CESSNA ONR / STUDENT DESIGN/BUILD/FLY COMPETITION

An AIAA Student Activity

The 2001 Cessna/ONR Student Design/Build/Fly competition was held at the Webster Field Annex of the Patuxent River Naval Air Station over the weekend of 21-22 April. Twenty seven teams from the United States and three foreign countries; Canada, Italy and Turkey, competed under near perfect weather conditions to see whose aircraft could transport the greatest total payload over the prescribed course within a 10 minute period. Of the 27 teams attending the fly-off competition, 22 made at least one successful scoring flight, with many teams making multiple flights during the two days of competition.

The goal of this year competition was to complete as many sorties as possible within a 10 minute period. Each sortie required a different payload, either as many pounds of steel or as many tennis balls as the aircraft could carry. Aircraft had to carry a minimum payload of either 5 lbs. of steel or 10 tennis balls on each sortie, and were allowed to carry up to 100 tennis balls.

The competition spanned two days of flying, with the flight queue filled with aircraft waiting for there turn to make a competition flight. During the competition there were a total of 105 multiple sortie flights periods.

The total score for each team was comprised of their flight performance on their best three flight, there 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.

The final results showed a close battle between the two teams from Oklahoma State University and the team from California Polytechnic State University at San Luis Obisbo. In the end, the California team was not able to match the top team from Oklahoma State, Team Orange, better known by it's aircraft's nickname of "Shamu". There were many close scoring battles for the remaining positions also; including the very close competition between the two entries from the United States Military Academy at West Point. The final positions and scores for all of the competing teams are listed in the table below.

School	Team	Score	Position
Oklahoma State University #2	OSU Orange/shamu	4976.86	1
California Polytechnic State University-SLO	Steel & Balls	4864.68	2
Oklahoma State University #1	OSU Black	4243.73	3
University of California-San Diego	TLAR II	3927.10	4
Utah State University	A	2880.90	5
Miami University	mU2	1988.80	6
University of Illinois	University of Illinois	1819.22	7
University of Southern California	Scorpion	1691.31	8
Mississippi State University	Carbon Goose	1411.02	9
La Sapienza	Flying Centurions	984.61	10
West Virginia University #1	Almost Heaven	837.69	11
Middle East Technical University	Anatolian-Craft	754.07	12
University of Arizona	Aircat 2001 CR2	719.26	13
United States Military Academy #1	Black Knight	698.54	14
United States Military Academy #2	TTF Technologies	611.49	15
Georgia Institute of Technology	"Georgia Tech" -or-	585.77	16
University of Central Florida #1	Knightmare	308.42	17
East Stroudsburg University	Double Trouble	296.52	18
Clarkson University	Knight Hawk	132.48	19
SDSU	Full Monty	56.29	20
West Virginia University #2	Lothworks	26.61	21
University at Buffalo	Undecided	18.75	22
Queen's University	Absolute Toque	1.26	23
University of Texas at Austin	The GPA killer	1.09	24
Virginia Polytechnic Institute & State Univer	Hokie Bird 5	1.03	25
United States Naval Academy	NAVY	0.93	26
ISTANBUL TECHNICAL UNIVERSITY	The Bosphorus Blue	0.08	27
Wichita State University	No Excuse	0.08	28
Syracuse	Der Gabelschwanz Te	0.08	29
City College of New York #1	City Hawk	0.06	30
City College of New York #2	City Viper	Withdraw	31
City College of New York #3	City Eagle	Withdraw	32
MIT	Great Balls of Fire	Withdraw	33
Tennessee Tech University	Golden Eagle	Withdraw	34
UCLA	Grand Master B II	Withdraw	35
University of Central Florida #2	Fly By Knight II	Withdraw	36
UT Arlington	No Payne No Gaines	Withdraw	37

The success of the competition required the efforts of many individuals. A special 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. Special thanks is due to the competitions corporate supporters, the Cessna Aircraft Company and the Office of Naval Research. And the final and biggest thank you must go to our hosts for the weekend, the personnel of Webster Field, VC-6, and the Patuxent Naval Air Station under the command of Captain P. J. Hovatter.

Overall the 2001 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.

See you next year - Greg Page: Contest Director

2000/01 AIAA Foundation Cessna/ONR Student Design/Build/Fly Competition

Proposal Phase Design Report

Oklahoma State University orange team

March 13, 2001

1 Executive Summary

The Oklahoma State University Orange Team began searching for an optimum design for this year's Design/Build/Fly competition by evaluating the mission requirements and determining the critical constraints on the design. The objective of this year's contest is to carry alternating payloads of as much steel as possible and as many tennis balls as possible for each sortie. The aircraft design must meet the following criteria as established by the rules of the contest.

- Wingspan of a maximum of 10 ft.
- Aircraft must be propeller driven and electric powered with a commercially available motor from the AstroFlight or Graupner brushed motor families
- Aircraft must use shrink wrapped, commercially available NiCad batteries
- Battery pack must weigh no more than 5.0 lb
- Aircraft must weigh a maximum of 55 lb at takeoff
- Aircraft must be able to withstand a 2.5g load case

1.1 Summary of Design Development

The initial approach in solving this particular design problem involves a systematic mission analysis. This analysis integrates the contest rules with other physical limitations to find the performance needed to score well in the competition. This mission analysis program was then expanded to include propulsion, aerodynamic, and structural information. The program was then used to evaluate and define flight performance and mission strategy.

1.2 Major Development Process Areas for Final Configuration

The first major step in the design process for the Orange team was the development of the mission analysis program. This mission analysis lead to the preliminary configuration selection that allowed the team to move into the next design phase. The program was then refined to include experimental data obtained by the propulsion group through wind tunnel testing. While the refinement of the mission analysis was being done, the structures group began work on the major sub-assemblies of the aircraft. When the mission analysis refinement was completed and the design of the subassemblies finished, then the team moved on to construction of a prototype aircraft. This prototype allows the team to do flight testing to refine the design for the final aircraft.

1.3 Range of Design Alternatives Investigated

The team investigated configurations running from the more experimental to the common and well tested. Configurations considered included flying wing, canard, conventional, multi-engine, and twin fuselage. Different manufacturing processes were also investigated including conventional balsa and monokote and carbon fiber.

1.4 Design Tools used in each Phase

During the conceptual and preliminary design phases there were two major design tools utilized. The aerodynamics group centered on developing an involved and expandable mission analysis program. The propulsion team utilized wind tunnel testing of previous propulsion systems. Once this data was gathered, it could be integrated into the mission analysis program and used to develop an optimal propulsion system for the desired flight profile.

Once the data was gathered, the equipment desired and purchased or built and put through more wind tunnel testing. This allowed for further analysis and refinement of systems and techniques that can be implemented in the detailed design of the structure and systems.

The culmination of the detailed design phase is the building and flight testing of a prototype aircraft. Here any unexpected issues in systems integration and performance can be explored and addressed for use in the building of the final aircraft.

2 Management Summary

2.1 Team Architecture

The Orange team is divided into three technical groups, aerodynamics, propulsion, and structures. This technical group division helps the efficiency of the team as a whole. The team personnel were divided up into the technical groups based on their personal interests, their technical background, and the need in each area. Each group has an appointed leader who reports to the chief engineer. The chief engineer is responsible for facilitating communication between the groups and delegating responsibilities to each of the groups. Although not official members of the team, our advisors have provided very valuable information and guidance. Below is a chart illustrating the architecture of team management.

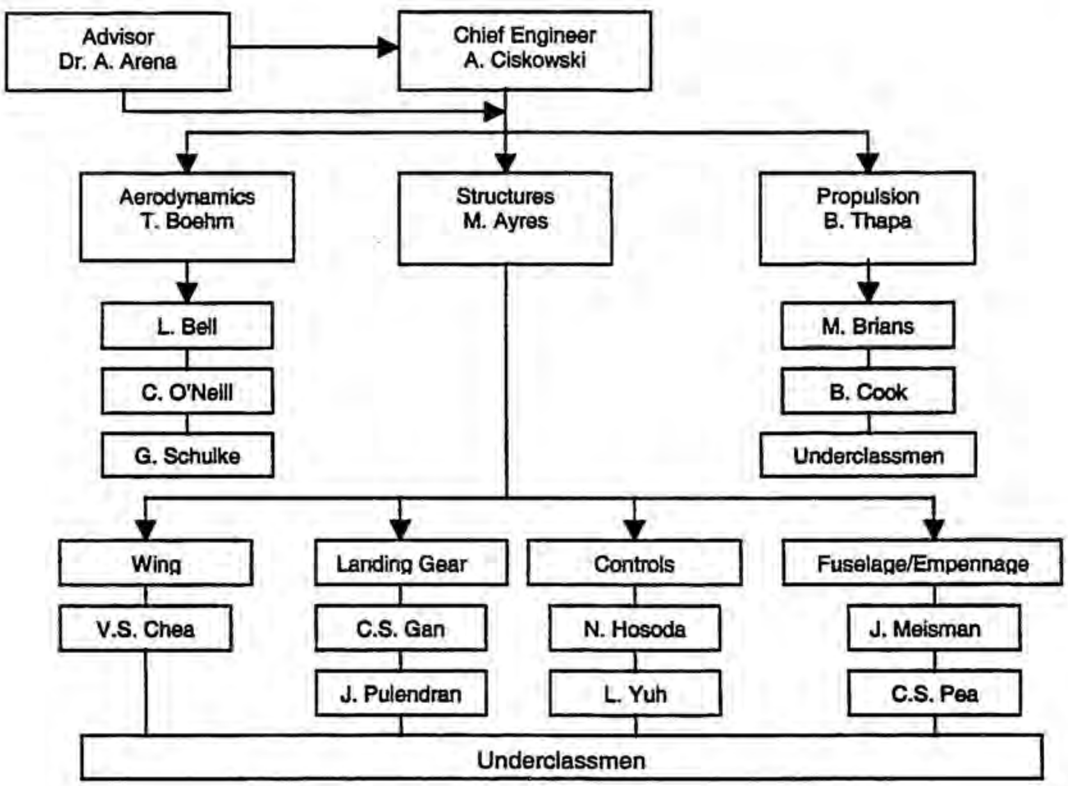


Figure 2.1-1: Team Architecture

2.2 Assignment Areas

The aerodynamics team is the primary configuration group. This team develops and refines a mission analysis program based upon the objectives and guidelines of the competition. The aerodynamics team is responsible for the external configuration and layout including airfoil design. This group is also responsible for the evaluation of aircraft performance in all phases of flight.

The propulsion team is responsible for testing and evaluating possible components of the propulsion system to supply data to the Aerodynamics group. They are responsible for motor, battery and propeller selection. This group is also responsible for working with the aerodynamics group to integrate the propulsion systems into the airframe. They are also responsible for the maintenance of the propulsion system.

The structures team is primarily responsible for the construction of the aircraft. This team conducts the stress analysis of the structural design. They are responsible for internal layout and component placement in the aircraft. In addition to the testing and analysis of all structural components, this group is responsible for preparing the construction drawings, and the selection of the construction method and materials.

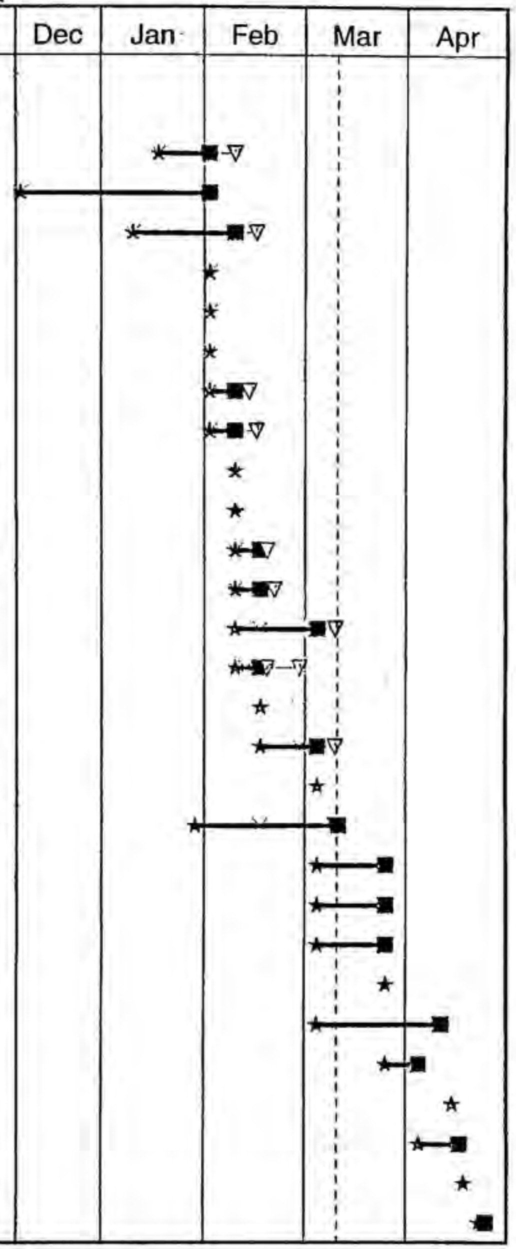
2.3 Management Structures

The design team meets formally twice a week as an entire group. During these team meetings, the chief engineer and the advisor bring to light any issues that require the attention of the entire team. These team meetings are also when the majority of the team brainstorming is conducted.

Aside from the team meetings, each technical group meets separately during the week. At these technical group meetings, the group leads address the issues pertinent only to their particular group. The technical groups also meet at other times each week to do the necessary group work such as wind tunnel testing. During construction the team will function more as a whole. The aerodynamics and propulsion groups will shift to help the structures team with construction of the aircraft. This is to minimize miscommunications between those responsible for the design and those responsible for the construction.

At the beginning of each week the chief engineer and the technical group leads meet to discuss the progress made during the previous week, the important issues for the upcoming week and make sure every group is aware of their schedule and upcoming deadlines.

Actual Start	Actual End	TASK	Projected Start	Projected End
12/15/00	12/15/00	Initial Team Meeting	12/15/00	12/15/00
12/15/00	12/15/00	Conceptual Design Begins	12/15/00	12/15/00
1/22/01	2/12/01	Concept Brainstorming	1/22/01	2/5/01
12/16/00	2/5/01	Optimization Development	12/16/00	2/5/01
1/15/01	2/18/01	Propulsion Reserch	1/15/01	2/12/01
2/5/01	2/5/01	Motor selection	2/5/01	2/5/01
2/5/01	2/5/01	Concept Selection	2/5/01	2/5/01
2/5/01	2/5/01	Preliminary Design Begins	2/5/01	2/5/01
2/5/01	2/16/01	Optimization refined	2/5/01	2/12/01
2/5/01	2/18/01	Aero & Structures:layout	2/5/01	2/12/01
2/12/01	2/12/01	Airfoil Selection	2/12/01	2/12/01
2/14/01	2/14/01	Detailed Design Begins	2/12/01	2/12/01
2/12/01	2/21/01	Design major components	2/12/01	2/19/01
2/12/01	2/23/01	Aero: sizing of control surfaces	2/12/01	2/19/01
2/19/01	3/12/01	Propulsion System Testing	2/12/01	3/7/01
2/15/01	3/2/01	Generate Construction Drawings	2/12/01	2/19/01
3/2/01	3/2/01	Prototype Design Freeze	2/19/01	2/19/01
3/2/01	3/12/01	Construction of Prototype	2/19/01	3/7/01
		Prototype Rollout	3/7/01	3/7/01
2/19/01	3/12/01	Prepare Proposal Report	2/1/01	3/13/01
		Prototype Flight Testing	3/7/01	3/26/01
		Refinement of Prototype Design	3/7/01	3/26/01
		Update Drawings	3/7/01	3/26/01
		Final Design Freeze	3/26/01	3/26/01
		Prepare Addendum Report	3/7/01	4/10/01
		Construction of Final Aircraft	3/26/01	4/4/01
		Final Design Rollout	4/13/01	4/13/01
		Practice Flights	4/4/01	4/15/01
		Aircraft Grounded	4/16/01	4/16/01
		AIAA International Competition	4/20/01	4/22/01



Projected Time

Actual Time

Figure 2.3-1: Orange[illegible]m Milestone Chart

3 Conceptual Design

The objective of the conceptual design phase is to identify and develop possible solutions. Governing physics and the competition rules are used to initiate and guide the design search. Design parameters are created to predict performance from each of the conceptual designs. Figures of Merit and Rated Aircraft Cost (RAC) are introduced to evaluate each conceptual design's effectiveness. Finally, a configuration is produced from the results of the conceptual design phase.

3.1 Mission Analysis and Strategy Development

In trying to get a feel for the breadth and depth of the design problem, a computer program was created to assist with aircraft selection and flight performance estimation. The ultimate goal was to allow the program to optimize the specified aircraft geometry with respect to score. The requirements of this optimization are twofold. The contest rules and governing physics must be satisfied. Second, selection of aircraft geometry with a high score is desired. The multi-dimensional design space defined by combination of contest rules and physics prevents an aircraft based on a heuristic or intuition from being an optimum.

The payload related score consists of the summation of total carried payload weight determined by the number of sorties and the individual payload weight carried for each. The 10-minute mission window and the available propulsion energy limit the total number of sorties possible during a mission. Overall aircraft weight is constrained by the contest rules at 55 pounds maximum with specific limitations on the minimum payload weights to be carried. The number of sorties and the overall payload weights are clearly coupled. It is expected that the overall best score will not result from either carrying a massive payload for one sortie or carrying a small payload multiple times. With the heavy lifting endpoint, the aircraft encounters the overall 55 pound gross weight limit. With multiple sorties, the payload reloading time overruns the total mission time.

In order to reduce pilot workload, the profile was broken into three constant power settings: full power, cruise power, and idle. Additionally, these velocities should be low enough for controllability while high enough to prevent overpowering wind effects. A cruise height of above 20 feet is needed for error recovery and spotting, while under 200 feet is needed to prevent excessively long climbs

Several parameters were set up as being key to competition performance. These parameters were all integrated into a system of equations using the constraints of the competition and aerodynamic principles to relate the parameters. This system was evaluated to the maximum score potential of a particular configuration.

A non-linear conjugate-gradient method was selected with the understanding that multiple guess values were required to avoid local maximums. A computer program was created which combined the conjugate-gradient method, relevant aerodynamics, propulsion performance, structural estimates and the contest rules. Six parameters were evaluated; steel payload weight, tennis ball payload weight, maximum battery power, wing area and the cruise powers for both steel and tennis ball sorties. All other variables were calculated involving the six evaluated parameters. Passing the variables to the relevant aerodynamic, propulsion or structural subroutine integrated performance estimates.

Limits on wingspan, total time, payload and takeoff distance were required to keep the aircraft within the DBF mission constraints. The RAC design criteria established in the DBF rules meant that the aircraft geometry heavily influenced the final score. Estimates of aircraft weight as a function of geometry were derived. The propulsion system was modeled through calculations of power, time and energy throughout the flight. Experimental values for the total energy available and efficiency obtainable were used.

Aerodynamic subroutines calculated takeoff, climb, cruise, approach and landing characteristics of the aircraft. Time, velocity and power for the segments were calculated. See the profile Figure 3.1-1.

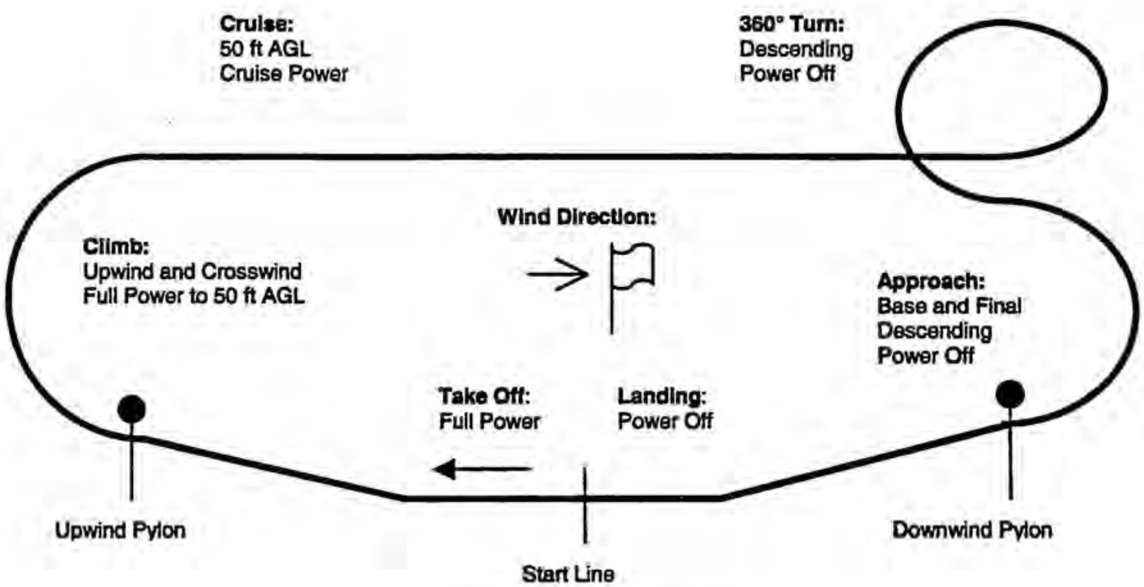


Figure 3.1-1: Flight Profile

3.2 Alternative Concepts Investigated

During the brainstorming activities of the team, no concept was judged or eliminated. This phase involved solely gathering every possible idea to give a foundation for developing the final aircraft. The next stage was then to begin selecting the most promising designs for further investigation. In this initial elimination, several parameters were used to aid in the selection of the most promising designs. Among them were our ability to build the concept with limited time and resources, the performance possibility of the aircraft, and the ability to compete in the mission assigned by the contest. A total of five possible configurations were selected for further evaluation. These five configurations are shown in Figure 3.2-1.

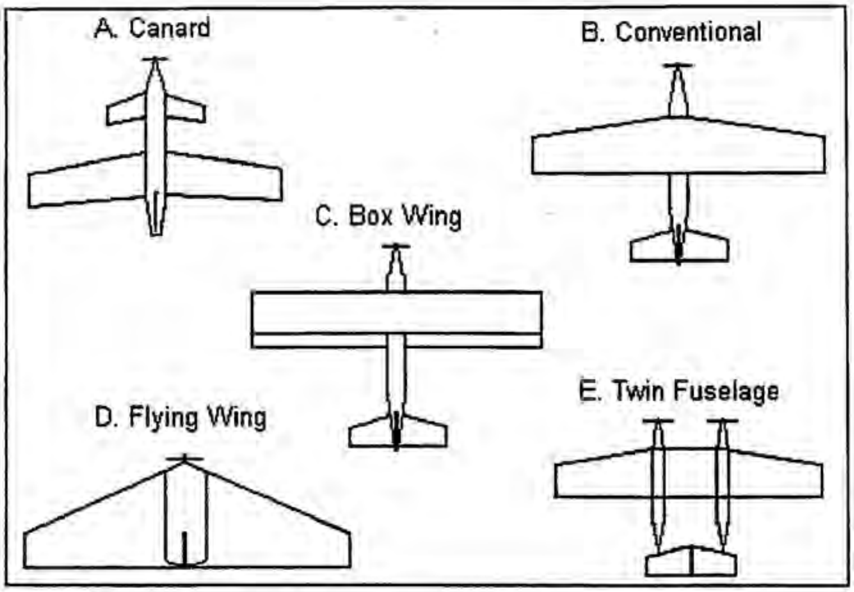


Figure 3.2-1: Preliminary Configurations

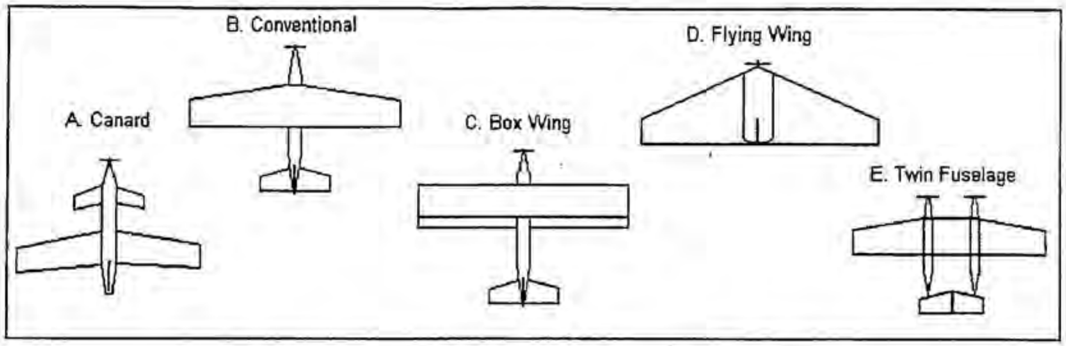
3.3 Important Design Parameters

Primary design parameters are needed to efficiently set the aircraft geometry and performance. The chosen parameters must be capable of entirely describing the dominant aerodynamic characteristics of an aircraft while still allowing for uncertainties. These design parameters are:

- Maximum Available Power
- Cruise Powers
- Payload Weights
- Number of Sorties
- Wing Area
- Take-off lift coefficient

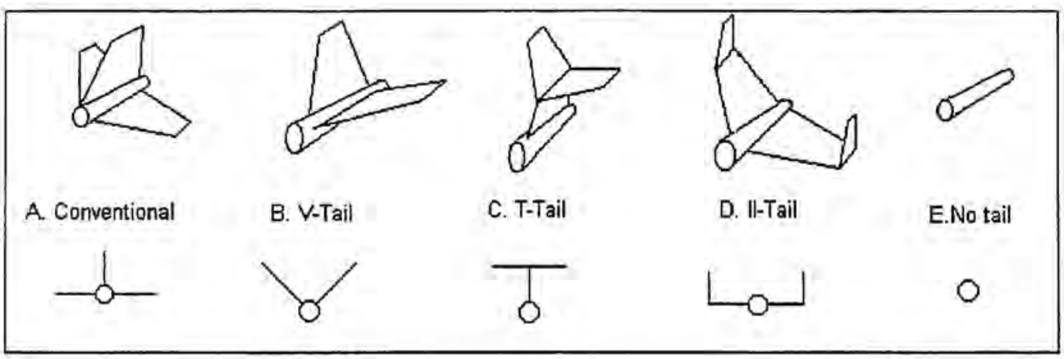
3.4 Figures of Merit

The main configuration was selected with the assistance of Figure 3.4-1. Once the main configuration was selected, several other decisions had to be made regarding the empennage type, shown in Figure 3.4-2, and wing placement. Table 3.4-1 below shows the FOM for the wing placement evaluation. Explanations of FOM usage are found in section 3.6.



Figures Of Merit	Weight Factor	A	B	C	D	E
SCORE / RAC	0.3	1	1	0	-1	0
Stability and Control	0.2	1	1	1	-1	0
Payload Capability	0.2	0	1	1	-1	1
Propulsion	0.1	0	0	0	0	1
Ease of Construction	0.1	0	1	-1	-1	-1
Production Costs	0.1	1	1	0	-1	-1
Total Score	1	0.6	0.9	0.3	-0.9	0.1

Figure 3.4-1: Configuration Figures of Merit Chart



Figures Of Merit	Weight Factor	A	B	C	D	E
RAC Impact	0.3	0	0	0	-1	1
Stability and Control	0.3	0	-1	0	-1	-1
Reliability	0.1	1	0	1	0	-1
Ease of Construction	0.1	1	-1	0	0	1
Servo Location	0.1	1	1	0	0	1
Production Costs	0.1	1	1	1	0	2
Total Score	1	0.4	-0.2	0.2	-0.6	0.3

Figure 3.4-2: Tail Figures of Merit Chart

Figures Of Merit	Weight Factor	High Wing	Mid Wing	Low Wing
Stability and Control	0.2	1	0	0
Payload Access	0.3	0	0	1
Ease of Construction	0.1	1	1	1
Fuselage Volume Affected	0.2	0	-1	0
Landing Gear Interface	0.1	0	0	1
Total Score	1	0.3	-0.1	0.5

Table 3.4-1: Wing Placement Figures of Merit Chart

3.5 Rated Aircraft Cost for Concepts

An important step in exploring the performance possibility of a concept in this contest is to evaluate the rated aircraft cost of the different configurations. The main breakdown of the equation for rated aircraft cost is shown in the table below.

Using the rated aircraft cost equation breakdown and information from the mission analysis program, the total rated aircraft cost for each of the five final designs considered can be evaluated. The results are as shown in Table 3.5-1.

Canard			RAC	5.076
MEW	13	13	100	1500
REP	1*30*1.2*36	1296	1	1296
MFRH	Wing	45	20	900
	Body	29	20	580
	Tail	20	20	400
	Systems	10	20	200
	Propulsion	10	20	200

Conventional			RAC	5.076
MEW	13	13	100	1500
REP	1*30*1.2*36	1296	1	1296
MFRH	Wing	45	20	900
	Body	29	20	580
	Tail	20	20	400
	Systems	10	20	200
	Propulsion	10	20	200

Box Wing			RAC	6.792
MEW	14	14	100	1500
REP	2*30*1.2*36	2592	1	2592
MFRH	Wing	66	20	1320
	Body	29	20	580
	Tail	20	20	400
	Systems	10	20	200
	Propulsion	10	20	200

Flying Wing			RAC	4.016
MEW	11	11	100	1100
REP	1*30*1.2*36	1296	1	1296
MFRH	Wing	45	20	900
	Body	7	20	140
	Tail	10	20	200
	Systems	9	20	180
	Propulsion	10	20	200

Twin Fuselage			RAC	5.696
MEW	15	15	100	1500
REP	1*30*1.2*36	1296	1	1296
MFRH	Wing	45	20	900
	Body	50	20	1000
	Tail	20	20	400
	Systems	10	20	200
	Propulsion	20	20	400

Table 3.5-1: Rated Aircraft Cost for Configurations Considered

3.6 Features that produced the Final Configuration

Once the mission analysis program had developed the performance needed to score well, there were a great many decisions to be made regarding the numerous possibilities for aircraft design. Although the program had answered many questions, it couldn't give all the information for more intangible things. In order to systematically make those comparisons, several categories were established under which to compare the designs. These categories were rated according to their estimated importance in getting the mission accomplished. In systematically making the decisions necessary to create this aircraft, several figures of merit were used. The main categories for comparison are given below.

Score / Rated Aircraft Cost: The first category for comparison was the configurations rated aircraft cost. Since the main goal is to perform as well as possible in the contest, obviously the higher score possibility for the configuration, the better. The lowest value for rated aircraft cost was desired in order to have a lower divisor of the points earned by amount of payload carried. The mission analysis program developed by the team was used in order to evaluate the score potential of each configuration.

Stability and Control: Stability and control issues were considered to be a very important category for evaluation as well. In order to perform well at the competition, the aircraft must be stable and easily controlled by the pilot. Although an unstable aircraft could have been used, it was shown in the mission analysis program that the aircraft was going to have to fly numerous sorties per mission and therefore an aircraft that could be controlled well was a must. After all, if the aircraft is difficult to fly and is not controllable then it won't be able to give the performance desired by the program.

Reliability: In order to compete with this aircraft in a situation that requires as many as eighteen flights to be competitive, the aircraft design must be reliable. While the aircraft's manufacturing processes and materials heavily affect this, there are also some designs where the performance is more reliable in a varying number of wind conditions and other situations. This concept is tied somewhat to the stability and control of the aircraft.

Payload Capability: Since the purpose of this aircraft is to carry payload, the ability of the basic design to do so was an important parameter for evaluation. The differences in the types of payload to be carried add another level of complexity. The tennis ball payload requires a volumetric consideration more than a shear lifting capability. The steel payload, however, would be limited only by what the aircraft can lift.

Propulsion: Before selecting propulsion system components several possible configurations were developed. These included single and multi-motor configurations. These configurations differ significantly in weight and RAC impact, so it was important to choose the configuration that would best serve the design goals. The single motor configuration gave the best score in the mission analysis program. The DBF battery specifications, 5 lbs of NiCad batteries, left the task of finding the battery type and brand that would give the performance developed by the mission analysis program. The selection process for the batteries was conducted through research into different products that were offered by several manufacturers. The propeller selection was aided by performance parameters that were dictated by the aerodynamics team later in the design process. The gearbox for the motor depends on the propeller size and was initially assumed to be between 2.4:1 and 3:1.

Ease of Construction: Even the greatest concepts are meaningless to us if they cannot be built. Because of this fact, as well as the relative inexperience of our team members in aircraft construction, manufacturing simplicity was deemed an important consideration. Given more time and resources, more complexity could have been allowed, but that was not the case in our current situational constraints.

Material Selection: As with any aerospace application, weight is a critical factor. With the recent innovations of carbon fiber used in building this type of aircraft, this was one of the first materials explored. The other major material considered seriously was balsa-monokote construction. Because of its strength to weight ratio, durability and ease of repair, carbon fiber stood out in the conceptual design phase.

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4 Preliminary Design

Once the main configuration decisions were made in the conceptual design phase, further evaluations and refinements were made in the preliminary design phase. Information was generated in the form of selection of fuselage shape and sizing of components. The mission analysis program was further developed to aid in the refinement of the design.

4.1 Design Parameters Investigated and Figures of Merit

4.1.1 Aerodynamics Group

The aerodynamics group's goal was to predict and then increase the competition score. Performance characteristics such as C_D, C_L, and S from the conceptual design phase were further investigated. Among the most important of these needed for design refinements were aircraft drag and airfoil performance.

Drag Calculation Overview

Drag affects aircraft during all operational phases. Drag analysis was performed for the fuselage, lifting surfaces and miscellaneous components with preliminary sizing information. Figure 4.1-1 shows the contributions of the various components to the overall drag coefficient

Fuselage: In order to consider many fuselage configurations, circular as well as elliptical formulas were found to compute effective diameters for the form factor equations available in Raymer's Aircraft Design: A Conceptual Approach. The analysis then calculated Reynolds Numbers and subsequent flat-plate skin-friction coefficients. These were multiplied by the fuselage form factor and wetted surface area to obtain the overall fuselage component.

This analysis was then developed to specify a diameter for the payload region and a separate one for the remainder of the fuselage. This ability helped to approximate the total fuselage drag in a single calculation versus multiple single diameter ones. This allowed a length for the payload area to be calculated using test data for an average volume displacement presented by the tennis balls.

Figure 4.1-1: Aircraft Drag Breakdown

Lifting Surfaces: The wing calculation assumed a flat-plate skin-friction coefficient utilizing parameters such as the location of the maximum thickness, sweep angle, and thickness to chord ratio. Using the polar plot for the specific airfoil used refined the drag calculation for the wing. The analysis of the horizontal and vertical tail was similar to the wing computation.

Other Components: Landing gear and upsweep of the aft fuselage to the tail were also addressed. For the landing gear, total frontal areas of the tires and struts were assumed using historical data. Both of these areas produced a significant percentage of the overall drag and steps could be taken to reduce the effect. For example, by adding a fairing to a strut and wheel the gear coefficient was reduced by 48%. In the case of fuselage upsweep, there was almost sixteen times the amount of drag generated by increasing the upsweep angle from five degrees to fifteen.

Interference factors on the individual aircraft components were estimated using historical data. Our assumptions included a moderately filleted low wing configuration that corresponded to approximately twenty percent interference, a horizontal and vertical tail of four percent, and fuselage interference to be negligible. Drag associated with gaps and voids between mating surfaces was not calculated for each individual component. Rather, a 7.5% additional drag is added to the component total to compensate.

Airfoil Selection

Through the use of the mission analysis program, the total aircraft weight, wing platform area, and design airspeeds were determined. From that program, it was discovered that the high take-off lift and cruise velocity were critical parameters. To achieve an airfoil of with these performance characteristics, special detail was given to the maximum coefficient of lift and low drag over a wide range of lift values.

Airfoil Analysis: For the scale of the aircraft and low velocities, the only airfoils analyzed were those tested for Reynolds numbers below 300,000. To achieve the highest accuracy, low-speed wind tunnel data was used. Majority of the information was taken from the websites of the University of Illinois at Urbana-Champaign's Low-Speed Airfoil Tests and the Nihon University Aerospace Student Group's Airfoil Database. Of the 1100 airfoils listed on the UIUC Airfoil Test Site, the selection was reduced to the following families due to their high lift capabilities: Eppler, Wortmann, Liebeck, Selig, and Selig-Donovan. By comparing the plots of coefficient of lift (C_L) versus coefficient of drag (C_D) and coefficient of lift versus angle of attack (α) provided by the Nihon University website, it was possible to conveniently analyze the airfoil data.

Airfoil Comparisons: The primary constraint on the airfoil selection was the high lift needed for take-off within 200 ft. Exploration of the mission analysis program showed that the coefficient of lift desired was greater than 1.5 and that the final score increased directly with the higher lift coefficients. Consideration was given to the addition of high lift devices (i.e., flaperons), but was nullified due to increased mechanics and construction complications. Preliminary comparisons of unaltered airfoils produced five choices that provided the greatest lift: the Eppler 423, Wortmann FX74-CL5-140, Wortmann 63-137, Selig 1210, and Selig 1223.

The second constraint of selection was the drag. Since the aircraft will operate for the majority of the time in cruise, airfoils were then examined to minimize drag and, hence, increase performance. Preliminary calculations given by the mission analysis program indicated that the aircraft would require a cruise velocity of 65 mph. Based on a 55-pound design weight, the aircraft required an airfoil with lift coefficients in the range of 0.6-0.8. Therefore, from the previous selected airfoils, the highest consideration was given to those airfoils that had the lowest drag coefficients in lift coefficient range of 0.8-1.0. Assuming lift coefficients within this range of values would account for the drag losses incurred by the actual aircraft. The polar plots for the three final airfoils considered are shown in Figure 4.1-2.

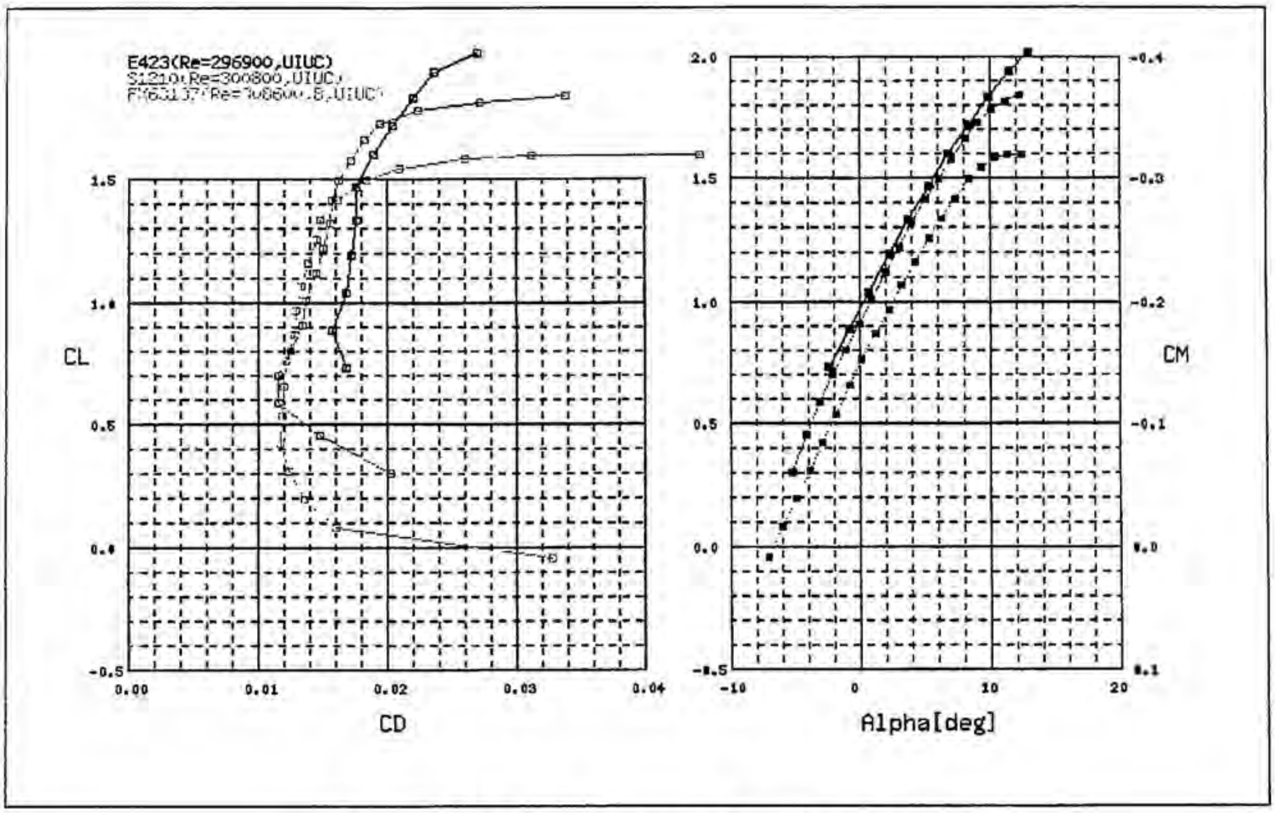


Figure 4.1-2: Polar curves for selected airfoils

It was evident from the polar curves that the drag increase due to increased angle of attack varied significantly for each airfoil. From the performance standpoint, the desired airfoil would posses a C_L vs. C_D curve that was very steep. With this qualification, both the Selig 1223 and Wortmann FX74-CL5-140 were eliminated from consideration. At this point, the Eppler 423 was the front-runner based upon maximum lift characteristics. However, further sensitivity analysis of the mission analysis program yielded a large correlation between increased drag and lower score. Although consideration of the high lift characteristics of the Eppler 423 airfoil was considered, the score reduction due to drag was too critical. The Eppler 423 airfoil was chosen for the main wing. Compared to the Selig 1210, the Eppler 423 is worth 2.5 points even with the 30% extra cross sectional area that increases weight. The Selig 1223 has an unfavorable drag curve forcing the cruise lift coefficient into the drag bucket. Since the scores are similar between the E423 and the Selig 1223, the added flexibility of the Eppler 423 is a benefit.

4.1.2 Structures Group

In order to investigate the important features of each component based on its function, the aircraft was broken down into sections. The wing, fuselage, tail section, landing gear, and speed loader all have figures of merit specific to their function in the airplane.

Fuselage: Preliminary design of the fuselage included examining designs for load carrying structures through the length of the fuselage. Several structural members were investigated for their ability to carry bending load through the fuselage. Figures of merit for these structural members include:

- Ability to carry bending moment
- Strength to weight ratio
- Ability to withstand torsion

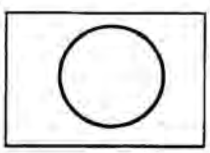
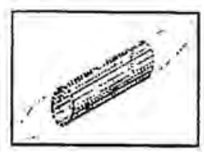
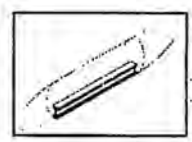
Figures of Merit	Weight Factor	 Reinforced Skin	 Longerons	 Stringers	 Keelson
Weight	0.4	-1	1	0	0
Bending Strength	0.35	-1	1	0	1
Connection Interface	0.15	1	-1	0	-1
Construction complexity	0.1	1	-1	0	-1
Rating		-.5	.5	0	.1

Figure 4.1-3 Fuselage Structure Type Figures of Merit Chart

Wing: With the chosen material for construction of the wing, the most important design consideration was the bending moment that would be carried. With an estimated aircraft weight of 38 to 55 pounds, the bending moment at the root chord could be up to 1650 in lb. The method of mounting the wing to the fuselage is also very critical. The section of wing mounted within the fuselage is only six inches wide over the ten-foot span. The following figures of merit are critical for the wing design and its connection to the fuselage:

- Strength of center section/attachment points
- Weight
- Manufacturability
- Ability to carry Bending Moment

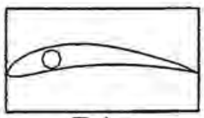
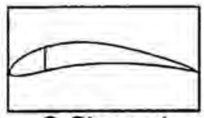
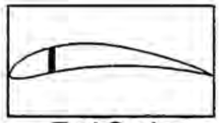
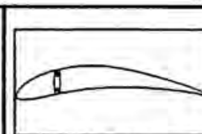
Figures of Merit	Weight Factor	 Tube Spar	 C-Channel Spar	 End Grain Balsa Spar	 Hybrid Spar
Weight	0.4	-1	1	1	0
Bending Strength	0.35	-1	0	1	1
Connection Interface	0.15	1	-1	0	1
Construction complexity	0.1	1	-1	1	0
Rating		-.25	.5	.85	.5

Figure 4.1-4: Spar Type Figures of Merit Chart

Tail: The tail must be mounted in a fashion that will allow incidence angles to be finely tuned before permanently mounting the tail into place. An ideal tail design would also allow the control linkages to be placed entirely within the fuselage to reduce drag. This would be accomplished by reducing the amount of carry-through structure through the fuselage. These things considered, the following are figures of merit for narrowing down a final tail design:

- Able to carry bending moment
- Ease of mounting stabilizers
- Small carry-through structure
- Transmission of torsion to fuselage

Speed Loader: The speed loader is an extremely mission sensitive component of the airplane. Not only does the speed loader design directly affect the sizing and design of the fuselage and carry through structure, but it also affects the Rated Aircraft Cost. If the speed loader is part of the fuselage, the weight of the speed loader is counted in the empty weight of the aircraft, increasing the rated aircraft cost. The design of the speed loader is also critical to minimize reloading time between sorties.

Landing Gear: The anticipated gross weight of the aircraft initially ranged from 38 to 55 lbs. Taking this into consideration, special attention had to paid to the landing gear. The goal was to design the gear to be as light as possible while providing sufficient strength to withstand the anticipated loads and retaining flexibility to reduce the force transferred to the aircraft structure. Due to the extent of taking off, landing and taxing that will be necessary to complete multiple laps, ground handling characteristics of the aircraft should also be exceptional.

Two gear configurations were initially considered: conventional landing gear, and tricycle landing gear. Without a nose wheel, the plane can pitch over if too much brake is applied. Conventional landing gear aircraft are naturally unstable on the ground. So, a conventional landing gear aircraft can be difficult to control when landing or taking off. The tricycle landing gear has the main wheels located at each side of the centerline behind the center of gravity, with a nose wheel on the forward centerline. The tricycle landing gear configuration is noted for ease of ground handling and is difficult to ground loop unless steered into a skid. Taking all of this into account, the tricycle landing gear was selected.

Four different designs were analyzed for feasibility. Figures of merit were used to aid the analysis process Figure 4.1-5. The main characteristics taken into account in the construction of the decision matrix was weight of the gear, drag contributions of the gear, ground handling, dependability and construction complexity. In conjunction with the design parameters investigated, weight and drag contributions were sized as the main priority. Relative importance was placed on ground handling and dependability of the gear while little importance was paid to construction complexities.

Since the ratings of the respective designs were extremely close, a decision had to be made by considering another factor is the ease of replacement. This was to be based solely on whether the design would be easily replaceable if the unforeseen happens. Since, three of the designs were struts, this meant that the main gears would have to be placed at the wing. If this were the case, it would be difficult to replace the gear if it were to fail. Replacement could only be made if a hatch were incorporated into the wing section. This could be avoided by using the bow type gear. The bow type gear will be attached to the wing but at a location where it will be located directly below the fuselage. Taking this consideration, the gear would be bolted directly from the fuselage. This configuration allows one to change the gear easily as there would already be a hatch incorporated into the design to accommodate the speed loader.

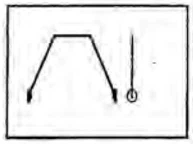
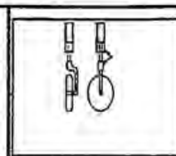
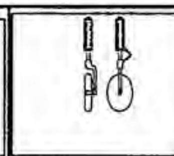
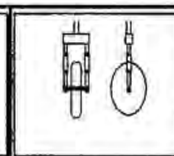
Design Criterion	Weight Factor	 Conventional Bow	 Non-expandable	 Expandable	 Two-Stroke
Weight	0.4	1	0	0	0
Drag	0.2	-1	0	0	-1
Ground Steering ability	0.15	-1	1	1	1
Dependability	0.15	0	0	0	1
Manufacturability	0.1	1	-1	-1	-1
Rating		.15	.05	-.05	0

Figure 4.1-5: FOM Decision Matrix used for Landing Gear Selection

4.1.3 Propulsion Group

The parameters that we considered to be important in the design of our propulsion system were power output, thrust produced, overall efficiency, and weight. The power produced by our system is very important because it is the determining factor in how many laps we will be able to complete. Therefore, we selected our batteries based on how much power we could get out of a 5lb pack. The thrust was also an important consideration because we are lifting up to 50 lbs over a length of 200 feet. Efficiency and weight also affect the performance of the plane, so we designed for maximum efficiency and relatively low weight.

The FOMs that we constructed aided in our propulsion system component selection. In each FOM the areas of consideration are weighted according to their importance. The alternatives are then rated on their level agreement with each area of consideration. The final score for each alternative is a result of the weights and ratings and gives us a good idea of which component best satisfies our needs. For the propulsion system selection we have constructed FOMs for the batteries and the motor. These figures, used in conjunction with analytical data and test results help us to choose the best possible propulsion system according to our needs.

4.2 Analytical Methods

4.2.1 Aerodynamics Group

Performance

The mission analysis program discussed in section 3.1 evolved throughout the design development. As shown in Figure 4.2-1, the process was iterative with further expansion for detailed design. The mission analysis program was modified to include experimental airfoil lift and drag as well as propulsion data. Sensitivity analysis was performed to verify the validity of initial assumptions and to select better flight routines. Inclusion of these functions allowed a more robust and versatile mission analysis program. Experimental low speed airfoil data was found in the UIUC low speed airfoil tests (http://amber.aae.uiuc.edu/~m-selig/). This allows the mission analysis program to use a piecewise best-fit line of the airfoil's polar plot. The modification of the mission analysis also allowed propulsion data such as batteries, speed controller, motor and propeller to be included.

Sensitivity analysis was performed to verify the validity of initial assumptions and to select the best flight routine and aircraft geometry. Important score increasing tactics were found from performing sensitivity analysis on the sortie laps, tennis ball packing, wing airfoil, wind velocity, weight, landing distance and cruise height.

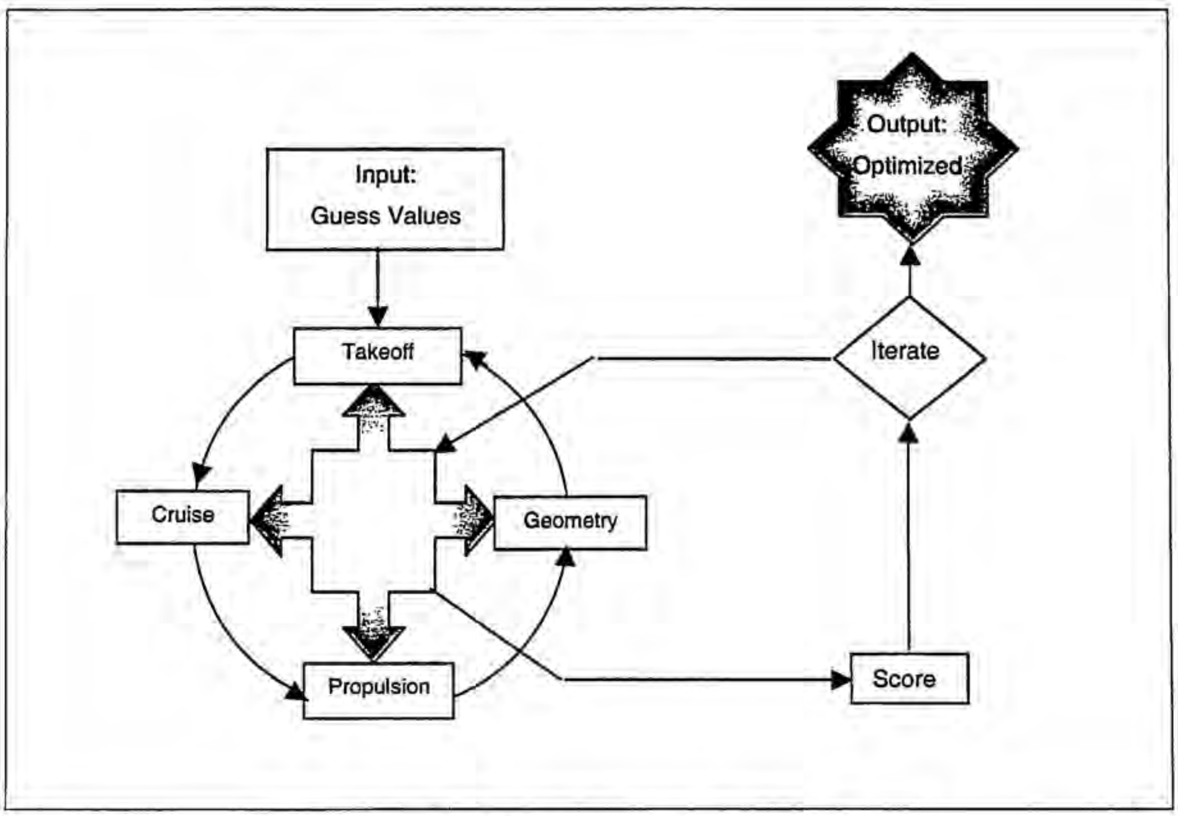


Figure 4.2-1: Mission Analysis Program Flowchart

Drag

The method used to estimate parasite drag was the component buildup method. This method sums the individual drag components of the aircraft by calculating a flat-plate skin-friction coefficient and a form factor that estimates the pressure drag due to viscous separation across the surface. Additionally, an interference factor is used to approximate the interference component between surfaces. Each component is comprised of the product of the drag factor, interference factor, wetted area, and form factor. The major components considered in this buildup method include fuselage, wing, horizontal tail, vertical tail, landing gear, and fuselage upsweep. Other drag components include induced drag from lift and trim drag.

4.2.2 Structures Group

Landing Gear

First and foremost, the location of the main and nose gear had to be calculated. Overturn angle and tip over angle had to be taken into consideration on deciding upon the location. It is noted that the minimum static load has to be greater than 6% of the aircraft weight. Calculations have shown that the CG could vary from 0 to 1 inch of the present center of gravity (25.4 inches from the nose of the aircraft). Taking these into consideration, calculations were performed by taking the sum of moments about the center of gravity. The following was obtained:

Distance between nose gear and main gear = 19 inches

Distance between nose of airplane and nose gear = 6.12 inches

The height of the landing gear is approximately 6 inches from the bottom of the wing to the ground. This distance was the minimum length needed in order for the propeller to have an appropriate clearance.

The main landing gear was based on 4g load for structural integrity where we assumed we assumed the worst case for the plane when it landed without flaring on one wheel. Calculations were also made to determine the dependability of the gear. A bolt-member analysis was conducted using Rotsher's pressure cone method for stiffness calculation to obtain the type and number of bolts that would be needed to secure the gear to the aircraft. Calculations show that two-¼ inch SAE Grade 1 steel bolts would give a factor of safety of 2.4 against fracture. Shear stress analysis showed that by placing the bolts at 5 inches apart, a factor of safety of 3.12 is obtained against shear.

4.2.3 Propulsion Group

In order to achieve the maximum performance possible from our system we incorporated an mission analysis spreadsheet created using MathCAD software. This analysis allowed us to get theoretical thrust, velocity, efficiency, and power results for any combination of batteries, motors, and propellers, before we ordered and tested any of our components. This was invaluable in the preliminary design phase because it gave us a good overall picture of what was happening with different combinations of components.

In addition to using theoretical analysis tools, both static and dynamic tests were performed. The static tests were bench tests performed using a dynamometer. The static tests included: battery endurance tests (at various fixed power settings and at variable power), motor parameter tests, installed performance tests (20%, 50%, and 80% blockage, as well as streamlined cowling effects), charge rate effects test, motor and battery cooling effects, and fuse effects. The dynamic tests were performed in the wind tunnel using the dynamometer. Wind tunnel tests included experiments on propeller efficiency, variations on takeoff, and predicted flight profile. The use of wind tunnel tests were highly beneficial because the combination of components of the propulsion system create highly coupled behavior that is complicated and difficult to predict using analytical methods.

4.3 Configuration and Sizing Data

4.3.1 Aerodynamics Group

Stability and Control

Stability and control issues include several aspects, such as tail size, fuselage size, wing parameters, and control surface size and deflection angles. There are two main categories and characteristics under which stability and control operates, longitudinal and lateral control which each contain static and dynamic considerations. Static stability is present if the forces resulting from a disturbance return the aircraft to the original state; dynamic stability is evident if the dynamic characteristics of the aircraft return it to an initial condition. The process by which it returns is dependent upon the aircraft center of gravity and forces resulting from the control surface deflections. All equations, approximations, and historical design values came from Raymer's Aircraft Design: A Conceptual Approach.

Longitudinal Stability

Initial Horizontal Tail Sizing: The first step in the stability and control analysis included sizing the horizontal stabilizer. This calculation came from approximating the distance from the wing quarter chord to the horizontal quarter chord to be sixty percent of the fuselage length. The provided aspect and taper ratios related wing reference area and tail volume coefficients.

Aircraft Pitching Moment: The next process comprises the governing moment equation for the entire plane. Moments for the fuselage and wing as well as the wing and horizontal tail lift moments were added to the moment resulting from the thrust.

The magnitude of the pitching moment derivative changes with the aircraft center of gravity. Power off neutral point is calculated by setting the derivative to zero. This defines the most aft aircraft center of gravity, also known as the aircraft neutral point, location to be determined for stable flight. The static margin is computed from the neutral point location. If the center of gravity is ahead of the aircraft neutral point, the static margin is positive and the total pitching moment negative. This results in a stable configuration as shown in Figure 4.3-1.

A static margin of 15% is assumed to be adequate in maintaining stability. Upon calculation and comparison, the sensitivity to the static margin is large. The difference between the center of gravity location and the wing aerodynamic center is approximately an inch, which is better defined as the difference between a stable and an unstable aircraft.

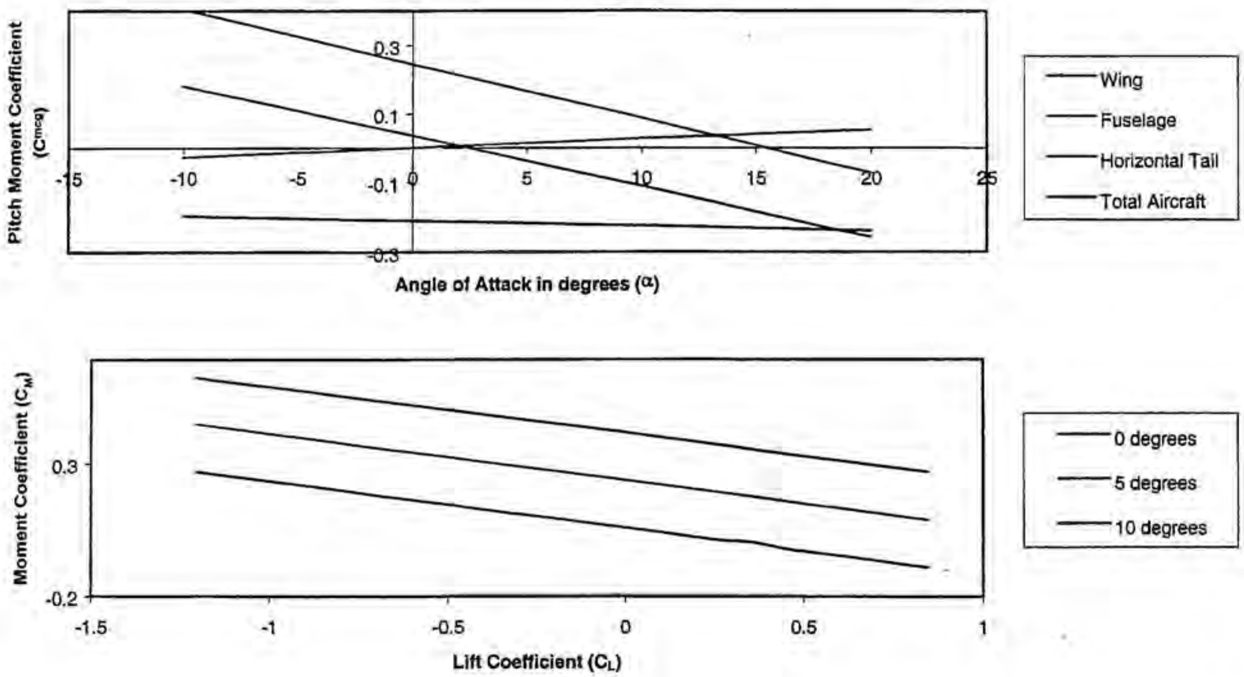


Figure 4.3-1: Pitching Moment and C_M vs. C_L at Changing Elevator Angle

Trim Analysis and Tail Resizing: The next step involved the trim characteristics at various angles of attack and elevator deflection. This is accomplished by calculating the total aircraft moment (C_M) and the lift coefficients (C_L). Since the weight of the aircraft and the wing platform area are previous defined, the total aircraft lift coefficient needed in cruise flight is a simple relation of aircraft weight wing area and dynamic pressure. The trim elevator deflection angle for a specific C_L is found graphically where the aircraft C_M is zero.

The initial C_M versus C_L graph does not provide a reasonable elevator deflection during cruise for the range of necessary aircraft lift coefficients. To adjust the trim graph, the horizontal tail area, elevator percentage of the horizontal chord, and horizontal tail incidence angle are adjusted until the elevator deflection is near neutral for the range of aircraft lift coefficients.

Takeoff considerations were also considered. Additional moments were added to the total aircraft pitching moment for the drag forces resulting from rolling wheel friction and the weight on the rear landing gear as the aircraft changes angles of attack. This takeoff check assumes that the aircraft has eighty percent of the takeoff thrust and can produce a positive aircraft C_M with elevator deflection.

After completing these two flight scenarios, the horizontal tail area and aspect ratio were set to allow the aircraft to fly with little or no elevator deflection. Although the horizontal tail platform area is significantly larger than the initial estimates, the short coupling of the fuselage, to reduce RAC, combined with a large wing lift force dictates that the horizontal surface will have to be significantly enlarged to provide a stabilizing moment.

Horizontal Tail Airfoil Selection: Once the horizontal tail specifications were determined, an airfoil had to be selected. Two types of airfoils that were considered: symmetrical and cambered. Symmetric airfoils provided the best drag solution and ease of construction. However, in the trim analysis the incidence angle was set at –5° to achieve the necessary tail lift; this does not provide for low drag in cruise because the horizontal tail is not aligned with the flow. Cambered airfoils provide the next option.

In selecting the airfoil, several parameters had to known and considered. Mean aerodynamic chord, lift coefficient in cruise, and required lift coefficient during takeoff dictate the airfoil operating parameters. The first consideration includes the mean aerodynamic chord of the horizontal surface. The Reynolds number over the airfoil decreases with the chord; in a low Reynolds number range the airfoil becomes less effective and efficient in providing lift. The surface is operating in a low Reynolds range but is not approaching the lower limit ineffectiveness. Thus, low Reynolds number airfoils are the major limitation in the horizontal selection.

Additional parameters include the lift coefficients needed in cruise and takeoff. These are found from the trim analysis by solving for the horizontal tail lift coefficient component. Many low Reynolds number airfoils were plotted on polar graphs using the data from the University of Illinois at Urbana-Champaign website. The airfoils that operated in the required C_L range were compared with the final having a combination of low drag coefficient and steep C_L versus C_D slope. The Eppler 387 airfoil was selected.

Aerodynamic Balance: From the drag, propulsion, and score mission analysis, the aircraft is optimized for low drag and high speed. Concern arises in the event of a steep negative aircraft angle of attack coupled with high velocity. In this situation, the elevator servo can become overpowered by the dynamic pressure and thus be ineffective at restoring pitch control of the aircraft. This can be catastrophic in the end results for both the aircraft and spectators. Therefore, an aerodynamic balance analysis was performed on the horizontal surface.

For this analysis the maximum speed of the aircraft was assumed to be twenty percent above the full throttle cruise speed with the elevator at maximum deflection. The servo torque is set equal to the difference between the moment from the elevator and aerodynamic balance surface resulting from dynamic pressure. From this relation, the span of the aerodynamic balance is solved in terms of horizontal tip chord, horizontal span, and elevator chord.

Lateral Stability

Aircraft Roll and Yaw Moments: Initial vertical tail sizing was conducted in the same manner as the horizontal with wing to tail relations. Historical aspect and taper ratios and distance between the center of gravity and mean aerodynamic chord of the vertical tail were used. The main driver in lateral stability calculations is the roll and yaw moments with the change in slide slip angle (β). Moments from the wing, fuselage, ailerons, and vertical tail are summed to complete the aircraft roll and yaw moments.

While the longitudinal moments can determine the aircraft center of gravity, these equations do not refine this estimate. These moments are less influential over center of gravity location compared to the aircraft pitch moment considerations.

The roll stability ensures the aircraft will restore a wings level position when disturbed. If roll stability is not present, the resulting roll moment will be in the same direction as the perturbation and thus reinforce the roll rate. When roll stable, the roll moment created by the sideslip will counteract the disturbance and return the aircraft to a wings level attitude. Roll moment is composed of several components; the most important being the dihedral effect from the wings, fuselage effect, and the vertical tail.

The greatest contributor to roll stability is the dihedral effect from the dihedral angle of the wings. For this low wing design, the fuselage is destabilizing while the vertical tail adds slightly to the stability. Stability is achieved through a negative slope, which can be seen in Figure 4.3-2.

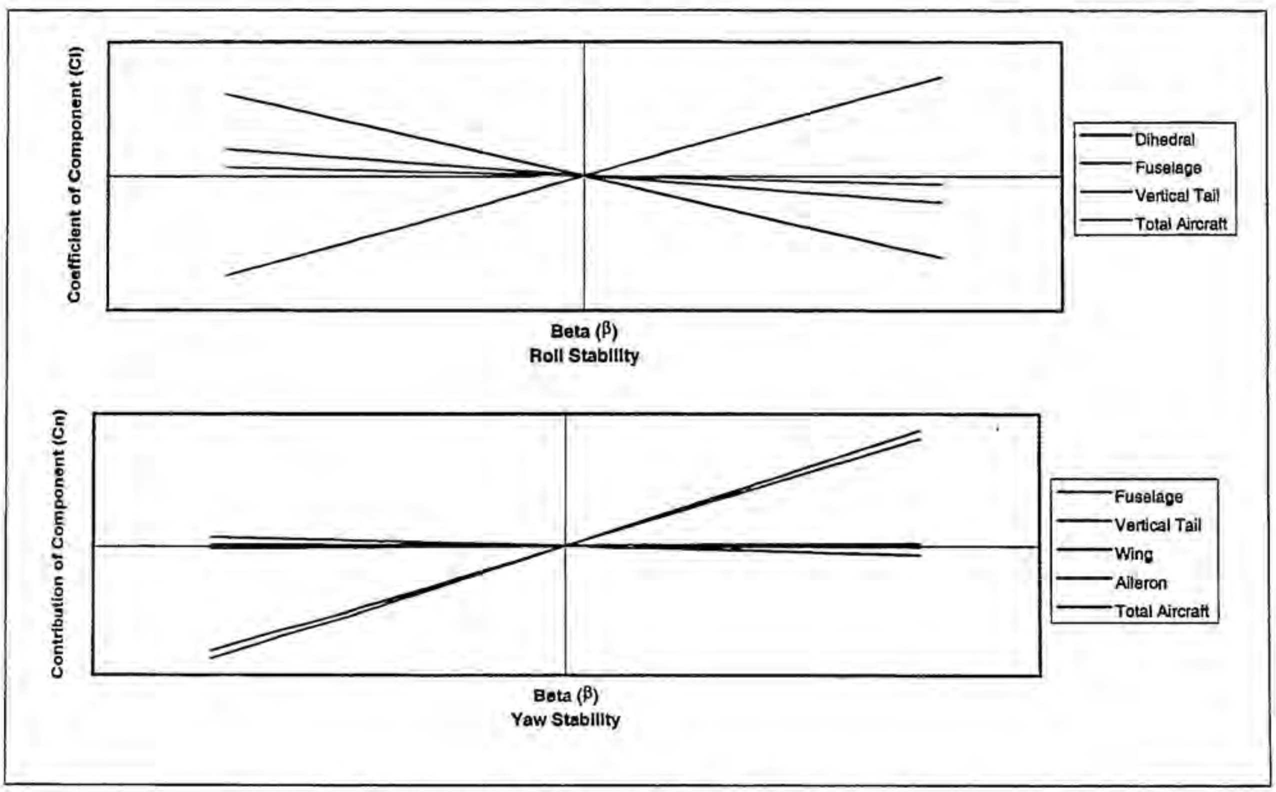


Figure 4.3-2: Static Stability Contributions

The yawing moment is comprised of components from the vertical tail, fuselage, wing, and aileron deflection. The major component is the lateral lift from the vertical tail. Rudder deflection increases this effect, which counteracts the destabilizing fuselage component. Yaw stability is characterized by positive slope, as can be seen in Figure 4.3-2, elevon deflection is arbitrary.

Rudder Sizing: Completing the lateral stability is the sizing of the rudder and elevon effectiveness. This is done simultaneously with the roll and yaw moment graphs. Since the size of the elevator has been set in the longitudinal analysis, only the effectiveness of elevon control can be computed rather than sizing considerations. Initial parameters were obtained from Raymer and these measurements were further refined for our specific aircraft through simultaneous solving.

4.3.2 Structures Group

Fuselage: The choices for fuselage structural members were a combination of bulkheads, longerons, keelson, stringers, and reinforced skins. A combination of longerons and bulkheads mounted to reinforce skin was chosen to fulfill all of the figures of merit for the fuselage. This combination provides bending moment and torsional strength, which will allow the aircraft to remain rigid.

The bending moment strength of a longeron is eight times greater than that of a stringer with a factor of two increase in weight. Torsion will be resisted by the bulkhead and reinforced skin interaction. The keelson was eliminated due to its method of reinforcement. A keelson would run the length of the plane through the floor of the aircraft, which in our design is interrupted by the recessed wing. This structure would have to be bent around the wing and would therefore be less effective than a longeron and much more difficult to construct. These factors, previously weighed out in section 4.1.2, justify the selection of longerons for structural reinforcement.

Wing: To carry the large bending moment in the center of the wing, a spruce spar separated by a foam block and covered with carbon fiber twill will be utilized. Balsa wood in the center section of the wing along the leading edge will strengthen the connection of the wing to the fuselage. This design choice allows a strong spar to carry the bending moment with the carbon fiber skin acting as a safety factor. The balsa wood center section allows a good surface to connect the wing to the wing to the fuselage with pins and bolts.

Other methods of carrying loads from the wing to the fuselage and well as carrying the bending loads in the wing were also considered. A center section composed of honeycomb was an alternative to the balsa wood, but did not provide a good surface for mounting pins to for a secure connection of the fuselage. Additionally, a spar comprised of concentric carbon fiber tubes was considered. The spruce spar was chosen over this design in order to provide a greater moment of inertia with less weight and utilize the weight added to the wing to the fullest. These factors were weighted in Section 4.1.2 to show the final selection of a hybrid spar composed of spruce and carbon fiber twill.

Tail: The chosen tail design is a strategically ribbed tail allowing carbon arrow shaft to be firmly mounted and then inserted through the fuselage for transmission of bending and torsion forces. The relatively small diameter shafts would allow the control linkages to pass through the rear of the fuselage. In order to firmly attach the stabilizers to the tail, the inboard sections will be made of balsa wood that will be shaped and glued to the fuselage surface.

The chosen design fulfills all of the figures of merit outlined for tail design. The carbon fiber arrow shafts provide a spar for the large tail as well as acting as carry through and attachment structure to the fuselage. Although a carbon fiber tube is not the optimum shape for carrying bending forces, it is an excellent compromise to perform all of the functions the tail components must satisfy as indicated by the figures of merit for the tail in Section 4.1.2.

4.3.3 Propulsion Group

Battery Sizing: The battery type that will be uses was predetermined by the competition rules, which stated that the power supply must consist of Nickel-Cadmium batteries, and must weigh 5 lbs or less. The battery selection was a very high priority for the propulsion team. Batteries designed for high voltage, with high capacity, and able to withstand the harsh conditions of high performance applications were needed. It was found that the Sanyo Fast Charge Batteries (R-Series) best suited the performance needs. Data for the different sizes of the Sanyo batteries is shown in Table 4.3-4.3-1.

Part Number	Size	Capacity (mAh)	Mass (g)	Price (US Dollars)	Capacity per Mass (mAh/g)
N-800AR	A	800	34	3.00	23.53
N-1300SCR	Sub-C	1500	52	2.25	25
N-4000DRL	D	4000	160	5.50	25
N-1250SCRL	4/5 Sub-C	1250	43	3.50	29.06
N-3000CR	C	3000	84	4.50	34.71
N-1900SCR	Sub-C	1900	54	3.50	35.19
RC-2400	**Sub-C**	**2400**	**54**	**5.50**	**44.44**

Table 4.3-4.3-1 Sanyo Fast Charge Battery Statistics – Provided by TNR Technical

After considering the performance curves for several types of batteries and looking at the manufacturers performance data RC-2400 battery were selected for the propulsion system. The RC-2400 had the highest capacity per gram and was relatively lightweight. At 1.9 ounces per cell the RC-2400 would allow us to have more cells in our battery pack than the higher capacity batteries and still provide us with 24.3 % more overall capacity.

Decision Factor	Weight	N-1900SCR	RC-2400	N-3000CR
Weight	.4	0	1	-1
Efficiency	.2	0	1	1
Capacity/Mass	.3	0	1	1
Cost	.1	0	-1	-1
	Total	0	.8	0

Table 4.3-2: FOM Decision Matrix for Battery Selection

Figure 4.1-3 demonstrates the maximum velocity attainable for different throttle settings limited by battery power and physical constraints of the aircraft. The chart verifies the power plant chosen will produce enough power to achieve the desired cruise velocity and then some. The intersections illustrate the maximum velocity at a given throttle setting for different loading conditions based on the propeller discussed in the next section.

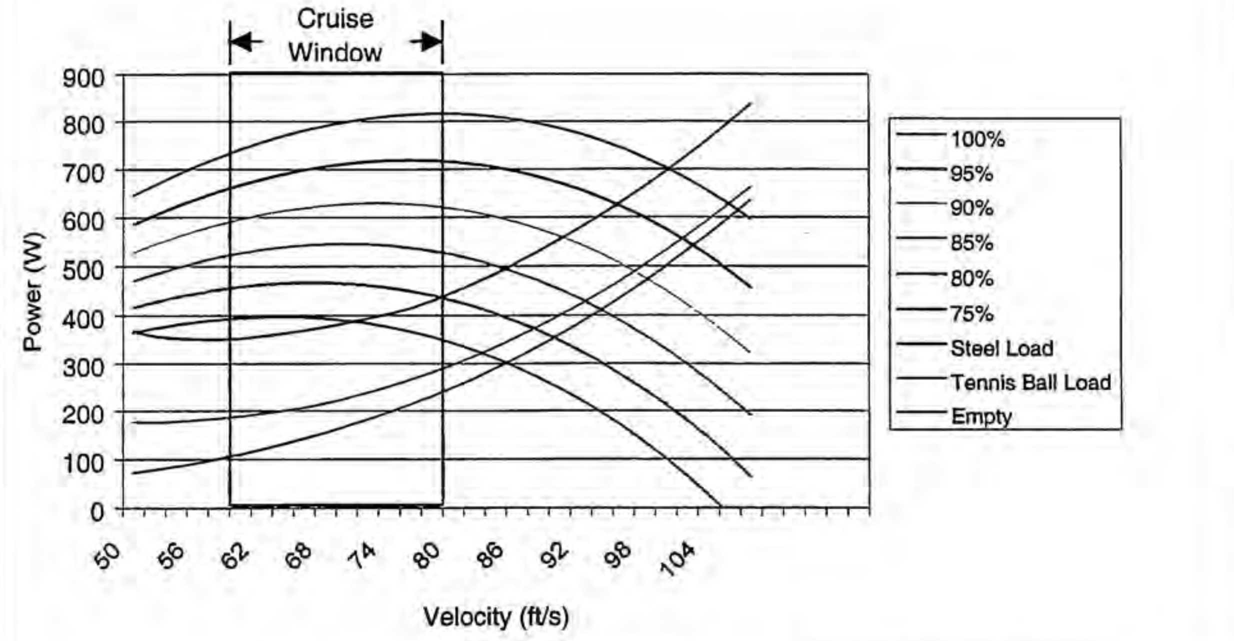


Figure 4.3-3: Power Available vs. Power Required for Various Throttle Settings

Motor Sizing: The AstroFlight Cobalt 60 (#661), 90 (#691), and 40 (#640) motors with Super Box gearing were examined. In determining which motor would best suit the performance needs, parameters including current, weight, power produced by the assembled propulsion system, and efficiency were considered. The initial design concept called for a propulsion system that could handle 1500 Watts of power, produce a minimum of 9 lb of static thrust, and have high efficiency. Analysis was begun with a conservative estimate of an overall efficiency of 40% based on historical data.

Basic motor performance analysis was performed using data supplied by the manufacturer. Theoretical analysis showed that the 40-size motor would not provide enough thrust for takeoff. High current (30 Amps) damage was also a concern. However, the efficiency of the 640 was slightly higher than for the other two motors. The 661's efficiency was quite close to the 640's efficiency and it produced much more thrust. It also had the ability to handle larger current and voltage inputs. There was a concern over the weight gained from using the larger motor, however this was determined to be acceptable for the thrust and efficiency combination of the motor. The 690 produced more thrust than the 661, however the efficiency dropped off. The 690 would require a larger and propeller and the combined weight and length issues would impact the RAC and other flight systems such as landing gear. Figure 4.3-4 shows the torque and efficiency curves for each of the three motors that we considered. Table 3.4-1 shows the other motor parameters that we considered. By making direct comparisons between the motors we were able to select the motor that would best serve our needs. It appeared to us that the 661 would be a good compromise between the 640 and 690 motors.

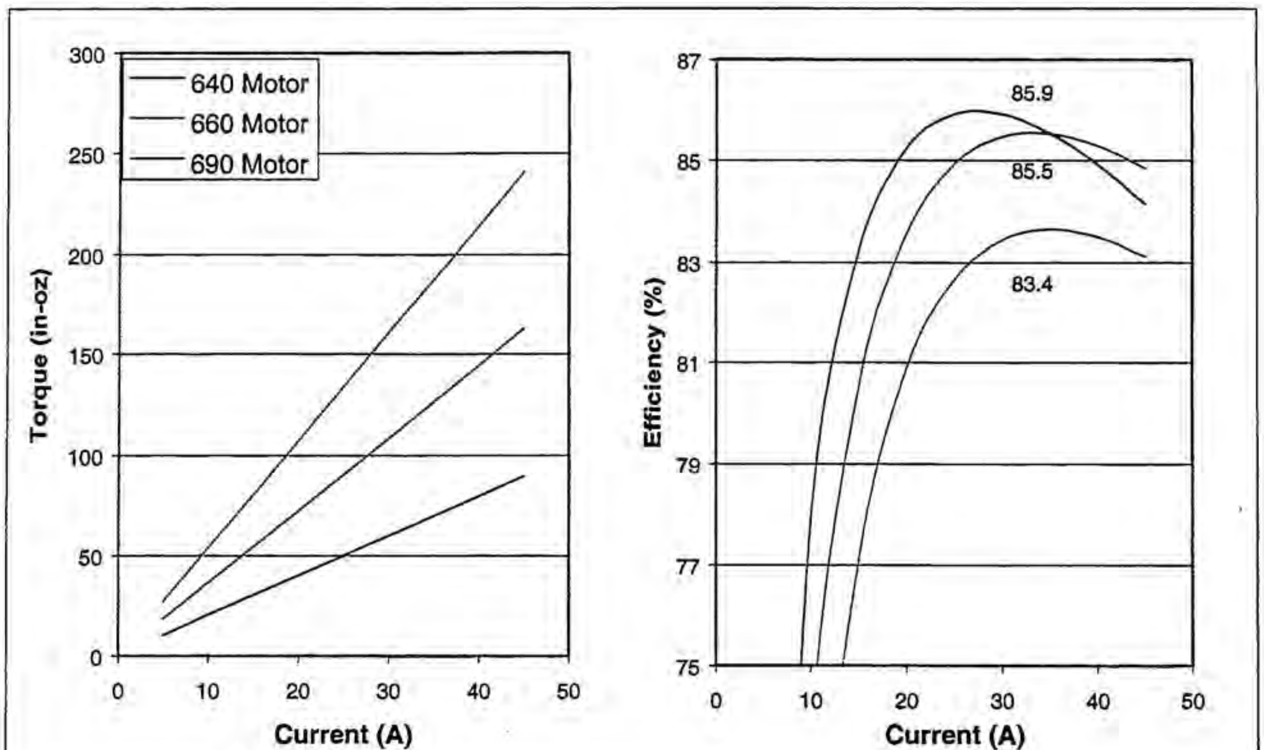


Figure 4.3-4: Performance for selected AstroFlight motors

Using the decision matrix shown in Table 4.3-2 and collaboration with the Aerodynamics team, we concluded that to get the best performance we should use the AstroFlight 60 size motor. The Aerodynamics team required about 1200 Watts of power for takeoff and climb and about 680 Watts for cruise, with approximately 9 lbs of static thrust for takeoff. They also wanted the efficiency of the propulsion system to be at least 40% based on historical data. Using MathCAD as our primary analysis tool we found that the AstroFlight 60 motor could meet all our criteria for the propulsion system to power our airplane.

Decision Factor	Weight	Astro 40	Astro 60	Astro 90
Power Output	.2	-1	0	1
Efficiency	.3	0	0	-1
Ability to Handle Current Load	.1	-1	0	0
Cost	.1	0	0	1
Weight	.2	1	0	-1
Availability	.1	0	0	0
	Score	-.1	0	-.2

Table 4.3-3: FOM Decision Matrix for Motor Selection

Propeller Sizing

The final component of the propulsion system to be selected was the propeller. The primary analysis of the propellers came from performance curves generated from experimental analysis. Using MathCAD to compute parameters for different P/D ratios, gear ratios, and propeller diameters we were able to determine thrust produced during different portions of the flight, overall efficiency of the system, current drawn, power available, and speed of the airplane. All of these parameters could be determined for both steel and tennis ball sorties. The graphs in Figure 4.3-5 show how the available power changes with P/D at different C_T, C_P, and η values. These graphs were used to determine what size and pitch needed for the propeller.

The values in Figure 4.3-6 gave us a baseline for further propeller testing and selection. Utilizing this information, we were able to obtain a reasonable range of propellers to then test dynamically. We performed flight profile tests and propeller efficiency tests on several types and sizes of propellers including wooden, APC, and carbon fiber as well as two- and three-bladed propellers.

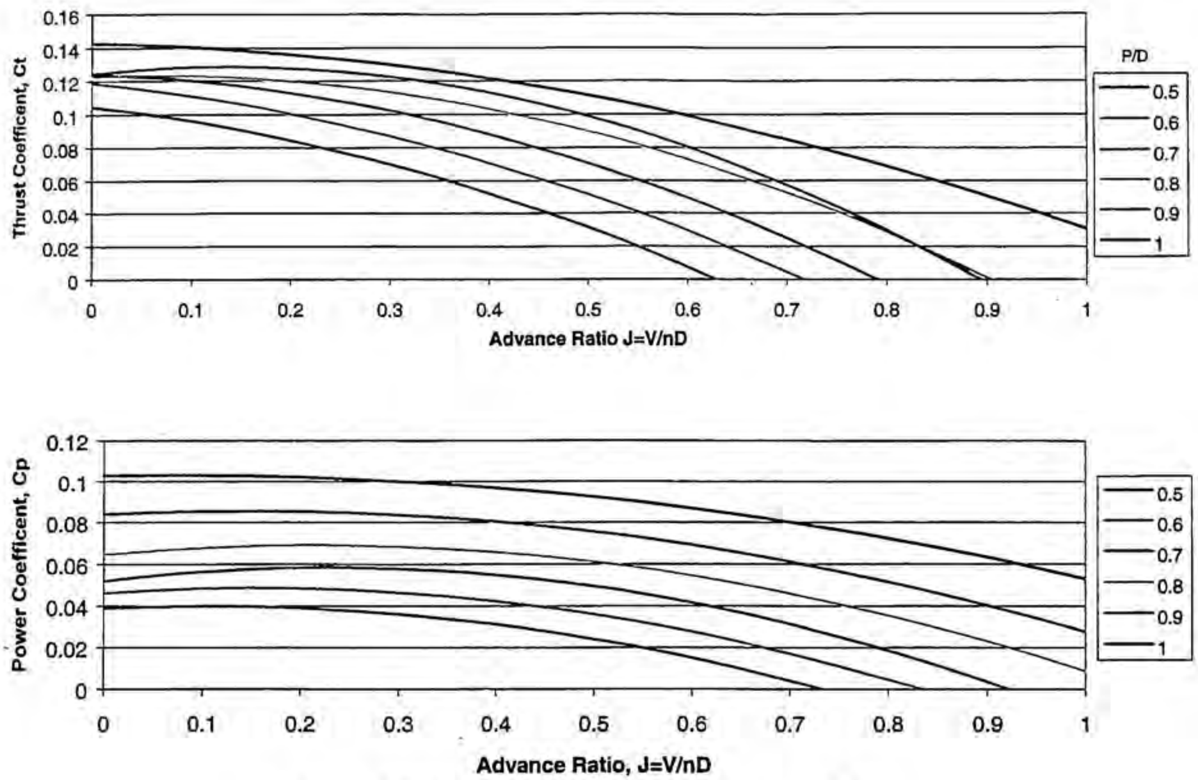


Figure 4.3-5: Advance Ratio vs. Thrust and Power

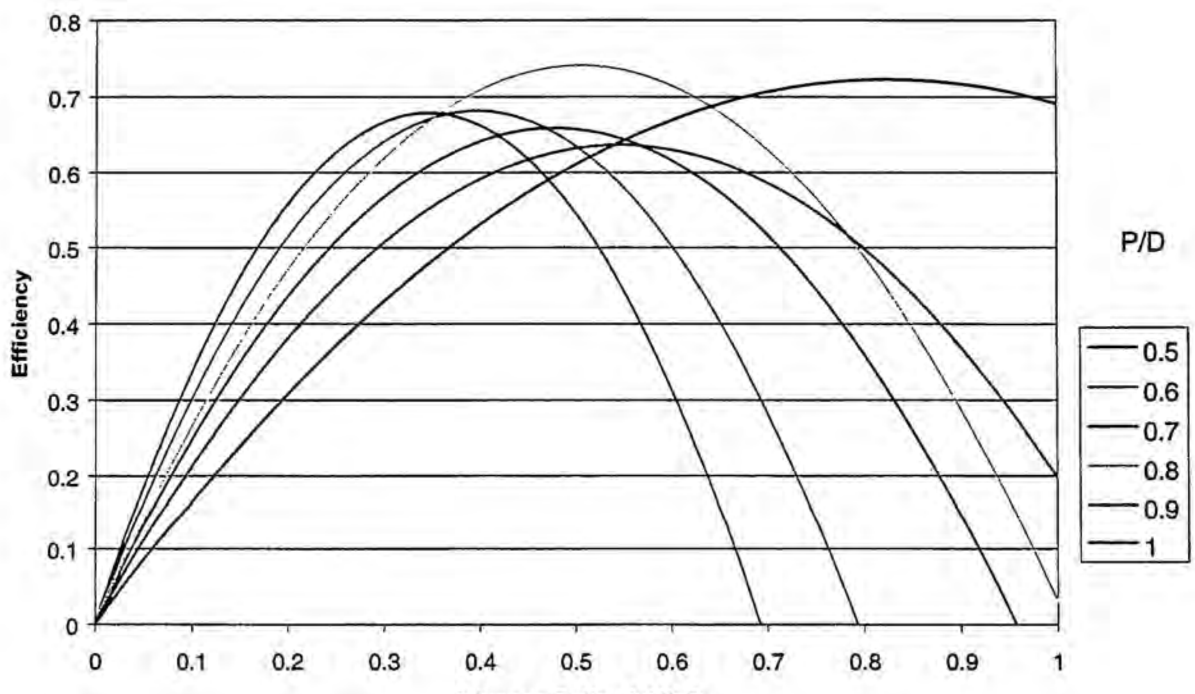


Figure 4.3-6: Advance Ratio vs. Efficiency

4.4 *Features that produced the Final Configuration*

4.4.1 Aerodynamics Group Summary

During the preliminary design phase the aerodynamics team analyzed the drag generated by the aircraft and explored methods for reducing the overall drag. Among those methods were wheel pants, careful selection of wing airfoil, and tail airfoil and incidence angle. The performance needed by the aircraft demanded a high-lift airfoil and the Eppler 423 was selected for the main wing as well as adding polyhedral for added stability. The short-coupled fuselage called for a moderate-lift, inverted airfoil for the horizontal stabilizer. For this the Eppler 387 was selected, with a negative incidence angle of 1°. A NACA 0009 airfoil was chosen for the vertical stabilizer. Aerodynamic balances were added to the horizontal stabilizer to aid in the servo's ability to overpower flight forces.

4.4.2 Structures Group Summary

Preliminary design yielded the following parameters for the final aircraft, which is to be built entirely from carbon fiber and foam. The fuselage skin will consist of a foam sandwich structure reinforced with longerons and bulkheads constructed from either foam or end grain balsa sandwich structures. The wing will consist of an airfoil cut from foam, covered in carbon fiber skin, and reinforced to carry bending moment with a hybrid spar constructed from a spruce and carbon fiber twill sandwich. The tail section will be structurally reinforced to carry bending moment, made simple to mount to the fuselage at the correct incidence angle, and transfer torsion to the fuselage through the use of carbon fiber tubes mounted in plywood ribs. The speed loader will consist of a cylinder mounted inside the center section of the fuselage to which a foam hatch will attach for fast removal. This hatch will be removable from the speed loader for placement in the fuselage without the speed loader itself. Finally, the landing gear will consist of a bow gear fabricated from multiple layers of carbon fiber and attached to the aircraft with two steel bolts.

4.4.3 Propulsion Group Summary

The propulsion system will consist of the 661 motor, 37 RC-2400 batteries, a gear ratio of 2.7, which was selected through analytical analysis and performance optimization, and a 22-inch propeller with a 12-inch pitch. This system provides 14 lbs of takeoff thrust with a cruise thrust of 3.59 lbs, a cruise velocity of 73.87 ft/s, and 800 Watts of cruise power for the steel payload. For the tennis ball payload the cruise thrust is 2.89 lbs at 72.45 ft/s and 600 Watts of power. The maximum power from the batteries is 1076 Watts and the usable power will last for 585.6 seconds. With our planned sortie runs we will use 89.7% of our energy. The power margin will allow us the option of making an extra lap if the conditions are favorable.

5 Detail Design

5.1 Performance Data

The mission analysis program was used to iterate through the possible optimum aircraft designs. The following aircraft configuration yielded the highest and most consistent score of 30.2 points with a 5-mph. wind.

A conventional configuration monoplane with the parameters shown in Table 5.1.1 below was found to be the optimum combination. This particular configuration is a maximum score while still providing for uncertainties in the flight profile. In particular, the mission profile is well under 600 seconds with battery power still available.

The number of steel and tennis ball laps ultimately determines the final score. Because the number of laps is influenced by the energy available, allowable time and individual sortie payload, flying the optimum profile is critical. The time requirement of ten minutes essentially prevents more than 8 total sorties. The maximum score without wind for each legal combination of steel and tennis ball laps is given in Figure 5.1-1. The optimal payload profile is 3 steel and 3 tennis ball laps.

Geometry	
Wing Span	10 ft
Wing Area	8.85 ft^2
Aspect Ratio	11.3
Fuselage Length	5 ft
Empty Weight	14.8 lb
Gross Weight	52 lb
Payload Fraction	72%
RAC	4.56

Performance	
Max Velocity	70 mph
Steel Cruise	50 mph
Tennis Ball Cruise	51 mph
Stall Speed (Gross Weight)	39 mph
Stall Speed (Empty)	23 mph
Glide Ratio (Gross Weight)	17.8:1
Steel Takeoff Distance	180 ft
Tennis Ball Takeoff Distance	60 ft
Landing Distance (Gross Weight)	240 ft

Payload	
Steel Wind 0 mph	26 lbs
Steel Wind 5 mph	32 lbs
Steel Wind 10 mph	35 lbs
Steel Wind 15 mph	35 lbs
Tennis Balls	100

Endurance	600 s.
Range	24000 ft.

Handling Qualities
Stable in all axis
Nose wheel steering
5 Control Surfaces
Brakes

Table 5.1-1: Aircraft Specifications

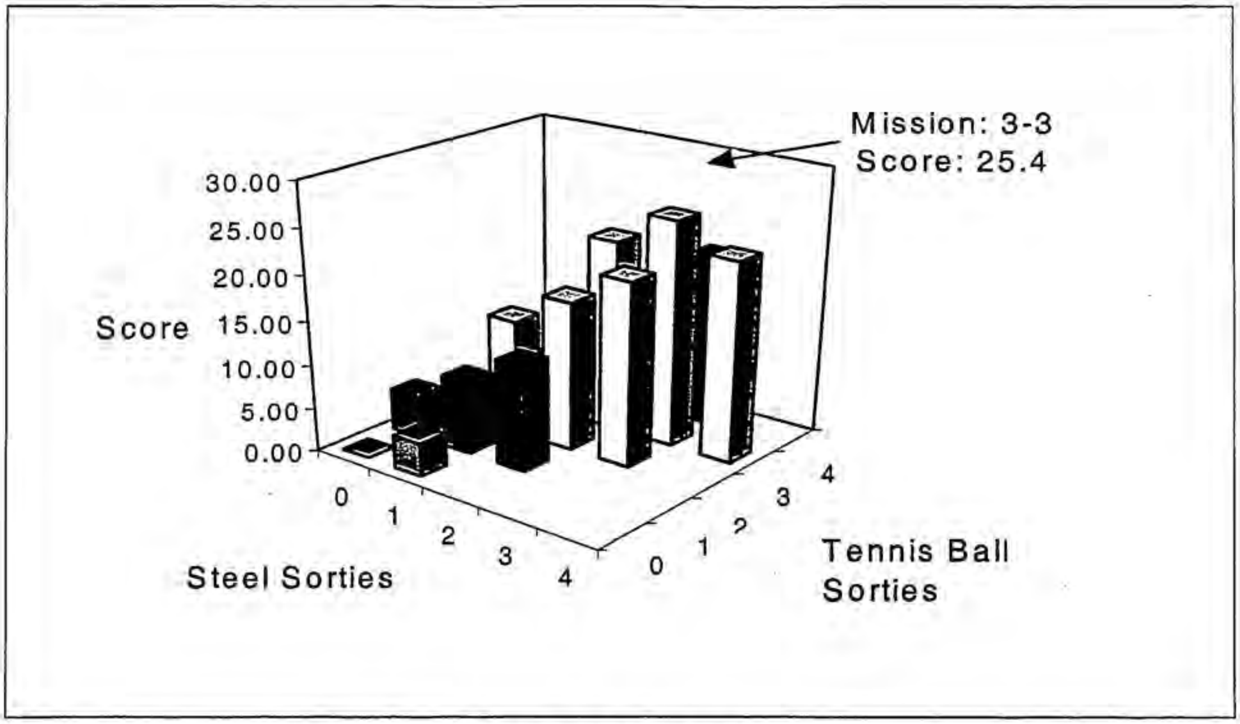


Figure 5.1-1: Sortie Mission analysis

The tennis ball packing influences drag and RAC. Because efficiently packing tennis balls requires certain fuselage cross section sizes, only the first four reasonable tennis ball cross section configurations were analyzed. The configurations were a 4, 7,10 and 14 ball cross sections. Small configurations required a long fuselage while larger configurations had excessive drag. Figure 5.1-2, the highest score was achieved with using a 10 ball cross section. From this analysis, the relationship between possible geometries and ultimate score has been constrained by the ultimate score.

There was strong sensitivity of the score to the empty weight. From Figure 5.1-2, a 1-pound change in empty weight yields a 1.3-point change. The increase is score is due to both the Rated Aircraft Cost and the ability to carry more payload. While decreasing the weight always increases the optimization's score, reality requires a structurally sound and durable aircraft. As expected, low structural weights must be balanced with a structural integrity.

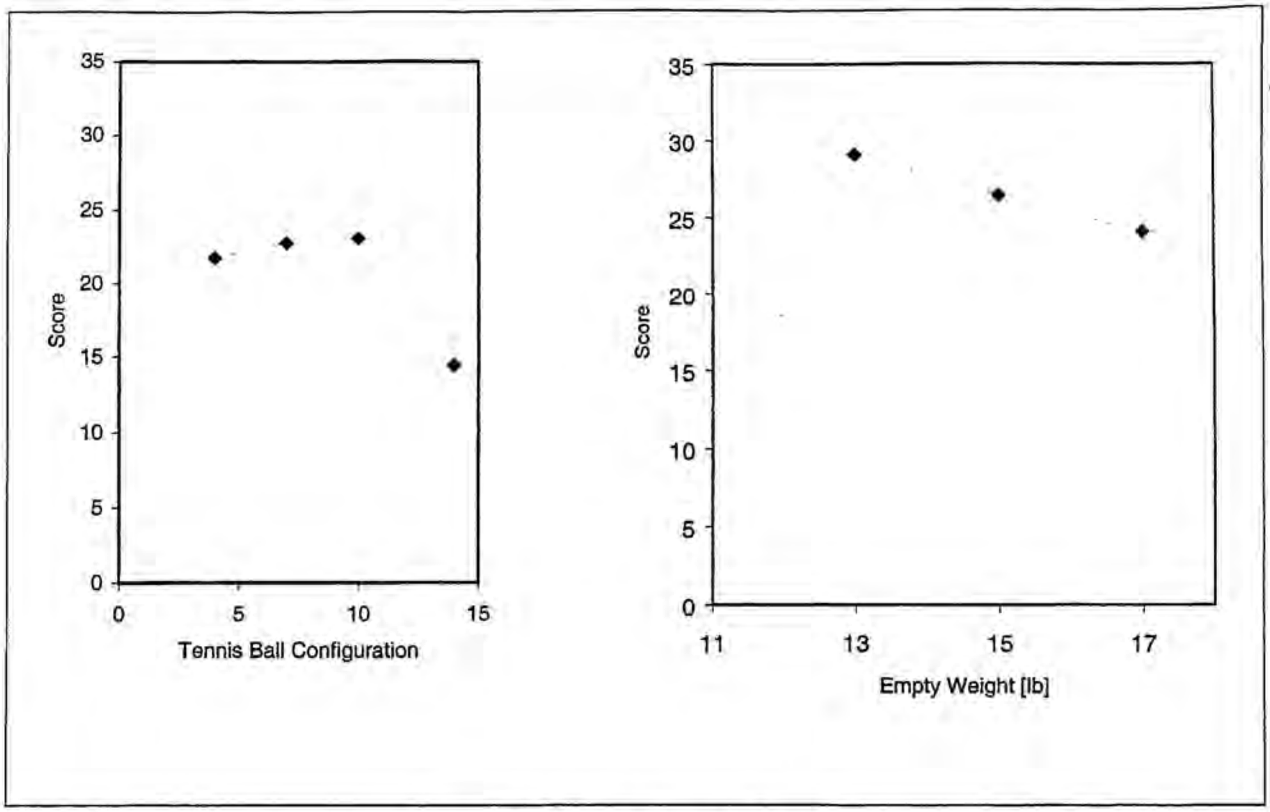


Figure 5.1-2: Configuration Trends

As expected, wind affects the score. An upwind takeoff increases the maximum steel weight due to the 200-foot takeoff distance constraint. Below the optimized wind velocity, the aircraft is unable to takeoff without exceeding the maximum takeoff distance. From Figure 5.1-3, the wind increase causes a near linear increase in score. Researching the historical wind speeds at Patuxent River, Maryland during the April competition time period yielded valuable results. A six year record of average wind speed yielded an 8.8 mph daily average wind speed. Partial data for the average low wind yielded a wind of over 5 mph. Wind strongly affects the score due to the takeoff distance limitation. For each 5 mph increment in wind velocity, the score increases by approximately 3 points. At a wind speed of 15 mph, the aircraft is at the 55-pound weight limitation with a score of 36. From the wind analysis, it was decided to optimize for a 5 mph wind.

Cruise height sensitivity analysis indicated that score was a weak function of cruise height. From Figure 5.1-4, the maximum score was nearly constant for heights below 100 feet. Although unexpected, having the score uncoupled from the cruise height will be beneficial to the pilot. The pilot will be able to choose a comfortable altitude and not be concerned with flying low.

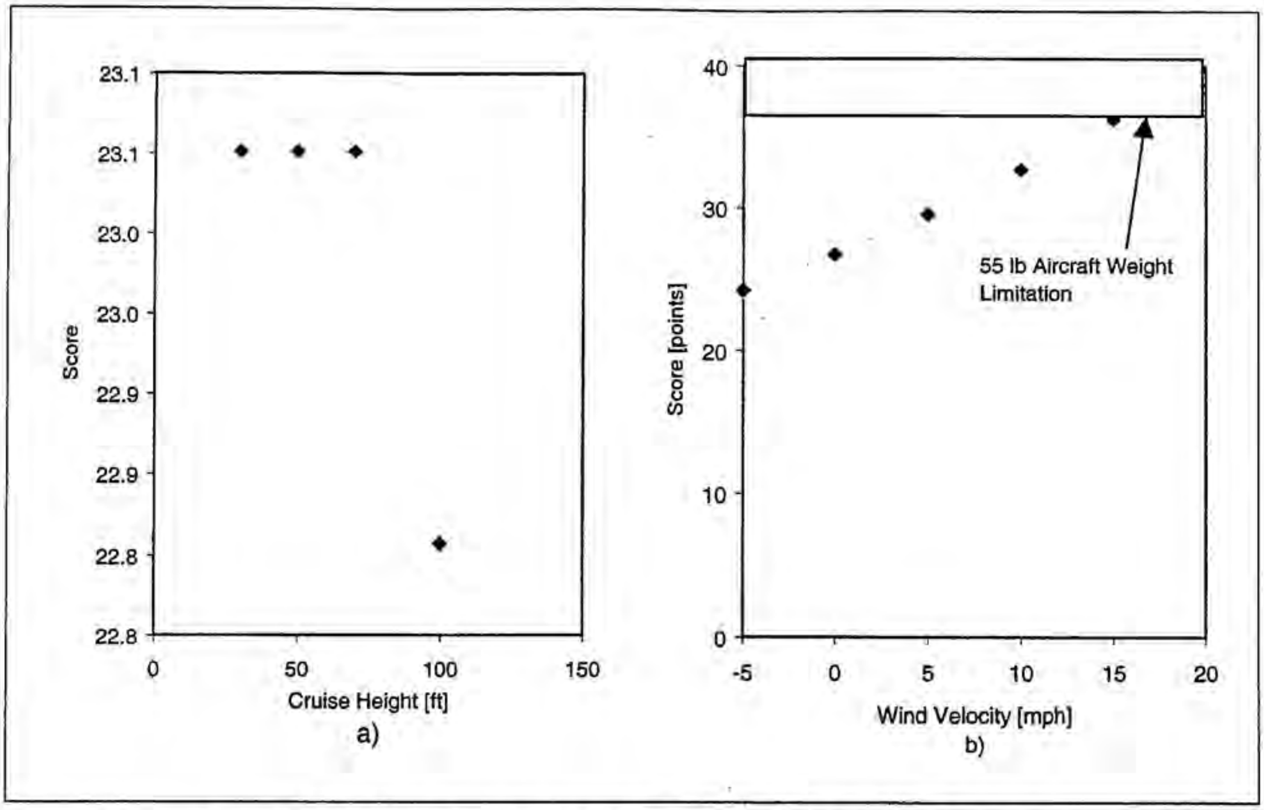
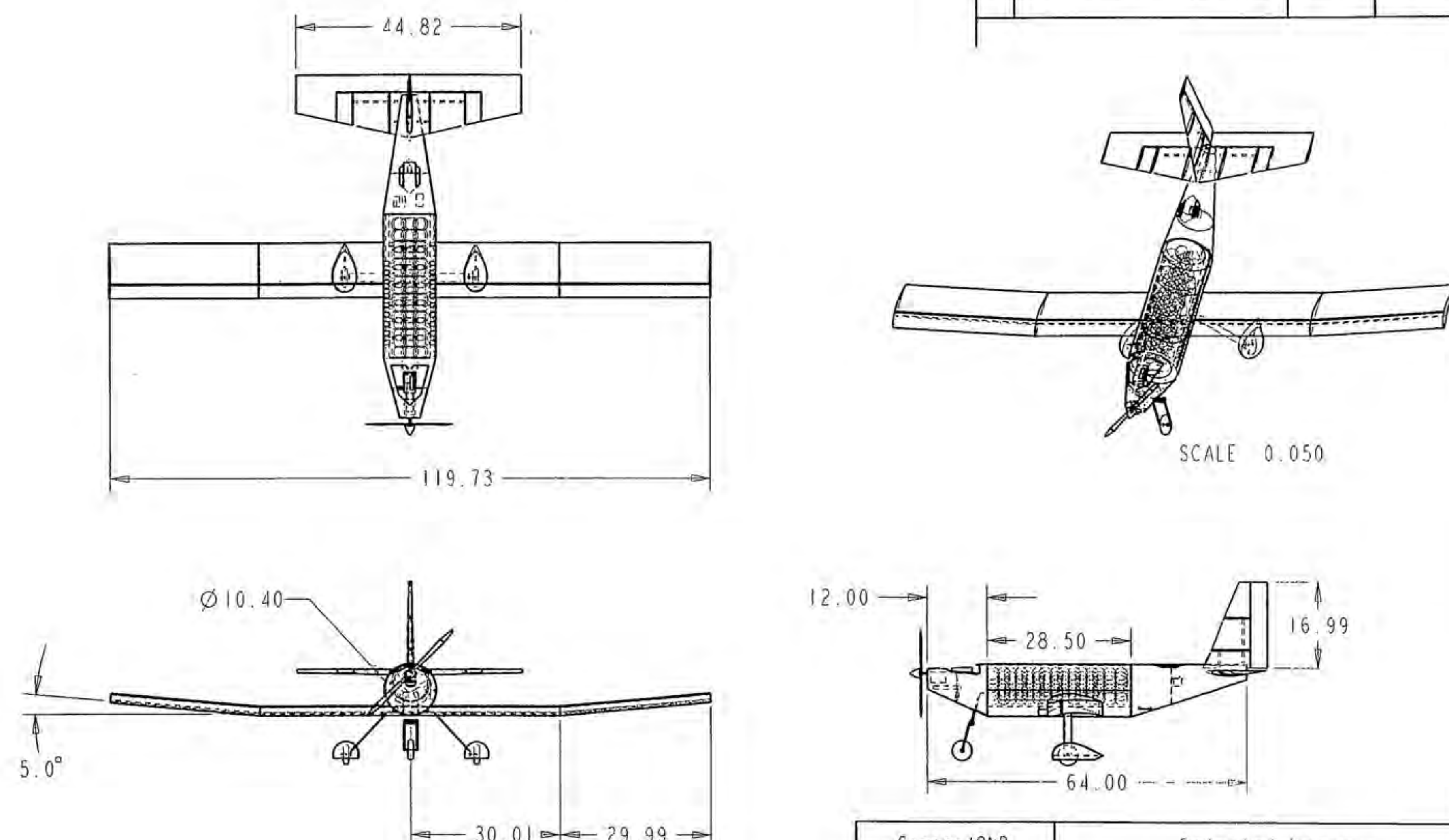
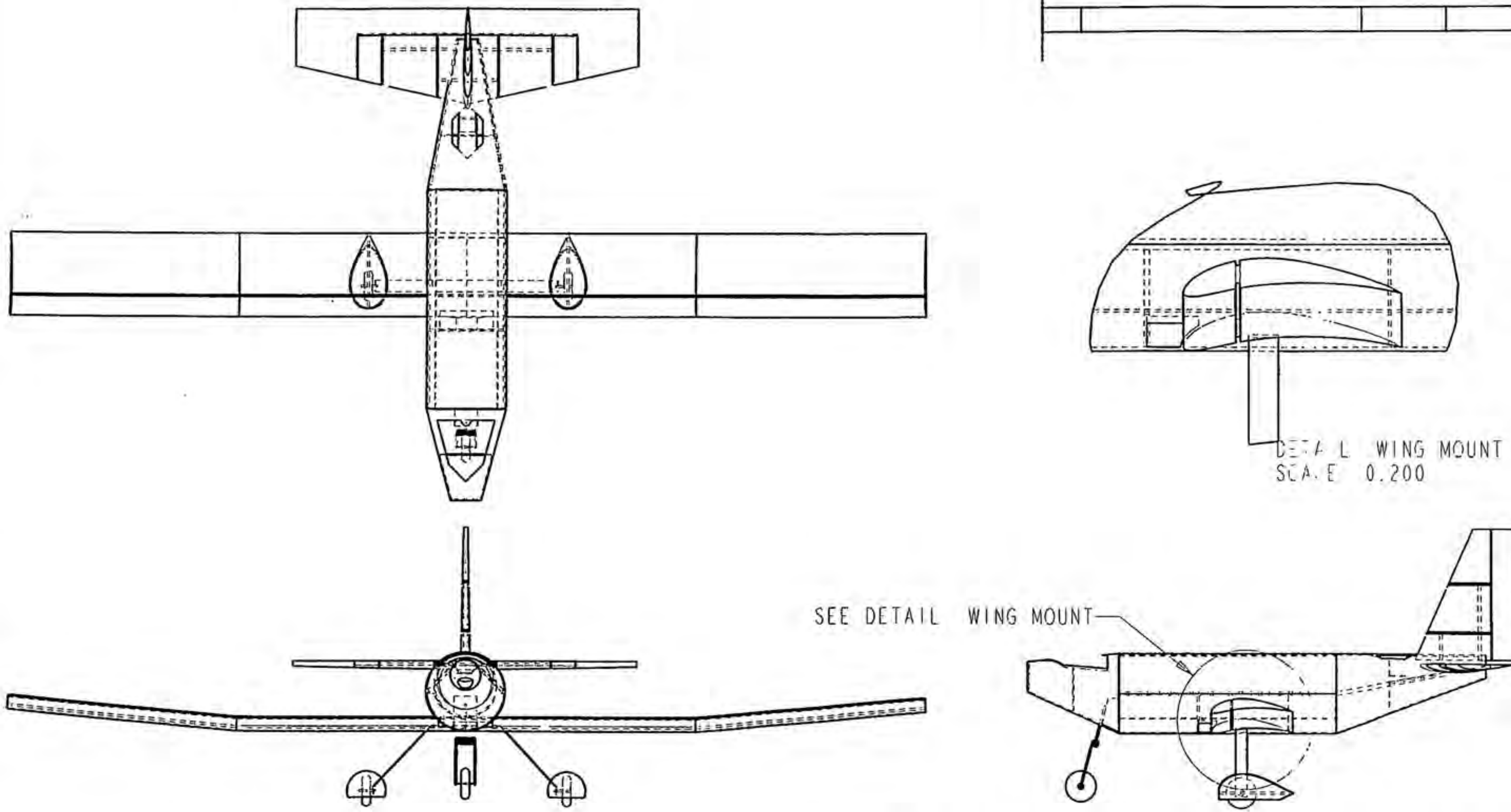


Figure 5.1-3 Score Sensitivity to (a) Cruise Height (b) Wind Velocity

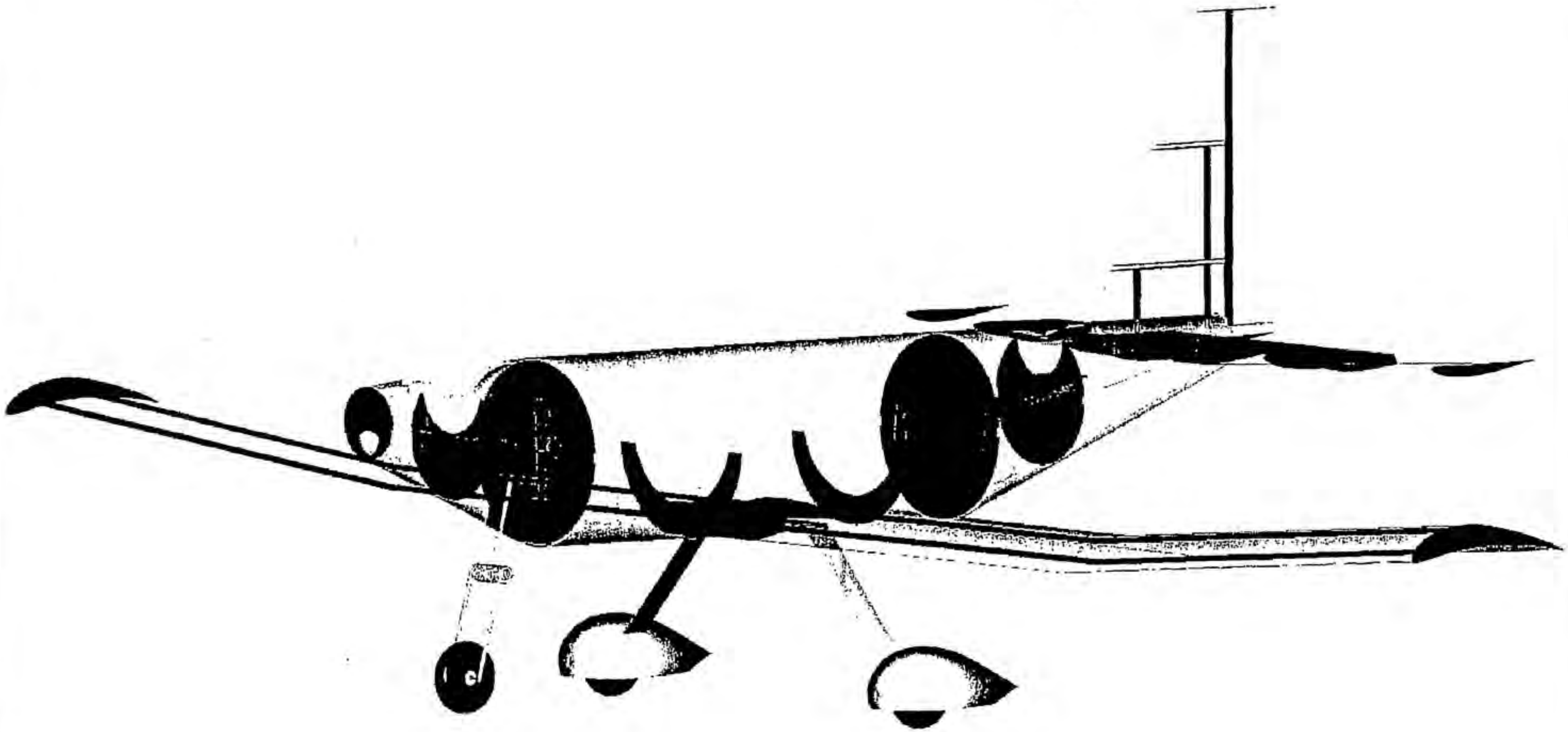
With the higher speeds predicted and required by the aircraft, landing and roll out distances need to be considered. An analysis of landing ground roll distance is disturbing. For this ground roll sensitivity analysis, the aircraft was loaded to give the highest score for the given wind. As discussed in the wind sensitivity analysis, an increase in wind also increases the carried payload weight. From Figure 5.1-4, a no-wind 20 lb steel payload sortie is estimated to take 650 feet to stop. A 10 mph wind with a 30 pound steel payload decreases the distance to 430 feet. Thus, an effective stopping method needs to be considered. To resolve this problem, an investigation into the effectiveness of brakes on ground roll distance was performed. The mission analysis program was modified to include braking during the ground roll. To test the brake sensitivity, a worst-case scenario of a zero-wind with the corresponding optimized steel payload weight was considered. Test results are given in Figure 5.1-4. With no brakes, the estimated roll out distance was 720 feet. With a braking force equal to only 10% of the aircraft weight, the ground roll distance was slashed to just over 270 feet. It appears that even small braking forces are effective in reducing ground roll distances. A brake system is beneficial.



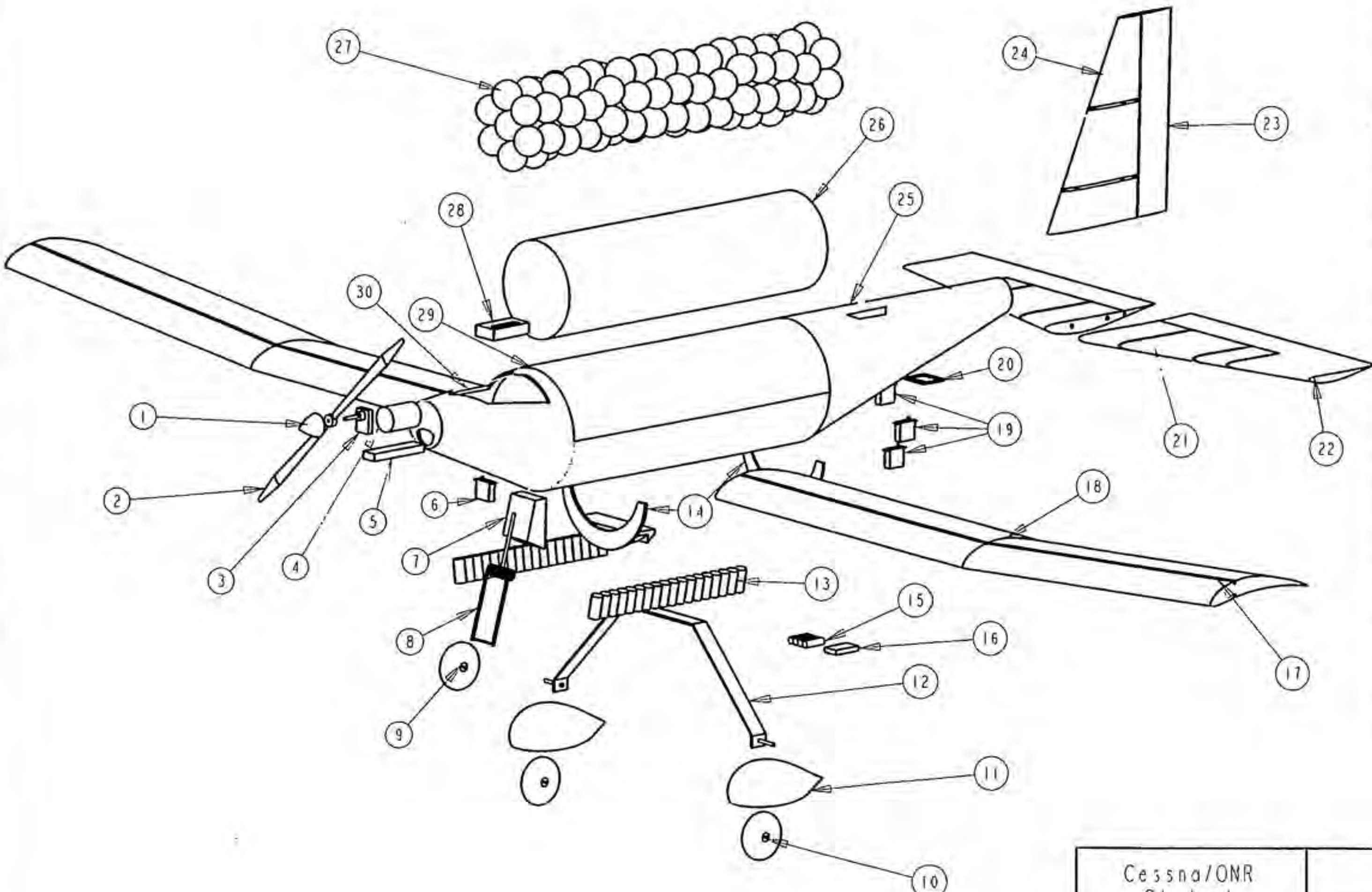
Cessna/ONR Student Design/Build/Fly Competition	Detailed Design	
	OKLAHOMA STATE UNIVERSITY	
2000/2001	13 Mar 2001	OSU Design Team



Cessna/ONR Student Design/Build/Fly Competition	Structural Assembly
	OKLAHOMA STATE UNIVERSITY
2000/2001	13 Mar 2001	OSU Orange Team

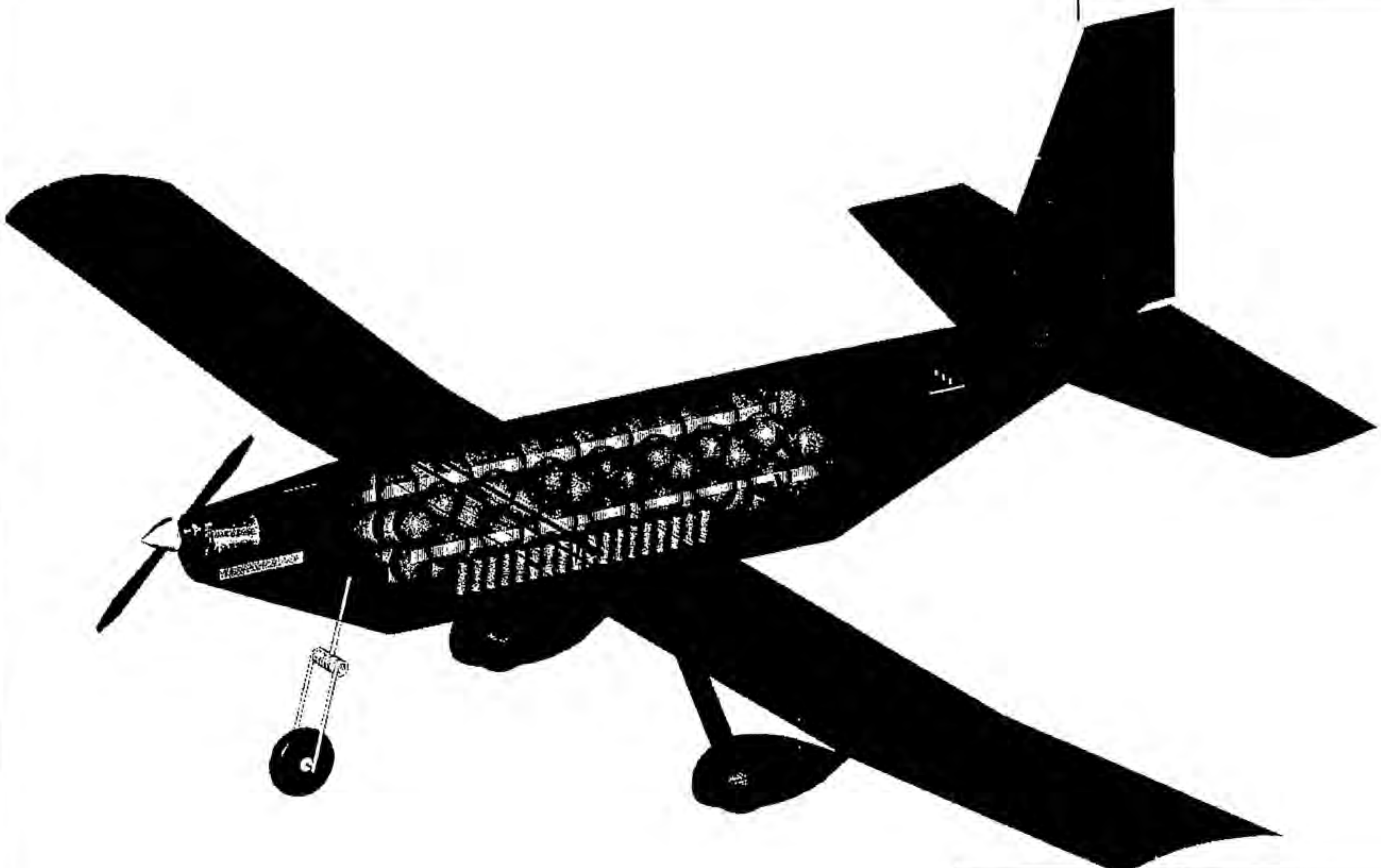


Cessna/ONR Student Design/Build/Fly Competition	Structural Isometric Rendering
	OKLAHOMA STATE UNIVERSITY
2000/2001	13 Mar 2001 · OSU Orange Team



No.	Item	Qty
1	Spinner	1
2	Propeller	1
3	Gearbox	1
4	Electric motor	1
5	Speed control	1
6	Nose gear servo	1
7	Nose gear mount	1
8	Nose gear	1
9	Nose wheel	1
10	Main wheel	2
11	Wheel pant	2
12	Main gear	1
13	Battery pack	2
14	Wing mount	1
15	Receiver Pack	1
16	Receiver	1
17	Wing spar	2
18	Polyhedral wing	2
19	Tail servos	3
20	Servo mount	1
21	Horiz. stabilizer	2
22	Elevator	2
23	Rudder	1
24	Vert. stabilizer	1
25	Rear hatch	1
26	Speed loader	1
27	Tennis balls	100
28	Wattmeter	1
29	Speedloader hatch	1
30	Forward hatch	1

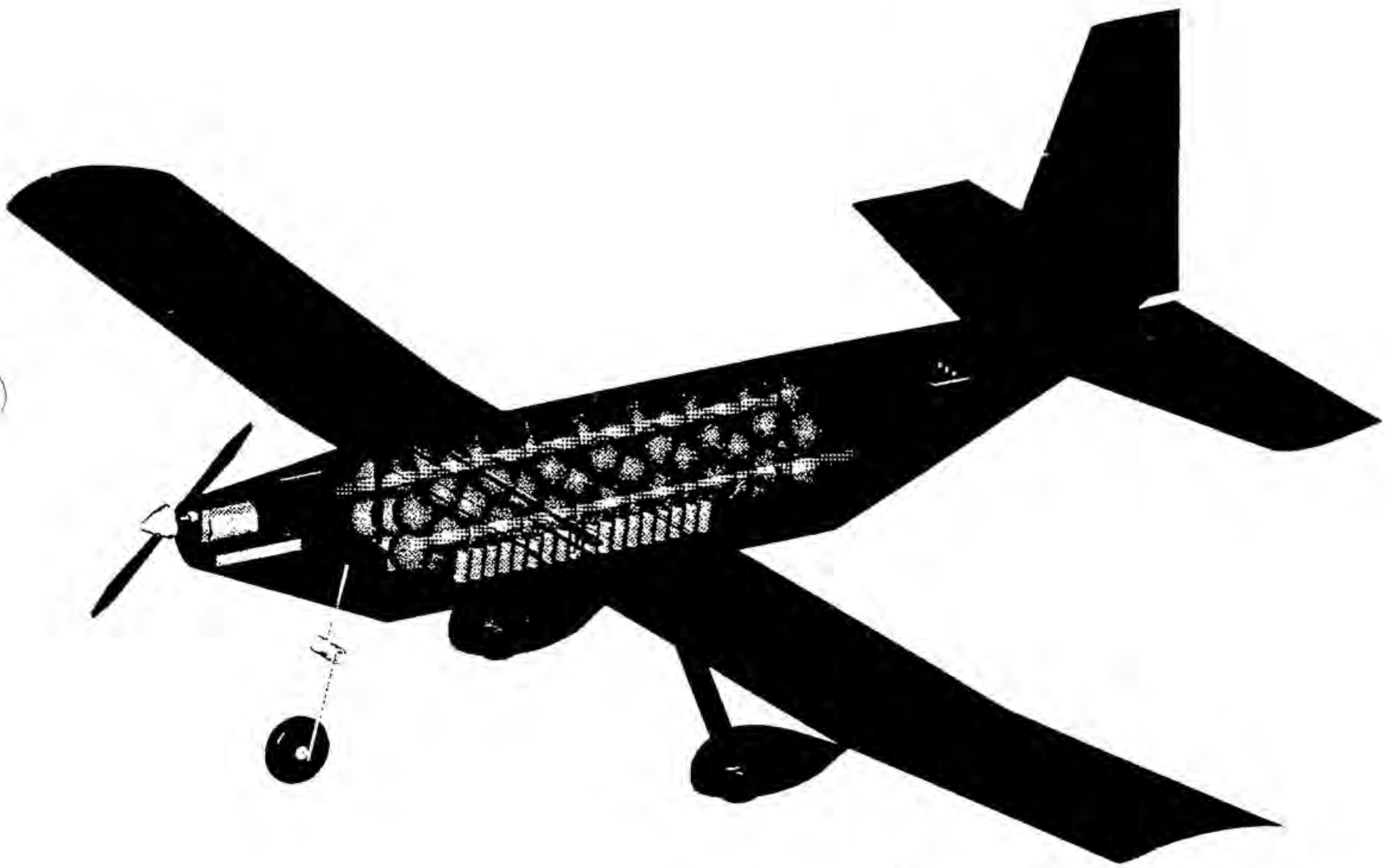
Cessna/ONR Student Design/Build/Fly Competition	Exploded View
	OKLAHOMA STATE UNIVERSITY
2000/2001	13 Mar 2001 — OSU Orange Team



Cessna/ONR Student Design/Build/Fly Competition	Isometric Rendering	
	OKLAHOMA STATE UNIVERSITY	
2000/2001	13 Mar 2001	OSU Orange Team

2000/01 AIAA Foundation
Cessna/ONR
Student Design/Build/Fly Competition

Addendum Phase Design Report



Oklahoma State University
orange team

April 10, 2001

ADDENDUM

8 Lessons Learned

8.1 Prototype Flight Testing Results

Once the prototype aircraft was assembled and deemed flight worthy, the flight-testing phase began. Pictures from the flight-testing phase are shown in Figure 8-1. The flight-testing phase compared the predicted and actual aircraft performance. Data obtained during flight-testing was analyzed to determine the aircraft's flight performance. The parameters assessed include stability and control, handling qualities, speed, lift capability, energy used per lap, endurance and were ultimately used to determine the final strategies.

Figure 8-1: Pictures of prototype flight-testing

Stability and control and handling qualities were verified by pilot opinion. As a result of the pilot's input, ailerons were incorporated into the final design in lieu of a rolling tail, which was included in the original design to reduce RAC of the wing. This change resulted in a reduction in the overall tail size.

After verification of the handling qualities as well as the stability and control of the aircraft, a speed run was conducted. The speed run was conducted at full power over a set course. The average up wind and down wind speed was then obtained. The resulting average speed was 70 feet per second. Since this was considerably lower than the predicted max speed of 102 feet per second, the calculations were reviewed and the aircraft inspected. Upon completion of this assessment, a difference in motor constants was discovered. Due to a delay in delivery of the contest motor, an old motor of the same size was used in the prototype. This motor, however, has different motor constants than that of the final motor ordered. These differences were overlooked prior to flight-testing. After discovering the discrepancies, adjustments were made to the calculations to verify the max speed for the test motor. Those calculations revealed a max speed of 71 feet per second.

Lift capability was the next parameter to be investigated. This evaluation was conducted by stepping up the steel weights from empty to maximum weight. Maximum weight successfully lifted was determined by two factors, take-off distance and climb rate. It was determined that climb rate was the limiting factor as to the amount of weight the aircraft can lift. As a result, the maximum gross take-off weight was determined to be 40 lbs instead of the 55 lbs originally predicted.

The energy used per lap was greater than predicted. For each tennis ball lap the actual energy used was 0.6 – 0.65 Ah compared to the predicted value of 0.44 Ah. There was less of a difference in the energy used per steel. This changed from 0.39 Ah predicted to 0.4 – 0.45 Ah actually used. These differences were a result of higher than predicted parasite drag on the aircraft.

Finally, with all the previously mentioned tests completed, multiple contest missions were executed. The results of these missions lead to the conclusion that a 3-3 mission was not possible due to the change in capacity limitation of the battery pack to be discussed in section 8.2.3. However, the information obtained lead to the development of two alternative strategies. The first strategy, 3 steel –2 tennis ball laps, involves light payload weight and low wind conditions. The second strategy, 2 steel – 2 tennis ball laps, evolved from the discovery of a sensitivity of the aircraft's performance to wind conditions. This strategy is utilized for wind speeds greater than 10 mph and was governed mainly by the aircraft's total time in flight.

The following chart summarizes the predicted versus actual flight-test data used for refinements on the competition aircraft.

Parameter	[illegible]	[illegible]
Max Speed	71 ft/s *	70 ft/s
Endurance	2400 mAh	2100 mAh
Energy used per tennis ball lap	0.44 Ah	0.6 – 0.65 Ah
Energy used per steel lap	0.39 Ah	0.4 – 0.45 Ah
Flight profile (low wind speed)	3 steel – 3 tennis ball laps	3 steel – 2 tennis ball laps
Flight profile (high wind speed)	3 steel – 3 tennis ball laps	2 steel – 2 tennis ball laps
Gross take-off weight	55 lbs	40 lbs

* Value for adjusted motor constants

8.2 Aircraft differences from the proposal design and justification

8.2.1 Aerodynamics Group

The sizing of the horizontal tail changed for several reasons. The major factor in this revision came from the necessity of ailerons to adequately control the aircraft under various loading and environmental conditions. Since ailerons are used for roll control, this requirement for the rolling tail sizing became unnecessary. The second factor in the resizing came from pilot response. The pilot determined the pitch stability and control to be overly adequate. Overall aircraft weight reduction comprised the final consideration.

Through analyzing the lateral stability and control equations, reducing the horizontal tail area by 6.5% did not significantly change the horizontal lift coefficient to require a new horizontal airfoil selection. This change brings the aircraft stability closer to 15% static margin and reduces the overall aircraft weight and drag coefficient. Parameters such as aspect ratio, taper ratio, percentage of the chord as elevator, and aerodynamic balance assumptions remained constant from the prototype.

8.2.2 Structures Group

There are several structural modifications to the design of the aircraft that have been integrated into the final airplane to be flown at the contest. These differences were introduced to improve the weight of the aircraft, the ability of the aircraft to perform the contest mission, and the crashworthiness of the aircraft.

The original design of the fuselage called for longerons that were eliminated due to the extremely strong fuselage skin that was obtained by the foam/carbon fiber sandwich. The extra bending moment that would be withstood by longerons was not required and would only have added weight to the aircraft. The fuselage was not over designed in all areas, however. The bulkheads in the prototype comprising the wing carry through structure delaminated after a few flights and were doubled before catastrophic failure occurred.

The wing was also modified from the original design. The spar, which was to be made of ¼" X ¼" spruce material, was increased to 3/8" material to yield an acceptable deflection under the maximum gross weight the aircraft will weigh in with at the contest.

The wing attachment to the fuselage was also modified from a combination of bolting and pinning to bolting only with nylon shear bolts. This modification was integrated to provide greater crashworthiness. The shearing bolts will not damage the leading edge of the wing or carry through structure in the fuselage in the event of abnormal loading conditions. In an effort to reduce interference drag, fillets were added at the fuselage to wing interface as well as the fuselage to tail surfaces interface.

The landing gear design was also modified. An aluminum two stroke nose gear called out in the design would have been difficult to manufacture and was therefore replaced with a commercially produced steel spring type gear that was tested on the prototype. This gear performed well and weighed very close to previous teams manufactured nose gear.

Finally, the speed loader configuration has also been modified. The original design of the speed loader specified that the fuselage hatch be mounted to the speed loader, allowing only one movement be required to replace the speed loader between sorties. This design did not take into consideration the weight and awkward nature of a payload carrying tube and the hatch will now be a separately removed from the aircraft allowing better handles be design to manipulate the speed loader into and out of the fuselage.

8.2.3 Propulsion Group

The propulsion system for the final design stage did not undergo any major changes other than the propeller sizing. After running several dynamic tests on the proposed system it was determined that a different size of propeller better served the needs of the design. The proposed propeller was a 22/12 and the final propeller was a 22/20, however there is still the option of changing out propellers to get the optimum performance at different wind speeds. The propellers that are useful for the projected wind and plane speeds are the 22/12, 22/20, 24/14, and 24/24 propellers. Having these propellers on hand will provide a wide range of options at the contest so that maximum performance of the airplane will be achieved. The only other change in the propulsion system was a minor adjustment of the gearing. Rather that using a linear gear system, which sometimes backs into the motor housing, a helical gear system is implemented. The rest of the components remain unchanged in the final design. However, the manufacturer's specifications on the battery packs stated an available capacity of 2400 mAh. After extensive testing both in the field and in the wind tunnel, it was determined that the battery packs would only safely supply 2100 mAh of capacity. This new capacity has since been used as the maximum capacity. This change in capacity means that a 3 steel – 3 tennis ball flight profile is no longer possible due to battery limitations. So, a change of the strategy originally developed, as discussed earlier in the prototype flight-testing results, was necessary.

8.3 Time and cost required to implement change/improvement realized

The intention of the team from the beginning was to build two planes, a prototype and a competition aircraft. In this way, the learning was done on the prototype and the changes, which would have been costly in both monetary means and time, were applied to the final aircraft from the learned lessons of the prototype. This allowed for all changes to be planned for in advance of the final construction and therefore not increase the cost of the project to realize the improvements.

The improvements realized from these changes between the prototype and final aircraft include a lighter structure, a stronger structure, and a structure that will perform the mission task better as indicated by each of the justifications listed for the improvements that were made. The most significant change between the prototype and the final aircraft is the tail sizing.

The time and cost of the change to the horizontal tail is minimal for several reasons, but very beneficial in the RAC calculations due to the resulting reduction in empty weight for the aircraft. For the prototype, a MathCAD program set up the lateral stability and control equations and coupled the appropriate graphs for the prototype sizing. Initial sizing came from a dependency upon a known platform area; resizing included modifying the platform area until the output and graphs produced an appropriate solution. The MathCAD program calculated all dimensions for the stabilizer, elevator, and aerodynamic balance.

Little production time was required to revise the construction drawings; the use of Professional Engineer allowed these dimension changes to be made easily. Foam templates were the only prototype hardware that could not be reused for the final. Structure ribs, spars, and foam cores had to be produced for the final aircraft regardless of design changes; the altered dimensions of these items did not add production time. Assembly of the tail and subcomponents occurred in the same manner as the prototype. The same materials were used in the final assembly of the tail and its subcomponents resulting in no monetary cost difference from the original design.

8.4 Areas for improvement in next design and manufacturing process

Although many improvements were integrated into the aircraft that will be flown at the contest, other refinements to the design and manufacturing process would be integrated into a future aircraft. Many items in the prototype were replaced with items made of lighter materials. This improved the rated aircraft cost by reducing the empty weight of the aircraft. There are still several components, however, that could be replaced in the final aircraft to further reduce the weight. These changes were not made in the final aircraft because of the time and cost of implementing them.

The first item is the blue foam used extensively in the structure of the aircraft. This material is excellent for forming aircraft components such as wing cores, but it is heavy. For a future aircraft, other foam types would be investigated to find one that is formable, light, and will stand up to the vacuum pressure needed to cure the carbon fiber/foam composite structures. Another item that is perhaps not ideal is the carbon fiber arrows that were used as spars in the tail section. Although these items allowed for easy tail section mounting to the fuselage, the circular shape is not ideal for a structural member designed to withstand bending forces. A future aircraft would integrate a lighter spar, possibly a lightweight carbon fiber beam, to replace these arrows.

The manufacturing process utilized in the final aircraft was not modified greatly from the prototype. This process was refined before manufacturing of the prototype by discussion with previous members of the team. There are, however, some improvements that will be passed on to next year's team. One improvement in the foam cutting technique that would be integrated into a future aircraft is to cut the airfoil cores over a shorter length of foam. This year's airfoils have a slight sag in the leading edge due to the wire lagging as it passes through the center of the foam. Another improvement is to use honeycomb in the fuselage skin as opposed to foam. The thin cross section of foam was difficult to cut over the conical sections of the fuselage due to the slower section of the hot wire melting additional material.

Flight-testing would remain a very high priority in the next design and manufacturing process. Actual results from the flights would be compared to the dynamic test results and the analytical data to how the plane performs in the air. This would require faster design and manufacturing processes, but cost will not be increased. Another area for improvement for the next design involves the battery charging methods. By changing the way that the batteries are initially charged the overall power output will be greater.

9 Aircraft Cost

The following pages are documentation of the rated aircraft cost for the OSU Orange team plane. First, Table 9-1 shows a complete account of the rated aircraft cost. Then, in Figure 9-1 there is a visual breakdown of the percentage contribution of each category to the total rated aircraft cost and another breakdown of each structure's contribution to the total number of manufacturing man hours.

Manufacturers Empty Weight			
Actual Weight (lbs)	*MEW Multiplier ($/lb)*		***RAC***
14 13.4	$100.00		**$1,400.00**
Rated Engine Power			
Number of engines	*Fuse Rating*	*Number of cells*	*Voltage per cell*
1	20	37	1.2
Rated Engine Power (watts)		888	
Rated Engine Power Multiplier ($/watt)		1	
Total Rated Engine Power RAC Contribution		**$888.00**	
Manufacturing Cost			
Wing			
Unit	*Quantity*	*Multiplier (hr/unit)*	*WBS Contribution*
Wing	1	15	15
Projected Area	8	4	32
Strut or brace	0	2	0
Control Surface	2	3	6
	Total Wing WBS Contribution (hrs)		53
Fuselage			
Unit	*Quantity*	*Multiplier (hr/unit)*	*WBS Contribution*
Body	1	5	5
Length (ft)	5.333	4	21.332
	Total Fuselage WBS Contribution (hrs)		26.332
Empennage			
Unit	*Quantity*	*Multiplier (hr/unit)*	*WBS Contribution*
Basic	1	5	5
Vertical Surface	1	5	5
Horizontal Surface	1	10	10
	Total Empennage WBS Contribution (hrs)		20
Flight Systems			
Unit	*Quantity*	*Multiplier (hr/unit)*	*WBS Contribution*
Basic	1	5	5
Servo or controller	5	2	10
	Total Flight Systems WBS Contribution (hrs)		15
Propulsion Systems			
Unit	*Quantity*	*Multiplier (hr/unit)*	*WBS Contribution*
Engine	1	5	5
Propeller or fan	1	5	5
Total Propulsion Systems WBS Contribution (hrs)			10
Total Manufacturing Man Hours			124.332
Manufacturing Cost Multiplier ($/hr)			20
Total Manufacturing Cost RAC Contribution			**$2,486.64**
[illegible]			

Table 9-1: Detailed breakdown of RAC

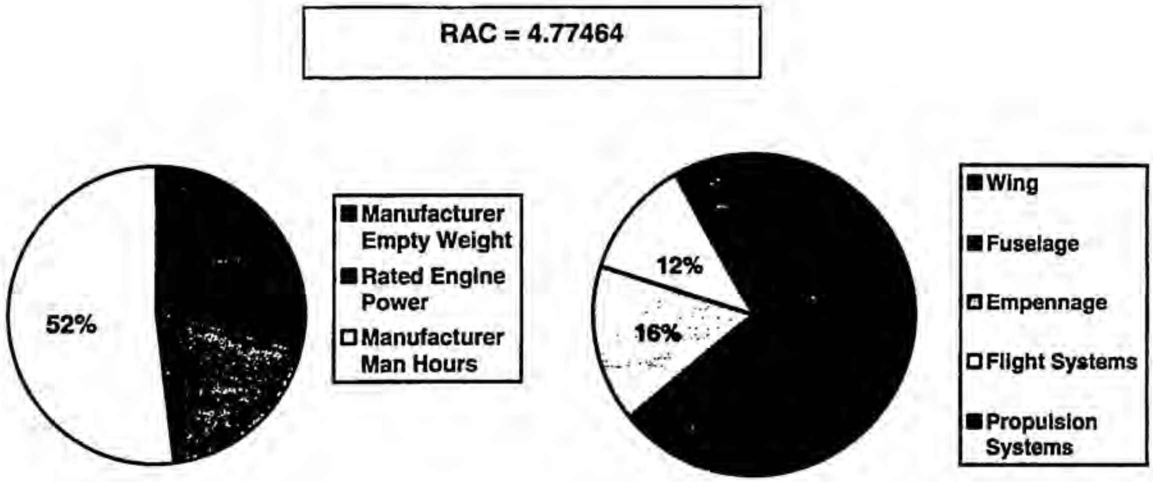


Figure 9-1: Visual Breakdown of Rated Aircraft Cost and Manufacturing Man Hours

Stealin' Balls

Design Report – Proposal Phase

2000/2001 Cessna/ONR/AIAA Student Design/Build/Fly Competition

Aeronautical Engineering Department

California Polytechnic State University, San Luis Obispo

Authors: John Asplund, James Bach, Jeremy Rocha, Francesco Gianinni

March 13, 2001

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1.0 Executive Summary

A team of students at Cal Poly San Luis Obispo has undertaken the task to design and build an airplane to participate in the Cessna/ONR Student Design/Build/Fly competition. The competition involves designing and building an electric powered, radio controlled airplane capable of flying tennis balls and steel around a designated course as many times as possible in a ten-minute time period. The payloads must be alternated between laps where a "Heavy Payload" lap is done with the steel and a "Light Payload" lap is done with the tennis balls. The maximum number of tennis balls that can be carried at once is 100. The score per lap for carrying tennis balls is 20 points where each tennis ball yields 1/5 of a point. The steel laps are scored one point per pound of steel carried. A sortie is defined as one complete ten-minute time period. The final score is a result of the average of three sorties. The final score is calculated by multiplying the average from the best three laps times the report score, divided by the Rated Aircraft Cost (RAC) of the configuration. The RAC is calculated based on the geometry of the aircraft, the empty weight, and the motor power required to fly the aircraft.

A variety of constraints were placed on the design team by the contest officiators. Each of the design constraints received consideration during the entire design process. The major design constraints are the time limitation of 10 minutes, takeoff distance of 200 feet, max wingspan of 10 feet, 5-pounds of batteries, and internally housed payload. At the beginning of the conceptual design, the design team investigated several possible airplane configurations. The team rated each airplane configuration based on a number of criteria referred to as the figures of merit (FOM). The initial FOMs for the conceptual design phase were used to screen competing concepts based on the contest design constraints and limitations within the design team. After possible concepts had been narrowed down to four airplane configurations, a detailed score analysis for each configuration was done using a Visual Basic program. These designs were a conventional low wing monoplane, a biplane, a tandem wing, and a flying wing. The program started with a takeoff model to size the wing and propulsion system for a given payload and gross takeoff weight. The best cruise speed, time required to complete the course, and RAC were calculated to find the scoring potential of each design. The sizing program evolved and grew more refined throughout each phase of the design process. The competing designs were also quantitatively evaluated to determine the most viable concept. The design that maintained a high scoring potential with inherently good handling qualities was the low wing monoplane. The low wing offers the accessibility to the payload while providing direct load paths to the primary structure for both in-flight and landing loads. The design is predictable and simple, with opportunity for innovation with the aircraft structure and landing gear. This configuration was selected for the final design.

During the preliminary design phase, the configuration was optimized to achieve the highest score using more detailed analysis. Because aircraft design is highly assumption driven, it relies heavily on the experience of the senior team members. The background of the team, and the enthusiasm for the project has contributed to the design and fabrication of a winning design. California Polytechnic State University, San Luis Obispo is proud to present Stealin' Balls, their 2000/2001 entry for the Cessna/ONR Student Design/Build/Fly Competition.

2.0 Management Summary

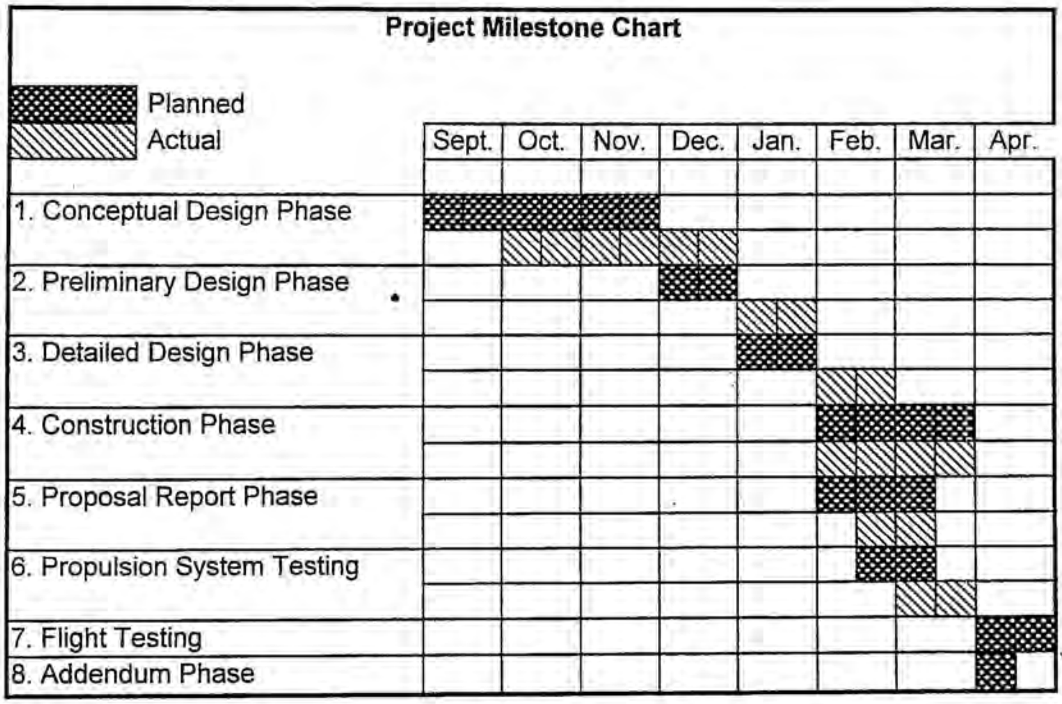
This years Cessna/ONR Student Design/Build/Fly (DBF) team at California Polytechnic State University (Cal Poly) is comprised of 20 students. The break down of the team is as follows:

- 2 graduate students
- 5 seniors
- 3 juniors
- 4 sophomores
- 4 freshmen

All but one of the students are Aeronautical engineering students. The other student is a mechanical engineering student. Initially, during the preliminary design phase, the group of students was divided up into several different design tasks. Younger students were paired with senior students to help teach them the design process. One group worked on the drag prediction and integrated it with the sizing routine in Excel. A second group worked on the performance including takeoff, landing and time prediction. The third group was responsible for propulsion and stability and control.

The two graduate students on the team, James Bach and John Asplund, have been involved in the DBF competition before. These two students orchestrated the design and construction of this year's airplane. The group leader, Jeremy Rocha, kept the team on schedule and motivated throughout the duration of the project. Jay Garcia, one of the junior team members, was responsible for fund raising and soliciting companies for materials. The less experienced members of the group participated in the design process while contributing to the report. The fabrication experience of the graduate students was used to train the underclassmen how to build using modern composite fabrication techniques. Passing the skills and knowledge down to the rest of the group is essential for the success of the future Cal Poly Design/Build/Fly teams. The milestones for the project are illustrated in Figure 2.1.

Figure 2.1 Project Milestones



3.0 Conceptual Design

3.1 Introduction

To determine the most viable configuration concept for the mission, several parameters were considered for comparison. Although there are many concepts that could complete the mission, the following four were chosen for comparison:

- Flying wing
- Monoplane
- Biplane
- Tandem wing

These concepts were evaluated based on the specific requirements of the mission to establish the best design.

3.2 Design Parameters

The design parameters used to screen competing designs include: payload capability, cost effectiveness, scoring potential, handling qualities, time, and take-off distance. The most useful analytical tool used throughout the conceptual design was a program written in Visual Basic simulating the entire flight.

To better understand the nature of the mission, a simple model that predicted the flight time required to complete the mission was written. With the ten-minute time limit, a first cut analysis showed that flying faster did not necessarily increase the total number of laps completed. Figure 3.1 illustrates the trend of cruise speed versus total time. Using conservative values for landing and reload, it was found that completing six laps was a reasonable goal considering the ten minute time limit. In addition, the total energy required to complete the mission increases as cruise speed is increased. Therefore it does not make sense to fly faster unless the aircraft is clearly capable of completing another lap. With six total laps as the performance goal, sizing of the competing configurations was performed to evaluate each design quantitatively.

The relaxed takeoff distance requirement does not drive the design as the much as the other parameters. With the increased span limitation and a fixed amount of batteries, the takeoff requirement is not a parameter the design is exclusively optimized for like the previous years of competition.

The payload capability of each design directly drives the size and weight of the airplane. The rules state that payload must be carried internally and not within the wing. For this reason, a flying wing must be equipped with a large pod to enclose the payload. When considering the pod required to carry the payload, the benefits of the low RAC of the flying wing are overshadowed. To carry both the steel and the tennis balls, the aircraft must have a large cargo bay that is easily accessible. The tandem wing, monoplane and biplane all have similar internal volumes to contain the payload.

The cost of each competing design is an important factor when the total score is considered. Since the score is divided by the cost, the most cost effective solution could provide the highest scoring aircraft. A tandem wing concept is a very inexpensive aircraft because essentially half of the wing area is considered a tail and charged accordingly. A concept with a shorter fuselage or a lower empty weight could also provide an inexpensive aircraft. These factors were all taken into consideration when comparing the concepts.

The scoring potential of each design was carefully investigated using fundamental design parameters specific for the mission. These include payload quantity, climb rate, and takeoff performance. The scoring potential for each concept was evaluated by keeping these parameters constant and running each concept through a mission simulation. The results are illustrated in Figure 3.2. The tandem wing is capable of scoring slightly higher, with the conventional configuration scoring just half of one point less. The flying wing was close as well, and the biplane had the lowest scoring potential. Just considering the scoring potential alone and building the highest scoring airplane is not the best method for winning the competition. Handling qualities and configuration risk are important considering a competition of this nature. Based on past experience, good handling configurations can increase the chances for success immensely. Therefore, concepts that have potential handling issues maintain less merit than the conventional configurations. The tandem wing is a high-risk concept because the flying characteristics are not as well known as conventional designs.

Initially, the optimal payload weight carried by the design was not well understood. However, after comparing the 50, 75, and 100 tennis ball configurations, the score was clearly maximized by the 100 ball design. In addition, because the aircrafts size was optimized to carry a specific payload weight, the optimal steel payload weight was equal to the tennis ball weight. All configurations were sized accordingly.

3.3 Figures of Merit

To find the optimal design it is important to take into consideration all of the elements the competition presents. The factors considered when determining the best concept for the mission are listed below:

- Empty weight
- Score
- Handling

- Risk
- Energy consumption

These parameters are known as Figures of Merit (FOM). The competing concepts are put into a matrix to weigh the advantages and disadvantages of each. The rating for each FOM goes from A through F with A representing the highest rating and F representing the lowest rating. The FOMs are listed with the most important at the top descending to the least important. Refer to Table 3.1 for the concept evaluation.

3.4 Conclusions

There are many different configurations that can adequately perform the mission. However, the four concepts evaluated showed that the low wing monoplane was the best compromise when considering the entire mission. The monoplane is very predictable to build and fly, and can achieve a very high score. The energy consumption of a monoplane is higher than the flying wing but overall the design is optimal for the entire mission. In addition, the performance prediction for the aircraft is greatly simplified by using a standard configuration. The flying characteristics are very predictable for the monoplane so the risk involved is very low.

Figure 3.1 Lap Potential

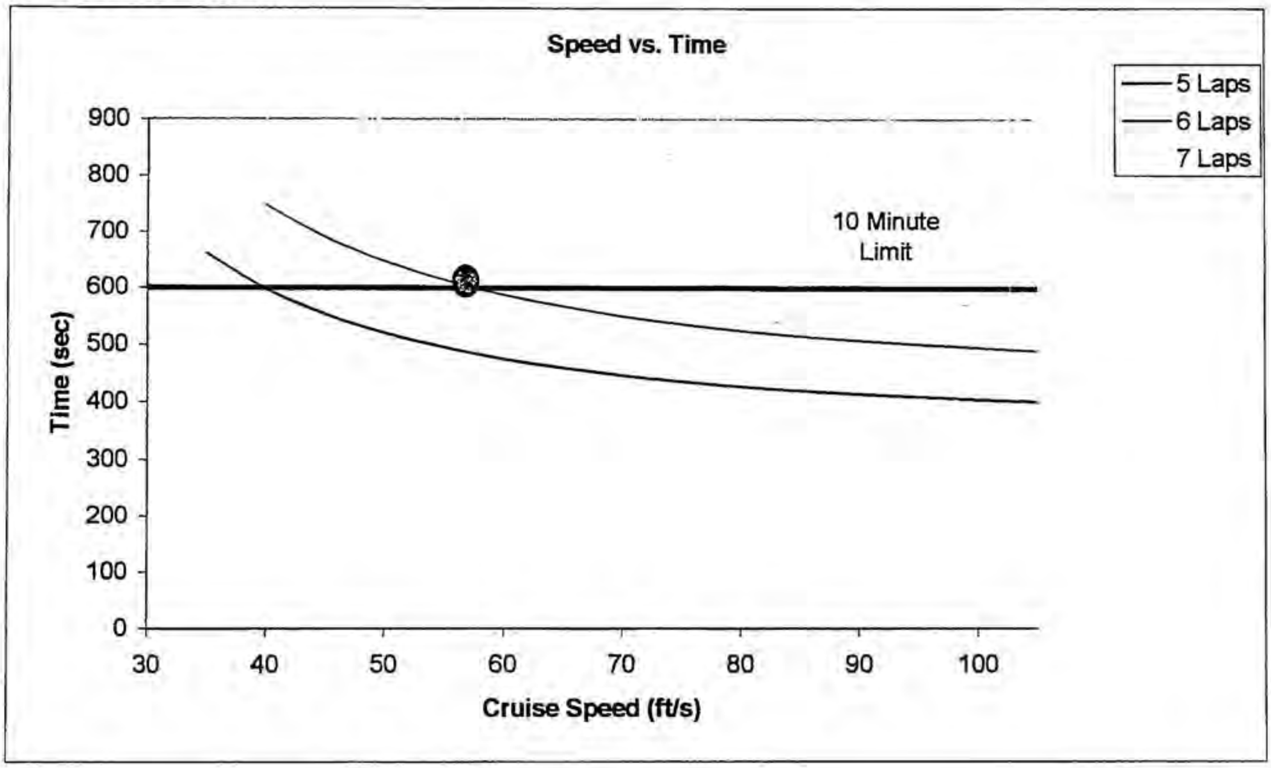


Figure 3.2 Concept Trade Study

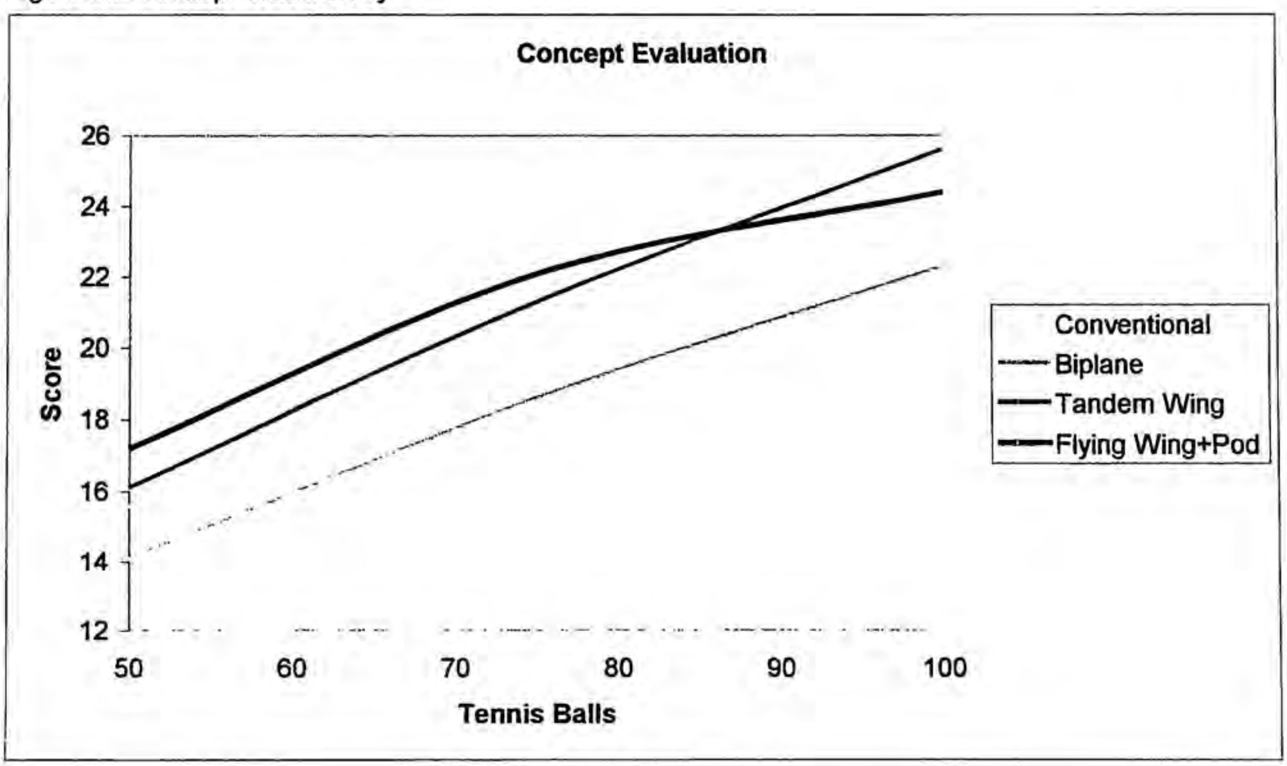


Table 3.1 Figures Of Merit

FOM/Concept	Tandem Wing	Biplane	Monoplane	Flying Wing
Empty Weight	C	D	B	A
Score	A	D	B	C
Handling	C	A	A	B
Risk	D	B	A	D
Energy	C	C	B	A

4.0 Preliminary Design

4.1 Figures of Merit

All preliminary design work was evaluated based of the following figures of merit:

- Impact on Maximum Score
- Handling / Easy to Fly
- Availability of Components/ Materials

Continually referring to these figures allowed each team member to make decisions that were best for the entire system. By understanding the sensitivity of each design parameter to the figures of merit, the team could focus their efforts optimizing the parameters that directly drove the design.

4.2 Design Parameters Investigated

The following areas of the design were investigated in detail because they had great impact on both the rated aircraft cost and the aircraft's ability to meet performance requirements:

- Mission Sizing
- Battery Selection
- Propulsion System Selection
- Wing Design

4.2.1 Mission Sizing

During preliminary design, it was determined that an aircraft sized for carrying the maximum 100 tennis balls with equal weight of steel would score the highest. This aircraft would be capable of completing six sorties within the ten minute mission time. It was also found that a conventional mono-wing configuration would be very cost competitive while minimizing the risk associated with exotic configurations. After this was determined, it was necessary to study the effect of wing aspect ratio, wing loading, and maximum lift coefficient on various aspects of performance including final score. This analysis was accomplished by tying together all the various tools for analyzing the design including models for propulsion system, aerodynamics, weight estimation, takeoff performance, cost, and tail sizing into a mission simulation program. The simulation flew the candidate aircraft though the mission to determine energy usage, time required and also checked key performance requirements. Different design parameters could be varied allowing their impact on the design to be understood.

Figure 4.1 shows the results of varying wing loading and aspect ratio on energy usage and final score for a 100 tennis ball, six sortie, monoplane configuration. Each configuration was sized with enough batteries to complete six sorties. A 10ft/sec minimum climb rate was imposed at gross weight on all the designs based on input from the pilot. If a design could not meet this requirement, a larger motor and or larger battery pack was substituted and the design was re-sized. Depending on the design, propulsion makes up 15-40% of the aircraft cost. Because propulsion system cost was directly rated to maximum power, designs that had more the 15ft/sec climb rate at gross weight generally were not very competitive due to high cost. These designs were reconfigured to allow acceptable climb rate with a more competitive cost. At high wing loadings, sizing for a fixed climb rate caused the aircraft to be incapable of meeting the 200ft takeoff requirement.

The results indicate that below aspect ratios of 8, the aircraft required more than the maximum battery weight of five pounds to complete 6 laps. Even though the empty weight of these aircraft was comparatively low, the poor cruise and turn efficiency make the designs incapable of completing the mission. Higher aspect ratio wings enable the aircraft to complete the mission with less than five pounds of Nickel Cadmium batteries (NiCds) at the expense of higher empty weight due to higher structural weight. Up to the span limit of ten feet, the trade between aspect ratio and final score was very soft. Additionally, input from the pilot suggested that he would prefer to fly a lower aspect ratio aircraft because of handling and it's lower sensitivity to gusts.

The design showed fairly high sensitivity to wing loading. For fixed climb rate, lowering wing loading caused the design to move further away from the zero wind takeoff constraint but also reduced maximum score. Wing loading did not heavily drive the mission energy usage for the aircraft. The final tradeoff for wing loading was between pushing the takeoff distance and pilot familiarity with "hot" airplanes versus achieving the highest possible score.

The final design configuration was chosen after carefully considering these factors with a wing aspect ratio of ten, and a maximum wing loading of 57oz/ft^2.

4.2.2 Battery Selection

The propulsion system must be powered by a NiCd battery pack no more than 80 oz in weight as described in the contest rules. Many different types and sizes of NiCd batteries were available. Essentially, by limiting the weight of the battery, the total energy available for propulsion is limited. Depending on the characteristics and type of NiCd cell used, it also limits the maximum power. Research on the characteristics of NiCd cells was conducted to choose the best battery to fulfill both the aircraft's energy and maximum power requirements. The voltage and maximum current capabilities of the available motors were also considered during the search.

Two styles of NiCd cells were investigated in detail, the high capacity Sanyo KR series, and the NiCd cells typically used for electric powered R/C models, the fast charge Sanyo R series. Table 4.1 shows the characteristics of each cell type.

The fast charge cells are designed to handle high maximum currents and therefore are well suited for typical electric aircraft motors. The specific power density of these cells is on the order of 25 Watts/oz. These cells have a specific energy density of approximately 1.2 Watt-Hours/oz. The Sanyo RC-2400 sub-C cell has the highest energy density and excellent power density compared to all of the fast charge cells.

The high capacity KR cells were considered because of their higher energy density (1.6 Watt-hours/oz). The compromises in the cell design which gave it high capacity also hurts the cell's internal resistance and their maximum discharge rate. Because of this, the power density is only half that of the fast charge cells, approximately 11 Watts/oz. Unfortunately, most of the design concepts required over 1000 Watts peak power to meet the minimum climb requirement, rendering the high capacity cells unsuitable. Additionally, NiCd cells delivering power at their maximum rate are not operating efficiently which significantly reduces their delivered energy. Most electric motors available for powering a 25 lb aircraft require 20-50 Volts input instead of the near 100 Volts that a series pack of high capacity cells would be. A combination parallel-series pack was considered to reduce the voltage and increase the current capabilities, but the difficulties in charging and maintaining balance between the cells eliminated this idea.

The final NiCd cells chosen for the design were the Sanyo RC-2400 cells because of their availability, performance, and the team member's familiarity with the cell.

4.2.3 Propulsion System

After initial sizing, it was determined that the aircraft needed a propulsion system able to run on approximately 28-34 NiCd cells, and be capable of handling 900-1200 Watts for takeoff and climb performance. Both single motor and twin motor configurations were considered. It was determined universally that for a given power requirement, the single motor solutions outperformed the twins in final score. This was caused by the additional cost of having more than one motor and speed controller, because for a given power level, two smaller motors, gearboxes and propellers weigh more than a single large motor. In addition, larger motors generally have higher efficiency.

To evaluate the performance of the motors available for the aircraft, a Visual Basic subroutine modeling electric motors was written. This model uses motor parameters including Kv (RPM/V), idle current, armature resistance, RPM and thermal limits to calculate output power and efficiency for DC motors. This program was used in conjunction with a propeller model and the mission simulation to compare the various motors offered by Astro Flight and Graupner.

Performance, reliability and availability were considered in selecting the motor. In the United States, the larger Graupner Ultra motors are not very popular nor are they regularly stocked by hobby vendors. The team members were also very familiar with the Astro product line and felt confident in their reliability. Although the Ultra motors significantly outperformed the equivalent Astro Flight motors, the lack of availability and absence of suitable gearboxes for the Graupner's caused the team to consider only Astro Flight products.

The power requirements for the design fell between two motors in the Astro product line, the Cobalt 40 and the Cobalt 60. Table 4.2 shows the four Astro Flight motors seriously considered for the aircraft. The performance model showed that the electrical efficiency of both these motors were comparable. Although the 60 is capable of running reliably at 1600 Watts input power, the motor is over half a pound heavier than the 40. The 40 is normally rated at 700 Watts input power, but after careful bench testing, it was found to be capable of running at higher voltages allowing it to be run at 1200 Watts for short periods. The mission profile calls for short periods at full power for takeoff and climb followed by extended cruise periods well within the normal power limits for the 40. The disadvantage of running the motor at such high power levels is premature erosion of the brushes and commutator. For a limited use contest aircraft, this was deemed an acceptable compromise. The team decided to use the Astro 40 8T with 3.1:1 gear reduction over the Astro 60 because of the significant reduction in empty weight and improved scoring potential.

The propeller performance model indicated propellers 14 to 15 inches in diameter with 12 to 14 inches of pitch would load the selected motor to the required 1175 Watts. The relatively high pitch of the propeller was chosen because of the aircraft's high cruise speed (60ft/sec). This pitch was a compromise between the advance ratio required for efficient cruise, and thrust required for takeoff and climb performance. Because of variation in efficiency and power absorption between different propeller manufacturers, the propeller model was calibrated using data gathered from the motor manufacturer and with testing performed with an Astro 40. Final propeller selection will occur during flight testing.

Early motor bench testing conducted with fuses indicated that all fuses failed at significantly higher current levels than their rated value. This was especially noted when the fuse was mounted within the propeller slip stream. Because the propulsion system cost is directly related to the current rating of the fuse, it will be possible to safely increase final score by choosing a slightly lower current fuse for use in the airplane.

4.2.4 Wing Design

Early in the design phase, it was recognized that the cost of the wing represented 20-40% of the total cost of the airplane. Wing cost was largely dependent on wing area, so in order to reduce cost as much as possible, the wing was designed to operate efficiently at high lift coefficients. This was a function mainly of the airfoil selected for the design. The wing plan form was also considered in the design because of its impact on handling, aerodynamic efficiency, and the maximum lift achievable given a constant wing area.

Several airfoils were compared using X-foil, an airfoil design tool. The design Reynolds number for takeoff was 200,000 while the cruise condition was Re(CL)^0.5 = 250,000. Figures 4.2 and 4.3 show the four airfoils considered for the design. Each of the airfoils shared similar physical characteristics, thickness ratios above 10% and camber above 3.6%. Maximum lift coefficient, drag at cruise (CL 0.8-1.2), usable CL range, and behavior near stall were considered. Of these requirements, the wing section drag impacted the maximum score the least. This was because the viscous drag was dominated by the contributions of the large fuselage and landing gear. The JA13 airfoil showed the best compromise between these factors and was chosen for the design. Figure 4.4 shows the airfoil with its characteristics.

The impact on score from wing plan form was also investigated in attempt to understand sensitivity. Constant chord, single taper, and multi-taper wings were evaluated. It was found that a constant chord, zero twist wing of aspect ratio 10 was only capable of achieving 88% of the lift of an elliptically distributed chord wing's maximum lift, before the root stalled. However, this ensures safe stall characteristics compared to the elliptical chord distribution. A compromise plan form was chosen which was capable of 95% the maximum lift to ensure reasonable handling. The expected lift distribution is shown in Figure 4.5.

4.3 Analytical Methods

Many of the tools used in the design process were created by the team members using Visual Basic or in Microsoft Excel. Nearly all the programs were written some time before work on the 00/01 Design/Build/Fly aircraft began and have been already well tested and validated. These programs include an electric motor/ propeller model, an aerodynamic model, a weight estimation spreadsheet, a longitudinal static stability model, takeoff and climb simulation, and a lift distribution spreadsheet. These tools were incorporated into a 00/01 DBF mission simulation model which included a cost model. To check validity, aircraft with known performance from the 99/00 competition were also run through the simulation successfully.

The only tool not developed by the team members was X-foil, written by Mark Drela. After learning to use X-foil, results from low Reynolds number wind tunnel testing performed at University Illinois Urbana Champaign were compared with data produced by X-foil. At the relatively high Reynolds numbers that this aircraft will fly at, the correlation between test data and predicted performance was excellent.

4.4 Features that produced the final configuration selection

During the design process, care was taken to identify the important aspects of the mission. By understanding the sensitivity of each design parameter, the team could focus their design efforts on the areas which would produce the highest scoring aircraft. Guided with experience and careful modeling of the system, the following features of the aircraft were found to be critical for attaining the highest score and were incorporated in the team's design. They are listed in order of importance.

- Low Empty Weight
- 100 Tennis Balls/ Equivalent Weight Steel Payload
- Conventional Design/ Conventional Handling
- Six Laps
- Aircraft Appropriately Sized for Mission
- Minimize Wing Area
- High Lift Airfoil

Because of these features, the Cal Poly 00/01 Design Build Fly should perform very well at the competition in April.

Figure 4.1 Mission Sizing

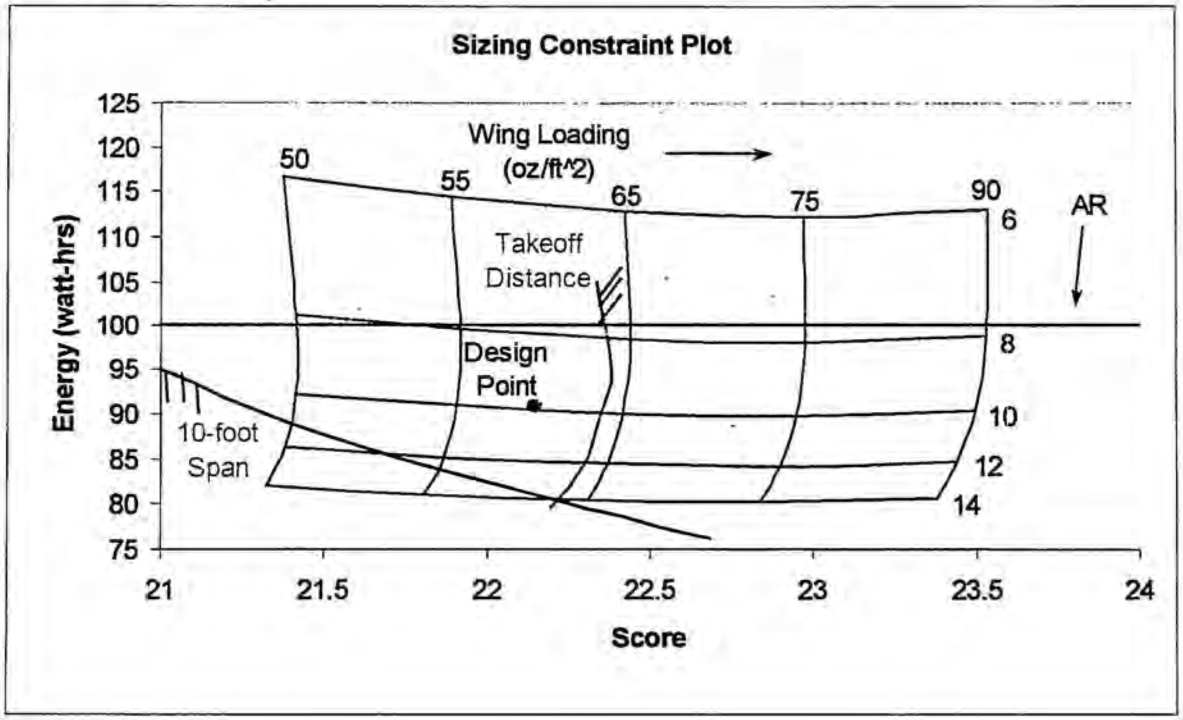


Table 4.1 Battery Pack Trade

NiCd Cell	*mAh*	*Weight (oz)*	*Package*	*Cells in 80oz Pack*	*mOhm /cell*	*Pack Voltage*	*Amps @ 1.0V/cell*	*Pack Max Watts*	*Pack Watt-hr*
KR-1100AAU	1100	0.85	AA	94	20	113	10	941	124
KR-1500AUL	1500	1.09	4/5A	73	16	88	13	917	132
KR-1700AU	1700	1.25	A	64	14	77	14	914	131
N-1250SCR	1250	1.50	4/5sub C	53	4.5	64	44	2370	80
RC-2000	2000	2.00	sub C	40	3.8	48	53	2105	96
RC-2400	**2400**	**2.15**	**sub C**	**37**	**3.6**	**44**	**56**	**2067**	**107**
N-3000CR	3000	3.00	C	27	3.3	31	61	1616	96

Table 4.2 – Motor Selection

Motor	*Weight (oz)*	*Gear Ratio*	*Propeller*	*Cells (RC2400)*	*Power In (Watts)*	*Max Efficiency*	*Cruise Efficiency*
Astro 40 8T	**14**	**3.1**	**15x12**	**30**	**1175**	**81%**	**77%**
Astro 40 10T	14	2.75	18x18	32	1225	80%	78%
Astro 60 11T	22.5	2.75	20x18	32	1300	82%	76%
Astro 60 7T	22.5	2.75	14x10	30	1300	84%	69%

Figure 4.2 Airfoil Comparison

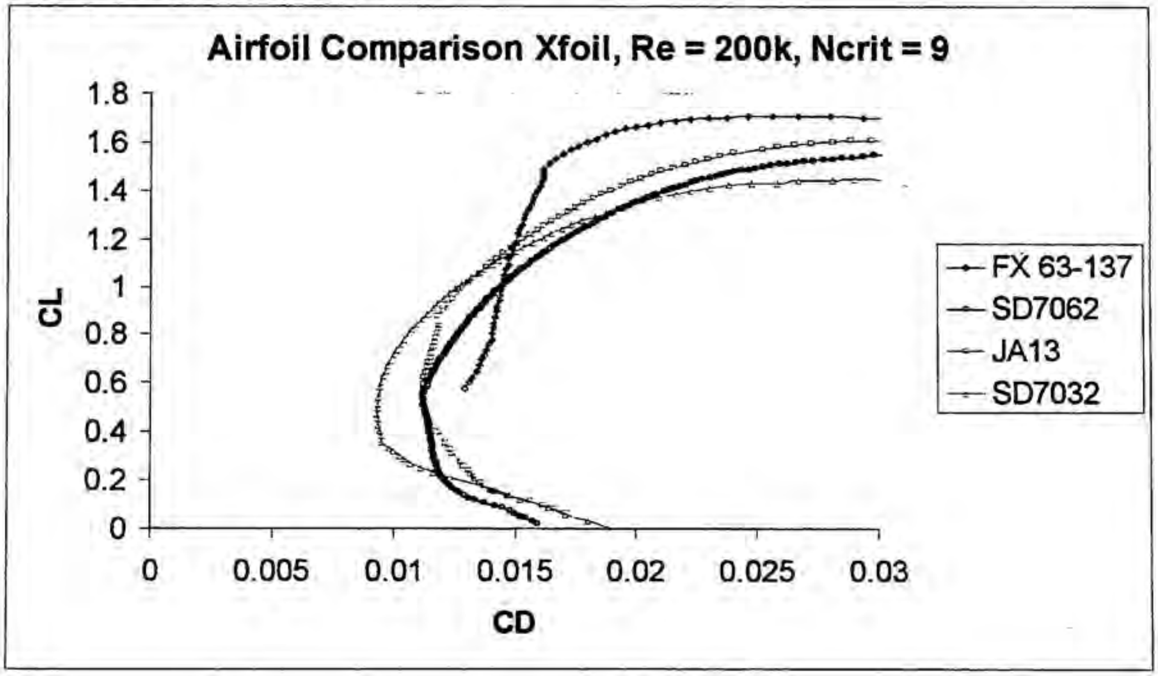


Figure 4.3 Airfoil Comparison

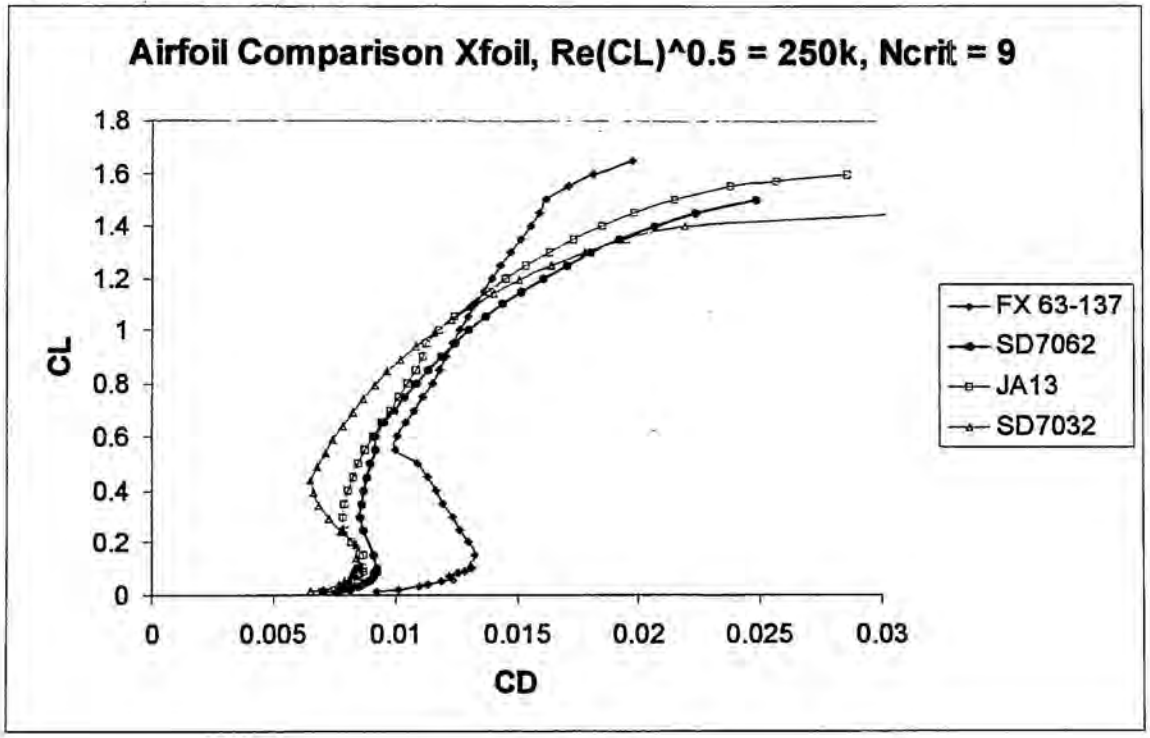
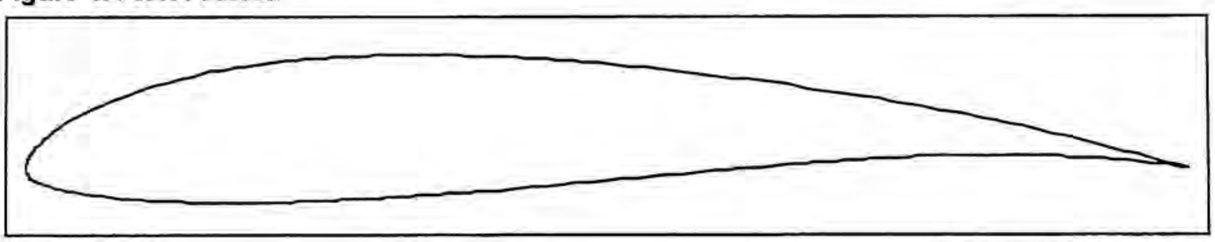
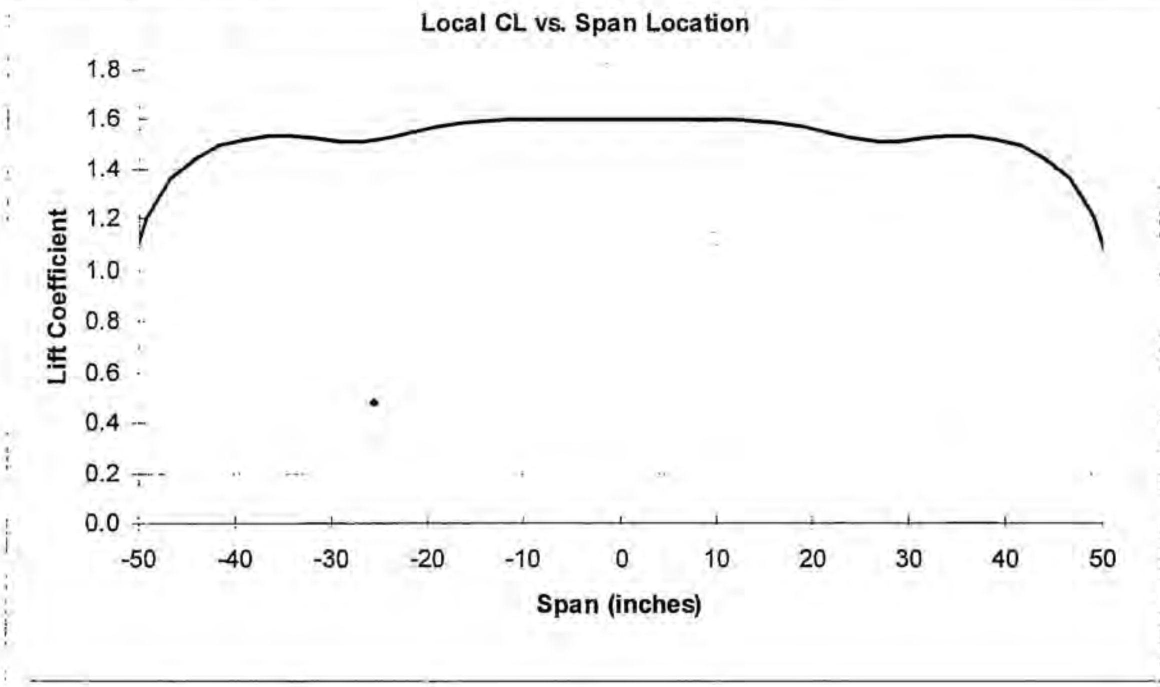


Figure 4.4 JA13 Airfoil



Thickness Ratio: 13.0%
Camber: 4.2%

Figure 4.5 Wing Lift Distribution



5.0 Detailed Design

The three-view layout is illustrated at the end of the detailed design section. The payload carrier is sectioned with the fuselage in the side view to display loading and structural detail.

5.1 Takeoff Performance

The final configuration's takeoff performance is listed in Table 5.1. These distances were calculated with a time based simulation using the equations of motion. The program incorporates a simple propulsion, aerodynamics and rolling resistance model to calculate the forces on the aircraft. This model was developed two years ago for the 98/99 DBF contest and has been validated over the past two years of competition.

Based on past team experience, it was decided to size the aircraft for comfortable takeoff margin with zero head wind. Too many teams in the past have wasted competitive scoring sorties because of missing the takeoff distance during the competition.

5.2 Payload Fraction

The final configuration's mass properties are listing in Table 5.2. The payload fraction for the design is nearly 50%. Careful structural design, the use of composites, and experienced builders allow the airplane to be very lightweight. The final scoring potential is heavily driven by manufacturers empty weight. Not only does it drive the aircraft sizing for performance, it makes up nearly one-third of the cost. Based on the type of construction seen on competing schools aircraft over the past few years of the contest, this aircraft should have a significant advantage because of it's lightweight and innovative structural design.

5.3 Payload Configuration

Several different packing configurations for 100 tennis balls were considered. The final configuration needed to cleanly fit within the fuselage of a conventional, low wing airplane. The stacking orientation that suited the design the best was consecutive rows of six balls arranged in a "six shooter" shape. The seventh ball of each row would be offset in between rows in the center of the "six shooter". This packing arrangement fits all 100 tennis balls with a 8.3 inch diameter cylinder, 40 inches in length.

For quick loading and unloading during a sortie, the tennis balls are carried within a "speed loader". The speed loader is constructed using thin transparent mylar wrapped into a cylindrical shape. The alternate payload, steel bars, which is a much higher density payload, easily fits within the cargo area designed for the tennis balls.

5.4 Performance

The drag model used for the design work calculates the viscous drag of each airframe component based on Reynolds number, shape, and area. For the wing and tail, airfoil drag polars generated in X-foil were used with a table lookup arrangement to estimate viscous drag. Induced drag was calculated using conventional methods. Because the team had no way to verify the drag estimates during the design phase, very conservative numbers were used. This decision was made to ensure that the airplane would have at least the expected performance, if not better than anticipated. Figure 5.1 shows the expected aerodynamic performance of the final configuration. Figure 5.2 shows the climb performance for the final configuration. The aircraft has an acceptable climb rate at gross weight of 15ft/sec.

During sizing, the available energy stored within the NiCd pack was estimated conservatively to allow some margin of error in performance estimation and for safety while flying the mission. It was assumed that the batteries would only deliver 85% of their rated capacity. This gave the 32 cell pack of RC2400 NiCds an energy storage of 80 Watt-hours. Table 5.3 shows the expected energy usage while flying the mission.

5.5 Propulsion System

The propulsion configuration for the design is a single nose mounted brushed DC electric motor. The motor chosen for the design is the Astro 40 8Turn Geared Cobalt motor with 3.1:1 spur gear reduction. Electrical power for the system is provided by 32 Sanyo RC-2400 Nickel Cadmium batteries assembled in series. The motor is throttled with an Astro Flight 204D electronic speed controller. The motor turns a 15-inch diameter by 12-inch pitch two bladed propeller at 7000 RPM. As part of the aircraft cost rules, a fuse must be placed within the motor system to determine the current capabilities of the motor. The expected full throttle input voltage for the motor is 35 Volts with a corresponding current of 35 Amps.

5.6 Landing Gear

The landing gear configuration is a tricycle design. The team manufactured a carbon fiber pre-preg main gear that attaches to the bottom of the wing. The steerable nose gear is a Robart RoboStrut 500 lightweight strut. The gear will use standard heavy duty R/C airplane rubber tires. Provisions are being made to fabricate a main wheel braking system if during flight testing, brakes are deemed necessary.

5.7 Mass Properties

A firm understanding of the location of the center of gravity (cg) is crucial for a good-handling airplane. The design allows for flexibility of component locations if initial weight estimates are not as predicted. Refer to Table 5.4 for a weight and balance table. The moments are all calculated from a datum plane one inch in front of the nose. The location of the cg corresponds to 35 percent of the mean aerodynamic chord (MAC).

5.8 Stability and Control

A good-handling airplane can greatly increase the chances for success in a competition of this nature. The emphasis on handling qualities is evident throughout the entire design process. One key performance trade off is obvious in the plan form of the wing. An elliptical lift distribution is optimal for generating lift efficiently. However, the trade-off is a decrease in overall stall characteristics. The plan form is tailored to get near elliptical lift distribution without sacrificing the handling qualities during stall.

The only penalty for a slightly oversized tail was an increase in structural weight and a minor increase in drag. For this reason, the vertical and horizontal tails were sized with generous tail volumes. The tail volumes are 1.25 and .075 for the horizontal and vertical respectively. The large horizontal tail allowed for a wide range of flyable CG locations while providing excellent pitch damping characteristics. The vertical tail provides control power in a strong crosswind. The large vertical also assists stall characteristics during gusty conditions by minimizing yaw excursions. Good ground handling qualities are key in minimizing time on the ground from touchdown to take-off. The conditions at Webster field are often unpredictable, so the chances of landing in a crosswind are high. Tail draggers classically have difficulty tracking straight without a lot of pilot skill. For this reason, the tricycle gear configuration is the best choice for a good-handling airplane.

The payload and batteries drive the location of the cg heavily. For this reason, the payload and the batteries are located on the cg. The size of the tail allows the cg to move while maintaining adequate control power and stability. The cg is located at 35 percent of the MAC to provide a design static margin of 10 percent. The lower than normal design static margin is a result of pilot preference.

5.8.1 Methods of Control

The aircraft is controlled in roll by deflecting the ailerons differentially to avoid adverse yaw. Dihedral provides spiral mode stability at the cost of poor crosswind performance. For this reason, the aircraft has no dihedral. The roll however is coupled to yaw by mixing the two controls electronically with the radio. The rudder provides yaw control and pitch is controlled by the elevator. Each control surface has one servo, refer to Figure 5.5 for the layout. The nose wheel is controlled by one servo that is electronically mixed to the rudder stick.

Three view and Payload Layout

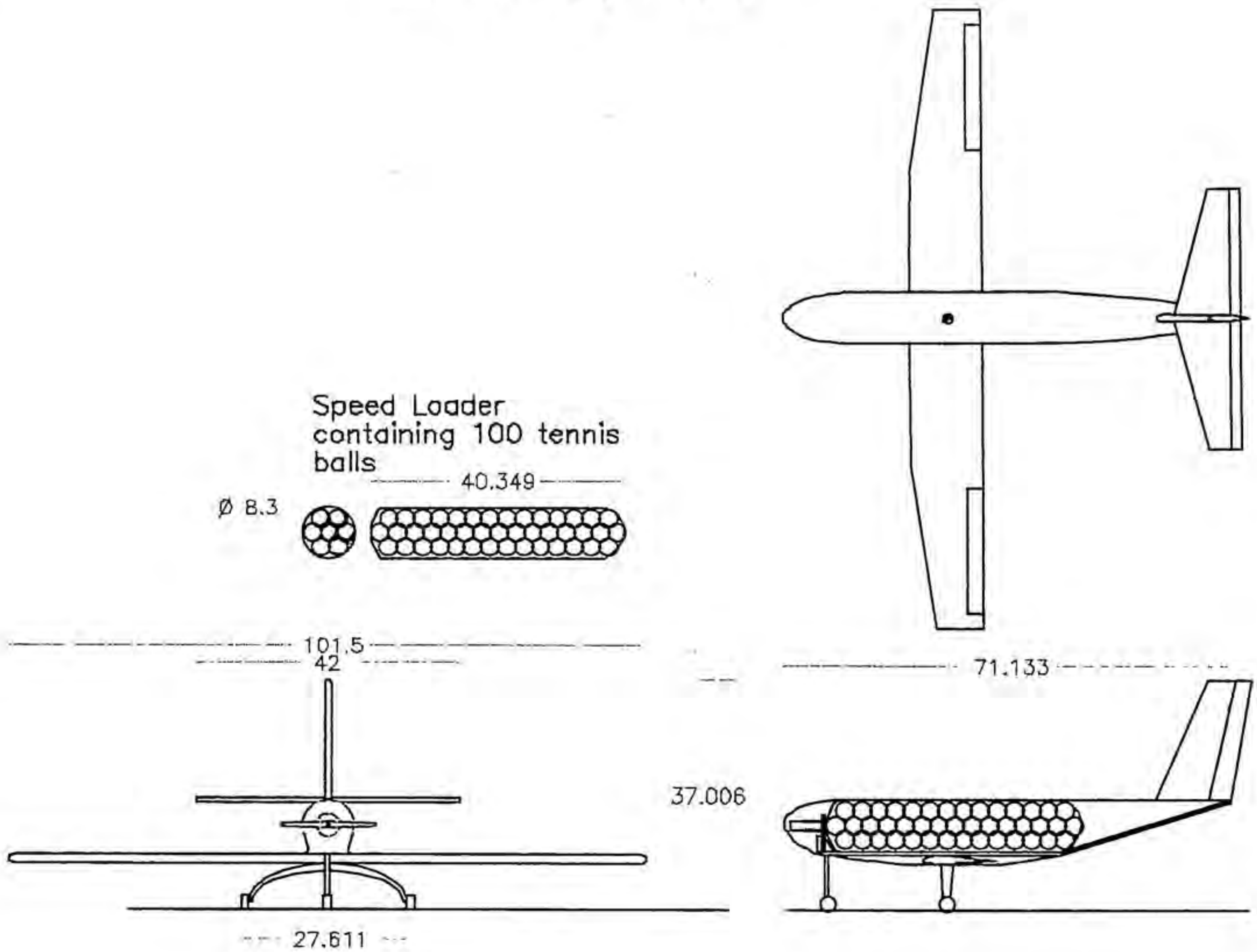


Table 5.1 Takeoff Distance

Head Wind (mph)	*Ground Roll (ft)*
0	160
5	130
10	90
15	40

Table 5.2 Weight Breakdown

Component	*Weight (lb)*	*Mass Fraction*
Wing	2.09	8.0%
Tails	1.17	4.5%
Fuselage	2.15	8.3%
Landing Gear	0.96	3.7%
Motor Sys	1.36	5.2%
Batteries	4.32	16.6%
Radio gear	0.98	3.8%
Empty Weight	13.03	50.1%
Payload	13.00	49.9%
Gross Weight	**26.03**	

Table 5.3 Flight endurance

Steel Lap	*Time (seconds)*	*Energy (Watt-Hr)*
Takeoff	7	2.7
Climb	9	3.5
Turns	17	4.3
Downwind	11	1.5
Landing	12	0.3
Lap Total	56	12.4
Tennis Ball Laps		
Takeoff	7	
Climb	9	3.6
Turns	17	4.3
Straight Legs	25	3.3
Landing	12	0.3
Lap Total	69	14.2
Six Lap Total	**375**	**80.0**

Figure 5.1 Aerodynamic Performance

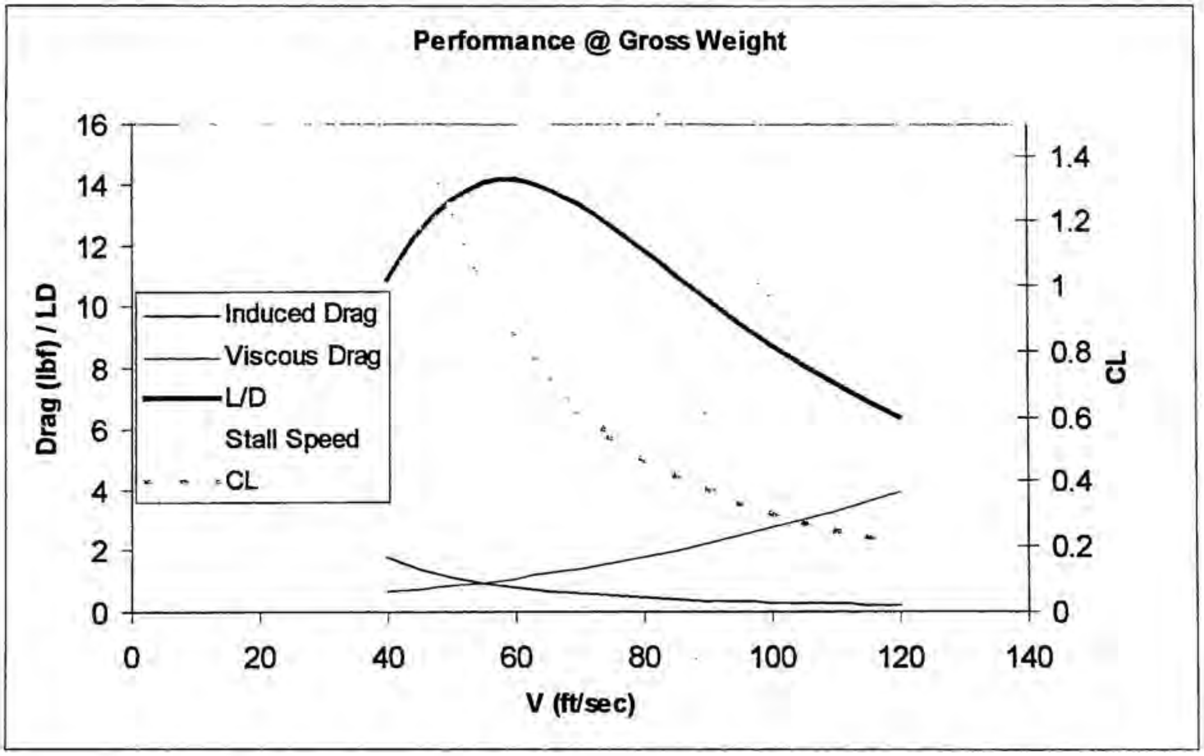


Figure 5.2 Climb Performance

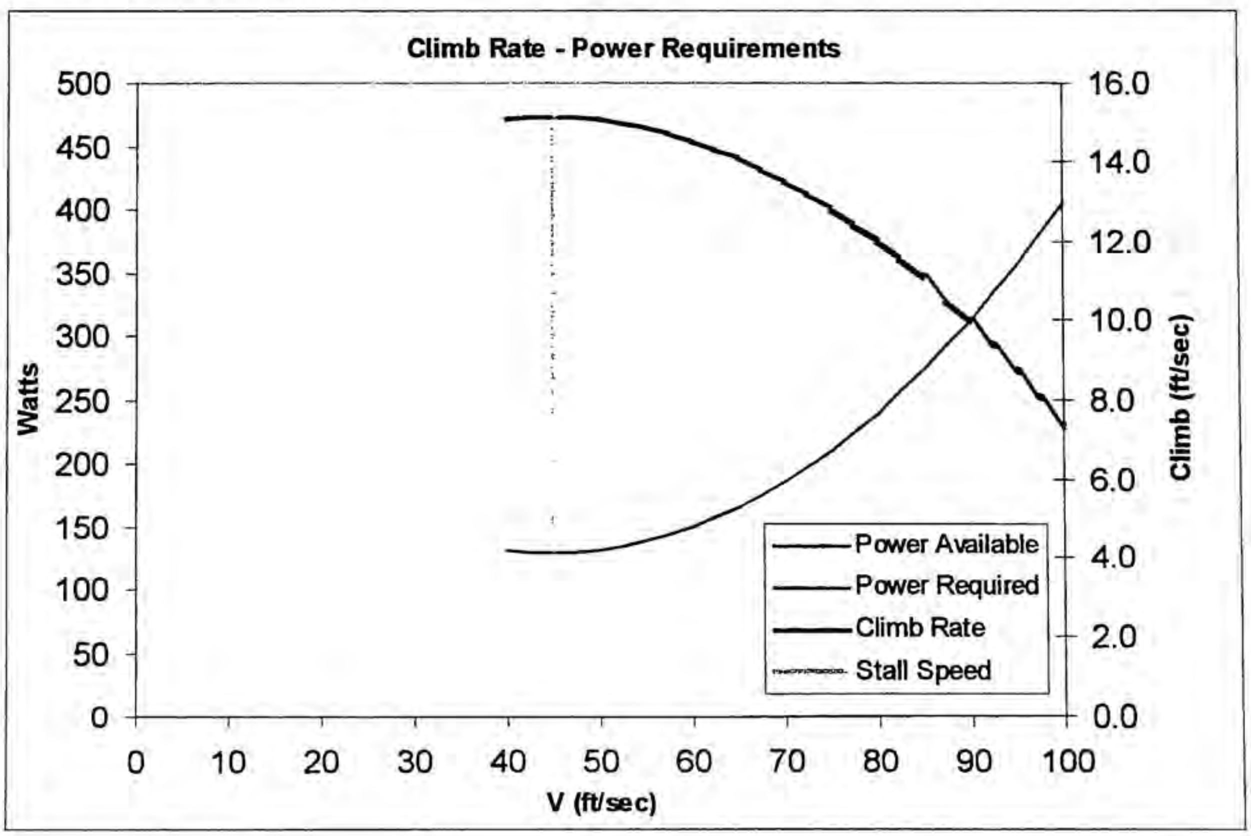


Figure 5.3 Systems Layout

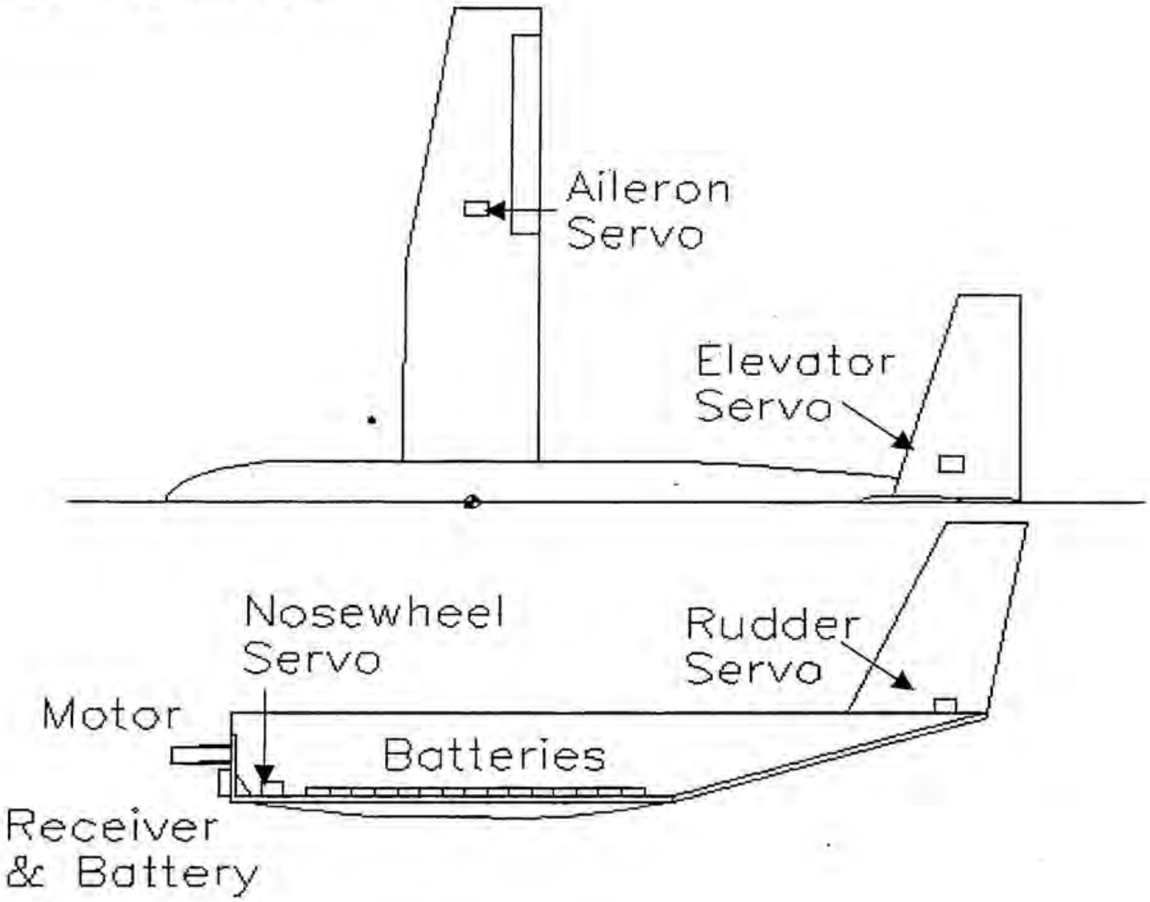


Table 5.4 Weight and Balance

Component	weight (oz)	arm (in)	Moment (in-oz)
Main Gear	*10*	*28*	*280*
Nose Gear	*5*	*8.7*	*43.5*
Horizontal Tail	*13*	*68*	*884*
Vertical Tail	*7*	*68*	*476*
Wing	*33.5*	*28*	*938*
Batteries	*69*	*27*	*1863*
Fuse	*34*	*30*	*1020*
Motor+Controller+Propeller	*22*	*5.2*	*114.4*
Radio Gear	*15.5*	*9*	*139.5*
Payload	*208*	*28.2*	*5865.6*
Totals	*417*		*11624*
CG Location	**27.9**		

6.0 Manufacturing Plan

6.1 Design Drivers

The figures of merit (FOM) driving the manufacturing of the aircraft include: ease of construction, durability, weight of materials, tooling, time, and availability of materials. The value of each FOM is illustrated in Table 6.1. The main goal of the manufacturing was to make the lightest possible structure to decrease the empty weight cost of the aircraft.

6.1.1 Ease Of Construction

The ease of construction was a driving FOM throughout the design. With experienced model builders on the team, there were many designs that could easily be discarded due to obvious construction complications. Experience in this area was what made the final design a feasible aircraft to manufacture. The entire aircraft took advantage of the simple manufacturing processes it was designed for. The fuselage is a simple foam core with a carbon keel and a Kevlar skin. The wing was hot-wired from Dow blue foam (density 1.8 lb/ft^3) and vacuum bagged using a carbon spar cap and a Kevlar skin as the primary structure for bending and torsion loads. Similarly, the horizontal and vertical tails are made from Dow foam with a carbon spar cap and Kevlar skin. The aircraft easily lends itself to post-design modifications due to the simplicity of the entire manufacturing process. In addition with a foam core structure, the aircraft is very durable.

6.1.2 Cost

There were two aspects of cost considered throughout the design and manufacturing of the aircraft: monetary value of the materials used and the Rated Aircraft Cost (RAC) stated in the rules. The cost of materials was a concern due to the minimal budget the team had to work with. The high-end materials such as carbon, Kevlar, and Rohacell were the most expensive part of the empty weight of the airplane. The systems including batteries, motor, controller and receiver were all very expensive items that were essential to the success of the aircraft as well. Overall very few compromises had to be made to stay within budget due to generous donations from Northrop Grumman and Lockheed Martin Skunk Works.

The RAC drove the design and manufacturing trades throughout all phases of the design process. The most effective way to build an inexpensive aircraft was to make it very light. All of the main structure was optimized for minimal weight to keep the RAC of the aircraft low.

6.1.3 Weight

Careful consideration for weight was taken while choosing the materials and methods for construction. The design takes advantage of hybrid materials that provide astounding strength to weight ratios. Carbon fiber is very strong, light and workable. The total empty weight of the aircraft needed to be kept to a minimum in order to maximize the payload fraction and keep the RAC down. Experience was definitely a factor in efficiently strengthening the heavily loaded areas of the aircraft.

6.1.4 Tooling

Various molding methods for manufacturing the aircraft were briefly considered, however, the time spent designing and constructing a mold for the wing and fuselage was better spent building the airplane. However, the landing gear was constructed using a male mold and pulling a female part due to the simple geometry. The entire landing gear construction process only represents a small fraction of the total work done on the aircraft.

6.1.5 Time

Manufacturing time was a driving factor throughout the entire design process. Cal Poly's participation in the Design/Build/Fly competition is extracurricular, so all the work had to be done during the team's spare time. The time allotted provided for very little experimentation with different construction techniques. The team decided to use the construction methods they were familiar with. The decision was made early to also simplify the construction of the aircraft. In so doing, many potential problems were completely avoided.

6.2 Manufacturing Processes

6.2.1 Wing

A molded wing was immediately rejected due to the lengthy process of fabricating the molds. Obechi was considered, however, the material introduced an increase in weight to achieve the same ultimate strength. Refer to Table 6.2 for the qualitative evaluation of the wing construction

The team had experience with wet lay-up, so it was decided to cut the airfoil from DOW foam with a Kevlar skin and a carbon spar cap. Using the lift curve slope generated from the plan form data of the wing, the lift force was known at any location along the span. The wing was divided up into one-inch increments so the lay-up schedule could be optimized. Since carbon gets very expensive as the thickness is decreased, 2.9-oz/yd^2 unidirectional (uni) cloth was used as the primary bending load member. With a fixed thickness, the width of the carbon was tapered to match the local moment of inertia required to prevent the wing from failing. With a thick airfoil, three layers of carbon at the root provided the strength to withstand 15 g's. Refer to Figure 2 for the lay-up schedule.

The foam was purchased from the local hardware store. The carbon and Kevlar were wetted out on two pieces of Mylar. The Mylar sandwiched the core, and was put into a vacuum bag. The panel was cooked at 100 degrees for 24 hours. The other wing panel and the tail are constructed using the same methods.

6.2.2 Fuselage

The fuselage is the area where the most innovation is evident. There are two parts to the fuselage: the skin and the keel. The keel absorbs all the flight and landing loads and provides hard points for the wing, tail, landing gear, payload, batteries, and motor. The keel is designed for both stiffness and ultimate stress. The keel is constructed using Rohacell foam (4.7 lb/ft^3) and carbon to create a sandwich. The Rohacell provides nearly ten times the compression strength of the Dow foam, and only weighs twice as much. The trade off is essential in making a strong, durable fuselage structure. The designed ultimate stress of carbon vacuum bagged over Rohacell is 100,000 lb/in^2. A trade study was done to determine the best combination of Rohacell and carbon to meet the load requirements. The moment of inertia increases with the square of the thickness of the Rohacell. However, the Rohacell is very heavy compared to a few layers of carbon. The best combination of foam and carbon is at the bottom of the bucket in Figure 6.3. However, Rohacell conveniently comes in 0.5-inch thick sheets. To save time and maintain accuracy, the 0.5-inch thick foam was used as illustrated by the red dot in Figure 6.3.

6.2.3 Tail Surfaces

The methods explored to build the tail include a wood build-up, balsa over foam, and balsa with fiberglass over foam. A foam core tail covered with balsa wood alone tends to be heavy due to the amount of resin required to adequately bond the wood to the foam. A wood built-up tail is very fragile and time consuming. Using the same construction method for the wing, the tail construction was simple and fast. The ranking of the processes is displayed in figure 6.4.

6.2.4 Landing Gear

The landing gear presented another area for innovation and optimization. Historically landing gear are about 10 percent of the total weight of the airplane. For a 30-pound airplane that is three pounds. With prior experience using carbon and foam, the landing gear was optimized for both ultimate stress and bending stiffness. The gear was sized for a pilot error on landing resulting in a 6-ft/sec impact with the runway. Assuming two inches of travel on the main gear, this sink rate is equivalent to a 10-g landing. The difficulty when sizing the gear was achieving the ultimate strength with two inches of deflection without failing. There are a few ways to meet both of these criteria. The gear could be designed for a constant stress by decreasing the moment of intertia from the root to the wheel. A tapered width landing gear is an easy way to decrease the inertia. Another way is to taper the thickness, or number of layers of carbon along the length of the gear. Both of these methods were employed to meet the design criteria. In addition, selecting the best combination of core and skin thickness could minimize the weight of the gear. Figure 6.2 illustrates the results of this optimization with the red dot representing the thickness with a factor of safety added for integrity. The nose gear was purchased from Robart due to its robustness and low cost.

6.3 Analytical Methods

The analysis method for screening the manufacturing processes was evaluation based on experience, instead of using elaborate models. A simple spreadsheet was created to compare the impact on wing weight using various densities of materials. The team has a good feel for what it takes to design and build a good aircraft. Simplicity proved to be the best analytical method available. By keeping the design basic, the skill-level required was significantly reduced. Time for construction and testing was always a factor throughout the project. Each construction technique was highly scrutinized to ensure ample flight test time to reduce the chance for pilot error.

6.4 Manufacturing Timeline

To complete the project on time, a tight schedule was adhered to throughout the design and fabrication processes. Milestone chart illustrated in Figure 6.3 was constructed to avoid the last minute crunch.

Table 6.1 Qualitative Ranking of The Figures Of Merit

Figure of Merit	Ranking		
	5	*3*	*1*
Ease of Construction	easy	medium	hard
Durability	High	medium	low
Weight	light	medium	heavy
Tooling	none	some	required
Time	short	average	long
Materials Avail.	easily acquired	average	difficult to find

Table 6.2 Ranking of The Figures Of Merit For The Wing

Figure of Merit	Ranking		
	molded	*obechi over foam*	*wet lay-up*
Ease of Construction	1	5	5
Durability	1	3	5
Weight	5	1	5
Tooling	1	5	5
Time	1	3	5
Materials Avail.	3	3	5
Total	12	20	**30**

Table 6.3 Ranking of The Figures Of Merit For The Tail Surfaces

Figure of Merit	Ranking		
	built-up	*balsa/foam*	*Kevlar/Carbon/Foam*
Ease of Construction	3	5	5
Cost	5	5	3
Weight	3	3	5
Tooling	5	5	5
Time	3	3	5
Materials Avail.	5	5	5
Total	24	26	28

Figure 6.1 Wing Lay-up Optimization

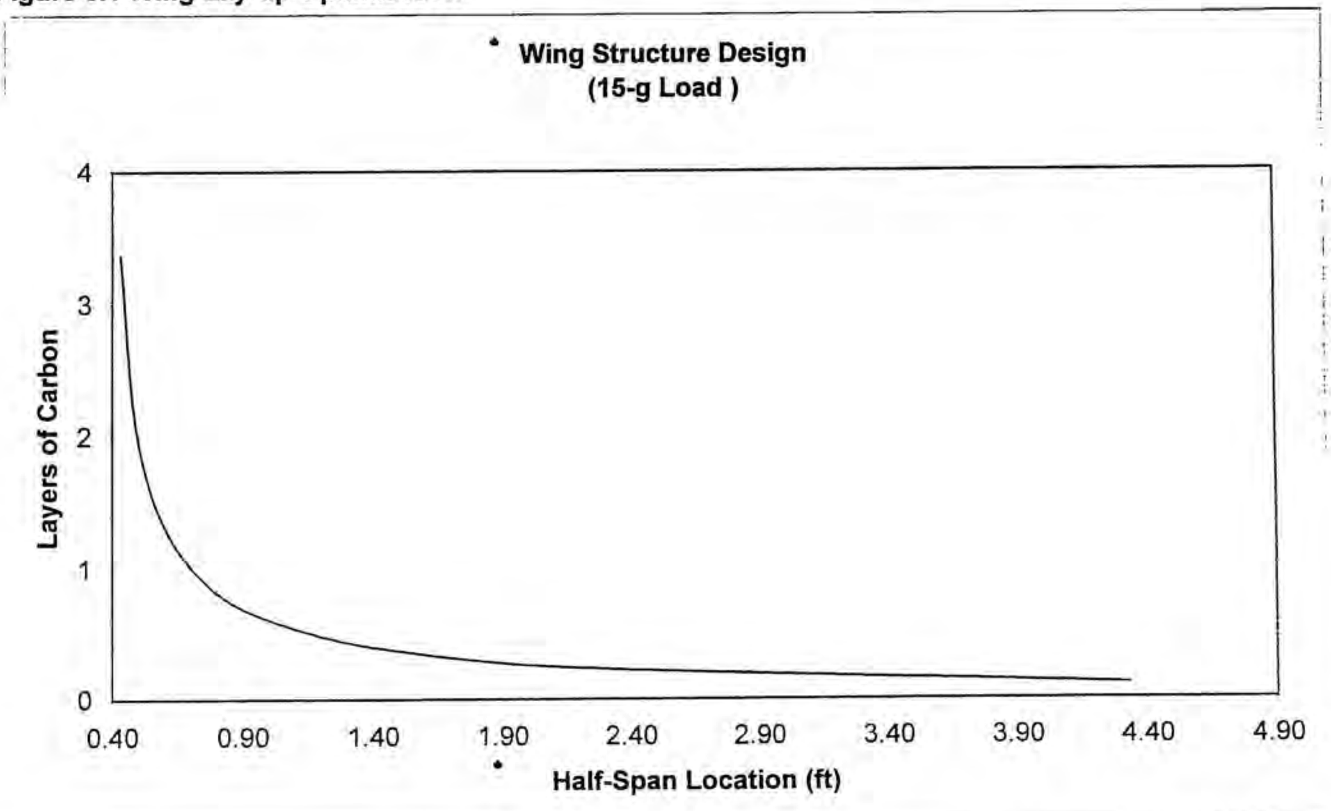


Figure 6.2 Keel Optimization

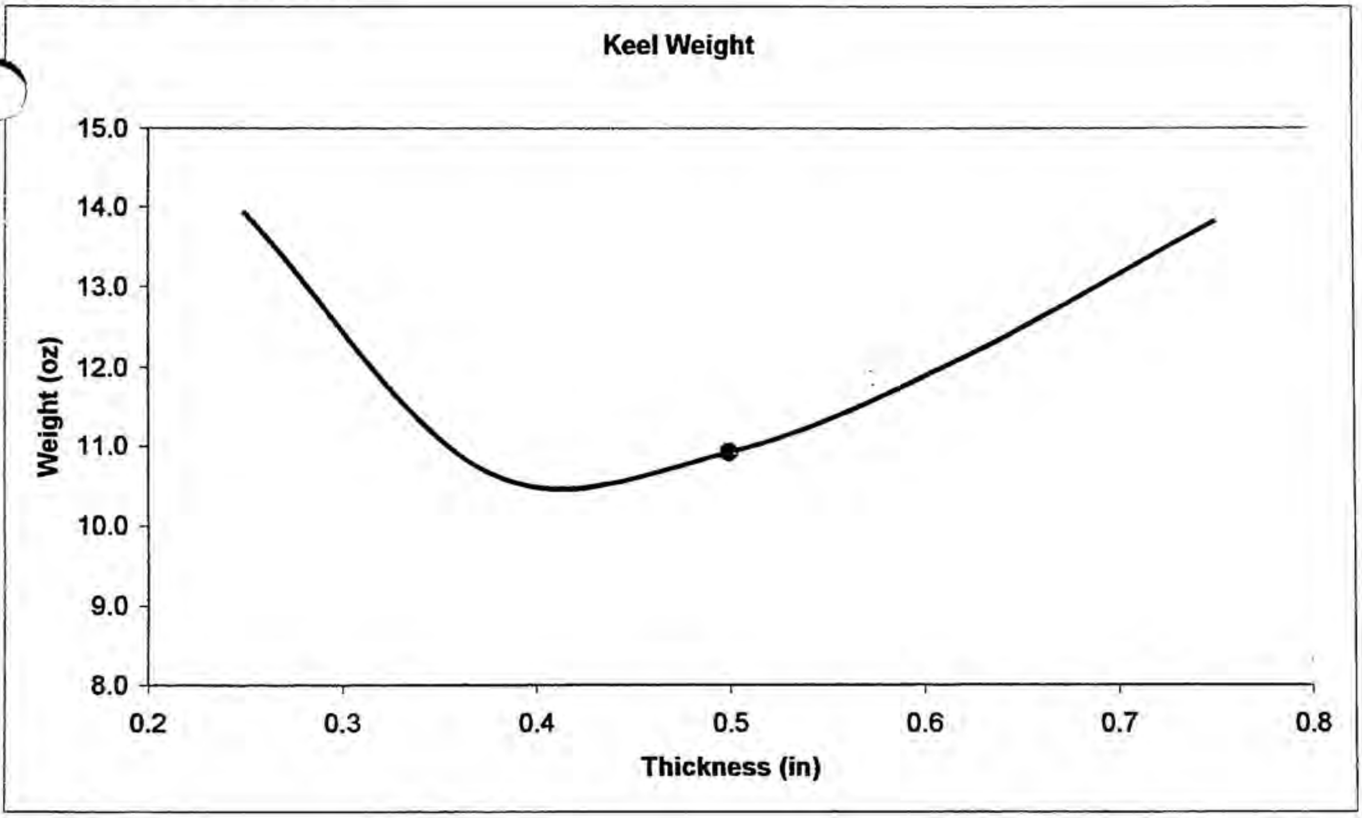
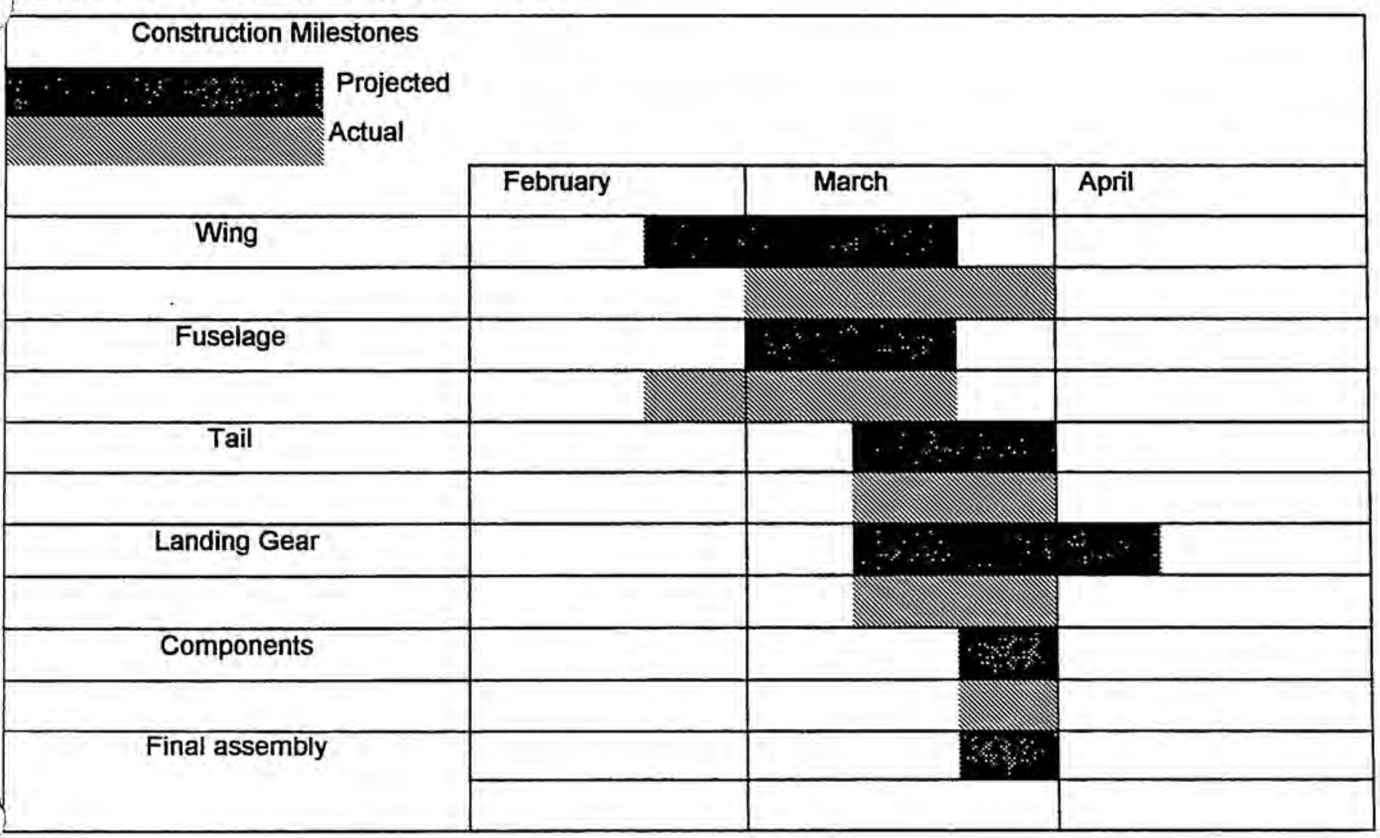


Figure 6.3 Project Timeline for Construction



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Stealin' Balls

Design Report – Addendum Phase

2000/2001 Cessna/ONR/AIAA Student Design/Build/Fly Competition

Aeronautical Engineering Department

California Polytechnic State University, San Luis Obispo

Authors: John Asplund, James Bach, Jeremy Rocha, Francesco Gianinni

April 9, 2001

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7.0 Addendum

Throughout the design process, there were several elements of the design that differed from the actual aircraft constructed. The main differences are a result of optimistic weight estimation for several main components of the aircraft. The weight increase had a large impact on other aspects of the design including the propulsion system, performance and cost.

7.1 Weight Estimation

Accurate weight prediction is essential when optimizing an aircraft for a competition of this nature. The performance of any aircraft hinges on the weight of the vehicle. For this reason, the weight prediction tools were tuned using existing model aircraft constructed using similar techniques. The result is a method that can predict the aircraft weight throughout the design process. However, as the scale of the aircraft increases the prediction methods deviate from the constructed aircraft because of several factors. These factors include higher than normal g loading, methods of construction and structural efficiency.

7.1.0 Structure

The airframe weight estimate without the motor was originally eight pounds. The actual weight of the aircraft was not as predicted for several reasons. There was a large deviation from the predicted weight estimation of the tail and fuselage. When building with carbon and Kevlar, it is often difficult to predict how much extra material must be added in highly loaded areas to avoid failure. There were also weight increases due to integration of the components of the aircraft. The hardware to mount the tail, landing gear, wing and the motor all contributed to increase in empty weight of the airplane.

The estimated weight of the tail was 10 ounces. The tail received extra material for integration to the fuselage that was not taken into consideration in the original weight estimate. There was also a small strip of Kevlar that provided enough torsional stiffness to prevent the elevator from flexing. The final weight of the tail is 16 ounces for these reasons.

The fuselage construction entailed hand-laid up Kevlar over white foam with a carbon and Rohacell main structure. The foam was hot-wire cut and sanded to shape. Kevlar was carefully laid over the foam and epoxy was applied. The foam provides great durability and is very easy to repair. However, the density of the foam and the total volume introduce a significant portion of the total weight of the fuselage. The weight reduction method was to remove all of the foam leaving a thin skin. Once completed, the team realized the foam provided added durability and strength for the fuselage. The weight prediction method did not account for added weight of the foam, causing a difference between the actual and predicted weight of the fuselage. As a result, the fuselage weight increased by nearly one pound.

7.1.1 Landing Gear

The other main component that contributed to the increase in weight is the nose gear. The mounting brackets and the down tube were both undersized in the original design, making the gear not adequate for the aircraft. In addition, the original tires that the airplane was sized with were several ounces lighter than the tires that are going to be used in the competition. After the modifications, the total weight of the nose gear increased by 0.5 pounds.

7.2 Aircraft Performance

During flight testing, key aspects of the final aircraft's performance were compared with the predicted performance. The areas tested and compared include climb rate, takeoff distance, power settings for cruise flight and endurance. These performance figures were critical for the aircraft to complete its mission.

At a gross weight of 30lb, the expected ground roll for the aircraft with half charged batteries and zero wind was predicted to be 170ft. Flight test data showed that when flown carefully, the aircraft easily left the ground before 200ft under those conditions. Climb rate at gross weight was also validated to be acceptable at approximately 10ft/sec half way through the battery pack.

An area where analysis did not match actual flight performance was cruise power and maximum endurance. During gross weight practice sorties, the aircraft would run out of batteries during the sixth payload lap. After careful consideration and testing, several parts of the design spreadsheet were identified to be slightly over optimistic. It was determined that the aircraft needed to use approximately 12% less energy, (11 Watt-Hrs) in order to complete the mission as intended. In an attempt to get back the sixth lap and improve performance, each contributor to system efficiency was ranked according to its impact. This list was then compared with the design teams ability to make changes to the aircraft in the short time remaining before the contest. The following changes were considered for improvement:

- Add NiCd cells
- Pilot smoothness
- Reduce payload capacity
- Reduce empty weight
- Propeller efficiency
- Speed controller efficiency
- Parasite drag clean-up

The easiest solution was to add more batteries to the aircraft. The maximum power could then be achieved by using a slightly smaller propeller at the higher voltage with less current. The difficulty with this solution was the Astro 40's ability to be run on high cell counts. The Astro 40 8T was designed to run on 26 cells but the team was planning on pushing the motor for the contest by running on 32 cells. In order to run on even higher voltage without destroying the motor, a higher turn armature would be required to keep the motor RPM down. These custom armatures are available, but not in time frame before the contest.

Energy can also be conserved during the flight by careful flying technique. By flying the course tightly and by making sure that the airplane is flown at best L/D speed, more flight time could be achieved. Practicing the course during the next few weeks before the contest will help minimize energy consumption.

Another possibility would be to reduce the payload capacity of the airplane down to 90 tennis balls and 11 pounds of steel. This certainly will allow the flight to be completed while posting a reasonable score with some performance margin.

Most of the unexpected losses in the propulsion system were found to come from speed controller partial throttle inefficiency. In the performance model, speed controller efficiency was estimated at 95%. All of the sizing trades were conducted with this efficiency. The team found that the during cruise flight, the power setting required was noticeably higher than anticipated. A static test was conducted to measure the actual efficiency to better understand the problem. To test controller efficiency, the motor and propeller combination were run on a bench power supply at the voltage and current expected for cruise. The propeller RPM was measured indicating a certain power level output from the motor. The motor was then run through the speed controller with the input voltage set at the full 36 volts of the battery pack. Then using the same propeller, throttle setting was then set to achieve the same propeller RPM. The input power was carefully noted and compared to the previous D.C. test. The team found that the input power increase by 17% when running with the speed control. This test indicated that the efficiency of the controller was only 85%. Unfortunately, no higher efficiency controller technology seems to be available for electric R/C aircraft. The team will investigate higher efficiency controllers for next years contest.

Both drag cleanup and propeller efficiency were considered to improve performance. However, the impact of improvements in these two areas was fairly small compared to other changes in the aircraft. In order to accurately measure propeller efficiency, careful testing would be required in the wind tunnel. This testing would be very useful, but compared to the relatively little time required to make other significant improvements to the aircraft, the team's time would be better spent in other areas. Reducing parasite drag by building wing root fairings, landing gear fairings, and an engine cowl could improve performance, but again is fairly time consuming for little gain.

7.3 Propulsion

During preliminary design, optimistic estimates of empty weight were used for sizing the aircraft. The power requirements for the design lied between two Astroflight motors, the Astro 40 and 60. Because empty weight drove the design strongly, the original design utilized the Astro 40, a smaller, lighter motor. The final aircraft empty weight was 35% higher than anticipated, the original motor no longer fit the power requirements for the design and the decision was made to switch the larger Astro 60. This decision was also made for reliability issues because the Astro 40 was not designed for kilowatt power levels. Another advantage of using the larger motor is its capability of running reliably on higher cell number packs, a solution to the endurance problem. Testing of the Astro 60 will be conducted in the following weeks to optimize the number of cells required to complete the mission.

7.4 Lessons Learned

The aircraft design process provides a platform for learning for every vehicle designed. Likewise, there were many valuable lessons learned throughout the design of this aircraft as well. Foremost is the optimism that can lead to underestimating weights. This seems universal in all aircraft design; the airplane comes out overweight and more power is needed to get the desired performance. The aircraft needed the larger motor, more cells and a bigger propeller because of the increase in weight. We also learned that taking chances with various aspects of the design could take valuable time away from completing the aircraft on time. If we had just chosen the larger motor from the beginning, the time spent trying to make the smaller motor work could have been used to optimize other components of the aircraft.

The competition is driving a design that is easily repairable and very predictable. Since three 10-minute flying periods must be completed, a well-handling airplane is essential for success. Flight test data has shown the overall handling qualities of the airplane are very predictable.

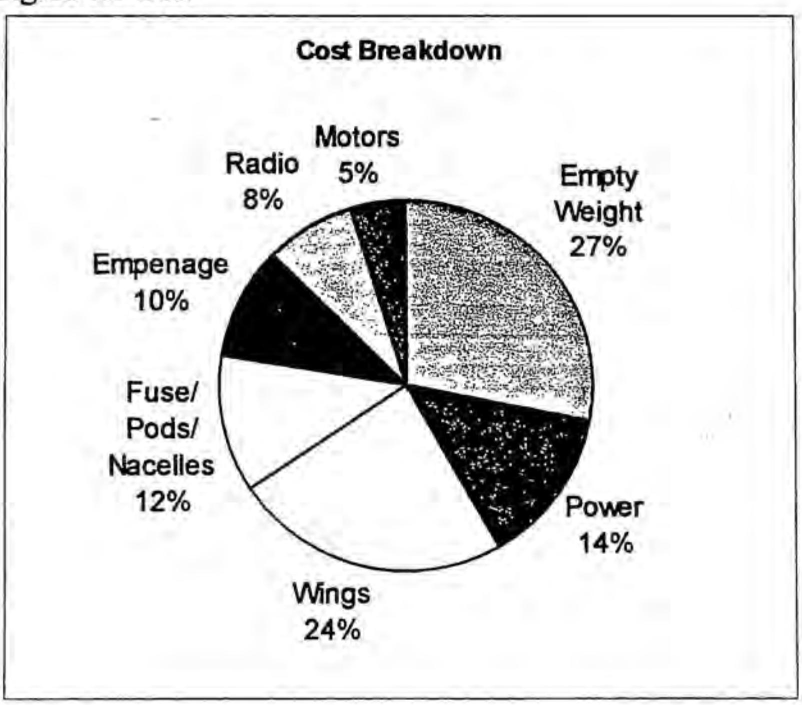
7.5 Cost

The rated aircraft cost (RAC) was a large design driver throughout the design process. The RAC was calculated using the supplied cost model in the rules for the competition. The RAC was carefully taken into consideration throughout the design process. The work breakdown structure (WBS) is provided in Table 7.1. According to the WBS, the majority of the man-hours are from the manufacturing of the wing. The percent breakdown of each portion of the cost is illustrated in Figure 7.1. The pie chart shows that the majority of the cost comes from the empty weight and wing area of the aircraft. The weight could have been shaved down to make a cheaper aircraft but the durability and survivability would have been compromised. The wing area was minimized for both efficiency and cost. The aircraft represents a highly optimized design for each aspect of the contest.

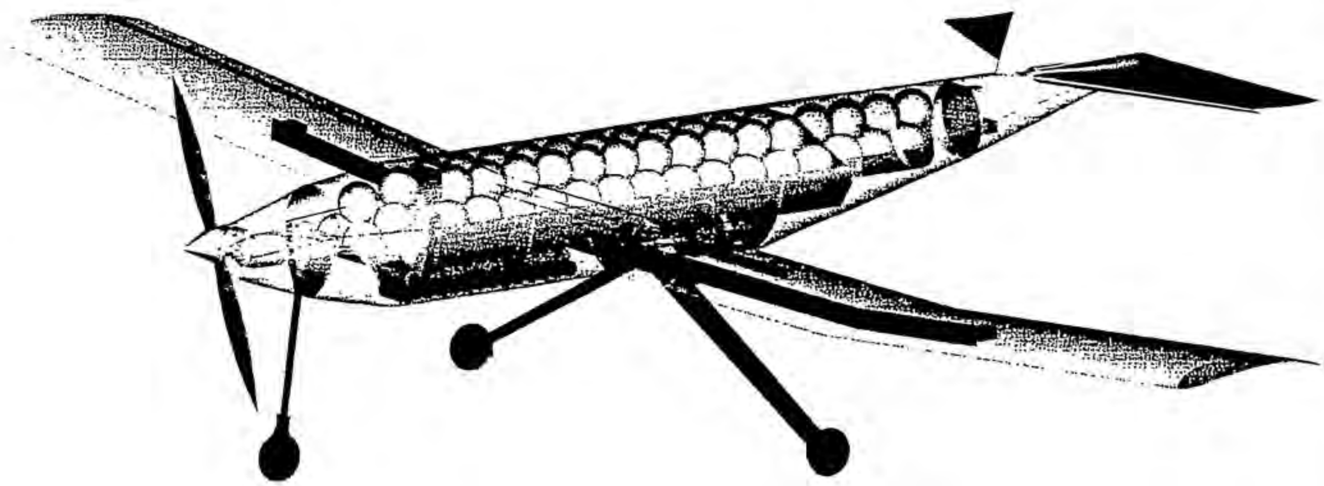
Table 7.1 Cost Breakdown

Empty Weight	11.7	lb	1170
Power	32	cells	576
Current	15	amps	
Wings	1	#	1004.8
Total Area	7.31	ft2	
Surfaces	2	#	
Fuse/ Pods/ Nacelles	1	#	500
Total Length	5	ft	
Empennage	1	#	400
Vertical	1	#	
Horizontal	1	#	
Radio	1	#	340
Servos	5	#	
ESCs	1	#	
Propulsion			
Motors	1	#	200
Props	1	#	
Total			**4.1908**

Figure 7.1 Cost



AIAA/CESSNA/ONR
Design/Build/Fly Competition



Design Report - PROPOSAL PHASE

OSU Black
Oklahoma State University

March 13, 2001

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1.0 Executive Summary

This report summarizes the approach of taken by the OSU Black Team to design an aircraft that would perform well in the 2000/2001 Design/Build/Fly (D/B/F) contest. The goal of this year's competition was to design and build a propeller-driven, electric, unmanned airplane that would be capable of carrying both cargo volume and heavy payloads around a specified course. These payload laps, or sorties, must be completed within a ten-minute time period. The challenge of the competition is to design an aircraft that will achieve a superior score within the requirements set by the contest officials. The contest score specified by the contest rules is a function of the written report score, the total flight score, and the rated aircraft cost.

The first step the OSU Black Team took in the design process was to divide the team into technical groups, to review the AIAA DBF mission requirements and then define technical, operational and support requirements. The three groups – aerodynamics, propulsion, and structures-- then each conducted research into design, power, and construction techniques. During this "pre-conceptual phase," one of the biggest achievements of the aerodynamics group was the development of an optimization code. This code used iterative and nonlinear mathematical solving techniques to locate the best score within the contest constraints. The optimization program allowed the contest parameters to be analyzed in a way that decided the major components of the competition such as wing sizing, sortie configuration, payload sizes, power needed, etc.

After gathering research material, optimization analysis was performed. The teams collaborated to begin the conceptual phase where both non-conventional and conventional designs, capable of performing to the requirements of the optimization, were analyzed and compared. This analysis focused mainly on the aerodynamic properties and the manufacturing feasibility of each design. The Rated Aircraft Cost (RAC) was developed using the six primary design parameters. The team reviewed six alternative planforms, six wing placement options, three fuselage shapes, five tail configurations, five landing gear concepts, four propulsion layouts and four material/construction techniques.

At the start of the preliminary phase a monoplane with a polyhedral, low-wing, circular fuselage with a conventional tail was chosen as the optimum design. The propulsion group selected the battery and motor combination of the SR2400 and the Astroflight 661 motor. The aerodynamics group found the static stability and the control surface sizes. The structures group completed the stress and load calculations. The structures group started the technical drawings.

During the detail design phase, the major components were set, and the manufacturing processes were determined. The aerodynamics group completed the dynamic stability calculations and finalized detailed optimization. The propulsion group selected a propeller to take full advantage of the battery–motor combination within the design performance envelope.

The final design includes an Astroflight 661 motor with a 22-inch propeller and a power source made up of 37 cells of SR2400 NiCad batteries. The wings of the plane will be of polyhedral form using a Selig S1210 airfoil. The ailerons will be powered by 50 oz-in servos. The horizontal stabilizer of the plane will be made with an Archer 18 airfoil and an NACA 0009 airfoil on the vertical stabilizer. Both the elevator and rudder will use horns to reduce the hinge moments on the control surfaces. The fuselage will be built primarily out of foam-supported carbon fiber. The main landing gear will be bow type and built using carbon fiber. The nose gear will be a steerable aluminum Oleo strut design.

The projected mission performance for the steel payload is a take off roll of 160 feet, with a climb rate of 13 feet per second at full power and a cruise speed of 91 feet per second. The 100 tennis ball payload will have a take off roll of 95 feet, a climb rate of 17.5 feet per second and cruise speed of 92 feet per second.

2.0 Management Summary

2.1 Architecture of the Design Team

The design team, led by a chief engineer, is composed of three groups: aerodynamics, propulsion, and structures. Each group is composed of a lead engineer and component specialists; this arrangement can be seen in Figure 2.1. The structure of the team is designed with the chief engineer overseeing the progress of the overall project and the lead engineers' progress, and then the lead engineers oversee the progress of assigned specialists' tasks. Together, the chief and lead engineers assure that all aspects of design and construction are completed on schedule.

At the beginning of the project, the chief and lead engineers assigned certain tasks to each group. The aerodynamics group's first responsibility was optimizing the plane's design and performance within the contest parameters. As the conceptual design was finalized, the aerodynamics group worked with the structures and propulsion group to develop a design that meet the requirements of the contest and was acceptable from the standpoint of the other two teams. After the initial optimization is completed, the aerodynamics members break into integrated product teams with propulsion and structures. Here they conduct trade studies, size control surfaces, and suggest drag reduction methods.

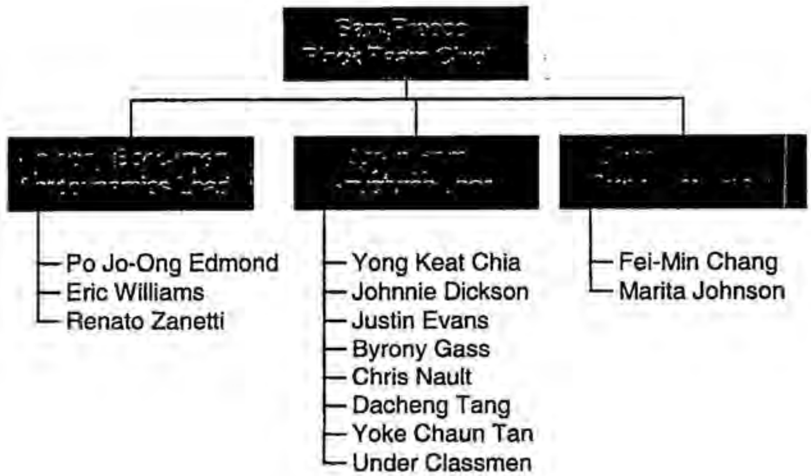


Figure 2.1: Architecture Chart for OSU Black Team

The propulsion group is responsible for the design of the motor, propeller, and battery configuration. After testing, researching, and experimenting, they work with the other two groups to check that the chosen power configurations are sufficient to propel the final design. The structures group is responsible for the structural integrity of the aircraft, lofting and manufacturing work. The integrated products teams are built around structural components of the aircraft such as landing gear, tail structure, and power sources.

2.2 Design personnel and Assignment Areas

The aerodynamics group is primarily responsible for the optimization code, and checking the relevant design constraints. They determine the aerodynamic loads and set design requirements, such as power from the propulsion system, maximum empty weight and component location. They are responsible for predicting and modifying the behavior of the aircraft.

The propulsion group's priority is supplying the plane with power. They have developed an efficient combination of batteries, motors, and propellers that best suites the aircraft. To do this requires knowledge of numerous testing, research, and experimental techniques.

The structures group is responsible for aircraft construction. In order to insure that a suitable and feasible plane is designed, they worked closely with the other two teams during the design phases. This is the group that must make sure that the plane is strong and durable enough to withstand the loads the plane will encounter. They performed structural analysis on the major components of the plane, decided on the appropriate construction methods, and produced the construction drawings.

Team Member	Title	Assignment Areas
Sam Preece	**Chief Engineer**	
Anthony Boeckman Po Jo-Ong Edmond Eric Williams Renato Zanetti	**Aerodynamics Lead**	Optimization/Stability Calcs CG Calcs/Tail Sizing Control Surfaces Airfoil Selection/Wing Sizing
Chad Stoecker Fei-Min Chang Marita Johnson	**Propulsion Lead**	Motors/Prop Fuses/Prop Batteries
Dustin Hamill Young Keat Chia Johnny Dickson Justin Evans Byrony Gass Chris Nault Dacheng Tang Yoke Chuan Tan	**Structures Lead**	Fuselage/Materials Construction/Jigs Landing Gear Design Tail Design/Servo Design Construction/Fuselage Tech Drawing/Landing gear Wing Structure Construction/Jigs

Table 2.1: Design Personnel and Assignment Areas

2.3 Personnel Assignments, Schedule Control, and Configuration Control

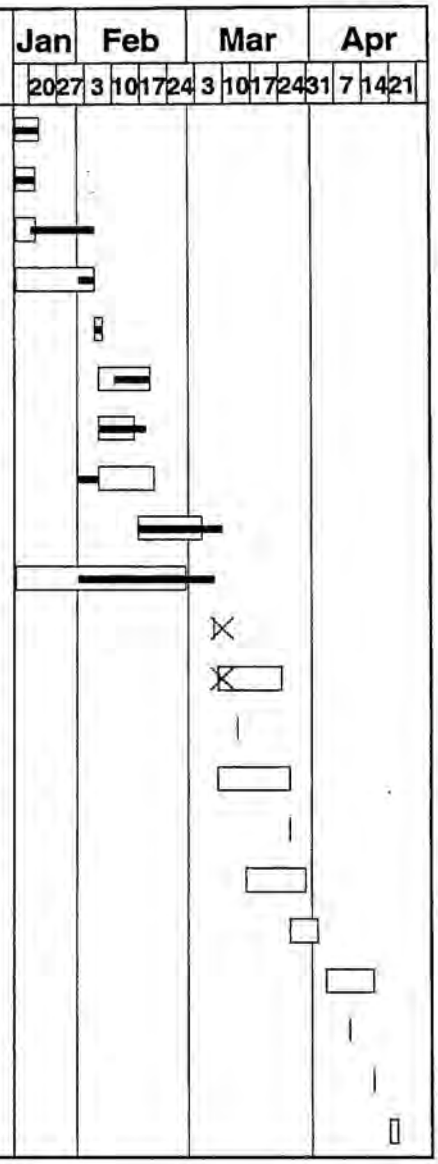
The team, since it was split into three independently function groups, had to be careful to stay on schedule and make the necessary progress. To do this effectively, a few methods were used. The chief engineer developed a milestone chart seen in Table 2.2 to plan out the project. At least once a week, the chief engineer and group leads would to discuss progress and developments. The group leads then each communicated with their groups to remain on schedule. This also controlled the configuration of the plane. With the leads in constant communication no overlapping of design occurred. Any configuration changes were always discussed during these meetings and then worked out between the involved groups, so each group was always aware of the developments of the others.

2.4 Milestone Chart

Table 2.2 documents the actual and predicted timing of the major events in the aircraft design. This table was used as a time management tool to coordinate the actions of the team. The three groups worked together to stay in the expected schedule seen in Table 2.2 to control configuration decisions in a timely manner.

OSU Black Team Milestone Chart

TASK	Planned Start Date	Planned End Date
Conceptual Sketches	1/16/01	1/22/01
Initial Optimization	1/16/01	1/21/01
Prelim. Technical Drawings	1/16/01	2/5/01
Progress Report #1	1/16/01	2/5/01
Configuration Freeze	2/5/01	2/7/01
Progress Report #2	2/6/01	2/19/01
Purchase Propulsion Parts	2/6/01	2/18/01
Purchase Construction Materials	2/1/01	2/20/01
Prototype Construction	2/16/01	3/9/01
AIAA Proposal Phase Draft	1/16/01	3/7/01
Prototype Roll-Out	3/7/01	3/9/01
Prototype Testing	3/8/01	3/24/01
AIAA Proposal Due	3/13/01	3/13/01
Progress Report #4	3/8/01	3/26/01
Final Design Freeze	3/26/01	3/26/01
AIAA Addendum Phase Draft	3/15/01	3/30/01
Final Aircraft Construction	3/26/01	4/2/01
Flight Testing	4/4/01	4/16/01
AIAA Addendum Due	4/10/01	4/10/01
Final Presentation	4/16/01	4/16/01
Competition	4/20/01	4/22/01



Planned Dates — Actual Dates × Actual Date

Table 2.2: Milestone Chart for OSU Black Team

3.0 Conceptual Design

3.1 Mission Analysis and Strategy Development

The mission profile this year for the AIAA Design/Build/Fly contest was to design a plane that can carry two different payloads on two different sortie courses. For the steel payload, or the heavy payload, the sortie is once around the course with a 360^{O} turn in the middle of the sortie. The plane must carry no less than five pounds of steel. For the tennis ball payload, or the volume payload, the plane must fly the course twice without the turn in the middle. The plane could carry no less than ten tennis balls and no more than one hundred. The rules require that the plane be power by an electric motor and have no more than five pounds of batteries. The plane must also take off with in 200 feet and complete all scoring laps within a ten-minute time limit. The wingspan is also limited to ten feet. The current that the motor could draw was limited to 40 amperes. The weight of the loaded plane was set at a maximum of 55 pounds. The flight score for a contest round is the sum of the number of tennis balls carried divided by five plus the weight of steel carried. The round's score is then divided by the Rated Aircraft Cost, RAC. The RAC is a penalty that relates the wing area, the length of the fuselage, the power available to the motor, the empty weight of the plane and the difficulty of construction. These rules and scoring functions set up the initial conditions that the team would use to optimize the plane design.

3.1.1 Pre-conceptual Design

Before any design work began, the aerodynamics group set to work analyzing the mission the contest created. In order to reduce the number of parameters compared and simplify the models for various elements in the RAC and the Score, the optimization model was expressed in terms of eight primary design parameters. Each phase of operation that the plane will see in the contest mission was then modeled. Next, the constraints of operation and contest rules were developed. This mission inspection was improved for each design phase. The model's accuracy and estimates improved as more design details were decided upon. The optimization codes allowed the aerodynamics group to perform sensitivity studies on operational and design constraints. The optimization drove the design through the engineering process and well past the detailed design phase.

3.1.2 Phases of Mission Operation

The aerodynamics group began by defining each phase of operation for the contest mission. A total of three phases were found to be important: takeoff, climb and cruise. These phases of operation are depicted in Figure 3.1. Although the plane only operates in three different phases, the model must account for the two different payloads and sorties requirements. Next, equations for power, velocity, and time were found for each of the flight conditions. All of these equations were functions of the optimization parameters.

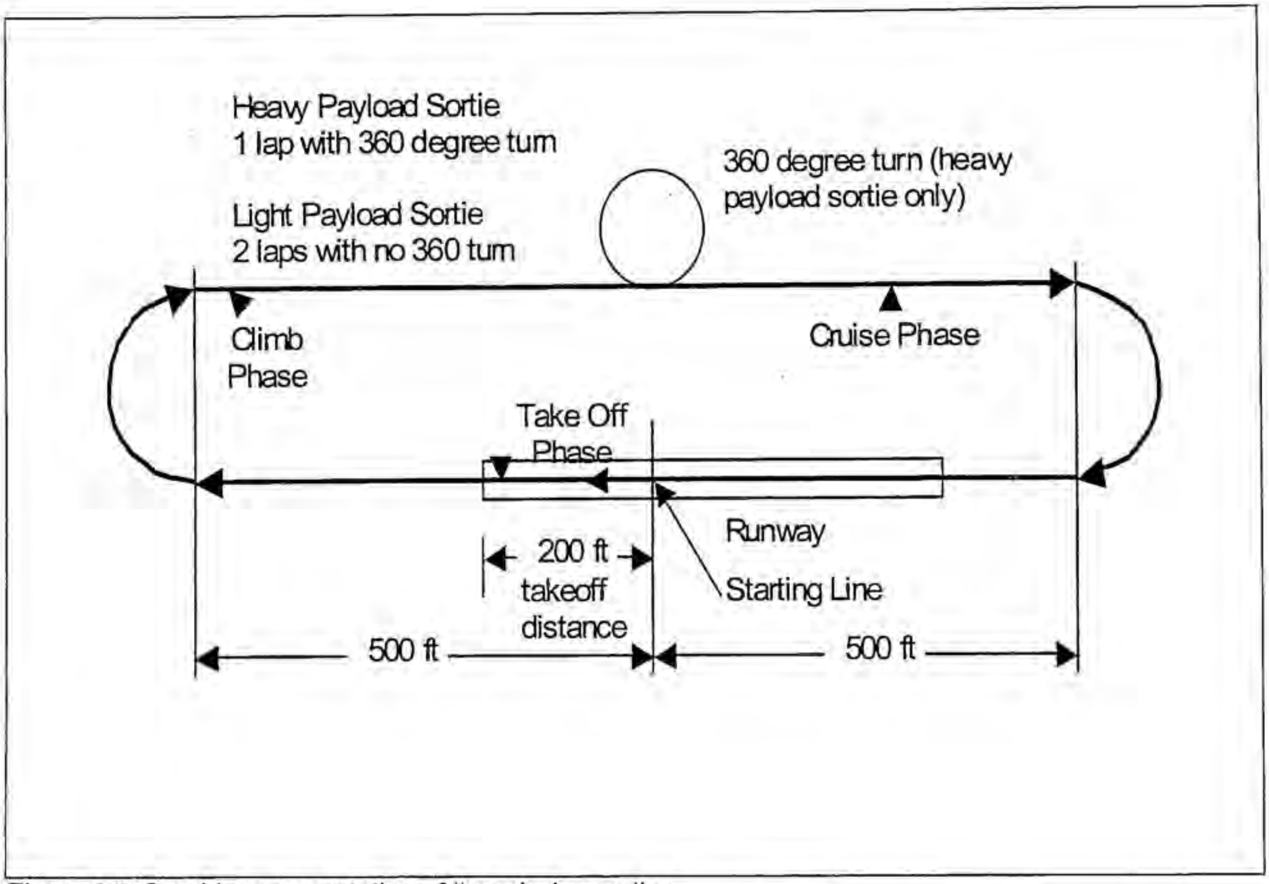


Figure 3.1: Graphic representation of the mission sorties

The first condition of flight was takeoff. The power used in takeoff was assumed to be the same for both the steel and tennis balls. This reduced the workload on the pilot and was found to be reasonable in the final optimal point for the conceptual design. The takeoff was modeled using the equation for takeoff from stand still to takeoff velocity. The model provided a check to make sure the plane could get off the ground within the distance limit and then found the time and energy used to do so.

The next phase of flight was the climb to altitude. Cruise altitude was an operational parameter set by the pilot and designers. This phase model assumed that the plane would continue at the takeoff power and lift coefficient until the cruise altitude was reached. The model of climb assumed that the plane could generate the power to climb, which should be the same as the power needed to take off. The end result of the model was the time required to climb and the forward velocity during the climb.

The last phase of flight was the cruise phase. This is the phase were the plane should spend most of its time during the contest round. The cruise velocity and power were found from the relationship between drag, lift needed to remain aloft, and the power needed to produce that lift. The model found the power used in the cruise phase and the time spent cruising.

To model the time spent on the ground changing payloads and the time spent taxiing from landing stop to the start line, a constant time per sortie flown was assumed.

3.1.3 Primary Design Parameters

To write the optimization code, eight primary parameters of the design were found. These limitations were found to be the core elements of the optimization during the early stages of design. These were not the only parameters, but were judged to be the parameters that mostly heavily influenced the design.

- Planform Wing Area, S: This parameter affects several areas. It and the coefficient of lift determine the amount of lift generated by the wing a given air speed. The wing area sets the weight of the wing. Wing area is also penalized in the RAC formula.
- Payload Weight of Steel, W_{ST}: The weight of the steel directly impacts the overall score of the plane. The weight of steel carried is summed and used as part of the score. In the code, it is used to check that the plane can generate enough power to get off the ground and have enough power to cruise around the cruise of the contest.
- Payload Weight of Tennis Ball, W_{TB}: The weight of the tennis balls effects much more than the score. The number of tennis balls is used to find the overall score, but the volume of the tennis balls has a greater influence. The tennis ball capacity determines the length of the fuselage, which in turn affects the weight of the plane. The weight of the tennis balls also affects the power drain in the tennis cruise laps of the contest run.
- Power to Take Off, P_{TO}: For the initial optimization, the power needed to take off was assumed to be the same for both the steel and the tennis ball runs. It was also assumed that the take off power would be the maximum power the plane ever saw. This value was set to the Rated Engine Power in the Rated Aircraft Cost formula. A limit, defined by the maximum allowable power in the contest rules, was set to constrain the optimization.
- Power to Cruise in the Steel Lap, P_{ST}: The power generated on a steel lap was used to find the energy used for the steel flights and check that the plane was capable of flight for the steel lap cruise.
- Power to Cruise in the Tennis Ball Lap, P_{TB}: The power generated on a tennis ball lap was used to find the energy used for the tennis ball flights and check that the plane was capable of flight for the tennis ball lap cruise.
- Number of Steel Laps, N_{ST}: The number of steel laps was used in the score to find the overall score for the steel payload runs. This number should be as high as possible, but is constrained by the energy stored in the batteries and the time allowed.
- Number of Tennis Ball Laps, N_{TB}: The number of tennis ball laps was used in the score to find the overall score for the steel payload runs. The number should also be as high as possible, but again, is constrained by the energy stored in the batteries and the time allowed.

3.1.4 Operational Constraints

The operational constraints this year were a combination of both the rules of the contest, and the realistic challenges of flying an airplane. The limits imposed by the contest rules include the time allowed, the endurance limits due to battery weight (energy storage capacity), and the take off distance limit. The realistic limits include the allowable range of C_L, the time on the ground needed for payload transfer, and the cruise altitude.

3.1.5 Sensitivity Analysis

Once the optimization code was written and tested, the aerodynamics group determined the sensitivity of the overall score, flight score and RAC, to several factors. The factors with the most importance were the number of laps of the two payloads. This can best be viewing in Figures 3.2 and 3.3. This analysis also found the importance of power available, the lift coefficient, the wind speed, and the drag. Although the drag has less drastic impact than the lift, it does affect the score. The importance of achieving a high lift on take off is offset by the ability to travel the course faster, fitting in more laps. The plane was also designed with a wind of five miles per hour. The wind speed has a drastic effect in the that it can gain three points of overall score for only five more miles per hour of wind speed. This sensitivity lead the designs to favor concepts that pushed the factors in direction with positive score effects.

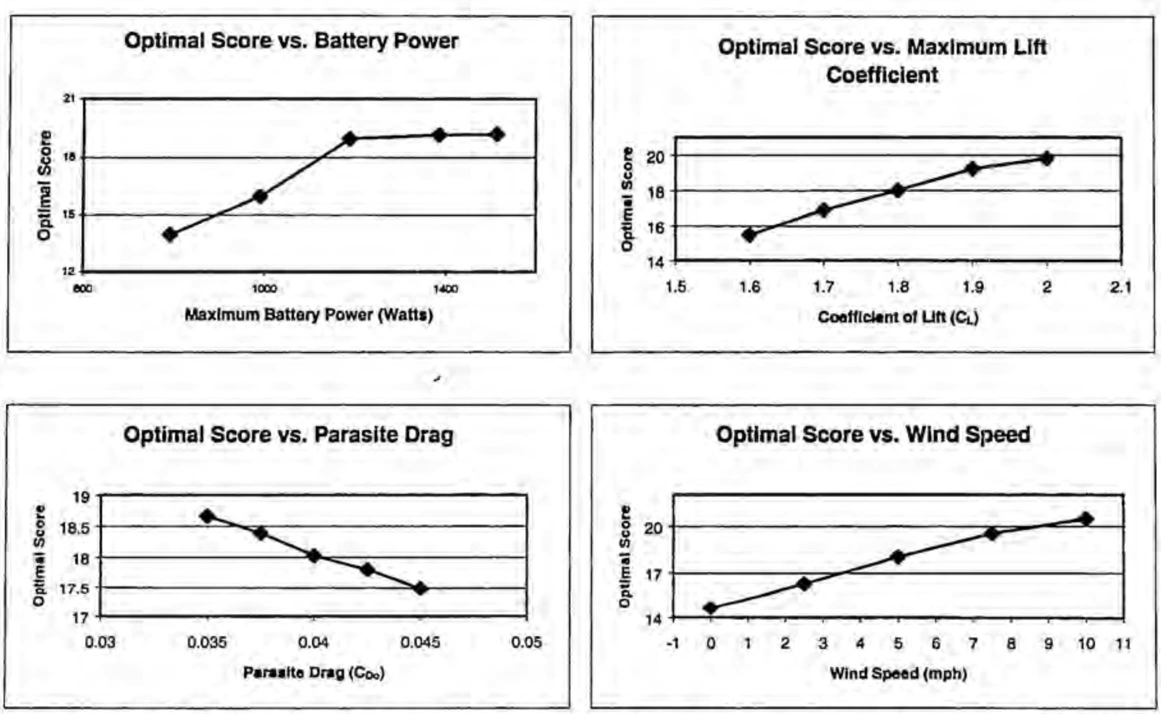


Figure 3.2: The sensitivity of the overall flight score to the Maximum Power out of the batteries, the coefficient of lift, the coefficient of drag and the wind speed.

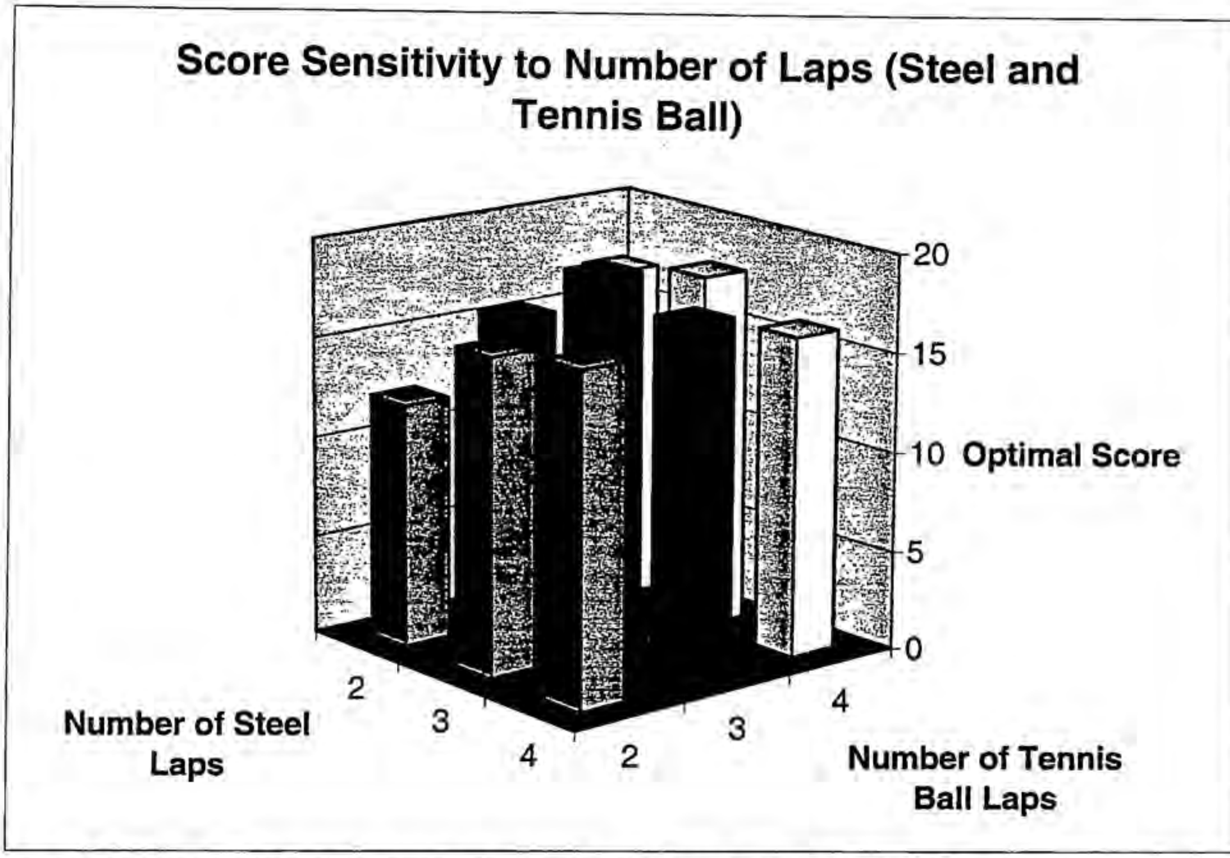


Figure 3.3: The sensitivity of the overall flight score to the number of laps or steel and the number of laps of tennis balls.

3.1.6 Strategy Development

The goal of developing the models of operation and performance for the aircraft was to optimize the overall score function. The code used a conjugant-gradient solving technique to local the maximum of the score design space. To insure that the global maximum was located, the optimization was run repeatedly with randomized initial guesses. The optimization used score as a function not only of the weight of steel and the number of tennis balls, but also of the maximum power used and the surface area of the wing, which are also factors in the Rated Aircraft Cost. The code searches for the optimal point within the design space set by the operational constraints. Once the optimal point was found for the operational and secondary design parameters, the optimization was analyzed for sensitivity to changes in these parameters.

3.1.7 Contest Strategy

The optimization produced a strategy and airplane to be designed by the team. The sensitivity showed some factors to be very important and others not. The code illustrated that a low drag, high lift plane would be required to reach the highest possible score. This data is summarized in Table 3.1

S	9.4 Square Feet	**P_{TO}**	1188 Watts
P_{ST}	669 Watts	**P_{TB}**	670 Watts
W_{ST}	13 Pounds	**W_{TB}**	13 Pounds
C_{LMAX}	1.8	**C_D**	0.04
N_{ST}	3 Laps	**N_{TB}**	3 Laps
Rated Aircraft Cost	5.143	**SCORE**	18.8

Table 3.1: Design Parameters Determined by Strategy Development

3.2 Alternative Concepts Investigated

As the real start of the conceptual design was the development of conceptual sketches by all members of

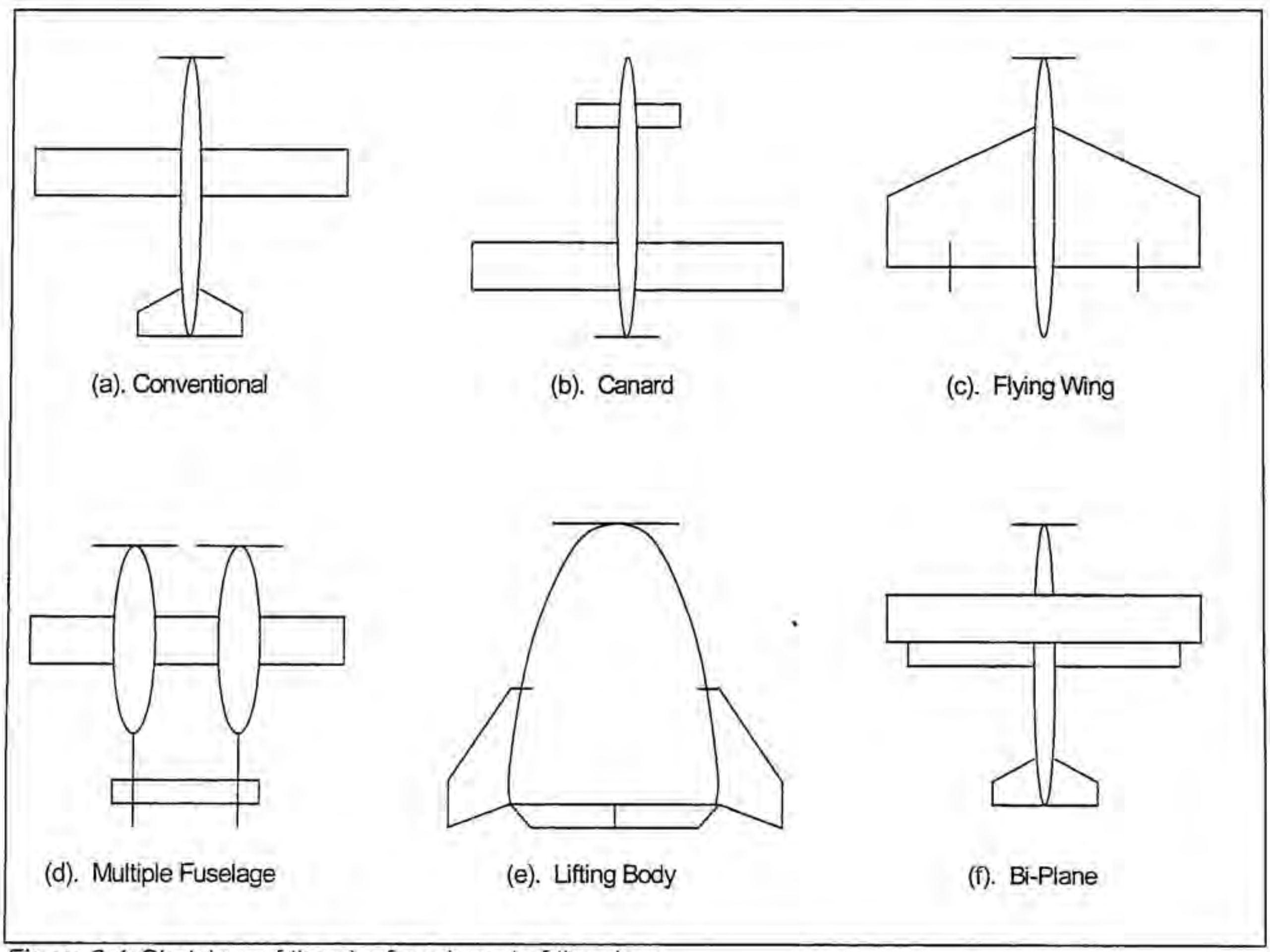


Figure 3.4: Sketches of the planform layout of the plane

the team. After all members submitted ideas for the final plane, the team began debating the merits of each concept and the producing new ideas by synthesis of several concepts and proposal of entirely new ideas. The result was a set of vaible concepts from which the final design would emerge.

3.2.1 Conceptual Planform Layout Alternatives

The following list of alternatives was considered for the general planform of the aircraft. The selection of planform concept affected all the following conceptual decisions. The sensitivity of conceptual design to the overall score was performed as the figures of merit were analyzed in section 3.3. Graphical representations of the concepts can be found in Figure 3.4.

a. Conventional Design: This is the standard configuration of most aircrafts. The wing can achieve maximum lift. The design is easily controlled, and combines low cost, quick construction and intuitive piloting. It has an average Rated Aircraft Cost (RAC). The conventional design is the standard to which the other concepts are compared.
b. Pusher prop with canard: The canard has good aerodynamics, but the wing can never achieve the maximum lift coefficient out of the airfoil for good stall characteristics. The increased weight in the rear due to motor placement also reduces the attractiveness of this design. The RAC is slightly higher than average because the wing area must be large enough to accommodate the lower C_{Lmax} of the canard-wing combination.
c. Flying wing: The RAC of this design warranted further investigation, but the difficulty in construction and control make the concept unattractive.
d. Multiple Fuselage: This design is simple to build, but has a high RAC, with two engines, two vertical tails and three fuselages.
e. Lifting body: This is a great design with very low RAC. Unfortunately, lifting body usually only works at high speeds and has extreme difficulty of control. This design would have a low RAC, the reason it was considered, but the gain in score was of no value since the problems implementation would overcome the advantages.
f. Biplane: This configuration provides a lot of lift, but has high drag and a high cost. The RAC is higher that of most designs due to wing struts and the added cost of having two wings, even if the area is the same.

3.2.2 Wing Placement Alternatives

The wing placement was next set of alternatives to be considered. The planform concept decision is a major the feasibility of all the following options. The selection of the wing placement option was made after the planform decision.

a. Straight Low Wing: The straight wing is the easiest wing to construct. The low placement allows easy access to a speed loader hatch. The low wing does not exhibit stable tendencies about the roll axis, making it unfeasible as a selection.
b. Straight Mid-Wing: This design would also be easy to produce, but the obvious loss of cargo space where the wing carries through is a negative aspect.
c. Low Wing Polyhedral: This wing is more complicated to build that a straight wing, but has the advantage of roll stability and leaving the upper surface of the fuselage open for speed loader access. The multiple joints do make the concept less desirable because of construction reasons and added weight.
d. Low Wing Dihedral: The low wing dihedral is also more complicated than a straight wing. It also has the advantages of roll stability and the open upper fuselage surface. Joining the wings to the base of the fuselage could be a problem, and the angles have to be precise.
e. Straight High Wing: design allows for roll stability and easy construction. The high wing limits the amount of hatch surface available to the speed loader.
f. High Wing Anhedrel: design has the same concerns and advantages as the high straight wing. The anhedral high wing does reduce the roll stability of the aircraft.

3.2.3 Alternative fuselage Concepts

The following three options existed for the fuselage shape: rectangular, hexagonal, and circular/oval. These three possibilities all have their own advantages and obstacles.

a. Rectangular: This design offers the easiest construction and payload configuration. It makes wing attachment easier but weighs more. Aerodynamically the rectangular shape is the worst out of the three because it has the most drag.
b. Hexagonal: A compromise between the rectangular and circular/oval configurations. Weighing less, this shape offers better drag performance than the rectangular shape while still having flat surfaces for easier construction and wing attachment.
c. Circular/oval: This shape has the lowest drag of the three candidates and has the lowest weight. Wing attachment with the circular/oval fuselage will be slightly more difficult.

The most important decision factors are drag and weight, based on the above information the circular/oval shape fuselage was chosen. The length of the fuselage was determined by the optimization and therefore was not addressed in conceptual design.

3.2.4 Alternative Tail Concepts

The aerodynamics and structures group worked together to determine the best tail configuration. The concepts listed here are those that were considered for the conceptual design. It is important to note that the tail decision was dependent on the planform selected.

a. Conventional Tail: This tail is the tail to which all other designs will be compared. Its advantages include simple construction, ease of servo placement, good overall performance.

b. T-tail: The T-tail puts the horizontal stabilizer further out of the prop wash and gives the opportunity to reduce drag due to surface area. The surface area of the tail remains the same as the Conventional tail. But with the relocation of the horizontal stabilizer, the root of the vertical tail has to be reinforced and the installation of the servo for the elevator becomes more complex.

c. Cruciform: The Cruciform tail adds complexity to both manufacturing and servo location. It adds complexity to both control surface implementation and construction concerns. The benefits are minimal considering it does not fully move the horizontal stabilizers out of the propeller wash. Overall this is not an attractive configuration.

d. Twin tail: The twin tail would offer slightly greater control and redundancy, but the increased RAC offsets this advantage. The increase in controls is also unnecessary. A single large vertical tail will likely produce less drag than the twins, since they have twice the number of sharp counters to interfere with the flow and are more area within the prop wash.

e. V-tail: A V-tail would require the same amount of surface area as a conventional tail, while adding complexity in construction and the need to mix the control channels.

3.2.5 Alternative Landing Gear Concepts

The only way that landing gear effect RAC is through the M.E.W. so gear weight was a concern. The major area of concern for landing gear is the load path incorporated with landing.

a. Bow type: A simple design that could be easily attached to the bottom of the fuselage. The load path associated with a bow type gear transmits directly to the fuselage. This is a definite advantage over the other types.

b. Oleo Shock-Strut: A design with one connecting point to the wing. A down side of the oleo is that the single strut must with stand longitude and lateral bending moments when landing, thus translating these loads to the wing.

c. Trailing Link: An effective design that eliminates bending moments in the wing. Some of the negative wing loading would still exist with this design. This design is also more complicated to build and design.

3.2.6 Alternative Propulsion Concepts

The conceptual Design of the propulsion system was driven by the rules of the competition, the rated aircraft cost, and the requirements realized from the score optimization. The rules of the competition restrict the design of the propulsion system to three main areas, motor selection, prop selection, and battery selection. Four different propulsion configurations were considered.

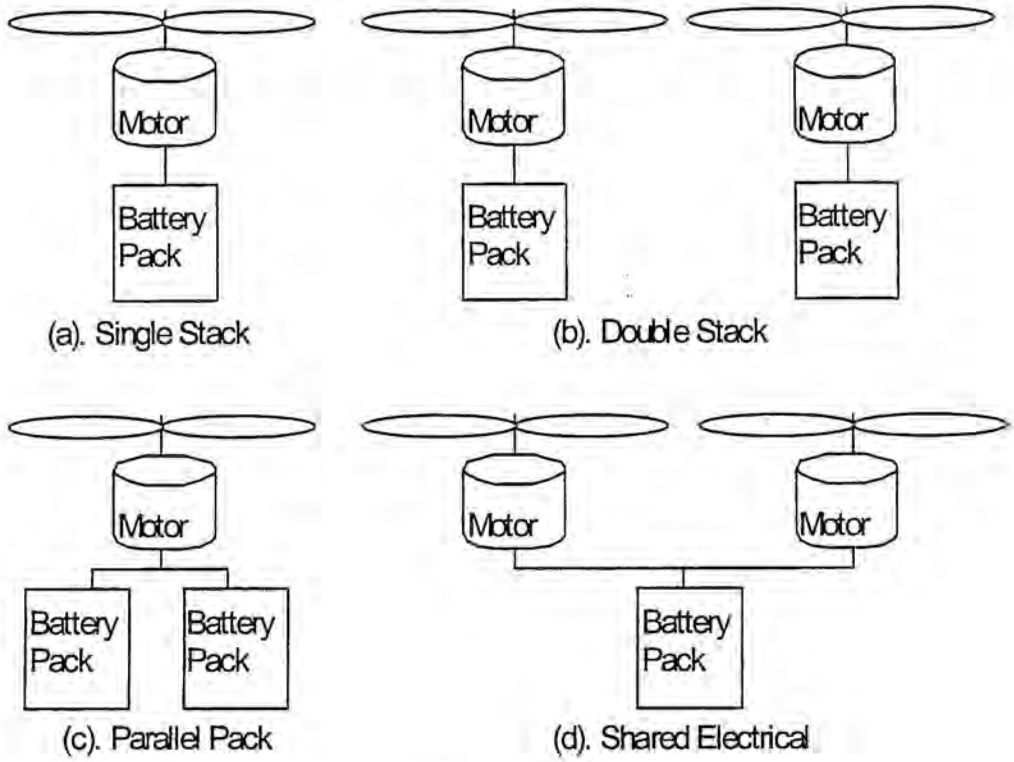


Figure 3.5: Motor-Battery-Propeller concepts investigated

a. Single Stack: A single motor, single battery, single propeller makes up the single stack. This configuration was the most basic choice for a propulsion system. It has the advantages of simplicity and low rated aircraft cost.

b. Double Stack: The double stack consists of two separate battery-motor-propeller systems. This was configuration was found to be a poor performer. The two engines doubled the rated aircraft cost. In addition, separate electrical systems meant two fuses, which again doubled the rated aircraft cost, resulting in a four-fold increase in the rated aircraft cost as compared to the single stack.

c. Parallel Pack: The parallel pack consists of two battery packs wired in parallel to a single motor connected to a single propeller. The primary advantage of the parallel pack is that a lot more batteries can be placed in the battery pack without resulting in a voltage that will damage the motor. Unfortunately, if the individual packs discharge at different rates causing the polarity of one of the packs could be reversed and ruining the entire battery pack. The advantages of wiring battery packs in parallel do not outweigh the disadvantages. There is no difference in the rated aircraft cost between the parallel pack and the single stack.

d. Shared Pack: The shared electrical system consists of two motor connected to a single battery pack. The shared electrical system is much better, in terms of rated aircraft cost, than the double stack but still results in twice the rated aircraft cost of the single stack because of the two motors. Two motors will produce more thrust and power. However, it was thought that enough power and thrust could be obtained from a single stack.

It was determined that all propulsion configurations could produce the power required to take off. Therefore the propulsion configuration was chosen on rated aircraft cost and simplicity. The simplest and the lowest in terms of RAC was the single stack. Therefore, the single stack was chosen as the propulsion configuration.

3.2.7 Material Selection Alternatives

The types of materials to use in the construction were considered. The 2 alternatives discussed here were the 2 that were best suited to size of the plane and the production capacity of the structures group.

a. Balsa and Monocoat: Balsa and monocoat are the conventional construction techniques for small scale aerial vehicles. The construction is straight forward, within the skill range of the structures group and low cost. Plane weight would be higher than other methods increasing the RAC.

b. Carbon Fiber: This method of construction is more expansive and more complicated. The skills to build with this technique were available to the structures group. The final plane will have a lower weight than the conventional construction, with lower RAC, and more durability.

3.3 Figures of Merit and Their Mission Features

Many figures of merit went into the design decisions. The following list of figures of merit were all ranked in Table 3.3:

- Rated Aircraft Cost: The plane was designed with the intent of winning the contest. For this reason, the best combination of RAC and total points was desired. The lower the RAC, the better the score was of the figure of merit.
- Cost: The budget was the limiting factor of the design. All the money used, was received in the form of sponsorship. Designs that can be constructed inexpensively were scored higher.
- Ease of Construction: The structures group must design a method of construction for all parts. Carbon fiber fabric reduces the complexity of most parts, but it requires more lead-time and greater skill. Concepts that are inherently difficult to produce in the workshop received low scores.
- Ease of Design: This figure refers to the knowledge base needed to successfully complete the design. The design is limited to the current understanding of the aerodynamic, propulsion and structures groups. Designs that required intense study of obscure areas of these fields were abandoned as too time consuming for a one semester project. Complicated designs that required knowledge of advanced subjects or operations were scored low.

- Aerodynamic Effects: This FOM accounts for the aerodynamics portion of the design. The design receives a high rating with low drag and high lift. Designs that would have had inherent problems generating lift or generating high drag receive low marks. Those design that are difficult to stabilize or balance also scored low.
- Durability: At the contest the plane will most likely land and take-off well over thirty times. The plane must withstand cyclic loading and stresses for extended periods of time. Designs with areas that are prone to destruction because of fatigue were scored low.
- Weight: The maximum total weight of the plane is 55 pounds. Every pound of empty plane weight was one less pound of payload the plane could carry. In addition, the empty plane weight was penalized in the RAC. Low weight planes received high scores.
- Ease of Repair: Although it is desired for the plane to never crash, planes with good repair characteristics were valued over a craft that is difficult to modify or fix. Aircraft with easy repair and modification earned high scores.
- Ease of Operation (Stability and Control): The plane must be flown by a pilot, placing too much load on the operator will result in impaired performance. A mediocre design that allows the pilot to focus on piloting can work better than a high performance configuration that relies highly on pilot performance. Easily flown crafts received high scores.

Figures Of Merit	Flying Wing, no fuselage	Flying Wing, with short fuselage	Monoplane	Biplane	Multi-fuselage, boom Tail	Lifting Body
Ease of Construction	-1	0	1	1	0	-1
Construction Time	0	1	1	1	1	-1
RAC	1	1	0	-1	-1	1
Durability	1	1	1	1	1	1
Ease of Operation	-1	-1	1	1	1	-1
Cost	0	1	1	1	1	0
Payload Capacity	1	1	0	0	-1	0
Aero Effects (Lift/Drag)	0	1	0	0	0	-1
Repair/Modify	-1	-1	1	1	1	-1
Access to Speed Loader	1	1	1	1	0	1
Total	1	5	7	6	3	-2

Table 3.2: Initial decision matrix for the overall configuration

The initial decision matrix, Table 3.2, was used to remove designs that the team found by simple comparisons to be least feasible, or desirable. The matrix used a simple comparison (–1,0, or 1) to find overall planform concepts that meet best suited the strategy requirements. The results were summed up and the best three of the six compared were analyzed in more depth for the final ranking chart.

3.4 Final Ranking Chart

The initial decision matrix for the planform layout of the aircraft was in Table 3.2. The final decision matrix, Table 3.3, uses a weighted score of each concept for each figure of merit that was found to vary substantially. After the planform layout was determined, the combinations of wing structure and fuselage cross-section were explored.

Figure Of Merit	Maximum Value	Monoplane	Biplane	Flying Wing, Short Fuselage
Maximum Value found by the optimization Routines	There is no "Maximum" Flight Score	18.92	15.9	19.7
Stability and Control Complexity of Design	5	5	5	2
Stability and Control Pilot Comfort	5	5	5	3
Difficulty of Construction, Construction Time	10	10	8	4
Total	20 + 19.7	38.92	33.9	28.7

Table 3.3: The final decision matrix for the plane configuration

The best configuration was found to be the conventional monoplane. The monoplane was easy to build and easy to control. The flying wing with a short fuselage design was found to be difficult to design and a high load on the pilot, who must also manage the power during flight. The Biplane design had potential due to increased lift, but the higher RAC, second wing penalty, reduced the scoring ability too much.

3.5 Features of The Final Conceptual Configuration

The final configuration of the aircraft was chosen based on the decision matrix in Table 3.3. The best description for the plane is a low polyhedral wing, circular cross-sectional fuselage with a conventional tail. The plane will use tricycle landing gear of either oleo or bow type. This configuration meets the requirements of the optimization, has a low weight estimate, and meets the demands of construction and time. The two-piece wing design has two semi-spans that will be built, permanently bonded together, and connected to the bottom of the fuselage by shear pens. Both of the semi-spans will have the polyhedral angle built into them, eliminating the need for secondary connections at these points. Also the connection to the fuselage would be simplified because the wings would not connect to the fuselage at an angle as with pure dihedral. This perpendicular joining design offers the simplest method of connection. As a result, it leaves the largest room for error in construction. The circular fuselage cross-section was chosen based on information from past experience and newly available tools. The standard configuration tail was chosen to reduce weight, keep low RAC and avoid control surface actuators placement problems.

4.0 Preliminary Design

As the preliminary design began, the focus of the optimization shifted from finding the best combination to improving the estimates and data for the optimization code. The propulsion group began to develop more accurate methods for finding power and the true energy storage of the batteries. The structures team developed better plane weight models and requirements for payload weights. As well as laying out the internal structure of the plane, and the volume needed to contain the tennis ball payload were refined. The aerodynamics group refined estimates of the drag coefficient and began a search to find airfoils that could meet or exceed the lift requirements found from the optimization. The aerodynamics group also began control surface sizing, found flight derivatives, and made initial center of gravity calculations.

4.1 Design Parameters Investigated

Moving from conceptual to preliminary design the parameters investigated changed slightly. The conceptual design decisions gave guidance to a refined set of parameters. The weight of tennis balls carried added a cargo volume, and the demands of the propulsion system helped define the battery, motor and propeller. The weights of the payload helped the structures group develop stress analyses for the fuselage and landing gear.

- Payload Weight, W_{st}, W_{tb} : Normally, the maximum payload weight would be considered a figure of merit. The optimization, however, used the payload weights to determine the best strategy. The weight of payload that can be handled was then a parameter for which the plane was designed rather than a positive feature of a potential design.
- Cargo Volume: Again, this parameter could be a figure of merit. The optimization found that the best score could be achieved with a plane carrying the maximum number of tennis balls; therefore, plane was designed to carry 100 tennis balls.
- Power – Take off and Cruise, P_{to}, P_{st}, P_{tb}: These factors were found to be important in reaching the optimal score. It was found from the optimization that power management is critical for a high score. The plane must be designed with the ability to hit the optimal power use at both the take off and cruise conditions.
- Surface Area of the Wing, S: This parameter was used to find the lift of the wing and minimize the Rated Aircraft Cost score. The optimization program determined the smallest wing that can generate the needed lift a takeoff within the limits of the lift coefficient. The lift coefficient was varied to find the sensitivity to the total score and the wing area.
- Lift Coefficient: For the optimization to produce results, a C_L must be supplied. Using experimental results for known sources and previous experience, a reasonable estimate to the maximum lift coefficient was determined. A sensitivity analysis to this factor was performed, and the C_L was found to be an important score factor.

- Drag Coefficient: This factor became important, as the speed of plane is critical to reaching the planned strategy. The right combination of coefficient of lift and drag is a goal of the preliminary design.
- Number of Laps, Endurance: The strategy developed from the returns of the optimization dictated the number of laps of both the heavy and light payload laps. Achieving both goals is necessary to reach the optimum score.

4.1.1 Operational Constraints

Due to the contest rules, the strategy and aerodynamics the following constraints were enforced.

- Cruise Altitude: This value was set by consolation between the designers and the pilot. The altitude of 70 feet was found to be acceptable to both.
- Time on Ground: This is an estimate of the time the plane is on the ground, including landing rollout, cargo exchange, and time needed to taxi to the starting line. This time was estimated from previous competition experience.
- Distance of Cruise: The contest officials set this value. The length was padded to allow for error.
- Runway Limit: A value of 200 ft was set by the competition rules, for the optimization a value of 180 ft was used to leave room for error.
- Battery Weight and Endurance: The weight of the batteries is another operational constraints set of the contest. The endurance of the batteries must be enough to hit the optimal score.
- Completion of planned strategy within time limit: Another design consideration is that the plan is able to complete the contest course with the intended number of laps of each payload with the allotted time.

4.2 Figures of Merit

Several figures of merit were used for the preliminary design. The following list describes each figure of merit and their respective missions features.

- Score Effects: This FOM is also a design parameter, but falls under the figure of merit as well. Every aspect of the plane takes into consideration the effect it will have on the overall score.
- Durability: In order to win with the current contest strategy, the plane will have to land at least 18 times before its finished. The plane will also be designed for transportation to the contest as well. As such it must be able to survive being dissembled and reassembled repeatedly.
- Ease of Modification of Design: The prototype airplane will be tested to reveal design defects and inadequacies. Design that allows the modification will be preferred over other designs.
- Weight: Each part will be considered for its individual weight penalty. The overall weight of the plane should be the optimal value or less.
- Ease of Construction: The difficultly in construction of complex fuselages, wings or tails was weighed. Designs that require skills or tools unavailable to the structure team were disregarded.

- Aerodynamic Effects: Designs with high lift and low drag were considered. For this contest, it was of great interest to reduce drag.
- Stability and Control: Designs with inherent control problems were valued less when compared to designs with well-known stability characteristics.
- Access Points for a Speed Loader: The different designs were compared for areas of access to the speed loader. Designs with ease and quick access to the cargo compartment were preferred.
- Cost: This is the one figure of merit that can place an absolute cap on a design. The cost of the projected design was considered in the materials selection and in the inherent cost of a design.
- Efficiency: The maximum motor-battery efficiency is obtained by multiplying the motor efficiency by the battery efficiency. The battery efficiency is found by taking into account the losses due to the internal resistance of the batteries. The motor efficiency is found by taking into account the copper and iron losses in the motor. A higher battery-motor efficiency results in a higher score.
- Total Energy: The total energy of the batteries is a measure of the total energy stored in the batteries. Sensitivity analysis shows that that an increase in total energy storage results in a higher flight score. The total energy does not have a direct effect on the rated engine power score.

4.3 Analytical Methods, Expected Accuracy, and Reasoning

During the design phases the team used both analytical and experimental methods to find results. The aerodynamics group used several analytical techniques to define the characteristics of the aircraft. The propulsion group used wind tunnel and static dynamometer test to characterize the propulsion system.

4.3.1 Analytical Methods

For the initial optimization, the aerodynamics group used simple models for take off thrust, take off distance, and climb rate. The steady and level flight of the cruise phase was model by solving the equilibrium equation between drag and thrust. As the optimization was refined with better data from the structures and propulsion groups, the equations were improved. The take off function more accurately modeled the thrust from the propeller. The drag estimate was refined from first order estimates (guesses from experience) to calculations from charts available in Raymer's *Aircraft Design, A Conceptual Approach*. The accuracy of drag predictions is difficult to quantify, as drag is difficult to model.

When the stability calculations were started the aerodynamics team chose to use the stability estimates in Nelson's *Flight Stability and Automatic Control*. The linearized flight equations of motion assumption were found to hold true for the team's design. The assumption of only small disturbances was valid, as the plane was designed as a cargo transport. The control surface estimates held under the same assumption. The approximations of the modes of flight vary due to how well the small disturbance theory holds. The estimates for the Short Period, Roll and Dutch Roll are known to vary no more than five percent with in the assumption envelope. The Phugoid motion can be inaccurate to about 30 percent, however the period of the Phugoid is large enough that the sortie will finish before the Phugoid can diverge. The Spiral estimate can be inaccurate as well. The Spiral mode is one that the pilot will damp out naturally by his control of the plane. If the prototype proves to be susceptible to any divergence, the aerodynamics group will reanalyze the aircraft and redesign to damp out the mode of motion.

4.3.2 Experimental Methods

Extensive experimental tests were conducted to determine the best propulsion system. The primary tool for testing was a dynamometer built for testing scaled propellers. This dynamometer could measure the thrust of the propeller, the torque of the propeller, and the rotational speed of the propeller. The second most important tool was an experimental quality wind tunnel. The wind tunnel has a 9 square foot cross sectional testing area and is capable of producing wind speeds of up to 120 ft/s depending upon ambient conditions.

Tests were conducted under static conditions, with the dynamometer outside the wind tunnel, and dynamic conditions, with the dynamometer in the wind tunnel. This data was then non-dimensionalzed to obtain the coefficient of thrust, the coefficient of power, and the efficiency versus advance ratio curves. In addition to this, extensive testing was done on last year's motor and battery pack. Thermocouples were used to measure the temperature of the batteries and the motor. The power, amperage, and watt-hour reading of the batteries were also recorded. The performance of the batteries and motors was tested under static and dynamic conditions at various discharge amperages. A test was also conducted to determine the effect of a blockage placed behind the propeller. This testing resulted in a marked improvement in this year's propulsion system.

4.4 Configuration Sizing Data

4.4.1 Airfoil Selection

The optimization assumed a maximum coefficient of lift, which was used during take off and climb. The optimization determined the lift coefficient at cruise for both payloads for velocity and power requirements. These considerations drove the airfoil selection. The first step in the airfoil search was to find candidates with sufficient maximum lift coefficient. It was assumed that the lift coefficient during take off is slightly less than 1.5, which is eighty percent of the maximum lift coefficient of 1.8. During cruise the lift coefficient varies from 0.7 to 0.9. The wing chord and airspeed yield a Reynolds number of 300,000. This search produced several possibilities. The candidates were screened to further reduce the number of possible airfoil.

In the optimization a total parasite drag coefficient of 0.04 was assumed. Because of other contributions such as fuselage, landing gear, tail, etc., the portion of the drag allocated to the wing/airfoil is less than

0.02. Many drag polar curves were compared while looking for reliable experimental data. When possible, airfoils with maximum lift coefficients, even those greater than 1.8, were analyzed. Airfoils that were unable to exhibit reliably predictable relationships between lift, drag, and stall point were removed as too great a risk. Furthermore, this extra lift will be useful especially in the prototype, which is error-prone, and might need more lift during take off. The airfoil that satisfied these requirements was the Selig S1210. Its characteristic drag polar is in Figure 4.1.

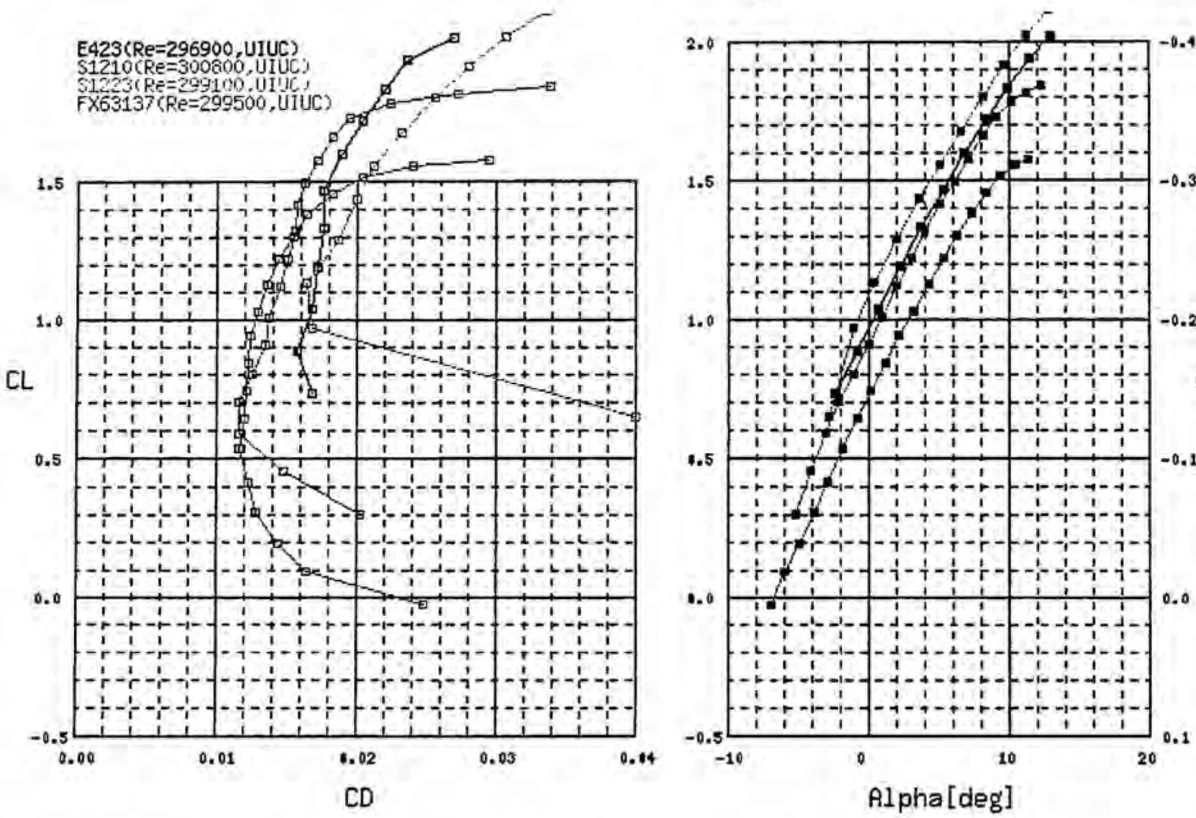


Figure 4.1: Drag polar Comparing four candidates for wing airfoil; the S1210 was selected due to its range of operation and low drag characteristics.

4.4.2 Pitch Stability

The calculations of the pitch stability determined the size of the vertical tail and its incidence angle. Aspect ratios of 4 and a taper ratio of 0.5 for the tail were chosen. These values correspond to the desired tail shape and rules of thumb from literature sources. This reduced the tail sizing variables to two unknowns, the root chord of the horizontal stabilizer and the coefficient of lift of the tail, C_{Lt}. The first calculation performed was the fuselage contribution to the pitch stability using an equation summing the contribution of each section of the fuselage to the overall stability.

Next, the non-dimensional location of the neutral point was found. The center of gravity was known from initial estimates and the position of the payload center of gravity. The stick-fixed static margin was used to find the tail chord and coefficient of lift necessary to achieve the desired pitch stability.

In order to find the tail chord and C_{Lt}, the stick-fixed static margin was set at 15% of static stability. This was a large value, as it was preferred to have a strong stability for the prototype. Literature on the subject stated a value of 0.05 to be adequate. The value would be reduced if the flight-tests showed it to be acceptable.

The next step was to find the necessary contribution of the tail to pitch moment coefficient at zero angle of attack, C_{Mot},. such that the plane is stable at the cruise angle of attack, α_{cruise}. To find the need downward force from the tail the moment must be found from all other sections of the airplane. Adding together the pitch moment from the wing, fuselage and thrust yields the total moment that the tail must counter. The airplane pitch coefficient at zero pitch angle was found to be a function of the tail's lifting force, even though it is directed down. With the force from the tail known, the pitch stability contribution of the other parts was found and added together to find the airplane's pitch moment coefficient versus pitch angle derivative, $C_{M\alpha}$. This value must be negative in order to achieve stability. With the values of C_{Mo} and $C_{M\alpha}$ known the pitch moment can be expressed as a function of the angle of attack and the force generated by the tail.

Our wing cruise angle of attack was –0.85 degrees, so C_M was set to be zero at that angle, in order to have the desired trim during cruise flight. The tail characteristics for a tail at an effective angle of attack of zero were found. The coefficient of lift for a tail with an aspect ratio of 4 was found to be 0.35. The size of the tail root chord was set to 9.5 inches. The selection of the horizontal stabilizer's airfoil is discussed in section 4.4.6.

The tail chord was oversized to 10.5 inches for two reasons. The first was to have a bigger security margin in the pitch stability. The second was to allow the elevators to be oversized as well. This increased the static margin to 0.206.

4.4.3 Yaw Stability

The size of the vertical tail was dependant upon acceptable yaw stability. With the shape of the fuselage and the wing already known their contributions to directional stability were calculated. Since the vertical stabilizer must be symmetric, the NACA 0009 airfoil was selected, as there is abundant information about the airfoil available. The vertical tail contribution was found to be a function of the vertical tail surface.

It was known that to have stability it was necessary to have a positive $C_{N\beta}$. The Literature suggested a $C_{N\beta}$ of 0.05 for the cruise velocity Reynolds number. A vertical surface of 1.13 square feet was found to be adequate to meet stability requirements. To account for any errors a 1.25 feet square vertical tail surface would be used.

4.4.4 Roll Stability

A similar approach as that used for yaw stability was used to find the polyhedral angle of the wing. The contribution of the low wing (without the polyhedral effects) and the fuselage was calculated. This value was found to be destabilizing as the low wing and fuselage work to promote instability. Then the contribution of the polyhedral angle of the wing (Γ) was calculated as a function of this angle. The wing had a polyhedral angle only toward the tip, from 2 to 5 feet on each semi-span.

Adding the coefficient of roll moment from the wing and body together resulted in the roll moment coefficient versus yaw angle derivative for the total airplane as a function of Γ, so setting this coefficient to a $C_{l\beta}$ of –0.05 multiplied by the maximum lift coefficient, easily found the polyhedral angle. An angle of 4.735 degrees was found and rounded up to a 5 degrees polyhedral angle.

4.4.5 Horizontal Tail Plan-form Layout Selection

When choosing the tail, various options for the plan-form layout were considered. The options included the straight tail, the straight tapered, swept tapered, and the straight swept. It was decided that the swept tapered style best fit the rest of the design requirements. This decision was based on the following criteria: weight, drag, ease of construction, and rated aircraft cost. When determining the best selection for the weight criterion, the weights of various options were computed. These weights were based on similar surface areas. The options ranked in as follows:

1. Straight Tail .374 lbf
2. Swept Un-tapered .374 lbf
3. Swept Tapered .44 lbf
4. Straight Tapered .44 lbf

The difference in weight is from the fact that, for a longer chord, more foam is required for the thickness needed.

When determining the best alternative in terms of drag it is necessary to look at the causes of drag on the tail surface. When trying to reduce drag, eliminating any portion of the tail that would cause flow separation was important. Gapless hinges were required to minimize the drag. The straight tapered tail would force the use of gapped hinges. This would be a high drag configuration. The swept un-tapered configuration required multiple control surfaces and would therefore raise rated aircraft cost. In the other two design configurations it was possible to use gapless hinges for the control surfaces. Overall, these two styles were preferred. The swept tapered was preferred over the straight tail because of the "Oswald Efficiency Comparison." The swept tapered had an Oswald Efficiency of .95 where as the Oswald Efficiency of the straight tail was .75. For the given performance characteristics, the swept tapered configuration was chosen. We decided to implement horns into the tail control surfaces in order to reduce the strain on the servos used to drive them. By creating a force balance between the horn and the control surface, the surfaces could be moved with less effort, making them more efficient.

4.4.6 Horizontal Tail Airfoil Selection

After strenuous consideration using a decision matrix, it was decided that adapting the A18SM (Archer) airfoil for the airplane was the most suitable. The selection of airfoil was based mainly on the aerodynamic considerations. The figures of merit (FOM) for the airfoil selection were the following: Cruise lift coefficient at zero angle of attack, cruise drag coefficient at zero angle of attack, thickness/weight, lift coefficient range.

First, based on the stability calculations, the tail required a lift coefficient of 0.35 at zero angle of attack. Only a few airfoils met the requirements. The airfoils were then compared to for drag characteristics. Airfoils were compared for lowest drag at zero angle of attack and for the range of low drag. As speed was critical to the optimal strategy, reduced drag had a high priority.

Weight of the tail was one of the critical aspects of design. The weight of the tail had two effects. The first was an increase in empty weight of the plane, which drove down the overall score. The second was center of gravity effects, because larger tails shift the center of gravity toward the rear of the plane. Weight was influenced by the maximum thickness of the airfoil. The cross-sectional area and volume of foam would be higher as the maximum thickness increases, therefore, the lower the maximum thickness of the airfoil, the more favorable it was to the overall plane design. Two candidates met these requirements, the A18SM and the BE50SM.

FOM	A18SM	BE50SM
Thickness/Weight	1	1
Cruise CL @ alpha=0 deg	1	0
Cruise CDo @ alpha=0 deg	1	1
CL Range at min CD	1	1
Total Score	4	3

Table 4.1: Decision matrix for the horizontal tail airfoil

The final condition was the operational lift coefficient range of the airfoil. Looking only at the difference between the minimum and maximum value was not enough. As mentioned earlier the airfoil would be operating at a lower lift coefficient region so A18SM was chosen for the horizontal tail. A comparison matrix between the two best options for the horizontal stabilizer airfoil is included in Table 4.1.

4.4.7 Flight Stability Derivatives

The following values are the flight derivatives of the preliminary plane design. Table 4.2 shows the longitudinal flight stability derivates, and Table 4.3 the lateral. Table 4.5 shows flight control derivatives in both longitude and latitude. A static margin of 0.204 was used, due to an upsized tail.

Wing		C_{Low}	0.731
C_{Mow}	-0.246	C_{Mac}	-0.227
$C_{L\alpha w}$	4.639	$C_{M\alpha w}$	-0.123
Fuselage			
C_{Mof}	-0.041	$C_{M\alpha f}$	0.589
Tail			
C_{Mot}	0.09	$C_{M\alpha t}$	-0.952

Table 4.2: Flight Stability Derivatives: Longitudinal, Static Margin: 0.206

$C_{N\beta fw}$	-0.031	$C_{N\beta v}$	0.097
$C_{N\beta}$	0.066	$C_{l\beta wf}$	0.011
$C_{l\beta d}$	-0.085	$C_{l\beta}$	-0.075

Table 4.3: Flight Stability Derivatives: Lateral

Longitudinal
$C_{M\delta e}$= -0.785 1/rad
Lateral
$C_{N\delta r}$=-0.035 1/rad $C_{l\delta r}$=0.021 1/rad $C_{l\delta a}$=0.097 1/rad $C_{N\delta a}$=-0.014 1/rad

Table 4.4: Flight Stability Control Derivatives, Static Margin: 0.206

4.4.8 Fuselage Shape Consideration

The payload of tennis balls is the driving parameter in determining the volumetric size of the fuselage, a study was conducted to see how ball configurations and the number of tennis balls for a given design affected two parameters, weight and fuselage drag. Figure 4.2 shows the different ball configurations considered including the configuration that the team chose, the Black Pack. It is shown only as the 100 ball configuration.

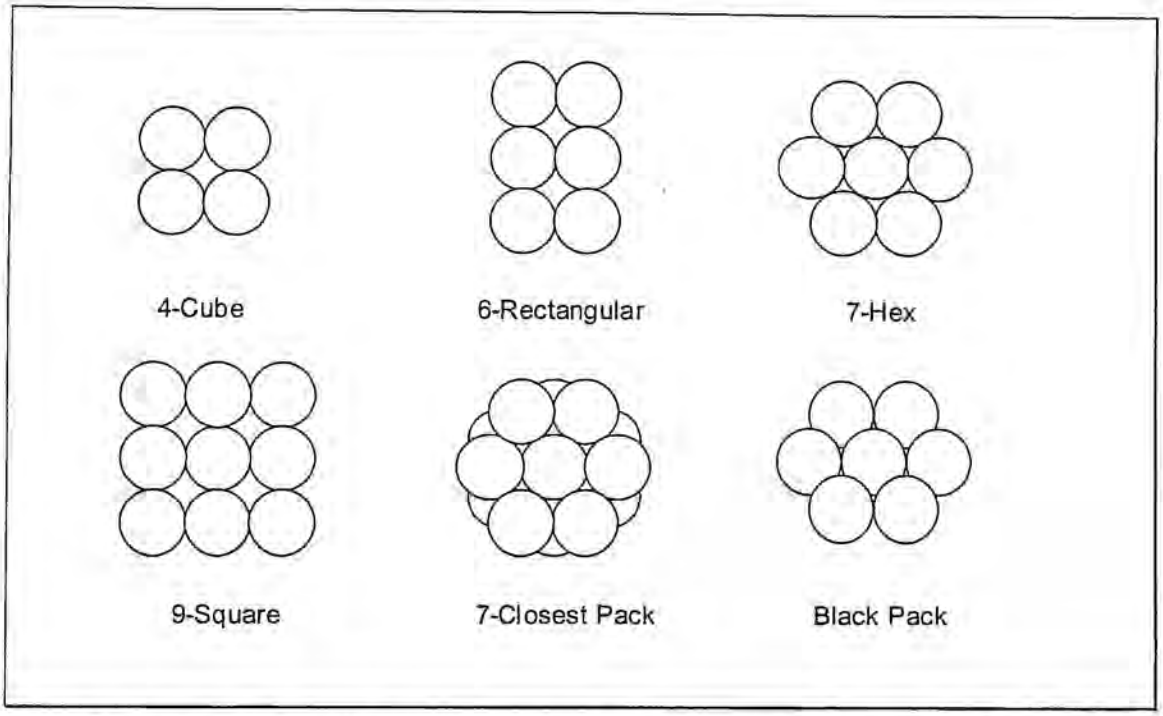


Figure 4.2: Fuselage tennis ball cross-section configurations considered in the trade study

From the team's aircraft optimization, the optimal tennis ball cargo stayed between 70 and 100 tennis balls for a payload. The trade study examines only this range for each configuration. The results of the skin weight portion of the fuselage configuration trade study are shown in Figure 4.3. The fuselage weight chart shows that as payload size increases, so do the fuselage size and the fuselage skin weight. The fuselage skin weight is based on the fuselage being constructed out of a carbon fiber/Styrofoam core sandwich. This construction technique weighs half as much as conventional construction. One result from the data is that the square and rectangular fuselages are not as light as rounded fuselages.

The fuselage drag coefficient, which was calculated using methods from Raymer (1999, p. 342-346), is nondimensionalized in terms of the wing area. An assumption made by the equations is that the entire fuselage experiences fully developed turbulent flow as a result of propwash. The results of this part of the trade study are shown in Figure 4.3. The results show the rectangular fuselages at real disadvantage with respect to the more rounded fuselages, also, the shorter length gained by the 7-Closest Pack configuration is better than the standard 7-Hex configuration.

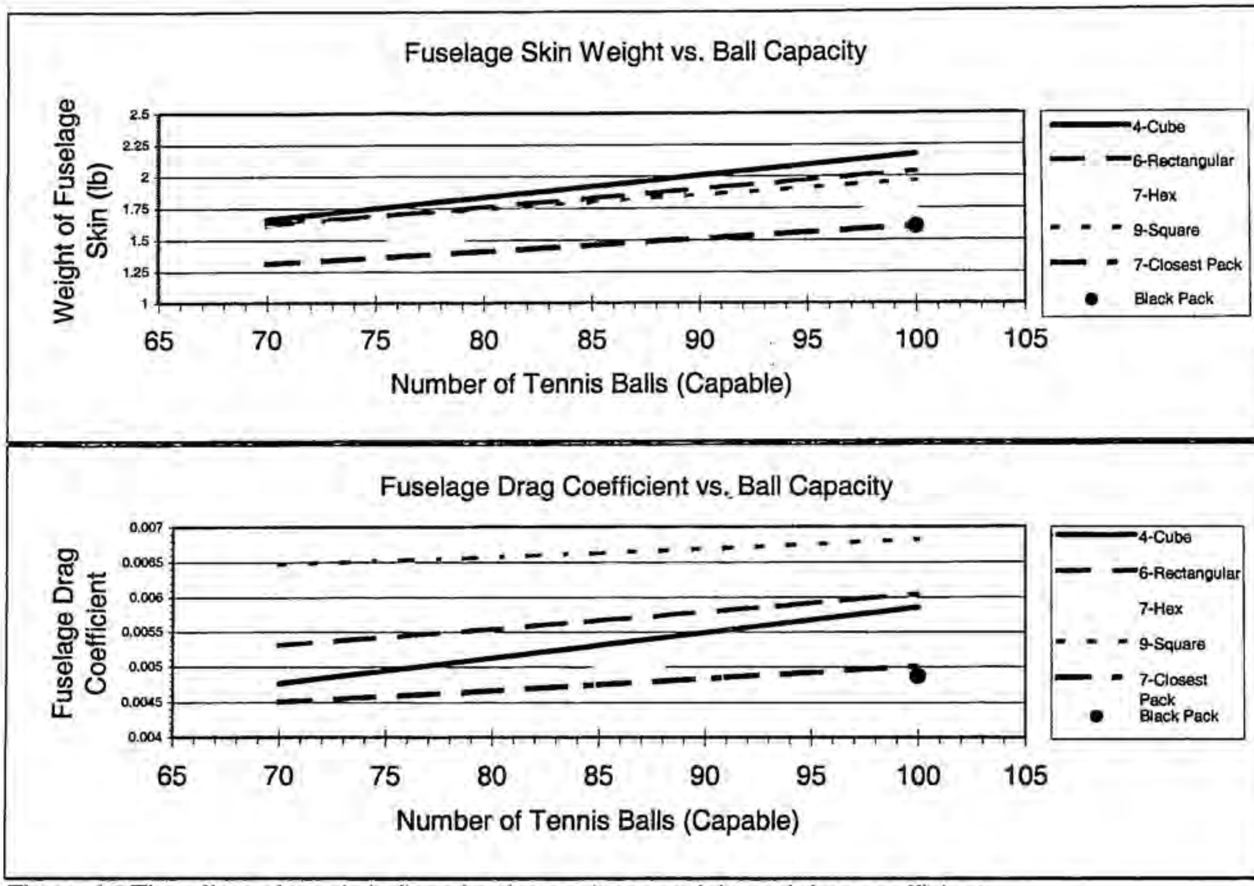


Figure 4.3:The effect of tennis ball payload capacity on weight and drag coefficient

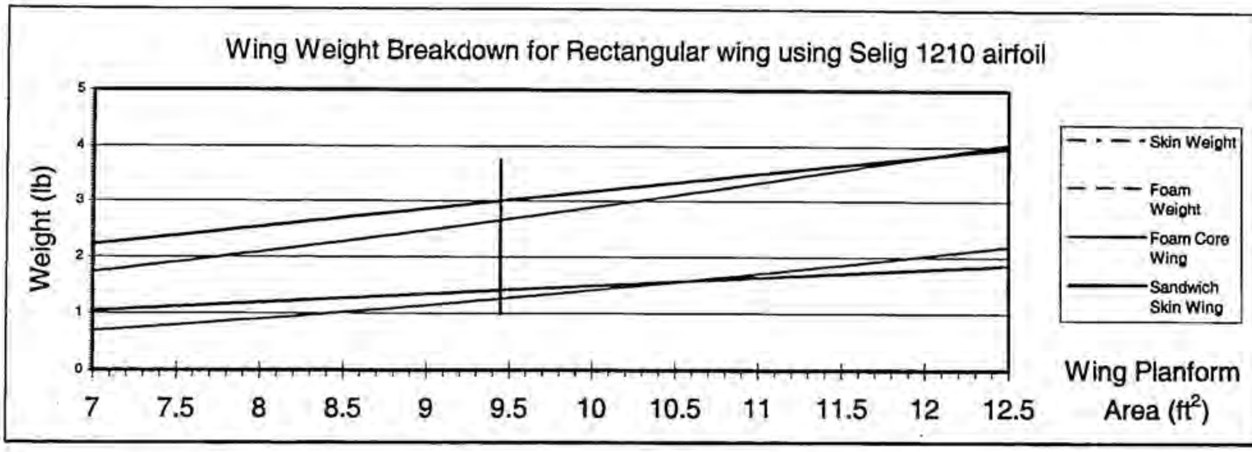
From this study, the team chose another variation in the 7-Hex configuration, the Black Pack configuration, which is a slanted back 7-Hex cross section. Like the 7-Closest Pack configuration, the Black Pack is a two-dimensional closest pack arrangement. This configuration was chosen over the 7-Closest Pack configuration because it eased the placement of batteries. Thus, it is no surprise that the 7-Closest Pack and the 2-dimensional closest pack Black Pack have nearly the same fuselage skin weight. Whenever the drag data from the two bodies were looked at, the similarities disappeared. The team's preliminary fuselage design has a slight advantage in drag coefficient due to its more economical use of fuselage volume for payload with respect to the 7-Closest Pack configuration because it incorporates a more efficient use of nose and tail space than the standard configurations.

4.4.9 Wing Structure Design

Two trade studies also resulted for the wing. The first was the wing construction study, which looked at the weight from the two leading candidates for construction. The first was a foam core construction with carbon fiber skin. Second, was a carbon fiber, foam, carbon fiber sandwich construction Figure 4.4 shows the varying of wing weight with respect to the wing area for a wing span held at a constant 10ft. For the optimized wing area value of 9.4 ft^2, the foam core wing came in 0.3 lb lighter than the sandwich.

For the Selig 1210 airfoil, the sandwich only begins to be viable at a wing area greater than 12 ft^2, and that is assuming that the sandwich skin is stiff enough carry the aircraft without an internal wing structure. From this trade study the team decided on using the foam core wing construction.

A second wing trade study preformed looked at the effect of wing taper on wing weight, assuming that the tapered wing and rectangular wing generate the same amount of lift. Figure 4.4 shows the wing dimensioning definitions used in the trade study. The foam core Selig 1210 airfoil was used to find the weights. The graph clearly shows that the rectangular wing maintains the lowest wing weight. This is a result that the wing foam weight increases as a result of the square of the chord. The weight saved at the tips is overshadowed by the weight increase at the root of the wing.



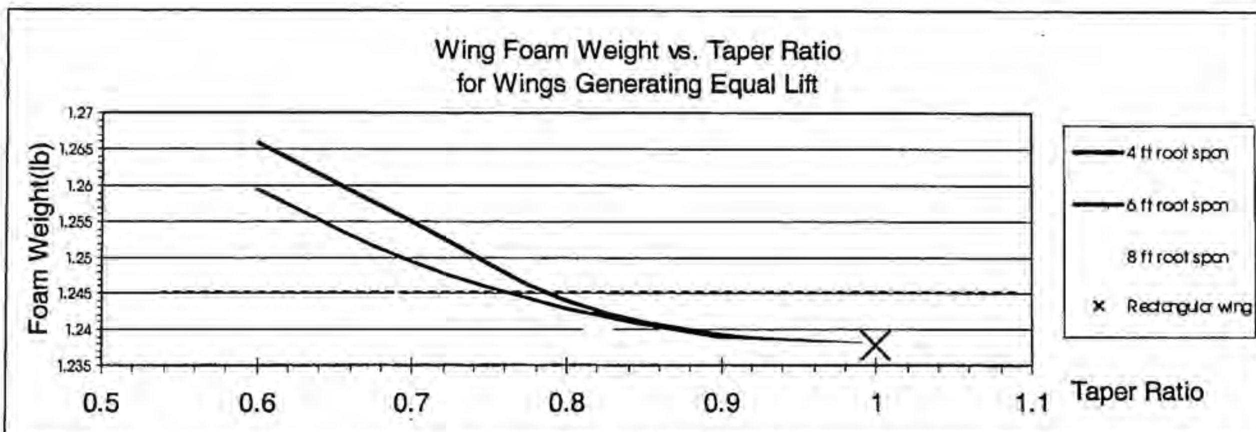


Figure 4.4: Results of the wing taper weight study. Each root span weight series is graphed with respect to the varying taper ratio.

With the method of wing construction decided a stress analysis was preformed. The maximum wing loading will occur during the tip test when the plane is placed on a test stand supported only at the tips of its wings. The highest stress levels occur at the wing roots. With an 150% load of 85 pounds, the stress on the top of a 1/8 in. thick main spar is 481.6 ksi. For reinforcement a second spar was added in the wing. To account for the high stress, the spar will be strengthened with a 3-ply C-channel reinforcement about the root of the wing with 0.4 in flanges. The maximum stress at the root of the wing with just the C-channel reinforcement would then be 87.3 ksi, less than one-fifth of the previous value. The most common carbon fiber failure mode is delamination from a second surface and buckling. Since the C-channel is one piece, that cannot happen.

4.4.10 Landing Gear Design

Several designs were considered to try to effectively transmit the loads induced while landing to the main structure of the plane. Options for a landing gear type that connected to the wing included connecting the gear to the main wing spar or building a load distribution system into the wing. While providing an effective path for the loads these designs could not alleviate some of the local problems such as crushing of the foam and delamination of the wing skin. For a bow gear these problems do not exist. A bow gear design was originally thought to be heavier and to produce more drag than a wing-mounted design.

After further research and investigating new construction techniques a solution was reached. A carbon fiber bow gear would be used. Properly designed this type of landing gear could solve all of the design challenges. The advantage to this designed lied in the fact that the carbon fiber arms of the bow could be made to any shape desired. This meant an aerodynamic, low weight design could be implemented. Realizing this technique allowed the team to create efficient main landing gear.

The nose gear presented a different challenge. Ground handling requirements dictated that the plane have steerable front landing gear. Another design requirement was to use front brakes on the plane, to make more efficient use of the time on the ground. With these requirements in mind an oleo type gear was chosen. To lighten the weight of the nose gear a carbon fiber tube was used for the main strut.

4.4.11 Motor and Battery Selection

The first task of propulsion preliminary design was to conduct an exhaustive search for different types of batteries. Information was compiled on approximately 135 batteries. Previous OSU planes used the Sanyo KR-2300SCE battery. These batteries were built to have an extended life and be used for very low amperage discharge rates. Experimental results showed that they had a very poor performance at a high amperage discharge rate. This degradation in performance can be avoided by using a high discharge type battery. It was also felt that a quick charge battery is needed to effectively compete in the contest. The type of application for which each battery was designed was also investigated. The high performance, "racing" batteries were all designed for applications that drained the voltage at a high current. For these reasons, a choice was made to consider only the quick charge, high performance, high discharge, racing batteries. This choice left 15 high charge/discharge, "racing" batteries for consideration. The four batteries that gave the highest total energy were selected as the top performers. The decision matrix in Table 4.5 was used to pick the best battery out of the top four performers.

The batteries were chosen on the basis of the total energy provided by all the batteries allowed by the 5 lb limit and the overall battery efficiency. These figures of merit are listed at the left of the matrix. The value for the total energy for the SR-2400 was significantly higher than any other battery. In addition it had the second highest battery efficiency. The SR-2400 was the top choice heading into the motor selection phase.

		Batteries		
Figures of Merit	SR-2400	SR-2000	SR-3000	N-RC2400
Total Energy (Kwatts-sec.)	435	372	415	373
Battery Efficiency (%)	90	90	92	86

Table 4.5: Battery selection decision matrix

The next step was to choose the motor. According to the rules, the motor must be a brush type motor. Oklahoma State University has a long relationship with Astroflight. Based on past experience, they have the best motors on the market, and worked well with teams from OSU. Astroflight is located in California, while Graupner is a German company. Choosing a motor from Graupner would add logistical concerns. In addition, Astroflight has always given OSU a substantial discount off the retail price of their motors. Astroflight was chosen as the manufacturer to purchase from.

Astroflight produces two series of motors, the "60's" and the "90's", which can operate at the planned voltages and currents. When comparing motors, the two important parameters are the mechanical power out of the motor and the motor efficiency. The mechanical power and motor efficiency for the different motors with the SR-2400 battery were compared in Figure 4.5. The 661 motor was a clear winner for both mechanical power out of the motor and the overall motor-battery efficiency. Figure 7 also shows that with the 661 motor, the highest mechanical power out of the motor is achieved with the SR-2400 battery. Thus the Astroflight 661 motor and the SR-2400 battery were chosen for our propulsion system. Having chosen a battery and motor, propeller and gearbox selection were next studied.

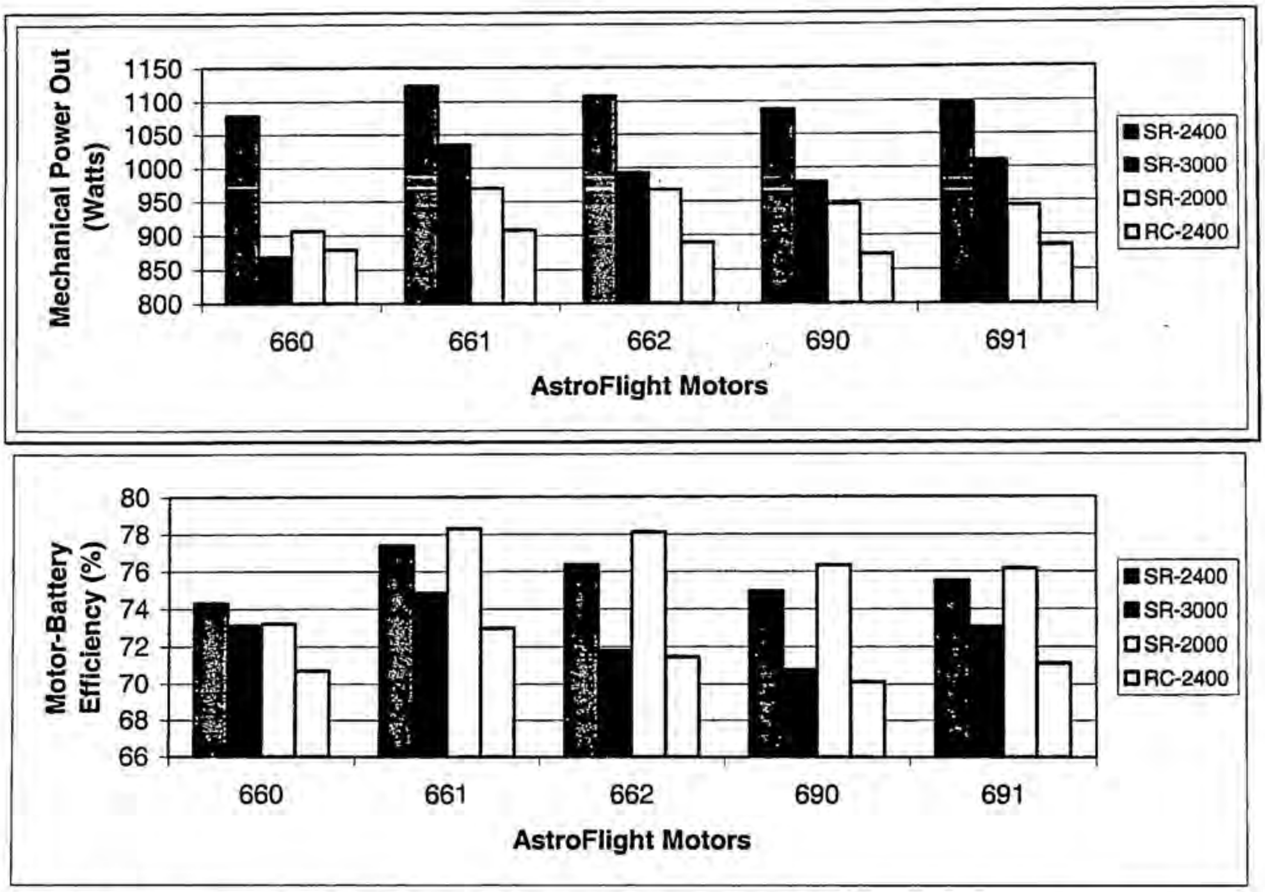
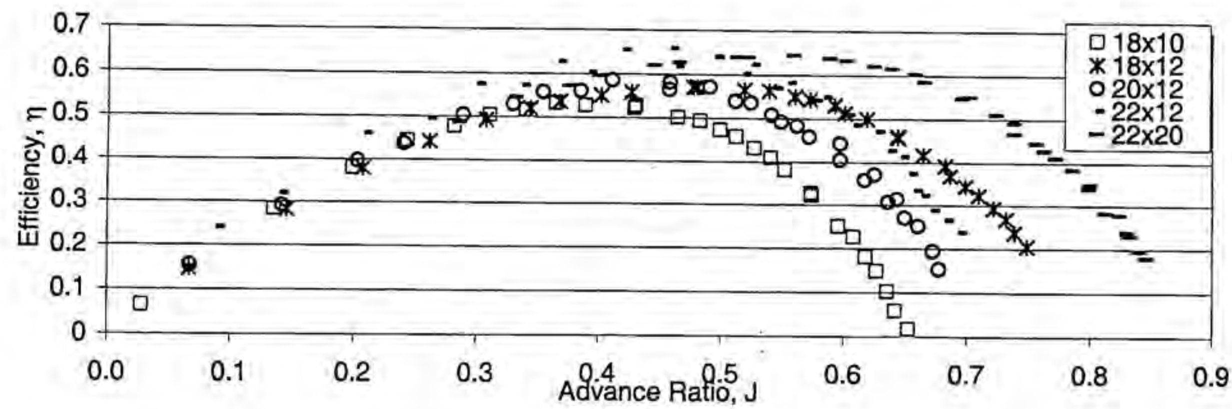


Figure 4.5: Mechanical Power Out of the Motor for Different Motor Battery Combinations (Top) and Motor-Battery Efficiency for Different Motor Battery Combinations (Bottom).

4.4.12 Propeller Selection

There were six main parameters for propeller selection--propeller size, propeller pitch over diameter (P/D) ratio, thrust from the propeller, torque from the propeller, power drawn by the propeller, and motor current used by the propeller. Extensive tests were conducted by performing experiments on twelve different propellers using a dynamometer and a wind tunnel to determine these six parameters. Data gathered from experimental tests was non-dimensionalized as four parameters--the coefficient of thrust (C_T), the coefficient of power (C_P), the overall propulsive efficiency (η), and the advance ratio (J). Graphing C_T, C_P, and η versus J gives definitive performance data of each propeller as shown in Figure 4.6. In addition, second-degree polynomials were fit to the C_T and C_P graphs. These polynomials were added to the computer models of the propulsions system. This allowed an accurate simulation of our propulsion system.



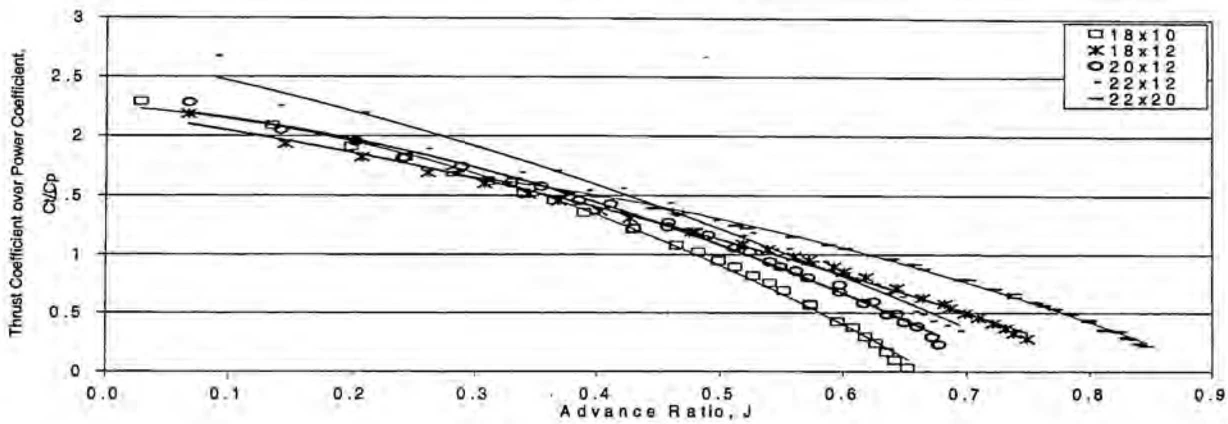


Figure 4.6: The overall efficiency of the propulsion system versus the advance ratio (Top) and the ratio of the thrust produced by the propeller to the power drawn by the propeller versus the advance ratio (Bottom)

Using the characteristic data from each propeller, the propeller size and the pitch over diameter ratio (P/D) were then iterated, to find the best propeller for the flight characteristics desired. Based on the optimization of the overall score, a 22-inch propeller was chosen.

Computer models, based on the experimental data, were then developed to give the performance of the propeller and the current drawn by the motor. The current drawn by the motor was used to estimate battery endurance. Using these models, iteration was begun to find the find the best gear ratio for the 22-inch propeller. That gear ratio was found to be 2.7:1.

4.4.13 Overall Configuration Sizing Data Summary

The preliminary design produced a plane that will reach the contest strategy requirements. The preliminary design is summarized in Table 4.6.

Wing Airfoil	Selig S1210
Wing Area	9.4 ft^2
Wing Span	10 ft
Wing Chord	11.28 in
Wing Angle of Incidence	0^0
Aspect Ratio of the Wing	10.68
Aileron Area	0.705 ft^2
Polyhedral Angle	5^0
Horizontal Tail Airfoil	Archer 18 (A18SM)
Horizontal Tail Area	1.75 ft^2
Horizontal Tail Root Chord	10.5 in
Horizontal Tail Span	31.5 in
Horizontal Tail Taper Ratio	0.5
Horizontal Tail Angle of Incidence	0^0
Elevator Area	0.45 ft^2
Vertical Tail Airfoil	NACA 0009
Vertical Tail Area	1.25 ft^2
Vertical Tail Root Chord	12 in
Vertical Tail Span	17 in
Vertical Tail Taper Ratio	0.5
Rudder Area	0.32 ft^2
Fuselage Length	71.6 inches (6 feet)
Battery	SR-2400
Number of Batteries (in 5 lbs)	37
Motor	AstroFlight 661
Propeller Diameter	22 inches
Gross Take Off Weight, no cargo	18.25 lb.
Gross Take Off Weight, Volume cargo	31.25 lb.
Gross Take Off Weight, Heavy Payload	39.25 lb.

Table 4.6: Configuration Sizing Data

4.5 Features that Produced the Final Configuration Selection

During the preliminary phase many of the details of the plane's configuration were finalized. An Astroflight 661 motor will power the plane, with a power source made up of 37 cells of SR2400 NiCad batteries. To propel the plane, a 22-inch propeller will be used. The wings of the plane will be of polyhedral form using a Selig S1210 airfoil. The ailerons will be powered by 50 oz-in servos, in order to control the plane. The tail of the plane will be made with an Archer 18 (A18SM) airfoil in the horizontal direction, and an NACA 0009 airfoil in the vertical direction. The plane body will be built primarily out of foam-supported carbon fiber. The main landing gear will be bow type and built using carbon fiber. The nose gear will be a steerable aluminum/carbon fiber Oleo strut design.

5.0 Detail Design

5.1 Performance Data

With the tail and control surfaces sized, the aerodynamics team began to investigate the modes of motion and general dynamic stability of the airplane. Other values that had been included in the optimization, but never explicitly stated are include here as well.

5.1.1 Take Off Calculations

During the optimization, the distance and time required to take off was calculated. The calculations took into account the ground effects and estimated friction forces on the wheels. After the design was finalized the take off calculations were checked for the performance unloaded. The values are summarized in Table 5.1.

Take Off Condition	Distance (feet)	Time (seconds)
Heavy Payload (21 lbs)	161.0	6.6
Volume Payload (13 lbs)	92.7	4.3
No Payload	24.0	1.5

Table 5.1: Take Off Values

5.1.2 Handling Qualities

Once the static stability was set, the team began the dynamic stability and modes of motion calculations. The Eigenvalues of the modes can be seen in Table 19, and the modes of motion are summarized in Table 20. The Short Period motion was found to have a period of 0.832 seconds with a damping ratio of 0.536. The Phugoid motion was found to have a period of 9.037 seconds with a damping ratio of 0.041. The Dutch Roll motion was found to have a period of 1.589 seconds with a damping ratio of 0.39. The rules of thumb from literature predicted that these values would be acceptable in flight

	Eigenvalue
Short Period	-4.0 ± 6.3i
Phugoid	-0.03 ± 0.69i
Roll	-17.7
Spiral	-0.01
Dutch Roll	1.544 ± 3.64i

Table 5.2: Eigenvalues

Mode of Motion	Frequency (Hz)	Damping Ratio	Period (seconds)
Short Period	7.55	0.53	0.83
Phugoid	0.69	0.04	9.03
Dutch Roll	3.95	0.39	1.58

Table 5.3: Motion Modes

5.1.3 Range

Since the plane was design to fulfill a specific mission profile, the range of the plane is best expressed in the number of laps of each type that can be completed. The plane is designed to handle 3 laps of the heavy payload, steel, and 3 laps of the light payload, tennis balls. The total distance flown should be approximately 24000 feet or 4.5 miles.

5.1.4 Endurance

Similar to the Range the endurance of the craft is best expressed in terms of the scoring laps that can be completed. Since the optimal score point is found by using all the energy in the batteries in the exact limit of the contest run time, the plane is designed for 10 minutes of flight.

5.1.5 Payload

The payload configuration of the plane was one of the primary design parameters during conceptual and preliminary design. For the volume payload, the plane was structured around a tennis ball configuration that contained the 100 tennis balls of the optimal score value. For the heavy payload, the structures teams designed to the maximum weight allowed under the contest rules, 55 pounds. This will allow the team to vary the amount of steel carried based on the conditions at the contest site. The tennis ball configurations can be varied as well, allowing the team to pursue any strategy within the limits of the plane's performance characteristics. The estimated Gross Take Off Weight, GTOW, for the tennis ball lap is 31.25 lb. The GTOW for the steel is estimated at 39.25 lb.

5.2 Component Selection and Systems Architecture

The configuration of the plane can be divided into two main sections: propulsion and structures

5.2.1 Propulsion Component Selection and System Architecture

As mentioned previously, the SR-2400 battery was chosen as the best battery for the flight conditions. After talking to SR, it was decided to buy pre-manufactured matched packs. Each packs consists of 37 cells and is separated into a front pack, consisting of 27 cells, and a rear pack, consisting of 10 cells. This pack configuration fits around the wing carry through structure under the payload area. The 37-cell pack gives a nominal operating voltage of 44.4 volts. The optimization called for take off current of 30 amps and cruise current of approximately 20 amps. Based on experimental results of last year's battery packs, it is felt that seven or eight take offs can be achieved. The motor chosen was the AstroFlight 661 motor. The factory-supplied gearbox of 2.7:1 will be used. The plane will have a cruise speed of 80 ft/sec so a high P/D ratio propeller was chosen. A higher P/D ratio results in peak efficiency at higher speeds. A 22-inch propeller was chosen.

5.2.2 Structures Component Selection and System Architecture

The Structural system of the plane consisted of the fuselage, wings, tail and landing gear.

The fuselage is designed to carry the required payload while weighing as little as possible. To do this it made the most efficient use of space to maintain a short length and small volume. The fuselage must also support the aerodynamics and propulsions systems. The shape of the fuselage was also designed to reduce frontal surface area and therefore drag.

The wings will be made of Selig 1210 airfoils; they are equipped with ailerons to control the plane while turning. Each wing also contains one main spar that was designed to with stand the tip test imposed at the beginning of the competition.

The tail horizontal the Archer 18 (A18SM) airfoil, and the vertical tail the NACA 0009 airfoil. The Archer airfoil was used to reduce the size of the tail required for the plane, this aided in both weight and drag reduction.

The final structural component is the landing gear. The demands of the landing gear are high. It was designed to withstand the rigors of landing the plane while being as light as possible. The final landing gear design has considered all the possible loads that could be induced upon landing had how to properly transmit these loads to the structure of the plane.

For a visual of the systems architecture please see the drawing package.

5.3 Drawing Package

To view the drawing package, please turn to the last five pages of the report.

5.4 Innovative Configuration Solutions, Manufacturing Process, Cost Reduction

Many of the design decisions were made to reduce cost, and are discussed in their respective sections. The rules of the contest actually aide in reducing cost since the plane is penalized for size and smaller planes are cheaper. Also, the more efficiently the plane was the built, the cheaper it should be. Most of the innovative design decisions had to do with the manufacturing process and can be seen in the following list.

- Ball Configurations: The seven ball hexagonal form was slanted to further reduce frontal area of the plane.
- Fuselage construction: The bottom of the fuselage was laid up as one piece for structural purposes. The wing carry through was also included in the fuselage mold so that it could be simultaneously laid up to reduce stress concentrations at the connection.
- Female molds were used to get a smooth surface on the outside surfaces of the aircraft.
- The polyhedral was laid up as one piece and baked at the correct angle into the wing. The ability to use the same mold for both sides guarantees the angles for both the right and left sides of the wing should be identical.
- The structural bulkheads and longerons were designed and built around the batteries to make efficient use of the space.
- Moving some of the tennis ball payload into the nose and tail of the plane reduced fuselage length.
- Carbon fiber bow main landing gear with a nonrectangular cross-section was implemented to reduce weight. Similarly a carbon fiber strut was used on the nose gear also to reduce weight.

All of these innovative configurations helped to make the manufacturing process more efficient or the plane less expensive.

6.0 Manufacturing Plan

6.1 Manufacturing Processes Investigated, Cost and Skill Required

Several methods of fuselage mold preparation were researched. For any of these methods, a master mold had to be made to either the dimension of the inner mold line (IML) or outer mold line (OML) of the aircraft. There were three processes investigated for making a master mold: Computer Numeric Controlled (CNC) machining a block of tooling dough/foam, wire cutting sections of foam, or carving tooling dough by hand.

By CNC machining a master mold, excellent tolerances and smooth surfaces could be achieved. This method would be very expensive because a CNC was not available to the team. This then would require subcontracting to make the molds. This makes the availability of this process very low.

The method of foam wire cutting had advantages in that the foam was inexpensive, and the skill level matched that of the team. The foam must be cut in sections using parallel templates. The parallel templates made it difficult to align all the sections and produce a blended shape. The cost of the foam was much less than that of the tooling dough, and CNC machining. This method also required less time to manufacture since foam sheets were readily available from local suppliers.

The carving of tooling dough by hand had advantages in that it is cheaper than CNC machining. Also, it can be made in one piece making the master more rigid than foam. Tooling dough was more expensive than foam, and the geometric tolerances would be difficult to maintain. Also, the team had no experience with tooling dough. After comparing the candidates with the decision matrix seen in Table 6.1, the decision was made to make the molds out of foam.

Figure of Merit	CNC	Tooling Dough	Foam
Cost	3	3	4
Skill Required	1	1	2
Availability	1	2	2
Total	5	6	8

Table 6.1: Manufacturing Decision Matrix

The next manufacturing decision involved two competing carbon fiber lay-up ideas. The team could either make a thin female mold out of fiberglass off an OML master, cut the OML master down to an IML master and lay-up the carbon fiber fuselage on an IML master plug with the flexible OML mold bagged to the outside for surface finish, or make a rigid female mold off an OML master and lay-up carbon fiber on the that OML female.

The OML female is the industry standard for many parts. For this purpose, the female mold should make the outside smooth to reduce drag. An OML female mold required additional backup structure in order for the mold to hold its dimension under vacuum.

During the detailed design phase, it was decided to use a foam core for much of the fuselage skin. This core was a thin cut off the OML foam master. For the structure the most critical dimension was the IML because speed loaders and batteries must fit precisely. Thus, an IML plug allowed the pieces to be laid-up from the inner dimension out. The designed 1/8-inch foam core between the outer and inner fuselage skins would be difficult to maintain precisely. Using an OML female mold, any core dimension errors would affect crucial internal areas. In the end, it was decided to lay-up carbon fiber on the male IML foam plug to save cost and time.

6.2 Process Selected for Major Component Manufacture

After the detail design was complete, final manufacturing plans were laid out. The team decided to use carbon fiber 2X2 3K twill as a sandwich with primarily foam core for the entire airplane. In order to use either pre-impregnated carbon fiber or a wet-lay-up fabric, a mold must be used to acquire the correct shape. The blended fuselage design required proper mold planning in order to achieve a light and durable structure and a smooth outer surface finish. The fuselage design involved four separate parts--bottom fuselage, motor cover, speed loader hatch, and upper aft section.

In mold design it is crucial to be able to remove the mold after curing it. Proper mold design and release techniques were planned as methods for this reason. For each of the fuselage parts, a separate mold was required. In order to secondary bond surfaces, a peel-ply material was purchased for use in the lay-up. Proper release films, breather, bagging films and sealants were all acquired for carbon fiber lay-up.

6.3 Individual Processes Investigated and Figures of Merit Used

For the manufacturing process, the figures of merit used were the main components of the plane. Each component, along with an investigated manufacturing process is discussed in the sections below.

6.3.1 Fuselage Process

The foam for the fuselage masters was to be first cut into accurate blocks of foam by gluing pieces together and wire cutting the blocks to dimension. By using templates cut to dimension and a rotating wire cut, rounded foam sections were used. These rounded sections were aligned on a tooling board. OML flexible fiberglass females were laid and cured on the OML plugs. An IML plug was aligned on a tooling board using the reduced sections. The fuselage parts were made by laying up carbon fiber, foam core, carbon fiber, and the flexible female molds on the IML plugs, and then vacuum bagging the entire lay-up.

6.3.2 Wing and Tail Process

The wing molds were easier to design since foam core was chosen for the airfoils. When cutting out the wing with templates, female cradles were formed simultaneously. To obtain a smooth finish on the wing, proper treatment of the foam cradles was important. It was decided to coat the cradles with a thin epoxy resin and wet-sand the surface until smooth. To increase efficiency, one female cradle was designed with the capacity to cook both wings using the same mold. In order to get the polyhedral bend in the wings, a five-degree foam angle block was cut and glued to the cradles. The wing spars were made in three parts. A "C" spar was laid up on a precision-machined piece of oak mold. Two additional rectangular spars were also added. Servo wire slots were cut into the wings with a hot wire.

Like the wing, the horizontal and vertical tails we made by cutting a foam core and female cradles. They were also composed of a single layer of carbon fiber on the foam core.

6.3.3 Landing Gear Process

The bow gear mold was made again with foam. The surface finish was acquired by applying epoxy and sanding both top and bottom mold surfaces. Several sets of bow gear were planned for spares in case of failure. The nose gear was machined out of aluminum and bonded with a carbon fiber tube.

6.4 Analytic Methods Including Cost, Skill Matrix, and Scheduling

As discussed previously, each of the different manufacturing methods had different costs, time to order and receive, and skill required. To CNC the molds would require a lot of time just to make the molds. This was time that would be better spent actually making the carbon fiber parts. Making the molds with tooling dough would require a higher skill than the team had available and would cost more. Foam cutting would requires a lot of time and the necessary tools, but fortunately OSU owns the tools needed.

6.4.1 Manufacturing Schedule

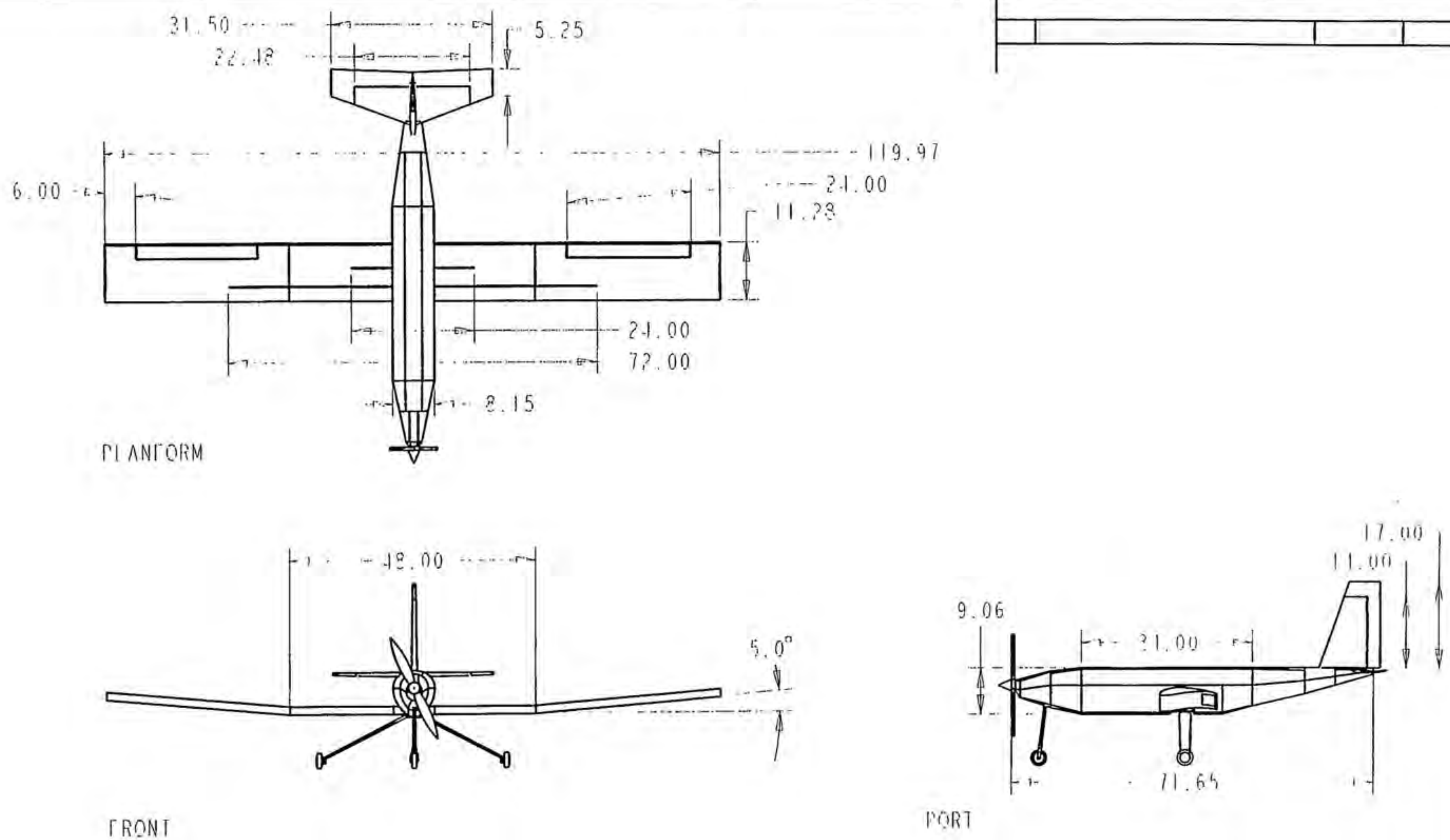
3/9/01

Task	Start	End	2001 February	2001 March
Purchase Supplies	2/5/01	2/7/01		
Female Wing Mold	2/17/01	2/17/01		
Fuselage Templates	2/21/01	2/21/01		
Speed Loader Templates	2/21/01	2/21/01		
Foam Fuselage Molds Cut	2/22/01	2/23/01		
Speed Loader Molds	2/22/01	2/22/01		
Cut Thin FOam Cores	2/23/01	2/24/01		
Surface Prep of Fuselage	2/24/01	2/24/01		
Mahcine Nose Landing Gear	2/24/01	2/24/01		
Wing and Tail Templates	2/25/01	2/25/01		
Wing and Tail Foam Cut	2/27/01	3/1/01		
Make Landing Gear Molds	2/27/01	2/27/01		
Fuselage Lay-Up and Cook	2/28/01	2/28/01		
Lay-Up and Cook Landing	2/28/01	2/28/01		
Lay-Up and Cook Speed	2/28/01	2/28/01		
Spar and Bulkhead Lay-Up	2/28/01	2/28/01		
Tail Lay-Up and Cook	3/1/01	3/1/01		
Landing Gear Drop Test	3/1/01	3/1/01		
Spar and Bulkhead Shaping	3/1/01	3/1/01		
Wing Lay-Up and Cook	3/2/01	3/2/01		
Edge Trimming	3/2/01	3/2/01		
Secondary Bonding	3/2/01	3/2/01		

Figure 6.1: manufacturing schedule for the prototype

6.5 Innovative Configuration Solutions, Manufacturing Process, and Cost Reduction

To reduce cost of manufacturing, the team made a few innovative decisions. In the past, OSU teams have a history of using carbon fiber, but usually their molds were not re-useable. This year, the molds were treated in a way that they would be reusable. Since the team will be building both a prototype and the final plane, this process will reduce both cost and time. The wing lay-up procedure is also time and cost reducing procedure (see Wing Process 6.2.3). Most of the cost-reducing, innovative details of the configuration have been discussed in the previous sections 5.4, 6.2 and 6.3.



THREE VIEW

OSU BLACK TEAM

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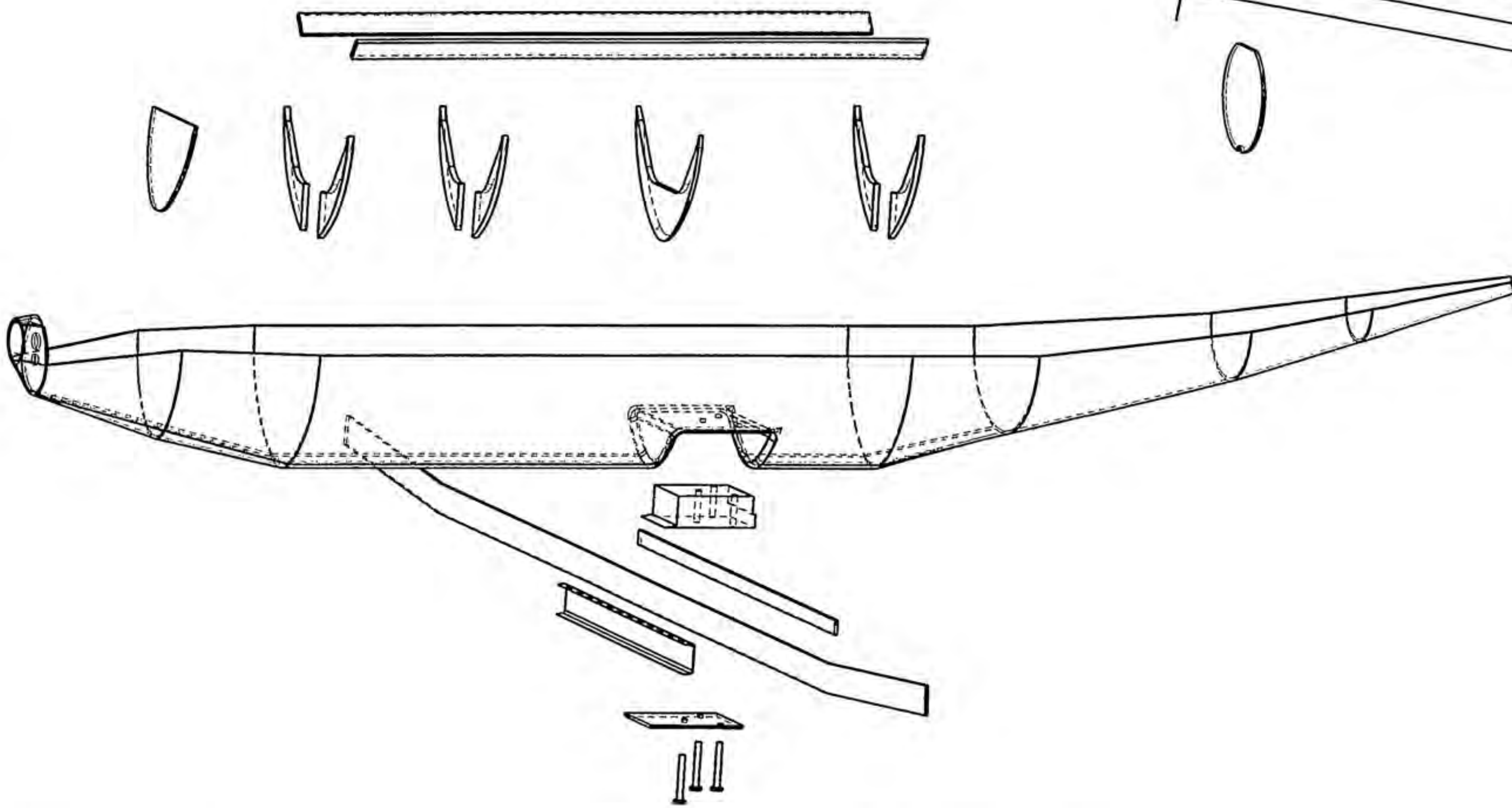

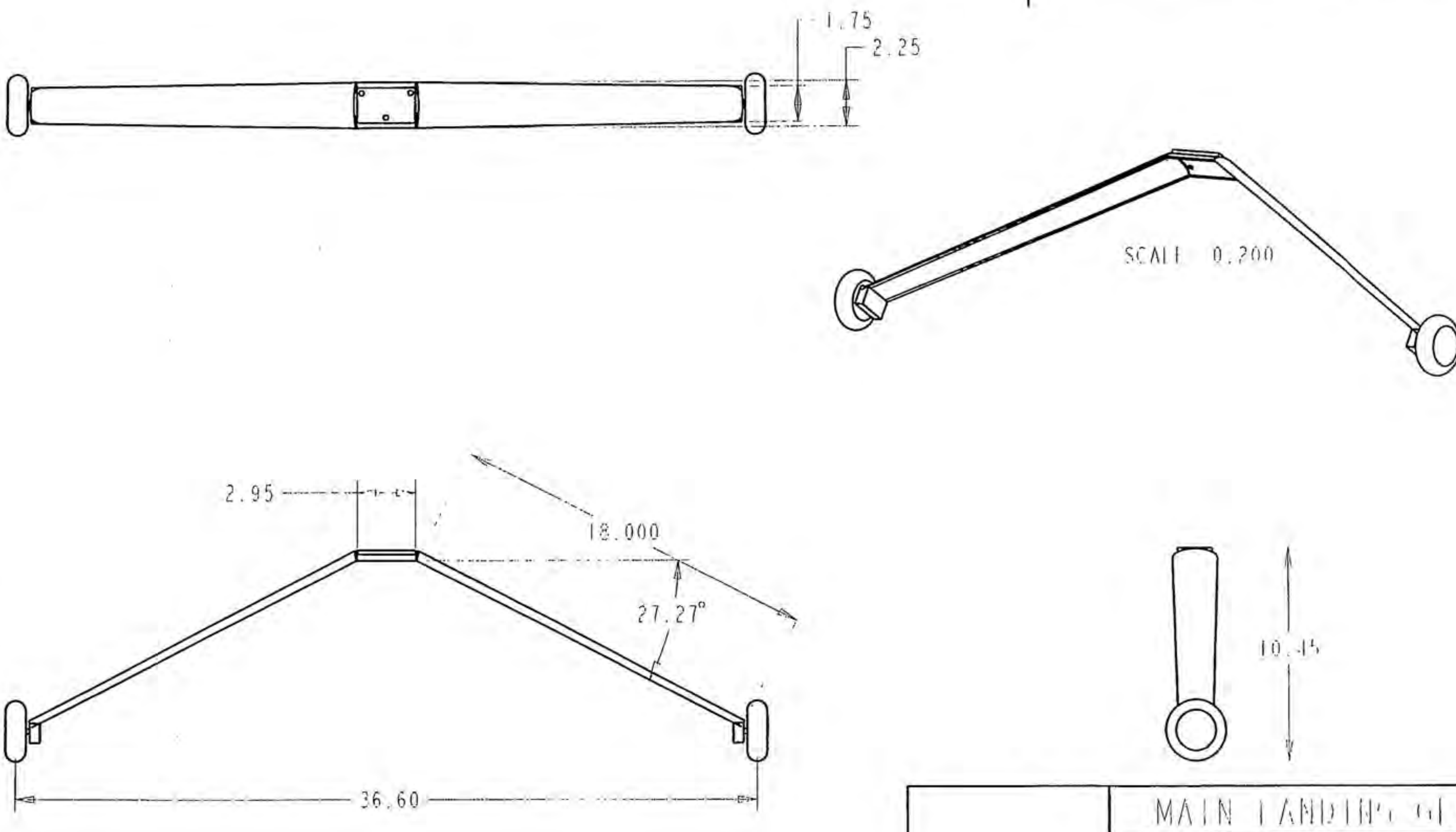

MAIN LANDING GEAR

OSU BLACK TEAM

SCALE