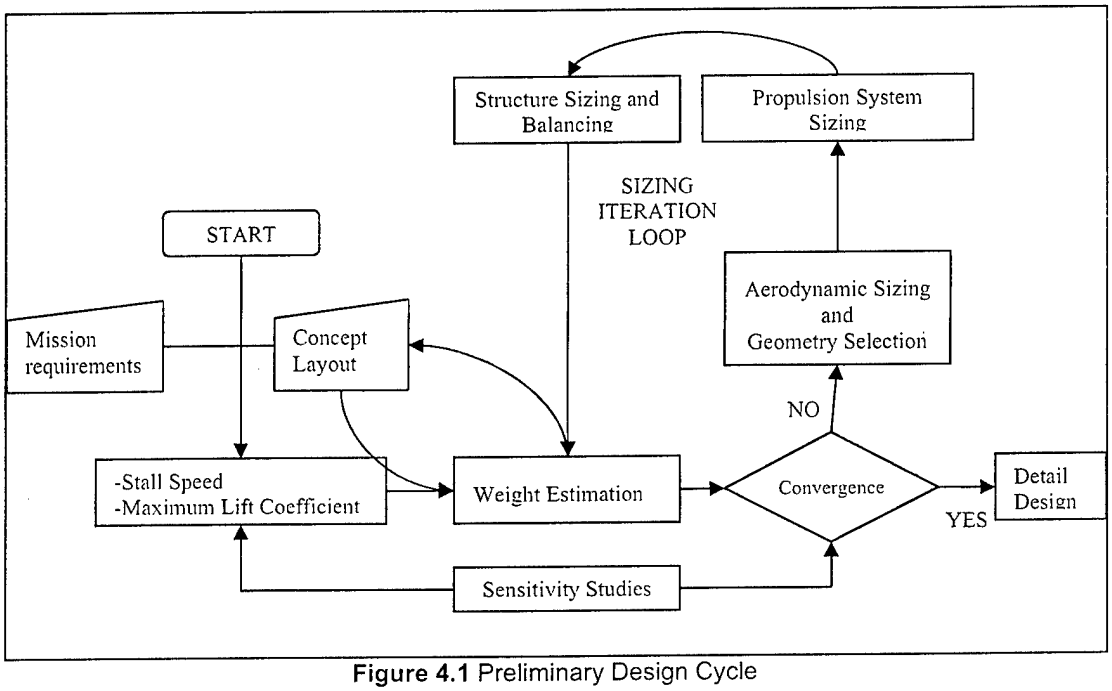


Figure 4.1 Preliminary Design Cycle

In the first mission, fire bomber, the aircraft is required to discharge 4 liters of water in 1000 feet and a 360° loop. A test was performed to determine the approximate time of discharging the water in the required distance. This time interval reveals the approximate 'minimum flight speed', i.e. the stall speed.

The remaining sections of this chapter are organized in the sequence as shown in the Figure 4.1, which is an illustration of the preliminary design cycle followed.

4.1 Selection of Stall Speed and Estimation of Maximum Lift Coefficient

As described previous part, the aircraft is required to discharge 4 liters of water in a given part of the mission profile.

The driven constraint for the stall speed is the water discharging part of the mission profile because the aircraft has to be slow enough to empty its water tank in this given distance. Therefore, in order to make estimation for the stall speed, the design team needed the approximate time to complete the above mission profile in which the water is discharged. As depicted earlier, the team performed a test to achieve this goal. The test was performed with a 5-liter-water bottle having one 0.5-inch-hole at the bottom and a air pressure supply hole at the top. Flight conditions were simulated by hanging the bottle from the window of an automobile cruising at a constant speed close to average cruise speed of the competitor aircrafts. After getting the approximate time of flight, the first crude estimate for the stall speed can be made easily. However it is not possible to fix the stall speed at the first estimate because other considerations, mainly those which are about the geometry of the aerodynamic surfaces, limit the stall speed. To see the limiting cases of stall speed, historical trend curves are examined. The two which are constructed with the date of past year competitor aircrafts are shown in figure 3.

It is known that stall speed is one of the main parameters in finalizing the geometry of the wing, because it directly affects the wing loading which is nothing but the weight of aircraft divided by the wing planform area. Take off ground roll distance also has an important effect on wing loading but in our case the effect of take off distance constraint on wing loading is not as considerable as that of stall speed. As seen from these discussions, the selection of the stall speed and configuring the wing is an iterative process. In order to perform this process, the design team generated a computer program named "PreSize".

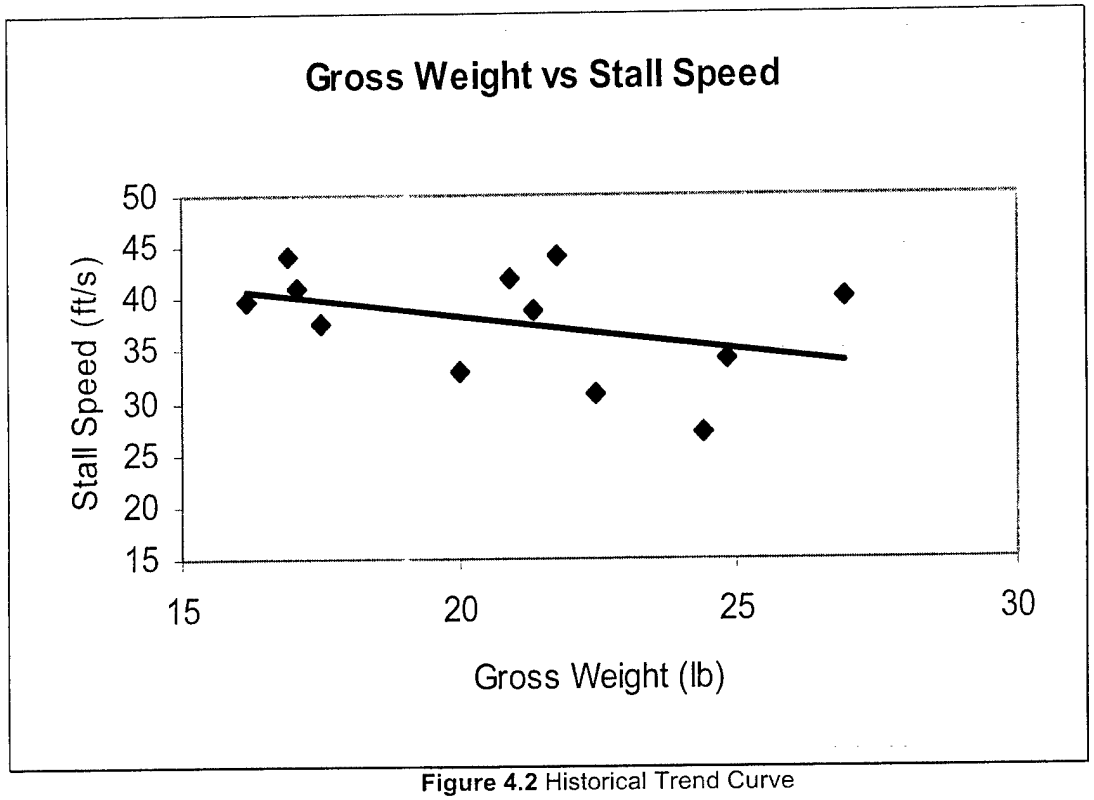


Figure 4.2 Historical Trend Curve

Another input parameter to the design cycle is a rough estimation of maximum lift coefficient of the wing, which is to be revised in preceding steps of the cycle. A value of 1.8 is selected as a first estimation of the maximum lift coefficient. However, in order to get the planform area an accurate estimation of the maximum lift coefficient should be made. This accurate estimation can only be made with the information set by wing airfoil, horizontal tail airfoil and flap configuration.

4.2 Initial Weight Estimation

Various methods of getting an initial weight estimation of the aircraft concept exist in the literature. Since the preliminary sizing of an R/C controlled airplane involves relatively simple and quick methods of iterations, one may not need a very close approximation of the actual weight at the beginning but may start with a crude approximation and improve it within the sizing iterations.

A quick way of performing initial weight estimation can be performed by assuming the take-off gross weight of the aircraft being equal to four times the payload weight. Knowing that the payload weight is specified as 8.82 lbs, the initial estimation of the take-off gross weight turns out to be 35.28 lbs. However, this estimation does not fit the data given by the competitor study which gives a lower value for take off gross weight. Therefore, the team agreed to use the competitor data instead of empirical estimations. A crude estimate can be made by examining the past years aircrafts and comparing the empty weight of these aircrafts with the payload weight. The competitor data is given below.

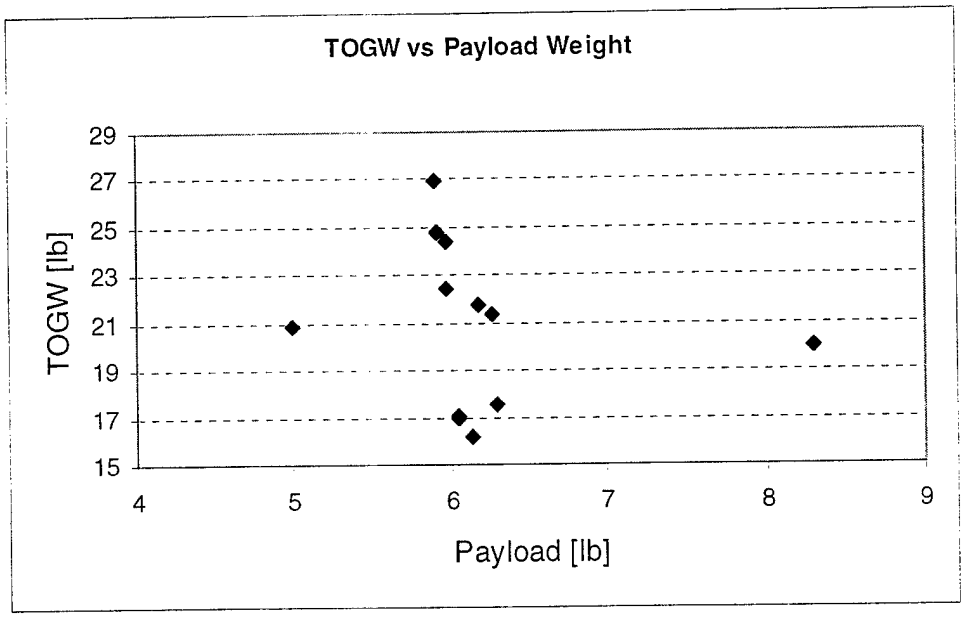


Figure 4.3 Historical Trend Curve for change in TOGW with Payload

There are variations in TOGW although payload weights are nearly equal. This trend can be explained by the dependency of the TOGW in the manufacturing technique and experience.

4.3 Aerodynamic Sizing and Geometry Selection

Aerodynamics mainly concerns the lift, drag and moment created by the components of the aircraft that are immersed to the airflow. The intention of this phase of design is to form and size the aerodynamic surfaces and the fuselage so as to maximize the lift-to-drag ratio of the aircraft at a reasonable range of rated aircraft cost. The analysis and selection of individual components of the aircraft is presented in the following sections.

4.3.1 The Wing

The wing of an aircraft is, without doubt, the most important component. It is responsible for maintaining flight in the first place. Thus the selection of other components relies upon the configuration of the wing. Design of the wing means determination of wing configuration. Wing configuration is determined by airfoil sections, wing planform area, aspect ratio, taper ratio, sweep angles, incidence angle of the wing, dihedral angle, wing twist and location of the wings on the fuselage. In the determination of wing configuration, manufacturability, i.e. the ease of manufacture, must also be kept in mind.

Having estimated the stall speed, maximum lift coefficient and the take-off gross weight of the airplane, the wing area can easily be calculated based on these estimations. In every sizing cycle (See Figure 4.1), the wing area is recalculated using the refined estimation of the aircraft weight. After a sufficient number of iterations, only minor changes are observed in weight estimation between successive steps.

Airfoil Selection Criteria

Airfoil mainly determines the maximum lift coefficient, therefore the planform area. It also affects drag characteristics and required power. The main challenges of this competition are to reduce planform area and power consumption rate. Therefore considerable amount of time and effort should be spent on the airfoil selection and analysis.

Any aerodynamic analysis should start with a clear definition of the flow field properties. The most critical flow field properties are Reynolds and Mach Numbers that are also known as similarity parameters. In our case of low-speed flow, lack of compressibility effects may allow us to neglect the influence of the Mach number on the flow field. However, Reynolds Number has a great influence on the flow field.

Also, the mission of the aircraft to be designed is a driven parameter for the selection of airfoil. This year's missions require a high lift yet low drag airfoil, because the aircraft is required to perform both low speed and high speed missions. High lift can be provided by high camber airfoils but attention should be paid to the flow separation characteristics of the airfoil which is very much affected by Reynolds Number.

The design Reynolds number for this project is in the same range with the birds and some of the small model airplanes, which is the range between 3E5 to 4E5. At this range of Reynolds numbers, a laminar to turbulent flow transition may occur. Airfoil performance is relatively low in this range according to Lissaman. Recently, there have been attempts to improve the performance of airfoils at such low Reynolds Number due to the developments on small remotely piloted air vehicles (RPVs). These RPVs have a wide range of Reynolds Number and are frequently required to operate at Reynolds Numbers significantly below half a million.

It can be stated that at lower Reynolds numbers the viscous effects are relatively large, causing high drags and limiting the maximum lift, while at the higher values (>70000) the lift-to drag ratio improves. The main reason of this is that the performance of low Reynolds number airfoils is dictated by poor separation resistance of the laminar boundary layer. At low Reynolds numbers, the boundary layer at the onset of adverse pressure gradient is still laminar and therefore unable to withstand significant pressure gradients. In laminar flow, as the lift is increased, the adverse gradients become more severe and laminar separation occurs; limiting the lift coefficient and significantly increasing drag.

As a conclusion, it turns out that flight at low Reynolds numbers is a problematic issue. The problems caused by the phenomenon occur at these flight regimes can be eliminated with a careful analysis and selection of airfoil section.

After analyzing the nature of flow field at low Reynolds numbers, the team started to decide on the alternative airfoil sections which are suitable for our application from the point of view of the above discussions. During the determination procedure of the nominees, wind tunnel data of large amount of airfoils were studied.

Candidate Airfoils

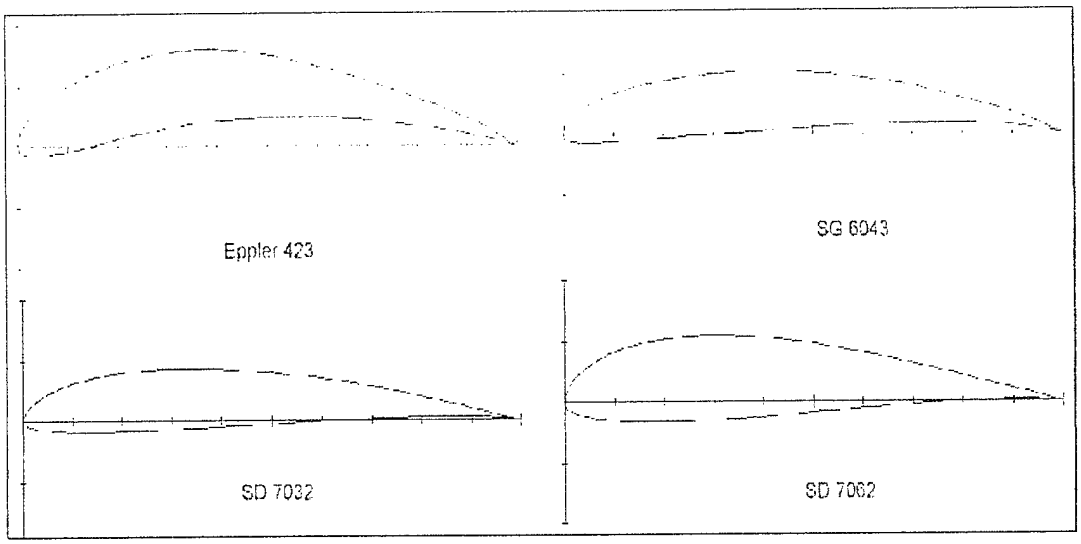


Figure 4.4 Candidate Airfoils

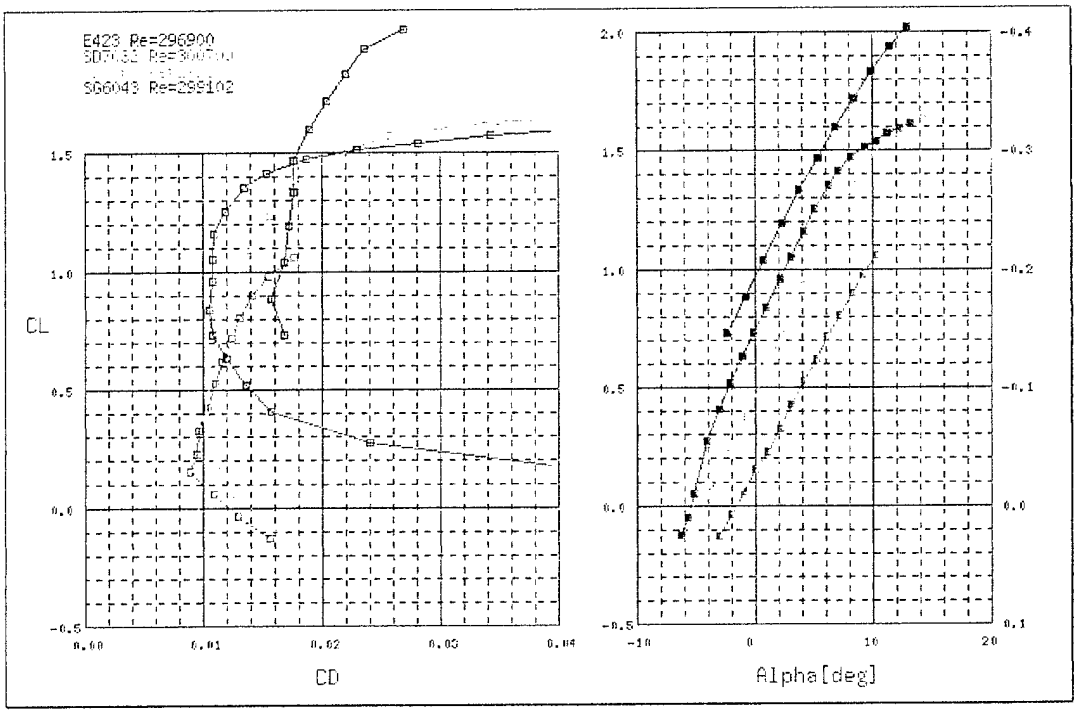


Figure 4.5 Wind Tunnel Test Data of the Airfoils

After the wind tunnel data are gathered, a number of competitor designs were made to determine which one yields the superior design compared to others. This analysis is performed by the use of computer program "PreSize". The results of this trade of analysis are given below.

Airfoil Section	Eppler 423	Airfoil Section	SD 7062
Wing Span [ft]	8.53	Wing Span [ft]	8.99
Wing Area [ft2]	8.956	Wing Area [ft2]	9.451
TOGW [lb]	22.5	TOGW [lb]	23
Airfoil Section	SG 6043	Airfoil Section	SD7032
Wing Span [ft]	9.09	Wing Span [ft]	9.25
Wing Area [ft2]	9.53	Wing Area [ft2]	9.677
TOGW [lb]	23.2	TOGW [lb]	23.4

Table 4.1 Comparison of Airfoil Sections

The results are clearly demonstrates that Eppler 423 section yields a lower value for wing area. This brings a decrease in both take off gross weight and rated aircraft cost. Therefore Eppler 423 is selected for our aircraft because of its high lift characteristics at low Reynolds numbers which bring advantages from the point of view of weight and rated aircraft cost. To see the behavior of Eppler 423 in detail, Navier-Stokes solutions of the section were studied whose results are presented in the next section.

Navier-Stokes Solutions

The Navier-Stokes solutions were performed with a Reynolds averaged Navier-Stokes solver. The turbulence model used was Spalart-Allmaras Model with 3rd order upwind scheme and structured viscous c-type grid.

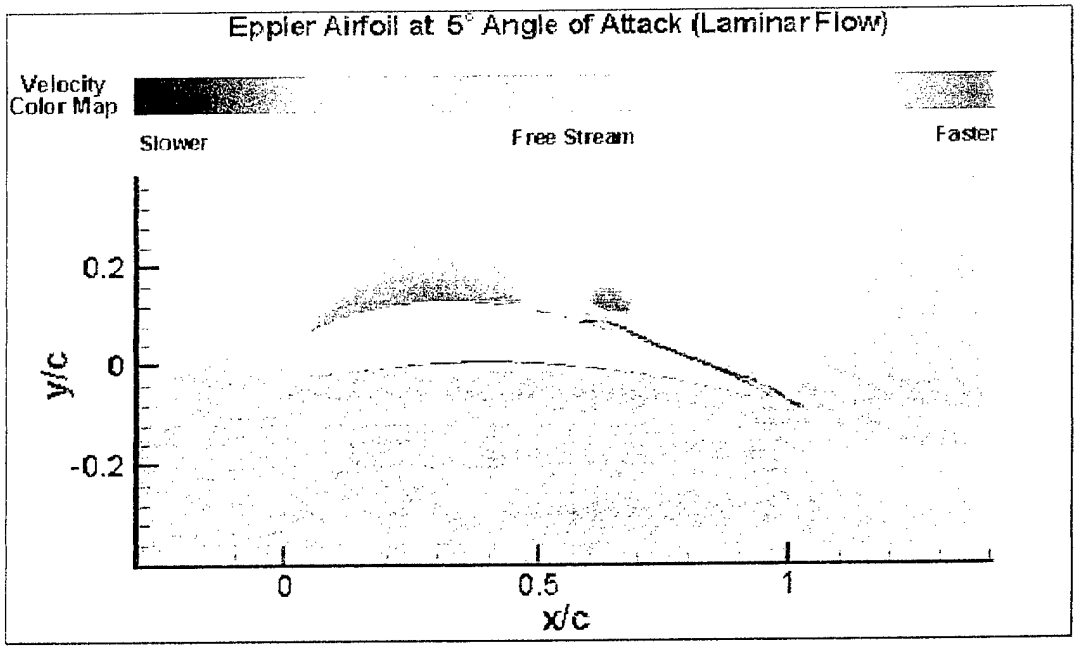


Figure 4.6 Eppler 423 NS Solution at 5^0 of angle attack, Laminar Flow

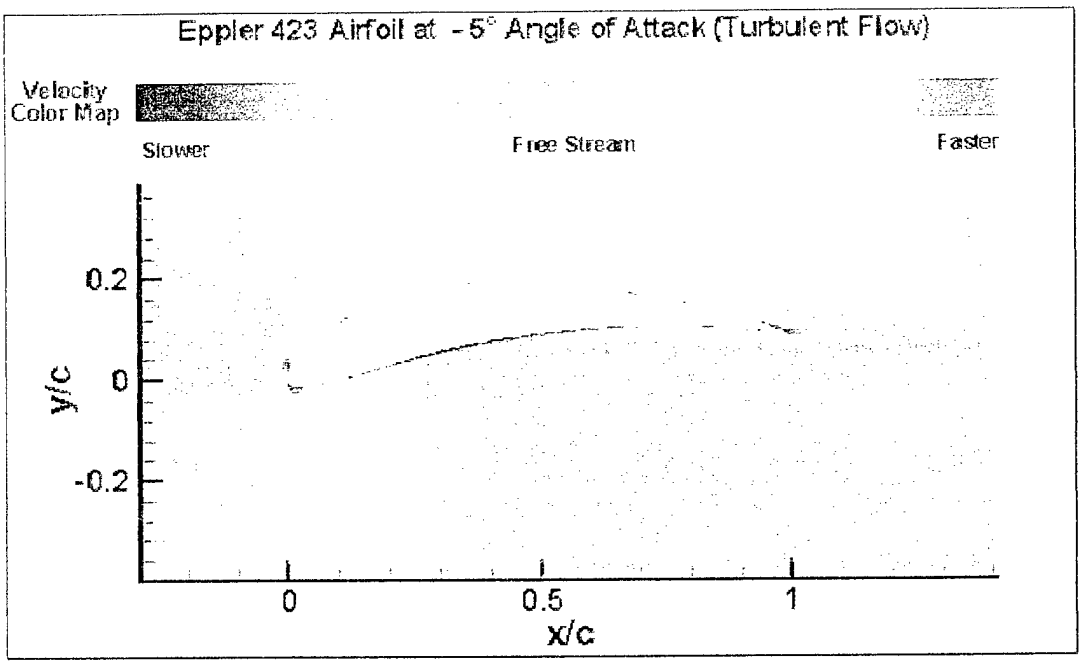


Figure 4.7 Eppler 423 NS Solution at 5^0 of angle attack, Turbulent Flow

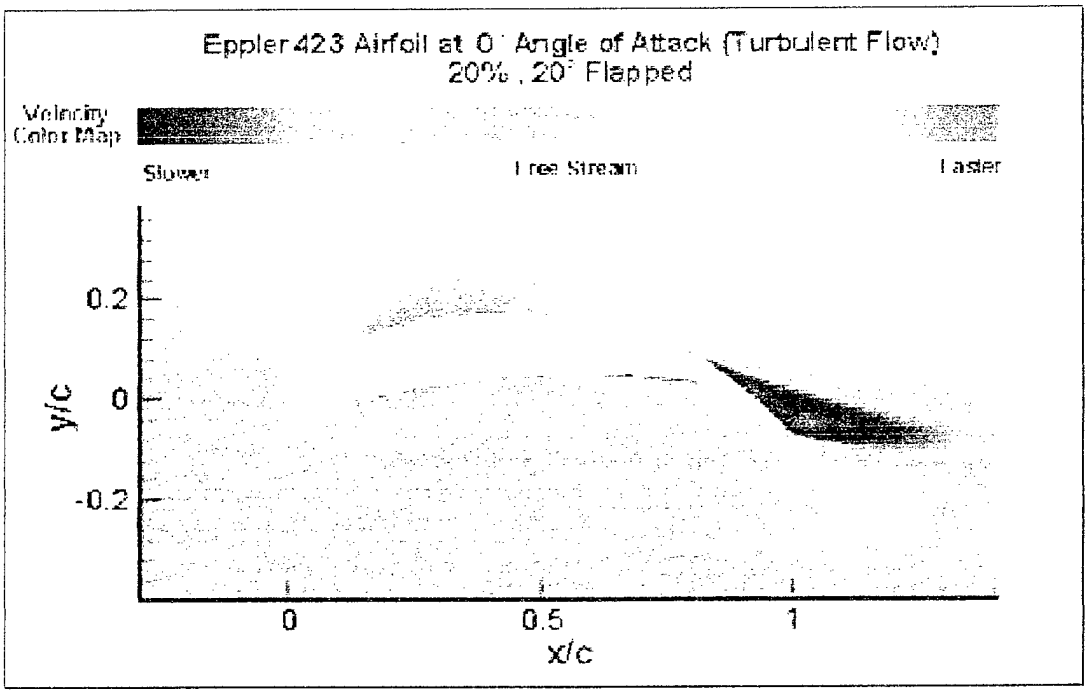


Figure 4.8 Flapped Eppler 423 NS Solution at 5^0 of angle attack, Turbulent Flow

The Navier-Stokes analysis revealed that turbulent flow has a much greater resistance to flow separation than laminar flow. This is due to the fact that, turbulent flow is more energetic than laminar flow. It has more energy to resist adverse pressure gradients and disturbances.

Geometry of the Wing

The key geometric parameters of a wing are aspect ratio, taper ratio, dihedral angle and section geometry.

Lateral stability requires either a high wing configuration in which the wing is located at the top of the fuselage or positive dihedral angle. In the conceptual design phase, it was already decided to have a high wing configuration in the light of figures of merit.

In the conceptual design phase, ranges of values for the aspect ratio and taper ratio were scored and selected. According to these choices, a straight wing with an aspect ratio ranging between 6 and 9 was considered as advantageous in terms of the prescribed figures of merit analysis.

The above considerations significantly narrow down the spectrum of alternatives concerning the wing planform geometry. In fact, the only parameter left to be determined is the aspect ratio. Ideally, the greater the aspect ratio is, the higher the efficiency of a wing, because a higher value for the aspect ratio decreases the 3-D effect of the finite wing. From the point of view of structural weight and stall characteristics, a low aspect ratio is desirable. This constitutes a trade-off problem to be solved by collaborative work of aerodynamic and structure groups. After a few steps of iteration loops (See Figure 4.1), the structure group has gained an idea of the wing structure and developed a model for wing weight at different aspect ratios. According to this model, the wing weight is calculated for different aspect ratios and presented in Figure 4.4. The figure reveals that the aspect ratio, which minimizes the weight of the wing, is estimated as 8.

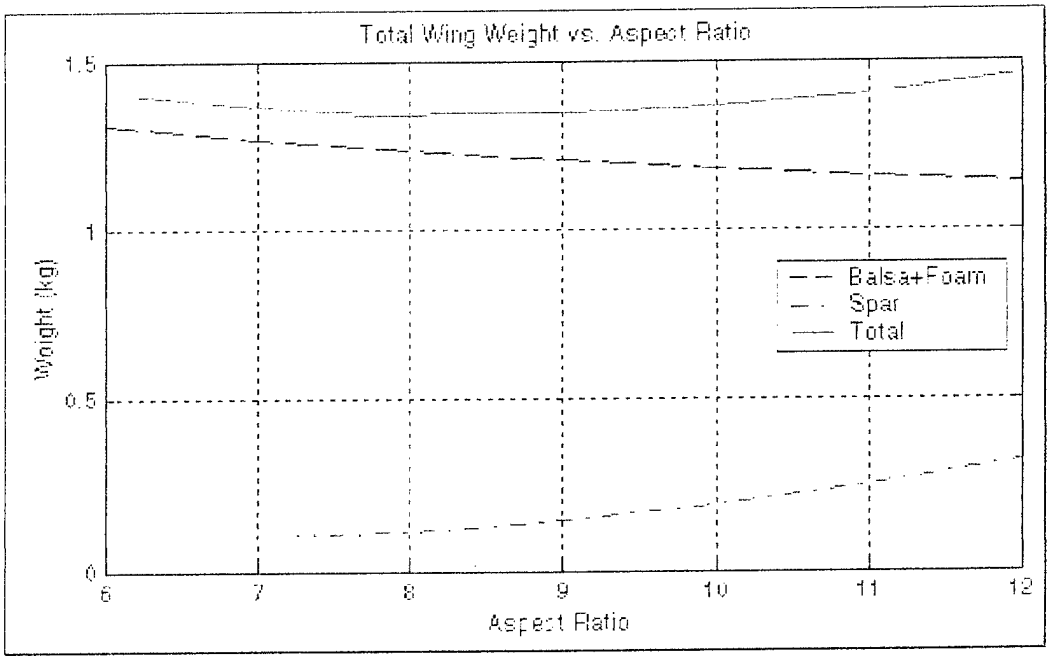


Figure 4.9 The variation of wing weight with aspect ratio

4.3.2 Empennage Group and Stability

The sizing procedure of the tail surfaces is a very complicated and problematic. Lots of problems need to be solved, such as the restriction on the horizontal tail span that it can not exceed the quarter length of the wing span, the designed tail have to overcome the high moment produced by the wing, because the length of the fuselage directly effects the Rated Aircraft Cost and the requirement of fitting all parts of the aircraft into a box which has specified dimensions. These problems can be solved by iteration on the design. Therefore, a program is developed to gain speed while making routine calculations. The program takes the center of gravity location of the aircraft and the data related with wing as input and calculates the tail boom length, the required tail incidence, tail surface area and the static margin of the aircraft.

Since design aircraft could not take off at the required distance, trailing flaps were decided to be placed. Also, since second mission of the aircraft will be high speed flight with no payload, excess lift will occur, thus for this mission reversed flap must be used.

Furthermore, flaperon configuration was decided to be used which is more advantageous than separate flap and aileron configuration. With flaperon configuration less number of servos are used, that is for separate flap and aileron configuration four servos are needed, however, for flaperon configuration two servos are needed. Therefore, flaperon configuration is more advantageous in the aspects of weight and RAC.

In order static margin not to alter after the water is dumped, the water tank c.g. was decided to coincide with the aircraft c.g, hence longitudinal stability was not going to change much. The negative effect of shaking of water on the stability of aircraft was also considered. In order to minimize this problem putting parallel ribs into the water tank was decided.

In table 4.2 one can see the results of iterations done and the major sizes of the horizontal and vertical tail and the location with respect to the fuselage, the incidence and the airfoil used for both stabilizer surfaces.

Empennage Specifications	
Distance from Wing ac to horizontal stabilizer ac [ft]	3.094
Horizontal stabilizer area [ft^2]	1.97
Vertical stabilizer area [ft^2]	0.861
Horizontal tail angle of incidence [degrees]	2
Airfoil section used for horizontal stabilizer	NACA 0009
Airfoil section used for vertical stabilizer	NACA 0009
Stability	
Static Margin (Empty)	0.094
Static Margin (payload)	0.083

Table 4.2 Estimated Empennage Specifications and the stability characteristics of aircraft for three different conditions.

4.4 PROPULSION SYSTEM

For a propeller driven airplane, major selection criteria for the propulsion system configuration is the static pitch speed to stall speed ratio. This ratio roughly describes the ability of the aircraft to accelerate near the stall speed. There are some regimes for static pitch speed to stall speed ratio such that each of them corresponds to different flight conditions. A static pitch speed to stall speed ratio of 2 corresponds to a critical flight regime and lack of thrust at typical flying speeds, a ratio of 2.5 is described as ideal for modeling purposes, whereas a static pitch speed to stall speed ratio of 3 is a value which is quite suitable for aerobatic flight characteristics. In order not to risk the airplane, it was decided to select a propulsion system configuration having a static pitch speed to stall speed ration near 2.5.

Another criterion that has to be fixed at the preliminary design stage is the number of motors to be used in the airplane. It is known that for a multiple motor airplane, propeller synchronization is a problematic issue. Moreover a multiple motor configuration will yield more weight and therefore more rated aircraft cost. For these reasons, the team decided to use a single motor configuration.

After making the above decisions, the team analyzed combinations of motors, propellers and batteries using the commercially available software "MotoCalc". The tables given below summarize the results of these analyses. First table presents the motor/battery/propeller combinations for the first mission, Fire Bomber, and the second table presents the MotoCalc results for second mission, Ferry Flight.

For the first mission, the important criterion is to have high trust to weigh while keeping the number of batteries minimum, because during Fire Bomber mission, the airplane will have its maximum take off gross weight. Shaded portions of the table 4.3 represent the suitable configurations.

		Astro Cobalt 60 11T#23						Graupner Ultra 3450-7			
Propeller	Number of Cells	Speed to Stall Speed Ratio	Static Thrust to Weight Ratio	Excess Thrust to Weight Ratio	best L/D Airspeed [Amps]	Propeller	Number of Cells	Speed to Stall Speed Ratio	Static Thrust to Weight Ratio	Excess Thrust to Weight Ratio	best L/D Airspeed [Amps]
20x14	28	2.32	0.35	0.13	17.5	20x14	28	2.72	0.47	0.24	28.7
	30	2.45	0.39	0.15	19.5		30	2.86	0.52	0.28	32
	32	2.57	0.42	0.19	21.8		32	2.98	0.56	0.33	35.3
	34	2.71	0.46	0.23	24.2		34	3.12	0.61	0.37	38.4
18x12	28	2.08	0.24	0.04	10.6	18x12	28	2.51	0.35	0.13	18.1
	30	2.21	0.27	0.05	11.9		30	2.65	0.39	0.17	20.7
	32	2.35	0.30	0.03	13.7		32	2.78	0.43	0.21	23.3
	34	2.47	0.33	0.12	15.4		34	2.94	0.47	0.25	25.9
16x10	28	1.79	0.16	0	8.1	16x10	28	2.19	0.23	0.02	9.2
	30	1.92	0.18	0	8.9		30	2.33	0.26	0.05	11.1
	32	2.03	0.20	0	9.7		32	2.46	0.29	0.08	12.9
	34	2.15	0.22	0.01	8.3		34	2.61	0.32	0.11	14.7

Table 4.3 Results obtained from analysis of different propulsion system configurations for the "Fire Bomber" mission

It can be seen from the results that two combinations were selected as candidates for final propulsion system configuration. The first candidate was 30x2400 mAh cells in series feeding a Graupner Ultra 3450-7 motor with 2.75:1 gear ratio driving a 2 blade wooden 18"x12" propeller and the other one was 32x2400 mAh cells in series feeding an Astro-Flight Cobalt 60 11T#23 Motor with 2.75:1 gear ratio driving a 2 blade APC 20"x14" propeller.

FOM	Number of Batteries	Availability of Motor	Availability of Spare Parts	Experience on motor	Weight	Weighted Average
Multiple	5	4	6	3	3	
Astro Cobalt 60 11T#23	10	12	12	12	12	11,55
Graupner Ultra 3450-7	12	10	9	8	10	9,95

Table 4.4 FOM analysis for two motor options

FOM analysis reveals that it would be better to select Astro Cobalt 60 motor instead of Graupner ultra 3450-7. The main reason for this is that the team has been experienced over the years of competition on Astro motors and also he spare parts for Graupner motors are not available to team.

Having decided to use Astro Cobalt 60 11T# 23 motor for the Fire Bomber mission, propeller configuration selection for the second mission was performed. During this analysis, number of batteries was still kept as a choice because analysis for the first mission might have been repeated for a small number of batteries depending on the results of analysis for second mission. It was observed that the aircraft has considerable excess power for the second mission and static pitch speed to stall speed ratio is more than 2.5 as expected. The team decided to keep the battery configuration as it is because even 30 cells give a lower performance for the first mission.

Analysis also revealed that for the second mission it is better to use a small diameter- high pitch propeller. 18x14 and 16x14 propeller configurations can be good candidates for the second mission.

		Astro Cobalt 60 11T#23			
Propeller	Number of Cells	Static Pitch Speed to Stall Speed Ratio	Static Thrust to Weight Ratio	Excess Thrust to Weight Ratio	at best L/D Airspeed [Amps]
18x14	28	2.4	0.42	0.24	14.3
	30	2.56	0.47	0.29	15.9
	32	2.71	0.51	0.32	17.5
18x12	28	2.08	0.41	0.20	12.1
	30	2.21	0.46	0.24	13.6
	32	2.35	0.51	0.27	21.8
16x14	28	2.47	0.28	0.14	10.5
	30	2.63	0.32	0.17	11.6
	32	2.8	0.34	0.16	12.8
16x12	28	2.14	0.28	0.11	9.1
	30	2.27	0.31	0.14	10.1
	32	2.42	0.34	0.16	11.1
16x10	28	1.79	0.27	0.06	7.1
	30	1.92	0.30	0.09	8.1
	32	2.03	0.33	0.11	8.9

Table 4.5 Different propulsion system configurations for the "Ferry" mission

In the selection of cell capacity, endurance was the main parameter. The most energy demanding flight mission of the present competition to be completed was the Fire Bomber mission simulated as 6-minute static run at full throttle. From the results of the analysis carried out, 2400 mAh batteries were selected to be the optimum choice for this particular configuration. Besides the limit on maximum battery weight, increasing capacity of cells yielded less flight time when compared to the theoretical value due to their effect on gross weight, whereas choosing a lower capacity was better when compared to the theory, yet unable to meet the 6-minute static run at full throttle condition.

4.5 Structural Design Analysis

In the structural part of the preliminary design phase of the aircraft, various methods were discussed to stiffen the structure of the aircraft, which was designed during the conceptual design phase.

Except the main bulkhead and tail box of the aircraft, remaining parts are open-section beams which can be examined analytically using flat plate buckling, bending, transverse and shear loading analysis. For this purpose, WeiSt program was used. It was developed by the past year's team in MS Excel and satisfied all our needs in the Preliminary Design phase. WeiSt helped us in calculating the stress values together with the weight values. Therefore accurate weight estimations were done during the design cycle.

To determine the optimum structural design for various parts of the aircraft, technique of FOM's is used

4.5.1 Figures of Merit

The figures of merit used during this evaluation are the same as the ones used and depicted explicitly in the conceptual design chapter (see Chapter 3).

4.5.2 Fuselage

For ease in manufacturing and maintenance, closed box structure was selected for the fuselage. Wood, metal, carbon-epoxy, E-glass-epoxy were the material choices but the carbon-epoxy was eliminated at the beginning due to the lack of manufacturing equipment.

A monocoque fuselage using E-glass-epoxy was appeared to be heavier than a metal or wood fuselage frame. Also it was meaningless to construct a monocoque fuselage if it does not carry significant amount of shear load.

There left two choices for the construction of fuselage frame: metal or frame. Although metal frame is more difficult to manufacture when compared to wood frame option, it would sustain much more rigidity than the wood frame.

Knowing these facts and applying the FOM evaluation the fuselage decided to be built up from metal. As the metal construction material; aluminum 2024-T3, which gives enough stiffness and less weight, was selected.

The aluminum frame was designed such that the only load carrying member would be the main bulkhead. According to the experiences that were gained last year, the main bulkhead was not designed as an open frame. The benefit of the closed frame bulkhead which is improved resistance to yield and buckling due to bending was used in this year's design.

4.5.3 Wing

In order to fit inside the given box dimension, the wing was designed as three segments. Maximum bending stresses due to the wing loading occurs at the mid section of the wing. Therefore the advantage of having no joints at the middle was used in three portioned wing. Making the use of the FOM evaluation, the optimized wing structure was obtained to be the foam rib structure with balsa skin.

Since the resultant lift force act on the quarter chord location, the spar is placed in the wing accordingly. The choices for the cross section used in the spar were: C beam, I beam, and the circular profiles. From the manufacturing point of view, circular profile was the best configuration, but when the weight was considered C beam was appeared to be the suitable choice. Having the experience from past year's aircraft, tempered aluminum 2024-T3 has a proven structural efficiency. A detailed stress and buckling analysis were performed and the thickness of the spar was estimated to have a maximum state of stress lower than the estimated buckling stress at the mid-section of the wing.

4.5.4 Landing Gear

The main landing gear was attached to the main bulkhead since the main bulkhead was the only load-carrying member in the aircraft's metal frame.

To prevent the additional drag the landing gear is positioned adjacent to the inside of the fuselage wall.

The choices for the material that would be used in the landing gear are steel and aluminum. The poor welding characteristics of the 20 and 70 series aluminum forced us to use steel. Variable cross section was used when designing the landing gear. Sufficient strength is obtained from a "T" profile, which was the lightest, however in critical regions, it is modified to an "I" and box profile for safety.

4.5.5 Empennage

Unlike the wing, no spar was used in the vertical and horizontal stabilizer to stiffen the structure because of the calculated low aerodynamic loads on these surfaces.

In the assembly of the horizontal and vertical stabilizers aluminum box was used. Via the box concept, vertical stabilizer, horizontal stabilizer and tail boom could be assembled easily. The cross sectional dimensions of the box are chosen such that the circular tail boom fits into it.

It was decided that a thin-walled circular 2024-T3 aluminum tube is the best material for this design which is light, stiff and having good aerodynamic characteristics. The thickness of the tail boom was selected such that it can provide sufficient stiffness to ensure that the tail incidence should not be changed due to tail loading and prevent excessive vibration.

CHAPTER 5 - DETAIL DESIGN

The main concern of the detail design chapter is of the documentation of the component and the systems architecture selection including the rated aircraft cost. Geometry, performance, weight and flight systems data of the sized aircraft were given in tabular form. In addition, drawings of the final configuration, size, structure and the system layout of the aircraft were included in this section.

5.1 STRUCTURAL DESIGN ANALYSIS

Detailed design of the structural system was investigated in the light of two main considerations; the construction, and the sizing and placement of the water tank into the aircraft. Since the amount of water carried had an exceptionally important role in the scoring of the aircraft this year, all design considerations were done to carry as much water as possible and to minimize center of gravity change after deployment of the water.

5.1.1 Fuselage

In the preliminary design phase, previously, the main bulkhead approach was preferred for the body frame. It was decided to use a closed section type structure, which is quite stronger than an open section, as a main bulkhead, since all of the elements of the aircraft such as wing, main landing gear and tail boom including the forces formed by these elements are to be carried by the main bulkhead. The cross section of the main bulkhead was chosen as a U profile, which is strong enough.

Since the main bulkhead was the main load carrying member of the fuselage, the wing was also attached to the main bulkhead by two bolts. In order to prevent the undesired movement of the water tank during the flight, the tank was supported by the main bulkhead from the sides and bottom. Moreover, its dimensions were designed such that it just fits into the main bulkhead.

The tail boom was connected to the fuselage by a box shaped beam extension which was supported by two stringers. This box was dimensioned and produced providing the circular tail boom to fit tightly into it. The tail boom was pinned inside it. This box shaped beam extension was to permit tail length to be adjusted for stability considerations

The engine was mounted by an aluminum support to motor mount bulkhead in front of the fuselage. The torsional effect of the engine was avoided by the supports placed under the motor mount. The battery pack was placed behind the motor mount bulkhead and under the water tank. The pack is supported along the edges and the bottom surface of the batteries is left open to enhance the circulation of cooling air in order to prevent the excess heating of the battery pack.

5.1.2 Wing

There are three possible production techniques for the manufacturing of the wing. The possible production techniques for wing are foam core with composite or balsa skins, the foam rib structure with balsa skin and the balsa rib structure. In the foam core applications, a block of foam is shaped to a desired wing shape and then covered with a skin of either balsa wood or fiber-glass/carbon composite.

Another alternative is described as foam rib covered with balsa wood, that is to say, the foam core is blanketed at certain sections of the wing in order to approach a rib structure which is advantageous in terms of its weight. The balsa rib structure is the classical rib structure application with the material being balsa wood. Note that, each wing structure alternative is considered to include at least a spar spanning the whole wing or a part of it. Also, film coverage will be applied in order to ensure a smooth surface finish.

As discussed in chapter four the wing was designed as three segments. An aluminum C shaped spar was placed at the quarter chord of the wing where the aerodynamic loads are known to be the maximum. To maintain the continuity of the spar all along the span, the cross-section of spars of the side-wings was dimensioned in such a way that 10 cm of these side spars fit tightly into the mid wing spar, and two bolts were used for connection of each spar. To reduce torsional effects of aerodynamic loads at the wing joints where balsa and foam continuity is disrupted, an 8mm circular wooden rod was placed near the trailing edge as a secondary extension.

The metal plate behind the motor mount was extended on top of the fuselage and bent over the leading edge. The mid-wing had two rod protrusions coming out of the leading edge which pass through the spar inside the wing. After inserting the rods into their corresponding holes, assembly of the wing to the fuselage was done by two bolts passing through the main rib.

Two wing servos were set in the foam ribs at the mid-locations of the flaperons. The pushrods of the flaperons servos are passing inside the wing to reduce the aerodynamic drag. The movement of the control surfaces was provided by using commercially available small plastic hinges. To improve the aerodynamic efficiency of the wing, transparency papers are used to seal the gaps between the control surfaces and the wing.

5.1.3 Landing Gears

The main landing gear was placed at the sides of the main bulkhead and fastened by means of two bolts at each as discussed earlier. Moreover, to prevent the torsional deflection, the parts closer to the connections at the main bulkhead was designed as a closed section, whereas at the tips its cross section turns to an I beam. A special consideration is given to the nose landing gear so that it can be adjusted in length using a set-screw. The rotation of the nose landing gear is provided by two distinct ball bearings.

5.1.4 Empennage

To assemble the horizontal and the vertical stabilizers perpendicular to each other, an aluminum box is used which has a length same as the root chord of the horizontal stabilizer. The cross section of the box is chosen such that the tail boom fits into the box. It is located between the two horizontal tail parts. The assembly sequence of the empennage is that first the tail boom fits into the tail box then the vertical stabilizer is attached through both the tail box and the tail boom.

5.1.5 Water Dumping Mechanism

In the water dumping system a servo controls the dumping of the water. A stopper with a gasket was mounted onto a push-rod passing through the hole on the main rib controlled by the servo. Initially, the servo presses onto the stopper placed on the dumping hole. When the servo is activated the push rod opens the hole and the water is dumped. To prevent the undesired movement of the water tank, it was supported by the main bulkhead. Furthermore, its dimensions were designed such that it just fits into the main bulkhead.

5.2 Propulsion

In the preliminary design phase, performance parameters of the aircraft were effective for the propulsive system design. Firstly, the team decided to use the engine from Astro Flight families among the two alternatives of engine selection that are Graupner and Astro Flight. Next, MotoCalc Electric Flight Performance Prediction Software was used to calculate the results of different configurations of the engine and the propeller. Astro Flight Cobalt 60 11T#23 (The optimum engine for the designed aircraft) was decided to be used. With the selected engine, a 20 X14 APC two bladed composite propeller was used. To produce the required level of thrust, 32 Ni-Cd cells were needed according to the results of the software stated above.

5.3 Rated Aircraft Cost Analysis

After the final configuration of the aircraft was determined, the rated aircraft cost was calculated. Table 5.1 shows the result given with the intermediate calculations. Contribution of each component to the total RAC is given in percentage as a bonus.

Airframe Weight [lb]	8,57160
Wing	
Wing Span [ft]	8,53020
Chord [ft]	1,04987
Number of Wings	1
Number of aileron	
Number of flaps	
Number of flaperons	2
Number of Spoiler	
Empennage	
Number of Vertical surface (with no active control)	
Number of Vertical surface (with an active control)	1
Number of Horizontal surface	1
V-Tail	
Propulsion	
Engine Weight [lb]	1,74165
Battery Weight (propulsion)[lb]	4,59000
Number of Engines	1

Body	
Length [ft]	5,05248
Width [ft]	0,60695
Height [ft]	0,60695
Control Systems	
Number of servo or motor controller	6
Battery Weight (control) [lb]	0,33069

Manufacturing Man Hours (MFHR)	191,78157
Manufacturing Cost Multiplier	$20
WBS 1.0 Wing(s)	104,55567
WBS 2.0 Fuselage	37,22591
WBS 3.0 Empennage	20,00000
WBS 4.0 Flight Systems	30,00000

Manufacturers Empty Weight (MEW)	15,23394
MEW Multiplier	$300
Rated Engine Power (REP)	4,59000
REP Multiplier	$1500
[redacted]	[redacted]

Table 5.1 Rated Aircraft Cost Table

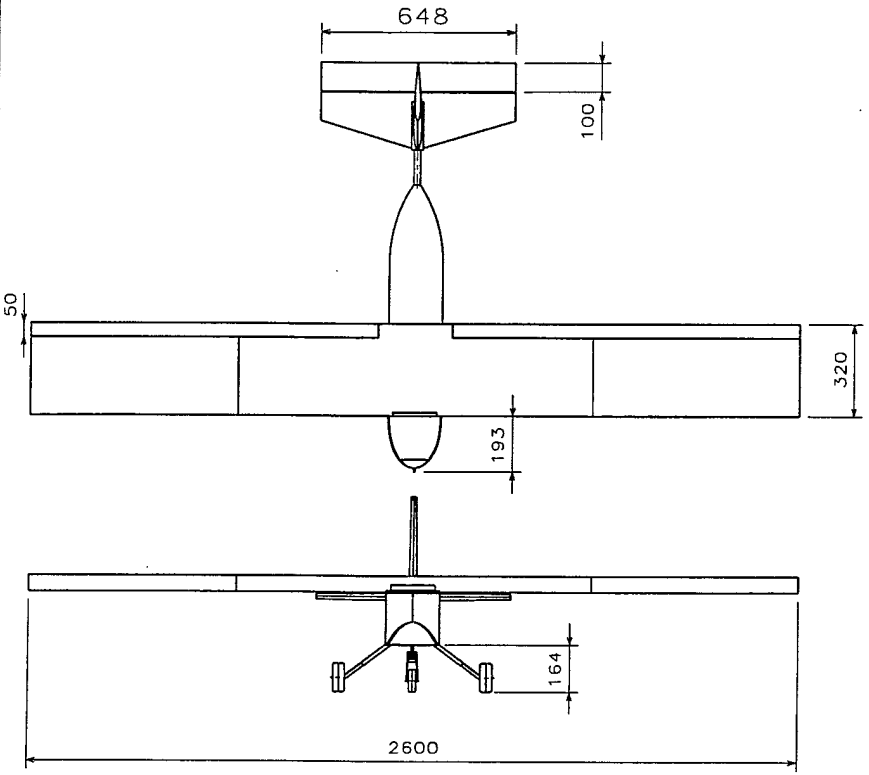
5.4 Sized Aircraft Data

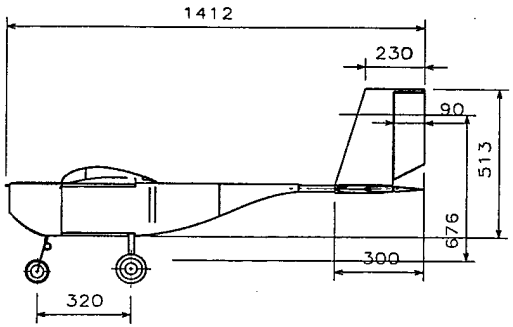
According to final configuration of the aircraft, geometry, performance, weight and system parameters are shown in Table 5.2 below.

GEOMETRY	Length	5.77 *ft* = 1.76 *m*
	Span	8.53 *ft* = 2.60 *m*
	Height	0.57 *ft* = 0.17 *m*
	Wing Area	8.93 ft^2 = 0.83 m^2
	Aspect Ratio	8.44
	Control Volumes	HT = 0.60 VT = 0.04
PERFORMANCE	$C_{L,max}$	1.8
	L/D_{max}	14.2
	Max. Rate of Climb	
	Stall Speed	31.17*ft/s* = 9.50 *m/s*
	Max. Speed	
	Take-off field length (empty weight)	
	Take-off field length (gross weight)	
WEIGHT	Airframe	8.82 *lb* = 4 *kg*
	Propulsion System	6.69 *lb* = 3.03 *kg*
	Control System	0.88 *lb* = 0.4 *kg*
	Payload System	0.12 *lb* = 0.054 *kg*
	Payload	8.82 *lb* = 4.0 *kg*
	Empty Weight	16.51 *lb* = 7.49 *kg*
	Gross Weight	22.48 *lb* = 10.2 *kg*

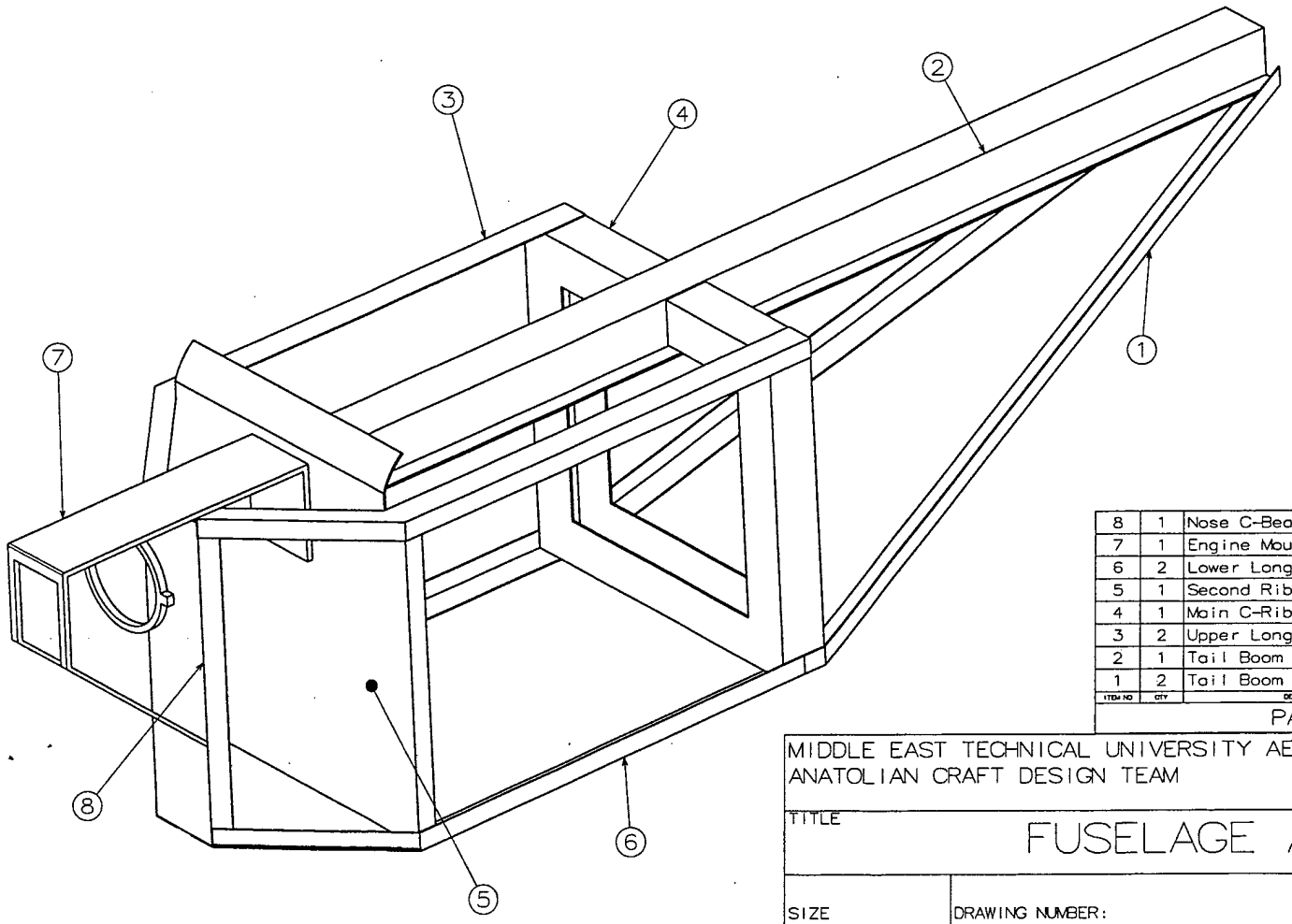
SYSTEMS	Radio	Futaba T9ZHP transmitter RI29DP receiver
	Servos	→Futaba S-3001 for Rudder, Flaperon and Water Deployment System →Futaba S-9202 for Nose Landing Gear →Futaba S-9204 for Elevator
	Battery Configuration	32xSR Batteries 2400 Max NiCd batteries (2400 mAh, 1.2V) in series
	Motor	Astro Flight Cobalt 60 11T#23
	Propeller	APC 20"x14"
	Gear Ratio	2.75:1

Table 5.2 Final Aircraft Data





MIDDLE EAST TECHNICAL UNIVERSITY AEROSPACE ENG. DEPT. ANATOLIAN CRAFT DESIGN TEAM		
TITLE: THREE VIEWS		
SIZE: A3	DRAWING NUMBER: METU_DBF04-001	DIMENSIONS ARE IN MM
SCALE	DATE	REV: OR
		SHEET



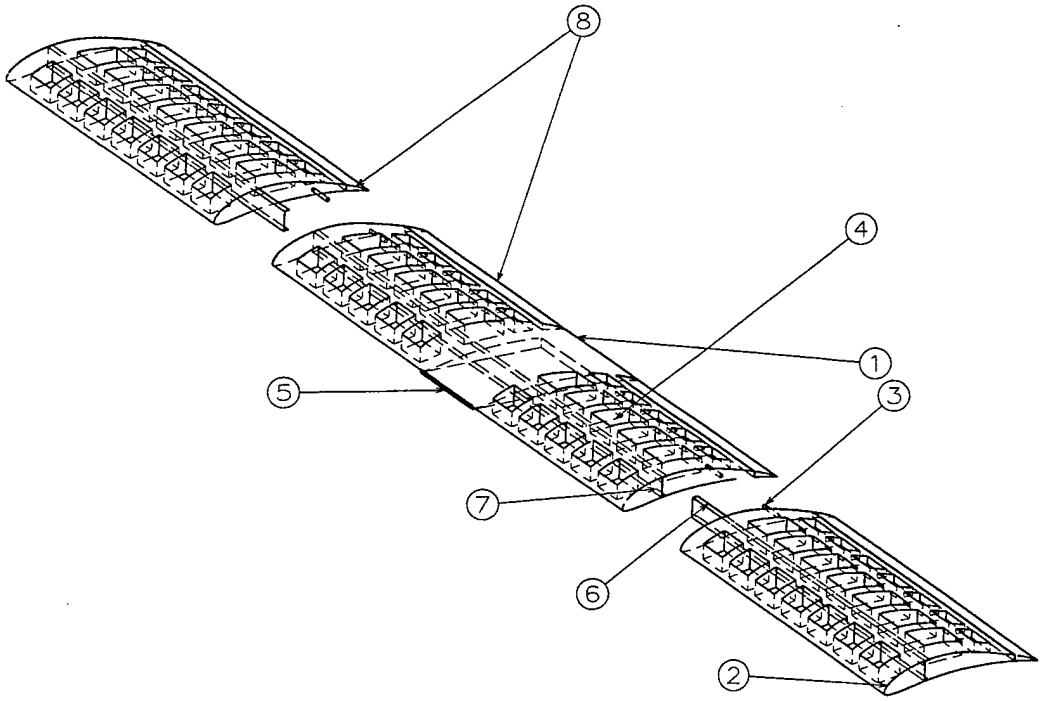
ITEM NO	QTY	DESCRIPTION	MATERIAL
8	1	Nose C-Beam	Aliminum 2024T3
7	1	Engine Mount	Aliminum 6061T6
6	2	Lower Longeron	Aliminum 2024T3
5	1	Second Rib	Aliminum 2024T3
4	1	Main C-Rib	Aliminum 2024T3
3	2	Upper Longeron	Aliminum 2024T3
2	1	Tail Boom Box	Aliminum 2024T3
1	2	Tail Boom Stiffieners	Aliminum 2024T3

PARTS LIST

MIDDLE EAST TECHNICAL UNIVERSITY AEROSPACE ENG. DEPT.
ANATOLIAN CRAFT DESIGN TEAM

TITLE: FUSELAGE ASSEMBLY

SIZE	DRAWING NUMBER:	DIMENSIONS ARE IN MM
A3	METU_DBF04-002	REV: OR
SCALE: 1:2	DATE: 02.03.2004	SHEET: 1 OF 1



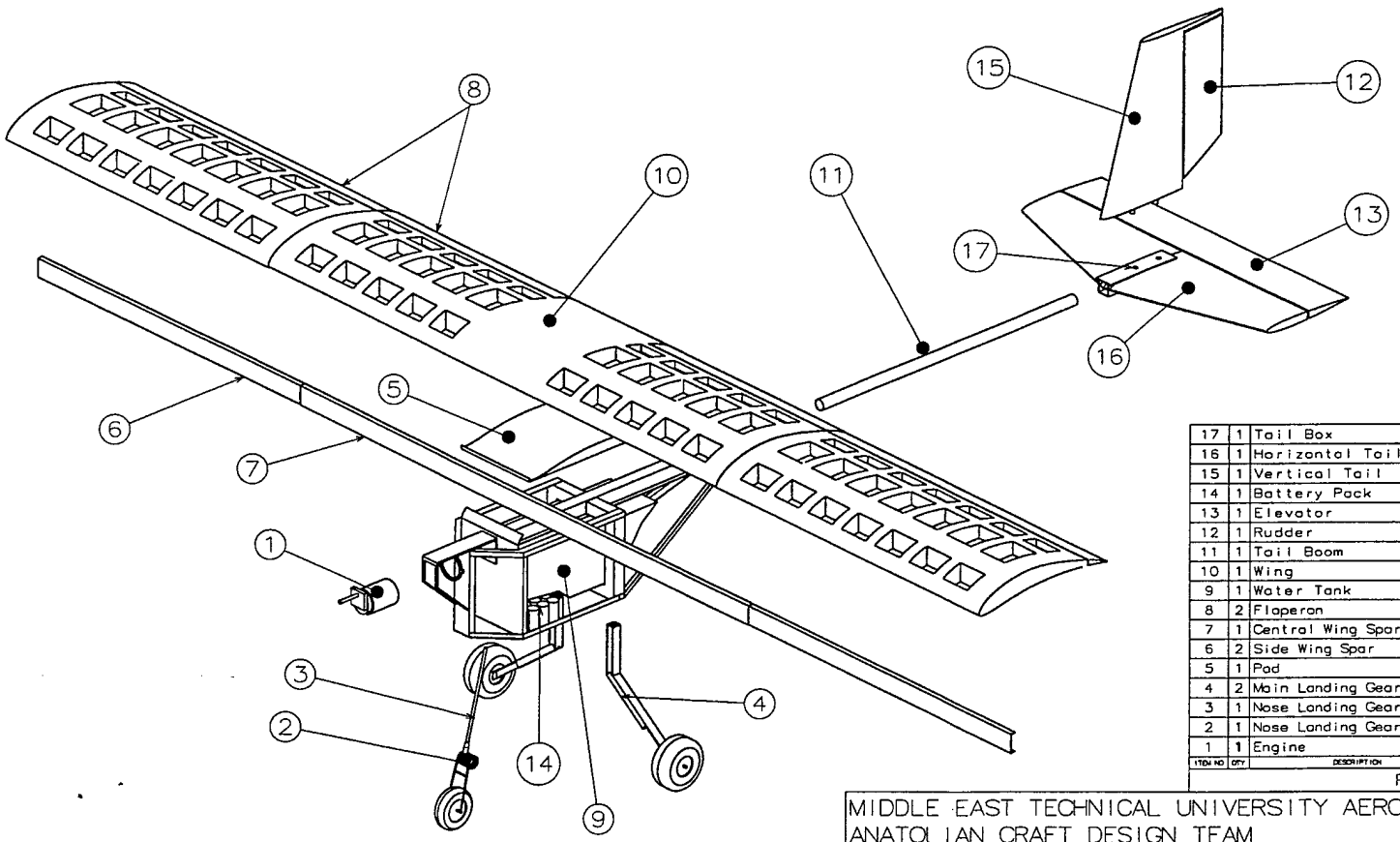
ITEM NO	QTY	DESCRIPTION	MATERIAL
8	2	Flaperon	Balsa Rib Structure
7	1	Central Wing Spar	Aliminum 2024T3
6	2	Side Wing Spar	Aliminum 2024T3
5	1	Pad	Foam
4	72	Lightning Holes	-
3	2	Torsion Pins	Wood
2	2	Side Wings	Foam-Balsa Rib Structure
1	1	Central Wing	Foam-Balsa Rib Structure

PARTS LIST

MIDDLE EAST TECHNICAL UNIVERSITY AEROSPACE ENG. DEPT.
ANATOLIAN CRAFT DESIGN TEAM

TITLE: WING ASSEMBLY

SIZE	DRAWING NUMBER:	DIMENSIONS ARE IN MM
A3	METU_DBF04-003	REV: OR
SCALE: 1:10	DATE: 02.03.2004	SHEET: 1 OF 1



ITEM NO	QTY	DESCRIPTION	MATERIAL
17	1	Tail Box	Aliminum 2024T3
16	1	Horizontal Tail	Foam-Balsa Structure
15	1	Vertical Tail	Foam-Balsa Structure
14	1	Battery Pack	SR
13	1	Elevator	Balsa Rib Structure
12	1	Rudder	Balsa Rib Structure
11	1	Tail Boom	Aliminum 2024T3
10	1	Wing	Foam-Balsa Rib Structure
9	1	Water Tank	E-Glass-Epoxy
8	2	Flaperon	Balsa Rib Structure
7	1	Central Wing Spar	Aliminum 2024T3
6	2	Side Wing Spar	Aliminum 2024T3
5	1	Pad	Foam
4	2	Main Landing Gear Strut	IS301 Stainless Steel
3	1	Nose Landing Gear Strut	Aliminum 7075T6
2	1	Nose Landing Gear Spring	Steel 1040
1	1	Engine	Astroflight

PARTS LIST

MIDDLE EAST TECHNICAL UNIVERSITY AEROSPACE ENG. DEPT.
ANATOLIAN CRAFT DESIGN TEAM

TITLE: MAIN ASSEMBLY

SIZE	DRAWING NUMBER:	DIMENSIONS ARE IN MM
A3	METU_DBF04-004	REV: OR
SCALE: 1:8	DATE: 02.03.2004	SHEET: 1 OF 1

CHAPTER 6 – MANUFACTURING PLAN AND PROCESSES

In order to choose the most efficient and suitable techniques for manufacturing the components of the aircraft, numerous manufacturing techniques were investigated after the completion of the design phase. Availability, required skill level, required time reliability and the costs of the manufacturing processes were the most important parameters of concern throughout the investigation. Afterward, a "Skill Matrix" was created to relieve the manufacturing plan, and manufacturing schedule was constructed.

The major limiting aspect was the availability of the manufacturing process. The team did not have the chance for obtaining some of the manufacturing techniques. Moreover, some of the processes required the team members to be well trained and skilled in order to apply those processes suitably. So as to decide which techniques suit to the team, a "skill matrix" was constructed as shown in the Table 6.1.

Wood working skills	Foam hot wiring	Balsa Sheeting	Fiberglass Sheeting	Servo Systems consulting	R/C Modeling	Metal Manufacturing Techniques	Heat Film Application	NUMBER OF TEAM MEMBERS
8	2	6	4	2	1	5	4	9

Table 6.1 Skill Matrix

A schedule was followed in order to build and prepare the aircraft for the flight tests prior to the competition. This also made it easier to work in a well-organized and efficient method. Owing to the limitations of the budget, the cheapest possible technique for the production of the components were selected. On the other hand, the accuracy and the quality of the production should be as good as the

expensive techniques. Another important criterion for an aeronautical product was the weight of the components. For that reason, lightest possible materials were selected in the production.

6.1 Figures of Merit (FOM)

With the purpose of assisting the team for choosing the most profitable techniques for the production of each component, the figures of merit (FOM) were built-up. The FOM contains availability of production, required skill level, required time, precision, strength, cost and weight of the process. The FOM in the final was achieved by using a weighted average with respect to the multipliers insinuating the precedence of a figure among others, scored as 1 up to 5. Scoring of each FOM was ranking from 1 up to 12 with higher score implying better characteristics of the related option.

Figures of Merit		Availability of Production	Required skill Level	Time Required	Precision	Strength	Cost	Weight	WEIGHTED AVERAGE
	Multiplier	5	3	2	5	4	2	5	
Fuselage	Bolting	12	12	11	11	7	11	10	10.50
	Welding	10	6	9	7	9	8	9	8.38
	Riveting	**10**	**10**	**12**	**12**	**12**	**12**	**12**	**11.38**
Wing Foam	**CNC Hot Wire**	**8**	**10**	**12**	**12**	**-**	**9**	**-**	**10.12**
	Hand Made Hot Wire	12	12	6	7	-	12	-	9.82
Landing Gear	**Welding Formed Profile**	**12**	**8**	**10**	**8**	**11**	**12**	**11**	**10.27**
	Pre-formed Profile	4	12	12	12	12	7	12	10.08
Fairings	Balsa Wood	11	8	9	12	10	10	11	10.46
	E-glass Composite	**10**	**10**	**12**	**10**	**12**	**12**	**12**	**11.00**
Control Surface Hinges	**Commercial Plastic Hinges**	**12**	**12**	**12**	**12**	**10**	**10**	**12**	**11.54**
	Co-axial Tubes	10	9	10	11	12	9	9	10.12
	Fabric Hinges	11	8	8	9	8	12	11	9.65
Wing Control Surfaces	**Foam with Balsa Skin**	**-**	**12**	**12**	**12**	**12**	**12**	**8**	**11.05**
	Balsa Rib	-	10	8	9	11	11	12	10.33
Tail Control Surfaces	**Foam Rib with Balsa Skin**	**-**	**12**	**12**	**12**	**12**	**12**	**9**	**11.29**
	Balsa Rib	-	10	8	9	10	11	12	10.14

Table 6.2 Figures of Merit

6.2 Manufacturing of Wings and Empennage

The most difficult components to manufacture were the wing and the empennage. Foam having 20 kg/m^3 density was used in the production of the wing. Using a CNC hot wire cutting machine the foam was cut to the shape of the chosen airfoil. In order to place the channel profile aluminum spar, the foam was cut along the span-wise direction at its quarter chord location. The foam was covered with a 1mm thick sheet of balsa wood to improve the strength of the wing and to preserve the outer

shape. Epoxy was used for gluing the balsa wood to the foam surface. To further reduce the weight of the wing and to place the servo motors of the flaperons, rectangular holes were cut out off the wing. Afterward, flaperons were cut from the wing, and hinged back. Next, wooden leading edge carrier rods were made and fixed to spar.

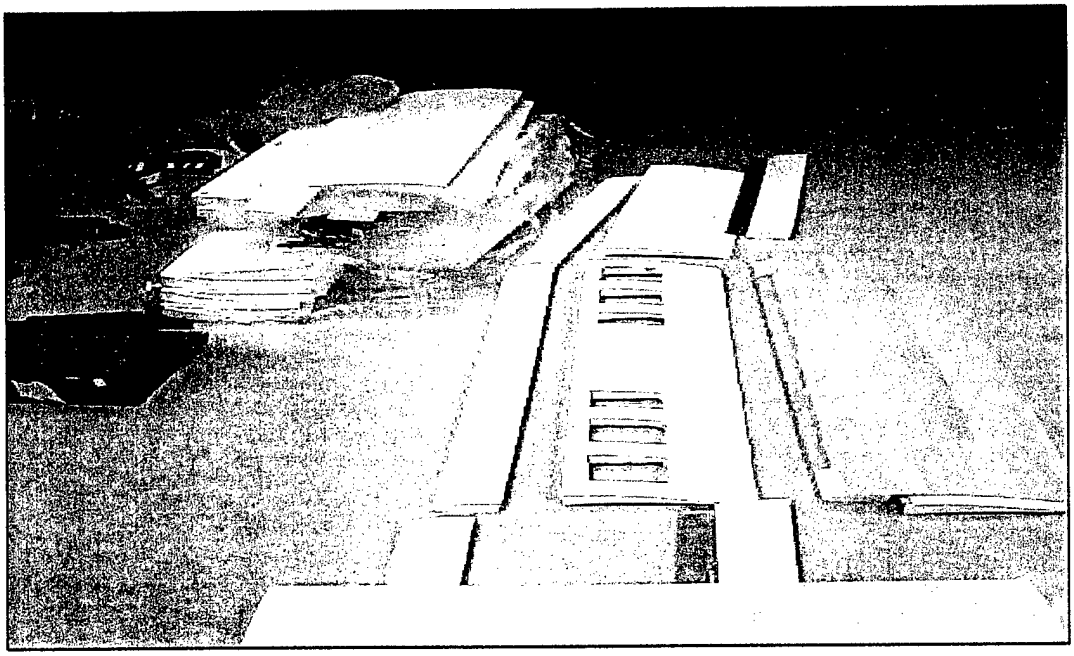


Figure 6.1 Wing Manufacturing Technique

Same method was followed in the production of the empennage. Using a CNC hot wire cutting machine the foam with 20 kg/m^3 density was cut, and covered with a 1mm thick sheet of balsa wood. By hand finishing, detailed final shaping of the tail was done. The two horizontal tail pieces and the vertical tail were connected to the tail box by means of the two wooden rods extending from inside of the tail pieces.

6.3 Manufacturing of Tail Boom

The aircraft is required to fit into a box with the interior dimensions 1x2x4 ft. When the tail boom is assembled to the main frame, the length of the fuselage becomes so high that the aircraft can not be fitted into the carrier box. Therefore, the tail boom was not fixed to the main frame instead it was manufactured in order to allow easy removal. The boom was fixed into the tail box with one of the rods of the vertical tail.

6.4 Manufacturing of Fuselage

Mainly, for joining the parts of the fuselage rivets were used instead of welding due to the lack of qualified welding technology of aluminum. The areas expected to have higher stress were strengthened via additional aluminum sheets. The box, which the tail boom was fixed to the main bulkhead, was constructed using rivets.

6.5 Manufacturing of Landing Gears

The springs for reduction of the impact force in landing were manufactured from steel wire with 3mm diameter, and it was fixed to the fuselage by means of a 7075 aluminum rod, a ball-bearing and a pipe clip. The main landing gear was produced from stainless steel with welding method using I and box profiles.

6.6 Manufacturing of Water Tank

The water tank was manufactured from e-glass and epoxy. Its shape was determined according to the placement of the batteries, shape of the fuselage and center of gravity location of the empty body. To prevent the shaking of the water in the tank, parallel ribs were put in the water tank.

CHAPTER7-TESTING PLAN

After design phase was completes, during the manufacturing phase, the static tests were begun to be performed, and after manufacturing was completed dynamic tests are going to be performed for the components as well as for the whole aircraft. Since we could not complete the manufacturing of all parts of aircraft before due date of report submission, some static tests and flight tests could not be performed. Reasoning, methodology and results of each one of these static experiments are listed below:

7.1 STATIC TESTS

7.1.1 Spar Test

The aim of this test was to observe the loading behavior of the spar of the wing. After the foam was fastened to the mid-wing spar with the length of 120 cm (3.94 ft) was simply supported at two ends. The load was applied at the center of the bar and it was gradually increased up to 10.5 kg (23.2 lb). It was expected that the mid-span deflected about 10mm, and the deflection measured as 8mm, also no plastic deformation occurred.

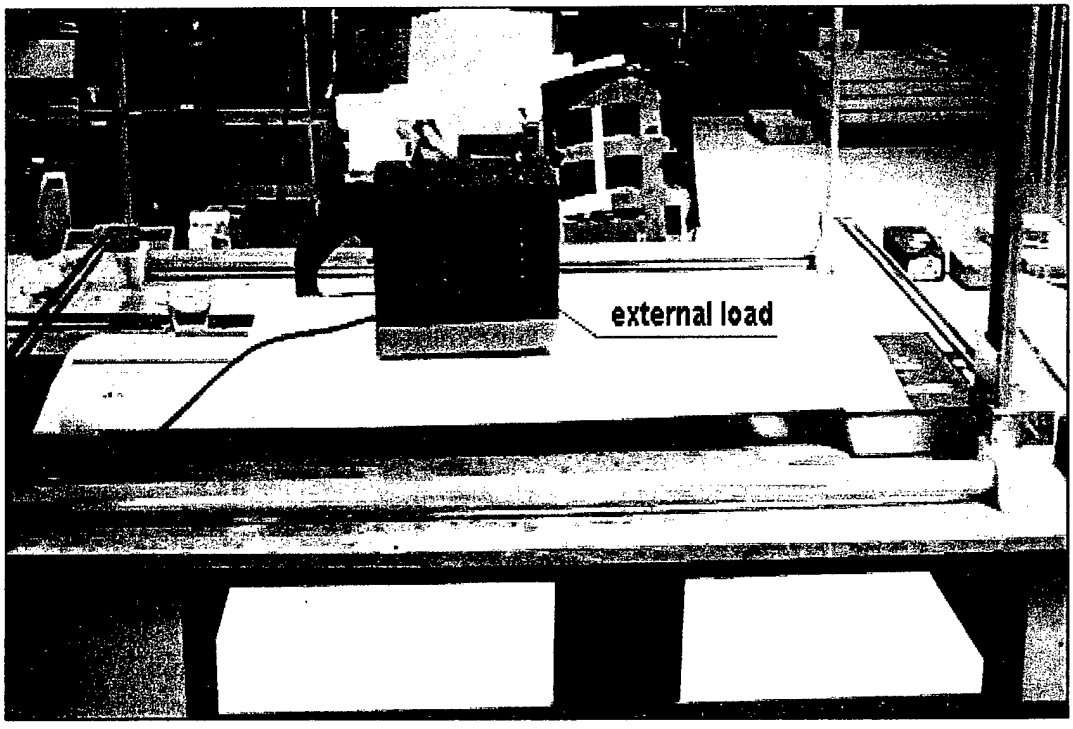


Figure 7.1 External load on the main wing

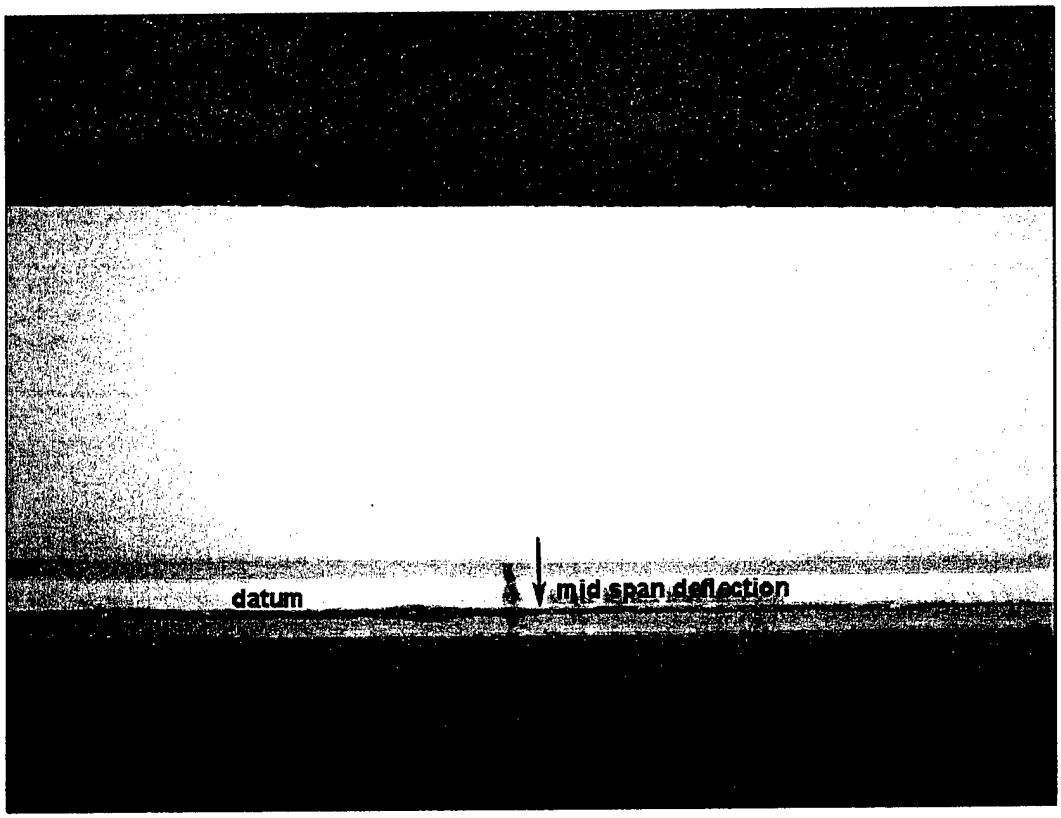


Figure 7.2 Mid-span deflection under external load

7.1.2 Landing Gear Test

The aim of this test was to observe the strength of the main landing gear. After the main landing gear was mounted on the body, 35 kg (77.16 lb), which corresponded to about 3.5-g landing load, was applied. As a result of the experiment, no plastic deformation occurred on the landing gear, that is the calculated stress values were not exceeded, and elastic deflections were observed as expected.

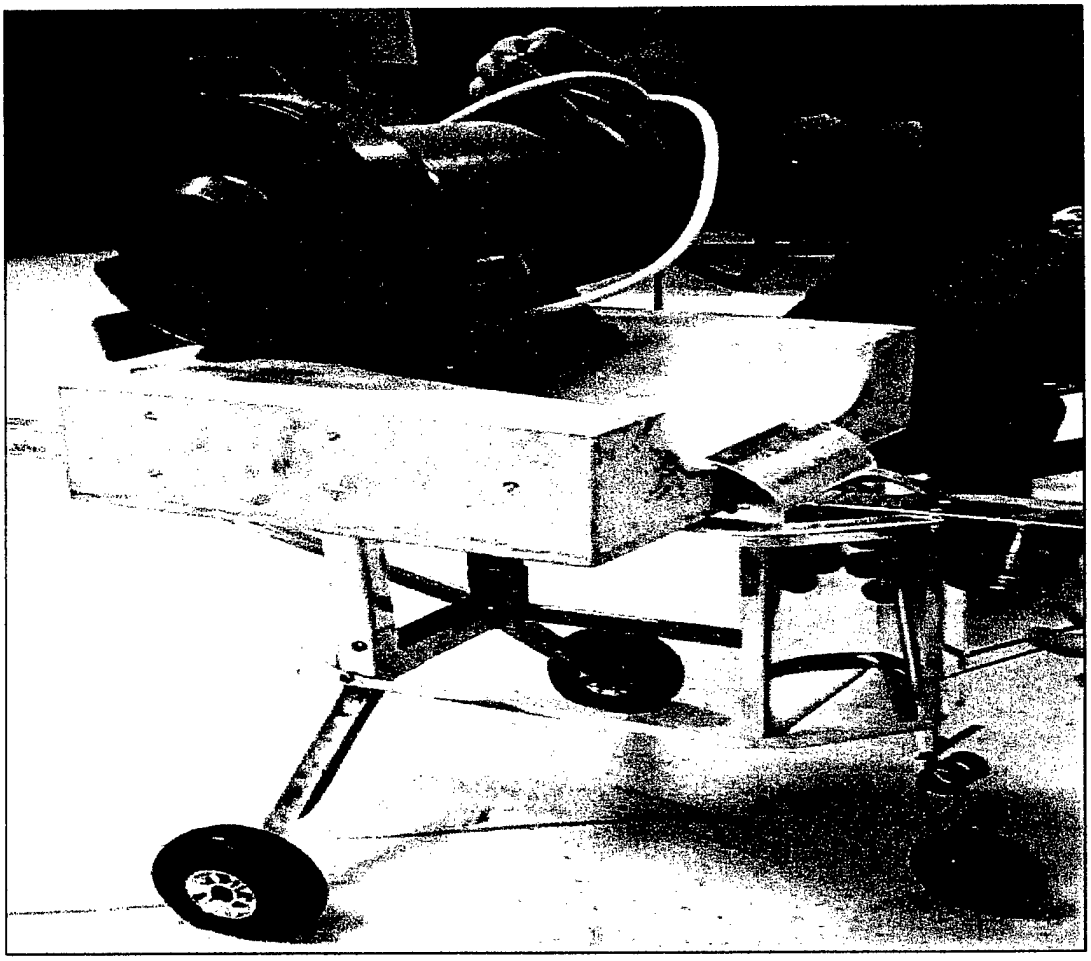


Figure 7.3 External loading on main bulkhead

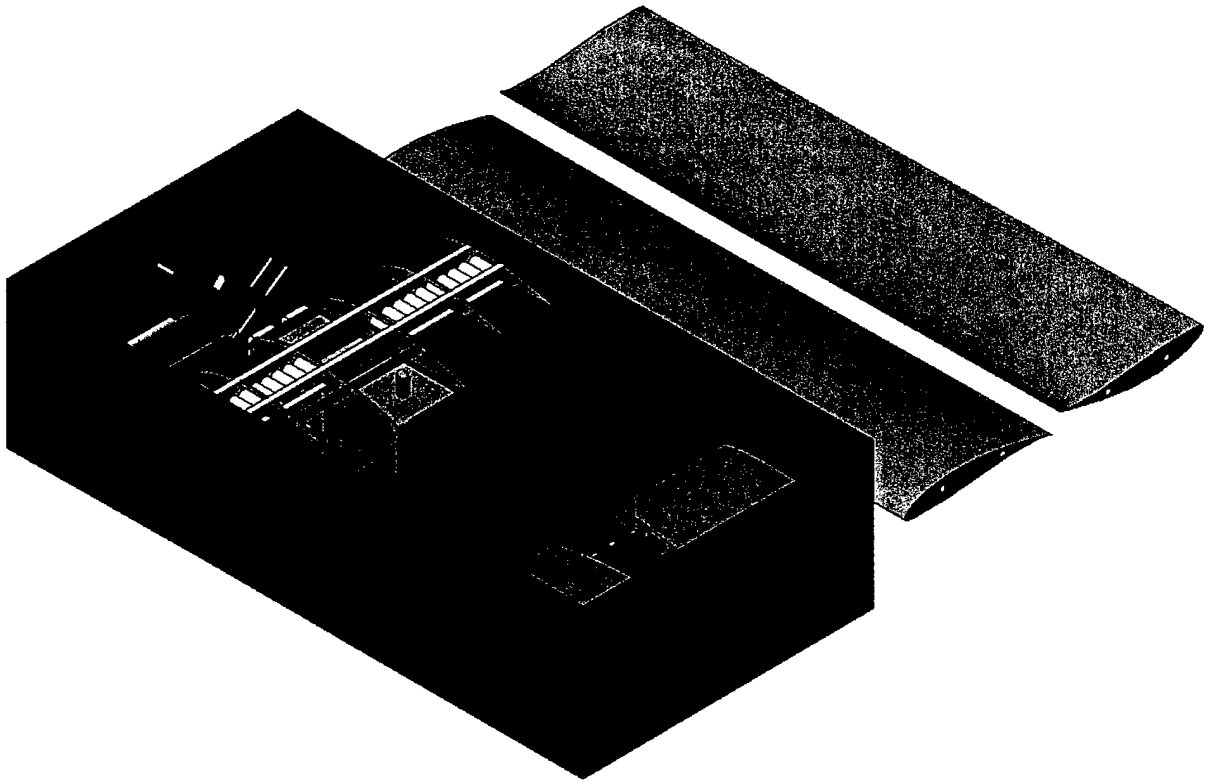
7.2 Flight Tests

Having completed the static tests, dynamic tests are planned to be proceeded in the design phase. Some ground tests for landing gear such as vibration behavior test on a runway and over turn angle are planned. Next, flight tests are planned to be performed. In the course of flight tests, it is planned that aircraft fly empty. During this test the efficiencies of control surfaces are going to be tested and if they do not work properly, they are going to be adjusted. Then, it will fly with 4 liters of water filled, and it will land with its tank full. In the end, both missions will be tested, and time will be recorded.

UCLA AIAA Aquanaut

Cessna / ONR Design Build Fly 2003-2004

Design Report



American Institute of
Aeronautics and Astronautics

Table of Contents

Executive Summary **4**
- **Conceptual Design** 4
  - *Conceptual Candidates* 4
  - *Payload Concept* 4
- **Preliminary Design** 4
- **Detailed Design** 5
- **Manufacturing Plan and Processes** 5
- **Testing Plan** 5

Management Breakdown **6**
- *Aerodynamics Team* 6
- *Propulsion Team* 7
- *Payload Team* 7
- *Structures Team* 7
- **Scheduling and Documentation** 7

Conceptual Design **9**
- **Problem Statement** 9
- **Conceptual Candidates** 9
  - *Conventional Aircraft* 9
  - *Tail-Boomed Aircraft* 9
  - *Swivel Wing Aircraft* 9
  - *Blended Wing Aircraft* 9
  - *Faceted Body Aircraft* 10
- **Conceptual Evaluation and Selection** 10
  - *Aircraft Weight* 10
  - *Takeoff Airspeed* 10
  - *Estimated Cruise Speed / Mission Time* 10
  - *Drag at Cruise Speed* 10
  - *Stability* 11
  - *Rated Aircraft Cost* 11
- **Tail Conceptual Evaluation** 12
- **Payload Conceptual Design** 13
  - *Payload Design Considerations* 14
  - *Deployment Schemes* 14
- **Landing Gear Configuration** 16
- **Initial Weight Estimate** 16

Preliminary Design **17**
- **Initial Wing Sizing** 17
- **Airfoil Selection** 18
- **Empennage Sizing** 21
- **Fuselage design** 22
- **Performance** 23
  - *Lift Prediction* 23
  - *Drag Prediction* 24
  - *Takeoff Performance* 25
  - *Max Speed* 25
  - *Rate of Climb* 26
- **Predicted Score** 27

Detailed Design **28**
- **Propulsion and Power Supply** 28
- **Static Stability** 31
  - *Center of Gravity* 31
  - *Longitudinal Static Margin* 31
  - *Yawing Static Stability* 31

Drawing Package 35
Manufacturing Plan and Processes **40**
Structure and Manufacturing 40
Wing Construction 40
Fuselage Structure 41
Tail Construction 41
Landing Gear Construction 41
Personnel Breakdown 41
Manufacturing Schedule 42
Testing Plan and Initial Results **43**
Static Tests 43
Dynamic Tests 43
Initial Testing Results 44
Payload Deployment Testing 44
Motor Test Stand Data 47
References **49**

Table of Figures

Figure 1: Management Overview. 6
Figure 2. Project Schedule. 8
Figure 3 Conceptual Sketch of Aircraft 11
Figure 4 Tail Configurations 12
Figure 5 Water Tank Dimensions 15
Figure 6 Takeoff vs. Wing Area 17
Figure 7 Airfoil Sections of Finalists 19
Figure 8 Drag Polar of Selig-Donovan SD7062 20
Figure 9 Drag Polar of Wortmann FX63-137 21
Figure 10 Horizontal Tail Sizing 22
Figure 11 3D Wing Lift Coefficient 23
Figure 12 Aircraft Drag 24
Figure 13 Maximum Speed 26
Figure 14 Rate of Climb 27
Figure 15 Payload Test Tank 45
Figure 16 Spar Specimen 45
Figure 17 Diagram of Spar Test 46
Figure 18 Propulsion Test Setup 48

Table 1 Conceptual Seclection Figures of Merit 11
Table 2 Tail Selection Matrix 13
Table 3 Initial Weight Estimate 16
Table 4 Empennage Specifications 22
Table 5 Center of Gravity Calculation 31
Table 6 Static Margins at Loading Conditions 31
Table 7 UCLA AIAA Aquanaut Data Specifications 33
Table 8 Rated Aircraft Cost 34
Table 9 Skill Matrix 41
Table 10 Manufacturing Schedule 42
Table 11 Initial Motor Test Data 47

Executive Summary

This report outlines the design and manufacturing of the University of California, Los Angeles AIAA chapter's submission to the 2004 Cessna/ONR Student Design/Build/Fly Competition. This year's mission requires an unmanned portable air vehicle be developed which is capable of performing two distinct missions. The aircraft has the option of performing a water-dumping in the air (Fire Fight) or simply flying laps (Ferry). The entry's total score is a combination of scores based on three criteria: report score, flight score, and rated aircraft cost.

Conceptual Design

This year's contest differs greatly from previous years. Instead of having to attach a payload that would drop in mid-air, the major task this year was to build a payload in the plane that could hold four liters of water and dump it over an area while in-flight. However, this year's relief comes knowing that there is no longer the obstacle of the assembly time.

Conceptual Candidates

The general configuration of the plane was the first to be decided on. Several choices were scrutinized. Each offered strengths and weaknesses for this competition. A conventional plane would be difficult to fit into the box. A flying wing would also have this weakness. Some designs offered a unique advantage. A swivel wing design would allow the plane to be quickly assembled from a simple folded form. Comparing estimated Rated Aircraft Cost, ease of construction, and aircraft performance, it was ultimately concluded that the conventional design offered a slightly more feasible approach to meet the competition specifications.

Payload Concept

The shear size of the required payload demanded that it be taken into account from the earliest stages of development. Deployment methods included a complicated valve which would open and close when signaled by the remote control. From a rated cost standpoint, it was determined that the deployment system should use at most one servo. A small team was tasked with construction and testing of the most promising design. Our goal became to find the simplest system possible. From several brainstorming sessions, different designs emerged. A fixed pipe system was deemed the most efficient as it used hydrostatic pressure to our advantage.

Preliminary Design

The team separated into different groups to perform their own specified tasks. The preliminary designs include initializing the wing size and choosing the airfoil to be used. The empennage sizing was done as well to determine what was best for the stability of the plane. Propulsion and power supply designs were created to determine the thrusts needed to propel our plane with its estimated weight. A

performance section was included in our preliminary design to calculate important figures such as drag, lift, speed, and others. Lastly, we focused on the static stability of the plane.

Detailed Design

Final flight and performance data was calculated and final sizing was done to the wing. Combining data with aspects such as propulsion, and structures, the aircraft specifications were set and finalized. Detail Engine requirements were calculated using a common R/C motor software package. Using this data, the proper engine, propeller, and gearbox were chosen from the available models. The fuselage would house the internal payload and thus was primarily designed around its dimensions. Surface area was minimized to reduce drag. After successful testing of prototypes for the payload systems, work began on implementing them into the aircraft and integrating it with the fuselage. With all of this work done, construction of the aircraft could begin.

Manufacturing Plan and Processes

Manufacturing of the vehicle was scheduled to begin at the start of the year. Completion was planned for mid-March. This would give the flight team enough time to test and modify the competition plane. Most of the fuselage would be built using a combination of balsa, spruce, and plywood. The use of wood allowed for high flexibility in component placing and ease of manufacturing and repair. Wings were constructed of foam cores and carbon fiber sheeting; this allowed for precise wing shaping. Carbon fiber spars would provide the necessary rigidity. Wire EMD cutting of foam allows flexibility in design of airfoils and is also relatively fast compared with alternatives.

Testing Plan

A series of static and dynamic tests are planned for the aircraft. The first of these have already been successfully run on the payload system. Wing loading and harsh landing testing is planned for immediate analysis. Dynamic tests will be performed in the future to determine flight characteristics and to uncover areas of necessary improvement or reinforcement.

Management Summary

Management Breakdown

Initial concept selection and preliminary trade studies were undertaken as a group. This led to a greater understanding of the challenges and design criteria amongst all members. The payload development, in particular, profited from the massive feedback. However, after the initial conceptual phase, the team was broken up into focused groups. The chief engineer/project manager oversaw the entire project and ensured that the design and building processes proceeding according to plan. The rest of the team was broken down into in aerodynamics, propulsion, payload, and structures sub-groups were led by

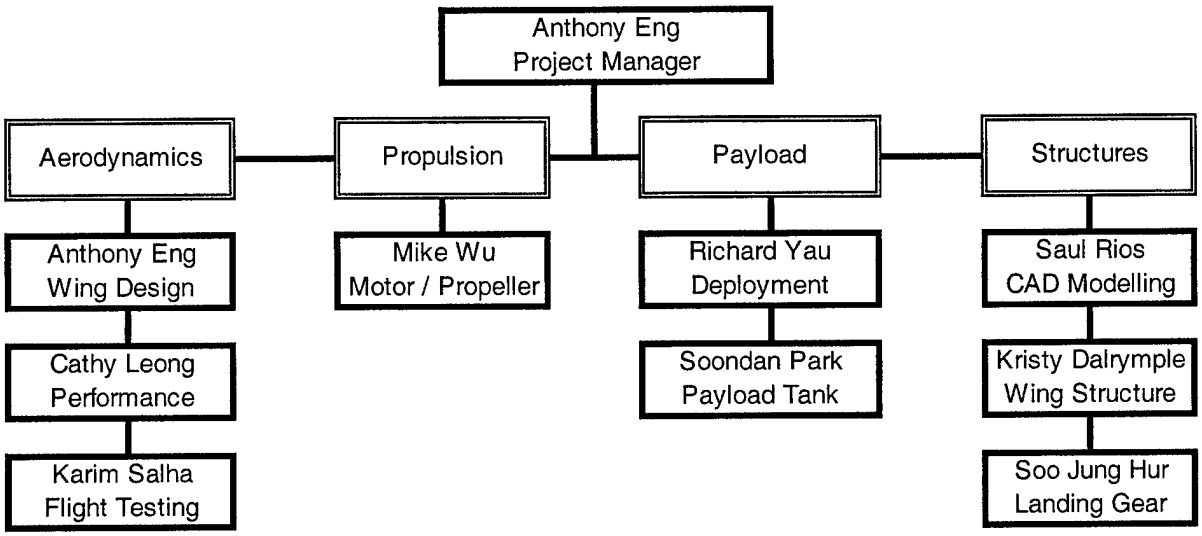


Figure 1: Management Overview.

Other team members included Adam Vogel, Andre Gondouin, Diana Chou, Douglas Cruz, Ronny Breslavsky, Tetsuya Hoshino, and Zach Siebers who helped out wherever needed.

Aerodynamics Team

The aerodynamics team developed mathematical models to evaluate the conceptual aircraft designs. Contest requirements and scoring were taken into account. A variety of planforms and taper ratios were considered. After deciding on the general layout of the aircraft, development moved on to the airfoils. Airfoil selection took into account aerodynamic and structural considerations. It was highly desirable to place the batteries in the wing, which placed constraints on airfoil selection. With the completion of the UAV, the aerodynamics teams became the primary flight-testing group. Their input will be used to improve the actual performance of the aircraft.

Propulsion Team

The propulsion team initially worked in conjunction with the aerodynamics team to determine the number and placement of engines. They then focused on optimizing the performance of the engine in conjunction with the batteries, propeller, gearbox, and speed control. After selecting a suitable combination of components, the team began testing systems to ensure physical requirements for current draw and thrust were met. The team also worked on the various servo and wiring requirements of the aircraft.

Payload Team

The payload team used the conceptual designs to build and test payload devices. Prototypes were constructed to test and optimize the payload deployment and attachment schemes. The internal payload tank was immediately built along with the corresponding deployment system under it. Since initial testing of the internal payload was successful, the team moved on to integrating the system into the rest of the plane.

Structures Team

The structures team was responsible for the fuselage and landing gear. The fuselage, above all, had to house the internal payload. They worked to minimize the size and weight of the sections while still maintaining their integrity. Materials selection and manufacturing fell within the realm of the structures team. The group also completed the final drawings of the aircraft. The team ultimately developed a central fuselage that matched the compactness requirements determined during the conceptual phase of the design process.

Scheduling and Documentation

It was decided early on that the UAV should be completed relatively early to allow for extensive flight-testing. As a result, design and construction had to take place at a quick pace. Milestones were set for every aspect of the project. Conceptual and preliminary design phases were scheduled to be finish by late December. Detail design and construction would be done by February. The design report would be completed along side the actual development of the design. Sections of the report were to be written shortly after the completion of the corresponding component. Figure 2 is a diagram of the predicted schedule.

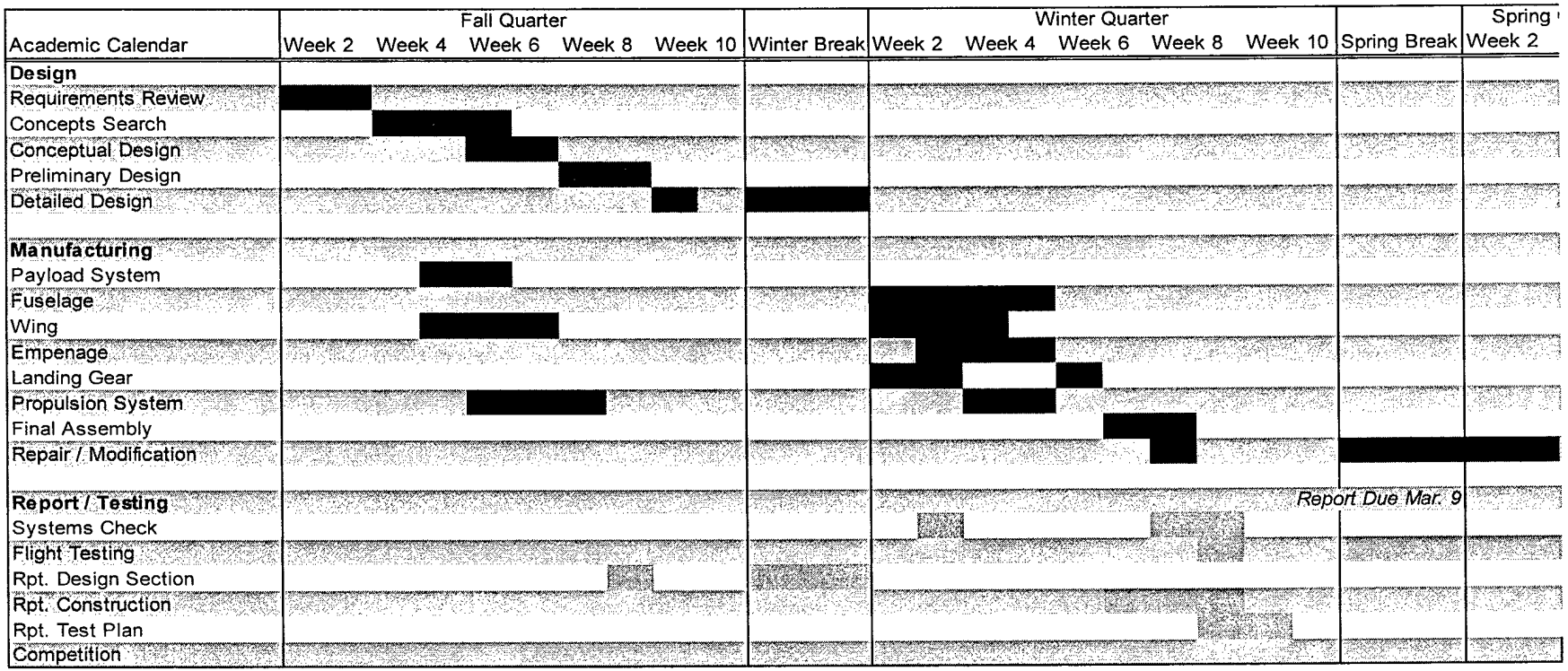


Figure 2. Project Schedule.

Conceptual Design

Problem Statement

The goal of the 2003 Cessna/ONR Design Build Fly competition is to build a UAV that quickly assembles out of a 4' x 2' x 1' inner dimensional box. The aircraft must carry 4 liters of water and deploy in flight'. The aircraft will fly one of two rated mission over a 1000-foot long course with a 360-degree turn in the opposing direction. The aircraft must take off in under 150 feet. The Fire Fight mission requires the aircraft to make two runs carrying 4 liters of water and deploying it in mid-air over an specified portion of the course. The Ferry mission requires the plane to fly 4 laps over the course, unloaded. The missions are scored on a combination of flight time, amount of payload deployed, and mission difficulty. The total score is the product of the report score and the highest flight score divided by the Rated Aircraft Cost.

Conceptual Candidates

Conventional Aircraft

A conventional configuration employs a long central fuselage that connects to both the wings and the tail. Aircraft of this type have been proven in the field and is by far the most popular configuration for modern airplanes. The major disadvantage would be its portability. The large fuselage could be difficult the fit in the required box. It was considered the baseline of what could be built.

Tail-Boomed Aircraft

This configuration is similar to the conventional aircraft except the tail is connected using one or two booms. The advantage of this configuration is that the tail can be designed to be removable and the structure of a full fuselage is not needed. This allows the design to save a significant amount of weight.

Swivel Wing Aircraft

The swivel wing concept was an attempt to cut down manual labor in the assembly process. It incorporated one or two wings that would be permanently attacked to the body and only needed to pivot into place. Its main disadvantage would be that the wingspan was limited. In order to fit in the box, the span had to be less than 8' with a cord less than 1'. Dual wings were also considered in the configuration, which would double the wing area.

Blended Wing Aircraft

A blended or flying wing design, which eliminates the need for a fuselage, would have several advantages. Its high lift and low drag were obviously in its favor. Another plus was its lowered Rated Aircraft Cost due to the limited number of control surfaces. On the other hand, there would be difficulties in carrying the payload internally as a blended wing design lacks sheer volume. It would also be more difficult to pack into the required box and more difficult to balance for stability.

Faceted Body Aircraft

The final concept considered was a faceted body, similar to the popular F-117 Stealth Fighter. The faceted shape, as opposed to the smooth surface of the blended wing, allows for the increased volume for the payload. It would require very little assembly and its flight would be very stable and hard to stall. Its disadvantages included a high induced drag and a high flare angle. Ultimately, a faceted body design would also have a low cruise speed and presented too many design challenges.

Conceptual Evaluation and Selection

Conceptual designs were evaluated using 6 criteria. They were selected to provide a quantitative comparison of the possible aircraft choices. The criteria reflect what was felt to be the most important aspects of an aircraft corresponding to the competition rules.

Aircraft Weight

A common model was used to estimate weight build-up of the wing, fuselage, avionics, and payload system. The model focused on the structural differences between the designs. Materials were approximated by using construction methods based on previous UCLA entries in the competition.

Takeoff Airspeed

An approximation was used to determine the airspeed required for the sized wing to provide the lift for takeoff. This is especially important because of the 150' takeoff requirement.

Estimated Cruise Speed / Mission Time

A first order approximation of cruise speed was used to estimate the total mission time. Mission time is the primary component of the total score. A high cruise speed / low mission time was a definite necessity.

Drag at Cruise Speed

Drag was approximated because of its effect on cruise speed and power requirements. It varied significantly because of the different body types of the concepts.

Stability

The stability was determined empirically from historical examples. Pitch, yaw, and roll stability were all taken into account. The conventional configuration was considered to be the best for this figure of merit.

Rated Aircraft Cost

The Rated Aircraft Cost is a figure specified by contest rules. It is a cost associated with the empty weight, rated engine power, and rated manufacturing hours. Manufacturing hours is determined by a set basis for rating surfaces and components.

Configuration	Baseline	Conventional Boom Tail	Swivel Wing	Blended Wing	Faceted Body
W/W'	1.0000	0.9538	0.9505	1.0842	1.0396
VLo / VLo'	1.0000	1.0152	1.1364	0.9242	0.9848
VC' / VC	1.0000	0.9780	0.8990	1.0988	1.0230
D70 / D70'	1.0000	0.9853	1.1176	1.1588	1.0000
Stability	1.0000	1.1397	1.1257	1.1438	1.1438
RAC	1.0000	0.9410	0.9619	1.0038	0.9990
Final Score	**1.0000**	**1.0021**	**1.0319**	**1.0689**	**1.0317**

Table 1 Conceptual Seclection Figures of Merit

Scores were normalized using the baseline configuration of the conventional aircraft. Lower final scores are better.

Based on the figures determined, the baseline configuration has the best final score. It is also determined that a high wing is preferable for better stability.

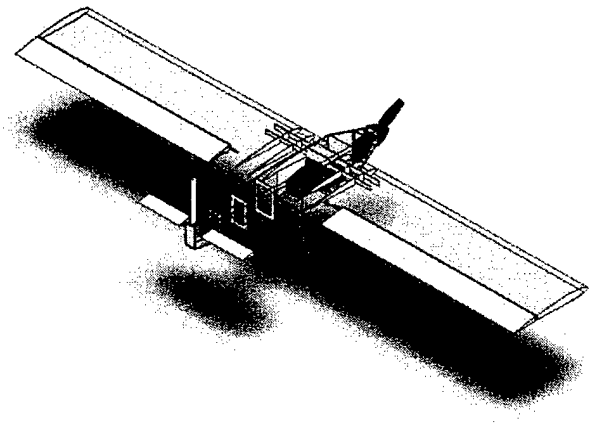


Figure 3 Conceptual Sketch of Aircraft

Tail Conceptual Evaluation

The tail's main function is to stabilize and control the aircraft. It affects trim, the balancing out of the moment created by the lift of the horizontal stabilizer to other existing moments about the length of the plane. This helps to keep stability by eliminating forces acting on the plane. The tail also helps to stabilize by regulating the flow around a plane, analogous to the fins on an arrow. Similar to a rudder on a boat, the tail also controls the direction of the plane. The horizontal stabilizers are used to change the plane's angle of attack, while the vertical stabilizer affects the way plane yaws. They essentially control the direction of the plane while the main wing provides for most of the lift force. There are numerous types of tails with its own advantages and disadvantages. The following are some of the most common variety.

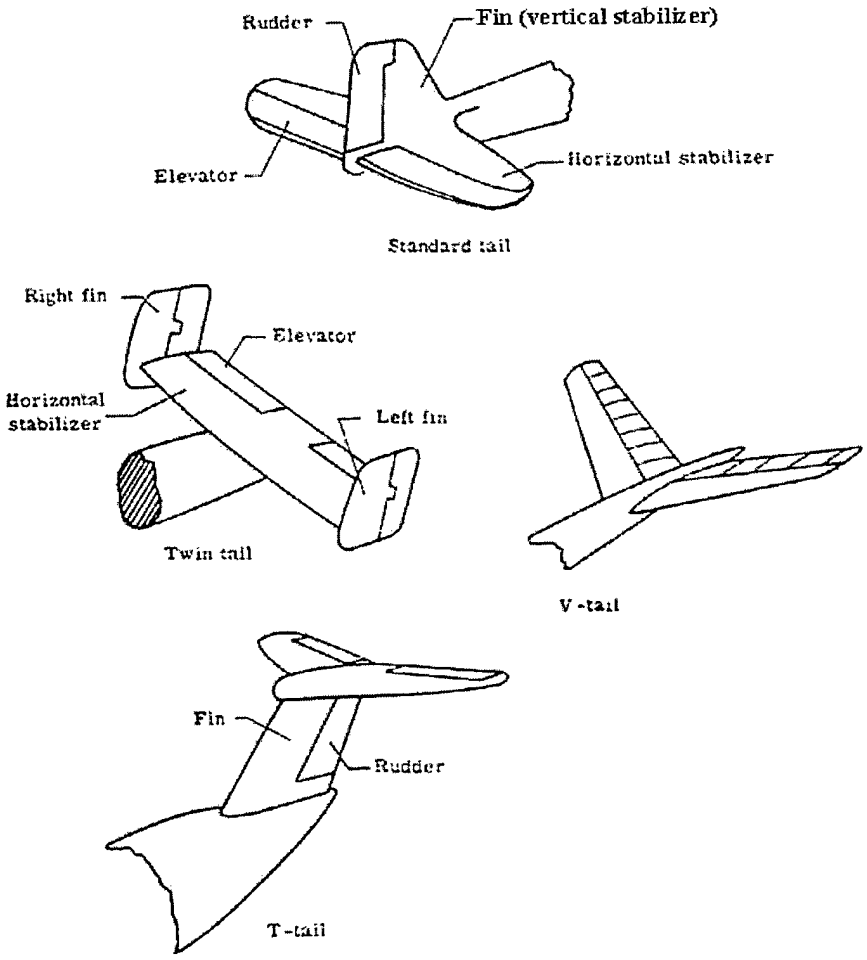


Figure 4 Tail Configurations

The conventional tail configuration for our UAV is found on over 70% of all planes in service. We chose this design due to its advantages in providing adequate stability and control at the lightest weight. It is the simplest to design using the electric motors that we have to radio control the plane.

We did not select the T-tail design for our UAV because it would have made our vehicle considerable heavier and complex. The extra materials needed to strengthen the vertical stabilizer to mount the horizontal tail did not compensate for the advantages of a T-tail. The horizontal stabilizer stays out of the wake due to its elevated position, making it possible to reduce its size.

The twin tail, also known as the H-tail, was another possibility for our design. However, the weight penalty and the complex nature in constructing it caused us to reconsider using that design. H-tails allows the vertical tails to be undisturbed by the wake of the plane. This helps to maintain better control of the vehicle without making the tail too big.

The V-tail was the design that we decided on last year's UAV. Instead of having a horizontal and a vertical component for control, we used the vertical and horizontal projection on the v-tail. This unique design allowed us to reduce the wetting area and the interference drag. However, the complication with setting the controls from previous year, especially the "adverse roll-yaw coupling" caused us to go with the more conventional method.

Three types were considered in detail. They are conventional tail without controllable rudder, conventional tail with controllable rudder, and a V-tail. The following matrix was developed to quantify the advantages and disadvantages of each choice.

Type	Rated Aircraft Cost	Advantage	Disadvantage	Score
Conventional with Rudder	20	High Control	High RAC	2
Conventional without Rudder	15	Simplicity	Less Control	1
V-tail	15	Lowest RAC with Yaw Control	Yaw-Roll Coupling	1

Table 2 Tail Selection Matrix

Rated Aircraft Cost is based on the number of surfaces. A v-tail is considered to be a vertical surface without control (5 hrs) and a horizontal surface with control (10 hrs).

Payload Conceptual Design

A major hurdle to this year's competition is the incorporation of a four-liter water tank as a payload. The deployment complexity of the required payload demanded that it be taken into account from the earliest stages of development.

Payload Design Considerations

1.) Payload should be secured and does not move/wiggle during takeoff, flight, and landing. While the payload is within the confines of the plane, it should not move, be loosely attached. Any such movement even if minor can affect flight. The 4 liters of volume should be placed in such a way to not change the center of gravity of the plane during takeoff, flight and landings.

2.) Water deployment should be as swift as possible. There should be minimal complications regarding the dumping of the liquid. A tube needs to be added to increase the flow of water out of the tank.

3.) Aerodynamics should take a minimal effect to the payload tank and its deployment piping. Aerodynamic characteristics need to be consistent before and after payload deployment. The tube needs to be attached and ran along one of the landing gear so that it does not stand out by itself to take on more drag.

4.) Water valve system needs to have remote-controlled open and close functionality. The valve leading to the deployment piping needs to be able to open, for the water to deploy, and to close without opening up the plane, to refill the water tank.

5.) Take off and landing is not hindered with payload piping attached. Attention to center of gravity is critical to a plane that flies and flies well. Since the payload makes up 9.5 pounds of the plane's total weight, it will definitely have an effect on plane acceleration, speed, maneuverability and stability. The released payload should also not hinder the landing gears or clearance issues when it is to take off again.

Deployment Schemes

In all the issues needed to be address later in the design stages, we started looking at how the water could exit the tank. First we considered what remote-controlled device could be used. We needed a valve that not only open for the water the exit, but also to close so a refill of the tank could be done without having to reach into the plane to manually close the valve. Many valves were considered. Most were either too heavy, or expensive. Then there was the complexity of the servo factor. We needed a valve that could somehow attach and function with the remote-controlled servo. In the end, we found a simple plastic valve that opens and closes when the valve handle turns 90 degrees. A servo arm glued to the valve handle so that the servo can control it.

To save on flight time and speed up the water deployment process, we turned to the Bernoulli's Equation.

$$p + \frac{1}{2}\rho V^2 + \rho g h = constant$$

where p is the pressure, ρ is the density, V is the velocity, h is elevation, and g is the gravitational acceleration

This equation helped us realize that by adding tubing to the bottom of the tank where the water exits and increasing the depth of the water height when it exits, the velocity of the water flow out of the system will be greater. A major concern we had with the extended tubing, of course, was the incredible amount of drag it would produce if it were to just hang down under the payload. As a solution, running the tubing to the side and down along the landing gear was our best way to reduce the air resistance. This increases the tube length but as long as the same depth of water height is kept, the exit velocity of water would remain the same. In order to do this, we even had to design our landing gear to meet the length requirements of the tube.

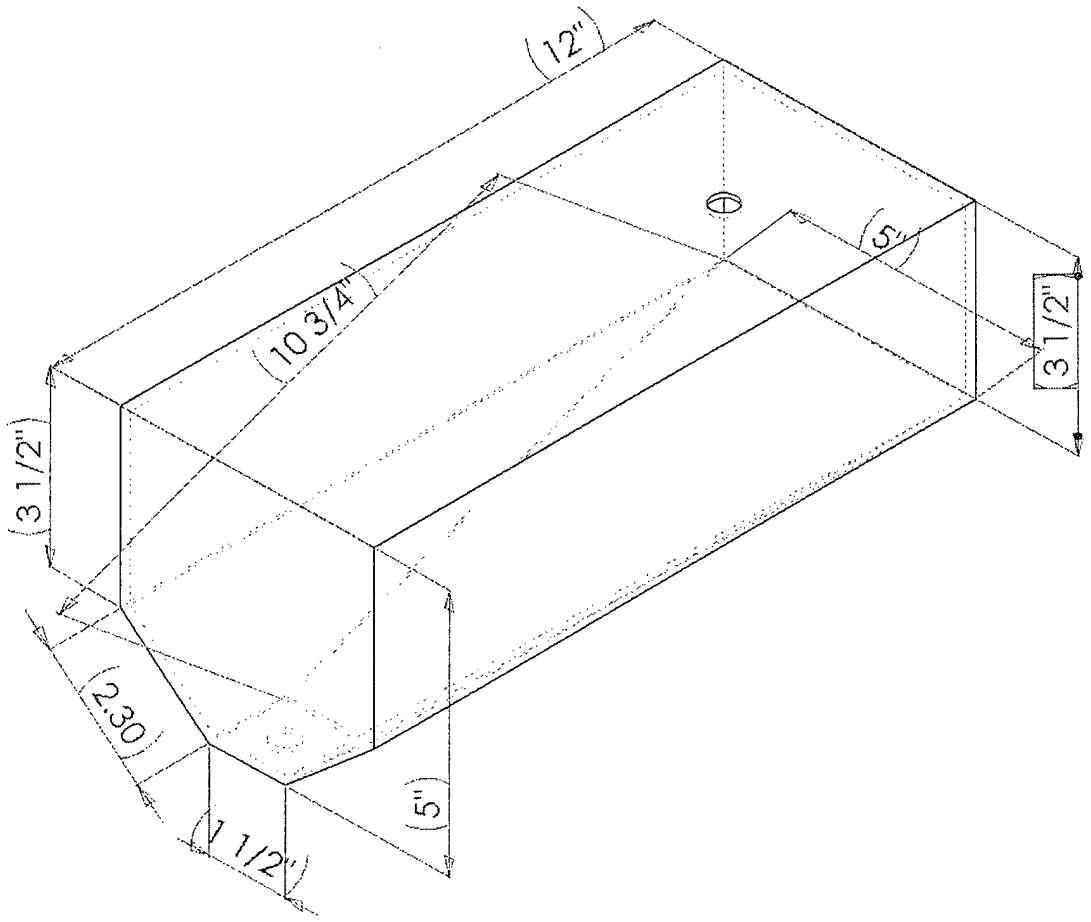


Figure 5 Water Tank Dimensions

Landing Gear Configuration

Like all aircraft, the landing gear configuration on this aircraft was chosen in accordance with its missions. The requirement of having the aircraft to fit inside a 4' X 2' X 1' box dictates that the gear would have to retractable or detachable. We considered three landing gear configurations. A tail dragger gear would allow the both the rudder and wheel to be controlled by the same servo, thus improving RAC. However, such an aircraft often only balances on the forward main gears, leaving the aircraft with stability issues during high-speed taxi. We were left with the bicycle or the tricycle gear types as the practical choices. The bicycle type gear required having the center of gravity of the aircraft to be place approximately half way between the front and the back gear. This arrangement would have made takeoff more difficult since no rotation could be attained. Furthermore, the almost equal distribution of the weight on the wheels would have required both the front and back wheels to be steerable. This not only complicates the construction process, but also requires a higher level of calibration to make sure all parts are lined up straight. For these reasons, the tricycle gear configuration was chosen. This type offers great stability during taxiing, ease of retraction and relative ease of construction. We determined the tricycle gear to be the best compromise for this design.

Initial Weight Estimate

In order to obtain an estimate for the aircraft, UCLA's 2003 entry into the Design/Build/Fly was used and scaled. From conceptual calculations, a rough estimate of thrust required to meet the takeoff requirement was found. Thus a weight of the required motor could be added to the approximation. Other parts were simply scaled by the size of the expected component. For instance, the 2004 wings would need to be approximated 30% larger to accommodate the increase in weight. The weight was therefore taken to be 30% more than the 2003 entry.

Initial Weight Estimate

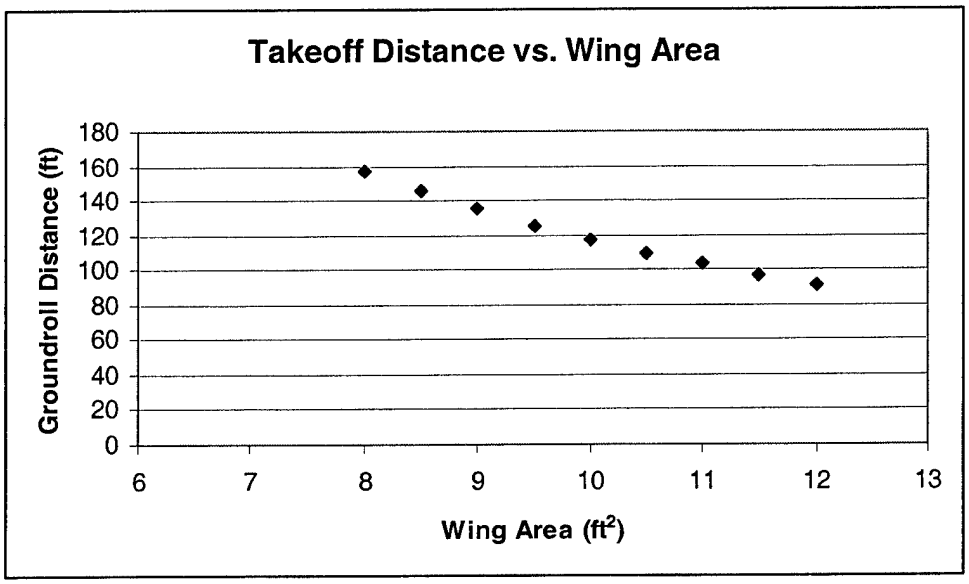
Component	Weight (lb)	
	2004 Aquanaut	2003 Bud-E
Payload	9.5	5
Battery	5	3.6
Motor	1.5	1.2
Wings	3	2
Landing Gear	2	1.5
Electronics	0.5	0.5
Fuselage/Tail	6	3.7
Sum	**27.5**	**17.5**

Table 3 Initial Weight Estimate

Preliminary Design

Initial Wing Sizing

The leading factors dictating wing size were the portability and takeoff requirements. Since the aircraft had to fit in the 4 x 2 x 1 foot box, wing sections could be no longer than 4 feet. This would require joints to connect the sections; thus leading to a reduced strength of the wing. Therefore, the number of sections should be limited. From this assumption, the wing should be no longer than 12 feet in span, limiting the number of sections to three. The aircraft also had to meet a maximum takeoff distance of 150 ft. Wing area and thrust are the primary factors in takeoff distance. Assuming the use of the Selig-Donovan 7062 airfoil (which had been used in previous years) and using preliminary thrust estimates, the planform area of the wing was optimized such that the aircraft could liftoff with at least 30 ft remaining on the runway.



Wing Area (ft2)	Stall Speed (ft/s)	Liftoff Speed (ft/s)	Takeoff Distance (ft)
8	44.78	53.52	157.7
8.5	43.44	51.52	145.7
9	42.22	50.46	135.1
9.5	41.09	49.11	125.7
10	40.05	47.87	117.3
10.5	39.09	46.72	109.9
11	38.19	45.64	103.2
11.5	37.35	44.64	97.1
12	36.56	43.7	91.6

Figure 6 Takeoff vs. Wing Area

Due to it's compliance with the takeoff distance (assuming an average thrust of 9 lbs), a wing planform area of 10 ft^2 was chosen. This would allow sufficient leeway in case of detrimental winds.

From previous experience, we had discovered that tapering of the wing was unnecessary for this aircraft. While, it would provide a slight reduction of induced drag, it also weakens the outer sections of the wing. With the extra loading applied to this UAV, the benefits of tapering the wing was decided to be negligible compared to the trade off in strength and manufacturability.

It was also clear that if a 10 ft^2 wing area was to be used, a span of 10 ft provided the best structuring of sections. A center wing carry-through of 2 ft could be permanently attached to the fuselage while two 4 ft outboard sections could also easily fit in the packing crate.

Airfoil Selection

The selection of a cross section for our wing relied on several factors. Lift and drag are obviously the most important. An airfoil with a high C_{Lmax} would allow for greater climb and takeoff performance. A low drag coefficient is desirable to minimize parasite drag. The lowest C_D should also correspond with our flight C_L, thus minimizing drag for cruise speeds. There are also several factors not directly related to performance, but with manufacturability. The airfoil would need to be thick enough to be able to hold battery packs in its interior. Thick airfoils are also easier to construct. Many high lift airfoils have drastic curls at the trailing edge. Such a feature requires advance manufacturing of this thin, curved segment.

The cruise speed of 70 ft/s would lead to a low Reynolds number. It is therefore important to obtain accurate low Reynolds number data. The most extensive freely available listing is maintained by the Nihon Aero Student Group. The majority of this data is obtained from the University of Illinois at Urbana-Champaign's low speed wind tunnel. The many airfoils in this database were analyzed and several finalists were chosen.

The majority of airfoils could not attain sufficient C_{Lmax} at the required Reynolds number. These would hinder the ability of the aircraft to takeoff in the required distance. Only a few dozen could achieve a C_L of 1 or greater at these low speeds. The exceptional airfoils were the FX63-167, SD7062, DAE11, and the SG6042.

From our payload deployment tests, it became obvious that the downwind segment of the flight course would take at least 14 seconds. Using this information, it was determined that the aircraft should cruise at 70 ft/s. Flying faster than this speed would only require a longer run as the payload could not be deployed quickly enough. A slower cruise, while allowing for a good safety margin, would be wasting time. Using this cruise speed and weight estimates, the cruise lift coefficient would be

$$C_L = \frac{2W}{S\rho V^2} = \frac{2 * 27lbs}{10ft^2 * 0.0023\frac{lb}{ft^3} * 70\frac{ft}{s}} = 0.48$$

This C_L of 0.48 is where the airfoil should therefore be optimized.

Takeoff and cruise speeds could be used to obtain Reynolds numbers. These values were 300000 and 450000, respectively. Maximum CL would have greater importance at takeoff (300000 Re), while minimum CD would be important a cruise (450000 Re).

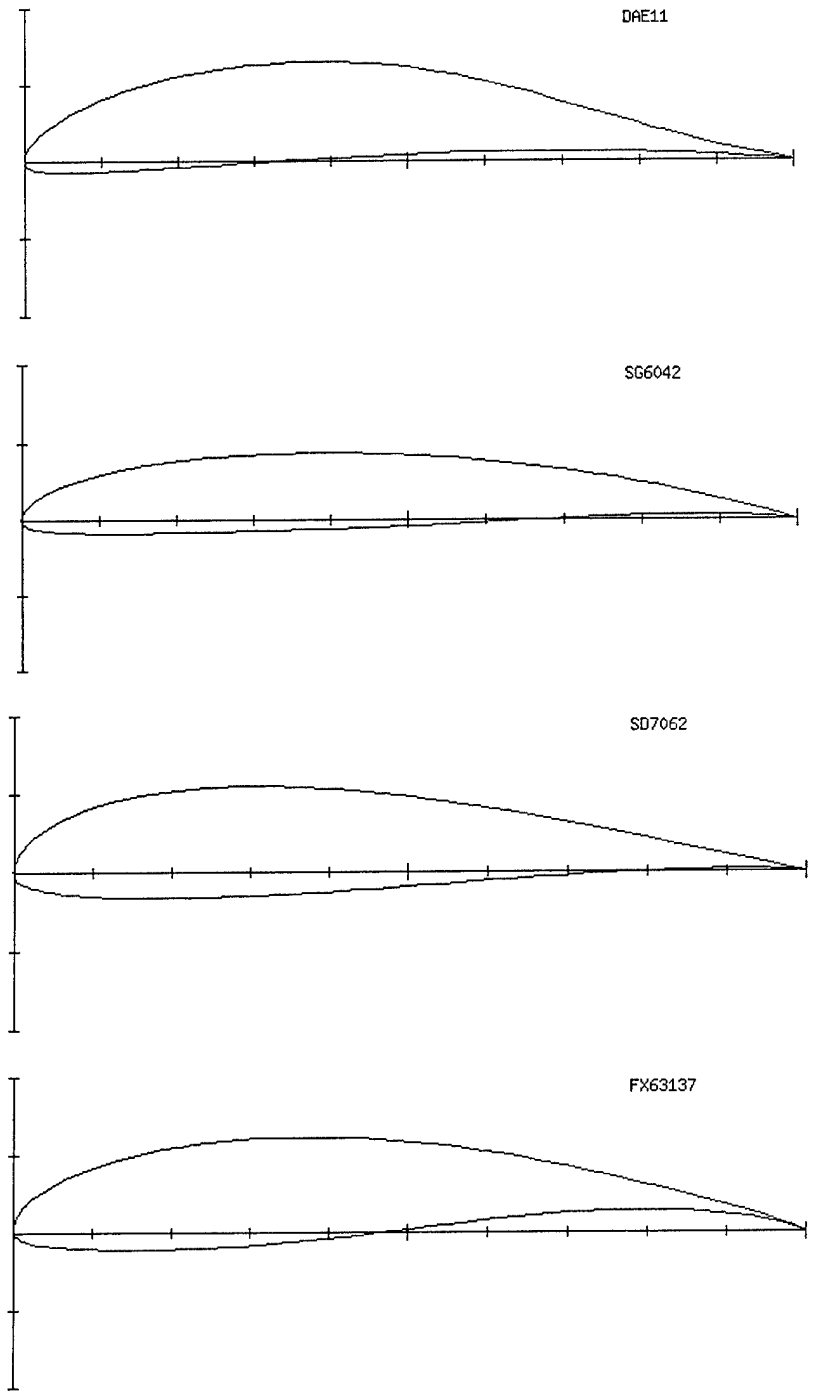


Figure 7 Airfoil Sections of Finalists

While the DAE11 exhibited a high maximum C_L of 1.65 and a moderate C_D of 0.01, the data was based entirely predicted performance. It was decided that the airfoil data should have physical experiments to back it. The DAE11 was therefore, eliminated.

The SG6042 attained a moderate C_{Lmax} of 1.45 and a low C_D of 0.008. While this would be great for high speed flight, the CLmax was lower than we were hoping to achieve. Since we had decided against using flaps for takeoff, our target CLmax was 1.5 or higher. Although the airfoil would not be used for this aircraft, it is likely that we may use it in the future.

The two remaining airfoils, the SD7062 and FX63-137, both achieved adequate C_{Lmax} of 1.5 and 1.6. Their respective drag polars and lift curves are shown below. While the SD7062 exhibits both high C_L and moderate C_D at the specified Reynolds numbers, the higher C_{Lmax} of the FX63-137 at lower Reynolds numbers were ultimately the deciding factor. It should also be noted that both airfoils achieve their minimum C_D at a C_L of 0.5. At our cruise Reynolds number, the C_D of both is approximately 0.01. The higher slope of the FX63-137 drag polar also means that drag would remain low if we required higher lift coefficients.

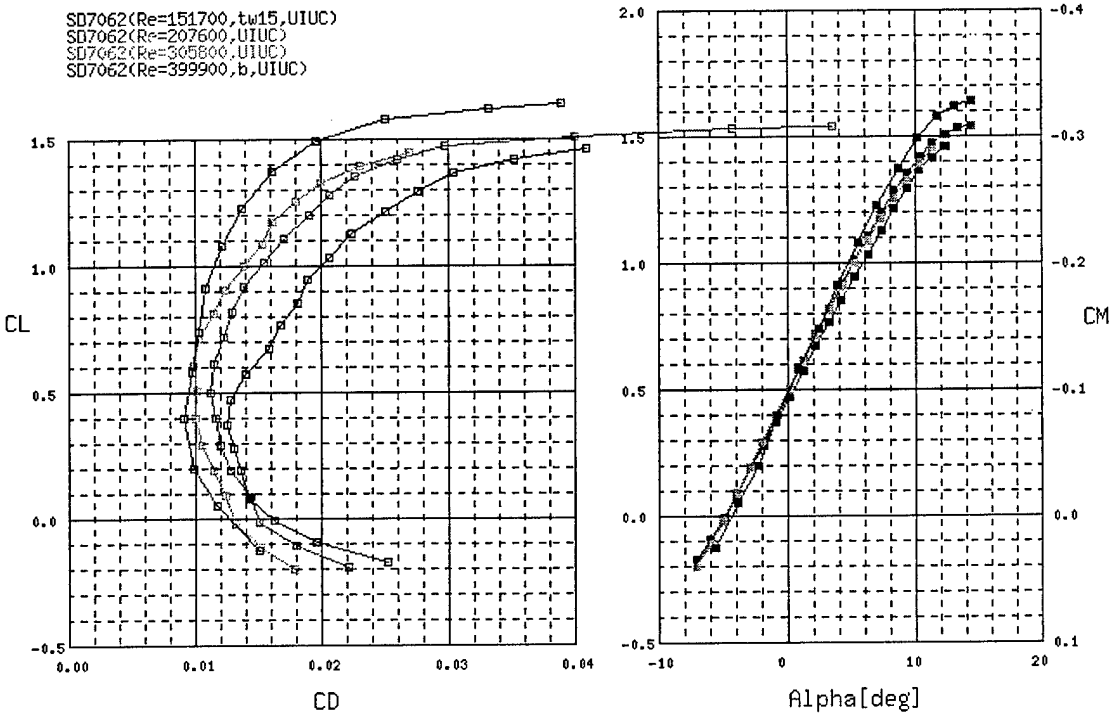


Figure 8 Drag Polar of Selig-Donovan SD7062

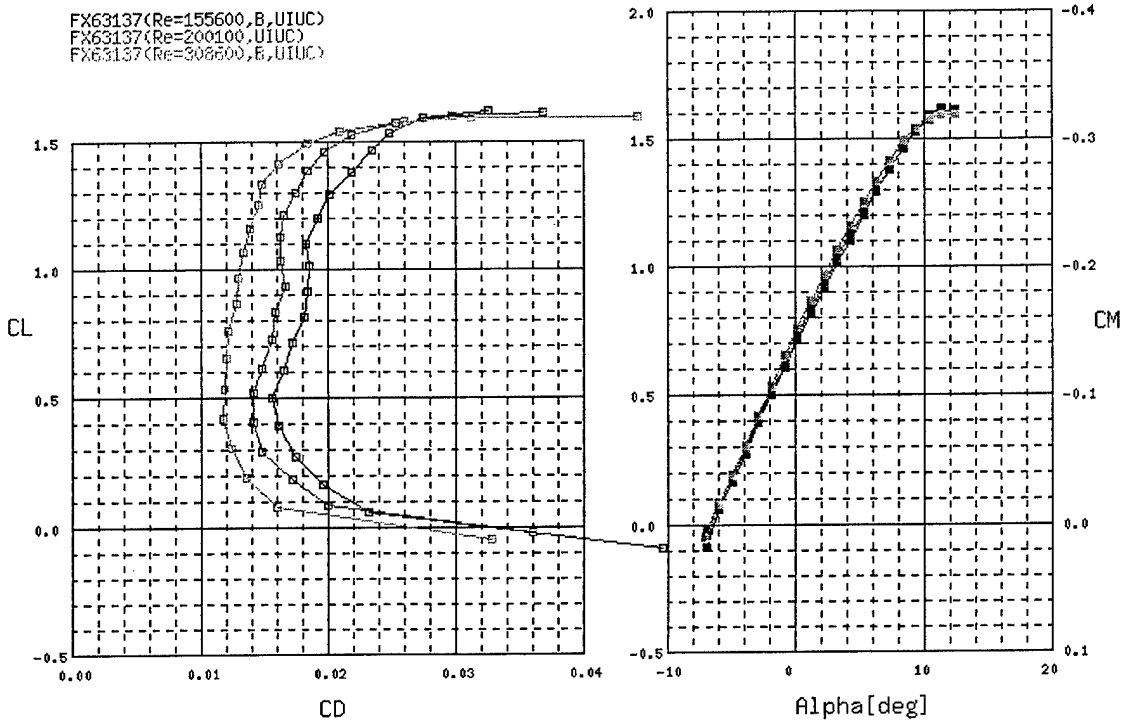


Figure 9 Drag Polar of Wortmann FX63-137

A concern with the FX63-137, however, is its trailing edge curl. This has the benefit of acting like a permanent moderate flap. The characteristic is harder to manufacture, as it thins out and curves at the same time. It was determined that, in the case of the FX63-137, the effect was within our capability to construct. The Wortmann FX63-137 was selected as the airfoil for the Aquanaut.

Empennage Sizing

The empennage is the main stabilizing component of the aircraft. Both an adequate moment arm and a large surface area are required for a horizontal stabilizer to be effective in balancing the forces of flight. The same is true for the vertical stabilizer. However, the portability requirements of the contest enforce restrictions on the size of these components. Either the fuselage and tail must fit within the specified crate or the tail would have to be detachable.

In order to decide whether or not a detachable tail was required for our design, a trade was used between tail moment arm and chord length of the horizontal stabilizer. The trade assumed that the tail would be fixed to the fuselage and that the tail span was restricted to 1.95 ft.

LE of Horizontal Tail (ft)	HT Chord (ft)	S_{HT} (ft^2)	Control Volume (Horizontal)
2.7	1.3	2.535	0.4397
2.8	1.2	2.34	0.4231
2.9	1.1	2.145	0.4036
3	1	1.95	0.3812
3.1	0.9	1.755	0.356
3.2	0.8	1.56	0.3279
3.3	0.7	1.365	0.297

Figure 10 Horizontal Tail Sizing

From historical data, an ideal control volume for a plane with our desired maneuverability would be 0.50. It is clear from this data that a conventional tail would be insufficient to stabilize and control the aircraft if the tail was fixed. However, the use of an all moving tail surface reduces the control volume requirement. The use of such a control scheme would only require a control volume of about 0.40. This, and the fact that construction of parts within a foot is cheaper, led to our decision to us a 1.0 foot chord all moving horizontal tail.

The vertical tail was sized in a similar fashion. However, it was clear that any vertical surface would have to be removable. There was simply no way of packing the fuselage with a vertical stabilizer intact. The final specifications of the empennage are shown in the table below.

Empennage Specifications			
Horizontal Tail		Vertical Tail	
Span	23.5 in	Span	14.7 in
Root Chord	12 in	Root Chord	14.0 in
Tip Chord	12 in	Tip Chord	5.6 in
Aspect Ratio	1.95	Aspect Ratio	1.5
Control Volume	0.3812	Control Volume	0.02

Table 4 Empennage Specifications

Fuselage design

Minimization of wetted area greatly affects friction drag, therefore tight internal packaging and a low fineness ratio would be the driving requirements for the fuselage (Raymer, 1999). Since RAC depends only on length of fuselage, it would be prudent to follow a low fineness ratio profile. To minimize pressure drag, a constant cross section fuselage was immediately ruled out. While it would greatly ease construction, the associated form drag from the increased equivalent frontal area and high boat tail drag would result in high power requirements and/or lower performance.

Knowing that a teardrop shape would offer the least drag penalty, it was now question of how close an approximation one should use. The only driving requirement is the need to fit the payload, motor and electronics in the structure. Profile drag is referenced to the maximum cross sectional area

(Raymer). To fit the payload inside, the cross-sectional area of a teardrop must be larger than that of a square. Therefore, to decrease profile drag, the fuselage should be shaped to fit the payload and no more. The Shorts Company produced a long line of aircraft with square cross-sections, including the Skyvan. In addition to increasing the profile drag, the acute angle formed between the high wing and the circular fuselage would impose an interference drag penalty greater than a high wing/square fuselage interface (McCormick). Interference drag effectively doubles as the angle decreases from 90 degrees to 60 degrees.

It was therefore determined that a rectangular cross-section would be ideal for our high-wing aircraft. The 90 degree angle between the wing and the fuselage would limit the amount of interference drag. The section around the water tank, which would be located under the wing, would take a rectangular profile sized to the minimum cross-section necessary for the payload system. Nose and tail sections would be contracted to the minimum cross-sections need for the motor and the tail structure.

Performance

Lift Prediction

The total lift is easily estimated for a straight rectangular wing. Approximations hold extremely well for such a wing. Following the procedures in references Raymer and McCormick, the predicted lift curve slope is shown below.

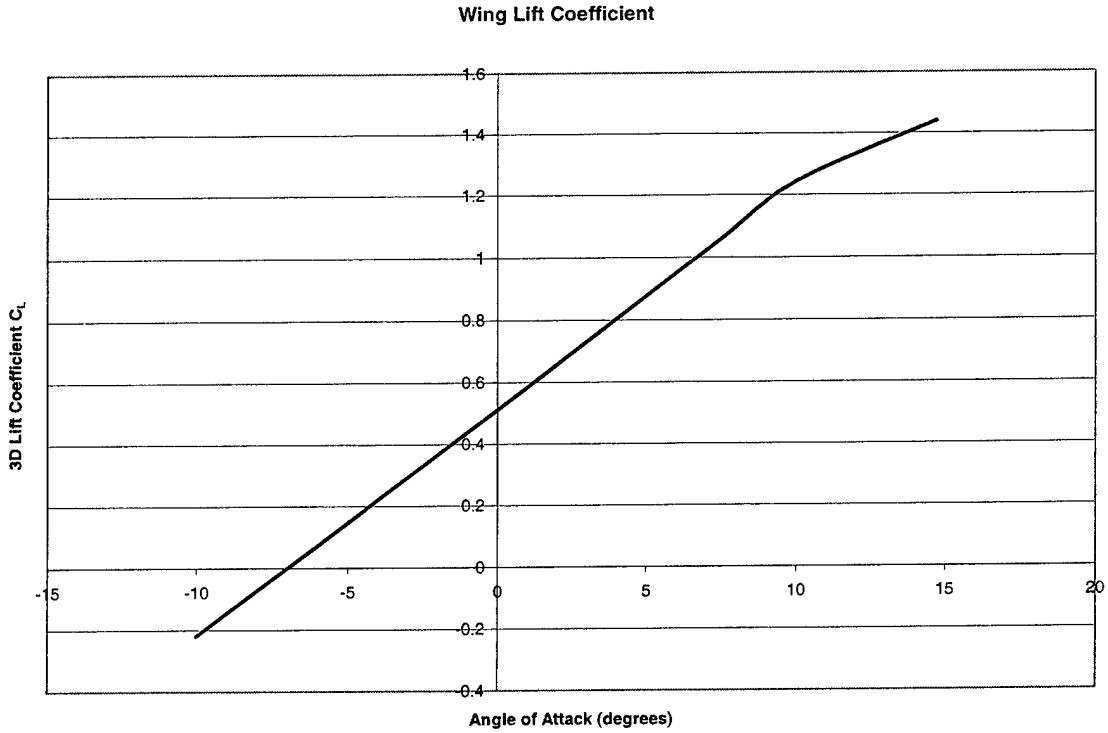


Figure 11 3D Wing Lift Coefficient

The maximum lift coefficient is reduced to 1.45 for the actual wing and the wing itself is able to achieve slightly higher angle of attacks than its corresponding airfoil section.

As predicted in our preliminary calculations, the ideal cruise C_L of this wing is 0.45. The preceding calculations hold and the aircraft will have a level cruising speed of 70 ft/s.

Drag Prediction

Drag estimation is a very time consuming task. For our estimation, we relied on a component method, whereby each component's drag is estimated and summed for the total aircraft drag. Airfoil structures, fuselage, payload deployment system, and landing gear were all taken into account. The breakdown between the different types of drag can be seen in the figure below.

An added safety margin was used for the total cruise drag to account for interference, leakage, and cooling drag which were neglected under the parasite drag.

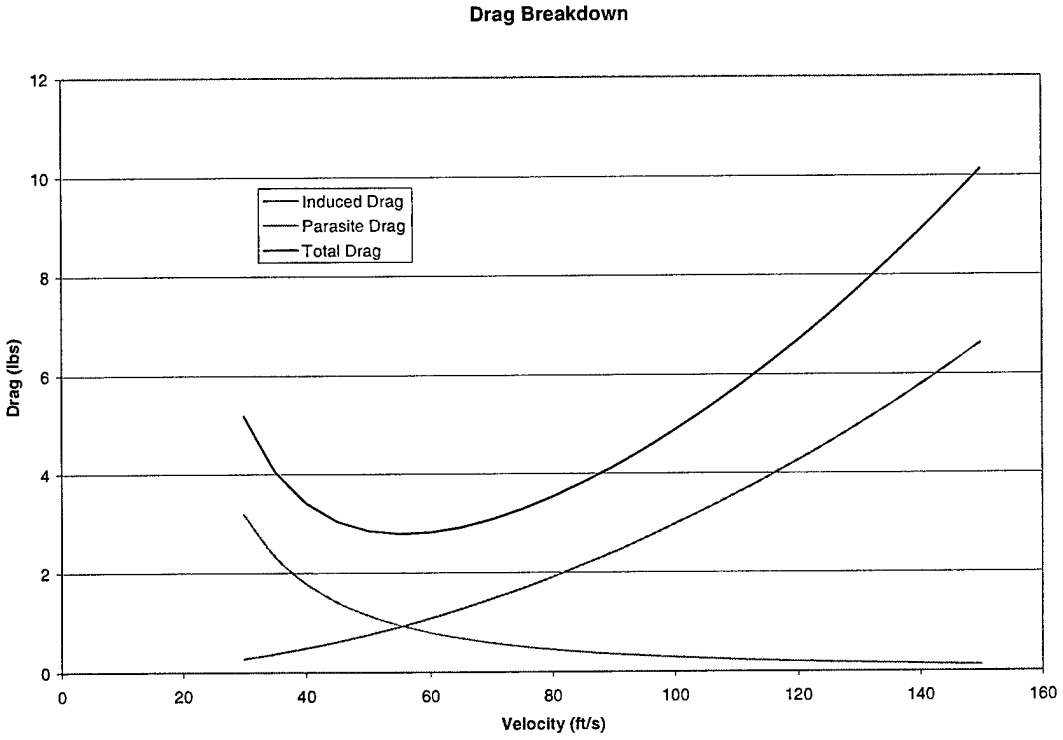


Figure 12 Aircraft Drag

Takeoff Performance

Using Hartmann's approximation for takeoff ground roll, and all of the updated specifications of the aircraft, the calculation is trivial. The first set of calculations is for the loaded case.

$$V_{stall} = \left(\frac{2 \cdot m \cdot g}{\rho \cdot S \cdot C_{L\max}} \right)^{1/2} = 38.13\, ft/s$$

$$V_{liftoff} = 1.2 \cdot V_{stall} = 45.6\, ft/s$$

$$\left(S_g\right)_{Loaded} = \frac{m \cdot V_{LO}^{\,2}}{2\left[T - D - \mu\left(m \cdot g - L\right)\right]_{0.707 \cdot V_{LO}}} = 117.3\, ft$$

Removing the payload, only the mass is decreased.

$$\left(S_g\right)_{Unloaded} = \frac{m \cdot V_{LO}^{\,2}}{2\left[T - D - \mu\left(m \cdot g - L\right)\right]_{0.707 \cdot V_{LO}}} = 44.7\, ft$$

Max Speed

Using component drag calculations, the total aircraft drag at cruise was calculated with respect to airspeed. Combined with the available thrust profile for our selected propulsion system, the maximum speed of the aircraft can be seen. From the chart below, the top cruising speed of the aircraft is the intersection between the two lines and is approximately 94 ft/s.

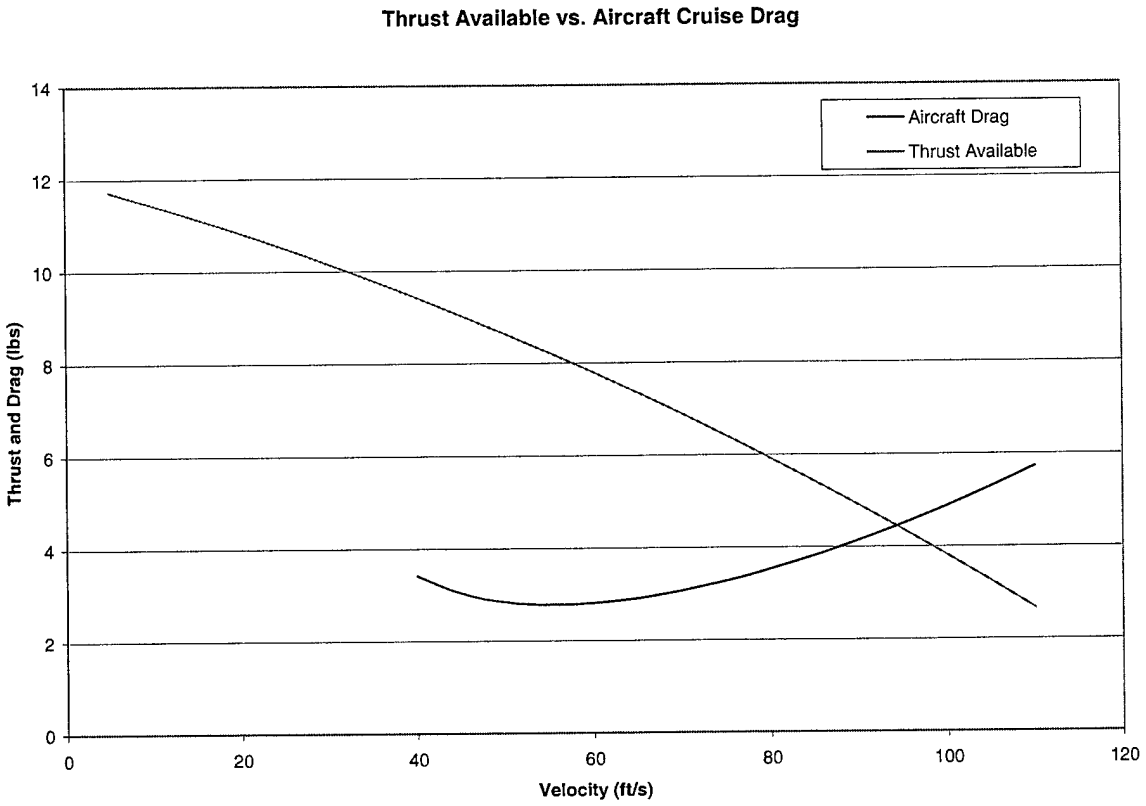


Figure 13 Maximum Speed

At this speed, however, the aircraft rate of climb drops to zero. This essentially means that the aircraft would be unable to maneuver. A more reasonable max speed of 90 ft/s was therefore set as the limit. It is also important to note that the weight of the aircraft does not significantly affect the max speed so the loaded and unload conditions are identical.

Rate of Climb

For low altitude radio controlled aircraft, the rate of climb does not depend on altitude. The air density simply does not change significantly over the range of altitude. It instead is dependent on the airspeed velocity that the propulsion system acts upon. Rate of climb is shown in the figure below compared with velocity for both loaded and unloaded conditions. A standard performance cap of 100 ft/min was imposed for operating standards. Maximum rate of climb for the loaded and unloaded aircrafts are 666 ft/min and 1027 ft/min, respectively.

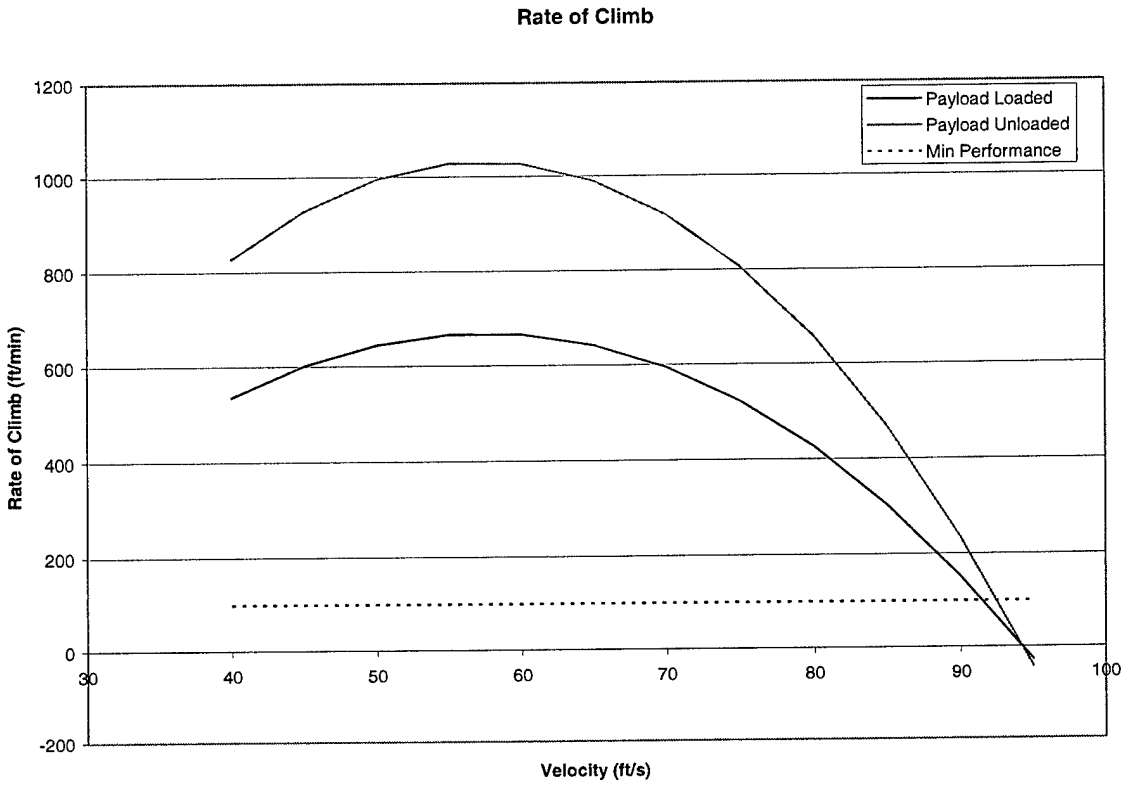


Figure 14 Rate of Climb

Predicted Score

Using all of the above data, and a moderate g loading of 2 for turning, it is possible to approximate the mission flight times. The Fire Fight mission can be approximated as a takeoff, a 720 degree turn, a flight of 1000 ft at 70 ft/s, and a landing. This is repeated with periods of time to reload the water tank. The Ferry mission can be approximated as a takeoff, a 2880 degree turn, and a flight of 7000 ft at 90 ft/s. Allowing for 30 seconds to refill the water tank, the flight time of the Fire Fight mission would be 2 min 30 sec and the Ferry mission would be 2 min 40 sec.

By applying the scoring formulas, UCLA AIAA Aquanaut would be able to achieve a total score of 74.57. This is assuming an average report score of 80.

Detailed Design

Propulsion and Power Supply

The process of choosing a propeller and engine requires first the specifications that are required of the engine and propeller. The propulsion for this aircraft is produced by the propeller and the electric motor. First we established the parameters of the motor. The engine/prop combination needs to produce.

At 23 mph (33.7ft/sec) needs 12 pounds of thrust;
At 34 mph (50 ft/sec) needs 9 pounds of thrust;
At 48 mph (70 ft/sec) needs 6 pounds of thrust;
At 55 mph (80 ft/sec) needs 5 pounds of thrust;
At 60 mph needs 4 pounds of thrust;

Upon viewing the specification then we begin evaluating the type of electric motor we need. Due to competition restrictions we have to either use an Astro Flight or Graupner electric brushed motor. The decision to use the Astro Flight family of motors was due to the fact that Astro Flight motors are easier to acquire and our ample experience with their motors from previous competitions.

The Astro Flight Cobolt series is best suited in terms of power consumption and rpm produced. Using a commercially available software package called MotoCalc, we ran several computer simulations with the Astro Flight Cobolt 60 and 90 series. Using both direct and geared we agreed upon the Astro Flight #661 motor with a gear box ratio of 2.75. The information and comparison is located in the chart below.

We can also see the comparison between the 60 and 90 series motor. The 90 series motor only performs well at higher airspeeds. Looking at the data, there is no significant performance advantage in a 90 series motor.

For the propeller blade we chose the 24x16 blade, however, most available 24x16 propellers are component based and require a substantially sized hub. Therefore the 22x18 propeller, which produces similar thrust to the 24x16 and is readily available in single piece composite construction, was chosen.

Looking at the data we can see that the 24x16 series propeller performs better than the other props at 50 ft/s. The 24x16 performs better than most propellers at lower airspeeds and still holds up in the higher speeds.

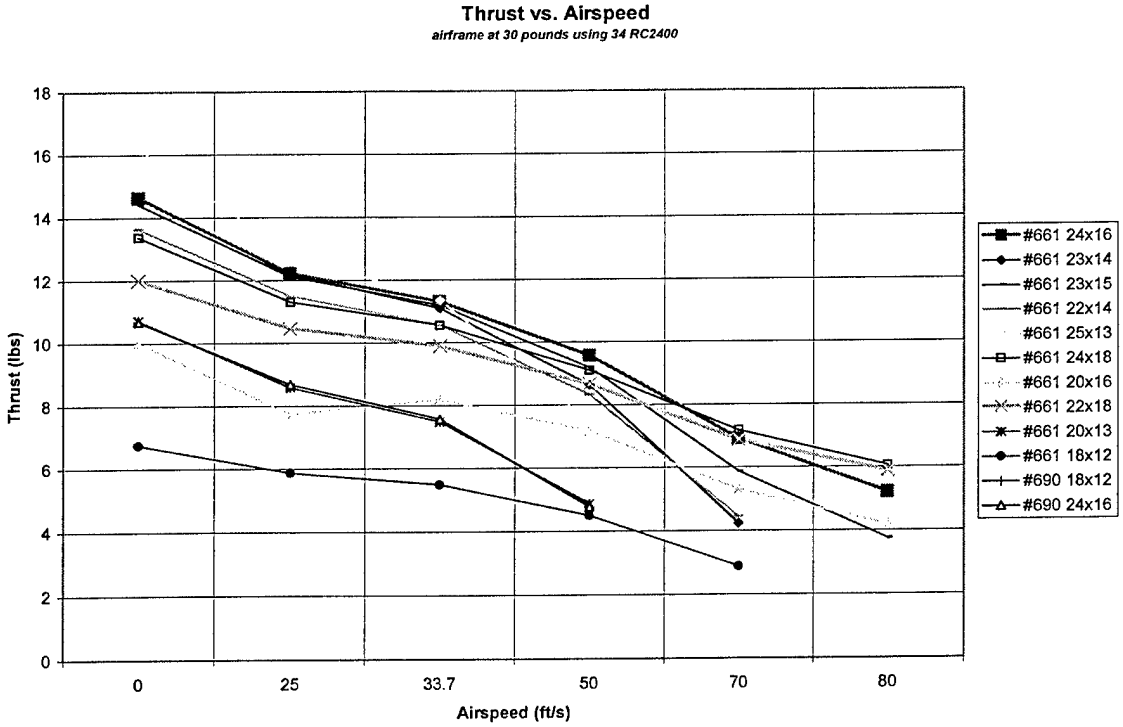
Then using MotoCalc we determined the number of batteries needed for the mission. Using the specification of maximum current draw of 40 amps and maximum weight of 5 pounds we came to the conclusion of using 34 cells of Sanyo RC2400 batteries in series. These batteries have 2400mH amps, which results in longer power life. 1700mH amp cells can run 100% throttle for 2:80 minutes and the 2400 cells can run at 100% throttle for 5:00 minutes according to the simulation.

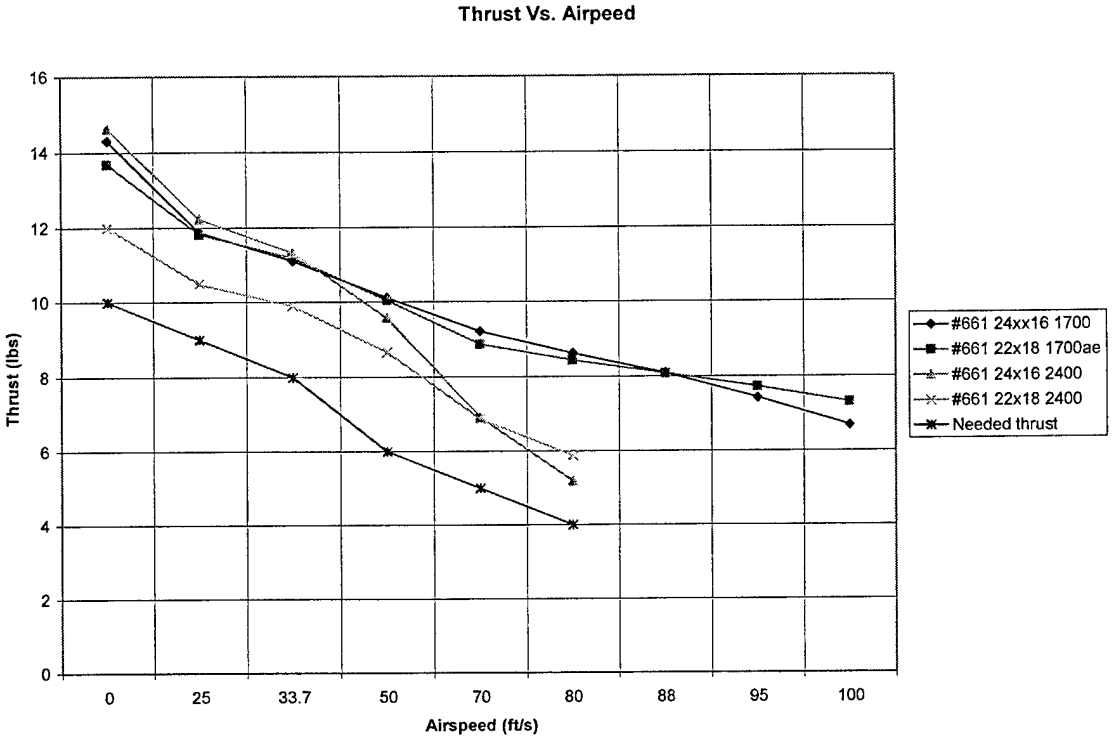
However we choose another battery pack as the alternative. Instead of using 34 cells we could use 50 cells of the Sanyo 1700AE batteries. This setup provides higher rpm for the engine, however the run time for this power pack would be short compared to the RC2400.

Using the equation (40 amps)*(total resistance/total voltage) we are able to find the maximum number of battery cells we could use before we reaching the limit of 40 amps. Then taking the maximum number of cells we made sure that the complete battery pack is under 5 pounds. After these two processes we got 34 cells for the rc2400 and 50 cells for the rc1700ae.

Using the above calculated figures and then plugging the different parts back into MotoCalc we are able to achieve a more accurate simulation of the thrust vs. speed. The improved and accurate version of the simulation is shown in the chart below.

In looking at the simulation results we can see that the combination of #661 24x16 and #661 22x18 with the RC2400 or RC1700ae batteries we are able to meet the specifications given by the designer.





Manufacturer's Specifications	
Motor	Astro Flight Cobalt #661
Constant	347 rpm/V
Idle Current	3 Amps
Resistance	0.103 Ohms
Weight	24 oz
Battery Specifications	
Capacity	2400 mAh
Impedence	0.0032 Ohms
Weight	2.08 oz
Gearbox	Superbox
Gear Ratio	2.75 to 1

Static Stability

Center of Gravity

Components had been sized earlier. By placing these in their locations with respect to the front of the aircraft, a reasonable estimate for the expected center of gravity is found.

Center of Gravity Calculation			
Component	Weight (lbs)	Location (ft)	Moment (ft*lbs)
Fuselage Structure	4.0	1.5	6
Front Landing Gear	0.2	0.9	0.18
Main Landing Gear	2.0	1.5	3
Wing	3.0	1.4	4.2
Empenage	1.0	3.4	3.4
Payload	9.5	1.4	13.3
Payload System	0.5	1.2	0.6
Propeller and Spinner	0.3	0.1	0.025
Motor	1.7	0.25	0.4225
Batteries	5.0	1.3	6.5
Control System	0.6	1	0.63
Total	27.77		38.2575
Center of Gravity (Loaded Config)		**1.377655744**	

Table 5 Center of Gravity Calculation

Longitudinal Static Margin

The static margin is a measure of the distance between the center of gravity and the neutral point of the aircraft. As long as the center of gravity is ahead of the center of gravity, the aircraft is statically stable. This produces a downward pitch moment which helps the aircraft recover from stall. Desirable values range from 5% to 15%. A small static margin ensures that the horizontal tail will have adequate control to rotate and stabilize the aircraft.

Static Margin		
Loading Configuration	Aircraft CG	Static Margin
Payload Fully Loaded	1.3777	5.8%
Payload Deployed	1.3660	6.9%

Table 6 Static Margins at Loading Conditions

Yawing Static Stability

As a quick check of yaw stability, an aircraft's vertical tail should be able to meet the following inequality. Though this is not an extensive calculation, it shows that the horizontal stabilizer is large enough to counteract a gust of wind.

$$S_{VT} \cdot l_{VT} \geq 0.7 \cdot Volume_{fuselage}$$

$$1.00\,ft^2 \cdot 2\,ft \geq 0.7 \cdot 1.23\,ft^3$$

$$2 \geq 0.861$$

UCLA AIAA Aquanaut

Geometry

Length	Height	Width	Span	Wing Area
47.5"	8.5"	6.0"	118'	10.0 sq. ft.

Control Volumes		Aspect Ratio
0.0200 (Vertical)	0.3812 (Horizontal)	9.5

Performance

$C_{L\,max}$	L/D_{max}	Max Rate of Climb	Stall Speed	Max Speed
1.45	15	11.1 ft/s	38.13 ft/s	90 ft/s

Takeoff Length (empty weight)	Takeoff Length (gross weight)
44.7 ft	117.3 ft

Weight

Airframe	Propulsion System	Control System	Payload System	Payload
10.4 lbs	6.7 lbs	10 oz	8 oz	9.5 lbs

Empty Weight	Gross Weight
18.2 lbs	27.7 lbs

Systems

Radio Used	Servos Used
Futaba T6XHS (controller) Futaba R149DP (reciever)	Hitec HS-81MG (flight controls) Hitec HS-805BB (payload)

Propeller Used	Motor Used	Gear Ratio
22"/18"	AstroFlight Cobalt 60 #661	2.75:1

Battery Config
34 cell Sanyo RC2400 (serial)

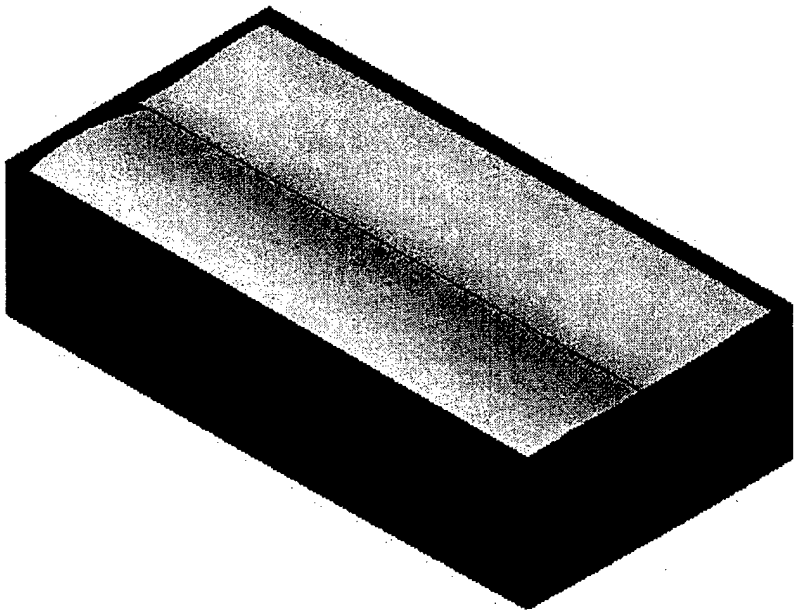
Table 7 UCLA AIAA Aquanaut Data Specifications

Rated Aircraft Cost

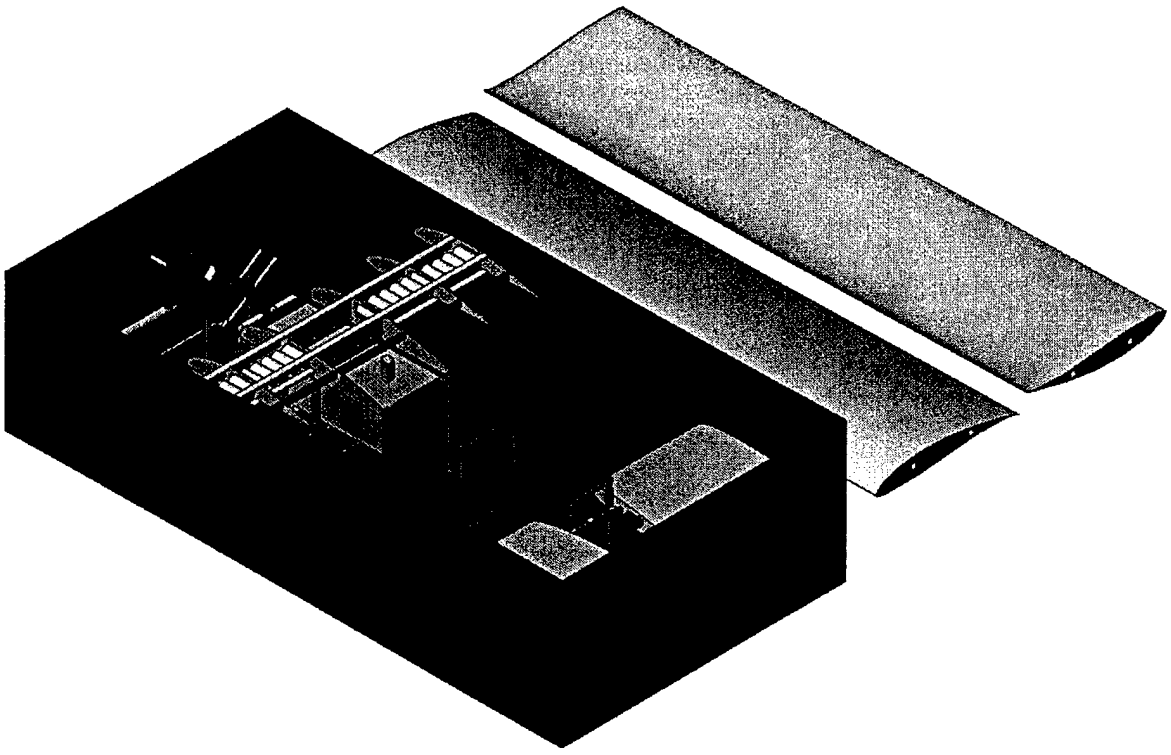
	Value	Multiplier	Total
Manufacturer's Empty Weight			
Aircraft Weight without Payload	18.2	$300	$5,460
Rated Engine Power			
Total Battery Weight	5 lbs		
Number of Engines	1		
REP	5	$1,500	$7,500
Manufacturing Man Hours			
Wing Span	9.8333 ft		
Max Wing Chord	1.0204 ft	10 hr/ft^2	100 hr
Control Type	1	5 hr	5 hr
WBS Wing			105 hr
Fuselage Length	3.958		40 hr
Fuselage Width	0.5		
Fuselage Height	0.708		
Fuselage Volume	1.40179	20 hr/ft^3	28.0358
WBS Fuselage			28.0358
Vertical Surfaces	1	10 hr	10 hr
Horizontal Surfaces	1	10 hr	10 hr
WBS Empenage			20 hr
Number of Servos	6	5 hr	30 hr
Number of Speed Controllers	1	5 hr	5 hr
WBS Flight Systems			35 hr
MFHR	188.04 hr	$20/hr	$3,760.72
Rated Aircraft Cost Total	**$16,720.72**	**1/$1000**	**16.72072**

Table 8 Rated Aircraft Cost

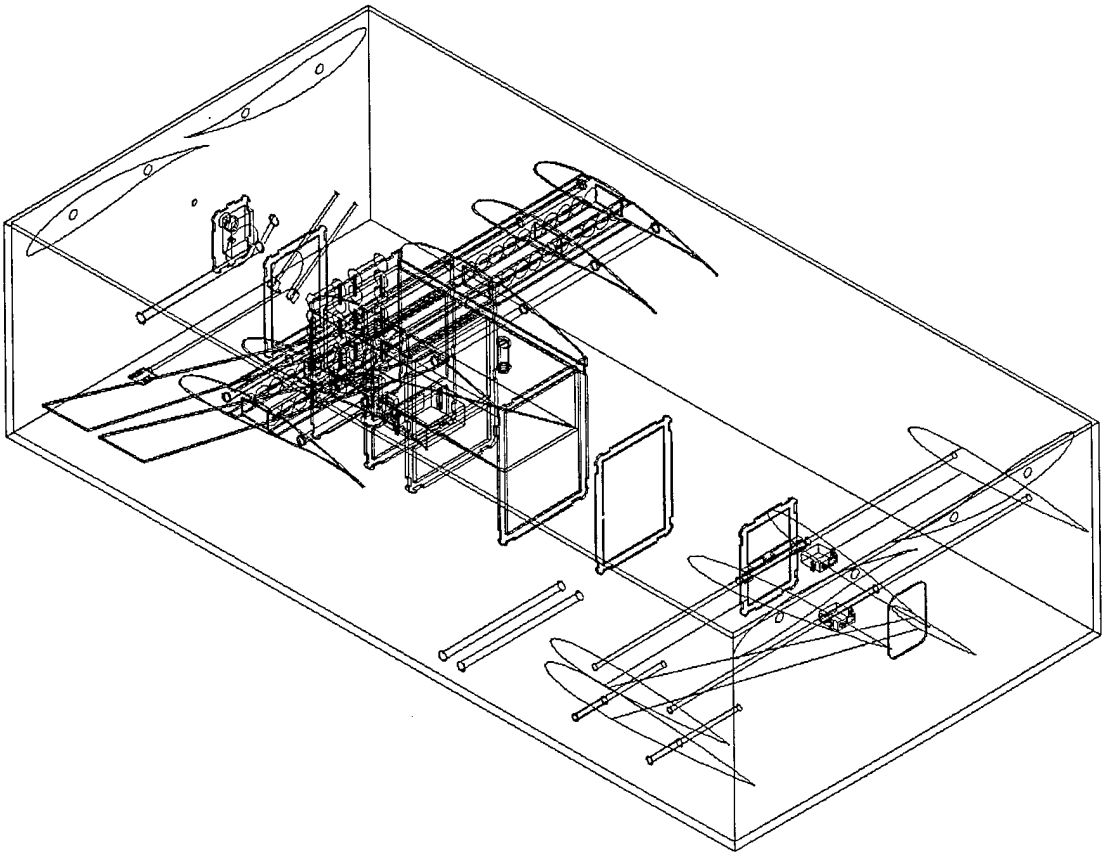
Drawing Package



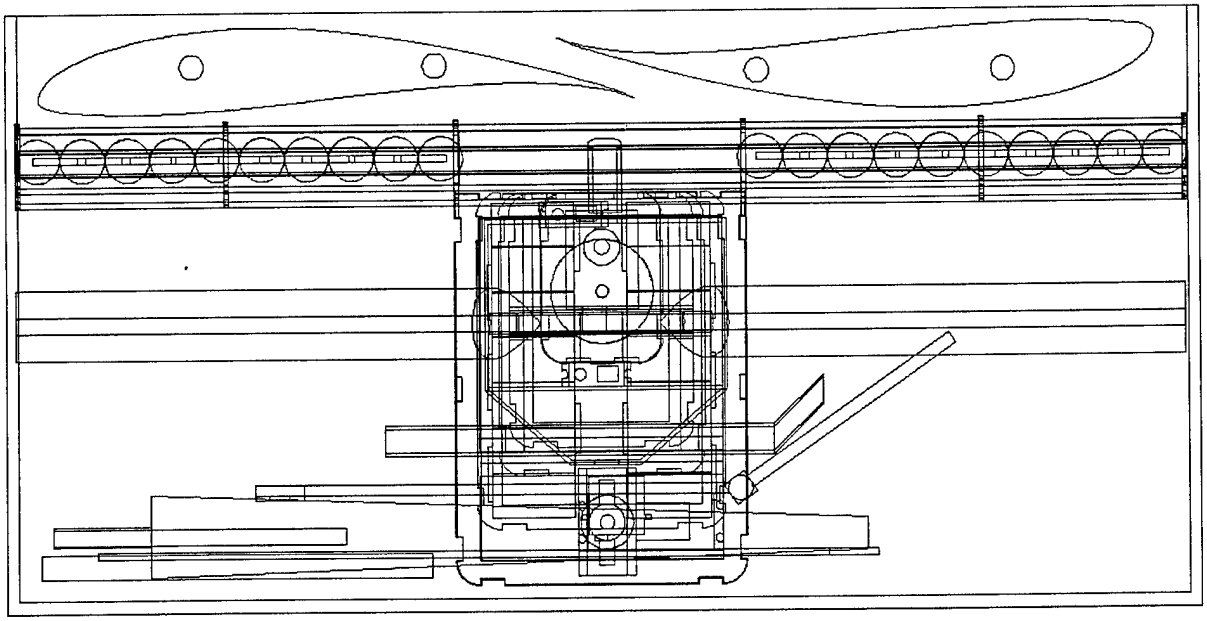
Aircraft completely packed in shipping crate



Layout in box with wings removed

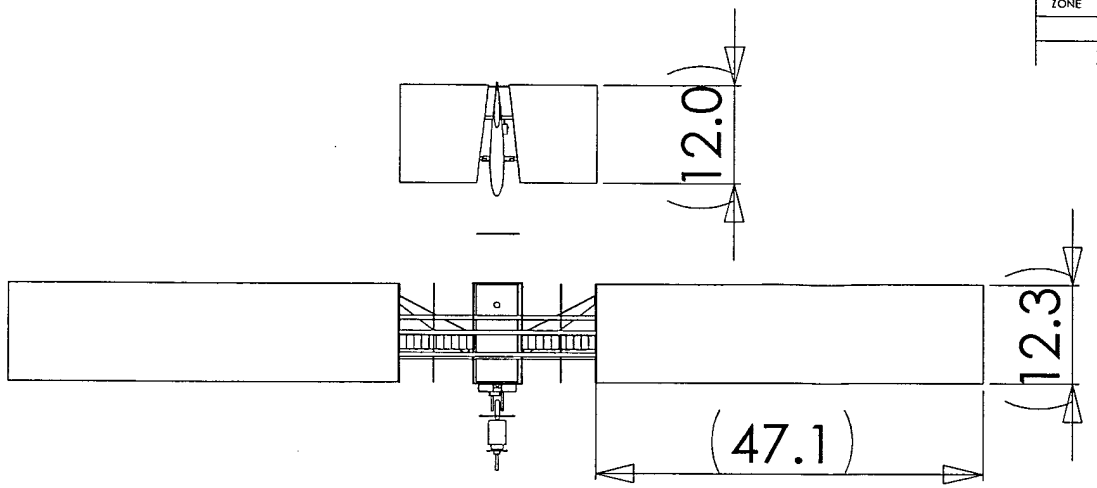


Transparent view of aircraft in crate (isometric)

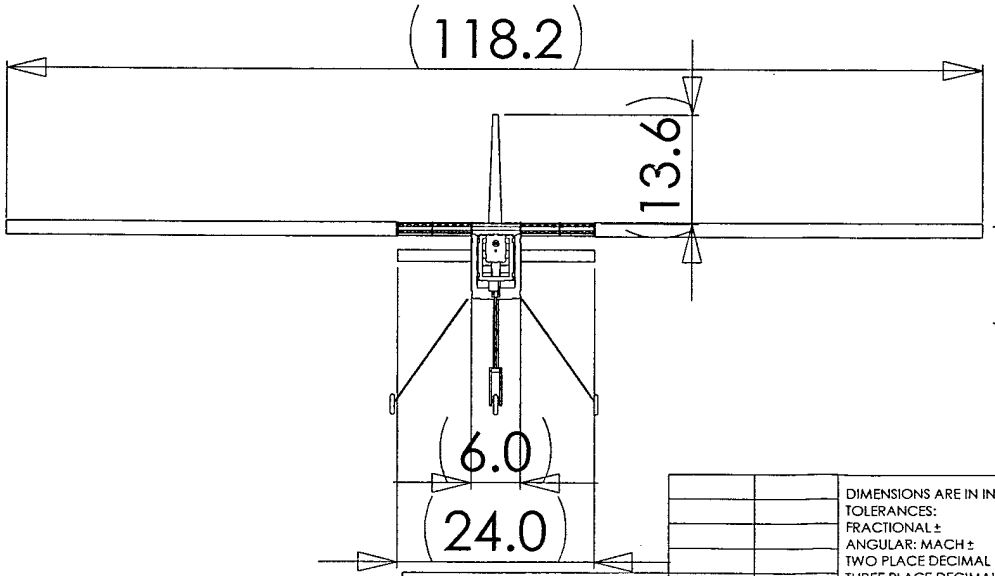


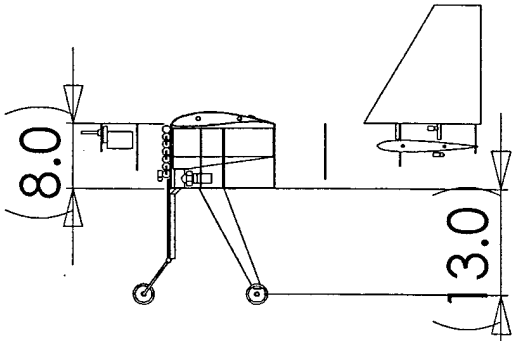
Transparent view of aircraft in crate (front view)

REVISIONS				
ZONE	REV.	DESCRIPTION	DATE	APPROVED



Note: Stringers and sheeting removed for increased visibility of underlying structure.





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NEXT ASSY	USED ON
APPLICATION	

DIMENSIONS ARE IN INCHES
TOLERANCES:
FRACTIONAL ±
ANGULAR: MACH ± BEND ±
TWO PLACE DECIMAL ±
THREE PLACE DECIMAL ±

MATERIAL —

FINISH —

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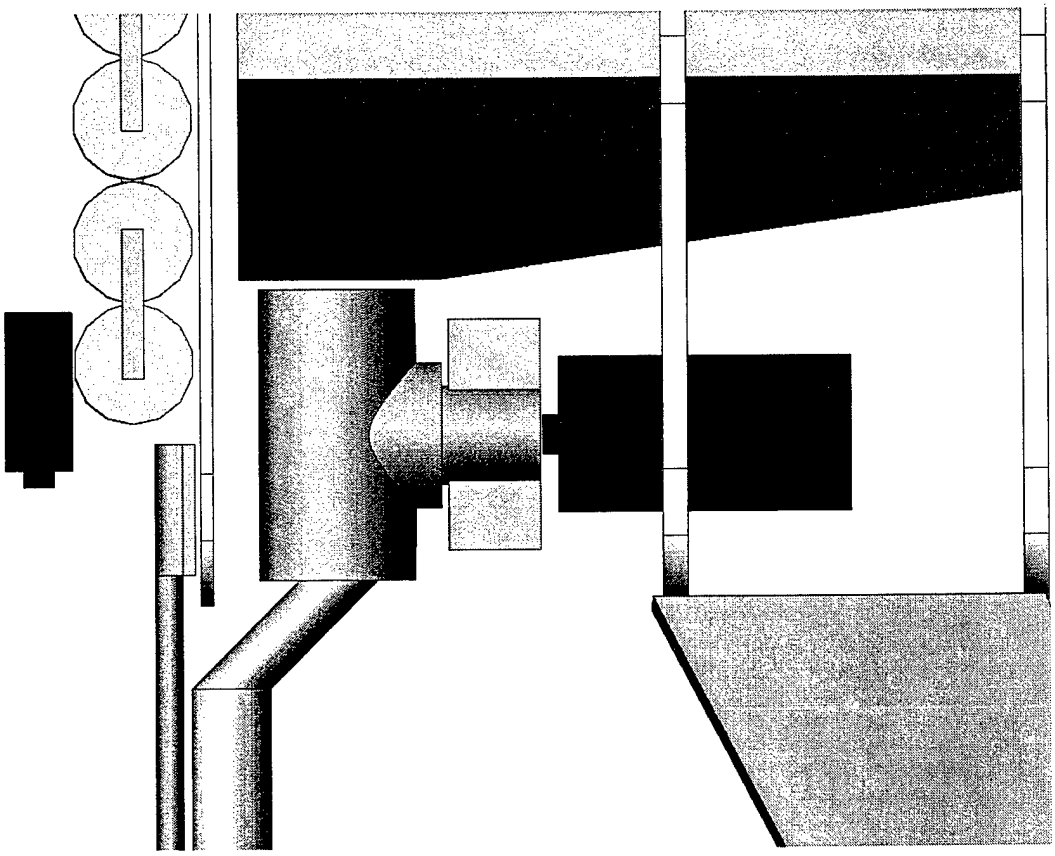
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MFG APPR.		
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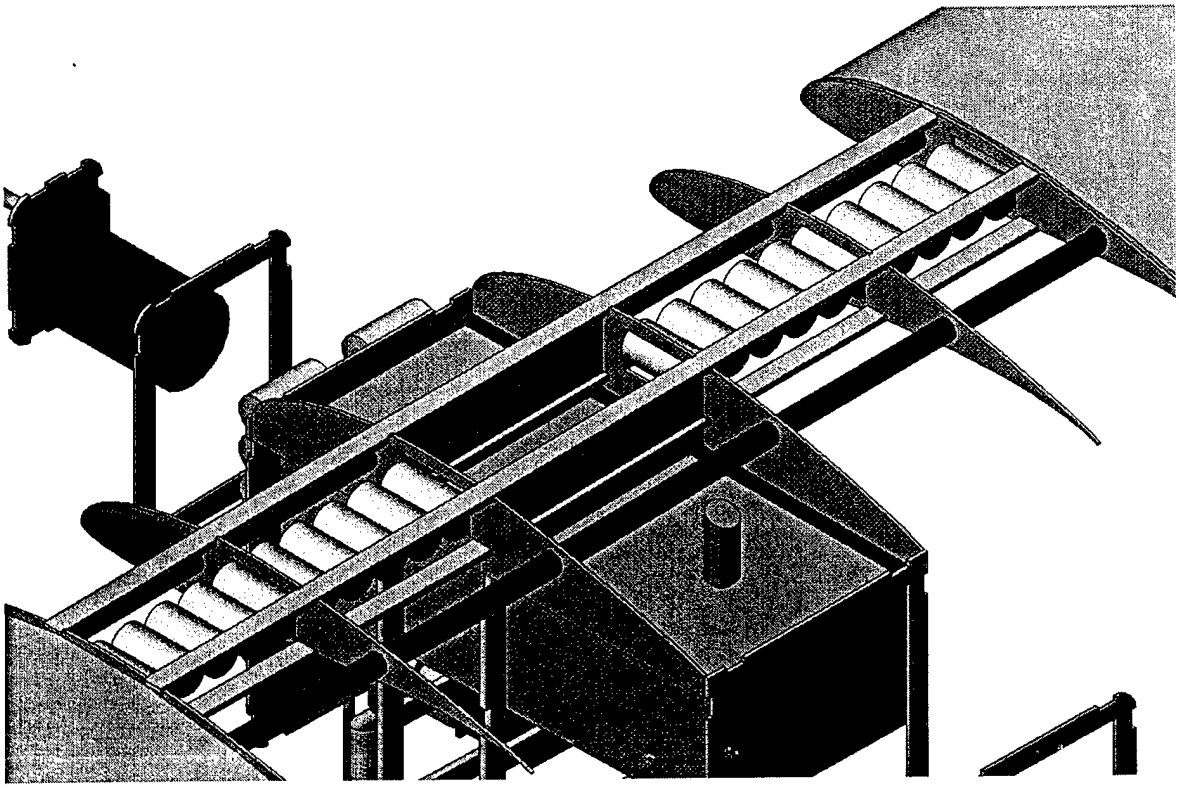
UCLA AIAA

Aquanaut
DBF 2004

SIZE	DWG. NO.	REV.
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Close-up of payload device. The payload system is based around a PVC ball valve. It is operated by a high torque servo. Attached to the valve is a vinyl tube that runs down the front landing gear.



Wing Carrythough Structure. The main structural strength comes from the two carbon fiber I-beams. The main load is transmitted through a carbon fiber tube that is directly connected to the front I-beam. A second carbon fiber tube is used as a guide spar. Also seen in this drawing is the inlet for the payload system.

Manufacturing Plan and Processes

Structure and Manufacturing

Wing Construction

For the final wing, issues such as how the outboard sections would attach to the center section now needed to be addressed. The center section was sized to have a 12.25" chord and a 23.5" span, reduced from 24" to allow clearance. Traditionally, not all this span would be taken up by wing. Part of the spar would be exposed so that the outboard sections could plug into it by means of concentric tubes. However, this overlap reduced the overall span of the wing. Therefore a new method was applied to the junction. The use of a separate joiner tube would allow the use of the whole span as an airfoil. The tube would first be plugged into the fuselage and then the protruding portion would fit into the outer wing section. This method would allow us to utilize the majority of the 10' span.

It now became critical that an efficient structure and manufacturing method be devised for the wing so that it would be light, easy to build and quick to assemble out of the box. Materials selection primarily drove the manufacturing methods available to us. The construction of carbon fiber or Kevlar ribs was initially considered. However it was rejected as being too labor and time intensive for initial wing fabrication while purchasing little structural improvement over more conventional methods. Indeed a composite wing would likely be far stronger than a non-composite wing, but as non-composite wings were seen to be strong enough this advantage was essentially negated. The remaining two manufacturing methods available were plywood frame with balsa skin, and spider foam cut into shape with hot wire EMD. Wood construction offered the advantages of having lighter weight and good geometric accuracy between design and finished product, but again came at the cost of long fabrication times and labor intensiveness. It was however the only manufacturing method which would allow placement of batteries and avionics in the wing as planned. Wood construction was therefore necessary for at least part of the wing. Hot wire EMD of foam was quick and relatively easy but as the wire frequently assumed some curvature during the cutting process it would alter the airfoil geometry a small degree. This deformation was deemed small enough to be acceptable and the quick manufacturing time argued nicely for this method. It was also quicker to fit control surfaces to a foam wing section. A hybrid wing structure was designed based on the comparative advantages of the different manufacturing methods. The entire center section of the wing would be built out of wood, allowing batteries to be stored in them, while the outer sections would be cut from foam.

The spars of the wing section underwent testing which is discussed under the testing section. These tests determined that we would use both cylindrical carbon fiber tubes and a carbon fiber I-beam for structural support.

Fuselage Structure

Since the payload system did not require any large holes in the fuselage, a conventional buildup of stringers and frames was possible. Cross-sections would be placed every 6" to form the shape of the fuselage. Additional frames were added under the quarter-chord position of the wing to transmit the loads from the wing to the fuselage and landing gear. Although carbon fiber was considered for the construction of the frames, it was ultimately deemed to costly for the minimal benefit. Birch plywood of 3/32" thickness was settled upon for the cross-sectional frames. Carbon fiber rectangular stringers would then connect the frames. This design ensured that the fuselage could successfully transmit loads as well as provide a rigid structure for the tail surfaces.

Tail Construction

Since the tail would be all moving, the construction would be much simpler. The surfaces would be cut out of foam in a similar fashion to the outer wing sections. The two halves of the horizontal tail would be joined at two points using carbon fiber tubes. The first is located at the quarter-chord point and is hinged with the fuselage. The second is located three-quarters from the leading edge and acts as the control rod. A servo is simply attached to move the control spar up and down to deflect the surface. The vertical tail would be constructed in a similar fashion.

Landing Gear Construction

Aluminum alloy 7075 of 1/8" thickness was chosen for the material due to its high strength as a basis for our landing gear. In order to add the flexibility required to absorb the impact from landing, the gear is hinged at the fuselage. This allows the brunt of the force to be transmitted to a reinforcing member, such as an oleo shock absorber. Testing of the landing gear setup with different members is set to carry out. It would compare the differences between spring devices and solid members. We are preemptively planning on employing 15-lb gas springs to absorb the forces.

Personnel Breakdown

Construction of the aircraft required a broad range of skills. In order to minimize manufacturing errors and maximize productivity, personnel were assigned tasks according to their experience. Skill levels were assigned to specific tasks in order to facilitate this process.

	Wing	Fuselage	Landing Gear	Tail	Propulsion
Structure	2	1	1	2	1
Control Surfaces	2	0	0	1	0
Sheeting	1	1	0	0	0
Electrical	1	1	1	1	2

Table 9 Skill Matrix

Higher values indicate greater difficulty.

Manufacturing Schedule

The manufacturing schedule was designed to create systems as quickly as possible. Simpler components such as the payload system were constructed early in the process. Other more complex components such as the wings were delayed. Building times were also scheduled around available time frames for the personnel. The earlier periods during the academic calendar, such as weeks 2 through 4, are times when class requirements are less constricting. Intense building was thus planned for those periods.

Academic Calendar	Winter Quarter Week 2	Week 4	Week 6	Week 8
Manufacturing				
Payload System				
Fuselage				
Wing				
Empenage				
Landing Gear				
Propulsion System				
Final Assembly				
Repair / Modification				

Table 10 Manufacturing Schedule

Testing Plan and Initial Results

Among the first of the completed components was the payload system. Respective components underwent static testing. Propulsion components underwent duration and thrust testing. With the actual aircraft completion, it would undergo a series of dynamic flight testing.

Static Tests

1.) Payload Deployment Test

Payload deployment system is tested for both time taken for all water contents to leave payload and also if the servo-valve combination works without any problems.

2.) Wing Tip Lift Test

The full size wings would be tested for structural integrity. Initially, the outboard sections are given tested on their ability to hold weight. With the completion of the aircraft, the fully loaded plane would be lifted by the tips of the wings. This simulates approximately 2.5Gs of force on the wing structure.

3.) Landing Gear Test

Landing gears were first tested on ability to hold the full weight of the aircraft. A drop test would be initiated afterwards from gradually increasing heights to a maximum of 3 feet to ensure structural strength and shock absorption.

4.) Motor / Battery Duration Test

The propulsion system would be put through a duration test. The motor is run at expected cruise power to determine the expected maximum flight time.

5.) Motor / Propeller Thrust Test

The motor was mounted on a self-made test stand that could move with minimal friction. Attached to the back was a linear spring force gauge that measured the amount of force produced by the motor when the propellers were attached. This test gave us the thrust figures we needed to determine if the plane could actually fly.

Dynamic Tests

1.) Fully Loaded Taxiing Test

The aircraft is simply powered up and taxied at high speeds to test the steering and stability of the landing gears.

2.) Unloaded Flight Test (Takeoff and Landing)

The aircraft is primarily tested for takeoff distance and low force landing. It takesoff and immediately returns to the ground.

3.) Unloaded Flight Test (Turning and Rate of Climb)

After successful takeoff, the aircraft is tested for maneuverability performance. Low g turns are initiated to test the sizing and deflection of control surfaces. Although rate of climb is difficult to measure for a UAV, the ability to climb is simply checked.

4.) Loaded Flight Test (Takeoff and Landing)

Similar to the unloaded test, except the payload is now carried.

5.) Loaded Flight Test (Turning and Rate of Climb)

Similar to the unloaded test, except the payload is now carried.

6.) Unloaded Flight Test (Flight Duration / Full Course)

The aircraft is put through the whole flight course to simulate the unloaded ferry mission. Four laps are flown with an opposing 360 degree turn on the down wind segment. This is a timed test to directly predict our performance in competition.

7.) Loaded Mission Flight Test (Internal Payload Deployment)

The aircraft is put through the whole flight course to simulate the loaded firefight mission. A lap is flown with an opposing 360 degree turn on the down wind segment. The payload is also deployed during the downwind segment. This is a timed test to directly predict our performance in competition.

Initial Testing Results

Payload Deployment Testing

A payload test was performed based on Bernoulli's equation. We tried different lengths of tubes and took the times for the 4 liters of water to exit the payload. The table below shoes that as the tubes became shorter, the time for deployment of the water increases significantly.

Tube Length (feet)	Deployment time (second)
2.0	17.84
	18.39
1.5	18.48
	18.20
1.0	19.14
	19.04
0.5	22.41
	22.53

The times from the table may appear to be slightly higher than satisfactory. But with an improved shape and an additional air intake hole (one that faces forward so the incoming air pressurizes the payload), the amount of time should be dramatically increased. We are expecting a deployment time of 14 to 15 seconds with the new improvements.

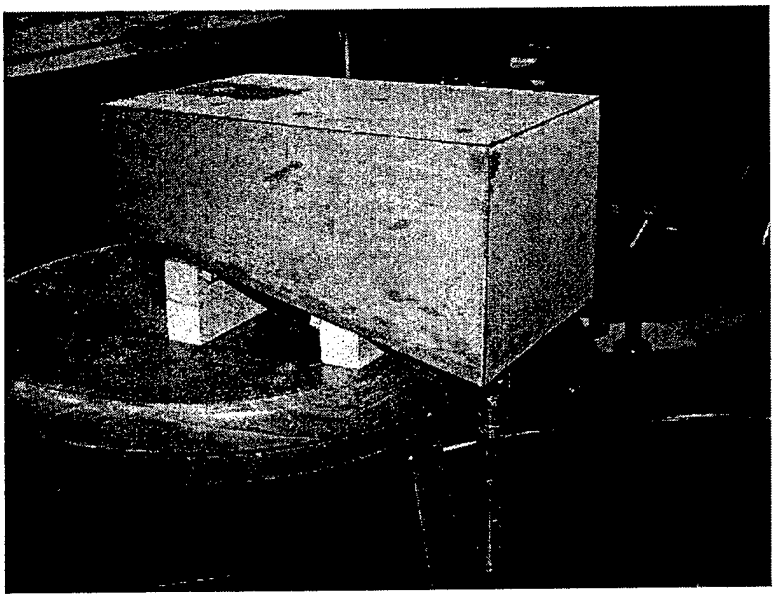


Figure 15 Payload Test Tank

Spar Testing

Our goal is to find the most efficient spar based on weight and strength to be used in the carry-through structure of the aircraft. Since the spars will run through the wing, it is necessary for the spars to be rigid enough to not deform in flight. This will ensure that the wings will not move in any direction other than what the pilot specifies.

We obtained several test pieces of beams that varied in composition and shape. Some of the compositions included: balsa wood, plywood, carbon fiber, and graphite. The different shapes include a plank, tube, rod, and I-beam.

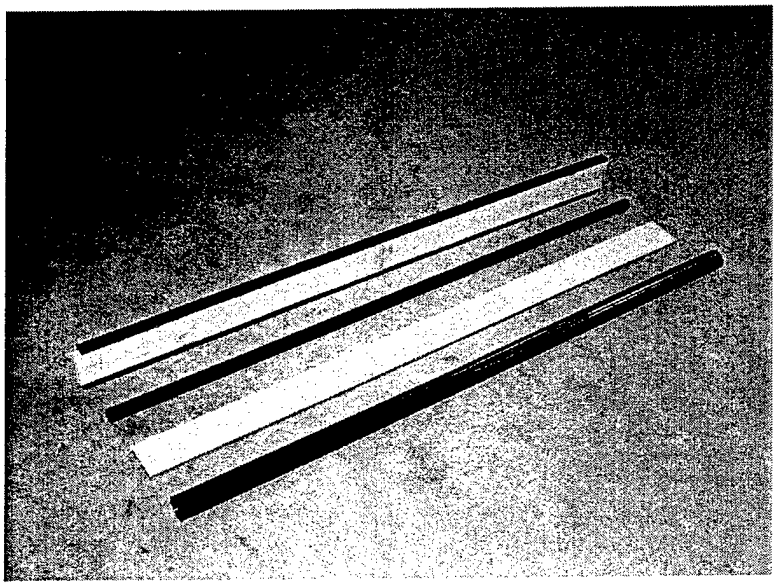


Figure 16 Spar Specimen

The test setup was designed simplistically for the purpose of gauging the moment force tolerated by the beams. We chose to test the torque tolerance since the moment force would be the dominating force on the spar while in flight compared to any compressive, tensile, or shear forces. Thus, the moment loads would cause the greatest amount of compressive and tensile stress in the beam. The diagram below illustrates the test setup where P_o represents the distributed counter-load and P represents the applied load. Simple free-weights were used to simulate these forces.

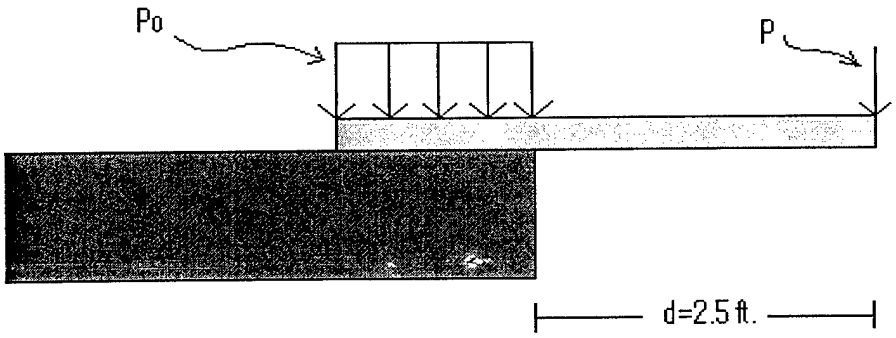


Figure 17 Diagram of Spar Test

Free weights were added to the bending beam until it fractured. Below is a table of each beam's description and tolerant moment load:

Test Spars					
Shape	Composition	Length (ft)	Width or Diameter (in)	Weight (oz)	Load at Fracture (ft*lbs)
Cylindrical Rod	Graphite	3	1/2	1.8	100
Cylindrical Tube	Carbon Fiber roll-wrapped	3	1/2	1.0	75
Rectangular Plank	Plywood	3	1 1/2	3.0	N/A
I-beam	Carbon fiber caps and balsa web	3	1 1/2	3.7	87.5

Our ideal spar should be rigid enough to maintain shape in flight, and withstand the maneuvering force of 5Gs while minimizing the overall weight of the aircraft. The graphite rod withstood the most force; however, it also deflected the most out of all the test spars. It exhibited a very low stiffness. The roll-wrapped carbon fiber tube was incredibly brittle and barely deflected before fracturing at 30lbs. The plywood plank could not be properly tested due to the torsion experienced in the beam when the load was applied. The I-beam barely deformed due to the rigidity of the graphite caps before the balsa web sheared off the graphite caps. This can be fixed by replacing the balsa with a stronger wood or even perhaps a full carbon fiber I-beam.

Based upon our specifications and tests, we will use an I-beam as our spars in the aircraft to keep the structure rigid and use graphite rods to join the wings to the fuselage to increase the overall strength. The combination uses both the strength of the graphite and the stiffness of the carbon fiber to our advantage.

Motor Test Stand Data

The motor test was performed on a self-built motor stand with a slide track attached underneath and a linear spring gauge attached at the back of the stand. When the thrust pulls the motor (sitting on the stand) forward, the spring gauge in the back measures the force that the motors pulls with. The motor was tested using 26 Sanyo RC 2400 cell batteries in serial configuration. Although this is less than our actual flight battery power, it was used as verification for our thrust calculations.

Propeller	Predicted Static Thrust	Measured Static Thrust	Thrust @ 5 mph
20/13 (full power)	7.140 lbs	6 lbs	6.123 lbs
15/13 (full power)	5.230 lbs	4 lbs	4.210 lbs
24/16 (75% power)	7.028 lbs	6 lbs	6.102 lbs

Table 11 Initial Motor Test Data

We found that the obtained thrust levels were consistently below predicted value. This is probably due to the fact that static thrust is hard to measure. None of the readings could be taken with completely stagnant air. The propeller would cause circulation in the surround air. Therefore, it was

decided that the data should be compared with a low airspeed calculation. The values then hold to prediction. We will continue motor tests as we acquire additional components of the aircraft and assemble the final battery packs.

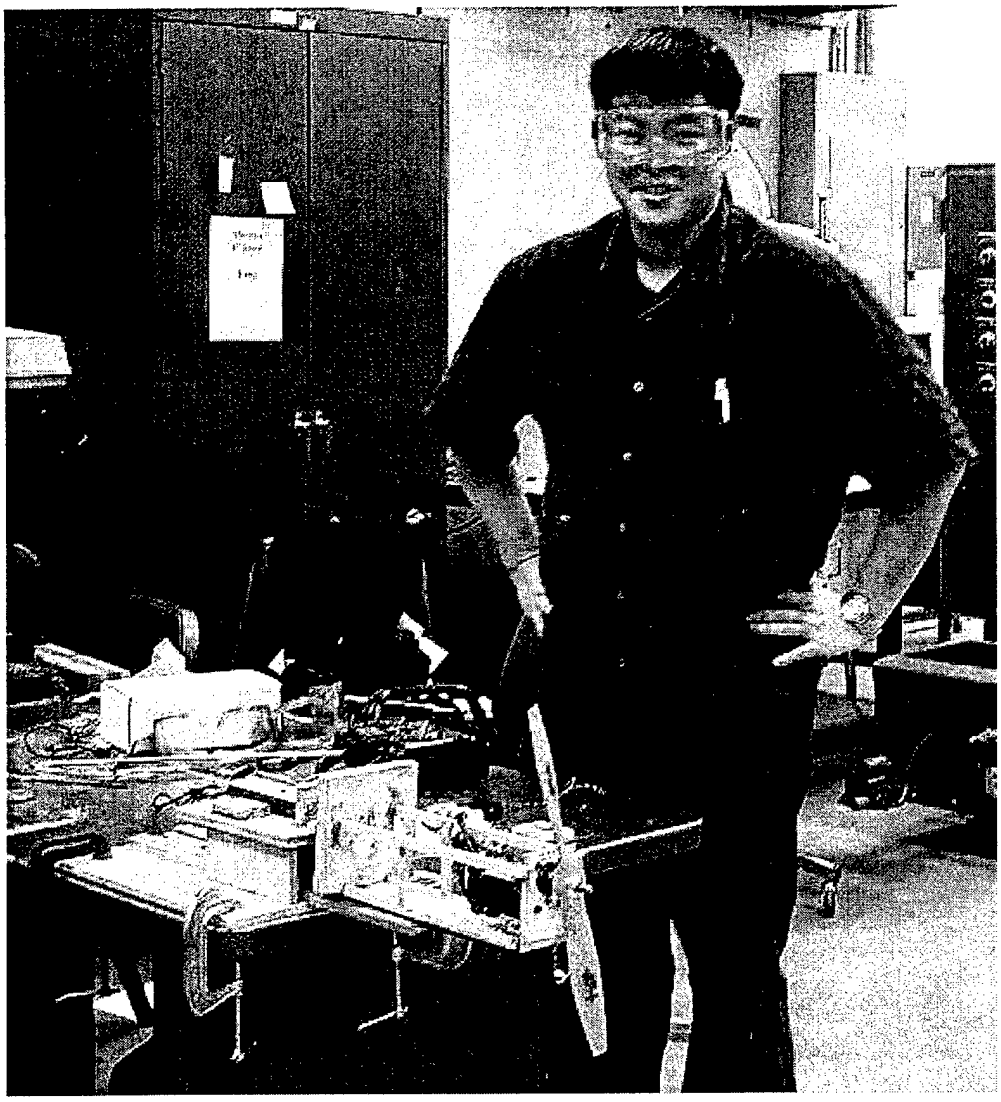


Figure 18 Propulsion Test Setup

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1. McCormick, B.W., *Aerodynamics, Aeronautics, and Flight Mechanics*, John Wiley & Sons, 2nd ed., 1995.

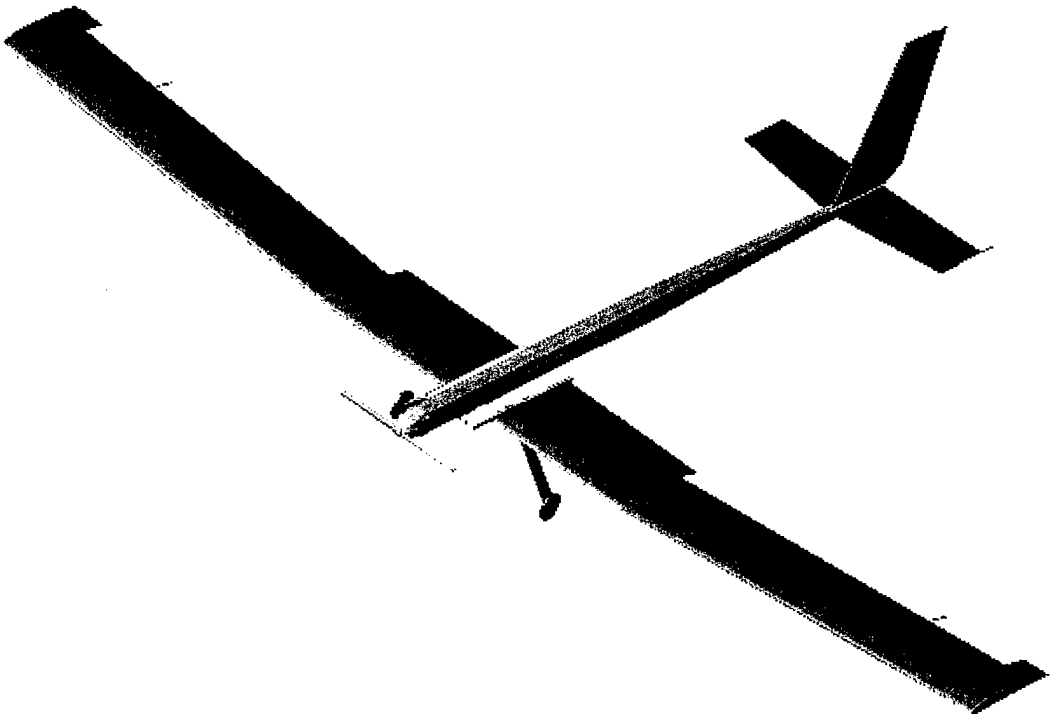
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AEROSPACE PROJECTS LABORATORY

Designing, Manfacturing and Testing of Death From Above for Cessna/ONR Student Design/Build/Fly Competition



California State Polytechnic University, Pomona
Saturday, March 06, 2004

Table of Contents

Section	Page
1.0. Executive Summary	1
1.1. Design Process	1
1.2. Manufacturing and Testing Process	1
2.0. Management Summary	2
2.1. Team Structure	2
2.2. Scheduling	2
3.0. Conceptual Design Phase	3
3.1. Mission	3
3.2. Rules	3
3.3. RAC	4
3.4. General Mission	5
3.4.1. Fire Fight	5
3.4.2. Ferry	5
3.5. Alternative Aircraft Configurations	5
3.5.1. Conventional	5
3.5.2. Biplane	6
3.5.3. Canard	6
3.5.4. Flying wing	6
3.5.5. Final Decision	6
3.6. Wing and Tail Configuration	7
3.7. Aircraft Weight Estimation	7
3.8. Structures	9
3.9. Propulsion	9
4.0. Preliminary Design Phase	13
4.1. Preliminary Wing Design	13
4.1.1. Preliminary Airfoil Selection	14
4.1.2. Preliminary Drag Estimation	16
4.2. Preliminary Tail Design	16
4.3. High Lift Devices	17
4.4. Payload Design	17
4.5. Structures	18
4.6. Propulsion	18
4.7. Preliminary Design Review	23
5.0. Detailed Design Phase	24
5.1. Wing Design	24

Section	Page
5.1.1. Airfoil Selection	25
5.2. Tail Design	26
5.3. Drag Estimation	27
5.4. Fuselage Detailed Design	27
5.5. Propulsion	29
6.0. Manufacturing Plan	33
6.1. Cargo Bay	33
6.2. Wing	34
6.3. Tail	35
6.4. Propulsion	35
6.5. Time Table	35
7.0. Testing Plan	37
7.1. Materials and Techniques	37
7.2. Components	38
7.3. Propulsion	39
7.4. Flight Test	39
7.5. Stability and Control	40
7.6. Aircraft Parameters	41
References	43

1.0. Executive Summary

The purpose of the present work, "Death From Above" (DFA), was to compete and win the AIAA Design/Build/Fly Competition.

1.1. Design Process

A Request for Proposal (RFP) from Cessna/ONR was found. The request requirements called for heavy-lift aircraft that could achieve the speed of a racer. A team was formed to determine the feasibility of a design. After determining a design was feasibility, the team was divided into four groups: aerodynamics, propulsion, research and development, and structures.

Several configurations were considered for the aircraft. The base design chosen was a conventional configuration because it gave stability during the missions for little drag increase. Individual aircraft characteristic were chosen based on trade studies and previous knowledge. The aircraft's first iteration characteristics were a low wing, tail dragger, short and wide fuselage, powered by a single engine. The structure was frame and beams. The aircraft was to be composite of foam, metal, and fiberglass.

As the design process progressed, major changes due to leaps in knowledge occurred. The short and wide fuselage because long and thin to meet payload and stability requirements. Battery placement moved from within the wing to outside the fuselage. The fuselage structure changed to monocoque where the skin was the primary structure. The aircraft was now composite of foam, metal, fiberglass and carbon fiber. The percentage of metal had decreased to less then 5% while the amount of composite material had increase because of weight concerns.

1.2. Manufacturing and Testing Process

Manufacturing of the aircraft began with testing materials because design and actual material usage doesn't always agree. The first objects to be constructed were fuselage structures because little previous knowledge was available to the team. A frame and beam structure was attempted out of aluminum but the material performed poorly during construction. Composite materials such as carbon fiber were used to construct simple shapes to test usability and found to be acceptable. Weigh concerns prompted sample components construction. The weigh predictions from the samples were found to be accurate at assembly.

Manufacturing of components began during the detailed design phase because it was necessary for scheduling. Most of the components were constructed without difficulty except for the wing, which experienced equipment errors. The components were statically tested to ensure they wouldn't fail in flight. No major problems were found.

Flight test were conducted after assembly of test plane. The test plane was not the final plane because some component need to redone to improve aerodynamic characteristic. The flight test proved the plane could meet requirements.

2. Management Summary

2.1. Team Structure

Death From Above design team was composed of six Aerospace Engineering Students from California State Polytechnic University, Pomona(CPP). This team was split into the four groups consisting of aerodynamics, propulsion, research and development, and structures. Due to the small size of the team, a project manager was deemed not necessary. The coordination efforts, scheduling, and budget management were done as a collective effort.

The aerodynamics group was responsible for the plane's configuration and aerodynamic models as well as the wing and tail design. The propulsion group investigated the various engines, battery configurations and proper propellers. The research and development group was responsible for researching construction materials, constructions methods used in the design and testing. The structures group was responsible for the structural configuration, design, analysis, and testing. In addition, stability and control analysis of the plane was a joint effort between aerodynamics and propulsion. Table 2.1 lists all the design personnel in their respective areas of specialty.

Group	Personnel
Aerodynamics	Negar Moinee
Propulsion	Peter A. Feher
Structures	John Rossi Mike Hernandez
Research and Development	Gilberto Rendon Dayton Roberts

Table 2.1. Chart of personnel in there assigned areas of specialty.

2.2. Scheduling

The team held types of meetings. Meetings were held twice a week to discuss important changes that affect the team. Weekly meetings were held in the presents of the faculty advisor in order to discuss the progress and future actions. Monthly meetings were held to determine deadlines for the tasks and milestones accomplished.

3.0. Conceptual Design Phase

Death From Above began with an evaluation of the rules and missions of the competition. Preliminary configurations were put forth to solve the problem. Each configuration was investigated for strengths and weaknesses, which include the score assigned to the configuration by the competition. An initial configuration was chosen. The configuration was further defined before analysis.

3.1. Mission

DBF was composite of one overall criterion, which was to achieve the highest possible score in the competition. The score was broken down into three catalogues: Rated Aircraft Cost(RAC), paper writing, and flight missions. The RAC was a score based on the dimensions of the aircraft such as fuselage volume, number of engines, and number of control surfaces. The RAC was a figure used to convert the aircraft's dimensions to a monetary value. The paper writing score was a measure of documentation skills possessed by the group. The flight mission score was the score obtained for completing the course.

Before flight missions could be undertaken, general requirements were imposed. The aircraft must fit into a box with interior dimensions of 2-ft wide by 1-ft high by 4-ft long, which limited the size of individual components. All payloads must be adequately secured using mechanical means, which doesn't include tape or Velcro. The maximum take-off distance is 150 ft for the aircraft. The maximum flight time was 10 minutes. After meeting these requirements the plane was allowed to attempt flight missions.

The first mission was a fire bomber mission. The aircraft was given the task of carrying 8 liters of water from the take-off point to the downwind leg of the course. To increase the difficulty, the aircraft wasn't allowed to carry more then 4 liters of water at anytime in the flight, which meant the plane, must be able to perform two laps around the course. The water was to be loaded from unmodified 2-liter plastic soda bottles. The dumping hole was to be 1/2-inch inner diameter. Time penalties were given for spillage or dumping at the incorrect time.

The second mission was a ferry mission. The objective was to complete the course in the shortest time possible. The course consisted of 4 complete laps. The aircraft was to remain unmodified from fire bomber mission.

3.2. Rules

The competition had many initial rules and additional rules were added as the design progressed. The initial rules were designed to limit the focus of the possible candidates because the judges were seeking to have a level-starting field, but it left room to vary the design. The first rule limited the aircraft to non-rotary wing and non-lighter-than-air which suggest that fixed wing aircraft were required. The aircraft was to weigh no more then 55 lb, which was an Academy of Model Aeronautics (AMA) rule.

The second rule prevents payload from being carried internal to the wing where the fuselage is considered to be the inner most 9 inches of the semi-span for a blended wing. The rule was ambiguous as to the meaning of payload but infers that it meant only the object being transported.

The next several rules were concerning the propulsion system. The main function was to maintain

safety at the event. The propulsion system was required to be a propeller driven by an electric motor. The propellers needed to be commercially produced and unmodified except for the tip, which could be clipped to change the diameter of the blades. Commercial ducted fan units were allowed. The engines were required to be unmodified Graupner or Astro Flight family of brushed electric motors.

The propulsion system was to be powered by over the counter NiCad batteries. A 5 lb limit was placed on the number of batteries to be used. This allowed different size batteries to be used but kept the upper energy they possessed nearly constant. The batteries were to be protected against the environment with shrink-wrap at contact points and "fully insulated" at disconnects. In addition, a separate battery pack was to power the radio control receiver and servos, to prevent the loss of control during a battery drain. Batteries weren't allowed to be charged during a flight mission. A 40 Amp limit was placed on the circuit powering the propulsion system, which was controlled, by placing a 40 Amp fuse between the positive battery terminal and the motor control. The fuse had to be an ATO or blade style plastic fuse.

The reminding rules were considering legal issues. The plane must have flown a least once prior to the competition to prove it was flight worthy. The group must also provide proof of this flight by an in flight photo at the competition. In addition the rated aircraft cost(RAC) for the aircraft must be completed and sign by advisor at time of inspection.

3.3. RAC

The rated aircraft cost measures the cost of an aircraft based on design and construction. The rated aircraft cost is divided into three catalogues: cost, power, and weight. The cost was based on the number of labor hours predicted by the dimensions, complexity and components used. The power was based on the aircraft's power requirements such as number of engines and number of batteries. The weight was the predicted weight of the aircraft. The RAC was determined by:

$$RAC = \frac{300 * MEW + 1500 * REP + 20 * MFHR}{1000}$$

where MEW was the aircraft empty weight, REP was the rated engine power, and MRHP was the manufacturing man hours.

The rated engine power was determined from:

REP = (1+.25*(# of engines –1)) * Total Battery Weight

where total battery weight was in lb.

The manufacturing man hours were determined from the total Work Breakdown Structure hours(WBS) and multiplying by 5 to convert to manufacturing man hours. Work Breakdown Structure hours for the wings were:

WBS = 10 * Wing Span * Chord * # wings + 5 * total control function multiplier

The wingspan and chord was in feet. The control function multiplier was 1 for ailerons, 1.5 for flaperons, 1 for flaps, and 1 for spoilers. The total control function multiplier was a summation of all control functions multipliers.

Work Breakdown Structure hours for the fuselage was:

$$WBS = 20 * Body\ Length * Width * Height$$

where the body length, width and height are in feet.

Work Breakdown Structure hours for the Empennage was summation of hours based on surfaces used. For every vertical surface entirely inactive, 5 WBS hours were added. For every vertical surface with activity control surfaces, 10 WBS hours were added. For every horizontal surface that was not a wing, 10 WBS hours were added. For a V-tail, 15 WBS hours were added.

Work Breakdown Structure hours for flight systems were 5 hours for each servo added.

3.4. General Mission

General mission requirements are those given to help set the basic requirements of the competition, not relating to a specific mission. The general mission requirements of the competition are that the aircraft must fit with a volume of 1ft by 2ft by 4ft. This volume is in a disassembled or assembled form. The aircraft must also take off within a 150ft, but is allowed to land on the runway at any point and roll off still receiving a score. Teams must choose to fly a mission for each flight, with a maximum allowable mission time of 10 minutes. The payload regardless of any mission must be secured through appropriate mechanical means.

3.4.1. Fire Fight

In the Fire Fight mission teams are required to load their payload containers with two 2-liter bottles of water as the mission recording time has begun, allowing for a maximum of four liters. The aircraft is then required to take off and release the water on the "downward" portion of the lap through an orifice no larger than half inch in diameter. More than one lap can be made to release water, but can only be released on the downward portion of the lap. Aircraft can then land and refill their tanks allowing for another run in which they repeat the same process. Inadvertent payload release will incur a time penalty of three minutes, while excessive spilling during loading of the aircraft will incur a one minute penalty.

3.4.2. Ferry

Aircraft will not be loaded with payload, but required to fly around the course completing four laps. For each lap, 360° turns must be made opposite of the base and final turns on the downwind leg of each laps. Like all missions the aircrafts performance will timed.

3.5. Alternative Aircraft Configurations

Various configurations were initially investigated for the design. Once enough information was obtained on each configuration figures of merit(FOM) charts outlining the areas of significance to the competition were made in order to compare them and find the one which is the most suitable. All aspects of the design were considered in these charts, such as structural design, propulsion systems and aerodynamics. The results of the FOM charts are shown in Table 3.1.

3.5.1. Conventional

The conventional design has a puller propeller on the nose of the airplane. The wing is located behind the engine. The control surfaces, both horizontal and vertical, are located at the end of the tail. The wing can be located at the top or at the bottom of the fuselage. It can have a tricycle landing gear or

a tail dragger configuration. The advantage of this design is its tried and proven concept, and good overall performance.

		Configurations			
Figures of Merit	Weighing Factor	Biplane	Canard	Conventional	Flying Wing
Stability	0.2	1	0	0	-1
Light Structure	0.25	0	1	1	0
Performance	0.15	1	0	1	0
Ease of Construction	0.15	-1	1	1	0
RAC	0.35	-1	0	0	1
Score	-	-0.15	0.4	**0.55**	0.15

Table 3.1. Configuration FOM Decision Matrix

3.5.2. Biplane

The biplane configuration is very similar to the conventional configuration. The difference is that instead a single wing it has two wings located above each other. This configuration provides higher lift capability at lower speeds.

3.5.3. Canard

The canard design differs from the conventional by the relative location of the wing and the control surfaces. The horizontal control surfaces are located at the front of the airplane. The wing is at the aft section and the vertical control surfaces are aft too. The propulsion system could be either a puller propeller at the nose or a pusher at the tail.

3.5.4. Flying wing

The flying wing design has no conventional fuselage. The entire airplane consists of the wing. There are no conventional controlling surfaces. The pros of this design are lower drag and simpler design. The cons are stability problems, and low maneuverability.

3.5.5. Final Decision

The canard at low speeds shows inefficiency due to the loss of lift due to downwash. The flying wing exhibits instability due to its aerodynamic center being placed too forward. It also has higher skin friction drag due to its large wetted area. The biplane due to its additional lifting surface has a higher RAC, which is undesirable for this competition and is generally used on acrobatic planes, which in this competition such capabilities were not necessary.

The V- tail was not an alternative configuration by itself, but a derivative the conventional or biplane design. A conventional design was chosen due to the light structure, ease of construction, low RAC as well as its handling qualities and performance.

3.6. Wing and Tail Configuration

The wing locations considered in this phase of the design were low, middle and high wing attachment. Figures of merit(FOM) crucial to the design and optimization of the performance were chosen in order to compare and contrast these three. The FOM's used in the comparison of these three models can be seen in Table 3.2.

		Wing Location		
Figures of Merit	**Weighing Factor**	**Low**	**Middle**	**High**
Landing Gear Mount	0.3	1	0	-1
Payload Loading	0.3	1	0	0
Stability	0.2	-1	1	0
Ease of Construction	0.2	1	-1	1
Score	-	**0.6**	0	-0.1

Table 3.2. Wing Location FOM Chart

In a low wing the landing gear is mounted on the wing saving a significant amount of weight, structure and complexity. The low wing possesses good crosswind as well as ground handling qualities. It also offers more cushion in the flare for the ground effect. However one of the disadvantages of the low wing is its need for wing fairings due to the interference drag at the wing/fuselage joint.

A high and middle placed wing on the other hand have better handling qualities in the lateral-directional modes; however, the landing gear mounts have less spacing and are mounted too close to each other and the mounting structure for the landing gear is more complex and heavy versus that found in wing mounted landing gear(found in low wings). Due to all the factors above and the rating from the FOM chart, the low wing configuration was chosen.

A similar FOM chart was devised for the tail section, in order to determine which configuration would be the most suitable for this mission. After a comprehensive study it was decided that a conventional tail would be the best suited configuration for the mission requirements. By use of an aft mounted horizontal tail, static longitudinal stability would be achieved and an aft mounted vertical tail would then assure aerodynamic stability and control.

3.7. Aircraft Weight Estimation

The weight of an aircraft is one of the underlying and most crucial factors in its design. The weight determines the lift needed and dictates the values of many other parameters used in the design. Conventionally planes are built around their weight and mission statements; therefore in this stage of the design an initial weight estimate was made. These estimates are shown in table 3.3.

A margin of 3 lbs was given initially in order to compensate for any underestimates of weight; therefore, the total weight of the aircraft was assumed to be 35 lbs.

Component	Weight (lbs)
Engine	2.125
Batteries	5
Cargo (4 liters)	8.815
Wing and Empennage	6
Fuselage	10
Total	**31.94 lbs**

Table 3.3. Initial Weight Estimates

One of the factors, deeply dependant upon the weight and also of high priority to the competition was the take off distance. Figure 3.1. illustrates this relationship and how it dictated our design goals and objectives.

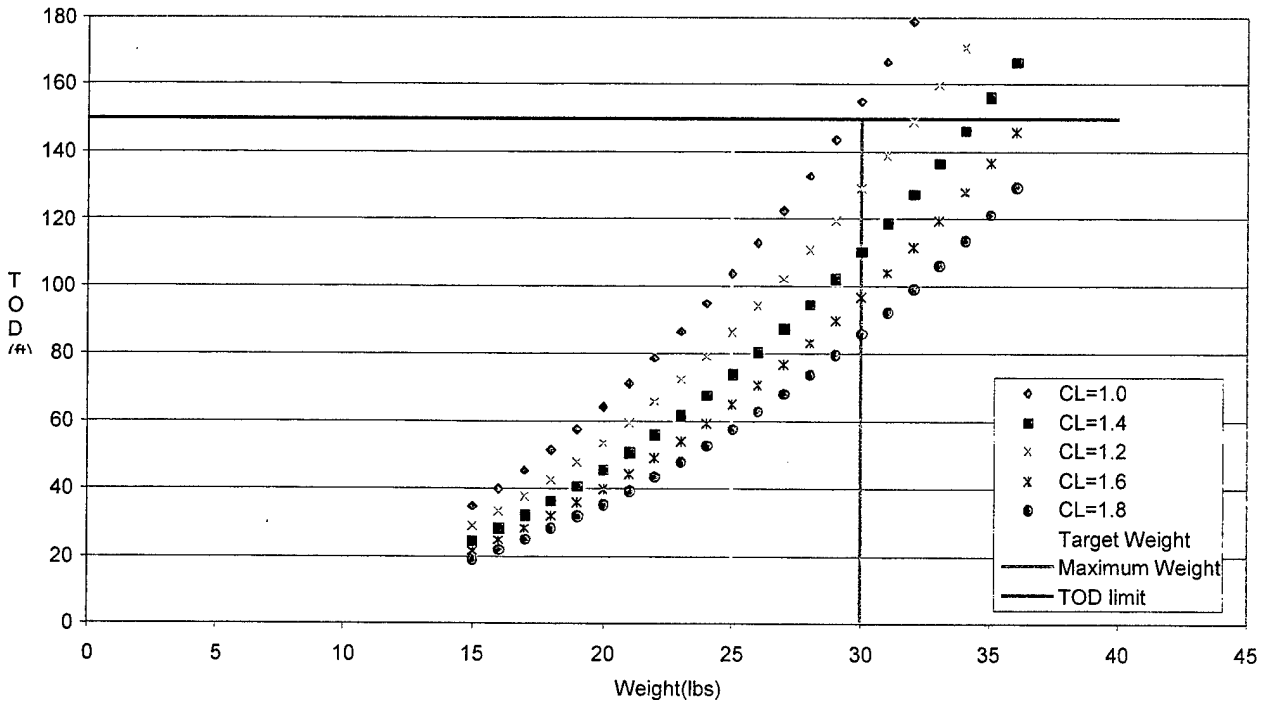


Figure 3.1. Take off Distance versus Weight

This graph was obtained in order to estimate the C_L needed for the wings, to meet take-off requirements. It also illustrated and highlighted the exponential relationship between take off distance and weight for various coefficients of lift. From this graph it was concluded that the maximum weight of the plane must not exceed 30 pounds and the optimum weight that we should design our plane to should be 25 pounds.

3.8. Structures

Preliminary conceptual ideas relating to the structure were based on configurations of interest and material availability. CPP previous year's entry was composed generally of spruce and balsa wood. Several different types of materials were considered. The most common considered for the fuselage included spruce, balsa, aluminum, fiberglass, and carbon fiber. For the wings material considered were the different types of foam, balsa, fiberglass, and carbon fiber.

Conceptual ideas for the construction of the wing were based on the previous year's experience. One conceptual design for the construction of the wing was to make the wing of balsa ribs and cover the wing with plastic film. Building on previous experience it was decided to bag foam cores with fiberglass. This would be less time-consuming and produce a wing less susceptible to damage. This would also make the production of flight control mechanism simpler, since they could be merely cut from the wing. An initial estimate for the weight of the wings for this configuration was 4lbs. This was based on measurements from the previous year's wing scaled by area.

Conceptual design for the fuselage began by looking at the volume required to hold four liters of water. This gave a base volume for the fuselage. Material selection and configuration began with examining the previous year's design. The previous year had a fuselage of balsa and spruce. Several different types of materials were examined including spruce, balsa, aluminum, fiberglass, and carbon fiber. The original conceptual design of the fuselage can be seen in the attached figure, which was composed of aluminum stringers and frames with the entire fuselage covered in fiberglass. This design would later turn out to be too heavy, giving the fuselage a weight of approximately ten pounds. The landing gear was also to be constructed of aluminum. The first weight projection for the aircraft was predicted to be around 35lbs. The prop propulsion was considering had a 26 inch diameter making the landing gear considerably tall, and adding more weight.

The control surfaces of the aircraft were also examined. With a wing made of fiberglass with foam cores, the ailerons, and flaps could be merely cut from the wing and attached to servos. With wiring being run through channels cut next to the spars. The vertical and horizontal tail would be flat plates. The tail was chosen to be flat plates to help facilitate the aircraft fitting within the box. These plates would be bi-directional carbon fiber, with the elevator and rudder attached at quarter chord on each respective surface.

3.9. Propulsion

The propulsion system was limited to propeller driven unmodified electric motors of the Graupner or Astro Flight families. Due to the proximity of the manufacturer and the easier access to engine data, the Astro Flight family of electric motors were chosen. The possible configurations were a larger single engine or two smaller engines.

Direct drive motors give higher speeds and lower thrust, while geared motors can spin larger propellers resulting in greater thrust, but sacrificing speed. The take-off distance for this competition is limited to 150 feet, and directly related to the thrust provided by the propulsion system. To meet the take-

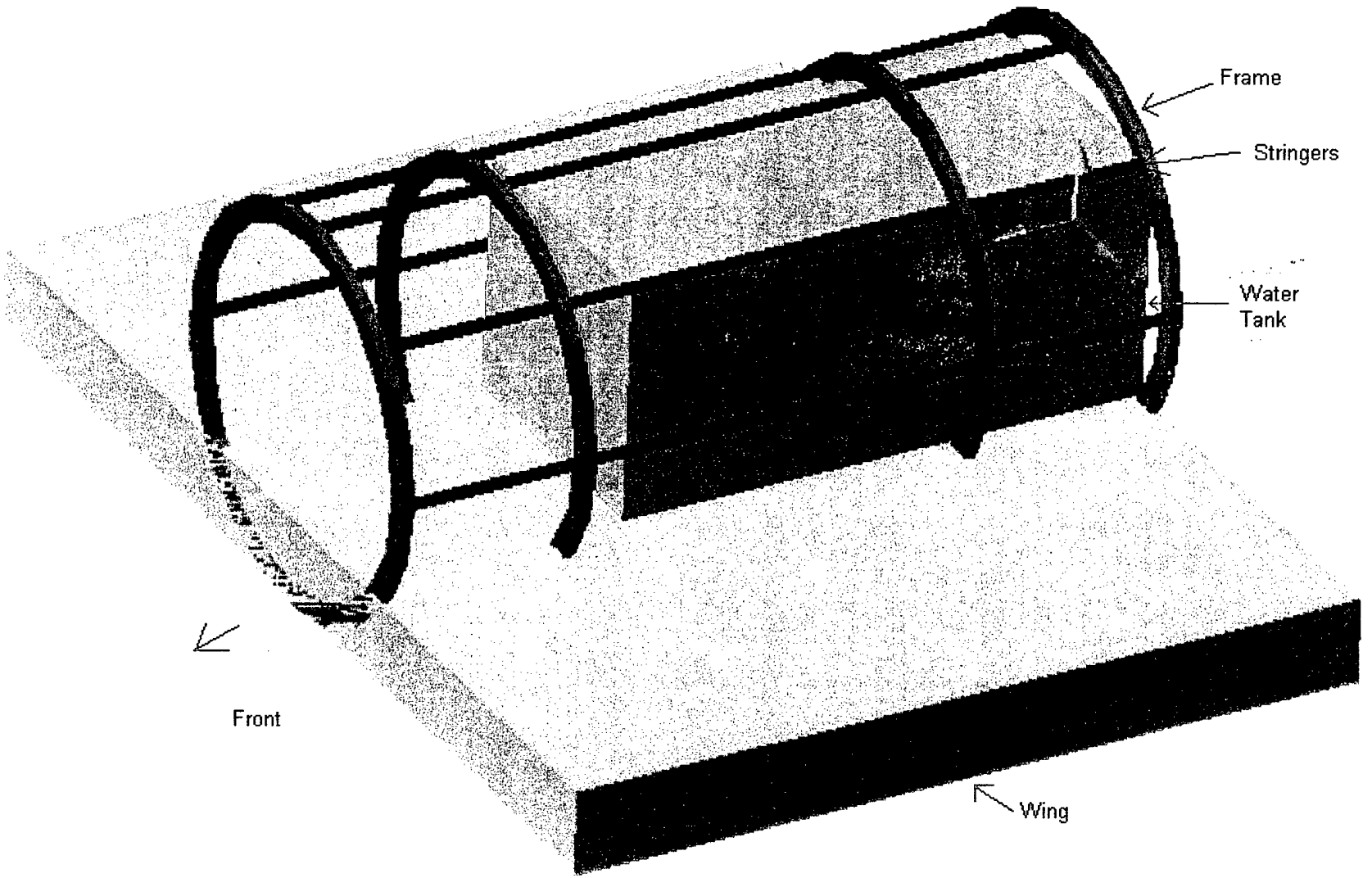


Figure 3.2. Original cargo area design for fuselage.

off constraint high thrust motor / propeller combination is required. This competition is primarily a speed competition, therefore high-speed motor / propeller combination is essential to win. These two contradicting requirements had to be met. The best motor / propeller combination is the one that provides the minimum amount of thrust necessary to make the take-off distance and maximizes the speed in-flight. Figure 3.6. is the general relationship of the expected gearing effect on thrust and speed, keeping the power output of the engine constant (Ref. 3).

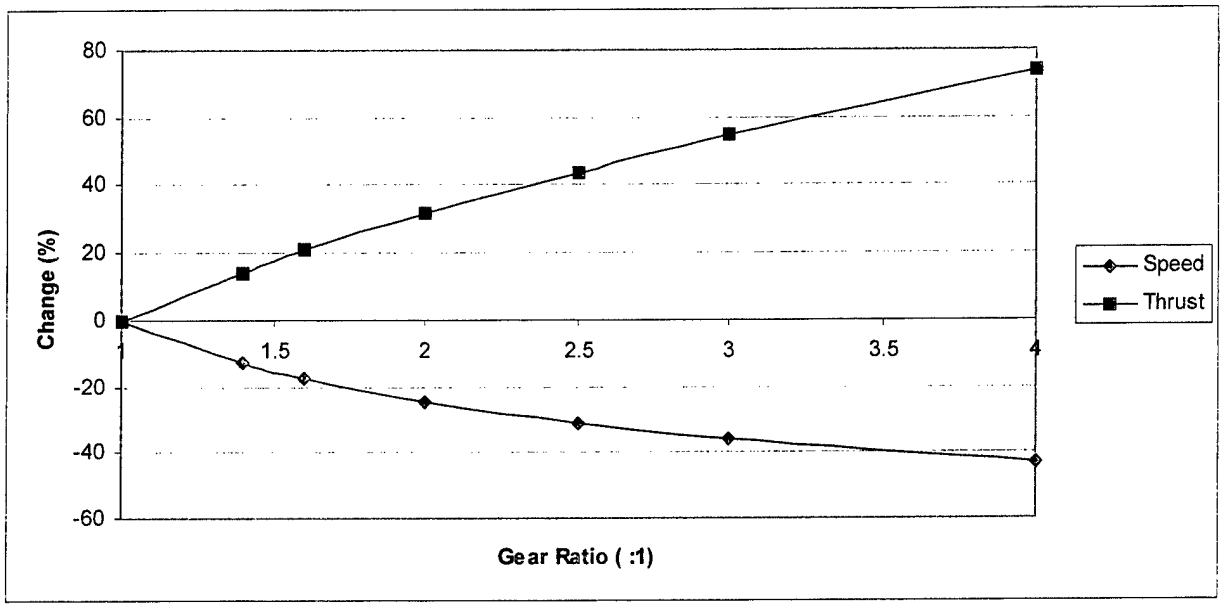


Figure 3.3. Effect of Gear Reduction

The two main constraints on the power available for the motor were the maximum 5 lb battery pack weight and the 40 Amp fuse in the line from the positive battery terminal to the motor controller. Sanyo® brand Ni-Cd battery cells were chosen due to their high performance.

The battery weight limits the capacity of the battery pack and/or the number of cells per pack. The battery pack weight is directly proportional to the number of battery cells which directly proportional to the cell capacity. With higher number of cells the pack voltage, and therefore the current, increases for a fixed motor resistance. The aircraft performance is dependent on the power provided to the motor. Electric power is the product of the electric current and the voltage with the necessary phase correction. The current is limited to 40 Amps by the competition rules, but the voltage is only restricted by the circuit elements' limitations.

The number of cells in the battery pack must be the maximum number of cells that will drive a current of 40 Amperes through the engine, without compromising the capacity of the battery pack. With the increased number of cells the engine power increases, but the total capacity of the pack decreases, keeping the battery pack weight constant. Figure 3.4. shows the possible battery pack voltage versus capacity for a few selected Sanyo® batteries in a series configuration.

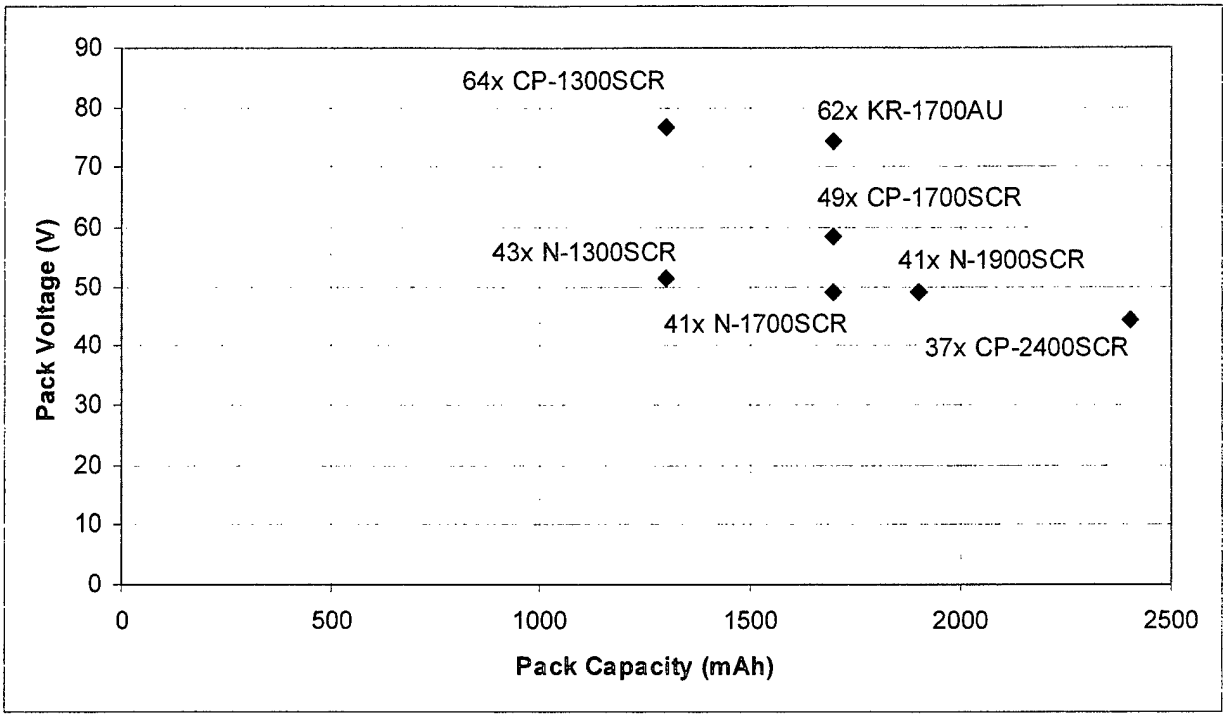


Figure 3.4. Pack Capacity vs. Pack Voltage of a 5 lb battery pack

For a single engine configuration the battery pack is directly connected to the speed controller through the fuse in the positive line.

The dual engine configuration is more complex, but provides redundancy. On the other hand, increasing mission critical parts will increase probability of mission failure. The benefits and the losses balance each other out, making it an equally rational design. For the maximum performance the two motors will have separate battery packs, each limited by the 40 Amp fuse. With this the total current delivered to the motors will be 80 Amps together. The two propellers can be either facing forward or can be arranged as a puller-pusher combination.

One motor driving two propeller configuration is not feasible. A belt system would result in a complex and less powerful propulsion system; therefore, it is not investigated further.

4.0. Preliminary Design Phase

In this phase the performance spreadsheet was refined in order to obtain the design parameters needed. The wing and fuselage were designed around those given values and optimized at this stage. The wing and fuselage sizing complimented by stability and control analysis determined the tail sizing. These parameters were then given to the structures group for structural analysis and testing. Lastly the propulsion group conducted further trade studies in order to determine which battery and engine combinations would be best suited for the plane.

4.1. Preliminary Wing Design

Preliminary wing design entailed the determination of wing area, sweep angle, aspect ratio and taper ratio that would meet the mission requirements. The initial parameter that was needed for the wing design was the wing area. This area was found using the performance spreadsheet. The limiting factors and parameters used in order to calculate the wing area were the take off distance, required lift at takeoff and sizing limitation due to the box. Figure 4.1. illustrates one of the design parameters used for the initial sizing of the plane.

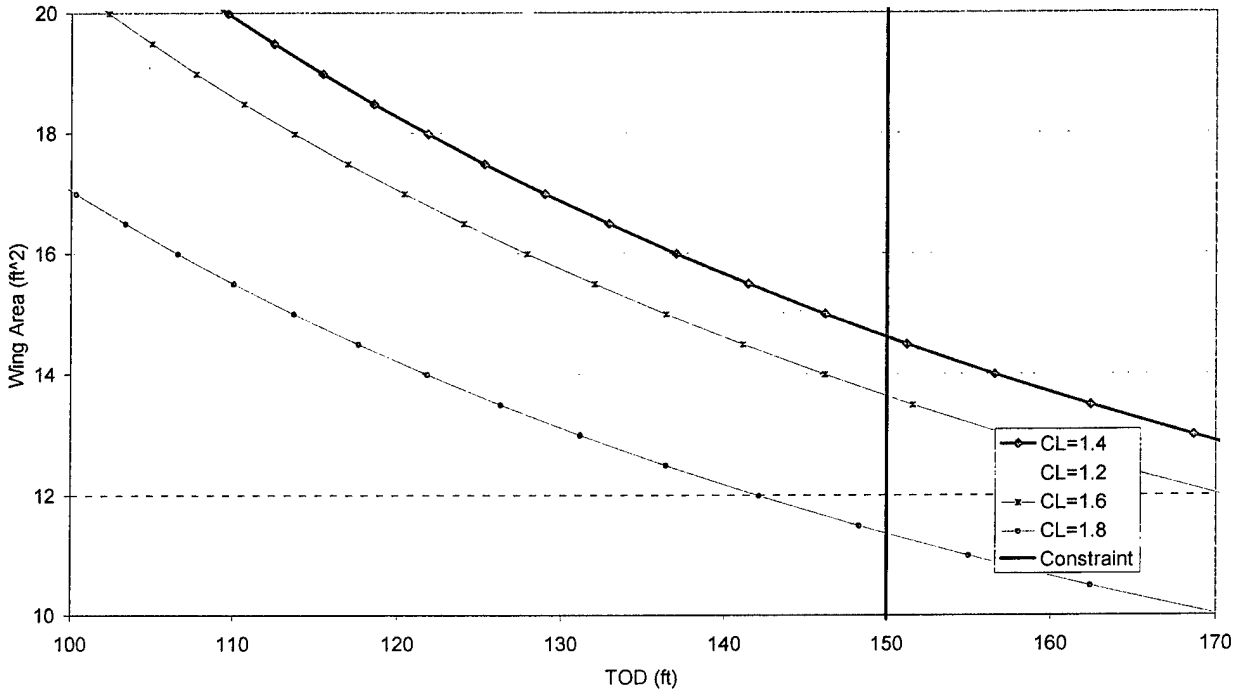


Figure 4.1. Wing area versus the take off distance

This figure illustrates the ranges of wing area at numerous values for C_L ,which would meet the take off distance requirement in the paper (Constraint is illustrated in Figure 4.1.1). After a comparison of the various parameters a wing area of 15.75 square feet was selected which would require a C_L of 1.4 to

meet take off requirements. A chord of 1.5 feet and a span of 10 feet were selected in order to obtain the optimum aspect ratio of 7.7.

4.1.1. Preliminary Airfoil Selection

The airfoil selection was the next step in the design. The performance spreadsheet was used to obtain a range for the coefficient of lift and drag, needed to meet the design goals. The six airfoils evaluated were the Clark-Y, S1223, DAE 51, SD6060 and the E423.

Computational methods aided by the program XFOIL were used in order to determine the airfoil parameters. These parameters were then compared against experimental results. The lift predicted from XFOIL showed fairly good agreement with the experimental data for low Reynolds numbers tested between ranges of 100,000 to 300,000(Ref. 9). The Reynolds number range of interest for this plane however is above that of the tested range, yielding similar and in some cases better results. Separation bubble effects are not predominant due to the fact that our range is outside of laminar range and in the realm of transition into turbulent flow (Re=506,500). XFOIL did however over predict the maximum lift coefficient. At some points there was missing data due to the fact that the solutions did not converge in the computations. The inaccuracy of the wind tunnel model could have also contributed to the discrepancies.

In low Reynolds number flows, the discrepancies in drag data is far more visible than that of lift data, due to separation bubbles that are not represented in XFOIL. As the Reynolds number approaches the transition Reynolds number this effect can be neglected which yields fairly similar results for the computational versus experimental drag data.

It can be concluded from the discussions above that XFOIL yields airfoil data that are fairly in agreement to experimental data, especially for the Reynolds number range of interest (506,500); therefore it was used to calculate all airfoil data.

The maximum coefficient of lift found from the $C_{L\alpha}$ curve and the corresponding coefficient of drag obtained from the airfoil drag polars were used in order to determine which airfoil was the best suited for this aircraft. A range of angle of attack was then calculated and chosen for the given airfoils. Sections of the $C_{L\alpha}$ curve and drag polars of various airfoils investigated in this stage are shown in figure 4.2 and 4.3 below:

Due to the high coefficient of lift range needed in order to meet the take off requirements and the desirable low drag, the Eppler(E423) was selected. This airfoil was a heavy-lift cargo model plane airfoil which has been shown to operate well at a Reynolds number of 500,000 (through wind tunnel testing) which was with in our given takeoff range for the Reynolds number. The angle of attack chosen, was eight degree which would result in a C_L=1.5. Another design criterion it met was the ease in construction it would offer due to its thickness. The Eppler 423 airfoil in comparison to the Selig S1223 offered less lift; however the S1223 also had a significantly higher drag coefficient and it had a noticeably thinner airfoil section, which would make it harder to construct.

A taper ratio was incorporated into the wings in order to approximate elliptical loading in the wings. Elliptical loading was desirable due to the fact that it reduced induced drag and according to Prandtl wing

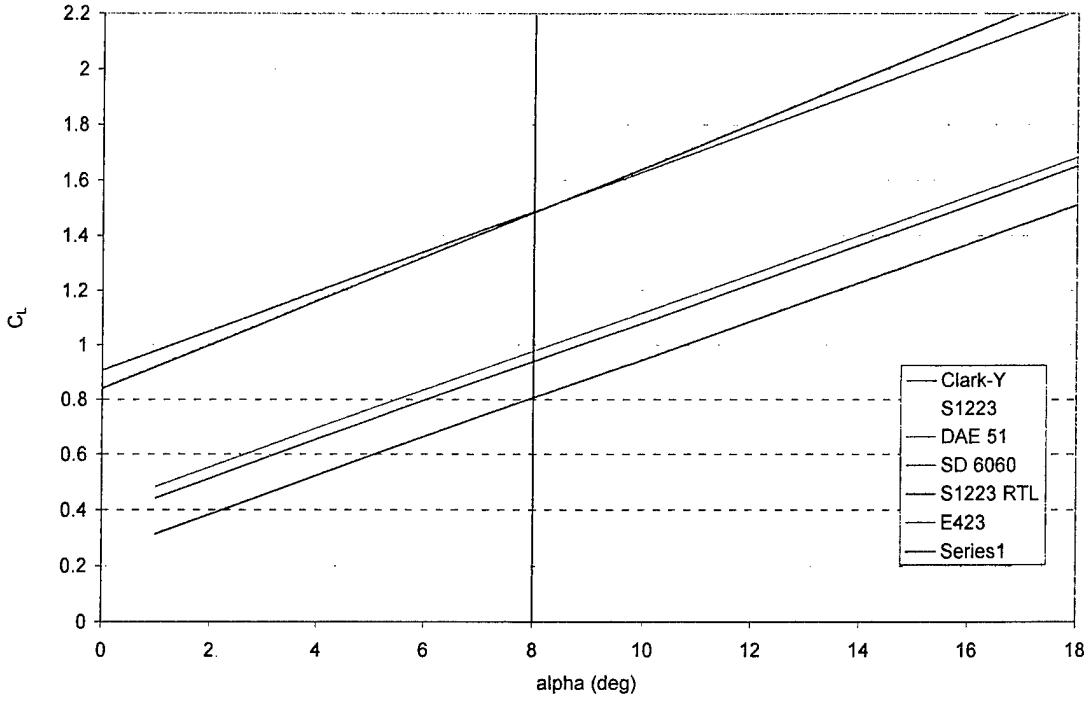


Figure 4.2 $C_{L\alpha}$ curve of the preliminary selected airfoils

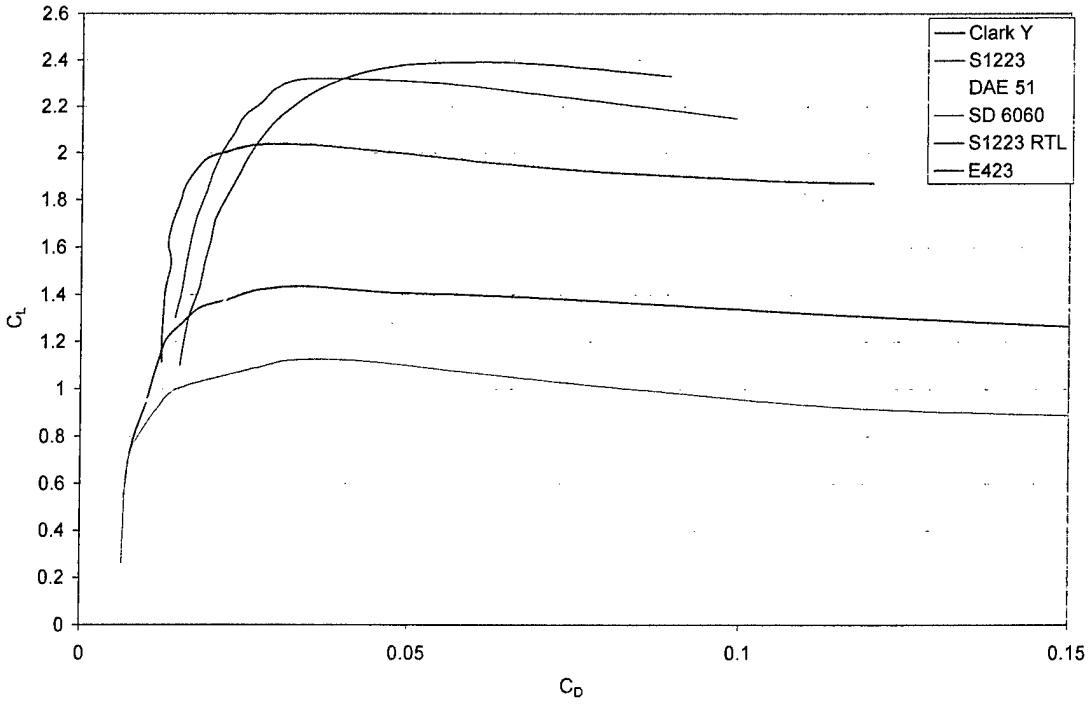


Figure 4.3. Drag polars of the preliminary selected airfoils

theory minimum induced drag is achieved when the lift is distributed in an elliptical fashion. Most untwisted and unswept wings have taper ratios ranging between 0.4-0.5; but for our configuration a taper ratio of 0.625 was chosen due to the fact that it came the closes to an elliptical shaped wing (similar to that of the Supermarine Spitfire) and it reduced less wing area, allowing for more lift capability. The length of the tapered section was 1.5 feet on each end.

Dihedral was placed into the wing in order to increase lateral stability. Two and a half degrees of dihedral was put at the outer wing panels (1.5 feet from the root, illustrated in the drawing package).

4.1.2. Preliminary Drag Estimation

Drag on the aircraft consisted of induced drag and parasite (profile) drag. The actual values for the wing profile drag were obtained from the drag polar (Figure 4.3). The estimation for aircraft component drag are shown in Table 4.1.

A/C Component	$D_{parasite}$
Wing	.007
Fuselage	.004
Landing Gear	.004
Total Parasite Drag	**.015**

Table 4.1. Preliminary parasite drag.

4.2. Preliminary Tail Design

The horizontal tail was designed to provide static and longitudinal stability while the vertical tail was to assure directional stability over the entire design range of the center of gravity(cg). The tail sizing was determined through volume ratios, mission and stability requirements. The rules dictated a horizontal tail span less than a quarter of the wingspan. This requirement and the box limitation limited the span of the horizontal tail to a maximum value of 2.75 feet. The optimum horizontal and vertical tail volume ratios were found through past data and experience and used in the performance spreadsheet to determine the tail sizing. The preliminary horizontal and vertical tail sizing can be found in table 4.2.

	Vertical Tail	Horizontal Tail
Root Chord	1.0 ft	1.2 ft
Span	2.5 ft	2.75 ft
Taper ratio	0.6	0.834
Volume Coefficient	0.30	0.55
Aspect Ratio	2.25	2.67

Table 4.2. Preliminary Tail sizing

4.3. High Lift Devices

Trailing edge flaps were chosen as the high lift device at this stage, due to their simplicity, ease of construction and control. High lift devices were needed due to the fact that the airfoil would not give enough lift in order to be able to meet the take off distance requirements. Twenty five percent of the chord was chosen for the flap length.

4.4. Payload Design

The payload was the primary focus of investigation because the aircraft was designed around it. The payload had been assumed to be 4 liters of water because the rules stated 4 liters was maximum capacity for the aircraft. The assumption failed to account for the speed of the aircraft during the downwind leg and the rate at which the water would leave the aircraft.

It was necessary to conduct several tests to determine the rate that water would move through the 1/2-inch opening. The first test conducted was to determine the baseline time for water leaving an unmodified bottle. The bottle was placed vertical and allowed to drain. The time was 22 seconds. The top of the bottle was cut open to allow air to enter and the time reduced to 16 seconds. The results showed that the water in an aircraft flying level could be done at a rate of 8 seconds per liter however it also should that hydrostatic pressure was required to achieve this.

A trade study was conducted to estimate the amount of water the aircraft should carry. The traded studied showed an aircraft flying at a top speed of 70 mph could drop the 4 liters with the downwind leg because the turn-induced acceleration which increased flow.

The problem with hydrostatic pressure in a model aircraft was a lack of space in the fuselage. The first thought was to design a tall column to maximize hydrostatic pressure but this increase the drag because the total area normal to the flow increased and the total surface area increased. The rated aircraft cost would have increased. Increasing the height of the fuselage was found to be dissatisfactory. To achieve the same the hydrostatic pressure, an alternative configuration was necessary. Reexamining the fuselage, the length was found to be an underused dimension. The length of aircraft was significantly larger because of the requirements needed to stabilize the aircraft. The horizontal column would take less room because the volume would be distributed along the length of the fuselage.

The effect of having a horizontal liquid payload was large amounts of center of gravity travel because the liquid exited aft of the center of gravity. The center of gravity travel needed to be minimized while still keeping the water distributed along the length of the fuselage to maximize hydrostatic pressure. A trade study was performed to determine the optimal distribution of water along the fuselage. The trade study showed that hydrostatic pressure was maximized when the water was distributed over a great distance because the angle of attack required was small, but the center of gravity travel was over 1 foot. A center of gravity travel of over 3 inches was unacceptable. The angle of attack required achieving the time 1 liter every 8 seconds desired was 5 degrees. It was noticed that the cross-section had a diameter of 1.5 inches, which was less, then the engine required. The engine required a 3 inches to fit in the nose of the fuselage. The fuselage diameter was increased to 3 inches to meet engine requirements. As a result the

tank for the payload was reduced to the length of 3 feet and the center of gravity travel was reduced to 2 inches. The angle of attack was increased to 10 degree, which was near the boundary of stall.

4.5. Structures

The weight estimate for the structure was determined to be too high during the PDR. Carbon fiber, which is more lightweight and stronger than aluminum, was chosen for construction of the fuselage. This material was donated to our school but required the construction of an oven to bake the carbon fiber. The original design was optimized for carrying a heavy load, but after the PDR the design changed for optimization of speed, leading to the complete redesign of the current configuration.

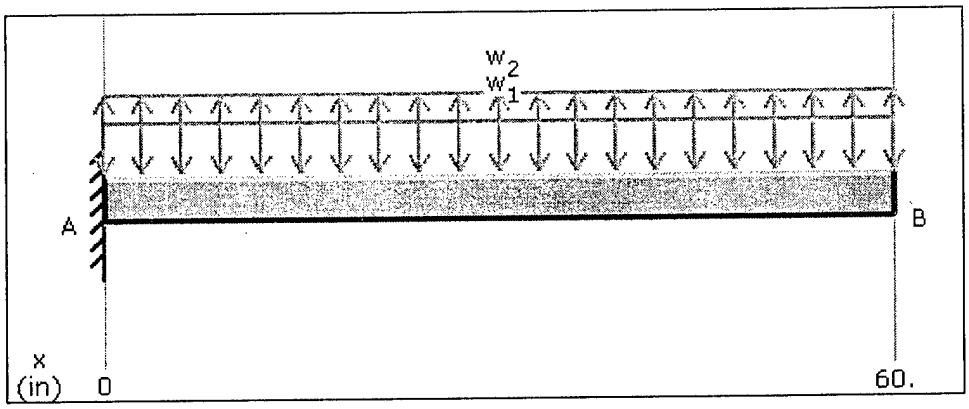
To help optimize for speed the outer shape of the fuselage was changed from a short bulky cargo area to a long skinny pipe like fuselage. This was also done to help facilitate the unloading of the cargo, so that at an angle of attack the long slender fuselage would impart more hydrostatic pressure. This would also help with stability as it would increase tail volume. To help facilitate for speed, the fuselage would now also carry less than the 4 liters it was originally designed to. The new tubular design would be five feet in length and three inches in diameter. The entire fuselage would be constructed of layered unidirectional carbon fiber overlapped at different angles.

After the PDR the shape of the wing was changed, however the structural design was not radically altered. The original design had a spar placed at quarter chord, and a second smaller spar placed at three quarters of the cord, this was help attach the wing to the fuselage. When optimizing for speed, the loads imparted on the structure during the turn made on the course would be increased. As a result the second spar in was increased in size match the quarter chord spar. The quarter chord spar spans the length of the wing, while the second spar only goes half the span. This would allow the wings to handle the turn in the course at higher speeds. The other significant change to the structure of the wing was made due to a selection of a different airfoil. The new airfoil was mush thinner than the previous one, while the spars and control mechanism still fit, the hard points embedded in the wing could no longer be embedded all the way in. As a result the bottom hard points were changed to a single a large plate to distribute the load to a much larger area.

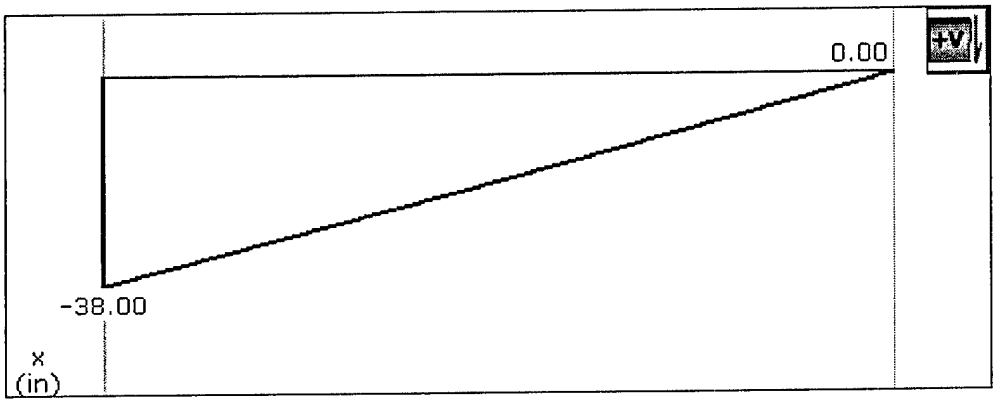
The landing gear design changed in size after the PDR. The reduction in weight from the original 35 lb to 25 lb, reduced the needed strength of the landing gear. Propulsions decision to use a smaller prop also reduced the needed height of the landing gear, further reducing the weight of the landing gear. The wing would be attached by being sandwiched between the fuselage and the landing gear. This would make only one total attachment necessary for the three components.

4.6. Propulsion

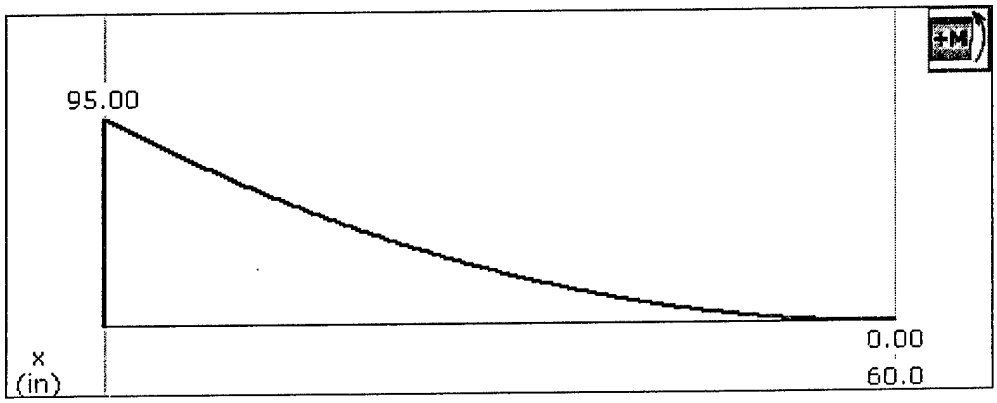
Take-off was identified as the most crucial mission phase. The take-off distance is limited to 150 feet. The propulsion system must provide enough take-off thrust to make the take-off distance. With increased flight speed the thrust developed by a propeller is decreasing. Take-off thrust was defended as the average thrust provided by the engine between zero and take-off speed. The aerodynamics team requested a minimum take-off thrust of 10 pounds. The take-off speed was 30 mph



4.4. Wing Loading over half span at maximum velocity of 120 ft/sec.



4.5. Shear Diagram in lbs at maximum velocity of 120 ft/sec.



4.6. Bending Moment Diagram in ft-lbs at maximum velocity of 120 ft/sec.

and the stall speed was 25 mph. The battery pack must contain enough charge to be able to power the propulsion for the entire mission. In flight the batteries hold slightly longer than on the bench. The trade studies were constrained on configurations that give at least three minutes flight time on the bench.

The single engine configuration is consisted of a single Astro Flight Cobalt 90 motor. The engine can be either direct drive or geared. The geared Cobalt 90 comes with the Astro Super Box, with a gearing ratio of 2.75 to 1. The manufacturer recommends the use of 36 to 40 Ni-Cd cells for this engine. The possible battery packs for these configurations could either use 36 to 40 Sanyo N-1900SCR cells. These cells hold 1900 mAh charge. This is the low endurance, high voltage, high performance configuration. The high capacity high endurance configuration is 36 Sanyo® CP-2400SCR cells. These cells hold larger amount of charge, 2400 mAh, that comes with a weight penalty per cell.

The dual engine configuration is assembled from two Astro Flight Cobalt 60 motors. The engines can be either direct drive or geared. The two engines must have the same configuration. Either both are direct drive or both are geared. The geared Cobalt 60 comes with the Astro Super Box, with a gearing ratio of 2.75 to 1. The manufacturer recommends the use of 30 to 36 Ni-Cd cells for this motor. The possible battery packs for this configuration could be either assembled from 30 to 33 CP-1300SCR cells. This is the high performance configuration. These cells can hold a store a charge of 1300 mAh. The high endurance configuration consists of 30 to 33 KR-1700AU cells. These cells hold a larger charge, 1700 mAh. These cells have the same voltage ratings as the CP-1300SCR, but have higher internal resistance. This results a smaller overall battery pack voltage. Table 4.9. contains the summary of these configurations.

	Single Engine Configuration		Dual Engine Configuration	
	36 - 40 N-1900SCR	36 CP-2400 SCR	30 - 33 CP-1300SCR	30 - 33 KR-1700AU33
Pack Capacity (mAh)	1900	2400	1300	1700
Pack Voltage (V)	43.2 - 48.0	43.2	36.0 - 39.6	36.0 - 39.6

Table 4.3. Battery Pack Configurations

To calculate the expected performance of these configurations the MotoCalc (Ref. 1) software was used. The results were used to create power available curves for the evaluation of the configurations.

Results not satisfying the requirements, more than 40 Amp current, less than 3 minutes run time, and inadequate thrust, were discarded.

To determine the number of batteries to use, a trade study was conducted. The number of N-1900SCR cells for the Cobalt 90 engine was varied from 36 to 40, holding all other parameters constant. The extra 4 batteries increased the aircraft weight by 1%. The gain was 15-20% increase in static thrust, and 5-8 % increase in pitch speed. This gave the conclusion, that the maximum possible amount of batteries should be used always. This eliminated the CP-2400SCR battery packs as a possible configuration.

The power required for level flight was constructed using Equation [6.29] of Anderson (Ref. 2).

$$P_r = \tfrac{1}{2}\rho_\infty V_\infty^3 S C_{D,0} + \frac{W^2/\left(\tfrac{1}{2}\rho_\infty V_\infty S\right)}{\pi e A R}$$

The power available curve was constructed from the thrust obtained from MotoCalc (Ref. 1) software. A sweep of different propellers was performed and the best results can be found on Figure 4.7.

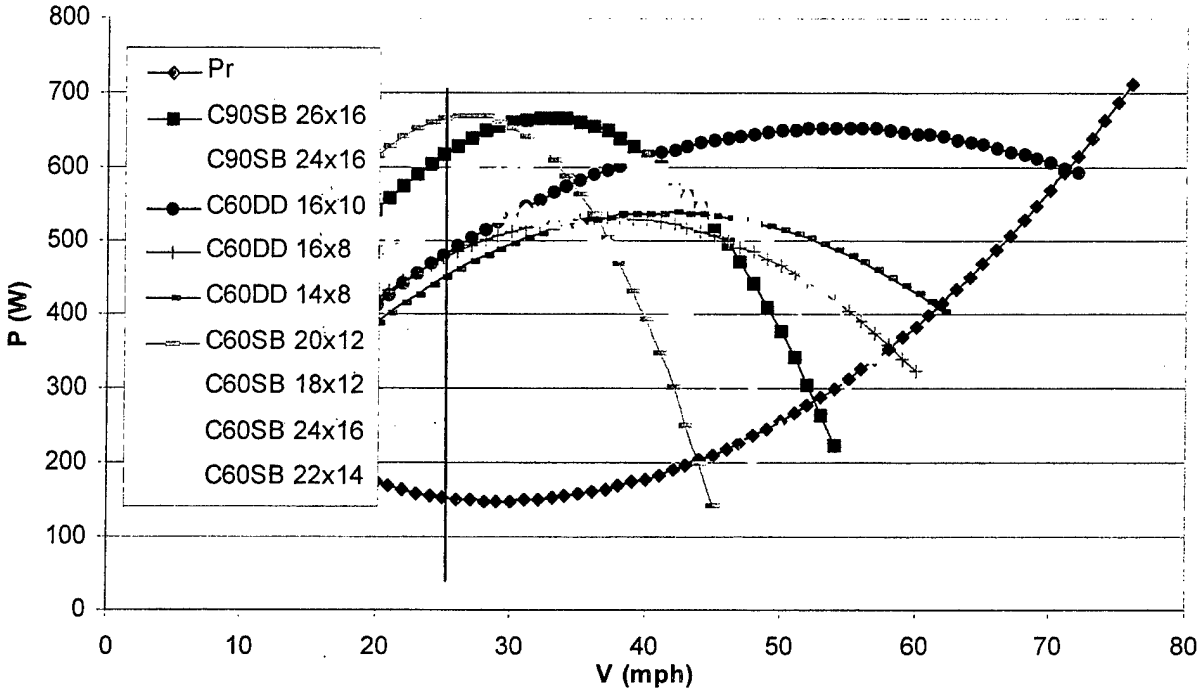


Figure 4.7. Power available and power required for various engine / propeller combinations.

The Astro Cobalt 90 direct drive engine had a too low take-off thrust. The dual Astro Cobalt 60 direct drive engine with 33 CP-1300SCR cells had a too short flight time. After the elimination of these two configurations and the evaluation of the chart three different designs were choosen for further investigation: the Astro Cobalt 90 with a superbox using a 26x16 propeller, the Astro Cobalt 60 direct drive motors with 16x10 propellers, and the dual Astro Cobalt 60 super box with 24x16 propellers. The excess power was used to evaluate the different configurations. Figure 4.8. shows the excess power curves for these three combinations.

The Astro Cobalt 90 with the superbox is the hight thrust, but low speed configuration. The direct drive Astro Cobalt 60 is the high speed low thrust combination. The Astro Cobalt 60 with the super box is a balanced configuration. Table 4.4. contains the performance summary of these configurations.

To evaluate the best design a figures of merit chart was constructed. Besides performance RAC, and complexity were taken into account, as well as performance. The Astro Cobalt 90 was taken as standad and the other tow configurations were compared to it. The figures of merit can be found in Table 4.5.

Based on this the final configuration was the Astro Cobalt 90 super box engine with a 26x14 propeller. The dual Astro Cobalt 60 engine with the superbox was a very weak design compared to the others. The dual engine direct drive Astro Cobalt 60 was only slightly worse than the Astro Cobalt 90. The

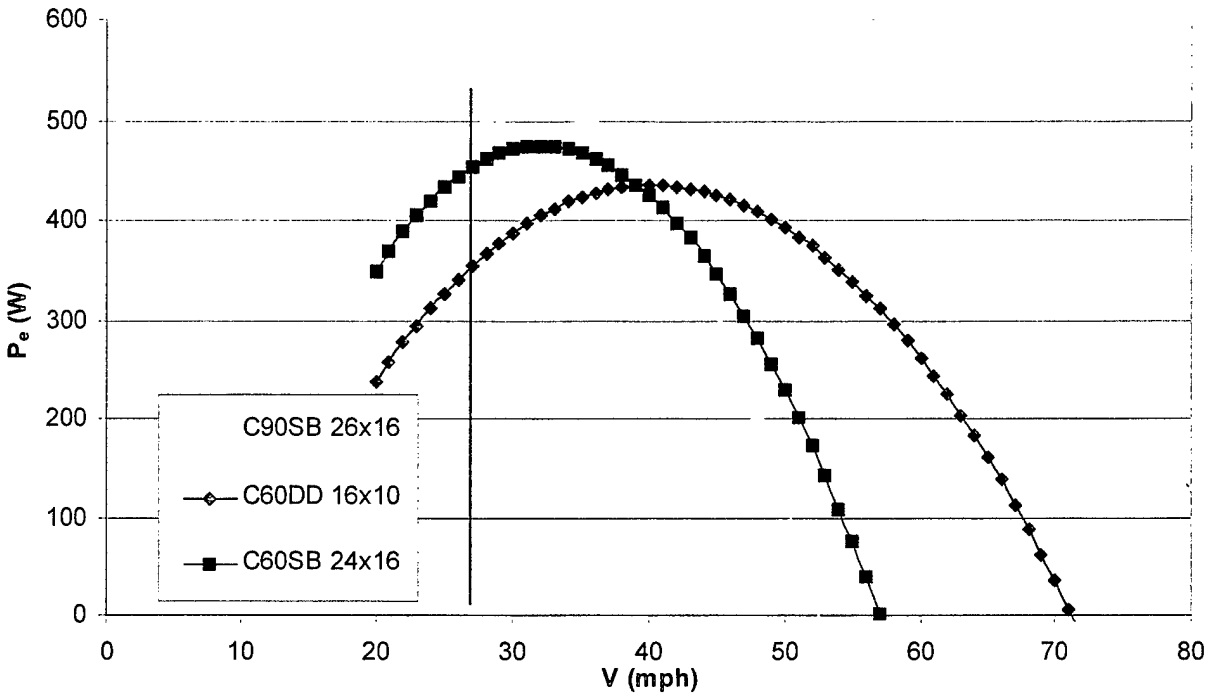


Figure 4.8. Excess Power.

	Take-off thrust (lbf)	Max Speed (mph)	Max Excess Power (W)	Curve Area (W-mph)
1 Cobalt 90 Super Box, 26x16 propeller	14.2	52.5	516.2	10560
2 Cobalt 60 direct drive, 16x10 propeller	11.0	71.5	466.6	15180
2 Cobalt 60 Super Box, 24x16 propeller	13.0	55.5	435.9	11170

Table 4.4. Expected Performance

		Weight	C90SB 26x14	C60DD 16x10	C60SB 24x16
Figures of Merit	Max Speed	0.5	1	1.37	1.04
	Max P_e	0.3	1	0.845	0.921
	Take-off Thrust	0.2	1	0.774	0.996
	A/C Weight	-0.3	1	1.02	1.03
	A/C RAC	-0.3	1	1.02	1.02
	Complexity	-0.2	1	2	2
	A/C Cost	-0.2	1	1.09	1.15
	Final Score		**0**	**-0.137**	**-0.25**

Table 4.5. Propulsion system evaluation.

speed advantage would give the direct drive Astro Cobalt 60 about 9 seconds advantage around the course, but the margin to meet the ten pounds take-off thrust is only ten percent. Since take-off is the most critical mission phase a margin of safety is more important to implement for this.

The selected propulsion configuration is a 40 Sanyo N-1900SCR cell battery pack, Astro Cobalt 90 motor with super box, Bolly 26x16 propeller. Table 4.6. contains the summary of the propulsion system specifications.

Battery Pack Weight (lb)	Battery Pack Voltage (V)	Motor Current (A)	Max Level Flight Speed (mph)	Estimated Flight Time (min)	Static Thrust to Weight	Max Speed to Stall Speed
4.9	48.0	38.9	52	4 - 6	0.49	2.1

Table 4.6. Propulsion system characteristics and expected performance

The battery weight is within the requirements. The estimated flight time is well above the requried 3 minutes. There is a significant flight time margin. The static thrust to weight ratio is relatively high. This aircraft will be able to take-off well under the required take-off distance and will be able to easily climb. The maximum speed to stall speed is less than the recommended 2.5. This means, that the plane will not be able to manuver too well and requires an experienced pilot to fly.

4.7. Preliminary Design Review

When the preliminary design phase had came to an end a review was set up with the sponsor, faculty advisor and peers. In this review it was found that the weight estimate was too conservative and that conservative estimate caused all of the parameters in the design to be overestimated. Consequently, the contest rules and missions were reviewed in order to determine the crucial parameters to the score and optimize the design in accordance to them. The aircraft had been configured for a heavy lift competition, while it should have been closer to a racer configuration. It was then that the second iteration began.

5.0. Detailed Design Phase

The second iteration of the design began here. The wing sizing, tail sizing, engines, batteries and structural configuration were all redesigned and is some cases completely reconfigured.

5.1. Wing Design

The initial parameter that needed to be re-determined was the wing area. The wing area is of great importance due to the fact that it is a determining factor for mission performance parameters such as the take off and landing field length, landing approach speed and initial cruise altitude capability.
The wing area was selected by use of a preliminary wing area selection graph (wing area vs. gross weight). The wing loading range was selected to be between 25-45 oz/ft^2 based on current trends and past experience. The corresponding wing areas for the wing loading range were then estimated and plotted as shown in Figure 5.1.

Given the wing loading the wing area needed in order to meet take off, cruise and landing requirements was calculated and plotted on the wing area selection chart (Figure 5.1.). Using these values it was found that there is no exact value of the wing area that could be calculated in order to meet all the requirements of the design mission, therefore by using past experience and data from current model aircraft a wing area of 9 square feet was selected as a compromise.

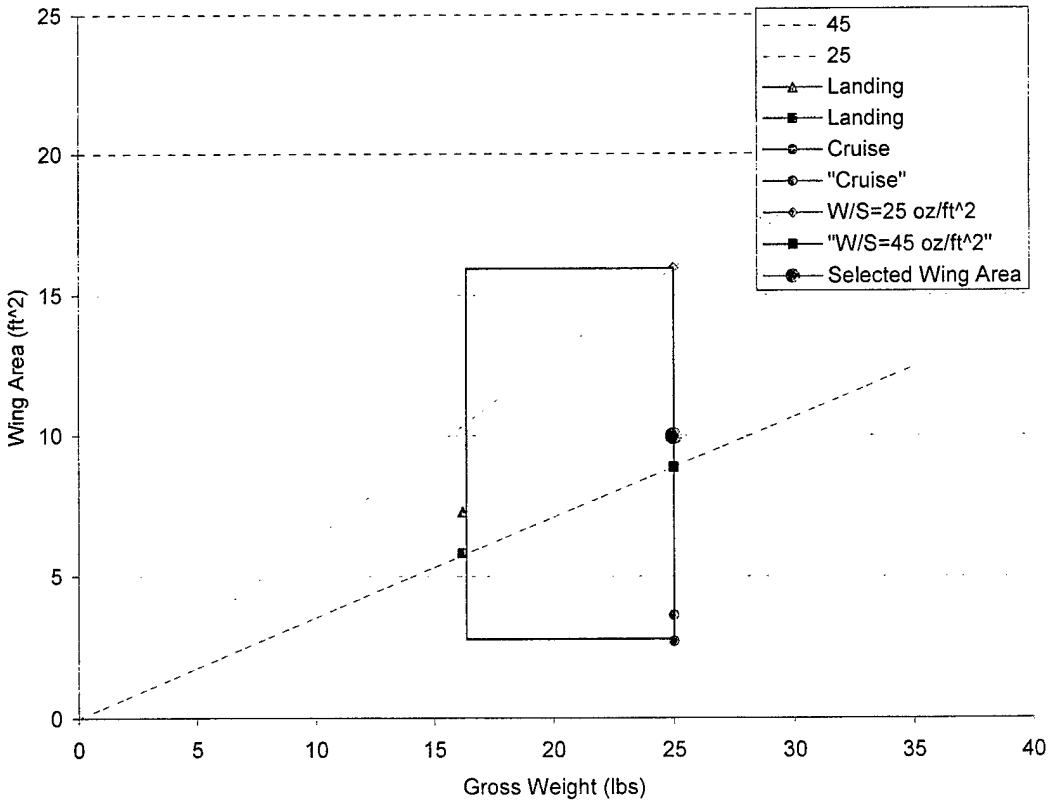


Figure 5.1. Preliminary Wing Area Selection.

Additional considerations also needed to be made for increases in takeoff gross weight and payload weight as the design matured; therefore a 10% increase in the wing area was chosen in order to accommodate the design changes as well as any additional structural weight that might be added in the construction phase. This resulted in a final wing area of 10 square feet. The preliminary wing area selection graph is shown in Figure 5.1.

In order to increase the aspect ratio to reduce induced drag and obtain better handling qualities a span of 10 feet and a chord of 1 foot was selected.

5.1.1. Airfoil Selection

In the detailed design phase, since the weight estimation had changed radically since the preliminary design review a new airfoil had to be selected. The new value for $C_{L\,(L/D)\,max}$ was found to be 0.8; therefore that value and the maximum C_L was used as an airfoil selection criterion. These airfoils included: RG15, NACA 2414, NACA 2415, M6, SD6060, SD6080 and the SD7037. These airfoils have high thickness ratios, low drag coefficients and reasonably high lift coefficients.

The SD7037 was chosen for this aircraft because it had the highest $C_{L\,max}$ as well as the lowest drag which can easily be observed in the $C_{L\,\alpha}$ and drag polar plots (Figure 5.2. and 5.3.) below.

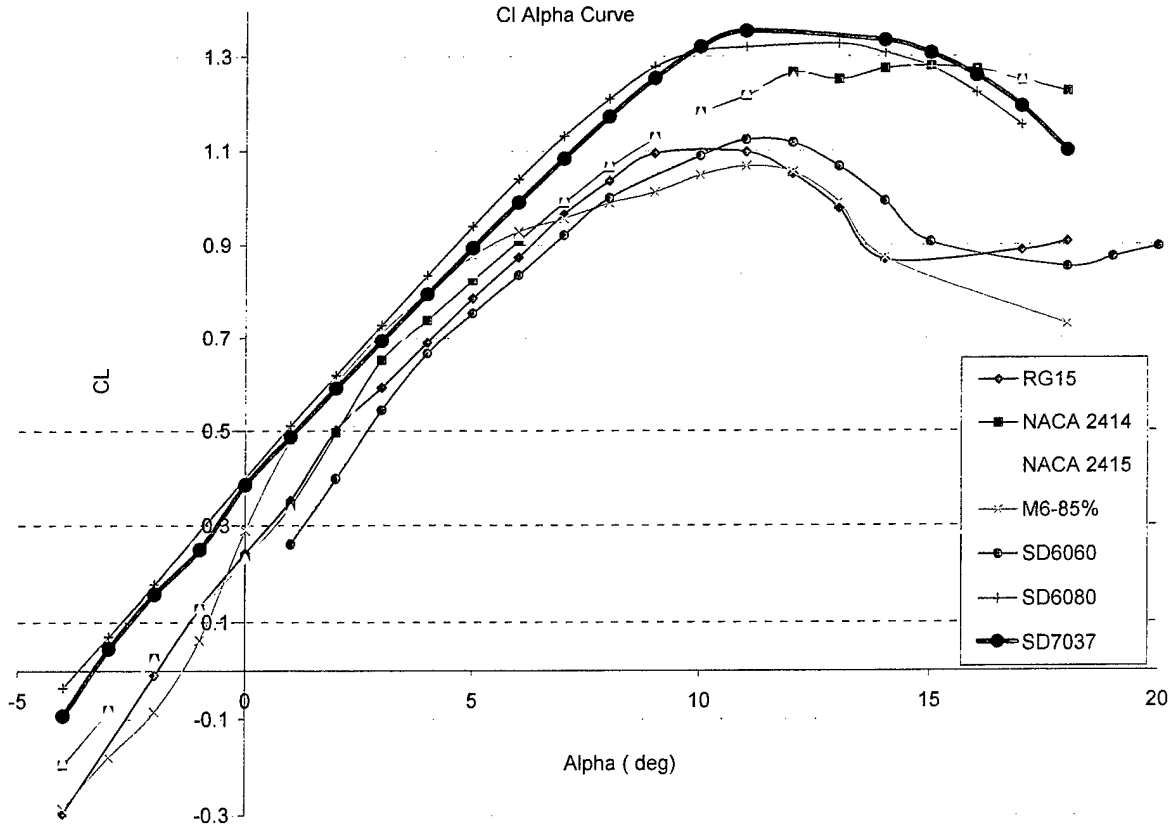


Figure 5.2. $C_{L\,\alpha}$ curve of selected airfoils.

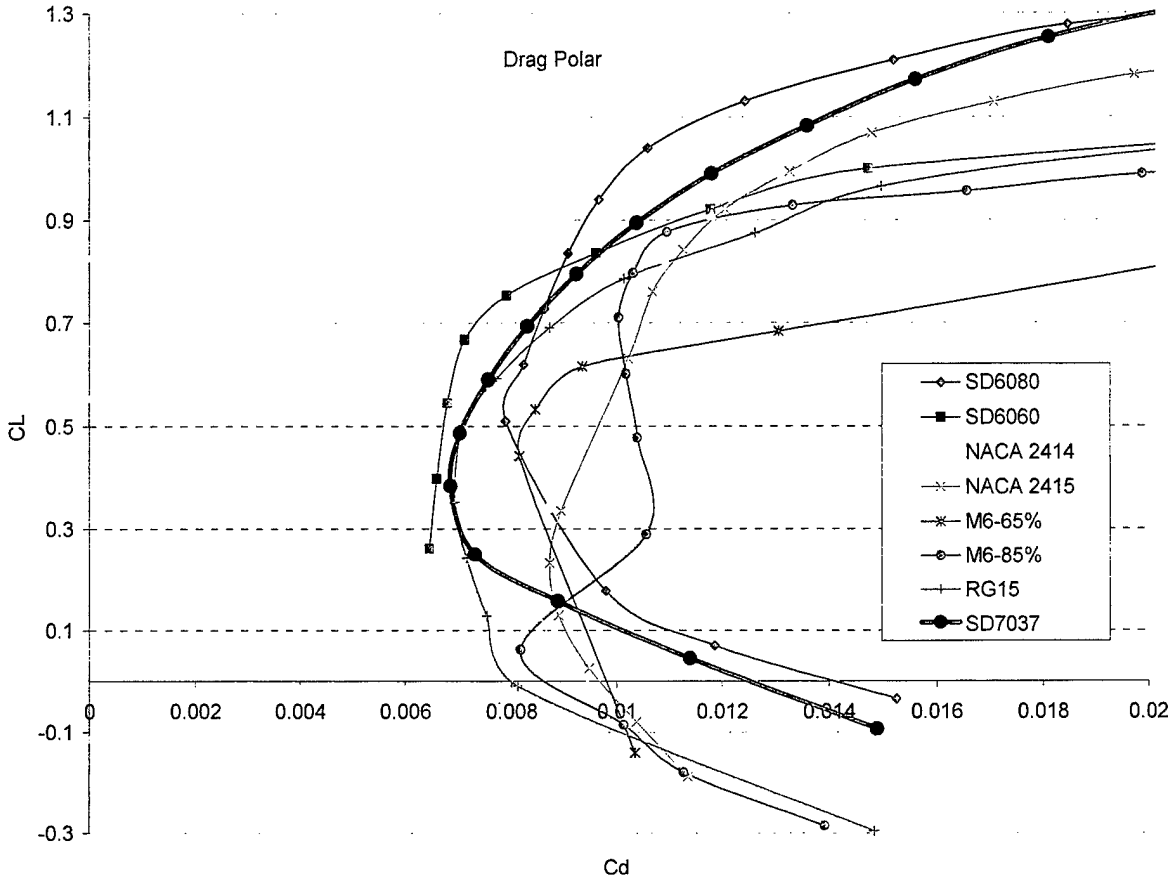


Figure 5.3. Drag Polar of selected airfoils.

It also met all the design and mission requirements. The only airfoil out of the ones selected which proved to have a smaller drag coefficient was the SD6060; but this airfoil did not offer enough lift and its $C_{L\,max}$ was considerably lower than that of the SD7037.

In order to assure lateral stability in the low wing design a dihedral of 5 degrees was incorporated into the wings outer 3.5 feet panels. A taper ratio of 0.8 at a taper length of 1 foot was put at the outer sections of the wing panels. This number was selected to prevent tip stall, maintain a higher Reynolds number for the airfoil and to avoid degradation and drag on the wing.

5.2. Tail Design

Static stability is essential to the flight of unaugmented aircraft. For remote controlled plane it is even more important to have a stable plane because the lag time is longer for the pilot to act. For similar reason a higher control authority is required. For a conventional airplane design, stability and control is dominated by the tail contribution.

The tail sizing was based on the required tail volume ratios. Once the preliminary results were computed, they were tested in a Matlab application written for this purpose. The base values required for

stability and control derivatives were chosen from existing airplane data. Using the Matlab application the exact tail areas and control surface areas were computed. The final tail sizing can be found in table 5.1.

	Vertical Tail	Horizontal Tail
Root Chord	0.834 ft	0.834 ft
Span	1.5 ft	2.0 ft
Taper ratio	0.6	0.834
Volume Coefficient	0.04	0.55
Aspect Ratio	2.25	2.67

Table 5.1. Tail sizing

The final computed stability and control derivatives could be found in Table 5.2.

5.3. Drag Estimation

Drag on the aircraft consisted of induced drag and parasite (profile) drag. The estimation of aircraft component drag coefficient, which would contribute the most to the drag are shown below in Table 5.3. Using the table above the drag at various flight conditions and velocities was computed.

5.4. Fuselage Detailed Design

For the final Design of the fuselage the carbon fiber tube design had been settled upon. This would involve construction of a three-foot carbon fiber tube. The tail and nose are detachable; this is to allow access to the interior. The tail and nose would telescope over the tube to allow the fuselage to fit within the box. The batteries are to be held exterior the fuselage in carbon fiber tubes. This will keep the batteries separate from the water within the fuselage. The tubes will be covered by a fairing for aerodynamics, but will allow some airflow past them for cooling. The batteries were placed on the exterior because their volume would have made it necessary to increase length of the fuselage, lengthening the fuselage would create a greater center of gravity change when the water was dropped.

Openings and attachments necessary to the fuselage have been placed where the structure telescopes. This is to make placing holes in the carbon fiber that would reduce its strength unnecessary. The nose attachment allows access to the engine and receiver, while the rear attachment allows access to the servos, which control payload dropping, the tail landing gear, and tail control surfaces. The half-inch hole for payload dropping is located where the tail and fuselage telescope together at the rear. The wires for the servos are embedded in the wings will be run out from the midsection of the wings and run underneath the fuselage into the receiver.

The attachment for the wing remains unchanged. The wing continues to be sandwiched in between the fuselage and the landing gear. The fuselage will be held in place with straps that go over the top of the fuselage and attach it to the same points that hold the landing gear to the wing. A fairing attached by tape will cover this. The wing is made up of three pieces, the outer section will be attached with carbon fiber joiners that fit within the spars. These joiners have dihedral angle of five degrees,

pointing each tip up five degrees. The outer sections of the wing will be held on with tape. This will allow for quick assembly and disassembly of the wings from the box.

	Stability Derivatives		Control Derivatives		Stability Coefficients		Dimensional Derivatives at Cruise		
Longitudinal	$C_{m,\alpha}$	-1.37	$C_{m,\delta e}$	-0.83	$C_{X,u}$	-0.068	X_u	-0.1	s^{-1}
					$C_{X,\alpha}$	0.17	X_α	26	s^{-1}
					$C_{Z,u}$	-0.52	Z_u	-0.78	s^{-1}
					$C_{Z,\alpha}$	-4.6	Z_α	-699	ft/s^2
					$C_{Z,\dot{\alpha}}$	-1.1	$Z\dot{\alpha}$	-0.72	ft/s
					$C_{Z,q}$	-3.7	Z_q	-2.5	ft/s
					$C_{Z,\delta e}$	-0.19	$Z_{\delta e}$	-29	ft/s^2
					$C_{m,u}$	0			
					$C_{m,\dot{\alpha}}$	-4.5			
					$C_{m,q}$	-16			
					$C_{m,\delta e}$	-0.82			
Lateral Directional	$C_{n,\beta}$	0.21	$C_{n,\delta r}$	-0.082	$C_{y,\beta}$	-0.53	Y_β	-81.3	ft/s^2
			$C_{l,\delta a}$	0.075	$C_{y,p}$	0	Y_p	0	ft/s
					$C_{y,r}$	0	Yr	0	ft/s
					$C_{y,\delta a}$	0	$Y_{\delta a}$	0	ft/s^2
					$C_{y,\delta r}$	0.21	$Y_{\delta r}$	31.3	ft/s^2
					$C_{n,\beta}$	0.21			
					$C_{n,p}$	-0.033			
					$C_{n,r}$	-0.15			
					$C_{n,\delta a}$	-0.0039			
					$C_{n,\delta r}$	-0.082			
					$C_{l,\beta}$	0			
					$C_{l,p}$	-0.76			
					$C_{l,r}$	-0.085			
					$C_{l,\delta a}$	0.075			
					$C_{l,\delta r}$	0.013			

Table 5.2. Computed Stability Derivatives.

A/C Component	Reference Area (in^2)	C_D
Wing		.007
Fuselage	7.07	.21
Main Landing Gear	6	.25
Rear Landing Gear	.375	.58
Total Parasite Drag		**1.05**

Table 5.3. Parasite drag coefficients at 80 mph.

5.5. Propulsion

The aircraft design was refined after the Preliminary Design Review. The propulsion design process was the same as for the preliminary design, but the inputs have changed. The three minutes constraint was removed. The number of cells was increased. This had a significant difference in the possible use of the Astro Cobalt 90 Direct Drive motor. The increased number of cells gave the necessary take-off thrust capability to this configuration. The design objective was changed from a heavy lifter to more of a racer design.

After further investigation of the battery cell characteristics the KR-1700AU was removed from the possible selections. This is a "high capacity" cell that is not suited for short duration large current drawing applications, such as this competition. This eliminated the dual Astro Cobalt 60 motor configuration, due to inadequate capacity of the largest possible battery pack.

The highest speed possible configuration that does satisfies the minimum take-off thrust requirement was the Astro Cobalt 90 motor. The optimal battery pack and propeller combinations were investigated during the final optimization process. To calculate the expected performance the MotoCalc (Ref. 7) software was used.

Only high performance Sanyo® SCR battery cells were examined for possible use. These are the CP and the N series. The battery cells with the performance required were the CP-1300SCR, CP-1700SCR, CP-2400SCR, N-1300SCR, N-1700SCR, and N-1900SCR. Table 5.4. summarizes the battery cells' weights and capacities.

The N-1700SCR and the N-1300SCR had about the same weight as the N-1900SCR, but smaller capacity; these were eliminated. The N series are fast charge batteries, but significantly heavier; they were eliminated as possible choice. The CP-1300SCR did not have enough capacity to provide power for the entire mission; it was eliminated. Using the CP-2400SCR with the maximum number of cells, to achieve the 5 lb battery weight, was 37. With this amount the pack voltage is too low to deliver enough power to meet the take-off thrust requirement. This left the CP-1700SCR as the only choice for the battery pack.

Table 5.5 contains the specifications of the CP-1700SCR battery cell. These values are provided by the manufacturer and subject to verification by actually testing the cells themselves.

Cell	Capacity (mAh)	Weight (g)	Max Number in a Pack
CP-1300SCR	1300	35	64
CP-1700SCR	1700	46	49
CP-2400SCR	2400	60	37
N-1300SCR	1300	52	43
N-1700SCR	1700	54	41
N-1900SCR	1900	54	41

Table 5.4. Sanyo® SCR Battery Cells

Type	CP-1700SCR
Capacity (mAh)	1700
Voltage (V)	1.2
Weight (oz)	1.62
Diameter (in)	0.87
Height (in)	1.3

Table 5.5. Manufacturer's Specifications

The maximum number of cells that a 5 lb battery pack of CP-1700SCR is 49 batteries. It is difficult to keep such a large bulk of batteries in one piece, and the new design did not provide enough room to carry the batteries internally. The battery pack is divided into two sub packs carried on either side of the aircraft. For weight margin, simplicity and to be able to balance the pack the use of even number of cells is desirable, 48 all together. The two sub packs are connected in series. Each sub pack is the twin stick style. Twelve cells are connected in a stick. Two sticks next to each other, connected in series form the sub pack. Each sub pack is 1.3 feet long. The sub packs have a flattened ellipse shape with 1.8x0.9 inches cross section. Each sub pack weights 2.43 pounds.

The power required and power available curves were constructed the same way as for the Preliminary Design Review. The push was toward higher speed with minimum allowable thrust. This made the Astro Cobalt 90 direct drive engine the preferred choice for motor. Figure 5.4 shows the different motor / propeller combinations investigated.

The best performance came from the Astro Cobalt 90 direct drive motor with a 14x10 propeller combination. Figure 5.5 shows the power available versus power required curves with the excess power superimposed on the graph.

This combination gives a significantly higher maximum speed, and still maintains the minimum required take-off thrust. The expected performance characteristics are tabulated in Table 5.6.

The battery pack weight is within the required five pounds limit. The voltage does not exceed the limitations of the individual components. The motor current is right on the border of the maximum allowed.

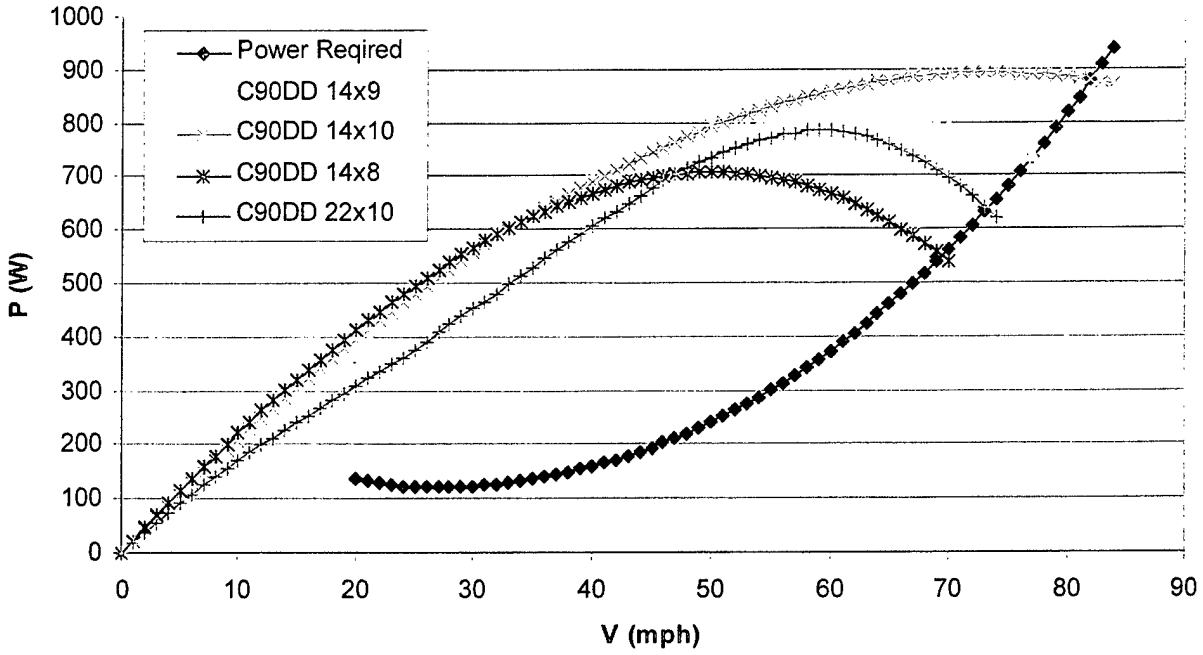


Figure 5.4 Power available and power required.

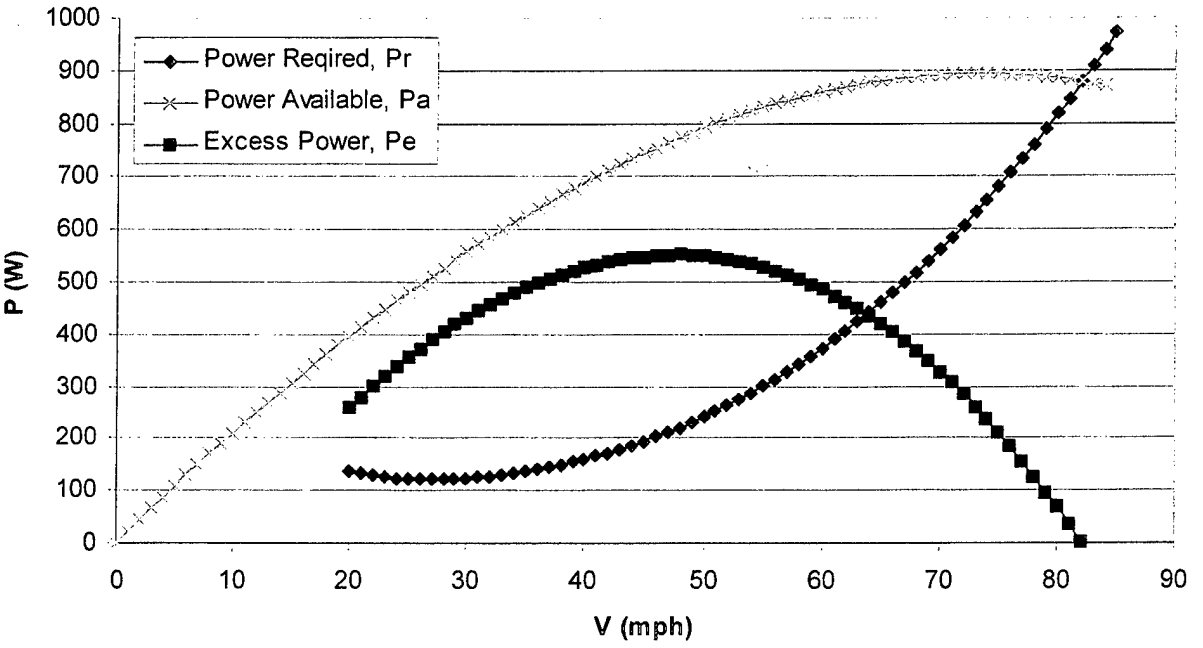


Figure 5.5 Power curves.

Careful flying is required. The mission modeling predicts a sortie flight time of one minute with this speed. With the two minutes and thirty seconds flight time there is a 25% margin in the batteries. The take-off thrust is adequate, although by only a small margin. The static thrust to weight ratio is quite high. This plane will have no problems accelerating or climbing. The maximum speed to stall speed ratio is above the recommended 2.5. The plane will be able to maneuver well and change direction fast.

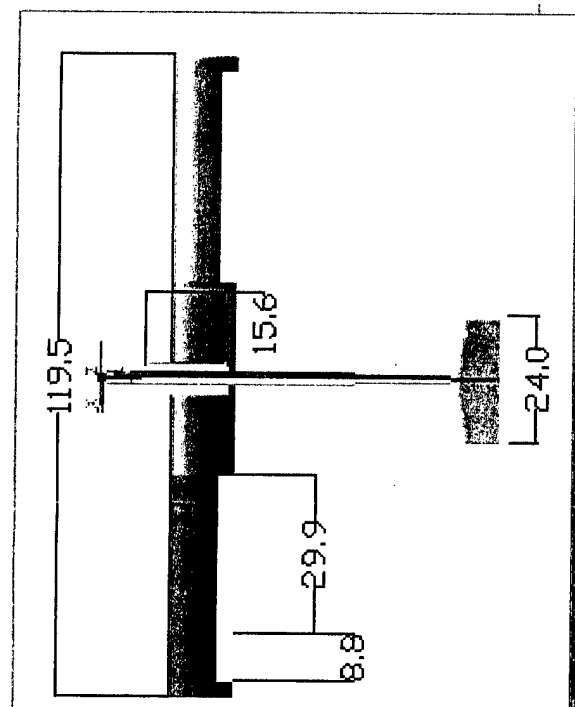
Battery Pack Weight (lb)	Battery Pack Voltage (v)	Motor Current (A)	Max Flight Speed (mph)	Estimated Flight Time (min)	Take-off Thrust (lb)	Static Thrust to Weight	Max Speed to Stall Speed
4.9	46.3	40	82	2:30	10.3	0.45	2.73

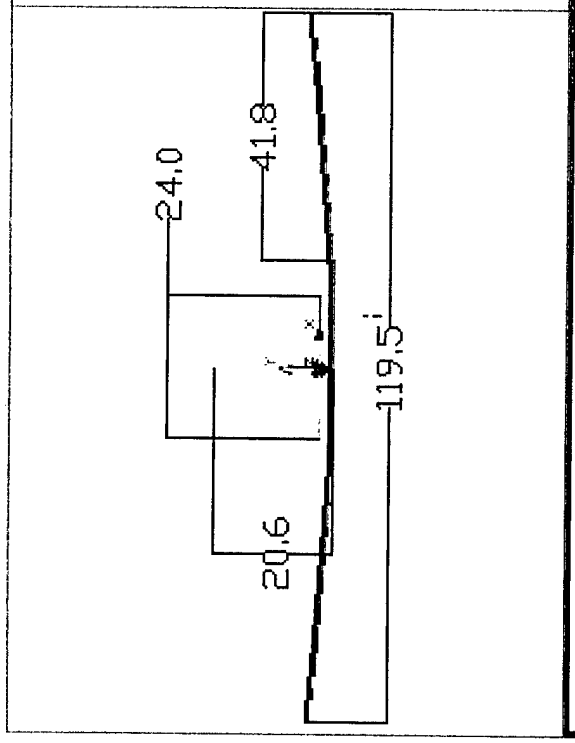
Table 5.6 Expected Performance.

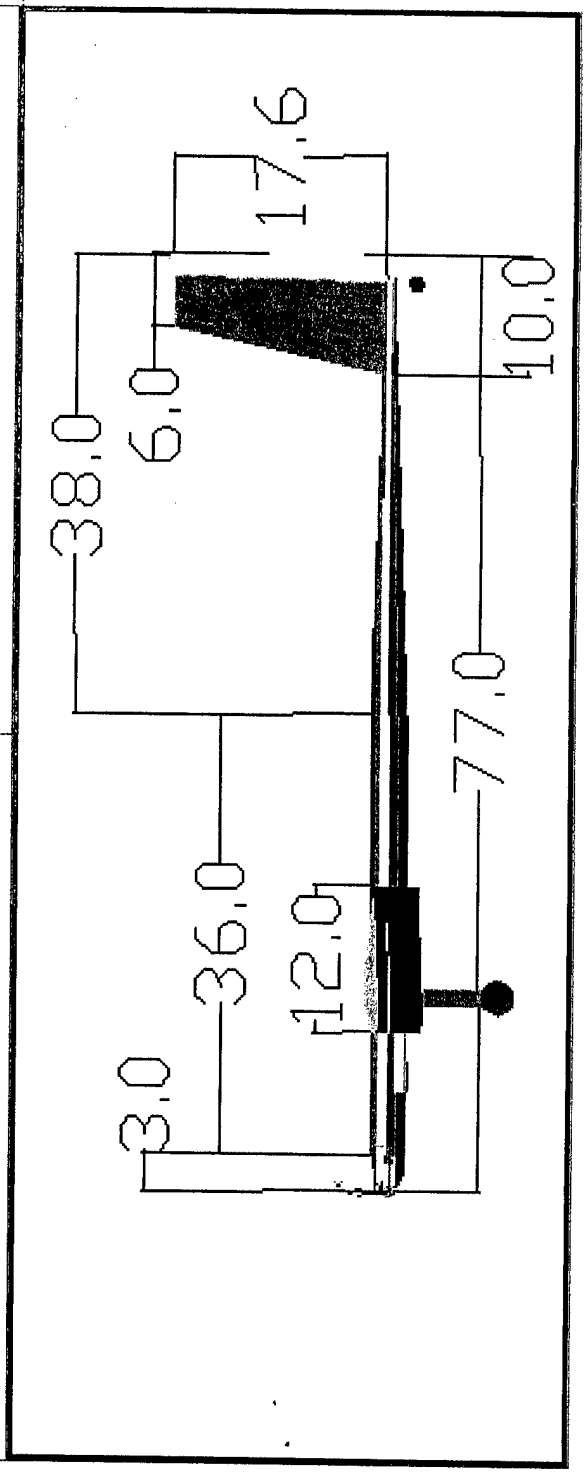
2004 Design Build Fly

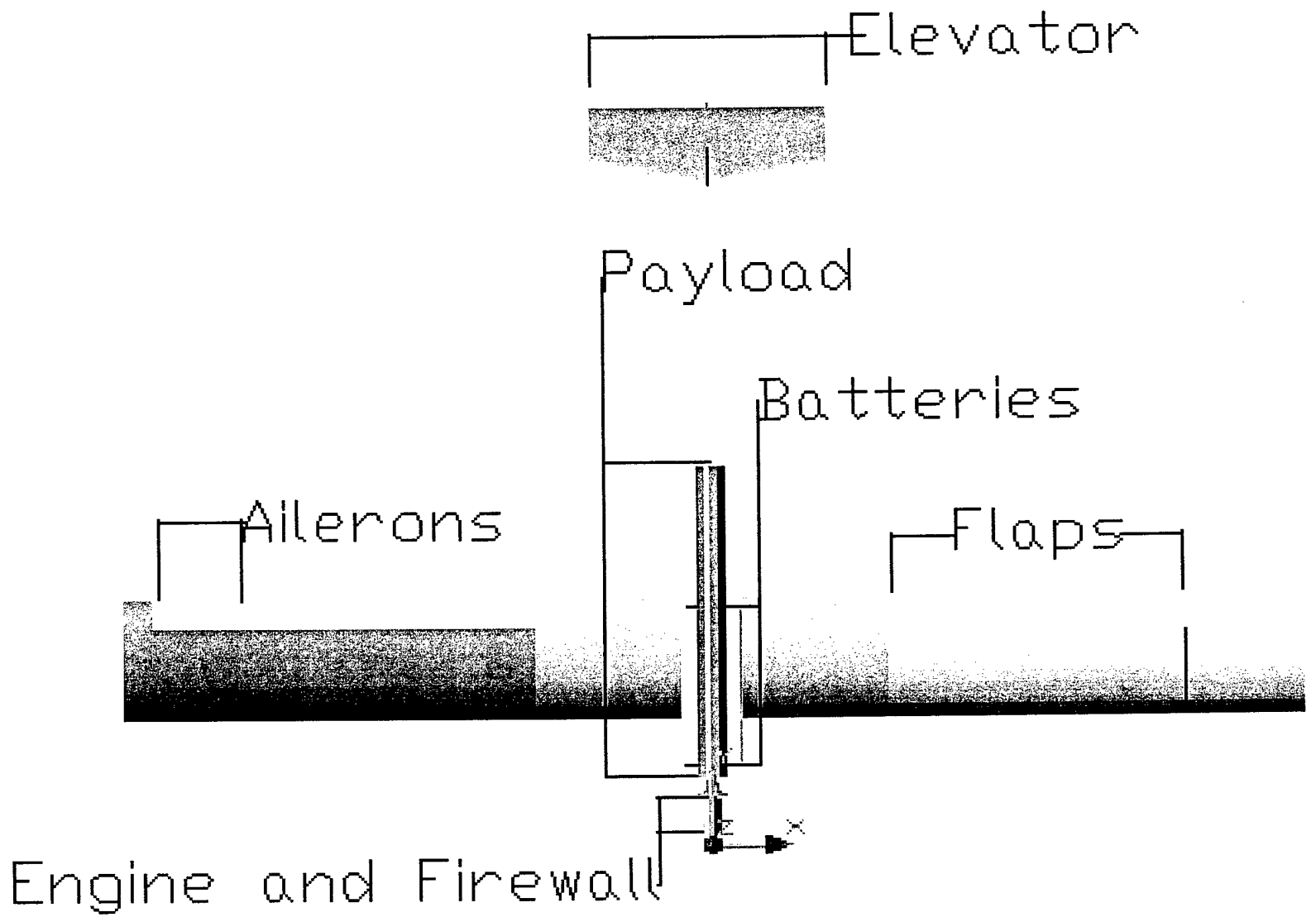
Cal Poly Pomona

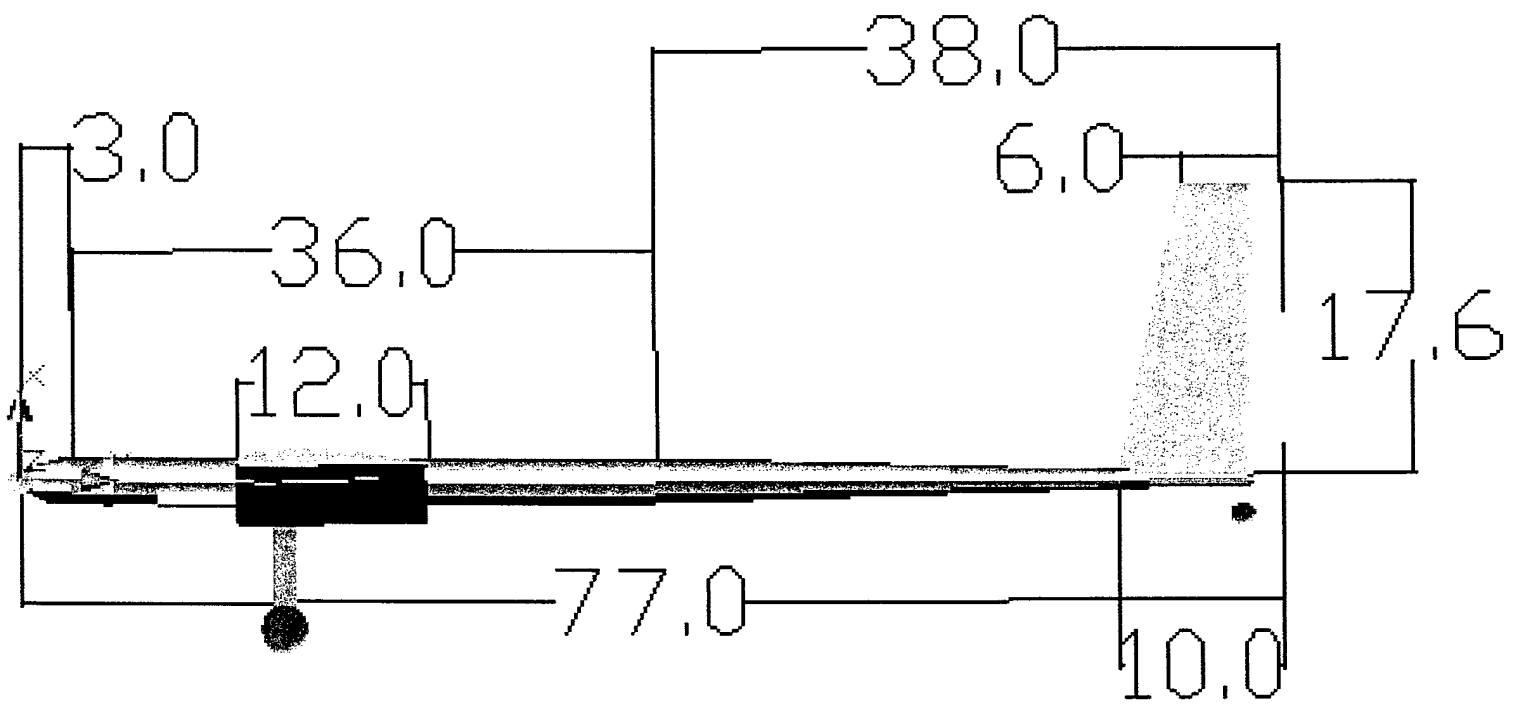
Death From Above!

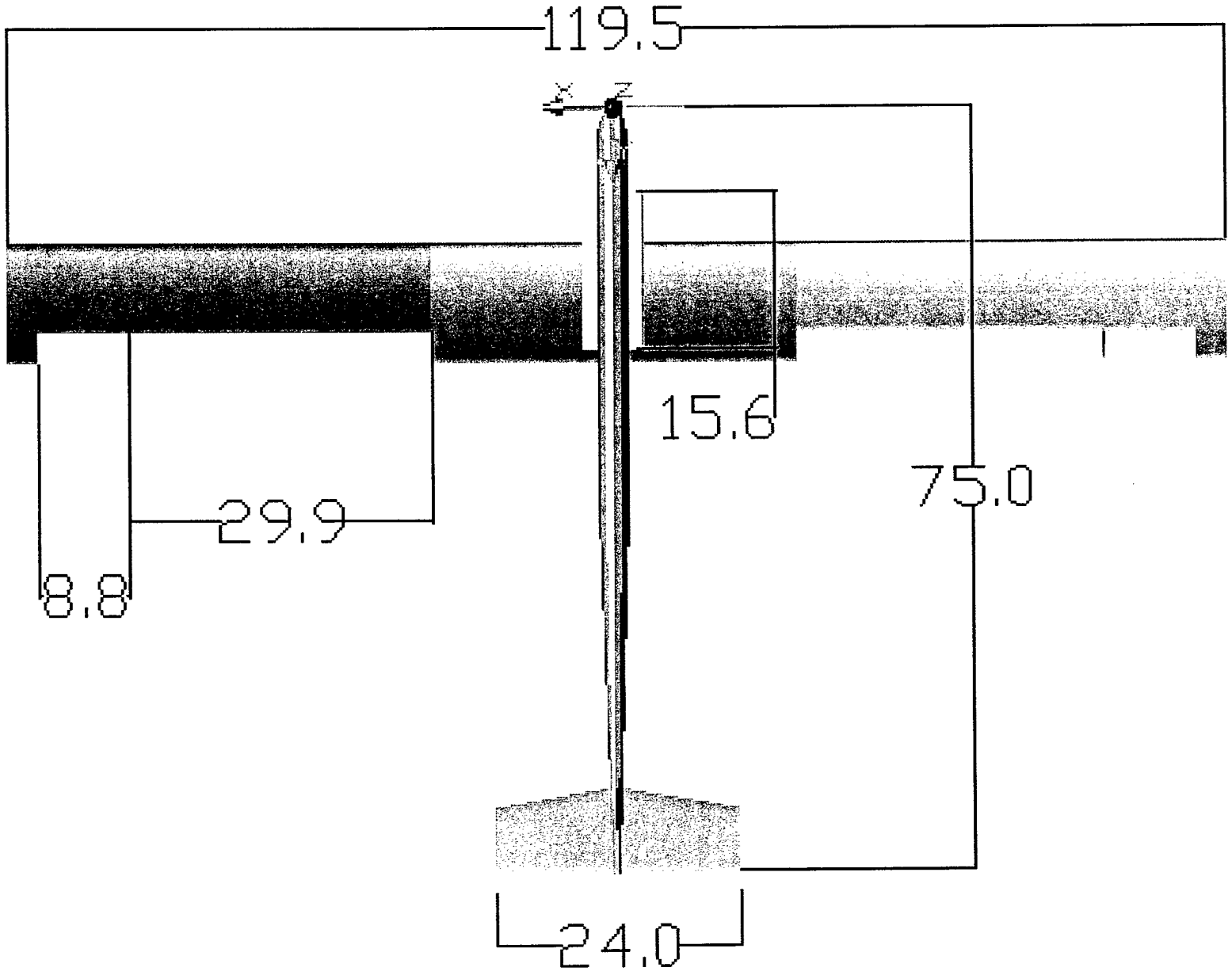


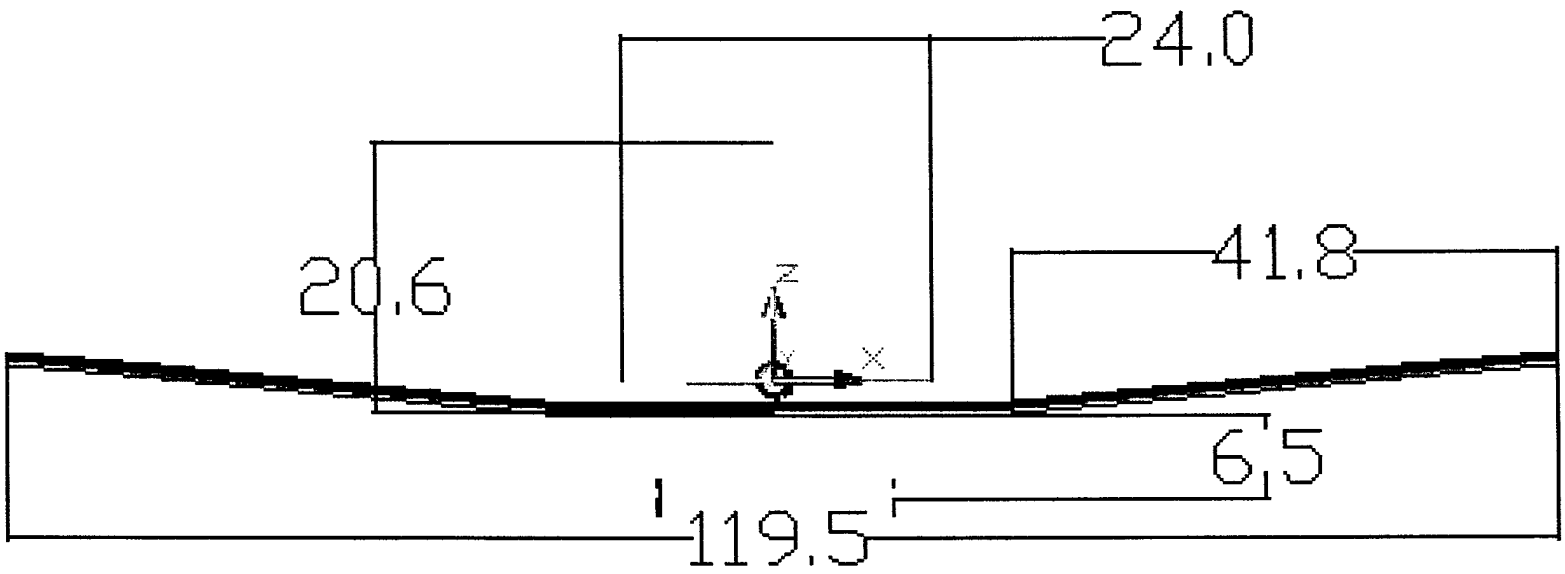





Elevator
Payload
Batteries
Ailerons
Flaps
Engine and Firewall


38.0
3.0
6.0
36.0
17.6
12.0
77.0
10.0


119.5
15.6
75.0
29.9
8.8
24.0


24.0
20.6
41.8
Z
X
6.5
119.5

6.0. Manufacturing Plan

The present work was design to be manufactured in four stages. The four stages were cargo bay, tail, wing, and assembly. Each stage was planned to take a week at most, but failures in materials performance and design changes occurred which changed the time period for milestones.

6.1. Cargo Bay

The first stage was manufacturing of the cargo bay because all other components would be attached to the cargo bay. Initial manufacturing specifications were a 3 feet thin walled cylinder with an inner diameter of 3 inches. The initial material chosen was prepreg fiberglass because of its light weight property to strength. The prepreg material required an oven to cure it properly, which required a mold to withstand 300 °F. The mold was designed from plaster of prais because it a ceramic material. The mold used for the ceramic mold was a 3-inch inner diameter plastic recycled water pipe.

The lack of an oven to cure the prepag meant that one needed to be obtained or built. Visits to local dealer of oven sellers showed it would be too costly to purchase one of the correct size. The decision was made to build an oven. The oven required three major components to build: a heating source, an insulator, and a thermostat. The size of the oven was a 2-foot by 4 foot by 1-foot wooden box because a box of that size was in the workshop. The heating source chosen were 200-watt light bulbs because they heat up air at a rapid pace. A light bulb was placed on three sides of box to generate the correct amount of heat. Aluminum foil lined the box to intensify the light in the box. A thermostat was necessary to turn the light bulbs on at the proper times. It was installed into the circuit feeding the light bulbs. To calibrate the oven, an oven thermometer was installed into the oven. The thermometer was reading 50°F less then the actual temperature in the chamber.

To test the oven, several test samples for the cargo bay were cured. The first sample to be cured was a 1-foot fiberglass cargo bay test section. It was allowed to cure for over 2 hours. After letting the sample cool down, the sample was found to be brittle and weak. A multiple layer fiberglass cargo bay did not show promise for farther investigation. The second sample to be tested was a 2-layer carbon fiber 1-foot section. It was given 2 hours to cure in the oven. After letting the sample cool down, it was found to have the desired properties. To increase strength farther, additional layers were added to the test sample. At 8 layers of carbon fiber, the weight of the carbon fiber didn't warrant farther investigation.

A firewall installation was necessary to support the engine. The initial idea was to install the firewall after curing but installation post curing required holes in the carbon fiber. The idea was rejected because cutting fiber weakens the structure. The firewall would be install during curing by using carbon fiber to hold it in place. Two molds were constructed of length 4 inches and 3 feet. The firewall was cut from aluminum to be 1/16 of inch larger then 3 inches. The firewall was sandwiched between the two molds. The 2-layers of carbon fiber cover only one of the molds. The reminding 6-layers covered the entire cargo bay.

The holes for loading and unloading of the water were a major concern. An improper placement of them could weaken the structure or not allow proper draining. A thin walled plastic tube of 1/2 inner

diameter was cut into four segments of 1/4-inch length. The first three segments were placed behind the firewall for loading and air venting. The reminding segment was placed on the bottom for water drainage. The fibers were placed to loop around the hole instead of being cut. The plastic bag for the tank would be installed at assembly stage. The cargo bay weighed 1.5 lb.

6.2. Wing

The second stage was manufacturing of the wing because it required a lead-time of a least 2 weeks. The dimensions of the wing were 10-foot wingspan, 1-foot root chord, and .8 ft tip chord. The taper section started 1 foot from the tip. The wing was to be composite of foam cores covered in fiberglass with a set of carbon rods traveling span wise. The white foam cores were purchased from Flying Foam because the precision need to cut foam wasn't within the group's abilities. The foam cores arrived a week later then expected with errors. The spar holes were smaller then the outer diameter of the spars because the spars had been incorrectly measured. The foam airfoil was slightly different from the template in a few spots.

The first task was to determine the correct airfoil shape of the airfoil shape for the root ribs. Plywood ribs were used to allow the fiberglass skin to attach to solid airfoil shape and allow attachment of the wings to not damage the foam on assembly. Tracing the foam drew a template for the ribs. The template was used to create 4 ribs for the wings. The tip airfoil was traced to create ribs for the two tips. The correct spar hole size was drilled into the ribs and a channel for the servo wire was also cut out next to the second spar hole. The ribs were glued to the foam and sanded to get the best fit. The ribs allowed the correct spar hole size to be cut into the foam. The servo wire channel was cut out at the same time.

The leading edge of the foam cores was a weak spot because it was the edge of the fibers. The foam cores had been designed with no leading edge for this reason. The leading edge was made of balsa wood to increase strength and withstand accidental collisions. The leading edge balsa was sanded to obtain the correct shape.

In the center section of the wing, it was necessary to reinforce the top and bottom for fuselage and landing gear attachment. On the upper surface of the wing, hard point holes were cut into the wing. The hard points were 1.5 inch by 1.5 inch by 1/8 inch and made of plywood. On the bottom surface, 1/16-inch hardwood was used to distribute the landing gear load. The hard points were glued in using epoxy. Before the wings could be bagged it was necessary to install the spars to prevent collapsing of the cores. There were two channels in the foam core along the entire span of the wing. They were approximately at 1/4 chord and 3/4 chord. The forward spar was to run the entire span of the wing because it was resist bending and twisting of the wing. The aft spar was to resist twisting between segments of the wing therefore didn't run the entire length of the wing. The spars were attached to the foam by using micro balloons and epoxy to fill in the gaps in the spar holes. The channel for the servo wires was filled with micro balloons and an aluminum tube to maintain the opening.

The wing was vacuum bagged and allowed to cure for 3 days. A problem was noticed at the end of 3 days. The wing had collapsed span wise over the spar holes. The problem was attributed to incorrect setting of the vacuum pump. It should have been set for white foam instead of blue foam. The rest of the

wing didn't contain voids or other problems. It was decided to use the wing as test wing. Additional wing would be made as needed.

The ailerons and flaps were cut out of the trailing edge of the outer sections of the wing. The foam was sanded to allow it to hinge. Horns were attached to the ailerons and flaps. Holes were carved into the wing for the servos. Servo wire was run through the wing to each of the holes. The ailerons and flaps were reattached using hinges. The servos were installed and connected to ailerons and flaps.

6.3.Tail

The third stage was manufacturing of the tail which used knowledge developed during cargo bay construction. The tail was 3 feet long with an inner diameter of 3 inch where it attached to the cargo bay. The other end was to be tapered to 1/2 inch to tip. The tip was replaced with a cap because it was necessary to gain access to the empennage attachment.

A mold was constructed with the proper taper to layer the carbon fiber. Carbon fiber was difficult to layer because cutting the fiber was a non-ideal situation. The layers were allowed to overlap some and then cut. By allowing the fibers to overlap, it enabled the fibers to transfer the load. Using 8 layers that were alternating direction of fibers add strength that was lost by cutting the fibers.

To attach the empennage and tail gear, it was necessary to place a hole in the tail. To maintain the structural strength of the tail, the hole was done with the same technique as the loading holes in the cargo bay.

The empennage consists of a vertical wing and horizontal wing. The 0012 airfoil was chosen to reduce drag. The empennage was to be attached by a joiner. A joiner attached the horizontal tail while the vertical tail attached to the tail gear through another joiner.

6.4. Propulsion

The team did not manufacture the battery packs. The battery vendors have all the equipment necessary to assemble the batteries. The needed twin stick style battery packs are ordered pre-made. The 12 cell long stick is too fragile by itself for flight. Special tubing is manufactured to hold them. This tubing is providing the support for the battery stick as well as providing attachment to the fuselage. The two 24 cell sub packs are on the two sides of the fuselage. Figure 6.1. shows the circuit diagram of this configuration.

The propellers are fabricated and balanced, for an additional fee, by the manufacturers. Balanced propellers are more expensive, but eliminate the need for balancing the propellers ourselves; something we have no experience base available.

The main power wires are high quality 13 gauge silicon wires. The fuse is blade style maxi slow blow. Astro Flight Zero Loss connectors are used for all the connections.

For reliability purposes only metal gear servos were used during construction. Due to the relatively thin airfoil micro servos had to be used. The wing servos, both for ailerons and for the flaps, are actuated by Hitec HS-85MG^{+} servos. These are connected to the receiver via standard three-color ribbon wire.

The rudder and the tail wheel is actuated by a single Hitech HS-645MG servo. The rudders are controlled by the same model servo. The payload release servo is a Hitech HS-545BB servo.

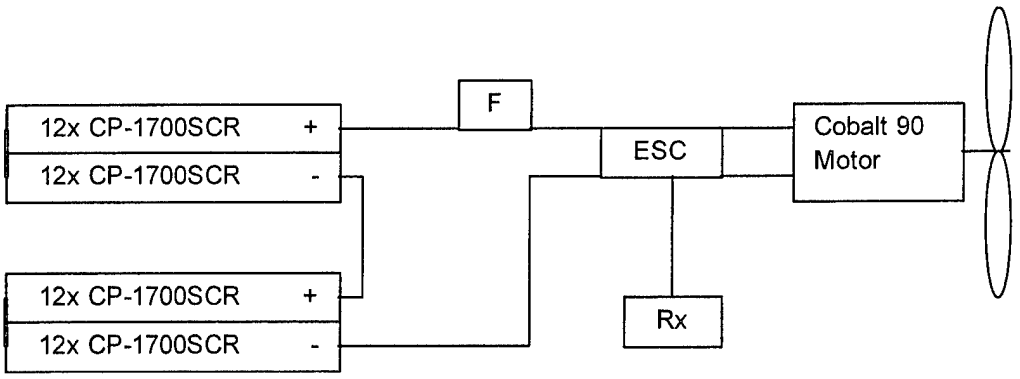


Figure 6.1. Circuit diagram of the propulsion system.

6.5. Time Table.

Stage	2-Feb	9-Feb	16-Feb	23-Feb	1-Mar	7-Mar
Cargo Bay						
Wing						
Tail						
Assembly						

Projected Actual

Table 6.1. Time Table for Manufacturing.

7.0. Testing Plan

The present work required many test of techniques and components. The testing was divided into 3 stages: materials and techniques, components, and flight-testing. Materials and techniques consist of learn how varies material performed under different conditions and how to properly handle them. Components consisted of learning how effective a component was performing it job. Flight test consisted of learning how the plane performed from engine start up to engine cut off.

7.1. Materials and Techniques

To achieve the best performance requires materials of the high qualities and lowest weight. The main trade off in building a racing plane was finding ways to keep the plane together while shedding weight. Almost all the materials used were considered raw materials because they were used for later systems.

The first test to be conducted involved the payload. The payload was considered a top priority because the plane would design around it. Tests were performed to measurer the speed that water exits a 2-liter soda bottle. It was found to 16 seconds standing up. This was the baseline value for loading the plane and emptying it. A pump was added to the system to accelerate the flow but it was found to slow it because of the constriction of the pump. The time was 22 seconds instead of 16 seconds. The decision was made to abandon the pump because it add time in unloading and setup. The next time introduced an air hose to the system. The air hose was found to give noticeable difference in time. The fail decision was to load the plane straight from the bottle.

It also influenced the length and height of the fuselage. Requirements stated that the plane could only be emptied by gravity. To maximize hydrostatic pressure, the plane was design to be long and short which allowed the angle of attack to control the height of the column of water.

The second test to be performed involved varies forms of aluminum. Aluminum is a very lightweight metal with high strength. The question was whether aluminum could be used for the structure. The structure was to be composite of frames and stringers. To test for possible frames usage, 1/8-inch aluminum plate was loaded with weights. The plate was found to withstand several pounds. The aluminum frame was promising. The next thing to be tested was rods and tubes of aluminum. The tubes were found to far to weak for structure usage. The rods showed promise after withstanding several pounds. Development of a new technique was necessary to assemble the structure. The rods were thread at the ends to be screwed into place. The technique showed promised but failed when the entire rod needed to be threaded.

The third test was for a new material. It was the first time prepag composite materials had been available to the group. Prepag composite materials are fiber composites already embedded in correct epoxy ratio that must be baked at 250 °F. This maximizes the strength to weight ratio of the material. The forms available were fiberglass and carbon fiber. The first test involved fiberglass because it was the lighter of the two materials. The fiberglass was baked in tube form for fuselage usage. The tube was found to be far to weak brittle for future usage. The carbon fiber was baked in tube form and was found to be

strong but not enough to make the aluminum. After adding more layers of carbon fiber, the strength increased while the weight remained reasonable. The prepag carbon fiber was found to be the idea materials for the fuselage.

7.2. Components

Testing of components occurred after manufacturing had completed them. The testing schedule varied greatly because many components were not ready. The components tested were fuselage, landing gear, tail and wings. The components were statically tested.

The fuselage was the first component to be manufactured and tested. The first test to be performed was a gravity test where the fuselage was dropped from 5 feet up. The test was design to test fragileness of the fuselage. The drop added no cracks to the fuselage. The fuselage was then resisted at one end and twisted at the other end. The test was designed to how the fuselage would act in a maneuver. The carbon fiber ring held it shape and was very difficult to twist. It was noticed that the ends were frayed. The ends were covered to prevent further fraying. The future fuselage would have additional carbon at the end to prevent the fraying. The final test was to secure one end of the fuselage and vertically load the other end. The test simulated loads from the tail and wings. The fuselage held it form against the loading.

The wing was the next component to be manufactured. The wing tested had indentation over where the spars had been. These were due to incorrect pressure setting on the vacuum. The wing was tested because the next wing would have more strength then this one and would not be done for a while. The wing was assembled and held up by it wing tips. There wasn't any noticeable bending. The wing was held at the tips and twisted in opposite directions. The wing twisted but didn't fail. Close inspection of the surface didn't show any void. The fail test was to load the center section of the wing with 20 pounds of weight. The wing was then held by the tips to perform a 3g loading test. The wing didn't give. The weight was increased by 10 lb but the wing didn't give. Failure test weren't performed because the wing was necessary for flight test.

The tail was the next component to be manufactured. The tail must resist bending and twist created by the empennage. The tail was secured at the end connected to the cargo bay and load vertically at the other end. The test was design to investigate bending. The tail was loaded up 40 lb because this was the max load to be on the tail. The tail maintained its shape. The next test was to twist the tail. The tail was able maintain it shape against the loading.

The final component to be tested was the landing gear. The landing was salvaged from a previous plane. It was known to be able to withstand 40 lbs of loading. The new plane was planned to weigh up to 20 lb. The landing gear was loaded in 5 lb increments. At approximately 50 lb the gear bends. A cross member is installed between the two tires. The cross member increases the loading up to 60 lb which meet requirements.

7.3. Propulsion

The motor and battery testing has three vital tasks. First, it is to verify the specifications given by the manufacturer, for example motor speed / volt constant of the engine, or internal resistance of the batteries.

The second objective of testing is to validate the results obtained from the program used to analyze the different configurations, for example static thrust or motor RPM. The third task is to fine tune the design to obtain optimal performance during actual flight conditions. Table 7.1. lists the various propulsion components and the information collected during testing.

Testing Plan				
Component	Measured Quantities			
Test Phase 1	Verification of manufacturer's specifications			
Battery	Internal resistance	Capacity		
Motor	Armature Resistance	Motor Speed/Volt	Motor Torque / Amp	No Load Current
Test Phase 2	Validation of analysis results using a 15x7 APC Propeller			
Battery	Run Time			
Motor	Static Thrust	RPM	Current	Voltage
Test Phase 3	Selection of best overall performance configuration			
Battery	Flight Time			
Motor	Take-off Distance	Maximum Speed	Rate of Climb	

Table 7.1. Information collected during testing

7.4. Flight Test

Flight tests don't begin on an airfield, but at the test facility. The testing schedule was static stand, electronics test, loading payload, unloading of payload, taxiing, touch and go, laps, and finally mission test. The process took many days because of weather and availability of pilot and airfield.

The plane was assembled in the workshop and allowed to stand on it own. The plane withstands it own weight. Since the plane is already assembled, the electronic system was tested. The flap servos worked properly. The ailerons servos worked properly however they didn't move the range required. The hinge area was sanded to increase the range. The plane was then loaded to max payload then allowed stand for an hour. The test serviced two functions: weight test and waterproof test. The plane was able to prove that it could hold the payload. A leak was detected and sealed up. The plane was reloaded and test again. No further leaks were detected. The servo to empty the tank was tested. The valve opened the gate but at a slow pace. The valve was found to be too tough for the servo. The valve was lubricated to

decrease the force required to turn it. It also was used many times to wear it down. The valve was reinstalled and found to operate correctly.

The plane was taken out to an open area for the taxi test. The plane was empty in the initial test. The engine was turned on and the plane was driven around the area. The plane didn't show any problems. The plane was then loaded with water to test max taxiing weight. The test showed no problems. The plane proved that it was ready for a flight test.

The first flight was a touch and go to test take off distance. The plane was able to meet the requirement but was empty at the time. The plane was then allowed to do a loop around the course. The flight was uneventful. The plane was tried at max speed and raced around the course. The plane was loaded with full payload. The plane went through touch goes proving again it could take off in the required distance. The plane completed a lap and landed successfully. The plane will be further tested to make it faster and lighter.

7.5. Stability and Control

Analytical tools from Reference(X) were used to ensure stability and good control characteristics of the aircraft. The first priority is to verify that the aircraft is stable and has adequate control. After confirming that the aircraft handles well the analytical results will be verified.

The actual stability and control performance will be evaluated by the pilot and the aircraft configuration will be adjusted accordingly. During the test flight the quantitative results of the aircraft's stability and control power obtained by analysis will be validated by qualitative opinion of the pilot. Using the test results the plane's configuration will be altered to obtain best overall performance. These results will be used to optimize the stability and control analysis procedure.

During the flight the pilot will evaluate the aircraft handling performance using the chart on Table 7.2. After the pilot completed the first flight test phase the score will be evaluated. Depending on how much time and resources remain the handling characteristics will be improved.

Pilot Evaluation Sheet					Score
Is it controllable?					
No			Improvement Mandatory	Control Loss	1
Yes	Is pilot workload tolerable?				
	No		Improvement Required	Major deficiencies	2
				Control difficulties	3
				Inadequate performance	4
	Yes	Satisfactory without improvement?			
		No	Deficiencies Warrant Improvement	Tolerable deficiencies	5
				Moderate deficiencies	6
				Minor deficiencies	7
		Yes	No Improvement Required	Fair handling	8
				Good handling	9
				Excellent handling	10

Table 7.2. Aircraft Stability and Control Flight Test Pilot Evaluation Sheet

7.6. Aircraft Parameters

Table 7.3. lists the final aircraft parameters.

Performance	
Cl_{max}	1.35
L/D_{max}	10.36
R/C_{max}	0.233
V_{stall}	30 mph
V_{max}	82 mph
TOD, empty	55 ft
TOD, gross weight	130 ft

Geometry	
Wing span	10 ft
Chord Length	1 ft
Taper Ratio	0.834
Taper Length	1 ft
Wing planfrom Area	9.83 ft^2
Fuselage length	5 ft
Height	3 in
Aspect Ratio	10.26

Control Volumes	
Vertical Stabilizer	
Span	1.5 ft
Aspect Ratio	2.25
Root Chord	0.834 ft
Taper Ratio	0.834

Airframe	
Propulsion system	7 lbs
Control system	1 lb
Payload	9 lbs
Wings	4 lbs
Fuselage	4 lbs
Empty weight	16 lbs
Gross weight	25 lbs

Horizontal Stabilizer	
Span	2.0 ft
Root Chord	0.834 ft
Aspect Ratio	2.67
Taper Ratio	0.834

Systems	
Radio used	Futaba
Servos used	7
Battery configuration	Twin stick
Motor used	Astro flight Cobalt 90
Propeller	14x10
Gear ratio	1, Direct Drive

Table 7.3. Final Aircraft Parameters

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AIAA
University at Buffalo
Buffalo Bomber



Written By:

Gilbert Romanowski

Dan Reed

Steve Frankino

Frank Centinello

DeJuan Laguerre

Milovan Vugdragovic

Drabo Mebougna Baba

Drafters

Gilbert Romanowski

Mrynal D'Arcangelo

Table of Contents

1.0 Executive Summary	2
2.0 Management Summary	3
3.0 Conceptual Design	6
4.0 Preliminary Design	10
5.0 Detailed Design	15
6.0 Manufacturing Plan	19
7.0 Testing Plan	23

I. Executive Summary

The University at Buffalo's student chapter of AIAA (American Institute of Aeronautics and Astronautics) is proud to present its 2004 Design Build Fly aircraft. The aircraft put forth by this student engineering society is the culmination of hard work and planning begun many months ago. As an assembly of young, aspiring, aerospace engineers who created a model intended for flight, it would be a great honor to bear witness to its birth of purpose.

The DBF project began with decisions regarding conceptual design. It was up to the DBF team to compare and contrast the strengths of various configurations that would complete two designated missions. In doing this, a considerable amount of attention was delegated to the more difficult of the two sorties. After preliminary analysis, the fire bomber mission was given a difficulty rating twice that of the ferry mission. With that in mind, the latter mission will receive the most consideration when determining the effectiveness of each configuration.

The University at Buffalo DBF team advisors are Dr. Phillip Reynolds and Dr. Cyrus Madnia. Senior aerospace engineering student Gilbert Romanowski is the team's captain. Danelle Shrader is the team treasurer and treasurer of the university's AIAA club. Dan Reed is the team secretary. The DBF design team includes Alexey Ouzounov, Erik Peterson, Chris Nebelecky, Steve Frankino, John Fudella, Yuka Kono, Will O'Meara Wern Huan Yam, Marie Roedigor, Bennie Mwiinga, Bhaskar Reddy, Shreshth Joshi, Ashish Shah, and Rob Suto.

Individual tasks within the DBF team were divided among select individuals. Propulsion system research was the responsibility of Erik Peterson while battery research was overseen by Alexey Ouzounov. Detail design, stability and control were under the supervision of Gilbert Romanowski. Preliminary design was handled by Dan Reed and Gilbert Romanowski.

With respect to each individual task we have an ordering of projected start dates alongside the actual starting dates. As one can see from the provided table, most actual starting dates are the same as projected starting dates.

The following steps describe the team's step-by-step design process providing a functional, mission-oriented aircraft while having fun learning the complexities of flight and flight control.

II. Management Report

The U.B. AIAA design team consists of team Captain Gilbert Romanowski and Co-captain Alexey Ouzounov. They are senior aerospace engineering students and supervise the team's aircraft design, building, and testing. The team captains supervise twenty undergraduates in the construction phase. DBF team Secretary Dan Reed kept records for the team. Dr. Phil Reynolds was consulted during the preliminary design phase. U.B. AIAA Student Chapter President Frank Centinello and Treasurer Danelle Schrader supervise funding for the plane.

The following is a chart of management for the design team:

Title	Name
Advisors	Dr. Phillip Reynolds and Dr. Cyrus Madnia
Team Captain	Gilbert Romanowski
Team Teasurer	Danelle Schrader
Team Secretary	Dan Reed
Design Team Staff	Alexey Ouzounov, Erik Peterson, Chris Nobelecky, Steve Frankino, John Fudella, Yuka Kono, Will O'Meara, Wern Huan Yam, Marie Roediger,Bennie Mwiinga, Bhaskar Reddy, Shreshth Joshi, Ashish Shah, Rob Suto

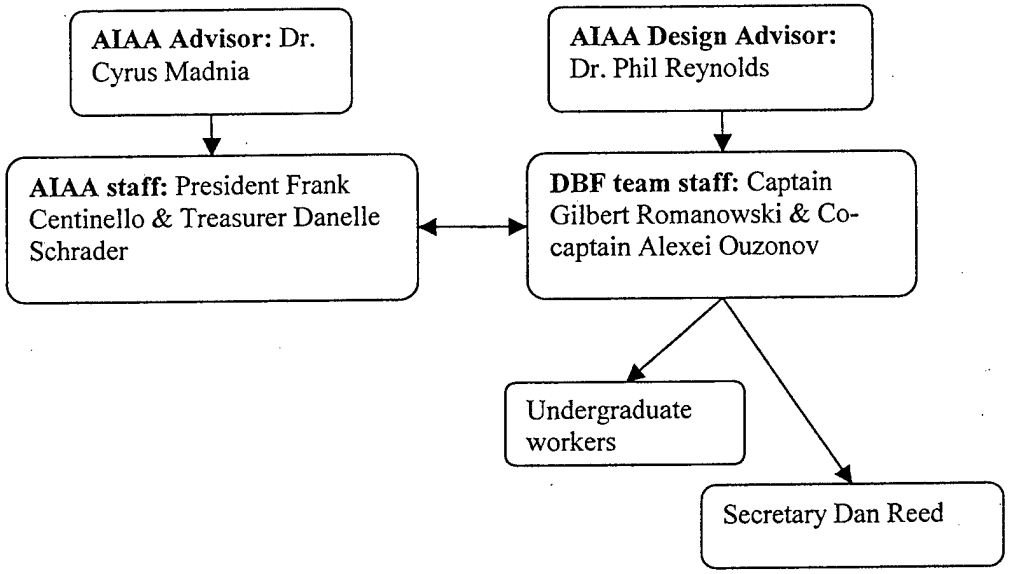
The chart below specifies individual responsibilities:

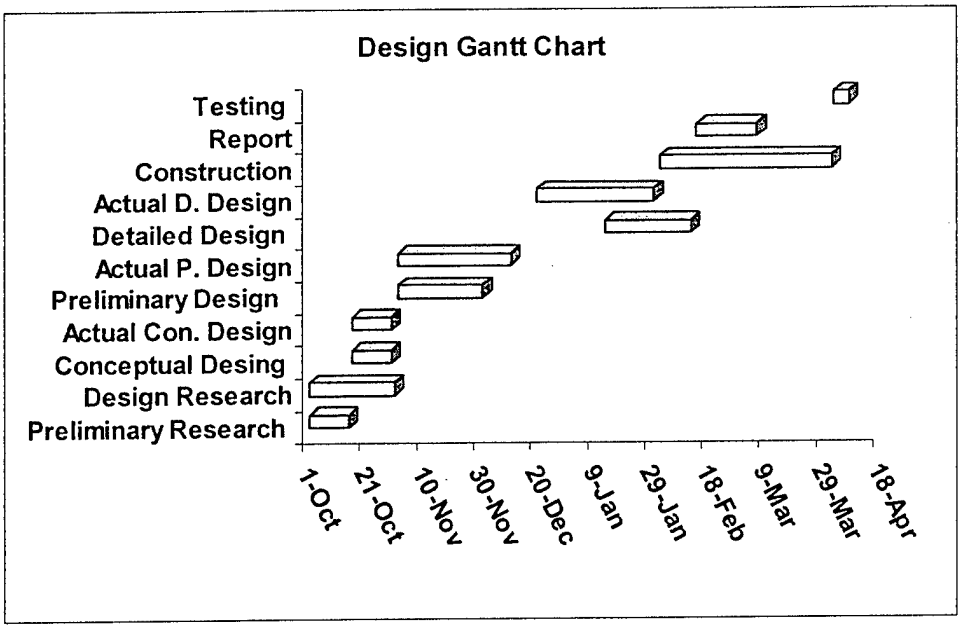
Task	Assigned To
Propulsion System Research	Erik Peterson
Buliding Material Research	Chris Nebelecky
Battery Research	Alexey Ouzounov
stability and Control	Gilbert Romanowski
Conceptual Design	Design team
Preliminary design	Gilbert Romanowski and Dan Reed
Detail design	Gilbert Romanowski
Construction	Design Team

The following milestone chart illustrates our time-table of work:

Task & Projected Start Date	Actual Start Date	Duration (days)
Estimated Research 10/1/03	10/1/2003	14
Actual Research10/1/03	10/1/2003	30
Estimated Conceptual Design 10/16/03	10/16/2003	14
Actual Con. Design 10/16/03	10/16/2003	14
Estimated Preliminary Design 11/1/03	11/1/2003	30
Actual P. Design 11/1/03	11/1/2003	40
Estimated Detailed Design 1/13/04	1/13/2004	30
Actual D. Design 12/20/03	12/20/2003	41
Estimated Construction 2/1/04	2/1/2004	60
Actual Report 2/14/04	2/14/2004	21
Estimated Testing 4/2/2004	4/2/2004	5

This flowchart shows the chain of command for the Design/Build/Fly team:




Design Gantt Chart
Testing
Report
Construction
Actual D. Design
Detailed Design
Actual P. Design
Preliminary Design
Actual Con. Design
Conceptual Desing
Design Research
Preliminary Research
1-Oct
21-Oct
10-Nov
30-Nov
20-Dec
9-Jan
29-Jan
18-Feb
9-Mar
29-Mar
18-Apr

III. Conceptual Design

The goal of the conceptual design was to weigh the strengths of various aircraft configurations to complete two designated missions, focusing on the more demanding of the two sorties. The fire bomber mission has been given a difficulty rating twice that of the ferry mission and will therefore receive the more consideration when determining the effectiveness of each configuration. Inability to perform the fire bombing mission would be detrimental to the flight score, while having a slower ferry speed will not be as significant.

Mission Requirements

The plane must be disassembled to fit in a 4 ft. x 2 ft. x 1 ft. box. In addition to the spatial requirements, the plane must be completed enough to be readily assembled.

The ferry mission requires a take-off, four complete laps, and a landing. The laps are only considered complete as the plane covers the required perimeter as well as performing a 360 degree turn opposite of the base on the downwind leg of each lap. There is no payload for this sortie.

The fire bomber mission entails the filling of the water bay, a take-off, a complete lap with a water drop, and a landing. This must be completed twice during the mission time. The lap consists of covering the required perimeter, the releasing of the water during the downwind portion of the lap only, and a 360 degree turn opposite of the base, also on the downwind leg of the lap. The water drop may only occur during the downwind portion and a penalty will be incurred for dropping outside the set area and for water left remaining in the water bay at the conclusion of the mission. The water bay should hold no more than 4 liters of water, while maximizing it to this limit to maximize the score.

Each mission requires a take-off that must be performed at no more than a maximum take-off distance of 150 feet. The take-off is deemed complete when the wheels leave the runway.

The final requirement is the performance of the airplane with the consideration of the Rated Aircraft Cost. The RAC effectiveness will determine the ability of the plane to perform rated against its complexity. Therefore, the goal is to minimize RAC while maximizing performance.

Configuration Analysis

The five major configurations considered were conventional, canard, bi-wing, flying wing, and diamond wing. These configurations were considered for accessibility to water bay, take-off requirements, payload weight, speed requirements for water dump, and RAC.

The diamond wing design provided adequate wing area and lift to accommodate a lower speed necessary for the water bay. However, the 4 large wings would make assembly

difficult as well as access to the water bay. More importantly, this configuration would raise the RAC considerably by doubling the number of wings.

The bi-wing design also provided adequate lift for the water dump, and much more accessibility to the water bay, but once again would result in a very high RAC compared with other designs.

The flying wing design provided adequate lift and speed requirements to meet the mission, but the control instability of the configuration raised concerns. The large wings would make the water bay less accessible and issues concerning engine placement were too much to choose this design.

The conventional setup, considered under many conditions (such as fuselage shape, T-tail, V-tail, landing gear, wing placement, etc...), seemed to provide an even tradeoff for most the requirements. With adequate lift and speed requirements, a manageable RAC and easy access to the water bay, the conventional setup would offer a workable design.

The canard tail, which is just a slight alteration from the conventional setup, was considered in high regard. While possessing equal lift, speed, accessibility, and RAC characteristics, we believe the canard tail to provide us our best chance to complete the mission with the highest score possible. The destabilizing moment of the canard allows the wing to provide the main stabilizing moment, which is opposite in a conventional aft tail setup. This moment accounts for the sensitivity in the airplane when using a canard surface. This would allow the wings to continually stabilize the airplane during the water drop and allow the canard to account for any center of gravity movement during this same process. The canard design also allowed us to reduce the size of the wing since the canard also serves as a lifting surface. The conventional tail would actually reduce lift, making it necessary for a larger wing lifting surface. The canard configuration also makes controlling a stall much easier than a conventional. If the canard is designed to stall before the main wing stalls, the natural tendency of the aircraft is to pitch nose down, thus the aircraft will pick up speed in the event of a stall, and the operator simply needs to pitch up to pull out of the dive.

Component Analysis

Various other arrangements of engines, landing gear, etc. have been analyzed to meet the same mission requirements

Weighing the number of engines comes down to the power requirement vs. RAC. While the RAC would increase with the additional engine, the 8+ lbs payload along with the 150 ft. take-off requirement deemed it necessary for the aircraft to complete the mission correctly and quickly.

Although the conventional prop arrangement offers the most reliable performance, ducted fans were considered for the source of thrust. However, the ducted fans required more rpm/volt than the allowed engine could output. Unfortunately, the thrust was not adequate and the idea was rejected.

The pusher versus tractor engine arrangement considered many factors. The structural requirements were basically equal, but the pusher causes less drag and allows a safer accessibility to the water bay. Also, the tractor causes more dirty air to hit the wing. So, in addition to the drag it would cause, this would be critical at the low speeds desirable for the water drop.

The engine mounting would provide less structure for the engine mounted below the wing. However, after selecting the pusher arrangement, the engine would be dangerously close to the flaperons. Also, the ground clearance becomes a major issue with the props selected with the mounting below the wings. Therefore, the engines would be mounted on the top of the wings.

The wing shape was chosen from a selection of straight, swept, and tapered wings. The swept wing would require more structure because it would have to support each wing with separate supports. This would also be more difficult to construct in the manufacturing process. The tapered wing provides a greater surface area with less induced drag due to the air sliding down the wing taper. The spar structure of the wing would be the same as the straight wing, with only a slightly more complicated shape.

The wing placements considered were bottom, middle, and top. While trying to keep the water bay at the center of gravity for stability purposes, it would not be ideal to have the structure required for a middle wing in the center of the fuselage. The bottom wing is statically unstable without a dihedral effect and such an effect would require more structure and therefore more weight. It would also affect the engine mounting that was selected to be on top of the wing. Therefore, the top wing, with more room for the water bay, would provide the best wing configuration for our requirements.

In choosing landing gear, both fixed and retractable were studied along with tricycle and tail dragger configurations. Because of the complexity of retractable gear, the drag produced by the fixed gear is acceptable. Tail dragging gear is less stable and draws the plane more from its path. The tricycle gear is sturdier on landing and self rights the airplane.
This analysis and decision-making is evident in more detail on the following rated FOM, Figures of Merit, Chart (Table 3.1). This chart reflects our ratings of the various configurations and components considered. The conclusion verifies the previous discussion

and yields a conventional airplane with a canard surface, two pusher props mounted on the top of tapered top wings and a fixed tricycle landing gear.

		Aerodynamics		Manufacturing			Performance		Stucture		Desire to build	Total
		Lift	Drag	Time	RAC	Ease	Stability	Control	Strength	Weight		
	Weighting	0.9	0.4	0.3	0.2	0.1	0.8	0.6	0.7	0.5	1	
Propulsion	Single	0	0	10	7	10	8	10	0	10	1	23.8
	Twin	0	0	7	4	6	10	10	0	5	10	30.4
	push	0	10	7	0	0	7	7	0	0	10	26.9
	Tractor	0	5	10	0	0	7	7	0	0	8	23.8
	Push Pull	0	7	7	0	0	10	10	0	0	5	24.9
Wing	Mono plane	7	10	10	10	10	10	10	10	10	10	52.3
	Biplane	10	5	5	5	5	10	10	10	5	1	39
	Flying Wing	8	9	8	9	6	7	6	10	8	5	41.2
Tail	T-Tail	0	9	9	10	9	10	10	8	8	2	34.9
	Canard	5	10	10	10	5	5	8	10	8	10	44.3
	V-Tail	0	9	7	8	7	7	7	7	7	8	34.5
	Tailless	0	10	10	7	6	4	4	8	10	0	25.6
	Conventional	0	9	10	10	10	10	9	10	10	9	44
Fuselage	Cylindrical/Box	0	9	5	7	5	0	0	10	5	0	17
	Trapezoid	0	10	5	9	5	0	0	10	6	0	18.3
	Blended	0	10	10	10	10	0	0	10	9	10	31.5
	Box shaped	0	4	10	10	10	0	0	7	10	0	17.5
Landing gear	Tricycle	0	5	7	7	8	10	10	10	7	8	39
	Fixed	0	5	10	10	10	0	0	10	10	9	29
	Retract able	0	10	2	2	5	0	0	8	2	10	22.6
	Tail dragger	0	5	9	10	10	7	6	8	10	0	27.5
Wing Config.	Straight	10	5	10	10	10	10	10	8	8	2	42.6
	Swept	7	10	7	7	6	7	9	10	10	8	45.8
	Top	0	7	10	0	10	10	0	10	0	5	26.8
	Mid	0	8	5	0	5	8	0	5	0	5	20.6
	Low	0	9	10	0	10	2	0	10	0	5	21.2
	Taper	9	9	8	9	9	10	10	10	9	9	51.4
Empannage	Single Boom	0	8	10	10	10	8	0	6	10	5	29.8
	Twin Boom	0	2	9	5	8	8	0	10	9	7	30.4
	Tapered	0	10	8	8	9	10	0	10	7	10	37.5

Table 3.1 Rated FOM Chart for Conceptual Design

IV. Preliminary Design

The preliminary design of the aircraft was based on the culmination of the ideas decided on in the conceptual design phase. During this phase a more analytical approach was taken to further develop the conceptual design. The most essential parameters were determined. The design parameters, which were of lesser consideration earlier, now play a vital role in determining the configuration of the airplane. Ideas discussed in the conceptual phase must be refined to obtain solid data on predicted airplane performance.

Wing

The search for the optimal airfoil involved researching a large number of different airfoils, designed for a Reynolds number of the same magnitude as the one for our aircraft, then virtually testing it for the full range of Reynolds numbers applying to the wing design. The polar graphs of the lift coefficient vs. drag coefficient were then compared with the Clark Y airfoil since it was the airfoil used in our previous designs, and a proven performer for small aircraft. Airfoils with shapes difficult to manufacture or insufficient thickness were discarded from consideration.

The Eppler 387 airfoil was determined to have the best shape for the wing, since it provides the most lift of the airfoils we considered, while retaining a very low drag coefficient of under 0.01 over a rather large range of angles of attack. The profile of this airfoil and its polar graph are shown in Figures 4.1 and 4.2 respectively. The lift coefficient at 0-angle of attack is 0.4, and the drag was 0.0061, which is very good for cruising conditions. The airfoil might still be required to have a small angle of attack at cruising speed depending on the final weight of the plane, in order to provide sufficient lift.

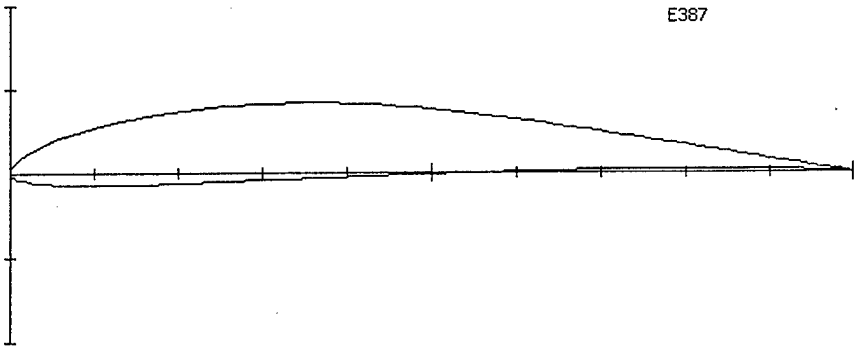


Figure 4.1 Eppler 387 airfoil

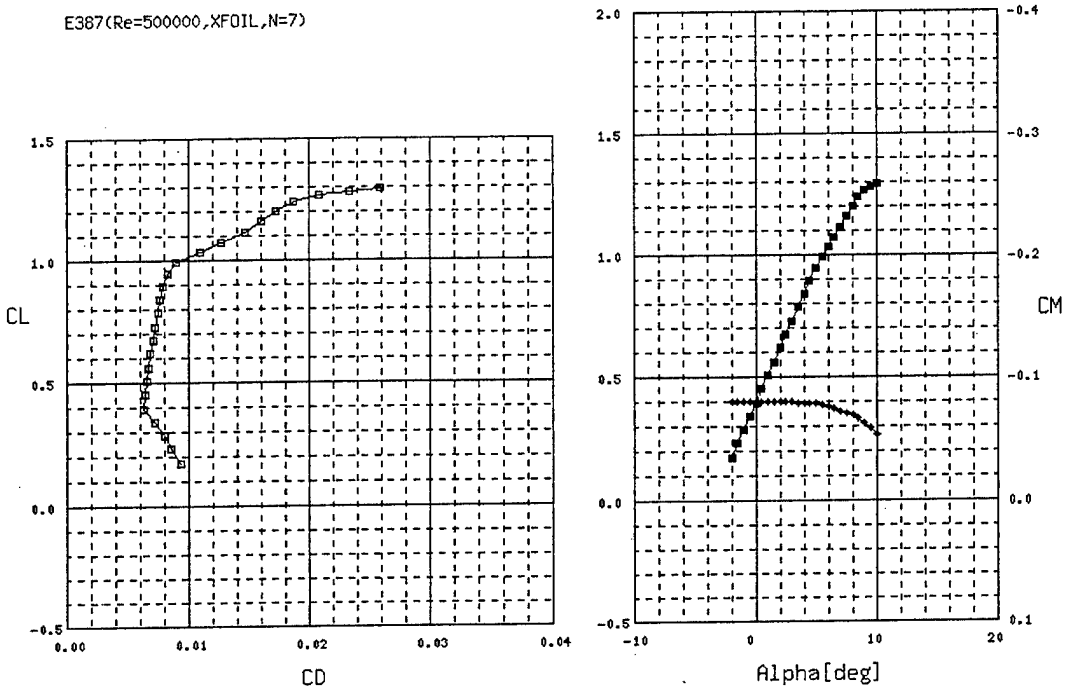


Figure 4.2: Eppler 387 lift coefficient vs. drag coefficient polar graph [6]

The Eppler airfoil is very robust. It retains a large lift coefficient over the full range of angles of attack, while keeping a considerably smaller drag coefficient at all times. The only disadvantage is the Eppler's rather abrupt change in drag at negative angles of attack and at about 6-degrees angle of attack. However, this was determined not to have too great of an affect in terms of control. Since the drag coefficient still remains lower than that of the other airfoils, it does not result in any relative performance losses. Also, for the most part, the plane should never require going into a negative angle of attack.

The optimal wing configuration was determined to be an 8ft wingspan, with root chord length of 24 inches; and a tip chord length of 12 inches. This results in a wing area of $9ft^2$, which equals an estimated wing loading of 54 ounces/ft^2. The taper ratio for this configuration is 0.5, which is a decent amount of taper, and results in a significant amount of drag reduction. A very good taper ratio would have been 0.2 or 0.3, however there was no way of achieving such a ratio without significantly decreasing the aspect ratio, having an impossibly large root chord length, or a structurally very weak and small tip chord. Since the aspect ratio has a greater influence on total drag, it was determined to be much more desirable to have it at 5.7 compared to a slightly smaller taper ratio.

Configuration	Canard
Wing Span	8 ft
Wing Area	9 ft^2
Root Chord Length	18 in.
Tip Chord Length	9 in
Airfoil	Eppler 387
Taper Ratio	0.67
Aspect Ratio	7.11
Wing Loading	53.3 oz./ft^2
Weight	30 lb
Motor	Astro Cobalt 90 (2)

Table 4.1: Sized Aircraft

Tail

Once the configuration of the tail is decided, the next step is to find the shape, area, and location of the vertical and horizontal surfaces. The size and location of the canard can be related to the dimensions of the wing to give the canard moment volumes, which in turn can be used to calculate stability. The range of vertical tail and horizontal canard surface coefficients are chosen by comparison with similar planes having similar configurations. While staying in the range, the higher end of the spectrum corresponds to higher stability and greater range for the placement of the center of gravity. The lower end of the spectrum corresponds to lower stability and gives a smaller margin of error in the placement of the center of gravity. A higher stability value is desirable, especially in heavy wind conditions, but it also makes the aircraft less maneuverable, thereby making the determination of the moment volumes an optimization problem. The equations describing this are:

$$S_h=V_h*S*c/x_h \quad \text{and} \quad S_v=V_v*S*b/x_v$$

S_h and S_v are the horizontal and vertical stabilizer areas respectively; V_h and V_b are the volume coefficients; c is the mean chord length of the wing equal to 1.125-feet; b is the wingspan at 8-feet; and x_h and x_v are the horizontal distances from the wings aerodynamic centerline to the tail surfaces' aerodynamic centerlines, both of which are equal to 26-inches; finally, S is the wing area at 9 square feet. The volume moment coefficients were picked to be the average of a series of 13 homebuilt aircraft, since homebuilt aircraft are of similar proportions to this airplane, even though this one is not designed for any passengers. From the above equations, the required horizontal surface area was determined to be 2.25 square feet and the vertical was 1 square foot. If the span of the horizontal stabilizer were to be kept under 27 inches, then the chord length would have to be about 13.5 inches in order to meet the 2.25 square feet area requirement. For ease of construction, the vertical stabilizer will be half of the tail span, and the same chord length and profile, thereby meeting 1 square foot area requirement.

The profile of the tail surfaces was chosen to be symmetrical. Even though extra lift is usually desirable, it would require the center of gravity at a location further back than its current one. Since the wing is in the wake of the tail, the flow over it is quite turbulent, which would cause the lifting force from the tail to vary, thereby varying the center of gravity's moment on the plane, thereby resulting in an unstable flight. The NACA 0012 airfoil was chosen as the tail profile due to its low drag coefficient, and the fact that it provides sufficient room for structural strength with a 12% thickness.

Fuselage

The main function of the fuselage for the mission is to hold the water tank, batteries and connect the necessary aerodynamic components. Therefore, the shape was designed to conform to the shape of these components. The result is a long flat fuselage which will also act as a lifting body.

The length of the fuselage was driven by the necessity for the canard to balance the pitching moments, and for the rudder to be at adequate distance from the propellers.

Propulsion

The propulsion system consists of a motor, propellers and battery pack. Selection of each was based on competition rules and considerations listed in the conceptual design section.

In addition, an issue was raised in regards to properly testing the static force output of the motor. This was resolved by testing the output along with the battery endurance.

Motor

The motor selection starts with the estimated required motor power. From practical experience, we decided that 40 watts per pound of total aircraft weight would give the power we needed to takeoff within 150 feet. For a conservative estimated aircraft weight of 35 pounds, this results in a motor power output of 1400 watts. We also wanted a single engine configuration because of its low rated aircraft cost contribution, and the success of single engine teams in previous competitions. These considerations resulted in the selection of the largest motor allowed, the Astroflight Cobalt 90. The motor performance specifications are listed in Table 4.2

Model	690	691S
Name	CO-90 Direct	CO-90 Geared
Gear Ratio	Direct	2.75 to 1
Arm Winding	10 turns #22	10 turns #22
Resistance	0.111 ohms	0.111 ohms
Speed	256 rpm/volt	93 rpm/volt
No Load	3.0 amps	3.0 amps
Best Battery	32 to 36 cells	36 to 40 cells
Max Amps	35 amps	35 amps
Prop Rpm	8,000/10,000	3,500 to 4,500
Best Props	13x8 to 16x8	20x14 to24x12
Power Watts	1100 to 1200	1200 to 1400

Table 4.2 Manufacturer's Specifications

This motor has a maximum power output of 1400 watts when connected to 40 cells. This motor was the only one selected from the two manufacturers with a power output in the range that we needed. However, the motor comes with or without a 2.75 to 1 gear ratio, so this aspect needs to be taken into consideration when selecting the propeller.

Batteries

The most important factor in the selection of the batteries is the capacity to weight ratio. Because of the 5-pound weight limit on the batteries, we must make the most use of that weight. Weight and capacity specifications were obtained for every battery from every major Nickel Cadmium battery manufacturer. This resulted in a list of 341 different Ni-Cd cells. This data was entered into a spreadsheet program and the cells were sorted by capacity to weight ratio.

Next, the cells that are manufactured specifically for high current discharge use were identified. This resulted in the selection of the DSC2100P cell manufactured by BYD. It can handle the current expected in our power plant system while still having a high capacity to weight ratio of 42 mAh/g. This cell is also an excellent choice because 42 cells result in a weight of slightly less than 5 pounds and a voltage output of 50. This is the best voltage for high motor power output.

Propeller

With the motor and battery fixed, a suitable propeller must be selected that satisfies the power output of the motor and the needs of the mission. A useful computer program called "ThrustHP" was utilized in the selection of a range of propellers to be evaluated for further testing. The motor manufacturer suggests several propeller sizes for both the geared and direct drive versions of the motor (see Table 4-2). An estimate of 1200 watts of max power output for both versions of the motor was used in the program and resulting rams, static thrust, and speed values were calculated for propellers of varying diameters and pitches. Another factor taken into consideration was the aerodynamic effect of a small propeller diameter spinning in front of a fuselage of almost the same area, which would be the case of the direct drive motor. This analysis led to the conclusion that the geared version of the motor would be used along with a propeller with a pitch of 24. This resulted in acceptable speed, rpm, and thrust values over a range of propeller diameters. The diameter can then be varied to increase the thrust and decrease the speed, or vice versa. Since the program is just a tool and the actual rpm and thrust values had to be determined experimentally, several propellers with diameters between 20 and 24 inches were purchased for testing. The final balance of thrust and speed can then be selected to ensure that the aircraft takes off within 150 ft. APC was selected as the propeller manufacturer because of positive opinions by other modelers as to their performance and strength.

V. Detailed Design

With the major components sized in the preliminary design phase, the goal of the detailed design was to create the specific dimensions and values to be used in the manufacturing of the aircraft. These values were then incorporated into the RAC calculation required for the competition. The RAC was determined to be efficient enough for the expected performance. The breakdown of the Buffalo Bomber's Rated Aircraft Cost is outlined in Table 5.1.

Table 5.1 Rated Aircraft Cost

A	300
B	1500
C	20

Item	Weight (lbs)
airframe	10
batteries	5
motors	6

MEW	**21**	**REP**	**6.25**	
MFHR				
	wing span	8		
	wing mean chord	1.125		
	# of wings	1	wbs1	100
	Flaperons	10		
	F. length	6		
	F. width	0.58	wb2	40.368
	F. height	0.58		
	vertical surfaces	10		
	vertical surfaces w/control Surfaces	10	wbs3	30
	Horizontal surface	10		
	Servo	7	wbs4	35
			MFHR	**205.368**

RAC	**19.78236**

For comparative purposes, the RAC of the Buffalo Bomber is significantly less than Max Drag, the school's fellow aircraft, due mainly to a slim, functional fuselage. The calculated values

for the Buffalo Bomber, based on the configurations decided on in the conceptual section and refined to be more mission specific in the preliminary design phase, are outlined in Table 5.2

Table 5.2 Sized Aircraft Data Table

Geometry		
	length (ft)	6.8
	span (ft)	8
	total hieght (ft)	2
	wing area (ft^2)	9
	Aspect Ratio	7
	canard Area (ft^2)	2.25
	Vertical Tail Area (ft^2)	1
Performance		
W_E		
	CL max	1.2
	L/D	6
	stall speed	9.6
	rate of climb	13.4
	field length (empty)	86.4
W_{TO}		
	CL max	1.2
	L/D	6
	stall speed	9.6
	rate of climb	13.4
	field length (gross) ft	100
Weight Statement		
	airframe (lb)	10
	propulsion (lb)	9
	control (lb)	2
	payload (lb)	9
	empty weight (lb)	21
	gross weight (lb)	30
Systems		
	radio	Futaba 9CA/9CH
	servos	S3104 1/4 Scale
	battery configuration	36 cell
	motor	Cobalt 90
	propeller	22" x 10"
	gear ratio	2.75 to 1

The Buffalo Bomber's geometry and systems are precise. The performance and weight parameters are estimated. Actual values for performances and weights will not be determined until after final construction and flight tests are completed.

The attached drawing package shows the visual representations of all the information presented up to now. The figures display a clean and thoroughly dimensioned orthogonal 3-view drawing of the aircraft, as well as a specific drawing of the water tank.

Figure 1 of the drawing package is the clean 3-view drawing of the Buffalo Bomber. This clearly depicts the location of the water tank and how the fuselage is designed specifically for it. It is easy to see the advantage of having the engines mounted on top of the wings for proper prop clearance from the ground.

Figure 2 of the drawing package is the dimensioned 3-view drawing of the Buffalo Bomber. This drawing displays the major internal and external components of the aircraft and their position on the aircraft. The dimensions give more of a feel for the clearance of the props as well as the balancing of the aircraft components around the water tank to properly place the center of gravity at the water exit.

Figure 3 is the detailed 3-view drawing of the water tank by itself. The shape of the water tank was designed to take advantage of the plane's center of gravity for the water release. The adequately sloped sides converge to a single point to aid the water flowing towards the exit as well as providing more weight closer to the center of gravity, to prevent shifting of the center of gravity.

VI. Manufacturing Plan and Process

Introduction

The design phase is only about half of the DBF project. The other half consists of manufacturing. In this section, five main areas of manufacturing are discussed: wing, fuselage, empennage, engine mount and landing gear. Also, a description of the figures of merit used is included.

Wing Construction

Members examined three alternatives for design methods in wing design: a foam airfoil with carbon fiber shell and spars, a hollow carbon fiber shell, and a more traditional balsa wood frame. The first and second options consist of the selecting and plotting of an airfoil in a CAD program. Next, we would have foam cut in the shape of the airfoil, and line the outside of it with carbon fiber. In the case of the second option, we would remove most of the foam by melting it and cutting it out. The traditional balsa frame would require more time and labor because many spares and ribs would need to be made.

The team decided that a foam wing with carbon fiber skin would be the easiest to create for the mission requirements. A foam core would eliminate the need of additional structuring (the additional layers of carbon fiber required) if the core was later removed. In other words, it is more efficient to leave the foam core in, as it provides adequate structure and is light enough to keep the overall weight down.

The first step in the process of creating the wing is the choosing of an airfoil. Once this is done, the wing planform and cross sections are plotted in AutoCAD so that a computer can control a device to cut the foam. After the wing shape is cut from the original block of foam, cylinders are removed so that carbon rods may be inserted as spars. After this, the surface of the wing is lightly sanded to serve the double-purpose of smoothing and making it more accepting of the epoxy for the next step.

Following the creation of the wing, a layer of carbon fiber is added for protective shielding and added structure. A carbon fiber skin greatly increases the wing's stiffness and resistance to low-speed puncture damage. For this step, carbon fiber cloth is cut in the general shape of the wing planform. Next, it is laid on the wing and 18-hour-drying epoxy is applied to the assembly. Any folds or rough areas are worked out of the cloth, and the assembly is placed into a vacuum bag so that for the drying time, the fiber will completely form to the wing shape and any unneeded epoxy will be channeled away from the wing. Finally, the dried assembly is removed from the vacuum bag and dry fringes are sanded down.

Now that the wing assembly is nearly complete, foam is cut from the assembly to make hard

points for connections to the landing gear, fuselage, and servo mounts. Balsa, or in the case of the landing gear, hard wood, is added to these voids. The ailerons are made of balsa wood and fitted to the trailing edge of the wing. Monocoat was then applied to the outside of the assembly to minimize friction.

Fuselage Construction

A similar design question was posed in the construction of the fuselage as for the construction of the wings. The decision was again between the traditional balsa wood frame and foam-core design. The foam core design was again chosen, but the construction process of the fuselage did differ from that of the wings.



Team members Steve Frankino and Gilbert Romanowski inspect the longitudinal carbon rods that will support the fuselage

First, a frame of carbon rods was created. Rods were cut for a structure meant to fulfill structural requirements and service hard points for wing, empennage, and landing gear attachments. The cut rods were then fastened to each other using carbon fiber and 18-hour epoxy.

Around this, a foam fuselage was cut and placed *around* the frame. As in the case of the wings, carbon fiber is applied to the fuselage surface with 18-hour epoxy, and is fastened with the use of a vacuum bag, and the assembly was monocoated.

Empennage Construction

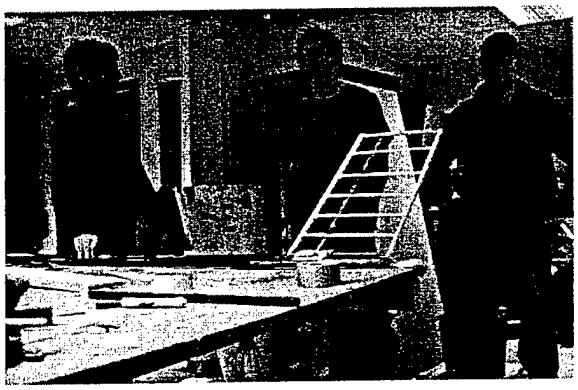
For the empennage, a balsa frame was chosen in order to give new team members experience with making balsa-framed aircraft parts. The system could have been made of carbon-fibered foam, but the experience was thought to be more valuable since the team will experience a change of leadership following this competition.

The tail is swept back and tapered. All but the root and tip ribs were simple balsa-cuttings. High stress areas of the tail, namely the spars and wing and root ribs were also made of balsa wood, but are twice as thick as regular ribs and a layer of carbon fiber is added to their surface with the use of the 18-hour epoxy and vacuum bag system. Ailerons, servo mounts,

and hard points for connection to the fuselage were then added. Finally, monocoat was added to the assembly to decrease friction.

Engine Mount

In the design stage, the team decided to build a plane with two motors in a pushing configuration. The added power will increase speed and controllability. A pushing configuration was chosen because of the advantage of conserving the air's momentum. In a pulling configuration, a propeller accelerates the flow, which, soon after loses momentum when it hits the fuselage. With a pushing configuration, the air flows around the plane, and through the propellers, which will maintain its momentum longer because there is no object that will reduce the speed of the flow.



Aditya Svaze, Will O'Meara, and Gilbert Romanowski behind the finished tail structure.

The engines are mounted on aluminum plates which are attached to a carbon fiber-enclosed wood board. That board is fastened to carbon rods that serve as the wing spars, to save on structural weight by combining components that require stronger support. Firewalls, to prevent the hot motors from melting the foam, are made of aluminum.

Landing Gear

The landing gear will have a tricycle configuration. The main gear is attached to hard points on the wings. The struts are made of aluminum and the tires are store bought. The nose gear will be purchased and made of aluminum with a built in shock absorber.

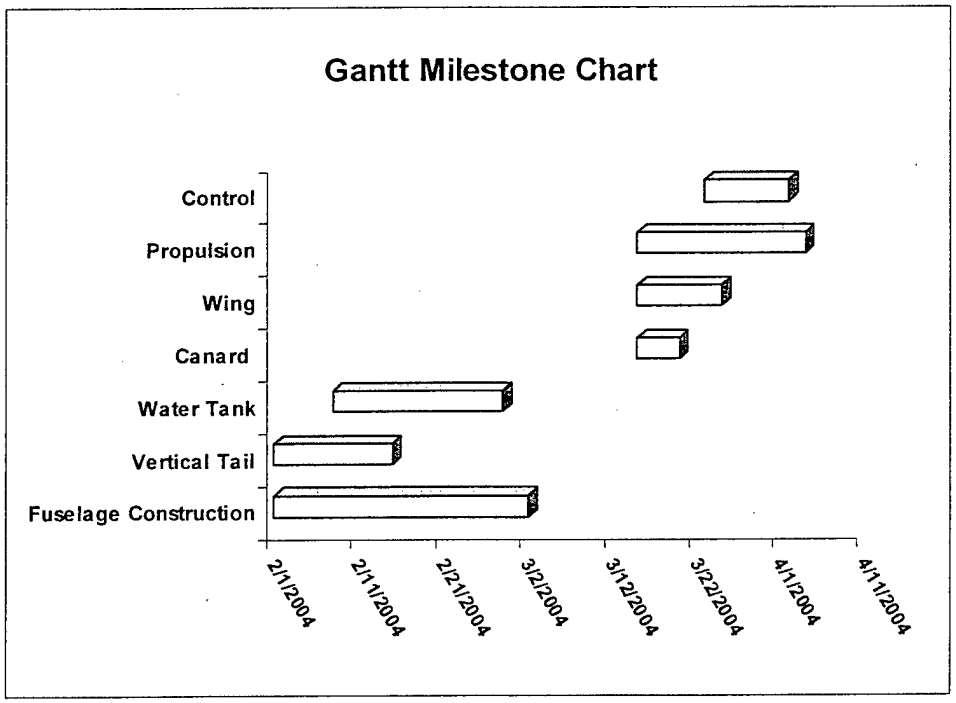
Figures of Merit

Figures of merit are important in deciding which direction to take in a manufacturing process. The attributes chosen for the figures of merit (Table 6.1) are cost, weight, strength, ease and desirability. Cost and weight were studied to find the lightest, cheapest materials. Strength refers to the given material's resistance to fatigue and damage with high load factors. The 'ease' category deals with the manageability of the work required for that type of material. Desirability relates the group's preference to the material. It is reflected in the tail design materials. For example, even though it was more difficult, the team decided to create a balsa-framed tail to give our newer members experience in wood framing.

The way the chart works is that each member voted on weights, which were averaged. Each member then voted on the score of each material for each piece in each attribute. This score was multiplied with the weight. These products were summed to give a material an actual score. The materials and methods with the highest scores won.

FOM Manufacturing							
		Cost	Weight	Ease	Strength	Desire to Use	
	Weight	0.2	0.8	0.4	0.6	1	total
Tail	Balsa	8	5	10	5	10	22.6
	Foam	10	5	5	3	2	11.8
	Carbon	5	8	6	10	6	21.8
Fuselage	Balsa	2	2	10	5	2	11
	Foam	10	8	8	3	8	21.4
	Carbon	5	10	5	10	10	27
Wing	Balsa	2	2	10	5	2	11
	Foam	10	8	10	3	8	22.2
	Carbon	5	10	5	10	10	27
Water Tanks	Plastic	8	5	2	5	5	14.4
	Carbon	5	10	8	10	10	28.2

Table 6.1 Rated FOM Chart for Manufacturing



VII. Testing Plan

The testing of the aircraft is a very important part of the design process. A thorough testing plan serves a dual purpose. The main purpose is to validate the airworthiness of the aircraft. It ensures that the analysis that took place during the preliminary and detail design phases is valid, and it also validates that the aircraft has been manufactured using the proper methods and quality of construction. The second purpose testing serves is to determine the handling qualities of the aircraft. The pilot should have an adequate idea of how the plane handles and if special attention needs to be given to problem areas. Flight testing can assist in honing in on the best possible handling qualities the aircraft can achieve. The tests performed were basic, yet thorough. Each test plays an integral part in assuring the survivability and optimal performance of the aircraft.

	STATIC AND GROUND TESTS	PURPOSE	COMPLETION DATE
1	Spar and Longeron Testing	To verify the strength of the carbon fiber rods	24-Jan
2	Battery Testing	To determine the endurance of the batteries with propellers attached	15-Mar
3	Static Thrust Test	To determine the maximum thrust output of the motors	15-Mar
4	Water Container Testing	To verify that the carbon fiber water container is water tight	16-Mar
5	Antenna Range	To determine the maximum effective range of the radio controller	17-Mar
	FLIGHT TESTING		
6	Full Power Ground Run	To determine acceleration and take off distance	2-Apr
7	Stall Throttle Setting	Determine the minimum throttle setting without incurring a stall	2-Apr
8	Pitch Control	Determine pitch characteristics of aircraft	3-Apr
9	Yaw Control	Determine yaw characteristics of aircraft	3-Apr
10	Roll Control	Determine roll characteristics of aircraft	3-Apr

Test 1: The **Spar and Longeron** strength test was performed to ensure that the carbon fiber rods could withstand the bending imposed on them. We considered that the maximum load that could ever act on the wing was 30 lbs at 4 ft. We placed a 30 lb weight at the end of the rod and it tested successfully.

Test 2 and Test 3: The battery supply will be tested for endurance while the static thrust is measured simultaneously. We will charge the battery supply, and place the throttle setting to the maximum power setting. We will connect power and take static thrust readings with a spring scale to determine the effect of battery fatigue on thrust. Thrust readings can be taken every 15 seconds until the batteries have been exhausted. A plot of Thrust vs. Time can then be generated.

Test 4: The **Water Container** test will simply consist of testing the container for waterproof and also testing the release mechanism.

Test 5: The **Antenna Range** will be tested. The plane will remain stationary while the controller makes inputs to the flight controls. The distance at which the inputs begin to dim will be over effective range.

Test 6: The **Full Power Ground Run** will test the acceleration of the aircraft on take off roll. The test will be performed three times: dry (no payload), ½ payload, and full payload. Time recordings will be made at intervals of 5 yards. A plot of Displacement vs. Time will be mad to determine the acceleration of the aircraft. The comparison of the three tests will allow us to determine the take off distance and also determine an optimal payload (with respect to take off).

Test 7: The **Stall Throttle Setting** test will determine the minimum throttle setting without incurring a stall. The conditions will be under no wind. A mark will be made on the throttle controller 15% above the "No wind throttle stall" setting. This is aimed to avert unintentional stalling of the aircraft, though wind will not have been considered. The test will be performed with both a full payload and no payload. The stall throttle setting with full flaps deployed will also be tested. This particular test will simulate a landing; however, the aircraft will be at a safe altitude so as not to crash under the stall condition. Once again the test will be performed with no payload (which is anticipated), but also with a full payload in the event of an electrical or mechanical failure which prevents the release of the water.

Dynamic Stability

The following tests will test the dynamic stability of the aircraft. Here we are looking to evaluate how well the aircraft responds to flight control inputs. We are going to evaluate the tendency of the aircraft to return to its initial trim position after inputs have been made.

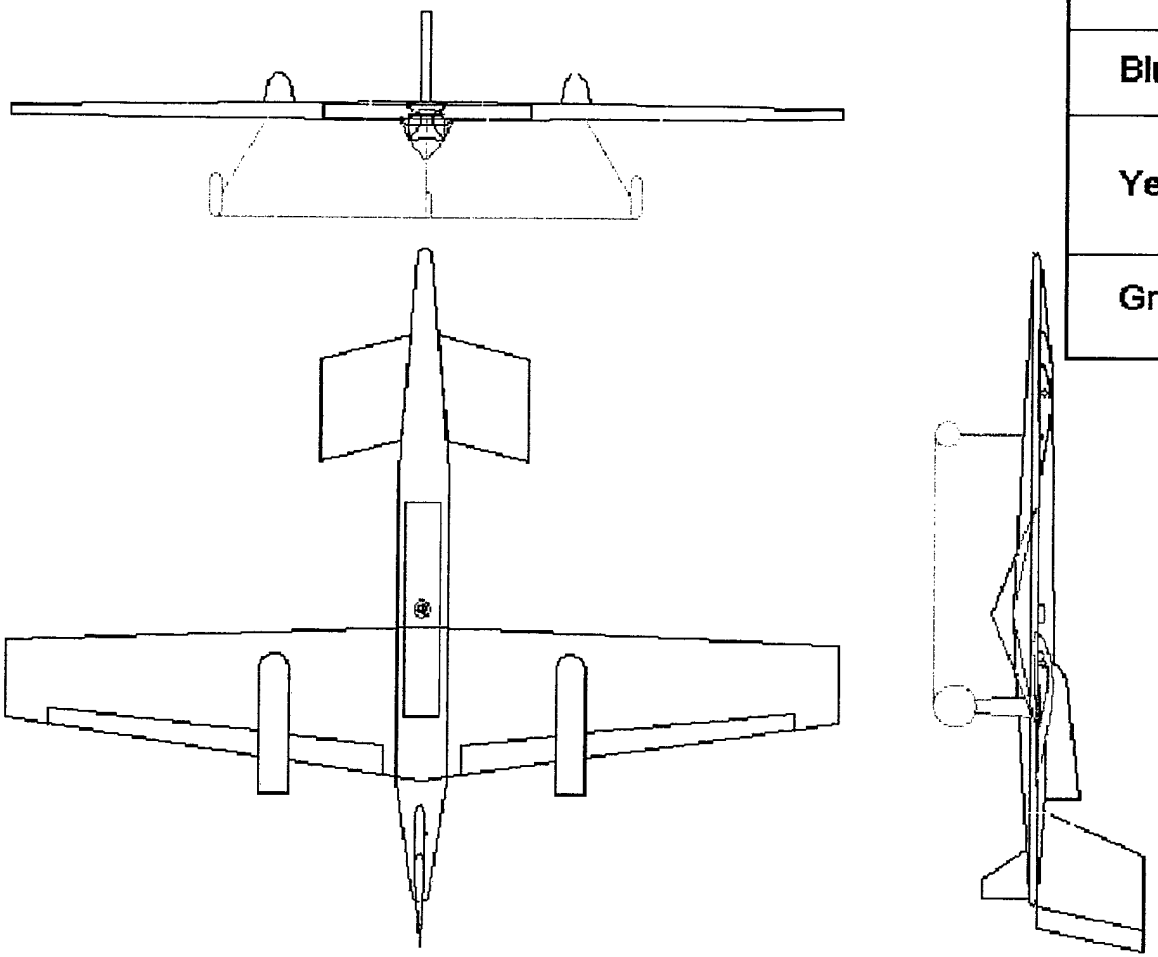
Test 8: The **Pitch Control** test will be the first of the stability and control tests to be performed. The maximum rate of climb will be determined by timing how long it takes the aircraft to reach an altitude of 40 ft. Once again the test will be performed with no payload and a full payload. The test will also include controlled oscillations and also step inputs to see if oscillatory motion can be induced. It is hoped that the step inputs only cause very mild oscillatory motion and that the aircraft settles quickly. General handling qualities such as responsiveness of the inputs and the amount of deflection required will be evaluated.

Test 9: The **Yaw Control** test will test the general handling qualities of the aircraft when rudder inputs are made. The likelihood to induce a stall after a prolonged rudder input will be tested. If rudder inputs cause too much or too little yaw motion, adjustments can be made by either changing the amount of deflection or changing the size of the flight control surface.

Test 10: The **Roll Control** test will test the time it takes to bank through 90 degrees. It will

also test the general characteristics of the roll motion of the aircraft. We will also attempt to estimate the minimum turning radius.

An optional test that may be performed is a single motor landing simulation. This test depends on our comfort level with the aircraft. It may be useful in the event that the battery supply is running low. The ability to maintain a controlled descent with only one engine may extend the endurance once the water has been discharged.



White - Outline of Aircraft

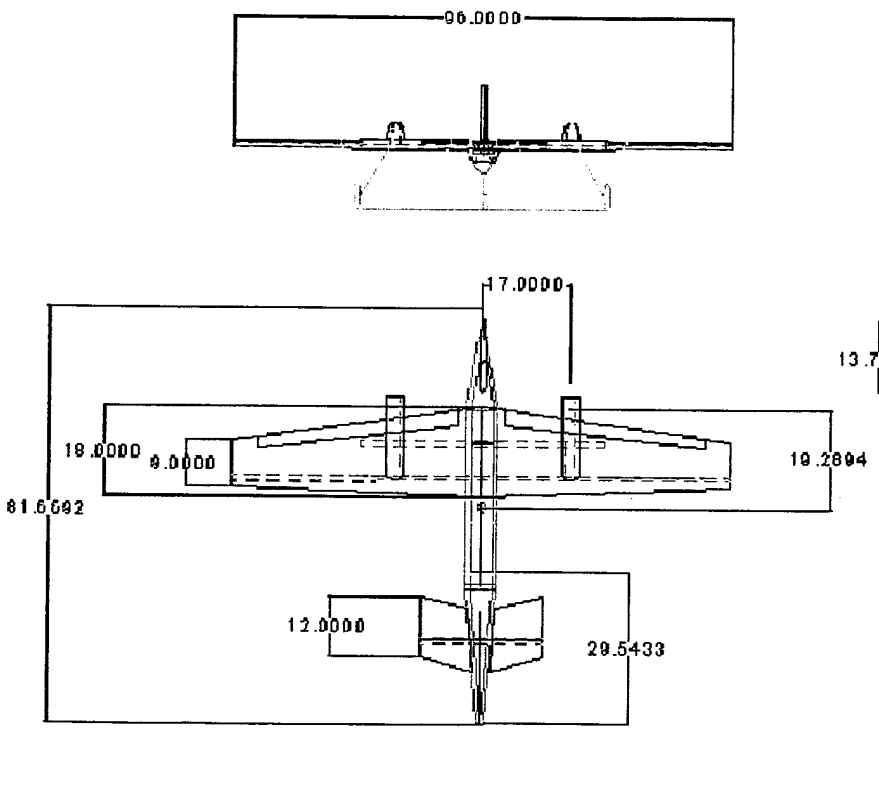
Blue - Water Tank

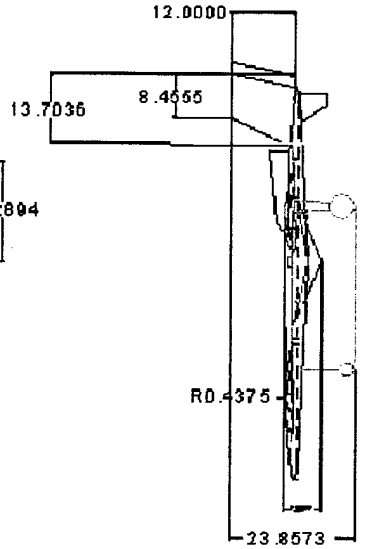
Yellow - Motors and Props

Green - Landing Gear

University at Buffalo	Buffalo Bomber	General 3-view	DBF 2003/04	Figure 1

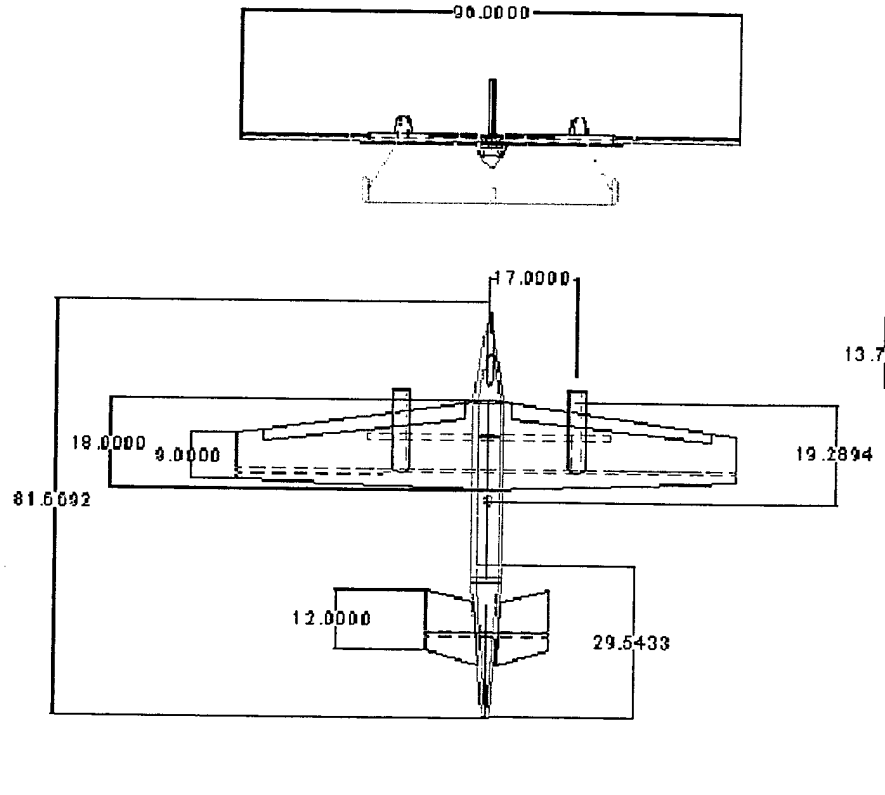
Color	Purpose
Purple	Structure
Yellow	Propulsion
Green	Landing Gear
Blue	Water Tank

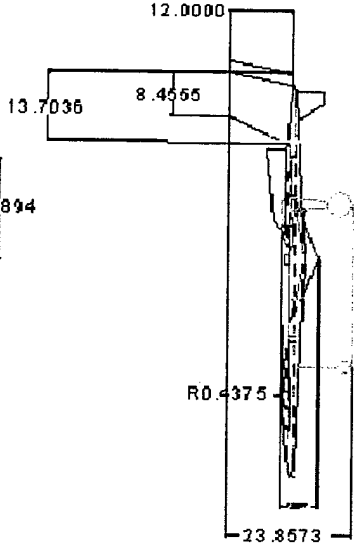




University at Buffalo	Buffalo Bomber	Detailed 3-view	DBF 2003/04	Figure 2

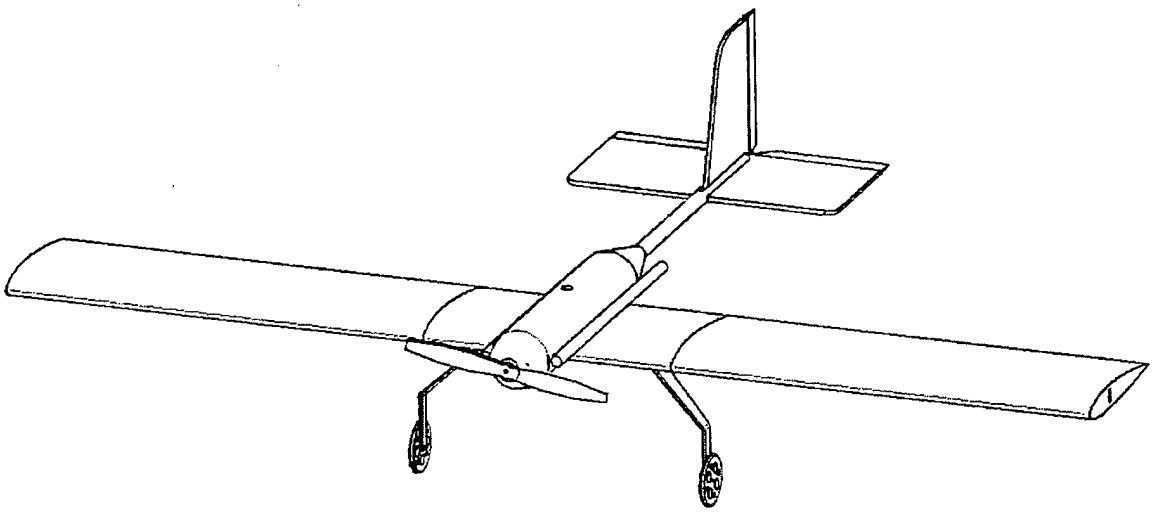
Color	Purpose
Purple	Structure
Yellow	Propulsion
Green	Landing Gear
Blue	Water Tank





University at Buffalo	Buffalo Bomber	Detailed 3-view	DBF 2003/04	Figure 2

2003/04 AIAA Cessna/ONR Student Design / Build / Fly Competition Final Report



"Swoop-A-Loop"

Miami University

Design Team:
Grant DeGeorge
Bob Petrochic
Kyle Riddle
Fabiano Santin
Sarah Schilling
Justin Soltani
Brian Sweeney
Todd Van Dyke

March 2004

Table of Contents

Title	Page #
List of Figures	iii
List of Tables	iv
1. Executive Summary	1
2. Management Summary	2
3. Conceptual Design	4
3.1. Mission Requirements	4
3.2. Figures of Merit	6
3.3. Conceptual Design Aircraft Configurations	9
3.4. Results and Conclusions	13
4. Preliminary Design	14
4.1. Design Parameters	14
4.1.1. Wing	14
4.1.2. Fuselage	21
4.1.2.1. Material	21
4.1.2.2. Reservoir	22
4.1.2.3. Entrance and Exit of Water	23
4.1.2.4. Ram Air	24
4.1.2.5. Placement and Location of Parts	24
4.1.3. Tail	25
4.1.4. Landing Gear	25
4.1.5. Propulsion System.	25
4.2. Testing and Analysis	26
4.2.1. Wing	27
4.2.2. Fuselage	27
4.2.3. Water Drainage	28
4.2.4. Prototype	31
4.3. Results and Conclusions	31
4.3.1. Wing	32
4.3.2. Fuselage	32
5. Detail Design	32
5.1. Component Selection	33
5.1.1. Wing	33
5.1.2. Fuselage	37

5.1.3. Tail 39
5.1.4. Landing Gear 40
5.1.5. Control Systems 39
5.2. Competition Aircraft Rated Aircraft Cost 41
5.3. Drawing Package 43
5.3.1. 3-View Drawing Aircraft Configuration 44
5.3.2. Aircraft Assembly 45
5.3.3. Exploded Aircraft Assembly 46
5.3.4. Fuselage Configuration. 47
5.3.5. Tail Assembly 48
5.4. Aircraft Specification Table 49
6. Manufacturing Plan and Processes 50
6.1. Process for Manufacture 51
6.1.1. Wing 51
6.1.2. Fuselage 53
6.1.3. Tail 54
6.1.4. Landing Gear 54
6.2. Manufacturing Milestone Chart 55
7. Testing Plan 55
7.1. Objectives 55
7.2. Testing Schedule 56
7.3. Testing Checklist 56
7.4. Component and Full Aircraft Testing of Prototype 57
7.5. Conclusions Static and Dynamic of Prototype 57
7.6. Component and Full Aircraft Testing and Conclusions of Final Design 57

List of Figures

Title	Page #
Figure 1 – Actual Gantt Chart	3
Figure 2 – Competition Flight Pattern	5
Figure 3 – Traditional Aircraft Configuration	10
Figure 4 – Blended Body Aircraft Configuration	11
Figure 5 – Tube Aircraft Configuration	12
Figure 6 – Airfoil Geometry	16
Figure 7 – Velocity Distribution	17
Figure 8 – Flow Field	18
Figure 9 – Lift-Drag Plot	19
Figure 10 – Wing Loading	20
Figure 11 – Stall Speed	20
Figure 12 – Baffle Design	23
Figure 13 – Ram Air Design	24
Figure 14 – Fuselage Drain Test: Setup 1	29
Figure 15 – Fuselage Drain Test: Setup 2	29
Figure 16 – Fuselage Drain Test: Setup 3	30
Figure 17 Thrust and Drag vs Speed (18/14 propeller)	34
Figure 18 Thrust and Drag vs Speed (18/12 propeller)	35

List of Tables

Title	Page #
Table 1 – Work Breakdown Structure	2
Table 2 – Breakdown of Rated Aircraft Cost Equation	7
Table 3 – Selection Matrix of Aircraft Configuration	13
Table 4 – Fuselage Material Properties	22
Table 5 – Fuselage Testing Results	28
Table 6 – Wing BOM	37
Table 7 – Fuselage BOM	37
Table 8 – Fuselage Length Based on 4-Liter Payload	38
Table 9 – Water Drainage System Evaluation	38
Table 10 – Tail BOM	40
Table 11 – Competition Aircraft Rated Aircraft Cost	43
Table 12 – Wing Manufacturing Plan	51
Table 13 – Fuselage Manufacturing Plan	54
Table 14 – Tail Manufacturing Plan	54
Table 15 – Test Schedule	56
Table 16 – Pre-Flight Checklist	57

1. Executive Summary

The purpose of this report is to provide a review of the steps taken and decisions made in the research and development of an RC model airplane for entry into the 2003/04 AIAA Design, Build, and Fly competition. This year's competition requires the designed aircraft to accomplish two missions set forth by the AIAA. The first mission is a heavy lift, slow flight "Fire Fight" where the aircraft will carry water and dump it during a specified flight path. The second mission is a high speed "Ferry" flight, where the plane will complete a specified flight path without a payload of water. From these missions, design requirement parameters have been based. These consist of the aircraft's range and payload, takeoff and landing distances, and maneuverability requirements.

For the competition, the design process was divided into three stages. These included the conceptual design, preliminary design, and finally a detail design stage of the final aircraft. Each stage required particular attention to different aspects of completing the competition. These aspects have allowed understanding of key elements of the competition, possible plane configurations, analysis of these configurations, and testing supporting the decision for the final aircraft design.

During the conceptual design stage, the focus was to look at the scoring aspects of the competition to how the total score could be maximized, while keeping the rated aircraft cost to a minimum. Analysis of the missions and RAC, which reflect the complexity and technology of the airplane design, was done through the use of FOM tables. The tables allowed the variable components of preliminary designs to be entered and have calculated results returned for evaluation. Based on these calculations, sketches were developed that aided in approximating wing and tail geometry, fuselage shape, and external and internal locations of major airplane components such as the engine, payload, landing gear, and batteries. These approximations have also allowed for an estimate of the weight and aerodynamics of the aircraft.

Analysis was significant during the preliminary design stage. Possible designs and component configurations were evaluated and tested to provide evidence of the best choice for the aircraft. The analysis and testing of different aspects of the plane, components could be strategically altered to find the best design to optimize flight performance while completing the competition mission requirements.

Throughout the detailed design stage of the project, an advanced framework was developed for the components of the plane that will be built for further analysis and experimental testing. This was done with actual flight testing through the construction of a prototype gas powered plane. During this stage, the manufacturing and testing plans were prepared and evaluated accordingly so the team's skills and effort could be best applied.

From the information gathered in all stages of design, the construction of a final aircraft design was completed. As a result of extensive planning, design analysis, and utilizing the information that was obtained, the best performance could be expected from the final aircraft design.

2. Management Summary

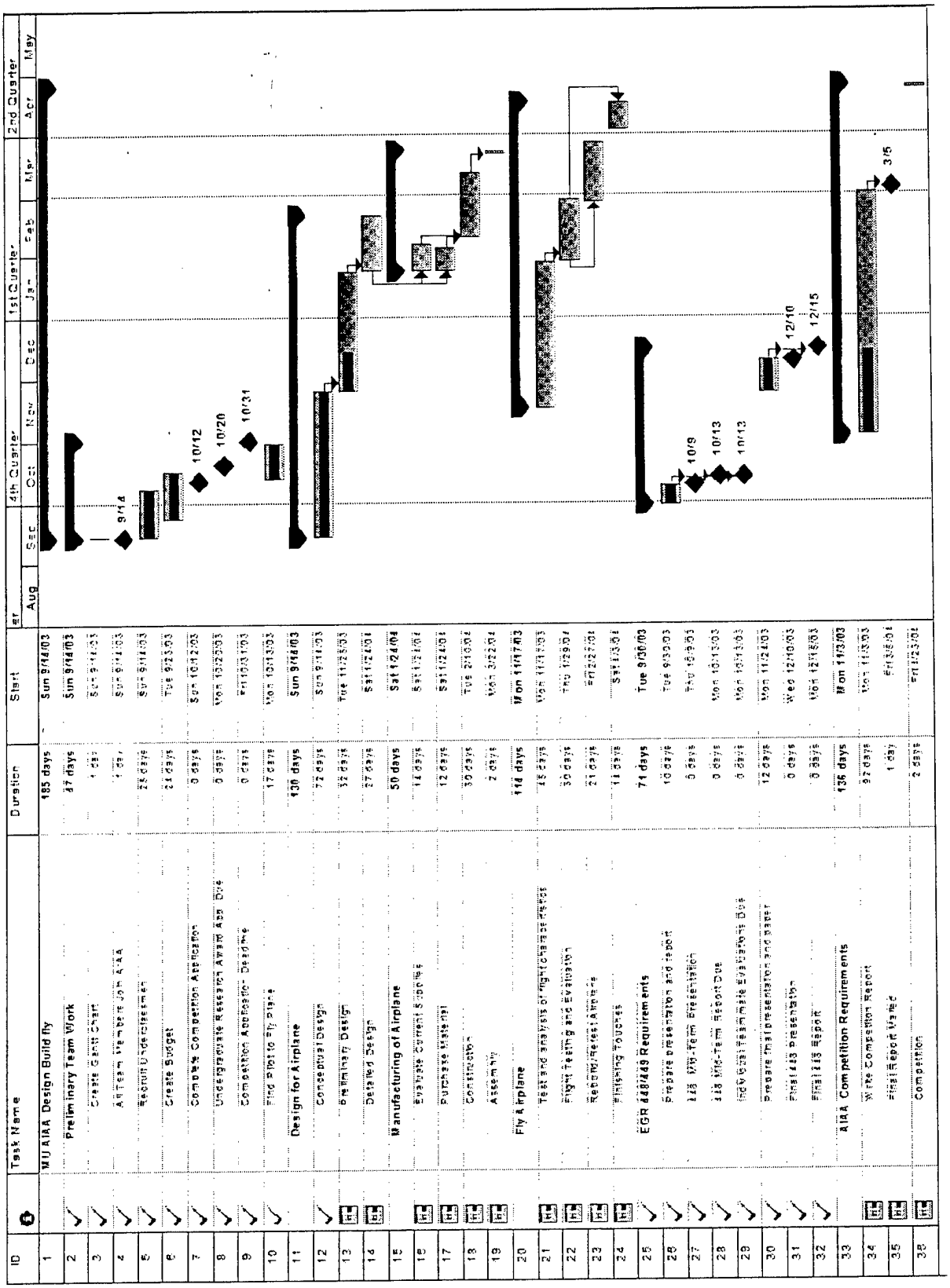
The design team for the competition consists of eight Miami University students in Manufacturing Engineering, Engineering Management, and Physics. Since none of the participating students were in aeronautical fields of study, extensive research was needed in each area of the plane to learn part usage, design considerations, and design strategies. Therefore, the work was broken down into parts of the plane to allow for in-depth research and mastery in every design area. Those research and design areas were as follows: wing, fuselage, propulsion system, tail and landing gear.

To better facilitate the scope of the project, the team also divided up into area directors. Area directors would be directly responsible for ensuring that their area of the project would be completed. All of the areas overlap to allow for every member to contribute and understand to each step of the process. Any member that is not an area director would be placed in each area as needed. The five major areas that were developed were the finance, design, construction, report and organization, and research and development. The work breakdown is shown in Table 1.

Table 1 – Work Breakdown Structure

Team Member	Year	Area Director	Research Director
Kyle Riddle	Senior	Finance and Work Organization	Propulsion System
Bob Petrochic	Senior	Research	Landing Gear
Todd Van Dyke	Senior	Design	Wing
Grant DeGeorge	Senior	Report and Team Contact	Fuselage
Sarah Schilling	Senior	Construction	Tail
Brian Sweeney	Junior		
Justin Soltani	Sophomore		
Fabiano Santin	Freshman		

The team met weekly with its advisors James Stenger, Richard Walker, and Tom Schroeder, to discuss the progression of the project and discuss any design issues. In addition, the team met once a week to work on analysis or construction of the design. Then, the area directors met weekly to discuss the progress of their area and what future work was needed. All work tasks were assigned as shown in the work breakdown structure in Table 1, while all goals and milestones were monitored on the team Gantt chart shown in Figure 1.



ID	Task Name	Duration	Start
1	MU AIAA Design Build fly	185 days	Sun 9/14/03
2	Preliminary Team Work	27 days	Sun 9/14/03
3	Create Gantt Chart	1 day	Sun 9/14/03
4	All Team Members Join AIAA	1 day	Sun 9/14/03
5	Research Undergraduate	26 days	Sun 9/14/03
6	Create Budget	21 days	Tue 9/23/03
7	Complete Competition Application	0 days	Sun 10/12/03
8	Undergraduate Research Award App Due	0 days	Mon 10/20/03
9	Competition Application Deadline	0 days	Fri 10/31/03
10	Ship Plane to Fly Place	17 days	Mon 10/13/03
11	Design for Airplane	136 days	Sun 9/14/03
12	Conceptual Design	72 days	Sun 9/14/03
13	Preliminary Design	52 days	Tue 11/25/03
14	Detailed Design	27 days	Sat 1/24/04
15	Manufacturing of Airplane	50 days	Sat 1/24/04
16	Evaluate Current Supplies	14 days	Sat 1/24/04
17	Purchase Material	12 days	Sat 1/24/04
18	Construction	30 days	Tue 2/10/04
19	Assembly	2 days	Mon 3/22/04
20	Fly Airplane	144 days	Mon 11/17/03
21	Test and analysis of flight characteristics	45 days	Mon 11/17/03
22	Flight Testing and Evaluation	30 days	Thu 1/29/04
23	Record/Review Analysis	21 days	Fri 2/27/04
24	Finishing Touches	11 days	Sat 3/13/04
25	EGR 448/449 Requirements	71 days	Tue 9/30/03
26	Prepare presentation and report	10 days	Tue 9/30/03
27	448 Mid-Term Presentation	0 days	Thu 10/9/03
28	448 Mid-Term Report Due	0 days	Mon 10/13/03
29	Individual Team Member Evaluations Due	0 days	Mon 10/13/03
30	Prepare final presentation and paper	12 days	Mon 11/24/03
31	Final 449 Presentation	0 days	Wed 12/10/03
32	Final 449 Report	0 days	Mon 12/15/03
33	AIAA Competition Requirements	136 days	Mon 11/3/03
34	Write Competition Report	92 days	Mon 11/3/03
35	Final Report Varied	1 day	Fri 3/5/04
36	Competition	2 days	Fri 4/23/04

Figure 1 – Actual Gantt Chart

3. Conceptual Design

3.1. Mission Requirements

The airplane must complete two different types of missions, which are the "Fire Fight" and the "Ferry." The requirements for both of these missions are described below.

Fire Fight - This mission focuses on a slow flight with heavy lift. The mission begins with an empty aircraft. The aircraft will complete two sorties. The sorties consist of the aircraft being loaded, taking-off, then dumping its water during the downwind leg, and return to land. The water will be loaded into the aircraft by the team using two litter bottles. Rules state that the maximum water capacity in the aircraft is four liters, with a maximum dump orifice of ½ inch. The water can only be dumped within the downwind leg between the turn markers. The aircraft must be empty upon landing after the final lap or else a three-minute penalty will be added to the total flight time. The starting and ending weight of the ground-based storage "tanks" will determine the weight of the water loaded. This mission has a difficulty factor of two. The single flight score for the "Fire Fight" mission is calculated by taking the difficulty factor multiplied by the pounds of water then dividing by the mission time.

Ferry - The Ferry mission is based on a fast flight with no water payload. For this mission, the aircraft must take-off, complete four laps, and land. The difficulty factor for this mission is one. The "Ferry" single flight score is the difficulty score dividing by the mission time.

In addition to the specific mission's requirements, many general competition requirements must also be met. The main specifications for the competition are:

Aircraft Storage - The aircraft is required to disassemble and fit into a 2 ft by 1 ft by 4 ft box. All payloads are required to be mechanically secured for flight. Therefore, the design of the aircraft needs to consist of parts and joints that can be easily assembled. However, this is a consideration for later on in the preliminary design due to the fact that the focus of the conceptual design is to select the best basic aircraft design, instead of choosing specific components of the design.

Take-off and Landing - For both of the missions the airplane must take-off within 150 feet. The wheels must be off the runway within this distance or a score of zero will result for that flight. This constraint will need to be analyzed due to the weight of the water for the fire fight mission. The aircraft must also land on the runway. It may however roll off after crossing the finish line.

Flight Pattern – Figure 2 illustrates the required flight pattern to be flown during the competition.

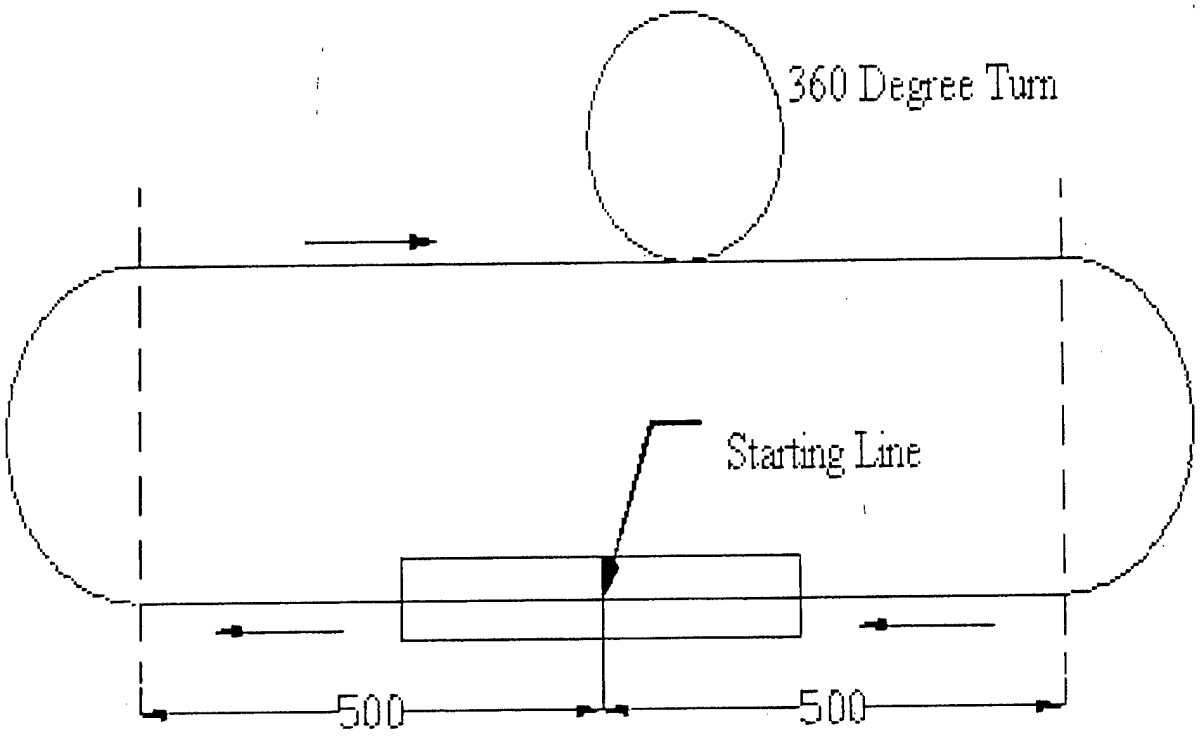


Figure 2 – Competition Flight Pattern

The flight pattern for both missions includes a right hand turn 500 feet from the starting line. Next in the downwind leg, the plane flies for 1000 feet with a single 360° turn in the direction opposite of the base and final turns. The plane then executes another right hand turn and lands. All laps in both of the missions must include the 360° turn. The aircraft can fly at any safe altitude as determined by the judges. The exact flight pattern shown in Figure 2 and altitude will be considered in the preliminary design since the conceptual design does not consider detailed aircraft features.

Rated Aircraft Cost - The rated aircraft cost (RAC) is an equation that takes into account all components of the competition aircraft and uses them to calculate a cost. The total team score is inversely proportional to the RAC of the designed aircraft; therefore minimizing the RAC is crucial to the overall competition score. The RAC factors in the complexity of the design. By simplifying the design the RAC can be lower.

Team's Overall Score - The maximum time for a mission is ten minutes. The team will have five flight attempts and the best single flight score of each type of mission will be summed for the total flight score. The winner is based on the team with the highest overall score calculated using the following equation:

$$\text{Overall Score} = \frac{\text{Written Report Score} * \text{Total Flight Score}}{\text{Rated Aircraft Cost}}$$

where the total flight score is the product of the Fire Fight DF and lbs water, divided by the mission time plus the product of the Ferry DF and mission time:

Total Flight Score = (Fire Fight DF * Lbs_Water / Mission_Time) + (Ferry DF * Mission Time)

The Rated Aircraft Cost, in thousands of dollars, is calculated from:

Rated Aircraft Cost, $ (Thousands) = (A*MEW + B*REP + C*MFHR)/1000

Where A, B, and C are contest-specified multipliers, MEW is the Manufacturers Empty Weight (the weight of the aircraft without load), REP is the rated engine power, and MFHR is the manufacturing man hours (determined from contest stipulations). The RAC equation is discussed in more detail in the *Rated Aircraft Cost* portion of this report found in section *3.2*.

3.2. Figures of Merit (FOM)

In order to effectively develop a sufficient variety of feasible conceptual design ideas for the aircraft, it was necessary to generate a list of critical FOM's. These FOM's allowed our team to not only develop conceptual design ideas, but also assisted in evaluating and selecting the best possible conceptual design. Twelve figures of merit were developed to aid in the conceptual design process. Each of the twelve FOM's are listed below and explained in detail.

Rated Aircraft Cost (RAC)

The RAC as given in the 2003/2004 rules, played a vital role in the conceptual design of our aircraft. The RAC is a divisor in the final score of the competition. As the RAC decreases, the final competition score will subsequently increase. The equation used to calculate the RAC incorporates several design parameters of the aircraft. The parameters of the equation given above are explained in more detail in Table 2.

Table 2 – Breakdown of Rated Aircraft Cost Equation

A =	Manufacturers Empty Weight Multiplier ($300)
MEW =	Actual aircraft weight in pounds with all flight and propulsion batteries, but without payload
B =	Rated Engine Power Multiplier ($1500)
REP =	Rated Engine Power ((1+.25*(# engines-1)) * Total Battery Weight (lbs))
C =	Manufacturing Cost Multiplier ($20/Hour)
MFHR =	Manufacturing Man Hours [Sum of Work Breakdown Structure (WBS) hours] 1) WBS Wing • 10 hours/square foot • 5 hours * control function multiplier Possible Control Function Multipliers ailerons = 1 flaperons = 1.5 ailerons + flaps = 2 ailerons + spoilers = 2 ailerons + flaps + spoilers = 3 2) WBS Fuselage • 20 hours/cubic foot (body length * width * height) 3) WBS Empennage • 5 hours/Vertical Surface with no controls • 10 hours/Vertical Surface with controls • 10 hours/Horizontal Surface 4) WBS Flight Systems • 5 hours/servo or motor controller

Actual Cost

The actual cost was estimated by totaling all estimated costs of all components of the aircraft. This FOM was useful in determining which conceptual designs were financially reasonable and feasible. Actual cost was utilized in the conceptual design as a quantitative FOM.

Weight of Aircraft

Weight was calculated by summing the individual weights of all aircraft components. Minimizing the weight of our aircraft was essential to not only lowering the RAC, but also to the aircraft's flight abilities. Weight was also considered a quantitative FOM.

Ease of Construction

Ease of construction was considered a qualitative FOM. Estimations of the skill and difficulty level of each conceptual design were conducted. This determination of difficulty aided in determining which designs were reasonable, given the team's design and manufacturing skills and knowledge.

Time of Construction

Much like Ease of Construction, this FOM was employed as a qualitative measure for the conceptual design. Given the time frame allowed for completion of the project, it was necessary to account for an estimated amount of construction time for each conceptual idea.

Takeoff Distance

Takeoff distant was a major determinant for which conceptual design idea to implement. The equation used to calculate the qualitative value of takeoff distance provided our team a means with which to estimate if our aircraft could feasibly get off the runway in the 150 feet required in the AIAA DBF 2003/2004 rules.

$$\text{Takeoff Distance} = \frac{(1.44\ W)^2}{(g \times \rho \times S \times C_{Lmax} T)}$$

where W^2 is gross weight, g is acceleration of gravity, ρ is the density of air, S is the plan form of the wing, C_{Lmax} is the max coefficient of lift, and T is thrust.

Water Reservoir Drainage Time

Quite possibly the most important consideration, this FOM is weighted twice as much as all other factors as set forth in the RAC. Therefore, twice as much emphasis was placed on the importance of this quantitative measure. Knowing the water will ideally flow out of any given reservoir at a velocity of:

$$V = \sqrt{2gh}$$

where V is the velocity, h is the height of the water in the reservoir, and g is the gravitational constant, it was possible to obtain a general understanding of drainage time for any given conceptual design.

Time & Ease of Filling Water Reservoir

Yet another qualitative FOM, knowing the approximate level of difficulty and time to fill the water permitted an accurate development and selection of conceptual design ideas. Because the water cannot be pressurized prior or during filling of the aircraft water reservoir, the ease and time of filling the reservoir became a key FOM.

Speed Capability

This FOM was necessary to account for the high speed Ferry mission. Though the aircraft must be capable of holding and carrying a sufficient amount of water, the aircraft must also be able to fly with a certain level of speed. Speed capability was preliminarily utilized as a qualitative FOM to estimate flight speed of various conceptual designs.

Flight Characteristics

Flight characteristics were considered as a FOM because each conceptual design must be capable of exhibiting steady flight. Whether empty, full, or in the process of draining water, the aircraft must maintain steady, controllable flight.

Drag and Aerodynamics

Given the various characteristics of different conceptual design ideas, it was essential to determine which would display the more aerodynamic flight. Excessive drag will slow the aircraft, more rapidly drain the batteries, and create less efficient flight.

Ability to Disassemble and Fit in Box

As indicated in the 2003/2004 rules, the aircraft must be capable of being disassembled and must fit in a 2' wide x 1' tall x 4' long box. The design of the aircraft must allow for an assembly that permits easy deconstruction of the plane to completely fit into the storage box. This FOM was determined through qualitative analysis of the basic structure of the conceptual design ideas.

3.3. Conceptual Design Aircraft Configurations

Based on what was established as important design concepts from the FOM's, three conceptual designs for the aircraft emerged. These designs are the traditional design, the blended body design, and the tube design. The main design difference in the three conceptual configurations is the fuselage and water reservoir shape and design. Each of these configurations contained characteristics that uniquely satisfied certain portions of the FOM's considered. However, the concept that should be chosen for further development in the preliminary design stage is the one that best meets the FOM's in total. The three conceptual design aircraft configurations are explained in greater detail below.

Traditional Design

This first conceptual design idea contains more traditional aircraft characteristics. This design employs a full body fuselage from nose to tail. The water reservoir is positioned within the front portion of the fuselage and holds the weight of the water as close to the center of gravity of the aircraft as possible. A bladder within the front portion of the fuselage will contain the necessary water and will assist in keeping the water from splashing and shifting. A vent located in the front of the aircraft will allow the water to

drain more quickly, forcing air into the water chamber and pressurizing the bladder. The wings are positioned on top of the aircraft and the water drains from the bottom of the aircraft at the center of gravity. This conceptual design idea compares closely to an actual fire fighting aircraft. The RAC is relatively large and the wing area is substantial to accommodate for the added weight of a full body fuselage. Figure 3 illustrates this traditional design idea.

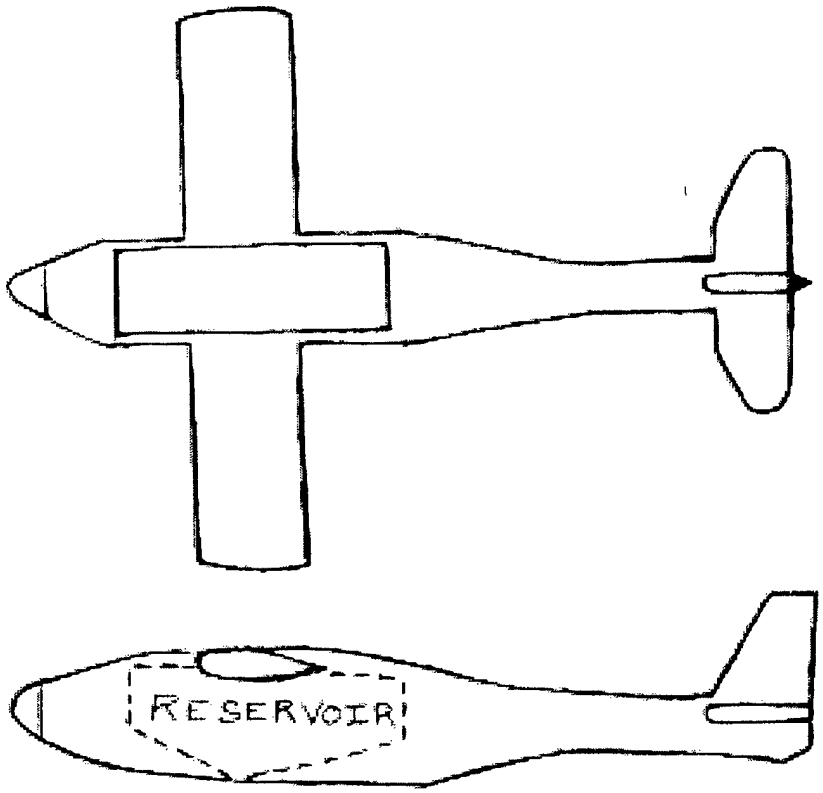


Figure 3 - Traditional Aircraft Configuration

Blended Body Design

The blended body conceptual aircraft configuration utilizes a more aerodynamic approach. The fuselage design is less bulky than the traditional design idea and therefore is a more economical design solution when looking at RAC. As stated in the 2003/2004 rules, "for a blended wing-body, body width is fixed at 18 inch." This design will fully utilize all 18 inches provided for the width of the fuselage and will therefore minimize the height of the fuselage. The water for the fire fight mission will be stored with blended body and wing area of the aircraft. The 18 inches provided for the width of the fuselage and an estimated 14-inch wing chord allows all 4 liters of water to fit into the wing/body configuration. Water is drained at the bottom of the blended body at the center of gravity of the aircraft. A bladder is again utilized to keep the

water from splashing and to minimize shifting. The blended body conceptual design is illustrated in Figure 4.

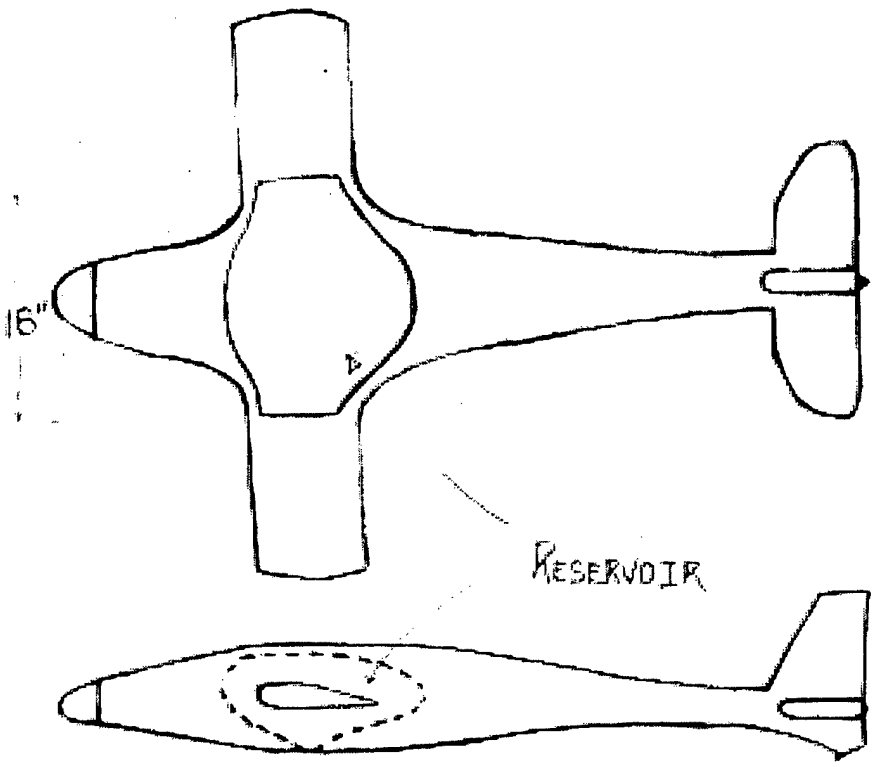


Figure 4 - Blended Body Aircraft Configuration

Tube Design

This final conceptual design idea employs the use of a rigid tube for the fuselage of the aircraft. The rigid tube allows the wings to be mounted in the middle of the fuselage. Spars run through the middle of the fuselage, providing the wings support in all directions. The water is stored within the tube and drains out of the back of the tube. To prevent the water from shifting within the tube, potentially throwing the aircraft out of steady flight, series of baffles are utilized inside the tube. Separate from the water storage chamber, the motor is mounted in the front of the tube. The tube is only as long as needed to store the water within a certain diameter of tubing. The end of the tube will lead into a tapered, streamlined portion of the fuselage, reducing the overall weight of the aircraft. The tube design is illustrated in Figure 5.

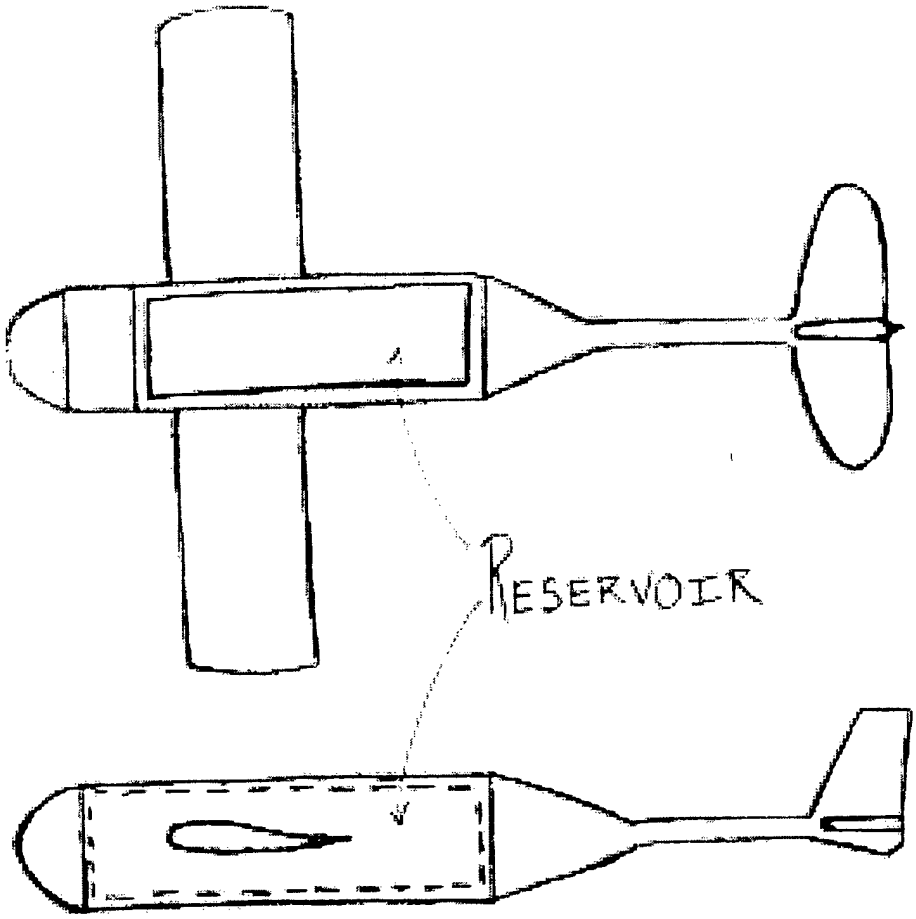


Figure 5 - Tube Aircraft Configuration

Selection of Aircraft Configuration

Through the use of a selection matrix, each of the three conceptual design aircraft configurations was compared to the twelve FOM's. Each FOM was assigned a weight of 1, 2, or 3. A '1' designates the FOM as the least significant, a '2' designates the FOM as moderately significant, and a '3' designates the FOM as critically significant in the design of the aircraft. These weights are used as multipliers in the selection process. Each of the three aircraft configurations was then assigned a rating between 1 and 3. Just as with the weights of the FOM's, a '1' indicates a low score and a '3' indicates a high score. The assigned weight for each of the FOM's was then multiplied by the rating for each of the respective conceptual configurations. A score was then obtained for each FOM of each conceptual design configuration. The scores were totaled and a final score for each of the three conceptual designs was

obtained, the highest score indicating the best concept. The matrix shown in Table 3 illustrates the selection and screening process.

Table 3 - Selection Matrix of Aircraft Configuration

Conceptual Design Ideas

		Traditional		Blended Body		Tube	
FOM's	*Weight (1-3)*	Rating (1-3)	Score	Rating (1-3)	Score	Rating (1-3)	Score
Rated Aircraft Cost (RAC)	*3*	1	3	3	9	2	6
Cost ($)	*1*	1	1	2	2	3	3
Weight	*3*	1	3	3	9	2	6
Ease of Construction	*2*	2	4	1	2	3	6
Time of Construction	*2*	2	4	1	2	3	6
Takeoff Distance	*3*	2	6	3	9	2	6
Drainage Time (x2)	*3*	2	6	1	3	3	9
Ease of Filling	*1*	3	3	3	3	2	2
Speed Capability	*2*	2	4	3	6	2	4
Flight Characteristics	*3*	3	9	2	6	3	9
Drag/Aerodynamics	*2*	2	4	3	6	2	4
Fit in Box/Disassembly?	*3*	2	6	2	6	3	9
TOTALS			53		63		70

As depicted in the matrix, the tube conceptual configuration received the highest conceptual design score with 70 quality points. The blended body scored the second highest with 63 points and the traditionally design placed last with 53 points. Based on this selection matrix, the tube conceptual configuration was chosen to be developed further in the future design stages.

3.4. Results & Conclusions

Through the use of twelve specific Figures of Merit and a development of three basic conceptual design configurations, it was possible to move forward in the design process. The selection matrix provided the ability to select the best of the three conceptual design configurations and move on with this specific design into the preliminary design of the aircraft. The tube conceptual design was chosen based on its highest selection matrix score. With a tube fuselage aircraft selected, the design was then further

developed in the preliminary design stage. The preliminary design of the aircraft provides a more detailed design of the aircraft and its components.

4. Preliminary Design

4.1. Design Parameters

To best understand how the conceptual design of the chosen plane should be further developed in the design stage, it was imperative to section the plane into its most important components. These components consist of the wing, fuselage, tail, landing gear, and propulsion system made up of the propeller, motor, servos, and battery.

4.1.1 Wing

When analyzing the wings, the first thing that needed to be taken into account is how much weight the aircraft intended to carry. The weight is going to change during the Fire Fight mission when the water is drained. Since the water will be draining in flight, it needs to be assured that it does not adversely affect the center of gravity of the airplane while it is in the air. The center of gravity on a classic airplane typically runs through the wing, which supports the aircraft during flight. Keeping this in mind, it is imperative that all the water that is carried needs to be focused around this point. In order to carry the water with the limited amount of power that is available, an appropriate amount of lift must be provided by the wing. Many different things govern this lift. First and foremost, wing span and chord length will ultimately govern how much lift is possible to create with the wing. The type of airfoil that is selected for the wing will also determine lift.

Wing Span and Chord Length

Using information obtained from the MaxCim web site, many different wing loading options for the plane were calculated. The wing loading is affected by total wing span, which is multiplied by the chord length to give square footage of the proposed wing. Using some simple math, a wing loading in oz/sq. ft is able to be calculated when an estimated weight of the airplane is added. For each wing loading up to 30 oz/sq. ft, there are suggestions for what size electric motors to use. When this was examined, it states that if you are "Over 30 oz/sq. ft - You're on your own!" Typical values that have been reached have been from 40 to 80 oz/sq. ft. One positive aspect is a motor is available to us which provides considerably more thrust than any of the motors referred to here.

Aspect Ratio

Another item governing wing selection is the aspect ratio of the wing. Aspect Ratio refers to how the shape of the wing is defined. A wing with a high aspect ratio has a very large wingspan with a small chord. These wing types provide less drag with slightly higher lift, whereas a wing with a lower aspect

ratio could have the same square footage but would be shorter with a larger chord. This wing would in turn have more drag and slightly less lift. With all great things there is a trade-off. It would seem that to maximize the performance of our wing, a design consisting of a very wide wing span with a small chord length is optimal. This brings in another consideration, which is the size of the box that the aircraft must for inside. Since the box is 4ft long, the plane must be designed to disassemble and fit into this confined space. While it is feasible to have wings that break down, to add in another section of wing that needs to be assembled and disassembled may bring into the picture more problems with structural integrity. This is something that is planned for further investigation. Additionally, formulas need to be derived to help analyze the stresses that would be put on the wing in these situations.

Airfoil Selection

Another criterion that must be evaluated is the selection of the airfoil. There are thousands of different airfoils that have already been designed and tested since Orville and Wilbur Wright proved to the world that man could accomplish powered flight 100 years ago. Airfoil selection requires many variables such as lift, drag, ideal angle of attack, and stall characteristics to be analyzed. While a thin airfoil is going to have less drag, and be quicker, it is going to have less lift and a higher stall speed. A thicker foil will have more lift but also more drag. Since lift is of greatest concern, due to the payload that must be carried by the aircraft, foils with thicker shapes that produce more lift were considered. One route is using a classic design that has good flight characteristics at a multitude of different airspeeds. One airfoil that fulfills these requirements is defined as an NACA 2412. This airfoil was defined in the 1933 Technical Report No. 460, "The Characteristics of 78 Related Airfoil Sections from Tests in the Variable-Density Wind Tunnel." The National Advisory Committee for Aeronautics, which was charted in 1917, became NASA with the National Aeronautics and Space Act of 1958. This airfoil is in the four-digit series of airfoils and each digit helps to define the overall shape of the airfoil. The first digit defines that the maximum chamber of the foil is 2% of the length of the chord and the second digit defines where this occurs, at 40% of the chord length measured from the leading edge. The last two digits signify the maximum thickness of the airfoil as being 12% of the total chord length. This airfoil has many redeeming qualities that still keep it in use today with many light aircraft and model planes alike.

Most of the research conducted for airfoil selection has been through the use of existing aeronautical knowledge. One of the main tools that were used for selecting the airfoil was a program called Javafoil. Javafoil is an on-line resource which allows a user to explore a variety of different airfoils while varying certain parameters. After the original research, it was concluded that using an NACA series airfoil with a maximum camber of 2% at 40% of the length of the airfoil was probably going to be the best course of action. With this criterion, the focus was on comparing three different NACA airfoils. The selection was narrowed to the NACA 2412, 2415, and 2418. It was initially thought that the 2412 would be the best choice based on its widespread continual use after seventy years in existence; however the main concern

was if it would have enough lift. Javafoil provided the illustrations of how each of these three airfoils would perform with their varied thickness. Figure 6 shows the first step in the analysis of the airfoil, defining its geometry.

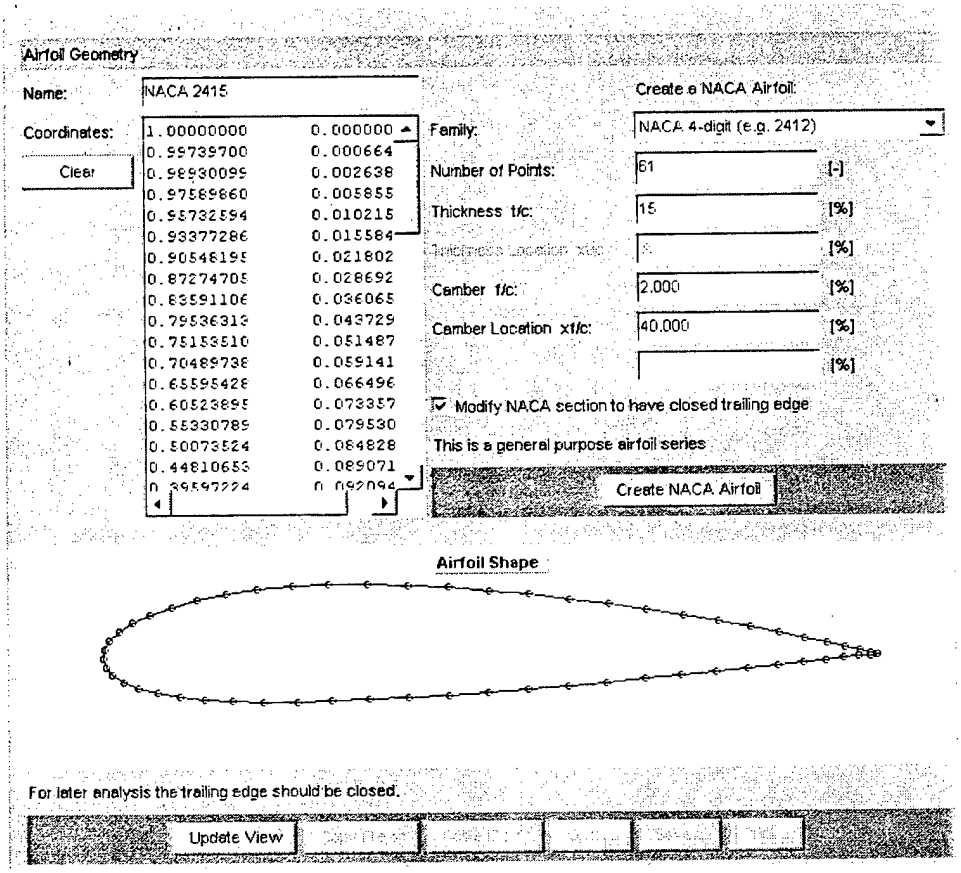


Figure 6 – Airfoil Geometry

After the design of the airfoil, there are several different aspects that can be analyzed, one of these aspects being the velocity distribution. The velocity distribution, as seen in Figure 7, shows the speed at which the air is flowing over the length of the airfoil, at various angles of attack. The velocity distribution is used to smooth out an airfoil to minimize drag. Since the airfoils have not been modified beyond their capacity, there is not much difference in the velocity distributions of the three designs.

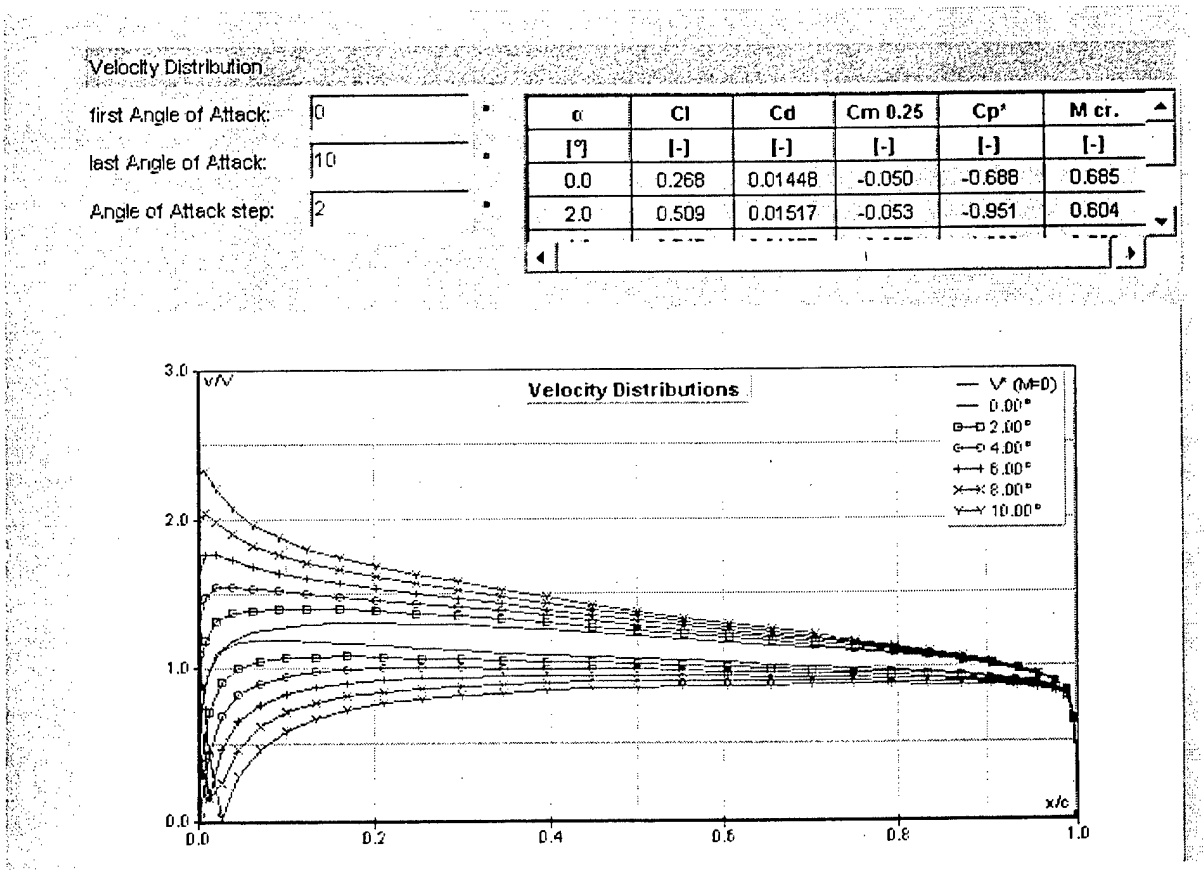


Figure 7 – Velocity Distribution

The flow field shows the actual pressure distribution and streamlines over the airfoil at various angles of attack. Figure 8 shows a NACA 2415 airfoil with a 10-degree angle of attack. This tool enabled the illustration and allowed the comparison of the pressure differences on the top and bottom of the airfoils at different angles of attack.

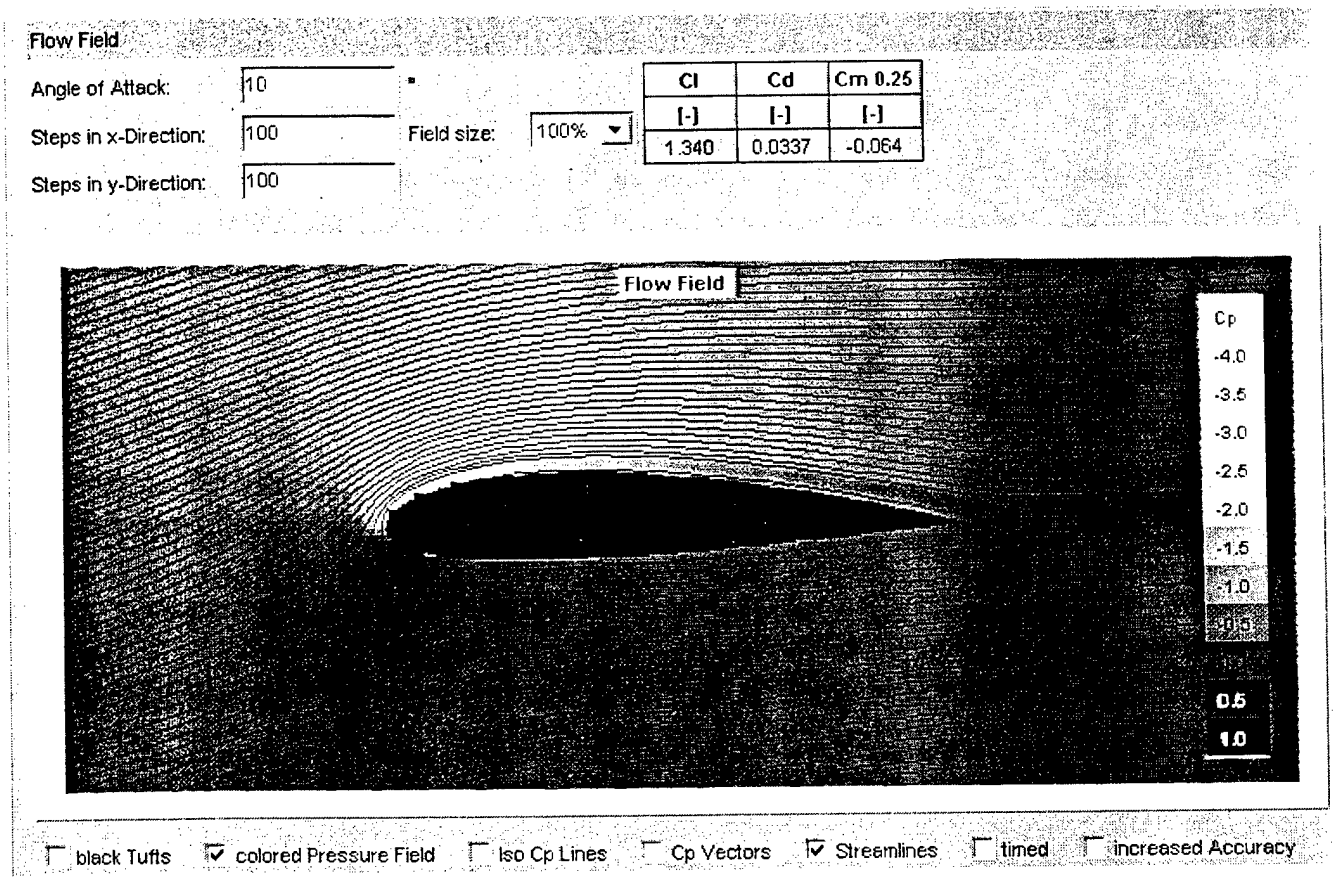


Figure 8 - Flow Field

By far the most important feature that was found using Javafoil was the lift-drag plot. Using this provided the information to find which of the airfoils being examined would give the best lift versus drag performance at various Reynolds numbers. It also allowed the exploration of which angle the airfoils would stop producing lift and stall out. Figure 9 shows the lift-drag plot for five different Reynolds numbers.

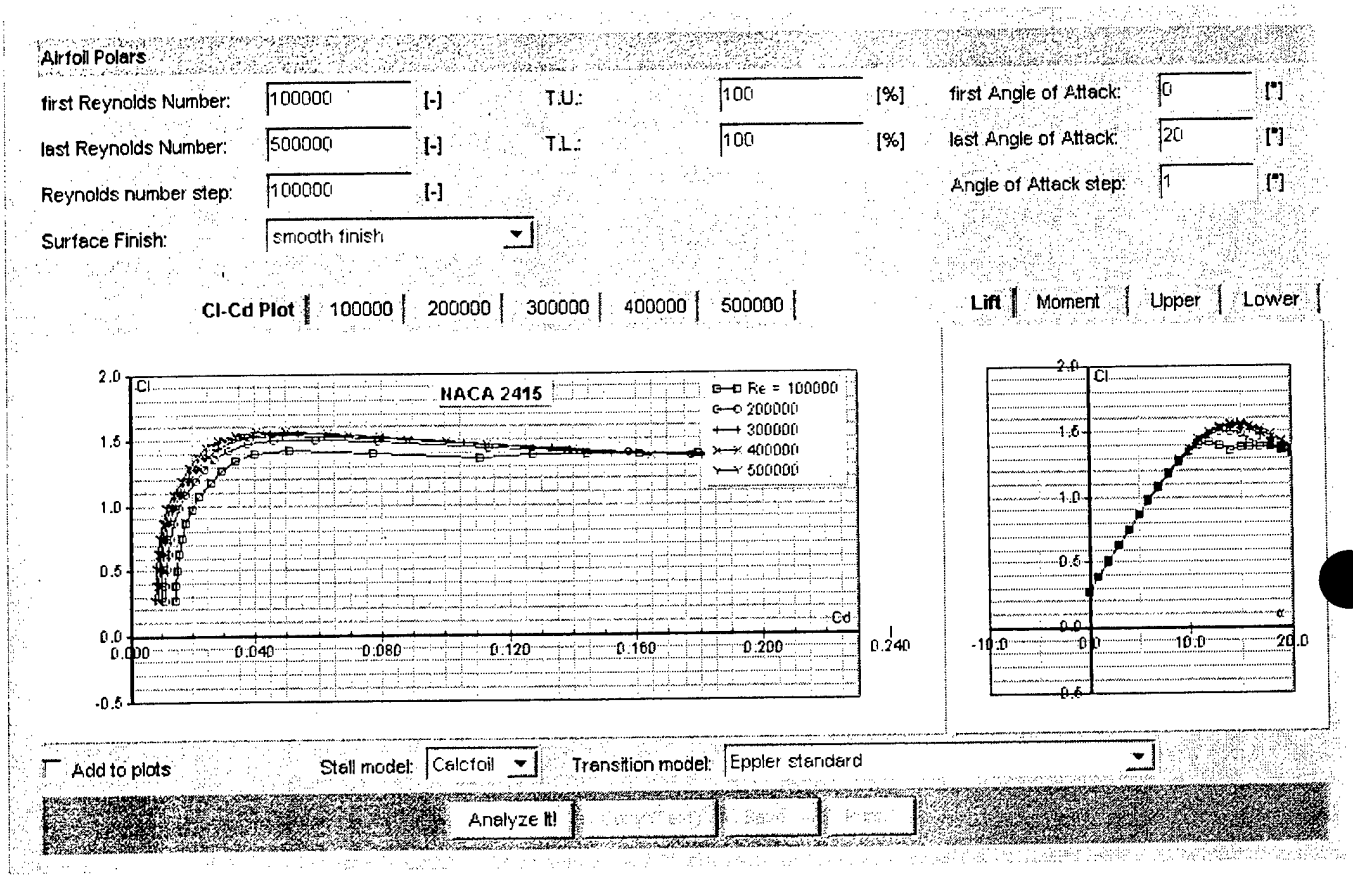


Figure 9 – Lift-Drag Plot

Wing Span and Chord length

In order to determine what would be the optimal design for the aircraft wing, the two main criteria considered were the wing loading and the stall speed. Values were calculated for wing loading by using an excel spreadsheet. A range of values for both wingspan and chord length were used and were plotted against each other. Figures 10 and 11 show the plots of wing loading in ounces per square foot and stall speed in MPH. Using these figures, the appropriate balance of speed and payload capacity could be evaluated.

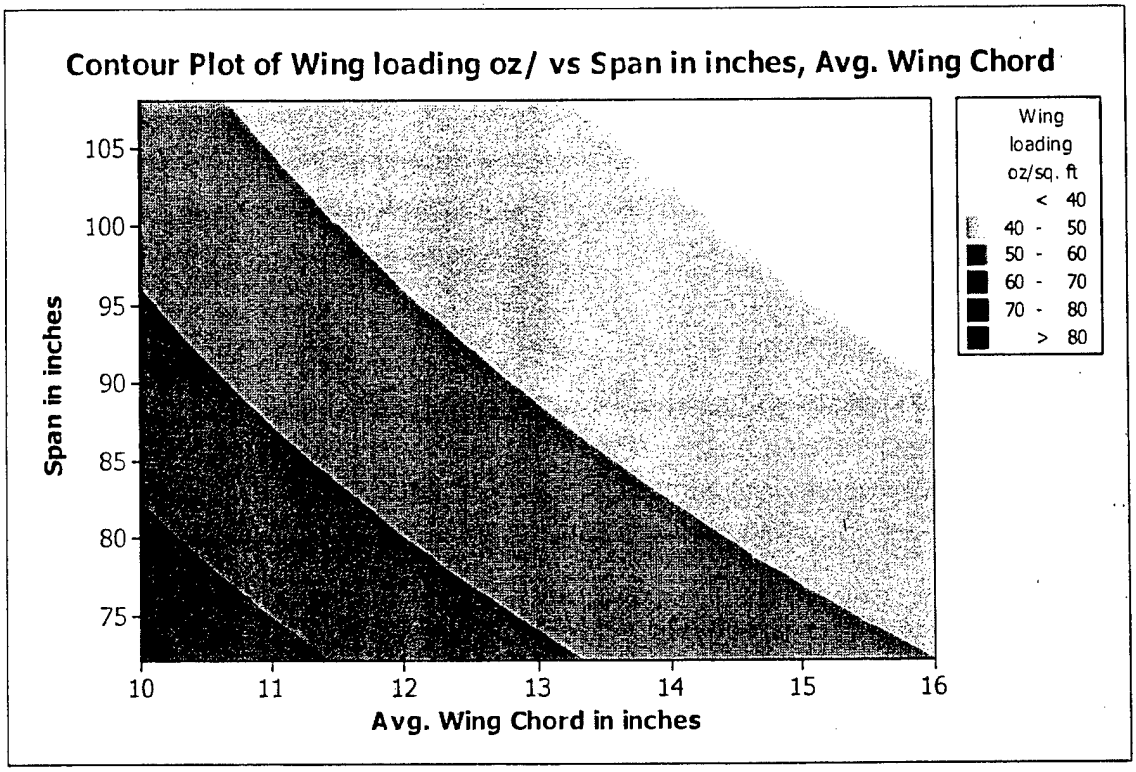


Figure 10 – Wing Loading

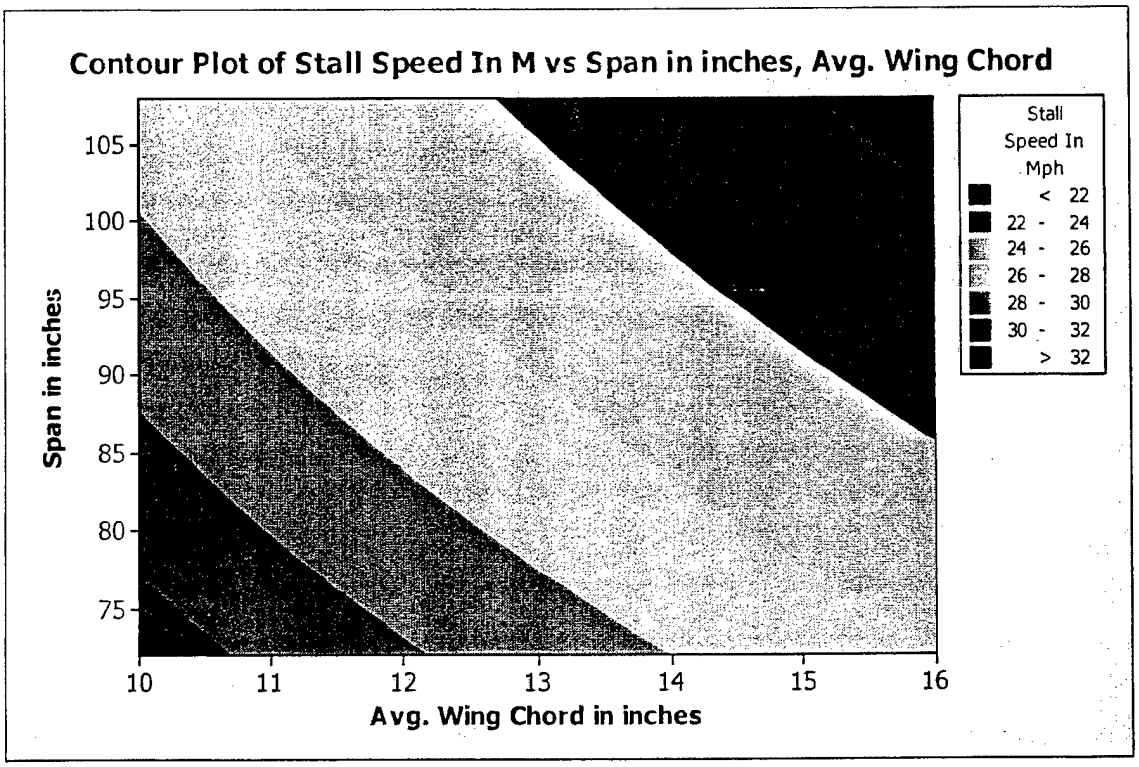


Figure 11 – Stall Speed

The information from Figures 10 and 11 enable the factors of stall speed and wing loading to be taken into account when determining the chord length. Based on this analysis, it was decided that it would be beneficial to have a chord length of approximately 14 inches which would give the aircraft a favorable stall speed of about 24-26 mph and a wing loading capacity of 40-50 oz/sq. ft

4.1.2. Fuselage

When first analyzing the shape of the fuselage, a predominately horizontal or vertical shape to house the water was considered. After considering various designs, it was believed in terms of RAC, ease of design, and speed of both plane and water drainage, that a cylindrical fuselage would be the best choice. Based on this decision, the fuselage design needed to allow for a stable and fast flight, as well as a quick release of the water.

4.1.2.1. Material

One of the initial steps towards a preliminary design of the fuselage included the selection of a material. In order to choose from a variety of feasible materials for the fuselage, it was first necessary to generate a list of critical properties the chosen material would need to possess. These properties allowed for the best possible alternative to be selected. The properties were as follows:

- **weight**: This value is desired to be as low as possible because the less weight the aircraft has, the lower the RAC is and the easier the plane will be to fly.

- **strength**: This value needs to be high because the more the fuselage can bend under stress without permanent deformation the better, especially since it is planned to make the fuselage a base structure for other components of the plane such as the wings, landing gear, tail, and as housing for our motor.

- **impact resistance**: The fuselage needs to be able to withstand the repeated impact loads that will occur when the plane lands. Additionally, in the case that the aircraft crashes during testing, the fuselage needs to have a high impact resistance so that it could remain intact and salvageable when building a new plane.

- **moisture impermeability**: Because the plane will be holding water, the material should not absorb any of the water that is placed inside the fuselage. Therefore a low percentage is desired.

- **machinability:** The fuselage will need to be easily machined in order to serve as a base structure for the other components of our plane. Many components will be affixed to the fuselage.

- **cost:** Money is always a concern and the material for the fuselage needs to be economical in order to fit within an allotted project budget. Cost was also utilized in the conceptual design as a quantitative FOM

Table 4 shows the numerical values assigned to each of the properties selected.

Table 4 – Fuselage Material Properties

	Polycarbonate	PVC	Acrylic - PMMA	Aluminum
Properties				
Weight (Density)	.043 lb/in^3	.0495 lb/in^3	0.0423	0.098 lb/in^3
Strength (Tensile)	10500 psi	6600 psi	8038 psi	43500 psi
Impact Resistance	14 ft-lb/in	17 ft-lb/in	0.5 ft-lb/in	-
Moisture Barrier	0.20%	3%	1.15%	0%
Machinability	1	1	2	1
Cost	$14.60/ft	$12.81/ft	$14.51/ft	$26.66/ft
1 – highest assignable value				

After analyzing each property of the materials in Table 4 it was concluded that polycarbonate would be the best choice for the material of the aircraft's fuselage. Aluminum was eliminated due to its high cost and weight. PVC did not have as high of a tensile strength as polycarbonate or acrylic and was heavier. This allowed PVC to be eliminated. Based on the two final choices, polycarbonate and acrylic, it was easy to eliminate acrylic due to its poor machinability and impact resistance. The lightweight, yet strong tensile characteristics, low cost, and ease of machinability supported the decision to use polycarbonate.

4.1.2.2. Reservoir

The main concern for the fuselage of the plane was its ability to hold close to 4-liters of water and allow it to drain quickly without drastically affecting the plane's flight characteristics. In order to enhance stable flight, the center of gravity needs to be held somewhat constant. To help keep the center of gravity constant, the water needs to be kept from rocking the plane back and forth and from running to the rear of the plane while it is being drained. To keep the center of gravity relatively constant, barriers were designed to put inside the tube to lessen the center of gravity from shifting, but still allow for the water to drain from the rear. The barriers will be baffles that have small half circles drilled out at their base. Figure 12 illustrates our initial baffle design.

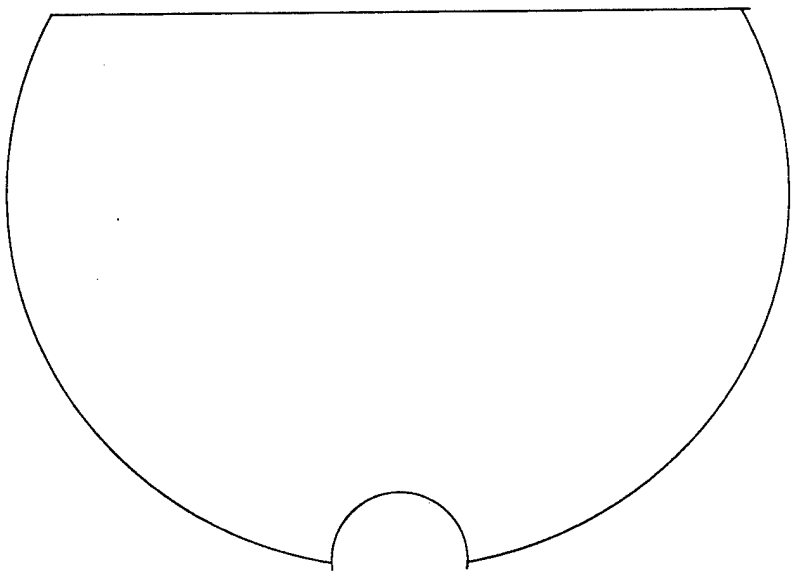


Figure 12 – Baffle Design

To allow for an even distribution of water throughout the length of the fuselage while draining, the hole at the base of the front baffle will be smaller than the rear baffle. This will hold an approximately equal amount of water at the front of the fuselage as the amount at the rear exiting the hole to better stabilize the center of gravity.

4.1.2.3. Entrance and Exit of Water

To allow for a constant flow of water exiting the plane an air input hole needs to be placed somewhere on the top front of the fuselage. This can be combined with the same apparatus used for the ram air design, which is discussed in section *4.1.2.4*. The water will be loaded into the plane by using the ram air hole in the top of the fuselage. This will require the design of a special input device for allowing the water to flow through the ram air hole and fill the fuselage. This device will consist of a flexible tube attached to a valve. The device will then be attached to the 2-liter bottles containing the water supplied at the contest event.

4.1.2.4. Ram Air

As stated by the AIAA rules for this year's competition, no pump or pressurized tank can be used to help empty the water tank. However, the tank vent may be aligned with the plane's onset air stream. This would provide very small amount of positive pressure to assist in draining the water, therefore lessening the drainage time. This will be beneficial because weight will be removed quickly, which will allow the aircraft to complete the mission route faster without expending additional battery power. A front and side view of the preliminary design for the ram air system is shown in Figure 13.

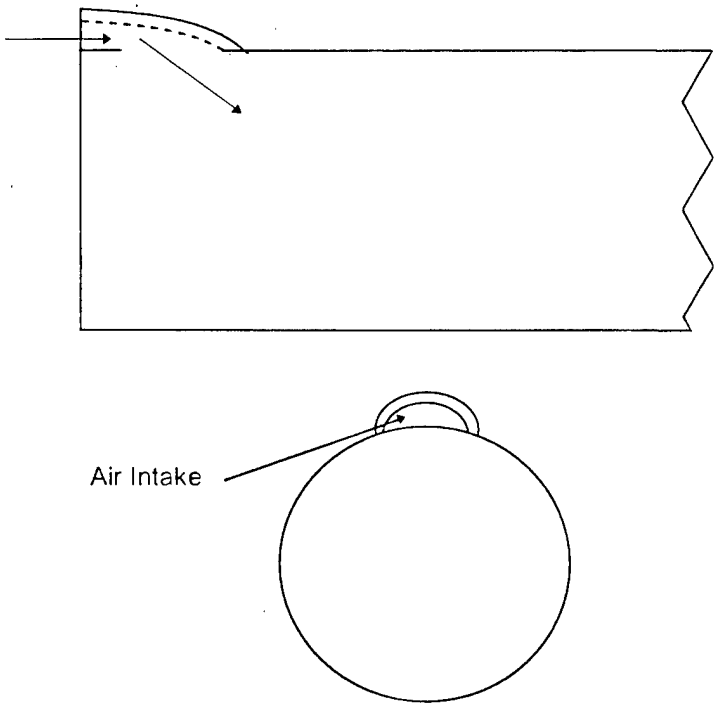


Figure 13 - Ram Air Design

4.1.2.5. Placement and Location of Parts

At this stage of the design the location of the parts of the fuselage were considered. As stated previously, the ram air intake vent will be located on the top front portion of the plane. The water exit hole will be located on the rear surface of the fuselage at the very bottom. This will allow all water to drain without having to tilt the nose of the plane drastically skyward. The motor, receiver, battery, fuse, and wires will be housed inside the fuselage at the front of the plane. A water-tight barrier will separate these system components from the water carrying section of the fuselage. There will also be a servo located on the outside rear of the fuselage to open and close the valve that releases the water being carried within the fuselage. There will be two baffles placed 8 inches apart that separate the fuselage into three equal compartments to allow the water to drain evenly.

4.1.3. Tail

The design of the tail of the aircraft cannot be fully considered until the preliminary designs of the other aspects of the plane are complete. Since all forces have to create a level moment about the center of gravity, the tail has to stabilize the plane by countering the moment from the front of the plane. Therefore, the tail has to be designed to create an equal and opposite force to that of the weight of the motor, propeller, and other system components located near the nose the aircraft.

Despite the fact that the design cannot be complete, the aspects of the design that are known and can be considered are the lengths from the center of gravity to the center of the tail and the area of the tail. By adjusting the distance from the tail to the center of gravity, the moment about the center of gravity could also be adjusted accordingly. Furthermore, modifying the area of the tail could change the force created by the tail.

Based on the initial observations just mentioned, the materials for the tail have been selected. A wheel will be used under the tail to help in control the steering of the plane and to absorb some the force when landing. To connect the tail to the fuselage, a carbon fiber rod will be used. A carbon fiber rod was selected because it is lightweight and high strength material, which is necessary to reduce weight and provide strength for maneuvers during flight and during impact on landing. The horizontal and vertical stabilizers will be constructed out of the same Styrofoam as the wings. A rudder and elevators will be made of balsa wood.

4.1.4. Landing Gear

There is one main design parameter that needs to be taken into consideration when designing the landing gear. This parameter is the strength of the landing gear. When the plane lands, the gear needs to withstand the forces that act upon them so that they do not bend, buckle, or fail. If any of these were to happen, significant damage would occur to the plane, specifically to the prop, motor and bottom of the fuselage.

While the landing gear is an important part of the aircraft, it is not a component that is critical to the selection of the preliminary design. Because of this, the design of the landing gear has not been fully analyzed at this point of the design stage.

4.1.5. Propulsion System

The aircraft propulsion system consists of components such as batteries, electric motor, propeller, and the motor gearbox. The propulsion system is a critical portion of the aircraft design. The exact specifications of the propulsion system rely heavily on the design of the remainder of the aircraft. Once these other aircraft specifications are determined, the selection of different variables for the propulsion

system components will be able to be finalized. Electricalc software will be utilized to assist in the determination of the exact parameters of these components. Decisions and considerations at this point are described below

Motor

Two motors have been obtained for the competition. These motors were not selected by the team, but rather left over from previous AIAA DBF competitions. Due to financial constraints, one of the two previously obtained motors will be used. To minimize costs, the aircraft will be designed around the performance requirements of these motors. Two motors used in previous years are available, and wither the Astro 90 or the Astro 60 will be used in the aircraft based on future analysis. If these two motors can not perform to the levels required by the remaining variables of the aircraft, a new motor must be considered and selected.

Propeller

The appropriate length and pitch of the propeller will be selected through the use of Electricalc. Based on the motor, exact wing loading, and weight of the aircraft, the propeller will be determined. These other aircraft variables must first be determined in order to select the most effective propeller.

Batteries

From previous AIAA DBF competitions, 3600 mAH (milliamp hour) batteries are available for use. Twenty-seven cells of these batteries have been obtained and will serve as a determining variable for the remainder of the propulsion system's component specifications. If it is determined that these batteries are not sufficient, new batteries must be selected and purchased.

Gear Box

A gearbox for the motor is an optional component for the propulsion system. If it is determined that the motor cannot generate enough thrust to get the aircraft into the air and keep the aircraft stable during flight, then this gear box option will be explored.

4.2. Testing and Analysis

Throughout the preliminary stage of the design, testing and analysis needed to be completed on the different subsystems of the aircraft. This would ensure that they will perform as expected when the subsystems are assembled in the construction of the final aircraft. Javafoil and stress calculations will be used in the analysis of the wing, a prototype fuselage will simulate water drainage, and the construction of a prototype plane sharing the flight and weight characteristics of the competition aircraft will be used to analyze flight performance

4.2.1. Wing

At this stage of the design process only mathematical calculations have been completed on the design of the wing and computer programs such as Javafoil have provided a better understanding of the forces that will be acting on the aircraft wing. However, a number of prototype airfoils have been constructed. This was done to not only gain experience constructing the wings, but it gave the ability to begin analyzing how the wings would connect to the fuselage. Support will run through the sides of the wing, which will connect to a base support on the side center of the fuselage. Carbon fiber rods were investigated due to their resistance to bending and structural support they will provide to the wings. Extra support will be needed in the wings to brace the amount of weight caused by the water payload during flight. This will also guarantee success for the structural verification test that will be conducted at the contest where all aircraft will be lifted with one lift point at each wing tip to verify adequate wing strength.

4.2.2. Fuselage

In order to test the preliminary design of the fuselage a prototype was constructed that resembled a model of what the final fuselage would look like. The outer shell of the fuselage was made of PVC piping that was 5" in diameter, 24" in length, and had a ¼" wall thickness. For visual purposes of viewing how the water would behave within the fuselage while draining, an opening 1" wide and 16" long was cut out of the top of the fuselage. This opening also provided the necessary opening to allow the water to be filled. Inside the prototype, two plastic baffles were placed to divide the prototype into thirds. On each baffle, a half circle was drilled on the curved edge to allow water to drain past the baffle. The size of this circle was approximately ½". The ends of the fuselage were sealed with circular plastic discs that fit on the inside of the fuselage. A ½" hole was drilled in one of these discs to allow for the water to drain out of the fuselage in accordance with contest rules. The outer edge of this hole was lined up with the bottom edge of the inner diameter of the fuselage.

The testing that was done using this prototype included three major setups.

1. Fuselage with no baffles
2. Fuselage with baffles
3. Fuselage with front baffle having a smaller size drain hole

In all of three setups mentioned, certain factors were held constant. 4-liters of water were used to fill the fuselage. The fuselage was held at an angle of approximately 3 degrees to simulate the angle of attack our plane would have during flight. During the draining period, the fuselage was rocked back and forth, which altered the angle of attack approximately 4-6 degrees. This was done to simulate the turbulence that may be encountered while the plane is in flight and dumping the water. Comparing setups 1 and 2 will help determine the effects of using baffles on drainage time.

Because there is a concern of the weight shifting towards the rear of the plane after the compartment at the front of the fuselage is empty, comparing setups 2 and 3 will help determine if the decreasing the hole radius to ¼ inch for the front baffle will affect the drain time. Decreasing this hole's radius will cause the water in the front compartment to drain slower than the middle compartment and therefore, keep a more balanced weight distribution throughout the entire drainage process. This design would still keep water pressure high on the exit hole located at the rear of the fuselage because all three compartments are connected. Therefore, the total time to drain the water should not be affected. The purpose of setup 3 is to see if the weight can be better balanced without affecting drain time.

4.2.3. Water Drainage

Table 5 shows the results from the three tests performed on the fuselage prototype in order to determine drain time and stability of the center of gravity.

Table 5 - Fuselage Testing Results

Setup	Time to Empty 4-Liters of Water (Sec)	Center of Gravity Stability*
1	55	Bad
2	42	Good
3	45	Excellent

*The center of gravity test was conducted only with the fuselage filled with water. No other components of the plane were attached.

During testing, the water level was monitored. Figures 14, 15, and 16 give a visual representation of the water level between each baffle and a rough estimation of the center of gravity at each setup marked with a star.

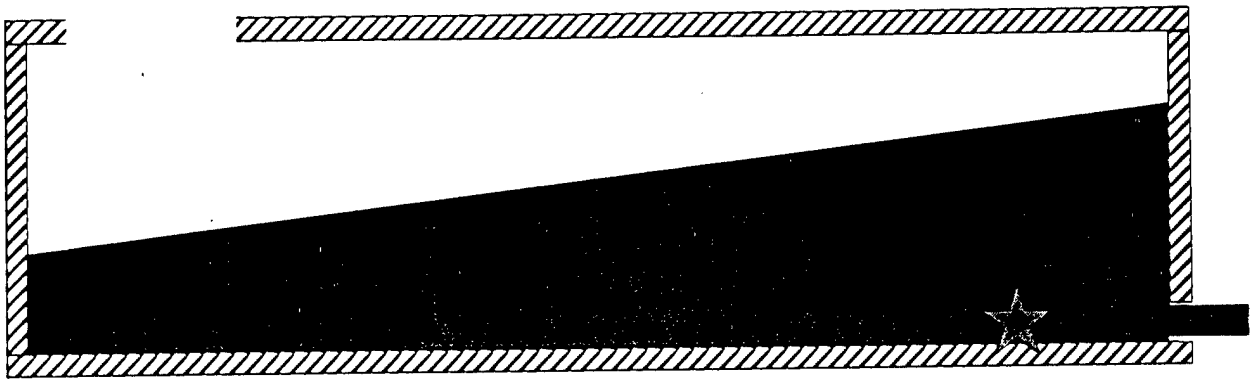


Figure 14 – Fuselage Drain Test: Setup 1

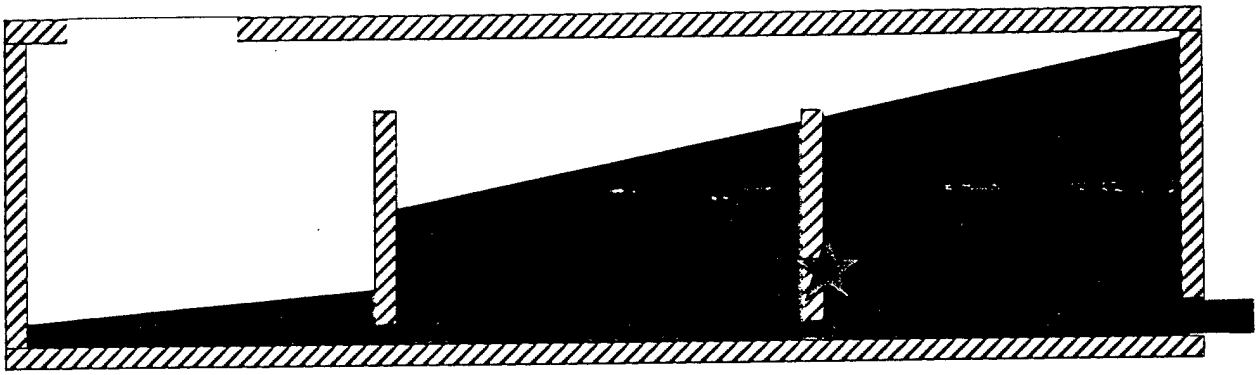


Figure 15 – Fuselage Drain Test: Setup 2

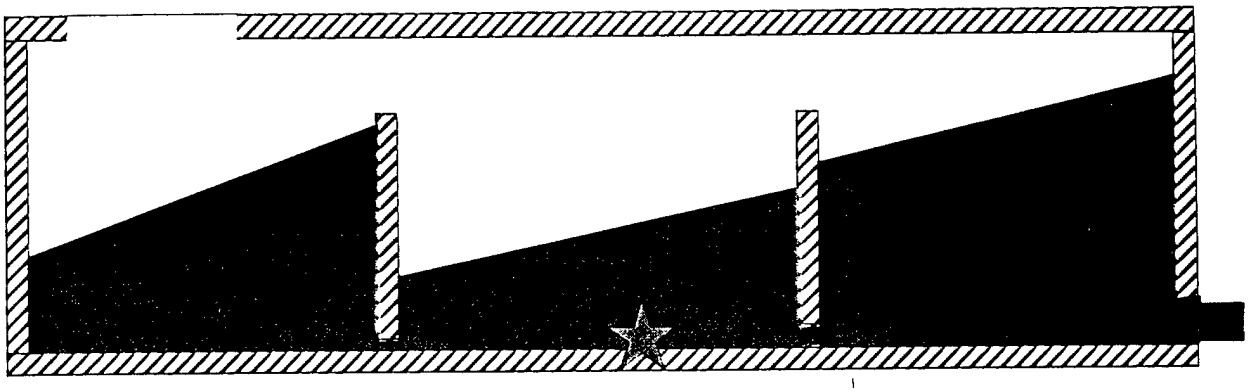


Figure 16 – Fuselage Drain Test: Setup 3

Based on the data shown in Table 5, decisions could be made on how the fuselage should be designed. In order to support the use of baffles in the design of the fuselage, the drainage times of setups 1 and 2 were compared. The time to drain 4-liters of water was 13 seconds faster with the use of baffles than when no baffles were used. This result agreed with the initial predictions that drastic water movement within the fuselage would slow drainage. Therefore, it was concluded that the use of baffles within the fuselage will help increase the speed at which the water is drained.

While testing setup 2, it was noticed that the compartment located at the front of the fuselage was completely empty before the rear compartment. This resulted in a very unbalanced fuselage due to the amount of water that collected in the rear of the plane during the end stages of the drainage. To combat this problem, the radius of the hole in the front baffle was decreased to a radius of about ¼ of an inch for setup 3. The problem of balancing the weight throughout the entire drainage process was assessed when the times of setups 2 and 3 were compared. The drainage time of setup 3 very close to setup 2 and the small difference in drain time can be attributed to slight variation in the rocking of the fuselage during draining. This supports the prediction that the smaller hole in the front baffle will not affect drainage time. While draining, the weight distribution of the water was also monitored. Because a member of the team was holding the fuselage and rocking it back and forth while the water was draining, the weight distribution could be reasonably determined while comparing it to the visible water levels inside

the fuselage. It was concluded that the water level in the first compartment near the front of the plane stayed much higher than setup 2. As a result, the weight between the front and rear of the fuselage is balanced and would allow a more stable flight for the plane because the center of gravity would be better maintained.

4.2.4. Prototype

A major part of testing and analysis was to build a prototype airplane. The two reasons for doing this were to provide an impression of how the final aircraft design would work, and to give the pilot of the aircraft more experience in understanding and mastering the competition flight path and the aircraft's performance while final design considerations and implementations were being performed on the contest airframe.

There are some differences between the prototype and preliminary design. The first difference is that the body will not be a cylinder, as in the preliminary design, but a piece of rectangular downspout. Using a gas engine instead of an electric motor is the second major difference in the prototype. While these differences may seem drastic when compared to our preliminary design, the prototype will still perform very similar to our actual design. With the wing on the prototype being the same airfoil and having the same wingspan that was chosen for our final aircraft design, it will be possible to confirm whether the selected design is sufficient for the final design.

A gas engine was chosen for the prototype airplane over an electric motor because it would give the pilot more time to fly the airplane and become better familiarized of its performance. If the electric motor were used, approximately only 10 minutes worth of flying time on a single charge of the batteries would be available. It takes approximately 45 minutes to recharge the batteries after they have been used. It was felt by the team that flying for only 10 minutes and then waiting for 45 minutes before flying again would be an inefficient use of time and would not give the pilot sufficient time to learn how the airplane flies. With the use of a gas engine a considerable amount of flying time can be obtained from one tank of gas compared to a single battery charge. It would also only take a few minutes to refuel the gas engine before the plane could be ready to fly again. With these advantages, the pilot will be able to have a significantly longer amount of time for flight testing than would have been available with the electric motor.

4.3. Results and Conclusion

To this point, various conceptual designs have been evaluated and narrowed to one preliminary design. This design will be refined further in the detail design. From the conceptual designs, the components have been chosen, such as airfoil, fuselage, and water reservoir that will best fit the design goals. The building of the prototype plane has provided practice in building a model airplane and will allow for faster and more skillful construction of the final aircraft design.

4.3.1. Wing

Using Javafoil to narrow the airfoil choices, it was decided that the NACA 2415 airfoil would be used for the aircraft. It was found that the extra thickness compared to the 2412 will give more lift, which is one of the main concerns given how much weight the aircraft will be carrying in the fire fight mission. The 2418 had an even greater amount of lift compared to the 2415, but it also had a greater drag coefficient compared to the 2412 and 2415.

Optimal design necessitates having the lowest stall speed and the highest wing loading. Based on this statement, it was decided that the aircraft wingspan would be eight feet and our chord length thirteen inches. Having the wing loading in the fifty to sixty ounces per square foot range is believed to be sufficient for takeoff and the early legs of the flight. After the aircraft begins to dump water the wing loading will decrease dramatically and reduce the stall speed, which will make landings easier.

The wings of the aircraft will be constructed out of foam. Two types of foam are currently available to us, blue insulation foam used in building construction and a less dense type of foam that is much lighter but more coarse in its makeup. The wing sections will be strengthened by inserting carbon fiber tubes through the length of them. The eight-foot section of wing will be divided into three sections. One is a central 18" section, which is permanently mounted to the fuselage. The second and third are outboard sections that will connect to the central portion by means of telescoping carbon fiber tubes. These tubes will go through the fuselage, thus reducing our rated aircraft cost since the height of the fuselage will be reduced. The leading and trailing edges of the wings will be made from balsa wood, which will strengthen these critical areas.

4.3.2. Fuselage

Using the results and observation from series of tests performed on the fuselage, it was concluded that we would use a baffle system in our fuselage. The fuselage would consist of two baffles with the baffle at the front of the fuselage having a hole radius of ¼" as opposed to the rear baffle which has a larger hole radius of ½".

5. Detail Design

After finalizing the components that would be included in the final design of the aircraft, it was necessary to optimize all aspects of the aircraft. Major areas of focus were to determine the needed power supply, keeping the plane as light as possible while maintaining the aircraft's strength, and eliminating drag and other factors that would impede flight. These areas of focus will allow for flight characteristics that would best meet the mission requirements that have been determined. The optimization of the final design was done through analysis of the prototype plane and through calculations that would predict the performance of the final aircraft design.

5.1. Component Selection

To better understand the characteristics and performance of the final aircraft, the aircraft was broken down into separate subsystems that could be analyzed in greater detail. These subsystems were the wing, fuselage, tail, landing gear, and propulsion system. As previously explained in the preliminary section of this report the components of these subsystems were examined using numerical calculations as well as programs such as JAVAFOIL, Electricalc, and RDS. In this stage of the design process, more testing was done in the form of test flights with the use of the prototype aircraft to compare our initial predictions with experimental data.

5.1.1. Wing

The wing of the aircraft must possess adequate wing strength to simulate a 2.5g load case. To ensure the weight of the plane would be able to be supported by the wings, the bending stress on the wing needed to be calculated. The calculations were as follows:

Wing Bending Stress

Using the formula:

$$\sigma = \frac{My}{I}$$

where, M = 25 lb (the estimated weight of the plane), y = 2 in (the max distance from the neutral axis to the surface of the wing), and I = .07968 in^4.

The bending stress (σ) in the wings is calculated to be 627.5 psi. Based on this result, the bending stress at 2.5g would be far less than the 65,000 psi that a carbon fiber rod is capable of supporting.

After selecting an airfoil and determining the major dimensions of the wing, a propeller needed to be selected so that the performance expectations could be determined. Propeller selection and performance characteristics were determined through the use of a software package called Electricalc. Using Electricalc requires that some estimates were made concerning various elements. Using the drag coefficient estimator provided with the program, it was found that a conservative estimation of the value is 0.08 for the entire plane. By trial and error, the two different props were selected for the respective missions. The main criteria used for selection were thrust at take-off speed, maximum cruise speed, amps drawn, and number of minutes at cruise speed. For the ferry mission, the plane needs a prop that will have a high max speed and thrust for take-of is not critical due to the lower weight. The firefight mission with payload needs a prop with greater thrust at the low end of the thrust-drag curve in order to lift the plane off the ground in the allotted distance and speed is not as critical. The take-off distance was estimated using the equation shown on page 8 of this report.

A reasonable max coefficient of lift was determined using Javafoil and thrust was estimated from the thrust-drag graph provided by Electricalc shown in Figures 17 and 18 for the two missions. Even though the amperage is fixed at 40A for the competition values were allowed to reach as high as 47.5A in analysis since estimations for drag and cell volts were conservative.

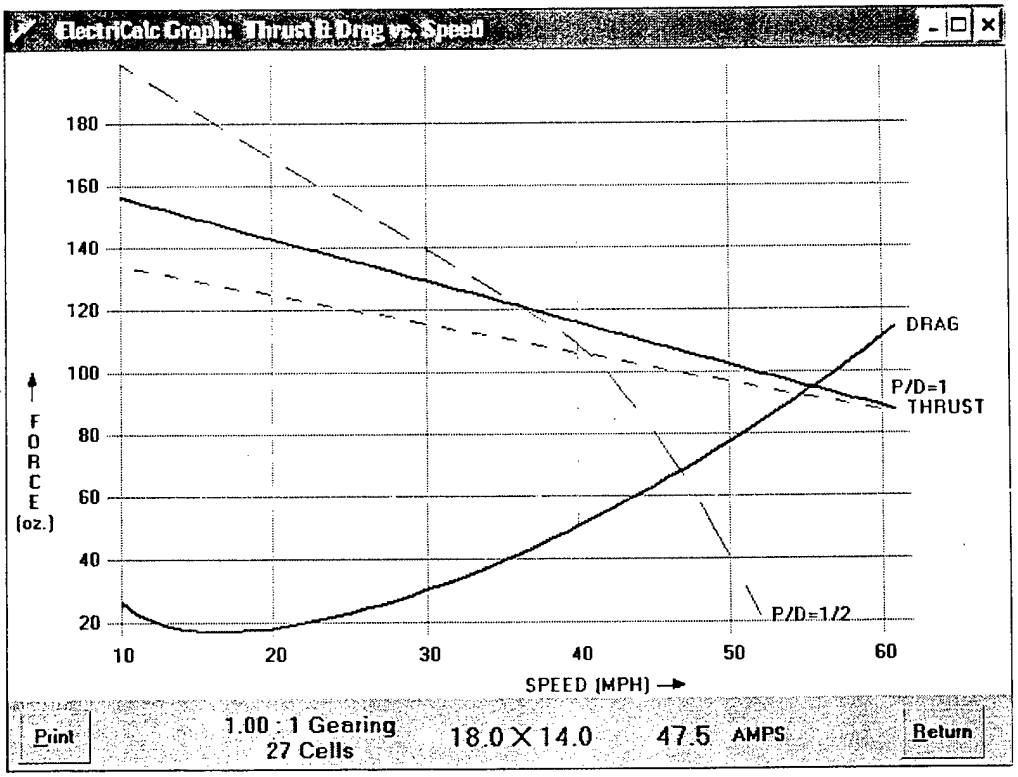


Figure 17 – Thrust and Drag vs. Speed (18/14 propeller)

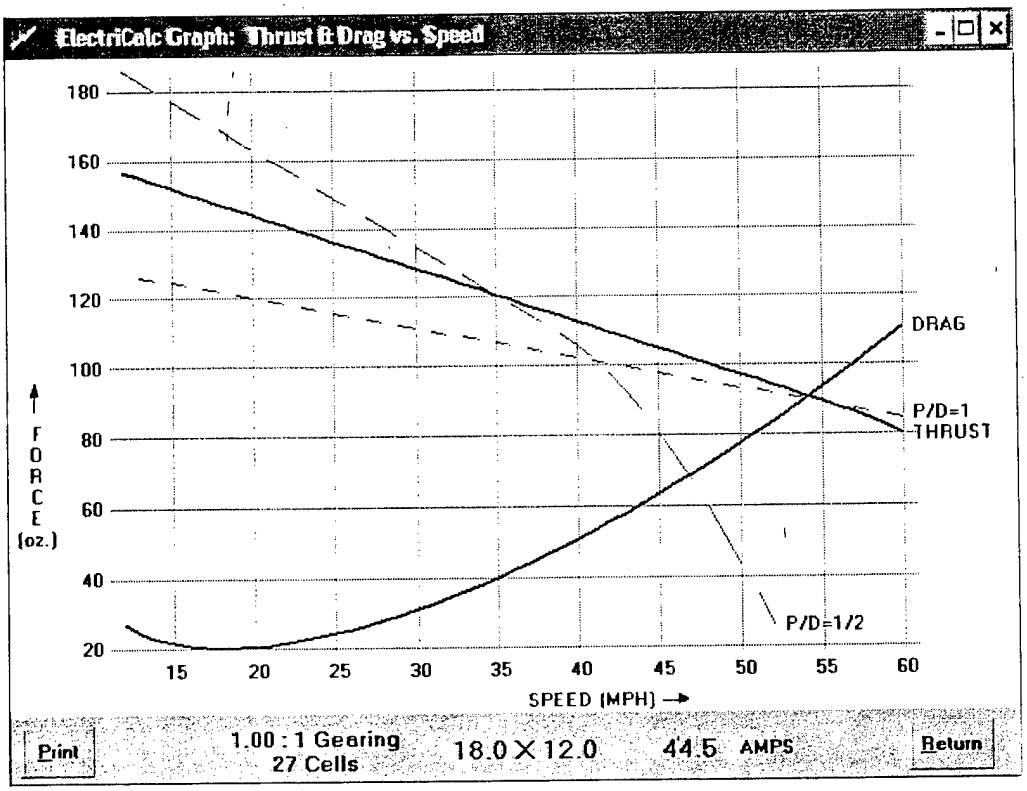


Figure 18 – Thrust and Drag vs. Speed (18/12 propeller)

In order to maintain longitudinal stability of the airplane while the water is draining, the control volume ratios of the tail must be sufficient to accommodate the variability of the center of gravity. The control volume ratio of the horizontal tail was calculated using equation 1 below and the control volume ratio of the vertical tail was calculated with equation 2 below:

1. $C_{HT} = (S_{HT} / S_W) \times (l / c)$
2. $C_{VT} = (S_{VT} / S_W) \times (l / c)$

where,

$S_{H,V}$ is the plan form of tail surface, S_W is the plan form of wing, l is length between center of gravity and quarter chord of tail surface, and c is the chord of tail surface.

Based on the result of .5715 for the C_{HT}, the control volume coefficient was within the design parameters of propeller driven simple flapped transport planes. The result of .27 for the C_{VT} is a number that fits within our design parameters.

Several forms of foam were evaluated for use in the core of the wing. After some research to see what was available, samples of two types of foam were acquired for testing, a white polystyrene foam and a

more dense blue polystyrene foam board commonly used for insulating houses. The lightweight white polystyrene was chosen for the core of the wing. This type of foam was chosen due to its high availability and low cost. This polystyrene foam was very easy to cut with a hot-wire and it was determined that the white polystyrene foam gave the best results in terms of airfoil shape and quality. The blue polystyrene was more dense and heavier and did not cut as easily as the white foam.

A leading section made of balsa matching the airfoil was attached with epoxy to the leading edge of the wing In order to make the wing more durable. The balsa section is lightweight yet a great deal stronger than the foam alone. This will make the leading edge more dent resistant and stiffen the leading edge of the wing to prevent twisting along the wing's horizontal axis.

Pieces of 1/4"x4"x4' balsa were used to construct the trailing edge and ailerons for the wing. Instead of using a traditional "built-up" design with balsa ribs, which would be a great deal lighter, the team chose to use solid balsa in order to save time. In order to cut weight from the design, the core of the trailing edge and ailerons was removed using a mill to cut parallel slots in the wood to achieve a lighter design.

The spar of our wing is constructed from carbon fiber tubes running the length of the wing. Although carbon fiber tubes are fairly expensive, the team chose to allocate a great deal of the budget to buying the tubes due to the extremely high strength and stiffness. The stiffness was necessary in order to prevent the tubes and foam from separating when the wing is flexed. If the spar was too flexible, when the plane is lifted from the wing-tips, the team feared that the foam would pull away from the tubes, losing structural integrity. In order to achieve maximum strength and stiffness, the tubes were aligned one on top of the other vertically, similar to an "I" beam of steel 30% of the way back from the leading edge. This format was adopted in order to achieve maximum strength with a limited amount of material. Another set of carbon fiber tubes was set 70% back from the leading edge in order to prevent the wing from twisting where it was attached to the fuselage. These tubes do not run the full length of the wing, only through the central portion. In order to connect the wing sections together, aluminum tubing was fit inside the carbon fiber tubes to make them telescope into one another.

Balsa sheeting was chosen to cover the wing making it ready for the final coat of Monokote. Balsa sheeting is very lightweight and the layers give the wing a great deal of rigidity. This type of covering was chosen instead of other methods of covering such as vacuum bagging due to the relative ease of construction and since it does not require the specific equipment that vacuum bagging requires. Vacuum bagging also requires a very smooth surface finish on the foam, which is not easily attained with the polystyrene that was used. Monokote was used to cover the wing in order to make it more aerodynamic and to add some color.

Table 6 – Wing BOM

Component	Material	Size	Quantity
Core of wing	Polystyrene	NACA 2415, 8' wingspan, 14" chord	1
Leading edge	Balsa Parabola	Length of wing sections	2
Ailerons and trailing edge	Balsa	1/4"x4"x4' Balsa	-
Structural spar	.6125" Carbon Fiber Tubes		
Cross members in spar	Aluminum Tubing	6	4
Covering wing	Balsa Sheeting	1/16" Thick	-
Outer covering	Monokote	Covers entire wing	-
Adhesive	Epoxy	-	-

5.1.2. Fuselage

The decisions made in the preliminary design stage relating to the design of the fuselage were more closely examined and fine tuned in this stage of the design. However, when choosing the component characteristics, more concern was directed towards how it would interact with the other subsystems of the aircraft during assembly of the completed aircraft. As mentioned in the preliminary design, the fuselage of the aircraft will be divided into two separate cavities. Of the two cavities, one cavity will be responsible for holding the water payload and the other will contain system equipment. This required a design for maintaining the center of gravity to preserve the stability of the plane while holding and releasing water and a water release system to dump the water on command.

To meet the requirements just explained, the main components of the fuselage where identified. They consisted of the tube which would house the water payload and system equipment, the baffles within the water cavity, the end caps to seal the water cavity from the systems cavity and the outside environment, and finally, the water release system which would be controlled by a servo. These components are shown in Table 7.

Table 7 – Fuselage BOM

Component	Material	Size	Quantity
Tube	Polycarbonate	20" length 5" outer dia.	1
Baffles	Polycarbonate	4.75" dia.	2
End Caps	Polycarbonate	4.75" dia.	2
Cork to seal water hole	Rubber	½" dia.	1

The main assignment of the fuselage is the holding and release of approximately 4-liters of water and therefore an appropriate fuselage size and a water release system needed to be designed and evaluated. After calculating the volume of 4 liters of water would need a fuselage of approximately 13.77" in length, the components that would be taking up volume inside the fuselage such as support rods and the baffles

needed to be considered. Table 8 shows the additional length we would need to add to the fuselage to accommodate the volume lost to different configurations of components in our final design.

Table 8 – Fuselage Length Based on 4-Liter Payload

Component Configuration In Fuselage	Original Length of Water Cavity (in)	Component Volume Inside Fuselage (in^3)	Length Needed to be Added to Original Fuselage Length to Maintain 4-Liter Capacity (in)	Final Fuselage Length (in)
4 Liters of Water	13.77	0.0000	0.00	13.77
No rods or baffles	14.00	0.0000	0.00	14.00
Rods and baffles	14.00	6.2845	0.12	14.12
Baffles and enclosed rods	14.00	18.8864	0.84	14.84
Baffles and boxed rods	14.00	12.6460	0.48	14.48

Based on Table 8, it was determined that a fuselage of approximately 15" in length would be needed to hold 4-liters of water.

The next main concern of the fuselage was during the water release and the ability of the valve to remain closed so that none of the water would leak from the fuselage premature to the valve being opened. To address this concern, three water release systems were designed and evaluated. The first was a type of hatch that would cover the ½" diameter hole and then slide out of the way when the water needed to be drained. The second design was a plunger system that used a rubber cork that could be pulled into the hole to seal the water cavity and then pushed out with a control rod when the water needed to be drained. The third involved the use of a solenoid valve to control water flow. The three separate systems were compared based on ease of construction, weight, cost, and mechanics which includes the ease of sealing the compartment and releasing the water. It was determined that certain FOM's were more important than others based on characteristics that will affect the RAC of the final design more adversely than others. The evaluations of the three considerations are shown in Table 9.

Table 9 - Water Drainage System Evaluation

FOM's	Scale Factor	Circular Hatch	Plunger System	Solenoid Valve
Cost ($)	x1	3	3	1
Ease of Construction	x2	3	2	2
Weight	x3	3	2	1
Mechanics	x4	1	3	3
TOTALS		22	25	20

*1 being lowest rating

The circular hatch seemed to be a suitable choice for the water release, but it was a concern that the hatch would not be able to provide the pressure needed to seal the hole while still being able to be slid out of the way by the power supplied by the servo. The solenoid valve also provided a suitable design, but the placement and mounting of a device that had a large weight and size would not be beneficial to the RAC of the final design. The plunger system provided a water release system that was very inexpensive, was moderately easy to construct, had minimal weight, and would assure reliability to seal the hole until the water needed to be released. Based on the highest total shown for each of the proposed designs, the plunger system was chosen as the best choice for implementation into our final design.

A ram air vent is to be used on the fuselage. In the preliminary design stage, placing the vent on the top of the fuselage was considered. However, after evaluating how the vent would be placed and its effect on the fuselage RAC, it was determined that a better location for the vent would be on the cone housing the motor with a tube that leads into the water cavity. The vent would need to be sealed during inspection. This design would serve the same purpose as well as keep our RAC unchanged.

During the construction of the fuselage and the components previously explained, it was essential to pay close attention to how the other subsystems would fit together. The fuselage serves as a base structure for the rest of the plane and the connection of the wing, tail, and motor to the fuselage must be considered.

The wing will contain carbon fiber rods that will be inserted directly through the length of the wing. Due to the low placement of our wing with respect to the fuselage, the rods will need to travel through the lower half of the fuselage. The back end of the fuselage will contain the water release system as well as a carbon fiber rod that will extend to the tail. Both of these components will be mounted to the rear of the fuselage and covered by a tapered cone. Modifications will also be done to the front of the fuselage. At this location, a motor mount must be constructed and bolted to the inside of the fuselage to hold the motor in place at the front of the plane.

5.1.3. Tail

The design of the tail is based on making it as lightweight and efficient as possible. The tail was constructed of balsa sheeting with removed interior sections to reduce weight. The leading edge of the tail was rounded down to create more of an airfoil shape and cause less air disturbance. In addition, the tail was covered with Monokote to allow for a more aerodynamic flow over the tail. The horizontal tail is at the base of the vertical tail, which is placed at the center of the fuselage. A carbon fiber rod connects the tail to the fuselage and the vertical and horizontal tail together. Table 10 details the material used in the tail.

Table 10 – Tail BOM

Component	Material	Size	Quantity
Horizontal Tail	Balsa Sheeting	270 inches squared	-
Vertical Tail	Balsa Sheeting	3/8" Thick	-
Elevator	Balsa Sheeting	95 inches squared	-
Rudder	Balsa Sheeting	3/8" Thick	-
Hinges	Plastic Pin Hinges	3/16" diameter	9
Monokote	Plastic Covering	-	As needed
Connection Rod	Carbon Fiber	1" diameter, 26" long	1

According to "Aircraft Design," the tail moment arm (TMA) should be 2.5 times the wings mean aerodynamic chord (MAC) and the tail area should be 20 percent of the wing area (WA). The TMA was calculated to be 35 inches. Using these values with the known wing design, the horizontal-tail area (HTA) of 270 inches square was calculated from the following equation:

$$HTA = \frac{2.5 \times MAC \times 20\% \times WA}{TMA}$$

The effectiveness of the elevator and rudder were also based on the suggested proportions in "Aircraft Design." The elevator area was suggested to be 35 percent of the horizontal tail's area, while the rudder was to be 30 percent of the vertical tail's area.

5.1.4. Landing Gear

When choosing the landing gear for the aircraft, a design was needed that would not buckle and fracture upon landing. For these reasons, the landing gear needed to be tested. Calculations were done to evaluate the amount of stresses and loads that would act upon it to make sure that it would still function properly on the aircraft design. After performing these calculations, it could be verified that the existing design would work on the airplane.

Landing Gear Bending Stress

Using the formula:

$$\sigma = \frac{My}{I}$$

Where M = 25 lb (the estimated weight of the plane), y = 2.59 in (the max distance from the neutral axis to the surface of the landing gear), and I =.0016 in^4.

The bending stress (σ) in the landing gear is calculated to be 40527 psi.

Landing Gear Maximum Torsion

Using the formula:

$$\tau = \frac{3T}{8ab^2}\left[1 + 0.6095\frac{b}{a} + 0.8865\left(\frac{b}{a}\right)^2 - 1.8023\left(\frac{b}{a}\right)^3 + 0.9100\left(\frac{b}{a}\right)^4\right]$$

Where T = 62.5lb-in, a = .625in, b = .125in.

τ = 2746.56 psi

Based on the results from these calculations and comparing to the tensile strength of the material used of 45,000 psi, it has been determined that the design of the landing gear will be sufficient in providing the aircraft with a design that will not fail during flight or landing.

In the preliminary design, the landing gear was to be made of aluminum and would have to be bent around the fuselage of the airplane. However, the landing gear assembly that was used on previous competition airplanes was available. It was decided that the landing gear could be placed on the bottom of the wing instead and the wing mount could be lowered on the fuselage. This new design allowed the use of the existing landing gear which saved considerable time and money on construction and also helped to lower our rated aircraft cost.

5.1.5. Control Systems

The detailed design of the control systems of the aircraft consist of a simple network of servos, control rods, radio receiver, radio controller, batteries for the radio receiver, the motor, and the batteries for the aircraft's power supply. These components were selected based on aircraft performance requirements set forth in the competition rules and regulations. The majority of these components, however, were selected because they were all readily available from previous years' competitions. Below are a breakdown of each system component and a brief description of why each respective component was selected.

Servos

Five JR NE-517 Remote Control servos will be used to control the two wing ailerons, the rudder flaps, the elevator flaps, and the release of water from the fuselage of the aircraft. These servos are standard remote control servos that were left from previous years competitions. They will adequately serve their intended purpose in controlling the aircraft. A sixth servo will be used to control the throttle of the aircraft.

Control Rods

Four control rods will be used in conjunction with the servos to control the movement of the wing's ailerons and the tail's flaps. The length of these aluminum control rods were individually selected based

on the required range of movement of the ailerons and the flaps. An additional control rod will be needed to control the water release system.

Radio Receiver & Controller

The Futaba Radio Receiver and Controller used in the aircraft to communicate the radio signals from the controller to the servos were selected because they were left from previous years competitions. Both the receiver and controller have remained in good condition and do not need to be replaced by newer or more sophisticated hardware.

Radio Receiver Batteries

The batteries for the radio receiver are Futaba NR-4J 4.8 VDC Nickel Cadmium batteries. Again, these batteries were left from previous years competitions and will adequately serve their intended purpose of powering the radio receiver and servos.

Battery Power Source

Twenty-seven cells of batteries will provide the aircraft with 3600 mAh (milli-amp hours). The cells are collected into two separate cylinders of nearly equal weight with thirteen cells in one group and fourteen cells in the other. The combined weight of all twenty-seven cells is exactly 5 lbs, meeting the design requirements of the competition.

Motor

Two motors were available for the final design of the aircraft. An Astro 90 and an Astro 60 Cobalt motors were remaining from previous years competitions. Due to the high power requirements of the aircraft, the Astro 90 Cobalt motor was selected for the final design.

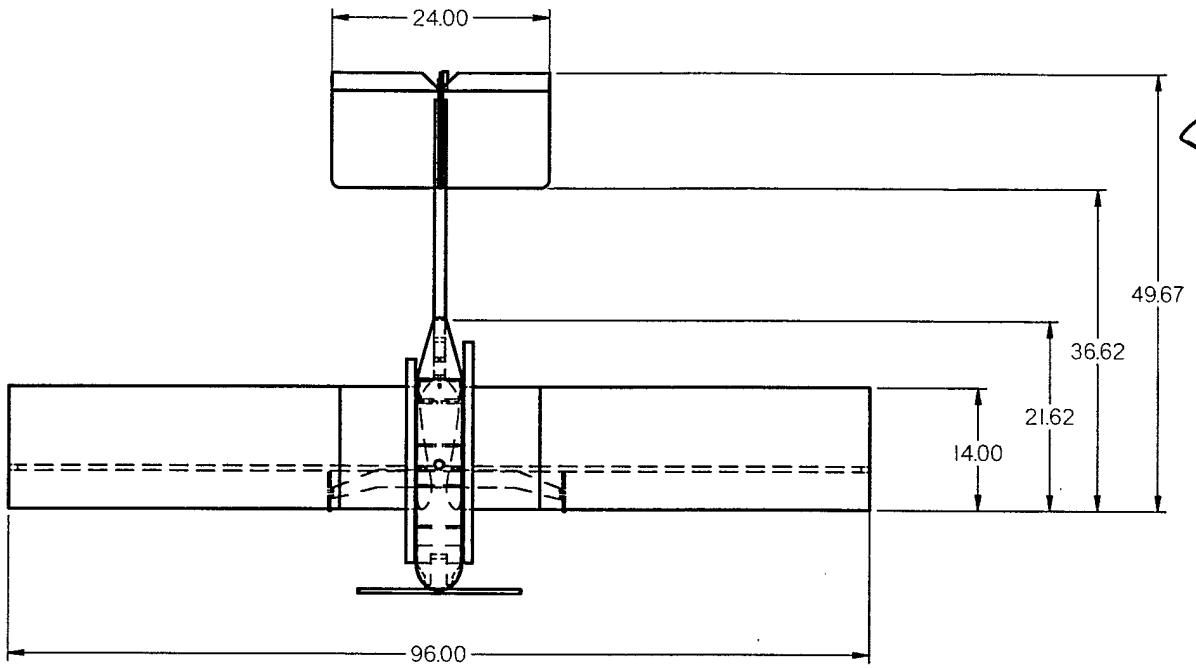
5.2. Competition Aircraft Rated Aircraft Cost

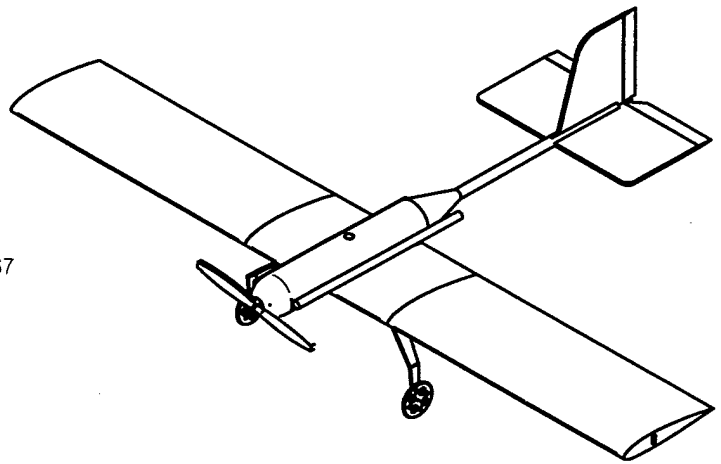
Table 11 – Competition Rated Aircraft Cost

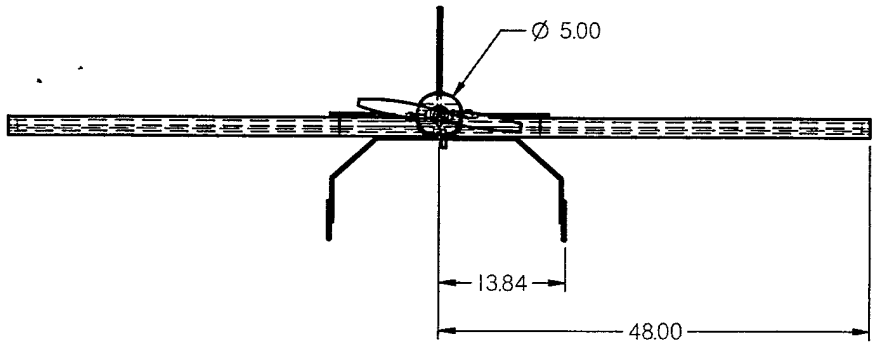
Rated Aircraft Cost	16.78911
Manufacturers Empty Weight	20
Rated Engine Power	5
Total Battery Weight	5
Number of Engines	1
WBS Wings	98.28
Wing Span (feet)	8
Chord	1.166
Number of Wings	1
Control Function Multiplier	1
WBS Fuselage	16.17538
Body Length (feet)	4.66
Width	0.4166
Height	0.4166
WBS Empennage	20
Vertical Surface w/o Control	0
Vertical Surface	1
Horizontal Surface	1
WBS Flight System	30
Number of Servos	6
Control Function Multiplier	
Ailerons	1
Flaperons	1.5
Ailerons and Flaps	2
Ailerons and Spoilers	2
Ailerons and Flaps and Spoilers	3

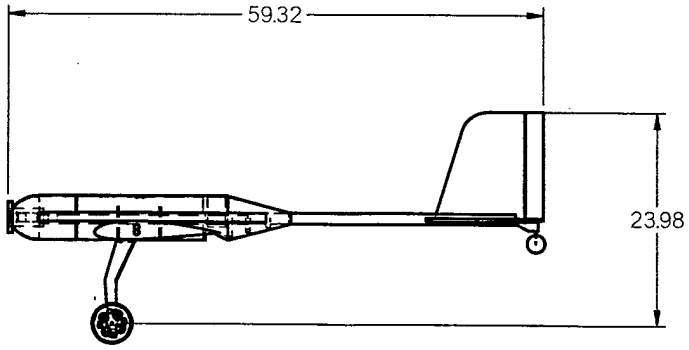
5.3. Drawing Packages

The following drawing package includes a 3-view drawing of the final aircraft configuration, aircraft assembly, aircraft exploded assembly, fuselage configuration, and tail assembly.

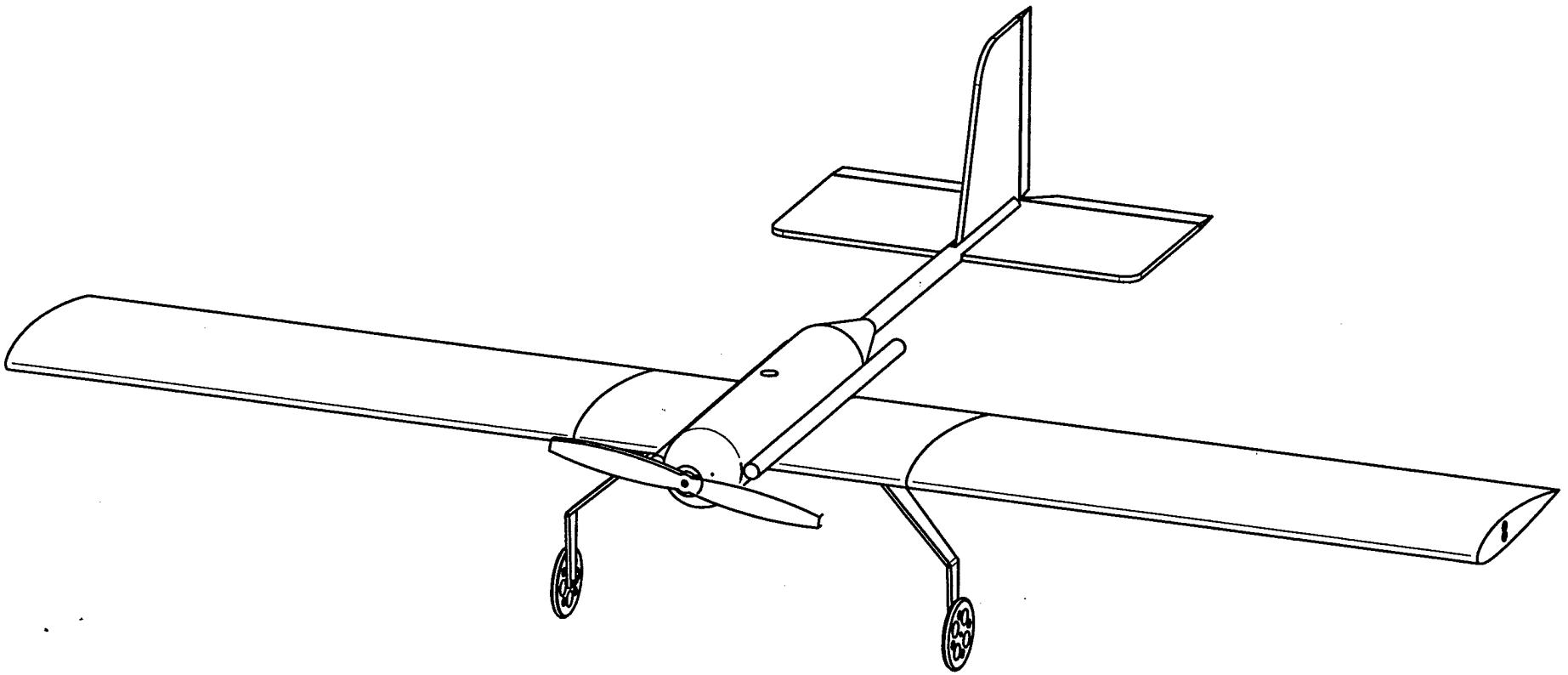




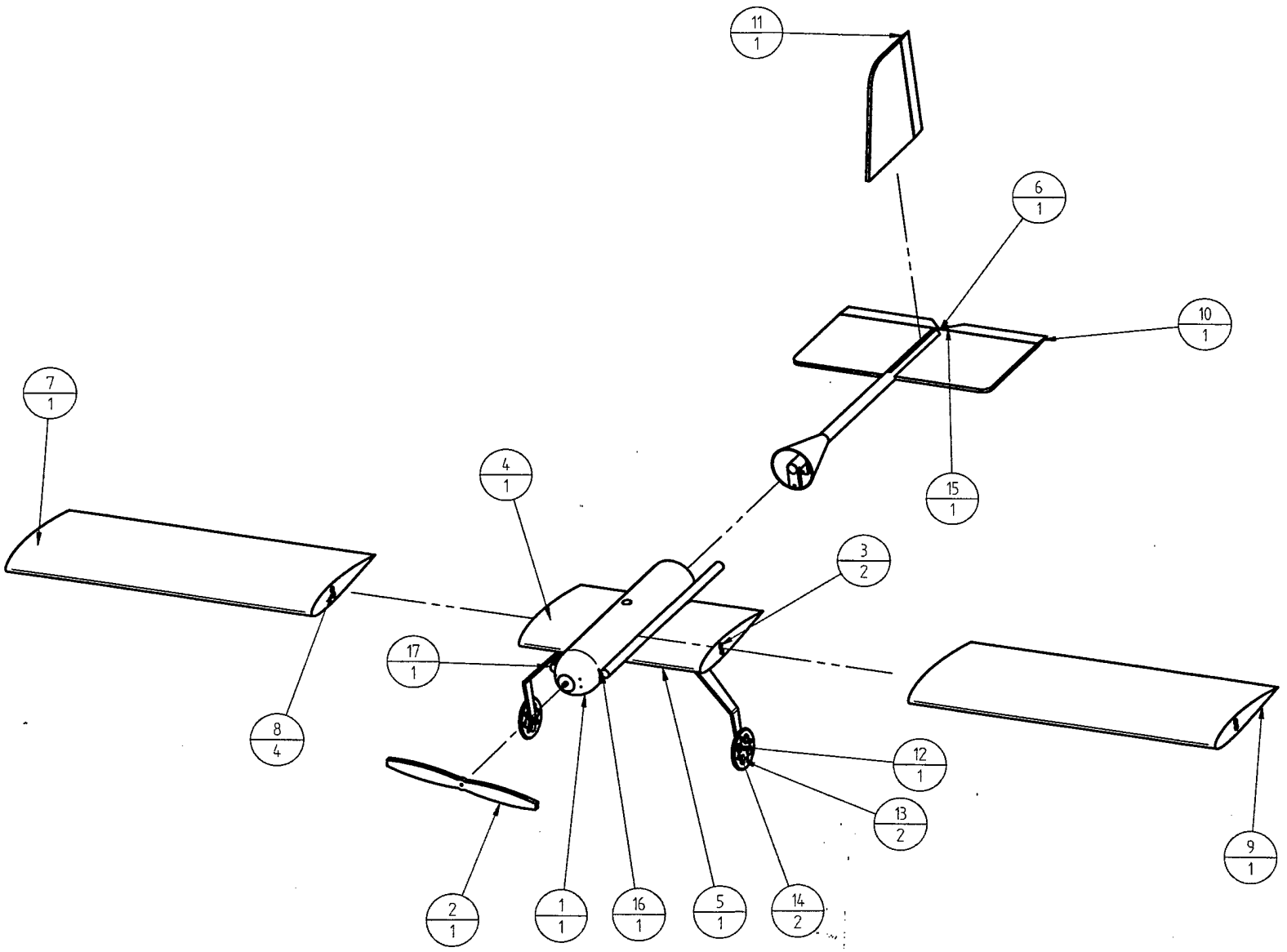




Unless Otherwise Specified:	Design Build Fly 2004
Dimensions are in inches	Miami University Oxford, Ohio
Tolerances are: Fractions: 1/16 Decimals: .XX = .01 .XXX = .005 Angles = .01	Title: Aircraft Configuration
Date: March 8, 2004	Drawing Number: 100A
THESE DRAWINGS ARE FOR REFERENCE ONLY.	Scale: 1:20 Sheet 1 of 1 page 44



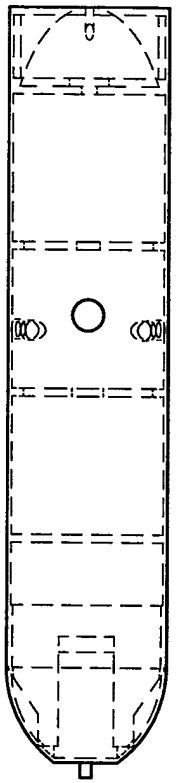
Unless Otherwise Specified:	Design Build Fly 2004
Dimensions are in inches	Miami University Oxford, Ohio
Tolerances are: Fractions: 1/16 Decimals: .XX = .01 .XXX = .005 Angles = .01	Title: Aircraft Assembly
Date: March 8, 2004	Drawing Number: 101A
THESE DRAWINGS ARE FOR REFERENCE ONLY.	Scale: 1:8 | Sheet 1 of 1 | page 45

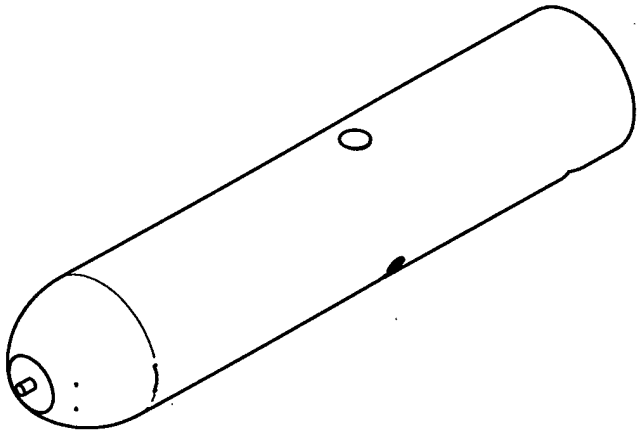


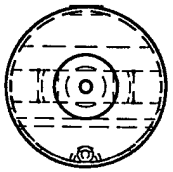
Item Number	Part	Quantity
1	Fuselage	1
2	Propeller	1
3	Carbon Fiber Rod - Center	2
4	Center Wing Section	1
5	Center Wing Section	1
6	Tail Extension	1
7	Outer Wing Section	1
8	Carbon Fiber Rods - Side	4
9	Outer Wing Section	1
10	Horizontal Stabilizer	1
11	Vertical Tail	1
12	Landing Gear	1
13	Landing Gear Wheel	2
14	Wheel Tread	2
15	Tail Wheel	1
16	Battery - Long	1
17	Battery - Short	1

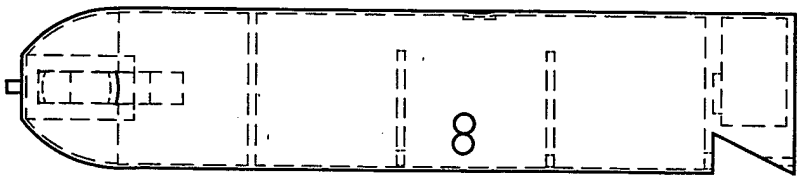
Unless Otherwise Specified:	Design Build Fly 2004
Dimensions are in inches	
Tolerances are:	
Fractions: 1/16	Miami University
Decimals: .XX = .01 .XXX = .005	Oxford, Ohio
Angles = .01	Title: Exploded Aircraft Assembly
Date: March 8, 2004	
THESE DRAWINGS ARE FOR REFERENCE ONLY.	Drawing Number: 102A
	Scale: 1:10 | Sheet 1 of 1 | page 46

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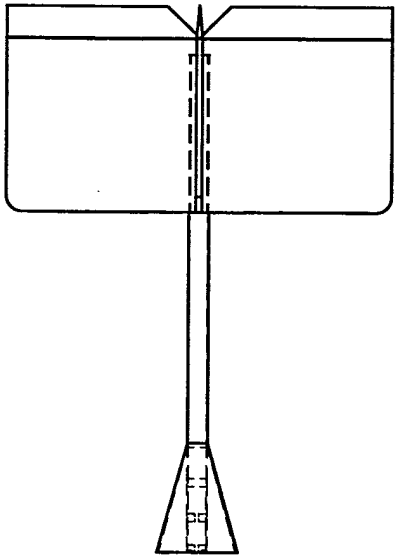


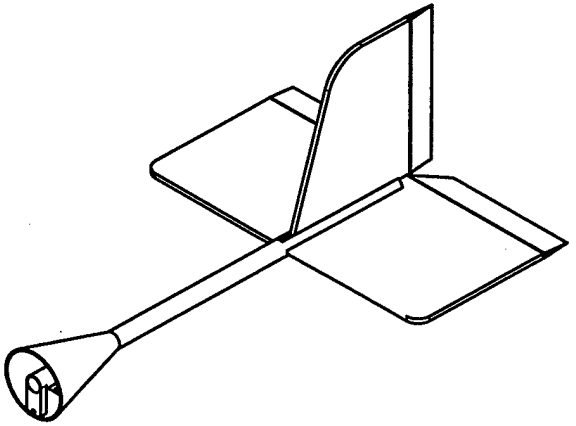


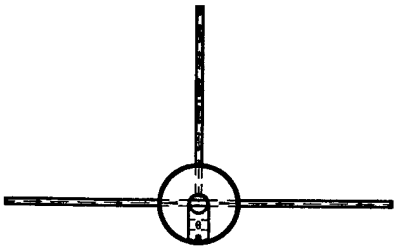


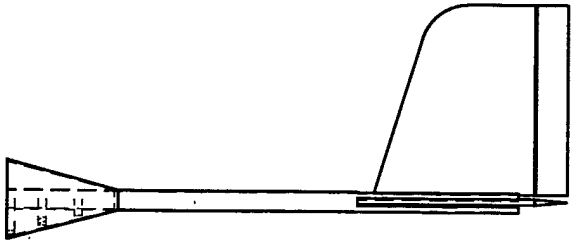
Unless Otherwise Specified:	Design Build Fly 2004
Dimensions are in inches	Miami University Oxford, Ohio
Tolerances are: Fractional: 1/16 Decimals: .XX = .01 .XXX = .005 Angles = .01	Title: Fuselage Configuration
Date: March 8, 2004	Drawing Number: 103A
THESE DRAWINGS ARE FOR REFERENCE ONLY.	Scale: 1:4 Sheet 1 of 1 page 47

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Unless Otherwise Specified:	Design Build Fly 2004
Dimensions are in inches	Miami University, Oxford, Ohio
Tolerances are: Fractions: 1/16; Decimals: .XX = .01, .XXX = .005; Angles = .01	Title: Tail Assembly
Date: March 8, 2004	Drawing Number: 104A
THESE DRAWINGS ARE FOR REFERENCE ONLY.	Scale: 1:8 | Sheet 1 of 1 | page 48

5.4. Aircraft Specification Table

Geometry

Length	56 in
Span	8 ft
Height	14.5 in
Wing Area	9.33 sq. ft
Aspect Ratio	6.86 in
Coefficient Volumes, C_{HT}, C_{VT}	.5715, .27

Performance

Propeller 1 (no water payload): 18/14

Max. Coefficient of lift	1.30
Max. Lift/Drag ratio	16.25
Max. Rate of Climb	820 ft/min
Stall Speed	22 mph
Max. Speed	55 mph
Take-off length empty	74
Take-off length gross weight	-

Propeller 2: (water payload): 18/12

Max. Coefficient of lift	1.30
Max. Lift/Drag ratio	16.25
Max. Rate of Climb	574 ft/min
Stall Speed	26 mph
Max. Speed	54 mph
Take-off length empty	-
Take-off length gross weight	145

Weight Statement

Component	**Weight (lb)**
Airframe	12
Propulsion system	7
Control system	.5
Payload system	.5
Payload	8.8
Empty weight	20
Gross weight	28.8

Systems

*Radio Controller	Futaba 10 Channel
*Radio Receiver	Futaba Radio Receiver
*Receiver Batteries	4 x Futaba NR-4J 4.8VDC (Ni-Cd)
*Battery Configuration	27 Cell Ni-Cd 3600 mAh
*Servos	5 x JR NES-571 Servos
*Control Rods	4 x Aluminum (variable lengths)
*Motor	Astro 90 Cobalt Motor

6. Manufacturing Plan and Processes

A primary task in completing the project is to manufacture and assemble the aircraft that has been designed. A well-planned manufacturing strategy is essential to accomplish the construction of the aircraft within the time, material, and resource constraints. For this reason, an outline was created with a plan defining the figures of merit pertinent to the manufacturing process. This plan includes the constraints just mentioned and a means of evaluating the process considered.

The following FOM's were chosen to aid in the decision of the construction techniques that will allow for the best procedure to complete the manufacturing plan. Using these FOM's and the process charts enabled for a detailed manufacturing plan for each component of the aircraft. During planning, if a procedure exceeded an allotted constraint of one of the following FOM's, an alternate procedure was then considered.

Material Availability

Material availability is a factor that must be taken into consideration when deciding on a manufacturing process. Material may need to be ordered and appropriate equipment capable of doing the machining must be available for use in order complete the desired process. If a material would take an extensive amount of time to order or equipment was not available to immediately use this constraint would be exceeded.

Required Skill Level

The required skill level for certain manufacturing processes may require some extra time in order to learn how to use the machine or handle the material in order to complete the process successfully. If a task was beyond the team's time capacity to learn in a timely manner, this constraint was exceeded.

Cost

Cost is always a factor when completing a project on a budget. It is essential to keep the manufacturing costs of the aircraft at a minimum to emphasize the importance of keeping the overall cost of production to a minimum. Disregard for cost would lead to a processing plan that may not be feasible due to limited funds. If a manufacturing process required an excess amount of money and could be done cheaper by other means, this constraint was exceeded.

Time

Time has been an important factor throughout the entire design project due to the deadline requirements. Keeping a pace that promotes the steady completion of the production of the aircraft will give the best opportunity to finish the construction in the required time. If it was necessary to spend a drastic amount of time on any single process, and as a result not allow for the manufacturing plan to continue as planned, this constraint was exceeded.

6.1. Process for Manufacture

For the construction of each of the subsystems of the aircraft, different manufacturing plans were researched and analyzed to allow for the best plan to be implemented. These choices were made for the wing, fuselage, tail, and landing gear.

6.1.1. Wing

The steps to manufacture the NACA 2415 wing of the aircraft are summarized in Table 12. Further explanation and reasoning is discussed in detail in the paragraphs that follow Table 12.

Table 12 – Wing Manufacturing Plan

	Process	Predecessors	Estimated Time (min.)
1	Cut aluminum templates		90.00
2	Cut sections of wing with hot-wire	1	180.00
3	Sand wing sections	2	30.00
4	Square-up wing sections	3	30.00
5	Cut wing into four sections	4	30.00
6	Attach wing sections with epoxy	5	60.00
7	Cut central section with hot-wire to match fuselage contour	6	60.00
8	Cut leading and trailing edges off wing with table saw	7	120.00
9	Cut balsa leading edge to length and attach to wing	8	120.00
10	Construct trailing edges and aileron by layering balsa sheets		180.00
11	Cut trailing edges and shape angle	10	120.00
12	Cut aileron to length	10	45.00
13	Mill slots in aileron to create ribs	12	60.00
14	Attach ¼" balsa to back surface of trailing edge of wing	8	60.00
15	Attach aileron to wing	13	15.00
16	Cut wing vertically for carbon fiber tubes	14,15	60.00
17	Route semi-circular channels for carbon fiber tubes	16	45.00
18	Cut channels in wing for servo wires	17	15.00
19	Mill slots in wing to create ribs		180.00
20	Cover wings with balsa sheeting	19	120.00
21	Route holes in wing for servos	20	60.00
22	Mount servos	21	30.00
23	Drill holes for hinges	14	30.00
24	Mount control horns	20	45.00
25	Monokote ailerons	24	120.00
26	Monokote wing	24	420.00
27	Fit ailerons to wing	25,26	25.00

In order to cut out the wings, aluminum templates of the airfoil were manufactured. These templates were then attached to blocks of foam of the appropriate length. An electrically heated wire was stretched between the templates and was pulled along the templates to create an accurate cut of an airfoil. This

method allows for the sections of foam to be cut with relative ease. Aluminum was used since it cannot be burned like wood templates can. In the design phase, templates made from plywood were used, but were quickly burned, resulting in bad cuts on subsequent wing sections. The concern that metal templates would act as a heat sink and impede the cutting of the foam proved to be unwarranted. After being cut, each section of wing was thoroughly sanded in order to make the surface as smooth as possible. Several sections of foam were attached end to end using epoxy in order to achieve a total wingspan of 8 feet. In order to make sure that the wing sections lined up correctly, a table saw was used to square up the edges before using the epoxy. The wing was then cut with a table saw into the four sections that would make up the wing, two for the central sections that will be permanently attached to the fuselage and the two outer sections that connect with telescoping tubes.

Once the four sections were cut, the central sections were cut using a hot-wire to allow the attachment of the fuselage. At this point, the leading edge and the trailing edge were cut off of the foam wings with the table saw in order to replace them with more durable balsa. The leading edge was a parabolic section of balsa that matched the profile of the wing closely enough that it did not need to be modified to match the profile. It was then cut to length with a hand saw and attached with epoxy.

The trailing edge and ailerons were constructed by layering 1/4"x4"x4' sheets of balsa in sections to achieve the correct length. In order to match the profile of the airfoil, the blocks of balsa wood were cut at an appropriate angle on the table saw. To complete this task, the foam section of the trailing edge that was cut off was used as a guide. These angled sections were then cut to length to fit the several sections of wing being built. In order to make the sections lighter a mill was used to cut out slots on the ailerons reducing the amount of wood left, leaving a rib-like structure with the same shape. The foam sections of wing had a 1/4" section of balsa attached along the backside surface, where the trailing edge was cut off. This section was cut using the table saw to match the profile of the airfoil and allow a solid point in the wing to attach the hinges for the ailerons.

After attaching the leading and trailing edges of the wing, with the exception of the ailerons, the wing was cut for the carbon fiber tubes with a hot-wire. A hot-wire was chosen for this cut in order to retain most of the foam in the cut. If a table-saw were used, an 1/8" would be lost in the cut when cutting the leading and trailing edges off of the wing. The cut was made 30% back from the leading edge, this point is where the wing profile is thickest and also where the center of gravity is going to be focused, thus necessitating the added strength of the carbon fiber spars.

A void for accommodating the carbon fiber tubes was made using a router table. Using a router bit that matched the outer diameter of the carbon fiber tubes allowed semi-circular channels to be cut along the length of the wing sections where the tubes could be sandwiched between the sections of foam for the

outer sections of wing. This process was more difficult for the central portion of the wing since it is permanently attached to the fuselage with carbon fiber tubes running through the fuselage. In the central portion of wing, the design incorporates one more carbon fiber tube placed 70% back from the leading edge in order to counteract the twisting moment of the wing. In order to manufacture this section, the tubes were first placed through the fuselage and then the three sections of wing were then attached with epoxy on each side of the fuselage. The multiple sections of wing being attached simultaneously made this step very difficult, yet resulting in a very strong portion ideal for attaching the two removable sections of wing.

Channels were cut in the wing to accommodate the wires for the servos before covering the sections with balsa. In an effort to reduce weight in the wing, a template was made that allowed for use of a hot-wire to cut out slots of the foam similar to the method described for the ailerons and trailing edge.

The wings were then covered in balsa sheeting with the use of an adhesive. Very thin layers of the adhesive are applied to both the balsa sheeting and the rest of the wing and ailerons and allowed to dry for 45 minutes. After drying, the balsa was laid on the wing at which point it is bonded instantly.

At this point the holes to accommodate the servos were routed into the wing and wooden blocks were mounted for attaching the servos. After mounting the servos, the ailerons were fitted to the wing and holes for the pin hinges were drilled. The points at which the hinges were attached to the ailerons were recessed into the ailerons in order to reduce the gap between the hinges and the rest of the wing allowing for better airflow around the wing. Before mounting the ailerons permanently, the control horns were mounted and the ailerons were covered with Monokote.

The final step in completing the manufacture of the wing was the Monokote covering. The Monokote is cut to size of the wing and then ironed onto the wing and shrunk with a heat gun to achieve a smooth, wrinkle free finish.

6.1.2. Fuselage

The construction of the fuselage will include mainly the manufacture of the water holding compartment. Table 13 shows the detailed steps in manufacturing the fuselage.

Table 13 – Fuselage Manufacturing Plan

	Process	Predecessors	Estimated Time (min.)
1	Cut polycarbonate tube to length		10.00
2	Drill holes for wing carbon fiber tubes	1	20.00
3	Cut baffles to size		20.00
4	Drill holes in baffles	3	10.00
5	Cut end caps to size		20.00
6	Affix baffles inside tube	1, 3	15.00
7	Affix end caps to front and rear of tube	1,5	15.00
8	Mill ram air valve in tube	1	10.00
9	Drill ½" diameter hole in rear end cap	7	5.00

6.1.3. Tail

The construction of the tail is centered on forming the tail shape and assembling the parts of the tail together. Table 14 illustrates the detailed steps in manufacturing the tail.

Table 14 – Tail Manufacturing Plan

	Process	Predecessors	Estimated Time (min.)
1	Cut balsa sheets to size		10.00
2	Curve leading and trailing edges	1	45.00
3	Mill out holes in tail	1	90.00
4	Mill out holes in tail for servos	1	60.00
5	Drill holes in trailing edge of tail for the hinges for rudder and elevator flaps	1, 2	30.00
6	Cut elevator and rudder flaps		20.00
7	Curve edges of elevator and rudder flaps	6	45.00
8	Drill holes in edge leading edge of flaps for hinges to connect to tail	6, 7	30.00
9	Apply Monokote to skin of tail	5, 8	30.00
10	Attach flaps and tail through hinges	5, 8	30.00
11	Cut carbon fiber rod to length		15.00
12	Cut slots in rod to support horizontal and vertical stabilizers	10	30.00
13	Fasten and glue stabilizers into the carbon fiber rod	9, 11	60.00
14	Fasten the carbon fiber rod to fuselage	12	30.00

6.1.4. Landing Gear

As stated in the component selection of the landing gear, the design chosen for the aircraft was an existing design that was already manufactured and available to the team. This component was designed by a previous Miami University AIAA team and was manufactured by an outside company because the proper equipment to manufacture the material was not available. However, for the company to manufacture the landing gear, it was specified that the aluminum needed to be cut to the proper length, width, and thickness. It would then be bent using a press to the proper angle specifications. The final step completed in the manufacturing process was to drill the holes at specified locations so that it could be mounted to the wing of the aircraft.

6.2. Manufacturing Milestone Chart

Component to be Manufactured	Date to be Manufactured by
Prototype	
Wing	1/20/04
Tail Components	1/27/04
Fuselage	1/22/04
Assemble Tail to Fuselage	2/5/04
Assemble wing to fuselage	2/10/04
Assemble landing gear to plane	2/10/04
Mount motor	2/10/04
Control System	2/12/04
Final Assembly	2/12/04
Final Design	
Wing	3/5/04
Tail Components	3/12/04
Fuselage	3/23/04
Assemble Tail to Fuselage	3/30/04
Assemble wing to fuselage	3/30/04
Assemble landing gear to plane	3/30/04
Mount motor	4/1/04
Control System	4/1/04
Final Assembly	4/1/04

7. Testing Plan

For the final design of the competition aircraft, it is necessary to allow time for testing and further development of the design. Testing provides the opportunity to ensure that previous analysis done on the design was accurate and if any factors of the plane were overlooked. If there are faults in the design, testing allows for adjustments to be made in order optimize the performance of the aircraft.

7.1. Testing Objectives

As mentioned at the conclusion of the preliminary design portion of this report, it was decided that a prototype aircraft closely resembling the final aircraft design would be built and tested. The advantages for completing the prototype aircraft were substantial. The prototype would provide an accurate impression of the performance of the final aircraft design. This can be true since the same airfoil and tail configuration would be used. The weight of the prototype could be gradually increased to mimic the water payload of the competition aircraft to understand how the aircraft would perform in the air. The prototype would also allow the pilot of the aircraft to gain more experience in understanding and

mastering the competition flight path and the aircraft's performance. This testing will help plan a better approach to the competition.

Also of importance is the time spent constructing the prototype airplane. This provided the first experience many of the team members had with building a model airplane. This enabled mistakes to be recognized and corrected for the future construction of the competition aircraft.

Additional stress testing will be completed on the landing gear and wing. Determination of optimal center of gravity location will be conducted on final completed aircraft. There will also be testing on factors concerning the flight of the prototype aircraft on the gas motor and during the actual flight.

7.2. Testing Schedule

The following schedule shown in Table 15 is an outline of the tests that were performed, there objective, and the dates in which they were performed.

Table 15 – Test Schedule

Test	Objective	Date
Landing Gear Test	Find approximate landing force that will be produced from aircraft	1/04
Wing Loading Test	Lift from wing tips and confirm structure can support weight of plane	1/04-2/04
Center of Gravity Test	Locate c.g. and adjust components so c.g. is located properly along wing for best flight performance	1/04-2/04
Gas Motor Test (prototype)	Run gas motor to determine approximate flying time	2/04
Flight Test	Monitor plane stability and performance to determine if adjustments need to be made	2/04-3/04

7.3. Testing Checklist

The importance of proper procedure during testing is considered in the following tables. Table 16 shows the pre-flight procedure that was completed before each flight to ensure all plane components were functional.

Table 16 – Pre-Flight Checklist

Center of Gravity			
Control Surfaces	Ailerons	Rudder Flaps	Elevator Flaps
- servos properly attached			
- servos functional			
- proper deflection			
Throttle - proper deflection			
Wing Loading Test - Lifting from wing tip without damage			
Fail Safe - Elevator up, Rudder right, Throttle off - Fuse Switch is on			
Construction - All components secure			

7.4. Component and Full Aircraft Testing of Prototype

Both static and dynamic testing of the prototype is essential to the success of the aircraft's performance. Though the prototype employs the use of a gas engine rather than an electric motor, many important variables and design criteria can be tested and evaluated. This was done by limiting the throttle of the gas motor to simulate the performance of the electric motor which will be used on the competition aircraft. Because the prototype aircraft was designed specifically to replicate the design of the electric competition aircraft, the testing of the prototype will provide much insight into the performance, strength, durability, and flight characteristics of the competition aircraft.

7.5. Conclusions Static and Dynamic of Prototype

The static prototype testing and the dynamic, in-flight testing of the prototype have proven effective measures of the competition aircraft's potential performance. Although not all design criteria for the prototype exactly emulate the characteristics of the competition aircraft, the testing will still prove useful.

7.6. Component and Full Aircraft Testing and Conclusions of Final Design

Testing of the final design involved things learned from the testing of the prototype. It was decided that overall aircraft weight could be reduced by making lighter wings. This was achieved by removing a substantial amount of Styrofoam from the wing by cutting holes in the wing structure. From calculations it was determined that overall strength of the wing would not be compromised due to the strength of the carbon fiber rods running through the wing.

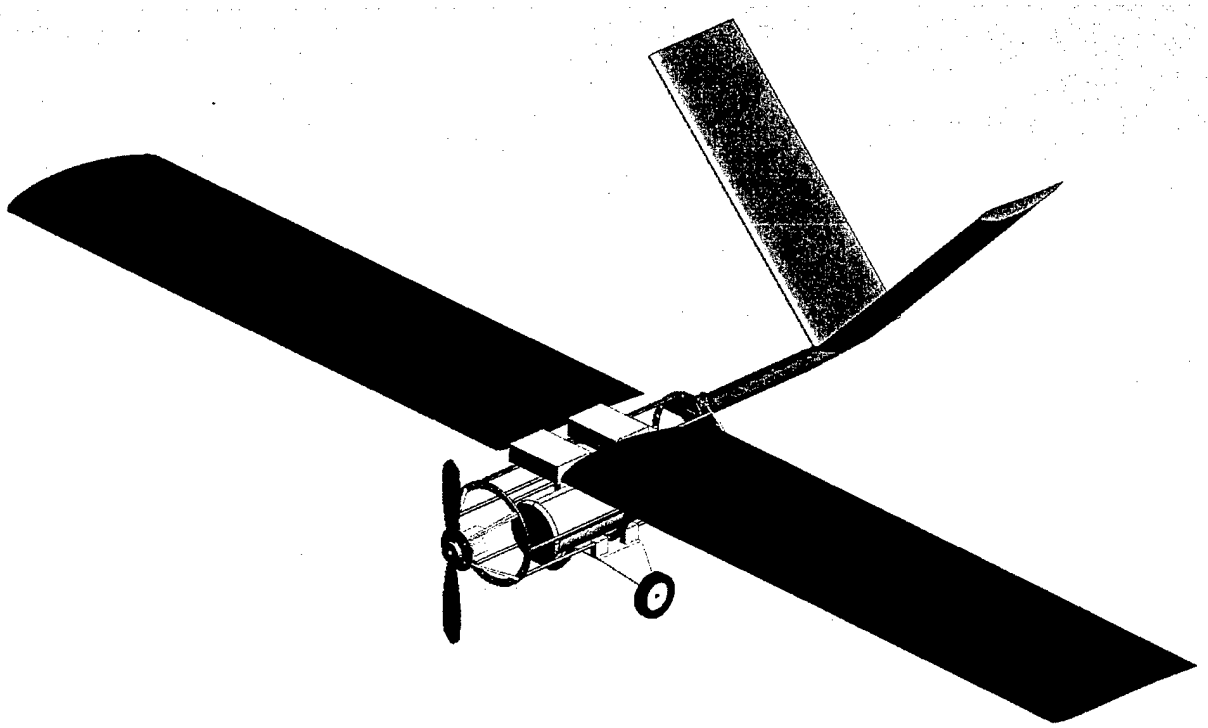
Also tested were the results of decreasing the angle of incidence of the wing. The prototype showed that the angle of attack was too large and caused the airplane to have a tendency to pitch up during flight. By decreasing the angle of incidence it was shown that the tendency of the aircraft to pitch up was reduced.

Further testing on the landing gear was not required for the final design. Since it is the same gear that was used on the prototype, it was shown that it would also work for the final design.

Also learned from the prototype was that the tail volume was too big and could be reduced. After calculating a new and smaller tail volume, it was implemented on the final design. Test flights showed that this was sufficient for the flight of the airplane.

UNITED STATES MILITARY ACADEMY
WEST POINT, NY

2004 AIAA DESIGN / BUILD / FLY COMPETITION



Team Boombatz Grandioso

Page #

1. Executive Summary **3**

2. Management Summary **6**

2.1 Design Team Assignment 6
2.2 Design Plan 6

3. Conceptual Design **10**

3.0 Introduction 10
3.1 Problem Statement 12
3.2 Design Parameters 12
3.3 Figures of Merit 13
3.4 Design Parameter Results 14
3.6 Conceptual Design Results 14

4. Preliminary Design **17**

4.0 Introduction 17
4.1 Wing 19
4.2 Electrical and Propulsion Systems 22
4.3 Fuselage 23
4.4 Landing Gear 24
4.5 Empennage 25
4.6 Center of Gravity Calculations 26

5. Detailed Design **27**

5.0 Introduction 27
5.1 Wing 28
5.2 Fuselage 29
5.3 Landing Gear 29
5.4 Empennage 30
5.5 Electronics and Propulsion 30
5.6 Specifications 30
5.7 Drawing Package 32

6. Manufacturing Plan **36**

6.0 Introduction 36
6.1 Wing 36
6.2 Fuselage 36
6.3 Landing Gear 36
6.4 Empennage 37
6.5 Payload 37

7. Testing Plan **38**

7.0 Introduction 38
7.1 Testing 38
8. References **40**

Team Boombatz Grandioso consists of six United States Military Academy cadets and was formed to design, build, and fly a radio controlled aircraft in the AIAA Cessna/ONR Student Design/Build/Fly Competition. The competition will be held on April 23-25, 2004, at The Cessna Pawnee East Field in Wichita, Kansas. The goal of Team Boombatz Grandioso is to design, construct, and fly a winning aircraft.

There are two mission profiles for the 2004 competition. Every team must fly both missions. The simplest mission is the Ferry mission. The aircraft must take-off without a payload, complete four laps with one 360 degree turn on the downwind leg of each lap, and then land. During the Ferry mission the airplane must complete a total of four turns. The second mission is the Fire Fight mission. The payload for this mission is four liters of water. This mission is flown in two sorties. Each sortie consists of a take-off, one lap around the course, with one 360 degree turn, a release of the payload on the downwind leg of the course, and a landing. In between the sorties another payload is loaded into the airplane.

To achieve the best score our team wanted to minimize flight time and maximize the amount of water carried. The mission score is the difficulty factor multiplied by the weight of water released and divided by the mission flight time. The maximum payload for each sortie is four liters. The maximum take-off distance is 150 feet. Additionally, the aircraft must be able to be broken down to fit in a two foot wide by one foot high by four foot long box. The airplane must also survive an inverted wing tip test. All of these factors were used to formulate the customer requirements, engineering requirements, and engineering targets that were used to develop the design.

The team's first step towards success was spearheaded by the Cadet In Charge (CIC) and the Assistant Cadet In Charge (ACIC). The CIC and ACIC began the conceptual design in August of 2003 and worked alone until the first of November. The conceptual design phase took the problem statement and project constraints and derived a conceptual design, using the five step engineering process. We used an Objectives Tree and a Pairwise Comparison to establish and weigh a list of customer requirements. Then a list of engineering requirements was established. Using a Quality Functional Deployment (QFD), we established a specifications list for the aircraft. In order to establish reasonable goals, the design was benchmarked against the *Nitro Airstrike* and the *Mega Stealth 2,* both of which are commercially purchased remote controlled airplanes. Three possible concepts were developed using a Morphological Chart. The conceptual design was selected using a numerical evaluation matrix to determine how well each concept could meet the customer requirements. The details of the conceptual design are in section 3.

On the first of November 2003 Team Boombatz Grandioso gained four members, bringing the team to full strength with six members. The CIC assigned each member a specific aircraft component. The six aircraft components were wing, payload, fuselage, empennage, landing gear, and propulsion. Provided with the conceptual design each member was able to refine and improve their specific component. The combination of the advanced components led to the preliminary design. The

preliminary design included Computer Aided Drafting Modeling and Analysis and Numerical Evaluation Matrixes. The preliminary design revised the conceptual design to include geographical shapes, dimensions, and weight assignments. The fine points of the preliminary design are in section 4.

Finally, the design teams brainstormed and developed concepts for each of their components and the connections involved. This went through a process of continuous refinement as manufacturing progressed. In addition, performance calculations were made based upon the preliminary design, allowing for the selection of propulsion components.

The final specifications of the Boombatz Grandioso Flier can be viewed in Figure 1.1.

GEOMETRY	
Length (feet)	4.5
Span (feet)	11.75
Height (feet)	0.75
Wing Area (Square feet)	17.625
Aspect Ratio	6.286
Control Volumes	Ailerons & V-tail
PERFORMANCE	
CLmax	1.23
L/Dmax	24.46
Maximum Rate of Climb (knots)	2.751
Stall Speed (knots)	16.353
Maximum Speed (knots)	53.44
Take-off Length Empty (feet)	17.94
Take-off Length Loaded (feet)	43.41
WEIGHT STATEMENT	
Airframe (lbf)	5.8
Propulsion System (lbf)	6.3
Control System (lbf)	1.8
Payload System (lbf)	0.5
Payload Empty Weight (lbf)	14.4
Gross Weight (lbf)	22.4
SYSTEMS	
Radio Used	Futaba FP-TP-FM
Servos Used	Futaba series (FR-S148, S-3003, FPR-1480F)
Battery Configuration Used	7 x 5 SR Batteries
Motor Used	Astro 90G
Propeller (nominal)	24 x 10 Menz Propeller

Figure 1.1 – Airplane Data

2.1 Design Team Assignment

Team Boombatz Grandioso consists of six United States Military Academy cadets and was formed to design, build, and fly a radio controlled aircraft in the AIAA Cessna/ONR Student Design / Build / Fly Competition. The competition will be held on April 23-25, 2004, at The Cessna Pawnee East Field in Wichita, Kansas. The goal of Team Boombatz Grandioso is to design, construct, and fly a winning aircraft.

The team's project advisor is CPT David Stringer. The team is divided into six separate entities responsible for one of the six sub - systems of the airplane. The CIC and ACIC are accountable for the conceptual design, guidance on the preliminary design and detailed design, design communication segment, and project management. Due to the small team size the CIC and ACIC are also responsible for an aircraft sub - system. Each member is responsible for the preliminary and detailed desing of their component, as well as the fabrication of their component. Figure 2.1 depicts the team organization and personnel assignment.

Personnel Assignment	
CIC/Wing	Brent Pafford
ACIC/Payload	Vincent Sayegh
Fuselage/CAD Specialist	Robert Crapanzano
Empennage	Thomas Jones
Landing Gear	Brad Smith
Power	Derrick Yohe

Figure 2.1 – Team Organization

2.2 Design Plan

The design plan was for the CIC and ACIC to execute the conceptual design and then brief the other four members to provide them with background information for their respective sub - system preliminary and detailed design. During the first two months the CIC and ACIC meet three times a week to create a conceptual design concept. Throughout the conceptual design phase the team leaders worked with the advisor, conducting in progress reports (IPR), to maintain on task. While in the conceptual design phase, the leadership created a milestone chart that spanned form September 2003 to April 2004. This chart (Figure 2.2) was formed in order to meet all contest requirements and to ensure that the airplane is complete and ready to fly on April 23, 2004.

Milestone Chart						
Monday	Tuesday	Wednesday	Thursday	Friday	Saturday	Sunday
September 1	2	3	4	5	6	7
Objectives Tree & Pairwise Comparison (Problem Statement & Research)						
8	9	10	11	12	13	14
Functional Decomposition						
15	16	17	18	19	20	21
Quality Function Deployment & Specifications List With Engineering Targets						
22	23	24	25	26	27	28
Morphological Chart & Development of Concepts (Brainstorm/Sketch)						
29	30	October 1	2	3	4	5
CAD Modeling & Analysis of Alternatives						
6	7	8	9	10	11	12
Numerical Evaluation Matrix						
13	14	15	16	17	18	19
Conceptual Design Complete						
20	21	22	23	24	25	26
Prepare To Brief Conceptual Design to Team & Advisory Committee						
27	28	29	30	31	November 1	2
			Electronic Entry Form Due		**Brief Conceptual Design to Team**	
3	4	5	6	7	8	9
IPR w/ Advisor	Form Generation					
10	11	12	13	14	15	16
Break Off Into Component Sections & Begin Preliminary Design						
17	18	19	20	21	22	23
24	25	26	27	28	29	30
IPR w/ Advisor			[redacted]			
December 1	2	3	4	5	6	7
8	9	10	11	12	13	14
Complete Preliminary Design			**Preliminary Design Brief**			
15	16	17	18	19	20	21
Term End Exams						
22	23	24	25	26	27	28
[redacted]						

Monday	Tuesday	Wednesday	Thursday	Friday	Saturday	Sunday
December 29	30	31	January 1	2	3	4
5	6	7	8	9	10	11
	Begin Manufacturing & Order: Wing, Landing Gear, Payload Tank, & Power Systems					
12	13	14	15	16	17	18
	Build					
19	20	21	22	23	24	25
	Build					
26	27	28	29	30	31	February 1
	Build					
2	3	4	5	6	7	8
	Fabrication Complete For: Tank, Empennage, Wing, & Fuselage					
9	10	11	12	13	14	15
	Place Payload Tank In Fuselage					
16	17	18	19	20	21	22
	Connect Fuselage, Empennage, Tank, Landing Gear, & Wing					
23	24	25	26	27	28	29
	Connect All Wiring To Control Surfaces & Power Assembly					Finish Report
March 1	2	3	4	5	6	7
	Written Report Due		Mail Report			
8	9	10	11	12	13	14
	Airplane Completed & Ready To Fly					
15	16	17	18	19	20	21
22	23	24	25	26	27	28
	Test Flights & Design and Fabrication Critique					
29	30	31	April 1	2	3	4
	Test Flights & Design and Fabrication Critique					
5	6	7	8	9	10	11
	Test Flights & Design and Fabrication Critique					
12	13	14	15	16	17	18
	Transport (Send) Airplane to Wichita					
19	20	21	22	23	24	25
		Leave For Wichita		**AIAA DBF @ Contest Wichita, Kansas**		

Figure 2.2 – Milestone Chart

Throughout the preliminary design stage every member focused on their specific sub - system. It was easier for each individual to do research and analysis for their section of the airplane instead of the two leaders focusing on the entire plane. Therefore, the preliminary design stage moved at a much more rapid pace. Furthermore, the analysis work during this stage was exceptional and set the entire team up for success during the detailed design stage and the fabrication stage. The preliminary design stage was overseen by the ACIC. He ensured that the team members stayed on task and produced valid results.

Towards the end of the first semester, the team began the detailed design of their components by brainstorming ideas and discussing their feasibility with the department technicians. The material ordering process began, and the budget was overseen by the CIC. The detailed design and manufacturing progressed almost simultaneously throughout the next semester. During this time, the team leaders oversaw the detailed design and construction, adjusted the timeline, worked on the design communication requirements, and formulated a new plan for design evaluation.

Report preparation started early in the first semester. The CIC was responsible for the compilation of the report and most of the report's context. The report was fifty percent complete at the end of the first semester, due to department policy which called for a semester capstone report. The report was completed through the preliminary design phase.

3.0 Introduction

The conceptual design phase is like the foundation of a house and like a foundation a weak conceptual design will create catastrophic failure in the future. Knowing that the conceptual design phase was extremely important we designed an initial aircraft configuration that would meet the customer requirements. The CIC and ACIC read the contest rules and parameters and then developed a list of customer requirements based on the nature and constraints of the contest. We used a pairwise comparison chart to weigh the customer requirements against each other. Next we used a functional decomposition to derive the functions required to meet the customer requirements. A morphological chart was made to create concepts that could accomplish the functions.

Conceptual designs were organized based off of the concepts in the morphological chart. We used a numerical evaluation matrix to choose the best conceptual design. The most helpful tool during the conceptual design stage was the Qualitative Functional Deployment (QFD). The QFD relates the qualitative figures of merit, engineering requirements, to the quantitative figures of merit, customer requirements. The engineering requirements are weighted based on absolute importance and relative importance. The absolute importance is based off of how well the engineering requirement relates to the customer requirement. The relative importance is the fraction of the engineering requirement with respect to the sum of all of the engineering requirements. We used the *Nitro Airstrike* and the *Mega Stealth 2* as competition benchmarks to determine acceptable engineering targets.

Customer Requirements (Whats)	Weight (Customer Importance)	Large L/D	Large W/S	Large T/W	Large e	Large AR	Small K	Small Cdo	Acceleration (high)	Velocity (high)	Unloaded Weight	Strenght	Minimize Conections	Minimize Parts	Minimize Electrical Conections	Related Airplane Cost	Cruise Thrust	Wing Planform Area	Assembly Time	Loaded Takeoff Distance	Battery Weight	Static Landing gear load
Direction of Improvement		↑	↑	↑	↑	↑	↓	↓	↑	↑	↓	↑	↓	↓	↓	↓	↑	↓	↓	↓	↓	↑
Fast	0.11	9	3	9	1	3	3	3	9	9	9	3	1	1	1	1	9	3	1	1	3	1
High AR	0.22	3	3	1	9	9	9	1	3	3	1	3	1	1	1	1	1	9	1	3	1	1
Light Weight	0.17	1	9	9	1	1	1	1	9	9	9	1	1	3	3	1	1	3	1	9	9	1
Low Cost	0.06	1	1	1	1	1	1	1	1	9	1	3	9	9	9	9	1	3	1	1	3	1
Large Wing Loading	0.11	3	9	1	1	3	1	1	1	3	9	1	1	1	1	1	1	9	1	3	1	1
Easy To Assemble	0.08	1	1	1	1	1	1	1	1	1	1	3	9	9	9	3	1	3	9	1	3	1
Durable	0.06	1	1	1	1	1	1	1	1	1	1	3	3	3	3	3	3	1	3	1	3	9
Easy To Control	0.03	3	1	1	1	1	1	1	1	1	1	1	3	1	1	1	3	3	1	3	3	1
Payload @ CG	0.17	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	1	3	1	1	9	1
Absolute Importance		2.44	3.72	3.06	2.61	3.06	2.83	1.06	3.50	4.17	3.94	1.89	2.11	2.39	2.39	1.56	1.89	4.39	1.61	2.89	2.83	1.28
Relative Importance		0.04	0.07	0.05	0.05	0.05	0.05	0.02	0.06	0.07	0.07	0.03	0.04	0.04	0.04	0.03	0.03	0.08	0.03	0.05	0.05	0.02
Competition Benchmarks																						
Units									ft/s^2	ft/s	lbf	ksi	#	#	#	$	lbf	ft^2	s	ft	lbf	lbf
Engineering Targets		18	1.5	0.5	1	7	0.05	0.04	15	30	10	50	12	8	5	###	9	36	60	115	3.8	21

Competition Benchmarks

△ Nitro Aerobatic
□ Mega Stealth 2

1 2 3 4 5

9 Strong Correlation
3 Medium Correlation

Figure 3.1 – QFD

Engineering Requirement	Goal	Weight
Wing Planform Area	> 15 ft^2	7.92%
Velocity (high)	> 30 ft/s	7.52%
Unloaded Weight	< 14 lbf	7.12%
Small W/S	< 1.5 lbf/ft^2	6.32%
Acceleration (high)	> 15 ft/s^2	6.32%
Large T/W	> 1	5.52%
Large AR	> 7	5.52%
Loaded Takeoff Distance	< 115 ft	5.22%
Small K	< 0.046	5.12%
Battery Weight	< 3.8 lbf	5.12%
Large e	> 0.98	4.71%
Large L/D	> 8	4.41%
Minimize Connections	< 12	3.81%
Minimize Parts	< 8	4.31%
Minimize Electrical Connections	< 5	4.31%
Cruise Thrust	> 9 lbf	3.41%
Strength	> 50 ksi	3.41%
Related Airplane Cost	< $3000	2.81%
Assembly Time	< 60 s	2.91%
Static Landing gear load	< 21 lbf	2.31%
Small CDo	< 0.04	1.91%

Figure 3.2 – Specifications List

3.1 Problem Statement

The mission of Team Boombatz Grandioso is to design, build, and fly a winning aircraft that is light weight and has a low rated aircraft cost. The design requires that the plane be disassembled and fit inside a 1 foot by 2 foot by 4 foot box. This means that the connections be temporary, but also strong and secure to ensure safe flight. The airplane must also be structurally sound to survive the inverted wing tip test. The most important of the aerodynamic considerations are the minimization of weight. The aircraft is designed to compete in both the ferry and fire fight missions.

3.2 Design Parameters

Before we selected any concepts for our aircraft we had to understand what functions the aircraft must perform. To do this we used a functional decomposition chart. The three main functions we came up with are interact with payload, interact with air, and interact with ground. Each function has several sub-functions. We then created concepts that would meet the requirements of the functions. The final step was to use a numerical evaluation matrix to select the best concept.

3.2.1 Interact with Payload

Interact with the payload is the first function. This function consisted of two sub-functions, containing the payload and deploying the payload. The concepts for containing the payload included square box, cylindrical tank, and a round trough. The trough concept was not realistic, because the water would splash all over the inside of the plane. The concepts for self-deploying the payload were to open a trap door, open a gate, solenoid valve, or pull a plug. Due to the massive weight and expense of a solenoid valve that concept was deemed not feasible.

3.2.2 Interact with Air

Interacting with the air has three sub-functions. The first sub-function is that the aircraft must produce thrust. The three concepts created to create thrust are a tractor propeller mounted on the front of the airplane, a pusher propeller mounted on the rear of the airplane, and a combination the two propellers. The second sub-function was that the plane had to maneuver in the air. We created two concepts to maneuver the plane; a v-tail configuration or a vertical and horizontal stabilizer. Finally, the most important sub-function for any aircraft is the production of lift. The two concepts for this function are either a cambered airfoil or a symmetric airfoil.

3.2.3 Interact with Ground

The function of interacting with the ground has two sub-functions. These are maneuvering and absorbing the impact of landing. The landing gear accomplishes both sub-functions. There were two possibilities for the landing gear configuration. A tricycle landing gear has the main gear aft of the center of gravity and a secondary landing gear on the nose. A tail dragger is the exact opposite in which the second landing gear is mounted aft of the main landing gear.

3.3 Figures of Merit

There are two types of figures of merit (FOM's). Engineering requirements are quantitative and customer requirements are qualitative. We used the FOM's to screen the competing concepts and choose the best one.

Customer requirements are qualitative requirements that the design must meet. The customer requirements that we developed were that the design had to be fast, have a high aspect ratio, lightweight, low rated aircraft cost, small wing loading, easy to assemble, durable, easy to control, and the payload mounted at the center of gravity. The most important customer requirement is the low rated aircraft cost. Not only is this part of the final score during the contest, but we are restricted to a budget. A lightweight aircraft is also important, because we want to be able to fly the airplane and get it off of the ground. To do this the airplane must be lightweight.

The engineering requirements for our aircraft are qualitative requirements. The engineering requirements are numerous. We chose our engineering requirements based on the customer

requirements and the contest rules. The most important engineering requirement was a large wing plan form area. This was so that the airplane would be able to create enough lift to take off. This airplane is carrying a large payload and will need a large wing section to produce lift.

3.4 Design Parameter Results

The results of our design parameters are displayed in Figure 3.1, which shows the quality functional deployment. The output of the QFD chart is the Specifications List. The Specifications List is the objective output of the design.

3.5 Conceptual Design Results

To organize our concepts we created a morphological chart. From the morphological chart (Figure 3.3) three concepts were selected. Figure 3.5 shows the numerical evaluation matrix used to select the best concept.

Means / Features / Functions	1	2	3
Fuselage	Swept Rectangular	Rectangle	Swept cylinder
Wing	Symmetric	Delta	Swept Symmetric
Empennage	V - Tail	Horizontal & Vertical stabilizers	
Cargo Bay	Square Box	Round Tank	
Payload Release Mechanism	Trap Door	Valve	Gate
Power Source	Battery Powered		
Landing Gear	Tricycle Gear	Tail dragger	
Control	Remote Control		

Figure 3.3 - Morphological Chart

From the morphological chart we chose three concepts. The concepts are provided below in Figure 3.4.

		Concepts	
Means / Features / Functions	1	2	3
Fuselage	Cylinder	Rectangle	Swept Cylinder
Wing	Cambered	Swept Symmetric	Symmetric
Empennage	Horizontal & Vertical stabilizers	V - Tail	Horizontal & Vertical stabilizers
Cargo Bay	Cylindrical Tank	Square Box	Round Tank
Payload Release Mechanism	Plug	Gate	Valve
Power Source	Battery Powered	Battery Powered	Battery Powered
Landing Gear	Tricycle Gear	Tricycle Gear	Tricycle Gear
Control	Remote Control	Remote Control	Remote Control

Figure 3.4 - Concepts

Design Constraints/ Objectives	Weight %	1	2	3
Fast	11.11%	0.3	0.4	0.8
		3.33%	4.44%	8.89%
High AR	22.22%	0.8	0.7	0.8
		17.78%	15.56%	17.78%
Light Weight	16.67%	0.7	0.6	0.9
		11.67%	10.00%	15.00%
Low Cost	5.56%	0.4	0.5	0.6
		2.22%	2.78%	3.33%
Small Wing Loading	11.11%	0.7	0.6	0.7
		7.78%	6.67%	7.78%
Easy To Assemble	8.33%	0.5	0.7	0.7
		4.17%	5.83%	5.83%
Durable	5.56%	0.8	0.7	0.8
		4.44%	3.89%	4.44%
Easy To Control	2.78%	0.8	0.6	0.7
		2.22%	1.67%	1.94%
Payload @ CG	16.67%	0.8	0.7	0.6
		13.33%	11.67%	10.00%
		66.94%	62.50%	75.00%

Figure 3.5 – Numerical Evaluation Matrix

To properly evaluate each concept they were weighted against each other and received a score between 0 and 1, with 1 being the best. The concept weights where then multiplied by the weight of the customer requirements providing our team with the best concept, which was concept number 3.

The parameters of the conceptual design can be viewed in Figure 3.6.

Conceptual Design	
Function	**How Achieved**
Interact with Payload	
- Contain Payload (Water)	**Cylindrical Tank**
- Deploy Payload	**Mechanical Lever**
	Gravity Deploy
Interact with Air	
- Create Thrust	**Tractor Propeller**
- Maneuver	**Horizontal & Vertical Tail Control Surfaces**
- Create Lift	**Cambered Airfoil**
Interact with Ground	
- Absorb impact and Maneuver	**Tricycle Gear**
Interact with R/C Controller	**R/C unit**
Interact with Users	**Remote Control**

Figure 3.6 – Conceptual Design Details

4.0 Introduction

The preliminary design phase is where we add the frame work to the foundation. Each member took their separate aircraft sub - system and worked on improving configurations, materials, and structures, each team drafted possible preliminary designs for their sub-design. We also began to create detailed CAD drawings to gain a feel for where all of the components were going to go and ensure that the airplane could fit inside the 1' x 2' x 4' box. The details of the preliminary design are in Figure 4.1

Preliminary Design	
Wing	Carbon Spars
	Foam Wing Covered With Monokote
	Ailerons For Control
	NACA 4412
Payload Container	Cylindrical Shape
	Composite
	Twist Off Cap For Filling
Fuselage	Swept Cylindrical Shape
	Top-Mounted Wing
	Holds Payload Container
Empennage	Trapezoidal Planform
	1/3 Planform Area of Main Wing
	V-Tail System
Landing Gear	Tricycle Landing Gear
	Large Wheels
	Shock Absorption
Propulsion	3 hp provides 30 mph
	Astor 90 Motor
	12 Inch Rotor Radius

Figure 4.1 – Preliminary Design Details

4.1 Wing

The wing was composed of a NACA 4412 foam cut out. This section discusses how we arrived at a NACA 4412 as our airfoil.

4.1.1 Construction

We want the wing to have a span of 11 feet and a chord of 1.75 feet. To have a wing this long survive the inverted wing tip test we will have to reinforce the wing with carbon fiber shafts. We also decided to cover the wing with carbon fiber and not monokote for added strength and to reduce drag. The core of the wing will be made of foam in the shape of a NACA 4412.

4.1.2 Planform Design

Assuming a cruise speed of 30 ft/s, in accordance with the engineering target, it was then necessary to further design the planform of the wing. We selected a rectangular wing because it was the simplest to manufacture, and the efficiency penalty for the rectangular wing was minor in comparison with the simplicity of design.

The second step was to select an aspect ratio. Our engineering goal was to have an aspect ratio of 7 or greater, comparable to heavy lift cargo planes. With the required planform, this would give us as eleven foot wingspan and a one and one half foot chord. We knew that since the wings would be mounted in two pieces, with the fuselage in between, that the effective span and aspect ratio would be greater, increasing actual performance. However, since we lacked the capability to model that, and since we wished to design conservatively, we ignored the effect that would have.

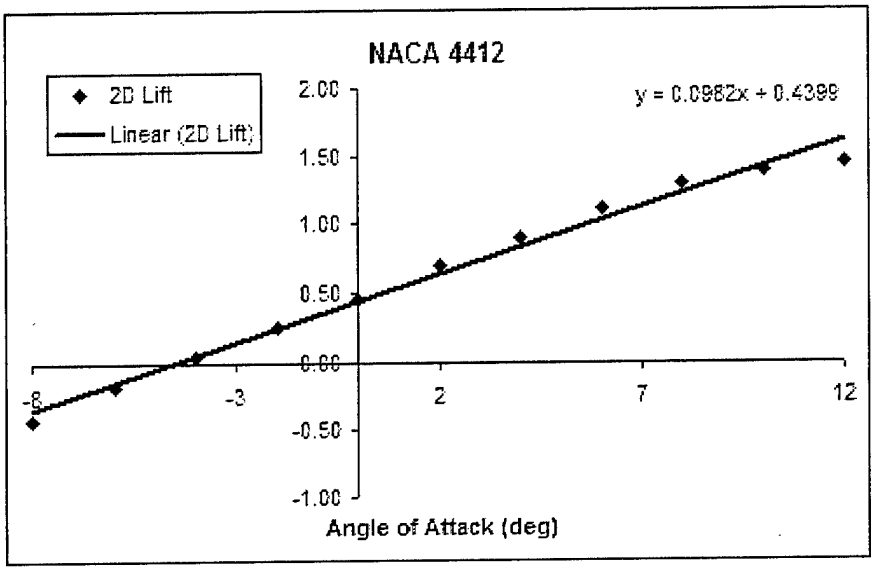
Finally, we decided to mount our ailerons outboard, using approximately 30% of the chord. This would give us a large amount of control authority with an unobtrusive aileron. Flaps were not a part of the design because R/C aircraft do not generally need them, and flaps would significantly increase the complexity of the design while also increasing the weight and the number of servos.

4.1.3 Airfoil Selection

Airfoil Selection was a difficult process. There were no easily accessible Airfoil Selection programs. The facilities at West Point were also not adequate for the manufacturing and testing of numerous airfoil sections. However, a solution was found. A two-dimensional design and analysis program was found on the internet. This program, X-foil, has the capability to get 2D airfoil data for a range of angles of attack for a given Reynolds number. Using our data, and analyzing a number of NACA airfoils the 2D analysis was performed. Then an Excel workbook was made that used aerodynamic relationships and planform data to transform the 2D data into 3D data, allowing us to choose an airfoil that best suited our design. Both the NACA 4412 posted the best results according to the 3D design program. The slow speed of the plane means that the leading edge of the airfoil must be thin. Because

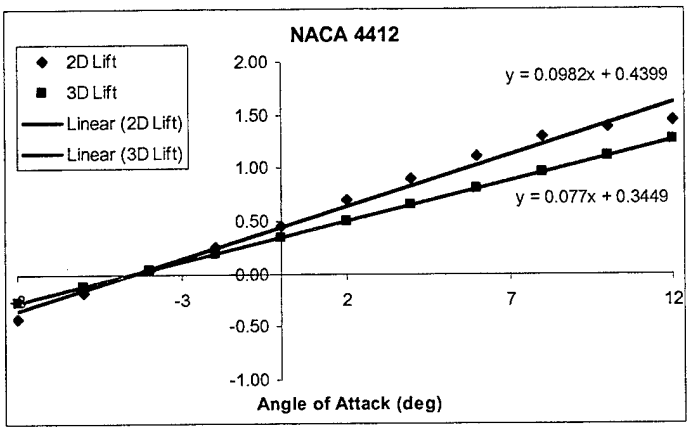
the NACA airfoil had both of these characteristics, the NACA airfoil was the natural choice. Figures 4.2 and 4.3 shows parts of the Excel 2D to 3D data conversion.

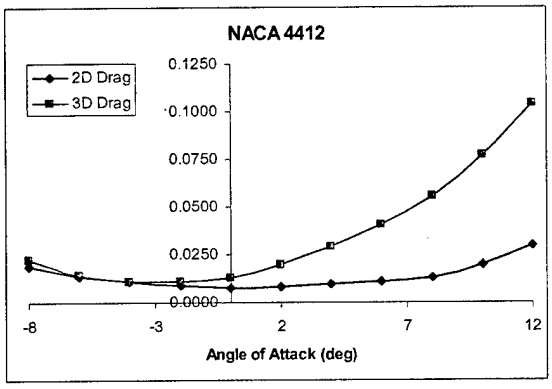
NACA 4412 Figures 4.5 and 4.6		
α	Cl	Cd
-8	-0.42	0.0191
-6	-0.18	0.0134
-4	0.04	0.0105
-2	0.26	0.0090
0	0.46	0.0072
2	0.69	0.0080
4	0.90	0.0092
6	1.11	0.0107
8	1.30	0.0130
10	1.39	0.0198
12	1.46	0.0297



ct	cr	b	AR	δ	e	Clα		CLα		αL=0		x intercept
in	in	ft				0.098	per degree	0.077	per degree	-4.480	degrees	0.345
18	18	11	7.3	0.06	0.9	5.626	per radian	4.412	per radian			

Figure 4.2 – 2-D Lift Curve





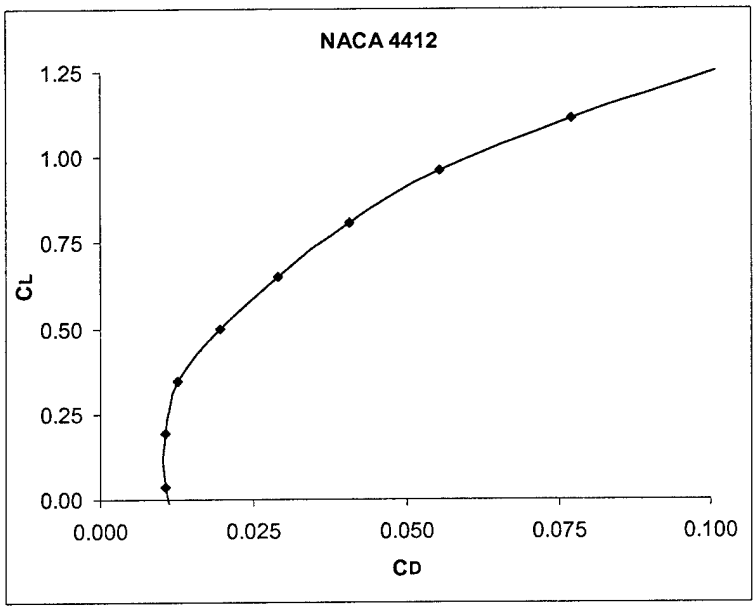


Figure 4.3 – Excel 2D to 3D Data Transformation : NACA 4412

The design coefficient of lift is 1.23 and will produce 30 lbf of lift at 16.353 knots.

4.2 Electrical and Propulsion System

In the conceptual design phase, we determined a single motor with a tractor configuration would be used for our airplane. Once this was determined, we conducted experiments and research to decide the number of batteries that we would require to allow our motor and propeller combination to provide the needed results for flight of our airplane.

4.2.1 Motor Selection

Direct from the AIAA rules, we are limited in the motors that we are allowed to use for this competition. The only motors that are permitted are Graupner or Astro Flight brushed electric motors. With this limitation imposed, we are given the opportunity to use multiple motors that are either gear or motor driven. After some quick research to determine the cost of a given motor, we have determined that it is in our best interest to use one of the gear driven Astro Flight motors that are available from previous AIAA teams because we have both the middle line motor and the most powerful Astro Flight motor. Thus, the motors that are available to us are an Astro 40G and an Astro 90G. After testing both of these motors, we determined that it is essential that our team use the Astro 90G motor based on the necessity for 11 pounds of thrust to be generated in order to get our large aircraft off of the ground. Although this motor will add a little more weight to the airplane, it will provide the necessary power to allow our airplane to take off and remain in flight.

4.2.2 Battery Selection

The battery selection for the aircraft is another crucial area for the success of our team. When focusing on this proponent of the airplane there are several restrictions and considerations that must be taken into account. Two of the important restrictions that the AIAA competition places on battery selections is that the battery pack can weigh no more than 5 lbs and the battery can not be changed or charged between sorties during a flight. This restriction becomes important to us because of the large size of our plane that will require a significant amount of energy throughout the duration of the flight. With this information in mind, our team has decided to order a new battery through SR Batteries that will allow us to create a battery that is as close to 5 lbs as possible without going over. Another consideration that was factored in prior to making our decision to purchase a new battery came from experiments with the available batteries showing that the batteries no longer attain the highest level of charge as a result of memory damage. Moreover, ordering a battery through SR Batteries also allows us to configure the battery in the shape that best meets the needs of our airplane design.

4.2.3 Propeller Selection

There are numerous propellers available on the market, thus proper selection of this key component of this electrical and propulsion system is essential. Given that the only restriction on propellers is that they must be commercially produced, our team conducted experiments on many

different propellers to determine which one would best fit our needs. The experiment consisted of placing all of the different propellers up to a Astro 90G motor with a Sorensen power supply to ensure that all propellers were subjected to the same conditions. The power supply was then set at 35 volts and pounds of thrust were measured using a force meter while the necessary amperage was measured through the power supply. After conducting the experiment for 15 different propellers of various pitches and diameters, the propeller that was determined to best meet the requirements to get our airplane into sustained flight is the 24—10 Menz Props. (Figure 4.4) This propeller produces the required pounds of thrust while requiring the least amount of amperage for a given voltage.

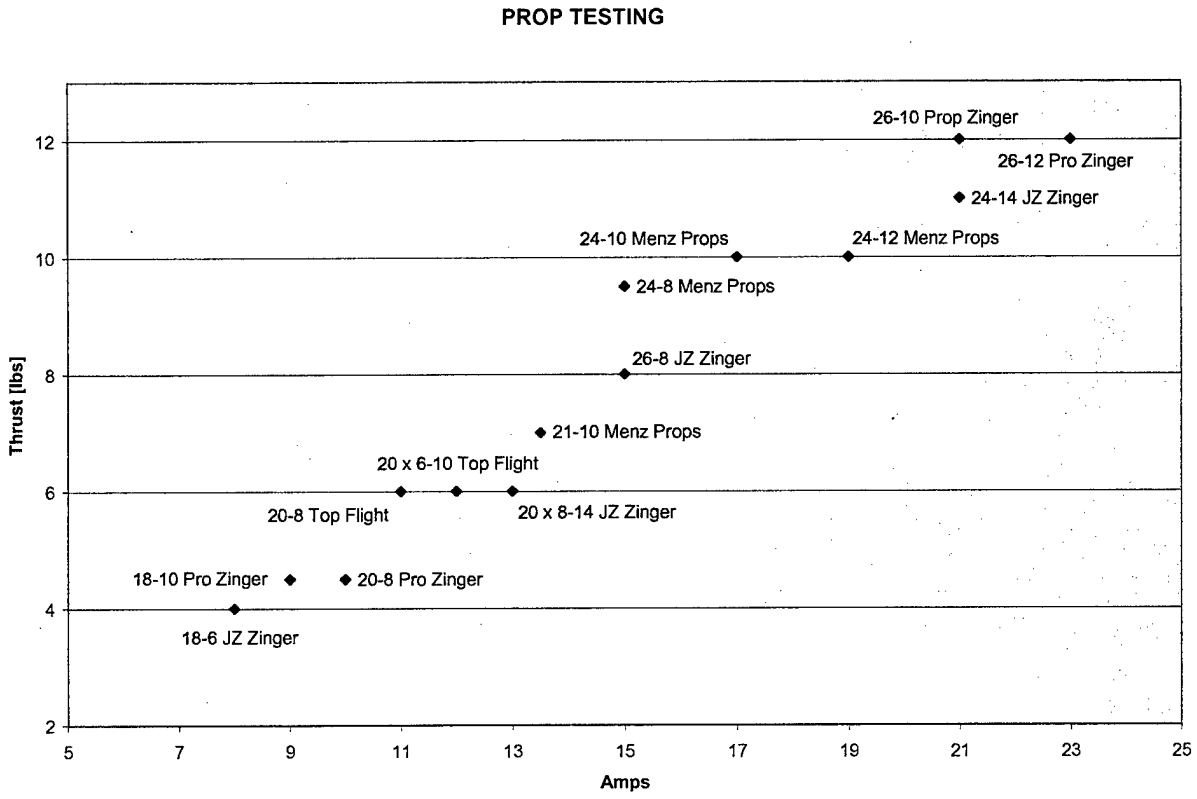


Figure 4.4 – Prop Testing

4.3 Fuselage

The fuselage team decided to use a cylindrical fuselage to meet the needs of the payload container. Also a cylindrical fuselage was the cheapest fuselage to manufacture.

4.3.1 Cylindrical Fuselage

The rationale behind the use of a cylindrical fuselage was strength, lightweight, and an existing design. The rings are aluminum T beams that are formed as a circle. Then holes are drilled into the ring to accommodate hollow aluminum spars than run between the rings. The spars in the main fuselage section will then accommodate the payload, wing, and landing gear. The front ring is modified to mount

the motor, and the rear ring mates with the tail boom. The spars and rings provide a significant amount of strength with little weight. Another benefit is that welding is unnecessary, and the connections can be moved along the length of the spars. The connections consideration was extremely important as the custom design meant that the center of gravity would be hard to predict.

4.3.2 Payload Delivery

Throughout the conceptual stages of the design, it was decided that it would be best to use a cylindrical tank to hold the payload. This is because having an emptying valve on the bottom of the tank would minimize the amount of the payload remaining in the tank after the valve is opened. Also, another concern was the fabrication of the tank and with a cylindrical tank it would be easier to find a tank that we could use instead of having to fabricate one from scratch. The preliminary design for the opening mechanism for the tank was to have a latch that when released would allow the payload to flow from the half inch diameter opening.

4.3.3 Wing Mount Design

The design of the wing mount was also very important. Strength of the wing mount was a very important consideration. The moments generated at the root of the wing are significant unless the main wing spar runs the full length of the span. Since the wing had to be easily disconnected, a brace design is not feasible. In order to accommodate the requirement that the wings disconnect in the middle yet still minimize the moment on the fuselage spars, a top-mounted quick-connect telescoping spar system was designed. In this system, the main wing spars slide into a short spar mounted on the top of the fuselage. Small clips that fit into holes drilled through both spars hold the wings in plane. The short spar absorbs the bending moments from both wings and transfers the lift to the top fuselage spars.

4.4 Landing Gear

The preliminary design for the front landing gear composed of the RF800 landing gear kit bought from Hometown Hobbies. Since this landing gear has never been tested on our planes, we decided to use this specific type. More importantly, the specifications on the RF800 given by the model airplane corporation claim it to be the largest and strongest front gear on the market. We feel confident that it will be able to withstand the maximum force of landing by our aircraft.

Once the fuselage is built, we will be able to construct the front landing gear, tailoring it to the exact fit that we need on the plane. During our preliminary design, we determined that the plane needed to rise off the ground by about 12 inches. This is the spacing we need to make sure the propeller clears ground level.

The preliminary design of the rear landing gear consisted of buying a pre-built model. We found that the largest landing gear available on the market was not sufficient for our needs. We needed more height on the rear landing gear for clearance of the propeller. Therefore, we moved into buying sheets of 7075 Aluminum. These are rated to be extremely strong and light weight. In order to reach a maximum

distance of 12 inches in height, we needed to angle the aluminum sheets at 60 degrees. Another important aspect of the rear landing gear is that we kept it as one solid unit. By doing this, we were able to increase the strength and durability of the landing gear.

The final component of the landing gear were the wheels. We decided to use wheels off of an umbrella stroller for our design. We did this because umbrella stroller wheels are rated at 175 lbs, well within our range of forces produced from landing.

4.5 Empennage

The empennage team also used a numerical concept evaluation matrix to define the preliminary design. The matrix determined that a "V-tail" design was necessary. This allows for several advantages, to include maneuverability, control, weight, and ground clearance. The empennage will be mounted using four arrow shafts connected to the aft ring of the fuselage. They will taper down in a pyramid configuration to a tube that will contain the wings. Two servos will be used to control each wing, and will be located in the rear tube.

4.5.1 Stabilizer and Control Surface Design

The airfoil selected for the ruddervators is the NACA 0012. This symmetric airfoil is relatively thin, which reduces drag, and decreases the overall weight. The control surfaces were designed to be 30 percent of the chord. The ruddervators were chosen to be 45 degrees from the ground, 90 degrees apart so that vertical and horizontal stability are proportional. The volume ratios for a ruddervator is equal to the wing planform area multiplied by the tail moment arm divided by the quantity of the wing surface area multiplied by the wing chord. The pre-selected (based on convention) horizontal and vertical tail volume ratios are 0.7 and 0.04, respectively. From our research, these values are adequate for providing stability to the aircraft.

4.6 Center of Gravity Calculations

The center of gravity location is extremely important for any aircraft. In an aircraft, if the center of gravity falls behind the center of pressure of the aircraft, the plane will become statically unstable. When a plane is statically unstable, any perturbation from the equilibrium position will drive the system further away from equilibrium. For a remote controlled airplane, stability is of particular importance because the aircraft is controlled from the ground, and there is a significant time delay between a perturbation and the corrective response directed by the controller. The static margin for an aircraft is defined as the distance from any point on the aircraft to the center of gravity divided by the chord subtracted from the aerodynamic neutral point, divided by the chord. The minimum static margin should be at least five percent of the chord length for an aircraft. For our aircraft, this yields a static margin of 0.9 inches. There are two possible weight conditions for our aircraft, loaded and unloaded. However, we designed the center of gravity of the payload to be centered at the aircraft's center of gravity. By doing this we

eliminate having to move the wing in any direction. Also the water delivery system is designed to drain at the aircraft's center of gravity to reduce any changes while in flight. Because the tail moves the neutral point behind the quarter chord of the wing, the placement of the wing should be between 10 and 11 inches from the front ring of the fuselage measured to the quarter chord of the wing. This should yield adequate stability when loaded while keeping the aircraft somewhat maneuverable when unloaded.

5.0 Introduction

In the Detailed Design section, we make materials selections and finalize dimensions. Additionally, there are many minor components that were not designed in the preliminary design phase. These sub - systems, which are necessary to the proper functioning of the aircraft and mission success, are designed as their need arises. The detailed design and fabrication phases occurred at the same time, with component designs being finalized just prior to their manufacture. This is because the techniques involved in fabrication affect the design of each component, and the actual geometry of other components affects the design of other components that still need to be designed. Because fabrication closely followed detailed design, the CAD drawings of small components is not available because their design exists only as sketches. However, major components have been drawn in CAD and are available in the drawing package at the end of the Detailed Design Section. The fine points of the detailed design can be viewed in Figure 5.1.

Detailed Design	
Fuselage	**Ring and Rib construction**
	7075 Aluminum Ribs
	Lightweight Carbon Fiber Skin
Payload Container	
	Two three litter bottles connected
	Swinging valve release system
	Screw on cap
Wing	
	Foam Core
	Two Carbon Fiber Spars
	Ailerons at 30% of the Chord
Empennage	
	Telescopic Tail Boom
	V-tail design
	Both sections act as control surfaces
Landing Gear	
	6061 Dual Main Supports
	Commercially Available Front Landing Gear
Power	
	Astro 90G Motor
	35 Volt Battery Pack
	24 x 10 Wooden Propeller
	Commercially Available Flight Pack and Controller

Figure 5.1 – Detailed Design Details

5.1 Wing

The wing design remained essentially the same in the detailed design phase. The changes and additions are detailed in the following sections.

5.1.1 Ailerons

The ailerons were made by using the last 30 percent of the airfoil chord. A hole was drilled in the front of the aileron which slid on an aluminum bar that ran the length of the span. The preliminary design originally called for two 7 series aluminum spars at 20 and 60 percent of the chord. The ailerons are also covered with a carbon fiber matrix and are operated by servos.

5.2 Fuselage

The detailed design of the fuselage did not change from the preliminary design due to time constraints. We felt that the preliminary design was very solid and would meet all of the customer requirements.

5.2.1 Motor Mount

The motor mount was designed in the detailed design phase. The motor mount was made to be place five inches in front of the nose ring and hold the motor so that the shaft was in the middle of the aircraft.

5.2.2 Rings

We decided to drill 24 holes into the aluminum rings. This allowed nearly any spar configuration to be used. The total number of spars to be used in the main section was set at eight, and their locations were fixed. A structural support was added to provide extra strength. Lightweight aircraft aluminum was used for the rings and the main section spars, but ultra-light 7 series aluminum was used for the rest of the spars. The 7 series aluminum has the advantage of being lighter, but is more brittle than aircraft aluminum.

5.2.3 Payload Delivery System

As we refined our design in the detailed portion of the design it was decided to use two three-liter soda bottles to fabricate the tank. By using two of the blunt ends of the bottles and connecting them and sealing them with a total length of approximately fifteen inches it would easily satisfy the four-liter payload requirement. For the emptying mechanism, we would use a bottle cap on the bottom of the tank with a half inch hole in it. We would add a second lid over the first cap and with a hinge and o-ring seal off the tank by holding it with a latch that could be unlatched to allow the payload to empty. The latch will be opened using servos.

5.3 Landing Gear

A method for mounting the front landing gear had to be designed during this phase. A crossbar was designed to hold the landing gear shaft and two servos while minimizing weight and ensuring

structural strength and staying unobtrusive inside the fuselage. The wheel used for the front landing gear is a soft, lightweight rubber tire that was available in the shop.

The rear landing gear will be one solid unit with a flat plate of aluminum at the top. We will have connectors specially made to connect the landing gear to the fuselage. Connecting the wheels only requires a simple washer, nut, and axle.

5.4 Empennage

Several changes were made to the empennage during the detailed design phase. Specifically, the mounting arrow shafts were substituted with a long aluminum pipe. The ruddervators will be tapered and mounted by its spars to the fuselage. The spars will be held in place with a cotter-pin, and the empennage will be mounted onto the fuselage using two flanges acting as sleeves for the shaft. The shaft will be adjustable to one foot forward or aft. This will be useful for optimizing our control surface sensitivity. It will be adjusted by securing two pins in holes drilled laterally through the mounting pipe. Again, two servos for the control surfaces and the antenna for the RC receiver will be used, however, they will be placed in the fuselage, so that the CG will not be impacted. The wings and piping will be covered in monokote for drag reduction and aesthetic value.

5.5 Electronics and Propulsion

With the batteries, motor, and propeller selected, the electronics and control system are set in place. The rest of the work on the electronics and propulsion system involves testing and installation.

5.6 Specifications

As previously mentioned, the specifications list provides the engineering targets and their respective weights as goals for the design. In order to rate our design, it is necessary to compare the engineering targets to the design data. When the plane is completed and the flights actually occur, the data can be taken for actual flight conditions, and the accuracy of our approach in calculating the design performance data can be evaluated. Many of the specifications were met, exceeded, or nearly met. This reflects the amount of research that went into the project during the conceptual design phase. However, some of the targets were not very good, and some of the requirements were poor choices.

GEOMETRY	
Length (feet)	4.5
Span (feet)	11.75
Height (feet)	0.75
Wing Area (Square feet)	17.625
Aspect Ratio	6.286
Control Volumes	Ailerons & V-tail
PERFORMANCE	
CLmax	1.23
L/Dmax	24.46
Maximum Rate of Climb (knots)	2.751
Stall Speed (knots)	16.353
Maximum Speed (knots)	53.44
Take-off Length Empty (feet)	17.94
Take-off Length Loaded (feet)	43.41
WEIGHT STATEMENT	
Airframe (lbf)	5.8
Propulsion System (lbf)	6.3
Control System (lbf)	1.8
Payload System (lbf)	0.5
Payload Empty Weight (lbf)	14.4
Gross Weight (lbf)	22.4
SYSTEMS	
Radio Used	Futaba FP-TP-FM
Servos Used	Futaba series (FR-S148, S-3003, FPR-1480F)
Battery Configuration Used	7 x 5 SR Batteries
Motor Used	Astro 90G
Propeller (nominal)	24 x 10 Menz Propeller

Figure 5.2 – Airplane Data

A	$300		WBSw	181.25		WBSf	22.5	WBSe	15
B	$1,500		Wing Span (ft)	11.75		Length (ft)	2		
C	$20		Chord (ft)	1.5		Width (ft)	0.75	WBSfs	25
MEW	14.4 (lbf)		Wings	1		Height (ft)	0.75		
REP	5 (lbf)		Area (ft^2)	17.625		Volume (ft^3)	1.125		
MFHR	253.75		$10hr/ft^2$	176.25					
			Ailerons	1					
RAC	$16.895		CFM	5					

Figure 5.3 – Rated Aircraft Cost

The total Rated Aircraft Cost for the Boombatz Grandioso Flier is $16,895. The coefficients A, B, and C are manufacturers empty weight multiplier ($300), rated engine power multiplier ($1500), and manufacturing cost multiplier ($30) respectively. The manufactured empty weight (MEW) was the actual weight of the airframe with out the payload at 14.4 lbf. The rated engine power is defined by the following equation:

$$(1 + 0.25*(1 \text{ engine} - 1))*5 \text{ (weight of battery lbf)} = 5 \text{ lbf.}$$

The Manufacturing man hours (MFHR) is defined as the sum of the prescribed assembly hours by the work breakdown structure (WBS). The WBS for the wing was 10 hrs per square foot of the wing surface area. With one wing having a wingspan of 11.75 ft and a chord of 1.5 ft the total area came to 17.625 ft^2. The total WBS for the wing was 176.25 hrs plus 5 hrs for the manufacturing of ailerons equaling 181.25 hrs. The WBS for the fuselage was 20 hrs per cubic foot of the fuselage. With a length of 2 ft, a width of 0.75 feet, and a height of 0.75 ft the volume of the fuselage was 1.125 ft^3, bringing the WBS for the fuselage to 22.5 hrs. As described by the rules the WBS for a "V" tail is 15 hrs. The WBS for the flight systems is 5 hrs times 5 servos totaling 25 hrs. Therefore the MFHR equals 253.75 hrs. The total Rated Aircraft Cost is defined by the following equation.

$$\text{RAC, \$ (Thousands)} = \frac{(A*MEW + B*REP + C*MFHR)}{1000}$$

$$\text{RAC} = \frac{[\$300*14.4 \text{ (lbf)} + \$1500*5 \text{ (lbf)} + \$20*253.75 \text{ (hrs)}]}{1000} = \$16,895$$

5.7 Drawing Package

Drawings of the detailed design are contained in figures 5.4 to 5.6. We did the drawings with ProDesktop, a computer aided design software program. The drawings are to scale; however, most fasteners and joiners have been left out for ease of viewing and component assembly.

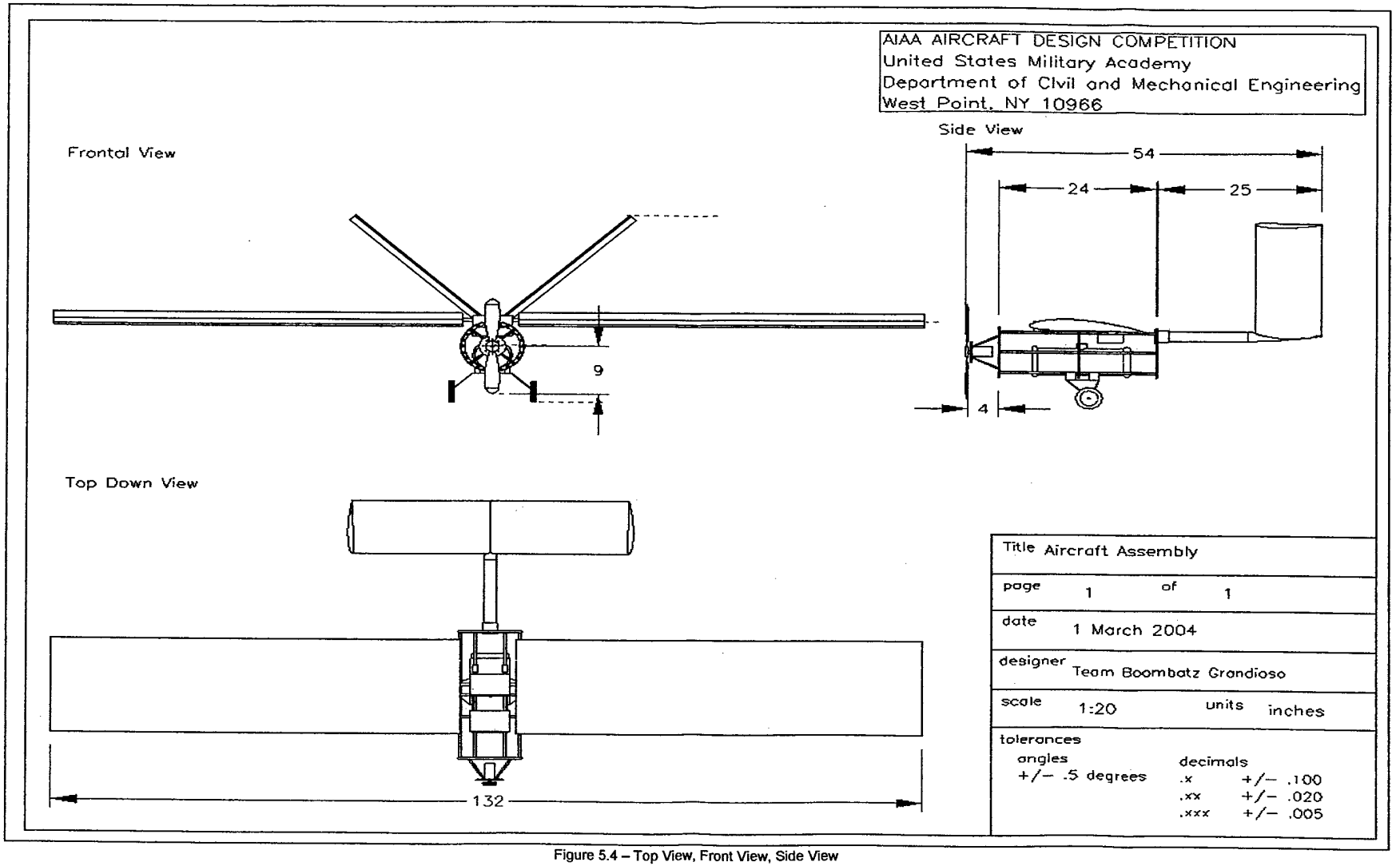


Figure 5.4 – Top View, Front View, Side View

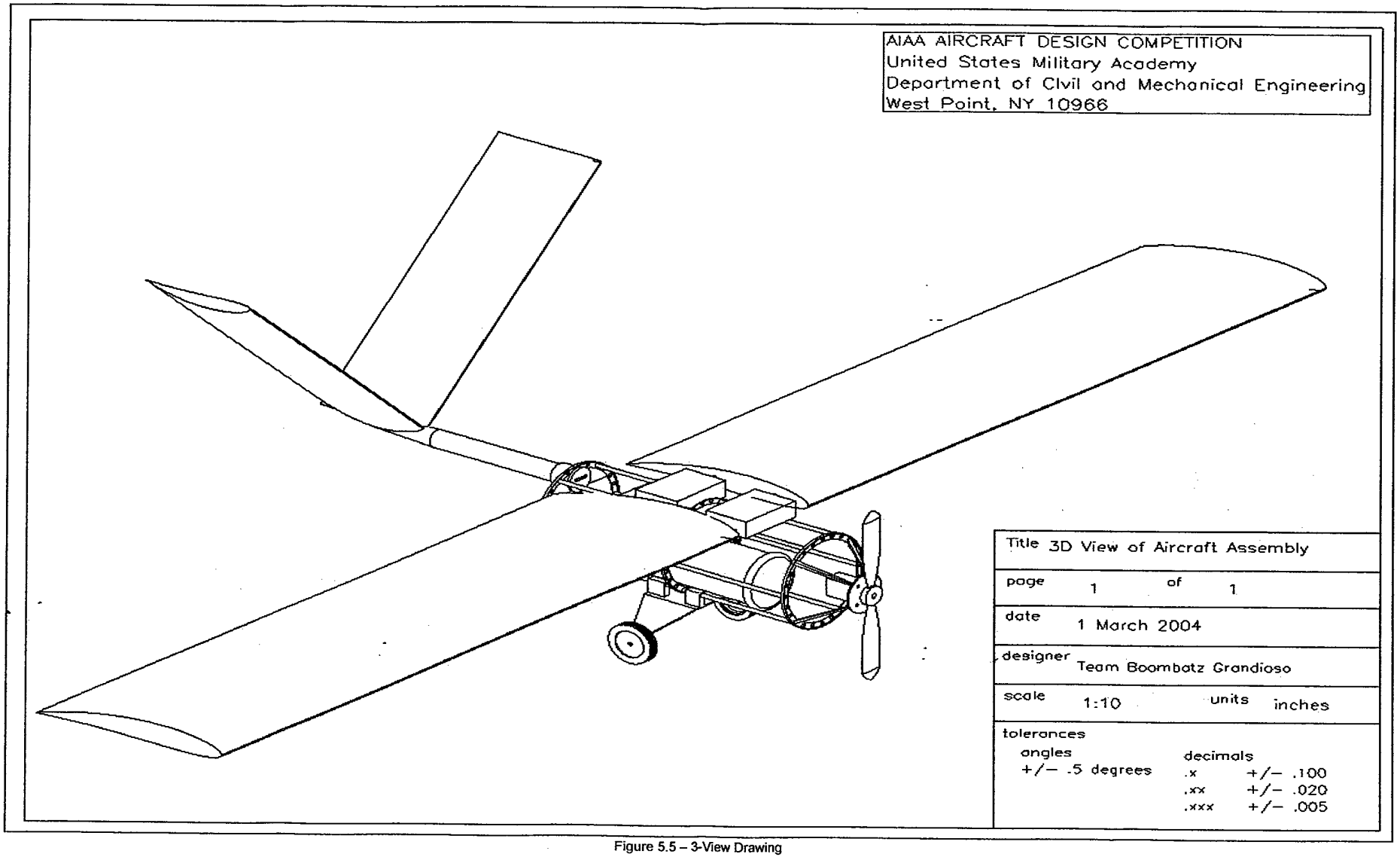


Figure 5.5 – 3-View Drawing

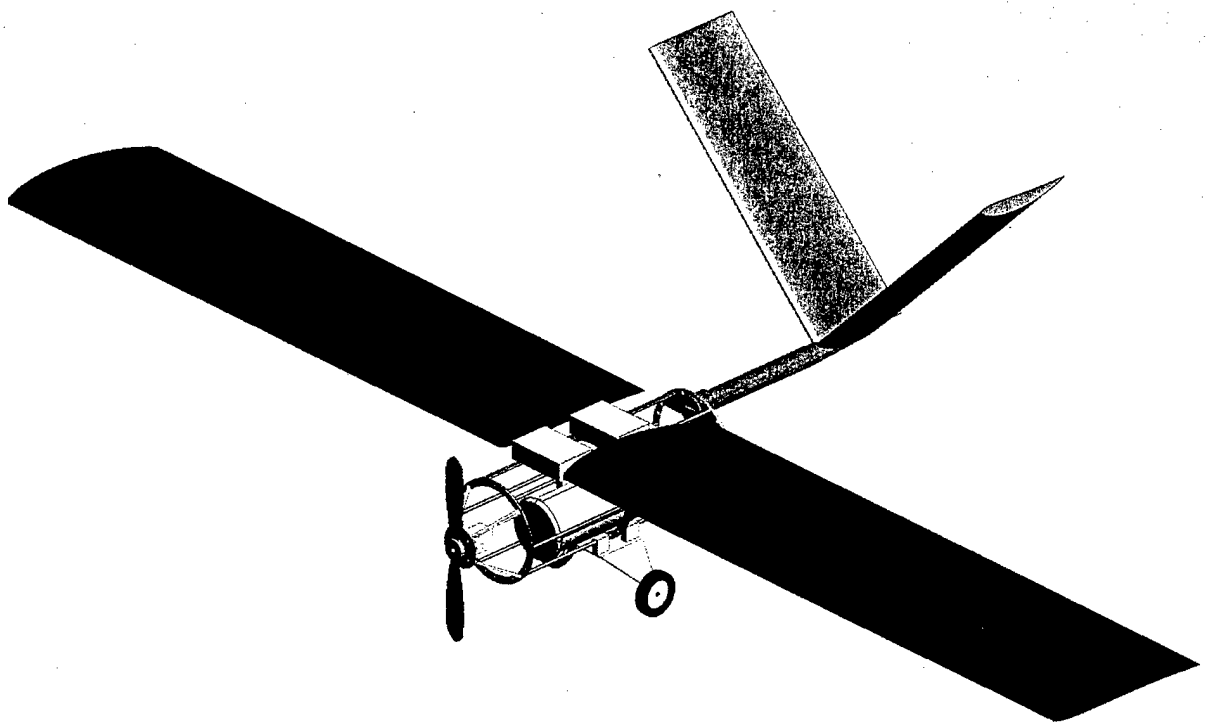


Figure 5.6 – 3-D View

6.0 Introduction

Our approach meant that the teams had to learn how to use shop equipment before fabricating a component. It also meant that the teams had to dedicate a lot of time to working with the department technicians on the detailed design and fabrication. When a detailed sketch was made from the concepts discussed between the team members and the technicians, the technicians would then instruct the team members on how to manufacture the component. The cadets would then manufacture the component. While this process took a long time, it meant that the quality of the product was much better than if we had tried to do everything ourselves to save time.

6.1 Wing

The manufacturing of the wing began with designing the necessary tools needed to fabricate the wing. A guide system was designed to ensure that perfect holes could be drilled into the foam core for the carbon spars. The wing was divided into 22 inch sections and then pieced together to form the eleven foot wingspan.

After the carbon spars where in place and the sections where connected a carbon fiber matrix was heated and sealed securely onto the wing using a vacuum system.

6.2 Fuselage

The fuselage is definitely the most complicated portion of the design, and the most work was put into its manufacture. Each component that was made was attached to the spars and clamped in place. In order to epoxy the aluminum, the surfaces have to be cleaned and etched. This cleans the bonding area of foreign material and cleans the aluminum oxide layer off of the top of the aluminum. This layer reforms quickly, so the process cannot be allowed to take too long.

The front ring was machined from a disc of aluminum. Holes were carefully drilled into the disk to accommodate the motor shaft and the screw mounts. Additionally, holes were machined into the disk at the precise angle that the spars needed to make to mount the front disk five inches forward of the front ring. These front spars were also cut to keep them flush with the hoop surface of the front ring.

The rear ring was also machined from a flat disk, but was much thinner than the front disk. A matching disk was made to mount the tail onto the aircraft. Once the tail is aligned, the two disks are screwed together.

6.3 Landing Gear

Assembly of the front landing gear did not take much time since it is a kit supplied with instructions. The front landing gear came with a solid aluminum extension rod with shock absorbers attached at the bottom to absorb the force of landing. Nuts and bolts are provided to attach the wheels.

7.0 Introduction

The team has designed a testing plan to be completed upon completion of the aircraft. At this time no in flight tests have been conducted to determine if any changes need to be made to the aircraft. The propulsions team has conducted tests on the battery, motor, and propeller to optimize the propulsion unit. During the preliminary design phase we calculated that 11 lbf of thrust was needed to fly the aircraft. The propulsions team used a static testing stand to test our system and prove that we could generate 11 lbf of thrust.

Other tests included structural tests of the carbon fiber spars that reinforce the wings. The spars were able to hold 50 pounds without failing. This was a very important test because we are relying on the carbon fiber spars to support the weight of the aircraft during the inverted wing tip test.

Further flight tests are scheduled to be completed in the spring when we get advantages weather. From the flight tests we will be able to correct any mistakes in our design and manufacturing.

7.1 Testing

Figure 7.1 relates the goals of our engineering requirements to what we actually achieved. Over all we did very well. The unloaded weight and thrust to weight ratio were the two biggest disappointments. However we need a five pound battery to achieve the majority of our goals. Therefore, our original goal of 14 lbf unloaded weight was a small underestimate. Furthermore we feel good about our results and we are confident that the Boombatz Grandioso Flier will achieve our mission goal.

Engineering Requirement	Goal	Weight	Achieved
Wing Planform Area	> 15 ft^2	7.92%	16.5 ft^2
Velocity (high)	> 30 ft/s	7.52%	
Unloaded Weight	< 14 lbf	7.12%	14.4 lbf
Small W/S	< 1.5 lbf/ft^2	6.32%	0.737 lbf/ft^2
Acceleration (high)	> 15 ft/s^2	6.32%	
Large T/W	> 1	5.52%	0.5
Large AR	> 7	5.52%	6.286
Loaded Takeoff Distance	< 115 ft	5.22%	43.41
Small K	< 0.046	5.12%	0.039
Battery Weight	< 3.8 lbf	5.12%	5 lbf
Large e	> 0.98	4.71%	
Large L/D	> 8	4.41%	24.46
Minimize Connections	< 12	3.81%	8
Minimize Parts	< 8	4.31%	5
Minimize Electrical Connections	< 5	4.31%	7
Cruise Thrust	> 9 lbf	3.41%	10 lbf
Strength	> 50 ksi	3.41%	
Related Airplane Cost	< $3000	2.81%	$16,895
Assembly Time	< 60 s	2.91%	
Static Landing gear load	< 21 lbf	2.31%	22.4 lbf
Small CDo	< 0.04	1.91%	0.042

Figure 7.1 – Test Results

The rear landing gear was manufactured out of solid 7075 Aluminum sheets. The exact dimensions and area were configured so that a clearance of at least 12 inches was given for the plane. Construction of the rear landing gear was difficult since we were working with 7075 Aluminum, a very strong metal in comparison to its counterparts. We had to manually bend the metal to create the desired angles on the sides. Careful consideration was taken to make sure that the gear is able to absorb the forces due to impact and that enough clearance was given for the propeller.

6.4 Empennage

The goal of constructing the empennage was to minimize fabrication. By ordering pre-existing parts, we were able to decrease construction time, however our options were more limited. The mounting of the shaft was the most difficult, as we were constrained to the current dimensions of the flanges/sleeves. We considered cutting the flanges and welding them onto the rear fuselage ring, which would be a full integration, however we did not need to because the pipe didn't interfere with the water container. In order to make the tail length adjustable, we drilled holes in the pipe and flanges every 3 inches, up to 2 feet. Problems consisted of interfering with the water container and allowing the rotation of the pipe mounting for trimming.

The mounting of the wings were done by staggering the spars at 90 degrees within the rear of the pipe. The arrow shaft spars were drilled such that a cotter pin could be inserted to hold the spars in place within the pipe. The most difficult part of this process was accuracy. The holes to mount the spars had to be exactly 45 degrees from the holes that mounted the pipe to the fuselage ring. Also, the angles for the spar holes had to be precise in order for the control surfaces to require the least amount of trimming. The servos and antenna were attached last inside the fuselage. To minimize drag, a cone was placed in the rear of the tail pipe, and monokote was applied to the wings.

6.5 Payload

In order to actually make the tank, the plastic bottles were easy to work with and seal. In addition to the detailed design, we added a second cap on the top of the tank so that it would be easy to fill the tank while inside the fuselage of the plane. By basically "building" around the payload, it was easy to incorporate the motor and battery within the fuselage to have enough room for all the necessary internal components of the aircraft. In order to activate the latch, a servo must be used to open the lid and release the payload. Also, the simplicity of the tank allowed us to concentrate funds on the wing, fuselage, empennage motor and batteries.

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AIAA Foundation

Cessna/ONR Student Design, Build, and Fly

2003-2004



Water Buffaloes

University of Colorado at Boulder

March 2004

Table of Contents

1 Executive Summary **1**
1.1 Design Summary 1
1.1.1 Design Alternatives 1
2 Management Summary **2**
2.1 Team Organization 2
2.1.1 Management Team 3
2.1.2 Aerodynamics Team 3
2.1.3 Propulsion Team 3
2.1.4 Structures Team 4
2.1.5 Payload Team 4
2.1.6 Documentation Team 4
2.2 Project Scheduling and Milestone Chart 4
3 Conceptual Design **5**
3.1 Competition Requirements & Scoring 5
3.1.1 Design and Performance Requirements 5
3.1.2 Competition Score 6
3.1.3 Rated Aircraft Cost 8
3.2 Design Parameters 8
3.2.1 Aircraft Configuration 9
3.2.2 Propulsion System 10
3.2.3 Payload Design 10
3.2.4 Structural Design 11
3.2.5 Landing Gear Configuration 11
3.3 Design Trade Studies 11
3.3.1 Tail Configuration 11
3.3.2 Structural Configuration 11
3.3.3 Propulsion 12
4 Preliminary Design **12**
4.1 Mission Models – Fire Bomber and Ferry 12
4.2 Estimates of Aircraft Parameters 13
4.2.1 Estimates of Lift 13
4.2.2 Estimates of Drag 14
4.2.3 Estimates of Stability 16
4.3 Design Optimization and Sizing Trades 17
4.3.1 Wing Design 17
4.3.2 Power & Propulsion 25
4.3.3 Fuselage Design 26
4.3.4 Water Tank 26



5 Detailed Design **29**
5.1 Components and System Architecture 29
5.2 Materials 29
5.3 Propulsion 29
5.4 Wing Design 29
5.5 Tail Design 29
5.6 Aircraft Geometry 30
5.7 Drawing Package 30
5.8 Performance 36
5.8.1 Lift and Drag 36
5.8.2 Stability 39
5.8.3 Velocity Profile 40
5.8.4 Rate of Climb 41
5.8.5 Take-off Field Length 42
5.9 Weight Statement 42
5.10 Rated Aircraft Cost 44
6 Manufacturing **45**
6.1 Figures of Merit 46
6.1.1 Availability of Materials 46
6.1.2 Cost of Materials: 47
6.1.3 Required Skill Level of Manufacturing & Ease of Construction 47
6.1.4 Turn Around Time 47
6.1.5 Strength and Reliability 47
6.1.6 Estimated Final Integrated Weight 47
6.2 Aerodynamic Surfaces 47
6.3 Tank Manufacturing 49
6.4 Fuselage & Master Molds 49
7 Testing **50**
7.1 Objectives and Schedule 50
7.2 Results 51
7.2.1 Tank Water Drop Test #1 51
7.2.2 Refilling Test 53
7.2.3 Wing Strength-to-Weight Test 53
7.2.4 Static Thrust Test 55
7.2.5 Battery Test 56
7.2.6 Thermal Tests 57
7.2.7 Flight Test 58
8 References **59**
9 Acknowledgements **59**

1 Executive Summary

This report outlines the design, manufacturing and testing for a remote control airplane to compete in the American Institute of Aeronautics and Astronautics (AIAA) 2003/2004 Design/Build/Fly (DBF) Competition. This year's competition is primarily focused on the idea of a water tanker. There are two missions that the aircraft will compete in, the Fire Bomber and the Ferry missions. Since this year's competition is heavily focused on the Fire Bomber mission, the aircraft has been optimized to perform as a water tanker. The design and fabrication of the aircraft has been significantly advanced compared to the entry submitted last year by the University of Colorado.

1.1 Design Summary

Team Water Buffaloes followed a standard three-step design process consisting of a Conceptual, Preliminary, and Detail Design Phase. Throughout the entire design process an Excel Workbook (DBFDesign), initially developed by the 2002/2003 team was used to evaluate design decisions on the aircraft. The workbook is an extensive collection of spreadsheets that provide an integrated model of the aerodynamics, structures, propulsion, mission performance, and competition score.

In the Conceptual Design Phase, it was decided to base the design and configuration off of last year's successful entry. To evaluate and suggest modifications to last years design, the team was split into subsystems consisting of Aerodynamics, Propulsion, Payload and Structures. The proposed modifications from each team were rated based on several factors including: the affect on individual mission scores, the overall competition score, the construction difficulty, and the ability to satisfy competition requirements.

Sizing and analysis of the aircraft began in the Preliminary Design Phase. Since the total time in the Fire Bomber mission is significantly more important to the final competition score than the Ferry mission by a factor of 17.6, a significant amount of the preliminary design was devoted to the design of the water drop and filling systems. Experiments were performed to determine a shape that minimized the drop time. Various other experiments in structures, manufacturing, and propulsion were used to further design decisions.

In the detail design phase, the DBFDesign workbook was extended to include an algorithm to optimize the design of the aircraft based on several input parameters. The winning aircraft was chosen as the configuration with the highest overall competition score that had a take-off distance less than 150 ft. Using this optimization algorithm, several thousand different configurations with varying wing span, wing chord, number of batteries, water volume, and propeller size were analyzed.

1.1.1 Design Alternatives

During the conceptual design phase, several modifications to the previous year's design were proposed and evaluated. The design alternatives were limited to component level modifications since the aircraft's basic configuration followed from last year's design. This was in contrast to a higher-level view, for example, considering a canard versus a standard configuration aircraft, or a puller versus a pusher. It was also chosen by the team to limit other possible modifications such as the number of motors and landing gear configuration. These modifications would have increased the design complexity and pilot workload.



In the Propulsion subsystem, the primary modifications were the motor size and gearing ratio. Other major components such as the number of batteries and propeller sizing were determined by satisfying the take off and mission constraints.

The Aerodynamics team analyzed the wing and stabilizer designs. Some of the different modifications considered for the wing included tapering, twisting, and lofting. The proposed wings were evaluated and compared based on their efficiency, lifting performance, and manufacturing complexity. The same metrics were used in comparing different horizontal and vertical stabilizer configurations. The three configurations analyzed were a T-tail, V-tail, and a standard configuration.

Another system analyzed was the design of the fuselage. Variations in the fuselage shape were dictated by the desire to maximize the Fire Bomber mission score. Design shapes of the fuselage and tank were evaluated in an iterative and parallel fashion to meet the design schedule, minimize the water drop time, and decrease manufacturing complexity.

Alternatives in the structural design and manufacturing method were also considered. Due to the in-flight failure of the balsa wing in last year's competition, a composite structure for the wing and fuselage was chosen to increase strength. Carbon fiber and fiberglass were the composite materials evaluated.

2 Management Summary

2.1 Team Organization

Team Water Buffaloes used a simple two-level pyramid structure for organizing into sub-teams, as shown in Figure 1. The top level consisted of the Management team including the project advisor and the systems engineer. The second level was comprised of a Documentation team and four technical teams: Structures, Propulsion, Aerodynamics, and Payload. A team lead was assigned to each of these groups and was responsible for the productivity of their team and communication with the systems engineer. Participating team members choose a primary and secondary team for the project.

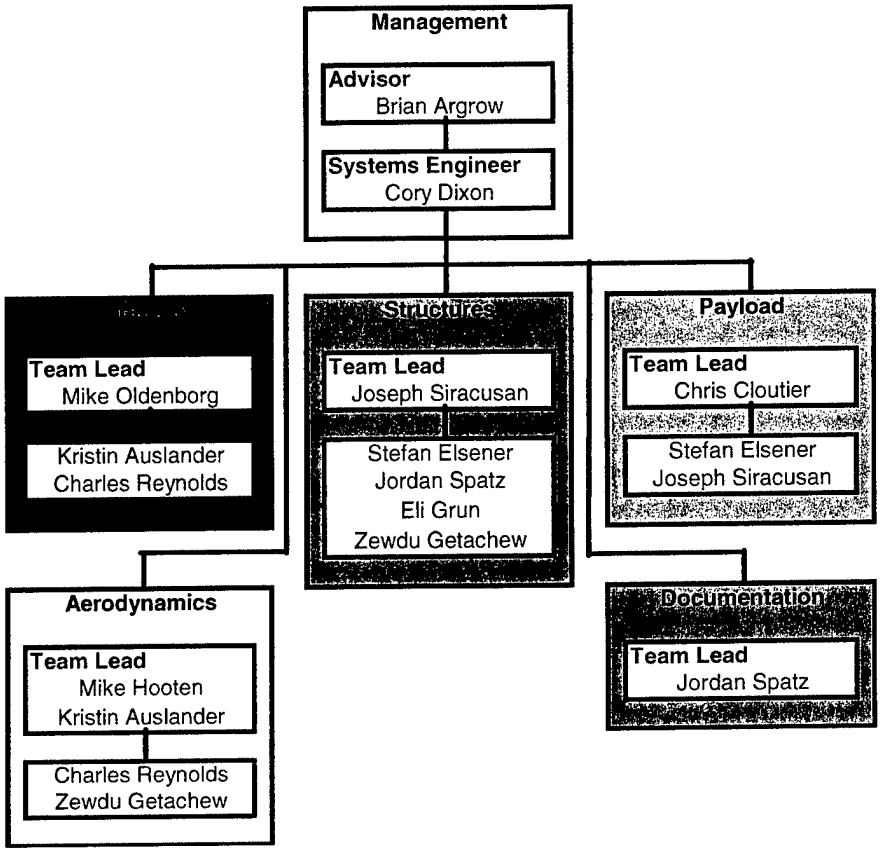


Figure 1: Team Organization Chart

2.1.1 Management Team

The task of the faculty advisor and systems engineer was to organize the team and to ensure successful progression throughout the aircraft design and manufacturing process. The primary administrative responsibilities of the systems engineer were to schedule and coordinate the various technical groups. The technical responsibilities of the systems engineer were to make final design and manufacturing decisions.

2.1.2 Aerodynamics Team

The aerodynamics team was responsible for the exterior design of the aircraft. This included shaping the fuselage, selecting airfoils, sizing the wing planform and tail assembly, and determining the aerodynamic stability and control derivatives. The decisions made by the aerodynamics group had immediate impact upon the propulsion system and structural design. Because of this, it was imperative that the aerodynamics team leads maintained communications with the other technical teams to assure a properly integrated design.

2.1.3 Propulsion Team

It was the task of the propulsion team to choose the motor, propeller, number of batteries, and configuration needed to meet the various mission requirements. During the manufacturing phase, the team was responsible for assembly and integration of the electronics into the aircraft. The propulsion



team was also tasked with performing experiments to verify the propulsion system model used in the aircraft design.

2.1.4 Structures Team

The structures team was responsible for the structural design and analysis of the aircraft and acted as the primary manufacturing team. To further the manufacturing techniques from last year, the team was tasked to examine various composite materials and fabrication processes. The structural design and materials used for the each component of the aircraft were selected based upon their analysis.

2.1.5 Payload Team

The payload team was tasked with designing the payload system on the aircraft and the filling apparatus used on the ground. Because the drop time and filling time heavily weighed on the final competition score, the payload team performed several tests to design an efficient deployment system.

2.1.6 Documentation Team

The primary task of the documentation team was to record and document the design, build, and fly processes. The team was also tasked with designing and maintaining the team website as well as organizing and editing the final design report.

2.2 Project Scheduling and Milestone Chart

At the beginning of the engineering process, the team planned out a schedule of the primary design and manufacturing milestones that had to be met to ensure the successful completion of the aircraft design. The design process was divided into conceptual, preliminary, and detail design phases. The manufacturing process was divided into prototyping, testing, and final construction phases. At the beginning of each week, the team had an organizational meeting to report the past week's progress and to present goals for the next week. These meetings provided a forum for inter-team discussions of problems and overall project status.

Figure 2 is a Gantt chart of the design and manufacturing milestones needed to meet the competition deadline. The chart shows the projected and actual timeline for each task. To allow for construction and testing of the aircraft before the report deadline, the design process needed to progress at a rapid pace. However, as shown in the chart, all three phases of the design process took longer than projected and delayed the manufacturing schedule.

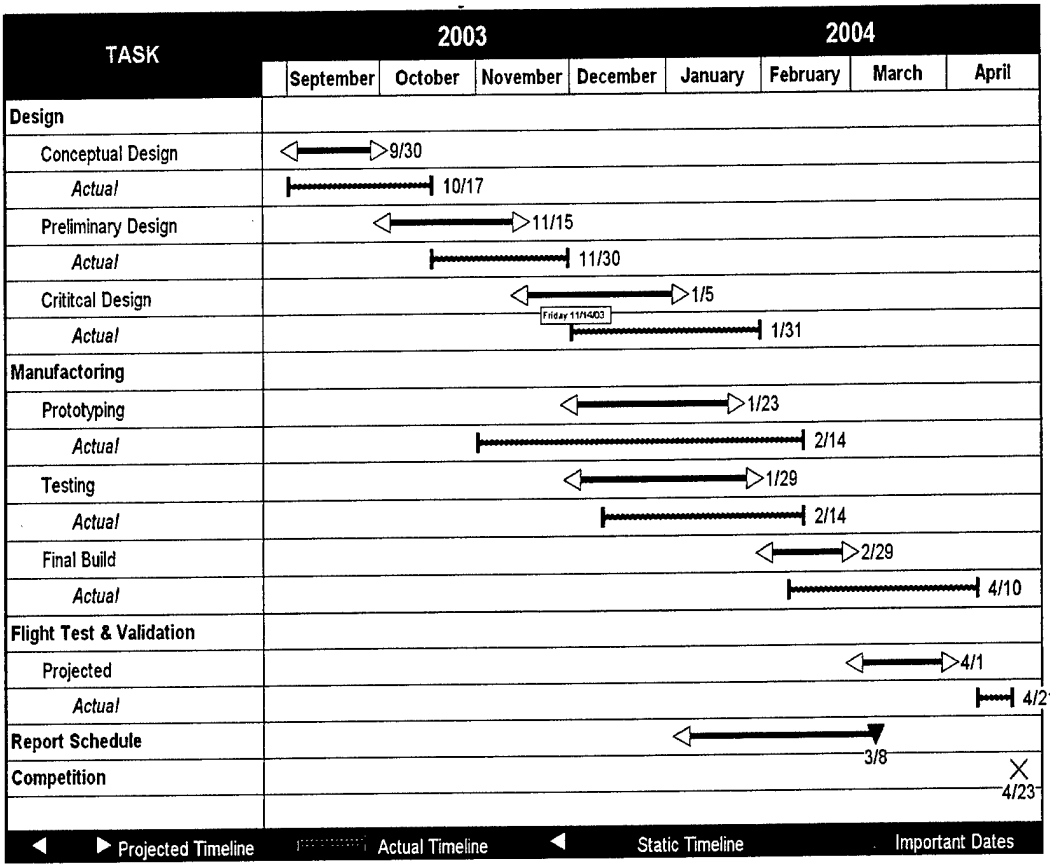


Figure 2: Project Gantt Chart

3 Conceptual Design

The goal for the conceptual design phase was to formulate a set of aircraft design parameters that would enable the team to choose an appropriate configuration to achieve the highest possible competition score. To minimize the number of design configurations, it was decided to build upon the design of last year's successful entry. This allowed for a more focused approach to the design process by narrowing the range of design alternatives.

3.1 Competition Requirements & Scoring

The basic idea of the competition was to design, build, and fly an electrically powered, remotely controlled, fixed wing aerial vehicle capable of optimally performing all mission specifications. Therefore, a fundamental understanding of the mission requirements and competition scoring was needed to obtain the highest possible competition score.

3.1.1 Design and Performance Requirements

There were a few fundamental competition requirements placed on the size and performance of the aircraft. The limit placed on the overall size of the aircraft was that it must fit in a 2-ft wide, 1-ft high, and 4-ft long box. For both missions, there was a performance limitation of a wheels-up takeoff distance less than 150 ft. The ability to achieve the takeoff requirement was affected by the wing loading and the maximum power available from the propulsion system. The propulsion system was limited to the use of a



brushed electric motor from Astro Flight or Graupner, Ni-Cd batteries, and commercially available propellers. The amount of power was also limited by the specification of a 40 A maximum current draw.

3.1.2 Competition Score

The final competition score was determined by the three factors related in Eq. 1.

$$SCORE = \frac{Written\ Score * Total\ Flight\ Score}{Rated\ Aircraft\ Cost} \quad (1)$$

Thus, to perform well a team needed to maximize the airplane's total flight score, minimize the rated aircraft cost (RAC), and perform well on the design report. The report score provided a measure of a team's ability to properly document the design and manufacturing process they followed. The RAC attempted to direct the design of the aircraft in certain directions by weighting different design parameters of the aircraft. It also provided a measure of the aircraft's complexity and size. The total flight score was the sum of the best flight score from each of the two missions: the Fire Bomber and Ferry missions.

3.1.2.1 Fire Bomber Mission

The Fire Bomber mission consists of two sorties where the aircraft performs as a water tanker. The flight score of this mission is given by

$$DF * Pounds\ of\ Water / Mission\ Time, \quad (2)$$

where the difficulty factor (*DF*) for the Fire Bomber mission is 2.0. The maximum allowed payload capacity was 4 liters, limiting the water weight in each lap to 8.8 lbs. The competition rules limited the exit orifice to a maximum diameter of one half-inch.

Each sortie consists of filling the aircraft from the ground storage tanks, taking-off in the required 150 ft, dumping the water during the downwind leg, and returning to land. The aircraft will be reloaded from two unmodified 2-liter plastic soda bottles and then fly a second sortie. Gravity or "pumped" loading of the aircraft is allowed, however the storage tanks may not be pressurized. The mission time will start with the aircraft initially empty and will stop when the aircraft comes to a complete stop after the second lap. The aircraft must complete a single 360° turn in the direction opposite of the base and final turns on the downwind leg of each lap.

Several penalties may affect the total mission time. Water may only be dumped during the downwind leg between the upwind and downwind turn markers. Early or late water dumping, including inadvertent release of water during a hard landing, will incur a 3-min time penalty per occurrence. Excessive water spillage during loading, as determined by the flight line judge, will incur a 1-min time penalty per infraction. Gross and/or deliberate spillage will disqualify the flight but will count as one of the teams' flight attempts. A 3-min time penalty will be added if the aircraft is not empty upon landing of the final lap. A time penalty of 3 minutes will be added for an incomplete second lap. If the second lap is not attempted or the 360° turn is not completed, the second lap will be timed as incomplete.

3.1.2.2 Ferry Mission

The Ferry Mission is the simpler of the two missions because there is no required payload to carry or deploy, which gives it a difficulty factor of 1.0. The aircraft is required to meet the take-off requirement of



150 ft, complete four laps, and land. As with the Fire Bomber mission, all laps flown must include a 360° turn in the direction opposite of the base and final turns on the downwind leg. The single flight score is

$$DF / MissionTime. \quad (3)$$

3.1.2.3 Competition Score Sensitivity to Mission Profiles

As there were only two possible missions that affected the competition score and the rated aircraft cost, it would have been difficult to design an aircraft that was optimized for both missions and the RAC. Therefore, it was necessary to determine the sensitivity of the competition score to each parameter; the flight time for each mission and the change in RAC. To easily compare these parameters the competition score was plotted as a function of unit change, where the unit for the two missions was the flight time and the unit for the RAC was the rated aircraft cost. Figure 3 illustrates the parameter affecting the competition score most was the flight time for the Fire Bomber mission. Therefore, it was decided to optimize the aircraft for that mission with a secondary focus on the RAC since that parameter had the second highest sensitivity to change. However, the flight score for the Ferry mission could not be ignored, since the best flight score for both the Fire Bomber mission and the Ferry mission are summed for calculating the competition score.

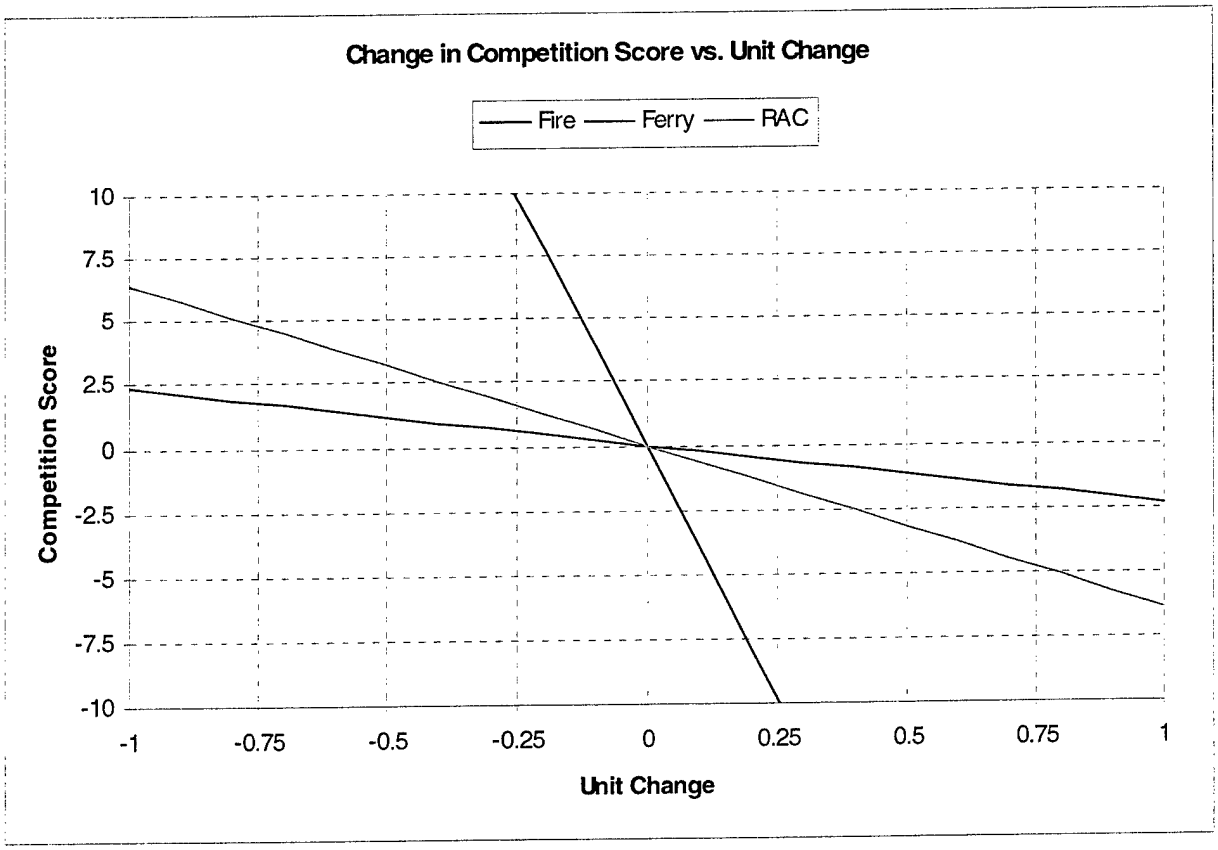


Figure 3: Competition Score Variation for the Two Missions and RAC



3.1.3 Rated Aircraft Cost

The rated aircraft cost is specified by the competition rules and provides a measure of the size and complexity of the aircraft. Since the total competition score is inversely proportional to the RAC, the RAC must be minimized without sacrificing mission performance. In thousands of dollars, the RAC is given as

$$RAC = (A * MEW + B * REP + C * MFHR)/1000, \quad (4)$$

where MEW is the manufacturers empty weight, REP is the rated engine power and MFHR is the manufacturing man-hours. The coefficients *A, B, C* are described and given in Table 1 and the Work Breakdown Schedule (WBS) to determine the number of manufacturing hours is shown in Table 2

Table 1: Rated Aircraft Cost Coefficients

Rated Aircraft cost (RAC) Breakdown			
Coef.	**Description**	**Definition**	**Cost Multiplier**
A	Manufacturers Empty Weight (MEW)	Actual airframe weight, lb., with all flight and propulsion batteries, but without any payload.	$300
B	Rated Engine Power Multiplier (REP)	Propulsion battery weight times a function of the number of motors.	$1,500
C	Manufacturing Man Hours (MFHR)	Prescribed assembly hours by work breakdown structure (WBS)	$20

Table 2: Work Breakdown Schedule for Manufacturing Man Hours

Manufacturing Man Hours: Work Breakdown Schedule		
Aircraft Component		**Man Hours per unit**
Wing	Area Control Surface	10 hr/sq. ft. 5 hr*type multiplier
Fuselage	Fuselage Volume	20 hr/cubic ft.
Empennage	Vertical Surface (no active control)	5 hr per vertical surface
	Vertical Surface (with active control)	10 hr per vertical surface
	Horizontal Surface	10 hr per horizontal surface
Flight Systems	Servo or Control Motor	5 hr per servo or motor control

3.2 Design Parameters

Based on last year's design, several design parameters where initially chosen to investigate various configurations. By analyzing several iterations, the set of design parameters was reduced to a minimal set by finding those parameters with the largest impact on the final competition score.

3.2.1 Aircraft Configuration

Because of the choice to further develop the previous year's entry, several design configurations such as canard and pusher style aircraft were immediately eliminated. The justification for this decision was due to the increase in complexity in the aerodynamic design for non-standard configurations, and from the lack of success of canard and pusher style designs in the competition. A major benefit from this approach is the carry over of design work and knowledge, allowing the team to focus on other areas of importance.

3.2.1.1 Fuselage

The length, shape, and overall design of the fuselage were dictated by the requirements to fit into the shipping box and carry a water tank. To allow for an increase in tail volume, a detachable tail was considered due to the restricted length of the shipping box. For the conceptual design, the configurations considered were a single fuselage body that fits in the box, or a multi-piece tail boom design allowing for removal of the tail section. The fuselage shape was also an important design consideration in that it affects several performance factors such as the total drag and available payload volume. The design could have also been chosen to help minimize the flight time of the Fire Bomber mission by utilizing dynamic pressure and Bernoulli's principle to obtain a pressure differential.

To reduce the total drag, improve efficiency, and design a faster airplane, a blended wing/fuselage shape based on the previously designed teardrop shape was chosen. To aid in the drop time of the water payload in the Fire Bomber mission by providing a pressure differential, it was decided to place the exit nozzle at the maximum thickness point of the aircraft and to add an air scoop behind the propeller to provide positive pressure.

3.2.1.2 Wing Design

The means to construct composite wings offered many more options for the design of the wing than was previously unavailable or unwarranted with other construction methods. The added versatility also required more decisions to be made early in the design process. Among the first was the choice of taper ratio. The optimal taper ratio of 0.35 to obtain a near elliptic lift distribution proved to be an unlikely choice. The major reason was that at the sizes we were predicting for the overall wing with a taper ratio at 0.35 would reduce the chord at the tip to a length where the Reynolds numbers would become unfavorable. In the same regard, the root chord would increase to accommodate the predicted planform area beyond a length that was practical. Also, the reduction with induced drag that would be a product of the taper ratio was believed to be negligible over the flight time that was predicted. Therefore, a rectangular planform was decided upon based on its practical chord lengths to achieve the projected planform area, and moreover, for its ease of construction.

The design of the wing planform last year was driven by the desire to build a reliable, efficient wing using standard balsa construction. This year, several construction techniques were considered allowing for many more design options than was previously available or warranted with balsa construction. Two of interest is the design of a lofted wing and adding washout. The benefit of a lofted wing and washout is that the span wise lift distribution can be made to follow a more elliptical distribution with a correct choice of airfoils, thus reducing the induced drag.

With the rectangular planform decided upon, an exploration of other means to reduce the induced drag needed to be explored. The versatility of foam-core composite wings allowed for the design of the wing to employ lofting; which is the use of a higher lift airfoil at the root and a lower lift airfoil at the tip. Another option that was available was the use of washout in the wing.

3.2.1.3 Empennage Configuration

Three different tail structures were initially considered: T-tail, V-tail, and a conventional tail. A conventional tail consists of a horizontal and vertical stabilizer, located aft of the center of gravity with the vertical stabilizer located above the horizontal stabilizer. A T-Tail resembles a conventional tail, except the horizontal stabilizer is located at the top of the vertical stabilizer. The T-Tail is more aerodynamically efficient than the conventional tail because the horizontal stabilizer is in a region of reduced downwash, but stiffness requirements of the tail assembly require more structure to support the raised horizontal stabilizer. Manufacturing of the T-Tail is also more complex due to the elevator control linkages that must be routed through the vertical stabilizer. A V-Tail uses two surfaces aligned in a V-shape to provide stability in pitch and yaw. A V-Tail can be lighter than a conventional tail but is more complex to analyze, build, and operate.

3.2.2 Propulsion System

The motor, batteries, and propeller are the three major components that make up the propulsion system. The design of the propulsion system must be such that it is able to provide enough power to meet the 150 ft takeoff distance and have enough energy to complete each mission. However, there are several competition rules that limit the design of the propulsion system. These primarily are a maximum current draw of 40 A, the requirement that the batteries be Ni-Cd and have a maximum weight of 5 lbs, that the motors are brushed motors from Astro Flight or Graupner, and that the propeller must be commercially purchased. The design parameters are then the choice of the motor size, the manufacturer, and the number of motors to use. Given these choices, the propeller and batteries were sized to meet the competition requirements. For the conceptual design, all brushed Astro Flight motors were considered using available motor specifications from the manufacturer. Since the 40-Amp fuse could limit the ability of a single motor to make the takeoff distance, multiple motor combinations were considered. Use of multiple motors, however, imposes a higher rated aircraft cost and increases the complexity of the design.

3.2.3 Payload Design

The payload design focused mainly on optimizing tank shape to minimize drop times. Initial concepts included pressure vessels, pneumatic pump, and ram air pressure using an air scoop. Once it was found that vessel pressurization other than ram air pressure was not allowed, the air scoop was chosen. Manufacturing was also taken into account. Several considerations were lightweight construction, smooth transition at the exit for maximum flow and off-the-shelf or in-house manufacturing.

Though the volume of water that the aircraft can carry only affects the Fire Bomber mission, the affect on competition score by reducing the amount of water that is carried was analyzed. Since at the conceptual design stage it would be difficult to fully predict a competition score, a sensitivity analysis was performed on the competition score based on a change in the amount of water volume that was dropped.



3.2.4 Structural Design

Though there is no easy way to relate the structural design to the final competition score, structural design decisions play an important role. This is a lesson that will not be soon forgotten after a wing structurally failed during the final scoring attempt in last year's competition. The design parameters for the structure where based primarily on material selection of either going with a balsa aircraft or with a composite aircraft using carbon fiber and fiberglass. Lessons learned from last year's experience directed the current construction towards composites. The reasons are that balsa construction is comparatively more time-consuming for building a prototype and final aircraft, and the significant increase in the strength-to-weight ratio that can be obtained.

3.2.5 Landing Gear Configuration

Landing gear affects the performance of an airplane in the air and on the ground. Though retractable landing gear offers the benefit of reduced drag, there are several reasons to choose fixed gear. The problems with retractable landing gear are increased weight, complexity, and reliability issues. Use of retractable gear also requires additional servos, further increasing weight and the rated aircraft cost. To improve ground-handling characteristics, a tricycle gear configuration was chosen.

3.3 Design Trade Studies

For those systems that had competing designs proposed, numerical Figures of Merit (FOM) were used to rate each proposal to aid in the final selection. If the design category was advantageous in a particular area it was assigned a positive one, neutrally beneficial was assigned a zero, and disadvantageous a negative one. Along with assigning numerical values to each proposed design, the categories were assigned percentage weights of importance. The FOM for a particular design was the sum over all weighted categories. The selection for the final design had the highest total sum.

3.3.1 Tail Configuration

FOM's were generated to determine the optimum tail configuration based upon overall complexity, RAC, aerodynamic performance, and stability and control as seen in Table 3.

Table 3: Tail Configuration

Tail Configuration					
Empennage Design	Complexity	RAC	Aerodynamic Performance	Stability & Control	Total
Weight	20%	20%	20%	40%	100%
Conventional Tail	1	1	0	0	0.4
T-Tail	0	-1	1	0	0
V-Tail	-1	-1	0	1	-0.2

Based upon the result of Table 3, the conventional tail configuration was selected.

3.3.2 Structural Configuration

Figures of Merit that dictated the structural configuration of the aircraft are shown in Table 4. The numerical FOM's categories include complexity, construction time, strength, ease of reproduction and reparability.



Table 4: Structural Configuration

Structural Configuration						
Construction Type	**Complexity**	**Construction Time**	**Strength**	**Reproduction**	**Reparability**	**Total**
Weight	20%	20%	20%	20%	20%	100%
Balsa	0	0	0	-1	1	0
Carbon-fiber-epoxy	0	0	1	1	0	0.4

The table illustrates the basis for the decision to choose a composite construction approach over balsa construction.

3.3.3 Propulsion

In the conceptual design phase, the main decision that had to be made was regarding the number of motors. As Table 5 illustrates, one motor was chosen due to the increased design and operating complexity of multiple motors.

Table 5:Propulsion Configuration

Propulsion Configuration				
Propulsion SubSystem	**Complexity**	**RAC**	**Avaliable Power**	**Total**
Weight	40%	40%	20%	100%
Single Engine	1	1	0	0.8
Multiple Engines	-1	-1	1	-0.6

4 Preliminary Design

4.1 Mission Models – Fire Bomber and Ferry

To estimate the flight time of each mission the mission profile was broken into phases. The distance, speed, and time of flight for each phase are listed in Table 6 along with the average flight speed, lap time, and total mission time. The speed on the runway is the average speed during the acceleration from zero to the takeoff speed of 54.4 fps. The same is true for the climb and landing regimes, where the speed listed is the average for that phase. For the Fire Bomber mission, the flight speed during the water deployment phase was the higher of two speeds; the stall speed of the aircraft or the speed needed to fully deploy the payload within the 1,000 ft.

Table 6: Breakdown of Mission Profiles for Fire Bomber and Ferry mission.

Fire Bomber			
Regime	**Length (ft)**	**Speed (ft/sec)**	**Time (sec)**
Fill			15.0
Runway	150.0	37.8	6.1
Climb	250.0	64.1	3.9
Turn 1	328.5	92.4	3.6
Drop	1157.1	68.1	17.0
Turn 2	328.5	92.4	3.6
Land	250.0	69.5	3.6
Taxi			5.0

Ferry			
Regime	**Length (ft)**	**Speed (ft/sec)**	**Time (sec)**
Runway	150.0	37.8	4.0
Climb	250.0	64.1	3.9
Turn 1	297.1	92.4	3.2
Straight	1094.2	92.4	11.8
Turn 2	297.1	92.4	3.2
Land	250.0	69.5	3.6



Using Eq. 2, the flight score for the Fire Bomber mission was estimated at 9.28 and from Eq. 3 the flight score for the Ferry mission was 0.54. The difference in flight score backed up the conclusions made based from Figure 3.

4.2 Estimates of Aircraft Parameters

4.2.1 Estimates of Lift

The lift required for the aircraft had to be estimated for two sets of flight conditions. The first was with the aircraft at cruise conditions and the second included both takeoff and landing with the maximum predicted aircraft weight of 22.7 lbs. The coefficient of lift for both sets of flight conditions was the descriptive tool for estimating the required lift, and is defined as

$$C_L = \frac{L_{req}}{\frac{1}{2}\rho V^2 S_w}. \tag{5}$$

In the case of cruise and for takeoff and landing L_{req} is equal to the weight of the aircraft. The atmospheric density, ρ in Eq. 5, was specified based on the standard atmospheric conditions for the altitude of 1,400 feet, which is the approximate altitude at the competition site. At this altitude, the density on an average day should be 0.00231 slugs/ft^3. The projected area of the wing S_w was estimated to be 9.5 ft^2. All that was left to derive the initial estimates of the coefficient of lift for the aircraft was the velocity.

The desired cruise speed was set at 100 fps to reduce the amount of flight time. With the chosen velocity, Eq. 5 was used to determine a cruise lift coefficient of 0.2. The velocity for takeoff and landing was based solely on the predicted stall velocity by the equation

$$V_{to} = 1.05 V_{stall}. \tag{6}$$

Therefore, the stall speed of the aircraft needed to be calculated at the predicted flight conditions. The stall speed for the aircraft is defined by

$$V_{stall} = \sqrt{\frac{2W_{to}}{\rho S_w C_{L_{MAX}}}} \tag{7}$$

where W_{to} was the takeoff weight of the aircraft. It was desired to have a slow stall speed to allow for a practical landing speed. From estimates of the maximum lift coefficient, takeoff weight, and wing area the stall speed was initially calculated as 44 fps. The predicted takeoff velocity was estimated at 46.2 fps, based on Eq. 6. With flight condition estimates known, the coefficients of lift for each were calculated with Eq. 7. Table 7 gives a summary of the results. With estimates for the maximum and cruise coefficients of lift, a search for airfoils to meet these estimates was initiated.

Table 7:Preliminary Estimates for the Coefficients of Lift at Different Flight Conditions

Estimated Coefficients of Lift			
Cruise		Takeoff	
With Payload at Max Speed	0.207	With Payload	0.971
Without Payload at Max Speed	0.423	Without Payload	0.594
During Water Drop	0.127		



4.2.2 Estimates of Drag

4.2.2.1 Total Parasite

The estimation of total drag for the aircraft was of extreme importance to the design. Preliminary estimates included the parasite drag of wings, fuselage, horizontal stabilizer, vertical stabilizer, landing gear struts, and landing gear wheels. The drag due to rolling down the runway was added for the total drag estimate to occur during takeoff.

4.2.2.1.1 Wings

The wing parasite drag was estimated as a function sensitive mainly to skin friction drag. Since the skin friction drag is dependent on the wetted area of the object in question, the wetted area of the wing had to first be calculated. The equation used to calculated the wetted area of the wing is given by

$$S_{wet} = 2.04(S_w - A_o) \tag{8}$$

where A_o represents the area of the wing that is obstructed by the fuselage. The multiplier "2.04" accounts for both sides of the wing as well as a factor of 1.02 to adjust for the curvature. With the wetted area known the next step was to calculate the skin friction coefficient with

$$C_f = 0.60515(\log_{10}(R_e))^{-2.58}. \tag{9}$$

Using the skin friction coefficient and the wetted surface area the equivalent parasite drag coefficient for the wing could be determined using the following relation:

$$C_{D_P} = K\frac{C_f S_{wet}}{S_w} \tag{10}$$

The term K is an empirical constant curve unique to the wing. The value for K used in the estimations for the parasite drag of the wing was 1.21, and was found using Ref 1.

4.2.2.1.2 Horizontal and Vertical Stabilizer

The method for estimating the parasite drag for both the horizontal and vertical stabilizers is identical to the method that was used in the estimation of the parasite drag coefficient for the wing. The K factors used for the estimation of parasite drag for the horizontal and vertical stabilizers were 1.198 and 1.199, respectively.

4.2.2.1.3 Fuselage

The estimation for the equivalent parasite drag for the fuselage was determined with the same method that was used for the wing. However, the wetted surface area for the fuselage required a much different approach than what was used for the any of the listing or stabilizing surfaces. The wetted area of the fuselage was estimated based on the surface area of a cylinder. Therefore, the equation that was used to estimate the wetted surface area of the fuselage is given by

$$S_{wet} = 1.02(\pi DL), \tag{11}$$

where D is an approximation of the predicted fuselage diameter and L is the approximation of the predicted fuselage length. Similar to Eq. 8, the surface area of the cylinder approximating the fuselage was multiplied by a factor of 1.02 to account for the curvature of the fuselage. The factor K for the



fuselage has a magnitude between 1.1 and 1.3, which is typical of most air transports. To remain conservative in our estimations of the total parasite drag for the aircraft, a value of 1.5 was chosen.

4.2.2.1.4 Landing Gear

The estimation of the parasite drag for the landing gear struts was composed of two separate estimations. The first was for the nose-gear strut, which was modeled as a slender cylinder (L>>D). For this condition, the friction coefficient is 1.2 (Ref. 1). The strut was also considered to be perfectly streamlined, which meant that K in Eq. 10 was equal to 1.

The estimation of the parasite drag for the main landing gear differed from the nose-gear strut in that the main was modeled as a flat plate. Like the nose-gear, the factor K was set equal to 1 to approximate a perfectly streamlined object. The equation for calculating the skin friction coefficient differs from the equation given Eq. 9 in that the flow is considered to be turbulent in the direction of the flat plate. Thus, an equation accounting for turbulent flow is used, resulting in

$$C_f = 0.455(\log_{10}(R_e))^{-2.58}. \tag{12}$$

4.2.2.1.5 Landing Gear Wheels

The wheels of the landing gear were the final component considered in the preliminary estimation of the parasite drag for the aircraft. Each wheel, one for the nose and two for the main gear, was modeled as a finite cylinder where the coefficient friction was approximated near 0.5, a slightly conservative estimate. Again, the factor K in Eq. 10 was set equal to 1.

4.2.2.2 Induced Drag

The two components that were considered for the estimation of the total induced drag were the wing and horizontal stabilizer. The coefficients of induced drag were calculated through a lifting line model used to estimate the lift produced by the wing, as described in detail in McCormick.

4.2.2.3 Rolling Drag

It was known that the drag due to rolling friction on takeoff would play a major part in the takeoff distance. This value is important in order to accurately predict the thrust required for takeoff. The estimation of this drag was based upon the simple static friction equation

$$F_f = \mu_f (W - L). \tag{13}$$

Here, μ_f is the coefficient of friction, estimated at 0.25 (Ref 4), L is lift, and W is the weight of the aircraft.

4.2.2.4 Total Drag

With the estimations for each component calculated, a total drag estimate was found. Table 8 displays the coefficient of drag estimate for each component and the total aircraft drag coefficient for both cruise and takeoff flight conditions.



Table 8: Estimates of Drag Coefficients for Each Component of the Aircraft.

Estimated Coefficients of Drag			
Cruise		**Takeoff**	
Wing	0.016	Wing	0.0076
Horizontal Stabilizer	0.0044	Horizontal Stabilizer	0.0048
Vertical Stabilizer	0.0008	Vertical Stabilizer	0.0008
Fuselage	0.0088	Fuselage	0.0082
Main Landing Gear Struts	0.0003	Main Landing Gear Struts	0.0003
Nose Landing Gear Strut	0.001	Nose Landing Gear Strut	0.001
Main Landing Gear Wheels	0.003	Main Landing Gear Wheels	0.003
Nose Landing Gear Wheel	0.001	Nose Landing Gear Wheel	0.001
Induced	0.0017	Induced	0.033
		Rolling	0.06

4.2.3 Estimates of Stability

The design needed meet two parameters for longitudinal static stability; $C_{M_0} > 0$ and $C_{M_\alpha} < 0$. A positive C_{M_0} is required so that the angle of attack for trimmed flight is positive. Figure 4 displays a plot of the coefficient of moment versus the angle of attack where the angle of attack for trimmed flight was determined to be 3.3°. Another factor that determined the static stability of the aircraft, as well as the extent of the stability, was the static margin. The static margin is defined by

$$SM = \left(\bar{x}_{ac_A} - \bar{x}_{cg}\right), \tag{14}$$

where the location of the aerodynamic center (Neutral Point) is determined as a function of the *tail volume coefficient*. The horizontal tail volume coefficient directly specifies the location of the neutral point through the relation

$$\bar{x}_{ac_A} = \bar{x}_{ac_{wf}} - \frac{C_{M\alpha_f}}{C_{L\alpha_w}} + \eta_h \frac{C_{L\alpha_h}}{C_{L\alpha_w}} \bar{V}_h \left(1 - \frac{\partial \varepsilon}{\partial \alpha}\right) \tag{15}$$

Figure 4 displays this variation of the neutral point in relation to the horizontal tail volume coefficient. It was from these relations that the static margin for this design was determined to be approximately 13%. This value met the design criteria that it be positive, representing the aerodynamic center of the aircraft (Neutral Point) as aft of the center of gravity, ensuring static stability. Another criterion that this static margin met was that it not be too large and positive, which would hinder effective maneuvering during flight and especially on takeoff.

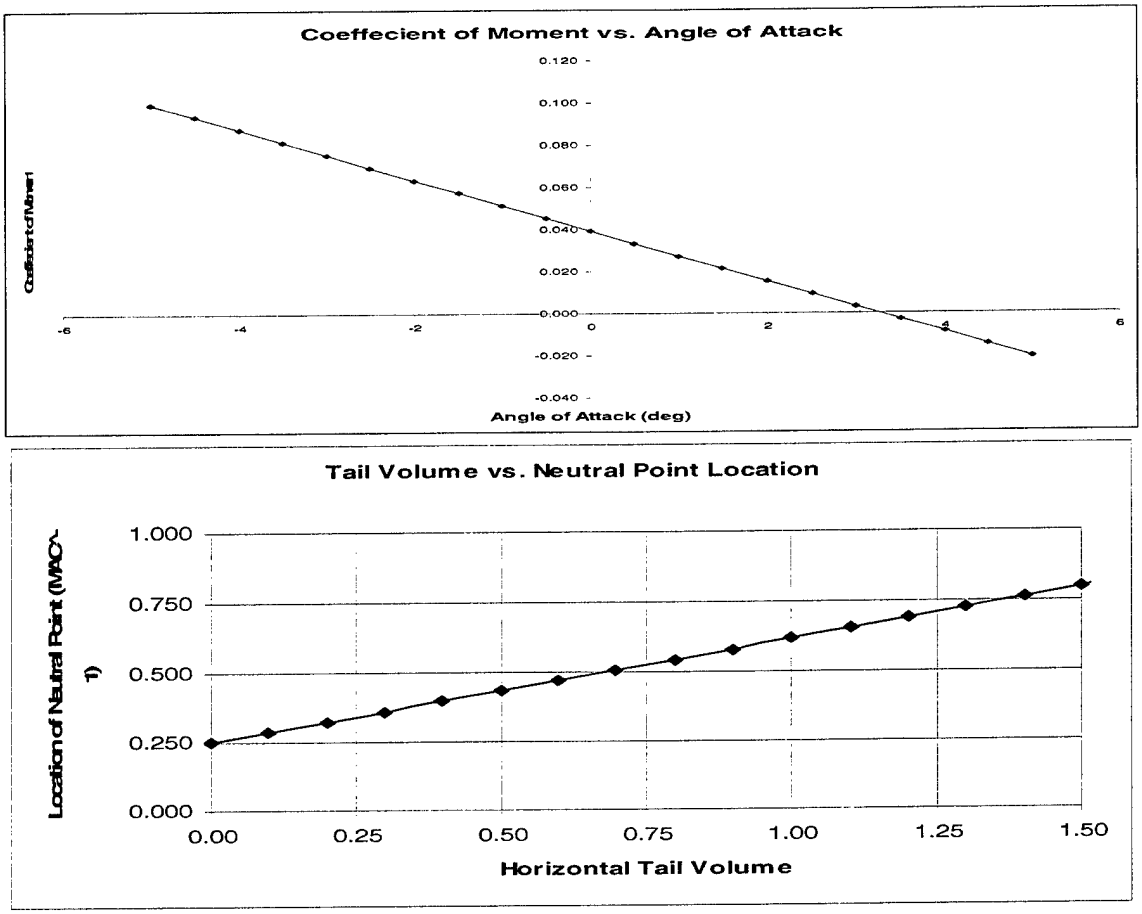


Figure 4: A (top) Coefficient of Moment vs. Angle of Attack. B (bottom) Neutral Point vs. Horizontal Tail Volume.

4.3 Design Optimization and Sizing Trades

4.3.1 Wing Design

4.3.1.1 Aerodynamic Wing Design

With the rectangular planform decided upon in the conceptual design phase, steps to reduce the induced drag were explored. The versatility of foam-core, composite wings allowed for the design of the wing to employ lofting; the use of a higher lift airfoil at the root as compared to a lower lift airfoil at the tip. This type of construction would also easily allow for washout to be built into the wing. Thus, the lift distribution of the rectangular wing can be made more elliptical with a proper choice of airfoil sections and washout.

Preliminary estimates suggested a wing design that could deliver a coefficient of lift at approximately 0.2 at zero angle of attack and a C_{Lmax} near 1.0 to meet the mission requirements. An extensive search for possible airfoil pairings that would meet the needed requirements was concluded after examining over 200 airfoil polar plots at Reynolds Numbers of approximately 300,000 from the Nihon University Aerospace Student Group online database. Key features explored in each airfoil were the desired C_{Lo}, C_{Lmax}, stall angle of attack, and minimum coefficient of drag. Based on these criteria the pairing of airfoils



that best fit the specified requirements were the Selig S3010 at the root and the Selig/Donovan SD8040 at the tip. Table 9 displays the basic characteristics of each airfoil and Figure 5 displays the plots used as the basis for their selection.

Table 9: Characteristics of S3010 and SD8040 airfoils

	Airfoils	
	S3010	SD8040
Thickness	10.32	10
Camber (%	2.79	2.64
C_{l_0}	0.3	0.22
α_0 (°)	-3	-2.1
$C_{l_{max}}$	1.2	1.08
α_{stall} (°)	10.1	11.9
C_{l_α} (/ °)	0.1103	0.117

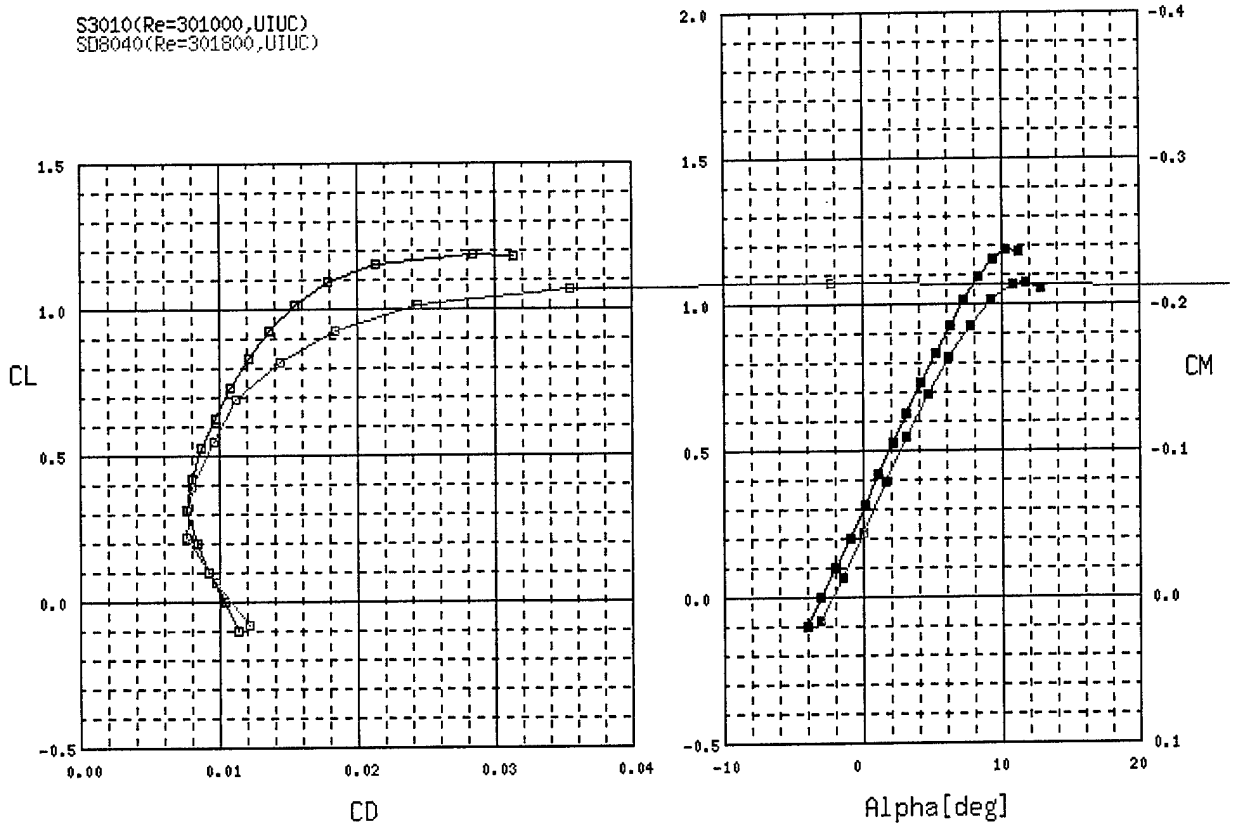


Figure 5: Polar Plots for S3010 and SD8040 Airfoils. (Courtesy of the Nihon University Aerospace Student Group Online Database.)

The key feature that made these two airfoils an ideal pair for our uses was the stall angle of attack. The S3010 stalls at nearly 2° less angle of attack than the SD8040. This characteristic would effectively avoid tip stall without adding any washout to the wing, which could reduce the wing's overall drag

efficiency at low angles of attack, shown in Figure 6. Another feature that weighed heavily on this choice of airfoils was the difference in the coefficient of lift between the two airfoils at any given angle of attack. The S3010 displayed a higher C_l than the SD 8040 at every angle of attack. This suggested that a lift distribution approximating an elliptic distribution could be attained with minimal taper, which would ease the manufacturing process.

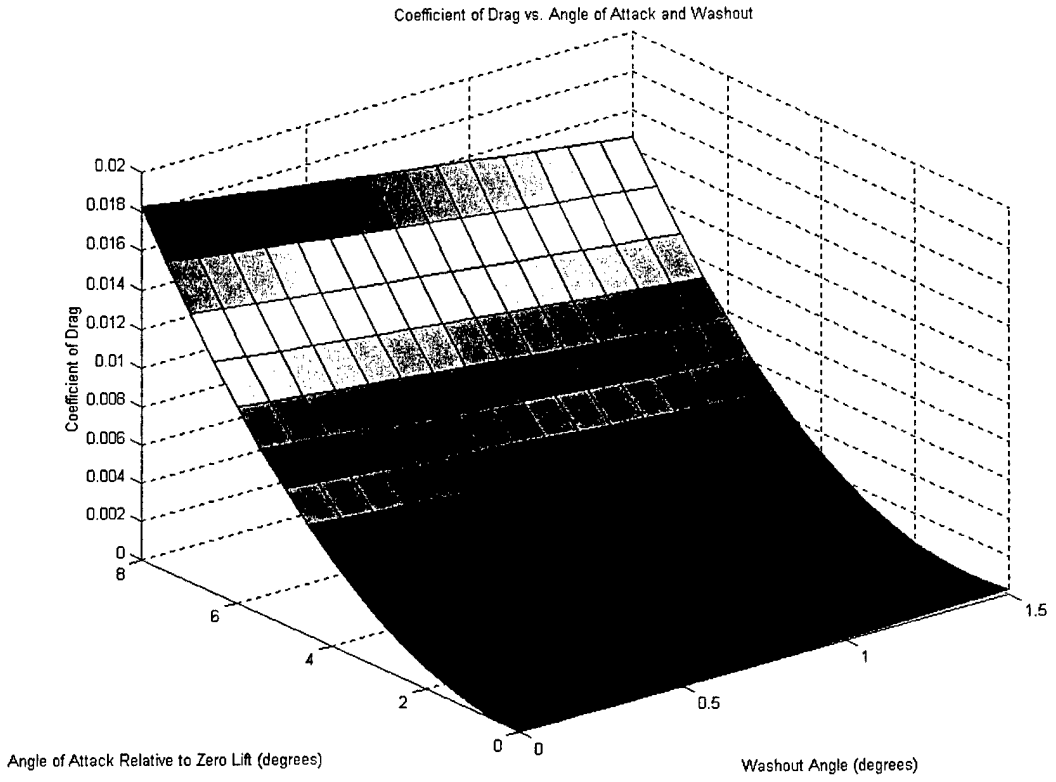


Figure 6: Variation of Induced Drag Coefficient vs. Angle of Attack Relative to Zero Lift and Washout Angle

4.3.1.2 Planform

The preliminary design of the wing was based on a practical limit of a 114-inch wingspan, which would allow a two-piece wing to fit into the shipping container with a 22-inch blended fuselage body. With a set wingspan, the effects of different taper wing areas were explored. Increased wing area was found to improve performance, but at the cost of higher RAC and a larger tail requirement. At this stage of the design, the target for tail length was such that the entire fuselage would fit into the container without disassembly. This meant that a wing with minimum area to meet performance requirements was desired.

4.3.1.3 Wing Loading

A lower wing loading would have resulted in a lower stall speed, shown in Figure 7, which would have been beneficial to landing speed. However, a lower wing loading would be sensitive to gusts, which are quite common at the competition location. Another factor that influenced the wing loading was the significant weight changes between the Fire Bomber and Ferry missions. As an upper limit a wing



loading of 40 oz/ft^2 was chosen. With the designed C_{Lmax} the stall speed was estimate was reaffirmed to be approximately 44 fps.

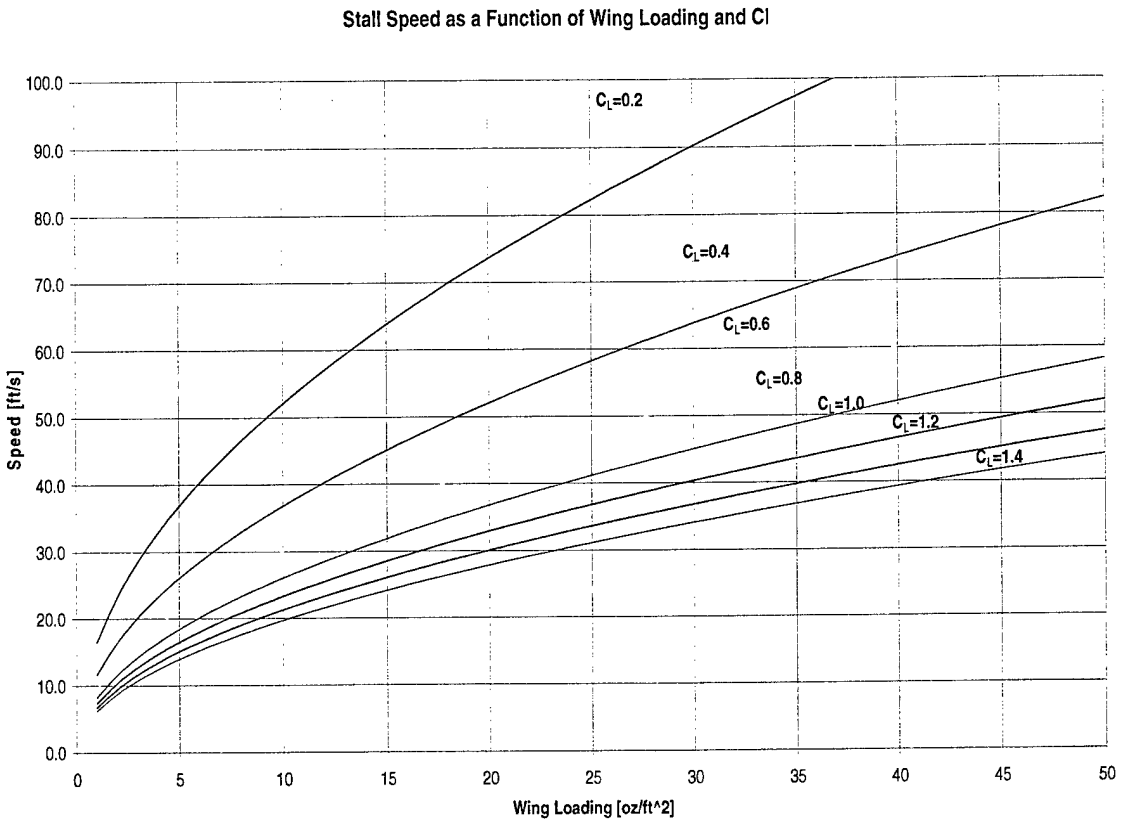


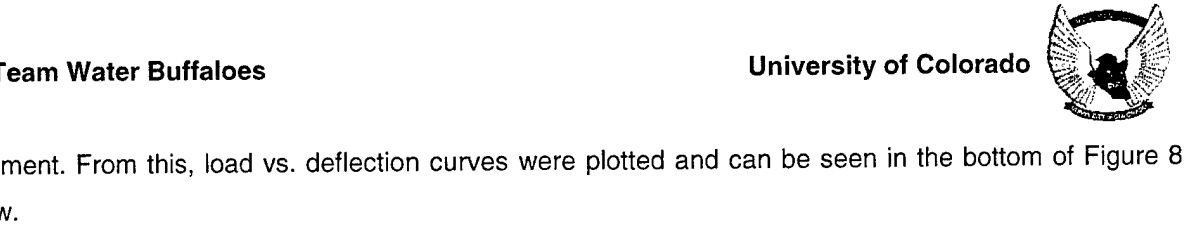
Figure 7: Stall Speed vs. Wing Loading

4.3.1.4 Structural Wing Design

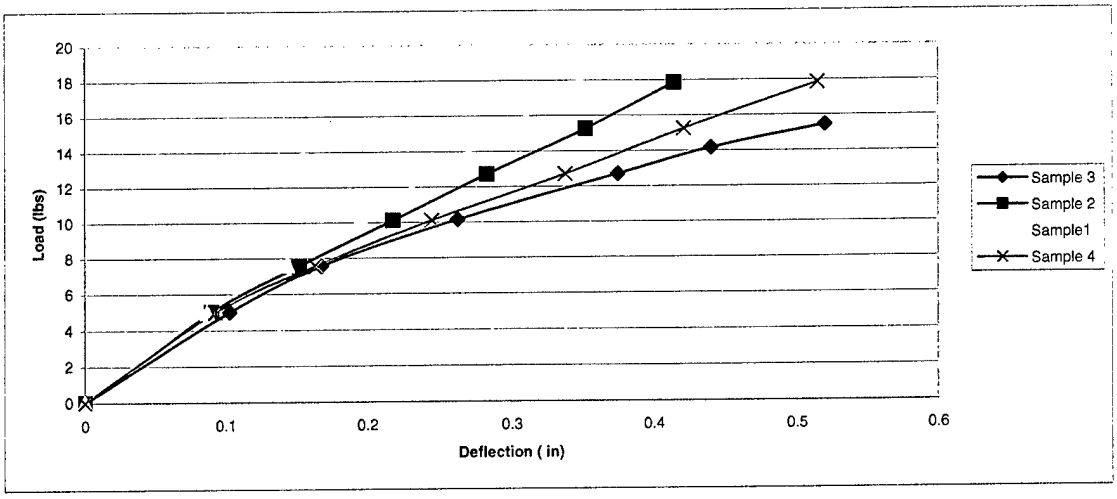
The structural behavior of foam core wings cannot be modeled very well by the usual means. FEM analysis does not very accurately predict the behavior of composite foam core wings and so other methods for analyzing the wings were found.

4.3.1.5 Foam Cores

To analyze the behavior of foam core wings it is necessary to have knowledge the structural behavior of foam. Properties of the white styrofoam were found on a manufacturing specifications page distributed by OPCO (Ref 8). The modulus of elasticity of both the foams was of particular interest for stress analysis. The white foam had a modulus of elasticity of 200 psi. The modulus of elasticity of the pink foam could not be found so an experiment was set up to do this. After consulting Professor Kurt Maute at the University of Colorado at Boulder it was found that measuring the deflection in a piece of foam of known cross-sectional area and length would suffice. The setup, seen in the top of Figure 8 below, was as follows: four 6 in^2 foam pieces were cut to a length of 22 in; the ends of these were clamped to a table on both sides; a load was placed on the center of mass and the deflection was measured at each load



increment. From this, load vs. deflection curves were plotted and can be seen in the bottom of Figure 8 below.



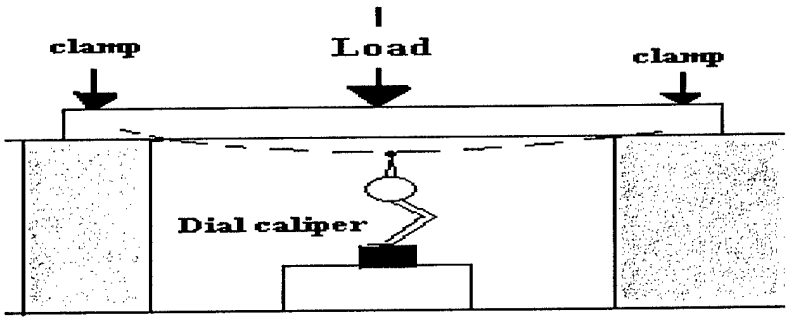


Figure 8: Top: Load vs. Deflection for Pink Foam. Bottom: Load vs. Deflection Test Setup.

From the load vs. deflection plot, the modulus of elasticity is defined by Reference 3 as

$$\frac{P}{v_{\max}} = \frac{48EI}{L^3}. \tag{16}$$

where the quantity P/v_{max} is the load-deflection curve slope, E is the modulus of elasticity, I is the moment of inertia, and L is the length of the beam. These quantities were known except for I, which was calculated using the following:

$$I = \frac{1}{12}bh^3. \tag{17}$$

Here b was the length of the segment of the foam beam being deflected and h was the height of the cross section. Once I was found, Eq. 16 could be solved for E, which yielded 188psi for the pink foam.

4.3.1.6 Distributed Load Analysis

The distributed load test was used to test the wings for structural integrity under the specific loads that might be seen by the wings. The load distribution over the wings was calculated using the lifting line

code. The load distribution was modeled by placing straps at evenly spaced locations along the wing and hanging pieces of conduit from it in specific locations as shown in Figure 9 below.

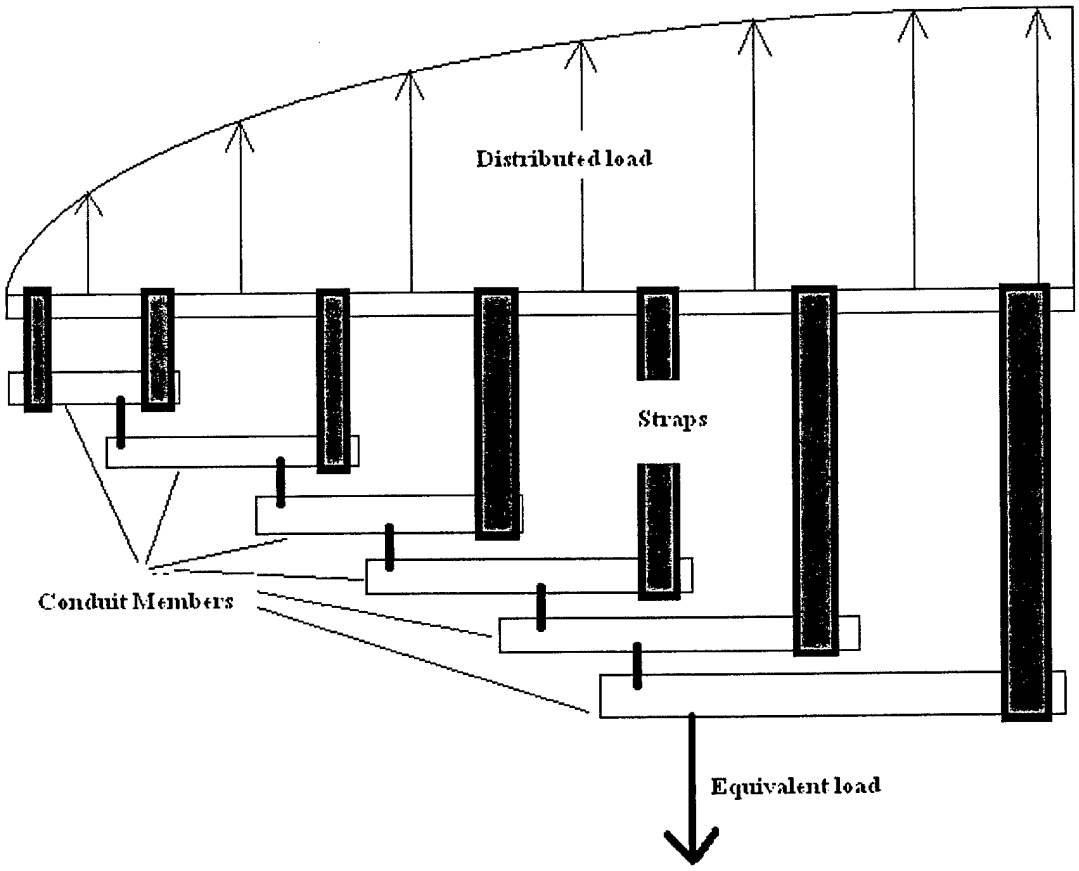


Figure 9: Distributed Load Test on a Wing.

The load curve was integrated around each strap to get an equivalent load at each strap. The load to be placed on the conduits was then calculated by summing the forces and moments about each member with reference to the end of the wing as shown in Figure 10. Iterating through the members, it was found that Eq. 18 could be used to find the distance between the all-thread and the straps, and ultimately to find the placement of the load weight.

$$x_n = \frac{T_{n+1}(n * C - x_n - x_{n-1} \ldots x3)}{T_n} \tag{18}$$

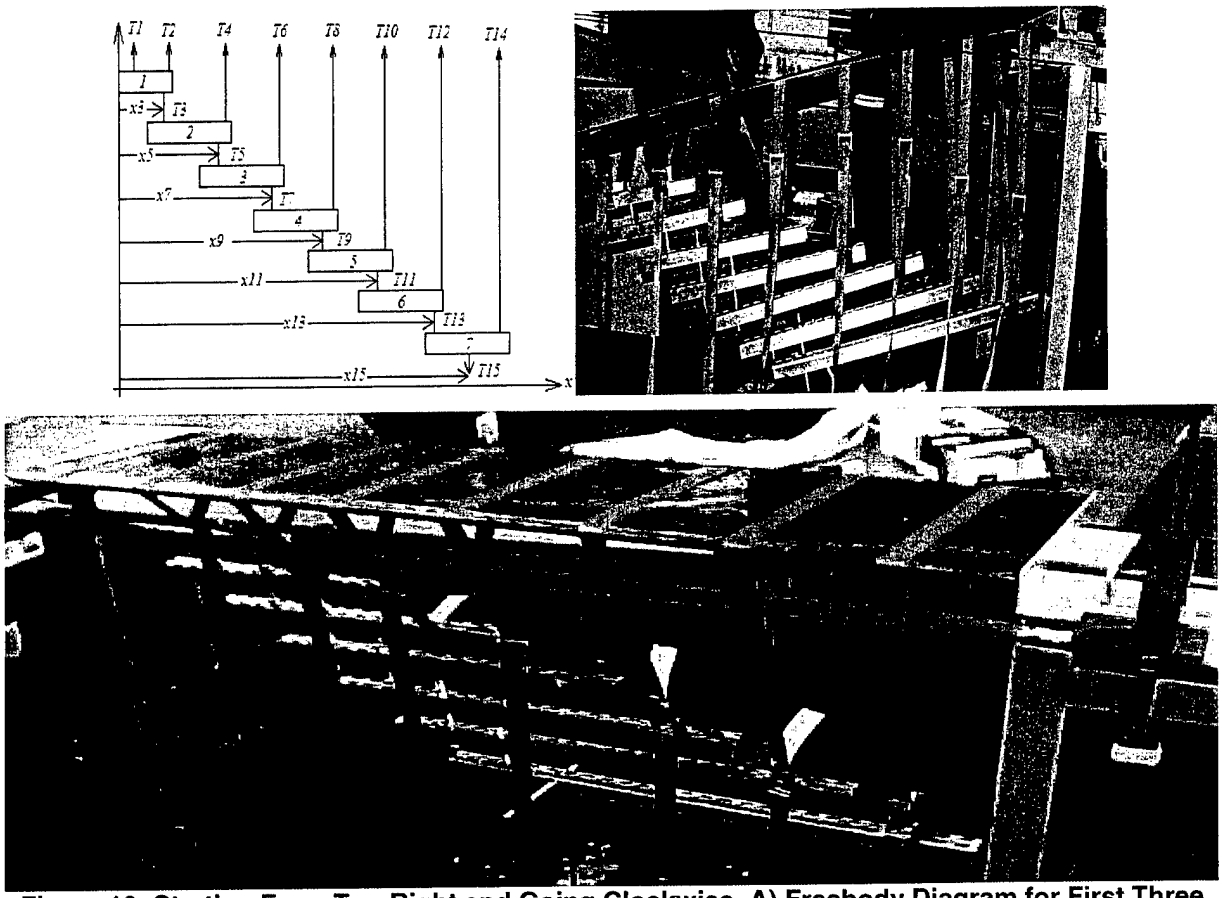


Figure 10: Starting From Top Right and Going Clockwise. A) Freebody Diagram for First Three Conduit Members. B) Wing in Bending Test. C) Distributed Load Setup.

With this information, the wing was then mounted to a piece of aluminum conduit by means of a wing spar embedded 9 inches into the wing root. This wing spar went through the conduit and was held down by a u-bolt. A quarter inch rod was also embedded into the wing near the trailing edge prevent the wing from rotating under the load. The mount was then secured to the table by clamps. Finally, weight was loaded and the deflections at 3 equidistant stations were measured until failure.

4.3.1.7 Layered Stress Analysis

A stress analysis was done on the wings to determine the shear stress between the layer of carbon fiber and the foam core. This was done to determine the wing deflection at which the layers would separate under shear. It was decided that the most efficient and effective way of doing this would be to model our wing with a rectangular wing that had the same deflection under the same loads. The lifting line code was used to get a lift distribution over the wing. Using the tip deflection from the wing in the distributed load experiment and the distributed load placed on it, a lift distribution was created for the model and imported into Matlab. The lift distribution data was separated into *x* and *y* arrays and a fifth order polynomial was then fitted to the data. The following equation was used to find the internal bending moment at the root of the wing:

$$M_z = \iint y(x)dx. \tag{19}$$

Here *y(x)* is the load distribution and M_z is the internal bending moment. The maximum stress would be at the root so Eq. 19 was integrated twice from zero to L, where zero was at the tip of the wing and L was the length of the wing. A diagram of this is shown in Figure 11 below.

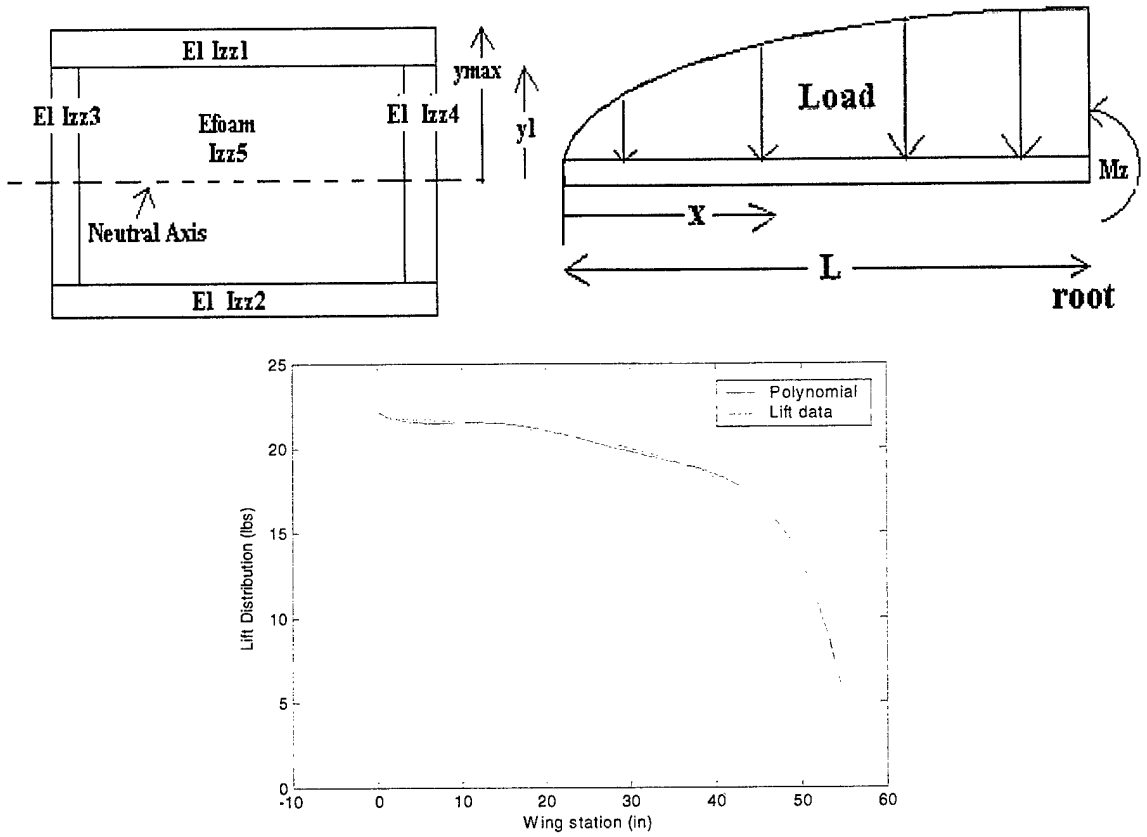


Figure 11: Clockwise from Top Left. A.) Model of Rectangular Cross-section. B.) Coordinate System of the Beam Model. C.) Lift Distribution and Polynomial Curve Fit.

In order to find the deflection at the tip of the rectangular model wing the following equation was used:

$$M_z = EI_{zz}\frac{d^2v}{dx^2}, \tag{20}$$

where *E* is equivalent modulus of elasticity of the beam, I_{zz} is the total moment of inertia of the beam and d^2v/dx^2 is the second derivative of the deflection. Integrating this twice with initial boundary conditions of $dv/dx(x=L)=0$ and $v(x=L)=0$ and evaluating at x =0 resulted in

$$v(x) = \frac{M_z x^2}{2EI_{zz}} - \frac{M_z L^2}{2EI_{zz}}, \tag{21}$$



This was then evaluated at different height values, which resulted in different EI_{zz} values, until the deflection matched that in the distributed load test. The shear stress was then calculated by the following equation

$$\tau_{xs} = -\frac{Q_{comp} V_y}{EI_{zz\,total} t}. \tag{22}$$

Here τ_{xs} is the bending shear stress, V_y is the maximum normal stress, which in this case is at the root, t is the thickness, and Q is the first area moment. The first area moment was then calculated for the top piece of the carbon fiber by:

$$Q_{comp} = \int_{y1}^{ytop} E_1 w_1 h_1 dy. \tag{23}$$

Here E_1 was the modulus of the top layer, w_1 is the width of the top layer, h_1 is the height of the top layer, y_{top} is the distance from the center of gravity (C.G) of the model's cross-section to the top to the top layer, and y_1 is the distance from the C.G. to the bottom of the top layer. Eq. 23 was then evaluated for an extreme condition of 150 fps flight speed and an 11° bank. It was found that there was a shear stress at the root of 316.5psi between the first layer of carbon fiber epoxy and the white foam. This does not even come close to the shear strength of carbon fiber epoxy of 72518 psi, but it does exceed the shear strength of white foam of 20psi (from density). This means that the white foam would shear under the load, most likely causing the wing to fail. However, due to the placement of a wing spar at the root it was concluded that this would not be a problem. Since there is no wing spar after the point at which the rib is attached the same study was done there yielding 143.76psi at the rib, which does not exceed the shear strength of the carbon-fiber epoxy, but still exceeds the shear strength of foam. From this analysis and the fact that the stations inboard of the rib were reinforced by the spar, it was decided to reinforce the spar section with another carbon-fiber epoxy layer. Another stress analysis study is to be done with two layers of fiber at the rib section.

4.3.2 Power & Propulsion

The power system was analyzed primarily based on meeting the takeoff constraint of 150 feet, minimizing the overall weight of the motor and battery supply, as well as maximizing the overall efficiency. As stated earlier in the conceptual design section, one motor was chosen in the tractor configuration. The motors available due to the competition requirements were by Astro Flight or Graupner. The Astro Flight 60 size was chosen for several reasons. During last year's competition, the 40 size was used in an attempt to have the lowest rated aircraft cost possible. This however, resulted in a flight speed that was too slow to be competitive with the other teams' flight scores. The 90 size was eliminated because of how much weight it would add to the aircraft, not only due to the motor itself but the additional battery weight that would be needed to power it. The DBFDesign was run for both the 60 and 90 size motors in order to verify this assumption. The number of batteries, rated aircraft cost, and competition scores were all compared for the same propeller size and takeoff distance constraint. The 60 size motor required 6 less



battery cells, had a lower rated aircraft cost by 1.3 points, and a competition score that was 6.6 points higher. Once the decision to use a 60 size was finalized, the Astro Flight was chosen over the Graupner because of prior product knowledge and success with that company.

4.3.3 Fuselage Design

The driving force behind the preliminary design of the fuselage is the same as those specified in the conceptual design phase. That is, the design of the fuselage shape follows directly from last year's concept, the design of the water tank, and minimizing parasitic drag. The length of the fuselage was chosen to be forty-four inches, allowing for a non-detachable boom structure and nose cone spinner. The height of the fuselage was made as small as possible based on the water tank size to minimize RAC.

4.3.4 Water Tank

4.3.4.1 Tank Location

The dimensions of the payload to be carried directly affect the overall configuration of the aircraft. Primary importance was the location of the payload and its effects on the center of gravity of the aircraft. Since the payload must be deployable, it must be positioned such that the center of gravity shift due to deployment will not exceed the requirements for longitudinal stability. The amount of shift is dependent upon the weight of the aircraft, the weight of the payload (fixed at 8.8 pounds), and the distance between the centers of gravity of the payload and aircraft. Zero shift occurs when the center of gravity of the box and aircraft are coincident. The initial design target was to place the center of gravity of the payload package longitudinally coincident with that of the empty aircraft.

4.3.4.2 Tank Design Analysis

Analysis of a simple cylindrical tank as seen in Figure 12 was done to formulate a theoretical emptying time for 4-liters of water.

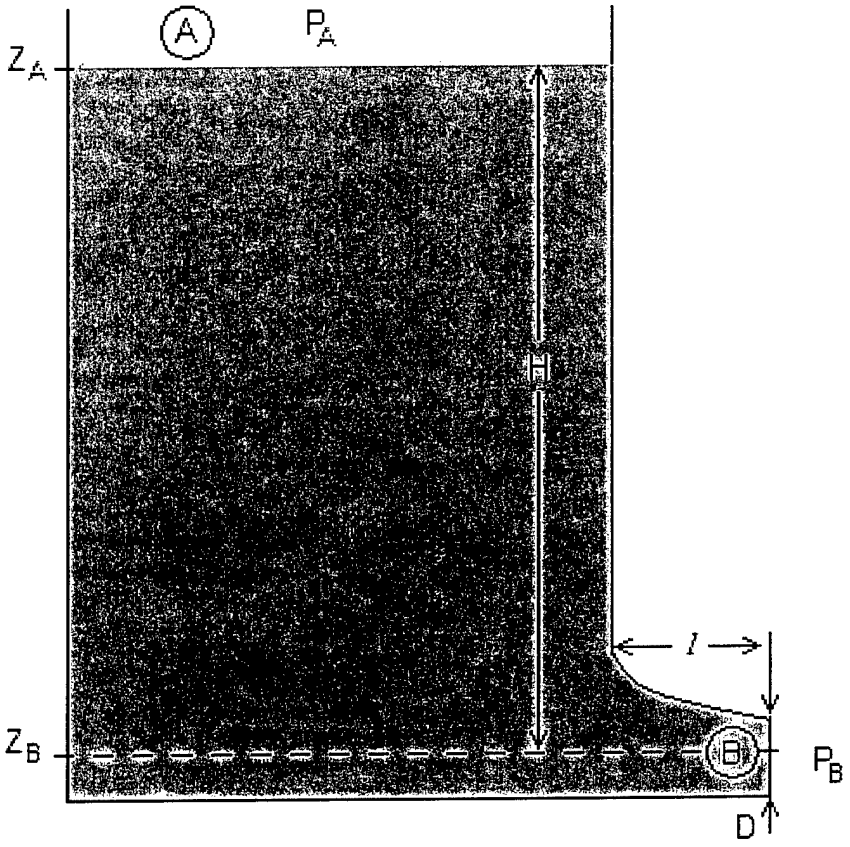


Figure 12: Tank Design Analysis

Applying Bernoulli's equation to Figure 12

$$\frac{P_A}{\gamma}+Z_A+\frac{V_A^2}{2g}=\frac{P_B}{\gamma}+Z_B+\frac{V_B^2}{2g}+h_t. \tag{24}$$

Here P and V are the pressure and velocity of the water respectively at the point of interest, Z is the height of the water from the bottom of the tank and h_t is the total loss. Rearranging Eq. 24 with the assumption that the pressure change between points A and B is simply dynamic pressure, the time averaged height of the water is one-half of the initial water height, and the total loss h_t is the sum of the head loss and an experimentally determined loss coefficient *K*, an equation for the water velocity at the exit during flight was obtained. This relation is shown by

$$V_B=\sqrt{\frac{\frac{1}{2}V_\infty^2+2gH}{1+\left(\frac{f\,l}{d}+K\right)}}, \tag{25}$$

where, f is the friction factor, V_∞ is the freestream velocity, g is the gravitational acceleration, l is the length of the exit tube, and D is the diameter of the exit orifice. The volume flow rate is then defined as:

$$\dot{V}=V_B A=\frac{Volume}{time} \tag{26}$$



The drop time was then found as a function of the freestream velocity by substituting Eq. 25 into Eq. 26 and solving for time.

$$time = \frac{Volume}{A}\left(\sqrt{\frac{\frac{1}{2}V_\infty^2 + gH}{1+\left(\frac{f\,l}{d}+K\right)}}\right)^{-1} \tag{27}$$

Using the values from the water drop experiment in Section 7.2.1 and the given values below, the loss coefficient and friction factor were determined.

$V_B = 5.18 ft/s$ $V_\infty = 0 ft/s$ $l = 0.083 ft$ $Volume = 0.141 ft^3$ $\nu = 1.059\times10^{-5}\, ft^2/s$

$H = 0.81 ft$ $g = 32.2 ft/s^2$ $d = 0.0417\, ft$ $A = 0.00136 ft^2$ $RN = \frac{Vd}{\nu} = 20{,}380$

The friction factor *f* was then found from a Moody chart in Reference 10 to be $f = 0.0275$. The loss coefficient *K* was then solved for from Eq 27 and found to be *K*=0.094. Figure 13 shows a *Matlab* plot of the theoretical drop time of 4-liters as a function of flight speed using the same geometry.

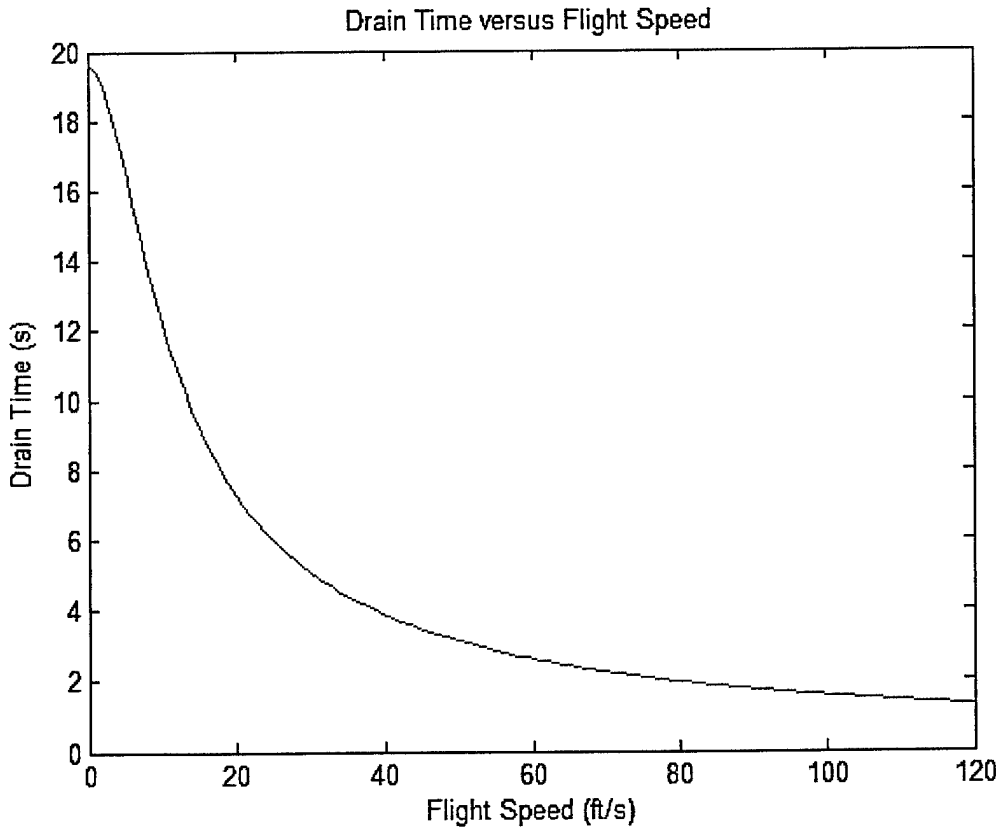


Figure 13: Drop Time vs. Flight Speed

Based on a design cruise velocity of 70ft/s, the drop time was found to be 2.4 seconds.



5 Detailed Design

5.1 Components and System Architecture

For the final analysis the aircraft design was broken into systems, such as the wing, fuselage, payload, and empennage assemblies. The wing and empennage were both optimized to meet the mission specifications through airfoil selection, sizing, and numerical computation methods. The specific numerical model used to analyze the performance of the wing design was the lifting line model, mentioned above. The fuselage was designed for an optimization of internal capacity for the payload system while reducing the wetted surface area to provide more favorable drag performance. The final payload system design focused on optimizing uniform and rapid water deployment. Likewise, the tank shape was also designed to efficiently fill the internal space of the fuselage. The fuselage and payload system, though separate systems, where integral to each other's final geometry.

5.2 Materials

The materials chosen for the final design of the aircraft were almost entirely composite. The wings and empennage employ a white foam core with the composite sheeting consisting of a single layer of 5.5 oz carbon fiber and a sacrificing layer of 1.4 oz fiberglass. The fuselage shell is made from the exact same composite structure as the wings with one extra layer of carbon fiber. Additional layers were added to other areas of high stress including the blended tail, where the tail section is mounted, and the nose cone since the motor was to be mounted there. The internal structural supports in the form of multiple bulkheads are made of 1/8 in plywood. The tank, to save weight, only consists of two layers of low density fiberglass and is sealed with epoxy. The reasons for these decisions will be discussed in more detail in the manufacturing section.

5.3 Propulsion

The final configuration of the propulsion system is an Astro Flight 60-sized motor with a direct drive, spinning a 17x14 propeller. The battery pack was chosen to be a 20 cell battery pack consisting of Sanyo 1700 SCR cells. From this combination, a static thrust of 8.4 lbf is expected. The maximum output power is 624 W, which with the drag estimate gives a maximum airspeed of 102 fps. The most efficient cruise current and speed for the propulsion system are 21.3 A and 70 fps, respectively. At cruise conditions, the battery is expected to last for 6.32 minutes and 2.5 minutes at full throttle.

5.4 Wing Design

The wings are joined to the fuselage with a wing-joiner tube. This joiner tube was chosen to be a 1/2 inch hollow carbon fiber rod that extends 9 inches into the wing, but did not extend fully through the fuselage due to space constraints with the water tank. A hollow rod was chosen over a solid one for the weight savings and because the added strength of a solid wing spar was not needed.

5.5 Tail Design

As with the design of the wings, the horizontal and vertical stabilizers were joined to the fuselage using a joiner tube. For the horizontal and vertical, the tube extends six inches into the span and has a diameter of 3/8 inch. By using plug-in-stabs, the size of the tail surfaces could be larger than the shipping box constraints would allow than if they were permanently attached to the fuselage.

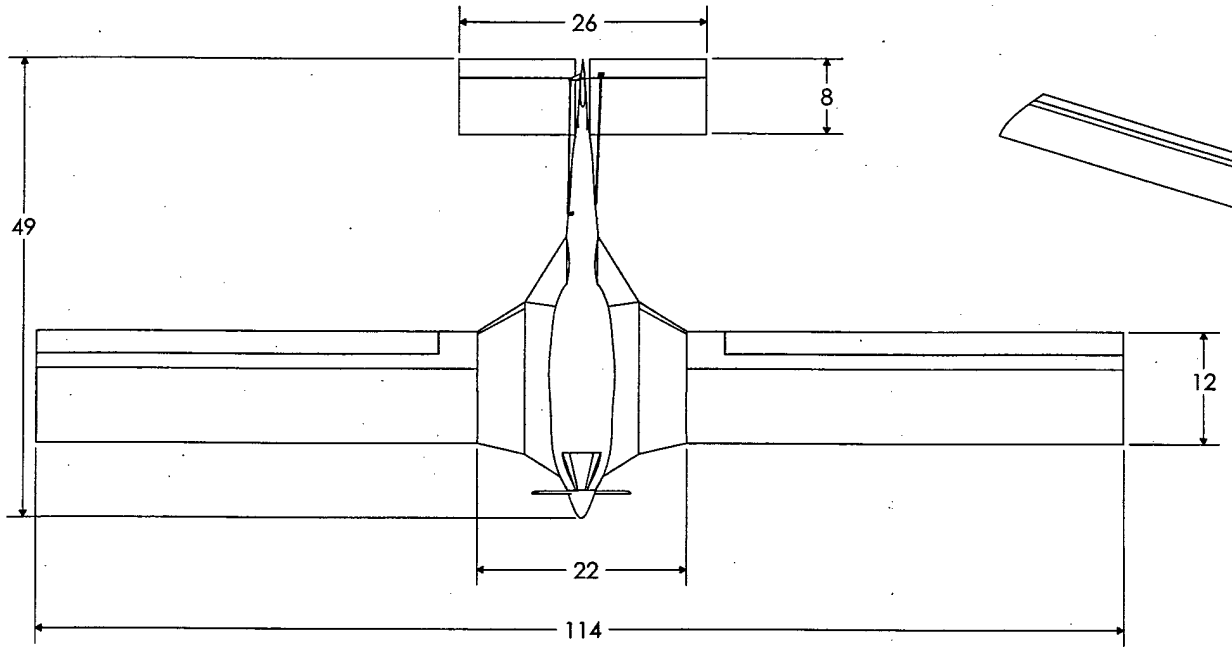
5.6 Aircraft Geometry

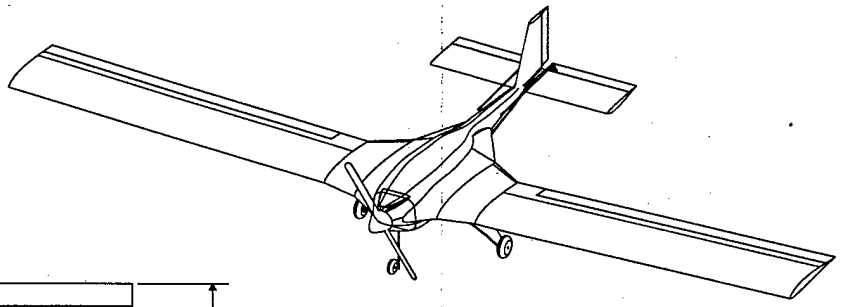
Table 10: Aircraft Geometry and General Performance Characteristics

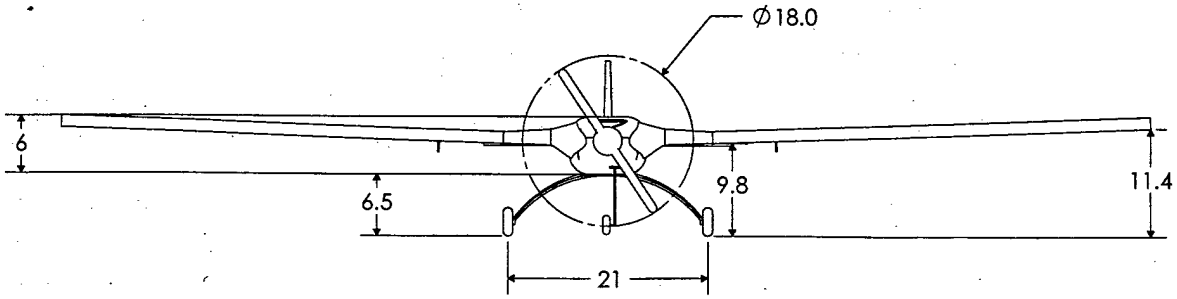
Aircraft Geometry and General Performance			
Dimensions, External:		**Dimensions, Internal:**	
Wing		Water tank width	11 in.
Span	114 in.	Water tank length	9 in.
Chord length (constant)	12 in.	Water tank height	5 in.
Aspect ratio	9.5	Water tank volume	3.97 in.
Dihedral	2.5 deg.	**Positions**	
Length overall	49 in.	Center of gravity	12 in.
Height overall	17 in.	Neutral point	13.85 in.
Fuselage maximum width	22 in.	Wing leading edge	7 in.
Horizontal stabilizer		Wing aerodynamic center	10 in.
Span	26 in.	Horizontal stabilizer leading edge	41 in.
Chord length (constant)	8 in.	Horizontal stabilizer aerodynamic center	43 in.
Aspect ratio	3.75	Vertical stabilizer leading edge	39 in.
Vertical Stabilizer		Payload exit orifice	12 in.
Height	8 in.	**Performance**	
m.a.c.	7.19 in.	Maximum level speed	95.81 ft./sec.
Taper ratio	0.56	Cruising speed	95.8 ft./sec.
Sweep angle	20.56 deg.	Economy cruising speed	61.4 ft./sec.
Landing gear height	6.5 in.	Stall speed at 1400 ft	42.05 ft./sec.
Propeller Diameter	17 in.	Rate of climb at 1400 ft.	
Areas:		Maximum gross weight	945.4 ft./min.
Wing, projected	1368 sq. in. (9.5 sq. ft)	Empty weight	1543.9 ft./min.
Ailerons (total)	112.5 sq. in.	Takeoff distance	139 ft.
Horizontal stabilizer	240 sq. in.	Endurance at maximum speed	2.6 min.
Elevator	52 sq. in.	Endurance at economy speed	5.8 min.
Vertical stabilizer	56 sq. in. (0.389 sq. ft.)		
Rudder	16 sq. in.		

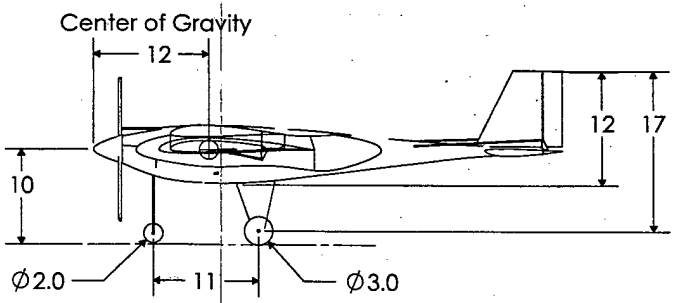
5.7 Drawing Package

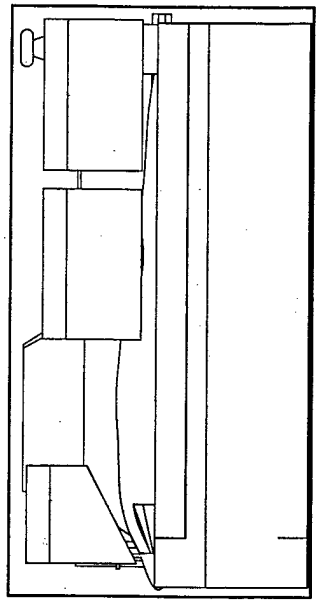
The following drawing package includes selected samples of the technical schematics used for the fabrication of the aircraft. Some of the drawings included are a three-view, a disassembled inside the shipping box and various zoomed in views.

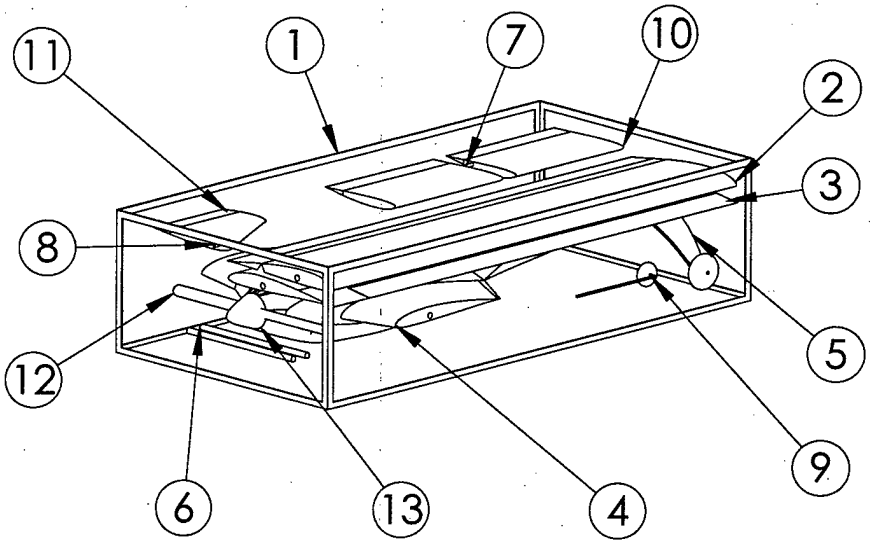

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114

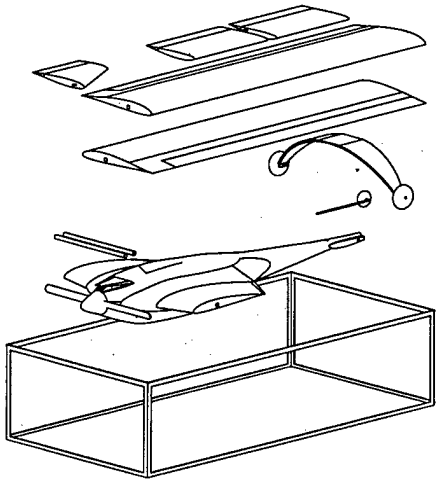


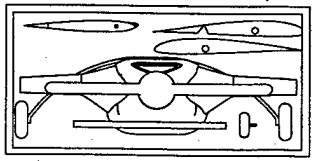

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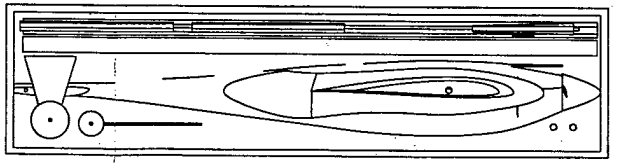

Center of Gravity
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11
Ø3.0



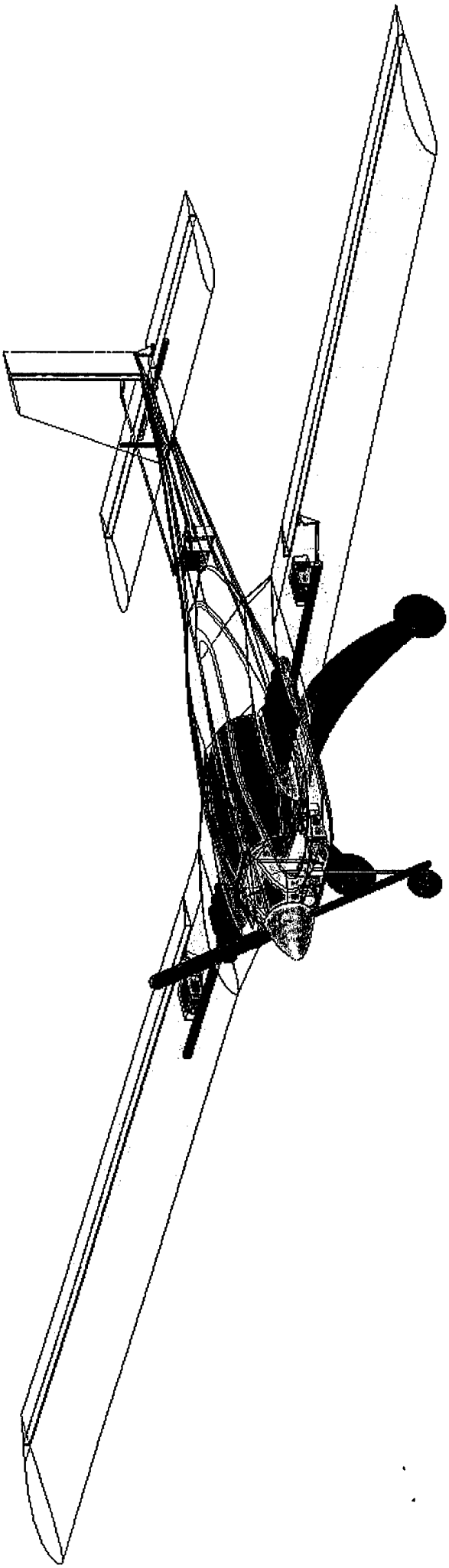


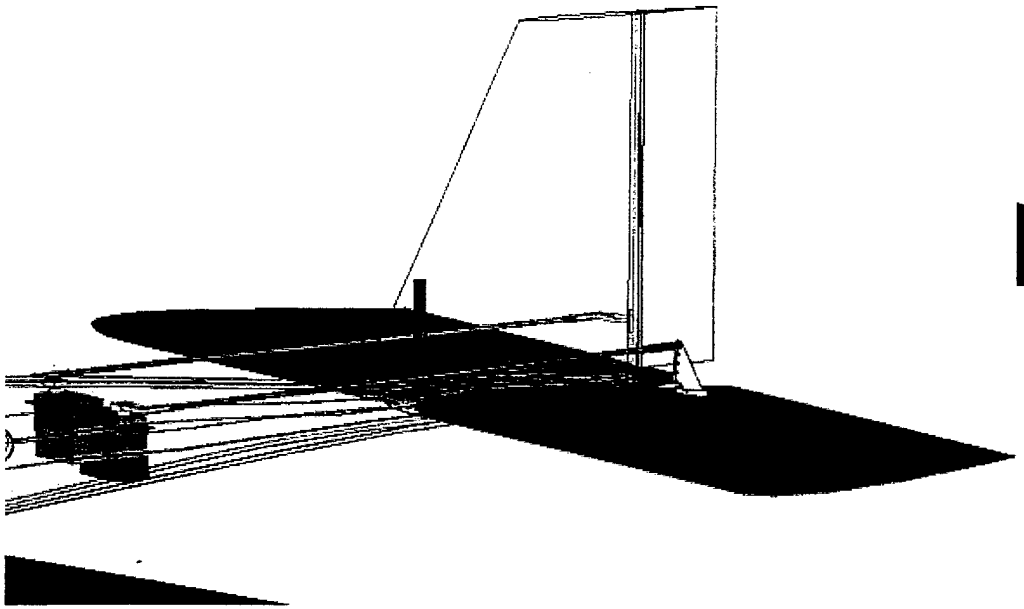


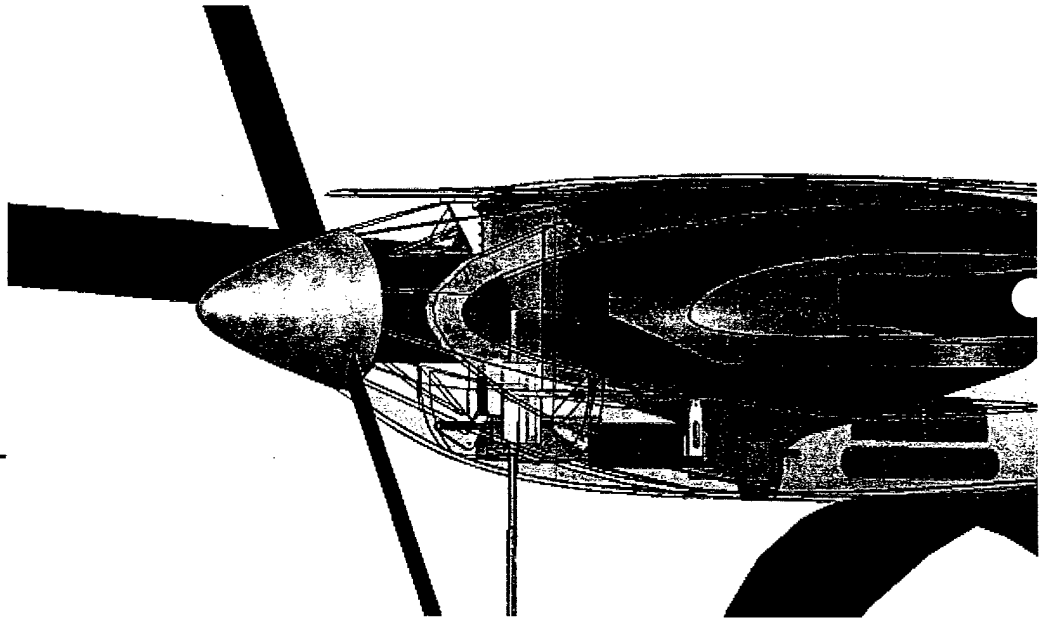


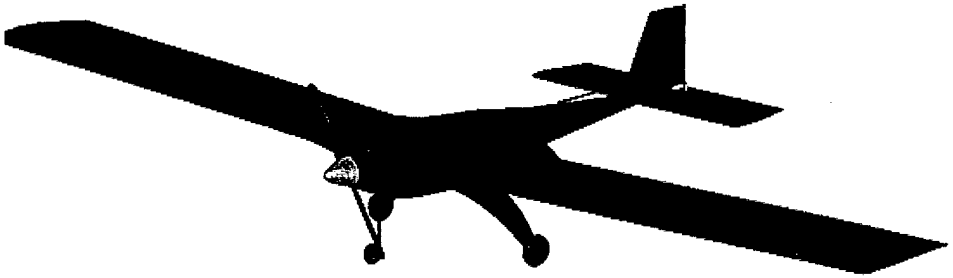


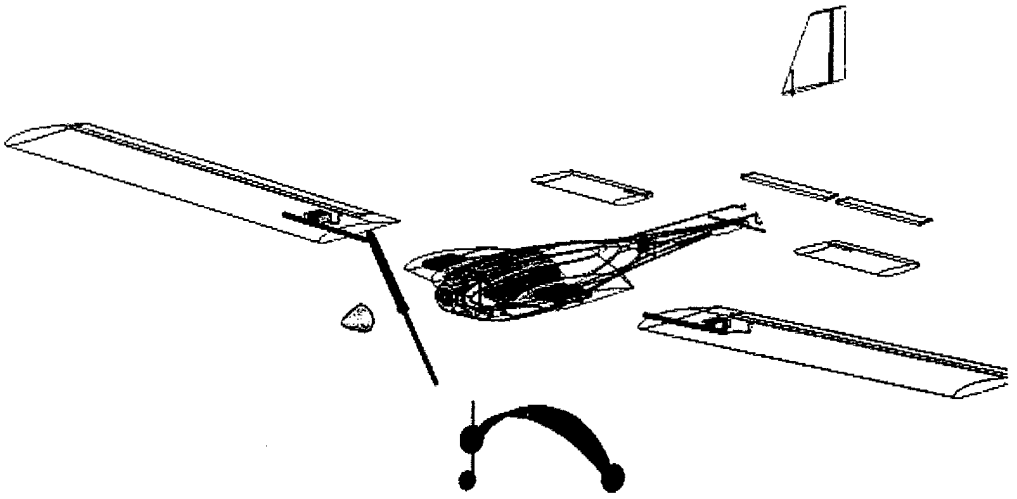
Water Buffaloe Parts	
ITEM NO.	Part
1	Box
2	Left Wing
3	Right Wing
4	New Fuselage
5	Landing Gear
	Landing Gear Frame
	Wheel
6	Wing Joiner Bar
7	Horizontal Joiner
8	Vertical Joiner
9	Nose Gear
	Nose Gear Frame
	Nose Wheel
10	Horizontal Stabilizer Assembly
	Left Horizontal
	Right Horizontal
	Elevator
11	Vertical Stabilizer Assembly
	Rudder
	Vertical
12	Propeller
13	Prop Spinner











5.8 Performance

5.8.1 Lift and Drag

After the dimensions of the wing planform had been finalized an algorithm for a *lifting line model* programmed through MATLAB was employed for final analysis of the wing's lift and drag performance. A detailed description of the lifting line model is given in Chapter 3 in Reference 2. The final results of the analysis are given in Table 11, and the coefficient of lift distribution for zero angle of attack is shown in Figure 14.

Table 11: Aerodynamic Performance Results Based on the Lifting Line Model.

Aerodynamic Performance		
Parameter	**α=0°**	**α=11° (approximate angle of attack for CLmax)**
Coefficient of Lift	0.1818	1.699
Induced Coefficient of Drag	0.00127	0.0482
Oswald's Efficiency Factor	0.982	0.95127

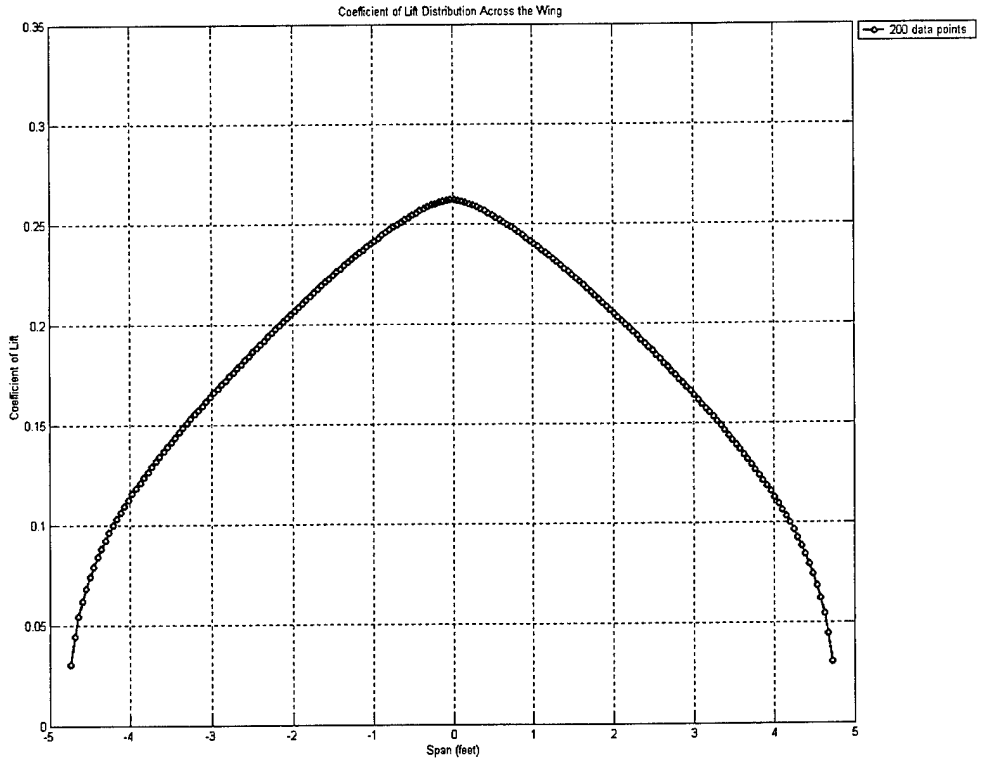


Figure 14: Lift Distribution Coefficient Across the Wing Based on the Lifting Line Model.

From Table ** it is evident that through the airfoils selected and the relatively high aspect ratio of the planform an *Oswald's efficiency* factor of 98.2% was achieved, corresponding to a nearly optimal induced drag performance.

The drag efficiency of the wing substantially reduced the total drag of the aircraft at cruise. The aircraft coefficient of drag was determined to be 0.036. The induced drag for both the wing and the horizontal stabilizer, with coefficients of drag at 0.0011 and 0.00074, respectively, comprised only 5% of the total aircraft drag. The contributions of all components of the aircraft that were considered in the total drag are displayed in Figure 15.

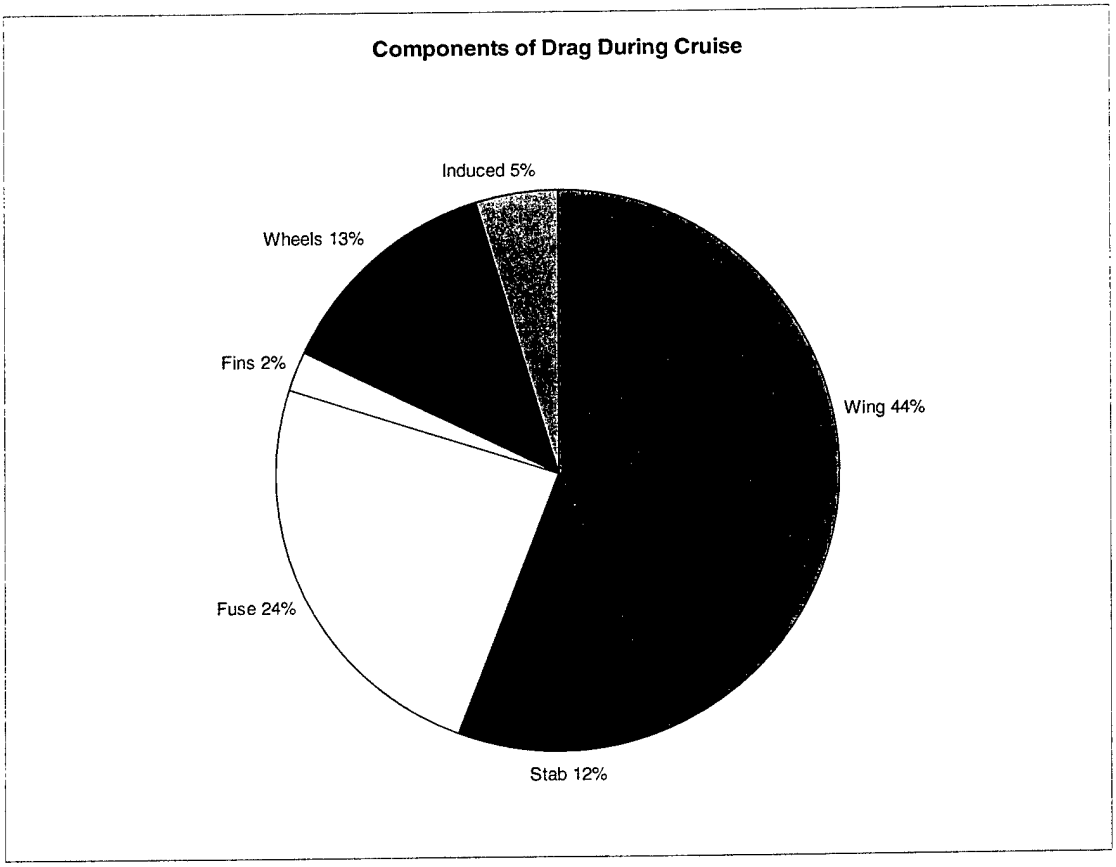


Figure 15: Contribution of all Aircraft Components to the Total Drag During Cruise

During takeoff the induced drag played a much larger role in the total aircraft drag. In takeoff conditions, assuming the maximum coefficient of lift at 1.17 for the wing resulting in an induced drag coefficient of lift of 0.048, the contribution made to the total aircraft drag increased substantially to 32% of a total 0.036, shown in Figure 16. However, a wing of lesser efficiency would have resulted in a much larger aircraft coefficient of drag during takeoff, which in turn would have increased the takeoff distance. The efficiency of the wing permitted a smaller wing area to be used and still meet the takeoff distance limitations of 150 feet.

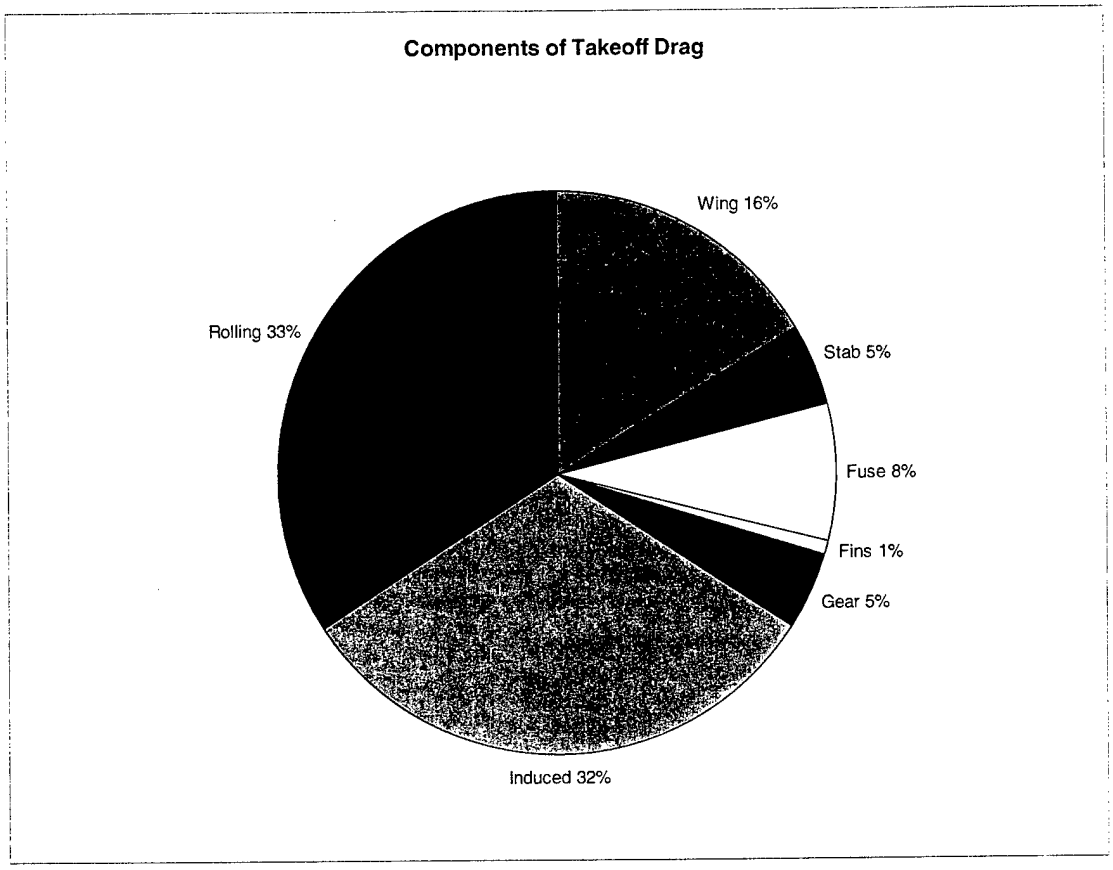


Figure 16: Contribution of all Aircraft Components to the Total Drag During Takeoff

The efficiency of the wing was also quite beneficial to the lift-to-drag ratio. The L/D_{max} for the aircraft was calculated as 14.43, using Eq below from Reference 1.

$$\left(\frac{L}{D}\right)_{\max} = \frac{b\sqrt{\pi e}}{2\sqrt{f}}. \tag{28}$$

Here, *b* is the wingspan in feet, *e* is the Oswald's efficiency factor, and *f* is the equivalent flat plate area, which was determined to be 0.327 sq. feet in conjunction with the plot displayed in Figure 17. At this lift-to-drag ratio, the velocity at which the aircraft would be traveling is 45.7 fps, determined from Figure 17.

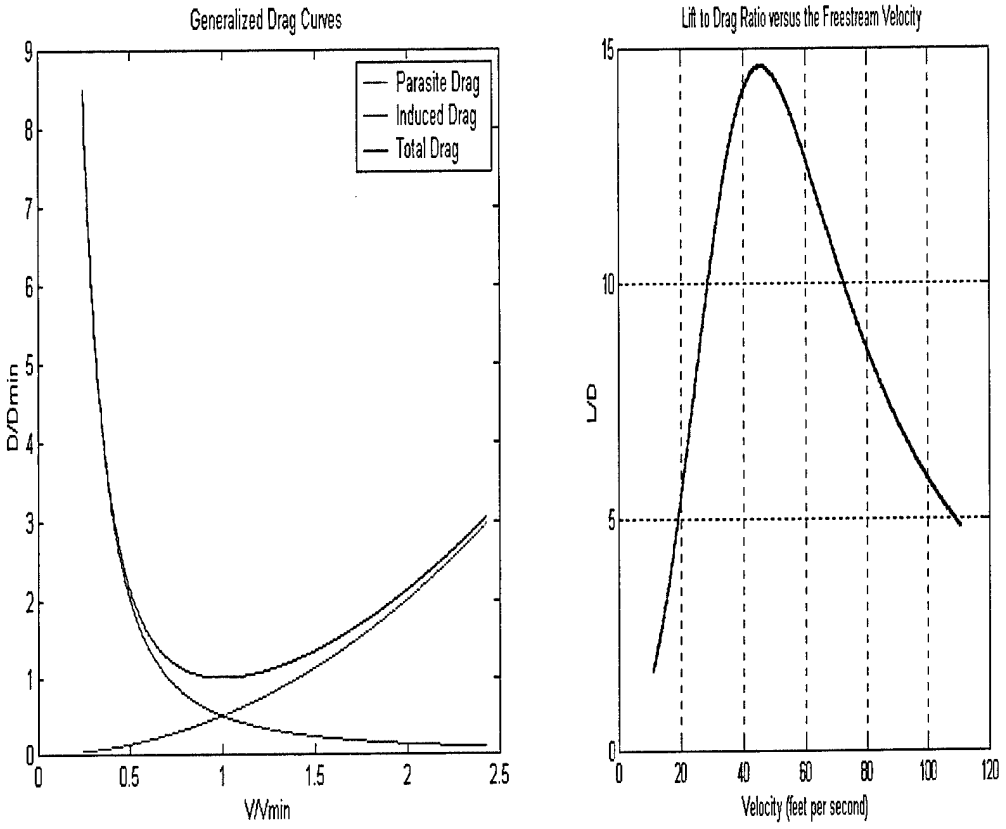


Figure 17: Generalized Drag Polar and Lift-to-Drag Ratio at Various Flight Velocities.

5.8.2 Stability

The aircraft performance was dependent on various aerodynamic coefficients and derivatives. These parameters, such as the coefficients of lift and coefficients of pitching moment, were determined through the methods and equations outlined in Reference 4. The derivatives calculated for the design of this aircraft at cruise conditions are given in Table 12. Some of the derivatives were given a value of 0, mainly for the reason that either those parameters were estimated to be negligible or no intention was made to implement the feature in the design. An example of the latter is any of the derivatives describing the effects of sideslip on the aircraft.



Table 12: Aircraft Derivatives of Flight at Cruise Conditions

Aircraft Derivatives			
Stability Derivatives		**Control Derivatives**	
C_{LoA}	0.1566	C_{Lih} (per deg.)	0
C_{Lowf}	0.182	$C_{L\delta e}$ (per deg.)	0.0071
C_{Loh}	-0.2	$C_{L\delta a}$ (per deg.)	0
$C_{L\alpha A}$ (per deg.)	0.0839	$C_{L\delta r}$ (per deg.)	0
C_{Lawf} (per deg.)	0.0782	C_{Mih} (per deg.)	0
C_{Lah} (per deg.)	0.0456	$C_{M\delta e}$ (per deg.)	-0.0027
C_{DoA}	0.034	$C_{M\delta a}$ (per deg.)	0
C_{Dow}	0.0098	$C_{M\delta r}$ (per deg.)	0
C_{Doh}	0.0077	$C_{\iota ih}$ (per deg.)	0
C_{Dov}	0.0073	$C_{\iota\delta e}$ (per deg.)	0
C_{MoA}	0.0235	$C_{\iota\delta a}$ (per deg.)	-0.0066
C_{Mow}	-0.033	$C_{\iota\delta r}$ (per deg.)	0
C_{Mof}	-0.0096	C_{Nih} (per deg.)	0
C_{Moh}	0.063	$C_{N\delta e}$ (per deg.)	0
$C_{M\alpha A}$ (per deg.)	-0.0087	$C_{N\delta a}$ (per deg.)	0
$C_{M\alpha wf}$	0.0065	$C_{N\delta r}$ (per deg.)	-0.0026
$C_{M\alpha h}$	0.0153		
$C_{\iota oA}$	0		
$C_{\iota\beta A}$ (per deg.)	-0.0575		
C_{NoA}	0		
$C_{n\beta A}$ (per deg.)	0.0076		

5.8.3 Velocity Profile

The stall speed of the aircraft was very important in terms of the minimum speed at which the aircraft could be flown during the water drop phase of the mission. Theoretically the aircraft should not be required to fly at a speed as low as the stall speed to expel all the water in the distance allowed. However, stall speed would also specify the minimum speed for landing. Two conditions were considered when calculating the stall speed of the aircraft. The first was when the aircraft had a full payload of water, which raised the weight of the aircraft from the empty weight of 13.9 lbs. to 22.7 lbs. These weights are inconsistent with the values mentioned the weight analysis; however, the purpose for using these values for the weights is to maintain a conservative estimate for the aircraft performance. With the full payload of water the stall speed is 42.05 fps. The other condition considered was for landing when the aircraft would

no longer be carrying a payload of water. With the aircraft weight at 13.9 lbs. the stall speed is 32.9 fps, which allows for a relatively slow landing speed, beneficial to the pilot.

At the opposite end, the maximum projected speed of the aircraft is 102 fps. This is the result of using a 17/14 propeller, which delivers a coefficient of power 1.11. However, this is attainable only at maximum power, which is obviously not the most efficient setting in terms of the available energy. Figure 18 shows a plot of efficiency versus throttle setting. From this, it is evident that the most efficient throttle setting is significantly less than 100%, but the efficiency does not drop off much from optimum for higher throttle settings. This allows for the aircraft to spend much of the mission at full speed without major concern with available energy.

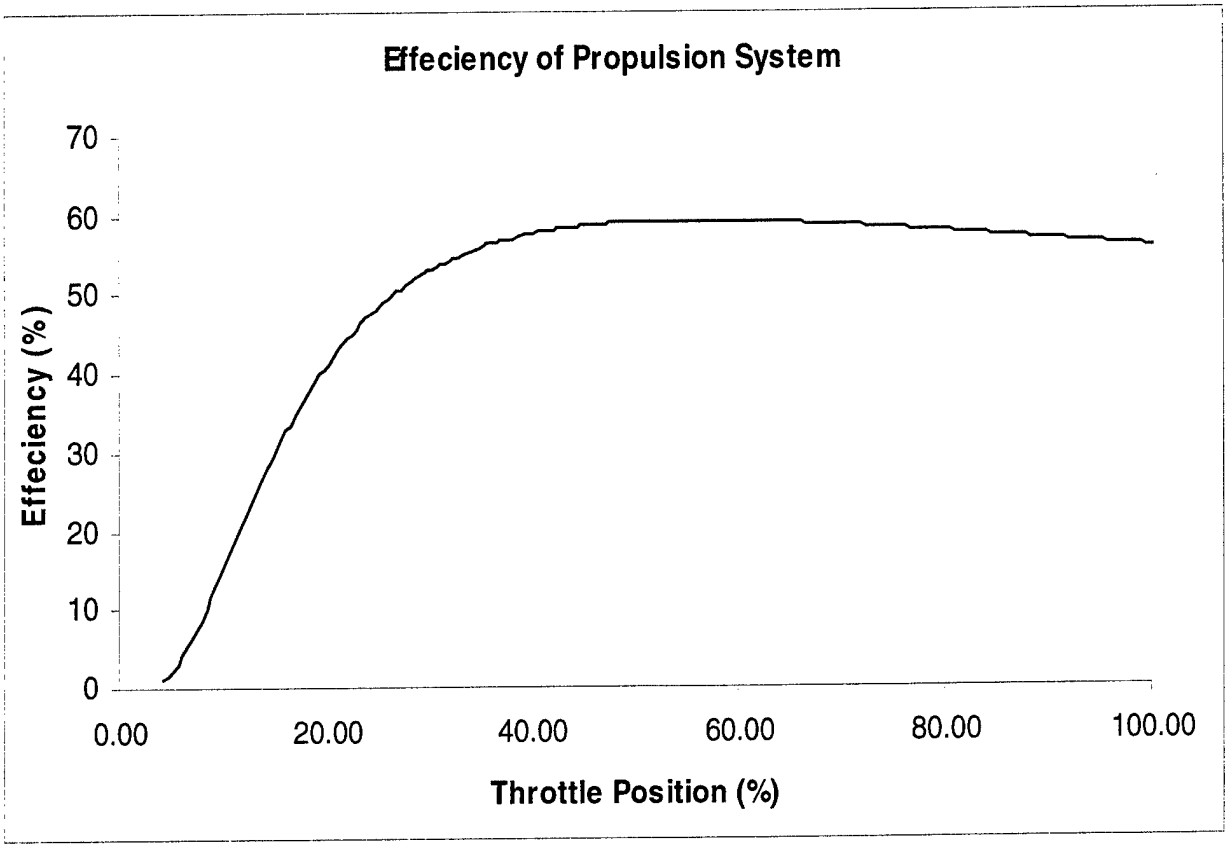


Figure 18: Power Efficiency vs. Throttle Position.

5.8.4 Rate of Climb

Eq. 29 was used to determine the climb rate for a propeller driven aircraft, (Ref 1). P_a is the available power, P_r is the required power, and W is the weight. Since the weight of the plane varies significantly between the Ferry mission and the Fire Bomber mission, the climb rate will differ for each mission as well.

$$R/C = \frac{P_a - P_r}{W} \tag{29}$$

The results of this equation are shown below in Table 13



Table 13: Climb Rate for Each Mission.

Climb Rates		
Mission	Fire-Bomber	Ferry
Maximum Power	624 W	624 W
Power Required for Cruise	139 W	139 W
Weight	22.7 lb.	13.9 lb.
Climb Rate	945.4 fpm	1543.9 fpm

A few assumptions had to be made in order for Eq. 29 to be valid. First, is that the required power is at cruise conditions. This value was chosen instead of the power required at take-off because it represents the worst-case scenario. In reality, excess power decreases with increasing airspeed, thus the required power will vary throughout the climb phase. The next assumption is that the aircraft will not climb to an altitude high enough to affect the propeller's effectiveness. At higher altitudes the density decreases, which lowers the maximum thrust available.

5.8.5 Take-off Field Length

To predict takeoff distance, Eq. 30 was numerically integrated to give velocity as a function of position along the runway. In order to ensure that the design will meet the takeoff requirements, the velocity at 150 ft must be greater than the required takeoff speed.

$$v(x) = \int_0^x \left(T - D - \mu_g (W - L)\right) ds \tag{30}$$

In Eq. 30, v is the velocity as a function of position, x is the position along the runway, T is the available thrust, D is the aerodynamic drag, and μ_g is the ground rolling friction taken 0.25 (REF 4). For the ferry mission the takeoff distance was predicted to be 53.6 ft. For the fire bomber mission, with an additional 8.8 lbs of weight at takeoff, the predicted distance is 138.2 ft. The final design of the aircraft was chosen because it gave the highest competition score with a safety margin of 10 ft for the takeoff distance.

5.9 Weight Statement

The weight estimate was determined by weighing each component individually. Certain components that could not be weighed, such as the fuselage, which had not been constructed yet, were estimated by weighing a sample piece of the material. The sample was measured giving a weight per unit area value. This value was multiplied by the wetted area to calculate a total weight. Other miscellaneous items, such as extra epoxy and plywood, were added based upon prior RC aircraft experience. It was important to determine the weight of each component in order to accurately place the center of gravity of the aircraft. This was especially critical to the design since the water tank had to be centered at that exact location. The total empty weight resulted in 12.686 lb, which was similar to the preliminary estimate of 13.9 lb. The results for the empty weight and gross weight are summarized in Tables 14 and 15, respectively.

Table 14: Weight Estimate for the Aircraft Without the Payload.

Components	Weight (oz/unit)	Quantity	Total Weight (oz)
Airframe			
Fuselage Carbon Fiber	0.01144	1430.4 in^2	20.592
Wing Foam Core & Ribs	0.01349	1150 in^2	15.521
Wing and Blended Section Carbon Fiber	0.01144	2700 in^2	30.892
Vertical & Horizontal Tail Foam Core & Ribs	0.01349	296 in^2	3.993
Vertical & Horizontal Tail Carbon Fiber	0.01144	296 in^2	3.386
Wing Joiner	0.171	28 in	4.787
Vertical & Horizontal Joiner	0.171	14 in	2.394
Landing Gear	10.56	1 gear	10.559
Nose Wheel	1.316	1 wheel	1.316
Tank	8	1 tank	8.000
Bulkheads	4	N/A	8.000
Motor Mount	2	1 mount	2.000
Electronics			
Receiver Battery	3.18	1 battery	3.180
Receiver	1.453	1 receiver	1.453
Servos	0.788	6 servos	4.730
Battery Cells	2.0106	20 cells	40.212
Propulsion System			
Spinner	2	1 spinner	2.000
17" Propeller	0.237	1 prop	0.237
Motor	25.397	1 motor	25.397
Push Rods	0.03451	48.5 in	1.674
Control System			
Nose Gear Bearing	0.17	1 bearing	0.170
Control Horns	0.0959	5 horns	0.480
Miscellaneous			
Miscellaneous plywood/balsa	8	N/A	8.000
Miscellaneous epoxy	4	N/A	4.000
Total Empty Weight in ounces			202.972
Total Empty Weight in pounds			12.686

Table 15: The Weight Estimate for the Aircraft With the Payload.

Components	Weight per unit	Quantity	Total Weight (ounces)
4 Liters of Water	35.2 oz/liter	4	140.8
Total Gross Weight in ounces			343.772
Total Gross Weight in pounds			21.486



5.10 Rated Aircraft Cost

The RAC coefficients, as specified by the competition rules, were presented in the conceptual design phase. With a final aircraft design, a detailed study of the RAC was performed. Table 16 shows the WBS parameter for the MFHR. From the table, it is evident that the wing area and fuselage volume were the major contributors to the total manufacturing man-hours of 207.5.

Table 16: Manufacturing Work Breakdown Aircraft Values

Manufacturing Man Hours: Work Breakdown Schedule				
Aircraft Component		**Man Hours per unit**	**Aircraft Parameter**	**Individual Hours**
Wing	Area	10 hr/sq. ft.	9.5	95
	Control Surface	5 hr*type multiplier	1	5
Fuselage	Fuselage Volume	20 hr/cubic ft.	2.88	57.5
Empennage	Vertical Surface (no active control)	5 hr per vertical surface	0	0
	Vertical Surface (with active control)	10 hr per vertical surface	1	10
	Horizontal Surface	10 hr per horizontal surface	1	10
Flight Systems	Servo or Control Motor	5 hr per servo or motor control	6	30

Table 17: Detailed RAC

Rated Aircraft Cost (RAC) Breakdown						
Coef.	**Description**	**Definition**	**Cost Multiplier**	**Aircraft Parameter**	**Man Hours**	**RAC**
A	Manufacturers Empty Weight (MEW)	Actual airframe weight, lb., with all flight and propulsion batteries, but without any payload.	$300	12.7	-----	$3.81
B	Rated Engine Power Multiplier (REP)	Propulsion battery weight times a function of the number of motors.	$1,500	2.6	-----	$3.84
C	Manufactureing Man Hours (MFHR)	Prescribed assembly hours by work breakdown structure (WBS)	$20	-----	207.5	$4.15

Table 4 shows the detailed RAC for the final design. With a MFHR given above, a REP of 2.6, and a MEW of 12.7, the final RAC was found to be $11.80. Figure 19 illustrates a component level view of the

RAC. In the final RAC, the aircraft empty weight and propulsion system are the dominant parameters, not the wing area and fuselage volume as shown in the WBS.

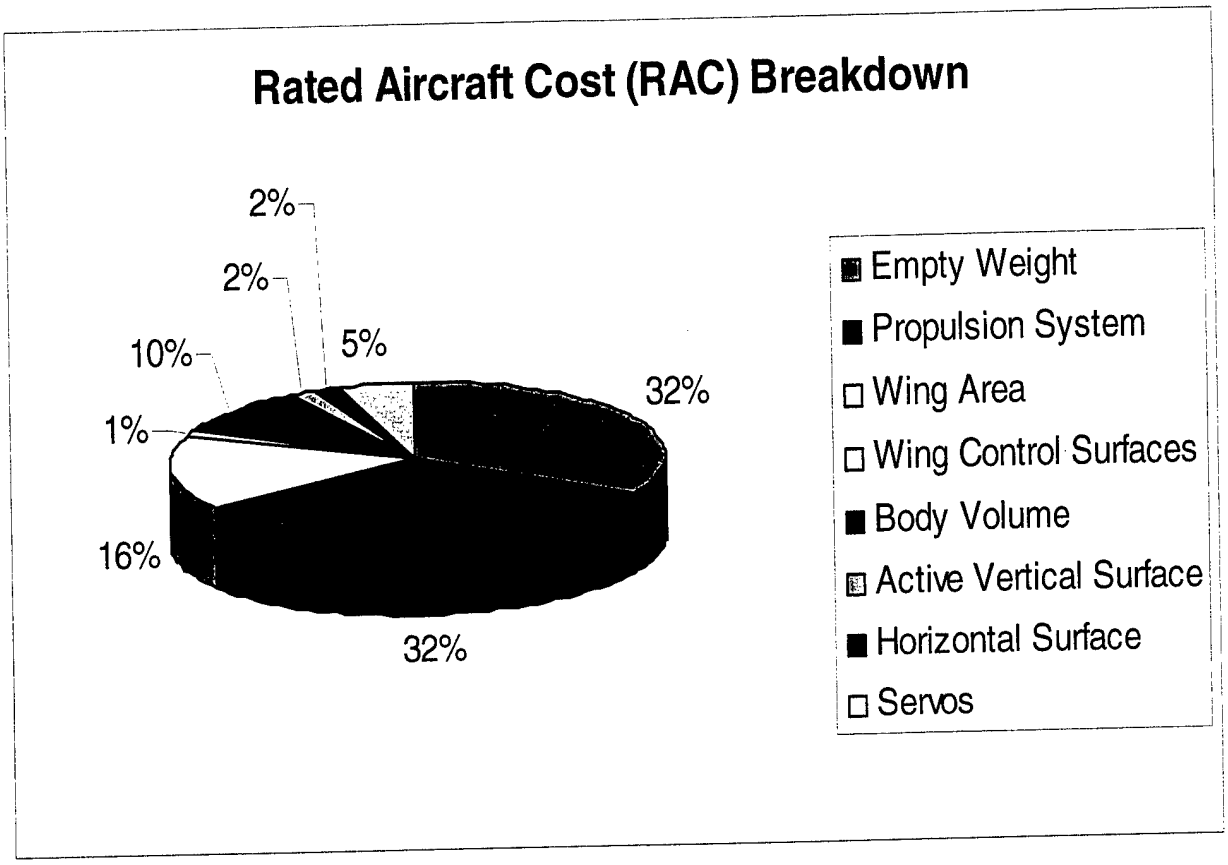


Figure 19: RAC Pie Chart

Table 18: RAC Sensitivity to System Change.

RAC Sensitivity to System Change	
System	Slope
Flap/Aileron Change	0.1
V-Tail Change	-0.1
Servo Change	0.1
Empty Weight	0.3
Battery Weight	1.5
Fuselage Weight	0.4
wing Area	0.2

6 Manufacturing

As composite materials were selected for both the fuselage and wing construction because of their strength to weight ratio, the manufacturing processes will focus on composite manufacturing, which includes fiberglass, carbon fiber, and low (white) to medium (pink) density foam. Manufacturing processes



were selected on the basis of repeatability, practicality, and cost. The primary components fabricated using a foam core include the wings, empennage, and control surfaces. For the fuselage, master molds are used to produce an accurate carbon fiber shell. The manufacturing of the water tank also uses a master mold process to ensure an accurate volume.

6.1 Figures of Merit

While composite manufacturing was initially selected for its strength-to-weight ratio and rapid turn around time of multiple components, manufacturing figures of merit were established to solidify their extensive use. Numerically figures of merits were assigned to the following categories: availability of materials (AM), cost of materials (CM), required skill level and ease of construction (RSL&EC), turn around time (TAT), strength and reliability (S&R), and estimated final weight of integrated manufactured component (EFW). If the manufacturing technique or material was advantageous in a particular area it was assigned a score of 1.0, neutrally beneficial was assigned a 0.0, and disadvantageous was assigned a -1.0. The categories were assigned weights based on their importance. The manufacturing process or material selected for the final design had the highest total weighted sum. The FOM for manufacturing is shown in Table 19, and each category is described in further detail in the subsequent sections.

Table 19: Manufacturing Figures of Merit

Component	Manufacturing Process	AM	CM	RSL&EC	TAT	S&R	EFW	Total
	Weight of Importance:	15%	10%	15%	20%	25%	15%	100%
Aerodynamic Surfaces	Balsa Structure w/ Balsa Sheeting	1	1	-1	-1	-1	0	-35%
	Foam Core w/ Balsa Sheeting	1	1	0	1	0	1	60%
	Melted Foam Core w/ Composite Sheeting	1	1	-1	1	-1	1	20%
	Foam Core w/ Composite Sheeting	1	1	0	1	1	1	
Fuselage	Balsa Structure w/ Balsa Sheeting	1	1	-1	-1	-1	0	-35%
	Balsa Structure w/ ShrinkWrap Sheeting	1	1	-1	-1	-1	-1	-50%
	Hallow Composite Shell	1	1	-1	-1	1	1	
Water Tank	Slighly Modified Off-Shelf	1	1	1	1	-1	-1	20%
	Shaped Aluminum or Plastic	1	-1	-1	-1	1	-1	-20%
	Hallow Layered Composite	1	1	-1	-1	1	1	
Master Molds	CNC from SolidWorks Drawing	1	1	-1	0	N/A	N/A	10%
	Handshape w/ no Drawing	1	1	-1	-1	N/A	N/A	-10%
	BulkHead Buildup+Handshaping	1	1	0	0	N/A	N/A	

6.1.1 Availability of Materials

Since the manufacturing process was initially placed on a tight timeline of only two-three months, availability of materials (AM) was critical. Therefore, the AM was assigned a positive one if it could be purchased and delivered off the shelf, un-modified within two weeks; assigned a zero if it could be



purchased and slightly modified upon delivery within three weeks; scored a negative one if the material was backordered or purchase-delivery-modifications took longer than one month.

6.1.2 Cost of Materials:

As this year's team obtained a Lockheed Martin sponsorship, cost of material (CM) was not heavily weighted in the selection of materials or manufacturing processes. Thus, most materials were ranked as either a positive one or zero if the product could be purchased off the shelf; the exception being the shaped aluminum or plastic water tank that would have required custom fabrication from outside sources.

6.1.3 Required Skill Level of Manufacturing & Ease of Construction

If a manufacturing process could be completed with limited prior experience the RSL&EC was assigned a positive one. A moderately skilled task, such as cutting foam wing cores or laying up composite material was assigned a zero. Finally, high skilled tasks that could only be accomplished by Level-2 or Level-3 machinist, such as CNC programming, were assigned a negative one.

6.1.4 Turn Around Time

Due to the possibility of incurring damage a short turn around time was desired for all aircraft components. Thus, if a manufacturing process took less than one month or could produce multiple components from one mold, the TAT was ranked as a positive one. If the process took two-three weeks but could only produce a one of a kind component it was rated as a zero. A score of negative one was assigned if the manufacturing process exceeded one month and could not produce multiple components.

6.1.5 Strength and Reliability

Final material strength and reliability was also a major consideration in selecting all processes. This lesson was learned from the 2002/2003 team's structural failure with a balsa wing. If a manufacturing process could produce a relatively strong, repeatable, and reliable final product it was rated as a positive one. If the process could be repeated reliably but did not meet the necessary strength requirements the procedure was assigned a zero. Finally, if the process could not produce the necessary strength it was rated a negative one.

6.1.6 Estimated Final Integrated Weight

The final weight of each component integrated into the design is critical and should be kept to a minimum. Manufacturing process that produce materials with a high strength-to-weight ratio were ranked as a positive one for the EFW. Methods with a one-to-one strength-to-weight ratio were given a zero, and lower than one-to-one strength-to-weight materials were given a negative one.

6.2 Aerodynamic Surfaces

The 2002/2003 University of Colorado DBF plane was built entirely from a build-up manufacturing process using balsa wood as the primary building material. However, lessons learned from this experience and generated FOM's, directed the current construction towards composites for the following reasons. Balsa construction makes more complex wing designs difficult to fabricate. Also, the monokote used for the wings' surface sags between the support ribs, altering the aerodynamic performance. Composite wings, on the other hand, do not suffer from this problem. In addition, composites have a higher strength to weight ratio as well as a faster fabrication time.

One way to further reduce the weight of the wings was to melt the foam core, creating a hollow wing structure. As can be seen in Figure 20, the process of melting the high-density blue foam out of a prototype water tank failed. This is because not all of the foam was completely removed, which left most of the foam as residue and weight on the walls. Therefore, composite wing manufacturing with foam cores remaining inside was selected. There are benefits to leaving the foam cores in place, such as producing a stronger wing.



Figure 20: Preliminary Tank Manufacturing With Foam Core Melted Out

The wing manufacturing process utilized a hot wire cutter to produce the inner foam mold. Plywood laser cut templates for the root and tip airfoils were used as cutting guides, providing a lofted wing. The lay-up process for the wings was quite simple, consisting of a single layer of carbon fiber to carry the load and fiberglass to provide a smooth finish. To increase the wing's strength and finish, the composite wings were vacuum bagged. Mylar strips were used during the bagging process to provide the final finish on the wing surface. The pressure inside the bag was maintained at a constant six pounds of mercury, for the low-density foam, for no less than 18 hours. To complete the wing after the bagging process, ailerons and servo hatches were cut out of the wing and it was trimmed to final dimensions.

6.3 Tank Manufacturing

The tank manufacturing process was similar to that of the fuselage construction, where a master mold was used to produce the fiberglass tank. The reason for this choice is due to failures in the first prototype lay-up, which used the "lost-foam" method. The most significant failures were that the tank did not maintain its proper shape, as shown in Figure 20 after the foam had been melted out, and that the volume capacity was not to the desired tolerances of 4 liters. By utilizing a master mold, the water tank can be made to hold a more accurate volume of water.

6.4 Fuselage & Master Molds

The fuselage will be manufactured by joining top and bottom pieces, which are produced from female molds. The male molds, also known as the master molds, were hand sanded to the fuselage shape out of tooling foam using cross-sectional templates obtained from the Solid Works model. The tooling foam was painted to provide a sealed, smooth surface to make the female molds. The final piece is produced by layering the desired amount of composite material into the female mold. For the fuselage two layers of carbon fiber are used for strength and stiffness and a single outer layer of fiberglass is used for a smooth finish. The mold and composites are then vacuum bagged to maintain the desired shape as the epoxy cures. Figure 21 shows an example of this process with the male (far right) and female (center) molds used to produce the University of Colorado's Raven aircraft (far left).

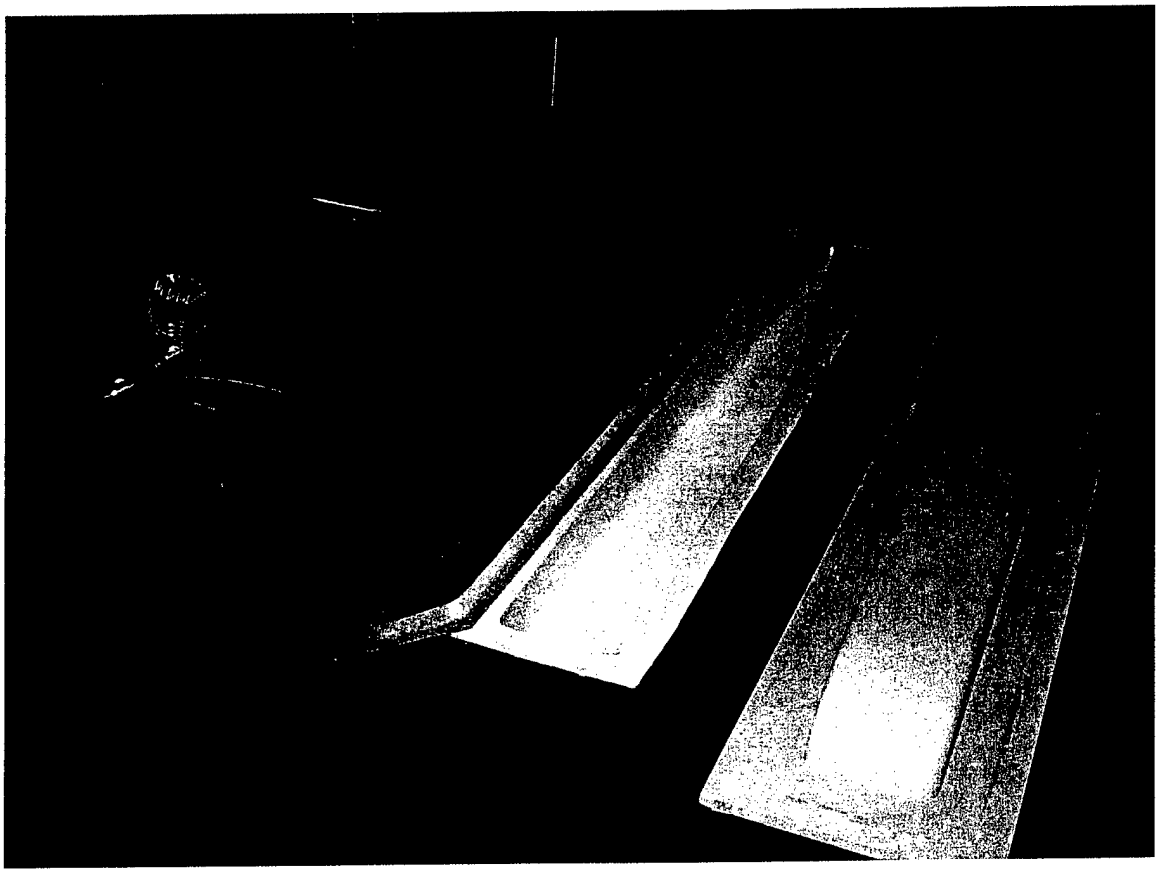


Figure 21: Raven Aircraft Produced at the University of Colorado With Male and Female Molds, Along With the Finished Fuselage Shell



7 Testing

Various tests were performed throughout the design and fabrication phases of the project to verify analytical models, manufacturing methods, and optimize aircraft performance. The main components that were tested include the water tank and refilling system, the wings, the motor, and the batteries. A flight test will be performed once construction is complete.

7.1 Objectives and Schedule

Table 20: Testing Objectives

Testing Objectives	
Test	Objective
Tank Water Drop Test #1	To provide an estimate of how long it would take to drop 4 L of water, and which tank design was most efficient.
Refilling Test	To determine how to minimize the time required to refill the tank using tow separate two-liter bottles.
Wing Strength vs. Weight Test	To determine which materials would provide the lightest wing while being able to sustain 8 to nine g's.
Static Thrust Test	To verify the analytical static thrust predictions using an inline load cell.
Battery Test	To see what was the least number of cells capable of providing enough power for a three minute flight.
Thermal Tests	To determine how much heat affected the motor and batteries' performance in order to decide whether an air scoop would be beneficial
Tank Water Drop Test #2	To obtain a better idea of how long the water drop would take in order to finalize the competition score estimate.
Flight Test	to verify that the aircraft performed as predicted through analytical models, and make adjustments if necessary to improve any design flaws.

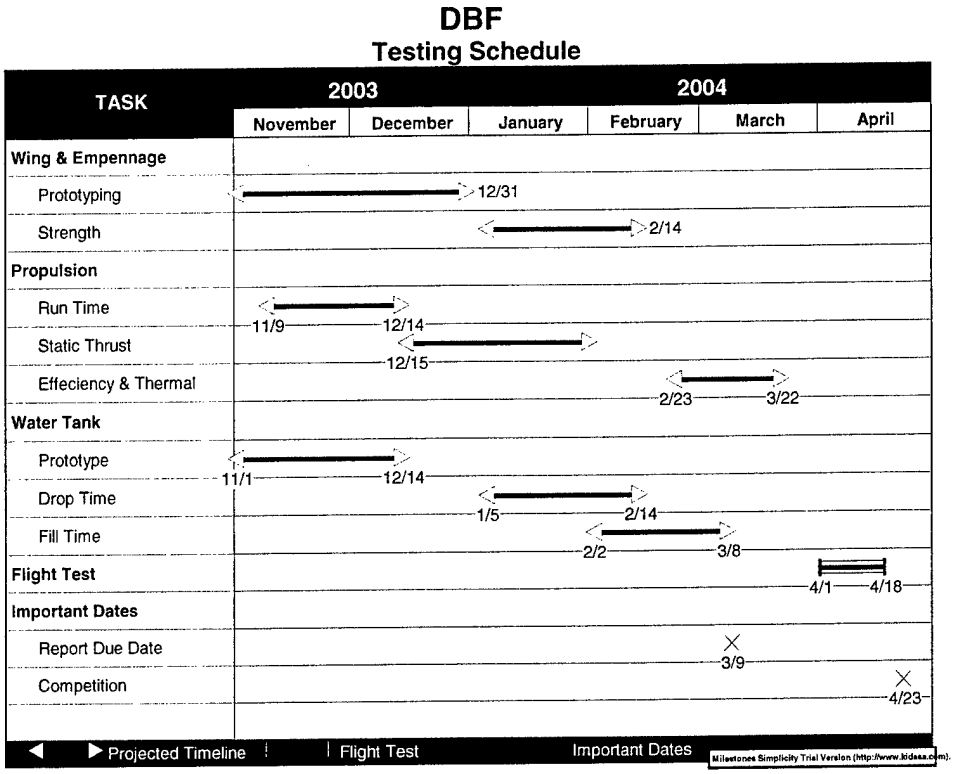


Figure 22: Testing Schedule

7.2 Results

7.2.1 Tank Water Drop Test #1

The purpose of this test was to get a physical sense of how long it would take to drain 4 liters of water out of a tank under atmospheric pressure through a half-inch orifice. Another objective was to obtain the drain time for a worst-case scenario. A half-inch hole was drilled through the bottom of a plastic bucket and then the hole was plugged. Four liters of water were poured into the bucket, the plug was released, and the drain time was record. This was done three times in three different configurations shown in Figure 23 a and b; upright with a one-inch inlet hole through cover, upright with no cover, and tilted sideways 45° with no cover. The results are shown below in Table 21.

Table 21: Drop Time Results for Flat Bottom Tank

Upright	1" Inlet
Drop #	Time (s)
1	53
2	52
3	51

Upright	Open to atm
Drop #	Time (s)
1	51
2	51
3	51

45 deg tilt	1" Inlet
Drop #	Time (s)
1	41
2	42
3	41

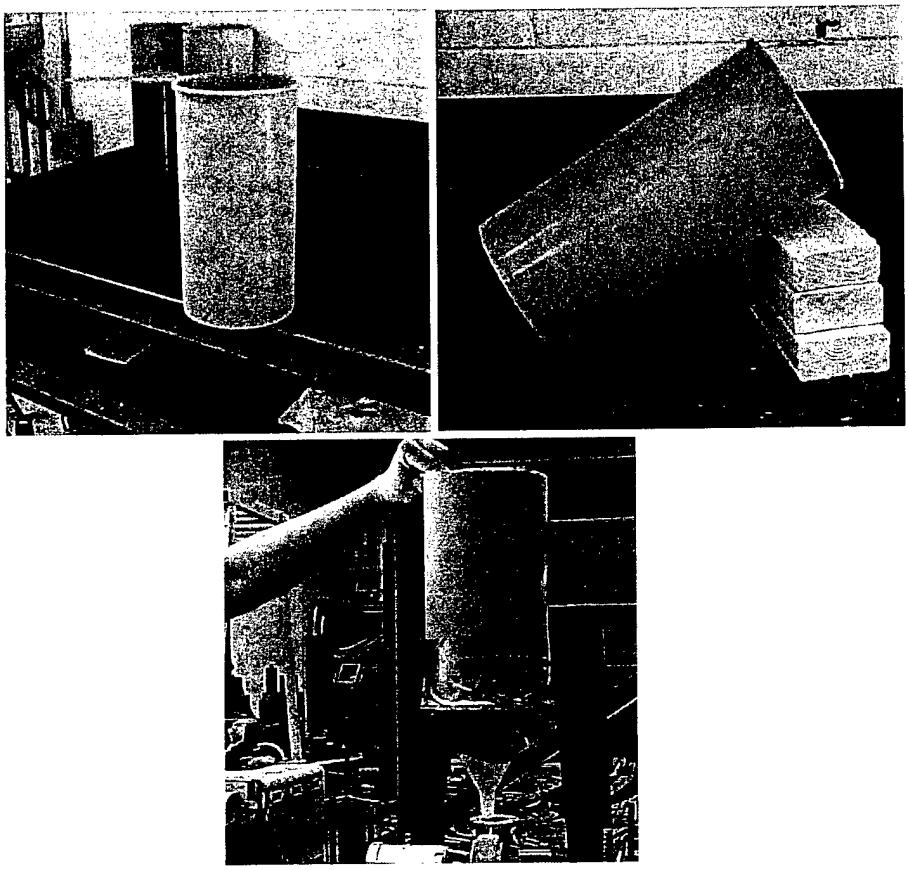


Figure 23: Preliminary Water Drop Tests. Starting from the Top Left, (a) Flat-Bottomed Tank, (b) Flat-Bottomed Tank Tilted 45 Degrees, (c) Tank With a Funnel

In an effort to further reduce the drain time, the next test utilized a bucket with a tapered exit orifice. Figure 23 c shows the tank used for this experiment. Four liters of water were placed in the tank and the drain time was recorded five times. It was noticed during this experiment that a vortex formed at the exit orifice about half way through the drain. This increased the drop time since the cross-sectional area of the exit hole was reduced due to the air flowing through the center of the funnel. An attempt was made to stop this vortex from forming by inserting a 0.635mil thick bar down into the taper. This attempt was successful and the drop time was reduced by three seconds on average as seen in Tables 22 a and b.

Table 22: Drop Time Results with Funnel

Upright 1" Inlet	
Drop #	Time (s)
1	53
2	52
3	51

Upright Open to atm	
Drop #	Time (s)
1	51
2	51
3	51

These results proved that having a tapered exit orifice was much more effective than having a flat-bottomed tank design and vortex control or an asymmetrical exit should be implemented to prevent a



vortex from forming. The drain time estimates were beneficial because they were used to estimate the airplane's minimum cruise velocity, using the drop distance and time.

7.2.2 Refilling Test

An experiment was performed to determine the drain time of a 2-liter bottle and estimate refilling time. With no air vent, the drain time was very slow and the flow exhibited 'chugging' behavior. To prevent this, three different sized hoses were tested as an air vent by insertion into the exit orifice. The drain time of each size hose was recorded, and the results are shown in Table 23a-c. The first three drop times averaged 17s, which seemed long. It was noted that air was not flowing well through the hose due to water getting into the line. For the next runs, the vent line was plugged at the outer orifice so water could not get into the line before draining.

Table 23a-c: 2-Liter Drain Results

a

Hose 1 I.D. = 0.25" O.D. = 0.375"	
Run	Time (s)
1	6
2	6
3	7

b

Hose 2 I.D. = 0.2" O.D. = 0.33"	
Run	Time (s)
1	8
2	6
3	7

c

Hose 3 I.D. = 0.33" O.D. = 0.425"	
Run	Time (s)
1	8
2	8
3	7

During the drop using hose #2, the 2-liter showed signs of compression due to suction. This was due to a lack of airflow from the vent line. From this and the drain times, it was concluded that a 0.25" inner diameter hose could be used to minimize the refill time so long as the outer diameter was as small as possible for the given inner diameter.

7.2.3 Wing Strength-to-Weight Test

When deciding upon which materials to use for the wing construction, the main goal was to have the highest strength to weight ratio. A composite wing made from carbon fiber, fiberglass, and a foam core was chosen due to previous experience in UAV wing fabrication at the University of Colorado. Tests were performed to see which combination of foam (low-density or medium-density) and fabrics (how many layers of carbon fiber and fiberglass) were the lightest while still providing the necessary strength. The apparatus setup was described in Section 4.3.1.6 of the report.

7.2.3.1 Wing #1 & Wing #2

Wing #1 Test Parameters: White foam, (1) 5.5 $^{oz}/_{yard}$ layer carbon fiber, (1) 1.4 $^{oz}/_{yard}$ layer fiberglass, hollow aluminum wing joiner. Wing #2 Test Parameters: Pink foam, (1) 5.5 $^{oz}/_{yard}$ layer carbon fiber, (1) 1.4 $^{oz}/_{yard}$ layer fiberglass, hollow aluminum wing joiner. For Wing #1 and Wing #2 the rib only comprised 1/3 of the chord length. Wing #1 failed at the rib after a loading of 30lbs, see top left of Figure 24. For Wing # 2, the hollow aluminum wing joiner was not strong enough so it started to bend as the weight was increased. This caused a shearing force, which cracked the plywood at the wing's root after a load of 50lbs. The lessons learned from the first test was to use ribs that extended across the entire chord length

to better distribute the load through the skin, and lower the vacuum pressure when using the white foam. It was noted from the second test that a stronger wing joiner is needed.

Wing #3 Test Parameters: White foam, (1) 5.5 $^{oz}/_{yard}$ layer carbon fiber, (1) 1.4 $^{oz}/_{yard}$ layer fiberglass, solid graphite wing joiner. While this wing utilized the same materials as the first wing, it was able to sustain 65 lb, 35 lb more than in the previous test. This proved that the stronger graphite wing joiner and full chord length rib were beneficial to the overall strength of the wing. Another thing that was noted during this test was that as the load increased, the induced torque twisted the leading edge down. This was important because the angle of attack will be reduced at high lift situations, thus decreasing the likelihood of stall. A lesson learned from this test is that additional buckling strength is needed in the area near the rib, see lower right of Figure 24. An extra layer of carbon fiber will be placed over the inner, top section of the wing for future tests. It was noticed that when the last weight was put on the test it vibrated the wing, causing more force to be put on the wing than would have been. This was probably what made the wing fail at the rib. This corresponds to the results of the layered stress analysis, which said that it might fail at the rib if the safety factor of 2 was exceeded, which it may have been with the vibration.

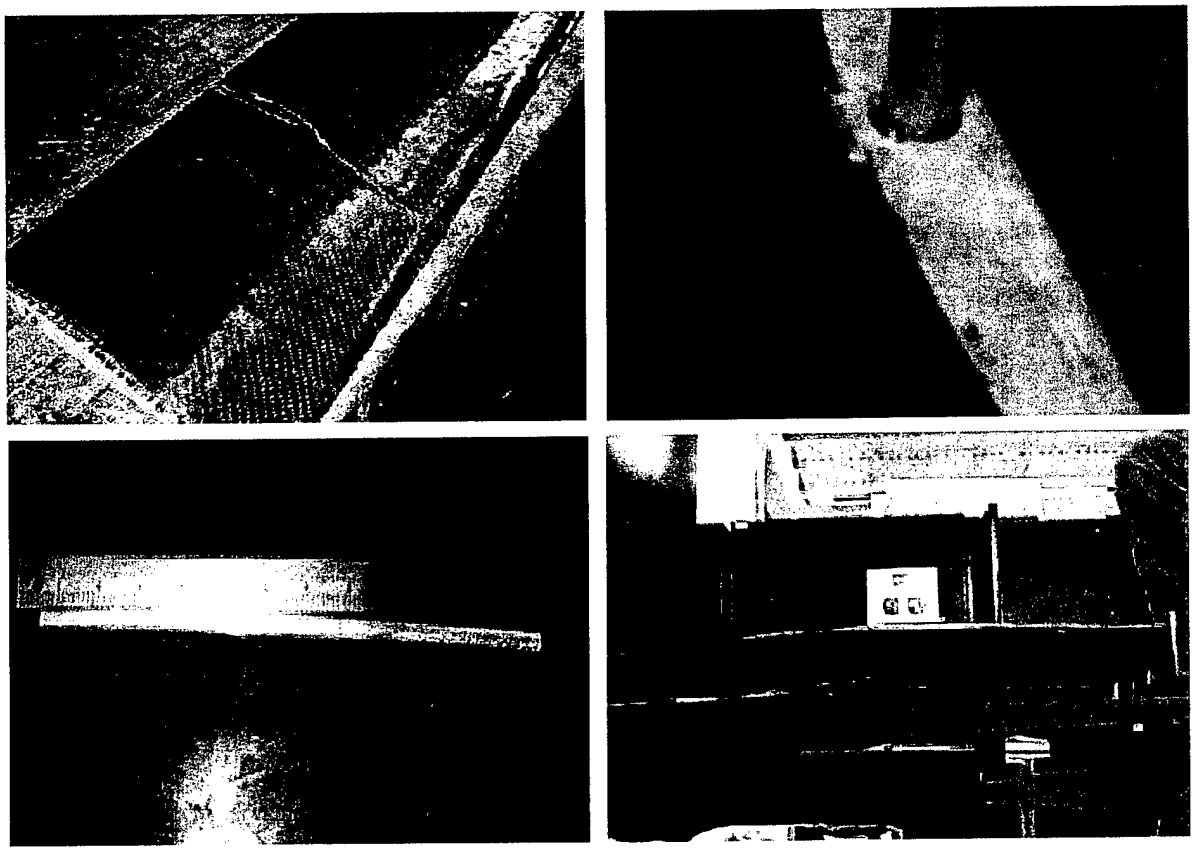


Figure 24: Starting at the Top Left –(a) Wing #1 Failure at the Rib, (b) Wing #2 Failure at the Root, (c) Bent Wing Joiner, (d) Wing #3 Failure at the Rib.

7.2.4 Static Thrust Test

The Astro Cobalt 60 motor with an APC 14 X 10 propeller was tested in order to verify the motor's static thrust. The theoretical and experimental results of power versus thrust and the experimental setup are shown below in Figures 25 and 26, respectively.

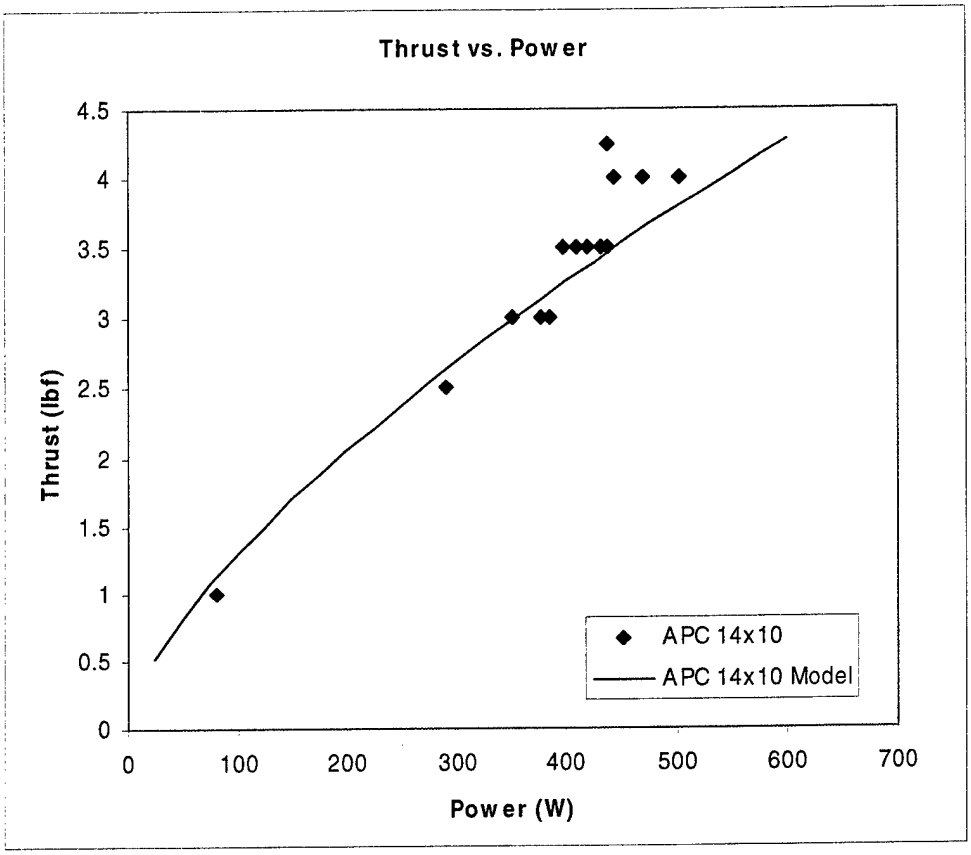


Figure 25: Thrust vs. Power for 60-Size Motor With an APC 14 X 10 Propeller.

There were a few limitations to the motor test depicted below. First, the test stand was not large enough to attach the designed propeller size, 17 X 14. Thus, a 14 X 10 propeller was used instead. This meant that the test was used only to validate the propulsion model, not to provide exact thrust values. Another problem with the test was due to the high altitude in Colorado. At lower densities there is less of a load on the propeller which decreases the current draw, (or power input in to the motor), and increases the RPM. Thus the motor will behave differently at competition where the altitude is lower by a few thousand feet.

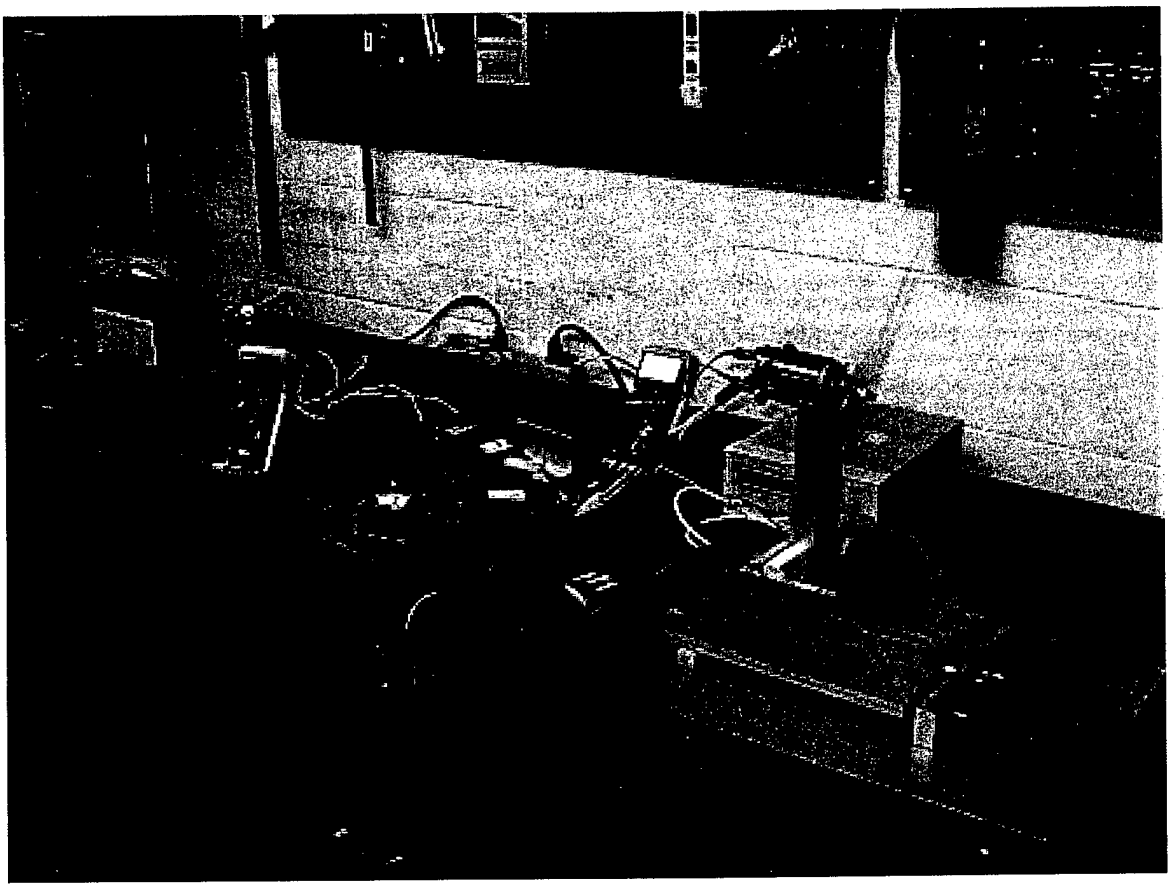


Figure 26: Test Stand and Testing Equipment for the Motor Thrust Tests.

7.2.5 Battery Test

Since batteries are such a large factor in the rated aircraft cost, the number of cells chosen for the design were minimized as much as possible. Therefore, it was important to test their life in order to ensure that they would last the duration of each mission. This test was accomplished by attaching twenty cells to a motor and running it at full throttle, (25 amps), until they no longer provided any power. Figure 27 displays the thrust as a function of time. The current that will be experienced in flight is more than 25 Amps due to the increase in propeller size. Thus, the experimental results were used only to verify the analytical model provided from (Ref. 6).

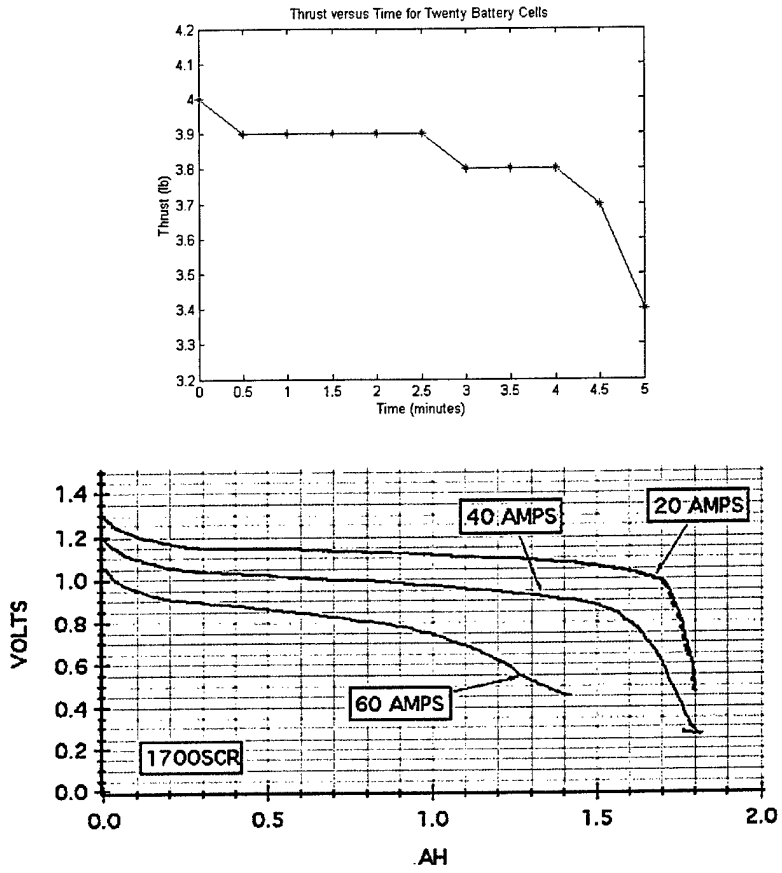


Figure 27: a (top): Thrust vs. Time for a Motor at Full Throttle with 20 Cells. b (bottom)Theoretical Prediction for Battery Life as a Function of Amperage.

Figure 27a shows that theoretically, for a 25 amp current, the batteries' power should drop off at approximately 4.1 minutes. This is similar to the value that was determined experimentally, 4 minutes, thus validating the model. At a 40 amp current, which represents the worst-case scenario for in flight, Figure 27b predicts a battery life of 2.55 min. This proves that 20 cells will most likely provide enough power for both the ferry and fire-bomber missions, which are estimated to take 2 and 3 minutes, respectively. Another thing to note from Figure ?? is that the batteries may not have been fully discharged after their previous use because the thrust drops at approximately 2.5 minutes. Instead, the batteries should have provided a constant thrust value until dropping off rapidly at 4 minutes.

7.2.6 Thermal Tests

Battery and motor efficiency decrease with increasing temperature. However, the flight times for both missions will be so short that these effects may be negligible. According to Utah State tests during the previous year, "Motor efficiency was independent of temperature, even after significant heating." (Ref. 7) Battery performance, however, will be tested in order to see how much of an improvement could be achieved with the addition of an air scoop. This will be done by placing the batteries in a wind chamber, and repeating the battery life tests previously mentioned. If these results show a significant improvement from those obtained without cooling, then an air scoop will be implemented.



7.2.7 Flight Test

While testing each component individually provides a lot of useful information, many problems and areas of improvement cannot be realized until a flight test is performed. Once construction is complete, the aircraft's first flights will consist of simple maneuvers to identify basic handling problems. Next, the airplane will be fully loaded and a few possibly problematic areas will be tested. These tests will be performed using a course with the same dimensions as those, which will be encountered in competition. One of the flight tests will be verifying that the aircraft can meet its 150 ft take-off distance requirement. If it cannot take-off within that distance, a propeller with a larger pitch and diameter and more batteries will be used in order to increase the static thrust of the motor. Alternatively, if the aircraft meets the take-off distance with ease, then a smaller diameter propeller will be used to increase the flight speed. Flying both the ferry and fire-bomber missions will also test the batteries. If the batteries cannot provide enough power for either of these missions, more cells will be added until it can do so with some room for error. Another test that will be performed is the water drop test. This will be done by first checking to see if the plane can fly slow enough to drain all four liters within the required drop distance. If this is successful, then it will be repeated, while varying the flight speed and where the 360° turn is performed, until the drop time is optimized. If the aircraft is unable to drop all the water within the required distance, the rated aircraft cost model will be re-examined to see if it is better to hold less than the full amount, or if it is better to make design changes to allow a lower flight speed. The last test that will be performed in order to make sure the aircraft is ready for competition is the inverted wing test.



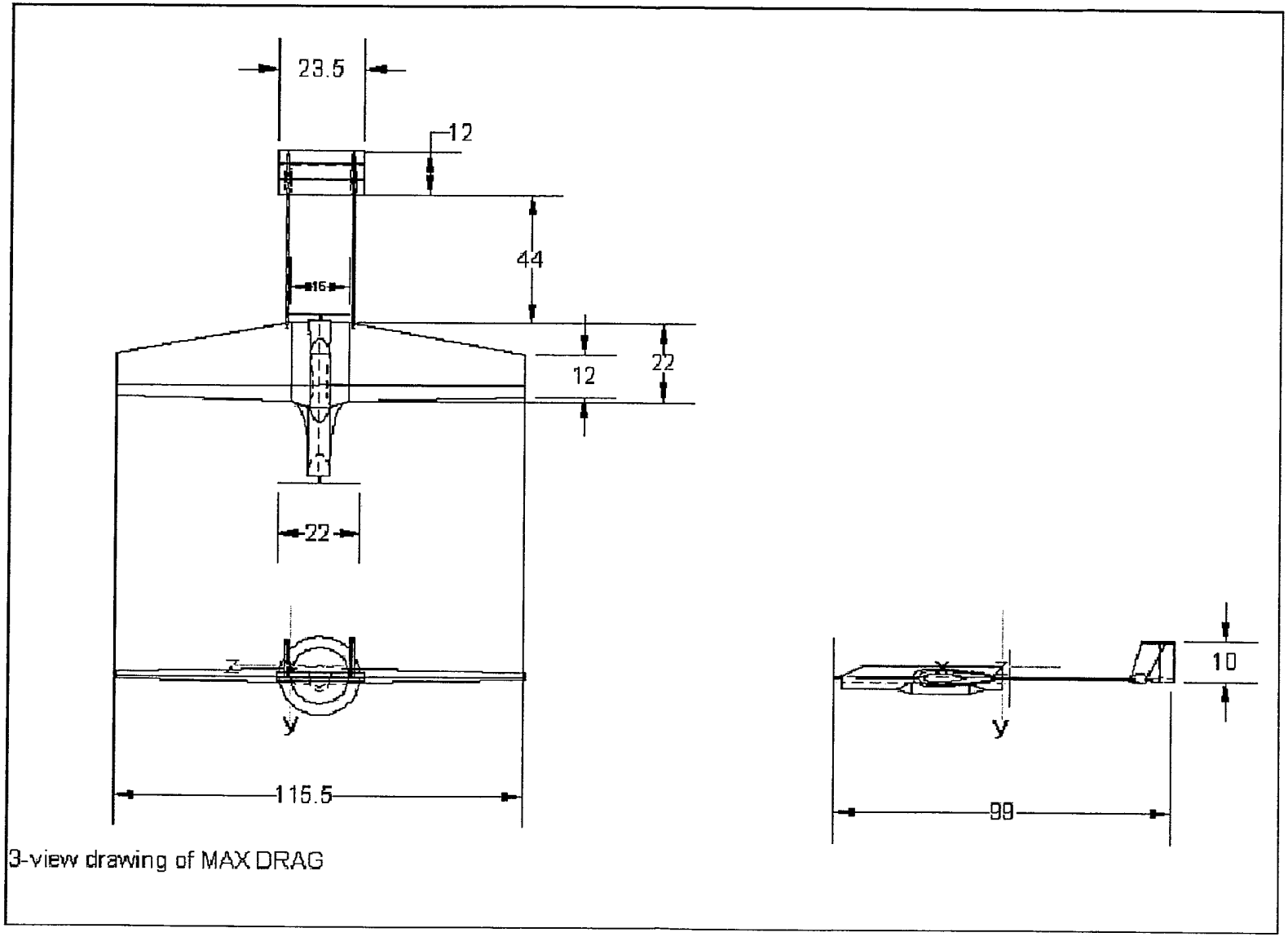
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- Professor Brian Argrow, University of Colorado-Boulder
- PhD John Crimaldi, University of Colorado-Boulder



3-view drawing of MAX DRAG

2003/2004 AIAA FOUNDATION STUDENT DESIGN BUILD FLY COMPETITION

Design Report by:
Cadet Brian Barnett
Cadet Brad Gates
Cadet Michael Morris
Cadet Dennis Meyer
Cadet Toby Wadsworth
Cadet Samuel Weickgenant-bitter

The Knight Wings
United States Military Academy
March 2004

Executive Summary

This paper details the development for one of the United States Military Academy's entries into the 2003/2004 AIAA Design Build Fly competition. The competition called for teams to create an aircraft that can complete both a fire bombing mission and a ferry mission. The project manager divided the aircraft up into five components and assigned a senior to head the development of each. The first step in the design process, the Problem Definition Phase, was to thoroughly analyze the problem and determine which tasks were most vital to mission completion. The next step, the Conceptual Design Phase, was to brainstorm different concepts in order to solve the problems identified in the Problem Definition Phase. These concepts were then evaluated against each other and the best ones implemented. The prevailing concepts were then compared against historical, empirical data and theoretical models in order to determine component sizes, orientations, and positions in the Preliminary Design Phase. The next step, the Detailed Design Phase, is to determine all the details of each component and how it will connect to the rest of the aircraft. At this point the design should be completed and is ready for the Manufacturing Phase. It is important to form and keep a manufacturing schedule as this phase is the easiest to get behind in. Once the aircraft is built, testing should be done to determine changes that you have the time and resources to make. If time permits this should become an iterative process. When testing is complete there is nothing left to do besides practice and compete.

Management Summary

As a part of the requirements for a degree in Aeronautical Engineering here at the Academy the seniors must complete a Capstone project. One of the many options available to us is an entry into AIAA's Design, Build, Fly competition. The six members of the team selected the competition as their Capstone project and then actively recruited the five underclassmen. In total, the team is comprised of 10 Aeronautical Engineering majors, 1 Systems Engineering major, and the pilot is a Military Science major.

The team was broken up into five sections, each headed by one of the senior members, with the remaining senior as the project leader. The underclassmen were then assigned to the sections based on anticipated need for additional labor.

Team Organization:

Project Leader – Dennis Meyer

Empennage – Toby Wadsworth

Assistant – Edward Russel

Fuselage – Brad Gates

Assistant – Heather Doyle

Landing Gear – Michael Morris

Assistant – Mac Barclay

Propulsion – Brian Barnett

Assistant – Ian Welch

Wing – Sam Weickgenant-Bitter

Assistant – Ryan Mcquarrie

Pilot – Mac Barclay

For the first three months of the fall term the team consisted of only the team leader, due

to the method used to administer and evaluate the senior Capstone projects here at USMA.

During the Problem Definition Phase the important deadlines for the project were identified. Working backwards from these deadlines the team developed a schedule in the form of a Gantt chart. This chart served as the metric by which we measured our progress. See Figure 2 for a copy of our Gantt chart.

Problem Definition

There are two mission profiles for the 2004 AIAA DBF competition: the Fire Bomber and the Ferry. The former requires the aircraft to be loaded with up to 4L of water, dump the water on the downwind leg of the course, land to be refilled, and repeat the process. The latter mission requires the aircraft to complete four laps around the course as quickly as possible with no cargo. Mission score is determined by dividing the mission profile's corresponding difficulty factor, 2 and 1, respectively, by the time used to complete the mission. Maximum flight time is 10 minutes.

The rules also impose several constraints on the design of the aircraft:

- Cannot be rotary wing or lighter than air.

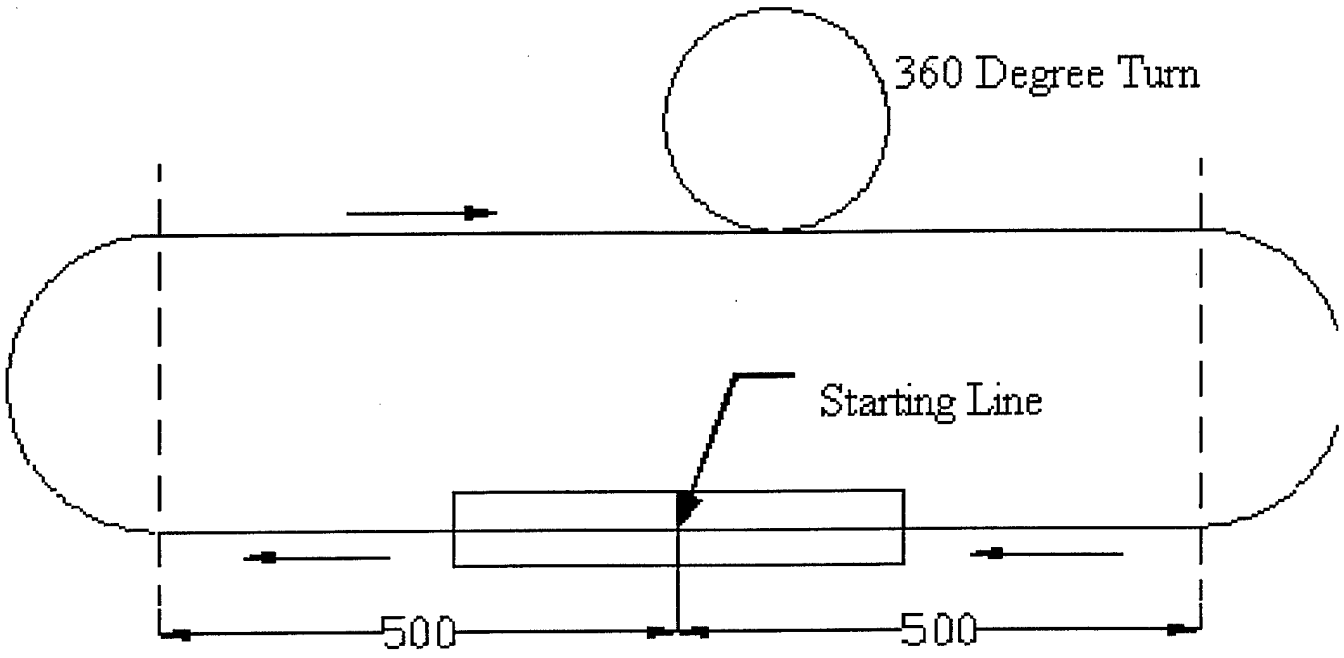


Figure 1: Course Diagram

	Task Name	Start	Finish
1	Project Duration	Tue 8/19/03	Sun 4/25/04
2	Problem Definition Phase	Mon 8/18/03	Thu 8/28/03
3	***Actual***	Mon 8/18/03	Thu 8/28/03
4	Conceptual Design Phase	Thu 8/28/03	Thu 9/18/03
5	***Actual***	Thu 8/28/03	Thu 9/18/03
6	Preliminary Design Phase	Thu 9/18/03	Fri 10/31/03
7	***Actual***	Thu 9/18/03	Fri 11/14/03
8	Detailed Design	Mon 11/3/03	Fri 12/19/03
9	***Actual***	Mon 11/17/03	Fri 1/30/04
10	Component Manufacturing	Mon 2/2/04	Wed 2/25/04
11	Aircraft Assembly	Wed 2/25/04	Wed 3/24/04
12	Aircraft Testing and Modification	Thu 3/25/04	Wed 4/21/04
13	Write Reports	Mon 2/23/04	Fri 3/5/04
14	Competition	Fri 4/23/04	Sun 4/25/04

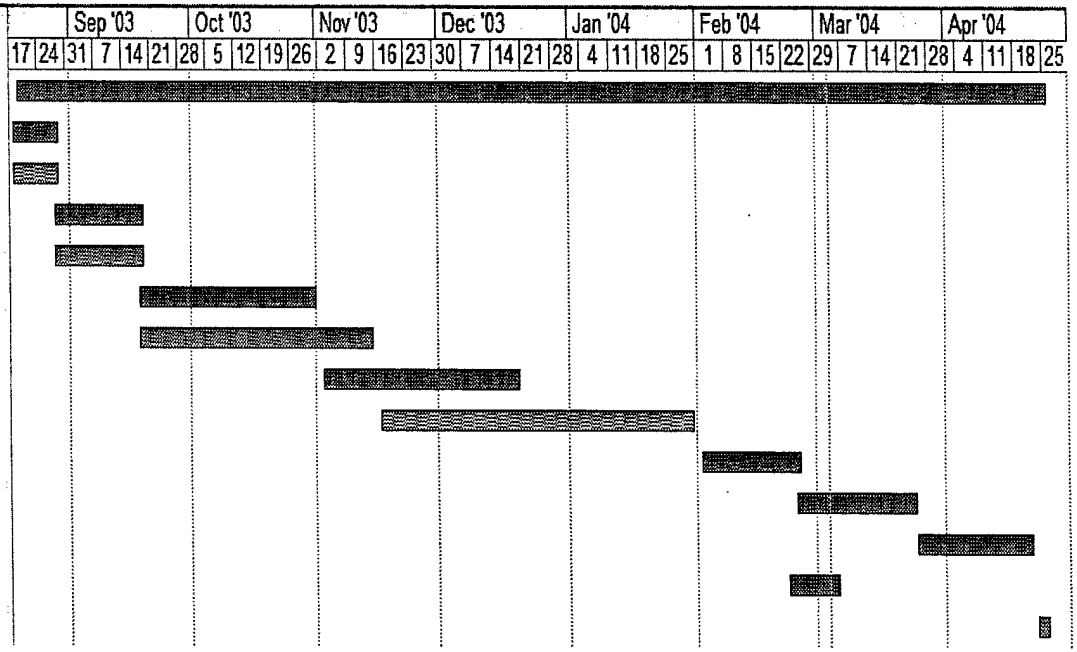


Figure 2: Gantt Chart

No payload may be carried internal to the wing proper.

- Must be powered by an unmodified, over the counter, electric motor of the Astroflight or Graupner classes.
- Maximum weight of the propulsion systems battery pack is 5 lbs. Batteries must be comprised of commercially available Ni-Cad cells.
- Maximum current draw of 40 amps, regulated by a fuse. Electrical connections must be keyed.
- Must survive both a normal and inverted wing tip test.
- Radio systems must be backed up by a fail-safe.
- Must fit inside a 4'x1'x2' shipping box.
- Must take off in under 150'.
- Total aircraft weight must be less than or equal to 55 lbs.
- Payload dump orifice must be less than 0.5".

The course configuration is depicted in Figure 1.[1]

After completing a complete analysis of the rules, constraints, and mission profiles, the team created an Objectives Tree to identify the design objectives for the aircraft. A copy of the Objectives Tree is included in Figure 3.

The tasks obtained from the Objectives Tree were then compared against one another using Pugh's Method in order to determine which are the most important. Table 1 summarizes the results of the numerical decision matrix.

Rank	Objective	Relative Importance
1	Provide control	13.15%
2	Fit in shipping box	12.21%
3	Take off in 150ft	11.27%
4	Forward speed	10.33%
5	Hold together in winds up to 30 mph	8.45%
5	Land safely	8.45%
5	Parts are secure	8.45%
6	Contain payload	5.60%
6	Initiate dump	5.60%
6	Allow payload exit	5.60%
7	Terminate dump	4.69%
8	Stable flight	2.82%
9	Protect payload	1.88%
10	Parts interface well	0.94%
11	Resists water damage	0.47%

Table 1: Top Five Design Objectives

The relative importance values attained from this analysis were used in order to determine where the majority of the team's effort and resources should be allocated. They were also used in the Concept Design Phase to help in deciding which figure of merit values to use.

Conceptual Design

Using the objectives attained from the Problem Definition Phase the team brainstormed figures of merit that could be used to judge the performance of different concepts. The objectives and FOMs were then compared in a Quality Functional Deployment chart (Figure 4) to find which FOMs were the most crucial to the design. Then research was done to find benchmark values for each FOM in order to guide the selection of engineering targets for our aircraft.

[1] Picture attained from the AIAA DBF website.

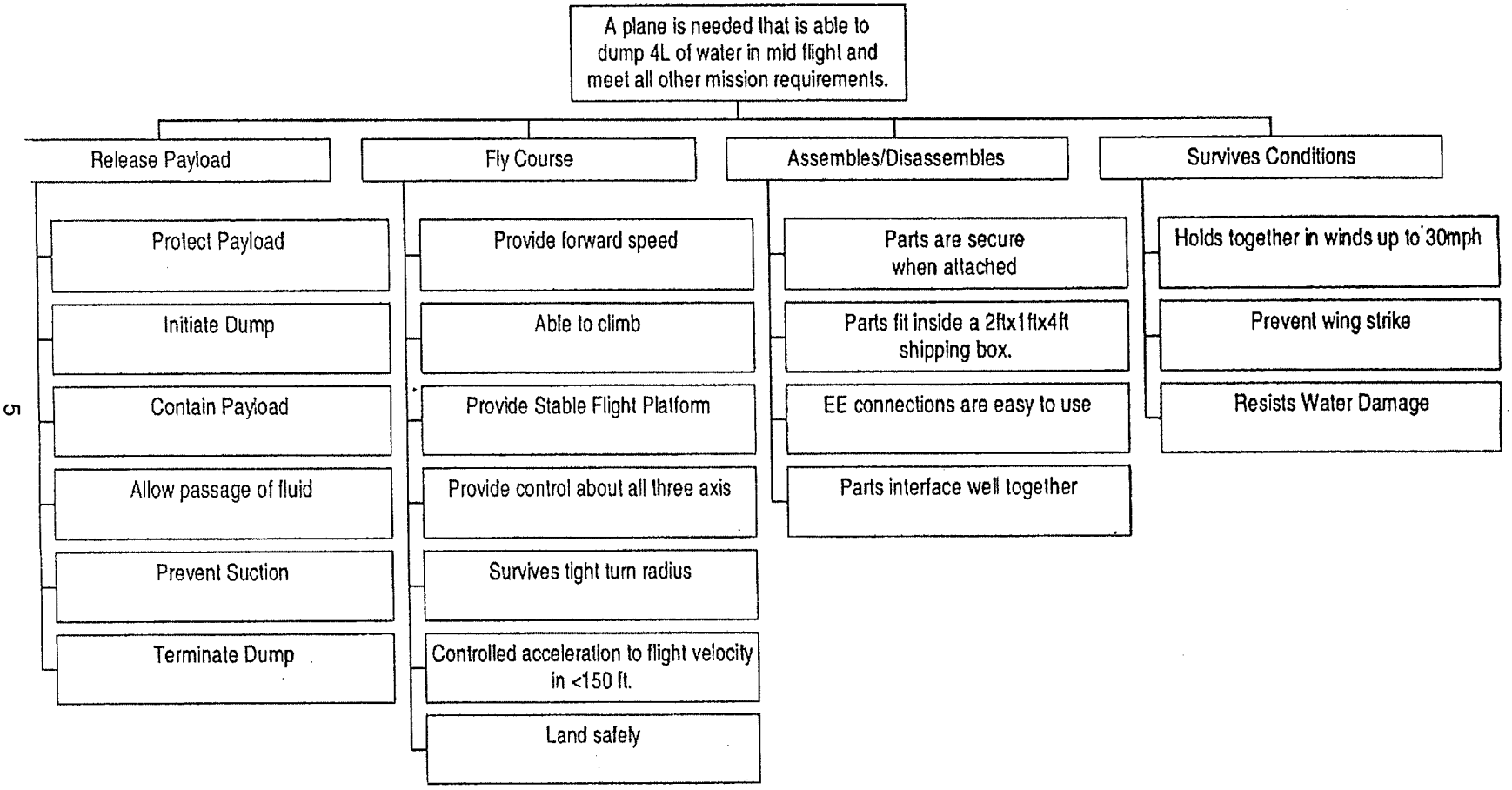


Figure 3: Objectives Tree

			1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
		Weight Factors (importance)	Cruise Speed	Stall Speed	(T/W)	Turn Radiusat Cruise Speed	L/D Max	Wing Loading	Static Margin (Loaded)	Static Margin (unloaded)	Payload Capacity	Time to Dump	Number of Motors	Max Flight Time	Number of Parts	Current Draw	Time to Disassemble	Time to Assemble
	Direction of Improvement		↑	↓	↑	↓	↑	↓	↑	↑	↑	↓	↑	↓	↓	↓	↓	↓
	Customer Requirement																	
1	Protect Payload	11%							3	3	9			9			3	3
2	Contain Payload	6%							9	3	9			9			3	3
3	Initiate Dump	6%							3	3	9	9		9			3	3
4	Terminate Dump	4%							3	3	9	9		9	9		3	3
5	Allow Passage of Fluid	6%							3	3	9	9		9	9		3	3
6	Forward Speed	10%	9	9	9	3	9	9			9			9	9	9	3	3
7	Take off in 150 ft	13%	3	9	9		3	9	9	9	9		3	9	9	9	3	3
8	Stable Flight	8%	3	3		9			9	9	9			9	9		1	1
9	Provide Control	12%							3	3	9			9	9			
10	Return to Ground Safely	8%		9					9	9	9			9	9			
11	Parts are secure	4%	3								9			3	9		9	9
12	Fit in Shipping Box	3%									9			3	3		3	3
13	Parts interface well	10%					3				9			9	9		9	9
14	Holds together in 30mph wi	3%									3		9	3	3		1	1
15	Resists Water Damage	5%									3	3	3	3	3	3	1	1
	Absolute Importance		1.61	2.79	1.94	0.56	1.29	1.94	3.47	3.13	2.91	1.52	1.67	5.56	4.58	1.94	2.56	2.56
	Relative Importance		0.04	0.07	0.05	0.01	0.03	0.05	0.09	0.08	0.07	0.04	0.04	0.14	0.11	0.05	0.06	0.06
	Competition Benchmarks																	
	Galileo - Rome University		95	36	0.35		14.3	2.43	0.16	0.16				176				
	2002 AIAA Entry		74	42		107										167		
	units		fps	fps	-	ft	-	lbf/ft^2	%	%	lb		lbf	lb		amps	min	min
	Engineering Targets		**90**	**38**	**0.4**	**100**	**60**	**2.625**	**0.2**	**0.2**	3	27	30	12	125	15	10	15

9 Strong Correlation
3 Medium Correlation
1 Weak Correlation

Figure 4: Quality Functional Deployment

Concepts Studied – Payload System

The payload is arguably the most important portion of the design. Without it you could only perform half of the missions available. Therefore it was the first component designed and all other components were to be designed around it, with few exceptions. The most important factors in deciding which concept to use for our payload system were weight and simplicity.

The first problem considered was how to regulate the flow of water between the container and the dump orifice. Two concepts were developed that would successfully accomplish this task. The first was a commercially available solenoid valve and the second was a manufactured pinch valve. The first option was by far the simpler of the two but proved to be prohibitively heavy. The second concept required considerable manufacturing however the low weight carried more of an advantage according to our design objectives.

The second problem considered was how to contain the water. Initially it was debated whether a hard shelled tank should be used or a free-form bag type container should be used. The latter option was considered heavily because it would naturally drain faster than the hard shell and would not require venting in order to prevent the formation of a vacuum. It was, however, eventually discarded for three reasons:

- The soft-shell itself could not be allowed to shift inside the fuselage and would therefore need some type of restraint system to restrict its movement.
- The movement of the CG during draining is far more difficult to predict and may cause the aircraft to become unstable at times.
- Soft-shells are inherently less rugged and susceptible to punctures and tears.

Using a hard-shell design would avoid these three problems and would only present the problem of vacuum build-up. It was then determined that the shape of the container would be set by the shape of the fuselage in order to use the least amount of fuselage space possible.

Concepts Studied – Fuselage

Historically there have been two fuselage shapes, square and circular. In the beginning days of aviation fuselages were typically square due to the prevalence of wood framework and the difficulty of shaping wood into a circle. As planes transitioned to metallic frames, circular fuselages became dominant. This evolution took place because a cylinder produces less drag than a square prism. For a given area the maximum distance between the centroid of the trailing surface and the outer edge is larger for a square than for a circle. This means that as the free stream passes the trailing edge of the prism, the flow near a corner is further from the vacuum formed at the centroid of the trailing face than it would be in a circular configuration. This would in turn cause the vortices at such points to

shed in a more violent manner, thus increasing drag. The corners also have another inherent flaw. As the prism moves through the air the edges cause vortices to form along the length of the fuselage, again increasing drag.

A cylindrical fuselage also provided a non-aerodynamic advantage over a square prism as well. In considering the machinery the team has available to work with, it was determined that circular supports are far simpler to manufacture than square supports.

Concepts Studied – Wing

The primary concern in the choosing a concept for the wing was the maximum lift over drag ratio ($\left(\frac{L}{D}\right)_{Max}$). The secondary concern is stability. Three basic concepts were considered for the wing: traditional wing configuration, biplane wing configuration, and a flying wing configuration.

The biplane configuration was quickly discarded because of its inherently low $\left(\frac{L}{D}\right)_{Max}$ value. Biplanes produce a high amount of drag due to the structural components between the wings and the interference of the upper wing's downwash. Additionally, the addition of a second wing does not produce enough lift to overcome this drag and therefore $\left(\frac{L}{D}\right)_{Max}$ is driven lower.

The flying wing configuration eliminates the need for a fuselage and drastically reduces the amount of drag produced. It has by far the highest $\left(\frac{L}{D}\right)_{Max}$ values of the three configurations. However, flying wing configurations are inherently unstable. It is possible to make them stable though the use of gyroscopes or other such sensors. This option would however, create computer programming problems which were determined to be beyond the skill level of the team. Additionally, evidence from past competitions suggested that these systems rarely work as intended, leaving the aircraft unstable anyway.

A traditional wing configuration provided a favorable compromise between $\left(\frac{L}{D}\right)_{Max}$ and stability. Additionally, it was determined that the aircraft would use a high-wing design in order to increase roll stability in cross-wind situations and namely to avoid wing strike upon take off and landing.[2]

Concepts Studied – Landing Gear

In choosing a concept for the landing gear the team considered three items: weight, stability, and simplicity. The landing gear has the lowest effect on our aircraft's performance and therefore spending as little time on it as possible while still keeping it light became important. The team opted to purchase commercially available landing gear because it would only require mounts to be machined and would save a great deal of time.

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[2] John D. Anderson, *Aircraft Performance and Design* (Boston: McGraw-Hill, 1999), 428-429.

There were two basic configurations available given our wing/fuselage configuration: tricycle and tail-dragger. While the tail-dragger configuration provides greater propeller clearance and wing incidence angle it is inherently unstable. This is because in this configuration it becomes necessary to place the CG aft of the main wheels. Conversely, the tricycle configuration is inherently stable because the CG must be placed fore of the main wheels. This difference in stability became the deciding factor in selecting the tricycle configuration.[3]

Concepts Studied – Empennage

There are three basic tail configurations that were considered: T-tail, V-tail, and conventional. In initial considerations the V-tail design was considered strongly because of its lower Rated Aircraft Cost due to it having fewer control surfaces. After initial research, however, it was discarded. This resulted from the V-tails complexity in manufacturing and control linkage. The T-tail configuration is the most efficient of the three considered, however, it requires a rigid vertical stabilizer and therefore a considerably heavier structure. It was determined that the advantage gained by the increased efficiency did not overcome the disadvantage incurred by the additional weight. The conventional tail was selected because of its moderate weight and ease of implementation.

Concepts Studied – Propulsion

The propulsion system is the most heavily regulated component on the aircraft. Because of this, the options to choose from were quite limited. Three options were studied: pusher prop, tractor prop, and ducted fan. Initially a ducted fan configuration was appealing because ideally it would be more efficient by eliminating the effects of the propeller tips. In order for this idealized model to be accurate the prop blades must interface with the duct with very exact tolerances. Most commercially available ducted fans are fabricated of plastic and wood, which deform under the loads imposed in flight. This requires the manufacturer to increase the tolerances between the blade and the duct and therefore reduces the advantage gained in efficiency. Furthermore, testimonies by model aircraft enthusiasts suggested that ducted fans were prone to blades breaking, relatively expensive, and had lower thrust to power consumed ratios than open propeller configurations.

The determining factor in choosing the tractor configuration rested in the weight distribution and its effect on stability. In the pusher configuration the aft engine placement shifts the CG aft, therefore a larger tail is required. Conversely, a tractor configuration shifts the CG forward, reducing the required tail planform.[4] According to our initial weight estimates our empty weight

[3] John D. Anderson, *Aircraft Performance and Design* (Boston: McGraw-Hill, 1999), 443.

[4] John D. Anderson, *Aircraft Performance and Design* (Boston: McGraw-Hill, 1999), 421.

was to be approximately 9 lbs. The weight of the motor alone was approximated at 34 oz, this is 17.7% of the empty weight of the entire aircraft. This would result in a massive increase in tail planform with only a slight gain in reducing skin friction drag. This configuration would also require us to use a twin boom tail design, which would increase the aircrafts weight considerably.

Preliminary Design

Weight Budget

During the Problem Definition Phase research was done into past DBF competitions and model aviation. One of the most important discoveries was that, in terms of weight, this year's competition is very similar to last year's. An average weight of three of last year's planes (USU, La Sapienza, and USMA) was the basis for our estimate for this year's weight. It was determined that a better use of materials, the lighter payload, and a smaller aircraft could reduce the weight from the average of 18.04lbs to 12lbs.

The total empty weight was subdivided into component weights using historical averages found in Aircraft Performance and Design by John Anderson.[5] Table 2 contains values of target weights for all aircraft components. For initial estimates the payload containment system was included as part of the fuselage.

[5] John D. Anderson, *Aircraft Performance and Design* (Boston: McGraw-Hill, 1999), 451.

Component	Target Weight (lbs)	% of Total
Fuselage	2.63	21.9
Wing	2.27	18.9
Empennage	0.61	5.1
Landing Gear	1.8	15
Propulsion	4.69	39.11

Table 2: Target Component Weights

Wing

The first component designed was the wing as it is the basis for much of the rest of the aircraft. It was decided early on that a rectangular planform would be used because it offers the most area for any given span and it is easy to manufacture. Tapered wings are more efficient than rectangular wings but it was determined that the higher C_L values created by a higher wing loading where more important than the efficiency gains created by tapering the wings. Our engineering target for wing loading was 2.625 lbf/ft^2 or lower. It was determined that the aircraft may end up with a higher weight than anticipated due to the ambitious nature of our weight goals. Therefore in order to keep the wing loading below the target value if the aircraft is overweight, the target was lowered to 2.25 lbf/ft^2. Constrained by the dimensions of the shipping box it was determined that the wing span would be 45 inches long with a three inch attachment spar to connect to the fuselage. Given a 21 lb aircraft the planform area required to result in a wing loading value of 2.25 lbf/ft^2 is 1350 in^2, resulting in a chord length of 15 inches.

The contest rules stipulate a penalty for returning to the runway without an empty

payload tank. Also, the Fire Bomber mission score is greatly affected by the amount of water the aircraft dumps. This creates a need for the aircraft to fly as slowly as possible on the upwind leg of the mission. At low speeds the problem of wing stall arises. This same problem is faced by commercial airliners when landing, they cannot land at cruise speed nor can they fly at landing speed. Commercial aircraft typically use a set of flaps to alleviate themselves of this problem. However, our research did not uncover any commercially available flap installation kits that would be rugged enough to handle the anticipated loads. Furthermore, it was determined that manufacturing our own flaps would be too difficult and time consuming. It was decided that the wings would instead use a flaperon, changing the equilibrium position of the flaps in order to allow for slower airspeeds.

The resulting design had a $(C_L)_{Max}$ of 0.95, $(L/D)_{Max}$ of 8.36, and a stall speed of 45 fps.

Empennage

The design of the tail surfaces is one of the least precise facets in aircraft design and is usually done using empirical data and historical trends. Average values for the volume ratios of the tail surfaces are: V_{HT} = 0.7 and V_{VT} = 0.04. These values are used in the equations used to drive our tail design:[6]

$$V_{HT} = \frac{l_{HT} S_{HT}}{\bar{c} S}$$

$$V_{VT} = \frac{l_{VT} S_{VT}}{b S}$$

In order to use these equations estimates had to be made for CG placement and tail length. Tail length was constrained by the size of the shipping box and the desire to keep the body in one continuous piece and was therefore limited to 15 inches. The further aft the tail was positioned the more effective it would be due to a greater moment arm and a decreased effect of the main wing downwash. Therefore it was determined that the tail length would be set at 15 inches. This would place the centroid of the tail approximately 38 inches aft of the CG of the aircraft. Entering this value into the equations for V_{HT} and V_{VT} yields values of S_{HT} = 373 in^2 and S_{VT} = 64 in^2. The chord was selected to be 12 in resulting in b_{HT} = 31 inches and b_{VT} = 5 inches. The resulting static margin for this design is was found to be 0.404 using the following relation:

$$\text{Static Margin} = \frac{x_{NP}}{\bar{c}} - \frac{x_{CG}}{\bar{c}}$$

Where x_{NP} is the distance from the nose to the neutral point and x_{CG} is the distance from the nose to the center of gravity. In this case x_{NP} was found by using the following equation:

$$x_{NP} = \left(\frac{x_{1-\frac{\bar{c}}{4}}}{\bar{c}} \right) \left(\frac{S_w}{S_w + S_{HT}} \right) + \left(\frac{x_{HT}}{\bar{c}} \right) \left(\frac{S_{HT}}{S_w + S_{HT}} \right)$$

This equation does neglect the planform area of the fuselage, however this was a good assumption because the change

[6] John D. Anderson, *Aircraft Performance and Design* (Boston: McGraw-Hill, 1999), 436.

in the projected area from nose to tail is very small compared to the planform areas of the wing and tail.

Propulsion

The following are the design constraints that were assigned to the propulsion system either by the competition rules or the overall design criteria: take off length 150 ft, propulsion system weights less than 4.69 lbs, 40 amp current draw, NiCd batteries, and Astro Flight or Graupner brushed motors. A SD 7003 airfoil was used for the wing sections and a computer program called MotoCalc was used to determine estimated characteristics of the wing and fuselage combination. All equations are from *Aircraft Performance and Design* text by John D. Anderson.[7] The following equation was used to calculate the stall speed.

$$V_{stall} = \sqrt{\frac{2 \cdot W}{\rho_\infty \cdot S \cdot C_{L_{MAX}}}}$$

This is the minimum speed the aircraft can fly while providing enough lift, and control. Based on the stall speed is the estimated lift off speed.

$$V_{LO} = 1.1 \cdot V_{stall}$$

The thrust required to lift off is found using the equation below. This is based on the thrust to weight ratio at the point of take off.

$$S_g = \frac{1.21\left(\frac{W}{S}\right)}{g \cdot \rho_\infty \cdot C_{L_{MAX}}\left(\frac{T}{W}\right)}$$

Since thrust decreases with velocity, we will calculate the thrust required by the engine to be at the lift off velocity. This will provide a margin of safety so we will not be disqualified for not meeting one of the constraints.

$$T = \dot{m}\left(V_j - V_\infty\right)$$

The power that is required to lift off and the power to perform a 15 foot per second climb at take off are calculated below.

$$P_{LO} = T_{LO} \cdot V_{LO}$$

$$\frac{R}{C} = \frac{P_{Excess}}{W}$$

In order to calculate the maximum speed of the aircraft the propulsion system must provide a thrust force that is equal to the total drag of the aircraft at maximum speed.

$$D = \frac{1}{2}\rho_\infty V_\infty^2 S \cdot C_D$$

After the performance parameters of the propulsion system were calculated MotoCalc was used to narrow down and view trends in motor combination and possible propeller combinations. After a motor was found the propeller, motor, battery combinations were varied until the best set for our mission parameters was found.

Table 3 shows the performance constraints on the propulsion system based on the overall constraints and minimum requirements for an aircraft that will fly.

[7] John D. Anderson, *Aircraft Performance and Design* (Boston: McGraw-Hill, 1999), Ch 8.

Design Constraint		Units
Stall Velocity	45.26	ft/s
Lift off Velocity	49.8	ft/s
Lift off Trust	5.39	lbf
Lift off Power	268.4	ft*lbf/sec
Excess Power Required	315	ft*lbf/sec
Min Power to climb @ L.O.	583.4	ft*lbf/sec
Drag @ Max Speed Required	3.162	lbf

Table 3: Performance Constraints

Based on the performance constraints and the overall size and power requirements the Astro Flight 661 electric motor was chosen.

The battery cells were selected in order to receive the largest power output while maintaining a minimum discharge time of 10 minutes. The result of this optimization was the selection of the Sanyo CP-2400SCR NiCad cells in a 30 cell pack.

Propellers were tested using a specially constructed thrust stand. Static and dynamic thrust was measured and plotted versus pitch for a given diameter. The best pitch for each diameter was selected and again the thrust was measure but this time it was plotted against diameter. The best of these propeller was selected, resulting in a 24x18 prop.

Landing Gear

The team had already determined that the landing gear would be commercially bought in order to save time and to allow the allocation of manpower to more crucial and complex components. Discussion of landings with our AMA pilot revealed that the landing gear should be able to handle at least a 2g landing without failing. This equated to a 42 lb static load. The team then looked for the lightest commercially available gear that fit the static loading requirement and found the Robart 159F but could not find a main gear that fit the requirements. However, the team pilot had a carbon fiber set that he felt would work well. He donated it to the team and after testing it proved that it could fulfill the requirements.

Fuselage

There were two requirements in deciding the size of the fuselage. First, the payload, propulsion, control system, and mounts for all the other components must fit inside of it. Second, it should be kept as short as possible to help reduce skin friction drag and so it will easily fit inside the shipping box. The upper limit on fuselage length was 20 inches in order to allow us to fit the whole aircraft body into the shipping box without disassembly. Using this and estimated sizes of the internal components it was determined that the fuselage needed to be at least 7.5" in diameter. The team decided to make the diameter 8" to allow adjustments to be made in the future. The design of the fuselage allowed us to quickly shorten it as long as it was not fully assembled. Therefore, it was made to be 18 inches long, the shortest length that could still accommodate all the internal components.

Attachment Points

Attachment points were paramount in our design considerations because of their propensity to create excess weight and their tendency to fail. A lot of care was taken to ensure that all attachment points were devoid of extraneous material. More importantly they were all designed to be adjustable. This was done with the understanding that designs are not always manufactured exactly as anticipated. Sub teams frequently discussed how they would bring their components together and whenever either would make changes they would communicate that change to the other sub teams affected.

Mission Model

One of our team's downfalls was the failure to create an accurate model of how changes in FOM values would affect our scores. The team relied on its knowledge of theory and its intuition to talk through the secondary and tertiary effects changes would have. It would have been much easier and more accurate to create a computer program that would take the changes as inputs and output raw data for analysis.

Detailed Design

The Detailed Design Phase was mostly material selection and fine tuning the design in order to make it more user friendly.

Wing

There were two options in deciding what materials to use for the wing: foam core or balsa spars. Balsa spars were eventually selected because a balsa framed wing is lighter than a foam core wing. In order to reinforce each wing a carbon fiber golf shaft was used. They are very light weight and are tapered as you move toward the wingtip. This provides an advantage as there is less material where the loads are lesser. The balsa spars were covered in a lightweight fiberglass in order to strengthen and stiffen the wing.

A fairing was fashioned out of foam and added between the wing and the fuselage. This is an important addition as without it the wing would experience tip effects in four areas rather than just two.

Empennage

The tail surfaces were constructed like the wings except they are reinforced by aluminum arrow shafts rather than golf club shafts. These shafts run through a central connecting piece and are pinned in place. The pins are removed and the tail disassembled in order to fit inside the shipping box.

Propulsion

The only change made to the propulsion system from the Preliminary Phase was the battery pack was split into two. This was done to allow us to make more minute changes in the placement of the CG. One pack was fitted in front of the payload

container and the second was placed behind the payload container. This allowed maximum flexibility in CG placement.

Landing Gear

The front landing gear was slightly shorter than the company had advertised which resulted in the creation of a plate that mounted the landing gear with half of the mounting plate protruding from the fuselage. In order to mitigate the amount of parasite drag this would cause a fairing was fashioned and attached.

The main landing gear was placed just aft of the CG to allow the aircraft to easily pitch nose-up and create a higher angle of attack for take off purposes.

Fuselage

It was decided that the fuselage would be covered in a lightweight fiberglass. This decision was made because many access panels were needed and the rigidity of the composite material allowed the panels to be cut only the needed size. Contrast this with having to cut the panel from the point of interest to the next structural member and then having to create and attach a panel of more rigid material.

Geometry data section

Total Length	46 inches
Wing Span	45 inches
Wing Chord	15 inches
Aspect Ratio	6
Height	16 inches

Weight statement section

Component	Target Weight (lbs)	% of Total
Fuselage	2.63	21.9%
Wing	2.27	18.9%
Empennage	0.61	5.1%
Landing Gear	1.8	15.0%
Propulsion	4.69	39.1%

Performance data section

Maximum Coeffiecient of Lift	0.95
Max Lift/Drag Ratio	8.36
Wing Loading	2.24
Stall Speed	45 fps
Static Margin	0.404

Systems data section

Radio	
Servos	
Speed Controller	
Battery Configuration	30 Sanyo CP-2400SCR
Motor	Astroflight 691
Propeller	24x18

Table 4: Sized Aircraft Data Tables

Table 5: Rated Aircraft Cost Worksheet

Manufacturers Empty Weight Multiplier	$ 300.00	
Rated Engine Power Multiplier	$1,500.00	
Manufacturing Cost Multiplier	$ 20.00	
		Contribution
Manufacturers Empty Weight	12	$ 3,600.00
Rated Engine Power		$ 6,375.00
Number of Motors	1	
Total Battery Weight	4.25	
Manufacturing Man Hours		$ 148.13
Wing		
Span	3.75	
Chord	1.25	
Wings	2	
Flaperons	1.5	
Fuselage		
Length	1.5	
Width	0.75	
Height	0.75	
Empennage		
Vertical Surface w/ Control	1	
Horizontal Surface	2	
Flight Systems		
Servos	6	
Motor Controller	1	
Total		10.123125

Manufacturing Plan

The biggest issue the USMA team had last year was a lack of manufacturing time. Consequently they were not ready in time to compete in the competition. This year manufacturing time was of paramount concern in the entire design process. Many alternatives were discarded because they would either take to long or were not feasible with the equipment available to us. Commercially available parts became invaluable in our mission to minimize manufacturing time. The only two components that required significant amounts of manufacturing were the fuselage and the payload system.

Fuselage

The fuselage was made out of aluminum because or its high strength to weight ratio. The supports for the fuselage were machined out of a plate and shaved down in places in order to save weight. The supports were then connected together using arrow shafts. These shafts were used because they are lightweight, strong, and easily attained. The entire fuselage is bonded together with epoxy. Welding it together posed a lighter option however, thin pieces of aluminum proved far beyond our skill levels.

Payload System

The payload system took the most manufacturing time as it was the most complex component on the aircraft. The valve had to be manufactured from scratch and was machined out of aluminum stock. In order for the valve to work correctly parts had to slide past one another. This created very low tolerances (±.0002") and time consuming processes.

With the exception of connection devices, the rest of the aircraft was generally made from pre manufactured parts and only assembly was required. This became more expensive than manufacturing everything would have been but it was worth it as we have a small team and manpower was at a premium. Figure 4 is a Gantt chart for the manufacturing process.

Test Plan

The first objective for our test plan is to make sure that all the safety systems are in place and working. This includes the propulsion system's cut off and the control unit fail-safe. This can be done as the units are installed but should also be tested again before each flight.

The next testing objective is to ensure that all flight-critical systems are functioning correctly. This includes the speed controller, each servo, and the motor. This should also be conducted before each flight.

Once the aircraft is safe and ready to fly, flight testing will occur. The primary objective of flight testing is stability. If the pilot is having a hard time keeping control of the aircraft then modifications will need to be made to the tail section and/or control surfaces. The secondary priority is finding out the radio settings for the various mission segments. It will prove vital to be just above the aircraft's stall speed as you dump the water. The downwind leg is 1000 ft

long, at 45fps it only takes 22.2 seconds to travel that far. Dumping 4L of water out of a half inch opening takes approximately 32 seconds. Therefore it is critical to go slow on the downwind leg in order to dump the most water possible. This leads into the thirst priority: finding out how much water the aircraft can dump. The penalty for returning without emptying the previous load is too severe to allow for error; therefore this must be tested thoroughly.

References and Acknowledgements

Anderson, John D. *Aircraft Performance and Design.* Boston: McGraw-Hill, 1999.

Major Patrick O'Brien for his tireless efforts as our team advisor and aeronautics instructor.

2003 La Sapienza Design team and AIAA *Aerospace America* because we used their publications as formatting templates.

	Task Name	Start	Finish
1	Building begun	Thu 1/29/04	Wed 4/7/04
2			
3	Fuselage frame manufactured	Thu 1/29/04	Mon 2/23/04
4			
5	Empanagge manufactured	Thu 1/29/04	Mon 2/23/04
6	Empanagge attatchment manufactured and installed	Mon 2/23/04	Wed 3/24/04
7			
8	Front Landing Gear ordered	Thu 1/29/04	Mon 2/2/04
9	Front landing gear mount and servos installed	Mon 2/23/04	Wed 3/10/04
10	Rear landing gear manufactured	Thu 1/29/04	Mon 2/23/04
11	Rear landing gear mount installed	Wed 3/10/04	Wed 3/24/04
12			
13	Propulsion finalized	Mon 2/2/04	Mon 2/2/04
14	Propulsion mount building	Mon 2/2/04	Mon 2/23/04
15	Propulsion mount installed	Mon 2/23/04	Wed 3/24/04
16	Flight control and wiring installed	Wed 3/10/04	Wed 3/24/04
17			
18	Valve manufactured	Fri 2/6/04	Fri 2/6/04
19	Tank Manufactured and valve mounted to it	Fri 2/6/04	Mon 2/23/04
20	Tank fittings manufactured and installed	Mon 2/23/04	Wed 3/10/04
21			
22	Wing Spars ordered	Thu 1/29/04	Mon 2/2/04
23	Wing assembled (no control surfaces)	Mon 2/2/04	Mon 2/23/04
24	Control surfaces and sheething added	Mon 2/23/04	Wed 3/10/04
25	Wing mounts manufactured and installed	Wed 3/10/04	Wed 3/24/04
26			
27	Aircraft testing and modification	Wed 4/7/04	Fri 4/16/04
28			
29	Competition	Fri 4/23/04	Sun 4/25/04

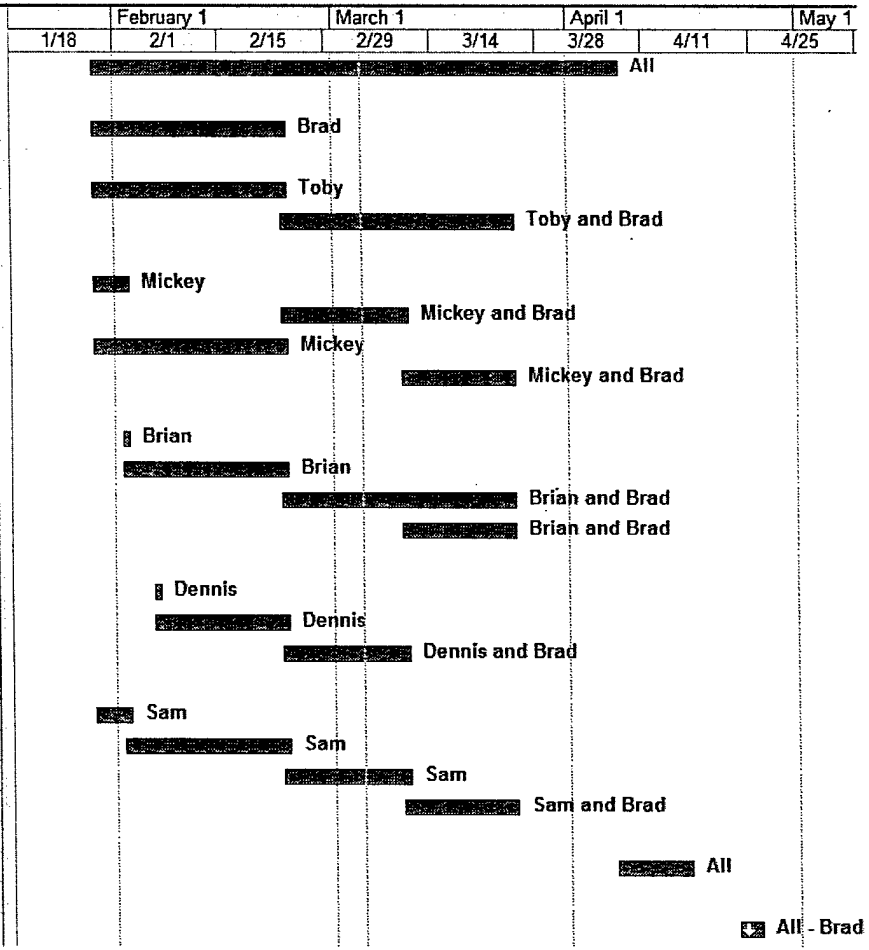
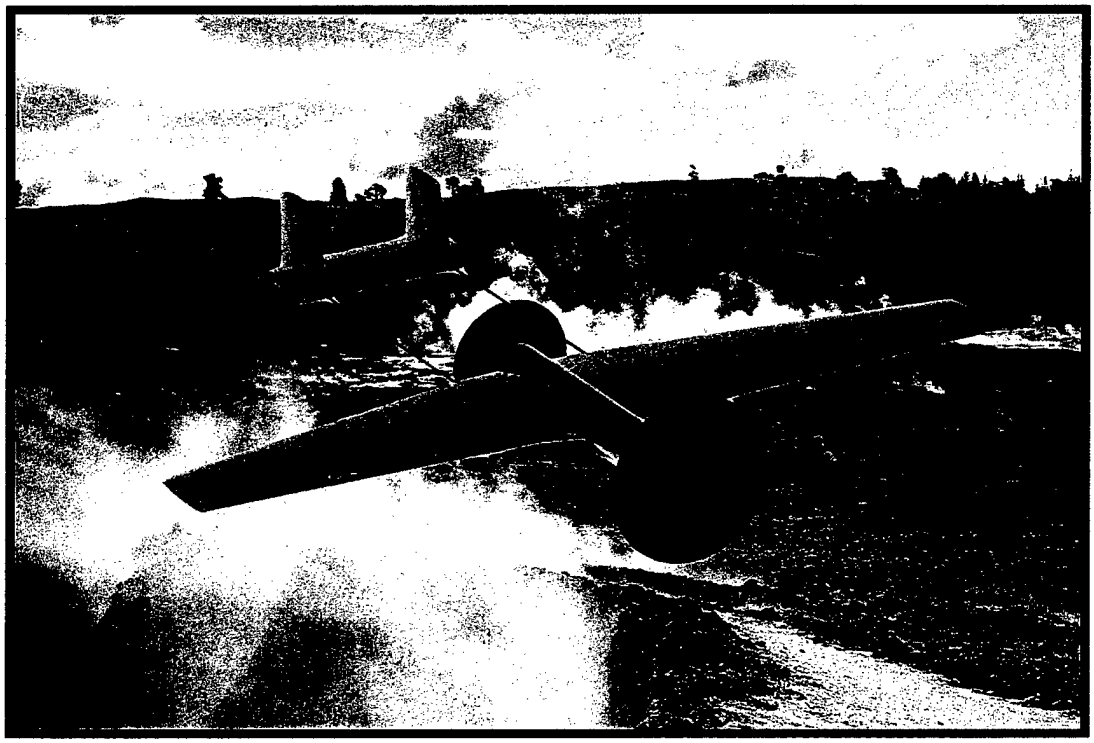


Figure 4: Gantt Chart for the Manufacturing Phase.

American Institute of Aeronautics and Astronautics

State University of New York at Buffalo Chapter

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



UB "MAX DRAG"

Design Report

Table of Contents

1.0 Executive Summary 2
- 1.1 Conceptual Design 2
- 1.2 Preliminary Design 2
- 1.3 Detailed Design 3

2.0 Management Summary 4
- 2.1 Architecture of Design Team 4
- 2.2 Configuration and Schedule Control 6

3.0 Conceptual Design 8
- 3.1 Wing 10
- 3.2 Tail 10
- 3.3 Fuselage 11
- 3.4 Propulsion 11

4.0 Preliminary Design 12
- 4.1 Wing 13
- 4.2 Tail 16
- 4.3 Fuselage 17
- 4.4 Propulsion 18
- 4.3.1 Motor 18
- 4.3.2 Batteries 19
- 4.3.3 Propeller 19
- 4.3.4 Motor Test Stand 20

5.0 Detailed Design 22
- 5.1 Wing 23
- 5.2 Tail 24
- 5.3 Fuselage 25
- 5.4 Propulsion 26

6.0 Manufacturing Plan 27
- 6.1 Wing 27
- 6.2 Tail 28
- 6.3 Fuselage 28
- 6.4 Engine Mount 29
- 6.5 Landing Gear 29
- 6.6 Material Selection 30
  - 6.6.1 Availability 30
  - 6.6.2 Cost 30
  - 6.6.3 Skill Level 30
  - 6.6.4 Time 30
  - 6.6.5 Strength 30

7.0 Testing Plan 33

1.0 Executive Summary

A team of students at The University at Buffalo decided to design and build a remote control airplane to compete in this year's Cessna/ONR Student Design Build Fly Competition (DBF). The AIAA sponsored competition will be held on the Cessna Pawnee East Field at Wichita Kansas, on April 23-25, 2004.

The competition involves designing and building a propeller driven, electric powered, unmanned radio controlled airplane capable of flying loaded with up to 4 liters of water and releasing them in mid-flight. The aircraft must fly a specified number of laps around a designated course in the shortest time possible. The airplane must be able to fly the designated course and complete one of the two possible missions within the ten-minute time period.

The goal of the team was to design and build an airplane that would achieve the greatest score. The total score achievable at the competition is based on the product of the written report score and the total flight score for the two different mission types, divided by the rated aircraft cost. The written report will be judged and assigned a score by AIAA officials. The total flight score is based on the gallons of water carried, and the total mission time left from the ten minute interval. The rated aircraft cost is a measure of the complexity and size of the aircraft.

1.1 Conceptual Design

The goal of the conceptual design phase was to evaluate the imposed design constraints as well as the experience of previous design teams to generate several possible aircraft configurations. Configurations were selected that could score highly and were manageable by the design team. The team utilized hand sketches and simple spreadsheet programs to compare different configurations and evaluate how changes in them would affect the final score. The major design constraints evaluated when generating the configurations were maximum takeoff length, battery weight, payload configuration, and total aircraft weight. The team's desire to build a smaller, faster aircraft compared to the previous years also influenced the configuration decision. The analysis resulted in a single engine monoplane (although this was later changed to a twin engine in the detailed design phase) configuration with a twin tail and a tapered wing. The full five pounds of batteries would be utilized.

1.2 Preliminary Design

The goal of the preliminary design phase was to further develop the single engine monoplane configuration. However the configuration was then changed to a twin engine, push-pull configuration, in order to increase the maximum speed of the aircraft and to cancel out the torques associated with a single propeller. The basic structure was developed and major components and materials were selected. The team utilized design software programs such as WinFoil, Pro/E and ThrustHP as well as spreadsheets to generate approximate dimensions and weights, and select possible components. Spar test sections were constructed and tested to validate the use of carbon fiber rods in the wings. Test

sections of the rod connections were also made in order to determine the best way to fit the rods together for increased strength as well for relatively easy assembly.

The weight of the loaded aircraft was estimated at 32 pounds. The wing loading was desired to be about 32 ounces per square foot, based on practical experience. The wing loading was used to select a wingspan of 9.5 feet, a root chord of 22.67 inches and a tip chord of 1 foot. The tail volume was selected based on the size of the wing. The basic structure of the wings, tail, and fuselage were designed using carbon fiber rods and foam. The overall aircraft design revolves around the ability to use a basic structure of easy-to-assemble carbon rods which would take on the main loads, and carbon-fibered foam to keep the external shape of the plane as well as to take on the shear loads. The required output of the front motor was calculated to be 1400 Watts, therefore the motor selected was the only one that fit in this power range, the Astroflight Cobalt 90. For the pusher motor, the Cobalt 60 was deemed sufficient to drive a 16-inch propeller in the un-geared configuration. The batteries were selected from a selection of virtually every commercial NiCd cell available, and the BYD DSC2100P was selected because of its high capacity to weight ratio as well as its ability to handle the high current discharge associated with the Cobalt 90. Finally, a motor test stand was designed and constructed to test different propellers and validate the preliminary choices made using the ThrustHP program.

1.3 Detailed Design

The goal of the detailed design phase was to refine the design so that manufacturing could begin. The final dimensions and structure were developed and all components and materials were selected. The team further utilized design programs such as Pro/E for 3D design as well as AutoCAD for 2D design. The motor test stand was used to test different propellers in order to select the best propeller based on real world static thrust performance data.

2.0 Management Summary

In the fall of 2003, a group of engineering students from the University at Buffalo assembled with the intent of designing and building an aircraft that would successfully compete in this year's Cessna/ONR AIAA Student Design/Build/Fly competition. The team consisted of 4 seniors, 2 graduate students, 2 sophomores, 6 freshmen, and was advised by two faculty members.

2.1 Architecture of Design Team

The team was designed and formed as a self-managed, interdisciplinary, dynamic group. This form was necessary due to the scope of the project and the number of people involved. The team's project managers were responsible for coordinating the administrative and technical efforts of the group. The development of the aircraft's design was divided into four teams as shown in Table 2.1. Each team concentrated in tasks relating to one area of the overall design. Membership in these individual design teams was based on expertise, interest, and team need.

Project Manager: Alexey Ouzounov & Gilbert Romanowski **Treasurer and Procurement: Danelle Schrader**			
Aerodynamics	**Power Plant**	**Structures**	**Control Systems**
Alexey Ouzounov	Milovan Vugdragovic	Dan Reed	Will O'Mearu
Bennie Mwiinga	Aditya s. Vaze	Milovan Vugdragovic	Gilbert Romanowski
Yuka Kona	Alexey Ouzounov	Yuka Kona	Marie Raediger
Dan Reed	Will O'Mearu	Chris Nebelecky	Bennie Mwiinga
Aditya s. Vaze	Danelle Schrader	Aditya s. Vaze	Erik Peterson
Erik Peterson	Dan Reed	Alexey Ouzounov	Yuka Kona

Table 2.1: Team Concentration Break Up

Actual and expected individual effort in each area of design and manufacturing is detailed in Table 2.2. A rating of 3 indicates maximum involvement by the team member, and a rating of 0 indicates no involvement.

	Alexey Ouzounov	Bennie Mwiinga	Yuka Kona	Dan Reed	Danelle Schrader	Erik Peterson	Gil Romanowksi	Milovan Vugdragovic	Aditya s. Vaze	Chris Nebelecky	Will O'Mearu	Marie Roedger
Conceptual Design	2	2	3	1	1	1	3	3	3	1	3	1
Design Parameters - Initial Phase	3	1	1	0	0	0	3	3	2	2	2	0
Figures of Merit - Initial Phase	3	1	0	2	2	1	2	3	2	2	2	1
Quantitative Analysis	3	2	1	0	0	1	3	2	2	1	1	0
Wing Analysis	3	1	1	0	0	2	2	0	1	1	1	1
Empennage Analysis	0	2	2	0	1	1	1	3	1	2	2	2
Fuselage Analysis	2	2	3	2	2	2	1	3	2	2	2	1
Power Plant Analysis	0	0	1	1	0	0	2	0	2	3	3	0
Numerical Analysis	3	1	1	1	2	0	1	0	1	2	2	1
Configuration Selection	3	0	0	1	2	2	2	2	1	0	0	1
Preliminary Design	2	3	3	2	3	1	2	3	3	1	2	1
Prototype construction and testing	3	3	3	1	2	2	2	3	0	1	1	1
Figures of Merit	2	3	1	2	2	2	1	3	2	0	0	0
Refinement of Design Parameters	3	2	3	3	2	2	2	2	1	2	2	2
Refined Numerical Analysis	3	0	3	3	3	0	0	1	2	3	3	0
Configuration Selection	3	2	3	2	2	2	1	2	1	1	1	1
Engineering Requirement Selection	3	1	3	2	3	2	2	2	0	3	3	1
Detailed Design	3	1	3	2	3	3	3	3	2	3	2	1
Configuration Optimization	2	2	3	2	2	2	1	2	2	0	0	2
Airframe	0	2	2	0	2	1	0	2	1	0	0	2
Propulsion Systems	0	2	1	3	0	0	2	0	0	1	1	1
Structural Systems	0	2	3	0	2	1	0	2	2	2	2	0
Control Systems	1	3	3	3	2	2	0	1	1	2	2	1
Final Configuration	2	1	2	2	2	2	2	2	0	3	3	2
Assembly Drawings	0	2	3	0	0	3	0	0	0	3	3	1
Performance Analysis/Optimization	1	1	2	1	3	1	1	1	1	2	2	0
Take-off and Clime	0	2	2	3	2	1	0	2	2	3	3	0
Range, Endurance, and Payload	1	2	2	3	2	1	1	0	3	3	3	0
Handling Qualities	1	1	3	1	3	0	1	0	1	3	3	0

Table 2.2: Team Personnel Assignment

	Alexey Ouzounov	Bennie Mwiinga	Yuka Kona	Dan Reed	Danelle Schrader	Erik Peterson	Gil Romanowksi	Milovan Vugdragovic	Aditya s. Vaze	Chris Nebelecky	Will O'Mearu	Marie Roedger
Manufacturing Plan	3	3	3	2	2	3	1	2	2	3	3	3
Figures of Merit	1	1	2	1	1	2	1	3	0	1	2	1
Manufacturing Process Selection	2	2	3	1	2	3	0	2	1	1	3	2
Detail Manufacturing Plan	1	2	3	2	1	3	0	2	2	3	2	3
Final Airplane Construction	3	1	3	3	3	3	3	3	3	3	3	3
CAD Design	3	3	3	0	1	3	1	2	1	2	3	3
Project Management	2	0	3	1	0	3	0	0	1	2	1	3
Documentation	1	1	3	0	0	3	0	1	2	0	0	2
Journal	0	0	2	0	2	3	0	0	1	1	0	3
Letter of intent	0	0	1	0	0	1	0	3	0	1	1	1
Project Report	3	0	3	3	3	3	0	3	0	0	1	0

Table 2.2: Team Personnel Assignment (Cont'd)

2.2 Configuration and Schedule Control

The team implemented a schedule to help visualize the project milestones and the time periods in which they were to be realized. The project managers were responsible for maintaining the schedule and ensuring that the milestones were being met.

Each week the entire design team met twice. During these meetings each individual team reported their progress. This provided a forum for the discussion of interdisciplinary problems, overall status, task assignment, and general administrative concerns. The majority of team productivity took place at other times during the week when groups met individually to discuss designs and ways that they could be improved.

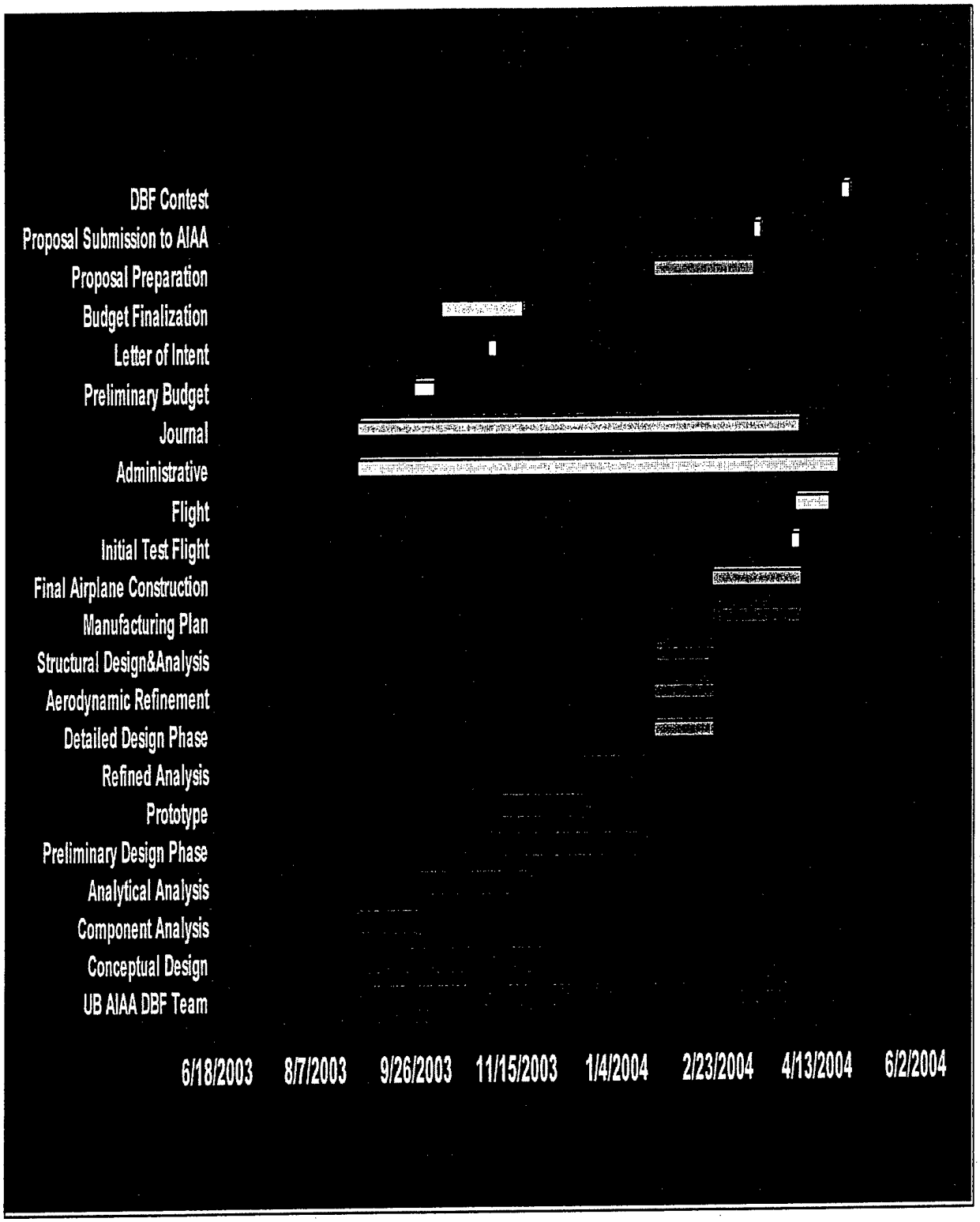

DBF Contest
Proposal Submission to AIAA
Proposal Preparation
Budget Finalization
Letter of Intent
Preliminary Budget
Journal
Administrative
Flight
Initial Test Flight
Final Airplane Construction
Manufacturing Plan
Structural Design&Analysis
Aerodynamic Refinement
Detailed Design Phase
Refined Analysis
Prototype
Preliminary Design Phase
Analytical Analysis
Component Analysis
Conceptual Design
UB AIAA DBF Team
6/18/2003
8/7/2003
9/26/2003
11/15/2003
1/4/2004
2/23/2004
4/13/2004
6/2/2004

Table 2.3 Gantt chart showing timeline for project

3.0 Conceptual Design

The conceptual design process began with the forming of small teams of people to research different components of the airplane as seen in Table 2.1. The most important factor in this part is to stay within the competition guidelines, which are as follows:

- Wingspan is unlimited, but the whole airplane must be able to fit inside a 4x2x1 foot box.
- Battery weight is limited to 5 lbs.
- Gross takeoff weight must be less than 55 lbs.
- Motors must be brushed, from Astroflight or Graupner.
- A single motor can draw no more than 40 amps.
- Airplane must take off within 150 ft.

A Figures of Merit chart (Table 3.1) was used to determine an optimized configuration for this stage of the design process. The factors that were taken into account were aerodynamics, performance, manufacturing, and structural integrity. A single engine monoplane with a twin tail and tapered wing was selected. This configuration coincided well with what the team had in mind for a simple, fast plane, that would be able to complete any of the three possible missions within the allotted time.

		Aerodynamics		Manufacturing			Performance		Stucture		Total
		Lift	Drag	Time	Cost	Ease	Stability	Control	Strength	Weight	
	Weighting Factors	8	5	7	5	9	8	6	7	9	
Propulsion	Single	N/A	N/A	10	10	10	10	10	N/A	10	440
	Double	N/A	N/A	7	4	6	10	8	N/A	5	296
Wing	Mono plane	7	10	10	10	10	10	10	10	10	614
	Biplane	10	6	7	5	7	10	10	8	5	414
	Flying Wing	7	9	8	9	8	7	6	10	8	493
Tail	T-Tail	N/A	10	9	10	9	10	10	8	8	510
	V-Tail	N/A	10	7	8	7	9	7	7	7	428
	Twin Tail	N/A	10	8	8	10	10	10	10	10	536
	Conventional	N/A	10	10	10	10	7	9	9	8	505
Fuselage	Cylindrical	N/A	8	7	7	4	10	N/A	10	8	382
	Blended wing	N/A	10	8	9	7	8	N/A	10	10	436
	Box shaped	N/A	4	10	10	10	10	N/A	7	8	431
Landing gear	Tricycle	N/A	10	8	9	8	N/A	10	10	9	424
	Tail dragger	N/A	10	8	10	10	N/A	6	8	10	423
Wing Configuration	Straight	10	5	10	10	10	10	10	7	8	574
	Swept	7	10	7	7	6	7	9	9	9	494
	Taper	8	9	8	9	8	10	10	10	10	580
Empannage	Single Boom	N/A	7	10	10	6	8	N/A	6	10	405
	Twin Boom	N/A	9	9	9	8	10	N/A	10	9	454
	Fuselage blended	N/A	8	8	8	7	10	N/A	10	6	406

Table 3.1: Figures of Merit Chart for Conceptual Design phase

3.1 Wing

The goal of the wing design is to find the best wing and airfoil configuration to meet the airplane's needs and be within reasonable construction restrictions. These include a wingspan of no more than 10 feet, and an airfoil thickness around 12% for structural strength. The total loaded airplane weight was estimated to be around 32-pounds, therefore the minimum lift provided by the wing in level flight must be equal to that amount. This would require a significant lift coefficient at close-to-0 angle of attack, which eliminates symmetrical airfoils from consideration.

A tapered, single wing design was determined to be necessary due to its structural strength and low induced drag. Due to this choice, the airfoil used would have to be efficient at a large range of Reynolds numbers, averaging at around 700,000. The airfoil must also be of a relatively easily manufactured shape, which would provide enough thickness at all points of the airfoil so as to not require any special structural materials other than the foam core and carbon-rod spars. Any other material requirements would increase production cost, and could also result in greater overall weight.

From that point on, the choices for the wing configuration were basically restricted to the degree of the taper, the aspect ratio, and the dimensions of the wing root and tip chord lengths.

3.2 Tail

The main function of the tail is to provide pitch and yaw stability for the aircraft. Several configurations were considered in an effort to minimize cost, weight and construction time. Conventional tail, V-tail, T-tail, twin tail, and tailless designs were considered. The tailless design was the first to be eliminated because the amount of analysis and fine-tuning required for such a design would be excessive for our team considering the design timeframe as well as computational requirements. It would provide very significant weight savings, however all control of the aircraft would have to be done through the ailerons, which would require software control in order for the flight to be stable. The T-tail design is an aerodynamically efficient design because the control surfaces are outside the wake region of the wing, however it results in a heavier tail since the vertical stabilizer has to be reinforced in order to support the horizontal one on top of it. The V-tail was discarded as well in favor of simpler construction, alignment and analysis of the more conventional designs. Finally the twin tail design was selected over the conventional tail, due to the fact that both vertical surfaces would be outside of the propeller wake, as well as the simulated antenna, which creates a very significant wake. Since the attachment to the fuselage is in the form of two rods going to the tail, the two vertical surfaces can be attached directly to those rods, thereby not requiring any extra structural reinforcement of the horizontal stabilizer in order to support them.

3.3 Fuselage

The word fuselage is based on the French word 'fuseler', which means, "to streamline." The fuselage is the main body of an airplane, customarily streamlined in form. A plane can do without a fuselage like earlier planes did. However, since we need to carry the battery packs, the cargo, and control instruments like servos, a fuselage is an essential part of our design.

The fuselage needs to be designed in such a way that minimizes construction time and is also highly optimized in a sense that creates drag values as low as possible without violating the rules of the mission. To meet the above-mentioned criteria, a sleek shape of fuselage is needed. The fuselage has to be designed with enough strength to withstand torque generated at the center of gravity by ailerons, rudder and elevator. Also the construction of the fuselage should be within the allotted budget. Strength is also a major area of concern. The fuselage acts as the main connection for the wings, tail, and landing gear. In addition, it carries the payloads, electronics and other various items. Composites will be used to great degree in this area in order to optimize the strength to weight ratio. Last but definitely not least, the fuselage has to either be able to fit completely inside of the box, or it has to be able to be disassembled in order for everything to fit inside the box.

3.4 Propulsion

The goal of the power plant design team is to select a configuration of motor, batteries, and propeller that best meet the needs of the airplane while meeting the design constraints. These constraints include required use of Nickel-Cadmium batteries, a choice of two specified motor manufacturers, a maximum battery pack weight of 5 pounds, a maximum current of 40 amps through a single motor, and a maximum takeoff distance of 150 ft.

The power plant design basically consists of deciding on several parameters relating to the motor, batteries, and propeller. The decisions relating to the motor include the manufacturer, size, and number of motors to be used, and additional gearing. These choices must be made while taking into consideration the estimated overall aircraft weight, the maximum takeoff distance of 150 feet, and the 40 amp current limit through a single motor.

The decisions relating to the batteries include the manufacturer, type, number, and configuration of the battery packs (such as series or parallel). These choices must be made while taking into consideration the maximum battery pack weight of 5 pounds, the voltage required by the motor selected, and the maximum current draw the motor will need.

The decisions relating to the propeller include the manufacturer, propeller diameter, pitch, and material. These choices must be made while taking into consideration the power output of the motor, and the desired top speed of the airplane.

After computing the figures of merit chart shown in figure 3.1 the design was simplified to a single motor. However, this still left many parameters to decide upon.

4.0 Preliminary Design

The design parameters, which were of lesser consideration earlier, now play a vital role in determining the configuration of the airplane. Ideas discussed in the conceptual phase must be refined to obtain solid data on predicted airplane performance.

		Aerodynamics		Manufacturing			Performance		Structure		
		Lift	Drag	Time	Cost	Ease	Stability	Control	Strength	Weight	Total
	Weighing:	8	4	7	5	9	8	6	7	9	
Propulsion	single	N/A	5	10	10	10	5	7	N/A	10	402
	push-pull	N/A	10	8	6	9	10	10	N/A	7	410
Wing	Monoplane	7	10	10	10	10	10	10	10	10	606
	Biplane	10	6	7	5	7	10	10	8	5	482
	Flying Wing	7	9	8	9	8	7	6	10	8	499
Tail	T-tail	N/A	9	9	10	9	10	10	8	8	498
	V-tail	N/A	9	7	8	7	7	7	7	7	398
	Tailless	N/A	10	10	7	6	4	4	8	10	401
	Conventional	N/A	9	9	10	10	8	9	10	10	517
	Twin-Tail	N/A	9	10	10	10	10	10	10	10	546
Fuselage	Airfoil	N/A	9	7	7	8	10	N/A	10	10	432
	Trapezoid	N/A	10	8	9	7	8	N/A	10	10	428
	Rectangular	N/A	4	10	10	10	10	N/A	7	8	427
Langing Gear	Tricycle	N/A	10	8	7	8	N/A	10	10	10	423
	Tail-Dragger	N/A	10	8	10	10	N/A	6	8	10	418
Wing Config.	Straight	10	5	10	10	10	10	10	8	8	578
	Swept	7	10	7	7	6	7	9	10	10	504
	Taper	9	9	8	9	9	10	10	10	9	581
Empennage	Single Boom	N/A	7	10	10	6	8	N/A	6	10	398
	Twin Boom	N/A	6	9	9	10	8	N/A	10	9	437
	Tapered	N/A	10	8	8	9	10	N/A	10	7	430

Table 4.1 Preliminary Figures of Merit

4.1 Wing

The tapered wing was chosen because it provides greater strength at the wing root where the twisting and bending moments due to the lift and drag on the wing are greatest. The majority of the lift generated by the wing would also be toward the center, creating a smaller moment about the center section of the wing, thereby reducing the amount of stress it would have to be able to withstand. Tapered wings also have a much lower induced drag, due to the smaller tip chord length, which is where the tip vortices occur. The main disadvantage of this design was the varying Reynolds number along the length of the wing, due to which the wing tips would have a lower Reynolds number and would therefore stall at

a lower angle of attack by a few degrees. This is highly undesirable since it would be impossible to make the airfoil shape of both tips exactly the same, so one would inadvertently stall before the other, and due to the large moment arm the tip has with respect to the fuselage, this would immediately result in a rather strong roll.

To avoid this, there were two choices, either have a wing twist so as to have the wing tips at a lower angle of attack by a few degrees compared to the root, or increase the wing tip's leading edge radius, which results in more drag, but increases the stall angle of the tip. It was determined that the manufacturing of the wing twist would be easier, especially since the basic shape could be easily cut out from foam, and also there was no exact way of determining the magnitude of the leading edge radius required to increase the stall angle by the desired amount.

The search for the optimal airfoil involved looking up a large number of different airfoils designed for a Reynolds number of the same magnitude as the range applicable to our aircraft, then testing it virtually for the full range of Reynolds numbers applicable to the wing design. The polar graphs of the lift coefficient vs. the drag coefficient were then compared with the Clark Y airfoil since it was the airfoil used in our previous designs, and a proven performer for small aircraft. Airfoils with difficult to manufacture shapes or insufficient thickness for structural strength were discarded from consideration.

The Eppler 387 airfoil was determined to be optimal for our needs, since it provides the same or greater lift coefficient as all other candidates, while at the same time retains a very low drag coefficient of under 0.01 over a rather large range of angles of attack. The profile of this airfoil and its polar graph are shown in figures 4.2 and 4.3 respectively. Its' lift coefficient at 0-angle of attack was 0.4, and the drag was 0.0061, which is very good for cruising conditions, although it might still be required to have a small angle of attack at cruising speed, depending on the final weight of the plane, in order to provide sufficient lift.

The stalling characteristics of this airfoil were average, and stall occurred at around 14-degrees at the wing root, and 12-degrees at the wing tip. The stall was not too abrupt since maximum lift occurs at around 10-degrees angle of attack, thereby providing somewhat of a feedback to the pilot about the aircraft's proximity to the stall angle. The thickness of the airfoil was 12%, which is sufficient for structural strength by providing enough room for the spars. The camber was 3.78%, and with the profile being of a rather simple shape, there should not be much manufacturing difficulty, with the exception of the wing twist. However the wing twist would have been required for pretty much any airfoil since there is no airfoil that would retain the same stall angle over the required range of Reynolds numbers.

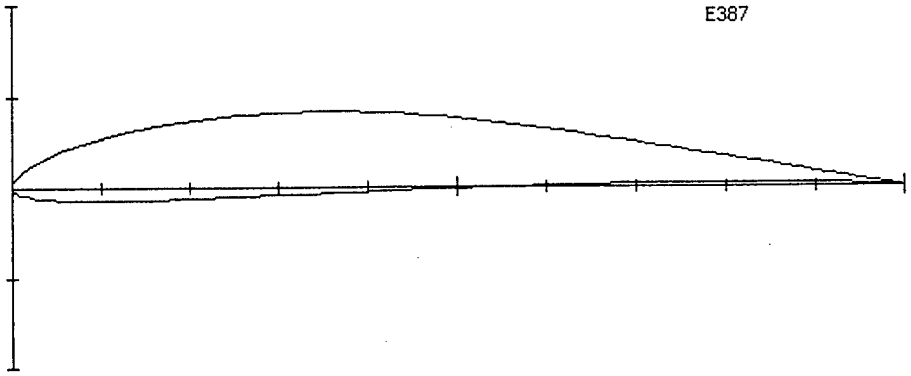


Figure 4.2 Eppler 387 airfoil [6]

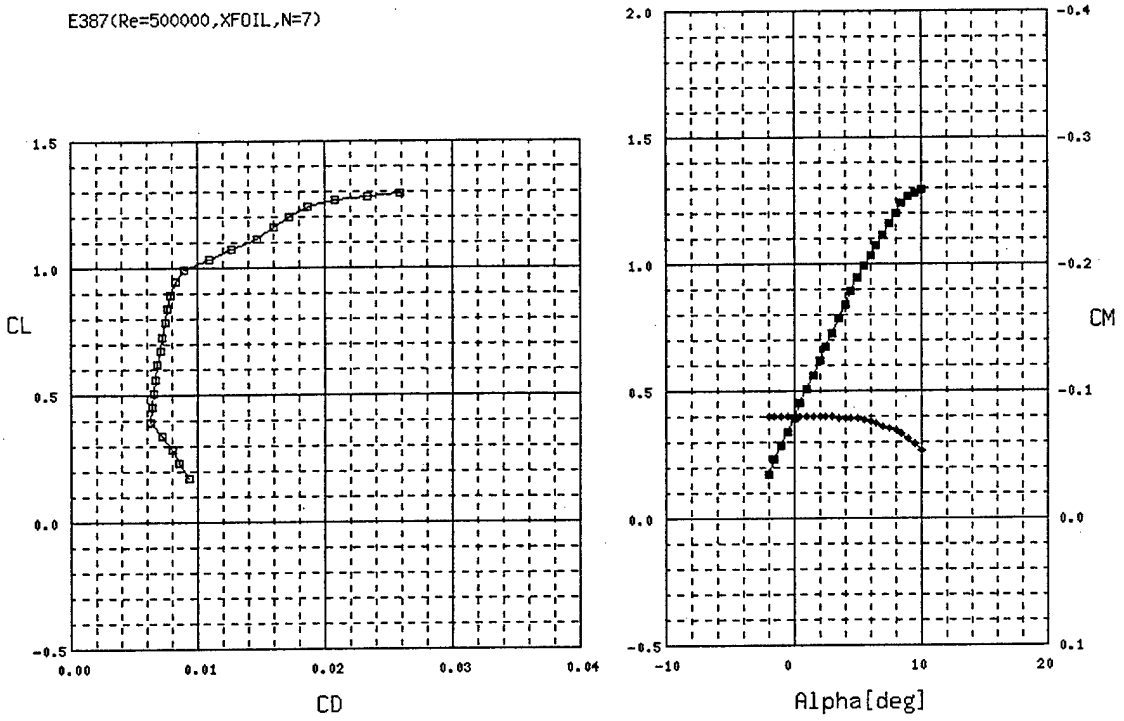


Figure 4.3: Eppler 387 lift coefficient vs. drag coefficient polar graph [6]

When compared to the Clark Y airfoil, the Eppler 387 was a definite improvement as can be seen by comparing their polar graphs in figures 4.3 and 4.4. It retains a larger lift coefficient over the full range of angles of attack, while keeping a considerably smaller drag coefficient at all times. The only disadvantage is the Eppler's rather abrupt changes in drag when it goes into a negative angle of attack as well as at around 6-degrees angle of attack. However this was determined not to have too great of an effect in terms of control, and since the drag coefficient still remains lower than that of the other airfoils, it

does not result in any relative performance losses. Also, for the most part, the plane should never require going into a negative angle of attack, unless an emergency occurs.

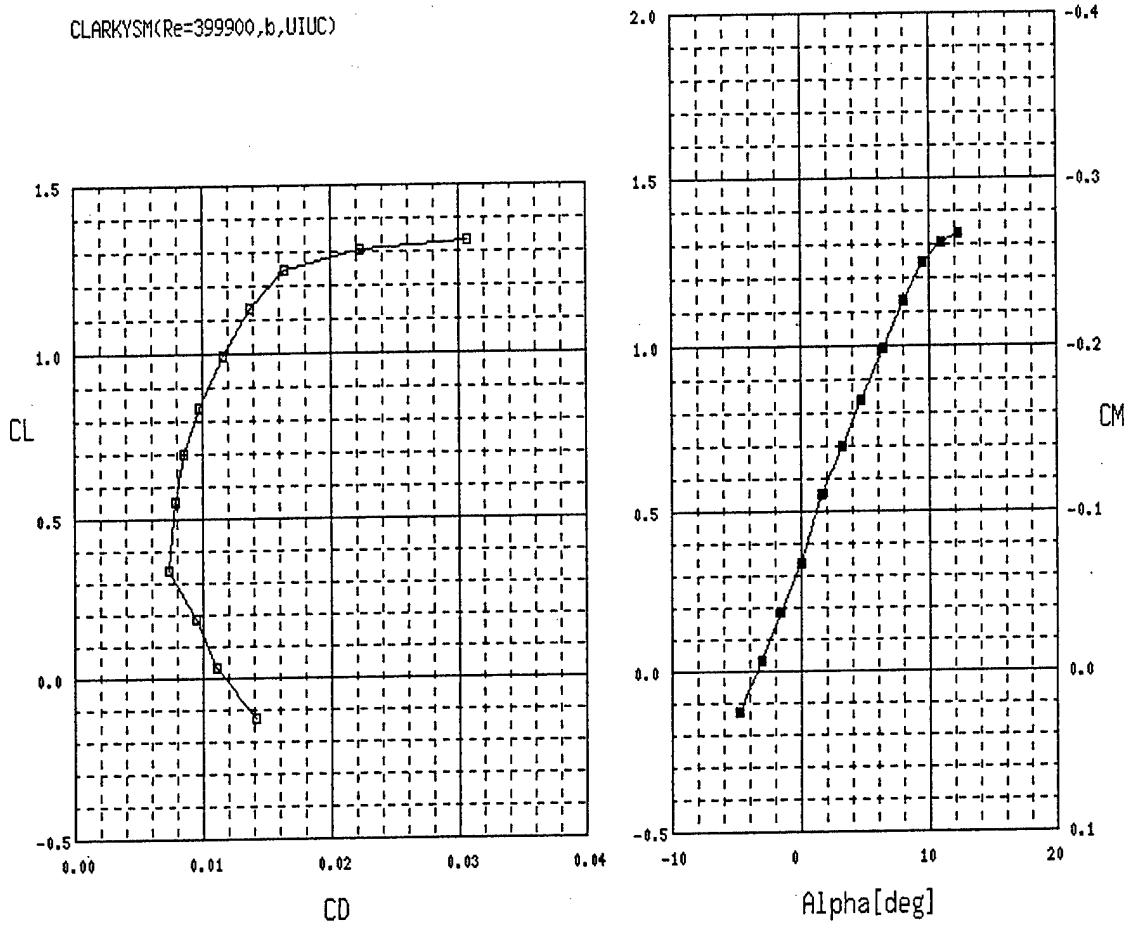


Figure 4.4: Clark Y lift vs. drag coefficient polar graph [6]

Since all of the data used for comparison of the airfoil performance was from virtual sources, it was necessary to determine the validity of this data when compared to real-world results. This was done by comparing the virtual polar graphs to those determined experimentally in the Langley Low-Turbulence Pressure Tunnel[6]. The experimental graphs matched almost perfectly the virtual graphs up to a Reynolds number of 460,000, and it was also stated in the technical report used that for larger Reynolds numbers, the experimental results match perfectly the theoretical, i.e. virtually determined, results.

The optimal wing configuration was determined to be: a wingspan of 9.5ft; root chord length of 22.67 inches (so as to have a taper ratio of 0.5); and a tip chord length of 12 inches. This results in a wing area of 13.5ft^2, which means an estimated wing loading of 36 ounces/ft^2. This number is easily achievable, however exact lift calculations for this wing design are still underway. The taper ratio for this

configuration is 0.5, which is a decent amount of taper, and results in a significant amount of drag reduction. A very good taper ratio would have been 0.2 or 0.3, however there was no way of achieving such a ratio without either decreasing significantly the aspect ratio, having an impossibly large root chord length, or a structurally very weak and small tip chord. Since the aspect ratio has a greater influence on total drag, it was determined to be much more desirable to have it at 5.7 compared to a slightly smaller taper ratio.

4.2 Tail

Once the configuration of the tail is decided, the next step is to find the shape, area, and location of the vertical and horizontal surfaces. The size and location of the tail can be related to the dimensions of the wing to give the tail moment volumes, which in turn can be used to calculate stability. The range of vertical and horizontal tail coefficients are chosen by comparison with similar planes with similar configurations. While staying in this range, the higher end of the spectrum corresponds to higher stability and greater flexibility for the placement of the center of gravity, whereas the lower end of the spectrum corresponds to lower stability and gives smaller margin of error in the placement of the center of gravity. A higher stability value is desirable, especially in heavy wind conditions, however it also makes the aircraft less maneuverable, thereby making the determination of the moment volumes an optimization problem.

In order to reduce the induced drag, the tail's aspect ratio should be quite large, but no greater than one quarter of the wingspan, since then it would count as a second wing, and seriously increase the cost of the airplane. For a wingspan of 115.5-inches, the horizontal stabilizer would have to be 28.5-inches. The surface areas of the horizontal and vertical stabilizers were determined using the following equations, which use volume coefficients that are typical for most existing aircraft of this size:

$S_h=V_h*S*c/x_h$ and $S_v=V_v*S*b/x_v$

Where S_h and S_v are the horizontal and vertical stabilizer areas respectively; V_h and V_b are the volume coefficients; c is the mean chord length of the wing equal to 19-inches; b is the wingspan at 115.5-inches; and x_h and x_v are the horizontal distances from the wing's aerodynamic centerline to the tail surfaces' aerodynamic centerlines, both of which are equal to 54.5-inches; finally, S is the wing area at 1944 square inches. The volume moment coefficients were picked to be the average of a series of 13 homebuilt aircraft, since homebuilts are of similar proportions as this airplane, even though this one is not designed for any passengers. The V_h used was 0.47, and V_v was 0.036. From the above equations, the required horizontal surface area was determined to be 316 square inches and the vertical one was 140 square inches. If the span of the horizontal stabilizer were to be kept under 28.5 inches, then the chord length would have to be about 11 inches in order to meet the 316 square inches area requirement. For ease of construction, the vertical stabilizers will be half of the tail span, and the half the chord length and profile for reduced drag, thereby meeting its' 140 square inches area requirement.

The profile of the tail surfaces was chosen to be symmetrical, since even though extra lift is usually desirable, it would require the center of gravity at a location further back than its current one.

Since the tail is in the wake of the wing, the flow over it is quite turbulent, which would cause the lifting force from the tail to vary, thereby varying the center of gravity's moment on the plane, and therefore resulting in an unstable flight. The NACA 0012 airfoil was chosen as the tail profile due to its low drag coefficient, and the fact that it provides sufficient room for structural strength with its' 12% thickness.

4.3 Fuselage

Since the construction of the fuselage should be possible with the available resources, simple geometric shapes were studied. The cylinder was found to be most optimized option for fuselage shape. The idea was discarded due to the accuracy and skills needed for construction.

The team members didn't want to spend time on constructing something that they weren't sure how accurate it will turn out. A rough cylinder shape would have contributed significantly to the drag due to irregularities on the surface. Therefore circular shapes were eliminated as a possibility for our design.

The team then decided to investigate quadrilateral shapes due to the ease of construction. Square, rectangles and trapezium shapes were analyzed using simple mathematical skills. Trapezium was found to be aerodynamically better than square and rectangle shapes on the bases of aerodynamics and on past experience of some of the members of the team who were involved in airplane modeling. There were some tradeoffs with that particular shape. It was difficult to taper the fuselage on the end. It was important to taper at the end because the back part wasn't really needed as it contributed to weight and drag and thus putting load on the engines, which resulted in lower net thrust. The empennage should be tapered to provide enough surface area for the horizontal and vertical stabilizers. Also with the slant sides, the part of the wing that gets attached to the fuselage needed a smooth slant. That meant spending more time on the construction.

The problem of fitting the fuselage inside the rectangular box was then considered, and it was decided that the fuselage should be no more than 4 feet long, with only the two carbon rods, that would make up the fuselage's structure, extending all the way back to the tail. Such a design would reduce the weight and drag of the fuselage. This meant that the fuselage would have to taper off rather quickly from a considerable thickness at the center of gravity, where the payload would have to be able to fit, especially since there is no way to have the payload between the wing rods, which must run through the fuselage unbroken. Therefore room for the payload would have to be made underneath or above the wing rods, extending the fuselage considerably below that point since the water tank would require to be placed in an area where there are no rods in order to more easily seal its inside surfaces. It was therefore decided that the fuselage would be of an airfoil-like shape made up of foam, with the carbon rods as its' structure. This would significantly reduce the drag, as well as resulting in some more lift, especially around the areas where it would taper-off into the wing airfoil. This design provides significant manufacturing difficulty, however the team feels confident enough in their skills to be able to at least sand down the foam to the right shape. The gains in aerodynamics would definitely be worth the effort required to manufacture this design. The middle of the fuselage would be in the form of a cylinder in order to fit the

two motors. The drag generated from that area is not of high concern due to the pusher motor in the back. Any boundary layer separation in front of the pusher propeller would only act to decrease the pressure before it thereby increasing the pressure difference across the propeller and resulting in greater thrust compensating for the added drag. The water tank would be permanently attached to the underside of the fuselage by carbonfibering them together. The actual container is a cylinder cut out of foam with two half cones in the front and back in order to streamline the flow around it.

4.4 Propulsion

The propulsion system consists of two motors, propellers and a battery pack. Selection of each was based on competition rules and considerations listed in the conceptual design section.

In addition, an issue was raised in regards to properly testing the static force output of the motor. This was resolved by using a motor test stand from previous years, which was machined and assembled by the team.

4.4.1 Motor

The motor selection starts with the estimated required motor power. From practical experience, we decided that 40 watts per pound of total aircraft weight would give the power we needed to takeoff within 150 feet. For a conservative estimated aircraft weight of 35 pounds, this results in a motor power output of 1400 watts. We also wanted a single engine configuration because of its low rated aircraft cost contribution, and the success of single engine teams in previous competitions. These considerations resulted in the selection of the largest motor allowed, the Astroflight Cobalt 90. The motor performance specifications are listed in Table 4.1

Model	**690**	**691S**
Name	CO-90 Direct	CO-90 Geared
Gear Ratio	Direct	2.75 to 1
Arm Winding	10 turns #22	10 turns #22
Resistance	0.111 ohms	0.111 ohms
Speed	256 rpm/volt	93 rpm/volt
No Load	3.0 amps	3.0 amps
Best Battery	32 to 36 cells	36 to 40 cells
Max Amps	35 amps	35 amps
Prop Rpm	8,000/10,000	3,500 to 4,500
Best Props	13x8 to 16x8	20x14 to24x12
Power Watts	1100 to 1200	1200 to 1400

Table 4.1 Manufacturer's Specifications

This motor has a maximum power output of 1400 watts when connected to 40 cells. This motor was the only one selected from the two manufacturers with a power output in the range that we needed. However, the motor comes with or without a 2.75 to 1 gear ratio, so this aspect needs to be taken into consideration when selecting the front propeller. For the pusher motor, the Cobalt 60 was selected due to availability of a motor from previous years as well as known performance with a 16x10 propeller and current draw characteristics.

4.4.2 Batteries

The most important factor in the selection of the batteries is the capacity to weight ratio. Because of the 5-pound weight limit on the batteries, we must make the most use of that weight as possible. Weight and capacity specifications were obtained for every battery from every major Nickel Cadmium battery manufacturer. This resulted in a list of 341 different Ni-Cd cells. This data was entered into a spreadsheet program and the cells were sorted according to capacity to weight ratio.

Next, the cells that are manufactured specifically for high current discharge use were identified. This resulted it the selection of the DSC2100P cell manufactured by BYD. It can handle the current expected in our power plant system while still having a high capacity to weight ratio of 42 mAh/g. This cell is also an excellent choice because 42 cells result in a weight of slightly less than 5 pounds and a voltage output of 50. This is the best voltage for high motor power output.

4.4.3 Propeller

With the motors and batteries fixed, suitable propellers must be selected that satisfy the power output of the motors and the needs of the mission. A useful computer program called "ThrustHP" was utilized in the selection of a range of propellers to be evaluated for further testing. The motor manufacturer suggests several propeller sizes for both the geared and direct drive versions of the motors. An estimate of 1200 watts of max power output for both versions of the front motor was used in the program and resulting rpms, static thrust, and speed values were calculated for propellers of varying diameters and pitches. Another factor taken into consideration was the aerodynamic effect of a small propeller diameter spinning in front of a fuselage of almost the same area, which would be the case of the direct drive motor. This analysis led to the conclusion that the geared version of the motor would be used along with a propeller with a diameter of 16. This resulted in acceptable speed, rpm, and thrust values over a range of propeller diameters. The diameter can then be varied to increase the thrust and decrease the speed, or vice versa. Since the program is just a tool and the actual rpm and thrust values had to be determined experimentally, several propellers with diameters between 20 and 24 inches were purchased for testing. For the pusher motor, the available space between the two rods that connect the tail is constraining the maximum propeller diameter to 16 inches. A smaller diameter would not capture enough of the flow around the fuselage due to the fuselage's size. In order for the propeller to accelerate the flow

enough to justify its use behind the other propeller that already results in a fast airflow behind it, a minimum pitch of 10 is required. The final balance of thrust and speed can then be selected with the help of static thrust measurements, especially with the two motors aligned one behind the other, to ensure that the aircraft takes off within 150 ft. APC was selected as the propeller manufacturer because of the amount of positive opinions of other modelers as to their performance and strength.

4.4.4 Motor Test Stand

An important aspect of our design process is being able to determine the thrust of our motor with the use of different propellers. In order to maximize this element in our design, we must be able to get accurate test results in the convenience of our own labs. Therefore, we needed to design a motor test stand to work with many different motor and propeller characteristics. Throughout the design of the test stand, many structures were considered and were finally narrowed down to one universal design. The following page illustrates the metamorphosis of the design from its conceptual phase to the final product.

The design of the stand was based on the two main functions it needed to allow for: unique motors (sizes, weights, etc.) and different diameter propellers. We needed a stand that could be used with many different formats and that would be durable enough to last for years to come. In order to measure the thrust results from each type of propeller, the stand is on wheels and is attached to a scale by a string. When the motor is powered up, we can see how much pull the scale receives and take our measurements from that.

The first design (Figure 4.2a) was intended for a specific motor and an overall propeller diameter. We then realized the value in making the stand capable of adapting to different motors. The original structure was then altered to the specifications in (Figure 4.2b). This stand has an adjustable upright portion, a sliding track-rod-system that can fit any size motor we might require for competition. With a few more changes we arrived at the design shown in (Figure 4.2c). This was brought about in the interest of simplicity. A single plate design with angled support seemed to be much easier than the rigid system of part (b). With this design we headed to the machine shop to consult with the shop attendants. The main concern brought up with our design was stability and weight. With the motor loaded, there was a concern that the stand would tip forward and become a safety hazard. Also, with a single plate system, the overall design may be too heavy to get accurate thrust results. Therefore, we retorted back to a form of design (b) as seen in the three-view drawing on page 14. This system is composed of angled aluminum beams, as opposed to single sheets, over the entire structure. The track with horizontally adjustable rods to hold the motor in place is simply attached between the two upright beams. This track in turn is capable of moving vertically up and down to get better results for smaller propellers.

After the motor test stand is completed, the motor will be tested using the chosen battery pack and various configurations of propeller diameter and pitch, as well as with or without the gearbox. From this data the best configuration resulting in desired thrust and speed can be chosen.

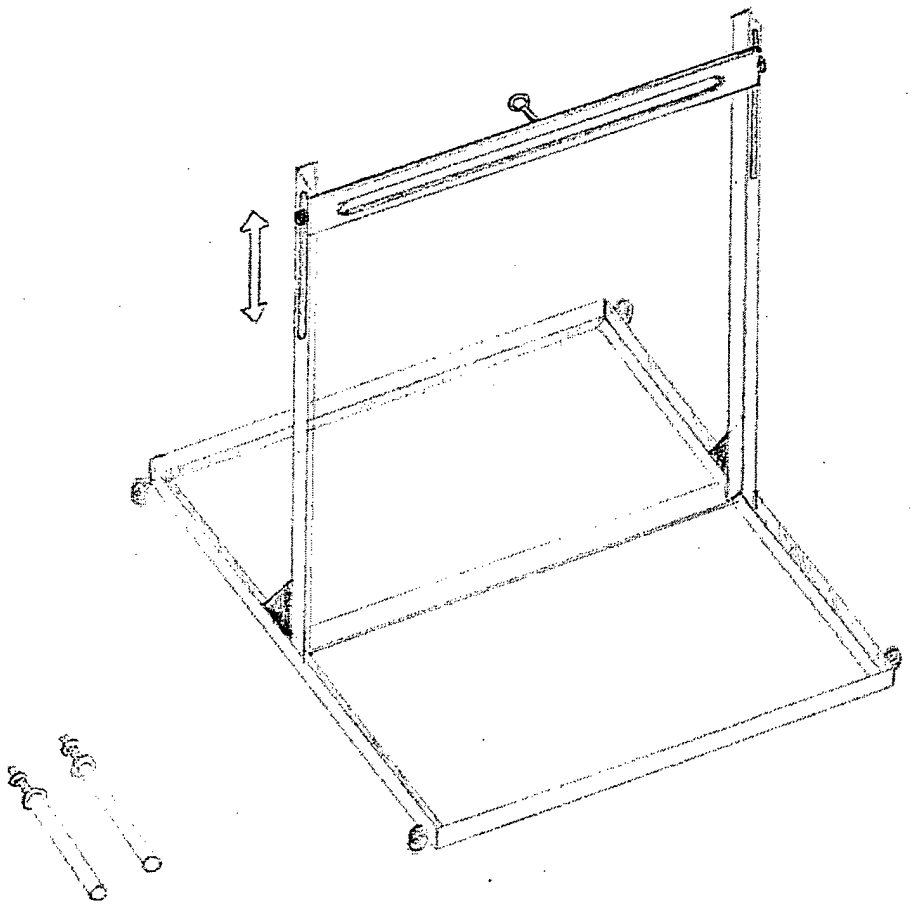


Figure 4.3 Final Motor Test stand design

5.0 Detailed Design

The results from the preliminary design phase are now refined and final dimensions and structure are generated. Figure 5.1 shows the final configuration of the aircraft. Table 5.1 shows the final rated aircraft cost.

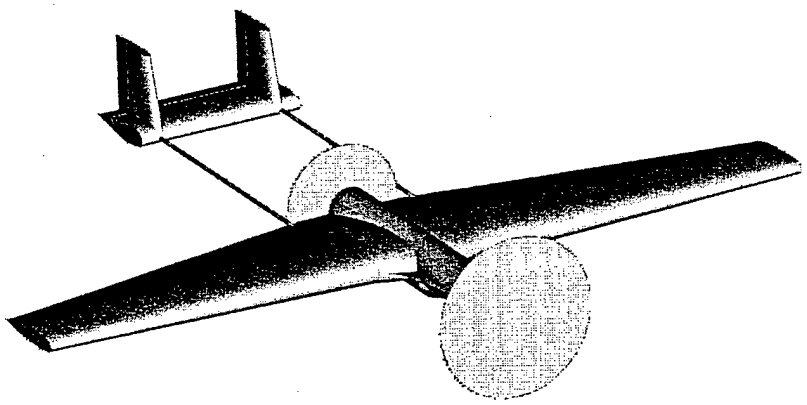


Figure 5.1 Exterior View of Final Design

Rated Aircraft Cost	$27,443		
		Cost:	Man Hours:
Airframe Empty Weight (lbs)	32	$9600	
Batteries Weight (lbs)	5		
Rated Engine Power	6.25	$9375	
Fuselage Total (ft^3):	7.92	$3168	158.4
Body Height (ft)	0.66		
Wing Span (ft)	9.5	$3800	190
Max Wing Chord (ft)	2		
Control Surfaces	3	$300	15
Body Length (ft)	8		
Vertical Surfaces (no active control)	0		
Vertical Surfaces (active control)	2	$400	20
Horizontal Surfaces	1	$200	10
Servos and Controllers	6	$600	30
Engines	2		
Body Width (ft)	1.5		

Table 5.1 Rated Aircraft Cost

5.1 Wing

The structure of the wing is comprised of a foam mould, in the shape of the desired wing size and profile stated in the preliminary design, which is covered in carbon fiber at key stress areas, which is basically the bottom surface of the wing, and has a spar going along its lower part, which is to take the main load of the wing. The spar consists of two carbon-fiber rods, which go the length of the wing, and are epoxied inside pre-cut holes in the foam of the wing. The rods are 7 inches apart, 4 foot long, and are joined to the rods in the other wing by means of two 22 inch long rods that go through the fuselage. These attachment rods are exposed 4 inches on each side outside of the fuselage and attach to the other rods using bolts that are epoxied at each end, and screw into nuts that are epoxied 2 inches deep inside the wing rods. The bolts and nuts on one side of the fuselage are counterclockwise, so that when being assembled, the crew only has to turn the rod clockwise with respect to the left wing for two turns, which will be more than enough to hold the wings tightly to the fuselage. This joint has been tested up to 25lb/ft length of rod loading without failure. The joining rods themselves are able to move freely inside round tubes of carbon fiber attached at the ends of the fuselage. The outside of the wing is only covered with

carbon fiber at the bottom surface. This is so as to save weight, since the epoxy used as the matrix for the carbon fibers is extremely heavy, and the smaller the area covered with it, the better. In terms of strength, this configuration will be about as strong as when the whole wing is covered, as long as the loads applied are aerodynamic and landing gear-applied loads, since these loads act mostly on the bottom surface of the wing. This should be the case most of the time, unless the plane crashes. When these loads are applied, the bottom surface will be in tension, and since carbon fiber is good in tension and not compression, as would be the case for the top surface, this configuration is about the same strength as when the top is covered, except it is about half the weight.

The wing thickness at the end of the wings is much smaller than at the root, and therefore cannot fit the larger size rods used in the middle of the wing. Therefore a telescoping design is used, where the larger, 0.68 inch rods extend up to 3 feet into the wing on each side, at which point a smaller, 0.635 inch-diameter, rod extends the structure. The extension is made by covering a 6-inch end section of the smaller rod with a carbon dust/epoxy mixture, and then placing that section of the rod inside the larger rod, creating a bond area stronger than any of the rods by themselves.

Since the fuselage is a variable airfoil shape, the wing meshes smoothly into the fuselage, while the spar rods go through it using the extensions, thereby creating a very strong, but easily assemblable structure, without the need for extra weight in the form of a connecting surface, and reduces the frontal profile area of the plane. The wing structure is being designed to withstand a 5-g load.

The ailerons start at the fuselage, and extend only 40-inches into the wing, since the wing becomes structurally weak at the tips and cannot support the moment created by the aileron. The servos controlling the ailerons are located 20 inches into the wing, and are directly connected to the wing spar rods, thereby not requiring any extra structure, or extra long push rods extending from the fuselage that would increase weight significantly.

The rear landing wheels are located 4 inches behind the center of gravity, attached directly to the wing spar, 1.5 feet from the centerline on each side. This configuration provides reasonable stability against roll during landing, and prevents the risk of the plane tipping back at landing, since the angle pitch required to accomplish that would have to be greater than 15 degrees, which is rare, and the tail would also touch the ground at that angle. This distance is optimal since if the rear wheels were to be placed further back, it would result in the center of gravity applying too great a moment about them, and requiring a large amount of horizontal stabilizer deflection in order to achieve liftoff, which would create unnecessary drag.

5.2 Tail

The tail's horizontal stabilizer is constructed out of a balsa wood rib structure, covered with monokote plastic fabric, and balsa sheeting at the higher stress area where the tail attaches to the rods from the fuselage. The ribs are connected by two balsa wood spars, one on the bottom of the rib and another on the top, and leading and trailing edge balsa wood pieces. The vertical stabilizers are

constructed from foam covered with carbon fiber wrapping around the rods that attach to the fuselage in order to secure them tightly, which has otherwise proven to be a problem in the past. The vertical stabilizers are 10-inches high starting from the top of the horizontal stabilizer which is 24 inches wide. All tail stabilizers are 12 inches long, which includes 4 inches for the actual control surfaces, providing ample stability and control for the plane considering the large lever arm to the center of gravity. The connection of the tail to the fuselage is made by the use of two more junction rods, which are 4 foot long, with the same design of bolts and nuts epoxied inside them as the wing connection rods. These attach to two other rods at the tail end, which are epoxied inside the tail under each vertical stabilizer, and two others epoxied at the ends of the airfoil section of the fuselage. The vertical stabilizers are attached to the rods by carbonfibering them together, with the horizontal stabilizer running between the rods.

The rib structure of the horizontal stabilizer is only 8-inches long, at which point the carbon rods, to which it is connected, end, and the elevator control surface starts. The elevator runs the length of the horizontal stabilizer and is 4 inches wide, which can provide a very large moment about the center of gravity, for greater control. The rudders and vertical stabilizers are constructed the same way and with the same dimensions. The servos controlling the horizontal and vertical stabilizers are located in the middle of the horizontal stabilizer, one at the top and one at the bottom to prevent interference between the push-rods. Both of the vertical stabilizer control surfaces are connected to the same servo.

5.3 Fuselage

The dimensions of fuselage were calculated on the basis of space needed for the water as well as for control instruments and the carbon fiber rod structure. The total length of the plane was found to be 99 inches, with the fuselage only accounting for roughly the front 43 inches, excluding the propeller shaft. The width along the centerline of the fuselage is 18 inches. The wing is placed at a distance of 22.8 inches from the front. The center of gravity of the airplane lies roughly 26 inches from the front. The front landing gear is 13 inches high and connected at 3 inches from the front, providing support directly to the front motor. The rear landing gear is a pair and fixed under the wing at a distance of 31 inches behind the front one. Aluminum is used to connect the motor to the carbon fiber rods. The motor is placed at 3.7 inches from the front end and in the middle of the rods. The length of the motor is about 4.7 inches. The fuselage front tapers off into the motor, while the back tapers off to the rear motor. The batteries will be placed on the top surface of the fuselage most likely near the motor. They can be adjusted along the length of the fuselage in order to have location of center of gravity at 26 inches from the front, which is the optimal location for a relatively stable flight on a tapered wing of these dimensions. The water tank, a half cylinder, is placed starting 19 inches from the front so as for the water to be centered at the center of gravity and not affect it once the water is released. Fairings of a half-cone shape are attached at the front and rear of the tank so as to guide the airflow around it more efficiently. The water is released using a servo attached to a valve from a water jug of half inch inside diameter, which when pulled out lets the

water flow out freely. The tank is refilled from an opening in the top, which also contains the receiver and servo battery pack.

5.4 Propulsion

Proceeding into the detailed design phase the motor and batteries were selected and a range of suitable propeller sizes were determined and purchased. With the motor test stand built the actual performance of these propellers was determined to aid in selecting the ones with the best balance between high takeoff thrust and high cruise speed. The best propeller was determined to be 20 inches in diameter and a pitch of 14 for the front motor and a 16 inch, 10 pitch propeller for the rear motor. At maximum power the front propeller spins at 4700 rpm, which is a little beyond what the motor manufacturer suggests but is still acceptable, while the rear motor spins at 14000 rpm. The total static thrust was about 14 pounds, which results in an estimated power to weight ration of .43 for a 32-pound aircraft. This should be more than enough for takeoff within 150 ft, but if it is insufficient we can simply switch to a larger diameter. Estimated cruising speed using these propellers is 40 mph.

The battery packs will consist of two shrink-wrapped sets of 20 cells connected in series which will result in a combined weight of just under 5 pounds. The pack is split up to maximize its use as a counterweight, two rows of 20 pushed all the way forward in the fuselage will push the center of gravity further forward than one long pack if the need arises.

With all elements of the propulsion system selected the total operating time can be estimated. The batteries have an estimated capacity of 2100 mAh at a discharge current of about 30 amps. The expected power draw is about 1200 watts, which results in 4.5 minutes of full powered flight, which is never the case for an actual flight, since it would rarely require full power other than at takeoff or emergencies.

Geometry:	
Length	8ft
Span	9.5ft
Height	1.25ft
Wing Area	13.5ft^2
Aspect Ratio	5.7
Control Volumes	
horizontal	179inch^2
vertical	79.5inch^2

Weight Statement:	
Airframe	17lb
Propulsion System	7lb
Control System	2lb
Payload System	1lb
Payload	5lb
Empty Weight	27lb

Performance:	
CL max	1.3
L/D max(wing)	65
max RoC	4.5fps
Stall Speed	18mph
Max Speed	40mph
T/O length	
empty	90ft
gross weight	110ft

Systems:		
Radio Used	Futaba 8-channel	
Servos Used	6 servos rated at 11kg.cm	
Battery Config	2x20 cells	
Motors:	Cobalt-90	Cobalt-60
Propeller	20x14	16x10
Gear Ratio	2.75/1	1/1

Gross Weight	32lb

Table 5.2: Aircraft Specifications

6.0 Manufacturing Plan

The design and numerical analysis of an airplane is only half of the task in this project. There is still the actual production of a working airplane. Many different methods of manufacture exist for the different components of this airplane; however, there is a certain degree of expense and skill required for some of these processes. In order to determine the best methods for constructing a lightweight, structurally sound, high performance aircraft, each part of the airplane is compiled along with the figures of merit in Figure 6.1

6.1 Wing

The process of deciding a construction method for the wing was the most time consuming in the project. This was due in part to the critical importance of a lightweight wing that could support the entire airframe in flight. There were three methods discussed in the conceptual design section for the wing: a traditional spar and balsa wood rib wing, a hollow carbon fiber shell, and a foam core with a spar. Through numerical analysis of the particular figures of merit, it was decided to construct a wing with a foam core and spar.

The first step in the manufacture of the wings was cutting the foam to the shape of the chosen airfoil. The foam that was used in construction had to be low density, for weight savings, but had to be strong enough to resist being crushed when it was placed in a vacuum bag for laminating, a procedure that will be discussed later. A polystyrene foam block measuring 9' x 2' x 1' is used to cut the shape of the airfoil. This block provides enough volume to create a one-piece foam core for each wing.

Next, airfoil coordinates are entered into AutoCAD to produce a three-dimensional model of the wing. A computer-guided machine with the hot wire attachment is used for the cutting of the foam blocks. This machine provides more accuracy and a smoother surface for applying a laminate. Though this method costs more, the time savings are significant due to the lack of sanding and surface preparation needed.

However, the block must first be cut in half to form two 4-foot blocks. The hot wire attachment used to cut the wing can only cut from the root to tip and not from tip to tip. This is due to the taper in the wing. The foam wing is then cut into two span wise sections and circular channels are dug out for placement of the spars. The detail of the cuts can be seen in Figure 6...

The next step is to combine the two sections of the foam core to produce a one-piece wing section. A thin bead of five-minute epoxy is applied between each section and they are pressed together to form a foam core with a carbon fiber spar. However, before the wing is laminated, all hard points and other internal structures must be built into the wing. This will provide secure mounting for the fuselage and the landing gear.

For mounting the ailerons, two 4 foot long pieces of ½-inch balsa is glued to the ends of the third section of the foam core. This provides a secure place to attach the hinges for the ailerons. After that,

pairs of 1/8-inch diameter carbon tubes are mounted on the front and back of the spar and fuselage mounting locations. These provide sockets into which the mounting pins are inserted.

After all of the internal structure are in place, the wing is laminated in carbon fiber and vacuum bagged. The carbon fiber fabric is placed on the wing and a sufficient layer of epoxy is applied. Next, a sheet of peel ply is placed over the wing before it is placed in the vacuum bag. The peel ply is fiberglass cloth coated with Teflon, which allows excess epoxy to be lifted away from the carbon fiber leaving a lightweight, smooth surface wing. The whole wing is left in the bag for twelve hours before it is opened. The wing is at maximum strength in approximately three days. The long curing time is due to the use of extra slow hardener in the epoxy, which provides higher strength than the quick cure hardener. This was chosen over Monokote and balsa laminate because it is lighter weight and provides increased structural rigidity for the wing. Also, the addition of team members with experience in laminating carbon fiber made this a useful method of construction.

6.2 Tail

The two main designs for the tail involved either foam cores with carbon laminates or balsa frame with Monokote covering. After building test sections for each design, it was decided that a balsa frame would be the best solution for the horizontal surface. The foam core section was much stronger than the balsa wood section, but it was twice as heavy. However considering previous experience with the vertical surfaces having trouble attaching to the carbon rods when made from balsa wood, the foam core and carbon shell was chosen for the vertical stabilizers.

The first step in designing the horizontal stabilizer is cutting out the ribs. A template, machined out of aluminum, is placed on a stack of $1/8^{th}$ in balsa wood and a band saw is used to cut out the ribs. This method is used to reduce the variance in the shape of the ribs. Even a small amount of deviation can severely decrease the control of the airplane.

Pieces of $1/8^{th}$ inch balsa, acting as the spars, are then run across the top and bottom of the evenly spaced ribs. After that, the leading edges and trailing edges of the stabilizers are glued on with CA. Next, before the Monokote is put on, carbon rods, which are the main connections to the fuselage, are mounted inside the horizontal stabilizer.

For the vertical stabilizers, aluminum templates were cut out in the shape of the inner and outer airfoils in order to guide the hot wire cutter over the foam. Once the foam was cut out, a balsa wood piece was placed at the rear edge to provide firm support for the hinges that attach to the control surfaces, and the whole stabilizer was then carbonfibered to the carbon rod and vacuum-bagged.

6.3 Fuselage

The construction of the fuselage is the most elaborate and unique part of the UB Bullship. Originally, a balsa frame with Monokote covering was thought to be the simplest and strongest design.

However, the aerodynamic qualities are not good. So, the decision was to use a foam core with a carbon fiber laminate. This provided high strength and the ability to produce more complex curves.

The foam core, which serves as the frame, is cut by the same method used in the wing construction. Various cuts are made in the foam with a hot wire and hot knife to form a groove for the wing mounts, an opening for filling the water tank and access to the batteries, and channels for the servos and control rods. Additional grooves are cut for balsa wood strips that act as reinforcements, which run around the outside of the fuselage at high stress areas.

The openings for the engines and receiver are cut using a hot knife, which allows us to make a tight fit. The channels for the batteries are not cut as tightly, so more air can flow over the packs for cooling. In addition, scoops are cut on each side of the fuselage so air can easily make it to the batteries.

The grooves for the servos are cut to fit the servos exactly. Three layers of fiberglass are then laminated on the inside of the grooves to provide secure mounting points. Also, channels are cut, so that the control rods may move the separate control surfaces with relative ease.

After the internal structures are in place, the whole foam core is laminated with a layer of carbon fiber and epoxy, and placed in a vacuum bag for a minimum of twelve hours. Markers are left on the surface so that the various openings can be found and the excess carbon fiber cut out after curing.

6.4 Engine Mount

It was determined in the Preliminary design that the UB MAX DRAG is a twin push-pull engine airplane, which made the decision to mount the engines in the nose and rear of the fuselage trivial. Therefore a design was needed that would mount the engine securely to the fuselage. The design team thought it would be possible to secure a firewall directly to carbon rods, which run in the middle of the fuselage starting from the wing connection rods to the front.

For firewall construction, $1/8^{th}$ inch is used. The plate is machined so it fits the engine perfectly, and has attachment holes on the sides that would be connected to boat wire clamps which consist of metal rings with rubber lining on the inside that fit perfectly and tightly on the carbon rods.

6.5 Landing Gear

The final landing gear design for this airplane is based on the tricycle style. However, the rear landing gear is mounted to the wing and not the fuselage for extra stability. The gear is attached to the hard points on the wing spar ensuring that the shock from landing is not transmitted to the fuselage. This design is more stable and fairly lightweight compared to using a hinge design that attached to the sides of the fuselage.

The nose wheel assembly used is commercially available and is rated for model aircraft of this size. It is controlled by a separate servo connected to the same channel as the rudder. This provides good ground handling and allows for easier hookup than routing a control rod from the rudder servo.

6.6 Material Selection

The following figures of merit provide the design team with the manufacturing characteristics of each component of the airplane. This was important in deciding how to go about construction the airplane outlined in the Detail Design section. The merit parameters are as follows: availability, cost, required skill level, time required, and strength.

6.6.1 Availability

This was extremely important to the design team when deciding what materials to use. There had to be material vendors available that could supply the team with the proper dimensioned and cost effective materials. A score of 0 was given if the product was not available and a score of 4 was given if the product was easily attainable.

6.6.2 Cost

The airplane built had to be cost effective. The team raised enough money to purchase some fairly exotic materials, but these had to be used in only the most essential components. A score of 0 is for and extremely expensive material and a score of 4 is for a relatively inexpensive material.

6.6.3 Skill Level

This year's team was fortunate to have the advice of many experienced builders. This allowed looking into processes that are somewhat more advanced. However, it was still important to stay within the scope of reality because reading a book on carbon lay-ups is different from actually doing it. A score of 0 is for expert skill level and a score of 4 is for beginner skill level.

6.6.4 Time

Time is always against a team in a design competition of this magnitude. The plan is to have the airplane built at least two weeks before competition. This allows enough time for testing the plane as a whole. All of the manufacturing has to be done relatively quickly in time for the competition. A score of 0 is for a very time consuming method and a score of 4 is for a fast manufacturing method

6.6.5 Strength

A strong plane is critical to surviving the sorties in the competition. This is where choice of material can make or break an airplane. A score of 0 is for a weak structure and a score of 4 is for structurally sound component.

		Availability	Cost	Skill Level	Time Required	Strength	**Total**
	Weighting	3	1	2	3	2	
Wing Structure	Balsa Wood Frame	4	4	3	2	3	34
	Foam Core/Carbon Laminate	4	3	4	3	4	40
Spar	Balsa Wood Core	4	3	3	3	4	38
	Carbon Rods	4	3	3	4	4	39
Fuselage	Balsa Frame w/ Monokote	4	4	4	2	3	36
	Foam Core/Carbon Laminate	3	3	3	4	4	38
Landing Gear	Spring Steel	4	4	4	3	4	41
	Carbon Fiber	2	1	4	4	4	35
Horiz. and Vert. Stabilizers	Balsa Frame w/ Monokote	4	4	3	4	4	42
	Foam Core/Carbon Laminate	4	3	3	2	3	33
Motor Firewall	Aluminum	4	4	4	4	3	41
	Steel	4	3	4	3	4	40
	Plywood/Carbon	4	2	3	3	4	37

Table 4.1 Manufacturing Figures of Merit

	Date Start	Date Finish
Fuselage		
Foam cutting	1/19/2004	1/20/2004
Engine mount	2/6/2004	2/12/2004
Hardware placement	2/1/2004	2/8/2004
Laminating	2/14/2004	2/15/2004
Tail		
Vertical Stabilizers	2/15/2004	2/17/2004
Horizontal Stabilizer	2/15/2004	2/17/2004
Wing		
Foam cutting	2/18/2004	2/19/2004
Spar construction	2/20/2004	2/23/2004
Hard point mounts	2/24/2004	3/1/2004
Laminating	3/3/2004	3/4/2004
Landing Gear	3/8/2004	3/10/2004

Table 6.2 Manufacturing Milestones

7. Testing Plan

The major testing objectives would be to determine the actual aircraft flying characteristics, so as for the pilot to know what to expect when applying various controls to the plane. This will include testing the actual maximum rate of climb, takeoff distances, carrying capacity, and possibly, if time permits, training by performing the actual missions on a model aircraft-dedicated airfield. These tests, however would have to wait until the basic static tests are performed to make sure the plane is relatively ready to even go on the runway. The main one of these would be whether or not the aircraft would be able to fit inside the 4x2x1 foot box. The wings and fuselage have already been constructed and tested to fit inside this box. Determining the actual center of gravity, would also be an integral part of testing, once construction is finished, since the placement of the battery packs depends its' location, so as to bring it to the design specifications. A wing-tip stress test would also be very crucial, since it would be better if the plane fails this test early instead of at competition. Also the final gross weight would have to be determined in order to make sure that the wing loading is not excessive, in which case serious modifications would be required.

Static testing has already been performed on the majority of the aircraft components. The strength of the rods in terms of what kind of moment they can withstand has already been performed, and they have been deemed to provide more than adequate strength considering the loads experienced in typical flights. The rod junctions have also been tested and deemed adequate. The motor/gear/propeller setup has also been tested in order to ascertain whether it would perform to the theoretical expectations. This was done using the motor test stand and a scale to measure the resulting output thrust.

Testing Schedule		
Item/Parameter	**Start**	**Finish**
Carbon Rod Strength	Sat 02/15/04	Sat 02/15/04
Rod Junctions	Sat 02/16/04	Sun 02/17/04
Propulsion Setup Testing	Sun 02/23/04	Wed 02/26/04
Completing Construction		**Sat 03/29/04**
Overall Volume Test	Sun 03/30/04	Sun 03/30/04
Final Gross-Weight Check		
Center of Gravitation Determination and Adjustment	Wed 04/02/04	Wed 04/02/04
Wing-Tip Stress Tests		
Functioning of Control Surfaces	Sat 04/06/04	Sun 04/07/04
Takeoff Distance	Sun 04/07/04	Sun 04/07/04
Rate Of Climb	Fri 04/11/04	Sat 04/12/04
Maximum Payload Weight	Sat 04/12/04	Sun 04/13/04
Maneuverability	Thu 04/17/04	Sat 04/19/04
Mission Requirements	Sat 04/19/04	Sun 04/20/04

Table 6.1 – Tentative Testing Schedule