SUMMARY OF RESEARCH

Title of Grant:

Support of AIAA Student Aircraft Design/Fly Competition

Principal Investigator:
Gregory S. Page

Report Period:
1 April 1997 - 31 December 1997

Grantee:
The American Institute of
Aeronautics and Astronautics Foundation
1801 Alexander Bell Drive, Suite 500
Reston, VA 20191-4344

Grant No.
N00014-97-1-0421

FIGURES 1,7,8,9,10,11,12,13,14 & 15

WERE INTENTIONALLY LEFT OUT OF REPORT

PER: GREGORY S. PAGE

(202) 767-5852

NAVAL RESEARCH LABORATORY

WASHINGTON, D.C.

AIAA Student Design/Build/Fly Competition
APA TC Education Subcommittee
Michael Selig, Point of Contact, APA TC

- Objectives: Establish a design/build/fly competition sponsored by the AIAA. Student teams will design, build and fly an unmanned R/C electric aircraft designed for maximum range on a limited battery (1996-97 rules). Slightly change rules from year to year to keep contest from stagnating to a one design contest.
- Inaugural Competition
  - Sponsors: ONR, Cessna, AIAA Foundation
  - BAI Aerosystems, Ragged Island, MD, April 26, 1997
  - 11 teams competing (20 registered in Oct 1996)
- 1st: University of Illinois at Urbana-Champaign ($2500)
- 2nd: Virginia Tech ($1500)
- 3rd: Texas A&M University ($1000)

Other Teams (in order of standing in competition)

- Utah State University
- West Virginia University
- Washington State University
- University of Calif, Los Angeles
- Syracuse University
- Polytechnic University, Brooklyn
- University of Alabama
- Rensselaer Polytechnic Institute

Special Session for Winning Teams at Atlanta

AIAA Design/Build/Fly Competition Awards and Presentations, Monday PM, Session 11-APA-9, Rm = ?

Changes for 1997-98 Contest Year

- Design Engineering TC joins Applied Aerodynamics, Aircraft Design, and Flight Test TCs in organizing competition. APA TC remains lead TC.
- Significant rule changes
  - Max lap contest (was approximately 1 min / lap)
    - → Max laps in 7 mins
  - No score for off-runway landing (effectively 0 laps)
    - → Off-runway landing yields no landing bonus
    - → On-runway landing yields 1 pt landing bonus
  - Ranking = Rept Score x Laps
    - → Ranking = Rept Score x (Laps + Lnd Bonus)
- Contest Location/Date: Hosted by Cessna Aircraft, Wichita, KS, April 25-26, 1998
- Awards Conference: World Aviation Congress (proposed)
- Web site moves → http://amber.aae.uiuc.edu/~aiaadbf

Information

- Contest web site: http://amber.aae.uiuc.edu/~aiaadbf
- Contest Admin: dbfadmin@euclid.nrl.navy.mil
- TC Representatives
  - Michael Selig (Applied Aero TC)
    m-selig@uiuc.edu
  - Greg Page (Aircraft Design TC)
    page@euclid.nrl.navy.mil
  - Chris Bovias (Flight Test TC)
    bovais@euclid.nrl.navy.mil
  - Design Engineering TC
  - Wil Vargas (AIAA & Student Activities Committee)
    wilv@aiaa.org

AIAA DBF Competition Team Composition 1996-97

University	Team Name	Names	Dept	Year
U of I - UC	RPR-1 Jack	Andy Broeren	ME	PhD
		Ashok Gopalarathnam	AAE	PhD
		Chad Henze	AAE	MS
		Benjamin Keen	AAE	Senior
		Sam Lee	AAE	PhD
		Christopher Lyon	AAE	MS
		Joshua Minks	AAE	Senior
		Patrick Boyssmith	AAE	Freshman
		Martin Klipp	AAE	Junior
		Chong Win Koh	AAE	Junior
		Pony Lee	AAE	Sophomore
		Shalim Mody	AAE	Freshman
		Vijay Rum	AAE	Freshman
		Patrick Schuett	AAE	Sophomore

In order of names signed on info sheet requested during the contest.
Expect name misspellings and incomplete lists.
Not all teams were reached....

University	Team Name	Names	Dept	Year
Virginia Tech	Virginia Tech	Jay Gundlach	AE	Junior
		Matt Orr	AE	Freshman
		Alex Roup	AE	Junior
		Greg King	AE	Sophomore
		Geoffrey Buescher	AE	Freshman
		Alex Knob	AE	Sophomore
		Rebecca Gasder	AE	Freshman
		Oleg Gologidov	AE	MS
		Michelle Werle	AE	Sophomore
		Bo Naaz	AE	Sophomore
		Chris Gunther	AE	Junior
		Mike Russell	ME	Junior

University	Team Name	Names	Dept	Year
Texas A&M	Country Tourers	Rip Rippey	AAE	Junior
		M. Shea Parks		Junior
		Jason Field		Junior

University	Team Name	Names	Dept	Year
Utah State U.	Utah State	Robert Strahl	ME	Senior
		James Furfaro	ME	Senior
		Ashley Childs	ME	Senior
		Mike Rigby	ME	Senior
		Matthew Erni	ME	Senior
		Kara Parkki	ME	Junior
		Dominic Florin	ME	Junior
		Shelly Barlow	ME	Junior
		Kirk Sorenson	ME	Junior

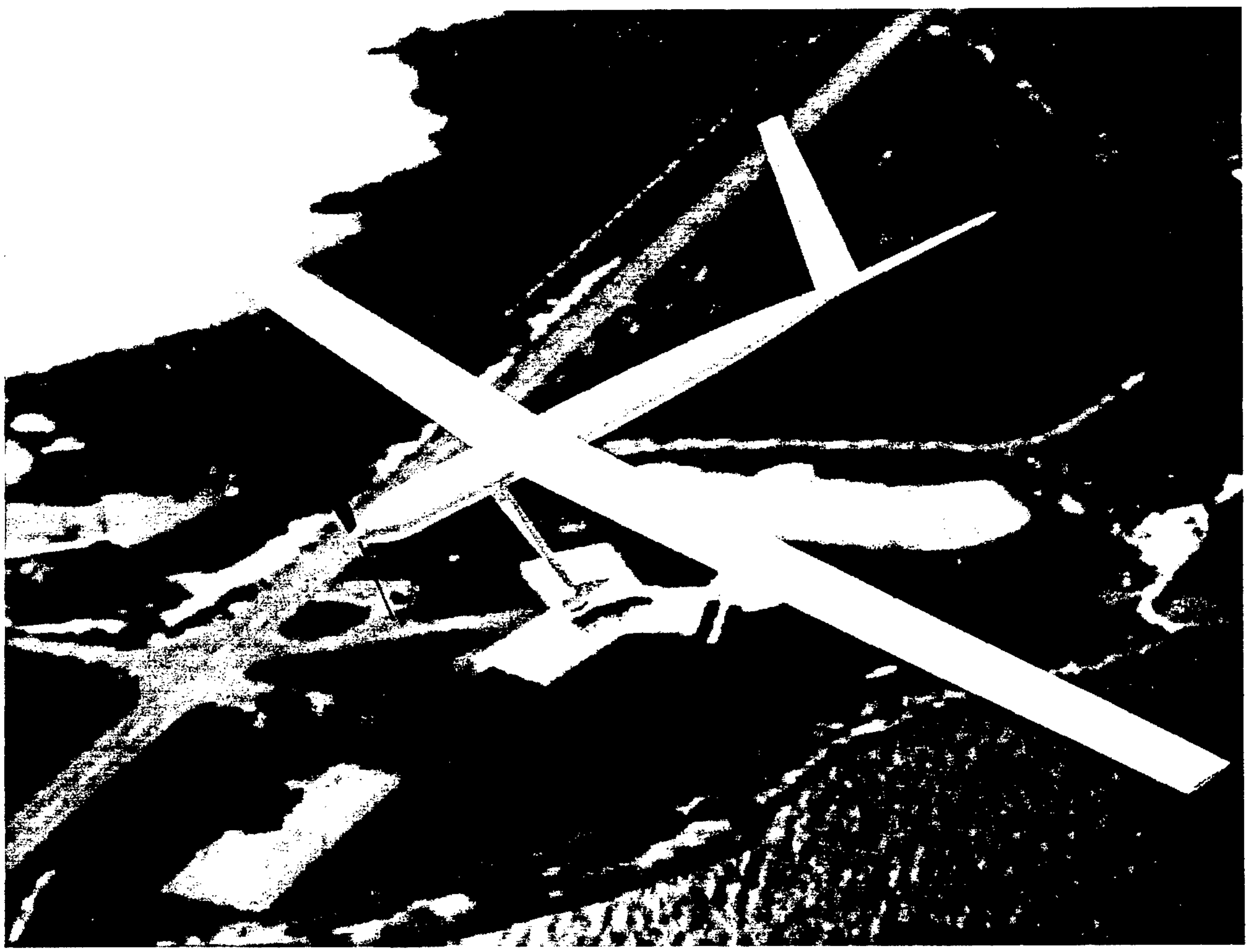
University	Team Name	Names	Dept	Year
W. Virginia U.	W. Virginia U.	Selios Pispitsos	MAE	Senior
		Elenor Taylor	AE	Junior
		Christi Kologzar	MAE	Senior
		Gearle Bailey	MAE	Senior
		Brandon Richards	MAE	Sophomore
		Ron H. Khoshneuiss-Ansari	MAE	Junior
		John Smith	MAE	Sophomore
		Steve Dick	MAE	Junior
		Pete Cooke	MAE	Sophomore

University	Team Name	Names	Dept	Year
UCLA	UCLA AIAA	Wayne Lu	AE	Senior
		Adam Huang	AE	Senior
		Jun Isobe	AE	Senior
		Franklin Meng	EE	Sophomore
		Hon Fai Vnong	AE	Senior
		Chris Silva	AE	Junior
		Ching-Yi Wang	AE	Freshman
		John Moreland	AE	Freshman
		Eric Prophet	AE	Senior
		Brian Luong	Mat. Sci.& Eng	Sophomore
		Michael Fuch	AE	Senior
		Tiry Pan	Bio. Eng.	Junior

University	Team Name	Names	Dept	Year
Syracuse U.	Syracuse U.	Sarah Benedict	AE	Senior
		Richard Hi Lee	AE	Junior
		A. Joseph Vinliquerra	AE	Freshman
		Kevin Bendowski	AE	Sophomore
		Jason Seklejean	AE	Senior
		Arun Chawon	AE	Freshman

AIAA Student Design / Build / Fly Competition

Virginia Tech Design Team
Design Report-Proposal Phase



Authors Christopher Gunther, John Gundlach, Alexander Roup
Group Members Geoffrey Buescher, Rebecca Gassler, Alex Knab, Matt Orr, Mike Russell, Michelle Werle

Table of Contents

Executive Summary	**1**
Management Summary	**1**
Conceptual Design	**3**
Preliminary Design	**4**
Detail Design	**6**
Manufacturing Plan	**12**
References	**17**
Appendix of Figures	**18**

Executive Summary

The evolution of the Virginia Polytechnic Institute and State University entry into the AIAA Student Design/Build/Fly Competition involved the collective effort of fifteen team members who developed two airframes. Preliminary design of Hokie Bird I began in September of 1996 and first flew in early December 1996. The performance and handling qualities of the first aircraft were evaluated over the winter semester break and a new configuration was developed. Hokie Bird II is a more sophisticated evolution of the first aircraft, with improved aerodynamics and structural design. Construction preparation for Hokie Bird II was begun in early February 1997. The preliminary design was finalized by mid-February 1997.

A range of design alternatives were including conventional, canard, displaced tail, and flying wing. The conventional design was chosen as the optimum compromise between performance, simplicity, and other factors. The displaced tail was another viable option, but the larger knowledge base with the conventional configuration steered the design selection in this direction. The other options proved to have inadequate performance relative to what was attainable with the conventional configuration.

Hokie Bird I had a three-piece detachable wing mounted on the top of the fuselage, and a conventional tail with elevator/rudder control surfaces. Ailerons were put on the outboard panels and flaps were located on the inboard wing panels. A tricycle landing gear with a steerable nosewheel was chosen for superior ground handling qualities over a taildragger configuration. The fuselage was simply a slightly rounded straight fiberglass shell with a carbonfiber tail boom. The wing and tails were made of Dow Blueboard foam cores covered with a combination of fiberglass and carbon fiber.

Flight testing of Hokie Bird I provided valuable experience with flying this type of aircraft. Many improvements for a second design were considered based on the series of flight tests. The handling qualities of this aircraft were considered to be acceptable.

Hokie Bird II is currently under construction. This design has the same wing span, wing area, and aspect ratio as Hokie Bird I, but several modifications were made to the design. The wing is in two-pieces with a single dihedral starting at the wing/fuselage junction. The wing is straight tapered, washout is used towards the wing tips, and airfoil sections were changed from the root to the tip. A V-tail is used to improve the aesthetics over the homely Hokie Bird I. The landing gear design was changed to a taildragger so that the drag in flight would be reduced, with an acceptable sacrifice of ground handling ease. Manufacturing of this airframe incorporates molds extensively. The fuselage is much larger than that of Hokie Bird I. The wing uses a fiberglass/Spyder foam sandwich construction with a spruce I-beam spar. The fibers of the skin are oriented at 45 degree angles to the leading edge so that the skin can resist torsion, while the spar can be used to resist bending moments. The wing construction of Hokie Bird II is much stronger and lighter than its predecessor.

The design software used was a modified version of a solar-powered aircraft design program written by John Gundlach in the Spring of 1996. The program can analyze electric propulsion systems, estimate aircraft component weights, and analyze aerodynamic performance in take-off, climb, turning flight, level flight, and landing. This program is versatile enough to perform analysis from preliminary design to detail design.

Management Summary

The Hokie Bird Design Team was divided into teams of design and construction groups. A team-oriented approach was chosen to facilitate team member interaction and ensure a broad knowledge-base was applied in each level of the design. The design team consisted of 3 primary groups: the empennage group, the wing group, and the fuselage group. Considering the lack of advanced knowledge of the team members, a junior coordinated each team's personnel and oversaw the group activities. The competition team organizational chart is shown in Figure 1.

The empennage team was primarily responsible for the selection of the tail configuration and design. A number of concepts were developed, varying from the traditional tail to a swept-forward V-tail. A swept rearward V-tail was finally chosen as the final design. The tails were designed to address any structural concerns. The proper sizing for control power and stability was performed. Construction techniques and procedures were developed and followed by the team members as well.

The wing team was an integral part of the overall plane design. The Selig Donnovan 7032 airfoil was selected and properly sized to adequately meet predetermined performance parameters. A structural analysis was done to ensure the wing could meet different failure mode criteria while maintaining an adequate factor of safety. A molded construction technique was chosen for increased accuracy and surface finish. Wing core, template, spar, and mold plug construction as well as final product fabrication were responsibilities of the wing team.

The fuselage team was involved in numerous areas of the aircraft design and fabrication process. First, the payload requirements were considered. All necessary cargo was then determined (payload, propulsion, electronics, internal structures). The fuselage dimensions were based on a "best fit" design for cargo capacity as well as location of wing and tail components. Propulsive parts consisted of batteries, motor, gear-box, propeller, and speed controller. Other components are the receiver, batteries, steel payload, microservos, and antennas. Wing and tail hard-point attachments were designed. A unique mold and plug construction technique was used for the fuselage (see Construction Techniques). Landing gear location and strength, prop clearance, and center of gravity location were also considered.

Each team met at least once a week for a summary of accomplishments and goal assessment for the upcoming week. Team leaders meet often to ensure component integration success and overall aircraft performance. A detailed mission statement and deadline timeline was established and followed at the beginning of the project. The milestone chart is shown below.

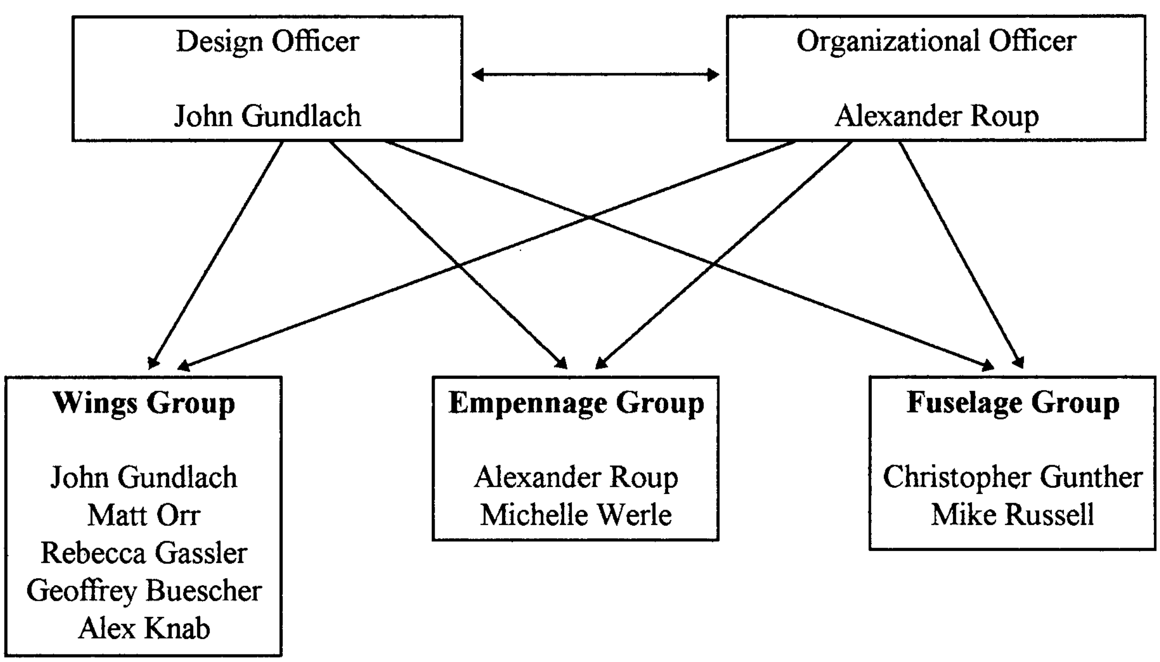
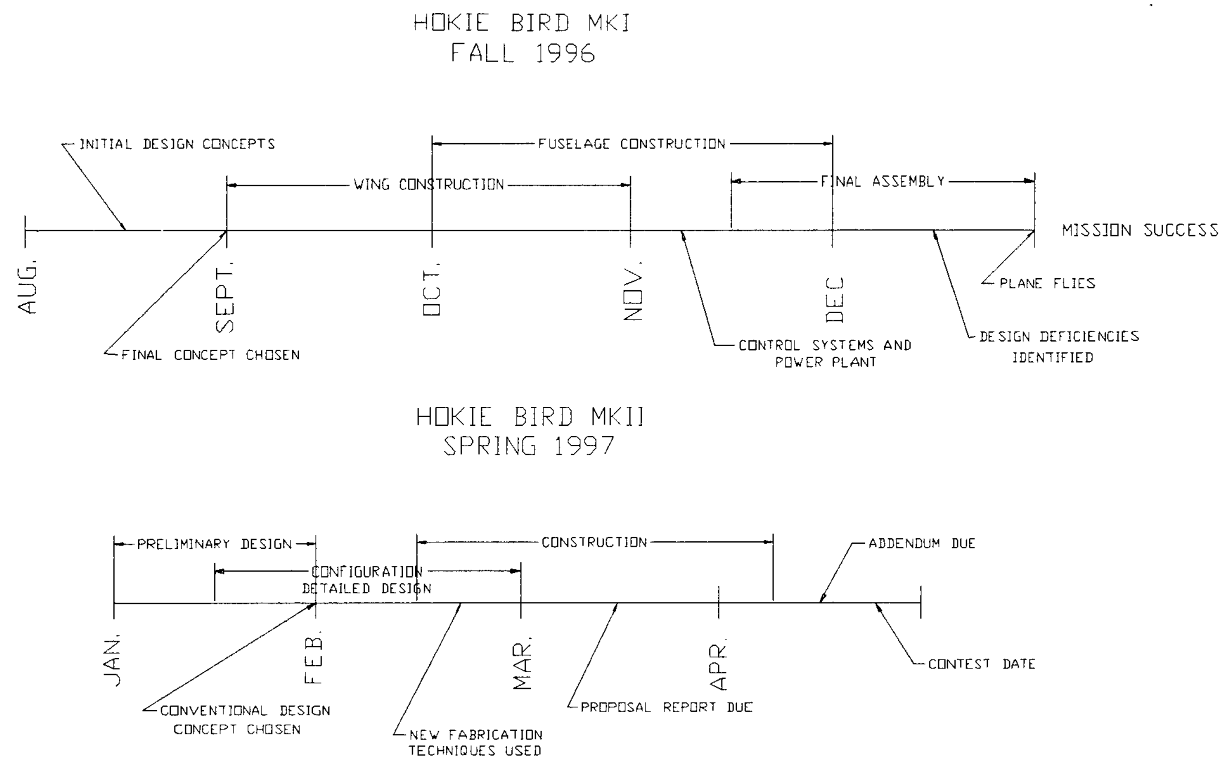


Figure 1 Management Diagram



Milestone Chart

Conceptual Design

Two separate conceptual design processes were performed for Hokie Bird I and II. Because Hokie Bird I was intended only to teach people how to build composite airframes and teach the pilots to fly this type of aircraft, a very simple conceptual design process resulted. The configurations evaluated for this design phase were conventional, flying wing, displaced tail, and canard.

In the conceptual design phase of Hokie Bird II, other configuration possibilities were reevaluated. Because the conventional design was successful in Hokie Bird I, it was the most developed design choice. The three versions of a conventional configuration are a conventional arrangement of wing/fuselage/tail, an inverted gull wing, and a low fuselage with a pod motor mount. The figures of merit of each configuration are listed in Table 1.

The flying wing is by far the simplest design to manufacture. However, the maximum attainable lift coefficient is much lower than that of a conventional design, which increases take-off and landing distances. Also, trim restrictions make the use of flaps impractical. Yaw control is difficult without the use of winglets, and if winglets were used, the design may as well be changed to a displaced tail. The lift to drag ratio for flying wings are typically lower than that of a conventional design, and the lift to drag ratio is the main parameter for constant power range performance.

A canard configuration offers great aesthetics and, if designed properly, is incapable of conventional stall. However, the canard has a lower lift to drag ratio than a conventional design, a lower maximum lift coefficient, and offers no reductions in complexity.

The displaced tail design, shown in Figure 2, offers the simplicity of a flying wing with the aerodynamic efficiency close to a conventional design. However, the torsional strength requirements of this configuration leads to a heavy wing structure. The fuselage would be nearly as large as with the conventional design. Unlike the conventional design, the short tail moment arm leads to large tail surfaces.

The conventional configuration is the optimum choice for this competition. The conventional design offers the best combination of aerodynamic efficiency, simplicity, and handling qualities. The pilots are most experienced with this configuration and their experience will increase chances of success.

The conventional wing/fuselage/tail configuration, shown in Figure 3, has several benefits, thus making it a popular selection for the majority of aircraft designs. Some of the most notable strengths of this configuration is its simplicity of construction, relative predictability of handling qualities, and ease of analysis. The thrust line is very close to or at the center line of the aircraft, making effects of changing power settings on aircraft trim very small. The main problem with this configuration is that the drag-producing landing gear must be large in order to allow ground clearance for the prop and for the tail at take-off rotation.

The inverted gull wing configuration, shown in Figure 4, is similar to the conventional configuration with the exception of the wing having a sharp anhedral towards the root which abruptly changes to dihedral after the landing gear. The benefit of this configuration over the conventional one is that the landing gear length is greatly reduced, which may improve the combined aerodynamic efficiency of the wing/landing gear combination. The drawbacks of this design include increased structural weight and complexity of manufacture.

The low fuselage with a pod motor mount, shown in Figure 5, is essentially a conventional design with a low, glider-like, fuselage with wheels protruding from its underside, with a motor pod above the fuselage. The benefits of this design include decreased landing gear weight and drag, and reduced take-off distance due to a strong ground effect associated with the low wing. Unfortunately, this significantly large ground effect will increase landing distance, and the wings will not be allowed to travel through a safe roll angle range without striking the tips on the ground. The aft portion of the fuselage would need to be angled upward to allow for rotation, unless a constant lift coefficient take off were allowed. This constant angle take-off and landing would result in a high take-off speed and landing speed. An angled tail boom would result in a high fuselage drag contribution.

Figure of Merit	**Value**	**Conventional**	**Inverted Gull Win**	**Pod Motor Mount**	**Displaced Tail**
Simplicity of Construction	20	20	10	15	18
Uniqueness	10	4	10	6	10
Expected Efficiency					
Due to Wing Configuration	15	15	10	15	15
Due to Powerplant/Propeller Configuration	15	15	15	12	12
Due to Landing Gear Configuration	15	10	13	13	8
Due to Empennage Configuration	15	15	15	15	13
Landing/Takeoff Performance and Reliability	10	10	10	6	10
Total	**100**	**89**	**83**	**82**	**86**

Table 1 Conceptual Design Figure of merit

Preliminary Design

The ELEC Fortran program was used for the preliminary design phase. This program is a modification of the SPA (Solar Powered Aircraft) program developed by John Gundlach in the Spring of 1996. ELEC can analyze electric propulsion systems, static stability, estimate aircraft component weights, The profile drag coefficients for the wing and tail sections were found by interpolating between lift coefficients and Reynolds numbers.

The propeller performance was found by using a technique demonstrated by the great Greg Page. Data for existing model aircraft propellers, such as coefficient of power and thrust at a given advance ratio, was used for analysis. Three propellers were selected, two of the three had the same pitch to diameter ratio, and two had the same solidity. A propeller was chosen based on available sizes, and its performance was analyzed by interpolating its pitch to diameter ratio and solidity with those of the three reference propellers. Initial sizing of propellers was done on a spread sheet to speed the process, but the same process was later performed within ELEC.

The electric propulsion system analysis involves the propeller, gearbox, motor, speed control, and batteries. Two scenarios must be evaluated; one for maximum power output, and one for less than maximum power output. For the first scenario, a current limit exists for either the motor, speed control, or battery system, and determines the maximum power that can be produced.

An electric propulsion system model was developed to analyze the energy drain of the battery during take-off and all other flight conditions. Propeller data was found using the propeller model outlined above. For the take-off condition, the minimum of the maximum allowable current for each of the components is used to determine the maximum power output achievable. For other flight conditions, the thrust required from the propeller is used to determine the propeller RPM. The power that the propeller produces is divided by the gear efficiency to determine the power required of the motor. The propeller RPM is divided by the gear ratio to find the motor RPM. This RPM is divided by the motors voltage constant to find the voltage at the motor leads. Next, a quadratic equation is solved for the current at the battery which accounts for the no-load current of the motor, the resistances of the motor, speed control, and batteries. The total power taken from the battery is the voltage of the battery multiplied by the battery current plus the square of the battery current multiplied by the battery resistance. The energy lost is simply the power taken from the battery divided by the time at which the power is at a given setting.

The electric propulsion system has to be a combination of commercially available batteries, motors, speed controls, gear boxes, and propellers. Because designing specialized parts for this system is against the contest rules, a true optimization is not possible. Instead, a trial and error method in which combinations of electric propulsion system elements must be implemented.

Take-off performance analysis involves the thrust produced by the propeller as the aircraft speeds up, decrease in rolling resistance as the aircraft produces lift, and changing drag coefficient as the Reynolds number increases. The take-off velocity is assumed to be 1.3 times the stall speed, and for simplicity, the take-off roll is assumed to occur at a constant lift coefficient associated with take-off speed. The climb past the 10-foot obstacle is also assumed to occur at the same lift coefficient.

The straight and level flight speed is optimized for the condition of minimum energy loss per distance. This roughly corresponds to a maximum lift to drag ratio but the effects of the electric propulsion system may alter the speed slightly. The optimum flight velocity is determined through an iterative optimization algorithm.

The radius of the turning flight is optimized for minimum energy loss per turn. The flight velocity entering the turn is assumed to be the same as for level flight, which leaves only radius as a variable. If the radius is made too small, the aircraft will require excessive power to maintain level flight. If the radius is too large, the power requirements will be lower but the turning time will be too great, resulting in a high energy loss. Optimization of turning radius is also performed through an iterative optimization algorithm.

Through construction and flight testing Hokie Bird I, several possible improvements were noted. These included aerodynamic, structural, and manufacturing modifications. First, the tricycle landing gear proved to have high drag, so alternative landing gear configurations were investigated. In the conceptual

The structural design of Hokie Bird I was a model of manufacturing simplicity. Foam core wings and male mold fuselage construction was used because maximum performance was not the primary concern. This design increased the airframe weight for the same strength over more advanced construction methods. This increased the power requirement to lift the payload and therefore decreased the range.

In the design of Hokie Bird II it was decided to use more advanced female molded construction techniques for both the wings and fuselage in conjunction with a more streamlined fuselage shape. The wings were designed with a stressed skin with fiber filaments angled at 45 degrees for resisting torsion. The spar design consisted of spruce spar caps for resisting bending and a balsa shear web for resisting shear due to bending and increasing the buckling strength.

Detail Design

Because ELEC is such a diverse program, both the preliminary and detail design stages used this software. The same procedures described in the previous section were used to evaluate the final configuration. A description of the final configuration is given in this section.

The Selig Donnovan 7032 and 7037 airfoils were selected for the wing of the Hokie Bird series. These two airfoils were two candidates of several low Reynolds number sections. Data from each airfoil was used in the performance program and the SD 7032 yielded the best compromise of total number of laps and a low take-off distance. Arfoils with a large trailing edge cusp such as the Wortman FX63-137 were not selected because construction would prove to be significantly more difficult than with lower-cambered sections and their performance was not the highest. The Hokie Bird I uses the SD 7032 exclusively on the wing while the Hokie Bird II uses the SD 7032 towards the root and transitions from the SD 7032 to the 7037 from midspan to the tip.

The final propulsion system included the following items: A Freudenthaler 14x8.5" folding propeller, a gearbox with a 2.5:1 gear reduction, an Aveox 1412/3Y brushless DC motor, an Aveox 120HV2 motor controller, and 21 Sanyo 1700 mAh cells. The selection was made from the results of take-off distance and total number of laps possible from the program. The above was the best choice of over thirty combinations evaluated.

Below is a listing of some performance estimations and vehicle characteristics.

Estimated Take-Off Weight	14.6 Pounds
Payload Mass Fraction	0.514
Wing Loading	2 Pounds/Feet^2
Take-Off Distance	149.7 Feet
Number of Laps	38.0
Lift to Drag Ratio	20.63
$C_L^{3/2}/C_D$	20.14
G-Loading	4.77
Cruise Velocity	48.5 Feet/second
Optimal Turn Radius	46.56 Feet
Energy Loss per Lap	1839 Joules
Wing Tip Reynolds Number	209751

The results of the stability and control analysis is summarized in the table below.

CG Relative to MAC	0.35
Neutral Point	0.505
Static Margin	0.16

$d\varepsilon/d\alpha$	0.2863	
$C_{M\alpha}$ Wing	0.5495	1/Radian
$C_{M\alpha}$ Fuselage	0.0545	1/Radian
$C_{M\alpha}$ Tail Horizontal Component	-1.4641	1/Radian
$C_{M\alpha}$ Total	-0.8601	1/Radian
$C_{M\alpha}$ dot	-4.882	1/Radian/Sec
C_{Mq} Tail Horizontal Component	-17.057	1/Radian/Sec
$C_{M\alpha}$ dot + C_{Mq}	-21.938	1/Radian/Sec
$C_{M\delta e}$	-1.192	1/Radian
$C_{L\delta e}$	0.286	1/Radian
$\Delta\delta e/ \Delta C_L$	0.135	Radians
$C_{n\delta r}$	-0.0401	1/Radian

Landing Gear Design Specifications

For the Hokie Bird, a tail wheel configuration was chosen for the final design concept. The landing gear was designed to meet the specifications in *Light Airplane Design*. The C.G. for the Design Gross Weight should fall inside a hatched area from 15 to 25 degrees of vertical from the main wheel contact point. The main wheel motion due to shock absorber deflection should fall between 5 and 15 degrees. Both criteria were met and demonstrated in Figure 6. Also, the tail wheel knuckle spindle axis should intercept the ground line ahead of the wheel contact point at a distance of at least one-tenth the tail wheel diameter. The tail wheel's shock absorber deflection must be within the cross-hatched area between the ground normal and a 45 degree offset. Ideally, it should lie 30 degrees from normal, and the plane was built accordingly. These specifications can be seen in the side view of Figure 6. The front view of Figure 6 demonstrates the necessity that the landing gear contact points lie outside a 25 degree sweep from the plane's center of gravity.

The track and wheel base of the plane were then determined. The relationship between track and wheel base is dictated by the Turnover Angle, which is determined by trial and error as shown in the top view of Figure 6. A detailed description of the technique may be found in *Light Airplane Design*. If the turnover angle is more than 60 degrees, the track or wheel base must be increased. The Turnover Angle of the Hokie Bird is exactly 60 degrees.

Some basic gear configuration specifications are as follows. The wheel track is 8.75 inches and the wheel base is 38.25 inches. The tail wheel has a diameter of 1 inch, while the main gear has a wheel diameter of 2 inches. Tires will be selected from commercially available products. However, hard-point attachments and struts will be machined from aluminum stock. The main gear will be fixed, but the tail wheel will be fully steerable by the use of a servo and actuator to enhance ground handling and taxiing ability. Retractable landing gear and the corresponding mechanisms were investigated, but it was determined that the benefits from using such a system were minimal. The increases in structural considerations, weight, complexity, cost, and loss of reliability ruled out the implementation of a retractable gear system. Instead, fairings will be built around the landing gear struts and the tires themselves to decrease drag.

a constant thickness of readily available wood. A 3/16 inch thick sheet was spliced, cut, and glued into a beam. Material properties for the Spruce were found in a table from Beer and Johnston. Failure criteria were 5.6 ksi in compression, 8.6 ksi in tension, and 1.12 ksi in shear, and a Young's Modulus of 1.5*10^6 psi was used. The primary design constraint of the beam is normal yield stress in compression; this value was used to determine failure of the wing under loading. The wing was assumed to be rigid in torsion. The wing skin (fiberglass) was assumed to carry all torsional loads to simplify the analysis. Von Mises' stresses were thus ignored. This is a reasonable assumption since the skin forms a closed torsion tube. The I-beam was designed to carry all the bending loads that may be incurred on the plane during flight and ground static testing.

The most critical design point is the wing root. The cross-section of the wing root portion of the beam can be seen in Figure 7. In order to minimize the weight of the I-beam, a tapered design was used. The thicknesses of the webs and flange were kept constant, but the width and height of the beam was varied. Since the thickness of the airfoil is 1 inch at the root and 0.5 inches at the tip, this is a necessary design criteria. The beam will be placed about the wing's quarter chord to minimize torsional effects due to non-centric loading.

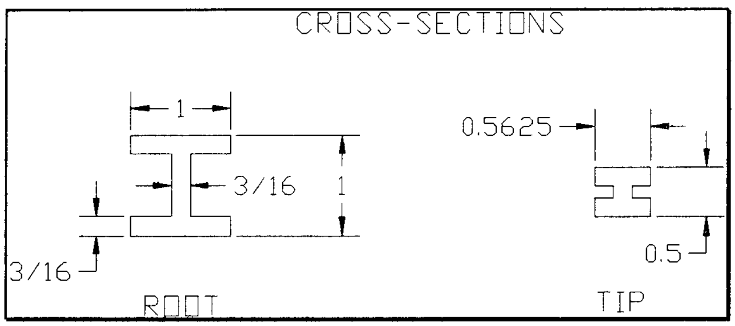


Figure 7 Spar Cross-Section

Two modes were considered for failure. The first is the mission requirement that the plane be supported at the wingtips and still be allowed to carry the fully loaded aircraft weight of 15 lbs (Figure 8).

The bending moment along the wing spar in this loaded condition can be seen in Fig. 9 as a linear relationship. Shear stresses along the wing span are equal to the tip load along the entire span (Fig. 10). The design condition is the normal stress when the beam fails under compressive yielding. This condition was analyzed using beam theory and a factor of safety using the given design is 1.47. The maximum value of the compressive stress obtained is 3798.4 psi and it occurs at 39.8 inches out from the wing root (Fig 11).

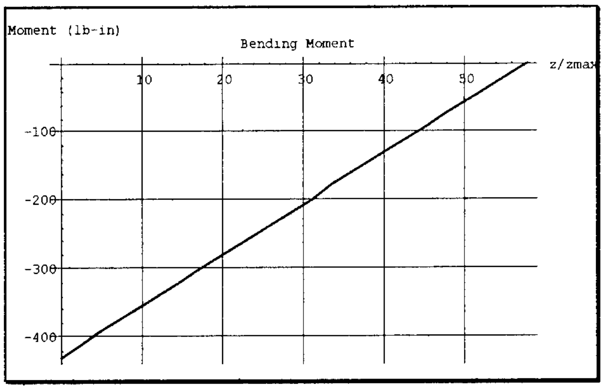


Figure 9

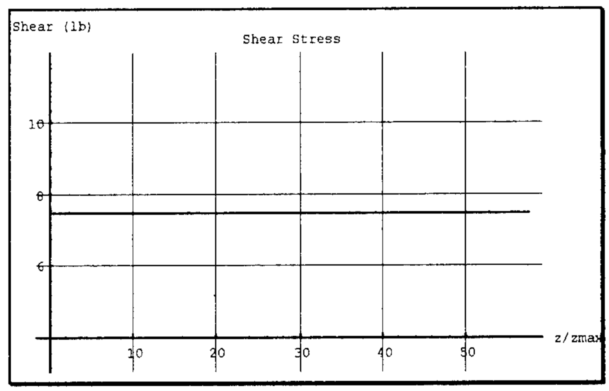


Figure 10

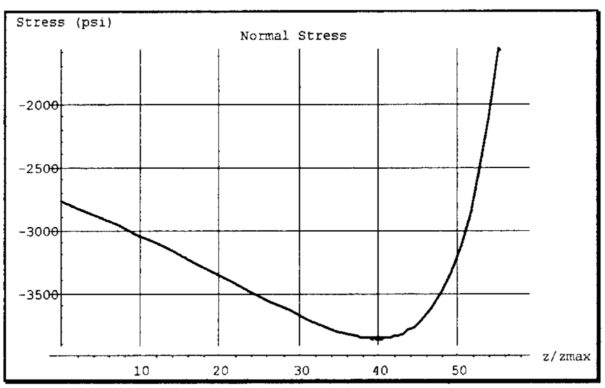


Figure 11

The next failure criterion was cruise flight. An elliptical lift distribution was assumed along the span of the wing (Fig. 12).. The tapered beam was then analyzed along the span to check for failure criteria. The bending moment curve can be seen in Figure 13, and shear forces are shown in Figure 14.

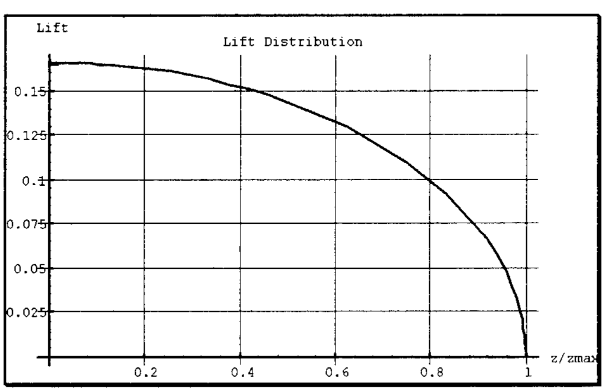


Figure 12

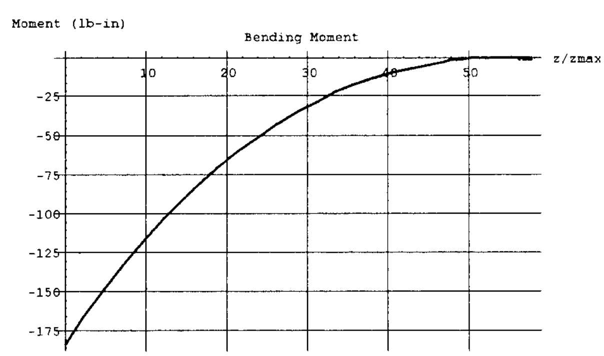


Figure 13

The normal stress plot (Fig. 15) shows the stress distribution. Note the maximum normal stress occurs at the wing root as expected. The maximum value of the normal stress is 1173.5 psi which corresponds to a factor of safety of 4.77.

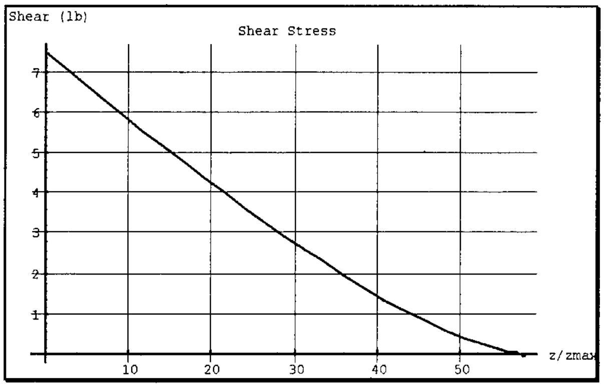


Figure 14

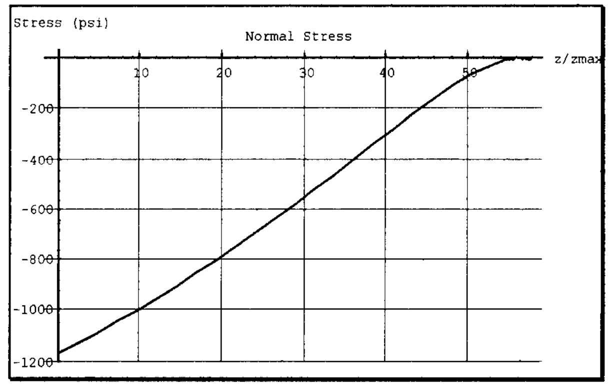


Figure 15

An important consideration is the weight of the spruce wing spar member. If a specific weight of 0.015 lbs/in is used and the dimensions of the I-beam are kept constant over the entire wing span (no taper), the weight of the spruce spar is 0.851 lbs. However, the tapered wing spar has a weight of only 0.326 lbs. A savings of 61.7% results from the design trade-off. This is a significant weight reduction and is achieved solely by tapering the height and width of the spar without modifying the web and flange thicknesses. This design is relatively simple and adequately meets mission requirements.

Fuselage and Empennage Structures:

The fuselage and empennage structures were designed more on past remote-controlled plane construction than anything else. The tails are not expected to experience loads that may endanger the structural integrity of the parts. The fiberglass construction should be adequately tough to resist any loading produced by aerodynamic forces, gusts, and control surface deflections. The fuselage was primarily designed to allow for fit of the cargo and payload needed. From previous plane construction and lessons learned from the construction of Hokie Bird I, the fuselage should be stiff enough to resist bending and torsional loads created by the tail and tail control deflections. Wing and empennage mounting attachments will be made of spruce and have pegged interfaces. The landing gear will also have balsa or spruce hard-point attachments on the fuselage. As a result, there should be no structural difficulties encountered in the fuselage during the aircraft service life.

The component integration of the payload and cargo was done to ensure a minimum fuselage volume to decrease drag and weight. Figure 16 shows the insertion of all major cargo components into the final design fuselage shape. The drawing is to scale based upon dimensions given in Table 2. Figure 16 also demonstrates the relationship between the cargo and CG location. The microservos will be imbedded directly in front of control surfaces throughout the aircraft.

Fuselage Components:

	Part	Dimensions (inches)	Quantity	Weight (oz.)
Payload	Steel Blocks	1 x 2 x 4.5	3	40
Propulsion	Speed Controller	2.1 x 1.5 x 0.4	1	2.2
	Motor	1.465 Dia. x 2.36 Long	1	10.2
	Gearbox	0.5 x 1.5 x 1.25	1	0.5
	Batteries	6.1x1.7x1.9	3	13.33
Electronics	Receiver	1.75 x 1.3 x 0.8	1	1
	Receiver Batteries	2 x 1.2 x 1.2	1	3
	Microservos	1.35 x 0.5 x 1.25	7	0.05

Total Weight	177.24 oz.

Table 2

Empennage Sizing

The tail sizing was calculated by picking vertical and horizontal tail volumes and a fixed fuselage length. The fuselage length was fixed at a conventionally proportioned length so that the fuselage design team could proceed with the fuselage design without having to resize. The vertical and horizontal tail volumes were picked from a set of data in *Light Airplane Design*. The required tail areas could then simply be calculated from the definitions of the vertical and horizontal tail volumes. A set of 13 tail configurations was considered for use. In these configurations, both conventional and V-tail designs of different aspect ratios and sweep angles were considered. These configurations were ranked in order of required tail surface area to produce the same tail volume coefficient, as shown in Table 3. The required surface area was the measure of performance used to determine the best tail design, with lower area corresponding to improve performance. The swept-back tail designs had reduced surface area because they allowed for an increased tail length for the same fuselage length. For this reason, a swept-back tail configuration was chosen. Two equivalent tail designs with the same surface area were identified, one with conventional vertical and horizontal stabilizers, and one with a V-tail. The V-tail configuration was chosen because it was considered to be more aesthetic and easier to construct.

Rank	Combinations	H Sweep	H AR	V Sweep	V AR	Total Area	% of Min
1	Vtail	20	5			202.4	100.00%
2	Horizontal + Vertical	20	5	20	2.5	202.9	100.25%
3	Horizontal + Vertical	20	5	10	2.5	206.2	101.88%
4	Horizontal + Vertical	20	5	25	1.5	209.8	103.66%
5	Horizontal + Vertical	20	5	-5	2.5	213.6	105.53%
6	Horizontal + Vertical	-5	5	20	2.5	217.3	107.36%
7	Vtail	10	5			217.4	107.41%
8	Horizontal + Vertical	-5	5	10	2.5	220.6	108.99%
9	Horizontal + Vertical	20	5	-5	1.5	222.3	109.83%
10	Horizontal + Vertical	-5	5	25	1.5	224.2	110.77%
11	Horizontal + Vertical	-5	5	-5	2.5	228	112.65%
12	Horizontal + Vertical	-5	5	-5	1.5	236.7	116.95%
13	Vtail	-5	5			244.6	120.85%

Table 3 Tail Configuration Rankings

Final Design

The final aircraft configuration chosen was a low-wing with a rearward-swept V-tail. Figure 17 gives a fully dimensioned, scaled depiction of this final concept. Major features include a wingspan of 118 inches, a fuselage length of 55.5 inches, and V-tails mounted with an inclination of 36.5 degrees. Two degrees of dihedral were added to the main wing and the tips of the wing were sculpted to increase efficiency. The CG location was determined to be 35% of the mean aerodynamic chord. This final conceptual configuration was ultimately used in the fabrication of Hokie Bird II.

Manufacturing Plan

Fuselage Construction

The unique design considerations of this project require the construction of a complexly contoured fuselage. The traditional fuselage fabrication technique calls for a balsa wood plug to be made by carefully cutting and sanding until the desired shape is achieved. Since complex curvature was called for in the design parameters, it was decided that a balsa wood plug would be to difficult to construct. This is because a lengthy amount of time is required for the filling material to dry. The filler would inevitably be required due to the complexity of the desired fuselage shape. Because filling and re-sanding would be inefficient and result in numerous waves, this method was deemed time-ineffective and inconvenient. These difficulties led us to search for an easier method. The final approach technique was found in a small article in *Motorcycle Consumer News* magazine which describes the use of clay for modeling.

The methods, clay types, and manufacturer's address given in the article led us to *Chavant, Inc.* of Red Bank, New Jersey. From their sample kit, which included fifteen different types of clay, two final selections were made, CM-70 and I-305. These clays were described as extra hard and very hard, respectively. Between the two types of clay, CM-70 was chosen because of economical reasons (CM-70 is over 25% cheaper). CM-70 is a dark brown formulation consisting of an oil-based clay. Its other main ingredients are wax, sulfur, and resin with other ingredients mixed in at unpublished concentrations. CM-70 has excellent qualities of adhesion, cohesion, and consistency and can be carved and extruded to an accurate, glass-like finish. The clay under proper conditions joins easily and will not show seams. It also does not crack or dry out because it is an oil-based product. After receiving the clay, different methods of working with the clay were tested.

The clay is hard at room temperature and not very moldable. When it is heated to approximately 150 degrees Fahrenheit the clay acts quite plastically and can be formed into the desired shape for a short time until it cools. In order to achieve this working temperature a heat source is necessary to warm the clay into a plastic usable state. Many heating devices were listed as usable in the *Chavant* literature, including purpose-built ovens with thermostats to regulate the required temperature. A restaurant bread warmer was also considered as an economical option; however, none were available for purchase. It was decided to improvise and build a crude but effective substitute to the more expensive and unavailable ovens described above. A substitute oven was fashioned from styrofoam insulation and a light bulb. A square box with a hinged lid was constructed with epoxy and styrofoam with an internal volume of one cubic foot. Next, a hole was drilled in the center of the top for a light bulb and another hole for the thermometer. Finally, we lined the inside and top with Reynolds aluminum foil to provide something to reflect the heat generated and keep the styrofoam from melting. With a 100 Watt Sylvania bulb, the heat required can be supplied by propping the lid open slightly.

In order to shape the clay ordered from *Chavant*, some of their basic tools (a rake, finisher, wire, and steels) with which to carve the clay were purchased. Before beginning to apply the clay, a basic frame of styrofoam and wood was made in the fuselage shape. The frame was constructed out of three pieces to which two sets of accurate templates were glued. These guides gave the general shape and dimension the model would take the form of. The first piece of styrofoam made up the section of the fuselage from the nose to the leading edge of the wing. This piece was made 2 x 3 x 14 inches long with the edges cut out to allow for the curvature of the fuselage. The second piece was made of a styrofoam piece 2 x 3 x 9 inches long. This middle piece makes up the part from the leading edge to the trailing edge of the wing. The final piece was made up of wood shaped to the approximate shape on a lathe. This piece makes up the last part of the plane from the trailing edge of the wing to the tail. Everything was assembled and glued together using five-minute epoxy. Before the epoxy dried, everything was trued and made as accurate as possible with pre-cut templates.

Finally, the clay was applied to the previously made form. After it was heated in the oven to about 150 degrees Fahrenheit, it was applied to the styrofoam and wood frame. The clay was applied by hand to get a rough shape of the fuselage and then smoothed with fingers before it cooled. This method allowed quick building of the basic form to the approximate dimensions of the fuselage. The plug was smoothed further with the rake by pulling excess clay off of high spots and adding hot clay to the low spots. The rake was used to pull clay alternately in directions parallel and perpendicular to the fuselage centerline. This gave the rough dimensions of the model. After getting the rough dimensions, further refinement was performed with the finisher by adding clay to low spots and removing clay from high spots. After the shape was achieved to satisfaction and the surface was made as smooth as possible, the final step was started, putting a finish on the clay plug.

The final step uses the steels to smooth out the remaining waves in the surface and to give the fuselage model a fine polish. The 0.010 millimeter steel was first used, and it was worked in opposing 45 degree directions from the fuselage centerline. This is done to take out the waves left by the rake and finisher, which are oriented from top to bottom and along the length of the fuselage. After the 0.010 millimeter steel is used, the model is redone with a 0.005 millimeter steel, giving the clay plug a mirror-like finish. After the clay plug is finished it can be given an even smoother surface, termed a "Class A" finish. This is achieved by painting the plug with a primer and then painting over it with a gel-coat paint to enhance the finish. This procedure fills in small gaps and other imperfections left on the clay by the previous processes.

Various methods may be used to construct a mold using fiberglass and resin, but a new product by *United States Gypsum Company* was chosen as a substitute. The product is called *Rayite* and it is a man-made gypsum material similar to that used in wallboard for house construction. *Rayite* was chosen due to its low cost and the relatively short amount of time required for the product to set. It also is lightweight, strong, and does not give off heat as a by product of its chemical reaction like fiberglass and epoxy resins do. *Rayite* comes in two parts: a surface coat, and a reinforcing coat. The surface coat is called MDM-S (model duplicating media-surface) and is first applied at approximately 1/8 inches thick directly on the surface of the plug. The surface coat is allowed to thicken (approximately 35-45 minutes from mixing) and then MDM-R (model duplicating media-reinforcement) is applied to reinforce the surface coat. This reinforcing coat is applied about 1 inch thick with a trowel.

In order to apply the *Rayite* it must first be mixed with non-contaminated water. The amount of water must be mixed in precise proportions with the powder. As a general rule five times as much MDM-R solution must be made as MDM-S solution. To make the MDM-S solution you must mix approximately 2.47 parts powder to one part water. For example, to make 1.25 pounds of mix, 0.89 pounds powder should be mixed with 0.36 pounds of water. This will give a consistency like that of pancake batter. The MDM-R solution requires approximately 3.03 parts powder to one part water. For example, to make 7.50 pounds of mix, 5.64 pounds of powder to 1.86 pounds water are needed. This will give you a consistency like that of dough. It is important that clean, clear water is used and that all powder is allowed to combine with the water before the mixture is stirred. The MDM-S dries to a strength of approximately 5000 psi in compression and the MDM-S dries to and approximate strength of 3500 psi.

The mold will be in two-parts since the mold will completely encase the plug. To make this type of mold a surface at the junction where the mold will be cut in two must be made. For our model, the fuselage will be split along the vertical axis. A piece of plywood will be cut to fit the plug almost exactly. The plywood sheet will be finished with a sealer such as polyurethane. Next, the plywood will be supported with some blocks so that the plywood's top surface will be level with the fuselage centerline. The gap between the plywood and clay will be filled with wax so that there is a solid surface to apply the *Rayite*. When everything is satisfactory, a layer of MDM-S will be applied to the clay surface. After it has thickened, a layer of MDM-R will be placed on the plug. After it has hardened, (approximately 1 hour) the mold will be carefully separated from the clay plug. This will give one-half of the mold. This mold half will be sealed with sealant and then the second half will be made.

The second half of the mold will be constructed similarly to the first half. Instead of using the plywood, MDM-S and MDM-R will be directly applied to the clay plug and previously made mold. To make the second plug the flange of the first mold must be waxed and then the clay plug will be placed into the first mold. Rayite can then be directly applied over the clay and onto the flange of the first mold. After waiting for an hour, the second mold should be done and ready to be sealed. Now there is a two part mold ready to be used to make the finished product.

After the mold is made the actual fuselage will be constructed. Various construction techniques and methods have been considered, but it has been decided that Spyder foam sandwiched between sheets of fiberglass will be used. After analysis it has been decided that single layers of 3 ounce glass will be used on the inside and outside of the fuselage. Between these 2 layers of fiberglass will be a single layer of 1/16 inch Spyder foam. Bulkheads will be located at strategic locations along the fuselage length with spars that will be embedded as needed. Small holes will be located to allow for positioning of microservos to move control surfaces. Cutouts in the bottom of the fuselage will be made to allow attachment of the wings. The nose of the fuselage will be strengthened so that it may bear the forces generated under full power of the motor. Hatches for the electronics and payload must also be cut out of the top of the fuselage to allow easy access to ease payload loading and removal.

For the fuselage lay-up, the molds must be waxed to allow easy removal of the finished fuselage. After waxing the molds, the epoxy will be mixed and then spread in a thin layer into the mold to allow for a smooth finished surface. After allowing the thin layer, or gel coat, to harden for an hour the first layer of fiberglass and stick to the middle layer of Spyder foam. The Spyder foam will be applied to the fuselage skin. The final layer of fiberglass will be applied over the Spyder foam and will be wet out with epoxy. Finally, the two completed fuselage halves will be connected with internal layers of fiberglass. After being joined, the fuselage will be reinserted into the mold to ensure a proper shape. A thin dowel will be run inside the fuselage to ensure that the composite sandwich is against the mold surface. The fuselage will be allowed to harden overnight and then it will be removed from the mold and trimmed to size.

Wing and Tail Construction

A molded construction technique was chosen for the wings. A simple core/fiber wing was used for the plane empennage. Very accurate wing sections were required for this project. Composite wing construction was chosen as the most convenient, cost-effective, accurate, and timely method available. Composite construction is the least labor intensive for the desired amount of accuracy. Drawings for the wing templates were generated with the Compufoil Professional Series computer program. This program has a large airfoil database, including the Selig SD 7032 and SD 7037 airfoil coordinates. Compufoil accounts for the thickness of the fiberglass and the loss of foam due to the cutting procedure and corrects the drawings accordingly. Templates were made by adhering these drawings to particle board and carefully sanding the patterns to shape.

A Tekoa Feather Cut foam cutting system was used to make wing cores from Dow Blue Board insulation foam. The foam cutter uses a high resistance nickel chromate wire which is heated upon application of a voltage across the two ends. This wire is put in tension across the ends of a bow and traces the path of the template to get the desired wing shape. The heated wire is pulled through the foam block by a system of weights and pulleys. Once the wire reaches a critical temperature, the wire melts the foam and can be pulled through, leaving a gap of nearly the same thickness as the wire diameter. The voltage across the wire and amount of tension used alter the cutting speed and surface finish of the foam. This method of developing the wing shape is very accurate and gives a smooth surface.

Any imperfections in the foam cores were filled with Dow vinyl spackling compound and sanded smooth. The foam wing cores were then prepped for a composite lay-up to make the mold. First, multiple layers of composite cloth were cut slightly oversized so that they could fully wrap around the airfoil section, leaving excess cloth. Mylar sheets were cut to the size of the airfoil section to fall just short of the leading edge, since the mylar cannot handle the extreme curvature. Part-All was used to wax the mylars and prevent sticking to the composite fiber and epoxy. Peel-ply is a fabric similar to fiberglass except it has the unique property of being able to absorb epoxy while still retaining the ability to separate from cured fiberglass fibers. Strips of this material were placed on the exterior edges of the mylar to act as both a hinge at the wing trailing edge and an excess epoxy absorber. The mylar gives the desired part a smooth finish and eliminates any waviness imperfections. West Systems marine epoxy consisting of resin and hardener was mixed to correct proportions. Composite cloth sheets were then placed one at a time on the mylar sheets and gently smoothed out. Each cloth was saturated with a coating of the epoxy, and excess epoxy was removed with a plastic applicator to minimize the weight and achieve a proper mass fraction ratio. The wing sections are thus very strong and failure resistant. The mylars covered with composite cloth were folded at the peel-ply hinge, and the trailing edge of the foam wing core was inserted slightly ahead of the fold-line in the mylars.

This assembly was then placed inside a vacuum bag and surrounded with bleeder cloth to make sure all the air was evacuated. Vacuum-bagging is a system where a composite lay-up is inserted into a sealed enclosure and a pump is used to evacuate air to a differential pressure of at least 16 inches of Mercury (Hg). This creates a suction on the part inside, effectively giving a uniform force over the entire surface. Excess epoxy then flows away from the part and ensures an even epoxy distribution. The composite cloth is also snugly attached to the foam core, giving a close fit.

protection and latex gloves were used whenever epoxy was handled. A gas mask with dusting chemical filters was worn when the fiberglass and carbon fiber wings were being sanded.

This procedure effectively ended the construction of the wing mold plug. For the aircraft tails, the procedure given above was followed. However, instead of producing mold plugs, the final tail sections were made. The smaller dimensions and ease of construction for the empennage was deemed to not require molded construction.

The wing plug was used to make a mold for final part construction. A procedure similar to that of the fuselage construction was followed. The plug was used to make two mold halves. These molds were utilized to make the final wing. The finish, accuracy, and weight savings of the molded wing construction justify the use this more complicated fabrication technique.

The wing spar designed for structural load carrying and ribs in the airfoil shape were constructed to a high degree of accuracy. One-half of the wing was then completed in one of the mold halves. Fiberglass weave (1.5 oz. cloth) was applied to the mold in directions normal and parallel to the span. Then cloth was applied in both 45 degree orientations to stiffen the wing in torsion. The wing spar and ribs were inserted into the half wing shell and affixed with epoxy. The opposite wing half was then built in the other mold half. The two portions were adjoined with epoxy along the spar and ribs and fiberglass hinges on the leading and trailing edges of the wing. The entire combination was allowed to dry and sanded to shape. This technique allows for a lighter and stronger wing than the simple foam-filled core wings that were used in preliminary design.

References

Beer, F. and Johnston, E., *Mechanics of Materials, 2nd Edition*. New York: McGraw-Hill, Inc. 1992.

Chavant Clay Modeling Products, Inc., 42 West Street, Red Bank, NJ 07701-1119

Etkin, B., and Reid, L. D., *Dynamics of Flight*. New York: John Wiley & Sons, 1996.

Rau, Fred., *Motorcyle Consumer News*, Volume 28, Number 1, January 1997.

Pazmany, L., *Light Airplane Design*. San Diego, Ca.: L. Pazmany, 1963.

Selig, M. S. et. al. *Summary of Low-Speed Airfoil Data*. Virginia Beach, Va.: SoarTech Publications, 1995.

Simons, Martin., *Model Aircraft Aerodynamics*. London: Argus Books, 1993.

United States Gypsum Company, Industrial Gypsum Division, 125 South Franklin Street, Chicago, IL 60606-4678

Figure 2

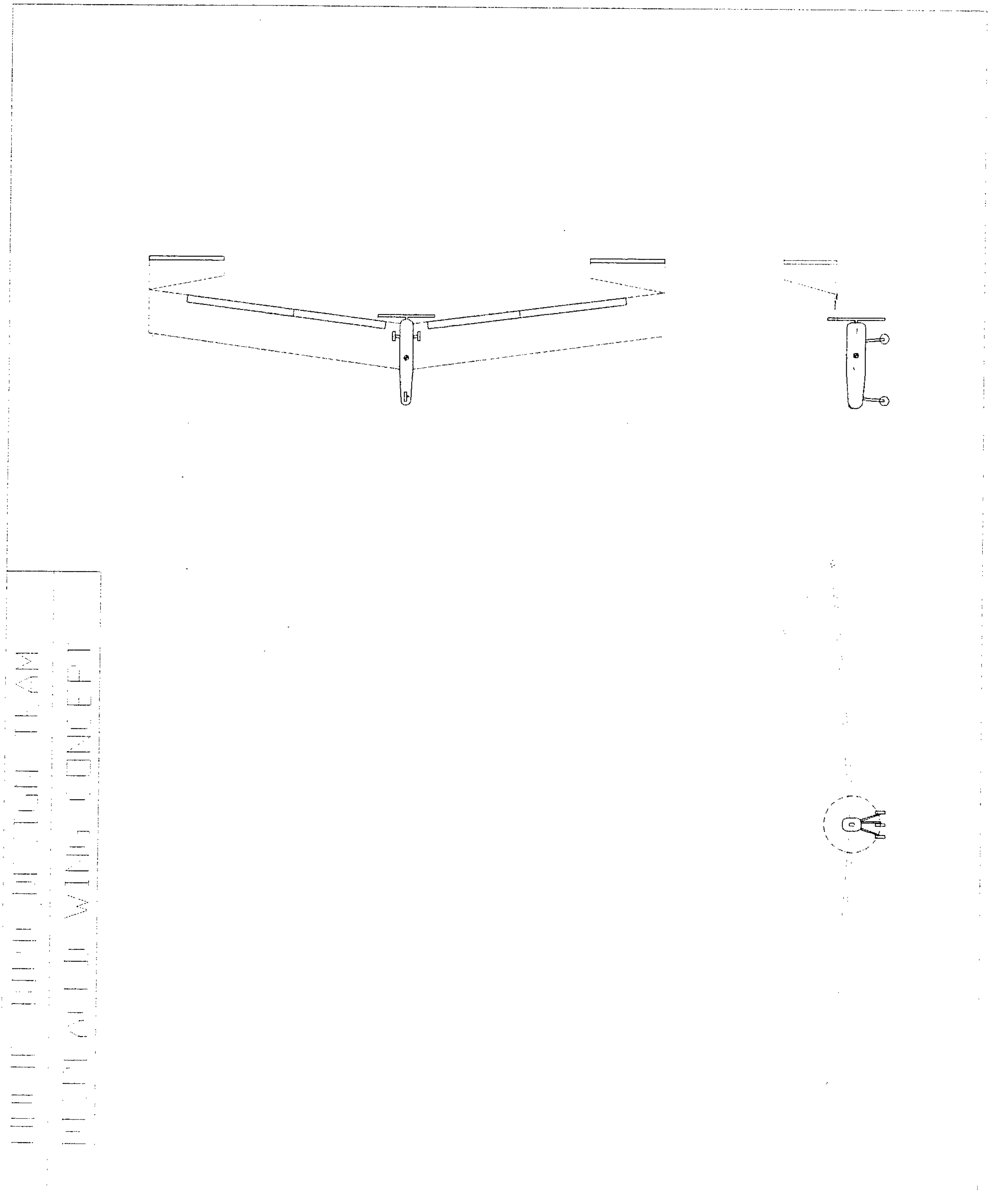


Figure 3

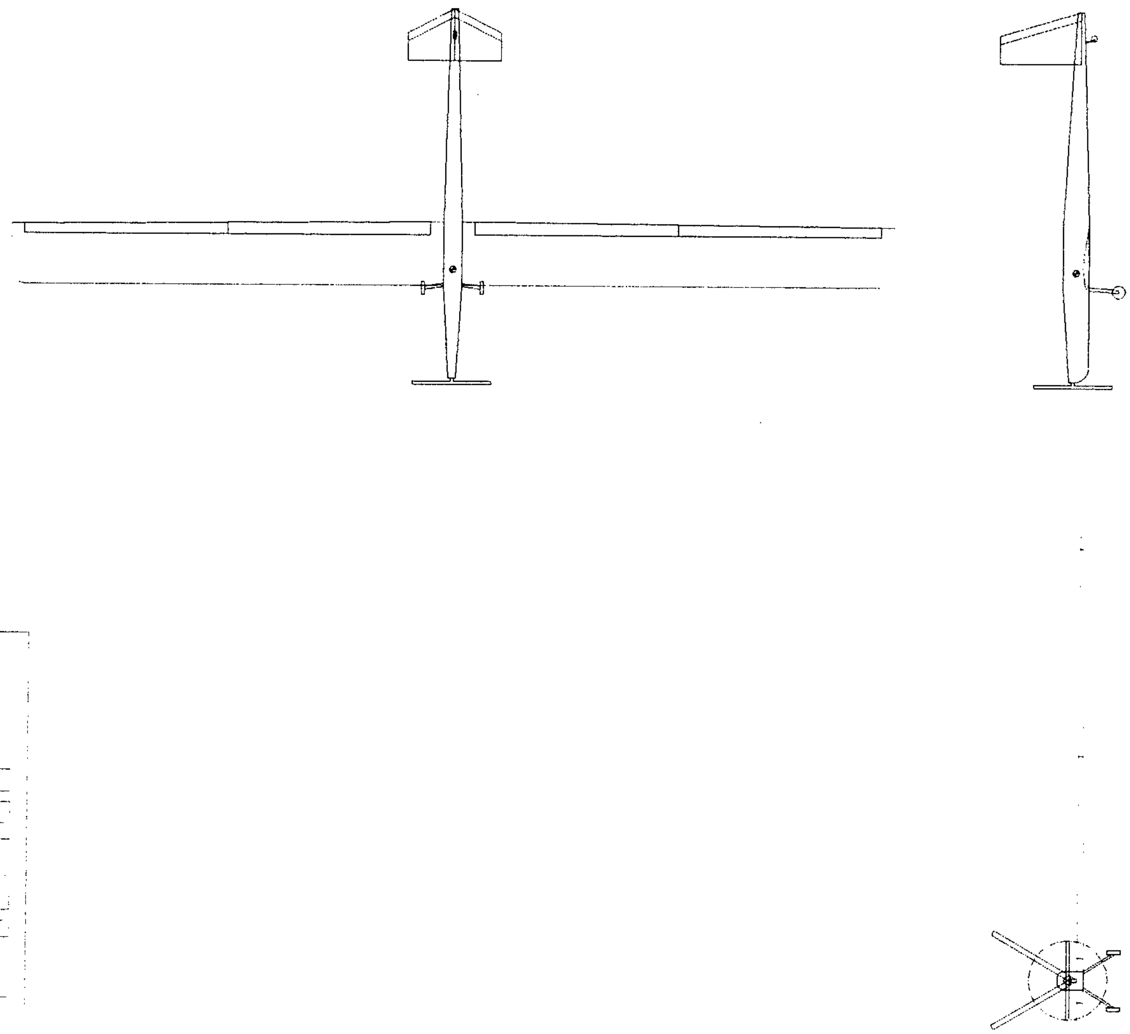


Figure 4

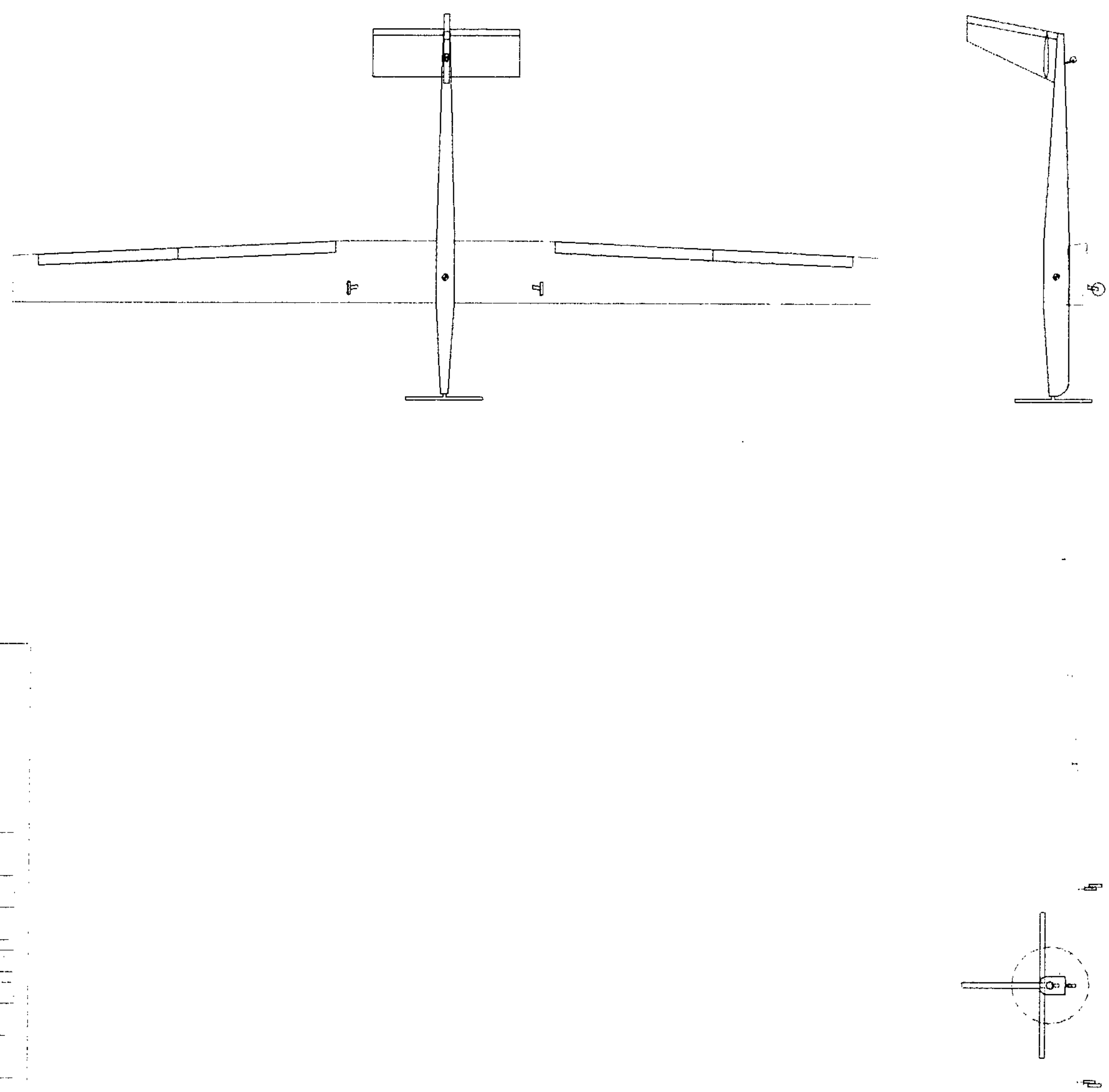


Figure 5

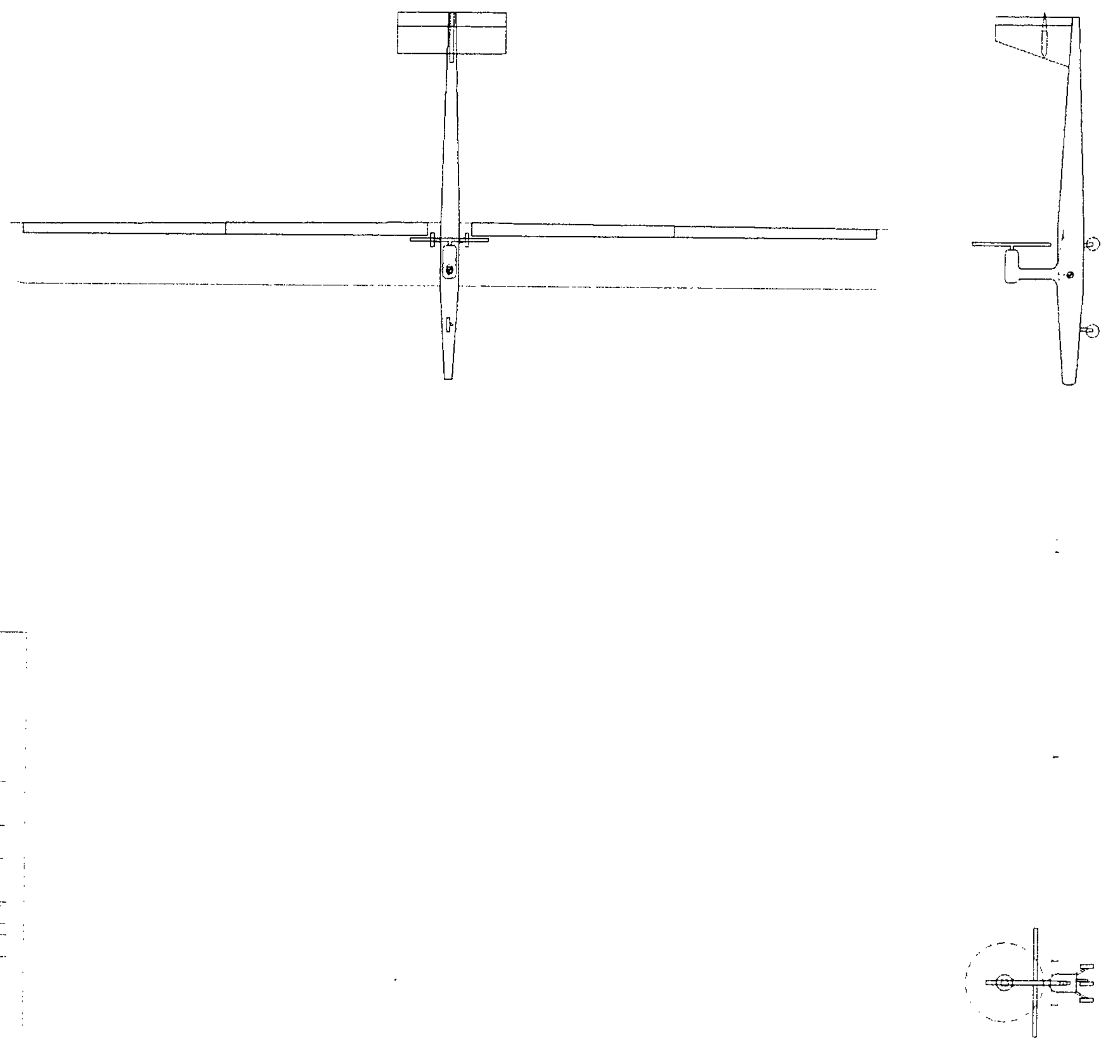


Figure 6

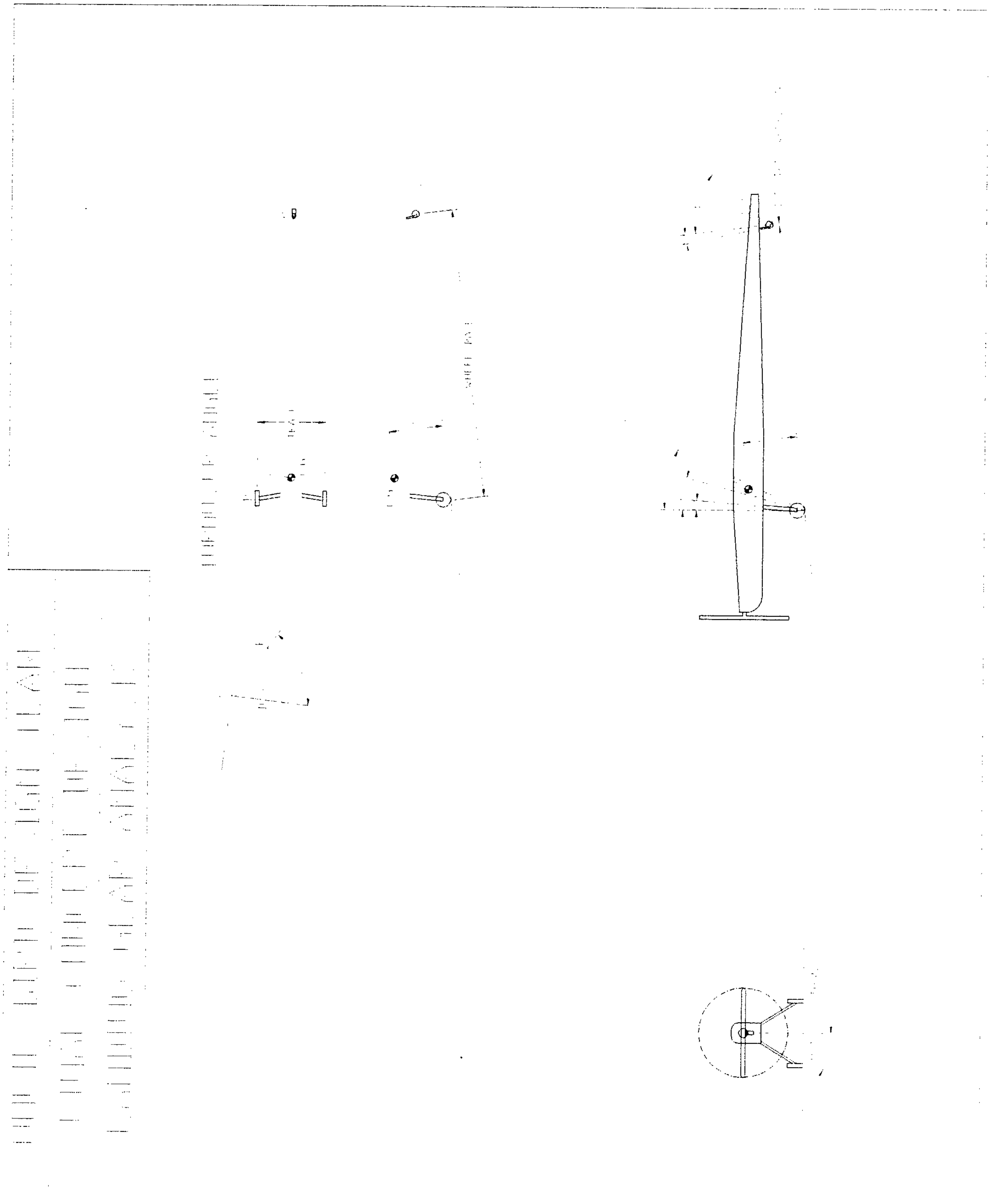
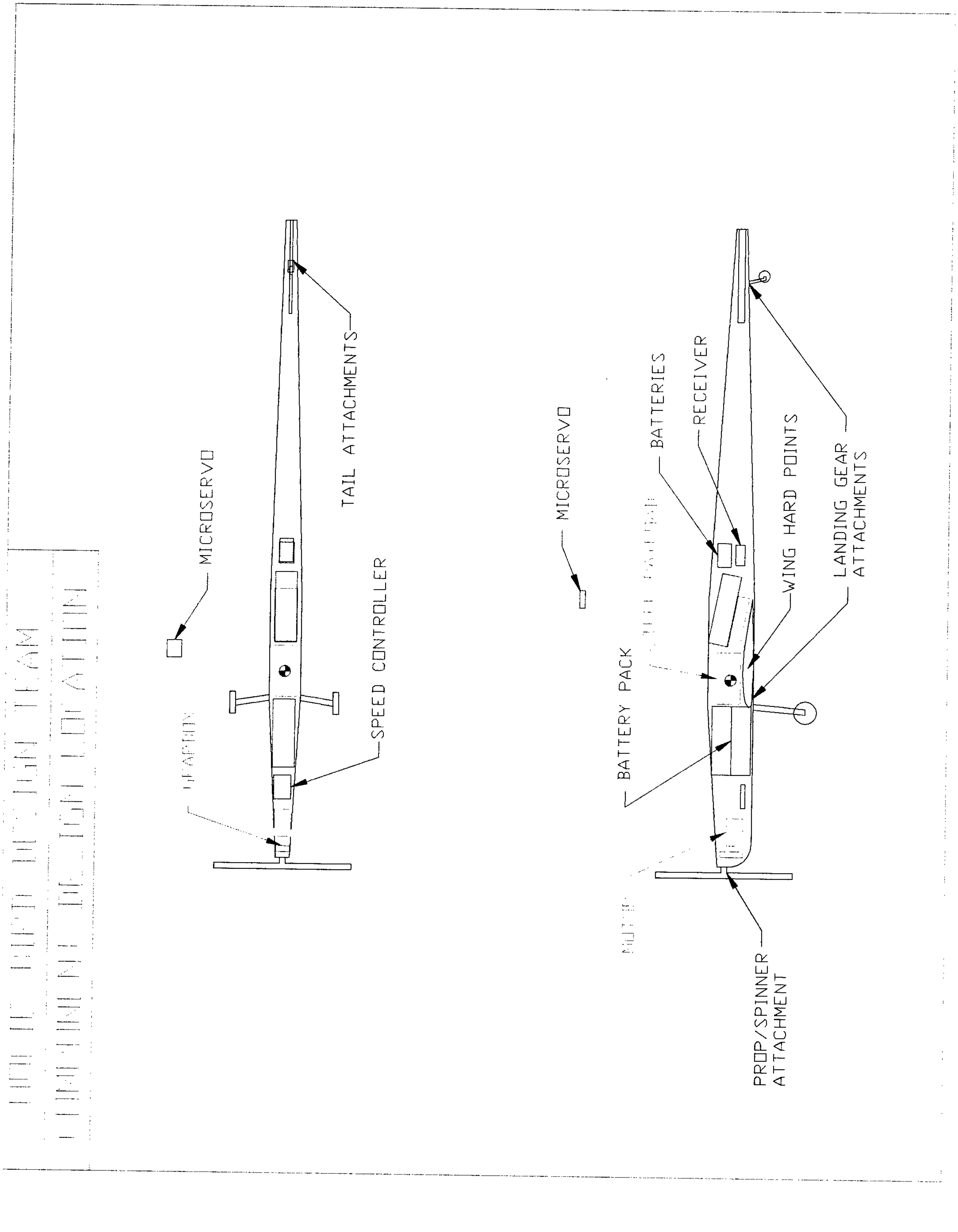
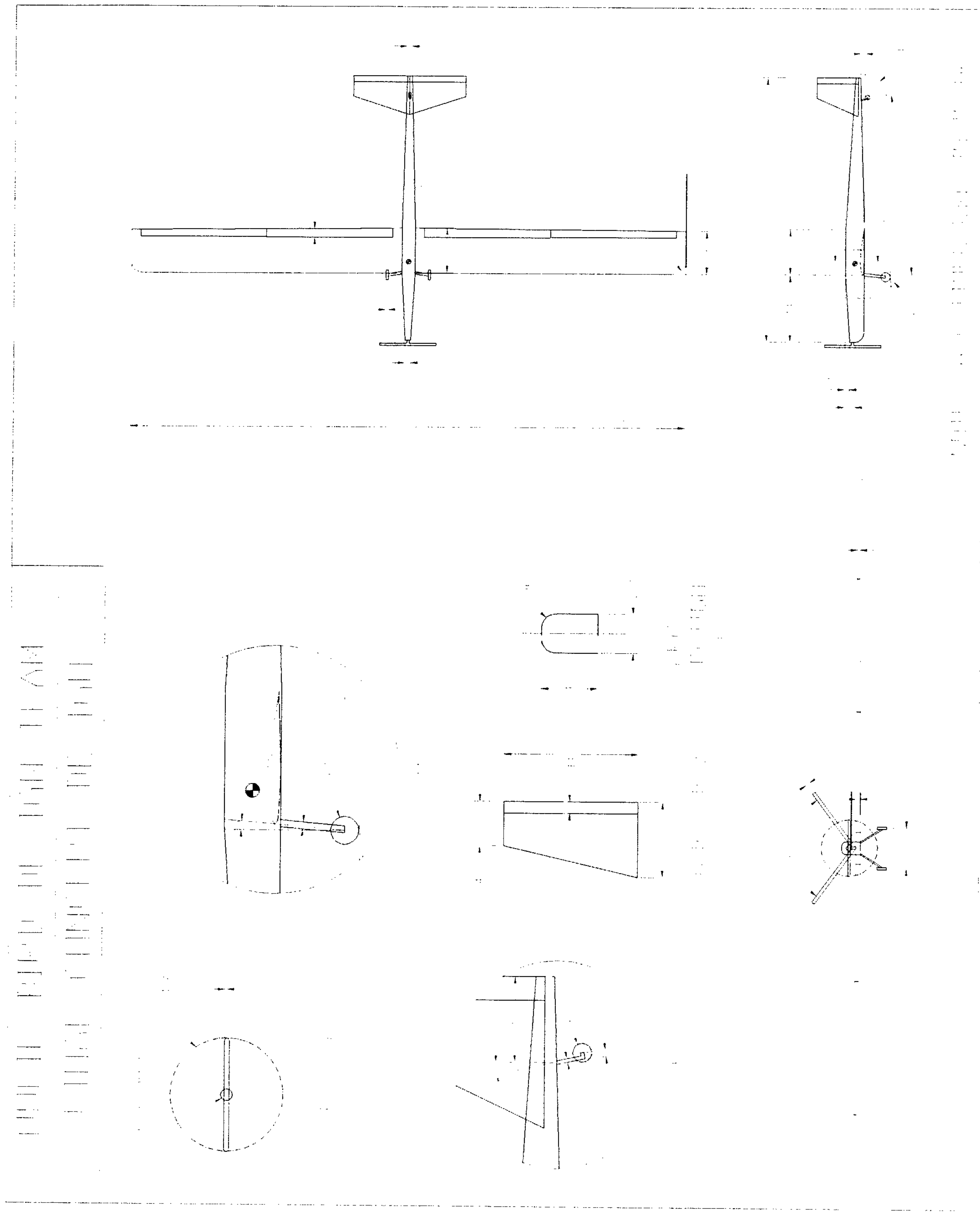


Figure 16



TAIL ATTACHMENTS
MICROSERVO
SPEED CONTROLLER
MICROSERVO
BATTERIES
RECEIVER
WING HARD POINTS
LANDING GEAR ATTACHMENTS
BATTERY PACK
PROP/SPINNER ATTACHMENT

Figure 17



GRAND MASTER B

UCLA AIAA Student Chapter

Proposal For the Design/Build/Fly Student Competition

1996/97

Executive Summary

Project Grand Master B was driven mainly by time constraints. Actual entry into the competition was very close to the deadline, that left little time for each design phases. Therefore all design features and innovations were done with following guiding principles: fast, cheap, recoverability. During each design phase the process that would garner the fastest results were given top consideration. Since the late entry also leaves little room for fundraising, priority was also given to inexpensive ways of accomplishing the task. Fast and cheap however, does not mean a shoddy design. In order to accommodate future design changes as well as possible damage to the plane, recoverability from ruin or alterations is an integral part of the design process.

Initial design was very subjective. The only design constraints were those set by contest rules. The only limitation was weight (including payload weight and battery weight). There were no restrictions regarding plane size, configuration, or type of conventional winged aircraft. Therefore, the main concern in the conceptual design stage was the configuration and planform of the aircraft. Many shapes were considered such as addition of canards, forward swept wings, and delta wings. Each member of the team drew concept pictures which were then weighed by their FOM. After weighing the FOMs from each members' design, the initial configuration was created.

After the general configuration of the plane was chosen, the initial performance estimations and vehicle sizing phase began. Performance estimations were mainly done from members experience with RC aircrafts. Major areas taken into consideration was the idea of a modular design for the aircraft so changes or repairs to the plane could be made quickly and efficiently. Also since this plane needs to survive a cross country trip

to the competition site, a modular design makes shipping easier. Rough blueprints were drawn in this stage to help visualize and size the aircraft. Design alternatives ranged from exotic materials for structure, to ducted fan propulsion.

During the Detail Design phase knowledge gained from aircraft design classes helped to finalize performance predictions. Final blueprints were created using traditional drafting techniques, including machinists conventions to assist in the manufacturing stage. Actual engine output and battery power obtained from vendor's catalogue were used to satisfy theoretical speed estimates and power requirements. Specific details for structures were included in the blueprint. Payload, and batteries were placed according to Center of gravity Calculations.

After the detailed design phase, the aircraft was completely defined and ready to be manufactured. Each phase of the design process was an intense brainstorming session that followed a general timeline for each development stage.

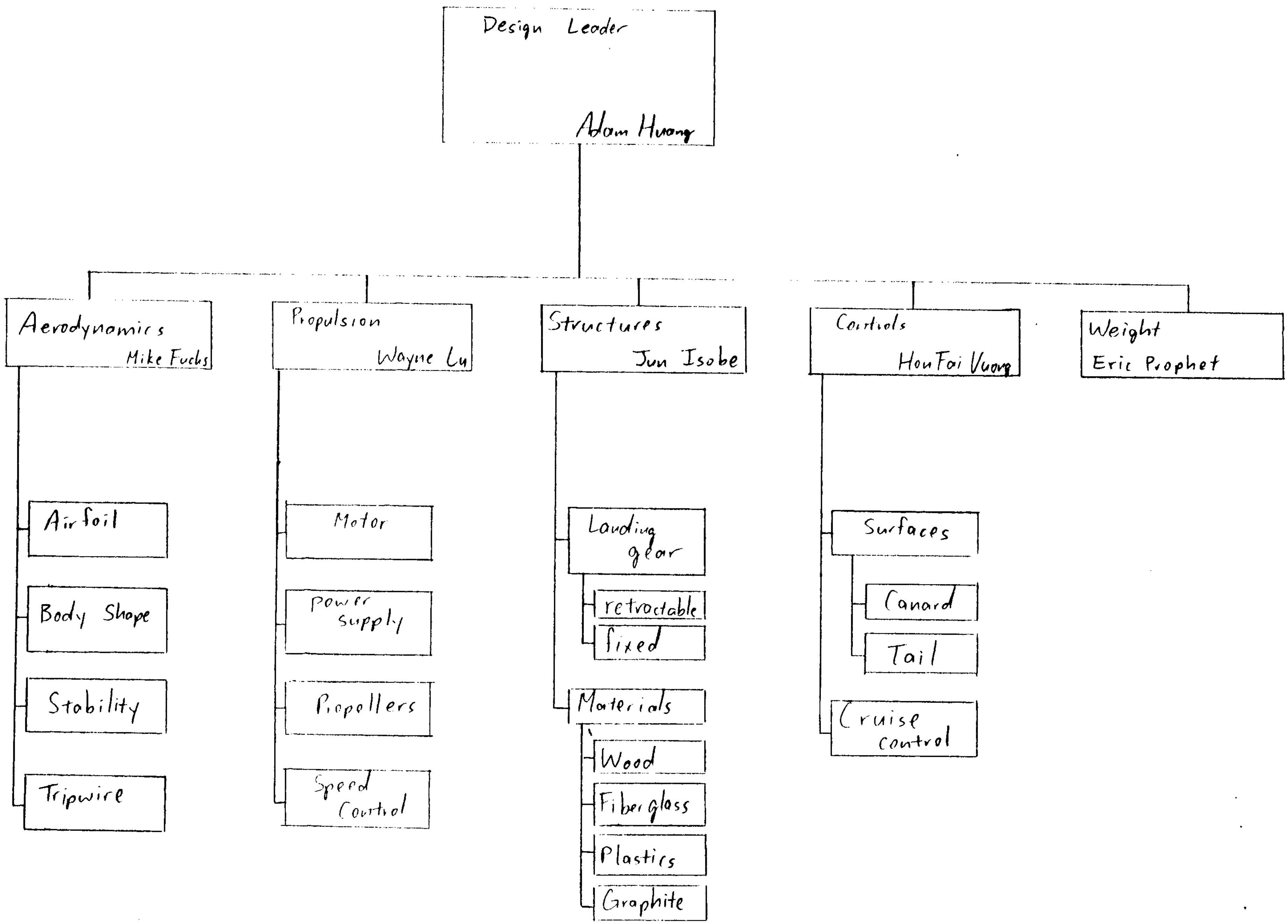
Management Summary

The architecture of the design team is based on the top-down team structure. A design leader heads the team, while groups below that concentrate on its own area of study. This team is broken up into five major areas of study: Aerodynamics, Structures, Controls, Propulsion, and Weights. Each area is headed by a section leader that reports back to the design leader. Each week the section leader meets with the design leader to work out the schedule for the coming week and also to review progress.

Section leaders are responsible for "Assignments of the Week" for members of the section. Members have the freedom to plan their own schedule so long as the assignments are completed by the following meeting. This provided an efficient team structure while still allowing student members to complete personal tasks.

Following the design stages the same teams were retained for the Manufacturing stage. While the general management architecture remained, schedule control was more relaxed, allowing student members to work on their perspective parts of manufacturing whenever time was available. This also gives more time to shop around for parts and materials.

Detailed list of design personnel for each study area as well as milestone chart are as follows.


Design Leader
Adam Huang
Aerodynamics
Mike Fuchs
Propulsion
Wayne Lu
Structures
Jun Isobe
Controls
HouFai Vuong
Weight
Eric Prophet
Airfoil
Body Shape
Stability
Tripwire
Motor
power supply
Propellers
Speed Control
Landing gear
retractable
fixed
Materials
Wood
Fiberglass
Plastics
Graphite
Surfaces
Canard
Tail
Cruise control

Project Grand Master B

Milestone Chart

Date	Official schedule	Actual Schedule
9/23/96	Quarter Begins	Quarter Begins
9/26	Instruction Begins	Instruction Begins
10/15	Notice of Intent to Compete	
10/30		Notice of Intent to Compete
11/1	Begin Conceptual Design Phase	Begin Conceptual Design Phase
11/9		Receive final rules from Competition Committee
11/14	End Conceptual Design Phase	
11/15	First Draft Conceptual Design Report	
11/16	Begin Preliminary Design Phase	
12/1	Begin research for pricing and material availability	
12/15	End Preliminary Design Phase Begin Detailed Design Phase	
12/16	First Draft Preliminary Design Report	End Conceptual Design Phase Begin Preliminary Design Phase First Draft Conceptual Design Report
12/22	Begin Manufacturing Phase First Draft Manufacturing Report	Begin research for pricing and material availability
1/1/97	End Detailed Design Phase First Draft Detailed Design Report Manufacturing Continues	End Preliminary Design Phase Begin Detailed Design Phase First Draft Preliminary Design Report

2/1/97

	First Draft of Proposal Report	Manufacturing begins.
2/15		End Detailed Design Phase First Draft Detailed Design Report
3/1	Final Draft of Proposal Report	First Draft of Proposal Report
3/17	Written reports for Proposal Phase due	Written reports for Proposal Phase due
3/23	Begin Flight Test	

Conceptual Design

The main purpose of the conceptual design stage was to come up with the general configuration of the aircraft. However, because many of the team members were unfamiliar with both RC aircrafts and figures of merit (FOM) of design concepts, this phase also doubled as a learning stage for those members.

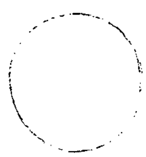
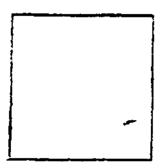
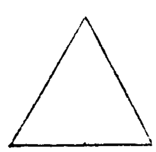
To facilitate the process of configuration selection, the aircraft was split up into four sections: body, wing, tail, and landing gear. This very general outline helps the members to focus on each section of the aircraft. At this time, many different possible shapes for each section of the aircraft were presented to the members. For example: possible body shapes include the cylindrical shape, the square box, a wing body shape, or a triangular tube. The wing is arguably the most important section of the airplane. Different wing placement positions were discussed, such as: a high, medium, or low placement with respect to the body, or bi-plane/tri-plane configurations. Wing shape was also taken into consideration. Members studied the straight, the forward-swept, the parabolic, and the delta shaped wing. Exotic additions such as winglets, fences, leading edge root extensions, and canards were also part of alternatives investigated. Tail configurations studied included single tail, T-tail, H-tail, and V-tail. The landing gear is very important not only because the aircraft needs to land, but often the majority of the drag force stems from a fixed landing gear. Concepts for fixed and retractable landing gear carriages were examined in detail

At this point, members were asked to draw what they thought the airplane should look like. This is both to get an idea of what each member's feel for the airplane is, as well as get a consensus of what members felt were important considerations to be taken into account in the final design. In designing their own aircraft, members were asked to keep in mind the design parameters: ease of manufacturing, ability to accommodate the payload, and innovative design configurations. All of the members were able to come up with very unique designs ranging from forward swept wings, to delta wings with ducted fan engines. It is from this pool of ideas that a preliminary configuration was chosen from.

While screening the different concepts, the three design parameters were also kept in mind. The process is not purely democratic. For example, most designs include a wing-body shape for the main body. The wing body shaped, similar to the shape used on the F-16, is a very aerodynamic shape. Its gently sloped contour will allow a higher percentage of laminar flows across the surface, as opposed to separation of boundary layers for shapes with edges. However, making a wing-body shaped aircraft is very difficult and time consuming regardless of building material. Therefore, a compromise was reached where a trapezoidal shape was adopted for the main body shape. For low Reynolds numbers associated with this competition, it was decided that wings with large aspect ratio is desired to reduce the induced drag. An additional reason for the high aspect ratio was due to the high payload to max gross weight ratio predicted. A canted wing section was later added to achieve better stability in turning flights.

Pusher engine configuration was selected since it is considered to be more efficient than a tractor engine configuration. A twin boom tail configuration was selected to accommodate the pusher engine configuration while maintaining static stability. The extended tail allows the wing to be closer to the engine, thus moving the aerodynamic center behind the center of gravity. By making the tail a high H configuration, there's enough room left in the back of the fuselage for a pusher engine. Finally, a fixed landing gear was chosen because of the difficulty in building a retractable wheel. There are commercial kits available, but then these assemblies tend to be very heavy and complex. Instead, fairings will be installed on the wheels to reduce drag.

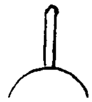
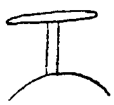
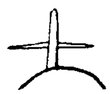
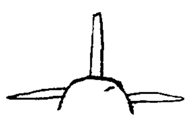
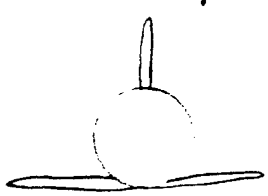
Aircraft Component Ranking Chart

Aircraft Component	Ease of Manufacturing (easy 1---5 hard)	Space Available (less 1---5 more)	Innovative Configuration (no 1---5 yes)
Body Shape			
 Cylindrical	3	3	2
 Box	1	4	1
 Triangular	2	1	3
 Wing Body	5	3	5

Aircraft Component Ranking Chart

Aircraft Component	Ease of Manufacturing easy 1---5 hard	Space Available less 1---5 more	Innovative Configuration no 1---5 yes
Wing Shape			
straight	1	1	1
swept	1	2	2
forward	2	1	5
Parabolic	5	2	3
Delta	2	2	5
wing placement			
high	1	1	1
mid	2	1	1
Low	1	1	1
Bi-Plane	3	2	2
Tri-Plane	4	3	3

Aircraft Component Ranking Chart

Aircraft Component	Ease of Manufacturing easy 1---5 hard	Space Available less 1---5 more	Innovative Configuration no 1---5 yes
Tail Unit			
 single	1	1	1
 T-tail	1	2	2
 Mid	2	2	2
 Mid Fuselage	1	2	2
 low	1	2	2
 V-tail	1	2	4
 H-tail	2	2	3

Preliminary Design

The Grand Master B's configuration had been set at a twin boom pusher powered plane. The next item that needed to be resolved was a sense of scale. The first element that was taken into consideration was the payload. Competition rules specify that the airplane must carry a 7.5lb steel payload. However, it did not specify what type of steel. Different types of steel has different density, and thus different volume for the same weight. The Weights group investigated a variety of steel, from ferritic steel at 7.50 g/cm^3, to Austenitic steel at 8.00 g/cm^3. To find a generic size for the payload, the average of the densities were used, resulting in a volume of 438 cm^3. The battery weight was constrained at 2.5 lb, together with the engine at around 1lb and the payload, this consists of 80% the weight of the main body. To accommodate this package, an initial estimate of the main body length was set at 3 feet. The basis for this length is so that the payload as well as other equipment in the airplane can be occasionally removed and replaced. Such a large aircraft not only allows much easier access in removing parts, but also leaves room for further development in the future.

A few criterias defined the size of the wing. First of all, the wing must have large enough a surface area to create at least 10 lb. of lift. Secondly, it must be strong enough so that 10 lb. of loading in the center will not cause failure of the wing. From these considerations, a total length of 10 ft was arrived at. Finally, for the remainder of the airplane, we set a gross weight limit of 20 lbs.

A NACA airfoil was selected to fill the requirement. The specific airfoil chosen was NACA 4915, and once that was determined another crucial bit of the performance parameters was established. For the wings, it was determined that a prefabricated leading edge was the best option, and there would be two spars made of spruce or similar material running along the span. One of these spars would be along the upper and one along the lower surface of the wing.

Due to the large size of the airplane, an engine with good performance was required to be found. The decision came down to two finalists, the Ultra 1800-3, and the Mega S-7. Decision was made to purchase the Mega S-7 because it was a more cost efficient engine and produced greater performance. The propeller diameter is fixed based on the engine chosen. Manufacturers usually included a suggested optimal propeller to be used with the engine. The engine chosen has a maximum thrust of 68 oz as listed in the vendor catalogue. Based on the weight estimate of 15 lb., the calculated cruise speed is 30 mph.

After deciding on the engine, the expected mission duration could be established. Under nominal power, a 6 to 8 minute flight was a reasonable expectation, but it would be difficult to make an educated prediction until the final configuration was determined. Unexpected difficulties in the landing gear or tail structure could dramatically effect the drag on the plane, and no estimation could be exact without that crucial data. In a worst case scenario, a flight time of approximately 4 minutes could be achieved with the throttle at its maximum position. The prospect of such a short flight is a rather frightening one, but it would still be an accomplishment to have such a project get off the ground.

In the initial stages of landing gear design, several options were discussed. The most intriguing option involved a suspension system to prevent damage from a hard landing. This suspension system would incorporate oil damped shock absorbers of the type found on remote controlled cars. This system would reside entirely internally, and therefore not incur any extra drag. The potential complexity and unproven nature of the design were two major reasons why it was abandoned. In addition, the weight and cost of such a system made a more conventional, bent wire assembly the better choice

An interesting device was suggested for improving the characteristics of the wing. This device utilized a tripwire in front of the leading edge. The wire is used to increase the turbulence of the flow over the wing, and thus increase the angle of attack that would induce stall. This is due to the fact that a turbulent boundary layer will separate later than a laminar boundary layer. Since flow separation results in drag and loss of lift a delayed flow separation will have final effect of less pressure drag.

The body is composed of ribs cut from 1/16" sheets of balsa holding on a skin that is made of 1/16" thick sheets of balsa for the sides and top, and a 1/8" thick sheet along the bottom. There are two cuts made along the bottom surface, one for the main landing gear and the other for the nose gear. Along the top, there are 3 access panels that are connected with hinges, and a fourth is created by the notch that accommodates the wing. The body also includes 1/4" by 1/4" spruce longerongs at the corners that connect to each rib and are bonded to all the surrounding surfaces. These increase the stiffness of the body and help transmit the loads that the plane encounters during flight and landings.

The tail booms were a potential source of trouble, and their design was the subject of some debate. A suggestion that arose regarding these booms was the utilization brass tubing. The advantages of such construction would be ease of manufacturing, relative cost efficiency, modularity, and strength without impeding the operation of the rear control surfaces. Since the tubes would require minimal machining, utilizing the tubing would be easy and take very little time. A thinner tube and a thicker tube are slide into one another, and then they are fastened using screws. This allows the easy disassembly of the rather bulky tail section for cross country travel.

The booms are hollow inside, allowing the servo controls for the tail surfaces to be placed in the main wing without potential obstruction. Although hollow, the brass is still rigid enough to support the booms, even at the length of approximately three feet. A disadvantage of the brass boom concept is that brass is a rather heavy material, and the weight is all aft of the center of gravity. Luckily, the pod lies almost completely forward of the center of gravity, and that allows us some cushion in that respect.

Detail Design:

This phase of the design was found to be the most difficult due to the level of skills required for a successful design. To most of the members, the theories and the mathematics involved were quite straight forward. However, field experiences with R/C aircrafts was considered to be extremely important. It not only provides an reality check against calculations, but some design topics really do not have any good theoretical relations. This section of the design is separated into three basic categories of aircraft performance, components, and structures. The performance section was accomplished mostly through theoretical methods while the structures and components section used field experiences as guidelines.

Due to the highly iterative nature involved in performance calculations, quick and easy programming is the key to a successful design. Equations obtained through research, class-works, and derivations were programmed into spread sheets and then iterated. Central to the performance calculation was the drag estimations. Theoretically, drag estimations were broken up into two sections: induced drag and parasitic drag. The induced drag was calculated through the Oswald efficiency correction method. The flat plate method was used to calculate the parasitic drag. However, it should be noted that "parasitic" drag is defined to be the total drag subtracting the induced drag. Since the free stream velocities involved in typical R/C aircrafts are quite low, wave drags and mach number corrections were eliminated from all design calculations. The resulting total drag was then used for engine sizing considerations.

Typically, in aircraft designs, the engine performance must satisfied numerous flight performances ranging from top speed to stall speed. The only specifications for this competition are the takeoff distance and range . This made engine sizing considerably easier. After the thrust required for takeoff was calculated, a few candidate engines were selected. Range performance is then checked with all possible battery combinations with their total voltage and ampere hours calculated.

The main variable that ties all of the performance calculations was the aircraft weight. Thus, engineering intuitions, experience, and extreme details used for weight calculation must be incorporated. All final design parameters must stay within the weight specified. This is the most important part since different final aircraft weight would heavily degrade the flight performance. The weight for the components used in the final design are as follows:

Components	Weights (lb.)
1. Payload Requirement	7.5000
2. Mega S-7 Electric Motor and Propeller	1.1500
3. Radio Receiver (JR)	0.0625
4. Receiver Battery Pack	0.2063
5. Servo-Motors (5)	0.4594
6. Speed Control (AstroFlight-204D)	0.0245
7. Battery Pack for Motor	2.3750
8. Landing Gear	0.7500

9.	Flight Structure	2.2500
10.	Control linkages and wiring	0.3000
11.	Maximum allowable misc. weights	0.9223
	Total :	**16.0000 lb.**

In terms of the structural design, the fuselage and the wing is analyzed separately due to different requirements associated with it. All structural designs were done with accessibility, strength, and weight taken into consideration. For the fuselage, it must have ease of access to the payload, radio receiver, and batteries. Four access points were designed for this purpose. Three hinged access panels are situated along the top of the fuselage. The top mounted wing will provide the forth access point when the wing is removed. Because of the high sink rate usually associated with low power R/C aircrafts, the fuselage must be able to absorb the remaining shock from the landing gears. A sixteen pound aircraft with a sinking rate of seven to ten feet per second is quite considerable. Structural reinforcement is incorporated on the main landing gear (tri-cycle configuration). Although structural integrity is heavily taxed by the access points, the structure was designed with enough rigidity to prevent anticipated flutter. A final note on the fuselage structure is that load calculations were done mainly by experiences with R/C aircraft and intuition.

The wing structure was analyzed with the assumption of the lift distribution following the planform area. Loading densities, shear forces, and moments on the wing was calculated at each spanwise position. However, a more useful illustration to check on the structural loads that the wing may be experiencing is by noting the aspect ratio and the wing loading. A ten feet wing span and 1.2 feet constant chord have an aspect ratio (AR) of 8.33. This is a very reasonable AR with the design wing loading of 21.3 ounces per square feet. Most electrical R/C aircrafts of similar ARs have wing loadings in the range of 20-24 ounces per square feet, while gliders typically have loadings in the 16-20 per square feet range with ARs of 12-14.

To further accommodate possible lift requirements in the aircraft design, a novel idea was introduced to the wing. The main wing section is a six feet span that centers on top of the rear fuselage. The outer two feet section on each section of the wing is designed to have quick attachment points with the main wing. This design allows the possibility of having a wing span that can be changed with ease when conditions dictates. For example, when more lift is demanded, the span can be easily increased with different outer wing sections. Furthermore, due to the lower moments and loadings near the wing tips, the structure there can be a lighter design. In fact, it is now planned to built several different outer wings to allow more flexibility in the competition (the best one to suit the competition day conditions will obviously be used). Thus, the ten feet wing span currently stated is the 'representative' span and may not be the actual one during the competition.

The engine selected was the 413 Watts, 68 ounce thurst Mega S-7 motors (16 cells). The endurance was estimated to be 4.2 minutes at maximum power, or 14 minutes at cruise. This translate to a linear range of 7 miles (36,960 ft-hopefully about 25 laps). The payload to weight ratio stands at 0.459.

Weight (lbs)	16
	16
	16
	16
Dimensions	
Wing	
Wing Span	10
Root Chord	1.2
Tip Chord	1.2
Wing Area	12
Horizontal Tail	
Tail Span	1.5
Root Chord	0.5
Tip Chord	0.5
Tail Area	0.75
Vertical Tail (2)	
Height	0.5
Average Root Chord	0.9
Vertical Tail Area	0.9
Fuselage (w estimated tail boom equilavent dimensions)	
Fuselage length	3.8
Fuselage Width	0.5
Avg "Square" Cross Section Area	0.25
Prop. diam	1

For Parasite Drag Coefficient

Wetted Areas:		**Frontal Areas:**	
WAW=2*1.02*SW	24.48	FAW	1.71
WAHT=2*1.02*SH	1.53	FAHT	0.1125
WAVT=2*1.02*SV	1.836	FAVT	0.108
WAF(Fuselage)	8.1	FAF	0.25
WG(Gear)	0.5	FG	0.2
TWA=	36.446	TFA	2.3805

Parasite Drag Coefficient

Cdo = (CP*TFA+CF*TWA)/ (Sc+Sw)

CP=0.04	0.04
CF=0.003	0.003

Parasite Drag ,CDo=	0.01705

Induced Drag:

Using Equation Cdi = Cl^2 / (Pi*epsilon*AR) =

B w =	10		
B t =	1.5		
W =	15		
F=	0		
1-F =	1		
Sw=	12		
St=	0.75		
Epsilon w	0.88		
Epsilon t	0.88	CD Par=	0.0170465
AR w =	8.3333333		
AR t =	3		

(F*W)^2	((1-F)*W)^2	St*epsilon t*ARt	Sw*epsilon w*ARw	Sw*Pi
0	256	1.98	88	37.699
0	256	1.98	88	37.699
0	256	1.98	88	37.699
0	256	1.98	88	37.699

Engine Thrust (static)=	Pi =	Prop Radius=	Engine HP
3.75	3.1416	0.5	0.3977
3.75	3.1416	0.5	0.4859
3.75	3.1416	0.5	1.0498
3.75	3.1416	0.5	0.4319

Velocity Required

	mph	fps	Density at S.L.	q, dynamic P	CD-I	CD	D (Thrust Required)	w/o CD-I
Stall	20	29.33	0.002378	1.02306844	0.07372517	0.09077167	1.114387569	
Cruise	30	44	0.002378	2.301904	0.014562996	0.031609496	0.873144316	
Top-Speed	84	123.2	0.002378	18.0469274	0.000236929	0.017283429	3.742953499	
Takeoff	24	35.2	0.002378	1.47321856	0.035554191	0.052600691	0.929907763	0.3014

Propeller Efficiency

w=.5(-v+sqrt(v^2+2T/(rho*A))	Eta-i	Eta-p
20.24794	0.59162122	0.50287803
16.573549	0.72638967	0.61743122
7.6710521	0.94138465	0.80017696
18.644705	0.65373187	0.55567209

Engine Required (This is not really needed for our case)

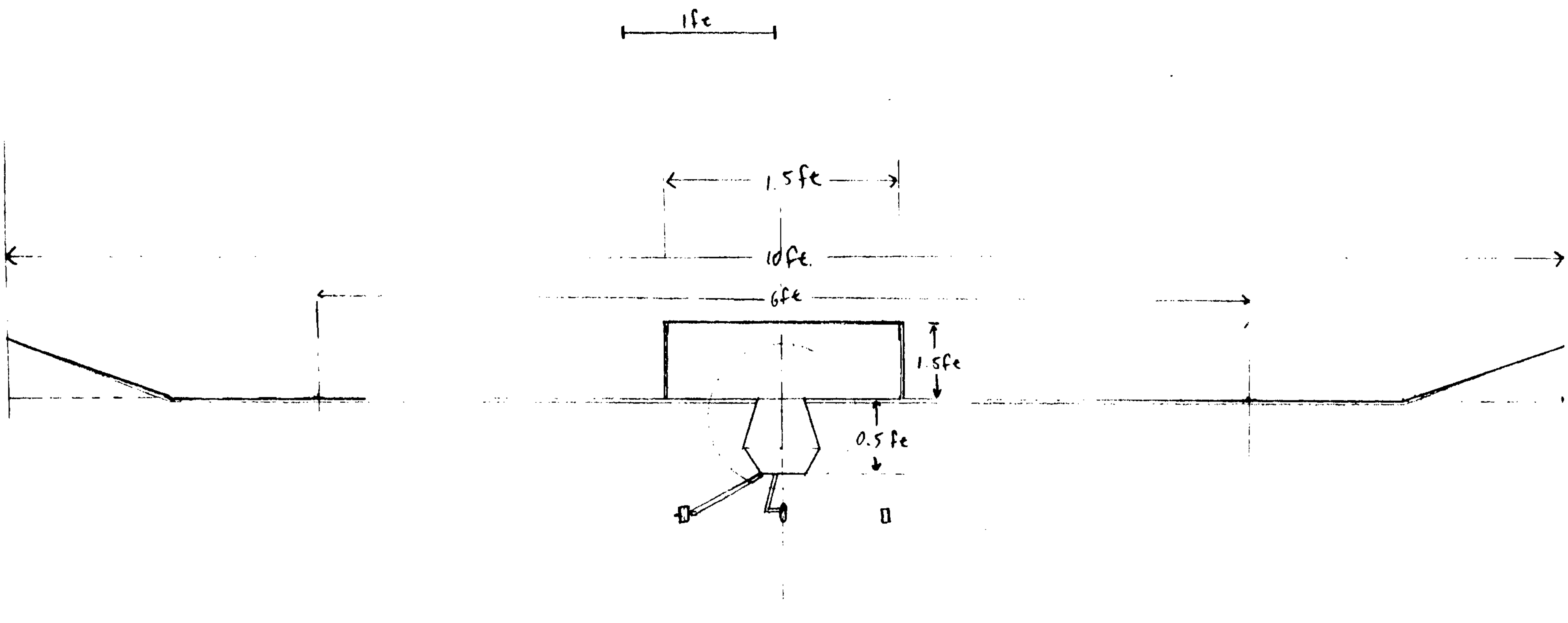
	T=D	HP
Stall	1.1143876	0.12
Cruise	0.8731443	0.11
Top-Speed	3.7429535	1.05
Takeoff	0.9299078	0.11

Lift	
Takeoff	16
Cruise	16
Max	16
.7TO	0

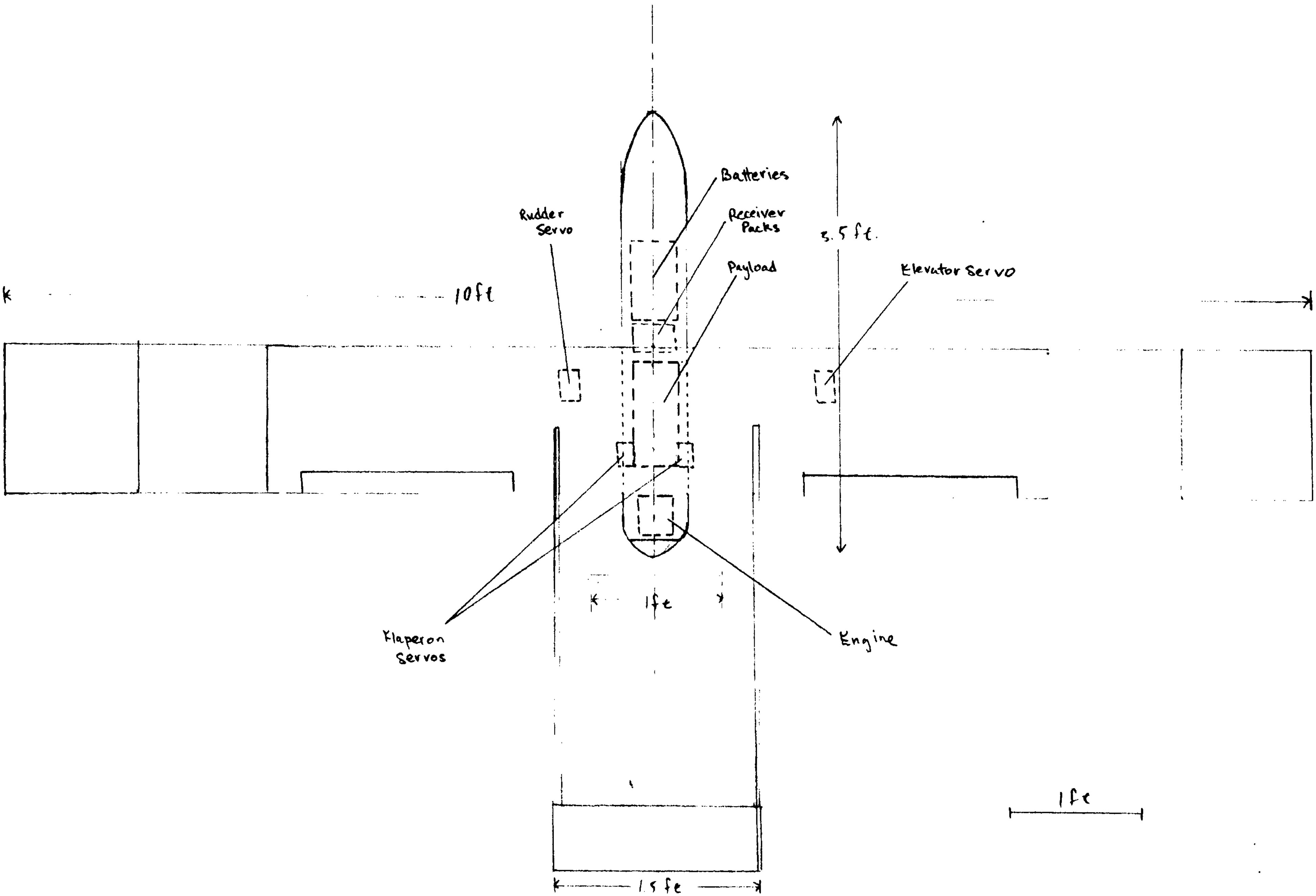
Takeoff	Thrust at TO	Rate of Climb (fps)	feet per min
	4	5.290289457	317.417367

Acceleration		
from stall	5.304170017	
from cruise	5.789672064	
Max	0.014181083	(should be zero)
Takeoff	5.652390733	(w/o induced Drag)
	4.387435627	(w/ induced Drag)

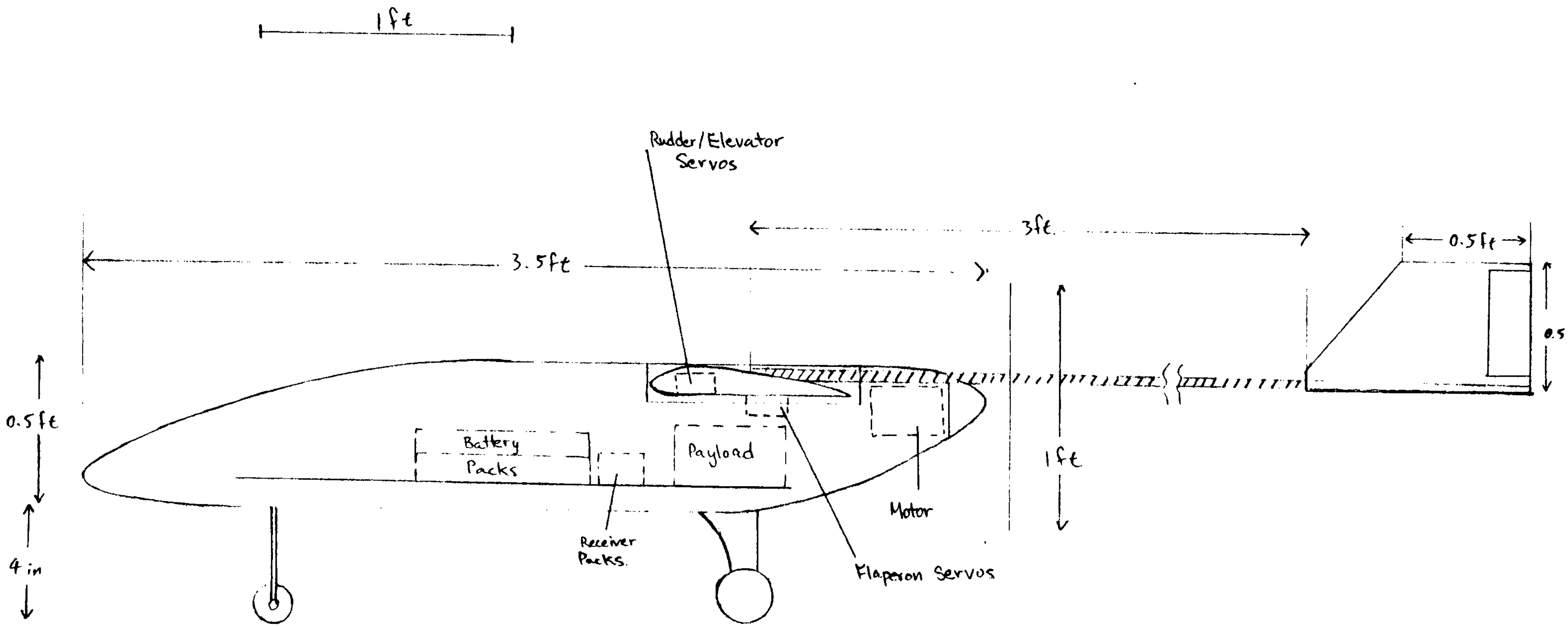
	w/o induced drag	w/ induced drag
SG	109.6031802	141.2032113
Roll Distance	17.6	17.6
Climb Distance	83.2124083	83.2124083
Takeoff Distance	210.4155885	242.0156196



1ft
1.5ft
10ft.
6ft
1.5ft
0.5 ft



Batteries
Receiver Packs
Payload
Rudder Servo
3.5 ft.
Elevator Servo
10ft
1ft
Flaperon Servos
Engine
1ft
1.5ft



1 ft
Rudder/Elevator Servos
3 ft
3.5 ft
0.5 ft
0.5 ft
0.5 ft
1 ft
Battery Packs
Payload
Motor
Receiver Packs.
Flaperon Servos
4 in

Manufacturing Plan

The major preflight manufacturing goals with Project Grandmaster B was to minimize weight, cost, and the engineering/production time. It was often a tug-of-war between these conditions. Production is doubly important for this team because of the short time available for actual construction of the plane. Engineering/production time is very expensive in the aerospace industry, and by including this additional criteria, the team got a more realistic sense of what actual industry deadlines are like.

A myriad of different materials and designs were considered to keep overall weight to an absolute minimum. Carbon fiber, fiberglass, and special plastics were considered along with more traditional materials such as various types of wood for use as structural members of the aircraft. Exotic material would certainly achieve higher performance then traditional materials. However, the team had no experience in dealing with advanced materials in model aircraft. Exotic materials are not only hard to come by, but extremely hard to machine.

Construction of the wing offered additional alternatives for construction material. Foam-core wings were very easy to manufacture and produces a high quality finished product. This concept also offered a very rigid structure for only slightly more weight. The only negative aspect of the foam-core was that it does not allow for much last minute design changes. This being the first built-from-scratch airplane for the team, major errors an oversights were expected. A material that allows flexible design as well as ease of emergency repairs was necessary. After examining all the FOMs for each different material, traditional balsa wood was chosen as the main structural material to be used. It is lightweight, has high relative strength, and most importantly, its ease of use and low cost made balsa still the best choice for this project. Given more time exotic materials might have more merit, but given the short available time the decision was made to stay with traditional wood.

The choice of using balsa wood for the majority of the plane made the manufacturing process much simpler. Balsa is relatively cheap to come by and is readily available in any hobby shop. Balsa is a soft wood, so that manufacturing with it does not require a high level of prior woodworking experience. Tools needed to manipulate balsa is also very simple tools. A saw and a knife is all that's needed to build most of the airplane. Manufacturing time was reduced by more than half if exotic materials had been chosen for the airplane.

The decision to use balsa as the primary structural element means that several high stress components, such as the landing gear assembly and the wing spars, would have to be fabricated using more robust materials. The main landing gear was designed to be modular, and incorporated in such a way that a violent lading would not result in unrecoverable damage to the fuselage. The material chosen for this purpose is aircraft grade birch plywood, which is an extraordinarily stiff material, although heavy, so its use would have to be judicious. The wing spars and longerons between fuselage ribs are either bass wood or spruce, depending on the availability of the materials. the wing spars are held together by 1/16" balsa sheets that connect the top spar to the bottom spar to increase rigidity.

There were two major factors in the production process which increased work efficiency significantly. First, a full-sized blueprint layout was drawn, so all the parts were configured and built to the drawn layout. This way, members can see the exact location where each specific part will be located on the finished product. Measurements were taken directly from the blueprint to avoid any confusion.

The second factor was the modular design used to separately integrate every major section of the aircraft. The same teams from the design stages were kept, and each responsible for manufacturing of their section. Each manufactured section was then assembled into the final product. A long term advantage of this setup is its reparability and transportability. In case the airplane was damaged, the affected section could be separated from the rest of the aircraft and repaired quickly. Our work group structure was based on this modular design.

The Aerodynamics team was in charge of airfoil selection, as well as construction of the wing and main body. The Aero team also worked with the Structures team in determining the placements of spars and load bearing members. The Structures team then had the task of building the landing gear assembly. After the main wing was completed, the Controls group stepped in to attach the control surfaces. The Controls group also handled construction of the tail section. The cruise control was placed with the help of the Propulsion team. The Propulsion group worked on the body after its assembly by the Aero Team, and is responsible for mounting the batteries, motor, and speed control as well as all other equipment that goes inside the body such as the propellor and cruise control. Each piece of the airplane that was finished was examined by the Weights team to look for superfluous structures. The Weights group is also responsible for sanding the surfaces of the all wood aircraft..

At first glance this workforce distribution looks deceivingly like a "production line" architecture. But upon closer inspection, two very distinct advantages stand out.

The benefit of a production line architecture of manufacturing is speed. Work going on simultaneously in different sections gets the job done faster. However, this team's architecture is such that there is still cooperation between each group. For certain sections different groups will have to work together. This helps the members to understand not only just one section of the airplane, but all parts of the airplane, and how each section affects the other. Hence production time is decreased without a corresponding loss in understanding of theories behind the project.

Project Grand Master B

Manufacturing Milestone Chart

Date	Official schedule	Actual Schedule
12/22/96	Manufacturing Begins Layout of main body cross sections	
1/15/97	Body panel installed	
2/1	Begin cutouts of Airfoils	Manufacturing Begins Layout of main body cross sections
2/15	Construction of main wing section	Body Panels Installed
2/22	Construction of Landing Gear	Begin cutouts of Airfoils
3/1	Construction of Tail Section	Construction of Landing Gear
3/8	Installation of aircraft components (servos, batteries, etc.)	Construction of main wing section
3/15	Final installation of remaining Aircraft components (engine, propeller, etc.)	Construction of Tail Section
3/23	Begin Flight Testing	Installation of Aircraft components (servos, batteries, etc.)
3/31		Begin Flight Testing

AIAA Student Design/Build/Fly Competition Addendum

aerospace engineering department

TEXAS A&M UNIVERSITY

Robert "Rip" Rippey III

M. Shea Parks

Jason W. Field

April 14, 1997

TEXAS ENGINEERING EXPERIMENT STATION

AIAA Student Design/Build/Fly

Competition

Addendum

Texas A&M University

Robert "Rip" Rippey III

M. Shea Parks

Jason W. Field

April 14, 1997

TABLE OF CONTENTS **Page**

TABLE OF CONTENTS	1
ADDENDUM	2
AIAA PROJECT PURCHASE INVENTORY	6

ADDENDUM

As with any aircraft design, lessons were learned by all of the team members. We were impressed with the finished aircraft and its handling qualities. However, we found the costs to be somewhat higher than originally expected, and simple changes could be made for overall improvement on the initial design.

After the decision was made to participate in the AIAA Student Design/Build/Fly Competition, we found ourselves conducting a thorough research of the market for electric-powered aircraft. With two team members active radio-control modelers, we were quite aware of the cost of building supplies and control systems. However, none of the team members could provide accurate cost estimates of the propulsion system required for the aircraft. At this point, we turned to a professor emeritus who was quite knowledgeable of manufacturers and suppliers of electric motors, speed controls, batteries, and chargers. His initial estimate for the entire propulsion package (motor, speed control, battery, and charger) was $500. He was primarily considering AstroFlight products. Feeling confident with our findings, we provided this figure to our faculty team advisor.

As our design progressed, we began looking into the more efficient, brushless motors. With the final decision made to purchase a propulsion system utilizing Aveox components, the price estimate jumped to $913.72 as shown in our AIAA project purchase inventory.

In addition, we needed to purchase a charger for the propulsion battery. As part of our ground support cost estimates, we included the 12-volt source battery and its charger. With all propulsion, control, structural, and ground support components included, our total investment in the project was $1476.56.

The final contest aircraft turned out to be almost identical to the proposal design. There were not any major changes to the control and performance areas of the aircraft such as the wing, tail, and control surfaces. The landing gear configuration was also left unchanged. However, there were minor changes implemented to the fuselage for overall propulsion system cooling. It was realized after the first two flights that it would be necessary to allow airflow through the fuselage in order to keep the propulsion battery pack and the motor controller cool. Without proper ventilation and convection cooling, both components simply heated too much and could have posed serious in-flight difficulties or malfunctions.

Ventilation and cooling for the battery pack was achieved by constructing an air duct below the battery pack location. Inlet and outlet openings were cut into the bottom of the fuselage, approximately 1 ½ inches forward and aft of the battery pack location. Then, coinciding with the openings, the battery pack was raised with a perimeter of balsa inserts to an offset of 1/8 inch from the fuselage floor.

In order to implement cooling for the motor controller, it was decided to move the controller from the inside of the fuselage to the underside to expose it to the freestream. A panel was cut into the fuselage floor in front of the landing gear, and the controller was braced into place. After both of these improvements, neither component reached extreme temperatures for the remainder of the test flights.

Proceeding with a proper design method, creating a second generation aircraft calls for areas of improvement in design. The two primary areas focused for upgrades are the overall structural design and in the manufacturing process. The change that would significantly improve the manufacturing process would be to make a construction jig for the building of the wing.

The structural design would have several minor improvements that would reduce the post building trial-and-error routine. A more efficient cooling system would be incorporated to allow optimal performance for the electrical and propulsion components as well as lending to a cleaner fuselage design. Cooling changes would have taken no more than four hours to implement during the manufacturing process, and would have required no additional cost. Provisions for internal ducting could be provided by simple pieces of balsa installed around the motor and speed control.

Heavier support around the tail wheel strut would need to be installed in order to safeguard against poor landings and transportation difficulties. To provide a more solid base for the tail wheel bushing to mount in, a small section of the fuselage bottom could be sheeted with 1/8 inch plywood. This action would take no more than an hour to cut and shape the plywood piece.

The overall aircraft structure would need to be made into more assembly sections in order to ensure easier transportation and storage. The 9- foot wing and front fuselage section has proved to hinder efficient transportation. The aircraft could be broken into two components by using a wing-to-fuselage joint instead of using a detachable tail. This modification would allow more efficient access to the propulsion battery and payload, and also ease in transportation. An additional eight hours would probably be required to install mounting blocks in the fuselage and wing.

The idea of using a two-piece wing is also a possibility. However, this would require a more thorough analysis and different construction techniques. With such a substantial change to be made to the structure, ten hours would be needed to carry out this improvement.

The final changes that could be added to a second generation aircraft would be to increase the size of the fuselage and its payload and battery storage area. It has been discovered that with all of the electrical components and utilities, little room is allowed for easy maneuverability and storage around cables and components. The exact size increase dimensions would also involve more thorough analysis. Time estimates could range from one to four additional construction hours, at no additional monetary costs.

In designing and constructing a second generation aircraft, these minor modifications would add to the success of this aircraft. The overall time and cost increases are extremely minimal and would allow margins for future changes.

AIAA Project Purchase Inventory

Item	Quantity	Cost
Propulsion System		
5 mm Propeller Adapters	1	$5.53
42mm Yokes	1	$9.72
40 mm Spinner	1	$4.23
Aero-Naut 15x9.5 Propeller, Carbon Fiber	1	$28.08
Astro Connectors	1	$6.75
NC20N1700HAF Battery Pack	2	$312.80
Aveox Speed Controller	1	$219.56
Aveox AV1412-2v w/RB4197INS Brushless Motor with Gear Box	1	$309.71
202 Solder 0.050 in.	1	$5.49
Snap Plug 16-14 Ga.	2	$2.98
PK2 60 Amp Glo Fuses	2	$5.98
M4x10 SKT HD Bolts, 10ct.	1	$2.89
	Total	**$913.72**
Aircraft Frame		
Misc. Building Supplies	1	$53.00
Nylon Bolts (4 ct.)	1	$1.55
Hatch Fasteners (2 ct.)	1	$1.69
Tail Wheel	1	$2.80
5/32 in. Axles (2 ct.)	1	$2.69
5/32 in. Wheel Collars (2 ct.)	1	$0.85
1/16 in. Wheel Collars (2 ct.)	1	$0.35
Main Gear (Donated by James Manufacturing Inc.)	1	$0.00
Tail Wheel Wire	1	$0.99
Covering TopFlite Monokote	3	$33.00
	Total	**$96.92**

AIAA Project Purchase Inventory

Item	Quantity	Cost
Propulsion System		
5 mm Propeller Adapters	1	$5.53
42mm Yokes	1	$9.72
40 mm Spinner	1	$4.23
Aero-Naut 15x9.5 Propeller, Carbon Fiber	1	$28.08
Astro Connectors	1	$6.75
NC20N1700HAF Battery Pack	2	$312.80
Aveox Speed Controller	1	$219.56
Aveox AV1412-2v w/RB4197INS Brushless Motor with Gear Box	1	$309.71
202 Solder 0.050 in.	1	$5.49
Snap Plug 16-14 Ga.	2	$2.98
PK2 60 Amp Glo Fuses	2	$5.98
M4x10 SKT HD Bolts, 10ct.	1	$2.89
	Total	**$913.72**
Aircraft Frame		
Misc. Building Supplies	1	$53.00
Nylon Bolts (4 ct.)	1	$1.55
Hatch Fasteners (2 ct.)	1	$1.69
Tail Wheel	1	$2.80
5/32 in. Axles (2 ct.)	1	$2.69
5/32 in. Wheel Collars (2 ct.)	1	$0.85
1/16 in. Wheel Collars (2 ct.)	1	$0.35
Main Gear (Donated by James Manufacturing Inc.)	1	$0.00
Tail Wheel Wire	1	$0.99
Covering TopFlite Monokote	3	$33.00
	Total	**$96.92**

AIAA Student Design/Build/Fly

Competition

aerospace engineering department

TEXAS A&M UNIVERSITY

Robert "Rip" Rippey III

M. Shea Parks

Jason W. Field

March 17, 1997

AIAA Design/Build/Fly

Competition

Texas A&M University

Robert "Rip" Rippey III

M. Shea Parks

Jason W. Field

March 17, 1997

TABLE OF CONTENTS

	Page
TABLE OF CONTENTS	2
LIST OF FIGURES	3
EXECUTIVE SUMMARY	4
MANAGEMENT SUMMARY	5
CONCEPTUAL DESIGN	6
PRELIMINARY DESIGN	10
DETAIL DESIGN	13
MANUFACTURING PLAN	16
Appendix A, Drawing Package	
Appendix B, Initial Design Aircraft Weight and Performance Estimation	
Appendix C, Sample Preliminary Neutral Point Location Calculations	
Appendix D, Sample ElectriCalc Program Output Data	

LIST OF FIGURES AND TABLES

Page

Table 1, Events in the Design Process 20

Table 2, Final Ranking for the Conceptual Design 21

Table 3, R.A.F. 32 Airfoil Specification 22

Figure 1, R.A.F. 32 Profile 22

Table 4, Actual Design Aircraft Component Weights 23

Table 5, Final Ranking for the Manufacturing Plan 24

Table 6, Events in the Manufacturing Process 25

EXECUTIVE SUMMARY

Designing an airplane for this competition proved to be a very challenging task. As a team, we were faced with the limits of our education and experience. We began by researching electric-powered radio-controlled airplanes to find out how different they were from common glow-powered airplanes. When put up to such issues as current draw, voltage, and motor gearing, the team found itself in need of advice from experienced personnel.

Meanwhile, we were brainstorming and coming up with great ideas that would surely let us win the competition. We thought that an electric-powered flying wing would be great for carrying weight around the specified course. We researched the history and design of tailless aircraft quite thoroughly. When reality set in, we realized that this was our first design that was to actually be built and flown. Furthermore, none of us had completed all the junior-level courses in the aerospace engineering curriculum.

The team chose to look at a conventional design that could be built quickly and inexpensively. A closer look at the design allowed us to lighten the structure as required. Landing gear could be a simple taildragger configuration to keep the overall weight down. Experience had shown that a light airplane usually handles better than a heavy airplane.

Optimization calculations finally put the exact dimensions on the wing, tail, and fuselage. We tried to make the aspect ratio of the wing as high as possible to get an efficient lift distribution. However, we knew that a relatively high aspect ratio would need a pretty sturdy spar to fulfill the structural verification requirement in the

competition rules. A spar assembly was constructed to assure the team that our construction techniques were adequate. At this point, we were ready to plug in the numbers and see how our design presented itself on paper.

Concerns began to turn toward transportation, motor selection, and structural requirements. With all this on its way, construction of the aircraft began. We chose not to wander in the direction of composite materials for manufacturing. We had all built balsa wood models, and our experience allowed us to construct our competition aircraft with ease.

MANAGEMENT SUMMARY

The design team consisted of three Aerospace Engineering students that had started taking junior-level courses during their introduction to the AIAA student design/build/fly competition. Robert "Rip" Rippey III controlled personnel assignments as the team leader. Rip focused his efforts on the construction of the aircraft and flight testing. With eight years of flying and building experience with radio-controlled aircraft, Rip helped guide the team through the conceptual and preliminary phases of the design process.

Jason W. Field concentrated on the preparation and maintenance of the propulsion system as well as providing landing gear. M. Shea Parks focused his efforts on producing the final drawing package and preparing numerical data for substantiating final design parameters. Together, Jason and Shea completed the detailed design of the aircraft.

Schedule control throughout the design process and construction was provided by Rip. As a visiting professor who had taught senior-level design courses, team advisor

Tom McElmurry was essential in keeping the developments of the aircraft moving smoothly. Rip's experience with radio-controlled aircraft proved to be a valuable asset in establishing configuration control.

Early in the development of the design team, a schedule was set for various elements of the design process. Table 1 shows the planned and actual dates of these events.

CONCEPTUAL DESIGN

During the conceptual design phase, the team investigated several parameters that were combined to yield the final configuration of the aircraft. To sort through these possible configurations, the team considered various design parameters that were felt to be important. Lift production, landing gear configuration, payload and battery accommodations, and propulsion systems were each analyzed in detail with each figure of merit.

Lift could be produced by a wing or a wing-body combination. The number of lifting surfaces and their locations were considered by the team to be the most important aspects for designing an aircraft that was to successfully complete its mission.

A landing gear was required to keep the aircraft controllable for a ground roll that could approach 300 feet. It was also necessary for the landing gear to allow the aircraft to be capable of a second flight with no service or repairs other than recharging the batteries and replacing the propeller(s).

The aircraft was required to accommodate 7.5 pounds of steel. Wings and fuselage(s) provided possible payload locations. Since a 2.5 pound battery pack could be of considerable size, provisions were considered for its location as well.

Electric motors were to be utilized to drive propellers or fan units for propulsion of the aircraft. The selection of propulsion units was restricted by AIAA to include only those commercially available to all participating teams.

The first figure of merit considered was handling. Since none of the team members had completely designed and flown an aircraft, flight characteristics played a large part in configuration selection. A conventional layout was chosen for the final design. By "conventional," the team desired an aircraft on which the main wings were located forward of the stabilizing surfaces. It was decided by this figure of merit that flying wings (tailless aircraft) and canard configurations would require more experienced design personnel to produce a successful, controllable airplane.

In the spirit of being competitive in achieving maximum range, the conceptual design was to have the lowest drag coefficient possible. With each wing tip and fuselage-wing junction adding to the overall drag created on the flying aircraft, a multiple-wing design was considered to be a poor candidate for this competition.

In order to reduce induced drag created by the wing, the aspect ratio was chosen to be as high as possible. Other figures of merit would provide a limit to the aspect ratio.

Elimination of all possible sources of drag warranted the fewest members of protruding landing gear. Since no components could be dropped from the aircraft at any time during the flight, a retractable landing gear seemed to be the best solution. The requirement of having the aircraft capable of a second flight with no service other than recharging the batteries eliminated the possibility of a permanent, spring loaded retractable gear.

Final flying weight was determined to be a high-priority figure of merit. Since the propulsion system was limited by a maximum battery weight of 2.5 pounds, weight of aircraft components played a large part in achieving selection of a landing gear configuration. By eliminating the need for a structurally sound nose gear, weight could be saved by using a tail dragger configuration. Electronic and mechanical components required for any type of retracting landing gear appeared to be a considerable contributor to final flying weight.

In order for the aircraft to meet the requirement of being lifted by the wing tips while fully loaded, wing structure would have to increase considerably for a high aspect ratio wing. For this reason, an aspect ratio of 10 was felt to be a comfortable limit to keep structural weight down.

The team also agreed that a single motor capable of flying the fully-loaded aircraft would probably be lighter than multiple motors of equivalent power output. In addition, the aircraft needed to have very few or no features that disassembled. Any assembly joint would have required accompanying structure that could add to the overall weight of the aircraft.

Ground stability and control was considered to be fairly important for takeoff and landing roll. This figure of merit led the team to select a main gear with two wheels as opposed to a single-wheel main gear with stabilizing gear on the wing.

For adequate control on the ground, a steerable tailwheel was to be employed. In addition, experience had shown that it was essential for the main wheel axles to remain perpendicular to the fuselage centerline to reduce the possibility of ground loops. For this reason, the design team opted for a sturdy aluminum main gear.

Transportation of the aircraft was a figure of merit also considered in selecting the final configuration of the aircraft. Since the team had agreed to drive to the flying site, minimal disassembly would be required. The aircraft was chosen to be separable just aft of the wing to satisfy this figure of merit. The extra structure required for such a separation point was predicted to be lighter than that needed for a detachable wing.

Finally, structural integrity was considered to be essential for any aircraft to complete its flight. Aside from g-loads during turns, loads were considered from unexpected hard landings, gusty flying conditions, and rough transport.

Ultimate selection of figures of merit stemmed from the ranking chart shown in Table 2. The features that produced the final configuration came from the importance of each figure of merit.

Lift for the aircraft was chosen to come from a single, one-piece wing attached permanently to a fuselage, ahead of the stabilizing surfaces (tail). The payload and battery pack were located in the fuselage near the wing. Instead of adding structure to the wing to fit the main gear, the team located the one-piece main landing gear on the fuselage. A single motor was chosen to drive a propeller.

PRELIMINARY DESIGN

During the preliminary design process the primary role and needs were established and debated. The primary focus of the design for the aircraft was the optimization for range. Three primary factors were used in order to efficiently size and develop the design of the aircraft. These factors were the maximum payload and performance weight mandated, maximum available motor and propeller thrust output, and handling qualities complementing the skill level and needs of the pilot. In determining the best combination for this focus, elements such as propulsion, power, aerodynamic theory, stability, and overall weight distribution were encompassed.

The initial concern in optimizing range and performance was determining the estimated associated weights of the components and payload capability order to reach a total aircraft design weight. As a design parameter, the empty aircraft, motor and accessories, battery cells, and control servos were estimated into an operational empty weight. Realistic values for building materials, motor components, and all accessories used in the weight parameter were thoroughly researched. This fraction of the total weight was restricted to be no more than 8.5 pounds (136 ounces). When allowing for 7.5 pounds of steel payload, the final operational weight restriction was determined to be 16 pounds (256 ounces). This parameter was essential to the design process in order to ensure proper testing and performance procedure. Estimated values are calculated and displayed in Appendix B.

Using the weight parameters, pilot control skills, and research results furnished from Senior Aerospace Engineering Design class students at Texas A&M University, appropriate parameters were set for determining the design wing layout. First, an acceptable wing loading of 2 lbs/ft^2 (32 oz/ ft^2) was incorporated. By using a final operational weight of 16 lbs, the chosen wing loading yielded a wing area of 8 square feet.

The next set of parameters that were essential to the wing design were aspect ratio and efficient preliminary drag coefficients. Initial calculations yielded several performance values based on the aspect ratio and drag coefficients. The aspect ratio and drag coefficients were determined by reviewing output curves from Mathcad programs incorporating performance theory, Breguet equations, and the design class research (Appendix B). Reviewing all of these sources in conjunction with the figures of merit stated in the Conceptual Design section, an aspect ratio of 10 was optimum for this type of aircraft. A value of 0.04 was estimated for the zero lift drag coefficient, 0.78 as the clean wing and fuselage efficiency factor, and 0.041 as the estimated induced drag coefficient.

Unswept wing taper ratio was then determined for the appropriate design of the wing layout. Using the theory of elliptical lift distribution on a wing and prior research, a taper ratio of 0.4 was incorporated into the design parameters.

With much of the wing design criteria determined, an airfoil was to be selected. Using previous design parameters, lift coefficient verses wing dimensions was plotted. This plot revealed that 0.521 was the peak airfoil lift coefficient for optimal cruise. This value was used in researching the most efficient airfoil for this aircraft. Through the process of researching airfoil tables and compilations, the Royal Air Force 32 airfoil was selected and configured into the wing design parameters (Table 3 and Figure 1).

The layouts of the empenage and control surfaces were determined by overall research and limits set by the design parameters produced form the wing layout. The resulting horizontal stabilizer area was 1.18 square feet, and the vertical stabilizer area was approximately 0.56 square feet.

The design of the fuselage and tail assembly was also produced by examining basic efficiency and clean performance criteria. The basic dimensions of the fuselage near the wing were 2 1/4 inches x 4 inches. The basic taper values and dimensions were determined by the sizes of the motor components, payload, batteries, and accessories. Preliminary neutral point location calculations were obtained, and approximate static margin values were between 13% and 15%. Appendix C displays examples of the neutral point location calculations.

The maximum available motor and propeller thrust output as a parameter of design was essential for determining the best propulsion system needed. Initially, sea level standard day conditions were used to calculate thrust required and thrust available estimates with the preliminary aircraft configuration (Appendix B). After preliminary calculations, 2 pounds of thrust was the determined to be the minimum required output. The calculations also revealed the thrust available to be at least 10 pounds. By applying the preliminary configuration and the SLK Electronics ElectriCalc program (Electric R/C model performance prediction software), the thrust values were referenced with many different motor, battery cell, and propeller configurations. Figure 2 displays the output parameters produced by the ElectriCalc program and the factors that could have been manipulated such as propeller size, motor make/model, and gearing. The preliminary results of the propulsion setup consisted of 19 N-1700SCRC rechargeable Ni-Cad cells, an AVEOX 1412/2Y electric motor, and an Aero-naut 15 x 9.5 carbon folding propeller. The motor used a gear reduction ratio of 3.7:1.

The performance estimates and vehicle sizing yielded an extremely clean and efficient aircraft design. This design was used to maximize range based upon power system and aerodynamic proficiency. The key parameters were the overall performance weight, wing layout, and the determination of initial performance envelope.

DETAIL DESIGN

Once the final configuration and design of the aircraft was thoroughly researched, tested, and altered (as necessary), the final performance envelope was calculated. Using many of the same methods incorporated for the preliminary design, takeoff and landing, climb rates, maneuverability, g-load capacity, range, and endurance performance were determined. The final specific weights and payload fraction were calculated for comparisons to the estimated values.

The takeoff performance was calculated using all applicable design parameters. The maximum takeoff roll distance was determined to be 300 ft when approximating a rolling friction coefficient of 0.02 and a liftoff velocity of 47.2 ft/sec. The maximum landing roll was given as 339 ft with a touchdown velocity of 49.3 ft/sec. These values could be easily manipulated using higher throttle settings and employing high-lift devices, such as flaps, during takeoff. Both takeoff and landing procedures would be manipulated and determined by the pilot.

The climb performance consisted of maximum climb rate, maximum climb velocity, climbing lift coefficient, and climbing drag coefficient. The maximum climb rate was 214 ft/min; this yielded a maximum climb velocity of 51.1 ft/sec. The climbing lift coefficient was 0.645 with a complementing climbing drag coefficient of 0.057.

The maneuvering and handling performance of the design aircraft consisted of the g-load factor, turning velocity, turning lift and drag coefficients, and bank angle. The g-load factor was approximately 1.9 g's with a turning velocity of 63.7 ft/sec. The resulting turning lift coefficient was 0.791 and the turning drag coefficient was 0.066. This maneuvering performance further yielded a turning bank angle of 58.3°.

The range optimization parameters resulted in a final design aircraft that was able to travel a maximum distance of 28,125 ft, or approximately 18 complete laps around the specified course. The time of flight endurance ranged from 5 to 13.5 minutes. The range and endurance values were produced using an average cruise velocity of 73.8 ft/sec.

The final weight distribution was determined from weighing the actual components. A complete breakdown of the final weights of all specific items is displayed in Table 4. The total aircraft weight with all components such as landing gear, battery cells, motor and accessories, control servos, and all building materials, excluding mandated payload, was 7.58 pounds (121.35 oz.). The contest parameter of 7.5 pounds (120 oz.) of steel payload resulted in a payload of 49.7% of the total performance weight of 15.08 lbs (241.35 oz.).

The type of electrical and control components were selected after extensive research and planning. The final selections proved to be the best components for the requirements of the design aircraft. For the control surfaces, four Futaba dual ball-bearing servos were used with the appropriate pushrod provisions. One servo was located on the fourth rib of each wing for control of each flaperon. Additional servos were placed in the tail boom assembly behind the trailing edge of the wing for rudder and elevator control. As a power supply, the 19 N-1700SCRC rechargeable Ni-Cad cells were stationed in the fuselage. The battery pack was positioned approximately at the predicted aircraft center of gravity location. The AVEOX 1412/2Y electric motor and speed control unit were placed in front of the leading edge of the wing, and all utility cabling was routed aft.

The landing gear assembly was made from 6061 T6 aluminum and was located ½ inch aft of the leading edge of the wing. The main wheels were 2 ½ inch aluminum racing wheels. The tail wheel was 1 inch in diameter and attached directly to the rudder.

Many of the design parameters were established on the basis of efficiency. In many instances efficiency was associated with overall economic costs. The design aircraft utilized great cost effectiveness. Some of the methods used in reducing the costs were restricting all main aircraft construction to balsa and plywood, utilizing only a few carbon fiber strips for overall reinforcement of the spar, and simple, yet strong, aluminum landing gear. The electrical systems and components were selected taking into account the overall costs. The results proved that the selections were the most cost effective and performance-matched for the detailed design aircraft.

MANUFACTURING PLAN

Before developing a manufacturing plan, the design team analyzed several radio-controlled models and UAVs to determine possible manufacturing processes. The majority of the radio-controlled models utilized balsa wood construction exclusively to obtain light structure. Plywood was used sparingly in these models to target areas potentially receiving high loadings such as motor mounts, landing gear mounts, and separation points. Some models had molded fiberglass shells for fuselage construction and balsa-sheeted foam core wings. These models seemed very durable, yet they were relatively heavy if care was not taken to eliminate unnecessary structure. An investigation of UAV construction showed a large utilization of carbon fiber and kevlar for primary structure. Aluminum parts were found in high-stress areas. While checking into these construction methods, the team determined figures of merit to be cost, material and facility access/availability, weight, time, and experience.

Cost was a factor in producing an aircraft that could be inexpensive to duplicate. All-wood construction was the most cost-efficient method to build the aircraft.

Balsa and plywood was easily accessible through local hobby shops. Fiberglass of various weights could be found at some hobby stores; however, the university had a decent selection. Carbon fiber was hard to find, but again, the university had unidirectional graphite for the team to use as needed. Aluminum was not readily available for the team, and manufacture of parts would have probably required the help of an experienced machinist.

Balsa construction proved itself to be very light as seen in many radio-controlled model airplane kits. If used efficiently, fiberglass could strengthen areas of high stress. However, when combined with resin, fiberglass was observed to add unnecessary weight. The team found carbon fiber usage to be very light for the strength it could add to an aircraft structure.

With time being a major factor in choosing a manufacturing process, the team was highly encouraged to utilize techniques and materials that did not demand a lot of preparation. For example, creating a fiberglass fuselage would have required a mold to be built before an actual fuselage was produced. Aluminum parts would also require substantial time to machine.

The team members agreed that experience was the most important figure of merit for determining a manufacturing process. All of the team members had worked with balsa and plywood at some time. Carbon fiber and kevlar had not been used by anyone in the team. In addition, fiberglass construction techniques had not been developed comfortably with any of the team members. Some level of machine experience was held by everyone in the team.

A final ranking of the construction techniques and figures of merit was determined as shown in Table 5. Balsa and plywood construction was chosen to be used throughout the manufacturing process of the aircraft. Aliphatic resin and epoxy was used exclusively to ensure sufficient glue joints. Structural requirements led the team to apply strips of unidirectional graphite along the caps of the wing spar, as detailed below.

Fabrication of the competition aircraft began with the wing. The one-piece wing had 1/4" x 3/8" balsa spars located top and bottom at 30% of the chord for the entire span of the wing. Due to the 9-foot span of the wing, 3-foot sections of the spar were spliced. Splices were completed by cutting each piece at approximately 25 degrees from the long dimension and gluing ends together with aliphatic resin.

With graphite strips capping the spars, the wide dimension (3/8") of the spars was placed tangent to the airfoil shape. The extra width allowed more bonding area for the graphite.

Balsa ribs 3/32" thick were cut to shape and glued to the spars. Since no dihedral was incorporated in its top surface, the wing was constructed inverted on a work surface to speed construction. With top and bottom spars in place, 1/16" balsa shear webs were applied in such a way to create a box spar. The shear webs had the grain of the wood oriented vertically to ensure the most efficient transfer of shear loads between the top and bottom spars. At this point, 3/8"-wide strips of unidirectional graphite was epoxied to the top and bottom of the box spar. Tests conducted by the team showed that two full-length (9-foot) strips and one half-length (4 1/2-foot) strip of graphite applied to the top and bottom of the box spar fulfilled the structural verification requirement as stated in the competition rules. The carbon fibers also kept the deflection of the spar assembly reasonably low when fully loaded with the expected final weight of the aircraft.

The leading edge spar used 1/4" square balsa wood sticks spliced using the same method mentioned above. In order to prevent the wing from twisting in flight due to the relatively high aspect ratio, a D-tube structure was formed by sheeting the entire span of the wing from the leading edge to the center of the box spar using 1/16" balsa wood.

The trailing edge of the wing was formed with 1/16" balsa sheeting. The sheeting was 1 1/2 inches wide on top and bottom for the entire span of the wing. Flaperons were cut from the trailing edge sheeting and hinged to a false spar installed in the wing. One servo was employed for each aileron to ensure sufficient control and mixing (flaperons).

Cap strips were glued to the ribs to create an I-beam structure as well as to provide adequate bonding area for the covering material.

Construction of the fuselage initiated with the completion of the wing. The fuselage sides were made of 3/32" balsa wood. In order to provide additional strength to the forward part of the fuselage, 1/32" plywood was laminated on the inboard sides of the 3/32" balsa from the firewall to the tail separation joint. The tail separation used four 1/16" birch plywood tabs glued inside the fuselage walls aft of the joint. Screws attached the tabs to the forward portion of the fuselage. Incorporating the forward part of the fuselage with the wing involved a simple bond between the fuselage sides and the two center ribs of the wing.

The firewall and landing gear mount were cut from 1/8" birch plywood. Support for the battery and payload was provided by a 1/8" lite plywood fuselage floor. The remaining area of the bottom and the top of the fuselage were sheeted with 1/8" and 3/32" balsa, respectively. Top and bottom sheeting employed cross-grain placement to efficiently transfer shear loads.

A 1/8" lite plywood cradle was installed to support the cantilevered motor during positive g loading conditions.

All tail surfaces were 1/4"-thick frames built flat on a work surface. Leading and trailing edges were sanded round to reduce drag and increase the effectiveness of the rudder and elevator.

The single-piece main landing gear was fabricated from 1/8" 6061 T6 aluminum. A local machine shop cut and bent the gear to our specifications. Four 1/4" x 20 nylon bolts threaded directly into the bottom of the fuselage to attach the main gear. The tail gear was bent from .078" music wire. Installation of the wire directly into the rudder ensured adequate ground steering at low speeds.

Ease of application warranted Top Flite Monokote to be used for covering the entire airframe.

Table 6 shows a milestone chart documenting the major events in the fabrication process of the aircraft.

Table 1, Events in the Design Process

Event	Planned Date	Actual Date
Optimization calculations for range	November 3, 1996	November 3, 1996
Configuration selection	November 8, 1996	November 7, 1996
Airfoil selection	November 15, 1996	November 25, 1996
Motor/battery selection	November 15, 1996	December 7, 1996
Detail design	November 22, 1996	December 18, 1996
List of materials ready	November 22, 1996	January 25, 1997
Main wing spar test	December 20, 1996	January 19, 1997
Start of construction	December 20, 1996	February 2, 1997
Report completion	March 8, 1997	March 13, 1997

Table 2, Final Ranking for the Conceptual Design

	Design Parameter			
Figure of Merit (importance ranking, out of 10)	**Lift Production**	**Landing Gear Configuration**	**Payload and Battery Accommodations**	**Propulsion**
Controllability (10)	10	1	4	5
Drag (7)	8	8	4	6
Weight (8)	2	8	6	8
Ground Handling (7)	2	10	2	4
Transport (3)	2	3	2	2
Structural Integrity (8)	7	8	8	6

Table 3, R.A.F. 32 Airfoil Specification

% chord	cr	ct	upper	lower	- lower
0.00	0.00	0.00	3.42	3.42	-3.42
1.25	0.07	0.04	5.56	1.96	-1.96
2.50	0.14	0.09	6.52	1.50	-1.50
5.00	0.28	0.17	7.84	0.88	-0.88
7.50	0.42	0.26	8.83	0.50	-0.50
10.00	0.56	0.34	9.72	0.30	-0.30
15.00	0.83	0.51	11.02	0.08	-0.08
20.00	1.11	0.68	11.92	0.00	0.00
30.00	1.67	1.03	12.98	0.30	-0.30
40.00	2.22	1.37	13.10	0.70	-0.70
50.00	2.78	1.71	12.46	1.10	-1.10
60.00	3.34	2.05	11.06	1.46	-1.46
70.00	3.89	2.39	9.10	1.60	-1.60
80.00	4.45	2.74	6.56	1.46	-1.46
90.00	5.00	3.08	3.60	0.92	-0.92
95.00	5.28	3.25	1.98	0.52	-0.52
100.00	5.56	3.42	0.12	0.00	0.00

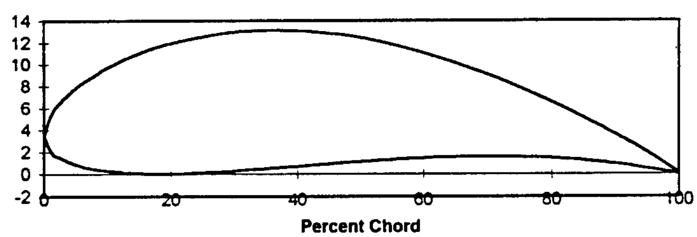


Figure 1, R.A.F 32 Profile

Table 4, Actual Design Aircraft Component Weights

Component(s)	Weight (ounces)
Landing gear with axles, no wheels	9.6
Main wheels (2)	1.8
Tailwheel and wire	0.3
Battery pack	39.1
Motor with speed control	17.3
Propeller and spinner	1.8
Complete wing, fuselage, no landing gear	60.8
Complete tail	9.5
Total, aircraft ready to fly	121.4
Payload	120.0
Total, fully-loaded aircraft	241.4

Table 5, Final Ranking for the Manufacturing Plan

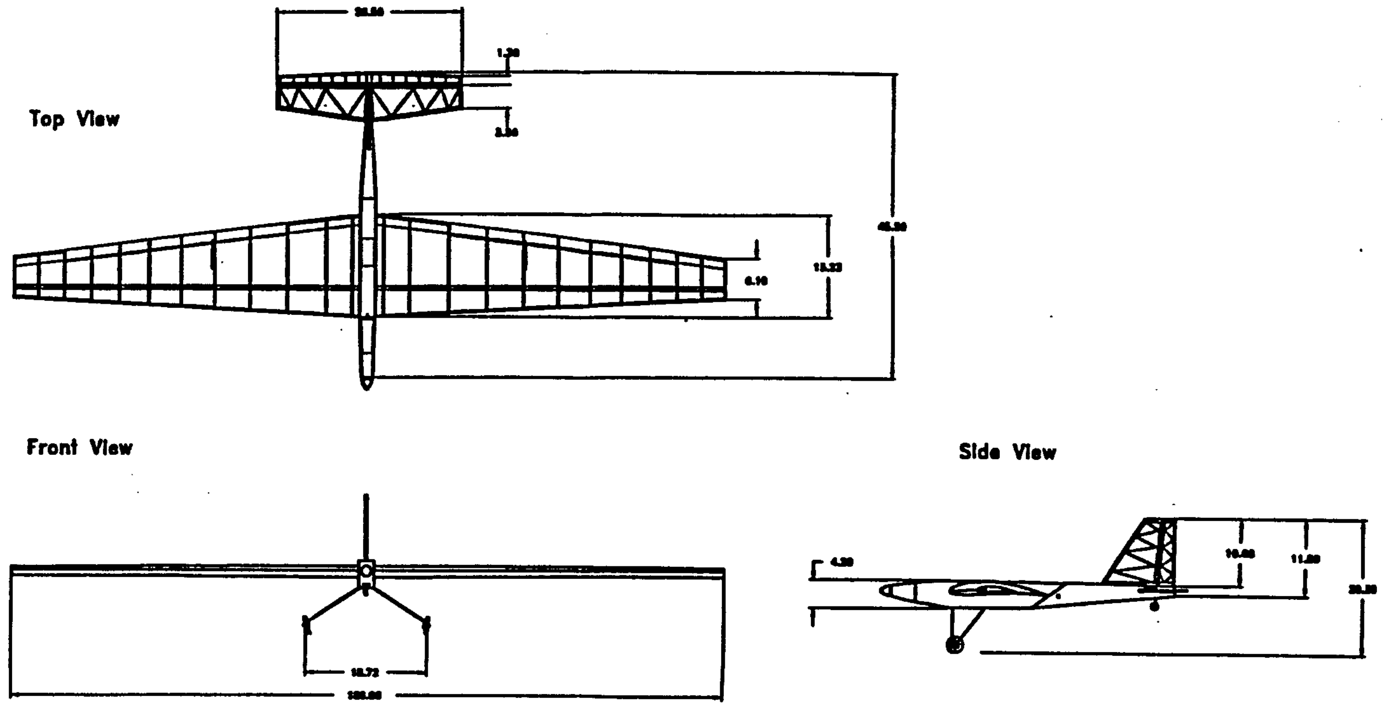
		Construction Process		
Figure of Merit (importance ranking, out of 10)	**Balsa/Plywood**	**Fiberglass**	**Carbon Fiber**	**Aluminum**
Cost (4)	9	6	4	4
Access/Availability (7)	10	7	6	4
Weight (8)	8	4	6	3
Time (8)	8	2	3	3
Experience (10)	10	5	4	4

Table 6, Events in the Manufacturing Process

Event	Planned Date	Actual Date
Drawings of aircraft ready	November 29, 1996	January 6, 1997
Main wing spar test	December 20, 1996	January 19, 1997
Completion of wing	February 7, 1997	February 14, 1997
Completion of tail	February 7, 1997	February 14, 1997
Aircraft ready to cover	February 18, 1997	February 20, 1997
Aircraft ready to fly	March 1, 1997	March 8, 1997

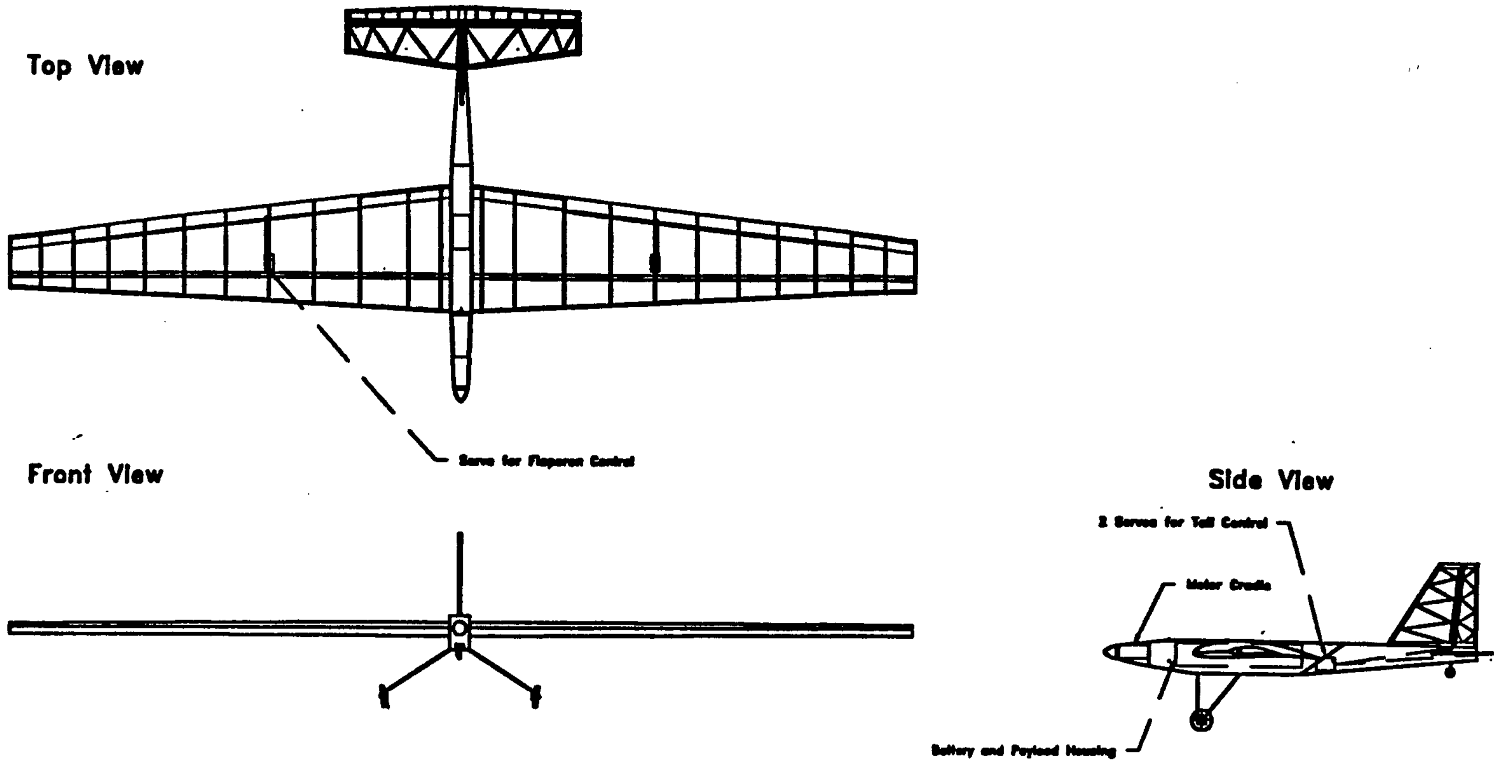
Appendix A

Drawing Package



General Size, Structure and Configuration

Figures Not Drawn to Scale



Primary Structure with Location of Propulsion and Control

Appendix B

Initial Design Aircraft Weight and Performance Estimation

INITIAL DESIGN AIRCRAFT WEIGHT AND PERFORMANCE ESTIMATION

SIZING WEIGHT ESTIMATES

WPL: SPEC PAYLOAD WEIGHT (lbs)
WEACC: ENGINE ACCESSORIES (BATTERIES, SPEED CNTRL, ETC.)
WPLSZ: PAYLOAD WEIGHT FOR AIRCRAFT SIZING, (WPL+WEACC)
PLFRD: DESIGN PAYLOAD FRACTION, (WPLSZ/WTO)
WEFR: DESIGN AIRCRAFT EMPTY WEIGHT FRACTION, (WE/WTO)
WE: DESIGN AIRCRAFT EMPTY WEIGHT ESTIMATE (lbs)
WTO: DESIGN AIRCRAFT TAKEOFF WEIGHT. (WTO= WPLSZ+WE)

$WPL := 7.5$ $\quad$ $WEACC := 3.5$ $\quad$ $WPLSZ := WPL + WEACC$

$PLFRD := 0.75$ $\quad$ $WTO := \dfrac{WPLSZ}{PLFRD}$ $\quad$ $WTO = 14.66667$

$WE := WTO - WPLSZ$ $\quad$ $WE = 3.66667$

SIZING AND CONFIGURATION VARIABLES:

WLTO: DESIGN TAKEOFF WING LOADING, lb/ft^2
AR: WING ASPECT RATIO
S: WING REFERENCE AREA, ft^2
b: WING SPAN, ft
CBAR: MEAN AERODYNAMIC CHORD LENGTH, ft
LAMBDA: WING TAPER RATIO

$WLTO := 1.5$ $\quad$ $S := \dfrac{WTO}{WLTO}$

$S = 9.77778$

$AR := 10$ $\quad$ $b := (S \cdot AR)^{0.5}$

$b = 9.88826$ $\quad$ $CBAR := \dfrac{S}{b}$ $\quad$ $CBAR = 0.98883$

DESIGN DRAG AND LIFT PERFORMANCE PARAMETER VALUES:

CDO: ESTIMATED DESIGN AIRCRAFT ZERO LIFT DRAG COEFFICIENT (CLEAN CONFIG.)
ΔCDOTF DELTA ZERO LIFT DRAG COEFFICIENT WITH TAKEOFF FLAPS
ΔCDOLF: DELTA ZERO LIFT DRAG WITH LANDING FLAPS
E: CLEAN WING EFFICIENCY FACTOR
ETOF: WING EFFICIENCY FACTOR WITH T.O. FLAPS
ELDF: WING EFFICIENCY FACTOR WITH LAND FLAPS
CLMXC: CLEAN MAXIMUM LIFT COEFFICIENT
CLMXTF: MAX LIFT COEFFICIENT WITH T.O. FLAPS
CLMXLF: MAX LIFT COEFFICIENT WITH L.D. FLAPS

$CDO := 0.02$

$\Delta CDOTF := 0.015$

$\Delta CDOLF := 0.025$

$E := 0.78$

$ETOF := 0.7$ $\quad$ $ELDF := 0.65$

$CLMXC := 1.0$ $\quad$ $CLMXTF := 1.3$ $\quad$ $CLMXLF := 1.6$

THRUST AVAILABLE ESTIMATES:

EXAMPLE TRAMVEL CALCULATION (TIGER ENGINE DATA)

TRAVSTE: SEA LEVEL, MAX THROTTLE, STATIC THRUST AVAILABLE, lb
TRASLPE: THRUST AVAILABLE vs VELOCITY CURVE SLOPE, lb/(ft/sec)
TRAMVELE: THRUST AVAILABLE AT MANEUVER VELOCITY, lb
VMAN: MANEUVER VELOCITY, ft/sec

$TRAVSTE := 10.4$ $\quad$ $TRASLPE := -0.0414$ $\quad$ $vman := 30$

$TRAMVELE := TRAVSTE + (vman \cdot TRASLPE)$ $\quad$ $TRAMVELE = 9.158$

DESIGN AIRCRAFT TAKEOFF PERFORMANCE ESTIMATE

$vm := 1.15$ $\quad g := 32.2$

$WTO = 14.66667$ $\quad AR = 10$

$RHOSL := 0.0023769$

$RHOTO := RHOSL$

$CLMXTF = 1.3$

$$vlo := vm \cdot \left(\frac{2 \cdot WLTO}{RHOTO \cdot CLMXTF} \right)^{0.5}$$

$vlo = 35.83285$

SYMBOLS:

- vm: LIFT OFF VELOCITY MULTIPLIER (V_{stall} X vm = V_{LO})
- RHOTO: DENSITY AT TAKEOFF ALTITUDE, lb sec²/ft⁴
- slo: TAKEOFF GROUND ROLL, ft
- h: WING HEIGHT ABOVE GROUND, ft
- MUR : ROLLING FRICTION COEFFICIENT
- PHI: DRAG AVERAGING FACTOR
- DAVG: AVERAGE DRAG, lbs
- LAVG: AVERAGE LIFT, lbs
- TWRTO: TAKEOFF THRUST/WEIGHT RATIO
- CLTO: TAKEOFF LIFT COEFFICIENT
- TASMX: MAXIMUM STATIC SEA LEVEL THRUST, lbs
- TALOFF: MAXIMUM LIFT OFF THRUST AVAILABLE, lbs
- TATOA: AVERAGE STANDARD SEA LEVEL TAKEOFF THRUST AVAILABLE, lbs
- vlo: LIFT OFF VELOCITY, ft/sec
- vloa: VELOCITY FOR LIFT AND DRAG AVERAGING CALCULATIONS, ft/sec
- CDTFA: DRAG COEF. FOR T.O. DRAG AVERAGING CALCULATIONS

CALCULATE THE AVERAGE TAKEOFF LIFT AND TAKEOFF DRAG:

$$CLTO := \frac{2 \cdot WLTO}{\left(RHOTO \cdot vlo^2\right)}$$

$CLTO = 0.98299$ $\quad S = 9.77778$

$vloa := 0.7 \cdot vlo$ $\quad vloa = 25.08299$

$CDOTF := CDO + \Delta CDOTF$ $\quad CDOTF = 0.035$ $\quad h := .5$

$$PHI := \frac{\left[16 \cdot \left(\frac{h}{b}\right)\right]^2}{\left[1 + \left[16 \cdot \left(\frac{h}{b}\right)\right]^2\right]}$$

$$CDTOFA := CDOTF + \left(\frac{1}{\pi \cdot AR \cdot ETOF}\right) \cdot CLTO^2 \cdot PHI$$

$PHI = 0.3956$

$D_{avg} := 0.5 \cdot RHOTO \cdot vloa^2 \cdot S \cdot CDTOFA$ $\quad D_{avg} = 0.38297$

$L_{avg} := 0.5 \cdot RHOTO \cdot vloa^2 \cdot S \cdot CLTO$ $\quad L_{avg} = 7.18667$

FROM THE ENGINE PERFORMANCE DATA, CALCULATE
THE AVERAGE THRUST AVAILABLE AT TAKEOFF VELOCITY

TRAVST: SEA LEVEL, MAX THROTTLE, STATIC THRUST AVAILABLE, lb
TRASLP: THRUST AVAILABLE vs VELOCITY CURVE SLOPE, lb/(ft/sec)

$vm = 1.15$ $\quad vlo = 35.83285$ $\quad TRAVST := 3.85$ $\quad TRASLP := -0.0414$

$$TAVGTO := \frac{TRAVST + (TRAVST + (TRASLP \cdot vlo))}{2} \qquad TAVGTO = 3.10826$$

$MUR := 0.02$ $\quad CLMXTF = 1.3$ $\quad WTO = 14.66667$

$$slo := \frac{vm^2 \cdot WTO^2}{32.2 \cdot RHOTO \cdot S \cdot CLMXTF \cdot \left[(TAVGTO) - \left[D_{avg} + \left[MUR \cdot \left(WTO - L_{avg}\right)\right]\right]\right]}$$

$slo = 113.53105$

POST TAKEOFF CLIMB PERFORMANCE:

CDTOF: TAKEOFF CLIMB DRAG COEFFICIENT
CDOTF: ZERO LIFT DRAG COEF. WITH TAKEOFF FLAPS
ETOF: WING EFFICIENCY FACTOR WITH TAKEOFF FLAPS
CLMDRG: CLIMB DRAG WHEN CLIMBING AT LIFT OFF VELOCITY WITH TAKEOFF FLAPS DOWN
TOCLMA: POST TAKEOFF CLIMB ANGLE, CLIMBING AT LIFT OFF VELOCITY
PTCLMR: POST TAKEOFF CLIMB RATE AT LIFT OFF VELOCITY
PTHRZV: HORIZ. VELOCITY, CLIMBING AT LIFT OFF VELOCITY
CLMTIM: CLIMB TIME TO REACH AN ALTITUDE OF 15 FEET
CLIMDIST: HORIZONTAL DISTANCE TRAVELED WHILE CLIMBING TO 15 FT ALTITUDE
TOTLDIST: TOTAL DISTANCE TRAVELED FROM TAKEOFF START TO CLEARING 10 FOOT OBSTACLE
TACLMB: CLIMB THRUST AVAILABLE

$$TACLMB := TRAVST + (vlo \cdot TRASLP)$$

$TACLMB = 2.36652$

$$CDTOF := CDOTF + \left(\frac{CLTO^2}{3.14 \cdot AR \cdot ETOF}\right)$$

$$CLMDRG := 0.5 \cdot RHOTO \cdot vlo^2 \cdot S \cdot CDTOF$$

$CLMDRG = 1.17814$

$$TOCLMA := asin\left(\frac{TACLMB - CLMDRG}{WTO}\right)$$

$$PTCLMR := vlo \cdot (sin(TOCLMA))$$

$$PTHRZV := vlo \cdot (cos(TOCLMA))$$

$PTCLMR = 2.90339$

$PTHRZV = 35.71503$ $\quad CLMTIM := \dfrac{15}{PTCLMR}$ $\quad CLMTIM = 5.16637$

$CLIMDIST := PTHRZV \cdot CLMTIM$ $\quad CLIMDIST = 184.51707$

$TOTLDIST := slo + CLIMDIST$ $\quad TOTLDIST = 298.04812$

DESIGN AIRCRAFT LANDING PERFORMANCE ESTIMATE

$WTO = 14.66667$

$RHOL := RHOSL$

$vml := 1.2 \qquad WLDM := 1$

$MURL := 0.02 \quad WLD := WTO \cdot WLDM$

$WLD = 14.66667$

$WLLD := \frac{WLD}{S} \qquad WLLD = 1.5$

$$vld := vml \cdot \left(\frac{2 \cdot WLLD}{RHOL \cdot CLMXLF}\right)^{0.5}$$

vml: LANDING TOUCH DOWN VELOCITY MULTIPLIER
vld: LANDING TOUCHDOWN VELOCITY, ft/sec
WLLD: LANDING WING LOADING, lb/ft²
RHOL: DENSITY AT LANDING ALTITUDE, lb sec²/ft⁴
sld: SPECIFIED LANDING GROUND ROLL, ft
MURL : LANDING ROLLING FRICTION COEFFICIENT
WLD: LANDING WEIGHT, lbs
WLDM: LANDING WEIGHT MULTIPLIER
CLMXLF: LANDING MAXIMUM LIFT COEF.
DL_{AVG}: AVERAGE DRAG, lbs
LL_{AVG}: AVERAGE LIFT, lbs
CLLD: LANDING LIFT COEFFICIENT
CDLD: LANDING DRAG COEFFICIENT

$vld = 33.70361 \qquad CLLD := \frac{2 \cdot WLLD}{\left(RHOL \cdot vld^2\right)}$

$$vlda := 0.7 \cdot vld \qquad CDLD := CDOTF + \left(\frac{1}{\pi \cdot AR \cdot ETOF}\right) \cdot CLLD^2 \cdot PHI$$

$DL_{avg} := 0.5 \cdot RHOL \cdot vlda^2 \cdot S \cdot CDLD \qquad DL_{avg} = 0.37003$

$LL_{avg} := 0.5 \cdot RHOL \cdot vlda^2 \cdot S \cdot CLLD \qquad LL_{avg} = 7.18667$

$$sld := \frac{WLLD \cdot \left(vml^2 \cdot WLD\right)}{CLMXLF \cdot g \cdot RHOL \cdot \left[DL_{avg} + MURL \cdot \left(WLD - LL_{avg}\right)\right]} \qquad sld = 497.85887$$

LANDING PERFORMANCE SUMMARY:

LANDING APPROACH VELOCITY, ft/sec	LANDING MAXIMUM LIFT COEFFICIENT	LANDING ROLLOUT DISTANCE, ft
$vld = 33.70361$	$CLMXLF = 1.6$	$sld = 497.85887$

DESIGN AIRCRAFT CLIMB PERFORMANCE ESTIMATE

$MNWTOR := 1$

$WTCLM := WTO \cdot MNWTOR$

$WTCLM = 14.66667$

$$WLCLM := \frac{WTCLM}{S}$$

$WLCLM = 1.5$

$RHOC := RHOSL$

$CLMXC = 1$

$TRAVST = 3.85$

SYMBOL DEFINITION:

VCLM :	CLIMB VELOCITY, ft/sec
MNWTOR:	MANEUVER WEIGHT/ TAKEOFF WEIGHT RATIO
WTCLM:	CLIMB WEIGHT, lbs
WLCLM:	CLIMB WING LOADING, lbs/ft^2
RHOC:	CLIMB ATMOSPHERIC DENSITY, lb sec^2/ft^4
CLCLM:	CLIMB LIFT COEFFICIENT
CDCLM:	CLIMB DRAG COEFFICIENT
TRCLM:	CLIMB THRUST REQUIRED, lb
CLMRATE:	CLIMB RATE, ft/sec
CLMXC:	CLEAN MAX LIFT COEF.
CLMANG:	CLIMB ANGLE, radians
CLMANGD:	CLIMB ANGLE, deg
VHORZ:	CLIMB VELOCITY HORIZ. COMPONENT, ft/sec
TRASLP:	THRUST AVAILABLE vs VELOCITY FOR TIGER ENGINE, lbs
TACLM:	CLIMB THRUST AVAILABLE, lbs

INPUT CLIMB VELOCITY, ft/sec, CHOICE AND THRUST AVAILABLE AT THAT CLIMB VELOCITY FROM ENGINE CURVES, lbs. VARY THE CLIMB VELOCITY UNTIL THE MAXIMUM CLIMB ANGLE IS DETERMINED.

$VCLM := 39$ $\quad TACLM := TRAVST + (VCLM \cdot TRASLP)$ $\quad TACLM = 2.2354$

$$CLCLM := \frac{2 \cdot WLCLM}{RHOC \cdot VCLM^2} \qquad CDCLM := CDO + \frac{CLCLM^2}{\pi \cdot AR \cdot E} \qquad TRCLM := \frac{WTCLM}{\left(\frac{CLCLM}{CDCLM}\right)}$$

$TRCLM = 0.85016$

$$CLMANG := \operatorname{asin}\left(\frac{TACLM - TRCLM}{WTO}\right)$$

$CLMANG = 0.09459$

$CLMANGD := CLMANG \cdot 57.3$ $\quad CLMANGD = 5.41995$

$CLMRATE := VCLM \cdot (\sin(CLMANG))$ $\quad CLMRATE = 3.68347$

$VHORZ := VCLM \cdot (\cos(CLMANG))$ $\quad VHORZ = 38.82566$

CLIMB RATE ft/sec	CLIMB VELOCITY ft/sec	CLIMB LIFT COEFFICIENT	CLIMB DRAG COEFFICIENT
$CLMRATE = 3.68347$	$VCLM = 39$	$CLCLM = 0.82981$	$CDCLM = 0.0481$

NOTE: CHECK FOR A CLIMB LIFT COEFFICIENT GREATER THAN THE MAXIMUM CLIMB DESIGN LIFT COEFFICIENT.

$CLMXC = 1$

ESTIMATE THE DESIGN AIRCRAFT THRUST LIMITED TURNING PERFORMANCE AND THE LIFT COEFFICIENT LIMITED TURNING PERFORMANCE

$MNWTRN := 1$

$WTTRN := WTO \cdot MNWTRN$

$WTTRN = 14.66667$

$RHOT := RHOSL$

$VTRN := 50.4 \qquad LDFT := 1.99$

SYMBOL DEFINITION:

LDFT:	TURN LOAD FACTOR, "g's"
VTRN:	TURNING VELOCITY, ft/sec
MNWTRN:	MANEUVER WEIGHT/ TAKEOFF WEIGHT RATIO
WTTRN:	TURNING WEIGHT, lbs
WLTRN:	TURN WING LOADING lb/ft²
RHOT:	TURNING ATMOSPHERIC DENSITY, lb sec²/ft⁴
CLTRN:	TURN LIFT COEFFICIENT
CDTRN:	TURN DRAG COEFFICIENT
TRTRN:	TURN THRUST REQUIRED, lbs
PRTRN:	TURN POWER REQUIRED, ft lb/sec
BKANG:	THRUST, LIFT COEF, AND LOAD FACTOR LIMITED BANK ANGLE AT DENSITY ALTITUDE
CLMXTRN:	MAXIMUM LIFT COEFFICIENT IN TURN CONFIGURATION

$$CLTRN := \left(\frac{2 \cdot LDFT \cdot WTTRN}{RHOT \cdot VTRN^2 \cdot S}\right) \qquad CLTRN = 0.98879 \qquad CLMXC = 1$$

$$CDTRN := CDO + \frac{CLTRN^2}{\pi \cdot AR \cdot E} \qquad CDTRN = 0.0599 \qquad TRTRN := \frac{LDFT \cdot WTTRN}{\left(\frac{CLTRN}{CDTRN}\right)}$$

$$TATRN := TRAVST + (VTRN \cdot TRASLP) \qquad TATRN = 1.76344 \qquad TRTRN = 1.76807$$

OVER A RANGE OF VELOCITIES, INCREASE THE LOAD FACTOR UNTIL THRUST AVAILABLE EQUALS THRUST REQUIRED OR UNTIL THE TURN LIFT COEFFICIENT EQUALS THE MAXIMUM LIFT COEFFICIENT. WHEN A LIMIT IS REACHED, RECORD THE LOAD FACTOR, THE THRUST VALUE AND THE LIFT COEFFICIENT VALUE, NOTING WHETHER THE LIMIT LOAD FACTOR IS DETERMINED BY AVAILABLE THRUST OR MAXIMUM LIFT COEFFICIENT.

$$BKANG := \left(\mathrm{acos}\left(\frac{1}{LDFT}\right)\right) \cdot 57.3 \qquad BKANG = 59.83804$$

TURN LIFT COEFFICIENT	TURN THRUST AVAILABLE	TURN VELOCITY
$CLTRN = 0.98879$	$TATRN = 1.76344$	$VTRN = 50.4$
CLMAX CLEAN	TURN THRUST REQUIRED	TURN LOAD FACTOR
$CLMXC = 1$	$TRTRN = 1.76807$	$LDFT = 1.99$

CALCULATE A MAXIMUM SEA LEVEL STANDARD DAY SPEED ESTIMATE FOR THE DESIGN AIRCRAFT

$WTSPD := 10$

$RHOS := RHOSL$

$VMSPD := 66.3$

SYMBOL DEFINITION:

SPDWTR:	MAX SPEED WEIGHT/ TAKEOFF WEIGHT RATIO
WTSPD:	MAX SPEED WEIGHT
WLSPD:	MAX SPEED WING LOADING
RHOS:	SPEED ATMOSPHERIC DENSITY
CLSPD:	MAX SPEED LIFT COEFFICIENT
CDSPD:	MAX SPEED DRAG COEFFICIENT
TRSPD:	MAX SPEED THRUST REQUIRED
TASPD:	AVAILABLE THRUST AT MAX SPEED
RHOS:	DENSITY AT MANEUVER ALTITUDE
VMSPD:	MAXIMUM SPEED AT DENSITY ATITUDE
VMXKTS:	MAXIMUM SPEED IN KNOTS

$$CLSPD := \left(\frac{2 \cdot WTSPD}{RHOT \cdot VMSPD^2 \cdot S}\right) \qquad VMXKTS := VMSPD \cdot \left(\frac{3600}{6080}\right)$$

$$CLSPD = 0.19577 \qquad CDSPD := CDO + \frac{CLSPD^2}{\pi \cdot AR \cdot E} \qquad TRSPD := \frac{WTSPD}{\left(\frac{CLSPD}{CDSPD}\right)}$$

$$TASPD := TRAVST + (VMSPD \cdot TRASLP) \qquad TASPD = 1.10518$$

$$TRSPD = 1.10149$$

OVER A RANGE OF VELOCITIES, INCREASE THE TRIAL MAX SPEED VALUE UNTIL THRUST AVAILABLE EQUALS THRUST REQUIRED THRUST VALUE.

RECORD THE THRUST REQIRED AND MAXIMUM SPEED VALUES.

THRUST AVAILABLE AT MAX SPEED	THRUST REQUIRED AT MAX SPEED	MAX SPEED, ft /sec
$TASPD = 1.10518$	$TRSPD = 1.10149$	$VMSPD = 66.3$
		MAX SPEED, knots
		$VMXKTS = 39.25658$

PERFORMANCE SUMMARY:

DESIGN TAKEOFF WT, lbs:WTO = 14.66667 DESIGN PAYLOAD WT, lbs: WPL = 7.5

DESIGN WING LOADING, lbs/ft^2: WLTO = 1.5 WING ASPECT RATIO: AR = 10

CLEAN A/C MAX LIFT COEF:CLMXTF = 1.3 CLEAN WING EFF. FACTORE = 0.78

CLEAN ZERO LIFT DRAG COEFCDO = 0.02 WING AREA, ft^2: S'= 9.77778

TAKEOFF PERFORMANCE:

TAKEOFF WEIGHT (lbs)	WING LOADING (lb/ft^2)	C_{Lmaxto}	TAKE OFF DISTANCE (ft)
WTO = 14.6667	WLTO = 1.5	CLMXTF = 1.3	slo = 113.53105

LIFT OFF VELCITY ft/sec	OBSTACLE CLEARANCE DISTANCE
vlo = 35.83285	TOTLDIST = 298.04812

LANDING PERFORMANCE

LANDING WEIGHT lbs	TOUCHDOWN VELOCITY ft/sec	LANDING ROLL DISTANCE ft
WLD = 14.66667	vld = 33.70361	sld = 497.85887

CLIMB PERFORMANCE
MAXIMUM CLIMB ANGLE

CLIMB RATE ft/sec	CLIMB VELOCITY ft/sec	CLIMB LIFT COEFFICIENT	CLIMB DRAG COEFFICIENT
CLMRATE = 3.68347	VCLM = 39	CLCLM = 0.82981	CDCLM = 0.0481

CLIMB HORIZ VELOCITY ft/sec	CLIMB ANGLE deg
VHORZ = 38.82566	CLMANGD = 5.41995

MANEUVERING PERFORMANCE

VELOCITY ft/sec	LOAD FACTOR	LIFT COEF	MAX LIFT COEF. CLEAN
VTRN = 50.4	LDFT = 1.99	CLTRN = 0.98879	CLMXC = 1

TURN BANK ANGLE deg	TURN THRUST AVAILABLE	TURN THRUST REQ'D
BKANG = 59.83804	TATRN = 1.76344	TRTRN = 1.76807

SPEED PERFORMANCE

MAX SPEED ft/sec	SPEED THRUST REQ'D	SPEED THRUST AVAILABLE
VMSPD = 66.3	TRSPD = 1.10149	TASPD = 1.10518

TAKEOFF PERFORMANCE SUMMARY:

TAKEOFF WEIGHT (lbs)	**WING LOADING** (lb/ft^2)	C_{Lmaxto}
WTO = 14.6667	**WLTO = 1.5**	**CLMXTF = 1.3**
TAKEOFF ROLL (ft)	**LIFT OF VELOCITY** (ft/sec)	**TAKEOFF THRUST** lbs
slo = 113.53105	**vlo = 35.83285**	**TRAVST = 3.85**

AFTER TAKEOFF CLIMB

CLIMB RATE (ft/sec)	**OBSTACLE CLEARANCE TOTAL T.O. DISTANCE** (ft)
PTCLMR = 2.90339	**TOTLDIST = 298.04812**

Appendix C

Sample Preliminary Neutral Point Location Calculations

NEUTRAL POINT LOCATION CALCULATION

SH:	HORIZONTAL TAIL AREA
S:	WING AREA
XACT:	HORIZ. TAIL A.C. DISTANCE TO M.A.C. LEADING EDGE
HACWB:	WING/BODY A.C.
AHT:	HORIZ TAIL LIFT CURVE SLOPE
AWG:	WING LIFT CURVE SLOPE
MAC:	M.A.C. LENGTH
DEDA:	DOWNWASH DERIVATIVE
HCG:	C.G. LOCATION IN % CHORD

$SH := 173.14 \quad S := 1152.56 \quad XACT := 30$

$AHT := 0.06 \quad AWG := 0.086 \quad MAC := 10.67$

$HACWB := 0.25 \quad DEDA := 0.26$

$$HN := \frac{HACWB + \left[(SH \cdot XACT \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot MAC \cdot AWG}\right]}{1 + (SH \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot AWG}} \qquad HCG := 0.30$$

$HN = 0.434 \qquad SM := HN - HCG \qquad SM = 0.134$

XACT	HN	STATIC MARGIN
30 in	0.434	13.4%
25 in	0.401	10.1%
24 In	0.394	9.4%
23 in	0.387	8.7%
22 in	0.38	8.0%
21 in	0.374	7.4%

NEUTRAL POINT LOCATION

CALCULATION

SH: HORIZONTAL TAIL AREA
S: WING AREA
XACT: HORIZ. TAIL A.C. DISTANCE TO M.A.C. LEADING EDGE
HACWB: WING/BODY A.C.
AHT: HORIZ TAIL LIFT CURVE SLOPE
AWG: WING LIFT CURVE SLOPE
MAC: M.A.C. LENGTH
DEDA: DOWNWASH DERIVATIVE
HCG: C.G. LOCATION IN % CHORD

$SH := 173.14$ $S := 1152.56$ $XACT := 21$

$AHT := 0.06$ $AWG := 0.075$ $MAC := 10.67$

$HACWB := 0.25$ $DEDA := 0.26$

Changed from Sheet 1/4

$$HN := \frac{HACWB + \left[(SH \cdot XACT \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot MAC \cdot AWG}\right]}{1 + (SH \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot AWG}}$$

$HCG := 0.30$

$HN = 0.39$ $SM := HN - HCG$ $SM = 0.09$

XACT	HN	STATIC MARGIN
30 in	0.459	15.9%
25 in	0.421	12.1%
24 In	0.413	11.3%
23 in	0.406	10.6%
22 in	0.398	9.8%
21 in	0.39	9.0%

NEUTRAL POINT LOCATION

CALCULATION

SH:	HORIZONTAL TAIL AREA
S:	WING AREA
XACT:	HORIZ. TAIL A.C. DISTANCE TO M.A.C. LEADING EDGE
HACWB:	WING/BODY A.C.
AHT:	HORIZ TAIL LIFT CURVE SLOPE
AWG:	WING LIFT CURVE SLOPE
MAC:	M.A.C. LENGTH
DEDA:	DOWNWASH DERIVATIVE
HCG:	C.G. LOCATION IN % CHORD

$SH := 173.14$ $S := 1152.56$ $XACT := 30$

$AHT := 0.06$ $AWG := 0.075$ $MAC := 10.67$

$HACWB := 0.15$ $DEDA := 0.26$

changed from sheet 2/4

$$HN := \frac{HACWB + \left[(SH \cdot XACT \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot MAC \cdot AWG} \right]}{1 + (SH \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot AWG}}$$

$HCG := 0.30$

$HN = 0.367$ $SM := HN - HCG$ $SM = 0.067$

XACT	HN	STATIC MARGIN
30 In	0.367	6.7%
25 in	0.329	2.9%
24 in	0.321	2.1%
23 in	0.314	1.4%
22 in	0.306	0.6%
21 in	0.298	-0.15%

NEUTRAL POINT LOCATION

CALCULATION

SH:	HORIZONTAL TAIL AREA
S:	WING AREA
XACT:	HORIZ. TAIL A.C. DISTANCE TO M.A.C. LEADING EDGE
HACWB:	WING/BODY A.C.
AHT:	HORIZ TAIL LIFT CURVE SLOPE
AWG:	WING LIFT CURVE SLOPE
MAC:	M.A.C. LENGTH
DEDA:	DOWNWASH DERIVATIVE
HCG:	C.G. LOCATION IN % CHORD

$SH := 173.14 \quad S := 1152.56 \quad XACT := 21$

$AHT := 0.06 \quad AWG := 0.075 \quad MAC := 10.67$

$HACWB := 0.25 \quad DEDA := 0.18$

Changed from sheet 2/4

$$HN := \frac{HACWB + \left[(SH \cdot XACT \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot MAC \cdot AWG}\right]}{1 + (SH \cdot AHT) \cdot \frac{1 - DEDA}{S \cdot AWG}} \qquad HCG := 0.30$$

$HN = 0.404 \quad SM := HN - HCG \quad SM = 0.104$

XACT	HN	STATIC MARGIN
30 in	0.48	18.0%
25 in	0.438	13.8%
24 in	0.429	12.9%
23 in	0.421	12.1 %
22 in	0.413	11.3%
21 in	0.404	10.4%

Appendix D

Sample ElectriCalc Program Output Data

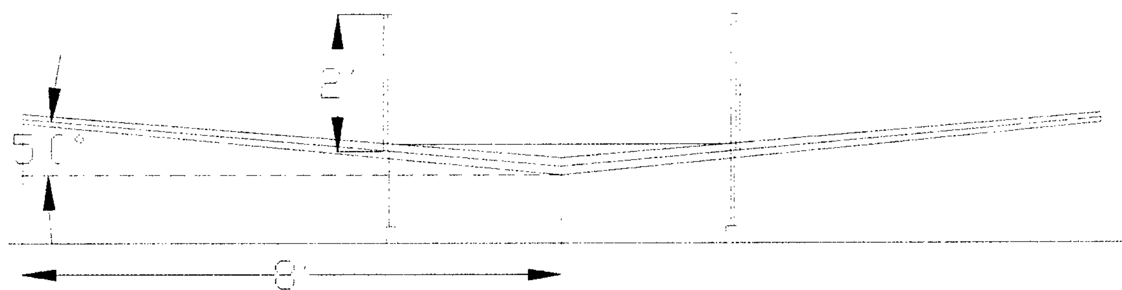
Setup	aggie1	aggie2	aggie3
Prop KRPM	3.64	5.15	6.52
Motor KRPM	13.49	19.07	24.11
Prop Watts	123.	347.	701.
Motor Watts	154.	406.	803.
Motor Amps	16.3	30.1	46.6
Motor Volts	9.5	13.5	17.2
Battery Amps	8.2	22.6	46.6
MAH	1950.	1950.	1950.
Minutes	13.6	4.9	2.4
% Throttle	50.	75.	100.
% System Eff.	66.	67.	66.
% Motor Eff.	82.	88.	90.
Prop Diameter	15.00	15.00	15.00
Prop Pitch	9.50	9.50	9.50
Pitch speed	33.	46.	59.
Plane weight	256.	256.	256.
Wing area	1152.	1152.	1152.
Wing loading	32.0	32.0	32.0
Drag coeff.	0.020	0.020	0.020
Watts/pound	10.	25.	50.
Climb rate	-15.	180.	1055.
Climb angle	0.	2.	13.
Thrust	11.	30.	76.
Drag	12.	21.	21.
Stall speed	21.	21.	21.
Max. speed	40.	57.	73.
Speed	41.	55.	55.
Motor	1412/2Y	1412/2Y	1412/2Y
Mfr	AVEOX	AVEOX	AVEOX
Kv	1475.	1475.	1475.
Rm	19.	19.	19.
Io	2.5	2.5	2.5
Gearing	3.70	3.70	3.70
Motor Config.	1	1	1
Prop Mfr.	thin carbon folder	thin carbon folder	thin carbon folder
K prop	1.31	1.31	1.31
K pitch	0.00	0.00	0.00
Cell Type	N-1700SCRC	N-1700SCRC	N-1700SCRC
Cell Count	19.	19.	19.
Cell Volts	1.20	1.20	1.20
Cell mohm	5.5	5.5	5.5
ESC mohm	15.	15.	15.

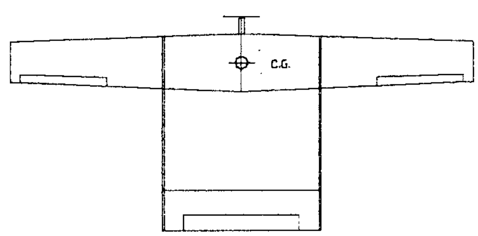
ElectriCalc 03-12-1997 20:34:44

in
degrees
MPH
oz
in^2
oz/ft^2
ft/min
degrees
oz
oz
MPH
MPH
MPH

DESIGN REPORT FOR

CESNA ONR/STUDENT DESIGN/BUILD/FLY COMPETITION





Polytechnic University
March 10,1997

Table of Contents

	Page
Executive Summary	1
Management Summary	4
Conceptual Design	6
Preliminary Design	14
Detailed Design	28
Drawings	38
Manufacturing Plan	45

EXECUTIVE SUMMARY

The competition required us to design, fabricate and demonstrate the flight capabilities of an unmanned, electric powered, radio controlled aircraft which provides the maximum range for a battery weight of 2.5lb. The aircraft must be able to take-off unassisted within a distance of 300 feet, clearing a 10 feet obstacle at the end of the runway. In addition, the aircraft is to carry a payload of 7.5 lb. of a specified shape and quantity. Once in air, the aircraft is to fly on fixed course completing as many laps as possible. The aircraft has to land and stop on a runway of length 300 feet.

Our design focused on maximizing the overall range with the amount of energy available. This requires 'stretching a Joule' of energy to as many miles as possible. We realized that a low speed, high endurance flight is not the design objective. Instead, efficient usage of the available energy to give the maximum number of laps is. Although, there was a limitation on the weight of the battery pack, there was none on the amount of energy capacity of the pack. Through research and calculations, we found the cells that would maximize the energy in the battery pack. Next, we attempted to find the most efficient way of draining energy at all phases of flight such as take-off, climb, cruise and maneuvers. This required analyzing each phase carefully and identifying all of the parameters that affect the energy consumption rate. Our analysis showed that for every wing loading, there exists an velocity that gives the best range. We also noted that the range available strongly depends on the wing loading. We concluded that an optimum W/S has to be found to 'stretch a Joule' as far as possible. Minimizing the drag on the UAV was considered of first priority in our design. Since without drag, once in air there will be no need to expend energy. Moreover, we realized that minimizing electrical losses in the propulsion system is as important as reducing the aerodynamic drag. After all, there is no use of maximizing the energy aboard and losing half of it due to low efficiency.

The configuration we selected only consists of a wing and a tail. This configuration is a hybrid between a pure flying wing and conventional wing-fuselage-tail aircraft. The flying wing has excellent drag characteristics and the conventional aircraft can be designed to have excellent handling qualities. Since our objective was to identify a balance between performance and handling qualities, we formed a configuration by borrowing the performance characteristics from a flying wing and the stability characteristics from a conventional aircraft. Through further research, engineering analysis and streamlining of parameters, we were able to design an excellent flying machine with adequate stability and excellent performance characteristics. We predict that our UAV will make at least 19 laps and will stay in the air for more than 17 minutes.

The overall design process was split into four phases, each with its own objective. The chart in figure 1.1 shows the tasks assigned to each of the design phases.

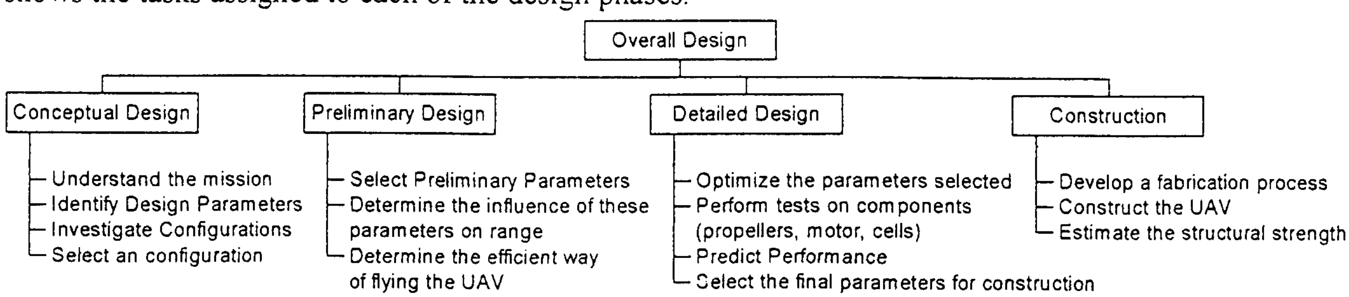


Figure 1.1. The four phases in UAV Design Process

Conceptual Design Stage:

The objective of this stage was to identify a configuration that would meet the mission requirements and give the best performance possible. During this stage, we identified the major parameters to be optimized to maximize the UAV range. The expression we used to recognize these parameters was Range = Velocity × Time, or in its full form $R = \frac{P_{batteries} \times \eta_0 \times K}{D} \times \frac{Amp - Hr \times 3600}{I}$. Where D is the drag, I is the input current and η is the propulsion efficiency, P is the power input. From this expression, we were able to isolate the parameters that affect the range. These are drag, efficiency, and the total energy stored onboard. Each of these items was thoroughly analyzed to find their effect on the overall range of the UAV. We found that the drag has to be a absolute minimum to get the maximum number of laps. Likewise, the propulsion efficiency has to be as high as possible to keep the energy resources as long as possible. We also found that the amount energy on board determines the flight time available and hence the range. Having identified the parameters to be streamlined, we investigated three different configurations including an all wing configuration, a canard configuration and conventional aircraft. Based on the conclusions and analysis, we selected a hybrid configuration between the conventional and the all wing configuration.

We utilized various design tools to reach a conclusion. The primary tools used at this stage were simple expressions, charts and graphs. Sources of information such as the Internet, library and encyclopedias were used extensively to gather information the configurations investigated. Brainstorming used throughout the design process to solve difficult problems.

Preliminary Design Stage:

Our objective during the preliminary design stage was to identify ways to maximize the range, and minimize the energy consumption rate. To do this, we analyzed each of the flight phases, take-off, climb, cruise and maneuvers, carefully to find most energy efficient way of performing these phases. We also analyzed the effect of parameters such as the wing loading, and aspect ratio on the performance of the UAV. We were able to select a set of preliminary values for the design parameters by the end of this design stage. Through research and testing, we also found the best cells to be used for the battery pack.

To assist us in our analysis, we utilized computer graphing software packages, simulation programs such the Aerocomp and Microsoft Flight Simulator. We also conducted simple experiments to draw conclusions on the issues at hand. A simple wind tunnel was built using cardboard to test the motor dynamically. We used table fans and a nozzle to get airspeeds up to 20 ft/s. We built a Balsa model of the configuration selected and tested it for handling qualities.

Detailed Design Stage:

During this stage, we found the optimum set of design parameters to maximize the performance. We used the same analysis as in preliminary design to select a refined set of parameters. A large part of this stage included experiments to select the best of the item being considered. The performance of the UAV was estimated through calculations and tests. A scaled model of the final configuration was built and tested for handling capabilities. The control surface areas required were also streamlined using this glider. The construction materials for the final design were also chosen during this stage. Some tests on different type of materials such as balsa, spruce and carbon composites. A software called Compufoil Professional was used to print out the airfoils.

Through our analysis, we were able to develop a unique design that has excellent performance characteristics and good handling qualities and required by the competition. As mentioned above, we expect to complete 19 laps and stay in air for more than 17 minutes.

MANAGEMENT SUMMARY

Our design team included members from various engineering fields. When the decision was made to work on the competition, we advertised throughout the school to inform any interested students about the competition. We tried to build a well balanced and a diverse team so that our design will be unique. The team we formed consisted of three senior engineering students, four juniors, one sophomore and two freshmen. Since the success of our design depended on teamwork, all of the members were made aware of their responsibilities and the importance of their participation in the project work. Most of the members in our team had enough leadership and management skills from their past activities as leaders. For example, our design manager held positions in the Student Council and in the Tau Beta Pi National Engineering Society. The team members were motivated and excited to work on a project to build an UAV. All of the members, their class status and fields of study are listed in table 2.1. We also found two engineers as our advisors for the project. Both of these engineers had ample experience in designing real as well as model aircraft. We requested the advisors not to step in until we requested help. This will ensure a design process completely managed, and organized by students alone.

Name	Class	Major
Pradeep Fernandes	Senior	Aerospace Engineering
Jonathan Katz	Senior	Mechanical Engineering
Ron Amster	Senior	Mechanical Engineering
Soufiane Toury	Junior	Aerospace Engineering
Wei-Jen Su	Junior	Aerospace Engineering
Khurram Butt	Junior	Aerospace Engineering
Igor Cherepinsky	Junior	Aerospace/Electrical Eng.
Stan Markelov	Sophomore	Aerospace Engineering
Nicole Massey	Freshmen	Aerospace Engineering
Sema Simsek	Freshmen	Aerospace Engineering
Prof. Sforza	Professor	Aerospace Engineering
Mr. Henry Prew	Engineer	Aerospace Engineering
Mr. George Myers	Engineer	Aerospace Engineering

Table 2.1. Team members

Team structure:

The design team was initially split into three teams, the aerodynamics team, structures team and electrical analysis team. Each team had a leader and enough members to conduct the team's business. These sub teams performed the engineering analysis at their own convenience and a general meeting was held once a week to discuss any issues that affected the whole design team. The team adhered to this structure as strictly as possible so that analysis could be done simultaneously and all of the members will be involved. During later stages of the design, the three teams were merged to form a single team for the construction phase. The merging was necessary at this stage because a change in one of the parameters affected the rest of the parameters. Hence, individual meetings could not be held. Chart in figure 2.2 shows the details of the team organization.

The construction team included all of the members. Each member assigned a specific task during the construction process. Since all of the members could not meet at the same time, e-mail and paper messages were used as a means of communication. A small group was formed to write the report as soon as the detailed design stage ended.

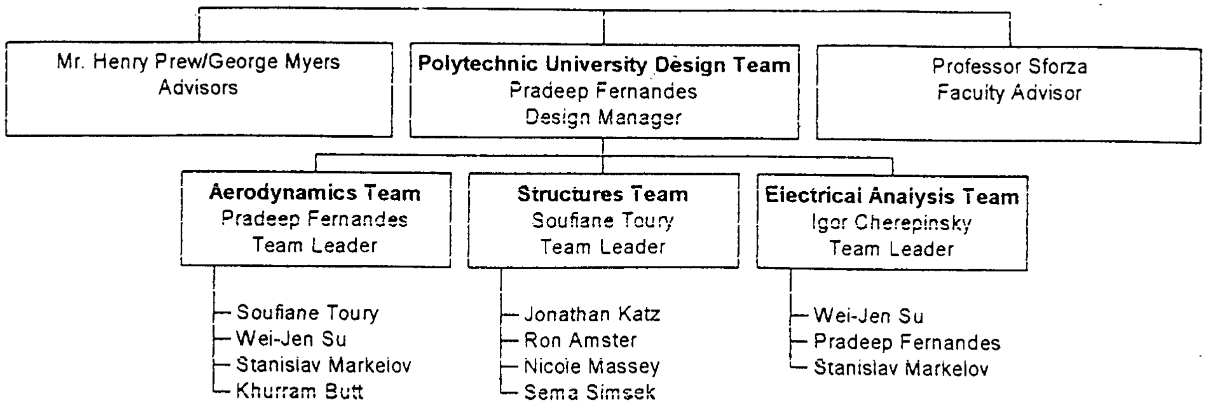


Figure 2.2. Design team structure

Assignment Strategy:

Assignments were based on personal interests and enthusiasm. The members were allowed to chose the teams they wanted to work for. Learning from each other and research was strongly encouraged. Freshmen and sophomores majoring in the aerospace engineering program were encouraged to join the aerodynamics team so that they can get an early introduction to the subject matter. Senior level students were asked to give as much information as possible to the underclassmen. Although winning the competition was the overall objective, gaining practical experience by putting theory to work was also considered a major objective. Members were allowed to join more than one team if they wished. Moreover, to build a transition for the future, juniors were given more responsibilities to prepare them for the future projects both in school and outside world.

Schedule Control:

To make all team members aware of the schedule and our status, a copy of the schedule was attached to the wall in the room where the design meetings were held. At the early design stages, each team was given a strict deadline to finish their business. If the team missed the deadline without a proper excuse, the team leader was held responsible to get the job done as soon as possible. At the end of a major design stage, a special meeting was held to discuss the scheduling for the next design phase. In developing the schedule, all of the members were requested to hand in a copy of their class schedule so that the design schedule will not conflict with the class schedule.

Project Schedule

Design Phase	Scheduled Start – Finish	Actual Start – Finish
Conceptual Design Stage	7/1/96 – 8/22/96 - 8 weeks	7/1/96 – 9/1/96 - 9 weeks
Letter of Intent to Participate	9/30/96	9/30/96
Preliminary Design	8/22/96 – 10/16/96 - 8 weeks	9/1/96 – 10/24/96 – 8 weeks
Research on UAVs	9/1/96 – 9/15/96 - 2 weeks	9/1/96 – 9/20/96 - 2 ½ weeks
Detailed Design	10/16/96 – 12/10/96 – 8 ½ weeks	11/1/96 – 1/27/97 – 12 weeks
Manufacturing Plan	1/1/97 – 1/21/97 - 3 weeks	1/27/97 – 2/10/97 - 3 weeks
Construction Phase (first model)	1/25/97 – 3/20/97	2/5/97 - Present

CONCEPTUAL DESIGN

The objective for this stage was to understand the mission requirements and to select an aircraft configuration. We analyzed the mission carefully until the major parameters that affected the UAV's performance were identified. We tried to come up with a configuration that satisfied the design parameters identified. Our analysis showed that the objective is to minimize the energy consumption rate. This also means that the energy on board should be maximized. Combining the above two items will give the maximum range and hence a maximum number of laps. Following this analysis, an extensive research was conducted to gather information on the existing UAVs, DC motors, batteries, construction materials and aircraft modeling accessories available. By the end of this design stage, we had selected a configuration that consisted only of a wing and a tail. Decisions were also made on type of motor and cells to be used.

Mission analysis:

An attempt was made to determine parameters that affect the energy consumption rate and the range of the UAV. We based our analysis on the simple expression below. We assumed that a straight-line flight would give the maximum possible range since energy is not expended in any maneuvers. The following expression will give an optimistic result for the range. But, for the purposes of comparing and selecting design parameters, this expression is extremely useful.

$$Range(R) = Velocity(V) \times Time(t) \qquad (1)$$

To maximize R, it is desirable to maximize both velocity and flight time. An attempt was made to see if this can be achieved for electric flight. Our analysis showed that both cannot be maximized simultaneously but they can be maximized independently. We then found the optimum combination of these two parameters to give the best range. This analysis is explained in the in the preliminary design section of this report. Expression (1) is simply used here to identify the parameters that affect the overall range.

$$Power\ Required(P_{req}) = V \times Thrust(T)$$

But for level flight $\quad T = Drag(D)$

Thus, $\quad V = P_{req}/D \qquad (2)$

Also $\quad P_{req} = K \times P_{batteries} \times Net\ Propulsion\ Efficiency\ (\eta_0)$

Where $P_{batteries}$ is the power drained from the batteries.

$K = 550/746 = 0.737$ is the conversion factor between ft-lb./s to Watts

$$V = \frac{K \times \eta_0 \times P_{batteries}}{D}$$

For electric flight, $\quad t = \dfrac{Amp\text{-}Hr \times 3600}{Current\ Drained(I)}$ [in seconds] $\qquad (3)$

From the above two equations, we realized that to get a high velocity a large current has to be drained from the battery pack ($P_{batteries}$ = Volts × Current, Volts are held constant). On the other hand, to get longer flight, current drainage should be kept to a minimum. Thus, ($V \propto I$) and ($t \propto 1/I$). Hence, $V \propto 1/t$. Thus the product of velocity and time has to be maximized for maximum range.

Substituting (2) and (3) in (1), gives

$$R = \frac{P_{batteries} \times \eta_0 \times K}{D} \times \frac{Amp - Hr \times 3600}{I} \qquad (4)$$

Using $P_{batteries} = Volts_{IN} \times Current_{IN}$,

$$R = \frac{Volts \times \eta_0 \times (Amp - Hr) \times K \times 3600}{D} \qquad (5)$$

From expression (4), to maximize the range, drag and current drainage have to be a minimum, and the propulsion efficiency and battery Amp-Hr have to be as high as possible.

Another important parameter is the power output per unit current input; $\frac{P_{batteries} \times \eta_0}{I}$. This is used as a selection criterion between different types motors available. It is desirable to have a motor that delivers highest power output per unit current input to maximize range.

Expression (5) reveals yet another term, *Volts × Amp-Hr = Watt-Hr,* which determines the best cells for the battery pack. It is desirable to have a battery pack with as high a Watt-Hr capacity as possible.

The following sections analyze each of the parameters identified above in more detail to recognize any other hidden parameters.

DRAG:

$D = \frac{1}{2} \rho V^2 S C_D$

$C_D = C_{D0} + K C_L^2$

$C_{D0} = f(\text{Surface area, skin roughness, profile})$

$$K = \frac{1}{\pi \cdot AR \cdot e}, \text{ where } AR = \frac{b^2}{S}$$

AR – Aspect Ratio; S – Reference Surface Area.

An alternative analysis shows the effect of yet another important parameter -- the lift to drag ratio (E).

$$D = \frac{D}{L} L = \frac{L}{L/D} = \frac{L}{E} \qquad \text{where } L = lift$$

The higher the E, the lower the drag will be. The maximum value of E can be obtained from the following expression:

$$E_{max} = \frac{1}{2\sqrt{KC_{D0}}}$$ [1]

Here again, K and C_{D0} have to be minimized to increase E_{max} and lower drag.

Table 3.1 shows our conclusions from the above analysis of drag.

Surface Area(S)	Minimize
Flight speed(V)	Minimize
C_{D0}	Minimize
K	Minimize
C_L	Minimize

[1] Hale, Francis., Introduction to Aircraft Performance, Selection and Design

AR	Maximize
B	Maximize

Table 3.1 Parameters to be optimized to minimize drag

The chart in figure 3.2, summarizes the parameters that influence the drag in detail. This chart will be used throughout the design to watch out for values of any parameters that might affect the drag.

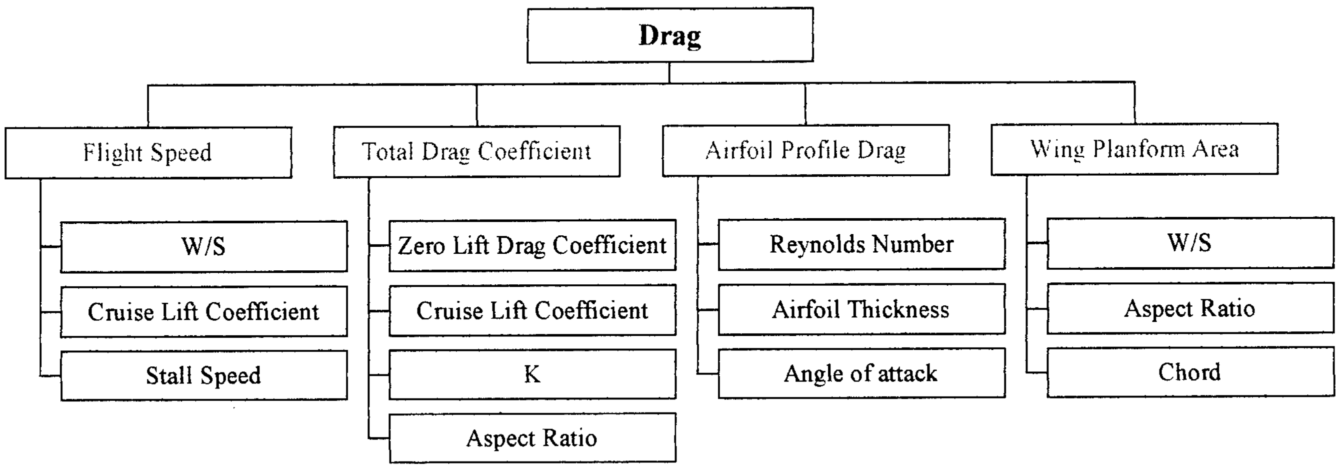


Figure 3.2. Parameters that influence the Drag on an aircraft

Even though figure 3.2 suggests that the flight speed be should be reduced to decrease drag, as discussed earlier an optimum value has to be found for this parameter. Moreover, the airspeed can not be reduced arbitrarily since the required lift and stall conditions. Preliminary design section of this report will address this problem.

FLIGHT TIME:

Flight time depends on the amount of energy available and the rate of energy consumption. It is very important to select a battery pack that allows the maximum energy storage onboard. The constraint on the battery pack is that the whole pack should weigh less than or equal to 2.5 lb. (40 Oz.). Thus, battery selection depends on storage capacity per weight of the cell. This leads to selecting cells that would give the maximum energy capacity in 40 Oz.

The flight time can be approximated by: $t = \dfrac{\text{Amp-Hr} \times 3600}{\text{Current Drained}(I)}$ *[seconds]*

It appears as if a battery with as high of a *Amp-Hr* as possible will be beneficial to maximize the flight time. But, previously it was shown that the actual parameter of interest is *Watt-Hr*. This again leads to selecting batteries with highest energy/weight of the cell.

We conducted a study on cells from different manufacturers. The objective of the study was to identify the cells that would give the maximum energy storage in a 40 Oz. battery pack. We found that a 2.0 Amp-Hr battery is the most desirable choice. This battery gives up to 62.5 Watt-Hr storage onboard. During the preliminary design, a more thorough research will be conducted to determine the discharge characteristics of this battery. The decision between different manufacturers will be made based on the operational characteristics of the battery.

Table 3.3a and 3.3b show the result of a study some nickel-cadmium batteries

SR Batteries					
Amp-Hr	**Voltage/cell**	**Weight/Cell**	**# of Cells**	**Voltage**	**Watt-Hr**
1.8	1.25	1.86 Oz	21	26.25	47.25
2.0	***1.25***	***1.86 Oz***	***21***	***26.25***	***52.5***
2.5	1.25	2.47 Oz	16	20	50
5.0	1.25	5.3 Oz	7	8.75	43.75

Table 3.3a. Nickel Cadmium cell characteristics for SR batteries

Sanyo					
Amp-Hr	**Voltage/cell**	**Weight/Cell**	**# of Cells**	**Voltage**	**Watt-Hr**
1.8	1.25	2.8 Oz	14	17.5	31.5
2.0	***1.25***	***1.6 Oz***	***25***	***31.25***	***62.5***
2.3	1.25	2.0 Oz	20	25	57.5
2.5	1.25	3.9 Oz	10	12.5	31.25
5.0	1.25	5.5 Oz	7	8.75	43.75

Table 3.3b. Nickel Cadmium cell characteristics for Sanyo batteries

The rate of energy consumption affects the duration of the finite energy resources. In our case, the consumption is directly proportional to the amount of drag the airplane has and the amount of lift required. Since the amount of lift required is equal to the weight of the UAV, reduction in weight is important. Also, rate of consumption depends on the flight profile. Minimum energy will be consumed if the UAV flies a constant flight path such as a straight-line path. Any turns or other maneuvers will require more energy. The atmospheric conditions, such as wind speed and turbulence will also affect the amount of energy drained from the pack. The chart in figure 3.4 summarizes these conclusions.

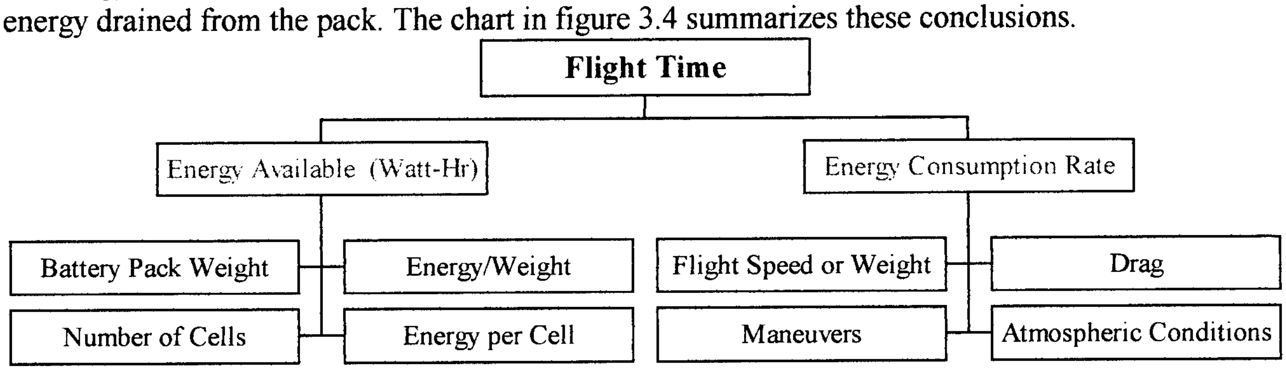


Figure 3.4. Parameters that influence the flight time.

PROPULSION EFFICIENCY:

Propulsion efficiency depends on all of the components that make up the propulsion system. The propulsion system of this UAV can be broken down in to the following components: motor, battery pack, propeller and wiring. The motor converts electrical energy into mechanical energy and needs to be very efficient. Propeller efficiency is equally important since it converts the torque developed by the motor in to thrust. During this design stage, an attempt was made to come up with a baseline for comparing and selecting different DC motors available in the market. We focused on the different types of motors available. That is, whether to use a brushed motor or a brushless motor. There was very little that could be done at this point on propellers since the competition required us to use a commercial propeller. The diameter and pitch of the propeller depends on motor chosen for the final design. Figure 3.5 shows the parameters that affect the propulsion efficiency.

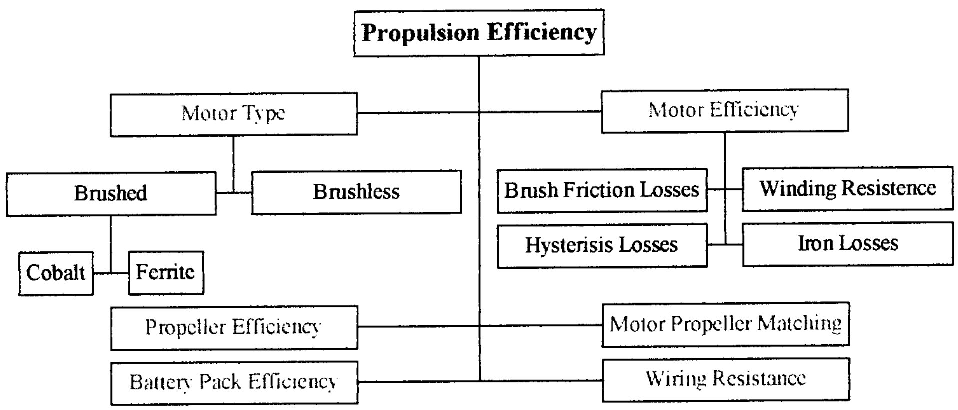


Figure 3.5. Parameters that affect the propulsion efficiency

As discussed before, a useful selection criterion between different types of motors besides the efficiency is the power output per unit current input (P_{out}/I). Our research on several different types of motor showed that a brushless motor is the most desirable choice for our purposes. Table 3.6 shows the results of our research. The data is shown for 20 Volts and 10 Amps.

Motor/Type	P_{out}/I *[Watts/Amps]*	*Maximum Efficiency*
Aveox 4609/Brushless	22	88%
Max15Y/Brushless	20	86%
Astro40G/Brushed	15	79%
Astro60G/Brushed	13.75	81%

Table 3.6. Comparison between different motors in terms of output power per unit current input

Selecting a Configuration:

The above analysis is summarized in the chart in figure 3.7. This every item indicated on this chart was used as a figure of merit in selecting a configuration. We considered the following three configurations:

- A conventional wing-fuselage-tail configuration

- An all wing configuration
- A canard configuration

Conventional aircraft have been successfully designed and studied over the years. This is the most commonly used configuration. There is plenty of information available on the performance, stability and design of conventional aircraft.

An all wing configuration was considered because of its low drag characteristics. A flying wing has only a wing and no other components such as the tailplane. Our research showed that there can be stability and control problems with this configuration. Moreover, the amount of information available on the design and optimization of flying wings is limited.

A canard configuration is similar to a conventional aircraft. The only difference is that the tail surfaces are located ahead of the wing. Our research showed that, the wing flies in the wake of the canard and is not as effective. Since the objective of our design is to make all components of the UAV as efficient as possible, this configuration was not further pursued.

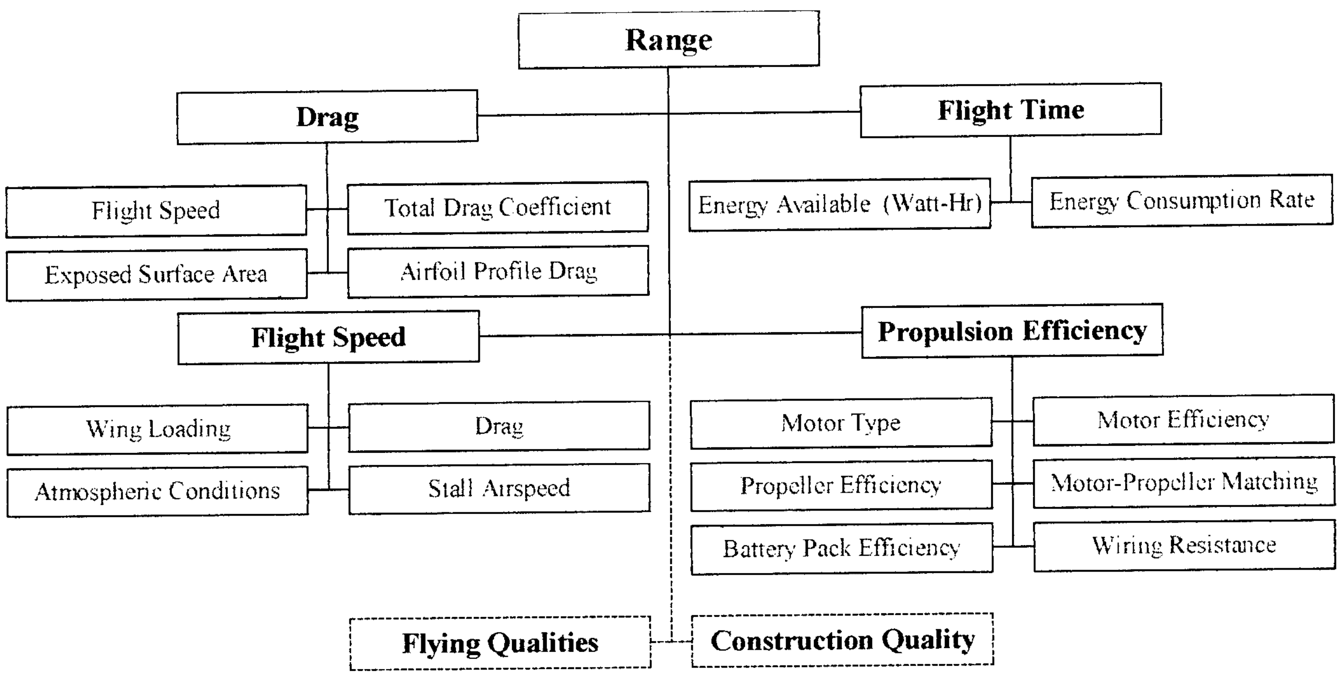


Figure 3.7. Detailed summary of parameters that influence the overall range.

The choice was between a flying wing and a conventional aircraft. We tried to perform some calculations to help us decide on one of these two configurations. Table 3.8 shows numbers obtained from our calculations. The calculations are shown relative to the conventional aircraft. For example, if the take off velocity required for the conventional aircraft is 1 ft/s, a velocity of 1.22 ft/s second would be required for the flying wing assuming a 25^0 sweepback at the quarter chord. For the reasons of stability, the flying wing normally requires a high sweepback angle. Since the wing only sees the normal component of the flow, having a high sweepback angle reduces the flow speed seen by the wing. To produce the same amount of lift, more power will have to be expended to speed up the flow over the wing. In the case of a conventional aircraft, a sweepback is not at all necessary. But, the flying wing has about 30 % less drag for the same weight and wing planform area compared to a conventional aircraft.

Based on this analysis, we fabricated a configuration by borrowing the performance characteristics (low drag) of a flying wing and the stability characteristics of a conventional aircraft. The resulting configuration consisted of a wing and a tail only – the two components that make up the basic aircraft. These two components would be connected to each other by two booms. The wing will provide enough storage space in the root section for batteries and payload.

The data in table 3.8 is based on a UAV weighing 25 lb. and a wing planform area of 15 ft^2.The runway length was assumed to be 300 ft.

	Required Sweep back	Take off velocity for the same weight	Thrust Required to take off	Drag	Stability
Conventional aircraft	0 deg.	1	1	1 lb	Excellent
Flying wing	> 25 deg	1.22	1.12	0.7 lb.	Might have problems.

Table 3.8. Data used to compare a flying wing and a conventional configuration.

During the preliminary design stage, an optimum set of design parameters will be found to maximize the performance of this configuration.

PRELIMINARY DESIGN

The objective of the preliminary design stage was to select a set of useable design parameters. These parameters will be optimized to give the best performance possible at a later design stage. During the selection of the design parameters, we particularly focused on the design considerations identified during the conceptual design stage. The overall goal of the design was to maximize the number of laps with available energy, which directly translates into maximizing the range. We based our analysis on comparing the effect of various parameters on range and flight time. An attempt was made to identify the trend of range and flight time with varying values of wing loading (W/S), aspect ratio (AR), cruise speed (V) and total energy available. The W/S was considered the base parameter and the rest of the parameters were related to the W/S. Hence, selecting a W/S would determine a power to weight ratio (P/W), wing planform area (S), a runway length and the maximum coefficient of lift required.

The information obtained from research on existing UAVs was used extensively at the beginning of our analysis. A database was compiled on the characteristics of existing UAVs to assist the team in selecting and comparing design parameters. The first sizing parameter obtained was the weight of the UAV. A rough estimate of the total take-off weight was calculated with the information already available for the UAV. The calculations showed that the UAV might weigh somewhere between 20 - 25 lb. The analysis used is shown below:

$$W_{Take\text{-}Off} = W_{Structure} + W_{Payload} + W_{Batteries} + W_{Propulsion\ System} + W_{landing\ Gear}$$
$$W_{TO} = W_S + W_{PL} + W_B + W_P + W_{LG}$$

Known weights:
$W_{PL} = 7.5\ lb.$
$W_B = 2.5\ lb.$

Weights assumed through research:
$W_P = 2\ lb.$ *(2 motors for worse case)*
$W_{LG} = 1\ lb.$

$W_{TO} = W_S + 7.5 + 2.5 + 2 + 1$
$W_{TO} = W_S + 13$

Assuming $W_S/W_{TO} = 0.4$ *(from research)*
$W_{TO} = 0.4\ W_{TO} + 13$
$\mathbf{W_{TO} = 21.7\ lb.}$

For the purposes of calculations during this stage, we set the UAV weight to 25 lb. From the database compiled, the UAVs that weighed around 25 lb. were selected for further analysis. Table 4.1 below shows the characteristics of these UAVs

Name	Empty Weight [lb]	T.O. Weight [lb]	Wing Area [ft^2]	Span [ft]	W/S [lb/ft^2]	AR	Power [HP]	P/W [fps]	Prop Diameter	Max Speed [MPH]	Overall Length [ft]	Structure
FOA Sakatan	9.7	16.8		7.02			1.6	52.38		62	5.29	Balsa Wood
EERM SAM -B		20.9	11.85	10.33	1.76	9.01	1.6	42.11	11.8"	57	6.90	Glass fiber propeller
Flogger D	18	23	12	6.79	1.92	3.84	2.9	69.35		52	7.53	Polystyrene foam plastics, wooden stiffeners, plywood, alpatic resin, glass fiber cloth
Bae FLYBAC		24.2		6.00			1.8	40.91			4.81	
HDH – 10	15	25		5.00			2	44.00		81	5.17	Glass fiber covered foam plastics
	13	25	12.76	5.67	1.02	2.52	1	22.00	11"	38	4.00	Graphite re-incorced glass fiber/epoxy, plywood/PCV, Kevlar epoxy, Styrofoam
Hawker		25		5.00			2	44.0		81	5.17	

Name	Empty Weight [lb]	T.O. Weight [lb]	Wing Area [ft^2]	Span [ft]	W/S [lb/ft^2]	AR	Power [HP]	P/W [fps]	Prop Diameter	Max Speed [MPH]	Overall Length [ft]	Structure
								0				
GTS 7901 SKY-EYE		26.5		8.00			1.5	31.13		92	6.67	Balsa wood, plywood, expanded plastics foam, glass fiber and aluminum alloy
RCS Heron		30		9.83			10	183.33		120	7.00	
MicroDrone		30	10	9.00	3.00	8.10						Open-cell plastics foam, epoxy impregnated at hard points and covered with
NRIST	27.6	30.9	8.88	6.94	3.11	5.42	5	89.00		120	6.73	GRP with foam plastics core
IETS 7501 Shrike		31		10.00						138	7.58	Balsa wood, plywood, expanded plastics foam and glass fiber

Table 4.1. Data on commercial UAVs

Since the purpose of this research was to come up with preliminary values for some of the design parameters such as the W/S, wing planform area (S) and cruising speed (V), the available data was plotted against various design parameters and a design region was identified on each of the curves. Similar research was also conducted on the commercially available hobby airplanes. The plots used are shown in figure 4.2 a, b, and c below with the design region marked with a blue rectangle.

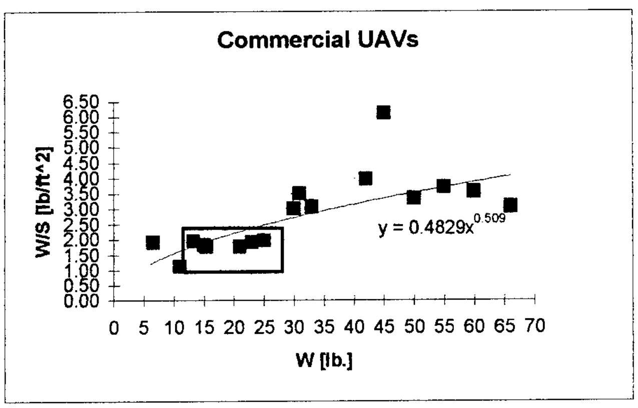


Figure 4.2a. Weight Vs. W/S for UAVs

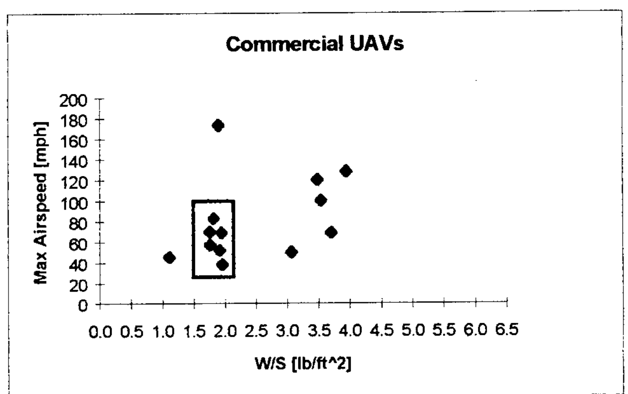


Figure 4.2b. Maximum Speed Vs. W/S for al UAVs

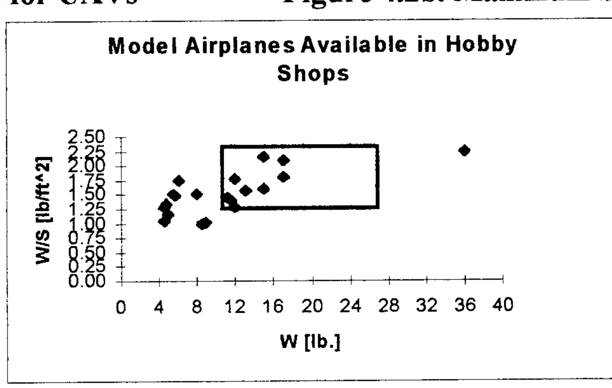


Figure 4.2c. Weight Vs. W/S for Hobby Airplanes

From the above plots, for a weight of 15 to 25 lb. the W/S falls in the range of 1.0 to 2.25 Likewise, the maximum speed for this range varies from 20 to 80 mph. From the table 4.1, it can be seen that the wing planform areas vary from 8 ft^2 to 12 ft^2. However, the UAV under design is unique in design. From the data in table 4.1, we realized that these UAVs are designed for high speed and general purpose. Most of the UAVs appear to be over powered. Almost all of them are powered by piston engines. Our design is unique in a sense that it requires the optimum usage of energy and, more importantly, requires us to fly over a fixed path. The only parameter that we were able to borrow from the above data is the upper limit on the W/S.

Effect of W/S on the performance:

Having determined these approximate limits on the W/S, a detailed analysis was performed on the effect of W/S on the performance of the aircraft. The effect of W/S on take-off, climb, cruise and turning performance was analyzed separately. The objective was to find the most energy efficient way of performing each of these phases. The optimum airspeed required for a given W/S was also calculated. Finally, the effect of W/S on the whole flight was determined by calculating the number of laps available at different W/S. The analysis required calculating the amount of energy available at the beginning of the cruise phase and the energy required for a complete lap. The following sections show a detailed analysis of effect of W/S on each of the flight phases.

Table 4.3 lists the figure of merit used to select a parameter during this analysis. These figure of merit were based on the requirements determined in the conceptual stage and competition requirements.

Figure of Merit	FOM based on
Runway length required	Mission requirement
Climb out angle required to clear 10 ft obstacle	Mission requirement
Energy required to take-off	Guideline set forth from conceptual design
Time required to take-off	Guideline set forth from conceptual design
Range	Guideline set forth from conceptual design
Number of laps	Guideline set forth from conceptual design
Flight time	Guideline set forth from conceptual design, Mission requirement (minimum 3 minute flight)
Turning radius	Guideline set forth from conceptual design, Mission requirement
Transportation requirements	All possible reasons

Table 4.3. Figure of Merit used in parameter selection

We had to make some assumptions to perform the analysis. These assumptions are listed in table 4.4 below. Some of the assumptions might appear unrealistic, but for purpose of comparing, the same two parameters these assumptions are suffice. Every effort was made, however, to make these assumptions as realistic as possible.

Parameter	Assumption	Comment
Aircraft Parameters:		
W/S	< 2 lb/ft^2	From research
Aspect Ratio (AR)	10	Realistic
Oslwald's Efficiency factor	0.85	Realistic
Zero-lift drag coefficient	0.025	Might too high
$C_{L,max}$	0.8 – 1.0	Realistic
Takeoff Speed	$1.2V_{Stall}$	Required
Atmospheric Density (ρ)	23.769×10^{-4} [lb-s^2/ft^4]	Standard Atmosphere, sea level
Runway friction factor	0.04	Concrete runway
Propulsion Efficiency (μ_o)	0.65	Realistic, conservative
Electrical Parameters:		
Input Voltage (V_{IN})	21.6 Volts	1700 MAH, 18 Cells.
Total Watt-Hr Available	36.72 Watt-Hr	Calculated

Table 4.4. Assumptions made during the analysis

Effect of W/S on Takeoff:

To analyze the take-off performance, the power to weight ratio (P/W), the current, and velocity required to takeoff were plotted against W/S. The following relation between P/W and W/S was obtained using expressions from elementary dynamics and the assumptions made above.

$$\frac{P}{W} = \frac{\left[1.2\left(\frac{2(W/S)}{\rho C_{L,\max}}\right)^{1/2}\right]^3}{2gS_{runway}} + 1.2\mu\left(\frac{2(W/S)}{\rho C_{L,\max}}\right)^{1/2} + \frac{1}{2}\rho\left[1.2\left(\frac{2(W/S)}{\rho C_{L,\max}}\right)^{1/2}\right]^3\left(\frac{S}{W}\right)\left(C_{D0} + \frac{KC_{L,\max}{}^2}{1.44}\right)$$

Assuming different runway lengths,. the curves in figure 4.5a and 4.5b were obtained.

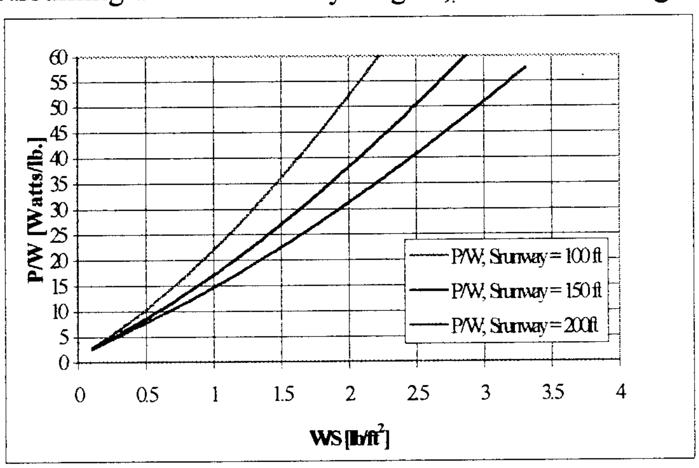


Figure 4.5a. P/W ratio required to take-off for

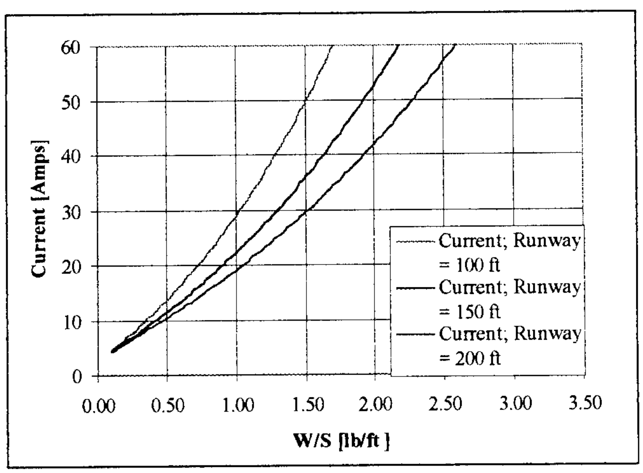


Figure 4.5b. Current required to take-off

different W/S **for different W/S**

To calculate the amount current required to take-off, the power required was related to the current drained from the batteries using the expression $P_{OUT} = \mu_o(I_{IN} - I_O)\times(V_{IN} - I_{IN} \times R_o)$[1]. This expression gives the power in watts in terms of the input current and motor efficiency parameters. It was assumed that the voltage of the battery pack is constant. This is a good assumption for Nickel Cadmium batteries during take-off. The expression above was solved for current *(I)* to obtain

$$I_{IN} = \frac{(V_{IN} + I_O R_O) - \sqrt{(V_{IN} + I_O R_O)^2 - 4R_O(I_O V_{IN} + P_{OUT}/\mu_o)}}{2R_O} \qquad (1)$$

Where I_{IN} is the current input (=current drained from battery pack)

$I_o = 1.2$ *Amps* the no-load current (efficiency parameter)

$V_{IN} = 21.6$ *Volts* the input voltage

$R_o = 0.060$ *Ohms* is the winding resistance

For the purpose of calculations, all of the efficiency parameters needed were obtained from a brushless motor manufacturer since it was already decided that a brushless motor would be the best choice for this project.

We also determined the most energy efficient way of taking off from the above analysis. Table 4.6 shows the energy used to take-off at different W/S and runway lengths. From the data we concluded that the UAV should take-off within short distance. Although this might require more power expenditure, the total energy used is less for a short take-off.

W/S [lb/ft^2]	Runway Length [ft]	Energy [Joules]
0.5	100	1044.5
1	100	1576.3
2	100	2628.84
0.5	150	1300.93
1	150	1832.6
2	150	2885.3
0.5	200	1557
1	200	2089.1
2	200	3141.7

Table 4.6 Energy required to take-off at different W/S and runway lengths

Effect of W/S on Range:

The range was plotted as a function of velocity with the W/S being a parameter. Three different W/S, 0.5, 1, 2, were investigated. Also, the current drain was plotted against cruise airspeed, on the same set of axes. To obtain the velocity versus range curve the following analysis was used.

[1] Boucher, Robert., Electric Motor Handbook

$$R = V \times t$$

$$R = \sqrt{\frac{2(W/S)}{\rho C_L}} \times \frac{Amp - Hr \times 3600}{I}$$

An arbitrary cruise velocity was assumed and based on W/S under consideration, the required C_L was calculated to provide the lift needed to maintain equilibrium flight. Using the assumptions made above, the drag at this velocity was calculated. For equilibrium, Drag = Thrust required. The thrust power (T×V) needed was found by calculating the drag power at this velocity. The amount of current drained was found using expression (1). The plot of range and current drain versus cruise speed is shown in figure 4.7

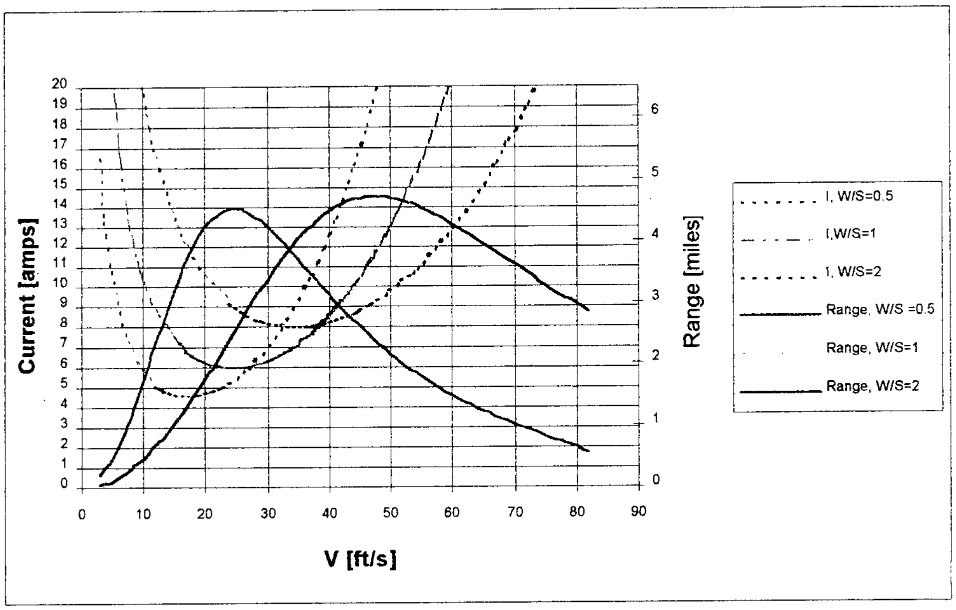


Figure 4.7. Variation of linear range and Current required with W/S

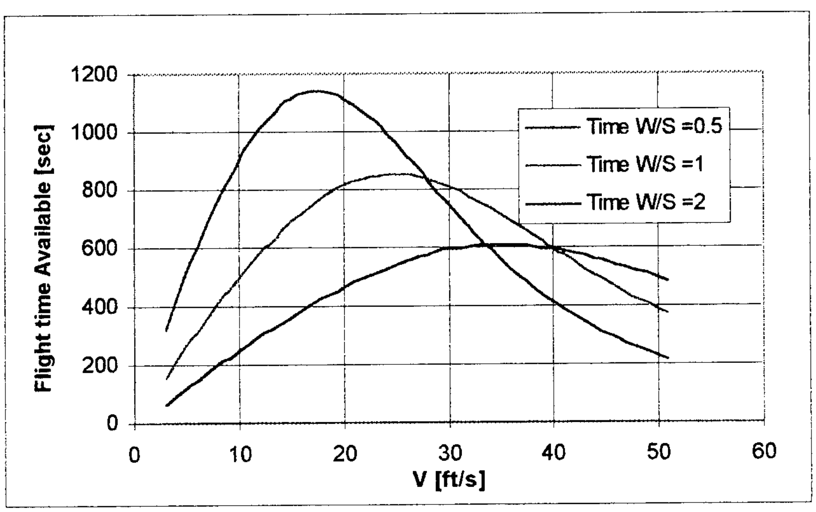


Figure 4.8. Flight time available Vs. W/S for a straight line flight

From the curves above, we concluded that, as the W/S increases the range also increases. Each W/S has a optimum velocity, which *does not* occur at minimum current draw. In addition, the maximum flight time does not occur at maximum range velocity. This can be seen from the above two curves by comparing the cruise speeds at the maxima of the two curves. The maximum range occurs at a higher speed than that required for the maximum time. Although it was enticing to select a W/S of 4 lb/ft^2 based on the range performance, we realized that further analysis was necessary to finalize the W/S selection. Effect of W/S on turning maneuvers was analyzed next.

Effect of W/S on turning performance:
The number of laps depends on the total range(R) available and the length of each lap (L_{lap}).

$$Number\ of\ Laps = \frac{R}{L_{lap}}$$

The length of a lap depends on the turning radius. As required by the competition rules, the distance between the upwind and downwind turn points is 700 ft. Then, the length of a lap is (L_{lap}) $= 2\times700 + 2\pi r$, where r is the turn radius. We realized that for a given total range, a smaller the radius would give greater number of laps.

To analyze the effect of W/S on the turning radius, the minimum turning radius required to make a 3g turn was calculated for each of the following W/S: 0.5, 1, 2. The 3g acceleration was selected for analysis because the competition requires that the UAV be able to sustain a 2.5g acceleration. We assumed that our UAV would be able to sustain a minimum of 3g acceleration. Figure 4.9 shows the plot of radius required to make the turn versus P/W ratio with W/S being a parameter. The acceleration experienced was also plotted on the same set of axes.

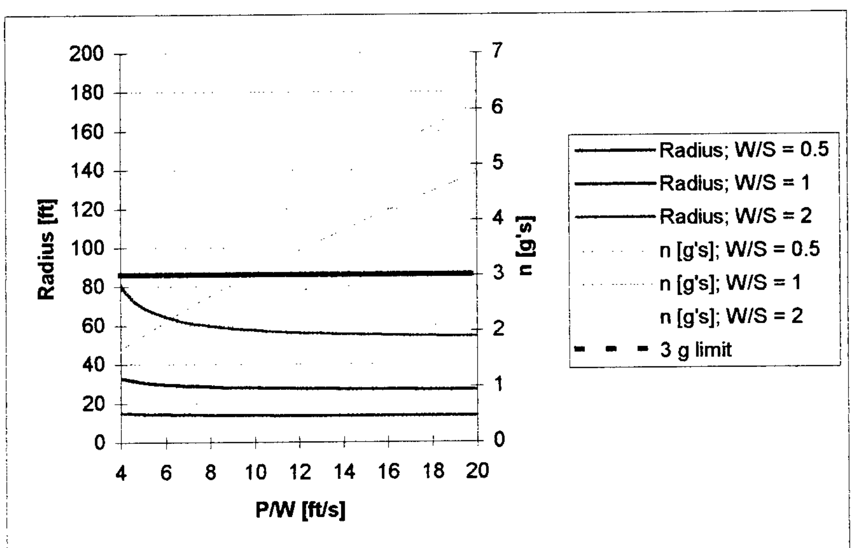


Figure 4.9 Turn Radius and acceleration vs. P/W

From figure 4.9, we concluded that for a given acceleration, the radius required to perform a turn increases with increasing W/S. Moreover, the P/W ratio required to perform the turn also increases with increasing W/S.

Putting it all together:
A logical conclusion could not be made by looking at the above curves individually. It was necessary to get a single picture of the influence of the W/S over the whole flight. This was done by determining the number of laps available for each of the W/S. From the data used to obtain the above curves, total amp-hrs required for 1 full lap (=(700 + πr) +(700+ πr)) was calculated. To determine the number of laps available, the total amp-hr available at the beginning of the cruise phase was divided by the amp-hr required for one full lap. Table 4.10 shows the data obtained. Figure 4.10 shows the trend of number of laps with W/S. From this data it can be seen that the maximum number of laps are obtained for a W/S of 0 (zero), which is not realistic. The lower limit was determined by the structural requirements since a low W/S, for same weight, requires a larger wing area.

W/S [lb/ft^2]	Cruise Speed [ft/s]	Minimum Turn radius [ft]	Length of each *loop* [ft]	Distance between turns [ft]	Length of each *lap* [ft]	Number of laps
0.5	25	14	86.96	1400	1586.96	14.00
0.85	31	24	148.03	1400	1648.03	13.24
1	34	28	173.79	1400	1673.79	12.79
2	48	55	347.90	1400	1847.9	10.11

Table 4.10. Number of laps available at each different W/S

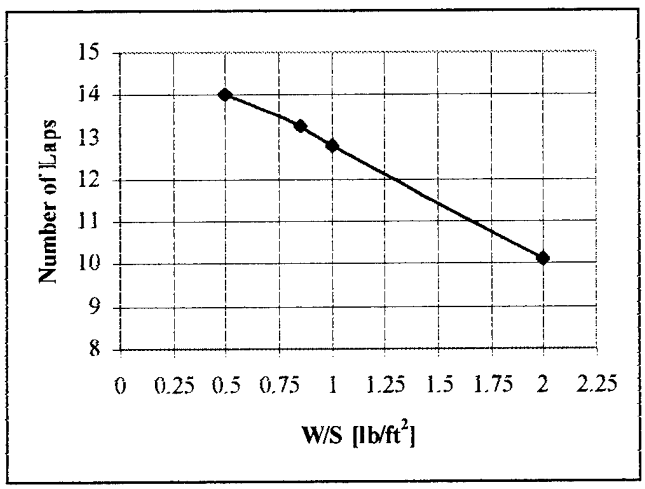


Table 4.11. Trend of Number of laps with W/S

Lower Limit of W/S:
To determine a realistic lower limit to the W/S, the span and chord required were plotted as a function of W/S with aspect ratio being a parameter. The curves in figure 4.12 are plotted for a wing with rectangular planform. If taper is introduced the span will be even bigger. The weight of the UAV was assumed to be

25 lb. It was decided by the construction group that each wing panel can be sectioned no more than once. This is a good decision since more than one section will require cutting the spar in more than one piece, which would create structural problems. Also, the weight of the UAV will increase due to the reinforcements required at the joints. Moreover, the standard longest piece of balsa, spruce or carbon composite spar available is 48 inches. So, if each panel were to have two sub-panels of 4 feet span, then each wing panel can be at the most 8 feet in span. Thus, the total span would be restricted to 16 ft. The above decisions were also strongly influenced by the transportation requirements. From figure 4.12, the lower limit of W/S for this UAV will approximately be 0.75 lb/ft^2 for an AR of 8, or W/S = 1 lb/ft^2 for an AR = 10. It was also noted that if the UAV weighs only 20 lb. instead, the W/S can be 0.75 lb/ft^2 at AR = 10. During the construction, every effort will be made to minimize structural weight. From the above analysis, the W/S was chosen to be 0.85 lb/ft^2 at an aspect ratio of 8.

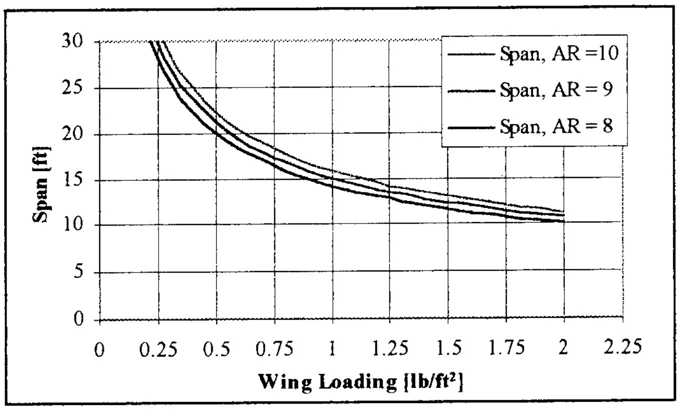


Figure 4.12 Span vs. W/S for different values of AR

The selection of W/S automatically sets most of the other parameters, which can be read or derived from the plots and equations used above. Table 4.13 summarizes the parameters selected.

Parameter	**Selected Value**	**Comment**
W/S	0.85 lb/ft^2	
AR	9	Needs further analysis
S	29.41 ft^2	Function of W/S and AR
Span	16.26 ft	Needs further analysis
Average Chord	1.8 ft	
Maximum coefficient of lift	1	
Stall speed	26.7 ft/s	
Take off speed Required	32 ft/s	
P/W for take off	14.3 [Watts/lb] or 6.3 [ft/s]	
Runway length required	150 ft	
Climb out angle required	3.8 deg.	
Cruise airspeed	32 ft/s	Equal to the take-off speed
Cruise coefficient of lift	0.73	

Table 4.13. Parameters selected based on a W/S = 0.85 lb/ft^2

Effect of Aspect Ratio on performance:
The aspect ratio plays a major role in the aircraft performance. Some of the major performance parameters influenced by the aspect ratio are the maximum lift to drag ratio, induced drag, wing lift curve slope, and the gliding ratio of the aircraft. The aspect ratio affects the size of the aircraft tremendously as discussed above. As noted in the conceptual design section, the maximum lift to drag ratio (E_{max}) should be as high as possible; this would also assure a high gliding ratio.

The gliding ratio, the horizontal distance (X) traveled divided by the altitude (H) lost, is identically equal to the lift to drag ratio (E). Maximizing E will maximize the gliding ratio. Figure 4.14 shows the effect of AR on maximum lift to drag ratio. With assumptions stated above, the maximum lift to drag ratio at an AR of 8 is 14.62. at and AR of 10 is 16.34; an increase of 12%. We decided to set the aspect ratio at 9.

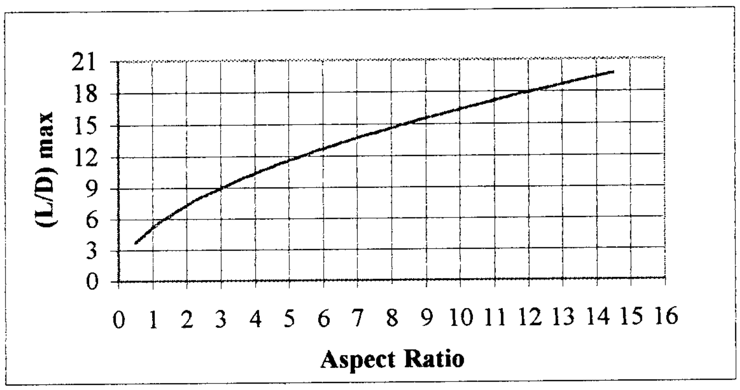


Figure 4.14. Effect of Aspect ratio on maximum lift to drag ratio

The remaining parameter of interest is the wing taper. Our research showed that a taper ratio of 0.4 will be most beneficial in terms of reducing the induced drag. This will also reduce some of the structural weight at the wing tip and lower the bending moment at the root section of the wing. But, introducing a taper this steep of a taper in the wing will increase the span. Since the aspect ratio selected is rather large, it was decided not to have any taper in the wing. Since there is no taper, the wing will be rectangular in planform.

Tailplane Design
The areas required for horizontal and vertical tail were approximated using the information available in the database created and from [ref 1]. The horizontal tail area was chosen to be 25% of the wing area and the vertical tail area was chosen to be 17% of the wing area. A further refinement of the area ratios will be considered in the detailed design phase. Based on the current W/S and weight, table 4.15 summarizes the values selected for the tailplane.

Component	Horizontal Tail	Vertical Tail	Comment
Area	8.25 ft^2	5.61 ft^2	Needs further analysis
Chord	1 ft	1 ft (?)	Needs further analysis
Span	8.25 ft	5.61 ft	Based on rectangular planform

Table 4.15. Preliminary parameters selected for the tailplane

Propulsion system selection:

Our preliminary calculations showed that the take-off might need about 25 Amperes of current at 20 Volts. From the research done during the conceptual design stage, the 2000 MAH cells were considered best for the pack because of the Watt-Hr capacity provided by these cells. We had to change this decision here because of their poor discharge characteristics. We selected our second choice, 2500 MAH cells, for the battery pack. However, we will test the 2000 MAH cells and 2500 MAH cells to verify our decision. If the 2000 MAH cells perform superior or equivalent to the 2500 MAH cells, we will use 2000 MAH for the competition. With the 2500 MAH cells, the battery pack included 15 cells. The total voltage available would be 18.75 Volts and the total energy available would be 46.88 Watt-Hr.

Since there was enough information available at this point about the take-off thrust requirements, a brushless motor rated at 600 Watts was purchased. Table 4.16 lists the motor characteristics. The motor was tested rigorously to determine the most efficient propeller.

Name	Aveox 1406/4Y
Max Power	600 Watts
Speed Constant (Kv)	1500
Torque Constant (Kt) [In-Oz/Amp]	0.901
Winding Resistance [Ohms]	0.060
No load Current [Amps]	1.2
Max Efficiency	89%
Dimensions	Weight = 6.9 oz. Length = 1.76" , Diameter = 1.5"

Table 4.16. Motor Characteristics

Our analysis showed that a thrust of 4.6 lb. was required to take-off. We assumed that a propeller that can deliver 25% more thrust than the required thrust should be adequate. Test results are shown in table 4.17. From this data, we selected the 15×10 and 16×10 propeller for a duration test. The final selection was based on the results of the duration test. We simulated a dynamic condition by using three table fans and constructing a nozzle out of cardboard. We were able to get airspeeds up to 20 ft/s from this setup. The exit of the nozzle was placed directly in front of the propeller during the test. Both of the propellers performed equally well under this condition. The 15×10 was chosen because the motor was able to give 1.6 lb. of thrust for 19 minutes with this propeller installed. The 16×10 lasted for 18 minutes. The table 4.7 shows the data recorded during the tests. An APC propeller was selected for the final configuration. We will test propellers from other manufacturers before making the final decision on the propeller.

Propeller	Current Drained	Throttle position	Thrust	Condition
13 × 8	25 amps	100 %	4 lb	Static
14 × 10	29 amps	100 %	4.2 lb.	Static
15 × 10	30 amps	100 %	6.6 lb.	Static
15 × 10	7 amps	40%	1.6 lb.	Dynamic
16 × 10	35 + amps	100%	8.8 lb.	Static

16 × 10	7 amps	35%	1.6 lb	Dynamic

Figure 4.17. Results of tests performed on the motor with different propellers.

Material Selection:

Having selected the preliminary parameters and geometry, the structures group determined that the Balsa wood will be the material of choice for constructing the airfoils. It was decided that the tail booms will be made up carbon composite material due to its strength to weight ratio. The diameter and length required were not yet determined. The main spar will also be made up of carbon fiber and the secondary spar will be made of spruce wood. Carbon fiber was selected for the main spar because of its strength to weight ratio compared to spruce, or basswood.

Preliminary Drag Analysis:

In order to predict the performance of the UAV, preliminary drag calculations were performed. Using the DATCOM method, we estimated the zero-lift drag to be 0.0247. The actual zero-lift drag coefficient will be slightly higher because the drag of the booms is not calculated in this table.

Component	Reference Length	Wetted Surface Area	Reynolds Number	K	C_f	KC_fS	C_{D0}
Wing	2	55.75	3.823×10^5	1.2	0.0054	0.3603	0.0132
Horizontal Tail	1.36	13.94	2.600×10^5	1.22	0.0058	0.0991	0.0036
Vertical Tail	1.19	9.84	2.275×10^5	1.195	0.006	0.0704	0.0026
Landing Gear							0.004
Interference 5%							0.0011
Sum							**0.0247**

Table 4.18. Data used for preliminary drag calculation

Summary:

A preliminary set of parameters has been selected at this point. Based on the tests performed, a motor and a propeller are identified. These parameters will be optimized in the detailed design stage. Some of the parameters selected in this stage will be changed to obtain better performance. Table 4.19 summarizes all of the parameters selected.

Parameter	Selected Value		Parameter	Selected Value
Weight	23 lb		**Aircraft:**	
W/S	0.85 lb/ft^2		Maximum coefficient of lift	1
Wing :			Zero-lift drag coefficient	0.0247
AR	9		Stall speed	26.7 ft/s
Area	29.41 ft^2		Take off speed Required	32 ft/s
Span	16.26 ft		P/W for take off	14.3 [Watts/lb]
Average Chord	1.8 ft		Runway length required	150 ft
Stabilizer:			Climb out angle required	3.8 deg.

Parameter	Selected Value		Parameter	Selected Value
Area	8.25 ft^2		Cruise airspeed	32 ft/s
Span	8.25 ft		Cruise coefficient of lift	0.73
Chord	1 ft		**Propulsion:**	
Fin:			Motor	Brushless/600 Watts
Area	5.61 ft^2		Propeller	15×10
Span	5.61 ft			
Chord	1 ft			

Table 4.19. Preliminary parameters

DETAILED DESIGN

The objective for detailed design phase was to optimize the parameters selected in the preliminary design stage. In addition, select an airfoil, calculate the stability parameters and predict the performance of the UAV. We started the analysis by refining the weight assumed earlier.

Detailed/Refined Weight Estimate:

A refined weight calculation was performed to estimate the final weight of the UAV, since most of the component weights were known from the preliminary analysis. This was also necessary to select the final parameters and estimate the performance of the UAV. The densities of materials to be used were either measured or obtained from manufacturers.

Component	# used	Length [inch]	Height [inch]	Thickness [inch]	Density of material [lb/ft^3]	WEIGHT [lb.]	Comment
Wing							
Ribs	52	20.4	2.05	1/8	7.40	1.123	Balsa, one every 4 inches
Spars (1)	2	192	0.5	1/8	91.24	1.5	Carbon Composite spar
Spars (2)	2	192	0.25	¼	24.00	0.17	Spruce
Covering	1	1	8030	1/16	8.00	2.323	Balsa
					Wing Weight:	**5.49**	
Horizontal Tail							
Ribs	15	16.2	1.62	1/8	5.50	0.156	Balsa, one every 4 inches
Spar1	2	60	0.5	¼	24.00	0.208	Spruce
Covering	0.5	1	2016	1/16	8.75	0.319	
					Horizontal Tail Weight	**0.73**	
Fin							
Ribs	12	18	1.8	1/8	5.90	0.17	Balsa, one every 4 inches
Spar	2	48	0.25	¼	24	0.035	Spruce
Covering	0.5	1	1440	1/16	8.75	0.23	
					Vertical Tail Weight	**0.46**	
Tail booms	2	70	1	1		1.1	
Landing Gear						1	
Motor + Controller						1.325	
Servos						0.75	
Payload						10	Includes batteries
Glue						0.75	
Misc.						0.5	
Total						**22.15 lb**	Includes [motor, payload, landing gear, wing, stabilizer and Fin]

Table 5.1. Detailed Weight Analysis

From table 5.1, the estimated UAV weight is about 22 lb. We set the weight of the UAV at 23 lb. from this point on. Due to this change in weight, most of the parameters selected in the preliminary stage were changed. Most importantly, the aspect ratio was increased to 9.5 and the span was decreased to its final value of 16 ft. Table 5.2 shows the changes in these parameters.

Parameter	Selected Value	Comment
W/S	0.85 lb/ft^2	
AR	9.5	
S	27 ft^2	
Span	16 ft	
Average Chord	1.7 ft	Needs further analysis
P/W	8.8 Watts/lb.	Available = 18.34 Watts/lb.
Stall Airspeed	26.7 ft/s	
Takeoff speed required	32 ft/s	
Runway length required	150 ft	
Climb out angle required	3.8^0	
Cruise Speed	32 ft/s	
Coefficient of lift required in cruise	0.649	

Table 5.2. Interim Performance Estimate

Airfoil Selection:

Having determined the wing geometry, we selected an airfoil next. From table 5.2 above, the cruise speed is selected to be 32 ft/s. The flow Reynolds number over the wing at this speed will be 346,636 based on the average chord selected. Research showed that most of the low Reynolds number airfoils behave very well above a Reynolds number of 300,000. In fact, at Reynolds numbers of around 100,000 the profile drag on most of these airfoils increases tremendously. At take off and landing, there is a potential for the UAV to reach speeds that might have a Reynolds number of around 100,000. To avoid the unnecessary increase in drag and high current drain, an average chord length was chosen which would keep the Reynolds number above 250,000 at all conditions. Figure 5.3 shows the variation of Reynolds number with chord at different velocities. The blue box indicates the design region necessary to keep the Reynolds number above 250,000. The average chord required was determined to be 1.7 ft

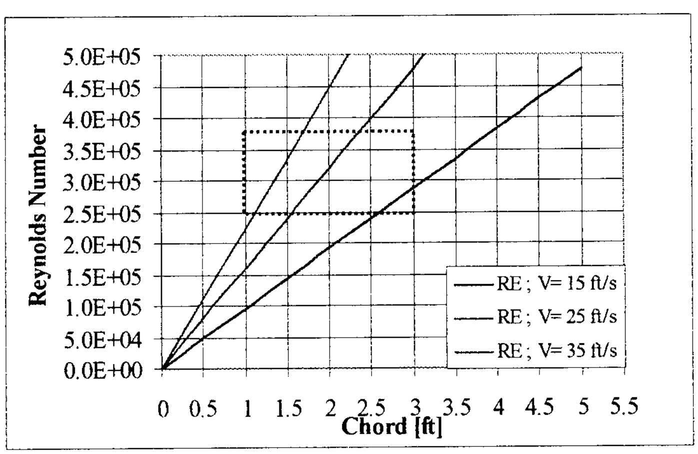


Figure 5.3. Variation of Reynolds with chord and Velocity

Our research selected two airfoils, the SD7032 and the K3111, from the *Summary of Low Speed Airfoil Data* by Michael Selig. The figure of merits used in selecting these airfoils were the airfoil lift to drag ratio, maximum coefficient of lift, stall characteristics and stall angle of attack. Our analysis earlier required an airfoil with minimum profile drag, a high maximum coefficient of lift and good drag characteristics above a Reynolds number of 250,000. Based on lift and drag characteristics, the SD7032 was selected, at this stage, to be the wing airfoil for this UAV. This decision might change later if the K3111 performs superior to SD7032. The SD7032 has a airfoil maximum coefficient of lift ($c_{\ell\, max}$) of 1.293, using the approximation $C_{L\ MAX} = 0.9\ c_{\ell\, max}$, the available $C_{L\ MAX}$ for the three dimensional wing is 1.16. The characteristics of the airfoils are shown in the table 5.4 for a Reynolds number of 300,000. The selection of SD7032 was based on all of the quantities listed. Since these two airfoils are so close in characteristics, test models will be built using both of the airfoils to decide on the best one for our application.

Airfoil	Maximum coefficient of lift	Stall Angle	Maximum airfoil lift to drag ratio	Thickness Ratio	Coefficient of drag at cruise	$\frac{\partial C_\ell}{\partial \alpha}$ [per rad]
SD7032	***1.293***	***10^0***	***66.67***	***9.96%***	***0.009***	***5.19***
K3111	1.1	10^0	60	11.3%	0.011	

Table 5.4. Characteristics of SD7032 and K3111 airfoils

The figure 5.5a and 5.5b show the coefficient of lift versus angle of attack and drag versus coefficient of lift for the SD7032 airfoil.

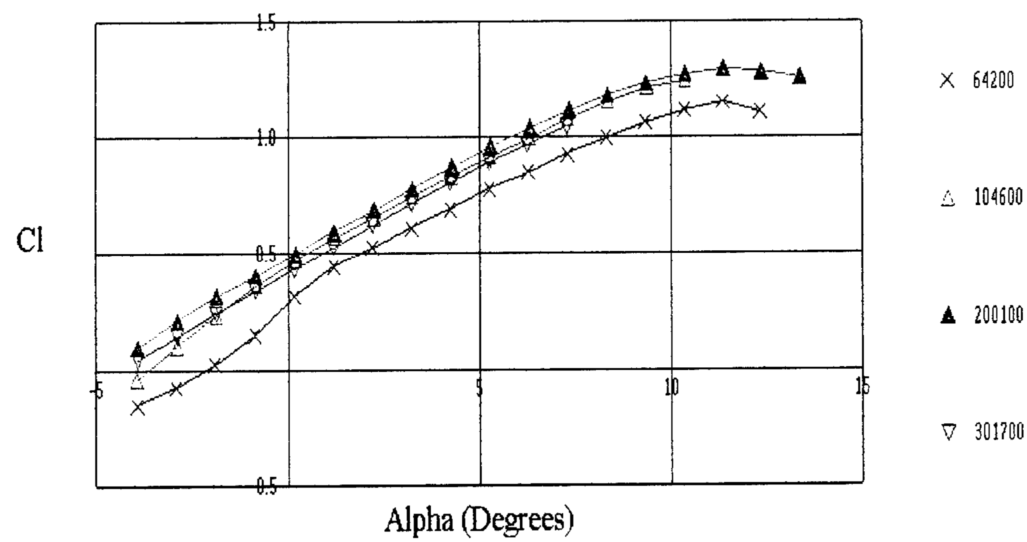


Figure 5.5a. Coefficient of lift Vs. Angle of Attack for SD7032

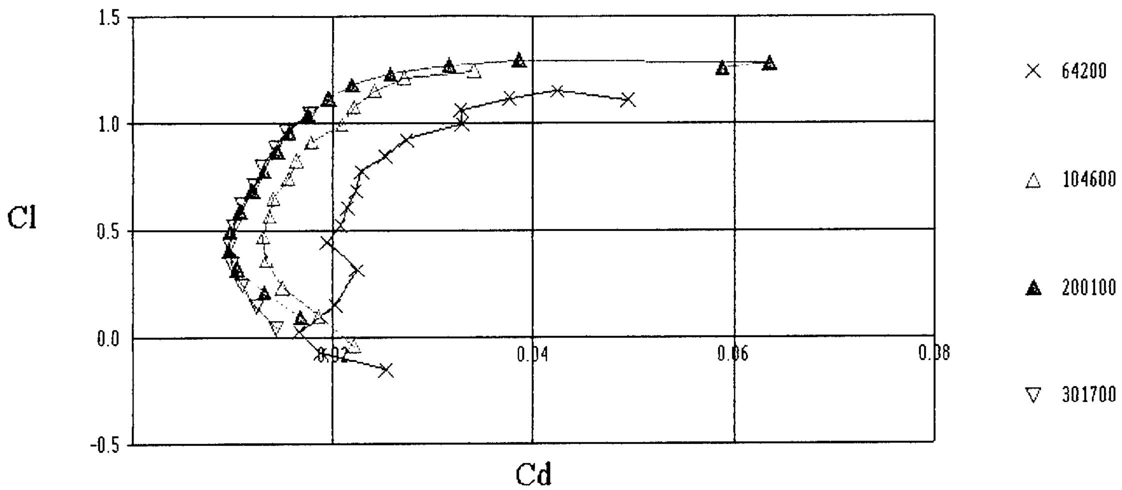


Figure 5.5a. Coefficient of lift Vs. Angle of Attack for SD7032

Wing Geometry Refinement:
The configuration chosen in the conceptual design requires that the payload and batteries be stored in the root section of the wing. Research showed that the volume required for a 7.5 lb. steel (density = 492 lb/ft^3) payload is approximately 26.33 in^3. This volume could be obtained by splitting the payload in to two parts each having the dimensions of 3.5"×2.5"×1.5". This would require a thickness of at least 2 inches at the root. Also, the batteries selected have the following dimensions: diameter × length = 1.02"×1.97". As determined earlier the battery pack has 15 cells, and if the pack were split in to two, each pack would have 7 cells. If the batteries were installed with their longer dimension on the floor and in two layers, the minimum thickness required would be 2.04 inch at the root section. Since the airfoil does not have the same thickness through the length of the chord, the airfoil at the root needed to be made at least 3 " thick. This would allow a thickness of more than 2 inches around the 30% chord point. To achieve this, the root chord had to be changed to 2' and the airfoil thickness ratio to 13 % at the root section only. Since this thickness is not required anywhere along the wing, a taper had to be introduced to reduce the chord length along the span. Since an average chord of 1.7 ft required for the Reynolds number requirement, the tip was chosen to be 1.4 ft. The refined wing geometry is presented in table 5.6.

Parameter	Value
S	27.2 ft^2
Span	16 ft
AR	9.5
Airfoil	SD7032 (9.97%), Root (± 6"): 13 %
Root Chord (C_r)	2 ft
Tip Chord (C_t)	1.4 ft
Taper ratio	0.7

Table 5.6 Modified Wing Geometry to accommodate the payload and battery packs

Tailplane Refinement:

We selected a symmetrical airfoil for the tailplane. The NACA0009 was chosen to be airfoil because of its drag characteristics around a Reynolds number of 300,000. Table 5.6 shows the characteristics of this airfoil at a Reynolds number of 300,000.

Airfoil	Maximum coefficient of lift	Stall Angle	Maximum lift to drag ratio	Thickness Ratio	Drag at Cruise C_L	$\frac{\partial C_\ell}{\partial \alpha}$ [per rad]
NACA0009	0.753	8^0	110.7	10%	0.02	5.79

Table 5.6. Characteristics of NACA0009 airfoil

The values selected for the horizontal tail and the vertical tail area during the preliminary design were changed due to changes in the wing geometry. In addition, the chord lengths and the planform of the tail plane were modified as discussed below.

It was decided that the horizontal tail would have a rectangular planform. This was necessitated by the configuration selected. Since there will be two booms coming from the wing, a swept back panel would require longer (heavier) booms. Moreover, the longest carbon composite booms available are 70 inches (5.8 feet) in length.

We decided to use a twin vertical tail configuration. The tips of the stabilizer are reinforced to attach the booms. Thus, tips are the strongest point of the stabilizer. To avoid further reinforcement at the center section to attach the vertical tail, we decided to make use of the already reinforced tips. In addition, with chord of 1 ft, a single vertical tail would require a span of 4.8 ft. This will cause problems in transporting the UAV. To avoid all these problems, the vertical tail was split into two panel each originating from the stabilizer tip. Each panel would a span of 2 ft and a sweepback angle of 25^0. The sweepback was introduced due to moment arm requirements. With these modifications, the root chord would be 1.36 feet and the tip chord would be 1 ft. Table 5.7 shows modified data for the tailplane.

Parameter	**Horizontal Tail**	**Vertical Tail**
Area	6.8 ft^2	4.81 ft^2
Span	5 ft	2 ft (each panel)
Root Chord	1.36 ft	1.36 ft
Tip Chord	1.36 ft	1 ft
Sweepback angle	0^0	25^0

Table 5.7. Modified Geometry for the Tail plane

Center of Gravity (C.G):

The center of gravity of the UAV with the payload aboard was determined to be located at 1 feet from the wing apex. Since we are required to fly the UAV without the payload, we decided to put the payload right on the center of gravity. Thus, removing the payload will no effect on the C.G. position. However, the C.G position required a design change since the moment created by the tail plane around the C.G. could not be balanced with the rest of the weight available. We had to construct a capsule to locate the motor 6 inches in front of the wing apex. This also helped reduce the noise created by propeller and increase the propeller efficiency slightly. Table 5.8 shows the data generated during center of gravity calculations. The static margin was determined to be approximately 12 %

Center of Gravity	1 ft from wing apex
Static Margin	12%
Aircraft Neutral point	1.2 ft from wing apex
Wing mean aerodynamic Chord ($\overline{C}$)	1.72 ft
Spanwise location of $\overline{C}$	3.76 ft from the wing centerline
Distance from wing apex	0.141 ft
Distance from wing apex to horizontal tail apex	5.4 ft
Distance from wing apex to vertical tail apex	5.6 ft
Horizontal Tail Volume coefficient (V_H)	0.708
Vertical Tail Volume coefficient (V_V)	0.060

Table 5.8. Data generated during the stability calculations

Washout angle:

We decided to introduce a tip washout of 2^0. There were two reasons to make this decision. First, the airfoil being used stalls around 10^0, most of the low Reynolds number airfoil we investigated have similar stall characteristics. Second, since there is no feedback from the UAV to the ground, there is no way of knowing the angle of attack and airspeed at all times. When the UAV is coming straight at the pilot, he might not realize if the UAV is close to stall or already stalling. By introducing a tip washout, even if the UAV stalls at the root, there can be adequate roll control to recover from any stall induced rolling.

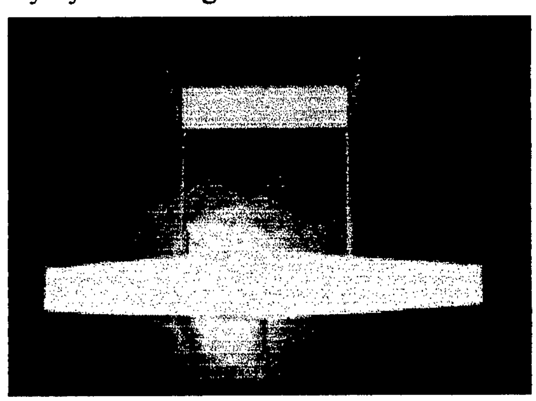
Landing Gear:

Our design requires two main gears on the wing and two small wheels, one at each of the vertical tails. Since the propeller being used is 15" in diameter, the main gear had to be 12" in length to provide adequate clearance for the propeller. The main gear was located directly at the point where the booms were connected since this is a reinforced section of the wing. The booms are attached 2.75 ft in from the wing centerline in the spanwise direction. The landing gear strut was attached so that line through the C.G. will make a 20^0 angle with the vertical measured from the main wheel location. This was done to protect the propeller during landing. Also, to prevent the aircraft from overturning, the main gear was attached so that gear strut will make 25^0 angle measure from a vertical through the C.G. The main wheel diameter required was estimated to be 3".

Dihedral:
We introduced a dihedral of 5^0 on the wing for couple of reasons. The UAV is required to make turns on every lap. As we found earlier, the smaller the turn radius the higher the number of laps available. There is potential for the UAV to experience some sideslip during the turns. The altitude lost has to be regained to continue with the flight, which requires more energy. Our research showed that, by introducing a dihedral, there will be some sideslip protection. The second reason was that the landing gear is attached 2.5 ft away from the center section. If there were no dihedral, at take-off and landing any disturbance might cause the wing tips to hit the runway. This can be avoided by either making the landing gear longer or by raising the tips using a dihedral. With a dihedral angle of 5^0, the tips are raised by 8.4 inches relative to the root.

Handling qualities:
To test the stability and handling qualities of the design, we constructed a scale model (1ft = 1 inch) of the final parameters out of balsa. Because of the scale used, the wing span was 16", and the root chord was 2". The model was tested for static by hand launching the model at different angles of attack. The model flew on a relatively straight line for an average distance of 25 ft when launched from 5 ft above the ground. The estimated speed was 8.3 ft/s. These tests showed that the C.G. is in the correct position and the full-scale model will be stable.

The same model was tested for dynamic stability by launching the model in a sudden but mild crosswind. Although the model went off the course, after a certain distance it stabilized and continued flying. To test the model for roll, pitch and yaw capabilities, control surfaces were put on the model. The model was launched at different angles of deflection of these control surfaces to analyze turn and pitch characteristics. Based on the results adjustments were made to the actual sizes of elevators and rudders. Since this scale model exhibited static and dynamic stability, we decided to build a full-scale model to be tested.



Model Used for testing

Final Parameters:
The following couple of tables give a detailed outline of the final parameters and performance predictions. We are now constructing a full-size model according to these parameters and test fly it to identify any parameters that need to be changed.

UAV geometry parameters:

Parameter		Value
Take-off Weight (W_{TO})		23 lb.
$W_{Structure}/W_{TO}$		0.45
$W_{Payload}/W_{TO}$	(Payload fraction)	0.33
$W_{Propulsion}/W_{TO}$		0.22
W/S		0.85 lb/ft^2
Wing:	Planform area	27.2 ft^2
	Root Chord	2 ft
	Tip Chord	1.4 ft
Taper ratio		0.7
Mean Aerodynamic Chord		1.72 ft
Aspect Ratio		9.4
Dihedral		5^0
Angle of Incidence		4^0
Tip Washout angle		2^0
Aileron:	Area	2.2 ft^2
	Span	3.7 ft
	Chord/Wing Chord	0.18
Horizontal Tail:	Area	6.8 ft^2
	Span	5 ft
	Chord	1.36 ft
Elevator:	Area	
	Span	
	Chord	
Vertical Tail:	Area	4.81 ft^2 (Total)
	Span	2 ft (each)
	Root Chord	1.36 ft
	Tip Chord	1 ft
	Sweepback angle	25^0
Rudder:	Area	
	Span	
	Rudder Chord/Fin Chord	
Overall Length		7.67 ft

Parameter	Value
Overall Height	2.83 ft
Main Landing Gear Length	10"
Main Gear Wheel Diameter	3.5"
Tail Gear Wheel Diameter	

Performance Parameters:

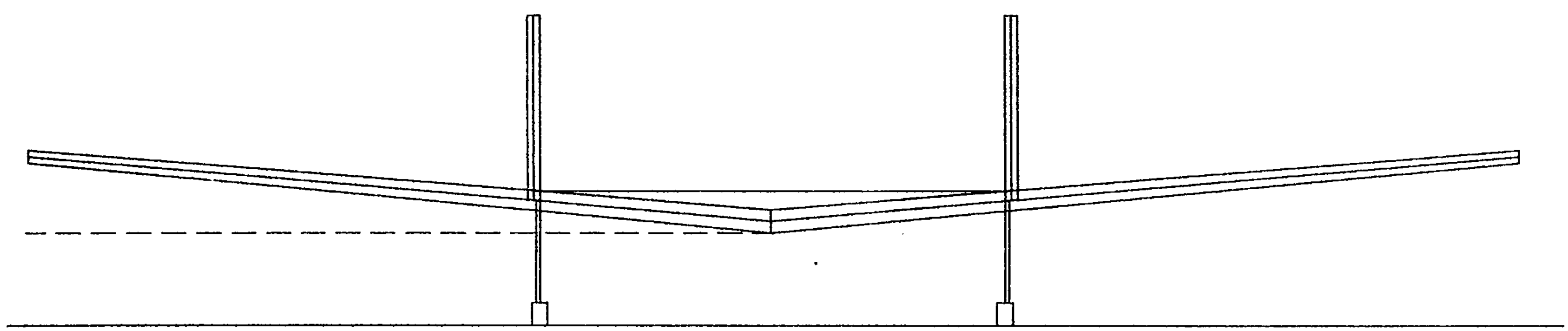
Propulsion System:	Batteries	2500 MAH/1.25 Volts
	Number of Cells	15
	Total Voltage(at take-off)	18.75 Volts
	Total Watt –Hours	46.87 Watt-Hr
	Motor type/Power	Brushless/600 Watts Max
	Max Efficiency	88%
	Propeller Diameter	15"
	Pitch	10

Take-off:	Runway Length	150 ft
	Velocity	32 ft/s
	Coefficient of lift	0.694
	Time to lift off	4.67 sec
	P/W	6.5 ft/s or 8.8 Watts/lb.
	Power	248.2 ft-lb/s or 336.68 Watts
	Current Required	20.0 Amps
Climb:	Angle Required	3.8^0
	Rate of Climb	5.68 ft/s
	Time to reach 200 ft	35.2 sec
Cruise:	Speed	32 ft/s
	Coefficient of lift	0.694
	Thrust required	1.46 lb.
	Power	46.45 ft-lb/s or 63 Watts
	Current Required	7 Amps

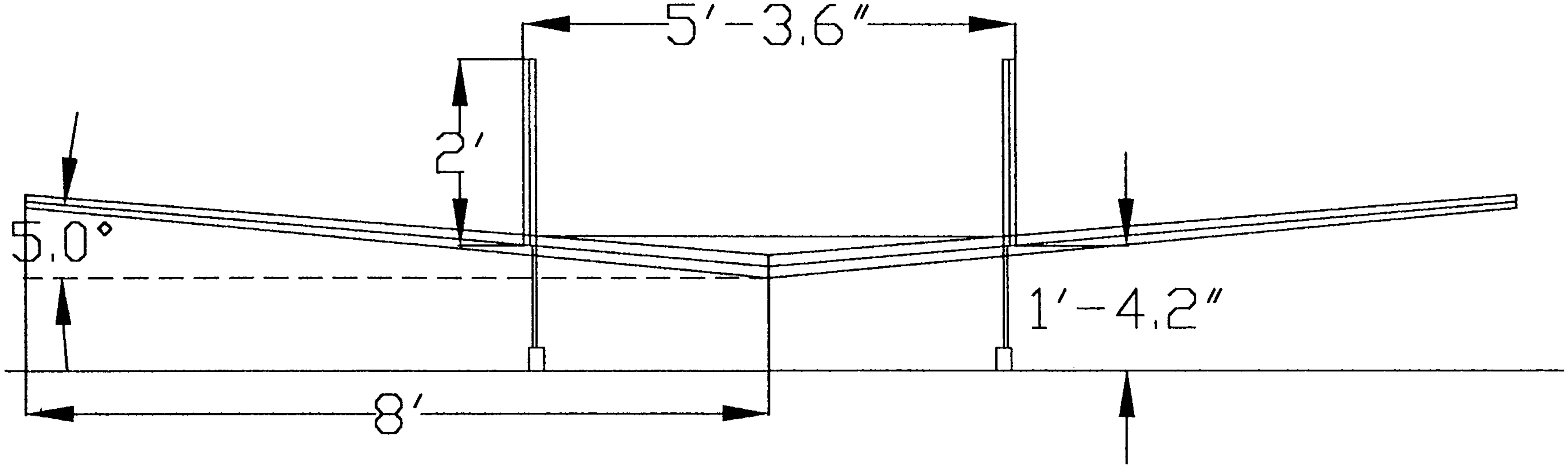
Performance Predictions:

Estimated Linear Range	36,220.8 ft (6.86 miles)
Time of flight	17.8 minutes
Length of a lap	1548 ft.

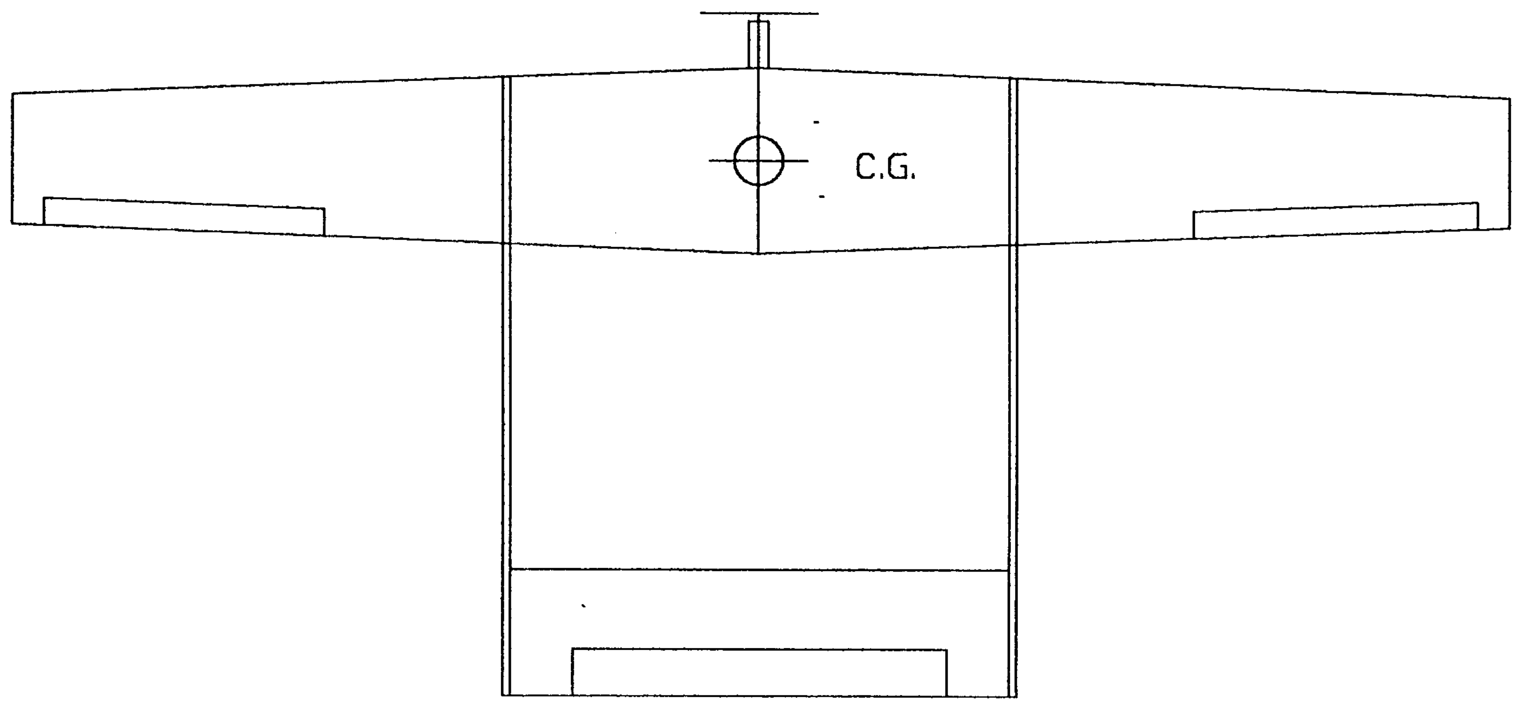
Number of laps	*19±1 (calculated assuming zero wind condition)*



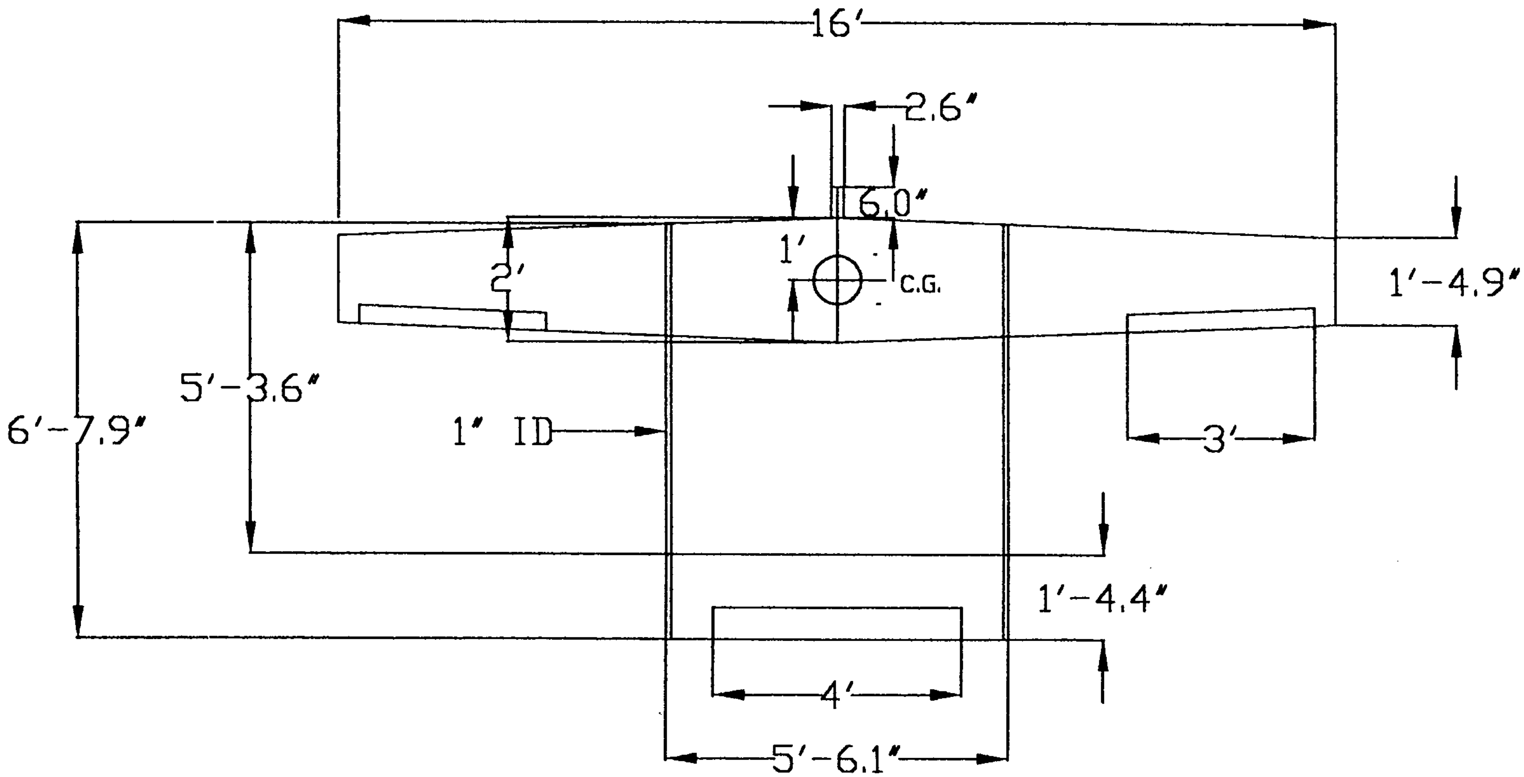
Front View



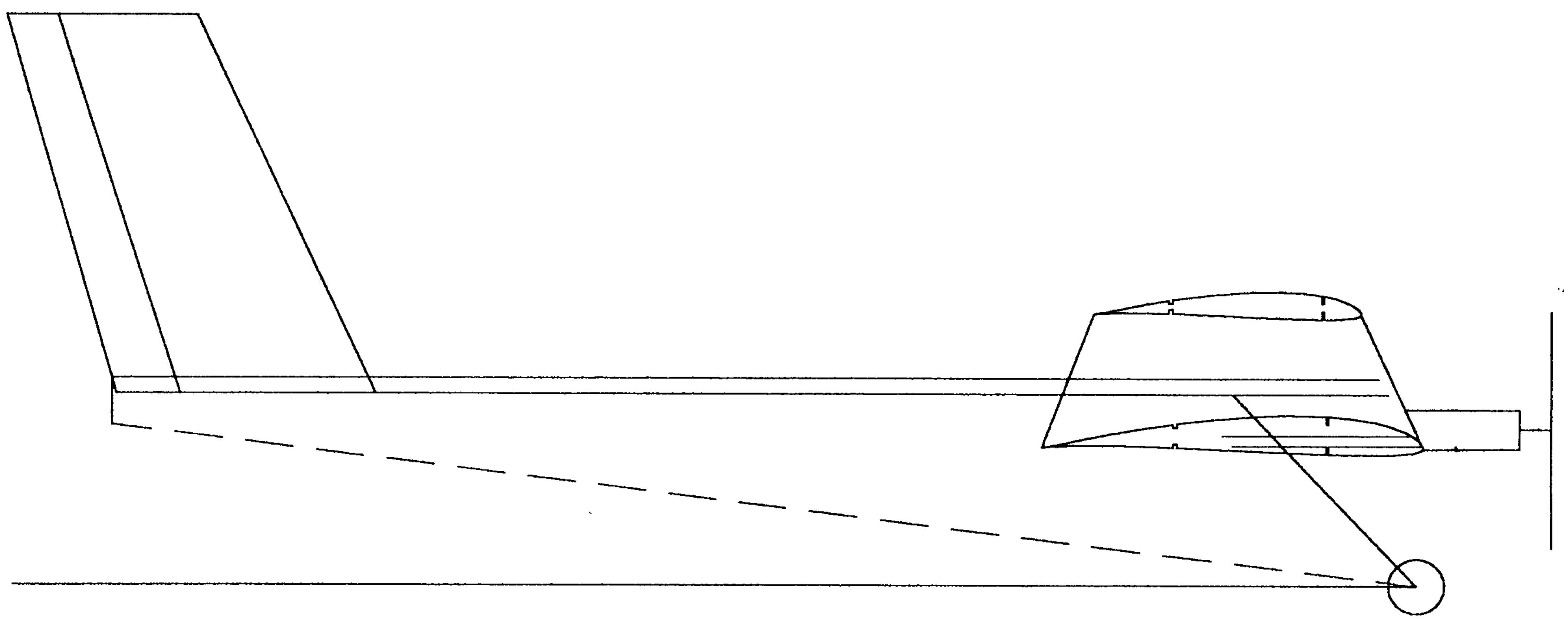
Front View with dimensions



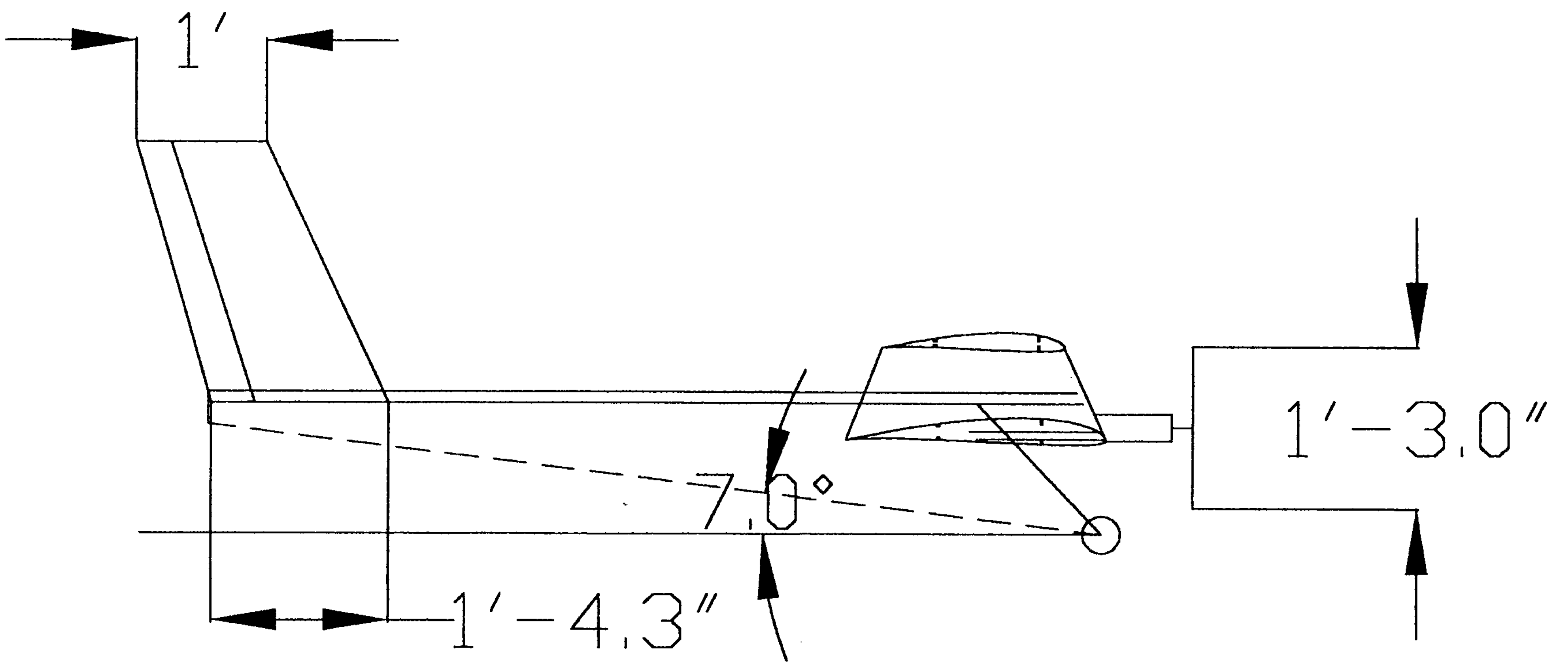
Top View



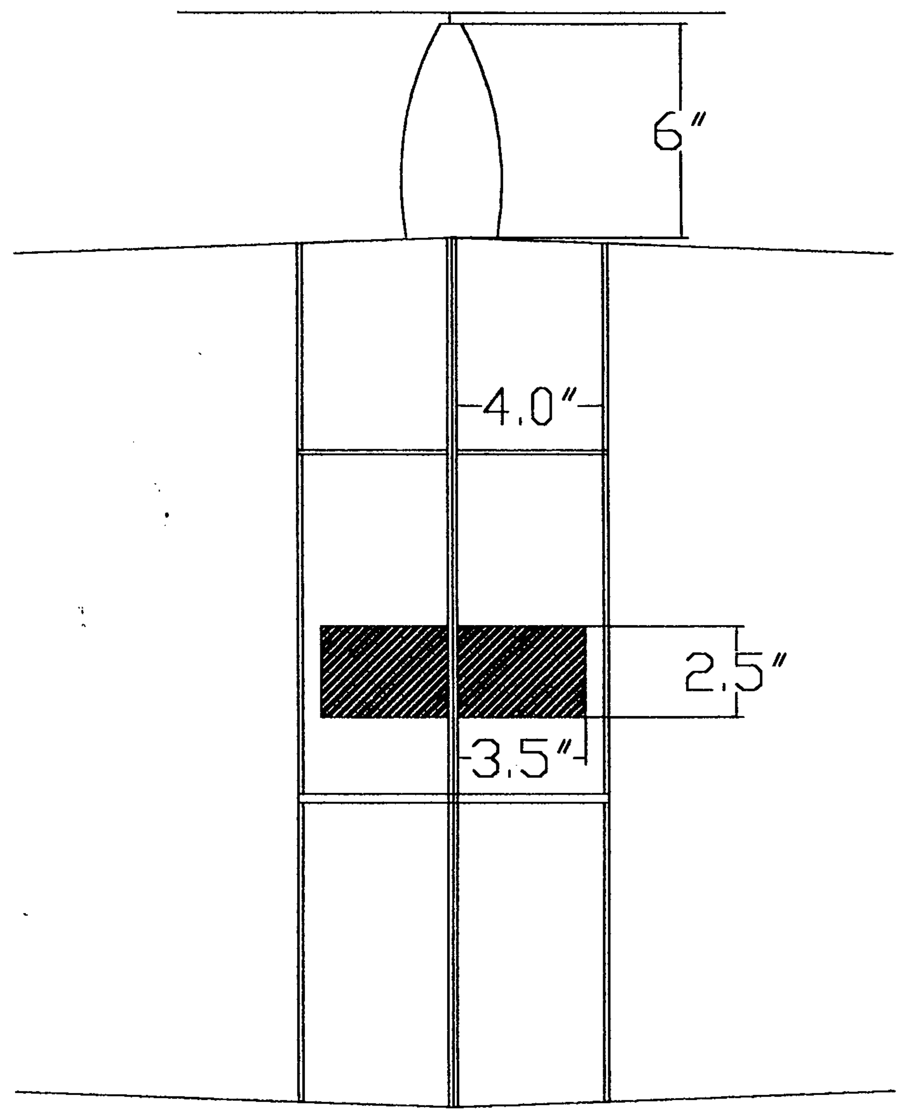
Top View with dimensions



Side View



Side View with dimensions



Center Section with dimensions

MANUFACTURING PLAN

We set a goal to make the manufacturing process as simple and standard as possible. During the development of the plan, emphasis was put on utilizing any item or equipment available commercially. If, for example, an item available commercially did not meet our needs exactly, we tried to modify our plan to make use of that item. A new and unique process might require new tools, extra cost and complicated instructions. The selection of a process depended strictly on skill, equipment and financial support available.

Since our configuration consisted only of a wing and a tail connected by two booms, the only components that required major construction were the airfoils and the wing jig. Due to the size of the UAV, we had to invent ways of connection and disconnecting various parts of UAV. There was no jig available commercially that could be used to build a 8 ft wing panel. But, we did find a ready-made jig to construct the tail plane. The processes selected to manufacture the airfoils and the jig was based on the figure of merits described in table 6.1.

Figure of Merit	Comment
Skill Available	8 out of the 10 people in the team had never constructed any model airplanes before
Equipment availability	Most of the equipment had to be shipped from either California or Texas, which required 5- 7 business days to arrive
Cost	We had a limited budget
Time	All of the team members were full time students with part time jobs
Personnel availability	All of the team members were commuters and were not available during weekends and in the evening.

Table 6.1. Figure of Merits used in selecting manufacturing processes

A second set of figure of merits used was related to the performance and accuracy of the construction. Table 6.2 lists these FOMs.

Figure of Merit	Comment
Weight	Any method of construction that would add more weight to the UAV is not desirable – Performance requirement
Accuracy	Accuracy in construction was considered very important since, for example, most of the low Reynolds number airfoils are extremely sensitive to their profile. – Performance requirement
Structural Strength	Any method that would either over load one part of the structure of create weak points in the structure is not desirable – Performance requirement

Table 6.2. Figure of Merits based on performance

Processes investigated to manufacture the airfoils:

1. *The Sandwich method*
   This process is very cost effective and requires little skill, and less time. All of the airfoils required for one panel can be manufactured at once. We used a software package called *The Compufoil Professional* to generate and print the root and tip airfoils for one of the wing panels. Aircraft grade plywood (1/16") was used to make templates for these two airfoils. The spar slots were also included in the templates. Balsa sheets equivalent in number to the airfoils required for a panel were sandwiched between the templates and secured tightly using nuts and bolts. Using the root and tip templates as the guides, the resulting block of balsa was sanded carefully until all of the airfoils were formed. This process produces uniform airfoils and the accuracy depends on the accuracy of the templates. Thus, Extreme caution was taken to manufacture the templates with high accuracy. If the template deviated from the printout by more than 0.25%, a new template was made. An error of 0.25% will change the chord length by 1/16 inch. This accuracy was considered necessary since the airfoil performance is extremely sensitive to the shape. Figure 6.3 shows one of the setups used.

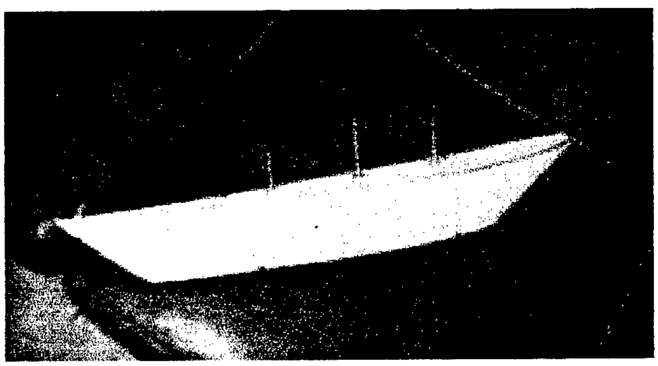


Figure 6.3. Sandwiched airfoils

2. *Trace each airfoil on Balsa and manually cut the airfoils.*
   This processes was not selected because of the time and skill required. Moreover, every single airfoil will be different in accuracy depending on the person who made it. Since time was a major concern for our team, this method was not selected.

3. *Manufacture the airfoils at a professional wood shop.*
   This process was not selected because of the cost involved. Since our model required more than 75 airfoils, the cost of having these manufactured professionally was more than we could support.

Manufacturing the Jig:
A manufacturing process was also necessary to select and construct a jig for the wing. Due to the size of the panels and chord, most of the conventional jigs were not suitable. The different types of jigs investigated are described below:

1. We chose to manufacture a jig that supported the under camber of every airfoil. Each airfoil would have a balsa sheet shaped exactly like the under camber of that particular airfoil. This jig is also ideal to introduce the tip washout on the wing. By progressively increasing the leading edge height of the

jig from the root to the tip, we were able to induce a 2^0 twist on the jig. Introducing this angle on the wing was just a matter of laying each airfoil on the jig and gluing the spar. This jig also provided enough support for each of the airfoils when the skin was being applied on the top and bottom surfaces. We realized that the jig should be perpendicular to the surface at all points. Provided that the surface being used is level, this type of jig meets the above requirement. The process selected to manufacture the jig was similar to the one used to make airfoils. The templates of the root and tip airfoil under surfaces were made out of plywood and the sandwich method was used. Figure 6.5 shows the assembled jig.

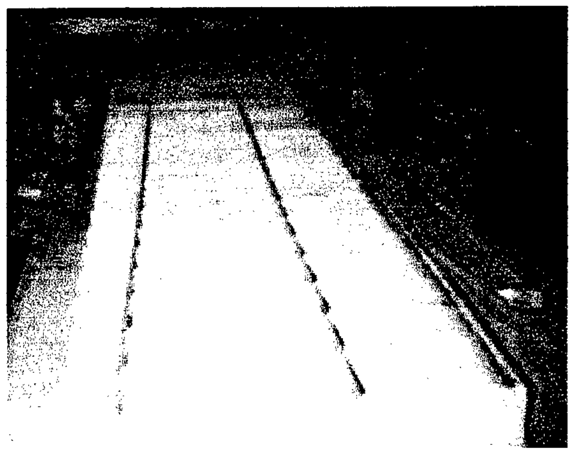


Figure 6.5. Assembled Jig

2. A second method investigated was to drill holes in to the airfoils and insert two long steel rods through the wholes. This method will suspend the airfoils in air supported by the rods. We did not select this method because, we realized that a 8 feet long steel rod will bend significantly if any pressure is applied. A deflection in the jig will be reflected on the panel being built. Since there is no way ensuring that the rods will be perfectly straight at all positions, this method was abandoned.

3. We also investigated a third method where the airfoils were supported at the leading and trailing edges only. This method was not suitable here because of the chord lengths involved. We did some experiments and realized that the spar locations and mid chord point needs support to keep the airfoils straight and perfectly vertical. This method was also not selected.

Wing Structure:

The wing had to be constructed in four sections. The room available for construction was only 14' × 13'. Since the total wing span is 16', there had to be at least two panels. For reasons of transportation, we decided to section each of the wing panels in to two. Thus, the wing would be split into four panels. Since the payload and the battery pack are stored in the center section, this part of the wing has to be the strongest. To avoid any structural problems during flight, we decided to permanently attach the two center panels. This was also necessary to introduce the dihedral angle. If the center panels were detachable, the stresses at connections will be enormous. Our analysis showed that the center section will have to support around 190 lb. if the UAV were to experience a 5 g acceleration. Thus, the wing now has two outer panels, each 4 feet in length and a center panel 8 feet in length. The center panel has the dihedral angle built in.

Connecting the wing panels:
To connect the outer panels to the center panel, we utilized a wing connector marketed by Byron Originals. This adapter is a aluminum spar with the following dimensions: 9"×1"× 1/8". The spar is permanently attached to the center section and protrudes out of the last airfoil. The outer panel has a female part, which is permanently attached in between the spars. To connect the panels, the spar is simply inserted in the outer panel the screws on the top are tightened to hold the panels together during flight. Each panel has two of these spars at each of (front and rear) spars. Figure 6.6 shows a skeleton of a wing panel

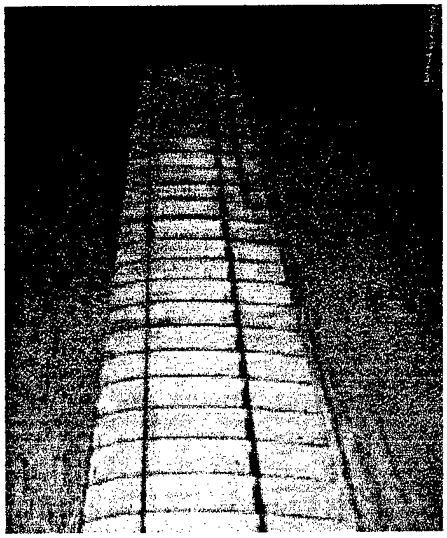


Figure 6.6. Wing panel sitting on the jig

We also investigated some other ways of attaching the panels such as using two carbon composite tubes through the center of the wing. This method was not used because it was more expensive compared to the above method. In addition, since the tubes have a circular cross section, we realized that they can not be attached to the spars properly because the spars have a flat lateral side. They will have to go through the ribs to have enough gluing area. This requires reinforcing the ribs since the balsa ribs won't be able to take the moments induced by the forces on the panel forces. This will increase the weight of the UAV, which was not desirable.

Connecting the tail booms:
Our configuration requires that the wing and tail be connected to each other with two carbon composite booms. We selected carbon composite booms with 1-inch internal diameter for each of the booms. The booms are connected at 2.75 feet from the wing centerline. The airfoil at this position is reinforced with 1/8 inch aircraft grade plywood. The booms are 5.8 feet in length and hence have to be detachable for transportation reasons. We decided that the booms would be attached from the top of the wing since the gap available between the rear spars was not enough to attach the booms through the center of the wing. A streamlined cover will be covering the boom. The cover has a fineness ratio of around 6. To secure the boom, we attached a piece of plywood to each set of spars and attached a PVC pipe with an internal diameter slightly bigger than the boom outer diameter. When the boom needs to be attached securely, two nuts and bolts were inserted through sides of the pipe and secured tightly. A similar arrangement was used at the horizontal tail. This way, the booms are completely detachable.

An alternative method discussed was to attach the booms at an angle to the wing. That is, the boom will not be perpendicular to the trailing edge but it will have an angle. This attachment will provide a longer distance along the chord. A longer attachment will distribute the stresses over a larger area. But, this type of attachment will require either cutting through the upper surface of the airfoils or making a bump on the surface of the wing at an angle to the chordwise direction. This might create turbulence and extra drag. Since this method might compromise the performance of the UAV, we chose to abandon this idea.

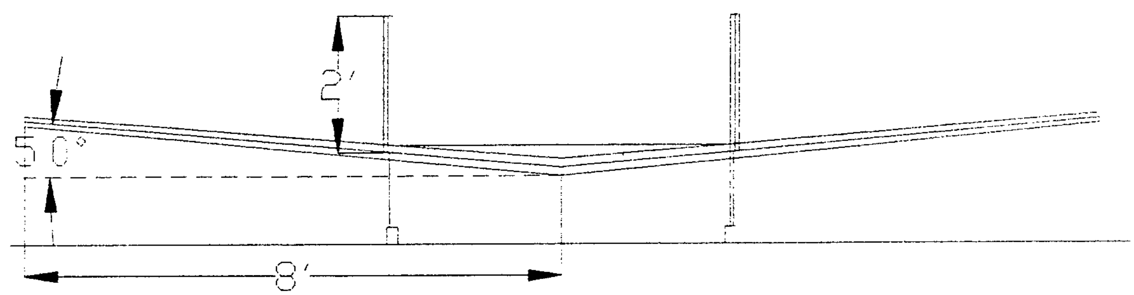
Tail Plane:
The tail plane was constructed on a ready-made jig. The horizontal tail had to be constructed in two panels and joined later. Each of the vertical tail panels were constructed using the same jig. Since the longest dimension in the tailplane is 5 feet (span of horizontal tail), we decided to make the whole tailplane a single structure. That is, the two vertical tails were permanently attached to the horizontal tail.

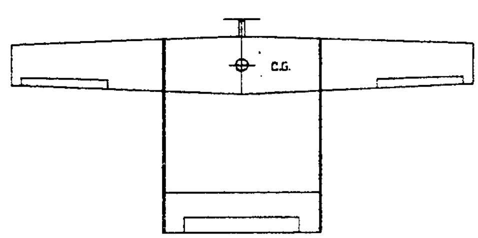
Motor capsule:
Since the motor had to be installed 6" ahead of the wing apex, we had to make a streamlined capsule for the motor. Our research showed that a body with fineness ratio of around 3 has minimum profile drag. The motor has a maximum diameter of 1.5 inches. To achieve a fineness ratio of 3, we set the capsule diameter to 2 inches. Since the motor required ventilation, the front of the capsule was left open. An opening was made at the end of the capsule to allow the air to escape. The capsule was attached to the wing using epoxy and carbon fiber laminates.

Landing Gear attachment:
We decided that the landing gear should be purchased from a vendor. We also decided to add a streamlined cover to the wheel to reduce the drag generated by the wheel. The landing gear will be attached at the position on the wing where the tail booms are attached. We decided to do so because this position is already reinforced due to the boom connections.

DESIGN REPORT FOR

CESNA ONR/STUDENT DESIGN/BUILD/FLY COMPETITION
Addendum Phase





Polytechnic University
April 10,1997

ADDENDUM

The final contest aircraft differs from the proposal design in several aspects. Most of these changes were necessitated by equipment availability, skill level and time availability. In addition, the amount of funds available influenced the final contest aircraft. There were no changes made to aircraft parameters such as the wing loading (W/S), wing plan form area (S) and aspect ratio. However, the actual aircraft weight differed from the calculated number. This led to changes in the wing loading and performance of the aircraft. With this change and a change in battery pack selection, the available number of laps increased to 21. The cost of manufacturing the aircraft matched very closely with the expected cost proposed in proposal phase of the report. An attempt was made to keep the airframe cost as low as possible during the construction phase. Some changes in the manufacturing process were implemented to reduce the cost. The following sections describe changes made and the reasoning used to implement these changes.

Final Weight analysis:

$$W_{Take\text{-}Off} = W_{Structure} + W_{Payload} + W_{Batteries} + W_{Propulsion\ System} + W_{landing\ Gear}$$
$$W_{TO} = W_S + W_{PL} + W_B + W_P + W_{LG}$$
$$= 7 + 7.5 + 3 + 1.3 + 1$$
$$= \mathbf{20\ lb.}$$

It was assumed in all of the calculations earlier that the aircraft would weight 23 lbs. But, the final contest aircraft weighs less than the proposed weight. Since the wing planform area was kept constant, the wing loading decreased. This is a desirable result, as described in the preliminary design stage.

Changes implemented during the construction phase:

We constructed the aircraft in such a way that it could be detached in as many pieces as possible to facilitate transportation. As proposed, the wing is spilt in to three sections. The center section is 8 feet in length, the two outer panels are 4 feet in length each. The outer panels are connected to the wing using two aluminum blades at front and rear spar locations.

Due to its relatively small size (5 feet span) the horizontal tail was made as one panel. In addition, the two vertical tails were permanently attached to the horizontal tail. The total length of this assembly would be 5 feet and height would be 2 feet (span of each of the vertical tails). We decided to use two tail wheels, each attached to one of the vertical tails. The tail wheels would be steered by the rudder control.

The tail booms are connected to the wing using screws, which advance into the built in threaded inserts in the wing and the horizontal tail. The booms are connected at 2.5 feet on each side of the wing centerline. The booms are connected to the under surface of the wing. For transportation, the booms can be detached by removing the screws used. A wooden disc is inserted into the carbon composite booms wherever a screw is inserted to protect the booms from being compressed.

Landing Gear:

It was mentioned in the proposal phase that the landing gear would be attached at the point where the booms are connected. It was decided that the landing gear be moved to the panel connection point to provide a wider spread of wheel base. The landing gear would be attached to an airfoil made of reinforced aircraft grade plywood. This airfoil would be slid on to the panel connector blades and secured with screws and bolts. This set up allows the removal of the landing gear for transportation purposes. To provide enough propeller clearance, the landing gear was attached in a way to make a 20^0 angle with center of gravity position. That is, the center of gravity would make a 20^0 angle with a vertical drawn at the position where the wheel touches the ground. The landing gear struts were purchased, which are 12.125 inches in length. The struts have a built in spring to provide adequate suspension during landing. Since the propeller chosen is 15 inches in diameter, we decided that at least 5 inches ground clearance is needed. To accommodate this constraint, wheels with 4 inches diameter were chosen. The landing gear is not retractable due to the weight penalty of retracting systems and low reduction in drag (8%). The wheels will be covered with wheel pants.

Payload:

The major change made during the construction phase was to move location of the payload from the wing center to the point of panel connection. This was deemed necessary to reduce the bending at the center of the aircraft. We calculated that the center section would have to support roughly 180 lbs. if the aircraft were to experience an acceleration of 5 g's. Reinforcing the center section to fulfil this requirement would increase the structural weight and airframe cost. By locating the payload at the panel connection point, when the aircraft is sitting on the ground, there are no loads at the center due to the payload. The weight of the payload is entirely supported by the landing gear since the landing gear is also located at this point. This support technique was also one of the reasons to move the landing gear to the panel connection point. When in flight, the payload actually unloads the wing since the payload tends to push down on the wing and the aerodynamic forces tend to bend the wing up. We decided to split the payload in to two sections each having the dimensions of 1”×2.5”×6.5”. Each section of the payload weighs 3.75 lbs. Since the point of panel connection is 4 feet away from the center, the payload would reduce the bending by at least 15 ft-lb. Thus, the total reduction in the bending moment at the center would be 30 ft-lb. The payload would be inserted into the airfoil section to which the landing gear is attached.

This set up also allows the removal of the payload in minimum time. The payload is readily accessible upon disconnecting the wing panels. In addition, the need for a door or hatch at the center section to access the payload is eliminated.

Propulsion System:

There were no changes made to the motor other than locating it ahead of the proposed distance. To balance the center of gravity, the motor needed to be placed 10 inches ahead of the wing apex. The motor was placed securely inside a 2.5 inches diameter PVC pipe, which was attached to the center of the aircraft. This motor capsule has a fineness ratio of 4. Also, the battery pack was constructed in such a weigh that it would slide through the motor capsule freely to balance the center of gravity.

Battery Pack:

In the proposal, the calculations were performed for a battery pack with 2500 MAH cells. This allowed only 15 cells in the battery pack in compliance with the competition rules. This battery pack would have a voltage of 18.75 Volts. We realized that the motor would have to drain more current at take-off to get enough power. As determined in the preliminary design stage, to drain more current, a bigger propeller would be needed. To avoid this problem, we decided to use 2000 MAH cells instead. The battery pack can contain up to 20 of these cells, which would produce a voltage of 26 Volts. In addition, a 2000 MAH cell was determined to be the best choice in the conceptual design stage. The reason a 2500 MAH cell was chosen because of the misleading information supplied by the manufacturer. This change in the cell selection led to a increase in the number of possible laps to 21. This is due to the increase watt-Hr capacity of the battery pack.

Improvements to be considered in the next construction phase:

Since the weight of the aircraft came out to be lower than expected, it is possible to implement a retractable landing gear system. There is definitely a reduction in drag by this implementation. The time and cost required to make this change will not be significant since the struts used for the landing gear are actually made for retractable system. Enough space is available to accommodate the retraction system at the panel connection point.

The manufacturing process can be made cheaper by having an accurate plan of items needed. Since most of our supplies came from either California or Texas, shipping cost on these items added up to more than $200. By having a plan that indicates all of the items needed, two or three bulk order can be placed to obtain all of the required items.

The control surface mechanisms can be made more drag free by using different procedures such as the ones used in commercial airliners. The current implementation has a significant gap ion the lower side of wing. This gap is required to provide for the deflection angle needed. This gap leads to turbulence and drag. Since reduction in drag is the main goal of our design, an implementation should be designed to close off any gaps on the wing and still provide for the needed control surface deflection. This implementation will be a time consuming since a design needed prior to the implementation. In addition, different non-conventional methods would be needed to cut the control surfaces on the wing.

Cost analysis:

The total cost of the UAV, including manufacturing and systems, is roughly $1900.00. Table attached shows a detailed break down of the actual cost for different components. The actual cost was slightly higher than the expected cost. This resulted due to the materials needed to reinforce the various connections, which could not be accounted for during the preliminary design stage. The actual cost is very close to the expected amount because, the selection of materials and manufacturing processes were based on the cost. In other words, a manufacturers price list was used to make decisions on different items.

Manufacturers List Price

Component	Cost	Description
Propulsion System		
Motor	$407.00	AVEOX Brushless Motor
Propeller	$12.00	APC 15 x 10 Propeller
Batteries	$125.00	2000 MAH cells SR cells (20)
Servos	$274.00	Heavy duty servos.
Radio	$209.00	Futaba 6XA PCM radio
Receiver	$160.00	PCM Reciever for fail safe mode
Structure		
Wing		
Ribs	$30.00	Balsa
Main Spar	$192.00	Carbon Composite
Secondary Spar	$5.00	Spruce
Horizontal Tail		
Ribs	$15.00	Balsa
Spars	$4.00	Balsa
	$4.00	Balsa
Vertical Tail		
Ribs	$12.00	Balsa
Spars	$2.00	Balsa
	$2.00	Balsa
Misc	$50.00	Plywood, Glue, Aluminum
Booms	$150.00	Carbon Composite, 1"ID, 70" in length
Landing Gear	$49.00	Suspension Struts
Wheels	$20.00	Du-bro 4" wheels
Wing panel Connectors	$60.00	Byron Originals wing panel connectors
Skin	$70.00	MICA film
Balsa Rite	$20.00	Fabric treatment formula for the skin
Skin for leading edge an	$25.00	1/16" Balsa
Misc	$30.00	
TOTAL	***$1,927.00***	

R.P.I. Team Flying Penguin
Attn: Prof. Rusak
JEC
110 8th Street
Troy, N.Y. 12180

AIAA Design Report

Proposal Phase

Sunday, March 16, 1997

Advisor: Prof. Z Rusak

Pilots: David Pitcairn, Joon Kim

Plane: Red Hawk I

1. Executive Summary

The RPI team found the development of the model to be extremely entertaining as well as informative. The group was well rounded with team members possessing extensive knowledge in R/C modeling as well as designing and analyzing full sized aircraft. The end result was a group of students and alumni that learned a lot about model airplanes and about each other.

From the beginning everyone decided that things should be kept elementary. The saying "keep things simple" initially came to mind. Therefore we wanted to avoid any fancy layouts such as canards/multiwing designs. A very conventional looking sailplane with conventional control surfaces was the way to go.

Obviously the most important component of the airplane was the wing. Much thought was made into the general configuration, size, and airfoil of our final wing. Our initial estimations had to be based on certain flight performance estimations. Did we want a wing that provided a lot of lift, or did we want a low drag airfoil.

Initially we had to be concerned with the 300 foot takeoff distance for if we couldn't get off the ground, all other calculations are pretty meaningless. Well drawing upon previous knowledge about models and gliders we decided that the runway length was not a limiting factor.

Because the goal is to complete the most number of laps we needed to design a plane that covered the greatest distance with the given payload. This is an important distinction to make from a maximum weight airplane. The ideal maximum weight airplane would have a very high lift airfoil like an E210 etc. These wings can lift a great deal of weight but with high drag and low cruising speed. Characteristics not ideal for a distance airplane.

We needed a wing that worked efficiently at higher speeds and with lower drag than a high lift airfoil. By looking at common model kits and the airfoils they use, and asking experienced modelers we were able to get in the "ballpark."

The battery had to be commercially available so we decided to pick the cell which allowed us the greatest energy density. The most commonly used cell in electric models is the NiCad C-cell which offer a capacity anywhere between 1200-2000 milliamp/hours. About 20 of these cells allowed us to stay under the weight constraint and the ability to arrange the cells neatly in the fuselage.

The motor also had to be commercially available. Our ideal motor had to put out sufficient power and be efficient. Electric models have recently started using brushless motors which are more efficient than older brushed motors. Gearboxes also allow us to spin larger, higher pitched, more efficient propellers. The Aveox Motor homepage has a on-line virtual dyno which helps with estimations.

The fuselage had to be as small as possible while still containing all of our components. Most high performance gliders use fiberglass fuselages which are more durable, stronger, and lighter than traditionally built up fuselages. Looking at other glider kit's help a lot with ideas about our own fuselage construction.

Most of our conceptual design drew upon the experience of our electric modelers as well as resources offered by many hobby companies, i.e. electric motor dyno's, airfoil performance programs. Preliminary calculations were done with equations learned in first year Fluids curriculum as well as first year Intro. to Flight courses and physics. Detailed design stage utilized extensive use of mockups, spreadsheets, and personal know-how.

2. *Management Summary*

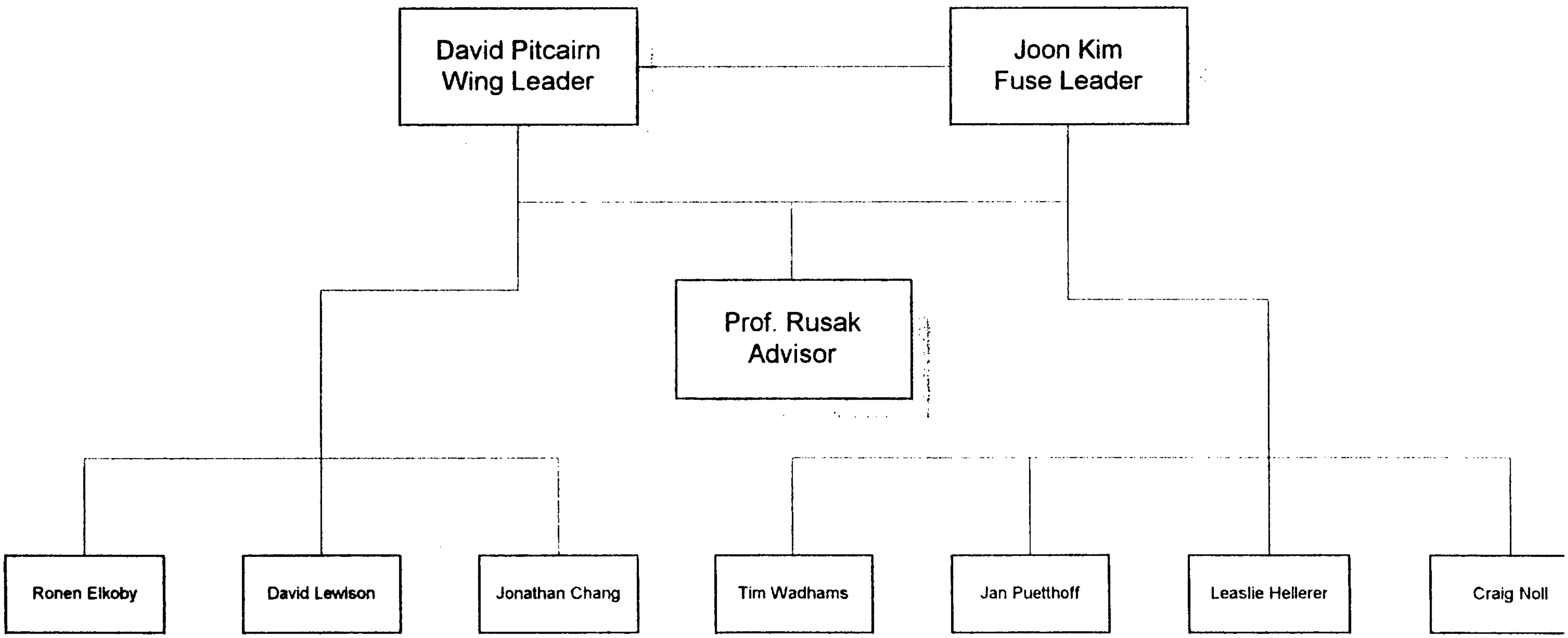
The management structure of the design team was created to be as simple and therefore as user friendly as possible. First, David Pitcairn and Joon Kim where designated as team leaders, with Prof. Rusak as the faculty advisor. The team leaders headed two separate design teams made up of the following people:

Wing: David Pitcairn, Ronen Elkolby, David Lewison, Jonathan Chang
Fuselage: Joon Kim, Leslie Hellerer, Jan Puetthoff, Craig Noll, Tim Wadhams
(Please refer to the Management Flowchart on next page for clarification.)

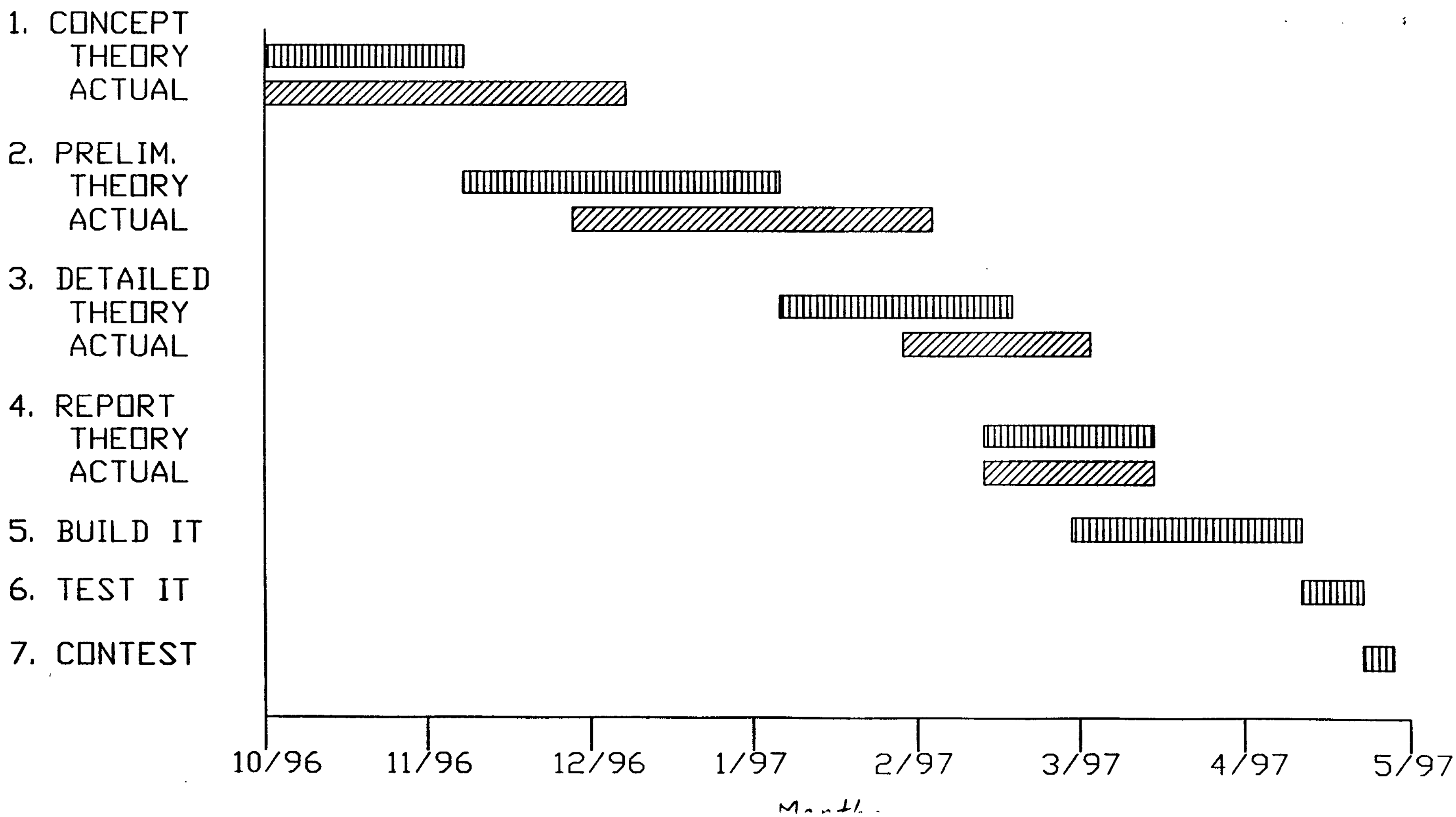
The team leaders worked together with the various other group members for the conceptual and preliminary design phases. Detailed design was done by the separate groups, with the respective team leader helping each of the members with their personal projects. Schedule control consisted of deadlines (refer to milestone chart) with a built in "margin of error", that allowed the team leaders to be lenient while making sure the project stayed on schedule. Configuration control was handled along the same lines. Since the overall configuration was fixed early on, the individual pieces were designed to fit within these limits.

Managment Flow Chart

March 16, 1997



MILESTONE CHART



3. Conceptual Design

Our first design decision was to decide on the overall aerodynamic layout of the airplane so that it could fit within the design parameter of maximum distance. The choices were multiwing, canard, flying wing or conventional layouts. The multiwing, with the mission of low drag, had little structural advantage. unneeded maneuverability, and a definite increase in drag. The canard had the efficiency and low drag advantage, but the complexity of such and undertaking along with the lack of experience with model canards prompted us to look further. The flying wing would have had a thick airfoil to house the various components and payload, therefor also creating extra drag. The conventional layout had the best compromise of efficiency, light weight, and design/build simplicity. This layout includes one fuselage with a motor in the nose.

Given the maximum battery weight criteria of 2.5 pounds, the object was to get the highest energy density possible. At first, we choose readily available 1700 mah scrc cells making up a pack of 20. The other choices were to use more lower capacity cells or less higher capacity cells. A pack of lower capacity cells would have more internal resistance, and due to the need to throttle back more, the motor efficiency would go down. The higher capacity cells would create a very bulky pack that would not fit in an efficient conventional airframe. Recently Sanyo released a new 2000 mah cell with minimal weight increase over the 1700 mah cell. This allowed us more power and runtime because we could still use a 20 cell pack and stay within the weight restriction.

There are many commercially available electric motors for models. Since the best motors are made in America, we refined our selection to include only the best brushed or brushless motor. This left us a choice between Astro(brushed) and Aveox (brushless) motors since they are the best/ most popular motors available. Though brushless motors are heavier and more expensive, they offer greater efficiency and the most potential for improved performance. We decided that the performance advantages made the added cost worthwhile, and subsequently choose the Aveox motor.

The wing planform would be of a conventional layout with ailerons for increased roll control. To achieve maximum distance, a low drag, high speed airfoil for Reynold's number about 200,000 was needed. Reynold's number of 200,000 is a well known standard in model aviation. As taught in all aerodynamics courses, a high aspect ratio wing is desirable for low drag, high efficiency flight. For the sake of the pilot's nerves, a small amount of dihedral could be added.

Wing position was also another consideration. Two choices were available to us, high wing and low wing. The high wing configuration is slightly more stable and would allow us to use existing fuselages. The low wing configuration would allow a shorter and lighter landing gear while still maintaining adequate propeller clearance. The truth of the matter is, most gliders are high wing to allow non-landing gear "belly" landings. The required landing gear eliminates this design function so we chose the low wing setup.

Sailplanes have four basic types of tail configurations. They are the conventional tail, T-tail, mid-tail, and V-tail. Due to the turbulent wake created by air flowing over the wing, we chose to use the T-tail configuration since it elevated the stab above this wake, thus making it the most efficient.

In conclusion, our team has reasoned that the most common airplane configuration is the most effective for this competition.

Conceptual FOM ranking chart

Wing Airfoil
1. Low drag
2. Adequate lift

Motor
1. Enough power for takeoff
2. Low current draw
3. Efficient

Battery
1. Best energy density
2. Small size

Wing Position
1. Propeller clearance (low wing)
2. Stability (high wing)

Wing configuration
1. Simple (conventional)
2. Enhanced properties of lift and drag (canard)
3. Improved strength (multiwing)

Construction
1. Lightweight
2. Strong
3. Quick to build

4. Preliminary Design

The first step in the preliminary design was to calculate the total weight of the aircraft. To do this an excel spread sheet was started that would include many parameters for both the preliminary and detailed design sections. Refer to Appendix A, Figure 1 whenever the excel program is mentioned.

First, approximate weights were entered for the major components and airframe. The airframe weight came out to be 7.0 lb. The expected accuracy of this weight is high due to the components having a fixed and measurable weight. The only variable is the airframe and landing gear weights. These weights were purposely overestimated for the performance predictions so they can only be lower. This meant the total weight, with a payload of 7.5 lb., was 14.5 lb.

Next, we tried to find the most efficient combination of wing planform and airfoil type. Using prior knowledge with gliders, we chose a 30 once per sq. ft. wing loading as a good starting point. Since this event is a distance event we wanted the least drag and therefore the least wing area possible. For this size airplane, you would not want to make it any greater than 30 ounces per sq. ft., so there were not really any other options. Knowing that the type of airfoils we would be using needed an aspect ratio of at least 10 to be efficient, but not more than 11 for structural and maneuverability reasons, we chose 10.5 a compromise. Now all we needed was the wing area before we could let the excel program do its thing. Based on the total weight and wing loading, we arrived at a wing area of 1,100 in^2. The excel program returned an optimal span of 107 inches. and a chord of 10.25 in. Other sizes were not investigated because this initial calculation seemed to be just what we wanted in regard to performance, ease of design, and ease of manufacture. These calculations are very accurate if we assume that the weight calculation is accurate.

The airfoil was selected using information from Dr. Selig's Home Page at Re = 200,000. The appropriate lift and drag data of likely airfoils were put into the excel program along with appropriate air properties to verify that the Reynold's Number was at least 200,000. It was, so the next step was to check to see if takeoff distance would be a problem. With a takeoff distance of around 100 ft for all of the candidate airfoils, we could discount that factor in the decision process. Since all the airfoils gave about the same performance we chose the Eppler 387 based on how well our Falcon 550 E flew.

In order to minimize the size of the fuselage, we looked into putting the weight in the wing. The internal volume of the wing was found to be sufficiently large to house a rectangular steal bar. Since the best place to put excess weight in an airplane is directly over the C.G., the most logical place was inside a rectangular wing spar.

The placement of the weight in the wing dictated using a zero dihedral wing at the root. In order to give the plane some stability, two breaks would have to be made in the outboard portion of the wing with some built in dihedral to be determined later. Now the question was whether to split the wing at the root or at the ends of the bar. The advantages of breaking the wing at the root were less complexity, easy manufacturability, and lighter weight. The calculated 2 ounce savings made the choice of a two piece wing obvious.

Another reason we did not need a large amount of dihedral was decision to use ailerons. The alternative was to use more dihedral with rudder control, but the pilot insisted on ailerons, which would provide increased maneuverability and safety.

A four foot fuselage gave us a good tail moment arm while still allowing an 8 inch nose. Eight inches was chosen to make sure we could balance the model without nose-weight, and in the event of a tail heavy aircraft, we could easily trim the excess length. This also gave a good tail moment arm since it was calculated that we would only need a horizontal stabilizer of approximately 5% of the wing area to maintain stability(see Appendix A, Fig. 9). Using our past experience and the desire for a less critical center of gravity, we decided to use 15%. This gave a stab. area of 165 sq. inches.

In order to elevate the propeller clear of the runway for takeoff, a landing gear was needed. This gear needed to be long because of the large diameter of the propeller. The three landing gear designs investigated were tricycle, taildragger, and monowheel. The drag and weight generated by each configuration decrease in that order. Since the ground handling qualities also decrease in that order, the best compromise was to use the taildragger gear, especially since the pilots prior experience with monowheels made him apprehensive.

The vertical and horizontal stabilizer airfoils needed to be low drag for maximum efficiency. The German F5B aircraft are highly efficient aircraft so we chose to just use the same airfoil they did. A simple search on the web found that they used the HD 800. For simplicity and light weight, a hinged elevator and rudder were chosen. The rudder was needed for landing and takeoff control with a tailwheel attached for maneuvering on the ground.

Due to the higher efficiency afforded by larger propellers, we chose to gear the motor. The gearbox needed to be a standard commercially available and easy to get component, so we just used the Robbe planetary gearbox that comes premounted on the Aveox motor.

The speed of the aircraft was projected to be around 60 mph. Being a distance event, a higher airspeed tends to be more efficient and is achieved with little loss of duration over a lower airspeed by pitching the propeller accordingly. To save the nerves and stay within the reflexes limits of the pilot, 60 mph was seen as a maximum.

In summary, the key features the distinguish the final configuration are as follows.

- Low wing
- Conventional configuration
- 107 inch. wingspan
- 1,100 sq. in. wing area
- Eppler 387 wing airfoil
- aileron roll control
- 165 sq. in. horizontal stab area

- "That looks about right" vertical stab. area
- HD 800 tail airfoil
- T-tail
- Single 4 ft fuselage
- Tricycle landing gear
- 2 piece wing
- payload in wing
- One geared motor

5. Detailed Design

The detailed design should first begin with takeoff predictions and calculations. Since we cannot even begin to become concerned with flight performance until we takeoff, we had better make sure we have enough power to get off the ground. An estimation could be made by a commonly used modeling formula which is 50 watts of power for every pound of plane. (Appendix A, Figure 6) Assuming a 14 pound model, 600 or 700 watts of full power is more than enough to get the model off the ground very quickly. Therefore if we got anywhere near this figure, the 300 foot runway limitation would not be a problem.

The handling qualities of the model must be ideal if we are to successfully maneuver the sailplane around the pylons and perform the necessary right and left hand loops. The configuration of the plane contains control surfaces for elevator control, yaw control, and roll control.

A rudder is necessary initially for good ground control and ability to steer during taxiing. A rudder also allows us yaw control during take-offs or landings in case of cross winds. Roll control and stability comes from the dihedral in the winglets and the use of ailerons. The dihedral should give the sailplane some "self-leveling" properties that make model airplanes more flyable. A well sized horizontal stabilizer makes a plane more stable and makes the position of the C.G. less critical. The brushless motor allows fully proportional throttle control for more precise use of power for cruise and climb.

Initial g-calculations were made by figuring out the centripetal acceleration around some radius turn with a constant maximum speed of 60 mph. (Appendix A, Figure 8) This gave us a good starting point for the g-loads that the airplane must support. Using this figure with some degree of saftey gave us a maximum g-loading of about 20. This would provide adequate strength to yank and bank at will.

To calculate the endurance and range of the plane we used equations based on our desired cruising speed, power needed for level flight, and current draw.

Using the virtual test dyno of the Aveox homepage, (Appendix A, Fig. 4 and Fig.5) we found that the current draw of our motor with a similar prop would be approximately 23.5 amps at full power. Assuming that in level cruise we will use half throttle, the current draw will be approximately 13.5 amps. This figure is further developed and proven using first year physics equations. Refer to Appendix A, Figure 8. This gives us a calcuated motor run time of about 10 minutes, giving a total range of 10 miles. Assuming one full lap worth of power is used for take off and clearing turns, the plane should still fly 19 total laps.

The payload fraction is approximately 1/2 the total weight of the airplane. We calculated the wieght of the airframe to be about 6 to 7 pounds including batteries.

The radio chosen was a Futaba PCM radio with a failsafe. This allows our plane to meet the saftey criteria as well as giving us the ability to use computer functions for mixing and optimizing our ability to contol aircraft effeciency. The servos chosen were HiTec hs-80mg metal geared micro servos. (Appendix A, Figure 10) These servos are light, take up little space, and have metal gears that are extremely difficult to strip. These are important characteristics that enable us to bury the servos in the wing and the tail. By burying the servos we simplify construction and hardware setup.

The ideal motor was obviously the most efficient one. Brushless motors offer increased effeciency as well as improved perfromance. Aveox makes many motors to fit a variety of applications. The use of a gear-box is also desireable because it allows us to turn more efficient larger diameter, higher pitched props. The motor had to be able to run on 20 cells with a gear box, output a maximum power of near 500 watts, and pull around 20 amps. Using the Aveox virtual dyno we tried different motor combinations until we got figures close to our estimation. A Aveox 1412/3y motor with a planetary gearbox match our disired figures the best.

Payload was put in the place with the most emply space which was the wing. Putting the payload in the wing allowed us to carry the weight without increasing our frontal area.

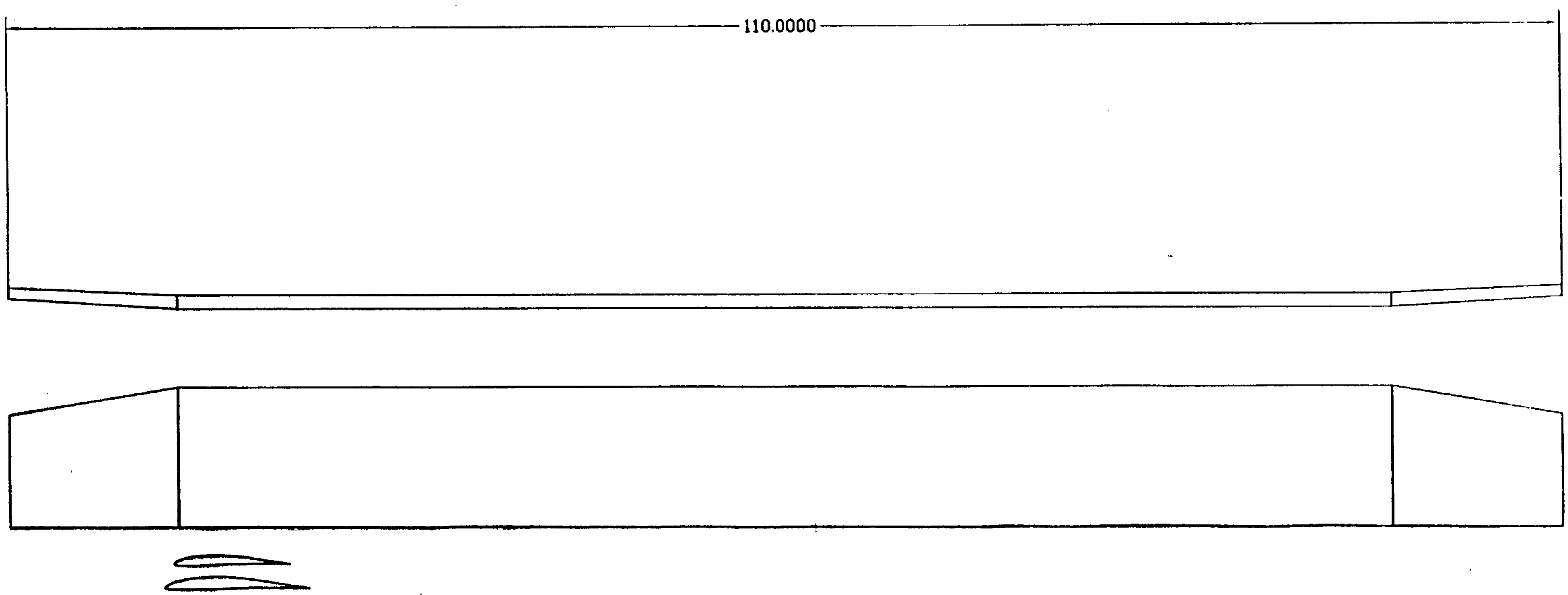
The foam core of the wing is half spyder foam to allow adequate strength for the shear loads. Spyder foam is used as the filler for the "D-Tube" area to act as a shear web while the white foam fills the rear half of the wing. Popcorn foam is very light and offers enough strength for this area of the wing. Carbon fiber fabric is layed on the front half of the wing to finish the strong "D-tube" section. Carbon fiber is stronger than normal fiberglass and with the spyder foam should act absorb all of our shear loads. The remaing areas of the wing are covered with regualr 2 oz glass cloth. 2 oz cloth is comonly used in models because it allows enough strength for the least amount of weight. 3.5 oz graphite cloth is used in the wing joiner layup becsue it is extremely rigid as compared to plane glass cloth. Details of constructions techniques are included in the manufacturing process.

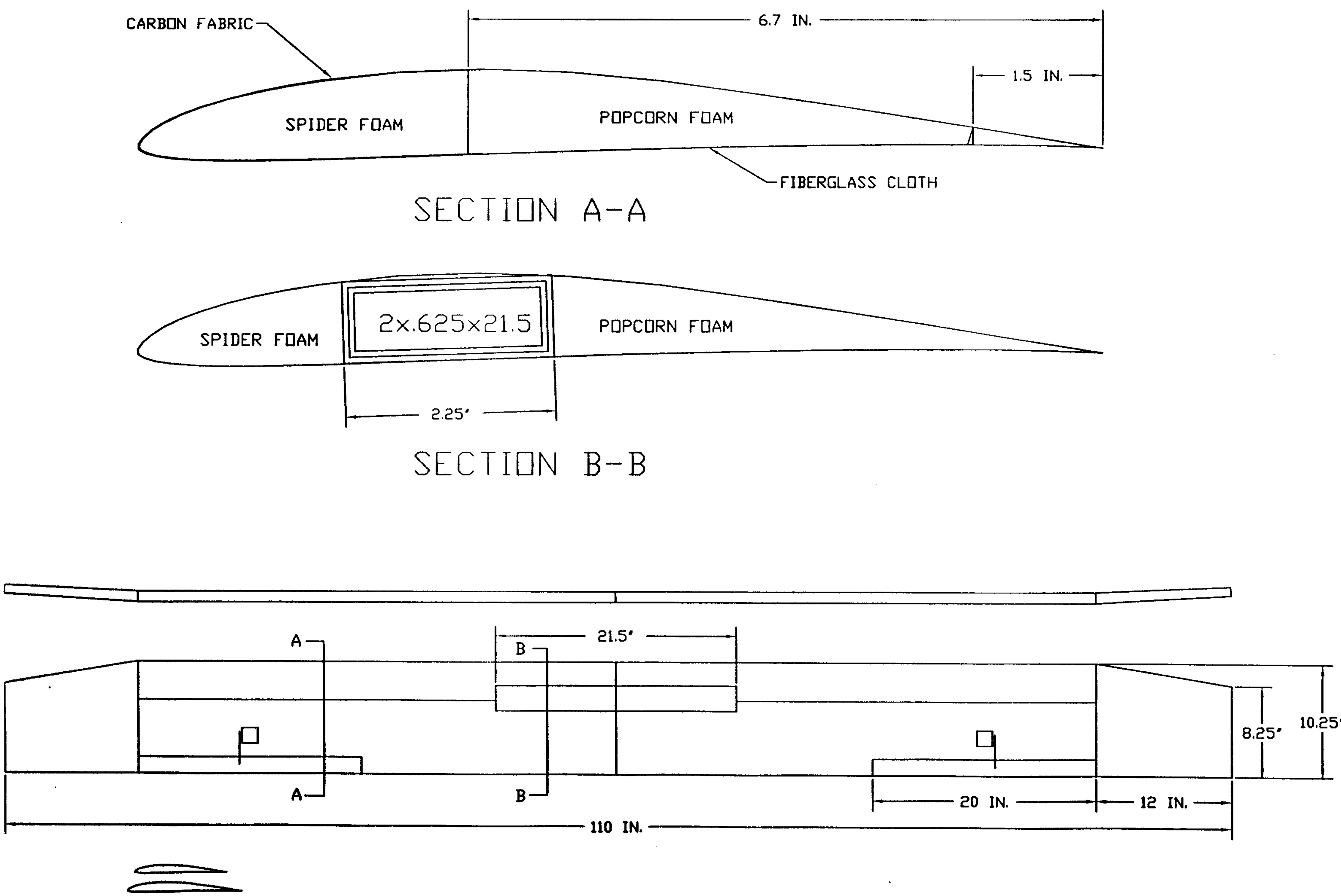
The fusalage is layed up with one layer of 2oz cloth and reinforced with kevlar and carbon tow. Kevlar is extremely light weight but has a high tension strength. Tensile strength is needed to keep the tail of the fuse from bending. Carbon tow is used in the wing area and really stiffens the exhisting layer of glass and kevlar. The wing will be exherting forces onto the fusalage at this point and flexability is not desireable.

Wheels need to be strong enough to support 14 pounds of weight but be as light and as thin as possible to minimize their effect on the plane. It was sugessted that wheels be made of circuit board and O-rings be glued on to them to obsorb some of the shock. Aslo the landing gear itself is going to be bent out of 3/16" piano wire. This size wire is usually used on planes that are about this large.

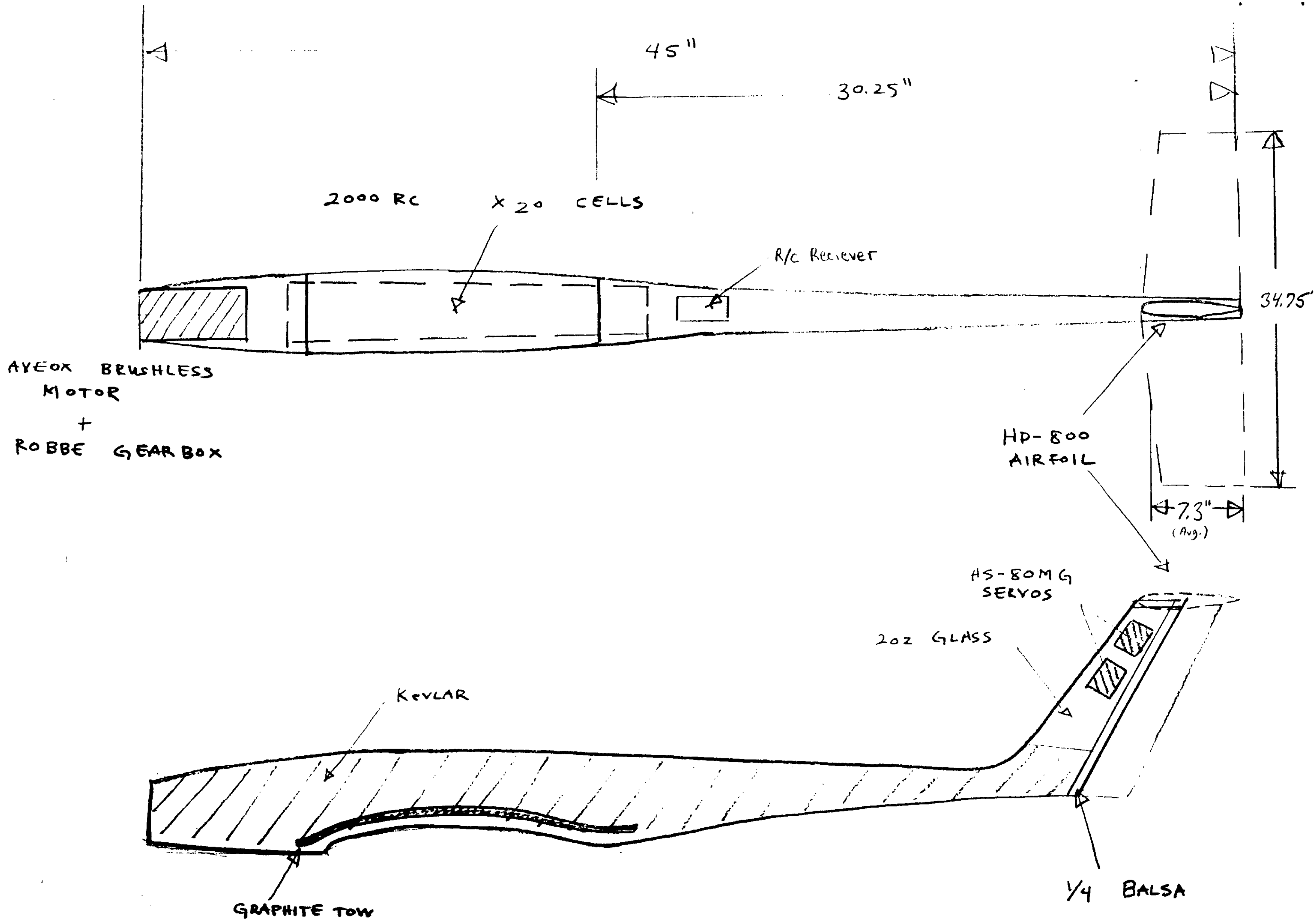
Holes will be drilled through the carbon fiber joiner and the metal bar to mount the landing gear and transfer the loads. This should tie in the landing gear into the strongest part of the airplane. This eliminates mounting blocks or brackets that add weight and complexity.

WING PLANEFORM

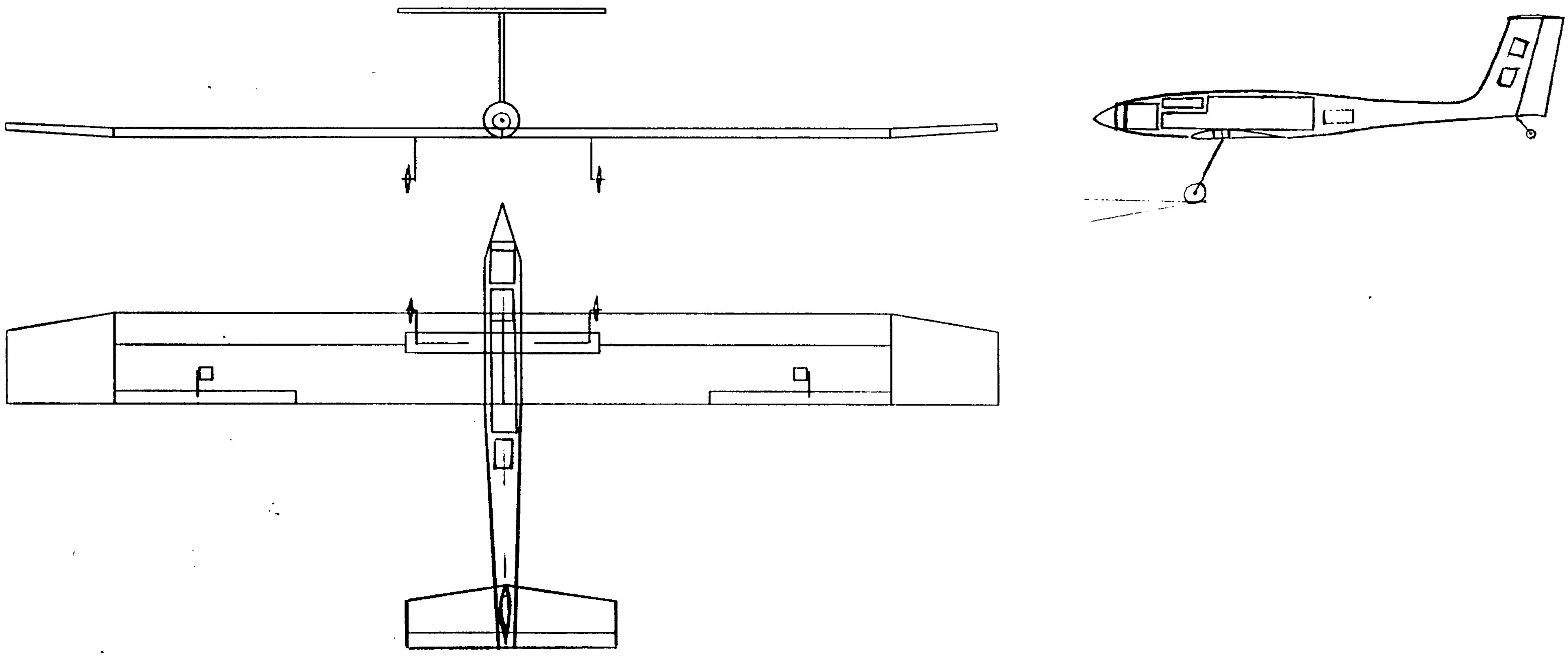




CARBON FABRIC
6.7 IN.
1.5 IN.
SPIDER FOAM
POPCORN FOAM
FIBERGLASS CLOTH
SECTION A-A
SPIDER FOAM
2x.625x21.5
POPCORN FOAM
2.25'
SECTION B-B
A
A
B
B
21.5'
8.25'
10.25'
20 IN.
12 IN.
110 IN.


45"
30.25"
2000 RC
X 20 CELLS
R/c Reciever
34.75'
AYEOX BRUSHLESS
MOTOR
+
ROBBE GEAR BOX
HD-800
AIRFOIL
7.3"
(Avg.)
HS-80MG
SERVOS
2oz GLASS
KevLAR
GRAPHITE TOW
1/4 BALSA

3-view



6. *Manufacturing Process*

The production process of our sailplane follows many traditional techniques used in many commonly available models. Our sailplane will be made of commercially available composite materials. The advantage of composite materials is its lightweight and unsurpassed strength. The use of composites gives us the greatest chance of building a durable sailplane that performs adequately.

The wing will be made up of a foam core construction with an epoxy glass skin. Many competition flyers have found this configuration to be extremely strong. Foam is cheap, lightweight, and can be easily found in many local hardware stores. Different shapes can be cut out of the foam using a hot-wire, allowing us the flexibility to cut out many different kinds of airfoils.

Our wing design utilized two different kinds of foam. The middle/main part of the wing is made of spyder foam and white foam and the winglets are made of just white foam. Spyder foam is a new product designed specifically for composite construction. Spyder foam has roughly twice the compressive strength of blue foam (Dow foam) and greater shear strength. We will use this spyder foam in the front half of our main wing section. The spyder foam in the "D-tube" section of the wing will act as the shear web, using its enhanced characteristics to absorb the shear forces. White foam or popcorn foam will be used in the rear half of the wing. White foam is extremely light and will provide adequate strength in areas of the wing that do not experience high loads.

After the two pieces of foam are adhered together with epoxy the ends are squared off and the templates for the airfoil are attached to the ends. The templates are made of thin plate 1/8" aluminum. The advantage of aluminum over wood is increased durability and higher accuracy near the sharp trailing edge. The foam core is then cut with the hot wire. The extra foam is saved for use in subsequent steps. This set of cores is marked as the left half and the process repeated for the right half.

Next, the carbon fiber wing joiner and joiner sleeve must be made. The steel bar is first covered with candle wax to act as a release medium. Then it is covered with 2 layers of 3.5 oz uni-web graphite cloth and epoxy resin. This carbon fiber sheath is our wing joiner. Then this steel bar and wing joiner are again painted with candle wax and covered with 2 sheets of 3.5 ox uni-web graphite and resin. This will be our joiner sleeve. After everything has cured, the whole assembly is heated to about 200 degrees F to allow the candle wax to melt and thus release the 3 components. Now the outside sleeve is cut in half. This method allows for a very accurate fit of the bar and the carbon fiber wing joiner ensuring the most effective transfer of forces.

After the foam cores are cut, the sleeves are epoxied into place, and the aileron servos are buried into the foam. Now the assembly is ready for the fiberglass and carbon fiber skin. Now let us look at one half of the wing. First 2 pieces of mylar, slightly larger than a wing half are cut out. On one piece of mylar, we lay the cloth for the top half which is comprised of a 4" x 45" piece of 3.5 oz uni-web carbon laid length wise, a 7" x 45" piece of 2oz fiberglass cloth laid lengthwise overlapping the carbon about 1 ", and a 21" piece of 1" kevlar tape that will act as the hinge for our aileron. This is all covered with a coat of 45 min resin to allow for adequate play time. The second piece of mylar for the bottom is exactly the same except no kevlar tape is used. Next, each piece of mylar, cloth, and resin are placed, mylar out-resin in, on the top and bottom of the foam core. This assembly is placed inside the foam cradle left over from the hot-wire cutting. Then this sandwich is placed inside a vacuum bag. After the resin has cured, the wing is trimmed and sanded. Now one inner wing half is finished and the same procedure is repeated for the other half.

The winglets are made in a very similar fashion. White foam is squared and hot wired to the airfoil templates, saving the extra foam for later on. Two pieces of mylar are cut out slightly larger than the foam core. Next, 2oz glass cloth is cut, laid on the mylar, and painted with resin. After the top and bottom halves are stuck on the core, resin side down, the assembly is stuck into the extra foam and placed inside a vacuum bag. At this point a 2 mm x 20 in slot is cut into bottom of the inner wing panels for the ailerons.

Now we must attach the winglets to the inner wing panels. The ends of the winglets and wing panels must be mitered to allow a tight fit. Then are attached with 1 inch 2oz glass tape and resin. After the appropriate hardware is installed for the ailerons and holes are drilled for the wing bolts, the wing halves are complete.

Two molds are needed to make the fuselage. One for the right half and another for the left half. Each mold is prepared with release compound before any cloth or resin is applied. First, one mold is fully laid with 2 oz glass cloth. Next, two 45 " pieces of 1 in kevlar tape are laid lengthwise down the entire fuselage, with the 2 pieces of kevlar overlapping by about ½ ". To reinforce the wing area, a 14" piece of 3/8 in carbon fiber tow is laid in the mold above the wing saddle. This is all painted with resin and allowed to cure. The same procedure is repeated for the other half of the fuselage. After the halves have cured in their molds, the excess fiberglass is trimmed off to an 1/8" above the mold. This lip is painted with resin on both molds and allows for a total of ¼" of overlap when we connect the fuselage halves. Now the 2 molds are lined up and bolted together and a balloon is inflated on the inside of the fuse which guarantees good adhesion between the halves. After the epoxy has cured, the molds are unbolted and the fuselage should release.

Micro servos are now epoxied into the tail to allow for rudder and elevator control. Wood is glued into the tail to allow hinging of the rudder and a surface to bolt on the horizontal stab. A round piece of plywood is epoxied into the front of the fuse to function as a motor mount and hardwood blocks are glued and tapped to act as wing bolt mounts. After the motor, battery, and receiver are installed, our fuselage is complete.

Although the use of composites is unfamiliar to most modelers, the process is not at all complicated. Traditional built up methods were investigated for use in the wing and fuselage but they had their disadvantages. Built up structures are simpler to construct because they don't necessitate the greater number of complex materials and tools found in composite building. Most models are built with just an X-acto knife, balsa, and CA glue. But built up structures take more time to construct and finish. It takes practice to build quickly and accurately. Built up planes require gluing, sanding, covering, and cutting. Mass producing built up structures would definitely not be ideal.

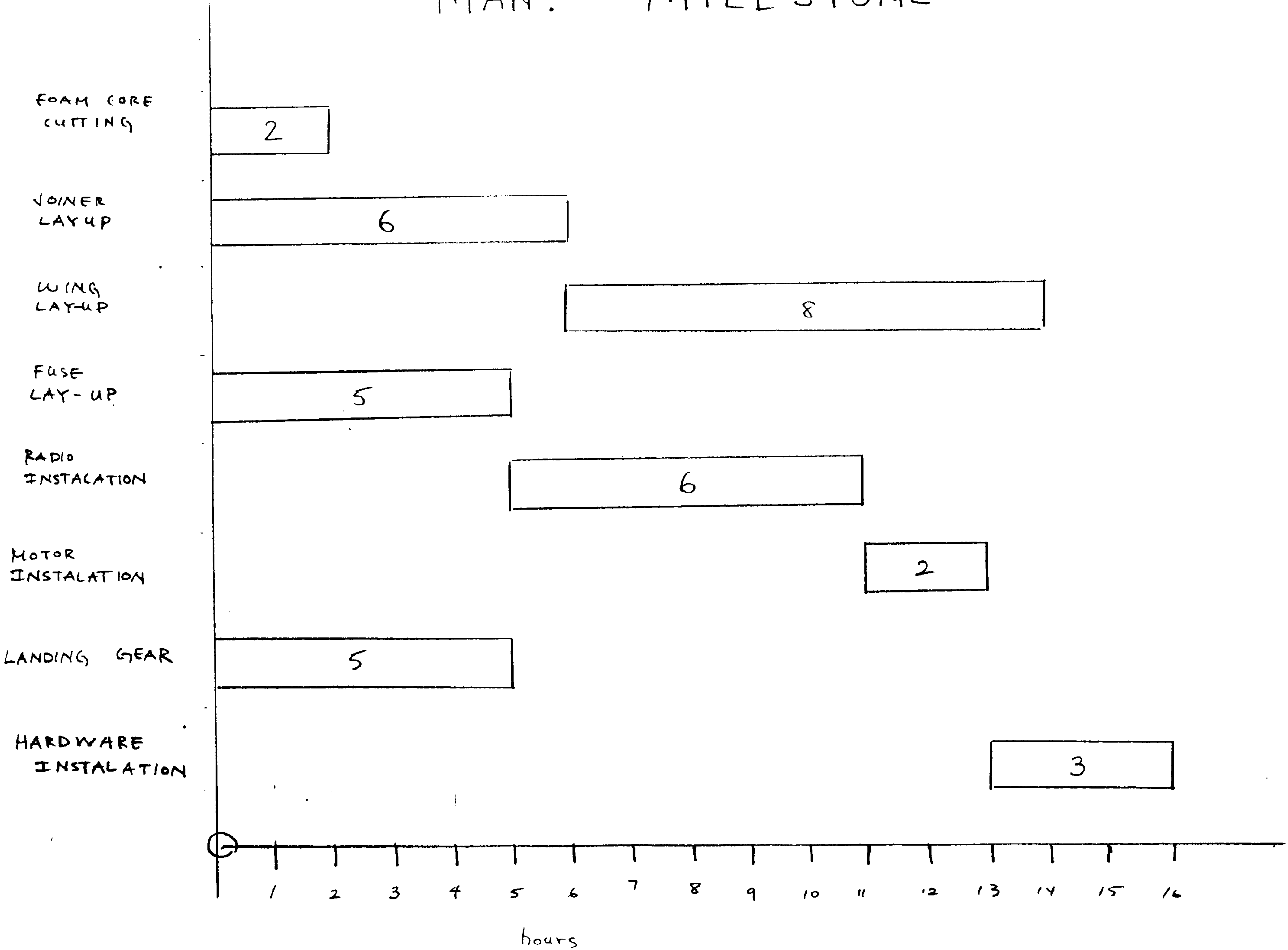
The process of composite structures lends itself to mass production. Molds can be used multiple times producing the exact same product every-time. Multiple molds can be used and laid up by one person simultaneously. Ideally one person with several fuselage molds can lay up 20 or even 30 a day. Each fuselage would be almost a carbon copy of the other one, coming out as straight and true as the molds. Fuselages can be made as fast as the epoxy can cure. According to our milestone chart, a complete sailplane can be made with about 30 man hours at $10 per hour for a total labor cost of about $300 per sailplane.

The estimated total price of all the composite materials, radio control gear, and related hardware was calculated to be 1,600 dollars. When the price of the labor is added in, the total estimated cost is 1,900 dollars. The price breakdown is as follows:

• Futaba 8UAP PCM radio	$590
• 4 HS-80MG @ $42X4	$168
•Syder foam	$ 24
•White foam	$10
•Epoxy Resin	$38
•Kevlar	$18
•Carbon tow	$4
•2oz. cloth	$8
•Uni-web carbon	$32
•Battery $8X20	$160
•motor/speed control/gear box	$510
•everything else	$40

The skill level required to make composite structures is very low. Basically, the modeler just needs to cut out pieces of fabric and paint them with resin. Built up structures are trickier to construct because attention must be paid to using the right amount of glue, so the model is not weak or heavy, joints must be tight, and attention must be paid to warping. In conclusion built up structures take longer to complete, are not as strong or as light, and do not have the quality control of molded composites.

MAN. MILESTONE



Appendix A

AIAA CARGO LIFT CALCULATIONS

			Gross W	Target Mass			
Fixed Weight (ounces)		grams	14.47775	0.4496196			
Motor&Gear	11						
Prop/hub	3						
Fuselage	10						
(20) 2Ah cells	40						
Speed ctrl/wirs	5						
Servos/Radio	9						
Ldg. Gears	5		Proposed Wing Area				
Wing	26.4		1100	in^2			
Stab	2.244		**Defined CARGO WEIGHT**				
Total	6.97775	lbs	7.5	lbs			
Cargo Box Calculations			Square Tube		lbs/ft	Pb filled	Lg@20# (in
Rho Steel	0.283		.625 X 2 X	0.043	0.97	9.1914765	26.111148
Rho Lead	0.407		1X2X	0.063	1.27	9.0622706	26.483429
			1X2X	0.083			
			1X2X	0.12	2.25	8.8404696	27.147879
POWER CALCULATIONS							
Prop Pitch	15						
Prop Dia	14						
Power In	RPM	Est. Eta	Power Out	Thrust(lb)			
77	2200	0.5	38.5	1			
532	5571	0.5	266	5			
Flight RPM		#DIV/0!					
Proposed Static Thrust			5	lbs			
Vehicle Prop Speed		75	MPH				
Estimated Speed		60	MPH	88	ft/sec		
PLANFORM CALCULATION							
	Wing		Area	Loading			
Constant Chord			1100	30.324305	oz/ft^2		
Center Section							
Span		35.46541					
Root Chord		10.85565					
Tip Chord		9.304842					
Span		107.4709					
Avg. Chord		10.23533					
Aspect Ratio		10.5					

Fig. 1

LIFT CALCULATIONS			**70 F**	**80 F**	**90 F**	**100 F**	
Kinematic Viscosity air			0.0001641	0.0001697	0.0001752	0.0001808	
Rho Air			0.0023453	0.002302	0.0022587	0.0022153	
Reynolds #			231610.44	224026.6	216923.66	210257.29	
				******70 F*******			
Airfoils	CLmax	Cd	AOA	CLplnfrm	CDplnfrm	Stall Spd	Takoffspd
SD7037	1.13	0.029	11	0.9492	0.0677095	28.134413	30.385166
SD7003-PT	1.1	0.0325	11.5	0.924	0.0691814	28.515484	30.796722
S3014-PT	1.15	0.02	11	0.966	0.0600919	27.888693	30.119789
SD7032-PT	1.32	0.0315	11.5	1.1088	0.0843213	26.030956	28.113433
E210	1.43	0.017	8.5	1.2012	0.0789916	25.009734	27.010512
E387	1.13	0.029	10.5	0.9492	0.0677095	28.134413	30.385166
Airfoils	CL@maxL/D	Cd	AOA	CLplnfrm	CDplnfrm	Cruise Spd	
SD7037	0.46	0.0098	10.5	0.3864	0.0162147	44.095896	
SD7003-PT	33.8461538	0.0325	10.5	28.430769	34.760476	5.1407019	
S3014-PT				0	0	#DIV/0!	
SD7032-PT	0.54	0.0115		0.4536	0.0203399	40.698669	
E210				0	0	#DIV/0!	
E387	0.48	0.009	10.5	0.4032	0.0159846	43.167457	
Airfoils	Ln	Takeoff Dist			Weight	14.47775	
		w/ 0 wind					
SD7037	0.27564642	102.1676					
SD7003-PT	0.29151515	105.7504					
S3014-PT	0.23584357	98.496					
SD7032-PT	0.29684371	88.34892					
E210	0.25115444	79.79402					
E387	0.27564642	102.1676					
				******80 F*******			
Airfoils	CLmax	Cd	AOA	CLplnfrm	CDplnfrm	Stall Spd	Takoffspd
SD7037	1.13	0.029	11	0.9492	0.0677095	28.397983	30.669821
SD7003-PT	1.1	0.0325	11.5	0.924	0.0691814	28.782623	31.085233
S3014-PT	1.15	0.02	11	0.966	0.0600919	28.149961	30.401958
SD7032-PT	1.32	0.0315	11.5	1.1088	0.0843213	26.27482	28.376805
E210	1.43	0.017	8.5	1.2012	0.0789916	25.24403	27.263553
E216	1.7	0.026	8	1.428	0.113611	23.152732	25.004951
Airfoils	CL@maxL/D	Cd	AOA	CLplnfrm	CDplnfrm	Cruise Spd	
SD7037				0	0	#DIV/0!	
SD7003-PT				0	0	#DIV/0!	
S3014-PT				0	0	#DIV/0!	
SD7032-PT				0	0	#DIV/0!	
E210				0	0	#DIV/0!	
E216				0	0	#DIV/0!	
#VALUE!	0.25115444	81.29608					
#VALUE!	0.31293414	70.42749					

Cont. Fig. 1

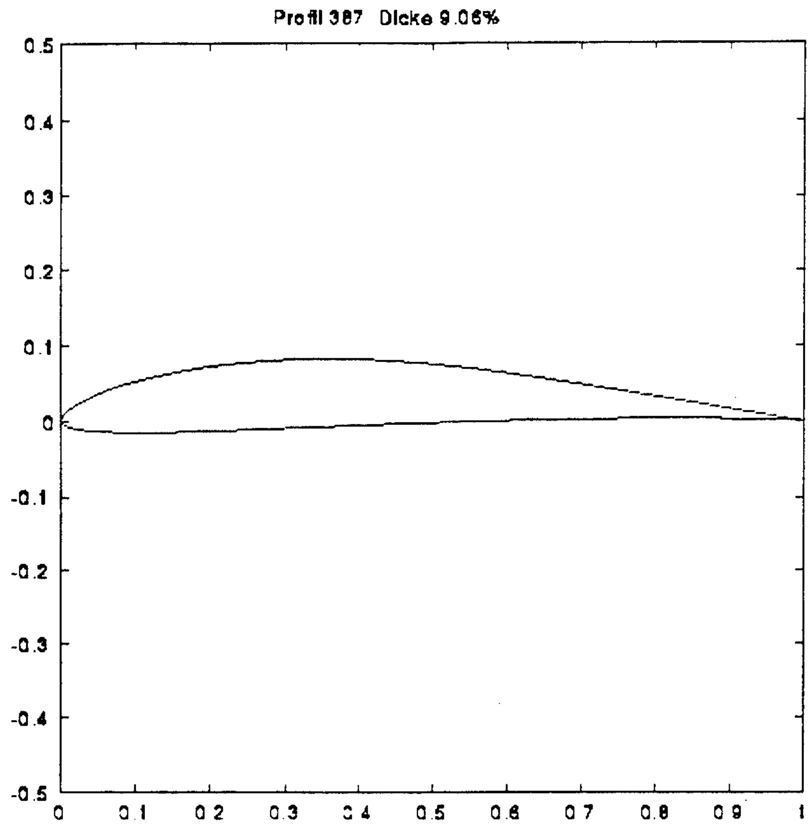
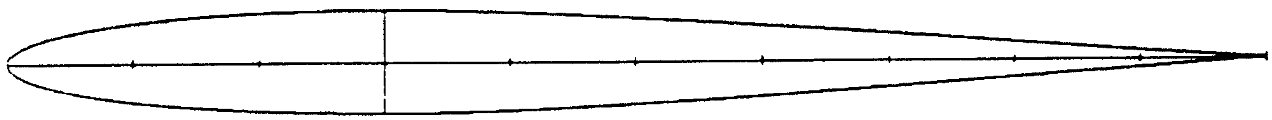
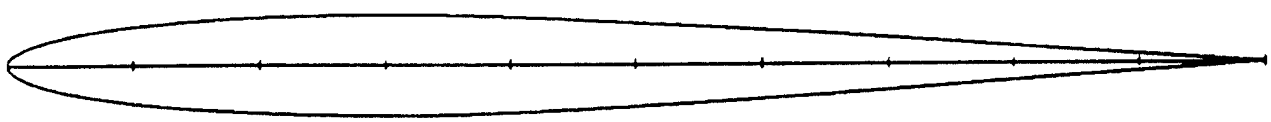

Profil 387 Dicke 9.06%
0.5
0.4
0.3
0.2
0.1
0
-0.1
-0.2
-0.3
-0.4
-0.5
0
0.1
0.2
0.3
0.4
0.5
0.6
0.7
0.8
0.9
1

Fig. 2



MAX THICKNESS = 8.0000 % AT X/C = .300

MAX CAMBER = .0000 % AT X/C = .117



HD800

Fig. 3

Virtual Test Stand Inputs

Open Circuit Voltage 1.25 Volts X 20 Cells = **25.00** Volts
Battery Resistance 0.0045 Ohms X 20 Cells = **0.0900** Ohms
Speed Controller Resistance = **0.010** Ohms

Motor Type **1412/4Y + PLANETA**
Motor, Unloaded **195** RPM/Volt
Motor Resistance **0.065** Ohms
Motor no-load current **0.7** Amps

Prop Type **Thin Folding Prop**
Propeller Constant **1.18**
Diameter **16.5** Inches
Pitch **15.0** Inches

Outputs

Motor: **4254** RPM
Current: **19.3** Amps
Voltage into Motor: **23.1** Volts
Power Input: **445** Watts
Power Output: **406** Watts
Efficiency of Motor Only: **91.1%**
Current at Max Motor Effeciency: **15.8** Amps

Back to Input Screen
Please don't forget to check out the Ecalc page for a much more extensive program for Windows 3.1 or greater.

Fig. 4

Virtual Test Stand Inputs

Open Circuit Voltage 1.25 Volts X 20 Cells = **25.00** Volts
Battery Resistance 0.0045 Ohms X 20 Cells = **0.0900** Ohms
Speed Controller Resistance = **0.010** Ohms

Motor Type **1412/3Y + PLANETA**
Motor, Unloaded **258** RPM/Volt
Motor Resistance **0.045** Ohms
Motor no-load current **1.6** Amps

Prop Type **Thin Folding Prop**
Propeller Constant **1.18**
Diameter **14.0** Inches
Pitch **15.0** Inches

Outputs

Motor: **5571** RPM
Current: **23.5** Amps
Voltage into Motor: **22.7** Volts
Power Input: **532** Watts
Power Output: **473** Watts
Efficiency of Motor Only: **88.8%**
Current at Max Motor Effeciency: **28.4** Amps

Back to Input Screen
Please don't forget to check out the Ecalc page for a much more extensive program for Windows 3.1 or greater.

Fig 5

Mega Pattern 24 Motor — power and duration for pattern planes!!

HLKM224 **Mega Pattern 24 Motor, 24 cells** **$239.00**
A bit heavier than Mega S motors. Low current draw with power for large pattern models. Order 4mm mounting screws HLRE716.

Motor:	Watts Out:	Volts:	Free RPM:	Load RPM:	Prop Size:	Amps:	Thrust Oz.:	Length Case:*	Motor Diameter:*	Shaft (mm):	Weight (oz.):
224	**686**	28.8	21.8K	10.8K	10x8	**47**	72	3.4	1.85	**5.0**	**14.8**

Motor/Gear No:	Gear Ratio:	Volts:	No Prop RPM:	Load Prop RPM:	Prop Size:	Amps:	Thrust:
224/GR1731	2:1	28.8	10.9K	5.3K	16x10	**47**	**97**
224/GR1731	2:1	36.0	13.6K	6.8K	15x8	**50**	**108**

How to choose an Electric Motor, Prop, and Reduction System

As with glow engines, experimentation is necessary. These following will get you close.

1. Choose the Electric Motor:

50 to 60 Watts per pound of airplane weight.
or...
Gives thrust greater than 1/3 of the airplane weight.

2. Choose Prop Pitch:

Pitch = $\frac{\text{Airspeed in MPH X 1805}}{\text{RPM}}$

• "Airspeed in MPH": Decide how fast you want to fly. Most RC airplanes fly between 30 and 60 mph. If you want to pitch the prop for a 30 mph climb and then fold the prop for glide put 30 here. If you want a 50 mph cruise speed with ability for a higher top speed put 50 here.

• "RPM": Decide how fast you will run your motor. If you have an on/off motor control put the Free Shaft RPM of the motor here. If you have a proportional speed controller and want to cruise at 2/3 throttle reserving full throttle for added power then put 2/3 of the Free Shaft RPM of the motor here.
(Note: If the Pitch suggested by this formula is too low and there are no props available in this Pitch, then you will need a gear or belt Reduction System to make the RPM number lower. See No.4)

3. Choose Prop Diameter:

Motor Watts:	Diameter, Inches:
10 to 50	5-6
50 to 100	7-8
100 to 150	8-9
150 to 200	9-10
200 to 300	10-12
300 to 400	12-14
Over 400	Select diameter to give about 1/2 of the Free Shaft RPM

If using a Reduction system multiply the diameters found above by: (.75 X Gear Ratio)

4. Choose a Reduction Ratio:

Reduction Ratio = $\frac{\text{RPM of Motor X Prop Pitch}}{\text{Airspeed in MPH X 1805}}$

PC program almost instantly tells you exactly how electric power, gear reduction, props, airplanes, sailplanes *(and much more) will perform!*

USRD0013 **Aero*Comp PC Program, 3.5" 720Kb floppy** **$84.00**
USRD0015 **Aero*Comp PC Program, 5.25" 360Kb floppy** **$84.00**

What if you could answer these questions, and get instant answers, and what if you could change the values instantly and see what the changes caused?:

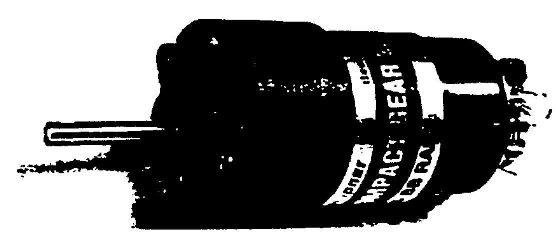
• What electric motor, prop, and battery will fly my P-51?, and how long will it run at full power? and how much thrust will I get?, and if I increase the prop to a 13-7 what will the thrust and running time be?
• Do I need gear or belt reduction with this motor on this airplane? Exactly what gear ratio would be best, and, if that's not available, then what prop should I switch to to make it work?
• There's a new electric motor that no one knows much about; I know a few specs for it, so how much thrust and flight time will it give me?
• How long will it take my airplane to climb to 600 feet? to 750 feet? and, what will the angle of climb be? What'll the Reynolds number be?
• That know-it-all at the field said that my Cub won't fly with my SPEED 600. Is he right or is he wrong?
• My motor control will burn up if I run a current of 50 amps through it for more than 3 minutes. What will the current (amps) be with an ULTRA 900, a 10x6 prop and 8 cells?
• What airspeed will I get in level flight with the motor wide open with this airplane, prop, and battery pack?
• What thrust can I get from a 14 x 8 prop running at 7400 rpm?
• What glide ratio will my electric powered sailplane get?
• What efficiency will my motor be running at?
• If I change from a 1" thick flatbottom airfoil to an undercamber what increase can I expect in glide duration on this sailplane?

We could hardly believe how well this program performed, and how easily with its Help screens. It runs on IBM compatible PC/XT/AT;s up to 586 type (no Mac's or Mac simulators), and takes 360K of disc and RAM space (a minuscule amount). It runs under DOS or Windows, including Windows 95. *Inputs:* Various electric motors (116 so far) or specs for a motor, or simulate any motor (and glow engines) by plugging in more cells, gear ratios, number of nicad cells, cell capacity, Propellers - no. of blades, diameter, pitch (to any sizes!), Aircraft type (mono, bi, triplane), wing - span, chord, thickness, airfoil, landing gear or not, runway, handlaunch, airplane weight, others. *Outputs (**almost instantly!**)* Full RPM of motor, prop RPM (geared or direct), current flow, input and output watts, motor efficiency, optimum gear ratio, wing loading, takeoff— distance, time and airspeed, cruise airspeed, rate of climb, climb angle, lift to drag ratio, glide duration, total duration, Reynolds number.

Geardrives, Belt Drives and Direct Drive Systems for Electric Flight

A Motor and Gear to power .25 size sport planes on only 8 cells!

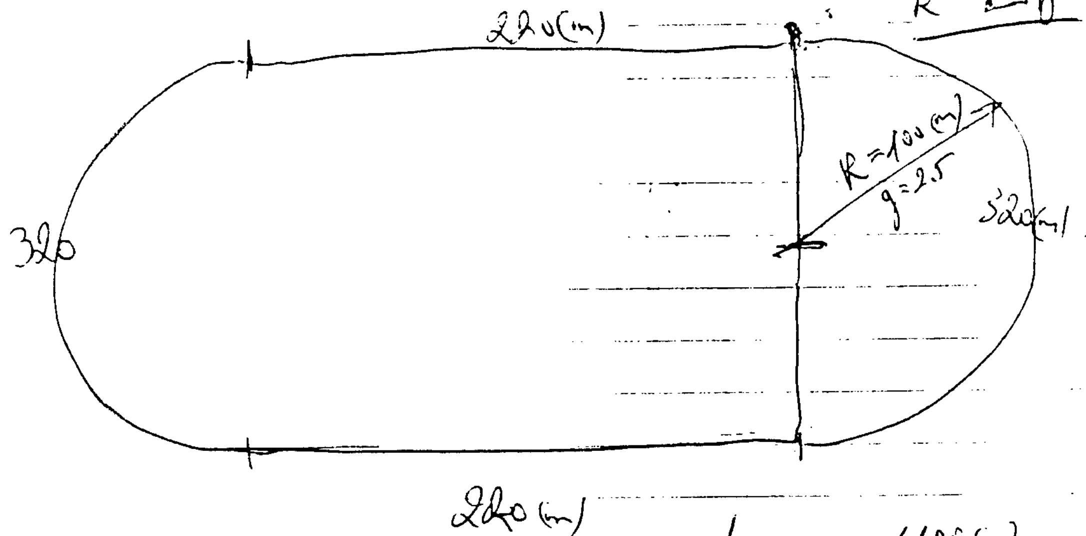
GR1722 **Compact Motor/Gear, 2.6:1, 500BB Race** $89.00
SALE to December 30, 1996 .... **$69.90**
Excellent Speed 500 BB motor with Gear Drive is 1.6" dia., 3.5" long (not counting shaft). 9.7 oz., ballbearings. Has 5mm shaft 1" long. Use GR1171 to attach regular airplane prop. The propeller shaft is only offset 3/8" from the center for easy installation in nose. Ideal as substitute for .25 engine in sport planes of about 4 pounds and 400 sq. in. wing area and in approx. 80" span electric sailplanes. Order 3mm screws HLRE290 for mounting.



Motor/Gear No:	Gear Ratio:	Volts:	No Prop RPM:	Load Prop RPM:	Prop Size:	Amps:	Thrust:
GR1722	2.6:1	9.6	10K	6.1K	11x7	**24**	**29***
				5.0K	13x7	**31**	**33***

*very conservative guesses from AeroComp3. Will out perform these numbers.

$\frac{V^2}{R} = \boxed{2.5}$



$L_{Lap} = 1100$ (m)

$\Delta t_{Lap} = \frac{L_{Lap}}{V} = \frac{1100 \text{ (m)}}{50 \text{ (m/s)}} = 22\text{ s}$

$\underline{C_D \approx 0.02}$

$C_D = C_{D_0} + K C_L^2$

$D = \frac{1}{2} \rho V^2 S_W C_D = \frac{1}{2} \cdot 1.225 \cdot 50^2 \cdot 0.75 \cdot 0.02 =$

$= 23 \text{ (Nt)}$

$\text{Work} = D \cdot V_{Lap} = 23 \cdot 50 = 1150 \frac{\text{Nt} \cdot \text{m}}{\text{sec}} \text{ (Watt)} =$

$= \underline{\underline{1.55 \text{ (Hp)}}}$

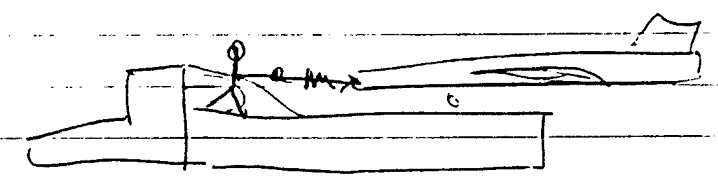
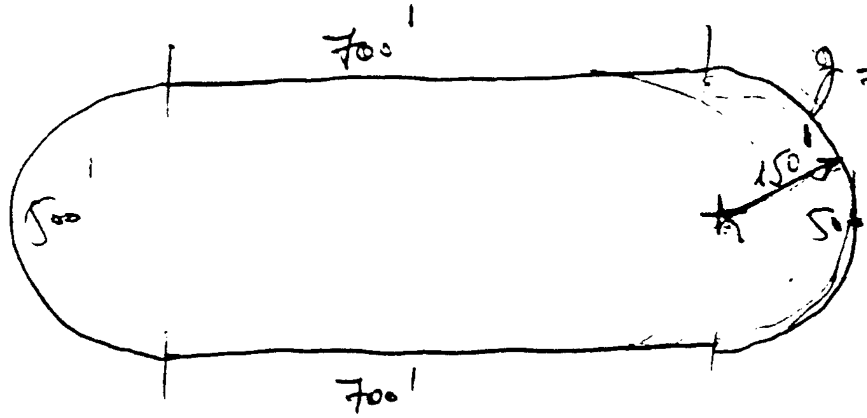


Fig 7



$$g = \frac{V^2}{R} = \frac{30^2 \left(\frac{m^2}{s^2}\right)}{50\,m} \approx \frac{900}{50} = 1$$

$$W = \frac{7\,kg}{0.6\,kg}$$

$$L_{path} = 2\times 700' + 2\pi\cdot 150' \cong 2400' = 720\,(m)$$

$$V = 60\,mph = 30\,(m/s) \qquad S_W = 10'\times 0.85' = 8.5\,(ft^2) = 0.774\,(m^2)$$

$$t_{lap} = 720(m)/30(m/s) \sim 25\,(s)$$

$$D = \frac{1}{2}\rho V^2 S_W C_{D_0} = \frac{1}{2}\cdot 1.225\left(\frac{kg}{m^3}\right) 30^2\left(\frac{m^2}{s^2}\right)\cdot 0.774\,(m^2)\cdot 0.015 = 10.8\left(\frac{kg\,m}{s^2}\right)$$

$$P = DV = 10.8\,(N)\cdot 30\left(\frac{m}{s}\right) \cong 325\left(\frac{N\,m}{s}\right)\ \frac{Joul}{s} = 325\,(watt).$$

Motor Power = 500 (watt)

~~= 20 cell · 1.2 (volt) · 2000 milliamps = 50 (watts).~~

$$\underset{(watt)}{P} = \underset{(volt)}{V}\cdot\underset{(amp)}{I} \Rightarrow I = 13.5\,(amps)$$

$$t_{flight} = \frac{2\,Ah\ 60\,\frac{min}{h}}{13.5} = 9\ \text{minutes}.$$

2/3

18 laps → 12 laps

30 laps → 20 laps

$$W = L = \frac{1}{2}\rho V^2 S_W C_L \rightarrow C_L = \frac{6(kg)\cdot 9.8\left(\frac{m}{s^2}\right)}{\frac{1}{2}\,1.225\left(\frac{kg}{m^3}\right)\cdot 30^2\left(\frac{m^2}{s^2}\right)\,0.774\,(m^2)} \simeq 0.14$$

$$C_{L_\alpha} \sim 5 \qquad \alpha = 1.6.$$

$$C_D = C_{D_0} + K C_L^2$$

$$0.01 + 2\cdot 0.02 \approx 0.05$$

Calculation of hor. tail size:

~~$40\# \rightarrow T = 54\,oz = 3.3\#$~~

$X_{ACW} = X_{CG}$.

$$\frac{X_{NP}}{C_{mcf}} = \frac{S_t\, l_t\, C_{L\alpha t}}{S_W\, C_{W}\, C_{L\alpha W}}\left(1 - \frac{2 C_{L\alpha W}}{\pi AR_W}\right)$$

$$C_{L\alpha W,t} = \frac{2\pi}{1 + 2/AR_{W,t}}$$

$AR_W = 10.5$ $\qquad C_{L\alpha W} = 5.23$

$AR_t = 4$ $\qquad C_{L\alpha t} = 4.2$

stability margin $SM = 20\%$

(35")

$$\frac{S_t \cdot 0.9(m)}{0.774(m^2)\; 0.26(m)}\frac{4.2}{5.23}\left(1 - \frac{2 \cdot 5.23}{\pi\, 10.5}\right) = 0.2$$

$$S_t = 0.08\,(m^2) \cong 0.8\,(ft^2).$$

15% of wing area:

165 sq. in.

$AR = 4.75 = b/c \qquad c = b/4.75$

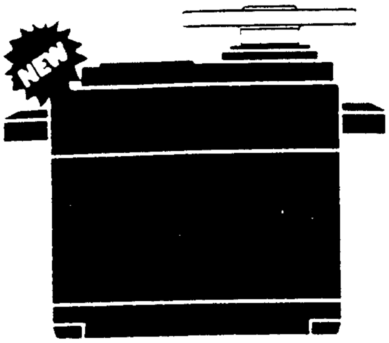
$S = 165 = b^2/c = \dfrac{b^2}{b/4.75} = 4.75\,b$

$b = \dfrac{165}{4.75} = \underline{34.75} \qquad c_{Avg.} = \underline{7.31}$

Fig. 9

SERVO

Shown actual size
Tested at 4.8Volt



$38.95

■ HS-525BB
PRECISION HI-SPEED SERVO

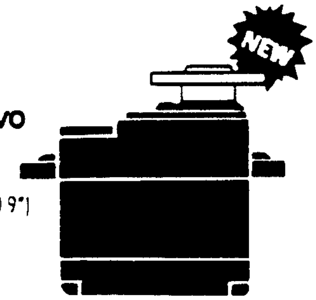
- 5 pole motor
- Ball bearing

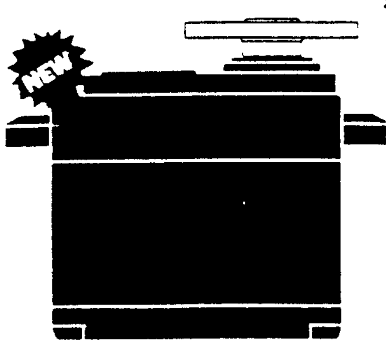
3.3kg/cm (46.2oz/in)
0.16sec/60deg
39.8x19.8x35.8mm (1.5x0.8x1.4")
44.5g (1.75oz)

$41.95

■ HS-60
SUPER MICRO SERVO

1.5kg/cm (21oz/in)
0.20 sec/60 deg
26x13x24mm (1.0x0.5x0.9")
14g (0.49oz)





$38.95

■ HS-545BB
PRECISION HI-TORQUE SERVO

- 5 pole motor
- Ball bearing

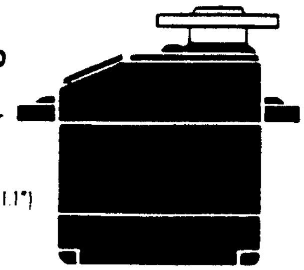
4.4kg/cm (61.6oz/in)
0.21sec/60deg
39.8x19.8x35.8mm (1.5x0.8x1.4")
44.5g (1.75oz)

$28.95 / $41.95

■ HS-80/80MG
MICRO FAST SERVO

- HS-80 Resin gear
- HS-80MG Metal gear

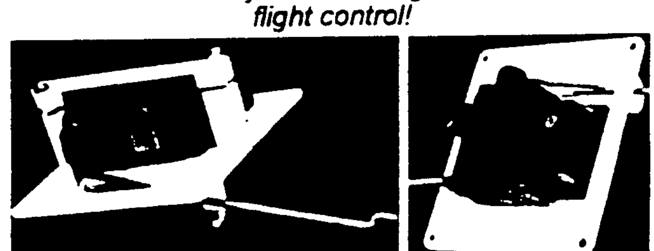
2.2kg/cm (30.8oz/in)
0.15sec/60deg
28x13.7x28mm (1.1x0.5x1.1")
HS-80 17.5g (0.62oz)
HS-80MG 21.5g (0.76oz)



You can install small servos in thin wings, and REMOVE them with these servo FLAT MOUNTS!

RA190 **Servo Flat Mounts, 2 pair** **$2.75**
About 1" long and 1/2" high. 2 pairs per pack with 4 screws and washers. Servo Flat Mounts allow the servo to lie with its side flat against the bottom so the servo installation will take up no more depth than the actual thickness of the servo itself. In a thin wing the aileron servos need to be mounted like this. Servos can sometimes be mounted by this method inside fins and horizontal stabs. You can shorten the vertical legs to the thickness of your servo. Insert a small screw and washer into the hole in the leg to clamp the servo in place. The screw can also hold the access plate or a fairing in place on top of the servo. Fits all "mini" and "micro" servos.

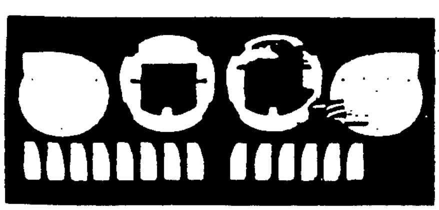
NEW! Mount your servos right next to the flight control!



RA200 **Surface Mount System for Servos, Pair** **$6.70**
2-3/8" square white plastic plates with molded-in exit fairings. Fasten your servo to them. When you mount the plastic plate in place you have installed your servo. Standard size servos mount in the included servo mount. To mount mini or micro servos order RA190 Servo Flat Mounts to these Surface Mounts.

RA210 Surface Mounts for Mini Servos, Pair **$6.70**
Same as RA200 but only 2" square. For mini servos up to 1-5/8" long (to ends of mounting tabs) such as Futaba S3101, Hitec HS80.

Aeileron/Flap Servo Mount



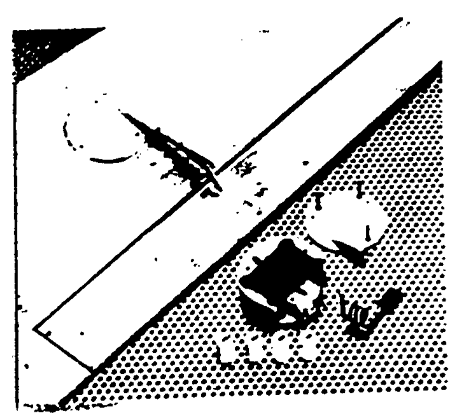
Servo Lock 1 and 2
This ingenious, all-purpose servo mount for wing-mounted servos solves the problem of installing and removing servos in the wing. The adaptors supplied cater for virtually all current wing-mounting servos.

$14.60

Servo Lock 1 **Order No. 8156**
50 mm ø x 14 mm for 13 mm servos

$14.60

Servo Lock 2 **Order No. 8157**
55 mm ø x 17 mm for 15/16 mm servos



C. 10

R.P.I. Team Flying Penguin
Attn: Prof. Rusak
JEC
110 8th Street
Troy, N.Y. 12180

AIAA Design Report

Addendum Phase

Sunday, April 13, 1997

Advisor: Prof. Z Rusak

Pilots: David Pitcairn, Joon Kim

Plane: Red Hawk I

7. Lessons Learned

Other than a change of the materials and configuration of the tail, the original design was not altered. The main areas of improvement consisted of the manufacturing process and mission profile. Implementation of these changes resulted in time and cost savings. During a second generation design approach, changes to the airframe and propulsion system would result in a superior final product. The only cost increase realized was a result of three additional hours added to the total build time.

The original design was not changed except for the tail's configuration and materials. The original configuration was a T-tail with a fixed stabilizer and movable elevator. The revised tail is of a cruciform configuration with a movable horizontal stabilizer. Replacing the original foam and fiberglass construction with wood, created a lighter structure that could be built quickly. If the plane was to be manufactured in large quantities, this change would not be cost effective. The original method takes less time once the equipment and templates are set up. However for a "one off" example, there is a cost and time savings due to the teams familiarity with that building method.

Overall, the team believes that the original design is sound, but there is room for improvement in the manufacturing process . Furthermore the mission profile was changed after the design was completed but the design still applies. The team believes that a slower speed may be more efficient for the competition but this will not be known for certain until further testing is completed(see Figure 1). While building the plug for the fuselage mold, the team experimented with a couple of different techniques. The first was to sand the fuselage shape out of foam and cover it with epoxy. This method created a very uneven surface that was hard to finish. By fiberglassing the foam, a uniform coating was obtained and the surface finish improved. This method could be improved upon in the future by shaping the plug out of wood and finishing it with paint. This would provide a better plug and consequently a better final product.

The time and cost to implement the manufacturing changes would be favorable. However the change in mission profile would not affect the total time and cost. By making the plug out of wood, the tedious and time consuming processes of repeatedly sanding and filling could be avoided. Also, foam, fiberglass, and epoxy are more expensive than wood.

When doing a second generation design, improvements would lead to a more refined final product. The improvements to the airframe could include a smaller, more dense fuselage, less tail area, and an optimized wing and control surface area. These improvements would create a lower drag and more efficient airplane with adequate handling qualities. The improvements to the propulsion system might include a different motor and propeller combination to meet more accurate specifications.

The actual costs compared favorably with the estimated costs. The only difference was the increase in build time. As a result of extensive sourcing during the design process, the expected costs still applied. The one change that slightly affected the final cost, was the decision to change the tail to a built up balsa structure. This did not change the cost of the materials but it would increase the cost of large production manufacturing due to the increased build time. This would add approximately three additional hours to the build time. With 33 man hours at 10 dollars per hour, the total labor cost would be increased to 330 dollars.

In conclusion, the experience gained during the design, and manufacture of the project will serve as a strong foundation for future teams.

Fig. 1-1 Estimation of # of laps and flight conditions

$$W = L = \frac{1}{2}\rho V^2 S_W C_L \quad , \quad C_L = \frac{W}{\frac{1}{2}\rho V^2 S_W}$$

$$t_{lap} \cong l_{lap}/V \quad , \quad l_{lap} = 2l_{min} + 2\pi R \quad , \quad R = \frac{V^2}{ng} \quad ,$$

$$D = \frac{1}{2}\rho V^2 S_W \left(\overbrace{C_{D_0} + K C_L^2}^{C_D =}\right) = \rho V^2 S_W C_{D_0} = \frac{2W}{C_L} C_{D_0}$$

$$\mathbb{P} = DV = \frac{1}{2}\rho V^3 S_W C_D \overset{\frac{kg}{m^3}\frac{m^3}{s^3} m^2 = Nt\frac{m}{s}}{=} V_{MAX}\, I\,(Amp)\,\eta \Rightarrow I_{(Amp)} = \frac{\frac{1}{2}\rho V^3 S_W C_D}{V_{MAX}\eta}$$

$$t_{flight} = \frac{a(Amp/hr)\cdot 60\left(\frac{min}{h}\right) 60\left(\frac{s}{min}\right)}{I\,(Amp)}$$

$$\#laps = \frac{t_{flight}}{t_{lap}} = \frac{a(Amp\cdot h)\cdot 60\left(\frac{min}{h}\right) 60\left(\frac{s}{min}\right)}{\frac{1}{2}\rho V^3 S_W C_D} V_{MAX}\eta \frac{V}{l_{lap}} =$$

$$= \frac{a\cdot 3{,}600\,(Amp\cdot s)}{\underbrace{\frac{W}{C_L}}\left(\frac{1}{2}\rho V^2 S_W\right) C_D} V_{MAX} \frac{\eta}{l_{lap}} = \underline{\underline{\frac{a\cdot 3600\, V_{MAX}\,\eta}{W\left(\frac{C_D}{C_L}\right) l_{lap}}}} .$$

For max # laps:

$V \to V_{MAX}$ (volt) for $2.5^{\#}$ of battery ; $V_{MAX} = 24$ (volts)

$a \to a_{MAX}$; best cells $a = 1{,}700 \div 2{,}000$ (mAmps · h)

$W \to min$; minimum weight $= 16^{\#} = 7.25$ (kg)

$\eta \to \eta_{MAX}$; $\eta_{max} = 0.5$

$l_{lap} \to min$; $(l_{lap})_{min} = 1{,}700' = 520$ (m)

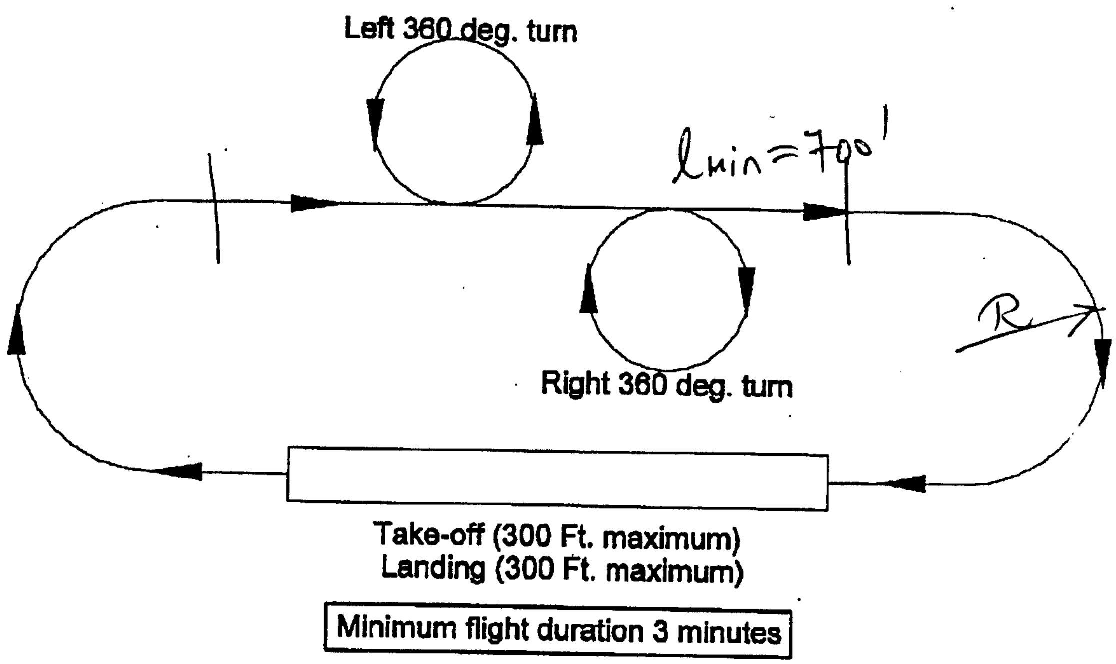
$$\frac{C_D}{C_L} \to \left(\frac{C_D}{C_L}\right)_{min} ; \quad \frac{\partial\left(\frac{C_D}{C_L}\right)}{\partial C_L} = -\frac{C_{D_0}}{C_L^2} + K = 0 \; ; \; C_L = \sqrt{\frac{C_{D_0}}{K}}$$

$$\left(\frac{C_D}{C_L}\right)_{min} = 2\sqrt{C_{D_0} K}$$

Assume: $K = \frac{1+\sigma}{\pi AR_W}$, $AR_W = 10.5$, $\sigma \sim 25\%$, $C_{D_0} \approx 0.03$

Fig. 1-2

AIAA Student Aircraft Design/Fly Competition
1996/1997 Contest Year



FLIGHT PROFILE

Fig. 1-3

$$\max(\#\,laps) = \frac{1.7\cdot 3{,}600\cdot 24\cdot 0.5}{7.25\cdot 9.8\; 2\sqrt{\frac{0.03\,(1+0.25)}{\pi\, 10.5}}\; 520\,(m)} = 29.$$

$$\#laps = \max(\#\,laps) - 2\ laps\left(\text{for maneuvers at the beginning} + 2\times 360^\circ\ \text{turns}\right) - 3\ laps\left(\text{for T/O and landing}\right)$$

$$= 24.$$

$$C_{L_{flight}} = \sqrt{\frac{C_{D0}}{K}} = \sqrt{\frac{0.03}{1.25}\pi\, 10.5} \sim 0.89.$$

$$V_{flight} = \sqrt{\frac{W}{\frac{1}{2}\rho S_w C_{L_{flight}}}} = \sqrt{\frac{7.25\cdot 9.8}{\frac{1}{2}\cdot 1.225\cdot 0.8\cdot 0.89}} \cong 13\ m/s$$

$$t_{lap} \cong \frac{l_{lap}}{V} = \frac{520\,(m)}{13\ m/s} = 41\ (sec),$$

Radius of turn at speed above and $n=2g$. $= \frac{13^2}{2\cdot 9.8} \sim 10(m)$ $\sim 30'$.

$$t_{flight} = 1{,}200\ (sec) \sim 20\ (min),$$

$$I_{amp} \cong 6\ (Amps),$$

To monitor during flight: time for a lap around $\underline{40\pm 1}$ (sec)!

MANUFACTURERS LIST PRICE

Component	Description	Quantity	Price
radio	Futaba 8 UAP PCM	1	$590
servos	HS - 80 Mg	4	$168
motor	Aveox 1412 3Y with gear box	1	$510
batteries	2,000 mah Sanyo SCRS cells	20	$160
foam	Spyder foam	2"x24"x48"	$24
resin	EZ-Lam	1	$38
tape	Kevlar	1	$18
carbon	tow	1	$4
cloth	2 oz Fiberglass cloth	1	$8.50
	Uni-Web Carbon cloth	1	$32
everything else	miscellaneous		$40
	GRAND TOTAL		**$1,593**

Fig. 1

Rensselaer Polytechnic Institute
Department of Mechanical Engineering,
Aeronautical Engineering & Mechanics
110 8th Street
Troy, NY 12180-3590

FAX #: 518-276-2623

TOTAL PAGES INCLUDING COVER SHEET 8

DATE: April 15, 1997

TO:

FAX #: 202-767-6194
Name: Gregory S. Page/B210
Address: AIAA DBF Contest
Kaman Sciences Corp., 2560 Huntington Ave., Alexandria, VA 2230
Phone: 703-960-1000

FROM:

Name: Prof. Zvi Rusak
Address: Jonsson Eng'g. Center, Rm. 4009
Rensselaer Polytechnic Institute, Troy, NY
Phone: 518-276-3036

MESSAGE:
Hi Greg, Here is our Addendum Phase Report.
Thanks for your help,
Zvi

R.P.I. Team Flying Penguin
Attn: Prof. Rusak
JEC
110 8th Street
Troy, N.Y. 12180

AIAA Design Report

Addendum Phase

Sunday, April 13, 1997

Advisor: Prof. Z Rusak

Pilots: David Pitcairn, Joon Kim

Plane: Red Hawk I

7. Lessons Learned

Other than a change of the materials and configuration of the tail, the original design was not altered. The main areas of improvement consisted of the manufacturing process and mission profile. Implementation of these changes resulted in time and cost savings. During a second generation design approach, changes to the airframe and propulsion system would result in a superior final product. The only cost increase realized was a result of three additional hours added to the total build time.

The original design was not changed except for the tail's configuration and materials. The original configuration was a T-tail with a fixed stabilizer and movable elevator. The revised tail is of a cruciform configuration with a movable horizontal stabilizer. Replacing the original foam and fiberglass construction with wood, created a lighter structure that could be built quickly. If the plane was to be manufactured in large quantities, this change would not be cost effective. The original method takes less time once the equipment and templates are set up. However for a "one off" example, there is a cost and time savings due to the teams familiarity with that building method.

Overall, the team believes that the original design is sound, but there is room for improvement in the manufacturing process . Furthermore the mission profile was changed after the design was completed but the design still applies. The team believes that a slower speed may be more efficient for the competition but this will not be known for certain until further testing is completed(see Figure 1). While building the plug for the fuselage mold, the team experimented with a couple of different techniques. The first was to sand the fuselage shape out of foam and cover it with epoxy. This method created a very uneven surface that was hard to finish. By fiberglassing the foam, a uniform coating was obtained and the surface finish improved. This method could be improved upon in the future by shaping the plug out of wood and finishing it with paint. This would provide a better plug and consequently a better final product.

The time and cost to implement the manufacturing changes would be favorable. However the change in mission profile would not affect the total time and cost. By making the plug out of wood, the tedious and time consuming processes of repeatedly sanding and filling could be avoided. Also, foam, fiberglass, and epoxy are more expensive than wood.

When doing a second generation design, improvements would lead to a more refined final product. The improvements to the airframe could include a smaller, more dense fuselage, less tail area, and an optimized wing and control surface area. These improvements would create a lower drag and more efficient airplane with adequate handling qualities. The improvements to the propulsion system might include a different motor and propeller combination to meet more accurate specifications.

The actual costs compared favorably with the estimated costs. The only difference was the increase in build time. As a result of extensive sourcing during the design process, the expected costs still applied. The one change that slightly affected the final cost, was the decision to change the tail to a built up balsa structure. This did not change the cost of the materials but it would increase the cost of large production manufacturing due to the increased build time. This would add approximately three additional hours to the build time. With 33 man hours at 10 dollars per hour, the total labor cost would be increased to 330 dollars.

In conclusion, the experience gained during the design, and manufacture of the project will serve as a strong foundation for future teams.

Fig. 1-1 Estimation of # of laps and flight conditions

$$W = L = \frac{1}{2}\rho V^2 S_W C_L \quad , \quad C_L = \frac{W}{\frac{1}{2}\rho V^2 S_W}$$

$$t_{lap} \cong l_{lap}/V \quad , \quad l_{lap} = 2 l_{min} + 2\pi R \quad , \quad R = \frac{V^2}{ng} ,$$

$$D = \frac{1}{2}\rho V^2 S_W (\overbrace{C_{D_0} + K C_L^2}^{C_D =}) = \rho V^2 S_W C_{D_0} = \frac{2W}{C_L} C_{D_0}$$

$$\mathbb{P} = DV = \frac{1}{2}\rho V^3 S_W C_D \overset{kg\frac{m^2}{s^3}m^2 = N\frac{m}{s}}{=} V_{MAX} \, I\,(Amp)\,\eta \Rightarrow I_{(Amp)} = \frac{\frac{1}{2}\rho V^3 S_W C_D}{V_{MAX}\eta}$$

$$t_{flight} = \frac{a(Amp\cdot h)\cdot 60\left(\frac{min}{h}\right) 60\left(\frac{s}{min}\right)}{I(Amp)}$$

$$\# laps = \frac{t_{flight}}{t_{lap}} = \frac{a(Amp\cdot h)\cdot 60\left(\frac{min}{h}\right) 60\left(\frac{s}{min}\right)}{\frac{1}{2}\rho V^3 S_W C_D} V_{MAX}\eta \frac{V}{l_{lap}} =$$

$$= \frac{a \cdot 3600\,(Amp\cdot s)}{\underbrace{\frac{W}{C_L}}_{}\left(\frac{1}{2}\rho V^2 S_W\right) C_D} V_{MAX} \frac{\eta}{l_{lap}} = \frac{a \cdot 3600\, V_{MAX}\,\eta}{W\left(\frac{C_D}{C_L}\right) l_{lap}} .$$

For max # laps:

$V \to V_{MAX}$ (volt) for $2.5^{\#}$ of battery ; $V_{MAX} = 24$ (volts)

$a \to a_{MAX}$; best ~~8~~ cells $a = 1{,}700 \div 2{,}000$ (mAmps · h)

$W \to min$; minimum weight $= 16^{\#} = 7.25$ (kg)

$\eta \to \eta_{MAX}$; $\eta_{max} = 0.5$

$l_{lap} \to min$; $(l_{lap})_{min} = 1{,}700' = 520$ (m)

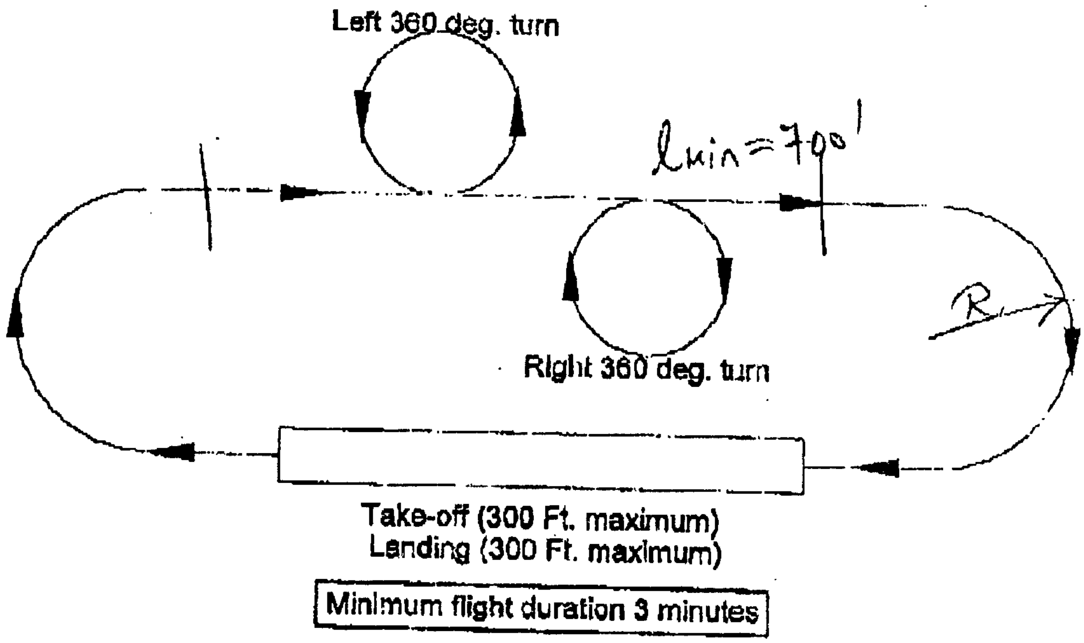
$$\frac{C_D}{C_L} \to \left(\frac{C_D}{C_L}\right)_{min} ; \quad \frac{\partial\left(\frac{C_D}{C_L}\right)}{\partial C_L} = -\frac{C_{D_0}}{C_L^2} + K = 0 \; ; \; C_L = \sqrt{\frac{C_{D_0}}{K}}$$

$$\left(\frac{C_D}{C_L}\right)_{min} = 2\sqrt{C_{D_0} K}$$

Assume: $K = \frac{1+\sigma}{\pi A\!R_W}$, $A\!R_W = 10.5$, $\sigma \sim 25\%$, $C_{D_0} \approx 0.03$

Fig. 1-2

AIAA Student Aircraft Design/Fly Competition
1996/1997 Contest Year



FLIGHT PROFILE

Fig. 1-3

$$\max(\#\text{ laps}) = \frac{1.7 \cdot 3600 \cdot 24 \cdot 0.5}{7.25 \cdot 9.8 \cdot 2 \sqrt{\frac{0.03\,(1+0.25)}{\pi \cdot 10.5}}\ 520\,(m)} = 29.$$

$\#\text{laps} = \max(\#\text{ laps}) - 2\text{ laps}$ (for maneuvers at the beginning + 2× 360° turns)
$- 3\text{ laps}$ (for T/O and landing)

$= 24.$

$$C_{L_{flight}} = \sqrt{\frac{C_{D0}}{K}} = \sqrt{\frac{0.03}{1.25}\pi\, 10.5} \sim 0.89.$$

$$V_{flight} = \sqrt{\frac{W}{\frac{1}{2}\rho S_w C_{L_{flight}}}} = \sqrt{\frac{7.25 \cdot 9.8}{\frac{1}{2} \cdot 1.225 \cdot 0.8 \cdot 0.89}} \cong 13\ m/s$$

$t_{lap} \cong \dfrac{l_{lap}}{V} = \dfrac{520\,(m)}{13\ m/s} = 41\,(sec),$ Radius of turn $= \dfrac{13^2}{2 \cdot 9.8} \sim 30'$
at speed above and $n = 2g$.

$t_{flight} = 1{,}200\,(sec) \sim 20\,(min),$

$I_{mp} \cong 6\,(Amps),$

To monitor during flight: time for a lap around $\underline{40''}$!

MANUFACTURERS LIST PRICE

Component	Description	Quantity	Price
radio	Futaba 8 UAP PCM	1	$590
servos	HS - 80 Mg	4	$168
motor	Aveox 1412 3Y with gear box	1	$510
batteries	2,000 mah Sanyo SCRS cells	20	$160
foam	Spyder foam	2"x24"x48"	$24
resin	EZ-Lam	1	$38
tape	Kevlar	1	$18
carbon	tow	1	$4
cloth	2 oz Fiberglass cloth	1	$8.50
	Uni-Web Carbon cloth	1	$32
everything else	miscellaneous		$40
	GRAND TOTAL		**$1,593**

Fig. 1

Washington State University
Chapter of the AIAA

Addendum Report

Coordinator:
David Darrow Jr.

Co-Coordinator:
Kevin Koller

President:
Joe Martinez

Advisor:
Walter J. Grantham

Addendum:

Deviations from original design:

Tail Section: The tail section was redesigned to provide a larger stabilization area as well as larger control surfaces. The horizontal stabilizer was upgraded from a fixed T tail with elevator to a flying stabilizer. The orientation with respect to the rest of the aircraft was unchanged. These improvements were made to improve expected control as well as certain aspects of aerodynamic efficiency.

Wing Sections: The wings were left unmodified from those described in the original report.

Fuselage: The fuselage was completely redesigned to accommodate components around the aircraft's center of gravity. Most components were moved forward to adjust for the moment about the c.g. created by increased tail weight. At the time of the original report, masses of components that had not been verified caused uncertainty in the original design and when these figures became available, the components were repositioned accordingly. The method of mold construction was also upgraded to incorporate CNC milling. This provided precise toleranceing and therefore a minimal size of the fuselage shell. The construction material was changed from a multi layer fiberglass carbon fiber composite to just carbon fiber. Carbon fiber's greater tensile modulus allowed for the required strength with fewer layers of cloth giving a further reduction in size and weight of the fuselage shell. The shell presented itself as easier to mold, as it could be cast over a single piece of machined foam and vacuum bagged with this light woven carbon fiber cloth and epoxy composite. Once the shell was formed, the main components were then inserted with minimal internal support required because of the strength and rigidity of the outer shell. A system for adjusting the angle of thrust was also developed and implemented. This would allow for flight trimming to be accomplished without increasing control surface drag. The steel payload was repositioned as close to the center of gravity as possible and equally fore and aft to insure the same c.g. with payload as well as without.

Electrical System: Originally 1100 mAh Sanyo racing cells were decided on to power the propulsion system. After completing some research, we discovered that Sanyo manufactured a higher capacity cell that maintained a similar weight to that of the racing cells. We provided the manufacturer with our predicted flight parameters. With this information, simulations were run on these new cells to verify that the cells could handle the large loads of a battery pack this size. The tests favored the new cells and it was decided to proceed with the new type of battery pack.

Construction Techniques: Efforts were made to lower the weight of the aircraft. Plywood, aluminum, fiberglass, and other heavy components were substituted for lighter carbon fiber wherever possible. In most cases, this did not increase costs because of large quantities of donated material available to student projects. Balsa was sandwiched between unidirectional carbon fiber to create composite replacements for heavier plywood bulkheads and other structural components.

Lessons Learned:

The manufacture of the aircraft proved to be far more difficult than the design process, report preparation, and preliminary evaluations combined. The large size of the plane and required efficiency called for construction techniques that had never before been attempted by anyone involved in the project. In the first stages of construction, less critical and less cost intensive components were built. This gave us some idea about how to proceed on the rest of the construction process. The first component to be completed was the tail, after being completely redesigned. The decision to design a larger tail that incorporates a flying stabilizer was made after a prototype tail was fabricated and it appeared to be undersized. Calculations showed that it should have been adequate, but unknowns such as the adverse affect on control of low speed airflow over a smaller control surface and overall low speed stability of the aircraft in possible high winds prompted the design change. The tail was then redesigned and rendered in three dimensions to check proportionality to the rest of the aircraft before construction proceeded on to the new and final lay out. The elevator servo was relocated to the top of the tail and changed to a standard size from a mini to increase torque. This allowed for a shorter steel pushrod that eliminated slop inherent in a nylon pushrod system. The new location also provided for greater accessibility and ease of modification to the pitch control system. As tail construction was completed, the design of the fuselage was also concluded.

The fuselage shell in itself posed the greatest challenge of the entire project. The original plan for the fuselage was to build a stringer and bulkhead system covered with monokote. It also called for the motor and drive train alone to balance out the weight of the tail assembly. With the increased weight of the larger tail assembly, the batteries had to be placed forward of the c.g. to maintain balance. The repositioning of the major components to positions forward of the wing created possible structural problems and problems with space available to place components was also encountered. The process for the manufacture of the bulkheads was never actually completely worked out either. One of the group members came up with a way to simplify the internal structure if the shell could support some of the loading. A structural shell called for the use of composites and a rethinking of the design of the fuselage. This new design required the construction of a positive mold to lay-up the composite materials. Detailed drawings were made in Pro-Engineer to determine the best placement for the components assuming minimal internal support and a shell with minimal surface area. Once placements were determined, the fuselage was drawn again in CADkey so that it could be transferred to MasterCAM so that the mold could be machined with the CNC mill. MasterCAM assigned surfaces to the splines created in CADkey and plotted cutting paths for the CNC mill to machine a positive mold plug out of medium density foam. The CNC mill process insured that high tolerances would be met and that the final shell would be symmetrical with respect to the centerline. The foam was then covered with lightweight carbon fiber cloth and vacuum bagged. An access hatch was cut in the cured shell and the foam mold was removed. This allowed structural members and components to be inserted and the electrical system finalized. Special attention was paid to the area where the tailboom assembly joined up to the fuselage and wingrod. This area distributed a large amount of the flight loading and needed to be especially strong. Once the fuselage shell was completed the components could all be installed and the aircraft was ready for final covering, finishing, and flight-testing.

Construction lessons:

Carbon fiber although light was discovered to be difficult and at times dangerous to machine. The thin nature of carbon fiber tubing allows it to flex when a lathe bit is applied. This made a boring operation on the tailboom extremely difficult. Sanding was the only option that had an affect on the carbon surface, but the large amount of fine dust posed health risks to the machinist. Good ventilation and masks were required whenever the carbon needed to be worked.

Computers were in most cases found to cause almost as many problems as the solutions they provided. In the machining of the fuselage, many obstacles were run into in the implementation of computer aided manufacture. The shape of the surface to be machined out of foam for the fuselage mold is quite complex. For this reason, the university machine shop had to be contracted to mill the foam mold. Preliminary drawings were created and finalized in version four of CADkey. They then had to be converted to version seven to be modified for MasterCAM. From Cadkey they were translated to IGES and then to MasterCAM Ge3 files. Once in MasterCAM, errors were discovered in the connections of lines and corrections were required in the original drawings before the shop could proceed. These irregularities inevitably were a result of repeated file transfer between software systems. Once corrected more walls were run into because of the shop personnel's unfamiliarity with the surface we required. For the most part, the technicians were learning as they fabricated the mold plug. This caused time delays but ultimately resulted in a higher quality mold than could have been produced by hand. Computer aided machining will be a much more viable option if one software package is available to us that will take the part from the drafting stage directly to CNC tool paths.

Delays in the shipment of parts also caused problems and by the end of the project, all parts were ordered at least one week or more before they were needed to insure their timely arrival. Most of the composite materials we used are only manufactured in large quantities. Hobby shops do carry many composite materials, but not of the sizes that were required for this project. Most companies had minimum orders when purchasing components such as carbon fiber tubes and cloth. Some of the chemicals required to form the carbon shell were classified as hazardous materials and could not be shipped by air and therefore would not have arrived in a timely manner. Substitutes had to be found in order to complete manufacture. Problems were also encountered with local vendors. The Pullman area is generally rural and selections of required materials were extremely limited. Approximately eighty-five percent of the materials were acquired from vendors throughout the country. To find these venders a great deal of time was spent on the Internet and in modeling or trade magazines. Once found; telephone correspondence, email, and fax were the only means of communicating requirements and ideas. This took a greater amount of time and effort than talking to someone in person.

Design Modifications for next year:

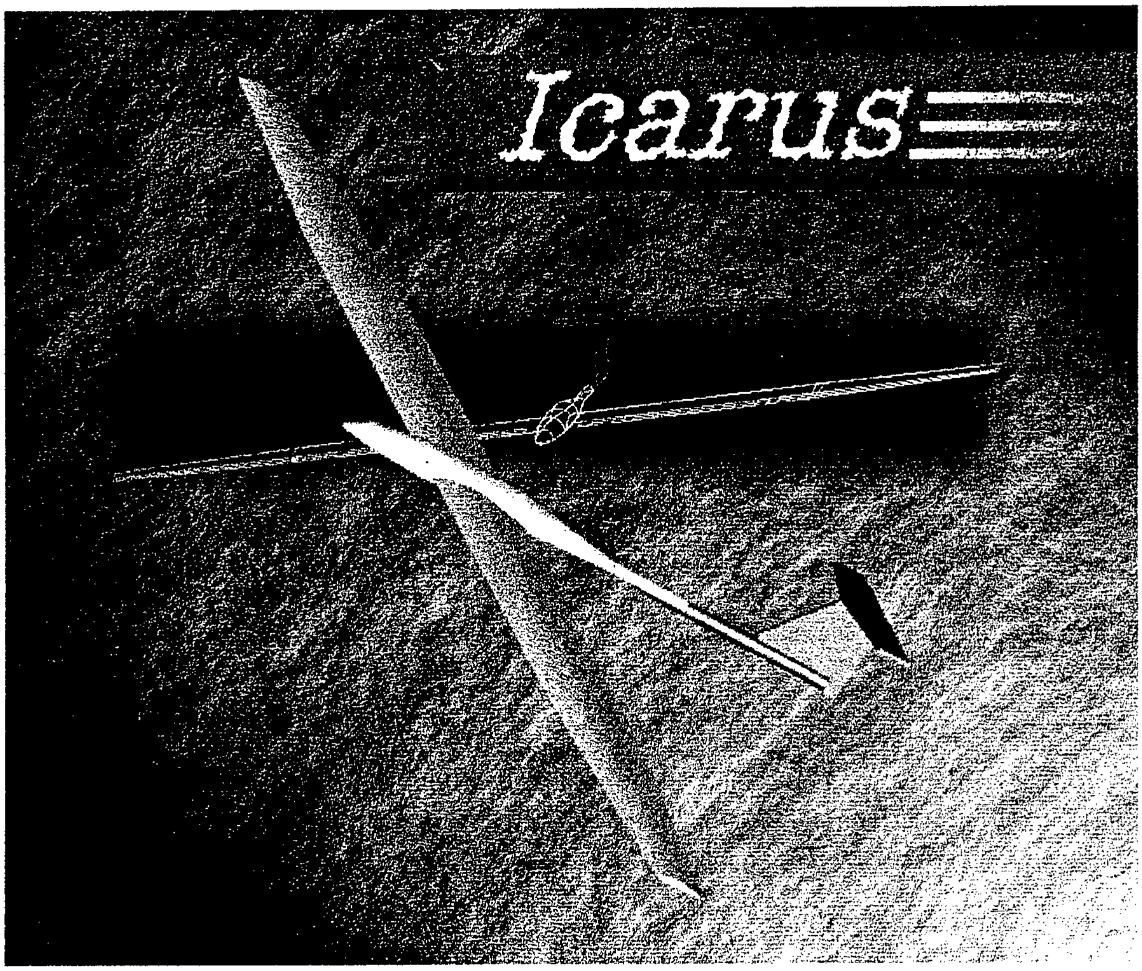
The overall design of this year's plane focused on simplicity of construction and proven concepts. Efforts were made to use proven aerodynamics in order to insure that the final design would perform within a predictable and manageable flight envelope. The plane is conventional in most design respects except for some of the construction techniques. Our entry for next year would incorporate a much lighter overall design. A flying wing presents itself as the best option to improve efficiency as well as weight. Other electronic equipment may also be incorporated next year to better track the aircraft and relay exact positions back to the ground. The aircraft built this year was designed with future competitions in mind. The motor, speed control, radio, batteries and other expensive components were decided upon based not only on the criteria of the plane this year, but also for a possible future larger more sophisticated machine. The overall design for next year's competition is expected to be radically different from that of the current and will undoubtedly have performance characteristics that far exceed those of the current configuration.

Name Change:

In the past weeks the name of the aircraft has been changed from Icarus to the Carbon Goose to more accurately reflect the appearance and composition of the aircraft.

1997 AIAA UAV Competition

Washington State University



Project Coordinator:
David Darrow

Assistant Coordinator:
Kevin Koller

President of the AIAA
Joe Martinez

Advisor:
Walter J. Grantham

TABLE OF CONTENTS

	page
Executive Summary	1
Management Summary	2
Conceptual Design	3
Preliminary Design	4
Detail Design	7
Manufacturing Plan	9
References	13
Appendix A	14
Appendix B	19
Appendix C	23
Airfoil Data	24
Appendix D	25

Executive Summary

The AIAA student chapter at Washington State University will design, fabricate, and demonstrate the flight capabilities of an unmanned, electrically-powered, radio controlled aircraft. Successful completion of this project could only be obtained through the basic engineering process of problem definition, research, brainstorming, implementation, and development. Once proper definition of the problem had been established, research into the theory of flight and aerodynamics was conducted by the group, leading to the preliminary concepts for the UAV aircraft.

Research of previous full-scale and model aircraft designs served as our examples for designing a proper airplane. The goal of the project was to design an aircraft with a balanced design possessing good flight handling qualities, practical and affordable manufacturing requirements, and providing a high vehicle performance. The development of our design was guided by the established rules and regulations provided by the AIAA Flight Test Technical Committee. These rules were as follows:

a) The aircraft had to provide the maximum range for a specified battery and a steel payload also take-off gross weight with payload had to be less than 55 pounds.

b) The airplane must be able to take-off over a 10 ft obstacle within a marked 300 ft. runway area. Also, the plane must complete as many laps of the flight course as possible with the available energy and land within the marked 300 ft. of runway area.

c) The plane will be propeller driven and electrically powered with an unmodified, over-the-counter model aircraft electric motor.

d) The weight of the NiCad batteries could not exceed 2.5 pounds and the plane must be able to carry a removable 7.5 pound steel payload. The payload could be segmented into no more than 3 pieces, each of which had to be rectangular in shape.

Through brainstorming, initial concepts considered for the project were: a) a flying wing, b) a high wing loading conventional aircraft, c) pusher prop design with high aspect ratio wings and twin tail booms, and d) an aircraft with a medium aspect ratio wing, puller-prop, small streamlined fuselage tail boom, and T-tail with flying stabilizers. Each of these concepts were analyzed for their advantages and disadvantages. Attributes from many different designs were combined and led to the development of a new design that exceeded the performance and versatility of previous aircraft.

The majority of the design tools used during all three phases were AutoCAD, Turbo Cad, and Pro/Engineer. Each provided visual graphics, as well as provided an outlet for creating and developing improved designs. Electricalc and Fluent were tools used in the later part of the design development, such as the preliminary and detailed design phases. These particular tools allowed the structure and design of the plane to be properly analyzed by the group.

Management Summary

Design personnel and assignment areas:

Members:	Assignments:	email addresses
David Darrow	Project Leader/Schedule and Budget Control/Pilot	99338379@mail.wsu.edu
Kevin Koller	Technical Design/Manufacturing Lead/Testing	kevink@wsunix.wsu.edu
Brian Nicholson	CAD Specialist/Progress Report	nicholso@wsunix.wsu.edu
Chris Coppock	Technical Specialist (Aerodynamics Analysis)	ccoppock@wsunix.wsu.edu
Joe Martinez	Technical Specialist (Aerodynamics Analysis)	jmartinez@mme.wsu.edu
Roger Manuel	Documentation/Report	rmanuel@wsunix.wsu.edu
Dave Cramer	Funding Acquisition/Public Relations	
Suresh Arumuganathan	CAD Specialist/Documentation/ Assembly/Report	suresh@mme.wsu.edu

The architecture of the design team of the WSU AIAA Student Chapter consists primarily of one project leader. Working alongside the project leader are the technical designers, CAD drawers, technical specialists, public relations, assemblers, and writers. Each member plays a vital role in the function and effectiveness of the whole team.

All members are given the freedom of making decisions regarding personnel assignments, schedule control, and configuration control. It is through this management structure that allows each member to work as a cohesive unit. Furthermore, in this type of management structure, the members are allowed to use their strengths more efficiently. Each member relays progress to the project leader on a weekly basis.

Milestone Chart for Construction and Testing of the UAV:

February 10	Order parts, Begin Tail assembly
February 17	Finish Tail Assembly Start Fuselage
February 24	Receive Wing Components and Build Fuselage
March 3	Finish Assembly of Major components, cover structure.
March 10	Finish main assembly, add landing gear.
March 17	Complete electrical System and have plane air worthy (spring Break)
March 24	Begin testing (no flight)
March 31	Begin Flight testing
April 7	Flight testing
April 14	Finish Flight testing, Pack Up and ship to Maryland
April 21	Wait for contest.
April 26	Contest
April 27	Rain Date

Conceptual Design - Configuration Selection

Several aircraft designs were looked at, with emphasis on the type of wing that was going to be used. The first type was a flying wing, this idea was turned down because it was not a well proven design and there would be little research about it. An aircraft with a high wing loading was looked at to give high speed flight but this design sacrificed efficiency. The design of a pusher prop, high aspect ratio wing with twin tail booms was looked at. This design seemed very hard to construct. Another type of design was the medium aspect ratio wing with puller prop, small streamline fuselage and tail boom, and a T-tail with flying stabilizer. This design gave the best balance of easy construction, efficiency, and cost effectiveness.

Important design criteria were using a proven design with available parts. Aircraft performance of lift, drag, and thrust were compared using software modeling programs to find the best configuration.

Motor selection had its factors for selecting the optimum design. Brushed motors are easily available, have a lower cost with a higher selection, but they have a lower efficiency. Brushless motors have a high efficiency and longer life and they are more reliable.

Design Matrix

	Proven Design	Efficiency	Construction	Cost	Performance in wind	Total
Flying Wing	3	9	6	8	8	34
High Wing Loading	10	2	8	6	9	35
Pusher Prop	5	6	4	1	5	21
Medium Aspect Wing	8	7	7	6	9	37

The final design criteria looked at configurations for airplanes that were based on a type of airfoil and power plant system that best fit the configuration. The categories were chosen because they were determined to be the most important constraints. We wanted a design that had information already available and proven to be dependable. Efficiency is a key to a winning design. With limited time to build and test the airplane a design that is easy to construct becomes vital. Cost and performance in high wind areas are two issues that the group felt must be a factor in choosing the best plane. An efficient aircraft in still conditions might not be able to perform in high head winds or cross wind conditions.

Preliminary Design - Performance Estimation and Vehicle Sizing

Each aircraft must

- Complete a take-off over a 10 ft obstacle within a marked 300 ft runway area.
- The aircraft may be of any size and configuration except rotary wing or lighter-than-air. Must be propeller driven and electric powered with an unmodified, over the counter model aircraft electric motor. May use multiple motors and/or propellers. May be direct drive or with gear or belt reduction. For safety, each aircraft will use a commercially produced propeller. Teams may modify the propeller diameter by clipping the tip.
- Must use over the counter NiCad batteries. Battery pack weight must not exceed 2.5 lb. Each aircraft will carry a removable 7.5 pound steel payload. The payload may be segmented into no more than 3 pieces, each of which must be rectangular in shape. (Wedges, cylinders or other "sculpted" shapes are not allowed).
- Aircraft and pilot must be AMA legal. This means that the aircraft TOGW (take-off gross weight with payload) must be less than 55 lb., and the pilot must provide documentation of prior RC experience and be a member of the AMA.
- Aircraft will then fly as many complete laps as possible over the specified course. The course will consist of two 180 degree turns at least 700 feet apart. (Turn spotters will be located 200 ft from either end of the take-off/landing zone.) On the downwind leg of the first lap the aircraft will make a level 360 degree turn to the right and a level 360 degree turn to the left. Both turns must be initiated after passing the upwind spotter, and be completed before passing the downwind spotter.
- Flight altitude must be sufficient for safe terrain clearance and low enough to maintain good visual contact with the aircraft. Decisions on safe flight altitude will be at the discretion of the flight line judges and all rulings will be final.
- Total flight time must be at least 3 minutes. No components may be dropped from the aircraft at any time during the flight. Upon landing, the aircraft must be capable of a second flight with no repairs or service other than recharging the batteries, and possible replacement of the propeller(s). Partial laps do not count.

Electrical propulsion simulation software (called Electricalc) was used to see how changing one component of the power system at a time effected efficiency, run time, and aircraft performance.

Variables included:

number of cells
size of cells
motor type and size
prop diameter
prop pitch
prop manufacturer
gear ratio

Aerodynamic analysis - Fluent Analysis
NUMERICAL ANALYSIS

The numerical analysis for this model was carried out using the Fluent software package from Fluent Inc. in Lebanon, New Hampshire. This package was chosen for its utilization of body fitted coordinates, good integration with CAD packages and good solution capabilities. The criteria for modeling this airplane was to set up a symmetric grid about the centerline of the airplane and model a half of it for reduction of computer time. The flow was then assumed to be incompressible due to the fact that the velocity of the airplane is fairly low, that is, about 12 m/s. The Reynolds Number of the flow over the wing was found by utilizing:

$$\mathrm{Re} = \frac{\rho\, U_{\infty} C}{\upsilon}$$

The resulting Reynolds Number was on the order 3×10^{5} which was still inside a laminar flow regime, thus no turbulence was modeled.

The process of implementing this solution system started by transferring the file from the CAD package into a meshing suite called Geomesh by ICEM CFD and Fluent. This program took the CAD drawing and set up a body fitted coordinate system on the airframe. The grid created maintained its parameters within safe values for the cells to be computationally sound. This grid had approximately 30000 cells. This grid was then exported to Fluent. The boundary conditions were to set the inlet at the front of the plane, a plane of symmetry across the centerline of the airframe, and pressure boundaries on all remaining exterior faces. The physical constants were set for air at slightly above sea level conditions ($P = 1\times10^{5}$Pa, $r = 1.2$kg/m^3, $n = 1.7\times10^{-5}$). The density was computed to allow limited compressibility effects. The boundary layer was not modeled as it would require a much smaller grid which would greatly complicate computation.

There were several different cases computed where the main varying factor was that the angle of attack was changed from -1° to 10° at a given interval. In the end it was assumed that an angle of attack of 2° was a fair estimate of regular flight and an angle of attack of 10° would be the takeoff condition.

The method of solution involved Fluent solving the finite difference forms of the continuity and momentum equations for a given grid cell. The SIMPLE algorithm was used to allow solution of a pressure field since there is no pressure term in the continuity equation. The equations were discretized according to standard convention and the Rhie and Chow scheme was used to eliminate the need for staggering the computational grid. For the solver a multigrid method was used to speed convergence, a convergence criterion of 10^{-3} order of reduction in normalized residuals was used to attain an accurate solution. Once solutions were found the data was analyzed using the EnSight post processing package. The figures generated in this report were generated from this package.

Included in this report are only the 2° and the 10° results. The results for pressure show that at 2° (Fig. A.1) that the pressure is fairly uniform over the body with the expected low pressure over the wing that generates lift. However, looking at the 10° case (Fig. A.2) we see that the pressure is reduced on top of the wing further and there is the addition of a high pressure at the leading edge of the underside of the wing. This creates more lift than the first case, however, more drag is also created. From these results it was found that while the drag would be high at takeoff with our airframe, the drag in level flight would be fairly low (noted by the lack of high pressure zones in Fig. A.1).

The next set of figures shows the airstream path over the fuselage and wing. These streaklines represent the path that a fluid particle would take over the body, the color coding shows velocity. In Figure A.3 it can be seen that the streaklines are fairly uniform over the airframe which implies low drag. It can also be seen by looking at the area behind the wing that there are few disturbances created. However in the 10° case we see that the flow over the wing area is very curved and upon inspection vortices are created at the tips of the wing. There is significantly more form drag in this case.

It seems that the numerical model gives a fair idea of the aircraft performance in flight and validates that the design will make a good airframe. It also allows optimization of the flight angles and allows the achievement of better flight characteristics by the operator since they understand the flow characteristics around the plane.

Detailed Design - Final Design, Drawings and Performance Predictions

The drag forces for the airplane were obtained by using Electricalc to find the drag coefficient of 0.034. From the drag coefficient the drag force was calculated. The equation for finding the drag force is $F_D=1/2\ C_D\ \rho\ V_0^2\ A_p$. At an angle of attack of zero and a velocity of 25 mph the drag force is 1.31lb. Other performance data was obtained using Electricalc. The climb rate is 239 ft/min at an angle of 10°. The thrust was calculated to be 55 oz. A stall speed of 16mph was obtained and a 29 mph maximum level flight speed. The motor has an efficiency of 91% and the system efficiency is 84%. The g loading for the airplane was supplied by the wing manufacture to be +8 and -6. The endurance limit is 9.6 minutes at full power. The endurance limit is for a simulated case using a power supply that will not be used for competition. To range will be found by flight testing the aircraft. The payload is 42% of the gross weight of the aircraft.

Design Summary

The following data was calculated from electricalc (a software package that we bought) to find the take off performance of the plane.

climb rate	239 ft/min
climb angle	10 °
thrust	55 oz.
stall speed	16 mph
max. speed	29 mph (level flight)
motor efficiency	91%
system efficiency	84%
endurance	9.6 min. continuous run time at 100% throttle
g loading	+8 / -6
range	not enough data until flight testing begins

Airplane Breakdown
Weight per Component

COMPONENT	WEIGHT (oz)
Wing	56
Horizontal Tail	4
Vertical Tail	5
Fuselage	27
Motor	17
Tail Booms	5
Batteries	40
Gear	6
Avionics	8
Payload	120
Gross Weight	288

Wing Analysis

Wing Loading	20.9 oz/ft^2
Aspect Ratio	20
Mean Aerodynamic Chord	9.25in
Root Chord	12in
Tip Chord	6.5in
Surface Area	2192.9in^2
Span	192in
Airfoil Section	E-203
Thickness to Chord	1.7in
Angle of Incidence	1°

Manufacturing Plan - Materials Selection and Fabrication Processes

This aircraft project incorporates many different material and manufacturing techniques in order to produce the most cost effective, light weight, and strong components possible.

The wings are constructed from foam cores with a carbon fiber beam spar system, spruce leading and trailing edges and obechi wood wing sheeting (this gives added strength over balsa with a reduction in cost over composites). The wing components were manufactured by Dream Catcher Hobbies. It was necessary to have these components manufactured from an outside source since tooling was not available at this university. Purchasing the required tooling would have greatly increased the cost of the project which was a major factor in the decision to purchase the components rather than build them from scratch.

The fuselage is constructed from composite materials including birch plywood, Styrofoam, fiberglass cloth, carbon fiber matt, spectra foam, and epoxy resin. Major components such as the reduction drive, motor, prop shaft, remote control receiver, electronic motor speed control, tail boom, and wing rod were placed in their positions relative to one another. These components were arranged to allow for the most compact and stream lined fuselage shell. After relative placement was determined an internal frame of plywood and composite materials was constructed to hold all of the components in place. Next detailed measurements were taken to determine the size and shape of the fuselage shell that would be needed. This was built in two pieces, top and bottom. The shape was created by carving and sanding Styrofoam to make a positive mold. This was then cut in to the respective top and bottom sections to form the two molds. The Styrofoam was relieved for the plywood reinforcement that was installed next. Fiberglass cloth was laid over the mold and epoxy resin brushed on. The shell was then heat cured by inserting it in a cardboard box that we heated with a hair dryer. This greatly reduced curing time. Once the shell was cured the foam mold was removed by carving and applying solvents to the inside of the structure. The surface irregularities were then filled and the shell was sanded to a smooth finish. Applying a light coating of enamel based spray paint to the outside surface finished off the fuselage structure.

The tail boom is a carbon fiber tube 43 in. long with 1.17 in. inner diameter and 0.035 in. wall thickness. This tube has a groove milled in the end of it to allow the vertical fin of the tail assembly to slide into it. This allows the tail to be removed from the fuselage / tailboom assembly for shipping. Once the tail boom was prepared it was inserted into the fuselage frame and glued into place.

The tail assembly utilizes a balsa built up structure that is covered with monokote plastic heat shrink covering. This material selection was chosen because of its light weight, ease of construction and low cost. A flying stabilizer was incorporated into the design for increased control response at the relatively low speeds that this aircraft operates at.

Parts cost	
composite materials for fuselage construction	$129.25
wing components	$475.00
carbon fiber tail boom and tubes used in fuselage frame	$70.00
wood for tail structure	$46.25
heat shrink covering, fillers, and paint	$112.00
radio gear and extra mini servos	$614.50
hardware	$76.50
adhesives	$43.00
motor and electronic speed control	$879.00
batteries	not purchased at this time
battery charger	not purchased at this time

Skill matrix

BUILDING TECHNIQUES	SKILL REQUIRED
fuselage construction techniques	
composite lamination through	
mold layup	9
foam core hollowed out for	
components and covered with	
fiberglass	5
traditional all wood construction	6
wing construction techniques	
foam core wings with wood sheeting	7
foam core wings with composite sheeting	10
traditional all wood construction	5
tail construction techniques	
traditional all wood construction	2
foam cores with wood sheeting	5
foam cores with composite sheeting	7

Scheduling time line

See Management summary

Manufacturing milestone chart

week of:	Feb.23	Mar.2	Mar.9	Mar.16	Mar.23	Mar.30	Apr.6	Apr.13	Apr.20
acquire wing components									
acquire power plant components									
acquire radio gear									
acquire tail boom									
acquire hardware									
acquire composite materials for fuselage									
define spatial relationships for components									
build fuselage frame									
construct fuselage shell									
mill tail boom									
construct tail assembly									
finish wing components									
install landing gear									
install radio gear									
covering and finishing									

References Cited

Jeppesen Sanderson Inc., 1992. Private Pilot Manual, Englewood, CO

McCormick, Barnes W. 1979. Aerodynamics, Aeronautics, & Flight Mechanics, John Wiley & Sons, Inc.

Parametric Technology Corp., 1994. Pro/Engineer User's Guide, Waltham, MA

Fluent Inc., 1995. Fluent User's Guide, Lebannon New Hampshire

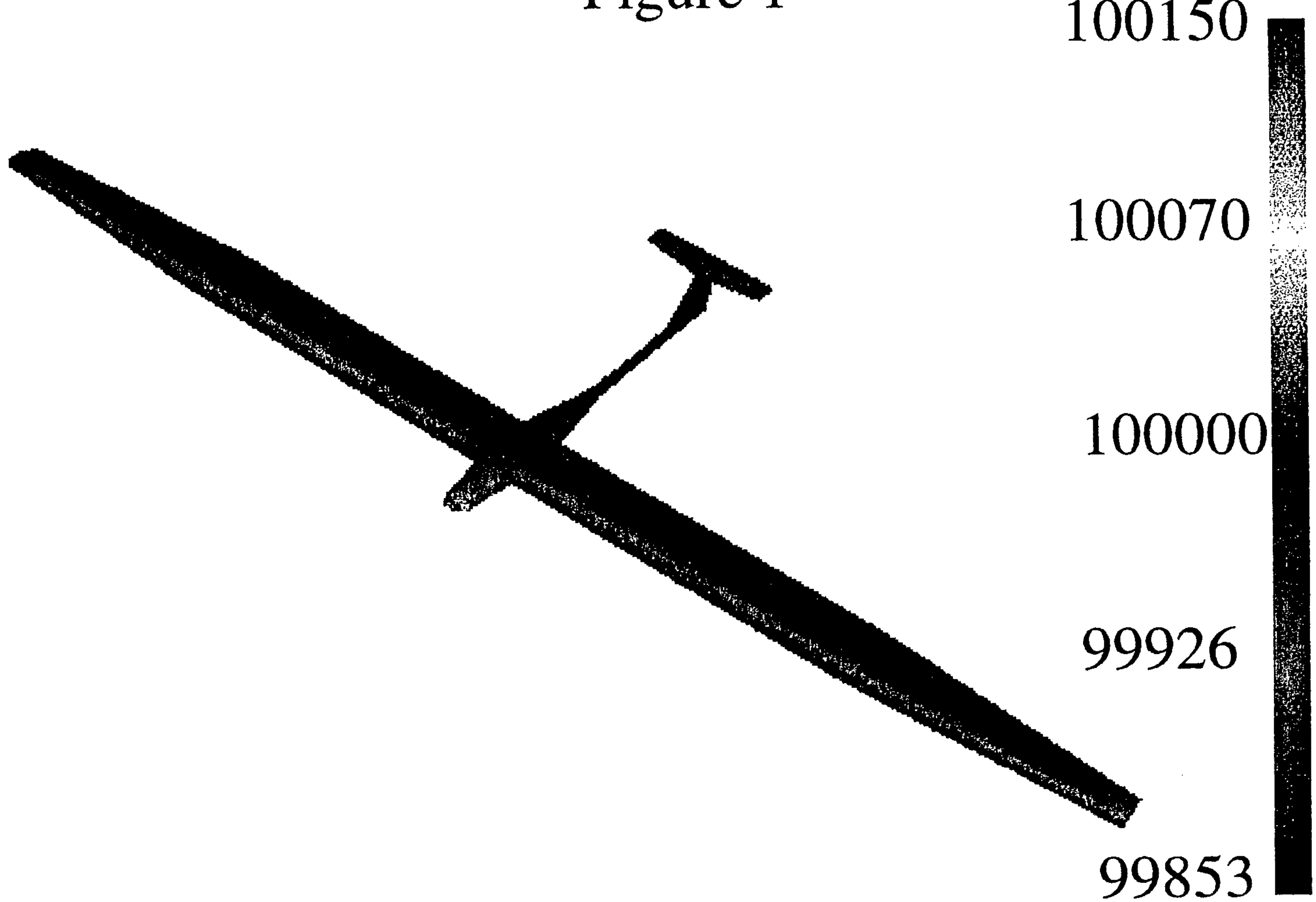
Hoerner, Sighard F. 1958. Fluid-Dynamic Drag, Great Britain

Bertin, John J., Smith, Michael L. 1989. Aerodynamics for Engineers, University of Texas at Austin, United States Air Force Academy

Appendix A

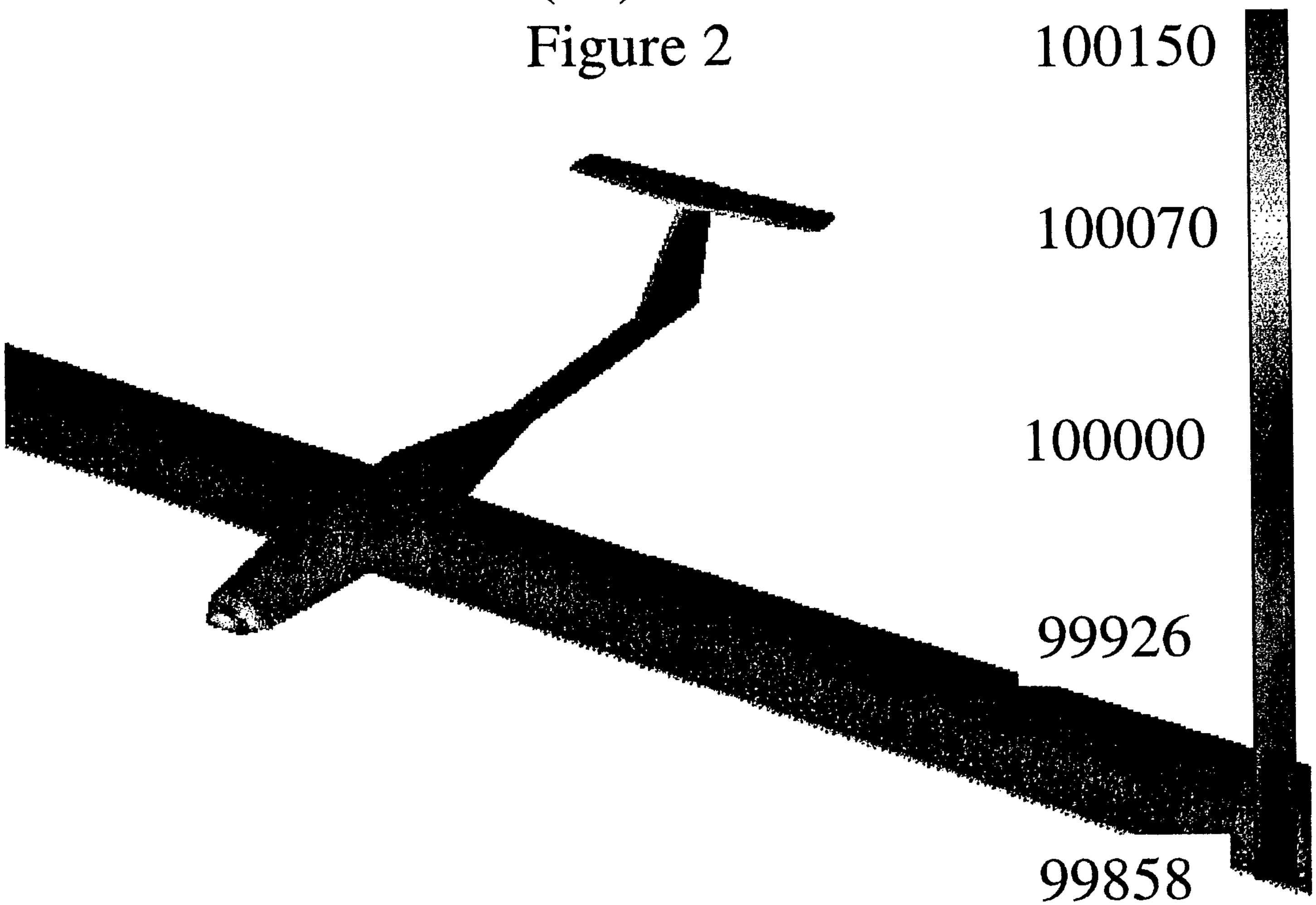
Fluent Analysis Drawings

Static Pressure (Pa) in Level Flight
Figure 1

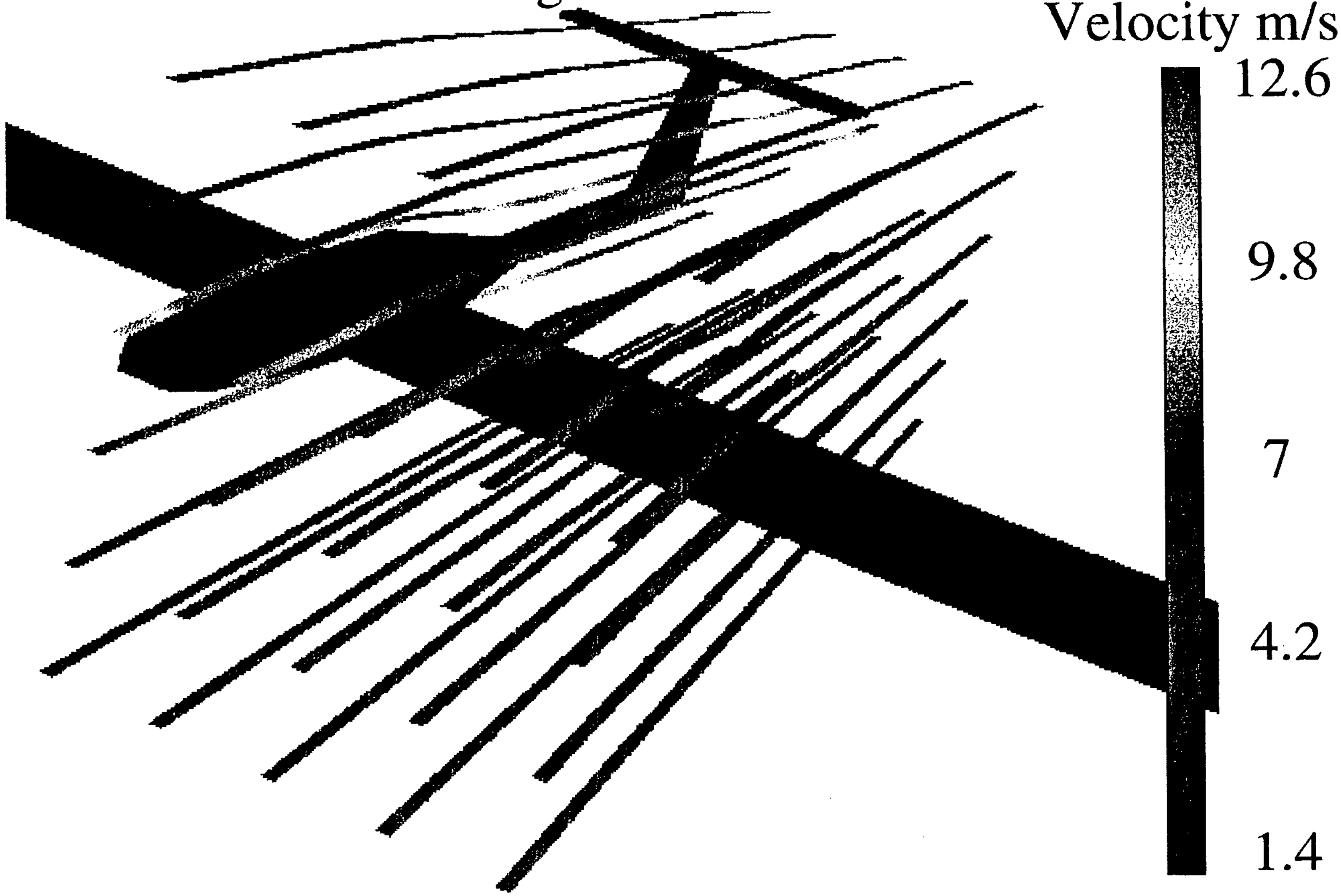


Static Pressure (Pa) at Maximum AOA 10°

Figure 2

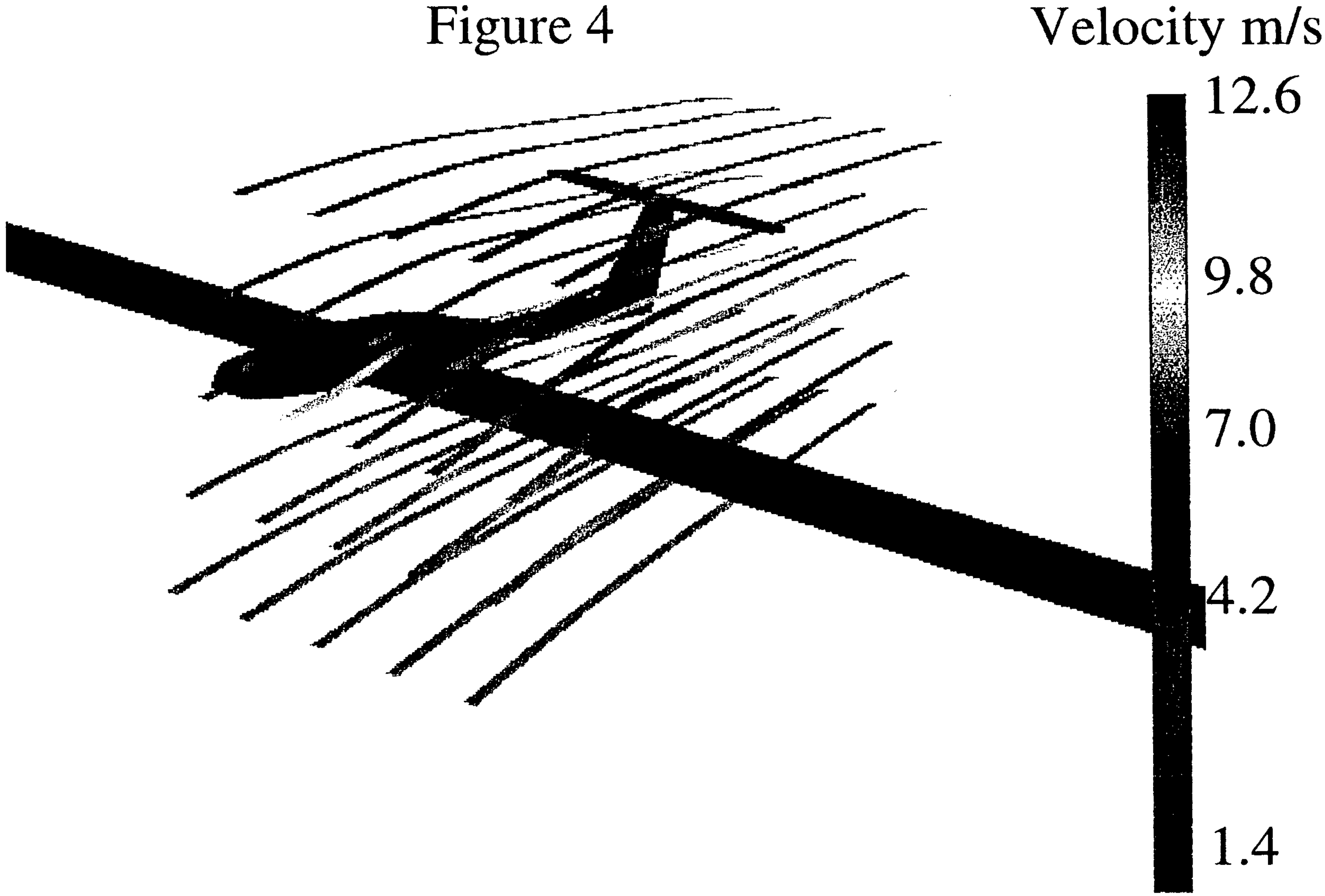


Streamlines Over Fuselage in Level Flight
Figure 3



Streamlines Over Fuselage at 10° AOA

Figure 4



Appendix B

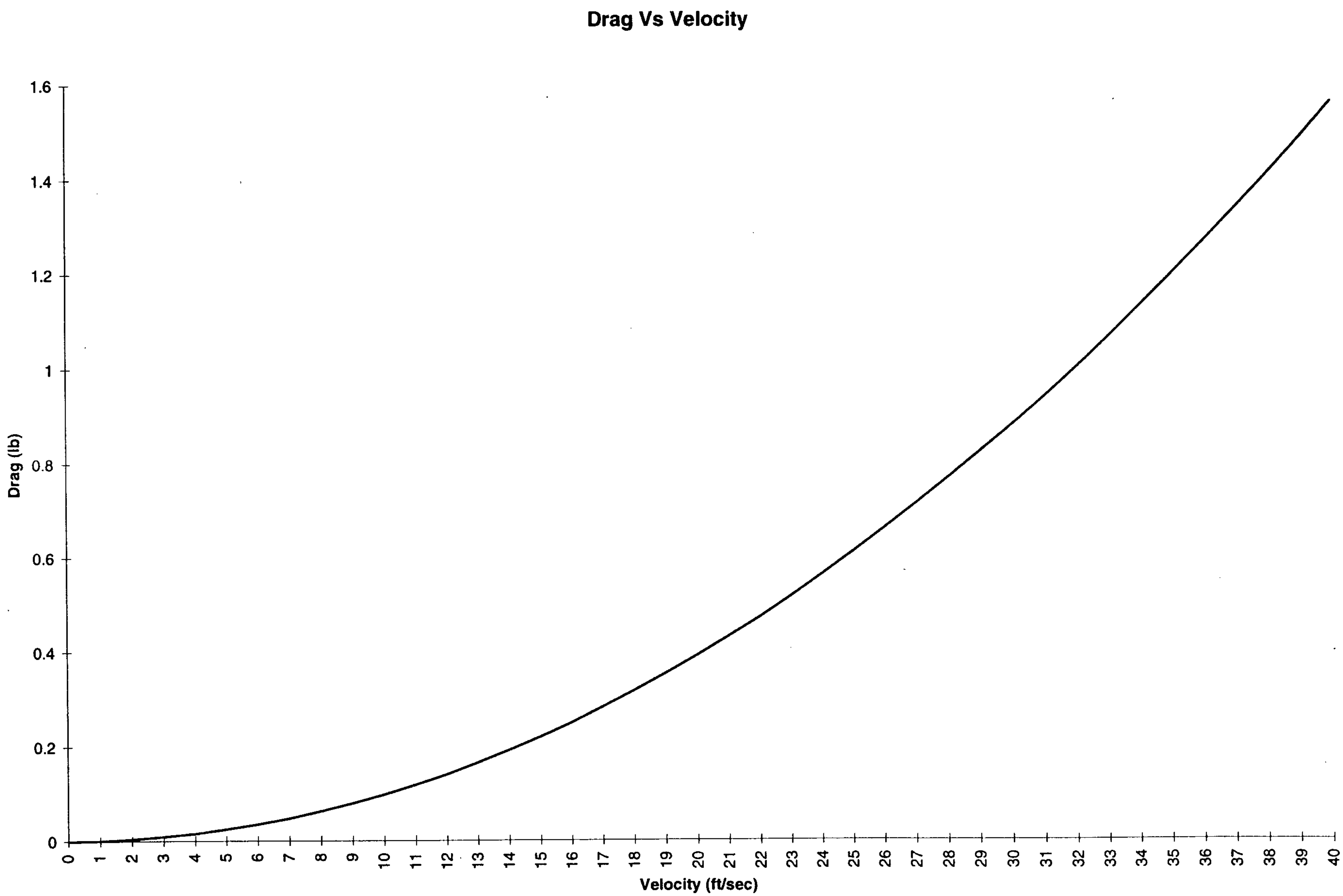
Calculations of the Drag Forces of the Airplane

Calculations of the Drag Forces of the Airplane

Drag Forces for Varied Angle of Attack

angle	0			5			10	
	Ap(in^2)	Ap(ft^2)						
wing	236	19.6666667		372.697	31.0581		516.397	43.0331
tail	14.25	1.1875		13.1315	1.09429		22.1632	1.84693
fuselage	18.06	1.505		20.0422	1.67019		25.0081	2.08401
total	268.31	22.3591667			33.8226			46.964
C_D	0.034							
V_o	25 MPH	36.666	ft/sec					
ρ	0.00257	Slugs/ft sec2						
F_D (lb)	1.3133024				1.98662			2.75851

F_D	V_o	
0	0	0
0.000976	1	0.000976
0.003906	2	0.003906
0.008788	3	0.008788
0.015624	4	0.015624
0.024412	5	0.024412
0.035153	6	0.035153
0.047847	7	0.047847
0.062494	8	0.062494
0.079094	9	0.079094
0.097647	10	0.097647
0.118153	11	0.118153
0.140612	12	0.140612
0.165024	13	0.165024
0.191388	14	0.191388
0.219706	15	0.219706
0.249977	16	0.249977
0.2822	17	0.2822
0.316377	18	0.316377
0.352506	19	0.352506
0.390589	20	0.390589
0.430624	21	0.430624
0.472612	22	0.472612
0.516553	23	0.516553
0.562448	24	0.562448
0.610295	25	0.610295
0.660095	26	0.660095
0.711848	27	0.711848
0.765554	28	0.765554
0.821213	29	0.821213
0.878824	30	0.878824
0.938389	31	0.938389
0.999907	32	0.999907
1.063377	33	1.063377
1.128801	34	1.128801
1.196178	35	1.196178
1.265507	36	1.265507
1.336789	37	1.336789
1.410025	38	1.410025
1.485213	39	1.485213
1.562354	40	1.562354



Drag Vs Velocity
Drag (lb)
1.6
1.4
1.2
1
0.8
0.6
0.4
0.2
0
0 1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19 20 21 22 23 24 25 26 27 28 29 30 31 32 33 34 35 36 37 38 39 40
Velocity (ft/sec)

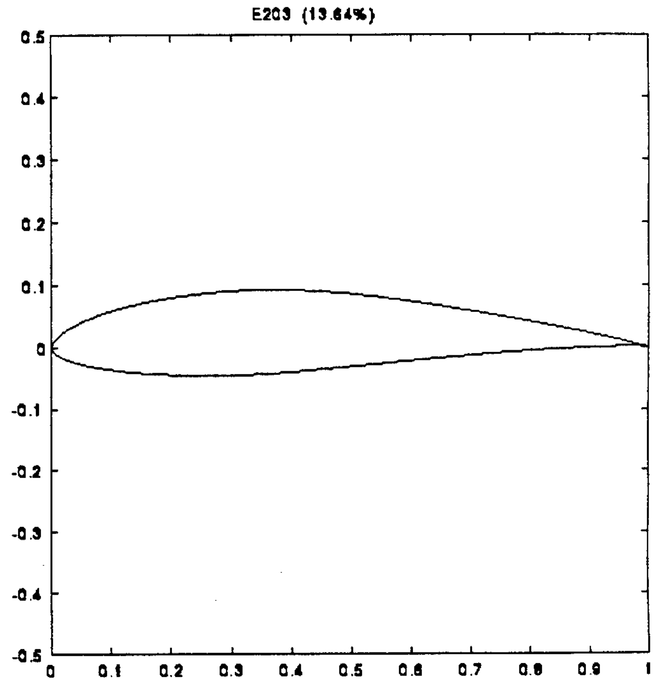
Appendix C

Data for the Eppler (E-203) low Reynolds number airfoil

Data for the Eppler E203 low Reynolds number airfoil

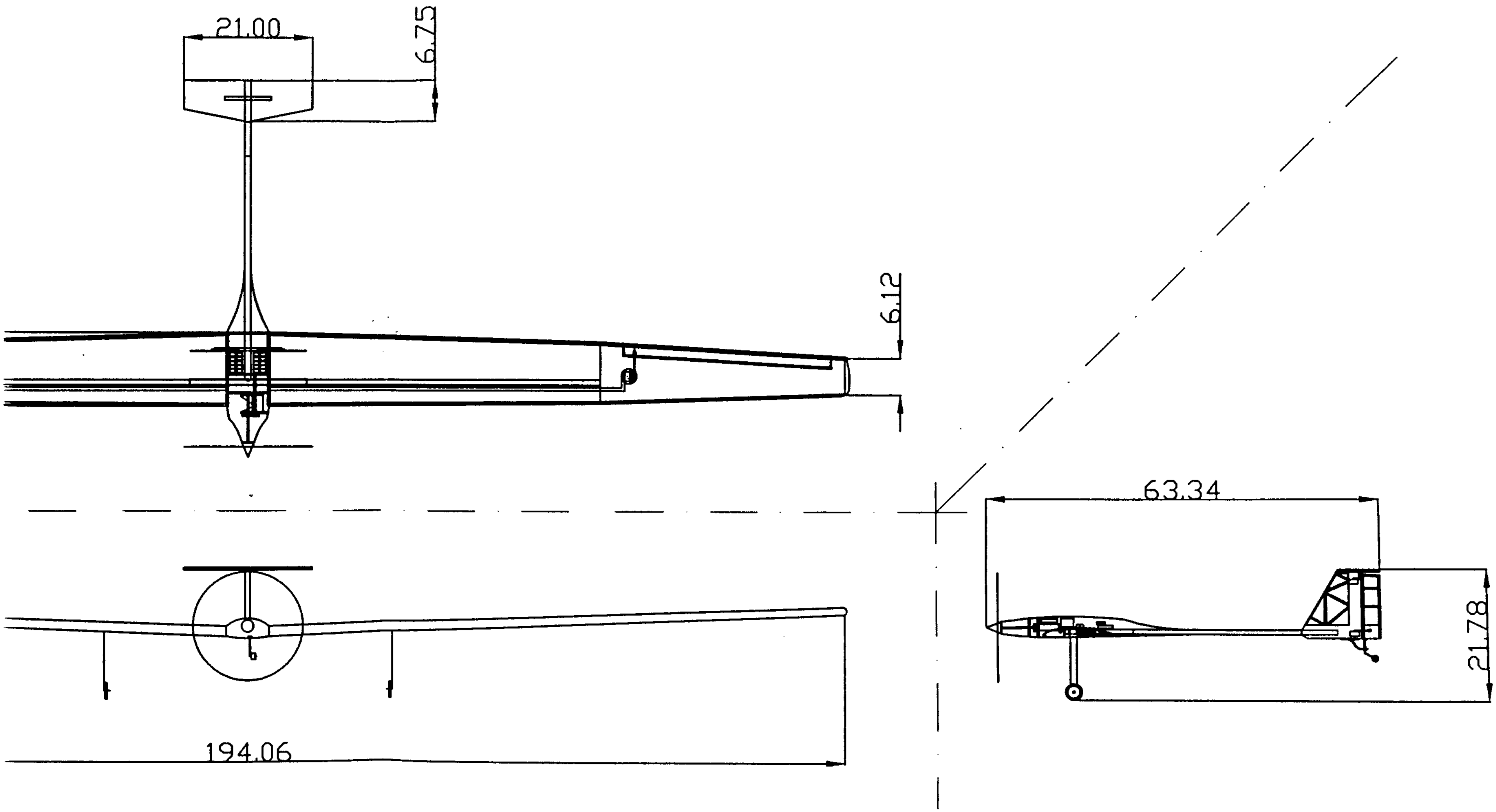
E203 (13.64%)

1.00000	0.00000
0.99645	0.00066
0.98625	0.00282
0.97030	0.00660
0.94932	0.01158
0.92361	0.01723
0.89325	0.02333
0.85854	0.02995
0.82000	0.03705
0.77819	0.04450
0.73369	0.05217
0.68712	0.05988
0.63910	0.06741
0.59028	0.07452
0.54130	0.08087
0.49264	0.08604
0.44460	0.08973
0.39748	0.09175
0.35154	0.09203
0.30704	0.09055
0.26419	0.08742
0.22332	0.08289
0.18486	0.07717
0.14920	0.07039
0.11671	0.06271
0.08772	0.05430
0.06250	0.04532
0.04128	0.03597
0.02427	0.02648
0.01161	0.01710
0.00344	0.00823
0.00002	0.00051
0.00258	-0.00625
0.01115	-0.01296
0.02471	-0.01953
0.04311	-0.02564
0.06616	-0.03113
0.09366	-0.03588
0.12534	-0.03979
0.16091	-0.04281
0.19999	-0.04489
0.24219	-0.04597
0.28702	-0.04602
0.33398	-0.04483
0.38286	-0.04210
0.43379	-0.03799
0.48643	-0.03314
0.54007	-0.02801
0.59395	-0.02289
0.64730	-0.01799
0.69937	-0.01350
0.74940	-0.00954
0.79665	-0.00620
0.84043	-0.00353
0.88005	-0.00156
0.91489	-0.00024
0.94441	0.00051
0.96813	0.00076
0.98561	0.00060
0.99636	0.00021
1.00000	0.00000

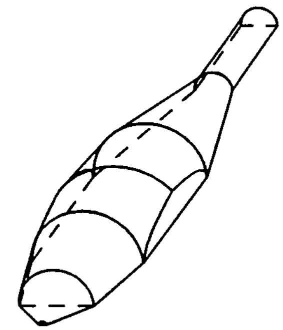


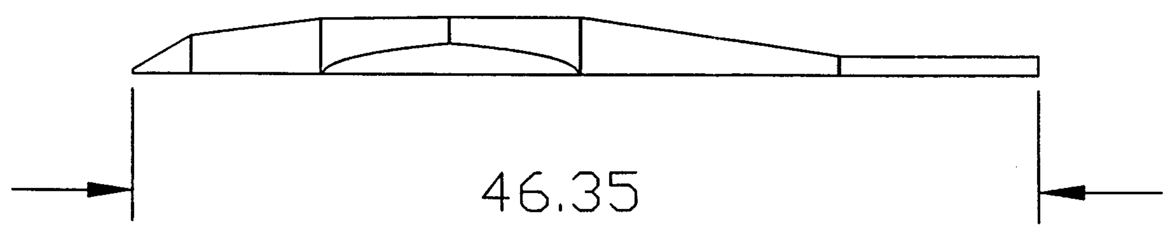
Appendix D

Fabrication Drawings



WASHINGTON STATE UNIVERSITY Mechanical Engineering Dept. WSU AIAA Student Chapter		AIAA UAV Project
UNLESS OTHERWISE SPECIFIED Dimensions are in Inches Break All Corners 0.010R All Fillets 0.015R Do Not Scale Drawing All Dim/Tol Per ANSI Y14.5	DRAWN David Darrow CHECKED ENGR APPROVED MATERIAL	Part Overall View of the PLANE SIZE B DRAWING NO. WSU - 007 REVISION 2 SCALE: SHEET: 1 OF 1

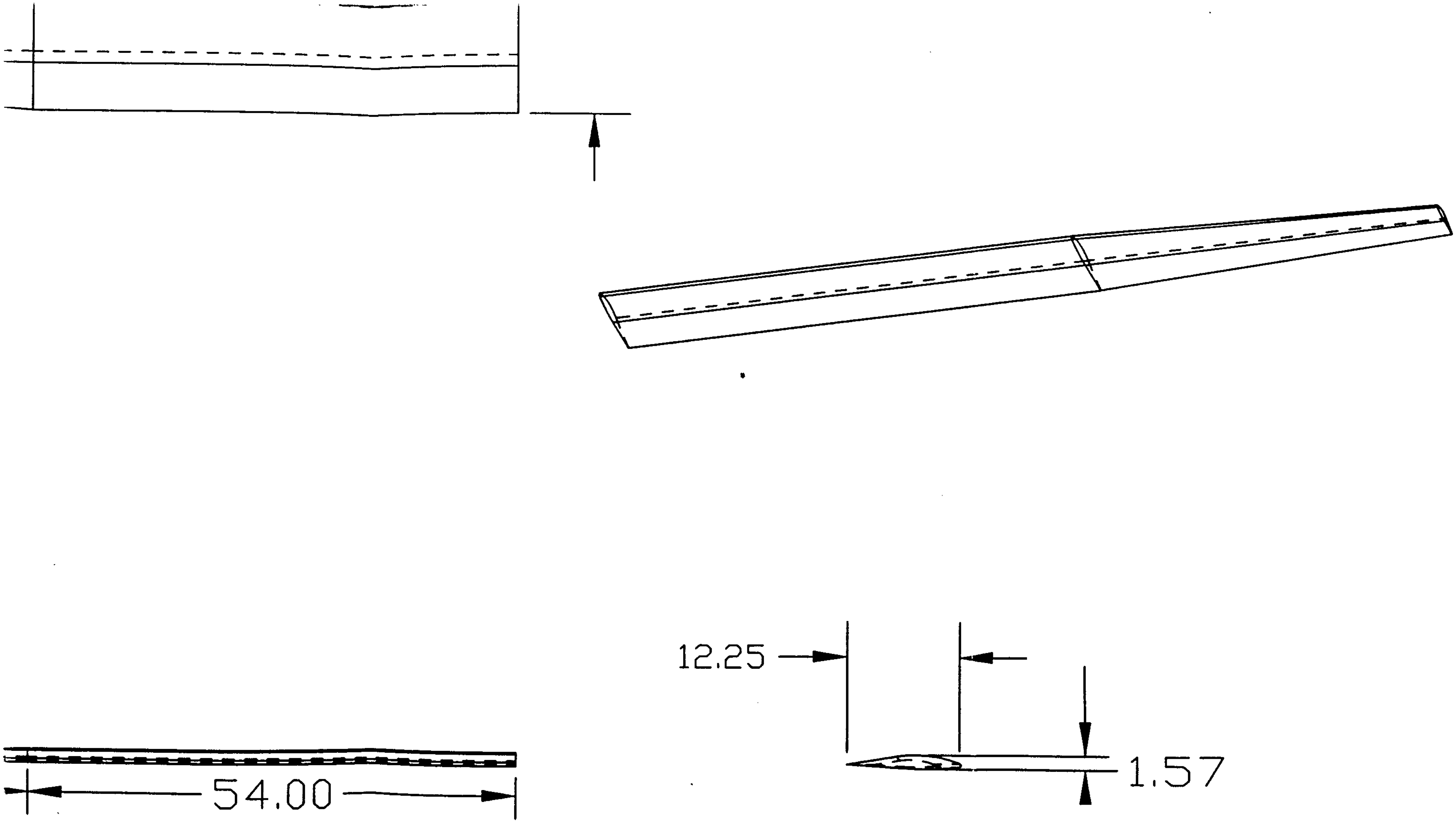




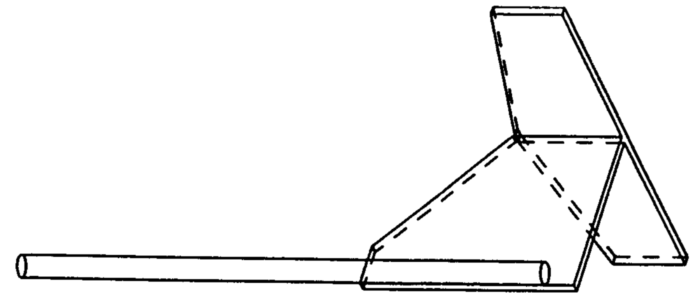
.00

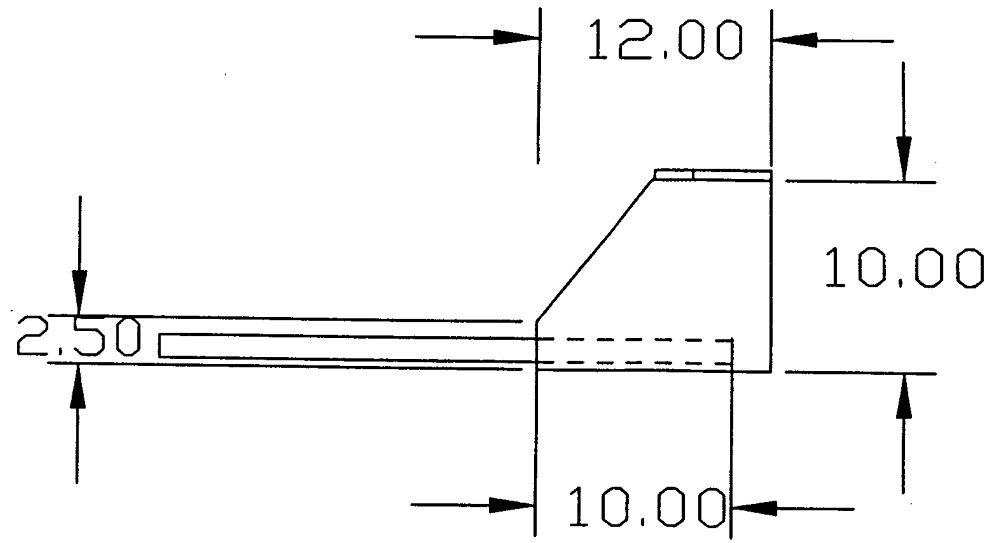
0

WASHINGTON STATE UNIVERSITY Mechanical Engineering Dept. WSU AIAA Student Chapter		AIAA UAV Project	
UNLESS OTHERWISE SPECIFIED Dimensions are in inches Break All Corners 0.010R All Fillets 0.015R Do Not Scale Drawing All Dim/Tol Per ANSI Y14.5	DRAWN Suresh CHECKED ENGR APPROVED MATERIAL	Part Fuselage Dimension	
		SIZE B · DRAWING NO. WSU - 005 · REVISION 2	
		SCALE:	SHEET: 1 OF 1



WASHINGTON STATE UNIVERSITY Mechanical Engineering Dept. WSU AIAA Student Chapter		AIAA UAV Project
UNLESS OTHERWISE SPECIFIED Dimensions are in inches Break All Corners 0.010R All Fillets 0.015R Do Not Scale Drawing All Dim/Tol Per ANSI Y14.5	DRAWN Suresh CHECKED ENGR APPROVED MATERIAL	Part Wing Panel SIZE B · DRAWING NO. WSU - 004 · REVISION 2 SCALE: · SHEET: 1 OF 1





WASHINGTON STATE UNIVERSITY Mechanical Engineering Dept. WSU AIAA Student Chapter		AIAA UAV Project
UNLESS OTHERWISE SPECIFIED Dimensions are in inches Break All Corners 0.010R All Fillets 0.015R Do Not Scale Drawing All Dim/Tol Per ANSI Y14.5	DRAWN Suresh CHECKED ENGR APPROVED MATERIAL	Part Tail Dimension SIZE B · DRAWING NO. WSU - 006 · REVISION 2 SCALE: · SHEET: 1 OF 1

Addendum Report from

Utah State University

for

the

AIAA Design/Build Fly Competition

Addendum

Changes Made in the Design

FUSELAGE: The fuselage will not be covered entirely with foam and then sculpted to an aerodynamic profile as originally proposed. Foam will be used as a transition from the front fuselage to the main fuselage section, but will not cover the entire length of the fuselage. The corners of the balsa wood structure will be sanded to a rounded shape, and monokote will then be stretched around the balsa wood framework. The steel will also be slightly longer than the original design calls for; a piece of scrap steel was found and was a slightly smaller cross-section than originally planned. This piece of scrap steel was used anyway, since it was very inexpensive. The overall fuselage dimensions remain unchanged. Foam will be used to fill the internal void space around the steel to fit it within the current sized fuselage.

WING DESIGN: Originally, the wing was going to be one solid piece. This plan was intended to avoid the weight of additional mounting which would be required if the wing were multi-sectional. Additionally, a one piece wing would be stronger than a multi-sectional one. However, the cost of shipping such a wing would exceed $800.00! For practicality, and for budget restraints, we built the wing in two pieces. This allows the wings to be packaged and taken with us on the airplane.

WING MOUNTING: The wing-fuselage interface has also changed. Originally, the wing was to be mounted by a plate with a dowel running through it at the wing spar, and a second plate near the trailing edge with two nylon bolts through it. The frontal plate (previously vertical) will be replaced with a horizontal plate identical to the one found on the trailing edge end. This too will secure the wing using two nylon bolts.

WING POSITION: The original position of the wing was largely determined from hand calculated approximations. During the last stages of design, an error was found in the computer program "airplane", which was used to stabilize the airplane. This made hand calculations necessary. Since the time that the proposal report was submitted, the error has been fixed, and more accurate calculations have become available. To stabilize the airplane, the wing has been relocated 3 inches forward of the originally determined position.

MOTOR SELECTION: The Astroflight FAI-05 motor was originally selected to power the plane because it is more efficient than its bigger brother, the FAI-15. However, in talking with Bob Boucher, Astroflight President, we have since decided to use the larger of the two motors.

Bob recommended using the FAI-25 or FAI-40 because they are rated for a higher power than the smaller motors. It is during takeoff that the highest power is drawn. During level flight, a lower power is used which is within the limits of even the smaller motors.

Since the larger motors are less efficient during level flight, the smallest capable motor should be used. Although takeoff exceeds the FAI-15 rated power by 25 Watts (the FAI-15 is rated at 375 Watts), it should not harm the motor to run it at this level for a short period of time. The FAI-05 is rated for only 325 Watts. The FAI-15 also provides a larger margin of safety in power available than the 05, and will reduce the risk of crashing.

BATTERY CONFIGURATION: Because of the change in the motor selection, it was necessary to alter the battery configuration. Instead of configuring the 16 cells 2x8 (2 parallel packs of 8 in series), we have chosen to configure all 16 in series. The FAI-15 motor is rated for 10 cells, and will run more efficiently in this manner.

Suggested Improvements

FUSELAGE: The main fuselage section, just beyond the steel payload, is oversized. The structure near the payload is required, but towards the attachment of the trailing edge of the wing, the fuselage could be tapered up to a point where the aluminum tube (going out to the tail) attaches. This would require approximately 10 hours of design and analysis time, and add approximately 2-4 hours to the construction time. Little variation in cost would be seen.

Alternately, a completely different fuselage design could be used. Ours is a square fuselage and an aluminum rod attaching the tail. The square fuselage creates additional drag. And the method of attaching the tail is an amateur one. A round fuselage would be aerodynamically cleaner and look nicer. There is, however, the additional challenge of attaching the landing gear and wing. The process of designing an alternate fuselage would require approximately 30 hours for someone who is already familiar with I-DEAS or a similar structure analysis program, and much longer for someone who is not. Also, a knowledge of materials available would be required. This design is not expected to cost more than the current one.

BATTERIES: The battery pack we have chosen provides 40 Amp hours of energy, which is the best we could find. There is the possibility of assembling a battery pack of 20 cells with *lower* energy, which would allow the speed controller to operate more efficiently. The cells would be assembled 2x10. It is unknown whether this higher efficiency would compensate for the lower energy provided.

Changes in Cost

The predicted costs were quite accurate for most items, since most items had been previously priced. One way in which the predicted cost differed from the actual is unpredicted materials. This includes the amount of glue needed. Glue costs between \$6-\$10 per container. We used four bottles totaling \$32. Also, we will buy and test several propellers. These have not yet been bought, and an exact price is not yet known.

A more accurate report of cost changes are not available at this time because the plane is still under construction, and materials are still being bought.

Table of Contents

1.0 Executive Summary 1

2.0 Management Summary 3

3.0 Conceptual Design 5

- 3.1 Batteries
- 3.2 Motor, Propeller & Speed Controller
  - 3.2a Alternative Concepts Investigated
  - 3.2b Analytical Methods Used for Rating FOMs
  - 3.2c Final Conceptual Design Configuration
- 3.3 Airplane Configuration
- 3.4 Fuselage

4.0 Preliminary Design 8

- 4.1 Optimum Flying Speed
- 4.2 Motor/Propeller
  - 4.2.1 Comparing the FAI-05 and FAI-15 Motors
- 4.3 Batteries
- 4.4 Airfoil & Wing Structure
- 4.5 Fuselage
- 4.6 Airplane Stability (Control Surfaces)
- 4.7 Wing Mounting
- 4.8 Landing Gear

5.0 Detail Design 15

- 5.1 Aircraft Details and Drawing
  - 5.1a Wing & Airfoil
  - 5.1b Fuselage and Landing Gear
  - 5.1c Tail
- 5.2 Aircraft Performance
  - 5.2a Take Off Performance
  - 5.2b Handling Qualities
  - 5.2c G Load Capability
  - 5.2d Payload Fraction
  - 5.2e Range and Endurance

6.0 Manufacturing Plan 19

Appendix A Aerodynamic Efficiency 21

Appendix B Battery Options 25

Appendix C Equations for Airplane Stabilization 26

Appendix D Motor Trade-Offs 34

Appendix E Airfoil Diagrams 43

Appendix F Fuselage Layout 46

1.0 Executive Summary

After reviewing the contest objectives and rules, we decided on our design objectives. These were the characteristics of the plane which we thought made the most difference, and which we knew we could design well. Our design objectives are listed below:

- Determine Optimal Flying Speed
- Select Airfoil Shape
- Design, Analyze, and Construct Wing
- Select Battery Size and configuration
- Determine Optimal Airplane configuration
- Design, Analyze, and Construct Fuselage
- Design and Analyze Wing Mounting
- Optimize Landing Gear Design

During the conceptual design, we investigated design alternatives, and listed possible advantages and disadvantages of each characteristic. Later, during the preliminary design, we determined the importance of each advantage and disadvantage. During the critical design, we calculated the actual values of each design parameter. These design objectives will be discussed below.

The optimal flying speed is the speed which requires the least energy per distance. To determine the optimal flying speed, we had to consider the performance of each component of the plane—the aerodynamic, motor, propeller, and speed controller efficiencies.

The desirable characteristics of an airfoil are high lift and low drag coefficients, and to maintain these coefficients over a wide range of speeds. Several airfoils for model planes were considered, and we found one which works well.

There were several options for the wing structure. During the conceptual design stage, we brain stormed over the possibilities, and narrowed our selection based on the attributes of each. The options which we considered were: the conventional balsa frame wing, a solid

Styrofoam wing, and Styrofoam wing with diamond shaped holes cut out to reduce weight, a Styrofoam wing with a reinforcement bar at the root, a fiber composite wing, and a Styrofoam wing with spars.

There are several batteries available. The primary characteristic was the battery pack capacity. Secondary characteristics were motor compatibility, battery internal resistance, total pack volume, cost, and the ability to provide the needed current.

The motor and propeller need to provide enough thrust, yet be as light weight as possible. In order to analyze the motor, we needed detailed characteristic of the motor. We found that Astroflight provided this needed information, but were not able to find this from other vendors. Therefor, we limited our selection to Astroflight motors. There are two types of Astroflight motors: Cobalt Sports motors, and FAI Competition motors. Many sizes are available in each type. For each motor is given an optimum propeller size, and this is the one we chose.

We considered two options for the airplane configuration: a wing/tail, and a wing/canard configuration. For the horizontal stabilizers, we looked at both symmetric and non-symmetric airfoils. We considered two options as well for the fuselage design: both a circular and a square structure.

There are three wing mounting positions: high, mid, and low wing. Each of these requires different mounting methods. Three methods which we investigated are: rubber band mounting, nylon bolt mounting, and fuselage hatch mounting.

We considered several landing gear options. We thought about buying off the shelf shocks, making our own shocks, using piano wire, and using aluminum sheeting. We also looked at using tricycle gear vs a tail dragger.

We used some commercially available software packages to aid us in this design: I-DEAS, MathCad, and AutoCad. We also used an airplane stability program available here at Utah State University called *Airplane*. We wrote a FORTRAN program to analyze the takeoff of our plane. And we used a wind tunnel to test the motor and propeller.

2.0 Management Summary

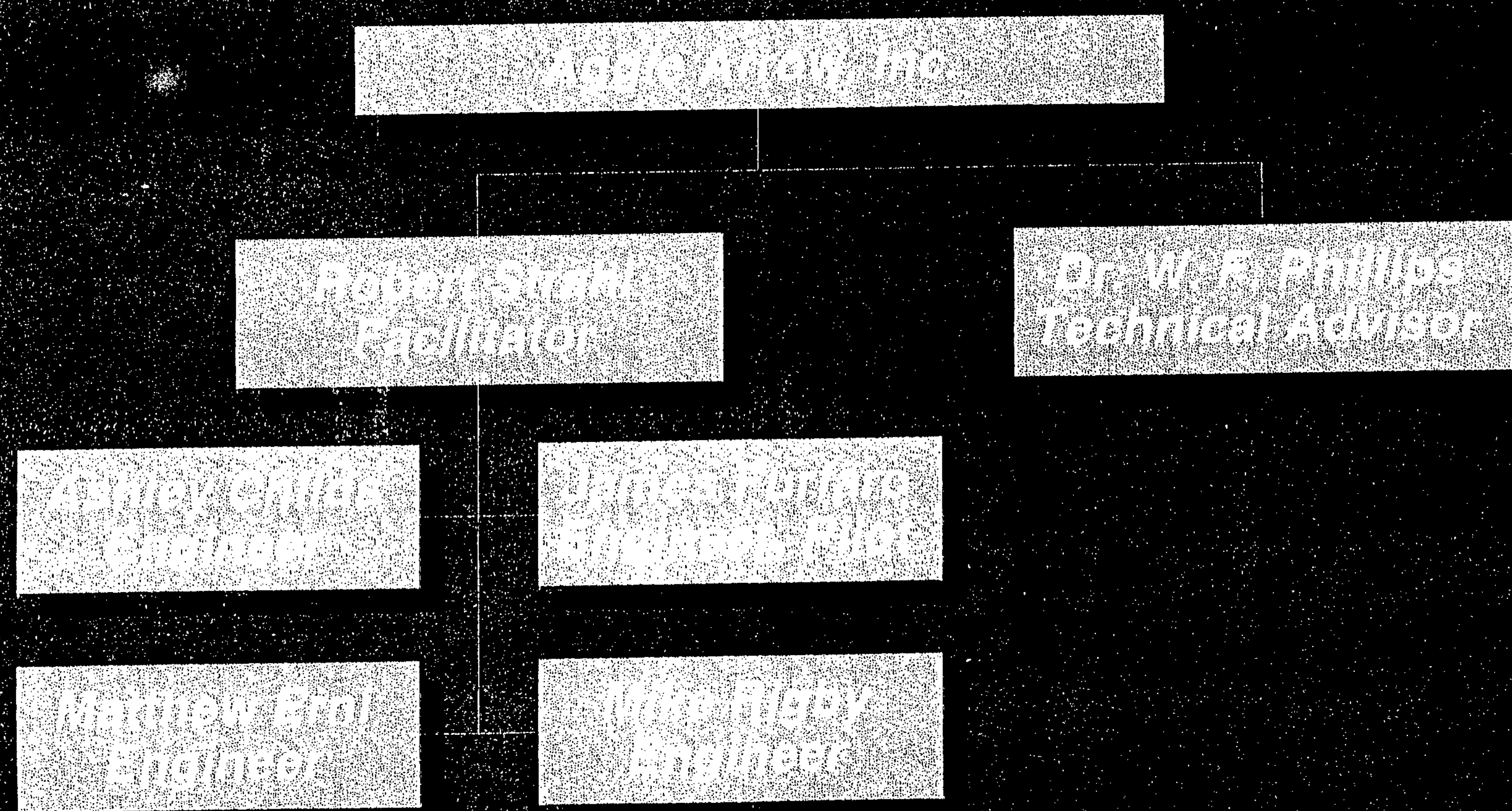
The names of our team members are shown in Figure 2.1. Robert Strahl managed the team, arranged meetings, wrote weekly reports and goals, managed the budget, and purchased the major components of the plane. His major design assignment was the battery selection and configuration. Ashley Childs was assigned to design and analyze the fuselage and landing gear. Matthew Erni was assigned to design the horizontal and vertical stabilizers, including airplane configuration; and to stabilize the plane. Mike Rigby was assigned to select the motor and propeller, calculate the efficiency of the entire plane and determine the optimum air speed. James Furfaro was assigned to select the airfoil shape, and to design and analyze the wing. The entire team will be involved in the construction of the aircraft, which begins shortly.

As a team, we set the goals which would help us to complete the project. A timeline of the major tasks is shown below in Figure 2.2. Each week we met with our faculty advisor, Dr. Phillips, to report our progress; he offered any suggestions which he had. Additionally, we met each week as a team to discuss goals and to coordinate our designs. The individual team members met together as needed.

Fig 2.1

Aggie Arrow, Inc.

Organizational Chart



3.0 Conceptual Design

3.1 Batteries

Since the batteries are what powers the plane, they will be discussed first. We obtained battery listings from two companies, B&P Associates, and SR Batteries. We organized the data on a spreadsheet, and determined the number of cells for 2½ pounds; and the total volume, capacity, and cost. We screened the many possibilities for high total capacity, which is our most desired characteristic. This reduced the selection to nine battery types. The final options are shown in Appendix B.

Once a motor is selected, the most compatible battery type will be chosen. This will be discussed in the preliminary design section.

3.2 Motor, Propeller & Speed Controller

Motor, propeller and speed control selection was based on parameters of competing concepts screened through figures of merit based on analytical calculations. This gave us a rough conceptual design on which to start our first integration of plane development.

3.2a Alternative Concepts Investigated

The alternative concepts investigated, and the figures of merit used, are shown in Appendix D. Gearing was an important parameter because it would increase the thrust, allowing the plane to clear the 10 foot barrier, and change the flying speed. The wing area sizing was an important parameter because it effects efficiency and ability to clear the barrier. The motor/propeller sizing was an important parameter because it needed to be sufficient to clear the 10 foot barrier, yet not so big that the plane would fly inefficiently. Finally, it was important to choose an efficient speed controller to minimize wasted consumed energy.

3.2b Analytical Methods Used for Rating FOMs

The analytical methods used for rating the FOMs were:

1) Using the aero-dynamic equations of flight on MathCAD for initial wing area sizing in order to know motor thrust required for steady level flight helping in narrowing motor/propeller selection and determining initial optimal speed.

2) Making motor/propeller power, thrust available, and pitch speed calculations on MathCAD to narrow the pool of feasible motor/propeller sizes capable of steady level flight and for use in the take-off analysis.

3) Take-off analysis using a FORTRAN program which employs a fourth order Runge-Kutta technique to determine the minimum motor/propeller size necessary for clearing the 10 foot barrier at the end of the runway. (Ballpark constants for Cd, e, AR, ρ, and wing area were assumed). Accuracy for all mentioned methods are only limited by the inaccuracy in the assumptions.

3.2c Final Conceptual Design Configuration

According to the FOM charts on competing concepts in appendix D, we chose the geared motor because it provided more thrust to clear the barrier, and a slower speed, which was closer to the optimum. The wing area was chosen initially as 14 ft^2 because it allowed for the majority of the motors which we were considering to clear the barrier and was the most energy efficient wing area for slower speeds. The motor/propeller size were determined to be in the medium range, or the 13 to 20 Volt Astroflight Sport Motors because they provided sufficient thrust to clear the barrier safely and didn't diminish efficiency caused by throttling back excessively. Finally, the HI-Rate Speed controller was chosen because it didn't have high efficiency losses during partial throttle, thus maximizing lap potential.

3.3 Airplane Configuration

We decided to use the conventional wing/tail configuration, rather than the wing/canard. There are advantages to each; the three deciding characteristics are discussed below.

There are two advantages to a tail plane. A tail plane is easier to balance; a canard makes the plane more aerodynamically unstable and harder to balance. The tail plane would also allow for a lighter fuselage because of the mass distributions. With this configuration, we are able to make the portion of the fuselage between the wing and tail lighter. If we used the canard configuration, the entire fuselage would need to be more sturdy, increasing the weight.

The advantage of a canard plane is that it has two lifting surfaces. A tail plane has the wing as one lifting surface; the tail however, must counter the moments and push down—providing negative lift.

3.4 Fuselage

The fuselage should be lightweight, yet strong, have an aerodynamic profile, and be able to store all the system components in any required configuration. Possible construction materials include balsa wood, monokote, light plywood, spruce wood, foam, aluminum, and fiberglass.

During the conceptual design, the specifics of the fuselage were not determined. The fuselage could not be designed to meet the above mentioned requirements until the final selection of the other major components was completed.

4.0 Preliminary Design

4.1 Optimum Flying Speed

To determine the optimal flying speed, we had to consider the performance of each component of the plane—the aerodynamic, motor, propeller, and speed controller efficiencies. Higher efficiencies require less energy per distance.

The best speed for aerodynamic efficiency was calculated for varying wing areas. Assumptions were made for the initial values of parameters such as aircraft weight, aspect ratio, maximum lift coefficient, coefficient of parasitic drag, and wing efficiency. The calculations are contained in Appendix A. The results are that a large wing area plane flying at low speed is more efficient than flying at a hight speed with a smaller wing area.

4.2 Motor/Propeller

After further analysis we decided to use a more powerful, lightweight FAI-Competition motor instead of the medium astroflight Sport motor because we could decrease the wing area from 14 to 10 ft^2 thus decreasing the weight of the plane, decreasing drag, improving efficiency and thus attaining more laps.

We choose to compare the FAI-05 and FAI-15 geared motors because they were the smallest of the Competition motors that would clear the barrier.

4.2.1 Comparing the FAI-05 and FAI-15 Motors

We decided to screen the selection of these two motors based on high motor/propeller/speed controller efficiency, low energy/distance consumption, and due to the fact that our battery configuration required use of either eight or ten batteries max in series (the FAI-05 using eight and the FAI-15 using ten).

The takeoff analysis for the motors was that they both attained an altitude of 10 feet in approximately 200 feet of runway and they both cleared the barrier at the end of the runway by over 12 feet.

Both gave us high efficiencies of about 65% and a max lap potential of 27 laps (7.67 miles) and a time of flight of 13.15 minutes, so we decided to choose the FAI-05. The FAI-05 used the least batteries (thus lower energy loss to internal resistance) and because it was provided with a little more voltage than it was rated for, it would stay in its efficiency range longer and theoretically gave us more laps than the FAI-15.

We chose the optimum speed for cruise to be about 35 mph because this gave us 10% extra throttle above stall in case of adverse conditions for safety reasons and still gave us low energy consumption of 11.5 J/m. This selection provided use enough information to size the wing for 35 mph with a wing area of 10 ft^2 and an aspect ratio of 13.3.

4.3 Batteries

Now that the motor has been selected (Astroflight FAI 15), we can choose the batteries which are most compatible to the motor, and which best meet our desired characteristics. We developed some figures of merit according to our needs, and weighted them accordingly. The trade-off study is shown in Table 4.1.

Table 4.1

Weight	0.35	0.2	0.1	0.05	0.1	0.2	
	Capacity	Motor Compatibility	Internal Resistance	Volume	Cost	Current	Overall % Satisfaction
B&P Battery Selection							
AA	92%	90%	29%	100%	55%	25%	**68.7%**
Sub C	89%	80%	72%	70%	100%	90%	**86.0%**
4/5 A	90%	100%	42%	80%	74%	60%	**79.1%**
A	90%	100%	43%	75%	76%	60%	**79.2%**
Long AA	90%	100%	30%	47%	61%	25%	**67.9%**
SR Battery Selection							
1800 Max	95%	80%	78%	48%	39%	90%	**81.2%**
2500 Max	100%	100%	100%	0%	44%	100%	**89.4%**
1250 Magnum	94%	100%	43%	74%	0%	40%	**68.8%**
1800 Magnum	95%	80%	78%	48%	43%	90%	**81.5%**

Capacity was the most important factor, because it limits the amount of energy available for flying. Motor compatibility and ability to provide the needed current are the next most important factors. The motor compatibility is based on the number of cells the motor requires and how the number of cells available can be split up to meet this need. The ability to provide current was only approximated. SR Batteries said that for the motor we wanted to run, at least a sub C or larger battery would have to be used. The internal resistance is inversely proportional (or related) to the battery cross sectional area; the smaller batteries have higher internal resistance. Low cost is important, however, we decided that the difference in battery capacity was worth it. Finally, the volume was not a major constraint, since the fuselage already had to be sized large enough to hold the batteries.

4.4 Airfoil & Wing Structure

In an attempt to establish general design criteria for the wing, energy equations describing the aircraft's path were graphed to accentuate trends which could be useful. Our analysis has shown that a larger wing area could fly more efficiently at a slower velocity then a small wing area at a higher velocity. Also the equations describing the energy necessary to complete a lap is a function of 1/AR, Therefore any increase in aspect ratio would decrease the energy required. From this information the wing was established with an area of 14 ft^2 and a wing span of 14 ft. These values represent our best estimate at the conceptual design phase, of the limitations of conventional balsa and lite-ply construction techniques for a wing that is to support an aircraft with an overall weight of 18 lbs.

To be able to better predict span and aspect ratio limitations with standard radio control construction techniques, mock-up spars were constructed and tested to failure. Data from these tests was useful in establishing appropriate materials and their thickness'. Results indicate that using 3/16" x ½" spruce stringers with shear webs constructed of 1/32" lite-ply would be sufficient to support the model in flight as well as during the wing tip test. For the analysis it was assumed that the entire loading would be carried by the wing spar.

To achieve the most efficient aircraft package, compromises were made concerning the interaction of the motor/propeller combination and wing/airframe components. It was discovered that the wing could sustain flight at velocities that would decrease motor efficiency. The wing established at the conceptual design phase operated most efficiently at a velocity of approximately 25 mph while the motor selected operated at speeds of 40 to 45 mph. Using curves describing the energy required to move the wing through the air and the energy consumed by the motor, a compromise was reached. The new wing design would have a wing area of 10 ft^2 and cruise at 35 mph.

An airfoil was selected to accomplish the task of lifting the aircraft using the following criteria: Low Drag, Wide Performance Envelope, and Coefficient of lift values sufficient to support the aircraft at a reasonable angle of attack with the predetermined wing area of 10 ft^2. Searches through data bases and airfoil books lead to the discovery of the Wortmann FX63b airfoil. The most prominent feature of this airfoil is the wide performance envelope attained. A wide range of lift coefficients is spanned by very little increase in the drag. At Reynolds numbers at 100,000 and below there is a significant increase in drag produced by the airfoil, Therefore it was used to stipulate the maximum aspect ratio that could be used. To ease construction difficulties, the ideal elliptical profile will be approximated with a 0.3 taper ratio. Considering Reynolds number, wing area, and taper ratio, the root chord and tip chord were established at 16.0" and 4.8" respectively. These dimensions constrain the wing.

4.5 Fuselage

As mentioned in the conceptual design, the fuselage should be lightweight, strong, have an aerodynamic profile, and be able to store all the system components.

To achieve the lightweight construction with sufficient strength, balsa wood could be used in either a circular shape construction (circular hollow ribs made of light plywood with balsa wood or spruce stringers), or a rectangular cross-sectional shape (constructed completely from balsa wood members in a truss configuration).

Air separation at the corners of the rectangular shape prevents this shape from satisfying the aerodynamic profile requirement; however, there are possible corrections which would convert the rectangular shape into an aerodynamic profile. These include the following: creating ribs over which monokote could be stretched in order to achieve a round shape, warping veneer around the rectangular shape, and gluing foam blocks to the outside of the rectangular structure and then sanding to the sculpted shape desired.

The circular fuselage would likely be lighter than a rectangular one, however there are many advantages to the rectangular fuselage which justify its usage: The rectangular shape would also be easier to construct than a circular one. It would be stronger. And the attachment of the landing gear and wing would be significantly easier.

Because of the above advantages, we have decided to build a rectangular truss fuselage, and coat it with foam.

4.6 Airplane Stability (Control Surfaces)

In order to stabilize the airplane, the masses need to positioned correctly. First, arbitrary positions for the center of gravity (CG) and major components of the plane were chosen. The moments about the CG were calculated.

A static margin of 13% was selected for a goal. Then the CG was moved to achieve this. Appendix C contains the equations used for this section of the report. MathCad made iterations and solving for variables easier.

Some important conclusions are stated below:

$X_{cg} = 19.94$ in $\quad X_{ac} = 19.36$ in $\quad$ Horizontal Tail Area $S_h = 1.0$ ft^2

The position of each component had to be determined to satisfy the above. These values were adjusted until it worked. Figure C1 in appendix C shows the layout of the major airplane components.

The area of the vertical tail, S_v, needs to be determined in order to calculate yaw stability. This value was calculated to be .759 ft^2.

4.7 Wing Mounting

Wing / fuselage interfaces were designed to efficiently transfer the loading to the wing spar from the fuselage. Removal of a portion of the wing's leading edge allows a shear pin to be mounted in the wing spar to transfer the loads by mating with a plate extending from the fuselage frame work. Resulting moments generated by the wing can easily be compensated for by locking down the trailing edge of the wing with nylon bolts extending into the fuselage.

4.8 Landing Gear

The landing gear could be installed in a tricycle gear configuration or as a tail-dragger. The tail-dragger may require a larger vertical control surface for ground control during landing, since these types of planes have a tendency to ground loop. It would, however, provide less drag and weight since a tail wheel is typically smaller than a larger wheel.

The landing gear could be constructed using several methods. First, it could use hydraulic shocks, which would be either off-the-shelf or made by the design team. Another option is springs, dissipating the landing energy through friction in the joint between the fuselage and the landing gear. Bent aluminum plating is another possibility for use as a landing gear. Piano wire is ruled out due to the large drag which would be encountered over a cylinder of the size required for such a heavy aircraft. Plastic gears are also ruled out due to strength considerations.

Hydraulic shocks would be heavy and expensive. If the design team were to build such shocks, it would be difficult and timely. Springs would work, but the construction would be difficult. Bent aluminum plating would be both light-weight and simple. The part can be cheaply bought and easily mounted. Therefor, this is the part which will be used.

The larges commercially available aluminum landing gear for model planes was rated for a 15 lb plane. Our plane will likely be under 18 lb, which is close. However, the 15 lb rating likely has a factor of safety. To determine whether the landing gear would be adequate for our plane, a hardness test was performed to determine what kind of aluminum the plate was. Then we calculated whether the plate was strong enough. The equations contained in appendix F show that the landing gear is strong enough.

5.0 Detail Design

5.1 Aircraft Details and Drawing

5.1a Wing & Airfoil

To further improve the characteristics of the wing, the airfoil was modified to match our specific operating conditions. The airfoil geometry was reproduced using an airfoil plot program that uses a logarithmic camber line to establish a uniform loading across the airfoil at zero angle of attack. Staying within the same family of airfoils and thus keeping the performance characteristics unchanged, the camber line was modified to produce the lift necessary to sustain level flight at the cruise velocity of 35 mph.

5.1b Fuselage and Landing Gear

The fuselage design, as explained below, is depicted with figures contained in Appendix F. A detailed layout of the fuselage is given in drawing F1. Storage layout of components is given in drawing F2-F3. The fuselage will be constructed in three sections. First, the front section which will house the motor, main batteries, receiver and receiver batteries, and the servos for the rudder and elevator. The front section design is indicated in drawings F4 through F7, with the motor mount assembly in drawing F9, and the front bulkhead in drawing F8. The main section will provide attachment of the landing gear, storage of the steel payload, and wing attachment. Main section drawings are given in drawings F10 through F15, with the wing mounting plate in drawing F17. The third and final section will consist of a hollow aluminum tube shown in drawing F16. Through the center will be two servo control rods, running out to the rudder and elevator. Analyses were performed on the fuselage to determine if the working loads would cause a failure of the structure. Since the joints in the structure are glued together, they are moment-carrying; therefore the structure is indeterminate and requires a finite element method to analyze. Structural Dynamics Research Corporation (SDRC) I-DEAS Solid Modeling and Analysis software was used to perform the finite element analyses. The finite element models of the two rectangular fuselage sections are shown in drawings F4 through F6 and

F10 through F12. Analysis of the wing mounting plate was performed by hand using MathCad 5.0, and is given in Figure F1 (wingplat.mcd). Analysis of the landing gear was performed by determining a worst case landing scenario and using this as the design criteria. A worst case landing scenario corresponds to a 6 inch drop onto the runway. This simulates a landing approach during which sudden stall causes the airplane to suddenly lose lift and drop to the runway. This calculation is given in Figure F2 (hardland.mcd). This drop scenario corresponds to a 6.8 g loading, and the fuselage and the landing gear must both be able to withstand it. The finite element model for the front and main sections yield maximum von Mises stresses of 1230 psi and 330 psi, respectively, for the worst case landing scenario. The strength of balsa wood was determined through testing to be 2300 psi, therefore proving the fuselage to be acceptable. The landing gear must not experience a material failure, as well as not deflect too much as to cause another component to scrape the runway. A calculation of the maximum deflection of the landing gear is given in Figure F3 (landdefl.mcd). This calculation yields a deflection of 4.3 inches in a worst case landing, which is acceptable. The aluminum tube section of the fuselage must withstand the worst case loading associated with it. These are induced by a sudden deflection of the elevator or rudder to a maximum of 45 degrees. These movements will induce a bending moment and torque on the tube, respectively. The moment coefficients were determined through other analysis. These calculations are given in Figure F4 (tubestress.mcd). The wing bolts which secure the wing at the trailing edge to the fuselage are exposed to a bending moment during turns, the worst case of which is a 90 degree bank angle. In this configuration, shown in the calculations in Figure F5 (bolts.mcd), the moment arm of 0.2 inches causes a moment from the weight of the plane. These bolts were also found to be acceptable, with a safety factor of almost 12. The tail wheel assembly is given in drawing F18.

5.1c Tail

The tail will be constructed with balsa ribs and coated with monokote. The horizontal and vertical surfaces are depicted in Figure 5.x, and sized as determined in the preliminary design phase.

5.2 Aircraft Performance

As will be explained in detail below, our aircraft is predicted to perform well. We have done our best to minimize the energy consumed by the aircraft, and yet still meet the design criterion to satisfy our design objectives.

5.2a Take Off Performance

5.2b Handling Qualities

Most full size planes have a static margin (SM) of 5% or less, even negative. This allows for a quick response. However, large planes have high moments of inertia when compared with model aircraft. Since our plane is small, we opted to use a SM of 13%. This should be a reasonable value for good controllability.

5.2c G Load Capability

5.2d Payload Fraction

Our plane is estimated to weigh 18 lb. This number might drop to 17 lb, depending on the accuracy of our predictions for construction. Our payload is the 7.5 steel blocks which our plane is required to carry. The payload fraction then becomes 7.5/18 = 42%.

5.2e Range and Endurance

If the assumption was made that the batteries would deliver their designed capacity (40 Amp hours), our plane would be predicted to fly 34 laps and maintain flight for 16.8 minutes. However, the batteries are predicted to, in reality, only deliver ½ to 2/3 of the rated capacity. The predicted range and endurance therefor becomes:

Range: 17 – 23 laps

Endurance: 8.4 – 11.2 minutes

6.0 Manufacturing Plan

Wing:

The wing will be composed primarily of a foam wing core from which a spar will be added and then the assembly will be sheeted with balsa and monokote covering. The wing will be divided up into 6, 23" sections which will be joined together before the balsa sheeting is applied. 23" is the practical span that can be cut with our present equipment. Airfoil profiles at chord locations corresponding to 23" segments will be printed out to scale and attached to 1/8" lite-ply sheets which will be used as template material.

Appendix A Aerodynamic Efficiency

written by A. E. Childs

1.0 Derivation of Equations

The work required to sustain steady, level flight must be minimized to achieve maximum range. The work required is equal to the thrust times the distance through which the aircraft travels. Various design parameters affect the thrust required; applying analytical techniques, optimum values for these parameters can be determined. Through this procedure, the work required for a specified distance traveled can be minimized.

An expression for the energy required (work and energy are synonymous and will refer interchangeably to the same quantity throughout this section) to complete one lap around the specified course will be developed. This expression will then be differentiated with respect to various design parameters to determine optimum values.

The energy required to complete one lap is given by

$$E = Fd = Td \tag{A1.1}$$

where E is the total energy required, F is the total force, d is the total distance traveled, and T is the thrust required.

The distance traveled for one lap consists of two straight flight sections of 700 feet each minus two end turns of equal radius as given by

$$ds = 2(L - 2r_1) \tag{A1.2a}$$

$$dt = 2\pi r_1 \tag{A1.2b}$$

where ds is the straight distance traveled, dt is the turning distance traveled, L is equal to the 700 foot specified course length, and r_1 is the radius of the turn the aircraft makes. The distance traveled during straight and level flight is equal to 2L, and the distance travelled during both 180 degree turns is equal to $2r_1$.

For any aircraft performing a steady, level, coordinated turn, the wings are banked at a constant angle from the horizontal. Summing forces, the weight must equal the vertical component of lift, and the thrust must equal the drag if there is to be no acceleration in the direction of flight or the vertical direction. The turning case will be considered first, since an aircraft flying in straight, steady, level flight have zero bank angle; the cosine term therefore equals one, and the lift equals the weight.

$$T = D \tag{A.3}$$

$$W = L\cos(\phi) \tag{A.4}$$

where T equals thrust required, D represents the drag, W is the aircraft weight, and L equals the lift. Combining (A1.3) and (A1.4), the following can be developed:

$$\frac{T}{D} = \frac{W}{L\cos(\phi)} = \frac{\frac{1}{2}\rho V^2 S C_D}{\frac{1}{2}\rho V^2 S C_L \cos(\phi)} \tag{A1.5}$$

where ρ is the air density, V is the velocity of the aircraft, S is the wing area, C_D is the coefficient of drag, and C_L is the lift coefficient.

The total drag force is equal to the parasitic drag plus the induced drag.

$$C_D = C_{do} + \frac{{C_L}^2}{eAR\pi} \tag{A1.6}$$

Combining (A1.5) and (A1.6), recognizing that C_L is a function of weight, velocity, air density, and wing area, and rearranging, gives the following equation for drag:

$$T = D = \frac{1}{2} C_{do} \rho V^2 S + \frac{2W^2}{\cos^2(\phi)\rho V^2 eAR\pi} \tag{A1.7}$$

Taking care to distinguish between straight flight and turning flight, the expression for drag in (A1.7) and equation (A1.2) can be substituted into equation (A1.1), resulting in the following expression for the energy required to complete one lap:

$$E = \left[\frac{1}{2} C_{do} \rho V^2 S + \frac{2W^2}{\cos^2(\phi)\rho V^2 eAR\pi}\right] 2\pi r_1 + \left[\frac{1}{2} C_{do} \rho V^2 S + \frac{2W^2}{\rho V^2 eAR\pi}\right] 2(L - 2r_1)$$

(A1.8)

Although this expression for energy assumes no acceleration in the vertical or flight path directions, the horizontal component of lift produces a constant acceleration normal to the flight path. This acceleration, which causes the aircraft to turn in a circle, is given by the following expression:

$$L\sin(\phi) = ma_R = \frac{W}{g}\frac{V^2}{r_1} \tag{A1.9}$$

where L times the sine of the bank angle is the horizontal lift component, m is the mass of the aircraft (which is equal to the weight, W, divided by the acceleration of gravity), and a_R is the radial acceleration--equal to the velocity squared over the radius of turn.

Solving (A1.9) for the turning radius, the following expression results:

$$r_1 = \frac{V^2}{g\tan(\phi)} \tag{A1.10}$$

Substituting this equation into (A1.8) to eliminate r_1 the total energy to complete one lap is given by

$$E = \left[\frac{1}{2}C_{do}\rho V^2 S + \frac{2W^2}{\cos^2(\phi)\rho V^2 eAR\pi}\right]2\pi\left(\frac{V^2}{g\tan(\phi)}\right)$$
$$+\left[\frac{1}{2}C_{do}\rho V^2 S + \frac{2W^2}{\rho V^2 eAR\pi}\right]2\left[2L - \left(\frac{V^2}{g\tan(\phi)}\right)\right] \tag{A1.11}$$

This energy equation is a function of weight, velocity, wing area, bank angle, density, wing efficiency, and parasitic drag coefficient. Initial guess values for weight, air density, wing efficiency, and parasitic drag coefficient can be assumed to some degree of accuracy. Velocity, bank angle, and wing area are therefore the main variables. In order to get a meaningful graphical representation to determine optimum values for these variables, this expression must be reduced from four-dimensions to three. To do this, the equation for the total energy to complete one lap can now be differentiated with respect to bank angle, phi, this derivative set equal to zero and solved for phi to obtain an expression for the optimum bank angle. This optimum bank angle expression can then be substituted back into the energy equation. The expression then is a function only of velocity and wing area. This can now be plotted as a three-dimensional graph, or as a two-dimensional graph of energy versus velocity for various different wing areas. These calculations are included in Section 2.0 of this Appendix, along with the resulting plots.

Appendix B Battery Options

Type	Cell Capacity (mAh)	Dia	Len	Wt.	Cells / 2.5 lb	Total Weight	Volume (in^3)	Capacity (Ah) / 2.5 lb	Price
B&P Battery Selection									
AA	900	.55 "	1.96 "	.96 oz.	41	2.46 lb	24.31	36.9	$ 174.25
Sub C	1700	.867 "	1.70 "	1.87 oz.	21	2.45 lb	26.84	35.7	$ 102.90
4/5 A	1000	.66 "	1.66 "	1.09 oz.	36	2.45 lb	26.03	36	$ 144.00
A	1200	.669 "	1.969 "	1.31 oz.	30	2.46 lb	26.44	36	$ 141.00
Long AA	1000	.56 "	2.55 "	1.09 oz.	36	2.45 lb	28.79	36	$ 165.60
SR Battery Selection									
1800 Max	1800	.90 "	1.69 "	1.86 oz.	21	2.44 lb	28.75	37.8	$ 199.50
2500 Max	2500	1.02 "	1.97 "	2.47 oz.	16	2.47 lb	32.79	40	$ 192.00
1250 Magnum	1250	.67 "	1.97 "	1.30 oz.	30	2.44 lb	26.53	37.5	$ 262.50
1800 Magnum	1800	.90 "	1.69 "	1.86 oz.	21	2.44 lb	28.75	37.8	$ 194.25

Appendix C Equations for Airplane Stabilization

SUM MOMENTS OF WING, TAIL AND FUSELAGE ABOUT CG OF PLANE

$$Cmcg_w = Cl_w \cdot \left(\frac{X_{cg}}{c_w} - \frac{X_{ac}}{c_w}\right) \cdot \cos(\alpha_w - i_w) + Cd_w \cdot \left(\frac{X_{cg}}{c_w} - \frac{X_{ac}}{c_w}\right) \cdot \sin(\alpha_w - i_w) \ldots$$
$$+ Cl_w \cdot \left(\frac{Z_{cg}}{c_w} - \frac{Z_w}{c_w}\right) \cdot \sin(\alpha_w - i_w) - Cd_w \cdot \left(\frac{Z_{cg}}{c_w} - \frac{Z_w}{c_w}\right) \cdot \cos(\alpha_w - i_w) + Cm_{acw}$$

$$Cmcg_t = \eta \cdot \frac{l_t \cdot At}{Aw \cdot c_w} \cdot Cl\alpha_t \cdot (\varepsilon_0 + i_w - i_t) - \eta \cdot \frac{l_t \cdot At}{Aw \cdot c_w} \cdot Cl\alpha_t \cdot \alpha \cdot \left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

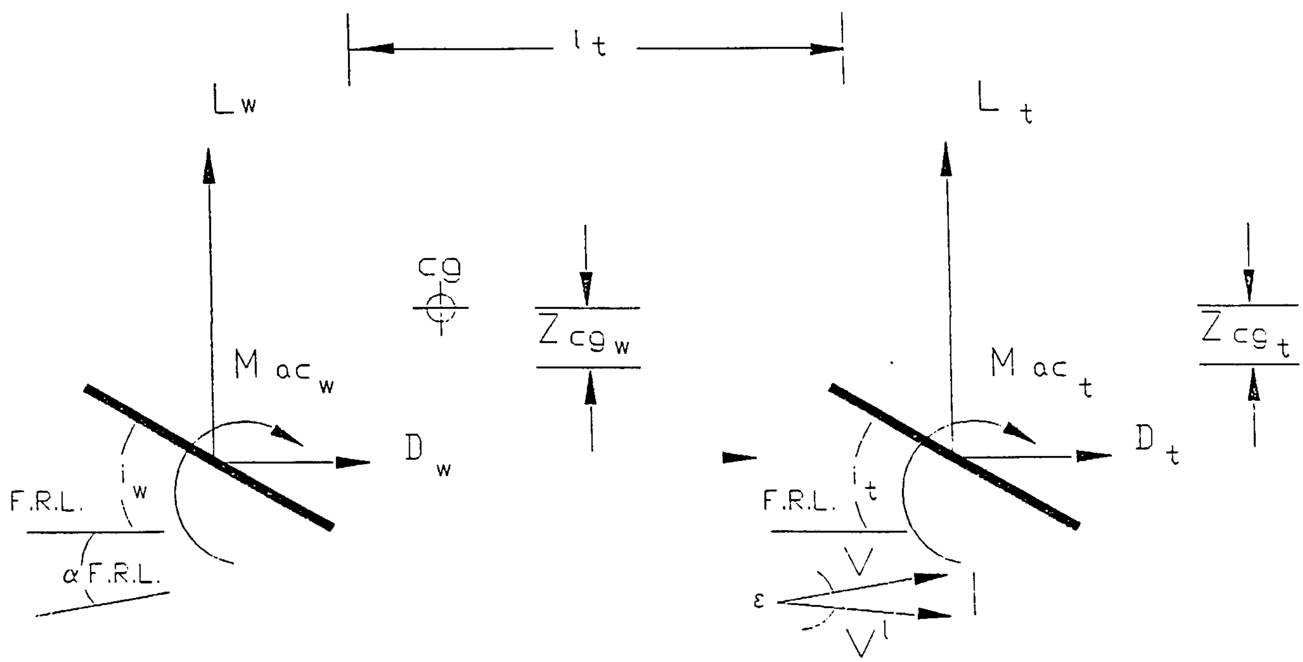
$$Cm_f = \frac{K}{36.5 \cdot Aw \cdot c_w} \cdot \sum_{x=0}^{lf} W_f^2 \cdot (\alpha 0_w + i_f) \cdot \Delta x$$

PLACE CG OF PLANE RELATIVE TO Xac OF WING TO ACHIEVE A SM OF 13%

$$SM = \frac{X_{NP}}{c_w} - \frac{X_{cg}}{c_w}$$

$$\frac{X_{NP}}{c_w} = \frac{X_{ac}}{c_w} - \frac{Cm\alpha_f}{Cl\alpha_w} + \eta \cdot \frac{l_t \cdot At}{Aw \cdot c_w} \cdot \frac{Cl\alpha_t}{Cl\alpha_w} \cdot \left(1 - \frac{d\varepsilon}{d\alpha}\right)$$

$$\frac{.13 \cdot c_w + X_{cg}}{c_w} = \frac{X_{ac}}{c_w} - \frac{Cm\alpha_f}{Cl\alpha_w} + \eta \cdot \frac{l_t \cdot At}{Aw \cdot c_w} \cdot \frac{Cl\alpha_t}{Cl\alpha_w} \cdot \left(1 - \frac{d\varepsilon}{d\alpha}\right)$$ *combine and set SM =13%*



CONSTANTS

$\rho := 1.225 \cdot \frac{kg}{m^3}$ $\alpha_w := 0 \cdot deg$ $Z_{cg} := .2 \cdot in$ $\Delta x := 1 \cdot in$

$V := 35 \cdot mph$ $W := 18 \cdot lbf$ $Z_w := .0485 \cdot m$ $W_f := .049135 \cdot m$

$Aw := 9.968 \cdot ft^2$ $Cd_w := .025$ $\eta := .8$ $K := .95$

$c_w := 16 \cdot in$ $Cm\alpha_f := 0$ $Cm_{acw} := -.15$ $i_f := 0$

$AR_w := 13.3$ $\alpha 0_w := -5.5 \cdot deg$ $Cl\alpha_w := \frac{.12727}{deg}$ $Cl\alpha_t := 2 \cdot \pi$

VARIABLES ITERATED ON

$\alpha_t := -2 \cdot deg$ $i_t := -2 \cdot deg$ $lf := 66 \cdot in$ $At := 1 \cdot ft^2$

$X_{ac} := 17 \cdot in$ $X_{cg} := 16 \cdot in$ $l_t := 50 \cdot in$

CALCULATIONS

$$Cl_w := -2 \cdot \frac{\left(\frac{1}{2} \cdot 2 \cdot \pi \cdot i_t \cdot \rho \cdot V^2 \cdot At - W\right)}{\left[\rho \cdot \left(V^2 \cdot Aw\right)\right]} \qquad Cl_w = 0.599$$ *clw for steady level flight*

$$i_w := \frac{Cl_w}{Cl\alpha_w} \qquad i_w = 4.704 \cdot deg$$ *incidence angle of wing*

$$\varepsilon_0 := \frac{2 \cdot Cl_w}{\pi \cdot AR_w} \qquad \varepsilon_0 = 0.029$$ *downwash from wing*

$$d\varepsilon d\alpha := \frac{2 \cdot Cl\alpha_w}{\pi \cdot AR_w} \qquad d\varepsilon d\alpha = 0.349$$ *downwash angle (rad)*

SET SUM OF MOMENTS = TO ZERO
SOLVE RESULTING EQUATION AND SM EQUATION SIMULTANEOUSLY
USING A MATHCAD SOLVE BLOCK TO YIELD Xcg and Xac

$X_{cg} = 16 \cdot in$

$X_{ac} = 19.36 \cdot in$

WEIGHT & POSITION SECTION

$X_{cg} := 19.94 \cdot in$ *Center of gravity position measured from nose.*

$W_m := .1984 \cdot kg \cdot g$ *Motor weight.* $X_m := 1.75 \cdot in$ *Motor position from nose.*

$W_s := 7.5 \cdot lbf$ *Steel weight.* $X_s := X_{cg}$ *Steel position from nose.*

$W_w := 2.0 \cdot lbf + 9.814 \cdot 10^{-2} \cdot lbf$ *Wing & 2 servos weight.* $X_w := 31.29 \cdot in$ *Wing position from nose.*

$W_t := .25 \cdot lbf$ *Tail weight.* *Tail position from nose is found in tube section*

$W_f := 1.0 \cdot lbf$ *Fuselage main section weight.*

$X_f := 19.64 \cdot in$ *Fuselage center of mass measured from the nose.*

$W_{lg} := 1.3 \cdot lbf$ *Landing gear weight.* $X_{lg} := 17.59 \cdot in$ *Landing gear position from nose*

$W_{svo} := .0445 \cdot kg \cdot g \cdot 2$ *2 servo weight.* $X_{svo} := 15.545 \cdot in$ *Servo position from nose*

$W_{rec} := .048 \cdot kg \cdot g$ *Receiver weight* $X_{rec} := 13.25 \cdot in$ *Receiver position from nose*

$W_{recbat} := .0935 \cdot kg \cdot g$ *Receiver battery weight* $X_{recbat} := 11.7 \cdot in$ *Receiver battery position from nose*

$W_b := 2.5 \cdot lbf$ *Battery weight.* $X_b := 7.45 \cdot in$ *Battery center of mass position from nose.*

$W_{rod} := \dfrac{13.8 \cdot gm \cdot g}{4 \cdot ft}$ *Control rod weight*

$W_{sc} := 25 \cdot gm \cdot g$ *Speed controller weight* $X_{sc} := 3.5 \cdot in$ *Speed controller center of mass from nose*

TUBE SECTION

$d_o := .5 \cdot in$ *Outer diameter of tube.* $t_{tube} := \dfrac{2}{32} \cdot in$ *Tube thickness.*

$d_i := d_o - t_{tube}$ *Inner diameter of tube.* $A_{tube} := \dfrac{\pi}{4} \cdot d_o^{\,2} - \dfrac{\pi}{4} \cdot d_i^{\,2}$ *Cross-sectional area of tube.*

$\rho_{tube} := 26.6 \cdot 10^3 \cdot \dfrac{newton}{m^3}$ *Density of aluminum.* $\rho_{tube} = 0.098 \cdot \dfrac{lbf}{in^3}$

$L_{tube} := 42 \cdot in$ *Length of tube, initial guess.*

$X_{tube} := X_f 2 + \dfrac{L_{tube}}{2}$ *Moment arm of the tube.*

$X_t := L_{tube} + X_f 2$ *Distance from nose to tail*

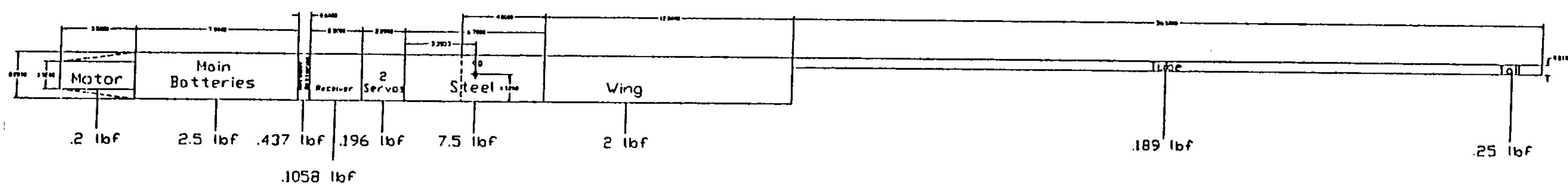
CALCULATION SECTION

given

$$W_m \cdot (X_{cg} - X_m) + W_b \cdot (X_{cg} - X_b) + W_{recbat} \cdot (X_{cg} - X_{recbat}) + (W_{rec}) \cdot (X_{cg} - X_{rec}) \ldots = 0 \cdot in \cdot lbf$$
$$+ (W_{sc}) \cdot (X_{cg} - X_{sc}) + W_s \cdot (X_{cg} - X_s) + W_{lg} \cdot (X_{cg} - X_{lg}) - W_w \cdot (X_w - X_{cg}) \ldots$$
$$+ W_f (X_{cg} - X_f) + W_{svo} \cdot (X_{cg} - X_{svo}) \ldots$$
$$+ (-1) \cdot (L_{tube} \cdot \rho_{tube} \cdot A_{tube} + W_{rod} \cdot L_{tube}) \cdot \left[\left(X_f 2 + \frac{L_{tube}}{2} \right) - X_{cg} \right] - W_t \cdot \left[(L_{tube} + X_f 2) - X_{cg} \right]$$

$L := \mathrm{find}(L_{tube})$ $L = 3.329 \cdot ft$ $L = 39.943 \cdot in$

$W_{tube} := A_{tube} \cdot \rho_{tube} \cdot L_{tube}$ $W_{tube} = 0.189 \cdot lbf$



Motor
Main Batteries
Receiver
2 Servos
Steel
Wing
Tube
.2 lbf
2.5 lbf
.437 lbf
.1058 lbf
.196 lbf
7.5 lbf
2 lbf
.189 lbf
.25 lbf

MAX MOMENT CREATED BY HORIZONTAL SURFACE

constants

$\eta := .8$ $\quad Aw := 9.968 \cdot ft^2$ $\quad c_w := 16 \cdot in$ $\quad Cl\alpha_t := 2 \cdot \pi$

$i_t := -2 \cdot deg$ $\quad l_t := 50 \cdot in$ $\quad At := 1 \cdot ft^2$

$\varepsilon_0 := 0.029$ $\quad i_w := 4.704 \cdot deg$ $\quad d\varepsilon d\alpha := 0.349$

calculations

$$Cmcg_t(\alpha_t) := \eta \cdot \frac{l_t \cdot At}{Aw \cdot c_w} \cdot Cl\alpha_t \cdot (\varepsilon_0 + i_w - i_t) - \eta \cdot \frac{l_t \cdot At}{Aw \cdot c_w} \cdot Cl\alpha_t \cdot \alpha_t \cdot (1 - d\varepsilon d\alpha)$$

$Cmcg_t(45 \cdot deg) = -0.576$

$Cmcg_t(-45 \cdot deg) = 1.036$

MAX MOMENT CREATED BY VERTICAL SURFACE

constants

$S_w := 9.67 \cdot ft^2$ $\quad l_v := 55.35 \cdot in$ $\quad S_v := 0.759 \cdot ft^2$

$\eta_v := .8$ $\quad Cl\alpha_v := 2 \cdot \pi$ $\quad \sigma := 0$

calculations

$$C_n(\beta) := \frac{l_v \cdot S_v}{S_w \cdot c_w} \cdot \eta_v \cdot Cl\alpha_v \cdot (\beta + \sigma)$$

$C_n(45 \cdot deg) = 1.072$

$C_n(-45 \cdot deg) = -1.072$

YAW STABILITY

$$Cn\beta_{wf} = -k_n \cdot k_{Rl} \cdot \frac{S_{fs}}{S_w} \cdot \frac{l_f}{b}$$

$$Cn\beta_v = \frac{l_v \cdot S_v}{S_w \cdot c_w} \cdot \eta_v \cdot Cl\alpha_v \cdot \left(1 + \frac{d\sigma}{d\beta}\right)$$

$$\eta_v \cdot \left(1 + \frac{d\sigma}{d\beta}\right) = .724 + 3.06 \cdot \frac{\frac{S_v}{S_w}}{1 + \cos(\Lambda_{c4w})} + .4 \cdot \frac{z_w}{d} + .009 \cdot AR_w$$

CONSTANTS

$S_{fs} := 9.78 \cdot 10^{-2} \cdot m^2$	$l_f := 66 \cdot in$	$X_m := 23.35 \cdot in$	$W_f := .097 \cdot m$	$d := .097 \cdot m$
$h_1 := .097 \cdot m$	$h_2 := .5 \cdot in$	$V := 35 \cdot mph$	$b := 11.5 \cdot ft$	$Cn\beta := .3$
$\nu := 1.4607 \cdot 10^{-5} \cdot \frac{m^2}{sec}$	$S_w := 9.67 \cdot ft^2$	$c_w := 104. \cdot in$	$l_v := 55.35 \cdot in$	
$Cl\alpha_v := 2 \cdot \pi$	$\Lambda_{c4w} := 2.3 \cdot deg$	$AR_w := 13.3$	$Z_w := .0485 \cdot m - .005 \cdot m$	

CONSTANTS

$\frac{l_f^2}{S_{fs}} = 28.735$ $\quad \frac{X_m}{l_f} = 0.354$ $\quad \sqrt{\frac{h_1}{h_2}} = 2.764$ $\quad \frac{h_1}{W_f} = 1$

$k_n := .0015$ *from fig 2.28*

$\frac{V \cdot l_f}{\nu} = 1.796 \cdot 10^6$

$k_{Rl} := 2.85$ *interpolated from fig 2.29*

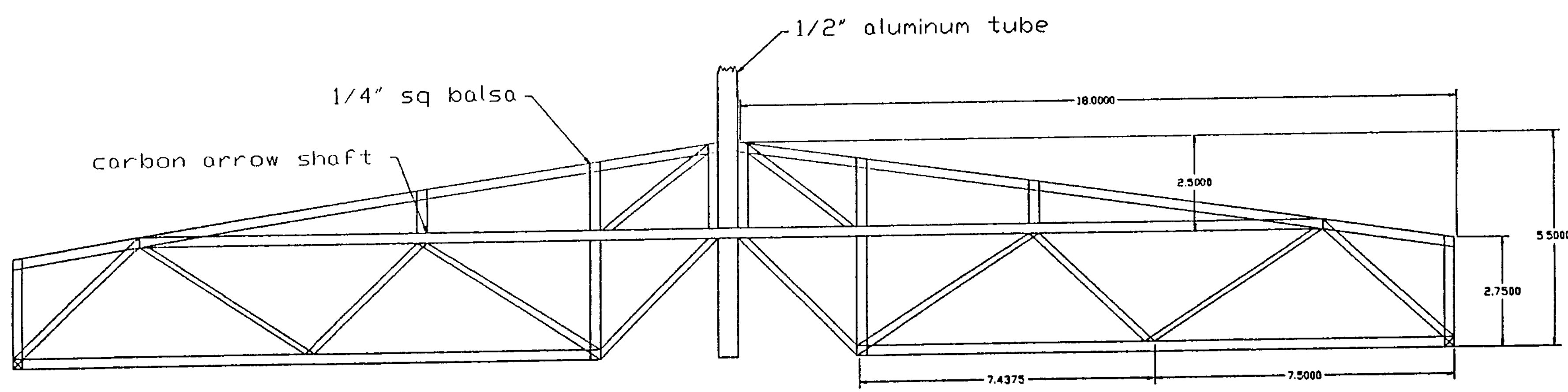
$S_v := 1 \cdot ft^2$ *initial guess*

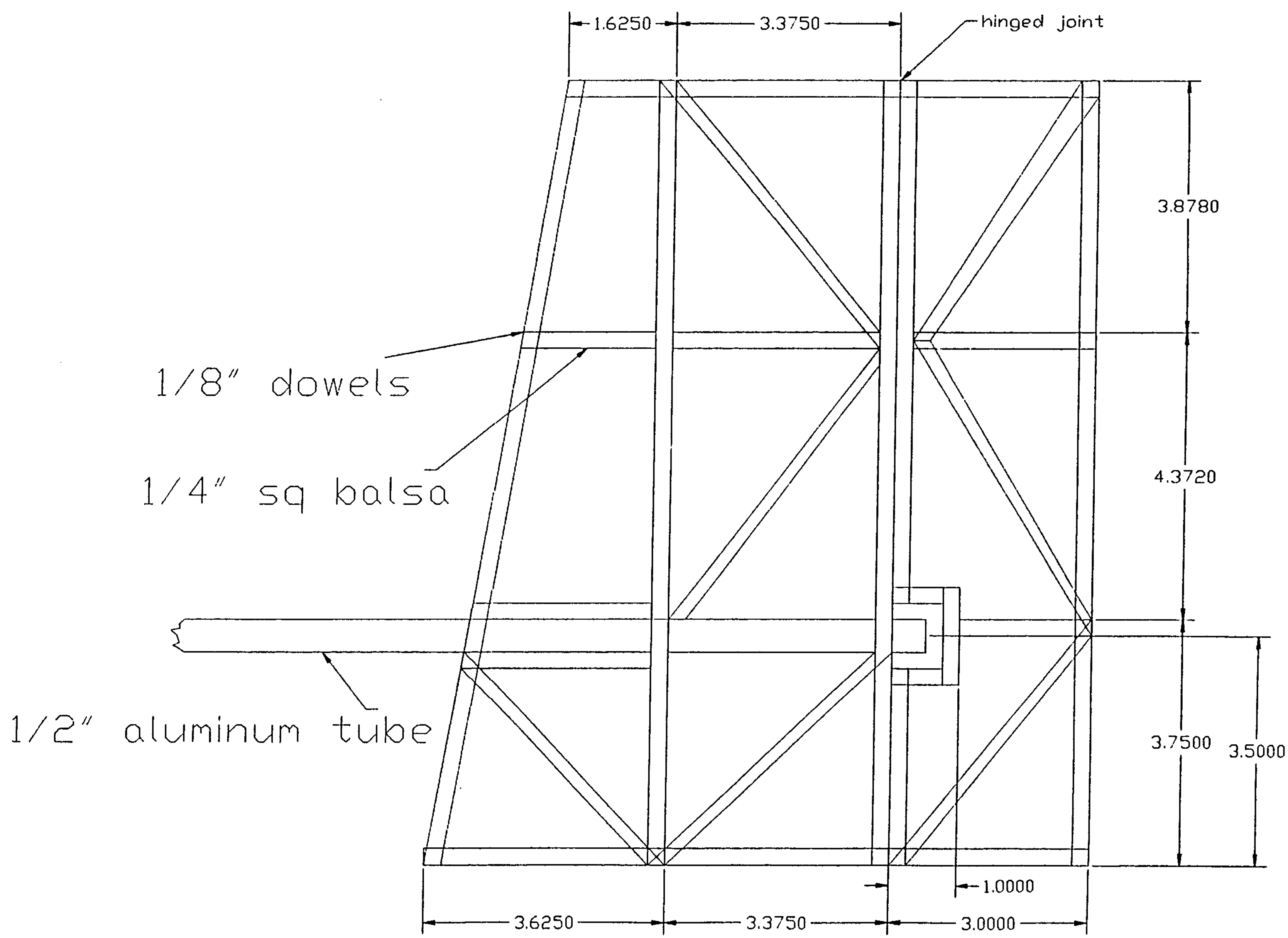
CALCULATIONS

given

$$-k_n \cdot k_{Rl} \cdot \frac{S_{fs}}{S_w} \cdot \frac{l_f}{b} + \frac{l_v \cdot S_v}{S_w \cdot c_w} \cdot Cl\alpha_v \cdot \left[.724 + 3.06 \cdot \frac{\frac{S_v}{S_w}}{1 + \cos(\Lambda_{c4w})} + .4 \cdot \frac{Z_w}{d} + .009 \cdot AR_w\right] = Cn\beta$$

$S_v := \text{find}(S_v)$ $\quad S_v = 0.759 \cdot ft^2$


1/2" aluminum tube
1/4" sq balsa
carbon arrow shaft
18.0000
2.5000
5.5000
2.7500
7.4375
7.5000


1.6250
3.3750
hinged joint
3.8780
4.3720
3.7500
3.5000
1/8" dowels
1/4" sq balsa
1/2" aluminum tube
1.0000
3.6250
3.3750
3.0000

Appendix D Motor Trade-Offs

Conceptual FOM Rating Chart

Mission Features:	Clear Barrier Maneuvers	Maximum Laps	Maximum Laps	Maximum Laps
FOMs	Thrust Available	Optimal Speed	Efficiency	Weight
Parameters				
Motor Gearing				
Geared	9	8	7	8
Ungeared	5	4	8	9
Wing Area Sizing				
10 ft2	4	n/a	9	9
12 ft2	7	n/a	8	8
14 ft2	10	n/a	7	7
Motor/Propeller Sizing				
Small	4	9	9	9
Medium	8	8	8	8
Large	9	7	7	7
Speed Controller Type				
Variable Resistance	8	4	4	8
LO-Rate	8	6	6	8
HI-Rate	8	9	9	8

7.5 Volt FAI -05 Geared Motor (Ratio 2.38) W = 18 lbf

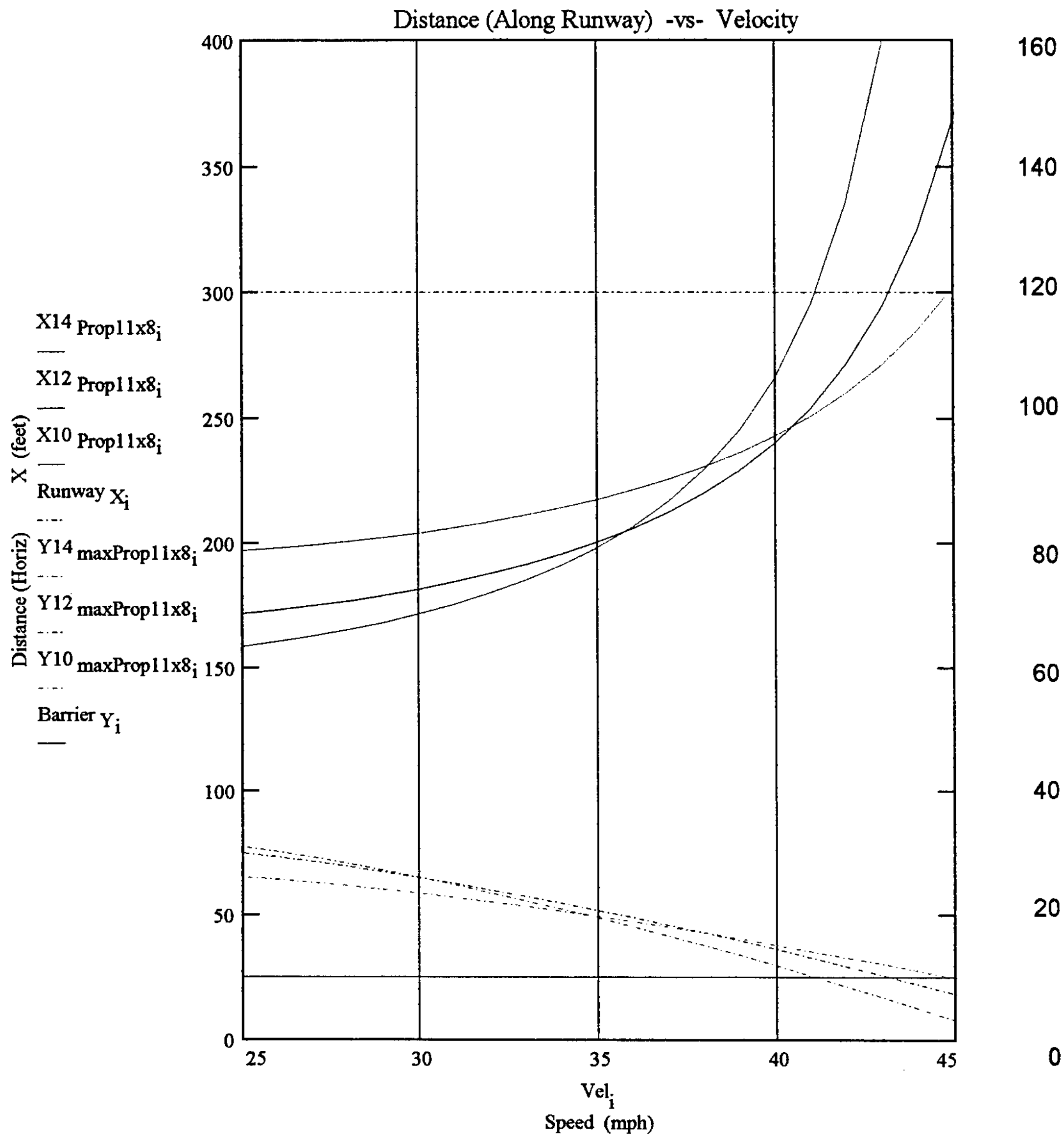
[8 Volts Supplied by 8 Batteries]

Throttle: 100% Wing Area: 10, 12, & 14

Max. Height for 300 Feet of Runway

Y (feet)

Distance (X) Traveled to Clear 10 Foot Barrier

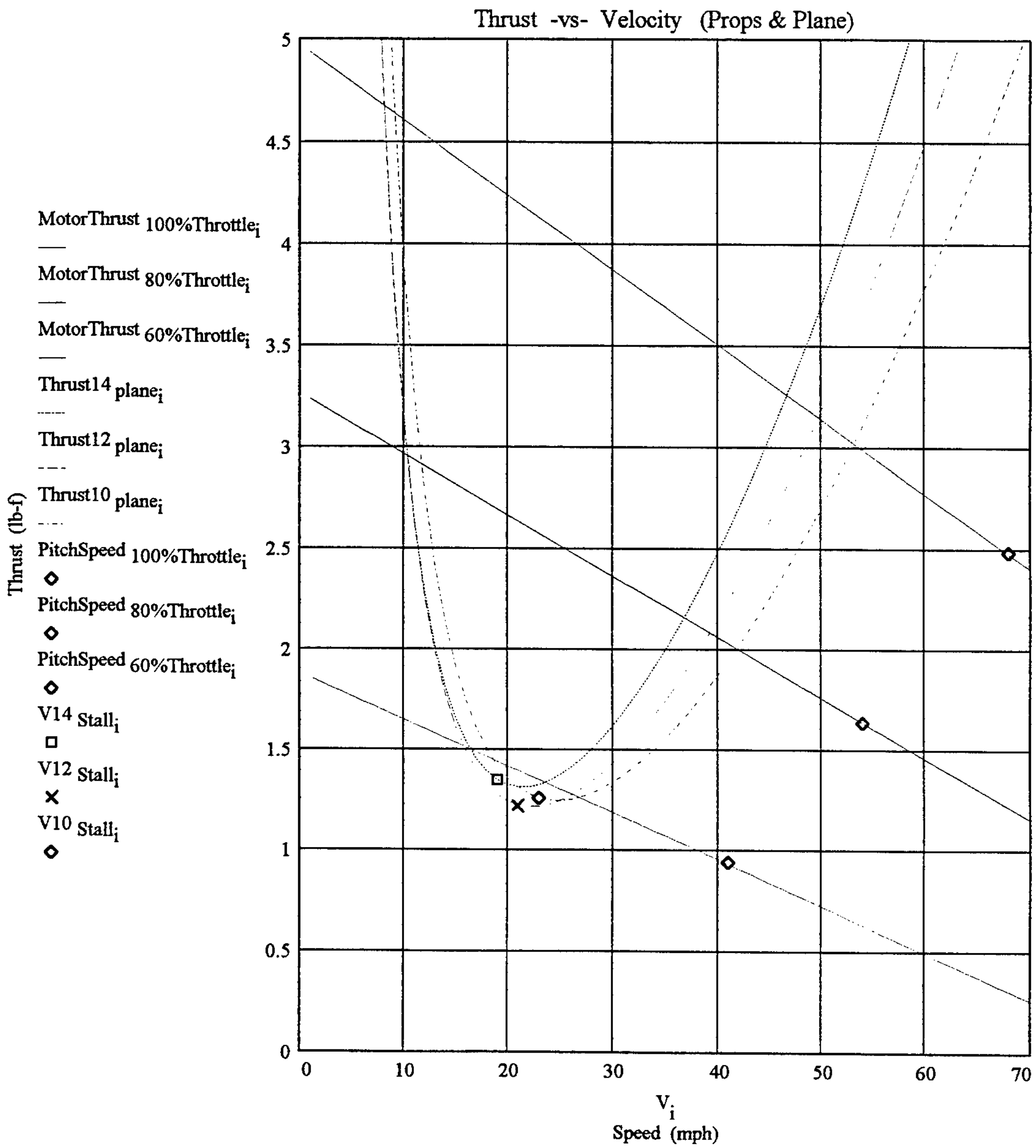


7.5 Volt FAI -05 Geared Motor (Ratio 2.38) [11x8 Prop]

8 Batteries (8 Volts)

Throttle: 100%, 80%, & 60%

Wing Area: 14, 12, & 10 ft^2

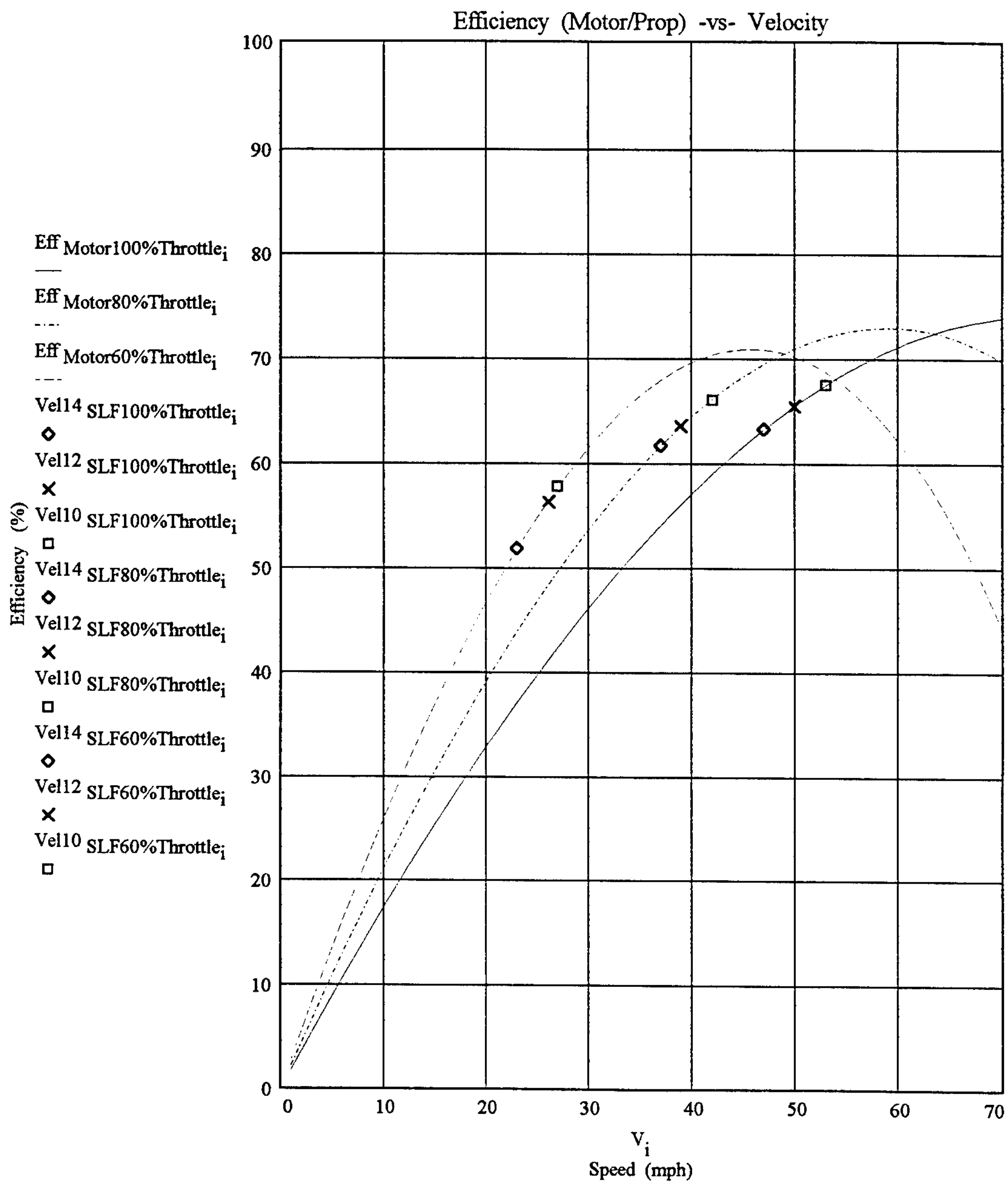


7.5 Volt FAI-05 Geared Motor (Ratio 2.38) [11x8 Prop]

8 Batteries (8 Volts)

Throttle: 100%, 80%, & 60%

Wing Area: 14, 12, & 10 ft^2

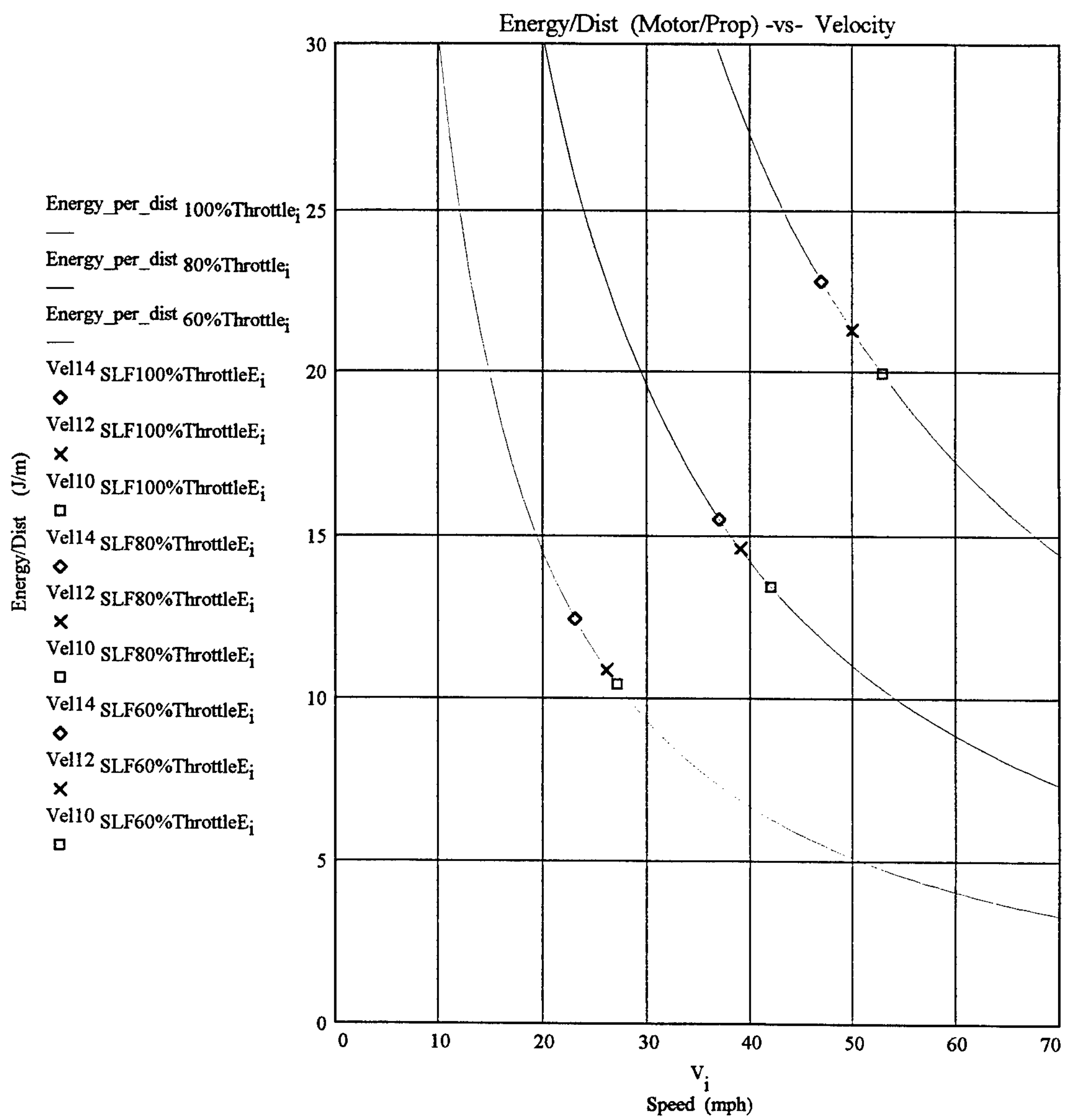


7.5 Volt FAI-05 Geared Motor (Ratio 2.38) [11x8 Prop]

8 Batteries (8 Volts)

Throttle: 100%, 80%, & 60%

Wing Area: 14, 12, & 10 ft^2



10 Volt FAI -15 Geared Motor (Ratio 2.38) W = 18 lbf

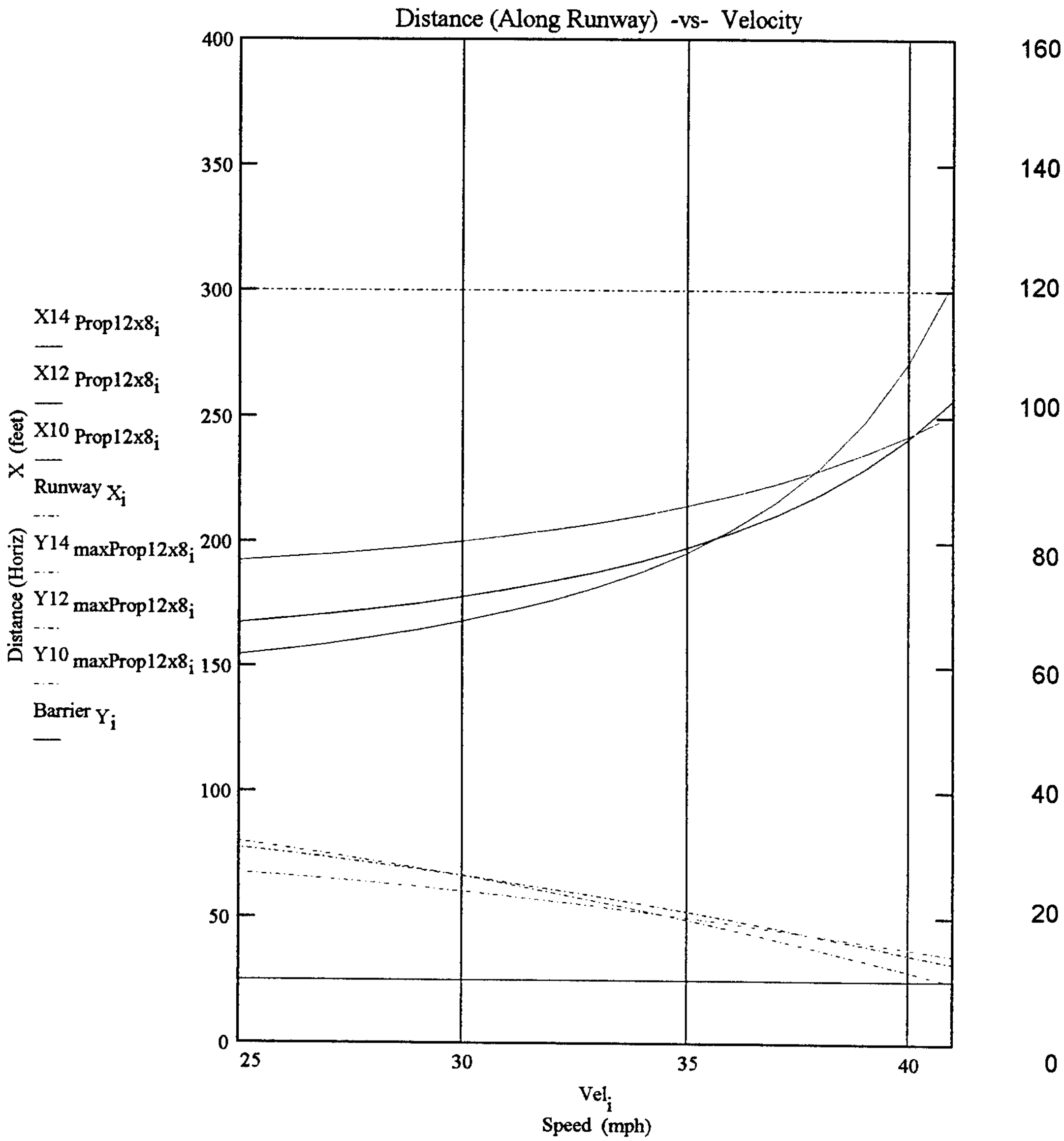
[10 Volts Supplied by 10 Batteries]

Throttle: 100% Wing Area: 10, 12, & 14

Distance (X) Traveled to Clear 10 Foot Barrier

Max. Height for 300 Feet of Runway

Y (feet)

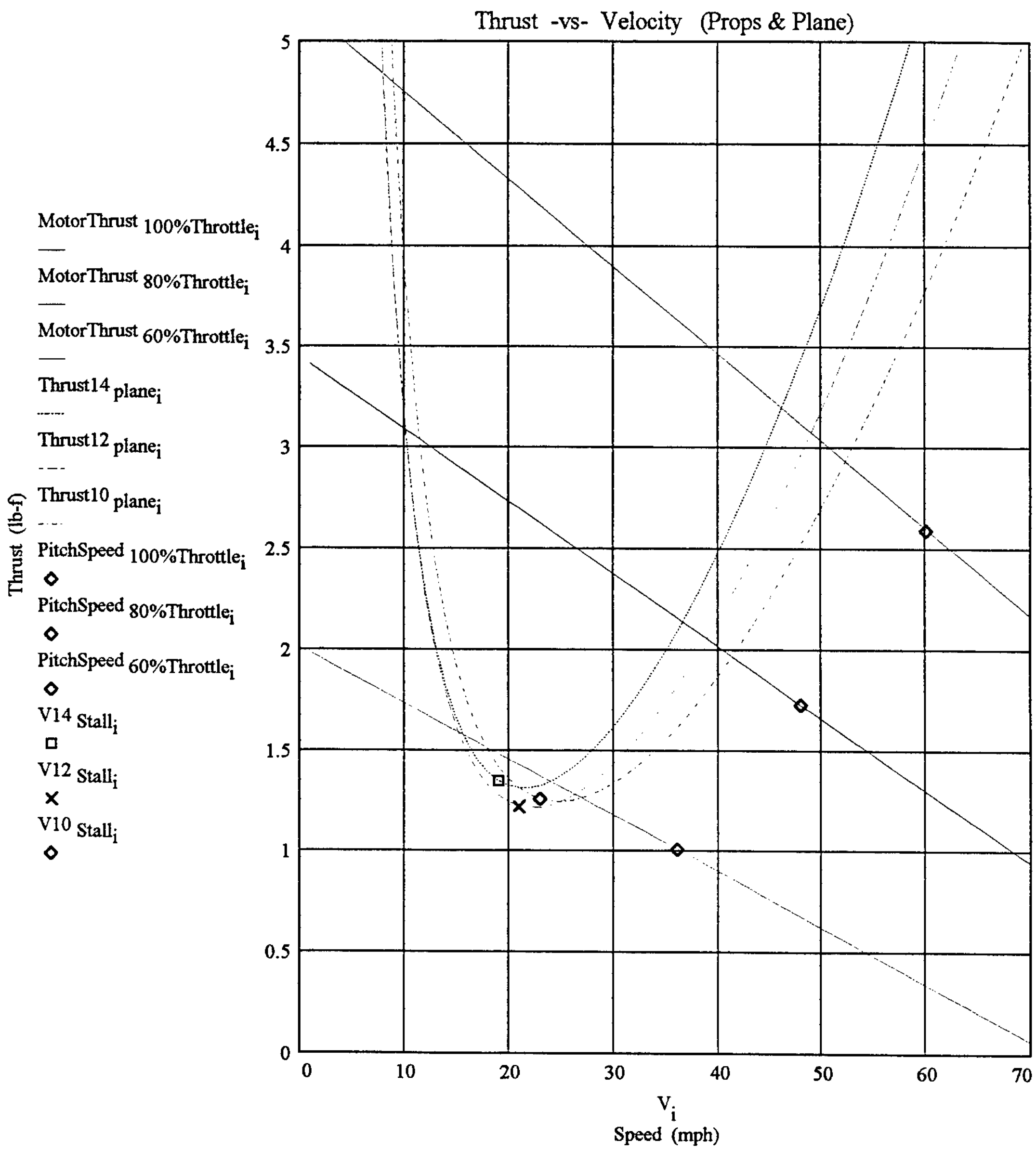


10 Volt FAI -15 Geared Motor (Ratio 2.38) [12x8 Prop]

10 Batteries (10 Volts)

Throttle: 100%, 80%, & 60%

Wing Area: 14, 12, & 10 ft^2

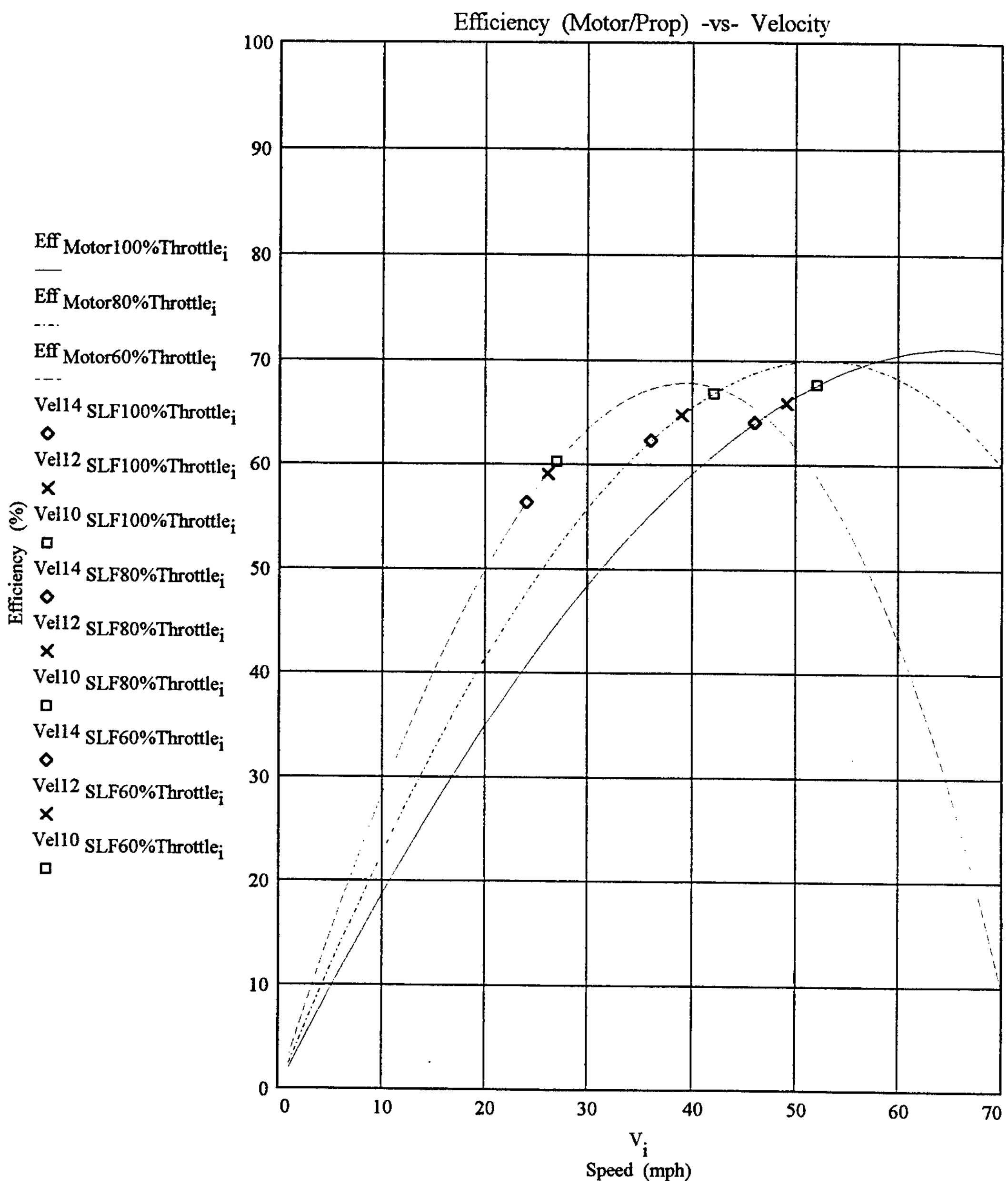


10 Volt FAI-15 Geared Motor (Ratio 2.38) [12x8 Prop]

10 Batteries (10 Volts)

Throttle: 100%, 80%, & 60%

Wing Area: 14, 12, & 10 ft^2

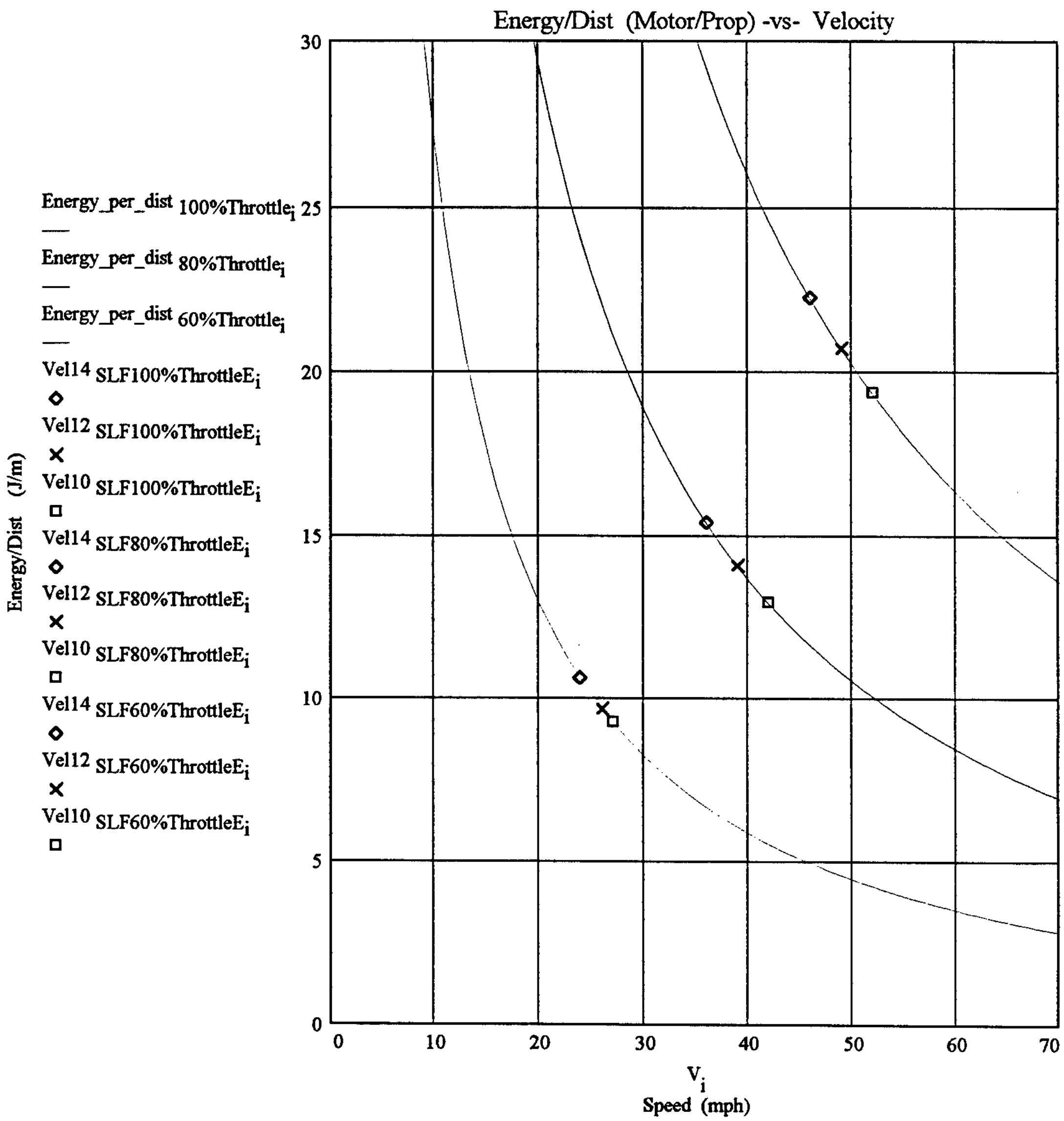


10 Volt FAI-15 Geared Motor (Ratio 2.38) [12x8 Prop]

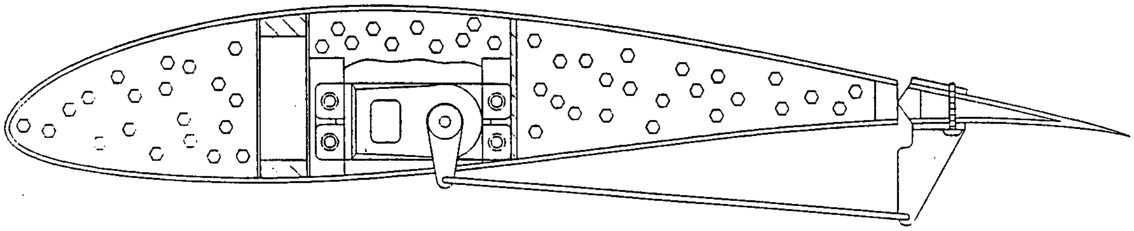
10 Batteries (10 Volts)

Throttle: 100%, 80%, & 60%

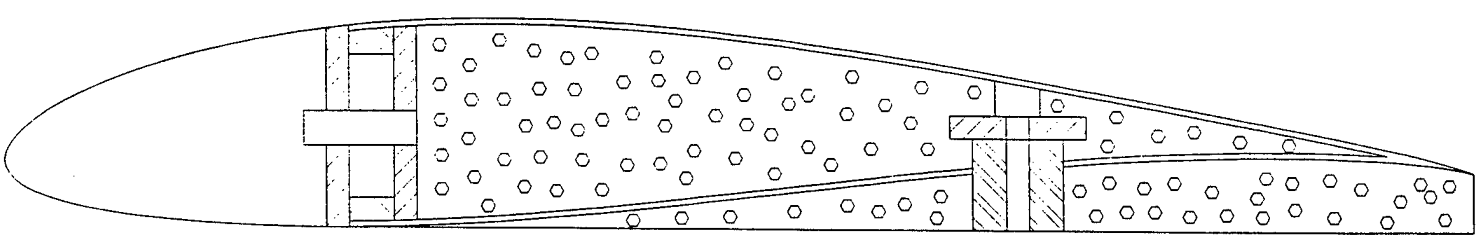
Wing Area: 14, 12, & 10 ft^2



Appendix E Airfoil Diagrams

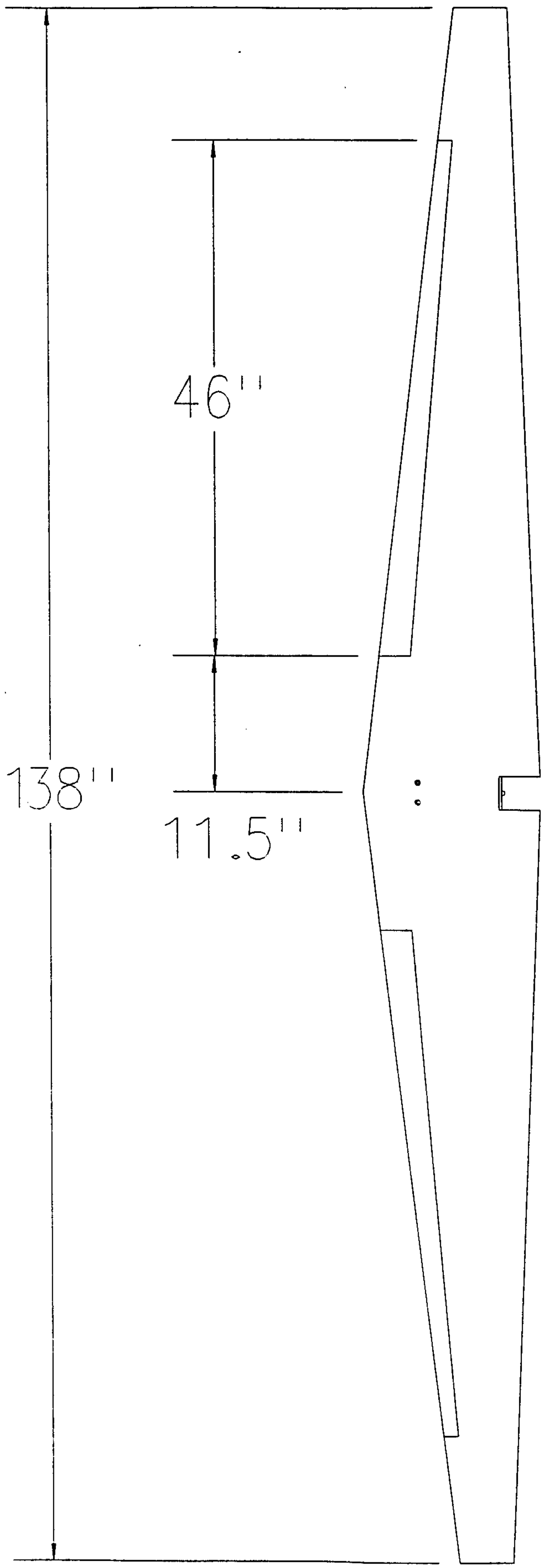


Aileron Servo Location



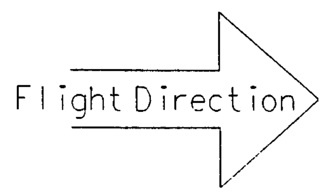
Wing Root Cross-Section

(Wing Mounting)



Root Chord
16''

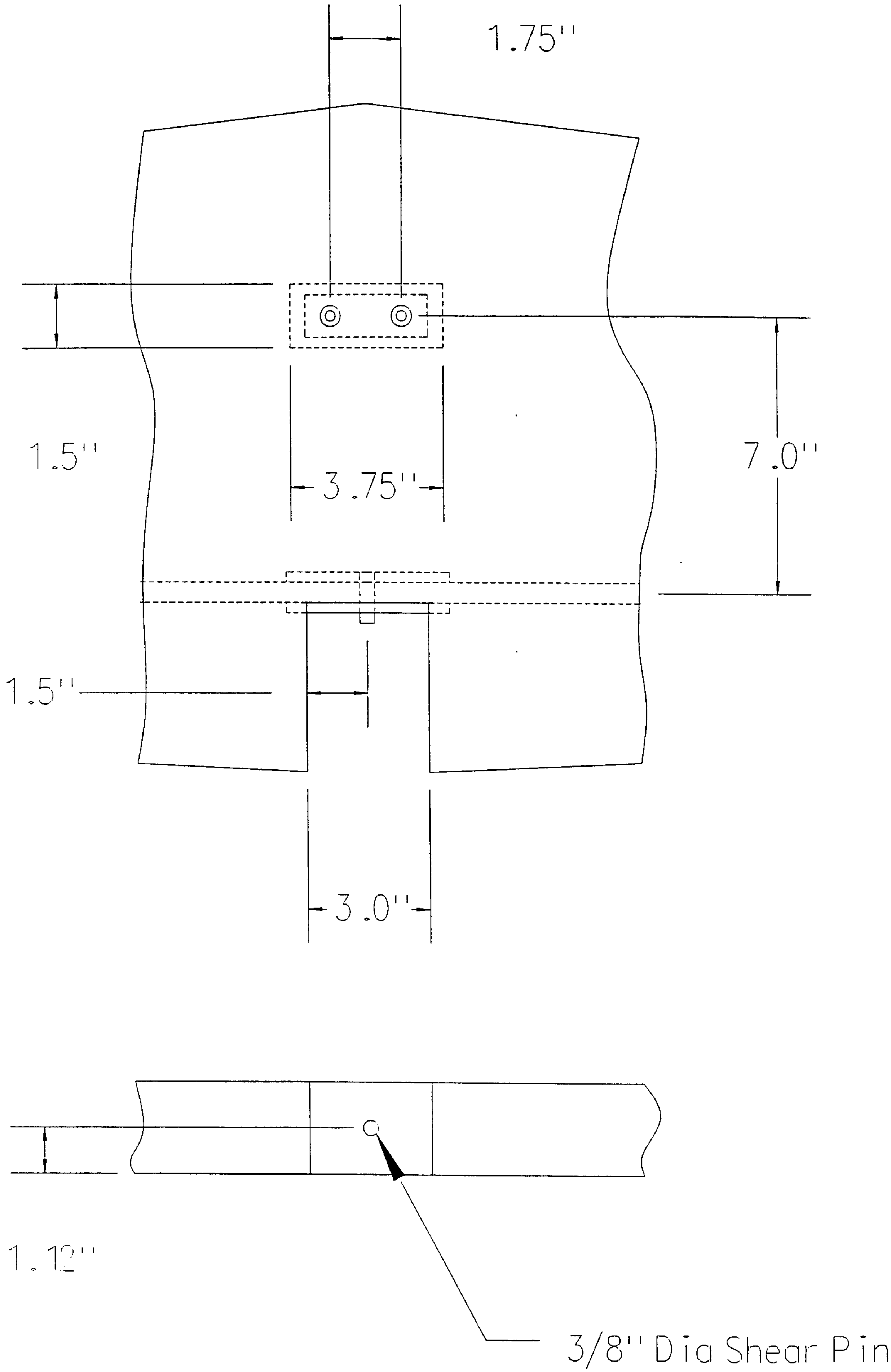
Tip Chord
4.8''



Ailerons

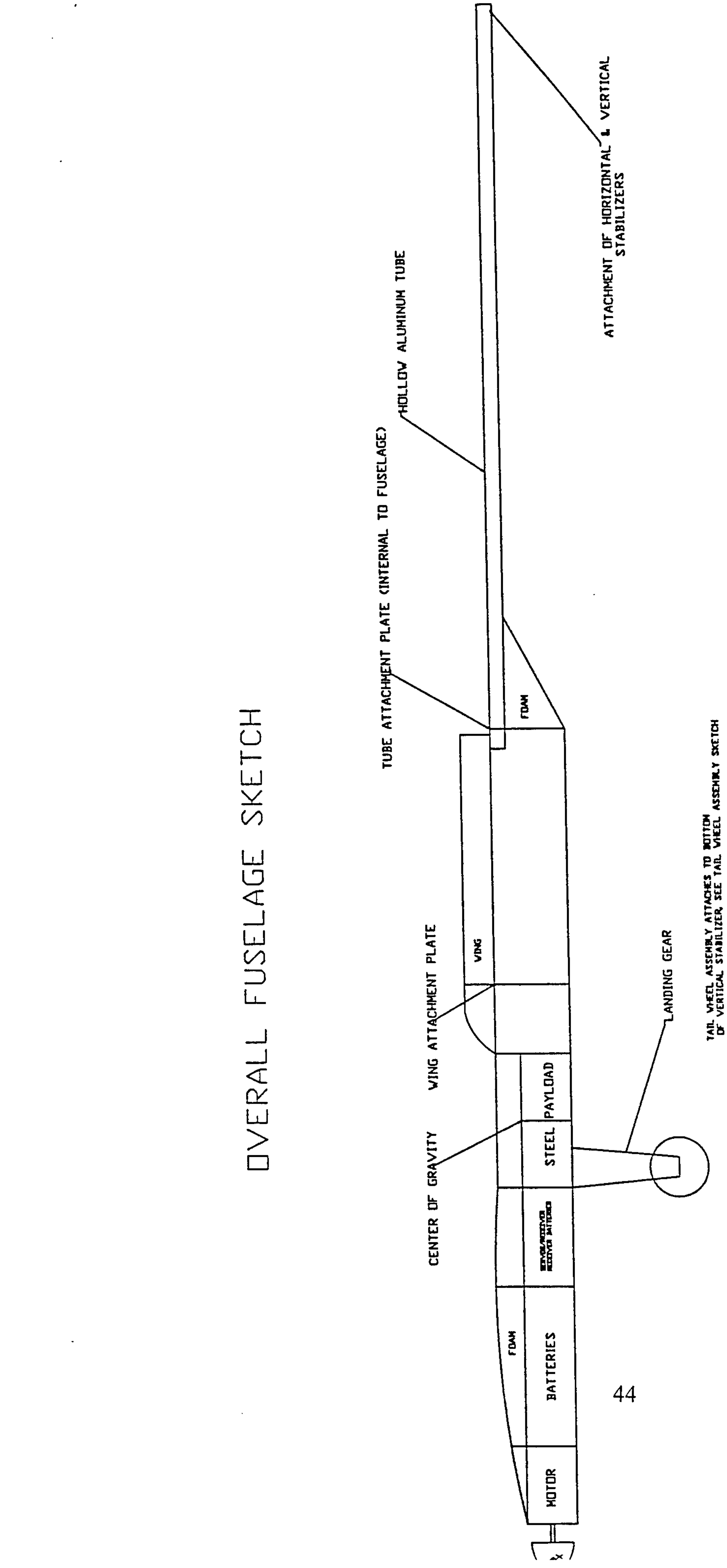
20% Wing Chord

12% Wing Area



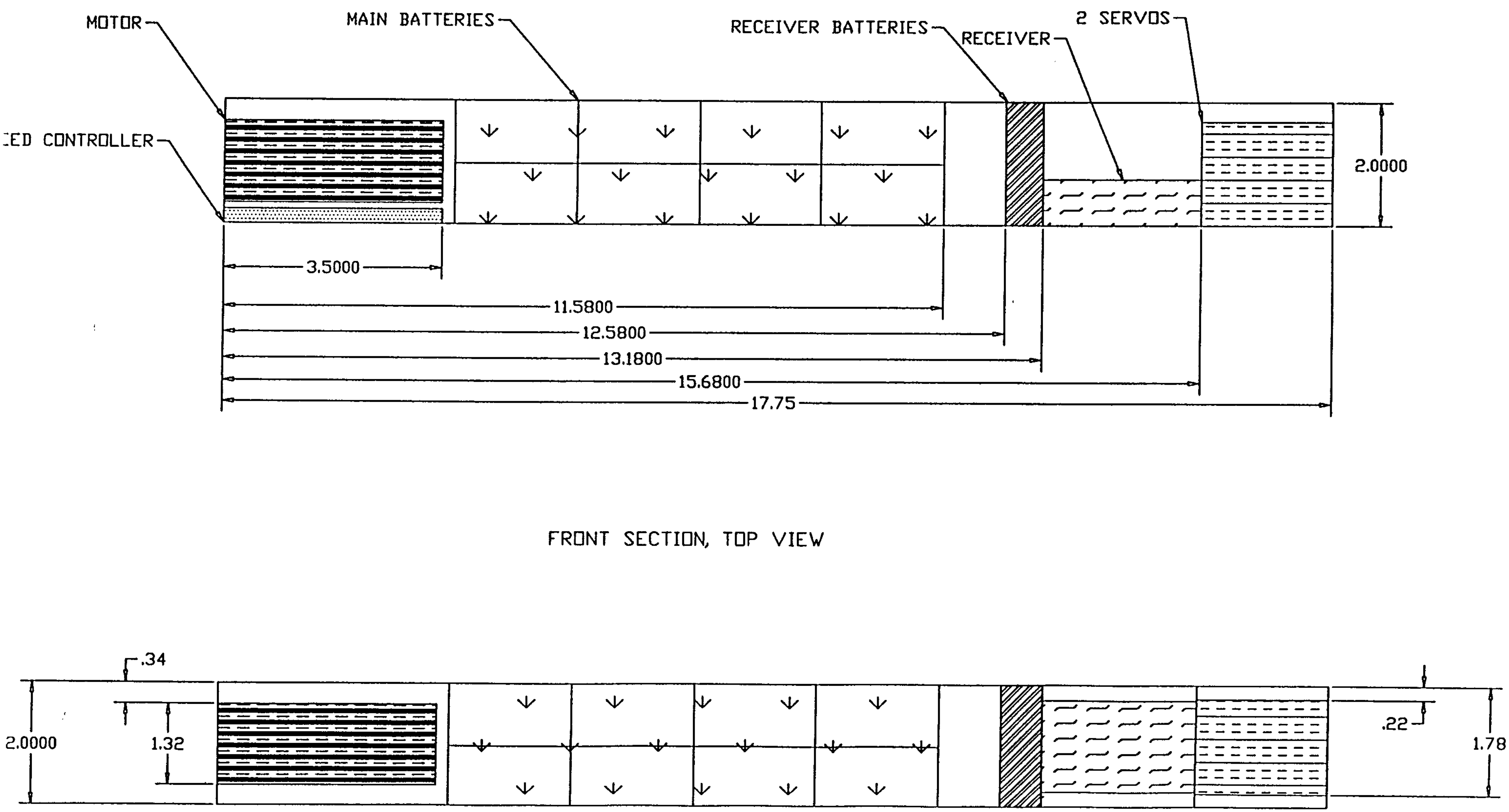
1.75''
1.5''
3.75''
7.0''
1.5''
3.0''
1.12''
3/8'' Dia Shear Pin

Appendix F Fuselage Layout



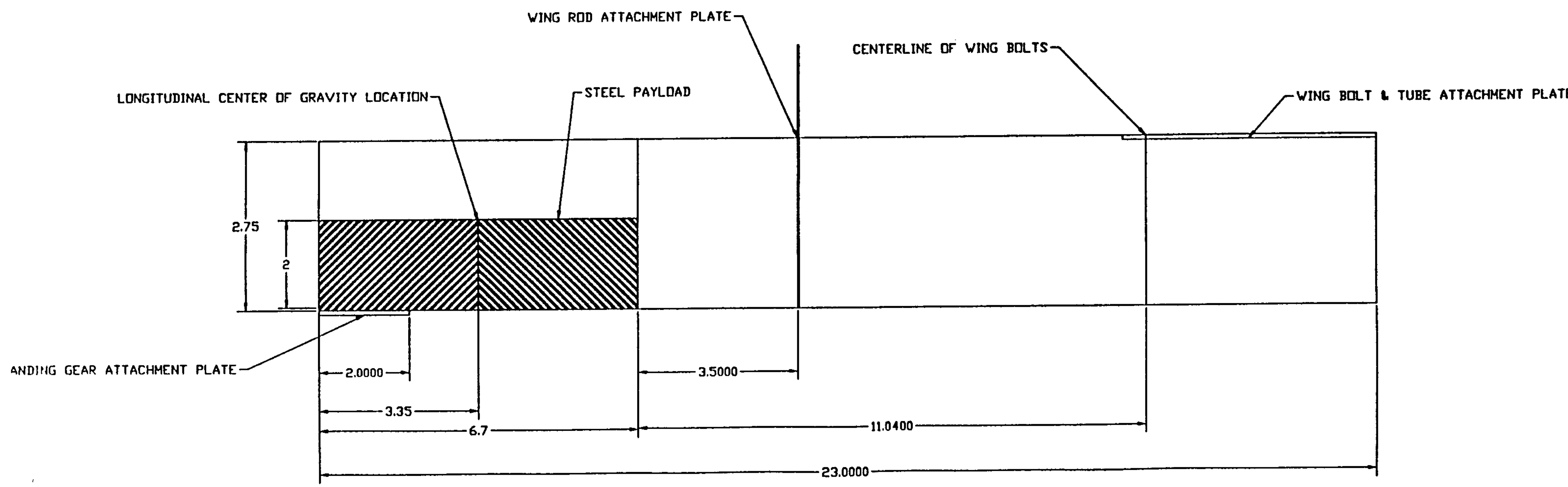
FUSELAGE STORAGE LAYOUT

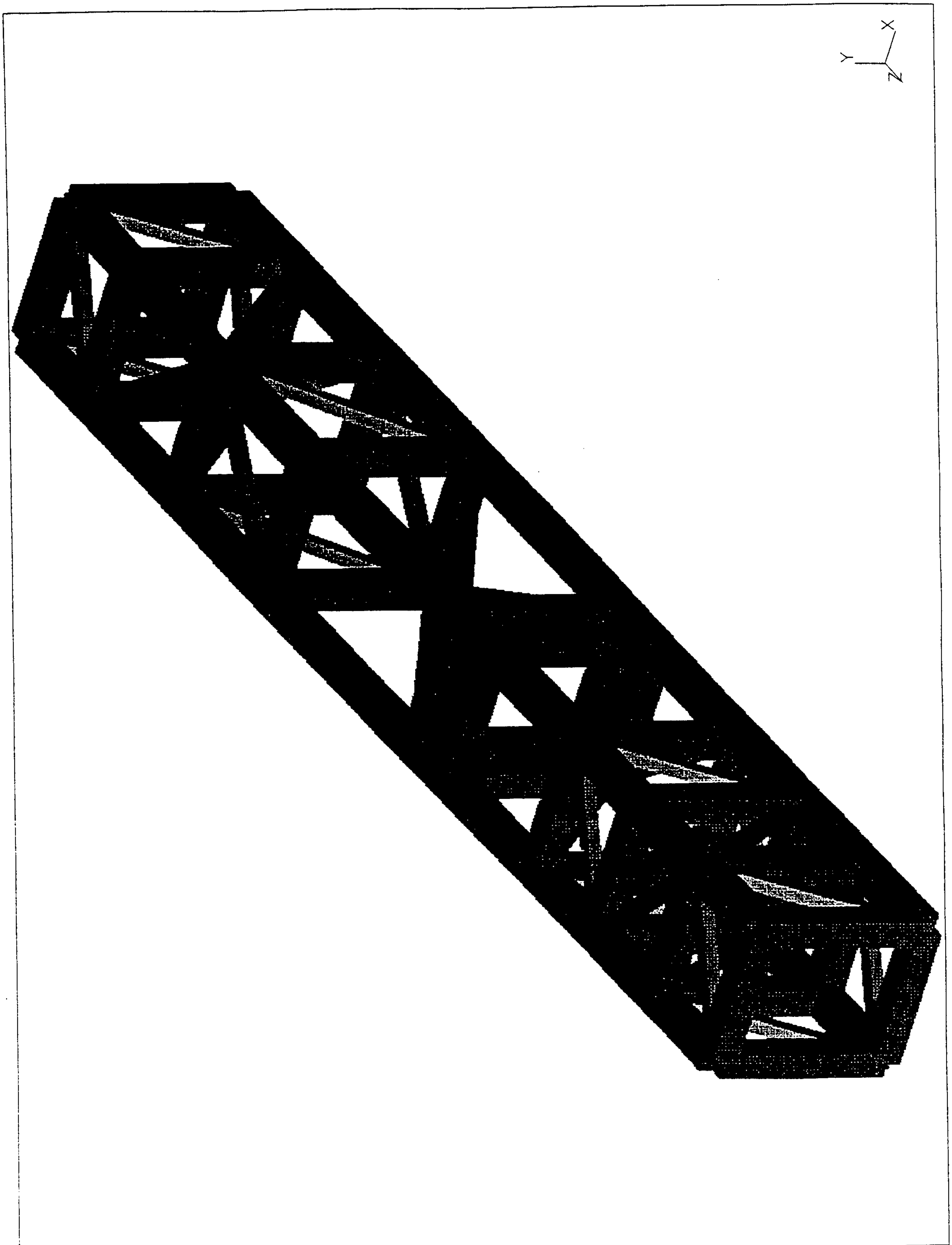
FRONT SECTION, SIDE VIEW
OUTER BOX DIMENSIONS INDICATE THE FUSELAGE INTERNAL CAVITY SIZE

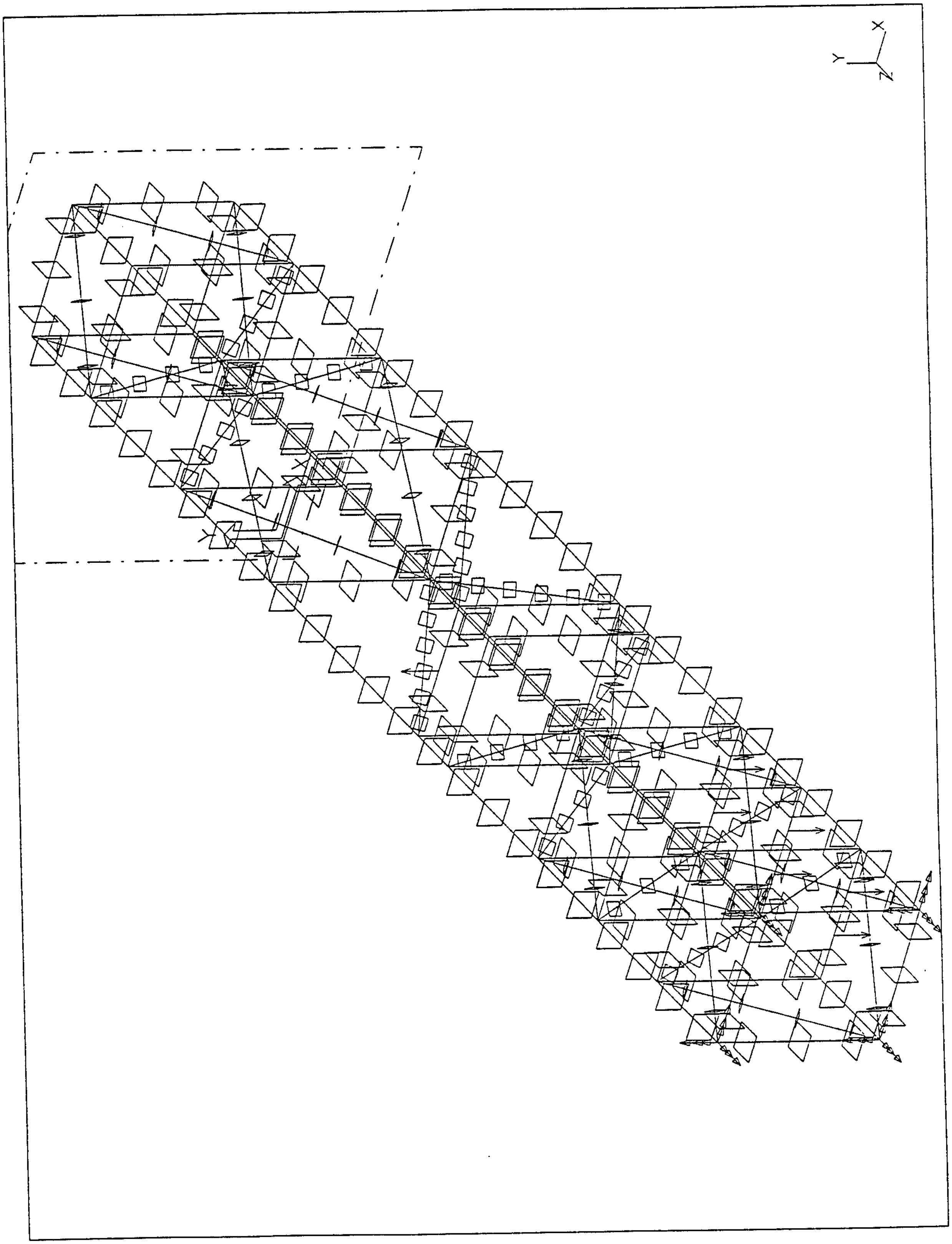


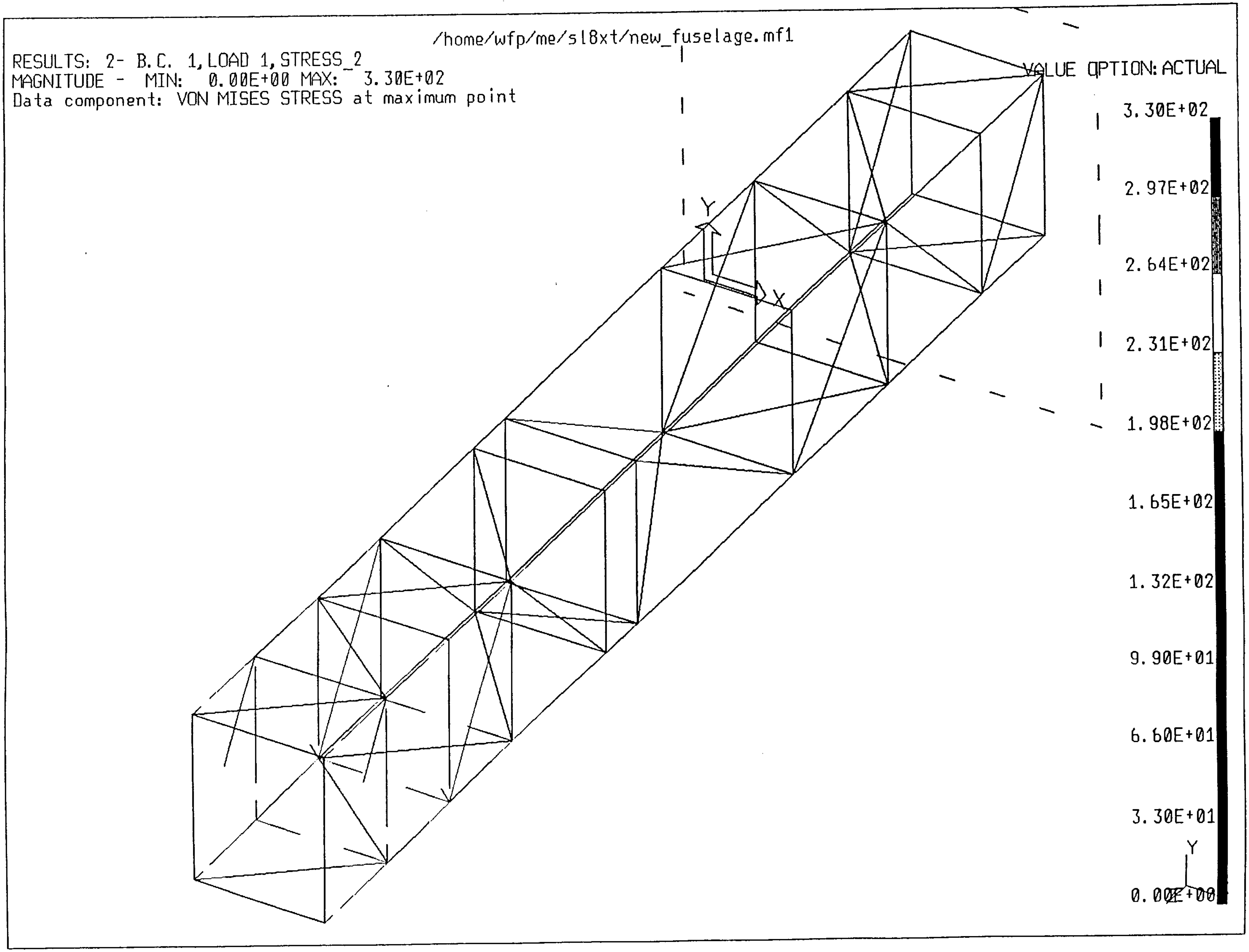
FUSELAGE STORAGE LAYOUT

MAIN SECTION, SIDE VIEW
OUTER BOX DIMENSIONS INDICATE THE SIZE OF FUSELAGE INTERNAL CAVITY



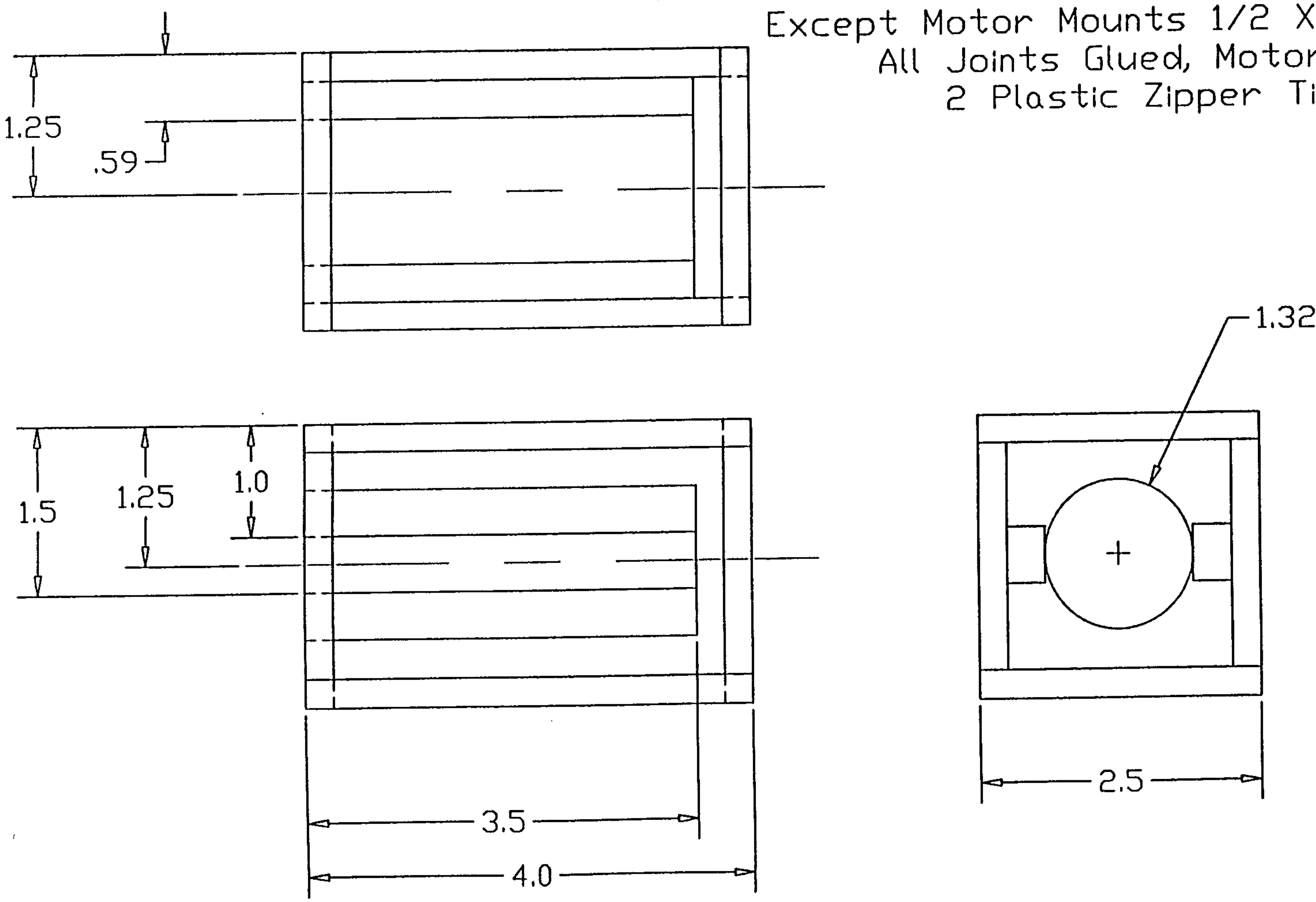

X
Y
Z


X
Y
Z

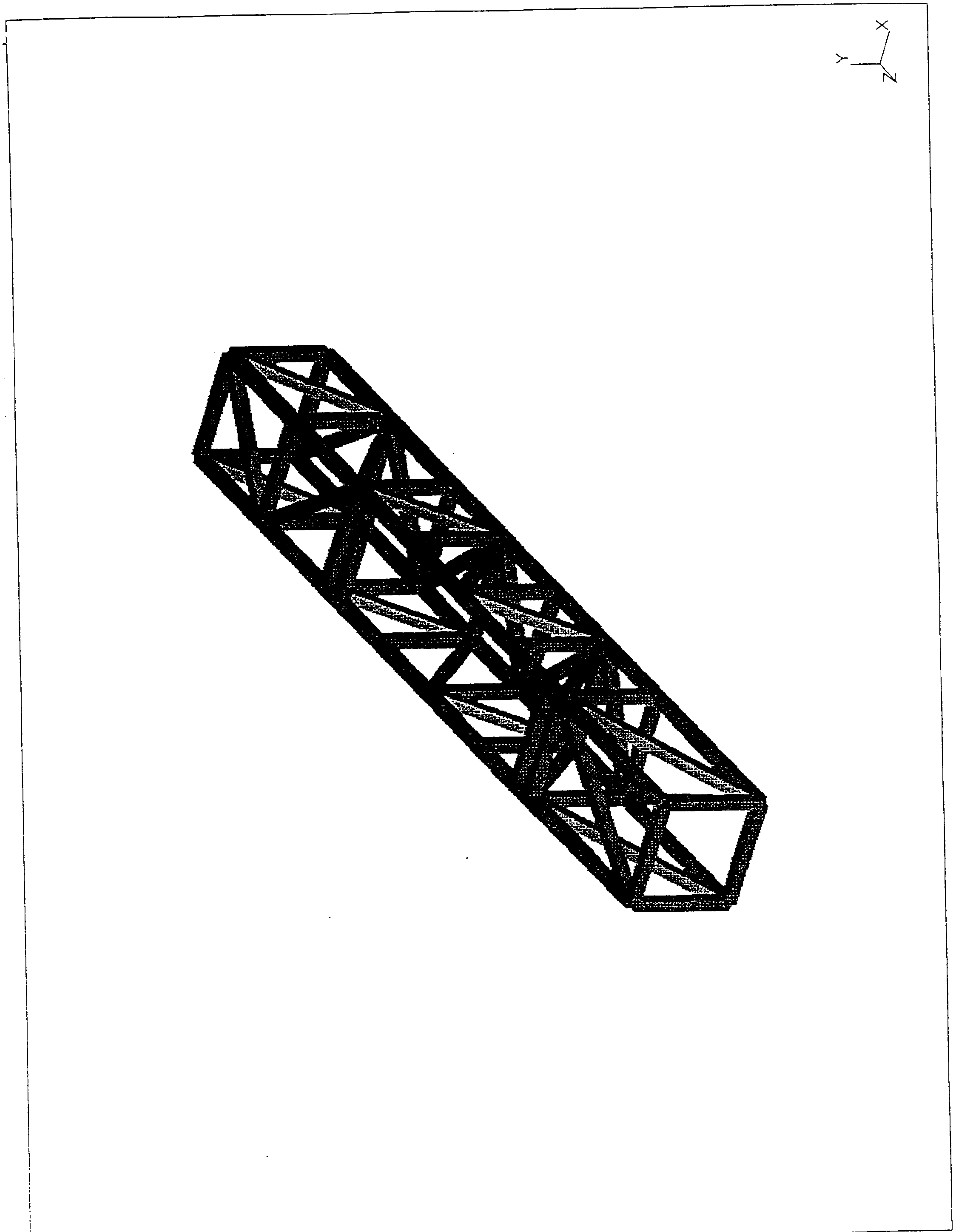

/home/wfp/me/sl8xt/new_fuselage.mf1
RESULTS: 2- B.C. 1,LOAD 1,STRESS_2
MAGNITUDE - MIN: 0.00E+00 MAX: 3.30E+02
Data component: VON MISES STRESS at maximum point
VALUE OPTION:ACTUAL
3.30E+02
2.97E+02
2.64E+02
2.31E+02
1.98E+02
1.65E+02
1.32E+02
9.90E+01
6.60E+01
3.30E+01
0.00E+00
Y
X
Y

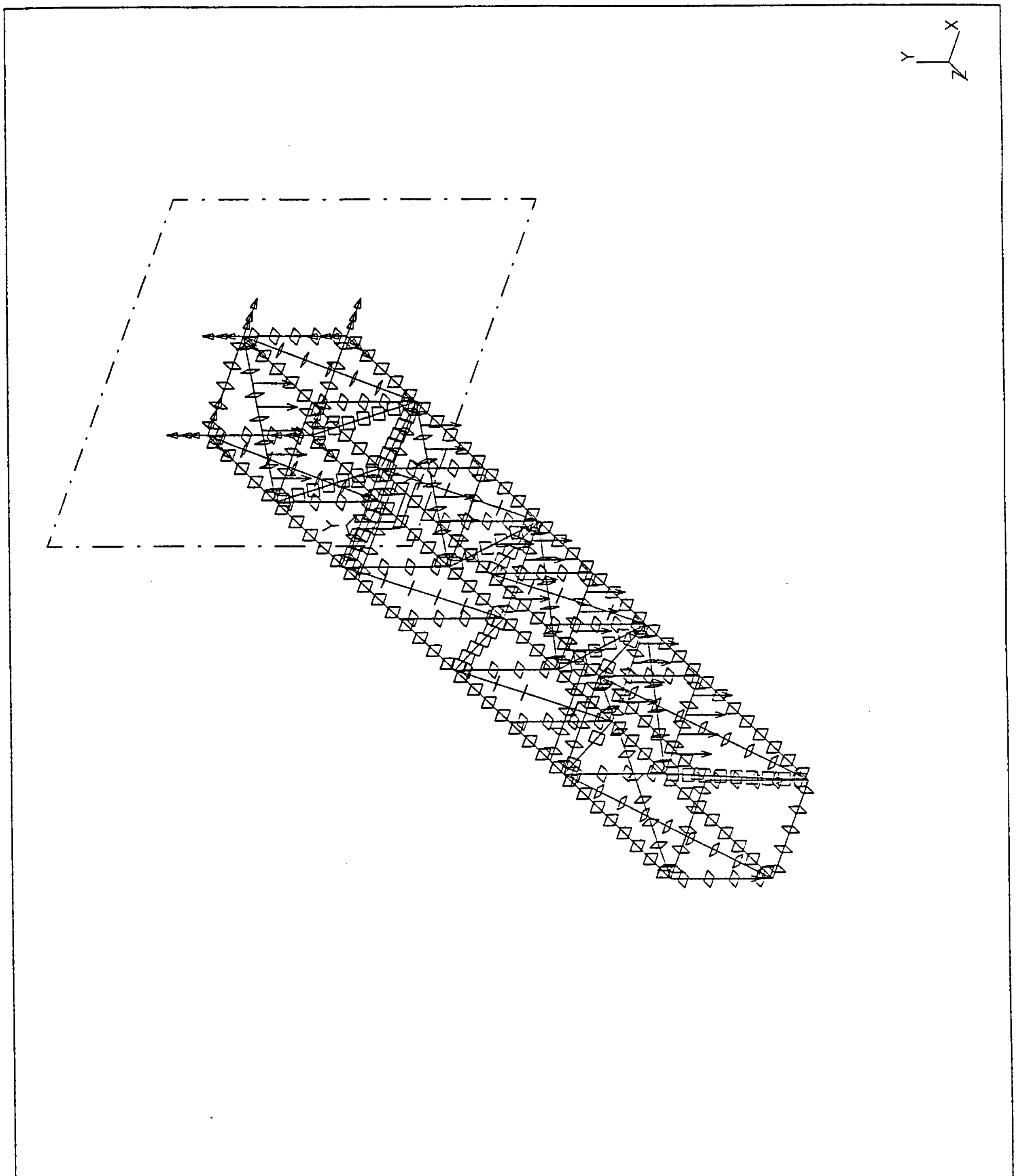
Motor Mount Assembly

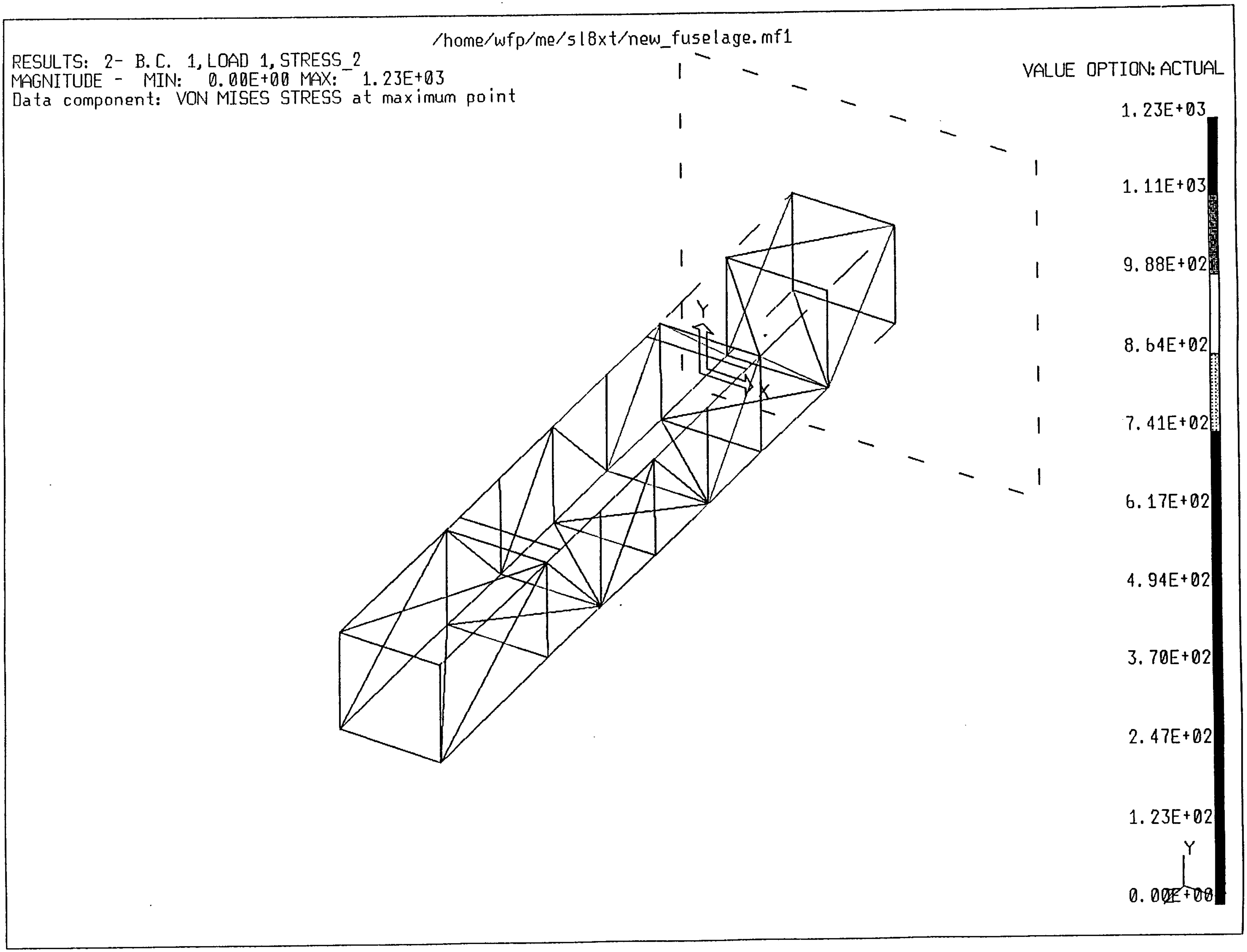
All Balsa Wood Members, 1/4 Inch Square
Except Motor Mounts 1/2 X 3/8 Inch Sanded to Fit
All Joints Glued, Motor Mount Secured by
2 Plastic Zipper Ties



***fuselage cross-members not shown for clarity


X
Y
Z


X
Y
Z


/home/wfp/me/sl8xt/new_fuselage.mf1
RESULTS: 2- B.C. 1,LOAD 1,STRESS_2
MAGNITUDE - MIN: 0.00E+00 MAX: 1.23E+03
Data component: VON MISES STRESS at maximum point
VALUE OPTION:ACTUAL
1.23E+03
1.11E+03
9.88E+02
8.64E+02
7.41E+02
6.17E+02
4.94E+02
3.70E+02
2.47E+02
1.23E+02
0.00E+00
Y
X

Wing Mounting Plate

Light Model Aircraft Plywood 1/32 inch thickness

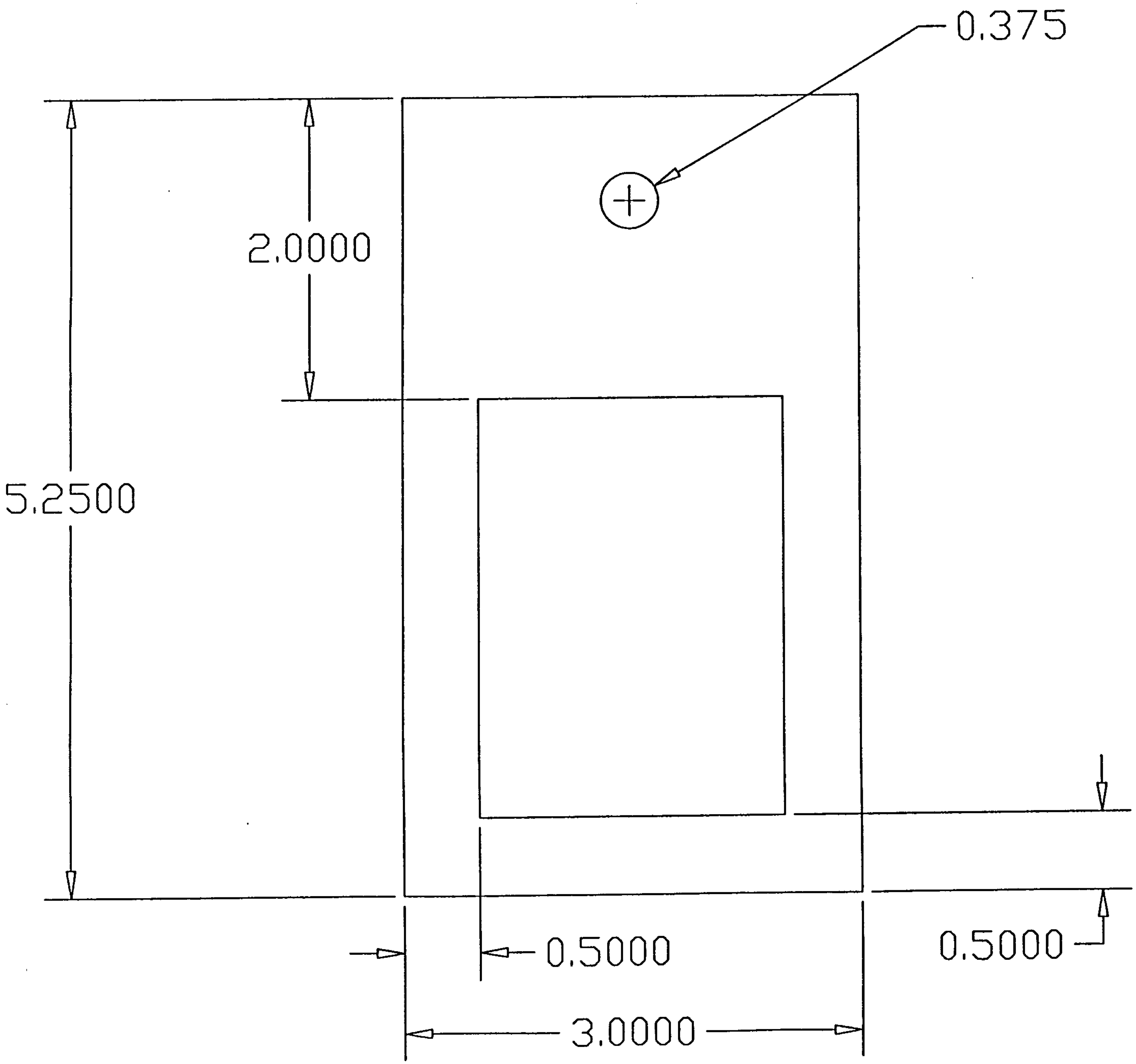


Figure F1

Ashley Childs | Aggie Arrow, Inc. | March, 1997

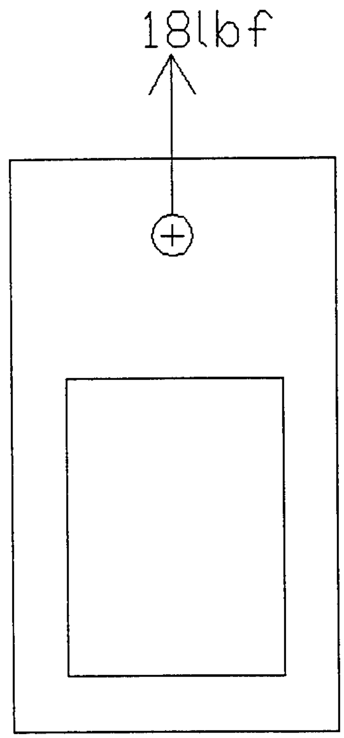


Diagram of the plate where the wing dowel will attach tot he fuselage. This calculation will show that this plate will not fail in tension with a stress concentration factor, nor will it fail due to the bearing of the dowel on the plate with the shown loading.

$w := 3 \cdot in$ — Diameter of the dowel connecting the wing to the fuselage.

$d := \frac{3}{8} \cdot in$ — Thickness of the plywood on the fuselage plate.

$t_{plywood} := \frac{1}{32} \cdot in$ — Maximum force exerted by the dowel on the hole.

$F := 18 \cdot lbf$ — Bearing area of the dowel on the plywood.

$A := d \cdot t_{plywood}$ — Tensile area of the plywood.

$A_{tens} := (w - d) \cdot t_{plywood}$

$a1 := \frac{d}{w}$ $\qquad a1 = 0.125$

$h := .5120 \cdot in - \frac{3}{8} \cdot \frac{1}{2} \cdot in$

$a2 := \frac{h}{w}$ $\qquad a2 = 0.233$

Figure F1

Ashley Childs | Aggie Arrow, Inc. | March, 1997

$K_t := 15$ — Stress concentration factor of the hole in the plywood, Figure A-XX, Design of Machinery textbook.

$\sigma_{bearing} := \frac{F}{A}$ $\quad \sigma_{bearing} = 1.536 \cdot 10^3 \cdot psi$ — Maximum bearing stress of the dowel on the plywood.

$\sigma_{tens} := \frac{F \cdot K_t}{A_{tens}}$ $\quad \sigma_{tens} = 3.291 \cdot 10^3 \cdot psi$

$\sigma_{max} := 5430 \cdot psi$ — Yield strength of the plywood, assuming Pine wood, from Table 26, Miner, Handbook of Engineering Materials, Wiley.

$SF := \frac{\sigma_{max}}{\sigma_{bearing}}$ $\quad SF = 3.535$

Figure F2

Ashley Childs — Aggie Arrow, Inc. — March, 1997

This calculation will determine the maximum force encountered during landing, in a worst case scenario. This will be determined from assuming that the airplane can be held at 6 inches off the runway and then stalled completely, to drop suddenly onto the ground from this height.

Assume that the landing gear behaves as a linear spring.

Sum Forces

$$-m\cdot g - k\cdot x - F_f = m\cdot V\cdot\frac{dV}{dx}$$

For conservatism, assume that the frictional force is zero.

$$-(m\cdot g) + k\cdot x = m\cdot V\cdot\frac{dV}{dx}$$

Integrate this equation by separation of variables.

$$-m\cdot g\cdot x - \frac{1}{2}\cdot k\cdot x^2 = m\cdot\frac{1}{2}\cdot\left(V^2 - V_o^2\right) + C$$

At zero spring deflection, V=Vo. And at maximum spring deflection, V=0.
Applying the first boundary condition, it is apparent that C=0.

Continuing for maximum spring deflection, where V=0:

$$-m\cdot g\cdot x + \frac{1}{2}\cdot k\cdot x^2 = -m\cdot\frac{1}{2}\cdot\left(V_o^2\right)$$

The potential energy of the plane at 6 inches off the ground is equivalent to the kinetic energy the plane would have as it hit the ground with an initial velocity.

$$mgh_o = \frac{1}{2}\cdot m\cdot V_o^2$$

Solve for Vo and plug this back into equation above.

$$V_o = \sqrt{2\cdot g\cdot h_o}$$

$$-m\cdot g\cdot x + \frac{1}{2}\cdot k\cdot x^2 = m\cdot\frac{1}{2}\cdot\left(2\cdot g\cdot h_o\right)$$

$$-2\cdot m\cdot g\left(x + h_o\right) + k\cdot x^2 = 0$$

$$k\cdot x^2 = 2\cdot m\cdot g\left(x + h_o\right)$$

Realizing that F=kx, and substituting this into the above equation, the expression for the maximum landing force is as follows:

$$F = 2\cdot m\cdot g\cdot\left(1 + \frac{h_o}{x}\right)$$

Figure F2

Ashley Childs Aggie Arrow, Inc. March, 1997

Assuming a maximum of 2.5 inches deflection of the landing gear (it is 5.5 inches plus an inch for wheels off the ground now), and a 6 inch initial height off the runway (from which the plane would be dropped):

$m := 18 \cdot lb$

$h_o := 6 \cdot in$

$x := 2.5 \cdot in$

$F := 2 \cdot m \cdot g \cdot \left(1 - \frac{h_o}{x}\right)$

$F = 122.4 \cdot lbf$

$Ratio := \frac{F}{18 \cdot lbf}$

$Ratio = 6.8$

Therefore, upon landing, forces of 6.8 times the weight of the plane could be encountered, in a worst case landing scenario.

The design of the fuselage must also withstand the maximum force encountered in landing, and the design criteria for the fuselage main body strength will be to withstand a 6.8 g loading.

Figure F 3

Ashley Childs | Aggie Arrow, Inc. | March, 1997

Landing Gear Deflection Calculation

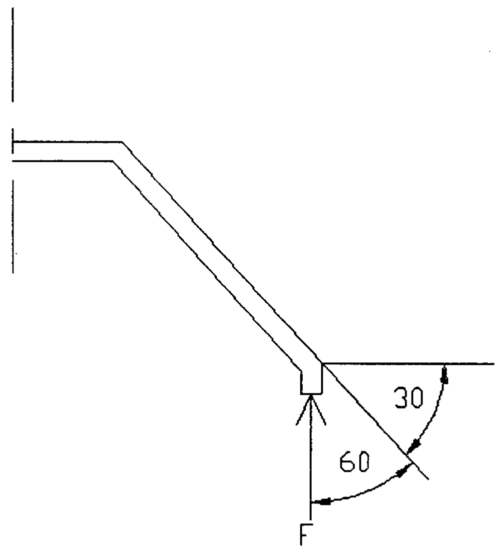
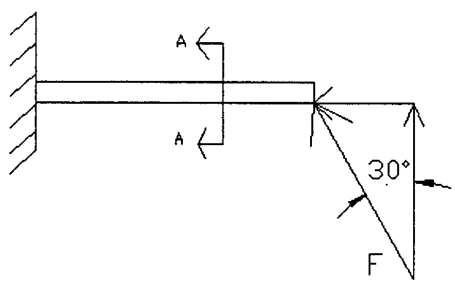
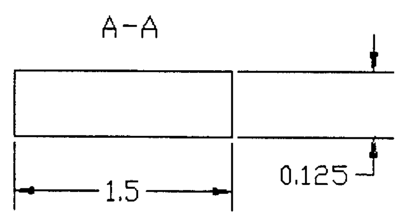


Diagram of half the Landing Gear. Force F is the landing force. The wheel is not shown and conservatively not included in the design. This could be modeled as a cantilever beam with a force applied at an angle, to determine maximum deflections of the landing gear during a hard landing. Testing was performed to determine the strength of the landing gear, which is not considered in this calculation. The landing gear is rated for a 15 pound airplane, and based on this, the testing, and probability of the hard landing scenario, the gear will not fail during a worst case scenario.

Cantilevered beam model of the landing gear with the force applied at an angle.





$F := \frac{18 \cdot lbf \cdot 6.8}{2}$ — Landing force encountered in a "hard" landing, half of this per side.

$P := F \cdot \cos(30 \cdot deg)$ $\quad P = 53.001 \cdot lbf$ — Component of the landing force causing bending and deflection upward. Axial deflections are neglected and not considered important.

$L := 8.5 \cdot in$ — Length of the angled side of the landing gear (length of cantilevered beam).

$E := 10.3 \cdot 10^{6} \cdot psi$ — Modulus of Elasticity of the landing gear.

$I := \frac{(1.5 \cdot in) \cdot (0.125 \cdot in)^{3}}{12}$ — Moment of Inertia of the cross-section A-A.

$\delta := \frac{P \cdot L^{3}}{3 \cdot E \cdot I}$ $\quad \delta = 4.315 \cdot in$ — Maximum deflection of the landing gear.

This deflection is acceptable since it will not bottom out and allow the fuselage to scrape. Also this is consistent with the assumption made in the hard landing force calculation.

Figure F3

Ashley Childs Aggie Arrow, Inc. March, 1997

This calculation will show that the tube used to connect the fuselage main section to the tail will not fail in two worst case scenarios: 1) Maximum Yawing moment caused by sudden in-flight deflection of the rudder to 45 degrees. 2) Maximum Pitching moment caused by sudden in-flight deflection of the elevator to a minus 45 degrees. Scenario one will cause a torque on the tube, and the second will cause a bending moment. This calculation assumes that when these moments are applied, that the airplane remains fixed and the tube is therefore a cantilevered beam.

$G := 26 \cdot 10^9 \cdot Pa$ — Material properties for Aluminum, Advanced Mechanics of Materials textbook.

$S_y := 103 \cdot 10^6 \cdot Pa$

$E := 71 \cdot 10^9 \cdot Pa$

$r_o := \frac{.75 \cdot in}{2}$ $\quad r_o = 0.375 \cdot in$ — Outer radius of the tail tube.

$r_i := r_o - \frac{3}{32} \cdot in$ $\quad r_i = 0.281 \cdot in$ — Inner radius of the tail tube.

$V := 35 \cdot mph$ — Level flight speed of the airplane.

$\rho := 1.225 \cdot \frac{kg}{m^3}$ — Density of air at sea level.

$S := 10 \cdot ft^2$ — Wing area.

$L := 12 \cdot in$ — Mean chord length of the wing.

$C_{yaw\_max} := 1.072$ — Maximum yawing moment predicted.

$C_{pitch\_max} := 1.036$ — Maximum pitching moment predicted.

$T := \frac{1}{2} \cdot \rho \cdot V^2 \cdot S \cdot C_{yaw\_max} \cdot L$ — Torque induced in the tube by the yawing moment.

$T = 402.86 \cdot in \cdot lbf$

$J := \frac{\pi}{2} \cdot \left(r_o^{\,4} - r_i^{\,4}\right)$ — Polar moment of inertia.

$\tau := \frac{T \cdot r_o}{J}$ $\quad \tau = 7.114 \cdot 10^3 \cdot psi$ — Maximum shear stress in the tube from the torque applied.

$SF_{torque} := \frac{\frac{S_y}{2}}{\tau}$ $\quad SF_{torque} = 1.05$ — Factor of Safety for the torsional failure of the tube.

$\theta := \frac{T}{J \cdot G}$ $\quad \theta = 0.005 \cdot \frac{1}{in}$ — Maximum rotation per unit length of the tube due to the torque.

$\theta_{max} := \theta \cdot 36 \cdot in$ $\quad \theta_{max} = 10.377 \cdot deg$

Figure F3

Ashley Childs | Aggie Arrow, Inc. | March, 1997

$I = \frac{\pi}{4} \cdot \left(r_o^4 - r_i^4 \right)$ — Moment of inertia about the axis of bending.

$M = \frac{1}{2} \cdot \rho \cdot V^2 \cdot S \cdot C_{pitch\_max} \cdot L$ — Maximum moment induced in the tube by the pitching moment.

$M = 389.331 \cdot in \cdot lbf$

$\sigma = \frac{M \cdot r_o}{I}$ $\quad \sigma = 9.481 \cdot 10^7 \cdot Pa$ — Maximum bending stress in the tube from the pitching moment applied.

$SF_{bend} = \frac{S_y}{\sigma}$ $\quad SF_{bend} = 1.086$ — Factor of Safety for bending failure of the tube.

$y = \frac{M \cdot (36 \cdot in)^2}{2 \cdot E \cdot I}$ $\quad y = 2.308 \cdot in$ — Maximum deflection of the tube.

Figure F4

Ashley Childs — Aggie Arrow, Inc. — March, 1997

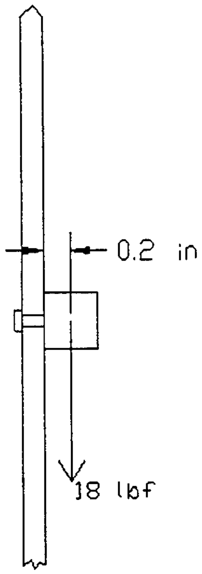


Diagram of the airplane banked at 90 degrees, the weight of the plane acting at the center of gravity, which is 0.2 inches from the bolt. The bolt is exposed to a moment and has a bending stress in it. This calculation will show that the bolt will not fail in this worst case scenario.

$F = 18 \cdot lbf$ — Force.

$r = .2 \cdot in$ — Moment arm of the force.

$M = F \cdot r$ — Moment induced in the bolts in a 90 degree banked turn.

$d_{bolt} = .25 \cdot in$ — Diameter of the bolt.

$c = \frac{1}{2} \cdot d_{bolt}$ — Distance to the outermost fiber of the bolt.

$I = \frac{\pi}{32} \cdot d_{bolt}^{4}$ — Moment of inertia of the bolts.

$\sigma_{bend} = \frac{M \cdot c}{I}$ — $\sigma_{bend} = 1.173 \cdot 10^{3} \cdot psi$ — Stress in the bolts in a 90 degree banked turn.

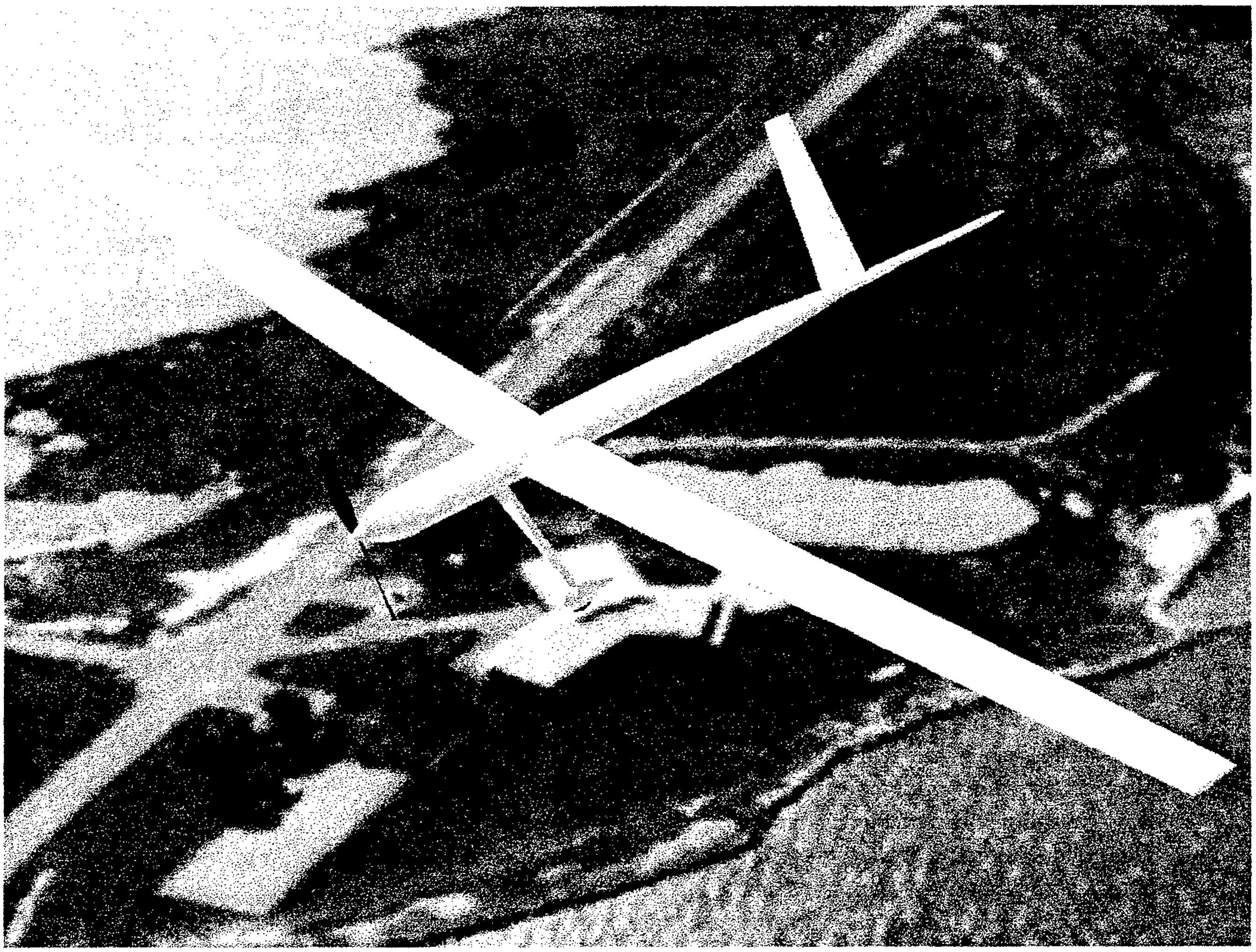
$\sigma_{max} = 13900 \cdot psi$ — Bending strength as determined by bending test of the bolts. Failure was not achieved and no permanent deformation occured for this stress level, therefore the yield strength is above this for the nylon bolts.

$SF = \frac{\sigma_{max}}{\sigma_{bend}}$ — $SF = 11.846$ — Safety factor on the bending failure of the bolt that secures the wing.

As a preclude to this report, I would like to explain that we did not finish our design soon enough to write this report. During the design, we did extensive analysis which we were not able to effectively explain here. So we wrote as much as we could, explaining the major areas of development. We would like to come and compete so that we can learn from the other designs, although we do not expect to win because of the report deficiencies.

AIAA Student Design / Build / Fly Competition

Virginia Tech Design Team
Design Report Addendum



Authors: Christopher Gunther, John Gundlach, Alexander Roup
Group Members: Geoffrey Buescher, Rebecca Gassler, Alex Knab, Matt Orr, Mike Russell, Michelle Werle

Addendum

Following the completion of Hokie Bird Mk.II several possible manufacturing process improvements were noted, a final cost analysis was completed, and future design modifications were suggested. This addendum addresses these and other issues that may affect the success of future endeavors.

The wing molds have room for improvement. First, due to time constraints, the plug was not made to the desired tolerances. Because the molded parts can not have a more accurate shape than the plugs, more effort should be concentrated in this area for future airframes. Pieces of wood wrapped in aluminum tape were used to divide the upper and lower half of the plug for mold construction. Small gaps between the wood and the wing allowed epoxy to seep through. The epoxy drainage made construction of the other mold half very difficult. Also, the epoxy used had bubbles present and caused voids in the surface of the mold used for part construction. The molds did not have enough layers of fiberglass to ensure rigidity. This lack of stiffness made the parts made in the molds warp slightly.

The flanges of the wing spar were changed from spruce to two layers of 5.6 oz. unidirectional carbon fiber. The shear webs were changed from spruce to vertical grain balsa wood with a thickness varying from 3/8" at the root to 1/8" at the wing tip. The first report mistakenly indicated that there was a single layer of 1.5 ounce/square yard plain weave cloth comprising the wing shell. What should have been said was that two layers of 1.5 ounce/square yard cloth with a layer of Spyder foam was used for the skins. The new wing construction technique used two 3 ounce/square yard plain weave fiberglass with fibers oriented at 45 degree angles. A layer of 1/16" thick Spyder foam rests between the two layers of fiberglass. The assumption of this wing structure is that the carbon fiber flanges will carry the tensile and compressive loads associated with wing bending moments, the shear webs will carry the shear loads from the wing spars, and the skin combined with the spar will resist torsion as a two-cell torsion tube.

The wing is made of one piece rather than the predicted 2 pieces. This was done because a one-piece wing is stronger for a given weight, is less complex, and has no slop associated with a wing rod. The available van allowed a one-piece wing, but if the aircraft had to be transported in a smaller vehicle, the wing would be required to detach into two or more pieces. The aircraft was designed to be competitive in the competition more than convenient to transport.

The wing aerodynamic design is sound and no major improvements are evident. The wing is the exact design specified in the earlier report. Perhaps a more extensive airfoil search would yield a more suitable section, but it is doubtful that significant gains could be made in this way. A better optimization routine could be implemented to specify the ideal twist, taper ratios, and taper change locations required to minimize induced drag at the design condition. The wing aerodynamic design used should be close to optimal given the chord and span constraints set.

The most significant modification that could be made to the design would be to change the tail configuration from a V-tail to a conventional one. A taildragger landing gear requires a steerable tailwheel for low-speed control on the ground. If a rudder were used, the tailwheel could be controlled on the same channel as the rudder with either the same servo or with a second servo. A V-tail requires that a separate tailwheel servo is mixed in though a separate channel than the rudder, leading to complex radio programming. Whether or not there is any drag reduction due to using a V-tail is debatable. The V-tail is less labor-intensive to build, but the complexity associated with the radio programming outweighs any advantages associated with this configuration.

Due to the experimental nature of the fuselage construction, some problems were inevitably encountered. There is a definite learning curve with the material and techniques used in the construction techniques used. The Chavant Modeling Clay is not the same as artist's clay, and it took much experimentation to get the proper tooling and molding methods. Dimensional accuracy was constantly a problem during the clay plug fabrication. Some fit-check templates were used to eyeball the overall straightness and size of the plug, but a more accurate procedure needs to be developed. The clay-working turned out to be more of an artisan crafting job than an exact science. However, with time, plug production approaches improved significantly. The core of the plug was Dow Blue-Board Insulation for ease of sizing and cost, but this did not give enough structural support for the outer layer of clay. New procedures involving wooden dowels and a lathe were used toward the completion of the plug. Overall, this technique seemed more accurate and provided a much better finish than the balsa core plug used in the building of Hokie Bird MKI.

The actual mold made of the American Gypsum Rayite worked extremely well. This method of mold creation was also new to the Virginia Tech RC Electric Airplane Team, but it resulted in a high-quality mold. A few corners were cut in the interests of time and money, but the mold material seemed to be extremely forgiving. The Rayite mold had several advantages over a fiberglass mold. It has a much higher stiffness, so mold deformation during part fabrication is not a problem. Also, the surface finish portion of the mold material gave a finer finish than any fiberglass mold used on the plane. Small imperfections in the mold were easily and quickly corrected with sanding and vinyl spackling. This ability alone is almost good enough to outweigh the use of a fiberglass mold. A two-part mold construction was used (the plug was split in the middle to make two half-molds). If a technique can be developed which involves only one mold, the construction process would be greatly simplified. Joining the two fuselage halves proved to be tricky but achievable. New methods of joining the halves should be developed to save weight and improve structural response. Stiffness in the completed part seemed to be an initial problem. This was corrected by incorporating a sandwich-type construction method with the molds. After a few layers of fiberglass were placed into the mold half, Spyder foam was placed along the longitudinal axis. More glass was then placed on the foam, forming a sandwich and resulting in a much stiffer and stronger part. The Spyder foam was used to give the fuselage the same characteristics as a much thicker fiberglass fuselage with less weight.

Once a finished fuselage was made, cut-outs, hatches, and mounting points created some problems. It was expected that the cut-outs would decrease the torsional rigidity of the structure, but the resulting decrease was beyond tolerable limits. Spruce bulkheads were then used to stiffen the structure. All cut-out corners were filleted to prevent tearing and delamination of the fuselage fiberglass. This was a lesson learned after the completion of Hokie Bird I. Commercially available landing gear, tires, and wheel shrouds were purchased to save time, but a better system could be made and custom tailored to the plane. Wing and tail attachment points were difficult to incorporate into the fuselage after it was finished. In future constructions, it would be much better to include the hard point attachments into the mold and make everything in one piece. Adding parts after the two halves were joined proved to be difficult and inefficient. Considering the many factors involved in the fuselage construction, the final product exceeded the specifications required by the original design goals.

The fuselage is not designed for adequate cooling of the batteries and speed control. A new fuselage design would incorporate cooling ducts and exits, or even use a radiator. The system would run much more efficiently if the batteries were kept cooler. The main concern is that the speed controller will overheat.

The time required to make the improvements suggested above would be slightly more than the time it took to build the first airframe. The manufacturing changes involve spending more time on the plugs and molds to attain higher quality. The design changes would require roughly half the time for the design of the competition aircraft. Gains from redesign and better molded parts would probably be modest. Most of the cost associated with these improvements is from composite materials and epoxy. This cost should be roughly $200.

The cost of all the materials, tools, and supplies combined is greater than that first estimated. The quantity of composite materials needed was grossly underestimated. Also, higher quality servos were used which were twice as expensive.

A thorough cost analysis is listed below. Some items are listed as a group and an approximate cost estimate for the group is given. Tools are listed separately from parts used for the aircraft.

Item Description	Qty.	Price	Total
Control Horns	2	$0.70	$1.40
Threaded Couplers	5	$0.75	$3.75
2-56 Clevises w/Rod	1	$7.79	$7.79
2-56 Solder Clevises	1	$4.99	$4.99
5/32" Wheel Collars	1	$1.15	$1.15
1' Tail Wheel	1	$1.49	$1.49
Lead Weights	1	$2.89	$2.89
Wing Seating Tape	1	$2.89	$2.89
Velcro	3	$2.99	$8.97
Landing Gear Straps	1	$0.80	$0.80
CA Hinges	2	$2.99	$5.98
¼-28 Wing Bolts	1	$1.15	$1.15
2-3/4" Wheels	1	$9.99	$9.99
Pull-Pull Cables	1	$5.29	$5.29
Semi-Flexible Pushrod	1	$4.29	$4.29
¼" Foam Rubber	1	$3.19	$3.19
2" Spinner	1	$3.29	$3.29
Futaba 7UAFGS	1	$300.00	$300.00
Mini, Coreless Servo	7	$61.99	$433.93
Trainer Cord	1	$13.79	$13.79
Servo Tape	2	$2.69	$5.38
Servo Extension Clip	5	$1.65	$8.25
Switch Mount	1	$2.59	$2.59
Charge Jack	1	$1.50	$1.50
Heat Shrink Tubing	1	$3.49	$3.49
Switch	1	$8.49	$8.49
Y- Harness	1	$10.29	$10.29
Extension	9	$4.99	$44.91
4-36 Cell Peak Charger	1	$119.99	$119.99
Medium CA	1	$7.49	$7.49
Accelerator	1	$4.49	$4.49
Thin CA	1	$7.49	$7.49
Kevlar Ribbon	1	$2.39	$2.39
5/32" LG Wire	1	$6.69	$6.69
Box-A-Balsa	1	$25.99	$25.99
Spray Paint	4	$7.49	$29.96
4x8' Dow Blue Board Foam	4	$25.00	$100.00

E-Z Lam Epoxy System 1.5 Gal	1	$92.00	$92.00
West Systems Epoxy System	1	$92.00	$92.00
Fiberglass	1	$100.00	$100.00
Carbon Fiber	1	$45.00	$45.00
Spyder Foam 1x4' x1/16"	6	$5.00	$30.00
19 Cell Battery Pack	1	$140.00	$140.00
Aveox Motor/Speed Control	1	$350.00	$350.00
Gearbox	1	$45.00	$45.00
Props/Prop Accessories	1	$35.00	$35.00
Parts and Supplies Total			$2,135.41
Assorted Tools	1	$300.00	$300.00
Vacuum Bagging System	1	$225.00	$225.00
Vacuum Bagging Supplies	1	$30.00	$30.00
Formica (Templates)	1	$20.00	$20.00
Feather Cut System	1	$149.50	$149.50
Thermal Generator	1	$119.50	$119.50
52" Bow	1	$39.00	$39.00
Hot Wire	4	$3.00	$12.00
22" Sanding Block	1	$6.49	$6.49
150 Grit Sandpaper Roll	1	$5.99	$5.99
Mighty Wire Bender	1	$16.49	$16.49
100 Watt Soldering Iron	1	$14.69	$14.69
¼"-20 Tap and Drill	1	$5.29	$5.29
Tap Handle	1	$4.69	$4.69
Tools Total			$948.64
Total			**$3,084.05**

This is the first time in several years that the Virginia Polytechnic Institute and State University Aerospace Engineering Department has entered a competition involving both design and construction of a flying aircraft. As a result, there was difficulty in fostering initial interest. Also, commitment from seniors was difficult because Virginia Tech requires that seniors be involved in a very time-consuming design project. Many of the underclassmen were not able to participate in the design portion of the project due to a lack of experience and knowledge. Possible solutions include using this competition as a senior design project or to incorporate it into the curriculum in another manner.

AIAA Student Design/Build/Fly Competition

Addendum

Submitted by:

The University of Alabama Chapter

Aircraft Name: TK421

4/9/97

Addendum Phase

The final phases of construction brought about many changes from the proposed design. These include changes to the spars, longerons, and tail surfaces. In the proposal phase, a clerical error was made regarding the size of the longerons. The longerons were made out of 3/8in. x 1/4in. spruce sticks. Additionally, the fuselage was changed to a diameter of 8 in. This eliminates the 9in. x 5in. ellipse. This ellipse would be more difficult to construct.

There were two placement changes in the final design. The first section of the rear spar was aligned such that it is parallel to the front spar. The second section of the rear spar is aligned with the front of the aileron. In the proposal the rear spar is aligned with the front of the aileron for the entire span of the wing. This change was made in order to more effectively share the wing loading with the front spar. Next, the rudder and the elevator placement was changed. In the proposal, the pictures show the rudder directly over the elevator. This would cause interference during deflection. To eliminate this interference, the horizontal tail was positioned such that the elevator begins at the end of the fuselage, and the vertical tail was positioned such that the back edge of the rudder meets the front edge of the elevator.

After some additional consideration, carbon fiber tape was added to the top and the bottom of the spars. This added only a minimal amount of weight, but significantly increased the margin of safety for the aircraft. This was done to more adequately accommodate the flight loads.

The tail surfaces were also modified for the final construction. The airfoil design for the tails was changed to a flat plate design. This design was chosen to help reduce weight and drag in addition to making the construction easier. Since the incidence angle was so small (0.3°) this angle was kept for the final design. The pilot will need to determine the proper trim conditions for flight.

Construction began on March 13, and was due for completion in two weeks. This estimate for construction time was not accurate. If this design is implemented in the future, a construction time of 3 weeks should be allocated. While this first construction took longer than 3 weeks, there were some areas in the construction planning which needed improvement. These areas included the tail and fuselage connections, payload mounting, and landing gear mounts. With the current plans, the aircraft should make a 3 week construction deadline. The total cost of the aircraft was $780.00. It was estimated that an extra $25.00 was spent on materials that were inadequate. This would give a total manufacturing cost of $755.00 for this aircraft. A complete list of the manufacturers list prices for all the materials used in this design can be found in Table 1. The cost would not change significantly if the proposed changes were made. The original cost was estimated to be around $600.00, however, the original estimated cost of the batteries and the miscellaneous equipment was not accurate.

Table 1: Manufacturers List Prices

Aircraft Plywood, Balsa, and Spruce:

SIG Manufacturing Co., Inc.	
1; Plywood Wingskin 1/64in. x 24in. x 48in. *	$38.25
1; Plywood Wingskin 1/64in. x 24in. x48 in.	$9.60
16; Balsa Sheets 1/8in. x 6in. x36 in.	$50.56
31; Spruce Sticks 1/4in. x 1/4in. x 48in.	$25.42
8; Spruce Sticks 1/4in. x 3/8in. x48in.	$7.84
1; Plywood Sheets 1/4in. x 6in. x 12in. (5-ply)	$2.25
1; Plywood Sheets 1/8in. x 12 in. x 48in. (Lite-ply)	$8.85
1; Plywood Sheets 1/32in. x 12 in. x 48in (3-ply) *	$15.95
*These items were damaged during shipping and were redelivered	
Model Box	
1; Plywood Sheet 1/4in. x 12in. x 4in.	$16.45
1; Plywood Sheet 1/4in. x 12in. x 2in.	$8.60

Adhesive, Epoxy, and accessories

SIG Manufacturing Co., Inc.	
1; Core-Bond (pint)	$9.95
TNT Hobbies	
1; 5 minute Epoxy	$6.49
2; CA	$6.58
Model Box	
1; CA	$5.49
1; Epoxy Brushes	$1.35

Materials for Wing Jig

Lowe's	
1; 8 foot 1x6	$5.48
1; 8 foot 1x4	$3.77
1; box of ¼ in. wing nuts	$5.42
26; Carriage Bolts	$3.90

Motors and Accessories

Hobby Lobby International

2; 10 in., 2 blade, pusher prop	$13.40
2; Graupner 600 FG3 MTR/GEAR	$51.95
2; Propshaft adapter	$26.40
2; Propshaft extenders	$7.40

Battery Packs

Cermark Electronics

2; 20, 1.2 Volts, 2000 mah batteries	$118.00

Servos/Pushrods

Hobby Lobby International

2; standard servos	$79.90
2; micro servos	$139.90
1; Complete Flexcable Pushrods	$5.99

Covering

Hobby Lobby International

6; rolls Superkote	$77.70

Sandpaper

Donated

*List Prices quoted from Crossroads Hardware

27; pieces of various grades	$.58-$.85 each

Miscellaneous

Hobby Lobby International

1; Clear Hinge Tape (30 feet)	$4.95
2; 3in Airwheels (pair)	$11.80
13; rolls of 1/4x48 in Carbon Fiber Tape	$35.75
1; pack of wheel locks	$4.99
2; Beckman Concealed Actuators	$19.60
1; pack Nylon Pin Hinges	$4.99

The following items were borrowed from the Aerospace Engineering and Mechanics Department. These list prices correspond to Hobby Lobby International:

1; Battery Charger	$37.90
1; Motor Controller	$69.00
1; Radio	$165.00

AIAA Student Design/Build/Fly Competition

Addendum

Submitted by:

The University of Alabama Chapter

Aircraft Name: TK421

4/9/97

Addendum Phase

The final phases of construction brought about many changes from the proposed design. These include changes to the spars, longerons, and tail surfaces. In the proposal phase, a clerical error was made regarding the size of the longerons. The longerons were made out of 3/8in. x 1/4in. spruce sticks. Additionally, the fuselage was changed to a diameter of 8 in. This eliminates the 9in. x 5in. ellipse. This ellipse would be more difficult to construct.

There were two placement changes in the final design. The first section of the rear spar was aligned such that it is parallel to the front spar. The second section of the rear spar is aligned with the front of the aileron. In the proposal the rear spar is aligned with the front of the aileron for the entire span of the wing. This change was made in order to more effectively share the wing loading with the front spar. Next, the rudder and the elevator placement was changed. In the proposal, the pictures show the rudder directly over the elevator. This would cause interference during deflection. To eliminate this interference, the horizontal tail was positioned such that the elevator begins at the end of the fuselage, and the vertical tail was positioned such that the back edge of the rudder meets the front edge of the elevator.

After some additional consideration, carbon fiber tape was added to the top and the bottom of the spars. This added only a minimal amount of weight, but significantly increased the margin of safety for the aircraft. This was done to more adequately accommodate the flight loads.

The tail surfaces were also modified for the final construction. The airfoil design for the tails was changed to a flat plate design. This design was chosen to help reduce weight and drag in addition to making the construction easier. Since the incidence angle was so small (0.3°) this angle was kept for the final design. The pilot will need to determine the proper trim conditions for flight.

Construction began on March 13, and was due for completion in two weeks. This estimate for construction time was not accurate. If this design is implemented in the future, a construction time of 3 weeks should be allocated. While this first construction took longer than 3 weeks, there were some areas in the construction planning which needed improvement. These areas included the tail and fuselage connections, payload mounting, and landing gear mounts. With the current plans, the aircraft should make a 3 week construction deadline. The total cost of the aircraft was $780.00. It was estimated that an extra $25.00 was spent on materials that were inadequate. This would give a total manufacturing cost of $755.00 for this aircraft. A complete list of the manufacturers list prices for all the materials used in this design can be found in Table 1. The cost would not change significantly if the proposed changes were made. The original cost was estimated to be around $600.00, however, the original estimated cost of the batteries and the miscellaneous equipment was not accurate.

Table 1: Manufacturers List Prices

Aircraft Plywood, Balsa, and Spruce:

SIG Manufacturing Co., Inc.	
1; Plywood Wingskin 1/64in. x 24in. x 48in. *	$38.25
1; Plywood Wingskin 1/64in. x 24in. x48 in.	$9.60
16; Balsa Sheets 1/8in. x 6in. x36 in.	$50.56
31; Spruce Sticks 1/4in. x 1/4in. x 48in.	$25.42
8; Spruce Sticks 1/4in. x 3/8in. x48in.	$7.84
1; Plywood Sheets 1/4in. x 6in. x 12in. (5-ply)	$2.25
1; Plywood Sheets 1/8in. x 12 in. x 48in. (Lite-ply)	$8.85
1; Plywood Sheets 1/32in. x 12 in. x 48in (3-ply) *	$15.95
*These items were damaged during shipping and were redelivered	
Model Box	
1; Plywood Sheet 1/4in. x 12in. x 4in.	$16.45
1; Plywood Sheet 1/4in. x 12in. x 2in.	$8.60

Adhesive, Epoxy, and accessories

SIG Manufacturing Co., Inc.	
1; Core-Bond (pint)	$9.95
TNT Hobbies	
1; 5 minute Epoxy	$6.49
2; CA	$6.58
Model Box	
1; CA	$5.49
1; Epoxy Brushes	$1.35

Materials for Wing Jig

Lowe's	
1; 8 foot 1x6	$5.48
1; 8 foot 1x4	$3.77
1; box of ¼ in. wing nuts	$5.42
26; Carriage Bolts	$3.90

Motors and Accessories

Hobby Lobby International

2; 10 in., 2 blade, pusher prop	$13.40
2; Graupner 600 FG3 MTR/GEAR	$51.95
2; Propshaft adapter	$26.40
2; Propshaft extenders	$7.40

Battery Packs

Cermark Electronics

2; 20, 1.2 Volts, 2000 mah batteries	$118.00

Servos/Pushrods

Hobby Lobby International

2; standard servos	$79.90
2; micro servos	$139.90
1; Complete Flexcable Pushrods	$5.99

Covering

Hobby Lobby International

6; rolls Superkote	$77.70

Sandpaper

Donated

*List Prices quoted from Crossroads Hardware

27; pieces of various grades	$.58-$.85 each

Miscellaneous

Hobby Lobby International

1; Clear Hinge Tape (30 feet)	$4.95
2; 3in Airwheels (pair)	$11.80
13; rolls of 1/4x48 in Carbon Fiber Tape	$35.75
1; pack of wheel locks	$4.99
2; Beckman Concealed Actuators	$19.60
1; pack Nylon Pin Hinges	$4.99

The following items were borrowed from the Aerospace Engineering and Mechanics Department. These list prices correspond to Hobby Lobby International:

1; Battery Charger	$37.90
1; Motor Controller	$69.00
1; Radio	$165.00

AIAA Student Design/Build/Fly Competition

Proposal

Submitted by:

The University of Alabama Chapter

Aircraft Name: TK421

3/13/97

Executive Summary

The first phase of the design was a conceptual phase. This phase included decisions about the overall configuration of the airplane with consideration given to the mission design parameters. A conventional, high aspect ratio wing planform and tail configuration was chosen. In the next phase, the preliminary design phase, components were sized for a given flight mission profile, payload, and maximum battery weight. The final phase of the design included a detailed analysis which was used to size internal structures. Once this phase was completed, the necessary materials were ordered and construction planning began. This planning included full scale detailed drawings used for construction of the various components of the airplane.

Several configurations were considered during the conceptual phase of the design. These included a canard configuration which utilizes the canard instead of the horizontal stabilizer, tractor versus pusher propulsion, high wing versus low wing, t-tail versus conventional tail, and tricycle versus bicycle landing gear. Design tools used for this phase of development range from faculty advice to use of design texts by Raymer, Roskam, and Bruhn.

During the preliminary design phase, a high aspect ratio wing was chosen to achieve the greatest efficiency. This was based on the design of the Gossamer Albatross in which range was the most important design parameter. Powered sailplanes also make use of high aspect ratio wings to produce a large amount of lift. The airfoils for the wing and tail were chosen from airfoils considered to have high lift to drag ratios for low speeds. An analysis was performed using computer software which included MathCad, Matlab, and Visual Basic. The analysis rendered the positioning and the sizing of the major aircraft components.

The major emphasis during the detailed design phase was placed on the sizing of internal structures such as the front and rear spars, frames, and longerons as well as rib and frame spacing. Decisions about construction breaks and component assembly were also made during this phase. Tools used during this phase included a variety of structures and mechanics text books.

Management Summary

The architecture of the design team changed as we worked through the different phases of the design. During the conceptual design phase, the team consisted of a chairperson who was responsible for organizing the team meetings, discussing team progress with the faculty advisor, and corresponding with the AIAA Design/Build/Fly contact. The team members worked in one of three areas of concentration. These areas were Cost, Airframes, and Battery and Motor. Each area of concentration met once a week, and there were weekly meetings for the entire team. Each area of concentration had a secretary who was required to attend the weekly team meetings. All other members were encouraged but not required to attend these meetings.

During the preliminary design phase, the team chose to have co-chairpersons. The cochairpersons met once a week to organize upcoming events. One chairperson was chosen to manage the design aspects, his responsibilities included guiding members through the calculations, drawings, and material ordering. The other chairperson dealt with the business aspects of the project. These responsibilities included organizing ideas for funding of the construction and travel, correspondence with the AIAA Design/Build/Fly contact, and organizing meetings. During this phase, there were biweekly meetings that all members were asked to attend. During these meetings, team members would learn about the current fund-raiser work on the design. Members were asked to do individual trade studies during this phase and to report their findings back to the team.

For the detailed design phase, the team began meeting three times a week as a whole. The cochairpersons continued to lead the design and business aspects along with meeting once a week. Besides the general meetings, several members worked in small groups during the week. This architecture has been maintained for the manufacturing phase of the project. During all design phases, members were given individual assignments to work on between team meetings. A milestone chart showing the planned and actual timing of several major elements of the design process can be seen in Appendix A.

Conceptual Design

Horizontal Control Surface

Initially a canard design was considered because airplanes with canards typically have less drag than airplanes with conventional tails. This is due to the replacement of the normal download that is carried by the tail with a lifting load on the canard. In addition to this advantage another advantage to the canard design is the stall characteristics. Typically, the canard stalls before the wing stalls giving the pilot an opportunity to adjust before stalling occurs on the control surfaces. Other known advantages such as a decrease of cabin noise is not applicable since the design is for an unmanned radio controlled aircraft.

The disadvantages to having a canard outweigh the advantages. While recoverable stalls are desirable in a design, there are many control related difficulties in such a design. When the wing stalls, only lift is lost; but when the canard stalls pitch control is momentarily lost. Canards make tight turns difficult especially in small thermals. More relevant to this competition, however, is the amount of runway that canard airplanes need. Normally, pilots will stall the airplane in order to land faster. However, if a pilot stalls a canard, pitch control is momentarily lost, and the nose dives toward the ground. In order to prevent stall on the canard while landing, the airplane's speed must be increased. This increase in speed may require 35% to 50% more runway. This increase in runway is also necessary for take off.

Propulsion

Propulsion was also considered during the conceptual design phase. Most conventional propeller aircraft utilize tractor propulsion. However, propeller wash over the wing is one major disadvantage for this type of propulsion. Propeller wash creates an area of flow disturbance causing turbulent flow conditions. These conditions cause an increase in drag which lowers the range of the aircraft. To eliminate propeller wash effects, a pusher propulsion system was chosen.

Wing Configuration

A high wing configuration was chosen over a low wing configuration for two main reasons. First, the motors are mounted on the wings limiting the space that the propeller has to spin. A high wing configuration provided ample space for the propeller to spin without striking the ground on take-off and landing. Without this space, longer heavier landing gear would be necessary to keep the propeller from striking the ground. Second, a high wing will allow for easy removal of the wings. To transport an aircraft with such a large span, the wings must be removable.

Tail and Landing Gear

The conventional tail was chosen over the t-tail configuration for this design. Through trade studies, it was determined that the t-tail is heavier than the conventional tail. This is due to the additional structure required to mount the horizontal stabilizer above the fuselage. One of the advantages of the t-tail is that the tail surfaces are less likely to stall when the wing stalls. The weight penalty that is incurred is greater than the advantage t-tails give. Two types of fixed landing gear (tricycle and bicycle) were considered for this design. The fixed tricycle gear was chosen over the bicycle gear configuration primarily due to issues involving landing handling qualities. Since the pilot's line of sight is not that of the nose of the plane, a straight, smooth landing is a great deal harder to achieve with bicycle gear.

Figures of Merit

A rating system of the individual concepts considered in the conceptual design phase was established. This system ranked the conceptual design and aided in the determination of the final configuration. The following percentages were assigned to each of the following categories:

1) Wing Configuration - 45%
2) Propulsion - 20%
3) Horizontal Tail Configuration - 10%
4) Tail Configuration - 10%
5) Landing Gear - 15%

The percentages were set based upon their overall importance to the final design. The competing design for each of these areas were ranked with respect to one another and a score was given for each design in the area. Appendix B contains the results of the ranking system used in the conceptual design phase.

Preliminary Design

The preliminary design process included an analysis which gave some indication of the potential performance of the aircraft in addition to sizing the components. The first part of this analysis includes a layout of the design and sizing of the main components. This included tail volume calculations, wing and tail sizing, fuselage sizing, and landing gear placement. The next step was a stability and performance analysis using zero lift drag, component weights, center of gravity, longitudinal static stability, trim calculations, power for take-off, and battery power. All data used in the analysis corresponds to homebuilt composite data.

Sizing and Placement

Using the statistical data for a two engine aircraft, the empty weight to gross weight ratio of 63% gave a gross weight of 20.27 lb.[1] To begin the preliminary analysis, a wing reference area was chosen to be 13 ft^2. The aspect ratio of the wing as well as the horizontal and vertical tail were chosen to be 12, 6, and 1.5, respectively. These values were chosen to maximize the range capability. Taper was introduced to the wing design to save weight. A cruise velocity of 60 fps and a maximum velocity of 66 fps were chosen to increase range and decrease the wing and tail areas. The fuselage was sized to accommodate the payload. The payload is 7.5 lb of steel fashioned into one rectangular piece 3" by 2" by 4.4". The payload's center of gravity is placed over the center of gravity of the aircraft. This was done to increase stability. MathCad, a mathematics software package, enabled changes to be applied to the entire design quickly and efficiently.

The wings were sized using the wing reference area and a taper ratio of 50%. The wing thickness was also tapered from root to tip. Airfoil thickness ranging from 14% to 15% of the chord was considered for aircraft with speeds below Mach .08. This range was taken from statistical data where thickness is given for the best performance at a given cruise Mach number. Tail sizes for the horizontal and vertical tail were estimated using the "tail volume coefficient" method. This method used a moment arm of 50% of the fuselage starting at the quarter chord of the tail and ending at the quarter chord of the wing.[2] The vertical tail reference area was calculated using the reference area and span of the wing. The horizontal tail reference area was calculated using the reference area and mean chord of the wing. Both used tail volume coefficients from statistical data.

The last step of the sizing process was the placement and sizing of the landing gear. First, the nose gear is placed 1/3 of the total distance from the nose to the center of gravity carrying 10% of the gross weight. The main gear placement along the long axis of the fuselage was calculated using statics equations. This resulted in a main gear placement of 0.58 feet aft of the center of gravity. With a required tipback angle of 15

---

[1] Roskam, J., *Aircraft Design Part V: Component Weight Estimation,* Roskam Aviation and Engineering Corp., Ottawa, Kansas, 1989, pp. 5- 9.

[2] Raymer, D., *Aircraft Design: A Conceptual Approach,* American Institute of Aeronautics and Astronautics, Inc., Washington D.C., 1992, p. 110.

degrees or more, the strut length was found to be 2.49 inches from the bottom of the fuselage.[3] Next, the wheel placement along the span of the wing was found based upon a maximum overturn angle of 63 degrees.[4]

Stability and Performance

The first step of the stability analysis is to find the center of gravity of the aircraft. To do this, the component weights must be calculated. Eight components were used including wings, fixed equipment (pushrods, servos, etc.), engines, fuselage, nacelles, nose gear, main gear, and empennage. Each was found using a ratio of component weight to gross weight from statistical data of a dual-propeller engine aircraft.[5] An aircraft center of gravity was calculated using centers of gravity for each component given by the "approximate weights" method.[6] A center of gravity was found to be 2.86 feet aft of the nose.

The next step of the stability analysis was the calculation of the longitudinal static margin. Using aerodynamic data of the wings, tail, and fuselage a neutral point of the aircraft was found to be 2.96 feet aft of the nose. For a stable aircraft, the neutral point must be aft of the center of gravity and the distance between the two should fall in a range of 5%-12% of the mean chord. The static margin was found to be 9.1%.

The final step of the stability analysis and the first step of the performance analysis was the calculation of trim conditions and elevator deflection. First, zero lift drag of the aircraft at cruise was calculated. The friction drags, form factors, Reynolds numbers, and wetted areas were calculated for the major drag contributors which include the wings, horizontal and vertical tails, fuselage, landing gear, and miscellaneous leak and protuberance drags. The sum of these drags make up the zero lift drag. Using this and other data, such as airfoil characteristics, wing and tail incidence, and sweep angle of the wing, elevator deflection and angle of attack of the aircraft were found to trim the aircraft. Three elevator deflections were used to calculate the total lift coefficients and moment coefficients at two separate angles of attack. These three cases enabled calculations for trimmed elevator deflection and angle of attack. An induced drag at trim was found to give a cruise lift to drag ratio of about 21.

Take-off within 300 ft. and over a 10 ft. obstacle required a certain amount of thrust. This thrust limited the motor selection to those motors capable of a minimum 1.5 lb. The Graupner 600 FG3 motor gives a thrust of 1.88 lb. with a high engine efficiency and a take-off ground roll of 146.45 feet. The total take-off distance was estimated to be 203.17 feet given a 10 degree flight path angle. A landing distance was calculated to be 149.11 feet. Power needed to achieve flight was calculated using a take-off speed 20% faster than stall speed. The motor can achieve the power needed for the takeoff with additional power available if necessary. A cruise power was calculated using a thrust at cruise given the gross weight and cruise lift to drag ratio. A power setting of 50% was required for cruise.

---

[3] Raymer, p. 232

[4] Raymer, p.233

[5] Roskam, p. 8

[6] Raymer, p. 398

Range for the aircraft was calculated using the ratio of the amps drawn to the amp hours stored in the batteries. Amps drawn at cruise are lowered from the amps drawn at take-off due to reduction in power from the speed controller. This calculation gives a time of flight of approximately 9 minutes which in turn gives a range of 32,000 feet. Using an estimated course distance of 1600 feet per lap, a distance of 20 laps was calculated. A complete list of all preliminary design characteristics can be found in Appendix C.

Figures of Merit

There were two major areas identified in the preliminary design phase as affecting important mission and design parameters. These areas are motor and battery selection which greatly affect the range of the aircraft. Therefore, the overall importance for these groups have been ranked very highly. The individual performances of the competing options were ranked and total scores are given in Appendix D.

The Graupner 600 series FG3 motors were chosen based upon issues such as cost, engine efficiency, and thrust requirements given by the final design. The decision to use a dual engine design over a single engine design was based upon trade studies showing that inexpensive light weight efficient engines with the thrust capability higher than 3 lb. were unavailable. The most important factor in range is the power available from the batteries. Pre-made packs were readily available but limited in voltage output and milliamp-hours stored. Custom packs were lighter but more expensive. The custom packs also gave voltage and higher milliamp-hours stored which were closer to the required values. Therefore, the individual performance of the custom packs was given a higher rating.

Detailed Design

An important step of any detailed design process is the optimization of the preliminary design towards a final design. Mathcad made it possible to change parameters and inputs such as wing reference area to maximize range. Maximizing range is the primary goal of the design. While motor selection and battery power are important to range, lift and drag qualities are more important. The airfoil selection is crucial to the lift and drag characteristics of the entire aircraft. Three separate airfoils, NACA 4415, 23015, 2412, were analyzed using lift curve slopes and drag polars. All three airfoils had a thickness to chord ratio of about 15% which was suggested for the best performance in low speeds. The NACA 4415 proved to be superior to the other airfoils having a cruise lift to drag ratio of 100+ thus the entire aircraft lift to drag ratio will be 20 or better at the cruise condition.

Other optimization steps taken included a re-examination of motors and batteries. In this case, batteries and motors were optimized to the highest efficiency possible. The sizing of all control surfaces, however, did effect the final design. First, aileron sizing affected the expected placement of the rear spar and original shape of the aileron. The aileron is more than capable of handling the maneuvers required at a higher maximum g load than is required. The rudder and elevators were sized in the same manner using the smallest possible area with the most effectiveness. Elevator sizing affected trim in a way that increased range. The change in elevator size along with the changes of incidence in the wing and horizontal tail gave a trim angle and elevator deflection of almost zero. This helped lower the drag while retaining the lift previously calculated. All control surfaces were given a maximum deflection of 10-15 degrees. This not only helped the overall performance but increased handling qualities of the aircraft. The placement of these surfaces can be seen in Appendices E1-E3.

The systems architecture includes four servos necessary to service all of the control surfaces. The ailerons require 2 servos to deflect them simultaneously in opposite directions. Pushrods connected to the servos will fit through one of three available holes in each of the wing ribs connecting to the aileron through the rear spar. The elevator and the rudder each require an additional servo which will be attached through the fuselage. Two battery packs, one for each motor, are mounted in the fuselage fore of the payload and are also connected through these holes in the ribs. The batteries were located such that the center of gravity moved forward giving a more stable aircraft design. Another smaller battery is also mounted in the fuselage and supplies power to the servos and receiver. The battery packs for the engines are below the 2.5 lb. requirement and allow for additional batteries to be used without a weight penalty.

One of the most important aspects of the detailed design is the sizing, placement, and material selection for the major components of the aircraft. The spars were sized by first determining the maximum bending moment along the span of the wing. This bending moment distribution was found by using the Schrink's approximation. Spruce sticks ($^1/_4$" by $^1/_4$") were chosen as the spar cap material with a light plywood used for the web material. The front spar was placed at the ¼ of the chord with no sweep. The placement of the rear spar was dependent on the aileron. Once the aileron was sized and placed on the wing, the rear spar was aligned with the hinge line of the aileron. This was done primarily to make the construction easier. When choosing material for the front and rear spar, several cross sectional areas for spruce sticks were analyzed. Applying carbon fiber tape along the top of the spar caps was also considered during this phase to increase the allowable stress in the spars. After analysis it was determined that the carbon fiber tape was not needed to help the spars carry the stress.

Balsa sheets were chosen as wing rib material. Balsa sheets are light and are easily used in the construction of the wing ribs and riblets. A rib spacing of 6 in. was calculated to prevent buckling of the spars. To keep the leading edge material from sagging between ribs, riblets were evenly spaced between the ribs.

Longerons were sized by first calculating the external bending moment on the aircraft. The internal resisting moment must equal this external bending moment, thus the longerons were sized to carry this moment. Eight longerons were evenly spaced along the cross section of the fuselage. Spruce sticks with a cross sectional area of ¼" by ¼" were used as longeron material.

The material chosen for the bulkheads and the fuselage frames is a heavier grade of plywood. The bulkheads were sized to carry the wing loading, the landing gear loads, and the payload. The 6 in. spacing of the fuselage frames was calculated by looking at the buckling of the longerons.

The final design has a gross weight of 20.3 lb. with the payload as 37% of the total weight. The wing has an aspect ratio of 12 and a span of 12.5 feet. The top, side, and front views of the airplane can be seen as scaled drawings in Appendix E1 -E3.

Manufacturing Plan

The first step in this phase of the design included decisions regarding the order of the manufacturing of parts. The structural sizing procedure stated in the detailed design section was used to determine the size and type of materials required for the manufacturing of the aircraft. Additional material was ordered based on statistical data obtained from current model aircraft.

An important aspect of the manufacturing plan is the selection of production break sections. It was determined that it would be necessary to have the wing separate at the root and detach from the fuselage. This would make transportation of the aircraft much easier. It was also decided that the wing would be the first part to be produced. A jig was built to aid in the construction of the wing. This jig can hold a wing with a semi span of up to 8 feet. Individual sliding blocks allow the rib spacing to be set to any specification. Once the ribs are aligned, the spars were attached at the specified points. This jig holds the ribs in place for the entire construction of the wing. After the spars are bonded to the ribs, the leading and trailing edge can be adhered. Plywood wing skins of $1/64^{th}$ in thickness were chosen as the leading edge material. The wing skins are very thin and can be cut and formed easily about the leading edge and the trailing edge of the ribs. However, before trailing edge could be applied, the nacelle hardware for the motors must be installed and reinforced. After these steps are completed, covering can be applied to the entire wing. Superkote is a plastic, iron-on, heat shrink material that was chosen to be used as the covering for the aircraft. It is able to shrink and adhere at low temperatures so that underlying surfaces are not damaged. It also shrinks quickly and tightly so that bubbles and wrinkles are eliminated. This material also adheres to itself so that overlapping joints are more permanent and aerodynamically efficient. These facts also make repairs relatively simple.

The control surfaces including the ailerons, the elevators, and the rudder will be constructed as individual components and then connected to the lifting structures with hinges and horns. The pushrods, connected to the servos, control the deflection of these horns and the deflection of the control surfaces. Hinge tape will then be applied over the hinge line to improve the aerodynamics of the control surfaces.

The fuselage will be constructed in much the same way as the wing was constructed. The frames of the fuselage will be lined up and the longerons placed in the notches cut in the frames. The horizontal tail will be built into the rear frames where the top of the frames take the shape of the airfoil. The vertical tail will then be connected into the horizontal tail. Within the bulkheads, a compartment will be constructed to house the batteries, servos, and payload. The compartment will also be used to anchor the wing to the fuselage. A side door on the fuselage will be connected to the bulkheads and give access to the compartment. After these steps are completed, the entire structure can be covered with the Superkote.

Figures of Merit

The two main categories that affected the manufacturing plan were cost and construction type. Some composite construction was considered for the manufacturing of the individual components. However, the composite costs are considerably higher for some structures. For instance, if carbon fiber tape had been used to reinforce the spars of the wing, the cost of the spars would increase by 100 %. Another composite feature considered in the design was the construction of a wing made entirely of structural foam. The structural foam wing was more than twice as expensive as the conventional balsa construction. Advantages to an all-foam wing include better aerodynamics and ease of construction. A significant downfall of an all-foam wing construction is the weight penalty incurred. The structural foam weighs 4-6 lb/ft^3 giving a total weight of the wing as great as 12 lb. The balsa and spruce construction yields a wing weight of approximately 6-7 lb., but it is much harder to construct. The cost issues were given a higher percentage of importance to the final design due to the fact that the team extremely limited budget. The results of the ranking can be found in Appendix F.

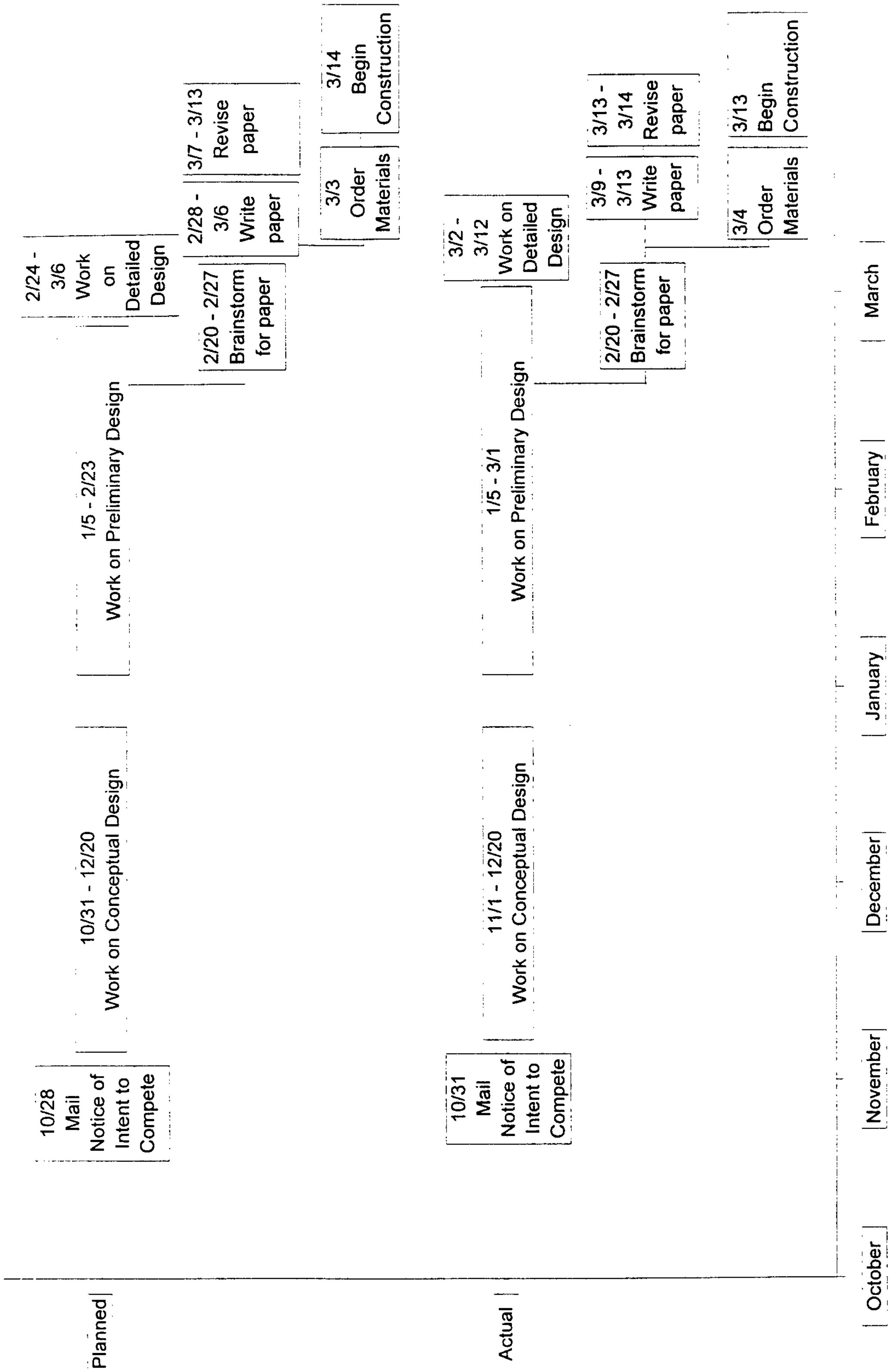
Time Line

The time line for the manufacturing plan was established. The following deadlines were assigned to the steps of the manufacturing plan:

- Order Materials - March 3
- Construction Planning - March 3
- Construction Beginning - March 14
- Complete Wings - March 18
- Complete Fuselage and Tail Combination - March 23
- Covering Surfaces - March 28
- Test Flights -April 9
- Adjustments -April 18

Currently, the design is on schedule with the building of the ribs and fitting of the spars for the wings.

Appendix A: Milestone Chart



Appendix B: Results of the Ranking System Used During the Conceptual Design Phase

FOM: Conceptual Design	Canard	Conventional Tail Planform	Chosen Design
Horizontal Tail Configuration - 10%	5	7	Conventional Tail Planform

FOM: Conceptual Design	Tractor	Pusher	Chosen Design
Propulsion -20%	5	6	Pusher

FOM: Conceptual Design	High Wing	Low Wing	Chosen Design
Wing Configuration - 45%	6	3	High Wing

FOM: Conceptual Design	T-tail	Conventional Tail	Chosen Design
Tail Configuration - 10%	3	4	Conventional Tail

FOM: Conceptual Design	Tricycle	Bicycle	Chosen Design
Landing Gear Configuration - 15%	6	4	Tricycle

Appendix C: Aircraft Characteristics

Gross Weight =	20.27 lb
Aspect Ratio =	12
Velocity at cruise =	40.91 mph
Stall Velocity =	21.25 mph
Lift to Drag Ratio =	20.57
Thrust =	1.88 lb
Landing Distance =	149.12 ft
Wing Loading =	1.56 psf
Coefficient of drag at zero lift =	0.0083
Induced trim drag =	0.0094
Coefficient of drag at cruise =	0.018
Coefficient of lift at cruise =	0.364
Oswald's efficiency factor =	0.706
Fineness ratio =	28.29
Wing sweep =	0°
Wing incidence =	3.1°
Tail incidence =	-0.3°
Taper ratio for the wing =	0.5
Aspect ratio for the horizontal tail =	6.00
Taper ratio for the horizontal tail =	1
Aspect ratio for the vertical tail =	1.5
Taper ratio for the vertical tail =	1

X position of the center of gravity =	2.86
Static margin =	0.91
Wing:	
Surface Area =	13 ft^2
Root Chord =	1.39 ft
Tip Chord =	0.70 ft
Average Chord =	1.04 ft
Span =	12.49 ft
Vertical Tail:	
Surface Area =	2 ft^2
Chord =	1.15 ft
Span =	1.731 ft
Horizontal Tail:	
Surface Area =	2.08 ft^2
Chord =	0.59 ft
Span =	3.534 ft
Fuselage:	
Length =	6.5 ft
Major axis =	9 in
Minor axis =	5 in

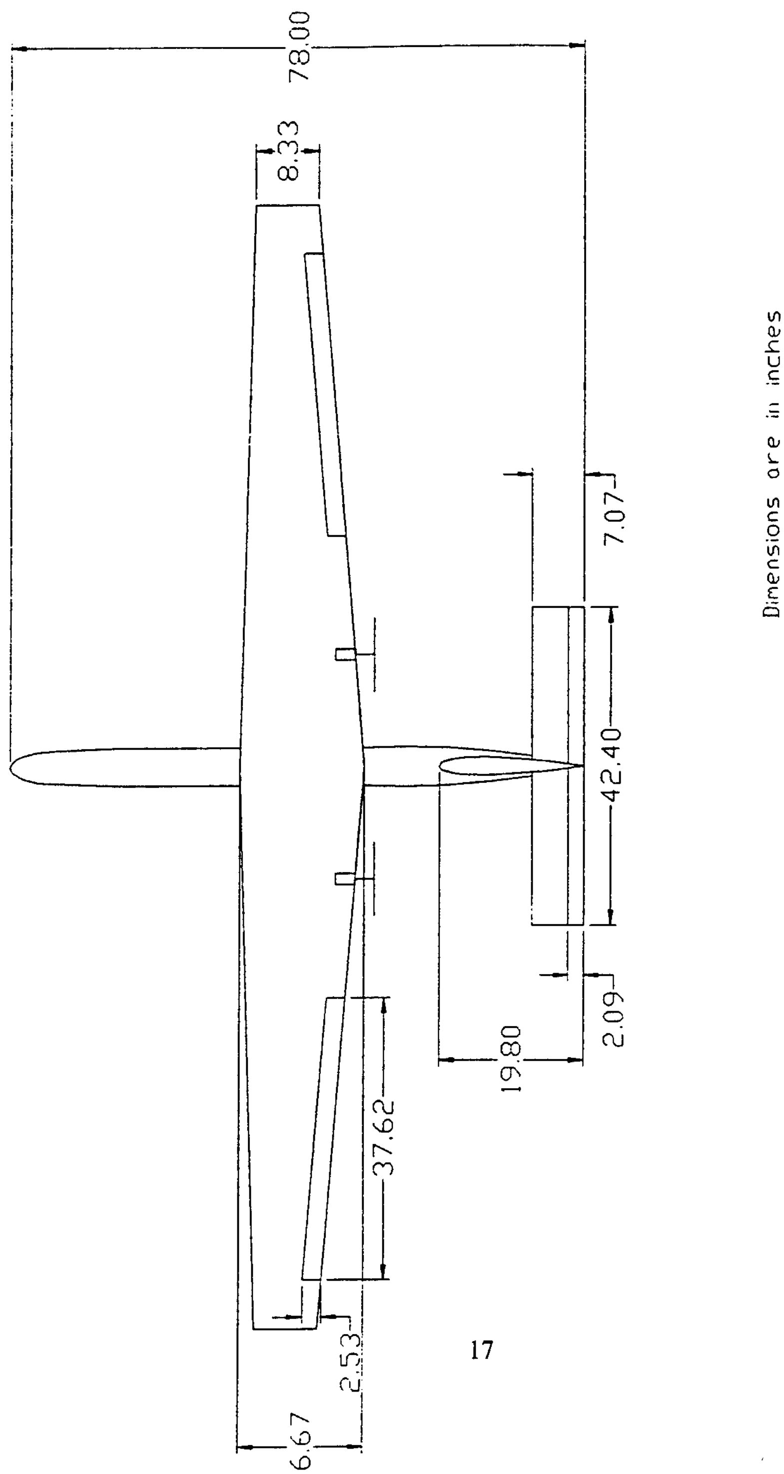
Appendix D: Results of the Ranking System Used in the Preliminary Design Phase

FOM: Preliminary Design	Graupner 500 Series	Graupner 600 FG3	Graupner 700 Series	Astro Flight	Selected Motor
Motors: 60%	4	8	5	3	Graupner 600 FG3

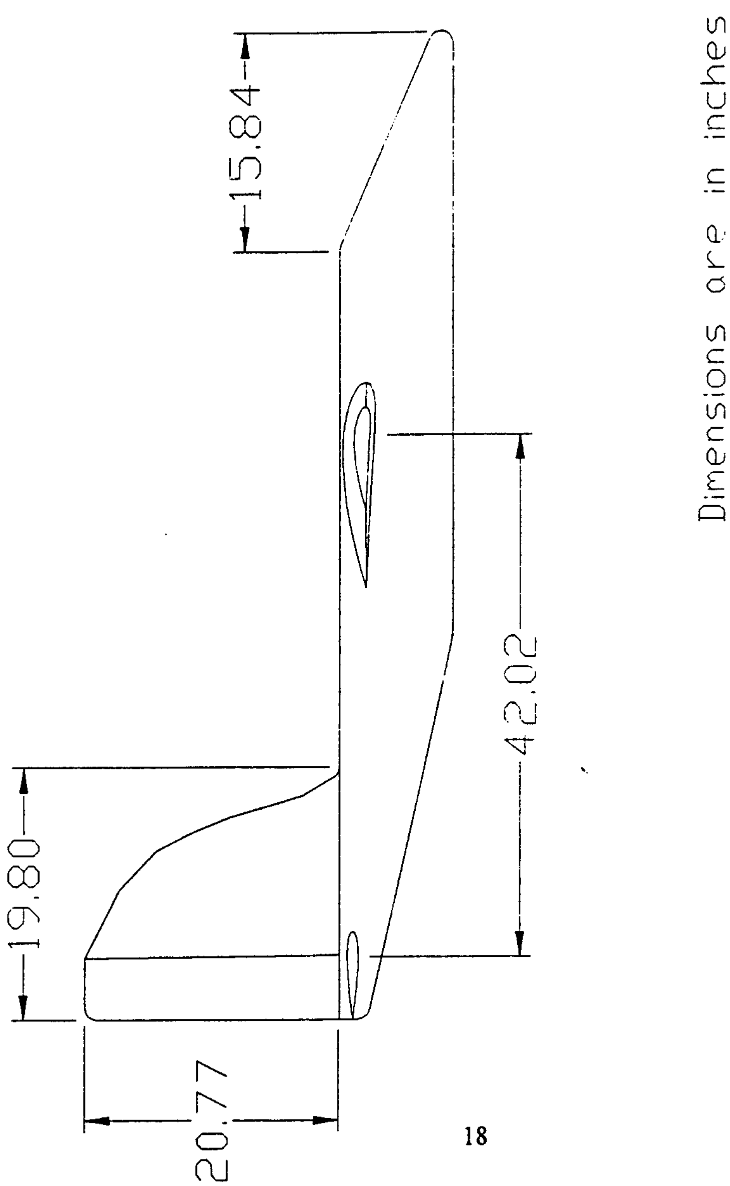
FOM: Preliminary Design	Single	Dual	Selected Design
Motors: 60%	3	7	Dual

FOM: Preliminary Design	Pre-made Packs	Custom-made Packs	Selected Design
Batteries: 40%	4	6	Custom-made Packs

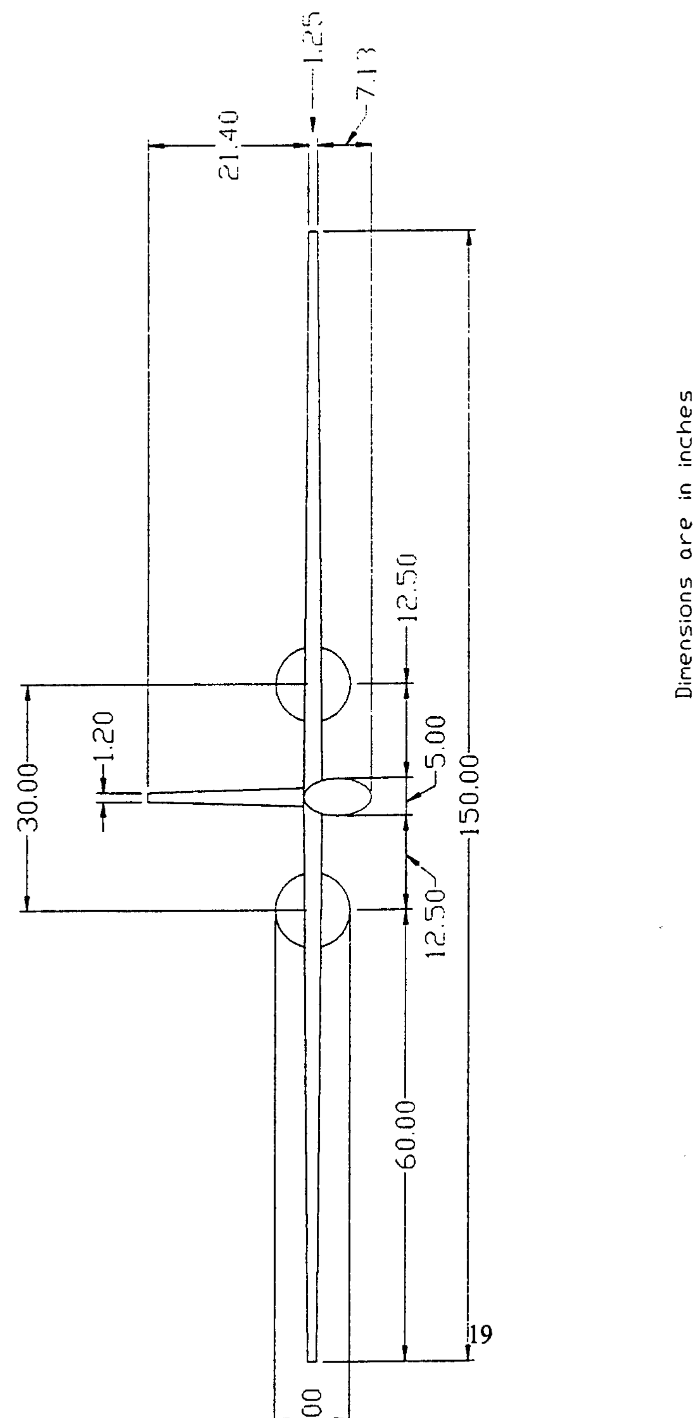
Appendix E1: Top View



Appendix E2: Side View



Appendix E3: Front View



Appendix F: Results of the Ranking System Used in the Manufacturing Plan

FOM: Manufacturing Plan	**Carbon Fiber Tape on Spar**	**All-Foam Wing Construction**	**Wood Construction**
Cost: 60%	2	6	9
Level of Difficulty: 40%	7	7	5
Total Scoring:	3.0	5.4	7.4

Chosen Design: Wood Construction

References

1) Beer, F., and Johnston, E. (1981). *Mechanics of Materials*, New York: McGraw-Hill Book Company.

2) Bruhn, E. (1973). *Analysis and Design of Flight Vehicle Structures*, Indianapolis: Jacobs Publishing, Inc.

3) Grosser, M. (1981). *Gossamer Odyssey: The Triumph of Human-Powered Flight*, Boston: Houghton Mifflin Company.

4) Nelson, R. (1989). *Flight Stability and Automatic Control*, New York: McGraw-Hill Book Company.

5) Raymer, D. (1992). *Aircraft Design: A Conceptual Approach*, Washington D.C.: American Institute of Aeronautics and Astronautics.

6) Roskam, J. (1989). *Airplane Design Part V: Component Weight Estimation*, Ottawa: Roskam Aviation and Engineering Corp.

Acknowledgments

We would like to thank the following persons for their valued input on this design:

Tim Connell
Dr. George E. Weeks
Dr. Tom Zeiler
Dr. Alton Highsmith
Dr. Amnon Katz
Dr. Beth Todd
Aerospace Engineering and Mechanics Staff
Jim Neidhoeffer
Dr. Charles Haynes
Dr. Kalmanje KrisnaKumar
Dr. Jim Cuttno

We would like to thank the following persons for their financial support:

HBI
Mary Anne O'Neill Inc.
Mr. Shackleford
Dr. William K. Rule

ADDENDUM

AIAA branch- Syracuse University

ADDENDUM -Aircraft Design

Since the original proposal, there have not been any design changes to the aircraft. Using a Clark-Y airfoil, the wing span is 84 inches. The fuselage is 50 inches in length. A Cobalt 640 G engine and Astroflight speed controller will power the aircraft.

The battery pack contains twenty 1700 SCRC cells. Weighing just under 2.5 lbs, the cells are assembled in two packs of ten cells each. Located directly behind the speed control, the two packs will lie side-by-side. This battery lay-out was selected since batteries heat up faster when stacked on top of each other. The batteries, the speed controller, and servos will be attached to a fiberglass panel inside the fuselage. Fiberglass was chosen because it is lighter than ply wood and stronger than balsa. This panel is located three inches from the bottom of the fuselage. The steel payload is located below the panel.

The payload consists of two 3.75 LB blocks of steel. Each block will sit vertically on either side of the fuselage. Placed inside, these blocks are centered around the CG of the aircraft. With this placement, the aircraft is expected to remain stable during flight. To accomodate the 7.5 lbs, that section of the fuselage is being reinforced with 1/4 inch plywood. The payload will be inserted and removed through a "doorway" section built into the fuselage.

One change was made in the manufacturing process. A wing jig was used for building the wing. The jig consists of two metal rods which mount to blocks of wood at either end. Two holes are drilled in each rib, and the ribs attached to the rods. The rest of the wing is then attached to the ribs and glued together. The purpose of using the jig was to aide in properly aligning all of the parts andto hold them in position while the glue dried.

To help those team members without experience in building Radio Control aircraft, a team member donated an instruction manual from another aircraft. By following the basic steps in the manual, our experienced builder was not required to always be present during the manufacturing process. This allowed for the plane to be built more quickly and accurately. The aircraft will be covered with Top Flight Monokote.

The majority of the materials and parts were purchased through Tower Hobbies and Superior Balsa and Hobby Supply. The aircraft will be flown using a Futaba 4 Channel radio donated by a team member. A private hobby store, Hobby City (located in Burtonsville, MD) is supplying the two battery packs at cost. Our electrical engineering department will build the battery charger. Miscellaneous parts were purchased at the local hobby store, Walt's Hobby Shop. Presented in Table A is the original budget proposal for the aircraft. Table B provides the actual expenses. Included in Appendix A are some additional performance calculations.

ADDENDUM -Lessons Learned

The biggest mistake made, was to design and build the aircraft in one semester. The first half of the year should be dedicated to the design of the aircraft, and the second semester for building and testing. To ensure that this happens next year, strict deadlines have to be set in September. They must also be enforced.

In addition, a more technical aspect of the design process should be followed. We went about the process by listening to suggestions from experienced builders and using an article from "RC Modeler" magazine. This article provided "basic" dimensions and proportions which should be used when designing the various sections of an aircraft. What should have been done was to first determine a "mission" profile for our aircraft. The profile consists of determining what flight characteristics the aircraft must possess. For instance, how well it should glide, the amount of lift needed, and the type of maneuvers it must perform. Five or six airfoils which fit these characteristics should then be selected for analysis. The analysis includes selecting a desired velocity, wing loading, and estimating the aircraft's weight. Using these constraints, the lift coefficients, drag coefficients, and moment coefficients are then calculated for each airfoil. The airfoil which best fits the desired profile is then selected for the design. From these results, the aircraft's aspect ratio and required wing span can be determined and the rest of the plane designed from there.

Table A: AIAA Design Build Fly Budget Proposal

- Four Channel radio and 3 servos	$150.00 *
-Engine Speed Control	$160.00 *
-Geared Engine	$185.00 *
-4 Battery packs	$80.00
-Battery Charger	$160.00 *
-Heat Gun and Iron	$40.00 *
-Monokote Covering	$10.00 per roll (atleats 2 rolls)
-Glue	$30.00
-Wood	$50.00
-Miscellaneous	$70.00

(miscellaneous includes such items as control rods, hinges, push rods, screws, etc.)

TOTAL	**$ 945.00**

Everything marked with an asterick is a one time expense, these items will be able to be used again, ensuring proper care. The batteries may also be reuseable for atleast next year also. This expenses total up to $ 775.00 of the $ 945.00.

Table B: Actual AIAA Design Build Fly Expenses

-Astroflight Engine	$185.00
-Speed Controller	$89.95
-Balsa, Ply Wood, and Glue	$140.00
-Two Battery Packs	$300.00
-Landing Gear	$40.00
-Miscellanous	$165.00
TOTAL	**$920.00**

Miscellanous items include propellars, control rods, hinges, push rods, monokote, etc.

Appendix

performance calculations

$C_L = .432$ $\quad C_{D0} = .0114$ $\quad S = 7\,ft^2 = 1008\,in^2$

$V_{max} = 65\,ft/sec$ $\quad V_{stall} = 34\,ft/sec$ $\quad V_{TO} = 40.8\,ft/sec$

$= 44\,mph$ $\quad = 23\,mph$ $\quad$ Chord $= 12\,in$

$K = 780$ $\quad AR = \frac{Span^2}{wing\,area} = \frac{84^2}{1008} = 7$

$Re = speed\,(mph) \times chord\,(in) \times K$

$= 44 \times 12 \times 780 = 411840$

$C_{L_{\alpha=0}} = 1.5$ $\quad \alpha_0 = 18°$

$$\alpha = \alpha_0 + \frac{(18.24 \times C_L) \times (1+T)}{AR}$$

T = planform adjustment factor = .16

$$= 18 + \frac{(18 \times 1.5) \times 1.16}{7} = 22.53 \text{ degrees}$$

$W/S = 2.85$

$C_m = -.0796$

$\sigma = 1$ density ratio of air

$$\text{pitching moment (in-oz)} = \frac{C_m \times \sigma \times V^2 \times S \times c}{3519}$$

$$= \frac{-.0796 \times 1 \times 44^2 \times 1008 \times 12}{3519} = 529.7\,in\text{-}oz$$

$$\frac{C_L}{\alpha} = \frac{1.5}{22.5} = .067 \text{ per degree } \alpha$$

C_L needed during flight $V = 30\,mph$

$$C_L = \frac{lift \times 3519}{\sigma \times V^2\,(mph) \times S\,(in^2)}$$

lift = weight

w/ payload $L = 15\,lbs = 240\,oz$

$$C_{L\,needed} = \frac{240 \times 3519}{30^2 \times 1008} = .930$$

$$\frac{.930}{.067} = 13.88 \text{ degrees}$$

For Clark Y $\alpha_{L=0} = -5$

$13.88 - 5 = 8.88^{\circ}$ wing set at in Fuse

w/out payload $L = 120$ oz

$$C_L = \frac{120 \times 3519}{1 \times 30^2 \times 1008} = .465 \qquad \frac{.465}{.067} = 6.94$$

$6.94 - 5 = 1.94^{\circ}$ $\sim 2^{\circ}$ wing set at 2° from centerline of Fuse

Stall angle is 18^{0} or 23^{0} from zero lift

$$a = 23 + \frac{(18.24 \times 1.5 \times 1.16)}{7} = 27.53^{\circ}$$ from zero lift

or 22.53^{0} AoA at altitude

ground effect height of wing from ground $= 6''$ Fuse ht
$+14''$ prop
$20''$

$\frac{20\text{in}}{84\text{in wing span}} = .24$ stall is reduced to .93 of value at altitude at .24 of wing span or at $.24 \times 84 = 20$ inches

$\alpha_{\text{gnd effect}} = .93 \times 22.53 = 20.95^{\circ}$

Load Factors $N = 1 + \frac{(1.466 \times mph)^2}{R \times G}$ $R \equiv$ turn radius

$G \equiv 32.2\ ft/s^2$

for 50 ft radius at 30 mph

$$N = 1 + \frac{(1.466 \times 30)^2}{50 \times 32.2} = 2.20$$

w/out payload $2.20 \times 120 = 264$ oz → wing lifts during Flight (turn)

$2.20 \times .465 = 1.023$ → C_L during turn

w/ payload $2.20 \times 240 = 528$ oz

$.930 \times 2.2 = 2.04$

must increase turn radius w/ payload

accelerate during turn and use up elevator

AIAA Design/Build/Fly
Final Design Report
Syracuse University

Executive Summary

The first step in the design contest was to set a goal for our team to achieve. This goal was to design and build a basic airpplane that will fly successfully while meeting all the requirements. Being the first year of the contest, and with our limited experience in the field of radio controlled electric aircraft, we decided this was the best way to begin. Next year, we will be able to combine our knowledge and experience to optimize the design.

The aircraft's design is based completely on its wing, being that the wing is what produces lift. Based upon the opinion from a couple of experienced designers/pilots at the local hobby stores, we elected to start with a six foot wing span, using the Clark Y airfoil. Preferring an Aspect Ratio between 5 and 7, the chord length was determined to be 1 ft. Since the aircraft does not need to be aerobatic and for simplicity, we placed the wing on top of the fuselage, making it a high wing aircraft.

The next step was to determine the lift the aircraft would produce, it's velocity, and required power. Including the engine, batteries, payload, etc; it was estimated that the plane would weigh 15 lbs. Using the Clark-Y airfoil data, we decided the plane would produce the desired lift of 15 lbs.

At this time, we decided to research what engines existed to ensure that one would fit our design and power requirements. The engine selected was the Astroflight FAI-40 geared. It was determined at the beginning of the design phase to use a geared engine and speed control. Preferably a 4:1 gear ratio, to complete more laps by staying in the air longer. Also, to adjust the engine to accelerate faster on take-off with the payload on board to ensure the plane gets in the air by 300 ft.

In our final design, we increased our wing span to 7 ft and kept the chrod at 1 ft. This was a result of serious consideration to the amount of lift the wing would produce, having to carry the 7.5 lb payload, and listening to other people in the hobby. The span increase does not make a big difference in the final weight of the plane, and the selected engine already has more than enough power for take-off.

The next step was to determine the dimensions for the rest of the aircraft. We found an article in the RC Modeler magazine which provided a brief article on design. It showed how to calculate the dimensions for the rest of an plane based on percentages of the wing's area. Our plane's dimension's are based on these percentages. The location of the wing on the fuselage was determined by calculating the aircrafts center of gravity. The spar of the wing was then placed at this location. The payload will be placed inside the fuselage vertically, centered around the center of gravity. It will be in two pieces, one on each side of the fuselage.

This year we kept everything as simple and basic as possible, because it was the best way to accomplish our goal. As a result, we only had two designs, the preliminary and the final. The configuration of the aircraft was set to be a high wing trainor style from the beginning. We elected not to investigate other configurations or design possiblities. Therefore, there is not a lot of detail in our desgin phases. It was decided as a team that this was the best way to approach the contest for the first year. By participating this year and building a successfull basic aircraft, we have been able to provide an opportunity for future students to gain experience in designing and building aircraft themselves.

Table 1: Team Memebers and Assignments

Member	Assignment
Kevin Bendowski	Experiencd Builder in charge of getting plane built
Sarah Benedict	Log keeper, calculations, builder, solicit sponsors
Arun Chawan	CAD plans, builder
Richard Lee	Aided in calculations, builder
Tim O'Donnell	Calculations, Treseaurer
Jason Seklejan	Team leader, design calculations, builder
Anthony Vinciquerra	Engine and Speed Control, builder

AIRPLANE DEADLINES

January 31 plans finished, final dimensions selected

February 7 engine/batteries selected and bought

February 14 radio and servos selected and bought

March 1 engine completed

March 17 report due

March 20 plane built

April 5 plane covered, engine & other parts placed in plane and ready to fly

April ? test flight of plane

April ? final report due

contest is April 25

GOAL to build a basic airplane that will fly successfully while meeting all the requirements.

MEETING LOG

Sat Jan 26
-finalized dimensions
-calculated CG with out engine, servo weights etc

Tues Jan 28
-discussed picking out engine, to finalize CG loaction and determine loaction of wing on fuselage
-listed al sections of aircraft to prepare buying the wood

Jan 30
-went to Hobby store to get engine info, ran into problem that battery weight will exceed limit, told should increase wing area

Feb. 2
-decided to increase span to 7 feet and chord to 1 ft, glider shaped body to reduce weight and better aerodynamics
-re-calculated V stall, lift drag and power etc
-found some engine specs, found an engine that produces more than required power, will call company to find more info
-question came up of 2.5 lbs per engine? will we need 2 engines
-need to know how much current, voltage, of battery to estimate how long it will run, with and without payload

Feb. 4
-learned how to calculate the power and # of laps expected
-calculated the CG excluding the engine, battery pack and servos, and payload
-battery pack will most likely be 21 cells at 2.46 lbs
-determined that steel payload has a volume of 24 in^3, it will be placed inside the bottom of the fuselage in a special compartment. Bottom of plane will have a door that opens to put payload in and out
-dimensions were finalized, plans made to determine MAC and picthing moment next meeting

Feb 9
-steel payload will be one piece, 6 inches long, 4 inches wide and 1 inch high
-discussed engine and speed control, must have heat sinks that will stick out the side of the fuselage to keep the engine from overheating
-weren't sure best way to compute pitching moment, decided to see professor and complete Tues, will deteremine if tail area is satisfactory at this time, then prepare to draw up plans
-will also assign parts of the report to be worked on, on Tuesday

Feb. 11
-according to R/C design book, best way is to make mock up of plane, with all the parts,or substitutes that weigh the same. Assigned people to brin the supplies, including the steel paylaod. The mock up will be done first w/out payload and then with. Location of wing behind the CG is preferred, more stable and helps to counteract nose down moment from engine and batteries.
-a budget was written up in detail
-a discussion of who goes to the contest was held, decided that preference will go into those who did the most work about 5-7 people interested, transportation will not be a problem as there are enough cars. Will look into how far away the hotels are to the flying field, another possibility is staying at Sarah's house, if isn't too far away.

-while doing mock-up Sunday, must determineif the wing generates more lift than the wait that will be in to nose of the fuselage.
-Fuselage dimensions were increased, it will be 5 inches wide, and 6 inches high, to support the bigger wing and to accommodate payload.
-Plan is to put the payload in it's own compartment, with openening doors in the bottom of the fuselage, it occurred to us, how to we ensure that the wood door can support the 7.5 lbs. On Sunday we will bring soem hrd wood and balsa to find out if they can support the steel. An idea was thrown around that we could reinforce the door with some composite materials, if we decide to pursue this idea, we will ask Dr. Davidson about it.
-A closer look must be taken at the report requirements to make sure we understand completly what AIAA ia asking for.
-would like to start buying wood, glue etc on Sunday so plans can be drawn up and start building
-I just had the question that we are planning to put the payload in the bottom of the plane, because it's forces will be pulling down towards the ground, is that really where we want it? We need to determine if it's vertical placement will have an affect on the planes lift.

Feb. 16
-made a mock-up of the bottom of the fuselage out of cardboard, actual size. Drew in approaximately where the engine, batteries, etc. will go. Also drew in structural rib locations, and decided how the airplane will taper in the back.
-Jason brought the block of steal, we decided it would be better to cut it into 2 pieces and have them placed on other side of the fuselage, placed vertically instead of horizontally
-calculated the power required for take-off with the payload, have 4 times more power than we need
-as far as location in the fusealge goes, the order would best be engine, then speed control, then batteries, this will cut down on the amount of wire we need, which will also be safer, the less wire the less likely to get tangled, servos will be located beneath the wing with the radio receiver, they are to be kept away from the engine and speed control in case they over heat, we will put some sort of hole in the fuselage or duct near the heat sink to help keep it cool
-also discussed use of aluminum fire wall as it will absorb more heat than a wood one will
-Joe is going to start designing a tray, that all the componenets can be mounted in so they won't move during flight
-Noted all the weights and and sizes of the parts so that proper moments and CG calculations can be obtained Tues, will also bring items of equivalent weights to put on our mock up and balance out the CG experimentally.
-prepared on paper the fuse dimensions so that plans can start to be drawn on CAD
-Tim and I made plans to write the report on what we have accomplished so far, to justify getting more money for the aircraft

Feb 18
-calculated the actual CG of the plane to be 10 inches back from the front of fuselage
-balanced out the mock airplane with parts that equalled the weights of the equipment going inside of the fuselage and got a CG of 13.5 inches back from the nose of the fuse.
-we realized that we did not have dimensions of the vertical stabilizer, we set those and determined the size of the rudder and how much aileron we would need on the wings
-made a list of the amount of plywood and balsa needed to build the plane, including what thicknesses, in one of my magazines, I have a price list of wood so I will estimate how much money we need to buy the wood next week
-Tim and I believe our best bet would be to go to Lockheed Martin and Carrier to see if they can donate some money since they donated to the conference

-plans have been made to write out exact dimensions such as where the fuse will begin to taper and the size of the ribs for the wing will be made out Sunday so that plans can be drawn up on CAD, if Arun and Kevin have time, they will try and get together before Sunday and start the fuse plans on CADD
-I will get up to the Dean's office tomorrow or friday and find out what we must do to solicit sponsors

Feb 23
-worked on completing the CAD plans
-checked the vertical stabilizer dimensions
-finalized the aileron, elavator, and ruder dimensions, wach are a certain percentage of the wing, vertical tail etc

Feb 25
-finalized CG location to 16 inches back from the front wing, based on calcualtions and book on RC design
-put together list of wood including size and type that would be needed for purchasing, including a price list
-made plans to get money this week
-realized some of the dimensions were wrong on plane, since based on % of wing area, we have been making a mistake and multiplying 84 inches by 1 ft , there is a untis problem, fixed the dimensions on the plans

March 2
-finisehd up plans on CAD
-had group members look at rough draft of parts of report that were completed, and got volunteers to work on other sections

March 4
-started to fix up plans for the report, i.e. putting in the dimensions
-performance calulations begun

Conceptual and Preliminary Design

From the start of the design phase, it was immediately decided to design a high wing trainer style airplane. This was done to keep things simple and because of time constraints. As a result, our preliminary and design phase were one in the same.

The majority of our aircraft design is based upon the wing design and size. A wing span of 6 feet was originally selected because it would provide just enough lift to get the plane off the ground with the 7.5 LB payload on board. Since the plane only needs to take-off, fly straight, make right and left turns, and land, a symmetric Clark-Y airfoil will was selected. As most Radio Control aircraft have an Aspect Ratio between 5-7, we chose a chord length of 1 ft. This gives an Aspect Ratio of 7 for our aircraft. Including batteries, engine, radio receiver, etc., it's estimated weight was 15 lbs.

The second step was to calculate the velocity to ensure a 6 foot wing span would work for our aircraft. Using the equation:

$$L = (1/2) * \rho * V^2 * S * Cl$$

we calculated a maximum take-off velocity of 64.5 ft/sec. The density was taken to be at sea level, L equaled the weight since the angle of attack was small, S was the wing area, and Cl was the maximum Coefficient of Lift for a Clark-Y airfoil at a thickness ratio of 11.7%. From researching the speeds of other Radio Controlled aircraft, our calculated velocity seemed accurate.

At this point, we proceeded to determine dimensions for the rest of the aircraft. The "Model Airplane News" November 1996 magazine, contained an article on designing radio control aircraft. In the article, it said to select a wing airfoil and size, basing the rest of the planes dimensions on the area of the wing. The horizontal stabilizer's total area was then 22% of the wing's area, giving it a maximum aspect ratio.

For the fuselage, a debate went on whether to have it shaped like a glider or like a traditional airplane. Glider advantages included: lighter in weight and less drag problems. These, we deemed the two important factors allowing us to complete more laps in the air. A traditional fuselage is easier to build and provided more room for all of the engine and radio components. The payload was the deciding factor, with a traditional fuselage there is more freedom in placement of the payload for better flight. We did not want to even try putting the payload on the wing somehow.

When first choosing the 6 foot wing span, it was a very conservative choice. We were told by experienced builders to start with 6 feet as a minimum. Worrying about this 7.5 LB payload being lifted into the air, it was decided to increase the span to 7 ft, while keeping the chord the same. This would provide more lift and not affect our estimated weight of 15 lbs. by large amounts. Using our previously calculated maximum velocity and the same equation, the generated lift was determined to be 20 lbs. The dimensions of our aircraft were adjusted accordingly.

It was now time to ensure that an electric engine(s) existed that would fit our size of aircraft and weight with the payload. After a visit to the local hobby store, we were informed that unless we were to buy a $450 .90 size engine, there wasn't one to fit our plane. In addition, the batteries themselves would weigh a minimum of 7 lbs. Two problems resulted, the battery weight exceeded the maximum 2.5 lbs and not being able to afford that expensive of an engine. Before panicking and starting a new design, we went back and calculated the power required for our current plane using the equation:

$$P = T * V$$

V was the previously determined velocity and T the thrust. Because the angle of attack would be small, the thrust used was equal to the drag. The drag was determined to be .693 lbs from:

$$D = (1/2) * \rho * V^2 * S * Cd$$

where Cd was taken from the Clark-Y airfoil data. The power required was calculated to be 23 Watts for take-off. Using the world wide web, we found an affordable engine from Astoflight that would fit the plane. We also contacted the company to ensure that it would do the job. This engine was the FAI-40 Geared, which puts out 800 Watts of power.

The rest of the design phase consisted of stability calculations. The center of gravity needed to be determined for placement of the wing and to ensure that our tail area was large enough. The center of gravity was calculated using two methods. This first was from basic static calculations by using volumes, weights, and the local CG's of each part of the aircraft. The other method by making a mock-up of the aircraft. Using cardboard, a mock-up plane was built to the actual dimensions. Then, items which weighed the same as the actual components to be placed inside of the fuselage were collected. These items were placed inside the mock-up fuselage and then the entire mock-up was placed on a fulcrum. By adjusting the placement of these components and the mock-up on the fulcrum, we found a point 14 inches back from the nose in which we were able to balance the plane. Besides determining the CG, this method had an advantage in that it aided in determining the position of the inside components.

The entire design of the aircraft has taken into consideration the 7.5 LB removable payload. It was taken into account throughout the majority of the design phase, particularly in the lift calculations. It was not taken in account for the CG calculation. Using the weight of steel and it's density, we determined the size required for this payload and decided it would be best to use two pieces. One would be placed inside one side of the fuselage and the other piece on the other side. Both of these pieces can be centered around the CG so that the aircraft will remain stable during flight with the payload on board. The pieces will be placed facing vertically inside of the fuselage, because horizontally they would change the CG more. Also, as the aircraft turns, the weight of the payload will shift properly and not affect flight.

As previously mentioned, the engine is an Astroflight Cobalt FAI-40 Geared. The battery pack will have a total of 21 cells and weigh 2.46 lbs. A speed control will also be used to help us get more laps out of the aircraft while in the air. A special design feature which we plan to include is to place the heat sink so that it sticks out of a small hole in the front of the fuselage to prevent overheating of the engine.

Detail Design-Final Dimensions

Once the wing of the aircraft had been placed and set on the fuselage, the dimensions of the aircraft were finalized. This was a vital piece of the design since the fuselage length and tailplane placement was highly dependent on it. As a result, the original wingspan of 7 feet or 84 inches was retained with two inches added to account for the rounded wingtips. The fuselage will have a 6 inch by 5 inch box cross-section, which will be rounded down for aerodynamic purposes. The 50 inch fuselage length was ideally selected to satisfy the structural rules found in the RC modelers handbook. These rules dictated how far from the tip of the fuselage aerodynamic center of the wing could be placed. The AC of the wing was placed one and 1 half chord lengths from the tip of the aircraft, which is about 18 inches. This makes the distance from the tip to the leading edge approximately 15 inches. Two and one half chord lengths is the distance between the AC of the main wing and the AC of the horizontal tail. This distance added up comes to a total of about 48 inches. For our design the extra three or four inches, seen in the 3D drawing, was added for sizing purpose and should not substantially change the original 50 inch design.

Wing design, which is also shown on the 3D figure, will consist of approximately 31, constant airfoil shaped ribs, about 2.625 inches apart. A layout of the wing spars and ailerons are also expressed in the 3D drawing. The horizontal tail span will be about 29 inches across based on a correlation between the main wing and horizontal tail aspect ratios. The tip chord length of the horizontal tail will be 5.2 inches while root chord is 7.2 inches. Based on the fact that the elevator control surface is approximately 35% of the total horizontal tail area, the elevator will be 2.3 inches wide. It will also extend from tip to tip other than a small break at the vertical tail for rudder clearance. Dimensions of the vertical tail were based on educated estimate with the height and base being fixed at 16.62 inches and 7.54 inches, respectfully. Thirty-five percent of this area accounts for the rudder control surface, which is 2.44 inches in width and also 16.62 inches in height.

Internal component configuration is very basic, with the Astroflight Cobalt FAI-40 geared, .781 pound engine placed in the nose in the typical fashion. A .119 pound speed control for speed variation, is placed directly behind the engine with a balsa wood firewall in between them. The 21 Nickel Cadmium cells, which will power the Cobalt engine, will be packed tightly together and attached 2 or 3 inches behind the speed control.

The remainder of the major internal hardware entails the radio receiver, the receiver battery pack and the control surface servos. This equipment, which is responsible for the control surfaces (ailerons, elevators, rudder), will be placed in the wing box area. Most of this equipment will be raised inside of the fuselage in order to accommodate the two 3.75 pound, steel blocks use for the payload testing. The undercarriage will be of tricycle configuration with the main gear being placed near the center of gravity. This is done to counter against a heavy impact on the structure during landing.

Detail Design-Performance Qualities

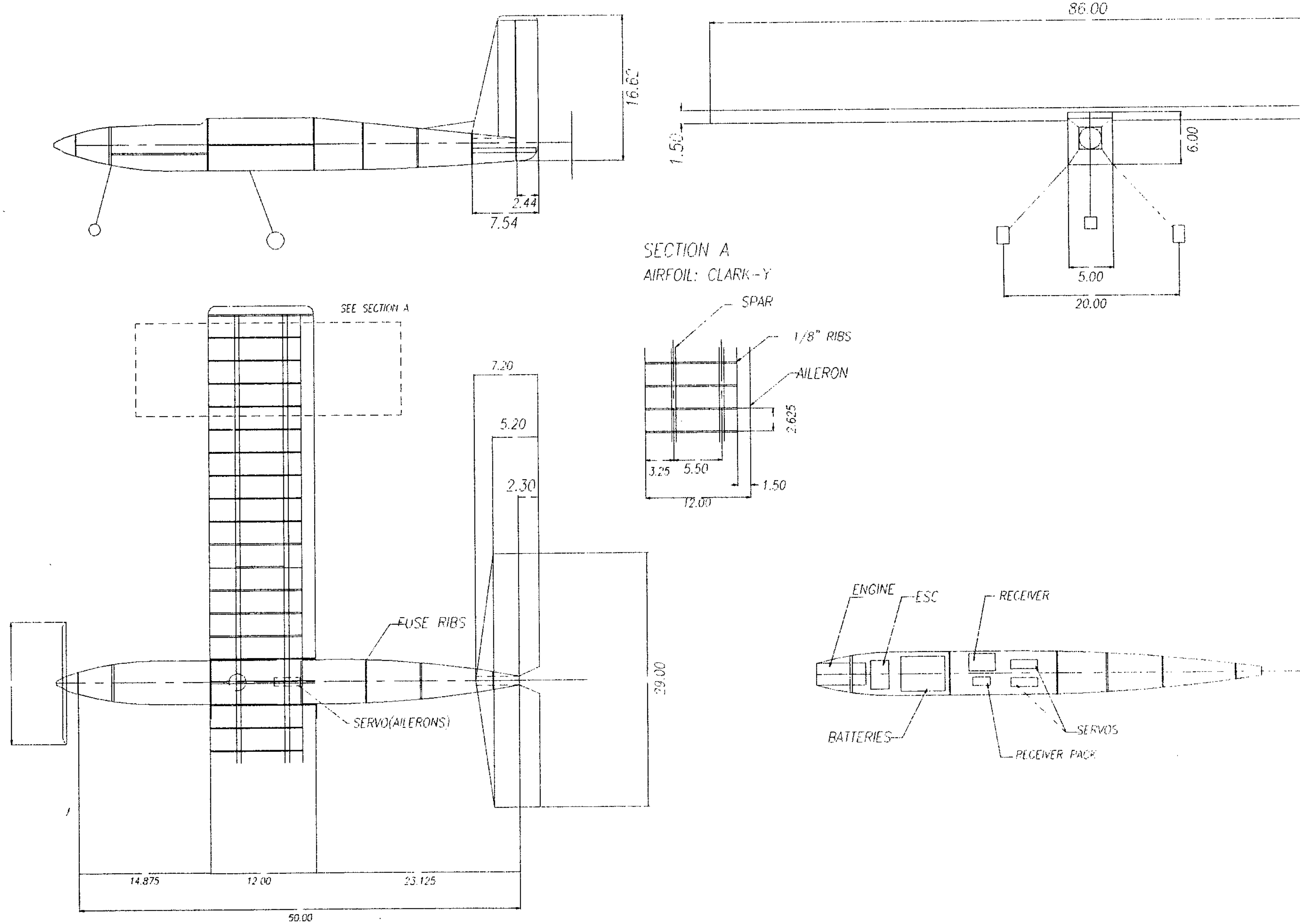
One of the more crucial performance characteristics of this aircraft, as in all aircraft, is it takeoff performance. It entails takeoff velocity, stall velocity, thrust required, power required and takeoff distance. Using the formula for velocity at steady, level, unaccelerated flight, we calculate the maximum speed attainable by our RC model. Using a weight of 15 pounds, a coefficient of lift of .43, a wing area of 7 square feet at sea level, the maximum velocity of the aircraft should be about 65 feet/sec. or 44 mph. Stall velocity, or velocity at maximum angle of attack, uses in its relation the maximum coefficient of lift, which is 1.5. The velocity at stall was calculated to be 34 ft./sec. or 23.2 mph. The equation:

$$V_{takeoff} = 1.2 \times V_{stall}$$

will then give us a takeoff velocity of 40.8 ft./sec. or 27.82 mph. The thrust required at altitude is calculated using the weight and the coefficients of lift and drag at zero angle of attack. This comes out to be .398 pounds of drag, or to put in typical RC model units, 6.36 oz. In this situation, thrust equals drag for steady, level, unaccelerated flight. Thrust required at takeoff will be approximately 2.16 pounds or 34.54 oz. through the use of a drag polar calculation. This translates into a power required of about 35 watts during flight at altitude.

The takeoff length of the aircraft based on a 15 pound airplane with maximum coefficient of lift at 1.5 will be 186.636 feet with the steel payload and 46.66 feet without the steel payload. Using the information received from the contest headquarters, the potential altitude for our design should be around 8,128 feet. The potential range is the potential altitude of 8,128 feet multiplied by the glide ratio, which is assumed to be 15. This gives a range of 121,920 feet or about 81 laps. Maneuverability and dynamic stability of the aircraft will most likely be average but satisfactory, due to the given wing placement and CG position.

Again, our goal for this contest was and is to design and build a solid aircraft which would fly under the given restrictions. It was not meant for us to design and build an acrobatic trainer model, since this would go beyond the scope of this contest. Therefore, we kept the external and internal configurations typical and the performance qualities modest.



16.62
2.44
7.54
86.00
1.50
6.00
5.00
20.00
SECTION A
AIRFOIL: CLARK-Y
SPAR
1/8" RIBS
AILERON
2.625
3.25
5.50
1.50
12.00
SEE SECTION A
7.20
5.20
2.30
29.00
FUSE RIBS
SERVO(AILERONS)
14.875
12.00
23.125
50.00
ENGINE
ESC
RECEIVER
BATTERIES
SERVOS
RECEIVER PACK

Manufacturing Plan

Our manufacturing method started with basic calculations that determined the size and dimensions of our aircraft. These calculations were based on the final weight of the aircraft, which includes the seven and a half pound payload, engine weight, battery weight, radio gear, etc. After the major calculations, we decided on the best dimensions to suit our needs. Using Autocad, we developed full scale plans to aid us in assembly.

Our building procedure starts with the assembly of the back horizontal and vertical stabilizers, as they are the easiest parts to assemble. This is a good place for inexperienced builders to get the feel of the building methods involved. These are going to be constructed out of solid one-half inch balsa wood for strength purposes and to counteract some of the heavier weights in the nose of the plane.

After the back wing and tail are built, we are going to move onto the wing. The wing is going to be built in two halves, then joined. This allows for easier handling, being the span is eighty-four inches, and more accurate dihedral measurements. There are many steps involved in the wing-building process. The first step is going to be to cut and shape the ribs to the exact *Clark-Y* airfoil shape. These are going to be made out of one-eighth inch balsa sheets, and will be hollowed out in the center to reduce weight and amount of wood needed, thus lowering our expenses. We will use the scraps for upcoming steps. Next, they will be positioned onto the bass-wood spars according to the wing plans. The two center ribs will be measured with a dihedral gauge to an exact two degree angle. Once the spars, leading edge, and trailing edge are all correctly positioned and pinned down about the ribs, they will be permanently glued with CA glue to start.

When everything dries, one-eighth inch balsa sheeting will be placed on top and below the ribs from the leading and trailing edges to the center of the spars. To even out all depressions, thin balsa strips will be glued on top and below each rib joining the sheeting. After the wing-tips are shaped and attached, the two halves will be joined with thirty minute epoxy. Fiberglass will then be wrapped around the joint.

The next major step is fuselage assembly. The fuselage sides are going to be made out of three one-eighth inch balsa sections. One will extend from the firewall to the leading edge of the wing, another extending the entire chord of the wing, and from the trailing edge to the rear. Once these three pieces are shaped and edge-glued according to the fuselage plans, a bass-wood piece, almost resembling the entire side, will be glued to the inside of each balsa side. This piece will contain cut-outs for the positions of the five fuselage ribs and the horizontal stabilizer. It will also give the majority of the strength to the walls. After the walls are dry, the one-eighth inch plywood ribs will be attached with epoxy to one side using a ninety degree angle for correct alignment. The ribs will be hollowed out to reduce weight, amount of wood, and cost. Then the other side will be attached in a similar fashion. Now that the fuselage is starting to take shape, the bottom may be sealed from the trailing edge of the wing to the end. This is going to be done using cross-grained pieces of balsa sheeting. This will provide sufficient strength and will be of negligible weight. Next, the firewall will be secured to the front of the fuselage. To this, the motor will be mounted. There is going to be a housing of one-forth inch balsa built around three sides of the motor. One side will be left open so the motor can be accessed and removed if needed. After the housing is sanded to shape, the bulk of the fuselage work is done.

The next step is to attach the elevator to the horizontal stabilizer. This is going to be done using plastic hinges, and toothpicks. The first step is to make even cuts in the elevator and the stab using an x-acto knife. Then the hinges are attached and glued with epoxy. After they dry, there will be two toothpicks inserted through the elevator and the hinges per hinge. This minimizes the danger of having a control surface rip out during flight. The rudder and ailerons will then be attached in the same fashion.

To attach the rear pieces to the fuselage, we are going to have to have to make cuts in the fuselage sides and top according to the plans. The horizontal stabilizer will be attached first, using thirty-minute epoxy to set it initially. Then balsa triangle stock will be put into place to ensure strength. The vertical stabilizer will be attached using the same method.

After these set, it is time to make the mounts for the wing, radio gear, and battery packs. The wing is going to be held on using rubber bands. The rubber bands will secure to wooden dowels that run through the width of the fuselage near where the leading and trailing edges of the wing will lie. Two of the three servos are going to be mounted on two elevated pieces of plywood, which will be positioned under the wing about three-quarters of the chord length back The aileron's servo will be mounted upside-down into the center of the wing. After these are secure, we will measure and cut the pushrods to the correct length, allowing some for z-bends. One of the two servos in mounted in the fuselage will control the elevator. The other will control the rudder and the nose gear simultaneously. We are going to have an electronic speed controller to control the motor's RPM. We chose one that will conserve the maximum amount of battery power possible, to extend flight time. The battery packs are going to be mounted behind the firewall leaving about four inches for the speed control. These are going to be placed on an elevated tray to allow room for the payload. There will be a removable hatch in the top of the fuselage to allow access to these components.

The payload is going to be placed on the base of the fuselage. There is going to be a reinforced trap door, on which the steel will lie. This piece will open at one end, and will be attached to the other end with hinges. This allows for easy access and removability of the steel. The positioning will be directly under the center of gravity, to minimize changes in flight.

The entire plane will be covered in Monokote, which shrinks and seals to wood using a heat iron. Monokote is a very strong material once attached. It is of negligible weight and can make many attractive designs. This is the best type of skin for a model airplane.

We have investigated many ways to build the airplane, and materials to use. With all of our combined knowledge, we decided that this method will provide the most strength for the weight, and most cost efficient. The bulk of our cost is the hardware involved. For example, a four channel radio, a motor with sufficient power, battery packs, a charger, and the electronic speed controller. Balsa wood is very light, easy to handle and shape, and provides sufficient strength. It is also extremely cheep. We are confident in the manufacturing method and design we chose, and believe it will perform competitively in the competition.

RPR-1 Jack
Design Report - Proposal Phase

AIAA Student Design/Build/Fly Competition
Ragged Island, Maryland

Department of Aeronautical and Astronautical Engineering
University of Illinois at Urbana-Champaign

March 14, 1997

RPR-1 Jack
Design Report - Proposal Phase

AIAA Student Design/Build/Fly Competition
Ragged Island, Maryland

Department of Aeronautical and Astronautical Engineering
University of Illinois at Urbana-Champaign

March 14, 1997

Table of Contents (continued)

6.0 Manufacturing Plan 44
- 6.1 Component Requirements and Design Drivers 44
- 6.2 Manufacturing Processes Investigated 44
  - 6.2.1 Wing 44
  - 6.2.2 Fuselage 44
  - 6.2.3 Empenage 45
  - 6.2.4 Tailboom 45
- 6.3 Figures of merit 45
- 6.4 Methods of Analysis Used to Screen Manufacturing Processes 46
- 6.5 Results: Manufacturing Processes for Final Design 46
  - 6.5.1 Wing 46
  - 6.5.2 Fuselage 47
  - 6.5.3 Empenage 47
  - 6.5.4 Tailboom 47
- 6.6 Construction Details 47
- 6.7 Cost Reduction Methods 48

References 52

Table of Contents

1.0 Executive Summary 1

2.0 Management Summary 3

3.0 Conceptual Design 6
3.1 System Requirements and Design Driver 6
3.2 Figure of Merit 7
3.3 Analysis Tools 7
3.4 Configurations Considered 8
3.5 Conclusions 8

4.0 Preliminary Design 12
4.1 Design Parameters Investigation 12
4.2 Analysis Tools 12
4.2.1 Cruise Model 12
4.2.2 Strength Testing 12
4.2.3 Propulsion System Testing 13
4.2.4 Stability and Controls Analysis 13
4.2.5 Turns Analysis 14
4.2.6 Take-off Analysis 14
4.3 Results of Analyses – The Prototype Aircraft 14
4.3.1 Gross Weight and Flight Reynolds Number Estimates 15
4.3.2 Cruise Model Results 15
4.3.3 Structural Analysis Results 15
4.3.4 Propulsion Testing Results 15
4.3.5 Stability and Control Analysis Results 16
4.3.6 Results of Turns Analysis 16
4.3.7 Results of Take-off Analysis 16
4.4 Preliminary Design Summary and Conclusions 16

5.0 Detail Design 23
5.1 Fuselage Design and Internal Configuration 23
5.2 Wing Planform Optimization 24
5.3 Landing Gear Selection 24
5.4 Propulsion System Selection 25
5.5 Airfoil Selection 25
5.6 g-Load Capability 26
5.7 Stability and Control Analysis 26
5.8 Take-off and Climb Analysis 27
5.9 Final Range, Endurance Performance and Payload Fraction 27
5.10 Conclusions 27

1.0 Executive Summary

This report presents the design process and results for the University of Illinois' entry into the first ever AIAA Student Design/Build/Fly Competition. The final design, RPR-1 *Jack* (Remotely Piloted Range) is the product of detailed aerodynamic and structural modeling and analysis. The design processes discussed in this report rely on both experimental data, acquired through testing and construction of a prototype aircraft (YRPR-1), and analytical modeling. The aircraft is designed to satisfy all of the Competition requirements and maximize the number of laps around the competition flight course.

The design process began with a consideration of several different configurations. A cruise model computer program was developed to quantify the performance potential of competing configurations. A flying wing configuration was analyzed in detail along with a conventional high-wing aircraft. The results of the conceptual design process indicated that the cruise range performance was nearly the same for both configurations. The high-wing configuration was selected because it was deemed more robust, i.e., better handling qualities over a larger flight envelope. It was also less complex to design and build. The conceptual design process also indicated that the final aircraft would be large. The 7.5 lbf payload combined with the 2.5 lbf battery limit meant that gross weights of 15 to 20 lbf were not unlikely.

The preliminary design stage produced many experimental and analytic studies of aircraft performance. Some of the design tools developed at this stage were: a more detailed cruise model, stability and control analysis, turns analysis and a takeoff analysis. Some of the experimental tools implemented were: structural testing of different wing construction methods, propulsion system testing and the construction of a prototype aircraft (YRPR-1). The construction of the prototype aircraft aided in gathering the necessary data for improving the analytical models and also revealed areas where more design work was required. Two such areas were the fuselage and landing gear.

The prototype aircraft suffered from insufficient power partly because of a large, high-drag fuselage and landing gear. These area were redesigned using computer-aided techniques and resulted in a configuration modification to a low-wing. The low-wing allowed for larger propeller clearance while using a shorter landing gear. The fuselage could also be made smaller since it was able to "rest" on the wing. Increased propeller clearance was important because propulsion system testing indicated that larger propellers would provide better range performance. The construction of the prototype and strength testing provided invaluable data in selecting the processes and materials used to build the final design.

The final design, RPR-1 *Jack*, satisfies all of the requirements described in the Vehicle Design Specifications and was also designed for maximum range performance. The predicted maximum range is 14.5 miles at a cruise velocity of approximately 40 ft/sec. When the turns in the flight path and the takeoff and climb are taken into consideration, the range is reduced to 12.5 miles. This is equivalent to 37 laps around the flight course at the contest site.

The University of Illinois at Urbana-Champaign is proud to submit this design to the sponsors of the AIAA Design/Build/Fly Competition.

2.0 Management Summary

The management structure of the project team was informal and was minimized to enhance group efficiency. The project coordinator managed the budget and also organized the weekly meetings so that these would be productive. Although, one person was designated as the chief purchaser, these duties were often delegated to others in the name of efficiency.

It was determined that a small group of people could function more efficiently in the design of a particular component and report their findings back to the entire team for final approval. Therefore, all of the major project tasks were addressed by dividing the team into subgroups and were allocated on a case-by-case basis as the need arose. Additionally, the team tended to divide itself between those interested in construction and those interested in design. The subgroups were chosen to reflect equal numbers of members from both groups. Thus, the builders could directly interact with the designers to ensure that feasible designs were established and communication was maintained. Also, it should be understood that often builders would become designers, rather than have the construction team accept an absolute design with no input on their behalf. A summary of team member responsibilities is given in Fig. 2.1. Note that many team members participated in several different activities. This contributed to better communication and more efficient functioning of the team.

The project team was assembled in May of 1996 and a milestone chart was developed shortly thereafter. Obviously, the project activities were limited during the summer. However, organizational and planning activities, as well as some conceptual design efforts were carried out over the summer months. The team in its present form was really assembled at the start of the Fall 1996 semester. The original milestone chart created during the summer of 1996 was modified in September to reflect the changes in the contest deadlines. This version is shown in Fig. 2.2. Overall, the actual timing of project milestones was at most one and a half months behind the planned date, with most lagging by one-half to one month. This was acceptable since the schedule was created with fairly flexible deadlines for most milestones. The events associated with the completion of each milestone are described in the following sections of this report.

Aerodynamics

- Chris Lyon
- Ashok Gopalarathnam

Performance, Flight Mechanics, and Stability and Control

- Chad Henze
- Marty Klipp
- Sam Lee
- Andy Broeren

Propulsion System Research and Testing

- Andy Broeren
- Chong Hin Koh
- Marty Klipp
- Vijay Ram

Fuselage Design

- Ben Keen
- Chad Henze
- Andy Broeren
- Josh Minks
- Patrick Schuett
- Ashok Gopalarathnam

Wing Structural Design

- Andy Broeren
- Chris Lyon
- Josh Minks
- Pong Lee
- Chong Hin Koh

Structural Testing

- Andy Broeren
- Patrick Boyssmith
- Shalin Mody

Figure 2.1 Subgroup organization diagram showing all team members and their area(s) of involvement.

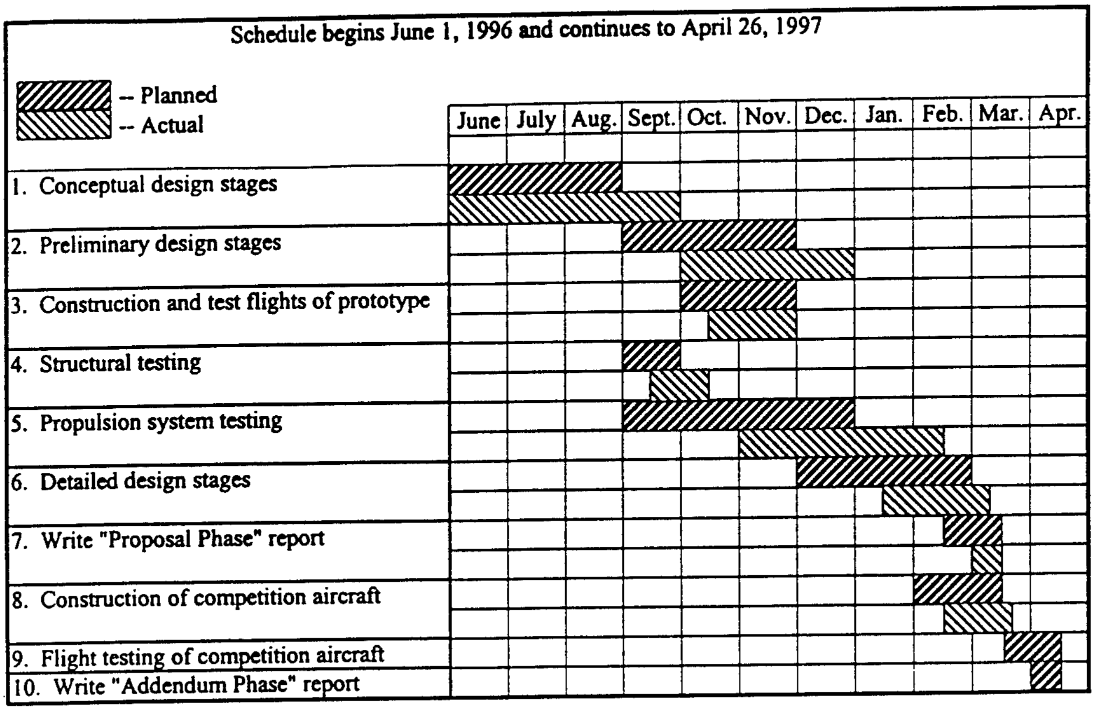


Figure 2.2 Project milestone chart showing the planned and actual timing of important activities.

3.0 Conceptual Design

3.1 System Requirements and Design Drivers

The aircraft system requirements are given in the 1996/1997 Rules and Vehicle Design Specification. Those identified as having the greatest impact on the conceptual design are:

- The system design objective is to achieve the maximum number of laps over the specified flight course with a 2.5 lbf battery weight limit.

- The aircraft must carry a 7.5 lbf payload, satisfy the takeoff requirement of clearing a 10 foot obstacle within the 300 foot runway area, and successfully land within the same 300 foot runway area.

- The design must be well balanced; offering high vehicle performance, good flight handling qualities, and practical and affordable manufacturing processes.

From these principal system requirements, the following design drivers were identified:

- maximum range; i.e., maximum number of laps including turns
- fixed payload of 7.5 lbf
- limited energy load; i.e., a maximum battery weight of 2.5 lbf
- 300 ft take-off and landing field length
- good flying qualities for the take-off, climb, cruise, turns and landing
- good maintainability; quick payload removal
- practical and low cost manufacturing processes

The maximum range requirement was identified as the principal objective of the conceptual design phase. It was felt that most of the available battery energy would be used in the cruise condition. To support maximum range studies, a simple cruise model computer program (cruise model, hereafter) was developed. This allowed for an efficient analysis of the impact on range of different aircraft configurations. The simple model did not include the effect of the turns because it was thought that turns would tend to limit the range of all configurations in essentially the same amount, at least to first order. Likewise, the takeoff/landing and other flight handling characteristics were not considered at this early stage since they are, in terms of energy use, minor requirements. The reasoning here was that these other requirements could be achieved regardless of the configuration and that these requirements would not change the principal characteristics of the selected configuration.

3.2 Figures of Merit

Using the design drivers mentioned above, figures of merit (FOMs) for screening competing concepts were established and the quantitative value judgments for each of these FOMs are summarized in Table 3.1.

- Range, or number of laps.

- Robustness. This FOM represents the "good flight handling qualities" requirement; the design should be able to fly well in a variety of weather or environmental conditions. This also includes the extent to which the aircraft can maintain adequate range performance at off-design flight conditions.

- Complexity. This FOM represents the level of difficulty in designing and building the aircraft and "the margin for error." For example, in fabrication, a few small errors in more complex configurations could have large consequences.

- Innovation. This FOM represents the potential benefit to the UAV community which could result from an innovative design. Also, more innovative configurations are generally looked upon more favorably for their utilization of unique design concepts.

3.3 Analysis Tools

As mentioned in Section 3.1, a simple cruise model was developed and a flow chart showing the operation of the program is presented in Fig. 3.1. Simple estimates for aircraft weight and drag characteristics were determined from available technical data and from experience. The induced drag, lift distributions and the wing/tail interaction were calculated using a vortex-lattice code. At this conceptual design stage, it was assumed that the aircraft could be trimmed and stabilized without serious penalties in range and that these penalties would scale proportionally for different aircraft sizes, so that relative comparisons could still be made.

The motor-battery-speed controller system plays an important role in determining the energy consumption, and hence, the range. The battery pack configuration is often thought to determine the total battery capacity. Modern electronic speed controllers operate based upon a switching rate or duty cycle which is proportional to the throttle position. At any throttle position, a switching rate is introduced which only draws energy from the battery in bursts proportional to the switching frequency. This switching rate determines the battery life. This assumes that the motor and speed controller efficiencies remain constant over the relevant range of voltages, a good assumption, especially for conceptual design calculations. As a result, a single, fixed battery pack configuration, with all cells connected in series, was selected for these studies. The cells having the highest capacity-to-weight ratio were chosen. The motor and propeller combination was selected to achieve the maximum propulsion efficiency at the design cruise condition. The selections were based upon a limited set of data taken by Roth[1] and were performed manually. This process demonstrated the need to conduct propulsion tests to extend the propeller and motor performance database.

3.4 Configurations Considered

During the conceptual design process, the configurations considered were: (1) conventional high-wing (selected as the baseline), (2) canard, (3) tandem wing and (4) flying wing. Only two of these, the conventional high-wing and the flying wing were studied in detail and compared. It was believed that considering two configurations which are very different would help to "bracket" the range of FOMs that would be similar for other configurations as well. Conceptual sketches of these configurations are presented in Fig. 3.2. and range versus cruise velocity plots are shown in Fig. 3.3. Table 3.2 shows the final ranking between the two.

For the conventional configuration, the maximum range estimate of 13 miles yielded a value of 3 for the "range" FOM. It was believed that the conventional configuration was the more robust, therefore it was awarded a "robustness" FOM value of 4. This was supported by running the cruise model in an "off-design" case which showed the range penalties to be small. The other FOM values in Table 3.2 are self explanatory.

For the flying wing configuration, the maximum range estimate was also found to be about 12 miles, also yielding a value of 3 for the "range" FOM. However, the flying wing configuration suffers in the "robustness" category as it is more sensitive to wind gusts and often suffers from poor directional stability.[2] Therefore a value of 1 was given for the "robustness" FOM. The other FOM values in Table 3.2 are self explanatory.

3.5 Conclusions

The high-wing conventional configuration was chosen for preliminary design. The main reasoning for this was the simplicity. A simple configuration would ensure that a reliable, "robust," aircraft could be designed and built to meet all of the requirements mentioned at the start of this section; for the immediate purposes the level of innovation was assumed secondary.

Table 3.1 Figures of Merit for Conceptual Design

Figure Of Merit	Ranking		
	5	3	1
Range	15-20 miles	10-15 miles	5-10 miles
Robustness	high	average	low
Complexity	simple	average	complex
Innovation	innovative		traditional

Table 3.2 Final Ranking of Conceptual Designs

Figure Of Merit	Ranking	
	Conventional	Flying Wing
Range	3	3
Robustness	4	1
Complexity	5	1
Innovation	1	5
Total	13	10

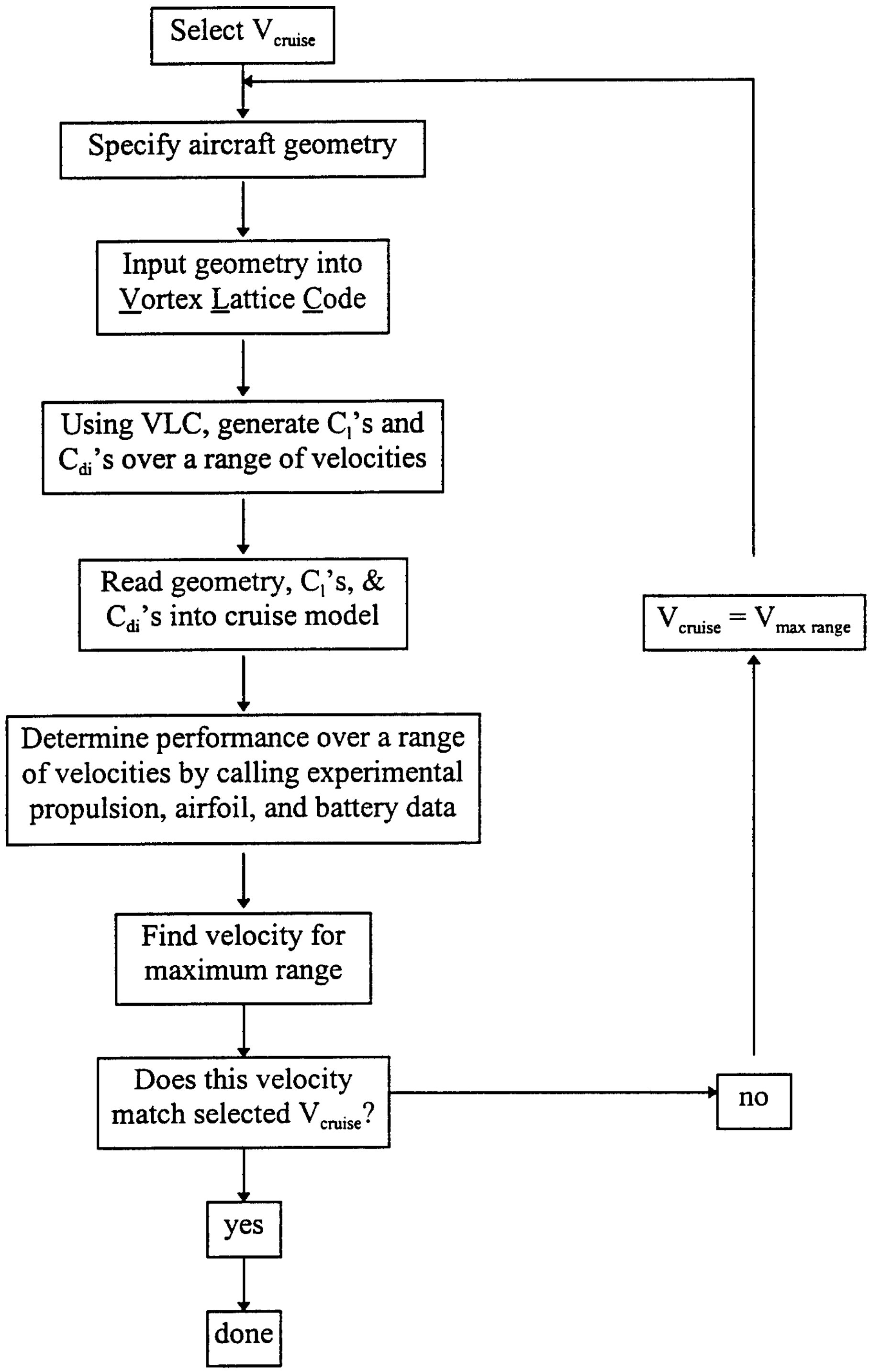


Figure 3.1 Cruise model program flow chart.

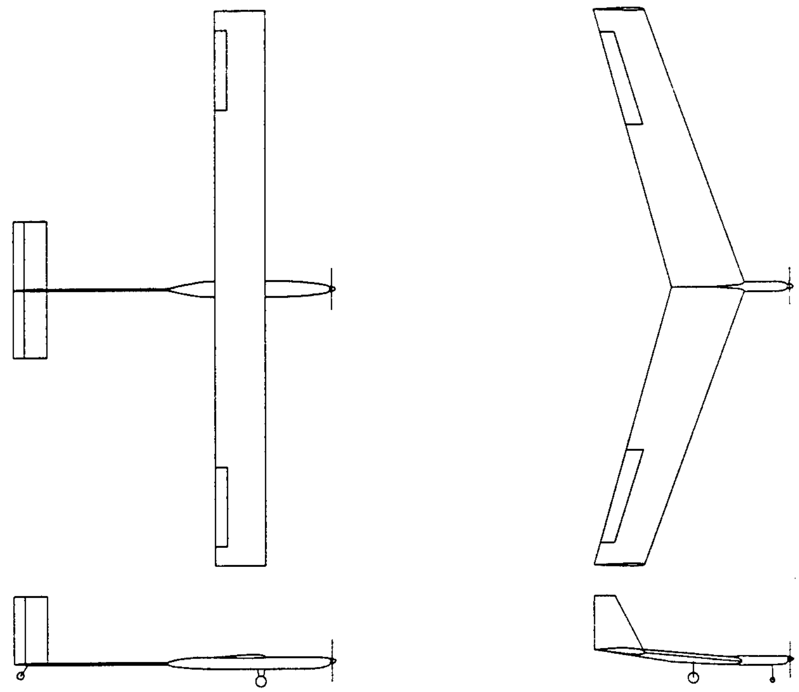


Figure 3.2 Conceptual drawings of the high-wing conventional and flying wing configurations.

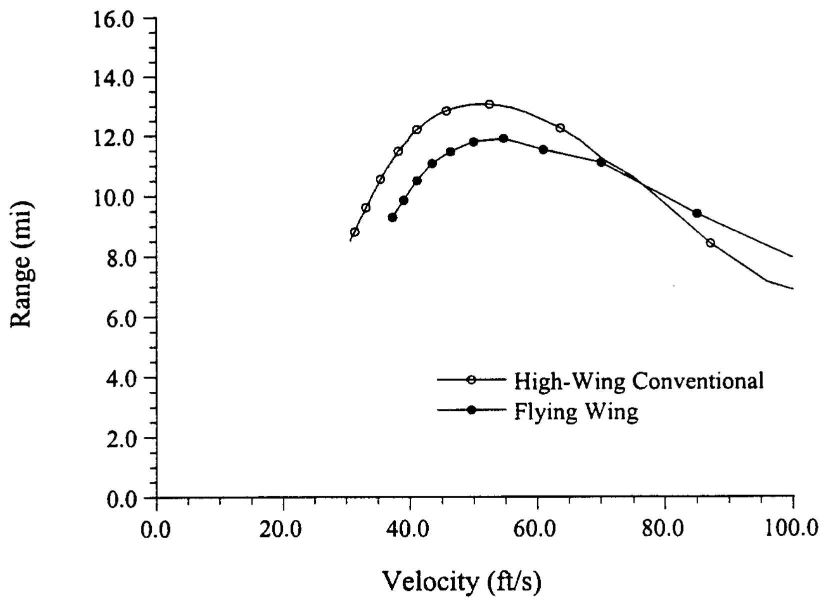


Figure 3.3 Cruise model predictions for the competing conceptual designs.

4.0 Preliminary Design

4.1 Design Parameters Investigated

The conceptual design work revealed a number of important parameters which were included in the preliminary design for vehicle sizing studies. These were:

- Estimate the gross weight of the aircraft and flight Reynolds numbers.
- Size the wing and horizontal tail for maximum range with stability satisfied.
- Conduct wing structural testing to get reliable strength, weight and cost data.
- Conduct wind-tunnel tests to get reliable propulsion system data.
- Build a prototype aircraft to get reliable aircraft sizing data, e.g. weight, construction accuracy, manufacturing processes, size of internal components, flight data, etc.
- Analyze the turn performance and its effect on the overall range.
- Analyze the take-off roll and its effect on the overall range.

Some aspects of the design were reserved for consideration in the final design. Since the fuselage is not required to house geometrically large payloads, it could be small in size and was therefore not considered. Likewise, with the landing gear, the loads were small enough to ignore at this stage of design.

4.2 Analysis Tools

4.2.1 Cruise Model

The cruise analysis (Section 3.3) was improved by including better estimates of the fuselage and landing gear drag and updated weight information and propulsion data. The cruise analysis was used to size the wing and horizontal tail, so as to yield optimum aircraft range.

4.2.2 Strength Testing

To support the aircraft design, it was determined that a structural evaluation of the wing should be conducted. Along with the strength data, the weights and complexity levels of each construction method could be evaluated. To facilitate this, several test wing sections were built. The construction methods are summarized in Table 4.1. All sections had a chord of one foot, a span of three feet and used the Clark-Y airfoil (airfoil type is addressed below). The various components of each test section were weighed and the sections were loaded to failure in bending to evaluate the strength. These data are summarized in Table 4.2. The data show that the foam core sections were much stronger, but also heavier than the others. On the other hand, the built-up section was both the lightest and the weakest.

4.2.3 Propulsion System Testing

The propulsion system testing was conducted to expand Roth's[1] initial data set. A test apparatus was designed and built to use in a low-speed wind tunnel here at the University of Illinois. The test apparatus supplied a controllable DC voltage to an AstroFlight Cobalt 40 motor, geared at 1.7:1. This motor was one of those tested by Roth and was selected for further testing because it provided the best range characteristics when used in the cruise analysis. The test sequence for a given motor/propeller combination was: (1) set the tunnel airspeed, (2) vary the motor voltage input using the manually controlled DC power supply, (3) record the tunnel flow data, motor voltage and current, propeller RPM, and thrust (measured by a strain gauge load cell). Since the only motor tested was the "40 size" AstroFlight, the test control variables were tunnel speed, motor input voltage and propeller size. A test matrix showing the propellers tested is presented in Table 4.3. The data reduction was performed to produce propeller efficiencies (see Fig. 4.1).. Calculation of the propeller efficiency was accomplished by using the motor efficiency supplied by the manufacturer.[3] It should be noted here, that the speed controller was left out of the test to allow for more accurate determination of the propeller efficiency. In reality, there are small inefficiencies in the speed controller, which will affect the overall propulsion system efficiency.

4.2.4 Stability and Control Analysis

In the preliminary design phase, the two major goals of the stability and control analysis were to assure longitudinal static stability and to size the control surfaces. The change in the pitching moment of the aircraft due to a change in angle of attack is the important parameter for static longitudinal stability. The location of the wing and horizontal tail surfaces, along with estimations for the change in moment due to the fuselage and the downwash on the tail determine the aerodynamic center of the aircraft—the point about which the pitching moment does not change with angle of attack. If the aircraft center-of-gravity (c.g.) is in front of this point, the change in pitching moment acts to oppose any change in angle of attack, and the aircraft is stable. The distance between the c.g. and the neutral point non-dimensionalized by the wing chord is referred to as the static margin, and is the standard measure of longitudinal stability. All of the control surfaces and the vertical tail where sized using suggestions based on aircraft historical data from Raymer.[4] At the preliminary design stage, the elevator effectiveness was also checked using a simple moment balance of all the forces acting on the aircraft.

4.2.5 Turns Analysis

The analysis of the aircraft's turn performance was examined in the preliminary design. Since the flying course consists of 180° turns on each end, the effect of these turns on the total range could be important. For this design, an optimal turn is one that consumes the least amount of energy. As the bank angle increases (and the turn radius and turn path length decrease), the aircraft must produce more lift to counteract the centripetal acceleration. This results in higher drag and thus the greater power requirement. The turn-analysis code allowed rapid studies of the effect of the turn speed and radius on energy use.

4.2.6 Take-off Analysis

The take-off analysis was performed to determine how much runway length the aircraft would require to take-off and how much energy would be expended on the take-off roll. A simple take-off model was developed to integrate the equations of motion down the runway using a simple fourth-order Runge-Kutta algorithm. The propulsion system testing provided the required data on thrust as a function of velocity and the aircraft drag polar, which was developed for the cruise model, was used. The take-off analysis did not include ground effects. This was done to predict the "worst case" scenario; ground effects should only "help" the take off, not make it worse. The effect of headwind was straightforward and easily implemented. The rolling friction force due to the landing gear wheels was modeled from experimental data obtained from various sources.[5] As a result, the take-off analysis was felt to account for all factors except for ground effects, and therefore should provide conservative estimates for the take-off performance.

4.3 Results of Analyses – The Prototype Aircraft

This section presents the results of the initial vehicle sizing calculations as they were incorporated into the prototype, YRPR-1. Where possible, the estimated design parameters and the actual values for the prototype aircraft are compared.

4.3.1 Gross Weight and Flight Reynolds Number Estimates

Preliminary estimates indicated a minimum gross weight of 15 lbf; a payload of 7.5 lbf, a battery pack of 2.5 lbf and 5 lbf for the airframe, motor and controls. However, the actual gross weight of the YRPR-1 was 18.51 lbf (see Table 4.4). The cruise velocity of the aircraft was estimated to be in the range of 30 to 60 ft/sec. Based upon standard atmospheric conditions, the corresponding Reynolds number range is approximately 190,000 to 380,000 per foot.

4.3.2 Cruise Model Results

The cruise model was used to select an optimum wing for the prototype based upon the 15 lbf gross weight estimate. The planform was 12 sq ft with a 12 ft span, as shown in Table 4.5 The baseline airfoil chosen for all analyses was the Clark-Y. While this seems like a poor choice in light of the recent advances in low-Reynolds number airfoil design, its low-Reynolds number performance characteristics are relatively good[6] and are shown in Fig. 4.2. This fact, along with its flat-bottom profile, 11.72% thickness, and gentle stall characteristics (see Fig. 4.2), made it a good choice to use as a baseline airfoil in the preliminary studies. The range results are presented in Fig. 4.3. A maximum range of 7 miles is indicated. The low range (compared to Fig. 3.3) is due to changes in fuselage drag modeling.

4.3.3 Structural Analysis Results

The structural design of the wing was quickly determined from a loads analysis and a review of the experimental data from the test section wings. This analysis showed that the built-up balsa construction had adequate strength while providing the lightest wing.

4.3.4 Propulsion Testing Results

A limited set of initial propulsion tests was conducted and utilized in the cruise model. For the prototype aircraft, the APC 13-8 propeller, AstroFlight Cobalt 40 motor combination was selected. This combination produced the best range in the cruise studies. The battery pack consisted of 22 Sanyo 1700 mAh cells and weighed 2.5 lbf. The propulsion system data are shown in Table 4.6

4.3.5 Stability and Control Analysis Results

The wing and stabilizer were located to give a static margin of 10-12%. The ability of the elevator to trim and control the aircraft in both landing and cruise conditions was then checked. The results are shown in the trim plot for cruise conditions, Fig. 4.4. The results of the stability and control analysis are summarized in Table 4.7.

4.3.6 Results of Turns Analysis

The effect of turn bank angle on the amount of energy used is shown in Fig. 4.5 and suggests a bank angle of 50° for minimum energy use. However, it is unlikely that the aircraft will be capable of maintaining this angle at cruise velocity.

4.3.7 Results of Take-off Analysis

The results of the take-off analysis, performed for no headwind, are shown in Fig. 4.6. The discontinuities in the lift and drag are due to rotation. The total energy consumed is 100 mAh, or 4.5 % of the total capacity. The results also show that there is no need for flaps or other high-lift devices to satisfy the takeoff requirement.

4.4 Preliminary Design Summary and Conclusions

A single flight attempt was conducted and resulted in a crash due to insufficient thrust. Through the crash of the YRPR-1, several design considerations became apparent. The lack of sufficient thrust reemphasized the need for good propulsion data and the importance of reducing drag in all aspects of the design. The final airframe weight, 60% higher than predicted, was also cause for a rethinking of the design. The exposure to these problems at an early stage in the process provided ample motivation to consider more aspects of the design than would have been considered otherwise. The YRPR-1 provided much valuable information in terms of weights and sizes of system components, strength and weight of a finished wing versus data from the initial experiments. Finally, the skill level of the construction team was quantified while providing them with some on-the-job training in aircraft construction.

Table 4.1 Test-Wing Section Construction

Section	Core Structure	Spars	Sheeting
1	balsa ribs	1/4" square bass w/ shear webs	1/16" balsa
2	extruded foam	two 1/4" square bass	1/32" balsa
3	beaded foam	two 1/4" square bass	1/64"birch plywood
4	extruded foam/cutouts	two 1/4" square bass	1/64"birch plywood

Table 4.2 Results of Strength Tests

Section	Section Weight (oz.)	Break weight (lbf)	Strength/Weight (lbf/oz)
1	6.4	107	16.72
2	8.9	216	24.27
3	10	179	17.90
4	8.9	> weight capacity	N/A

Table 4.3 Propeller Test Matrix

Manufacturer	Diameter (in)	Pitch
APC	12	6
APC	13	6
APC	13	8
Zinger	12	6
APC	13.5	10
Top-Flite	14	4

Note: All propellers were tested at airspeeds of 0, 10, 20, 30, 40, 50, 60 and 70 ft/sec, with motor input voltages in the 0-20 volt range

Table 4.4 YRPR-1 Weight Breakdown

Component	Weight (lbf)
Propeller	0.13
Motor and Gear Box	0.93
Motor Mount	0.19
Radio Receiver	0.14
Speed Controller	0.13
Main Battery	2.50
Servo Battery	0.28
Payload	7.50
Tail Servos	0.14
Main Gear	0.50
Wheel (2)	0.40
Fuselage	0.78
Tail Boom	0.30
Vertical Tail	0.20
Horizontal Tail	0.20
Tail Boom Mount	0.69
Wing	3.50
Total	**18.51**

Table 4.5 YRPR-1 Wing Data

Span	144 in.
Mean Aerodynamic Chord	12 in.
Area	1728 sq. in.
Aspect Ratio	12
Taper Ratio	1
Sweep	0
Airfoil	Clark-Y
Weight	3.5 lbf (with aileron servos)

Table 4.6 YRPR-1 Propulsion System Data

Motor	Astro Flight Cobalt 40
Gear Ratio	1.7 : 1
Propeller	APC 13-8
Cells	Sanyo 1700mAh
Battery Pack	22 Cells in Series

Table 4.7 YRPR-1 Stability and Control Results

Tail Attributes	Horizontal	Vertical
Area	189 sq. in.	122.5 sq. in.
Aspect Ratio	3.86	2.64
Taper Ratio	0.75	0.75
Airfoil	S8025	S8025
Control Surfaces		
Elevator	30% chord	
Rudder	40% chord	
Ailerons	25% chord, 40% span	
Locations	(measured from propeller line - no spinner)	
Wing MAC	9 in.	
Vertical MAC	69 in.	
Horizontal Tail MAC	69 in.	
Center of Gravity	10.5 in.	
Neutral Point	11.76 in.	

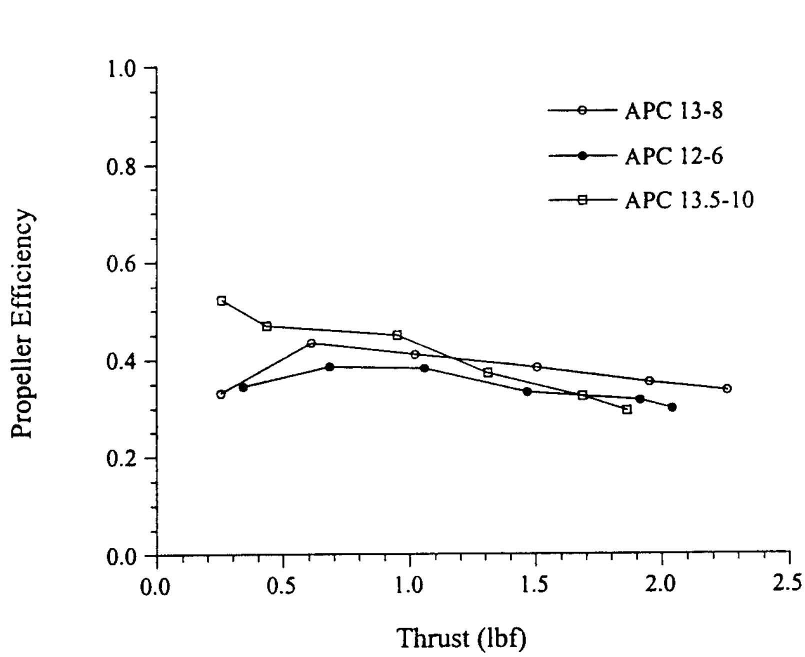


Figure 4.1 Propeller efficiencies at 40 ft/s airspeed for three of the propellers tested in the initial propulsion wind-tunnel tests.

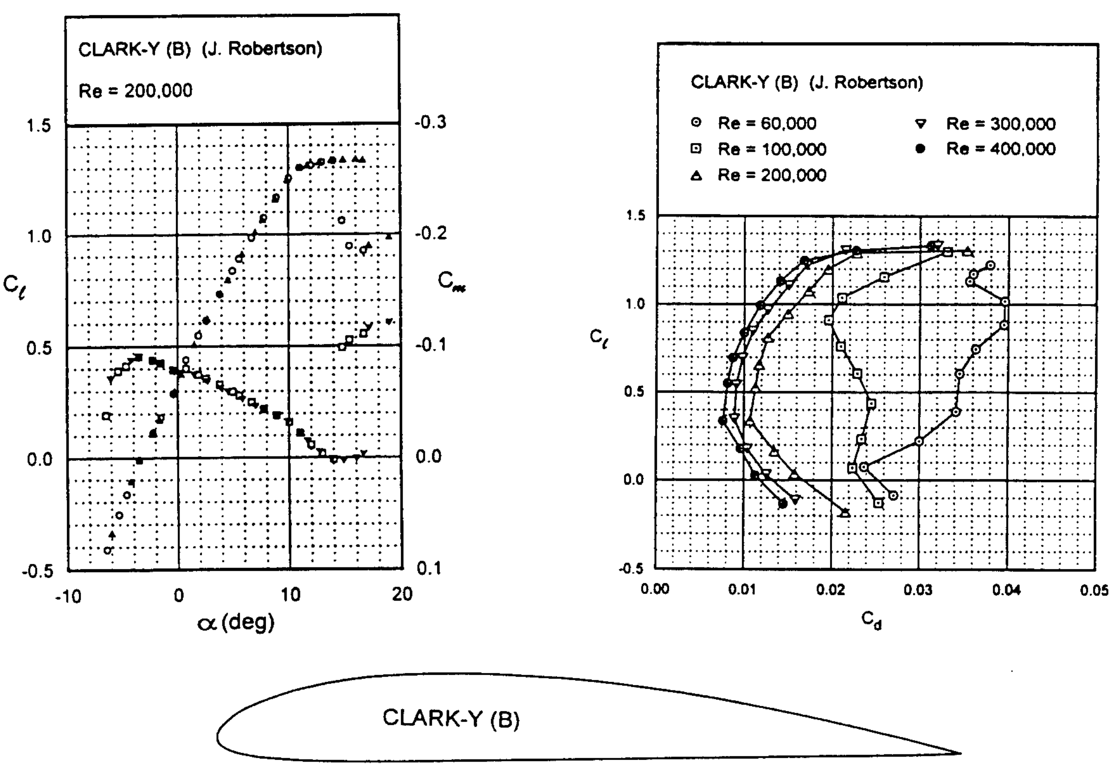


Figure 4.2 Airfoil profile, drag polar, and lift and moment curves for Clark-Y (after Ref. 7).

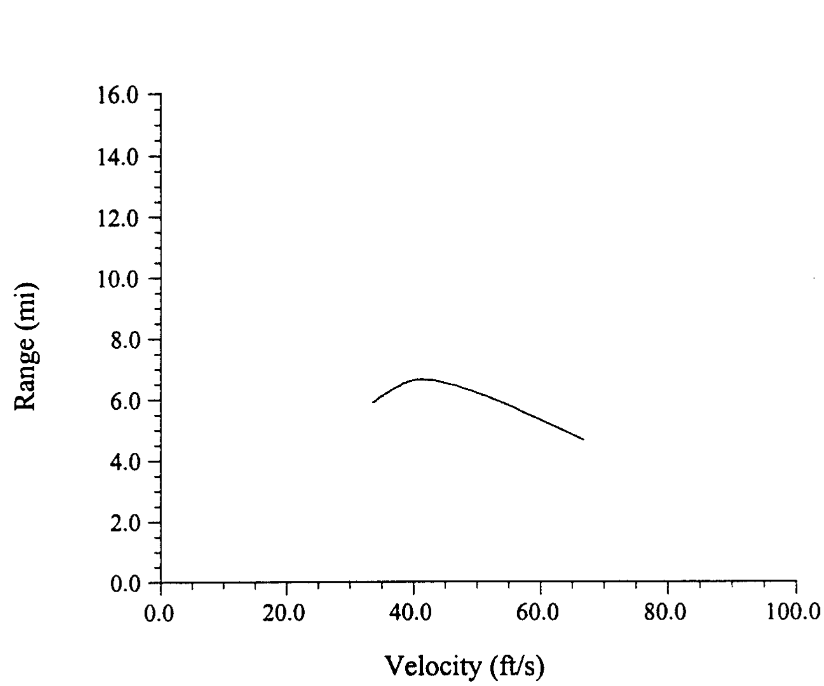


Figure 4.3 Cruise model results for prototype aircraft, YRPR-1.

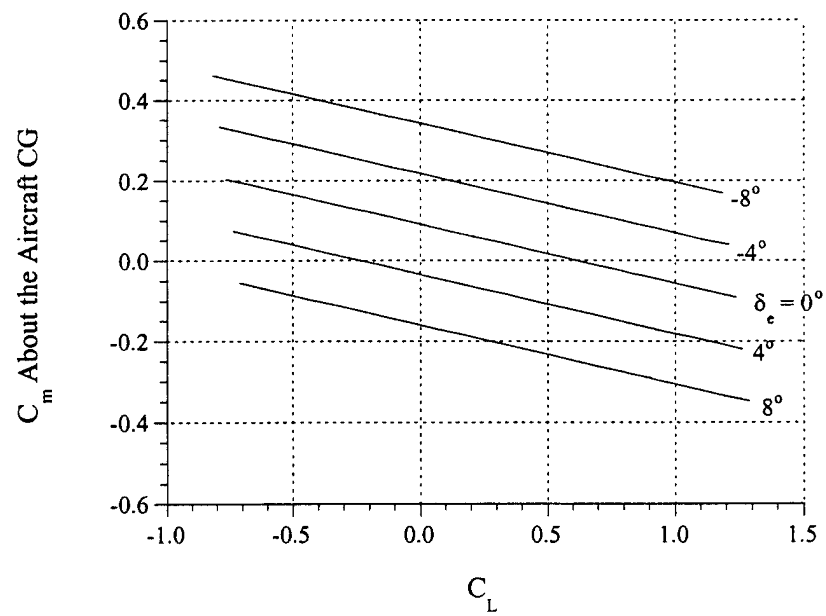


Figure 4.4 Longitudinal trim plot for prototype aircraft, YRPR-1, at cruise.

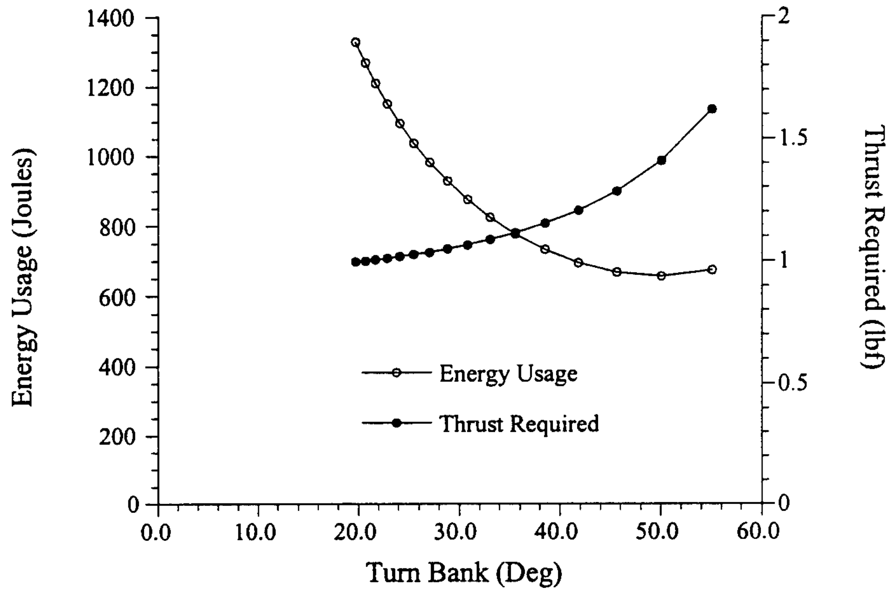


Figure. 4.5 Effect of bank angle on energy usage and thrust required.

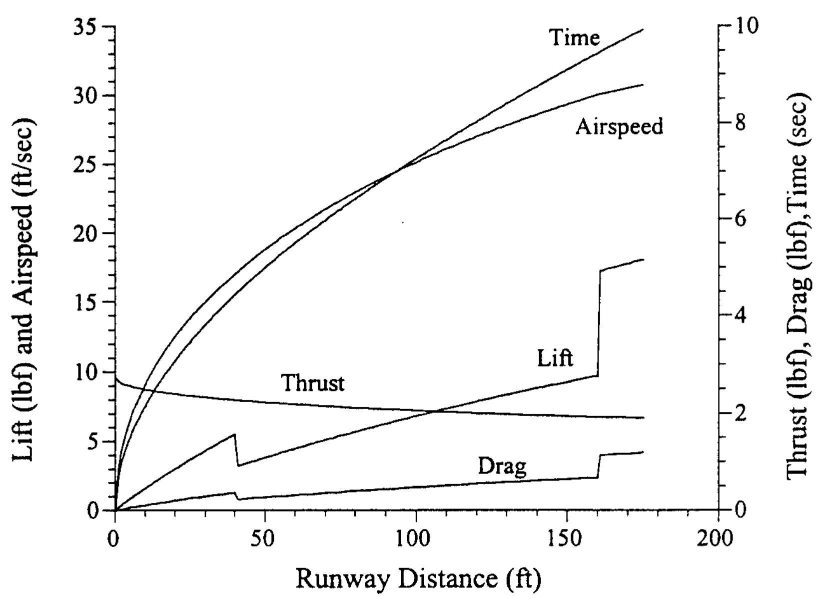


Figure. 4.6 Take-off performance for YRPR-1.

5.0 Detail Design

This section contains all of the performance data for the final design, RPR-1 *Jack*. The drawing package, showing the external and internal views is found at the end of this section.

5.1 Fuselage Design and Internal Configuration

The experience gained from building and flying the YRPR-1 emphasized the need to reduce the overall drag of the airplane. In particular, several important design drivers were identified for the fuselage:

- The fuselage had to be streamlined in shape, with the smallest possible frontal-area necessary to enclose all components.
- The center of gravity of the payload had to be very close to the empty aircraft c.g. so that the handling qualities of the aircraft would remain unaffected by removal of the payload (as required by the competition rules).
- All components, particularly the payload had to be easily accessible for removal and maintenance.
- The main landing gear had to be long enough to provide sufficient vertical clearance for a maximum 15 inch diameter propeller during take-off and landing.
- The fuselage had to be relatively simple and easy to build.

The need for sufficient ground clearance for the propeller and the desire to avoid the drag of long landing gear struts resulted in the low-wing configuration for the final design. As illustrated in Fig. 5.1, the low-wing has several advantages over the high-wing.

The positioning of internal components determined the final exterior shape of the fuselage. The final arrangement of the components is shown in the drawing package (internal view). The weight and balance data are presented in Table 5.1, and since the payload is located at the c.g. its removal will not change the aircraft c.g.

The final stage of the fuselage design involved generating a smooth outer fuselage shape. This was accomplished using a lofting software package called LOFTSMAN, which used conic curves (Raymer[4]) to define the geometry of the entire fuselage.[8] The entire geometry was mathematically defined in this process so that the shape of any arbitrary cross-section of the fuselage could be plotted, see Fig. 5.2. This simplified the construction process.

5.2 Wing Planform Optimization

The objective of the wing optimization for the final design was to maximize the range of the aircraft. The choice of wing span and area was important, since changes to wing area directly affect the zero-lift drag while changes in span directly affect the induced drag of the airplane. The optimization routine used the cruise model and another program written in MATLAB. As shown in Fig. 5.3, the MATLAB program used relations for variations of aircraft weight and drag with wing span and area to generate the input files required by the cruise model. The aircraft-weight variation with wing span and area (developed from the preliminary design database) is shown in Fig. 5.4. The drag variation was obtained by assuming that the sum of the drag contributions due to the fuselage, landing-gear, tail surfaces and interferences is independent of the wing size. The wing profile and induced drags were added to this sum.

The output of the optimization study is illustrated in Fig. 5.5, where the contours indicate the best range for each value of span and area. The model captures the essential features of the expected behavior. As the span increases and the area remains small, the range decreases (above a critical span, owing to increasing wing weight). For low spans and large areas, the range is also decreased, due to large induced drag penalties. From Fig. 5.5, a 14 ft span and 14 sq ft area wing shows an improvement over the 12 ft span 12 sq ft area wing of the prototype. Table 5.2 shows that the difference between these to cases is small, only 6%. This difference was found to be insensitive to the value of "fixed" weight and drag models. Therefore, small errors in estimating the design weight or drag would not significantly change the wing area and span relationship.

This optimization showed that the 12 ft span, 12 sq ft wing is near the optimum planform for this configuration, therefore it was selected for the final design. The experience and confidence that had been gained in building the prototype wing contributed to this decision. Also, larger size wings become impractical to build and transport.

5.3 Landing Gear Selection

The landing gear chosen for the final design consists of a single strut that was mounted in the wing on each side of the fuselage (see drawing package). The Robart RoboStrut system offers shock absorbing capability and variable length. These struts can also be faired easily resulting in reduced drag penalties. Figure 5.6 shows the loading on each strut as a function of vertical landing speed and the landing gear stance. Figure 5.7 shows the wing root bending moment as a function of vertical landing speed and landing gear stance. Based upon this analysis, the landing gear struts were positioned one foot from the fuselage centerline. Figures 5.8 and 5.9 show the landing gear load and the wing root bending moment, respectively, as a function of vertical landing speed for this stance.

5.4 Propulsion System Selection

The final battery pack was selected based upon the argument discussion in Section 3.3. Recall that the most desirable cells were the ones with the highest capacity-to-weight ratio. Sanyo RC2000, 2200 mAh, cells were selected and 19 of these were combined in series to form a battery pack weighing 2.44 lbf. The propulsions tests were extended from the preliminary design and larger diameter, higher pitch propellers were tested. These tests showed that the Zinger 15-12 has the best propeller efficiency of all propellers tested (see Fig. 5.10), therefore it was selected for use in the final design. Two types of electric motors were considered for the final design: brushed and brushless motors. Available literature stated that brushless motors had higher efficiencies and larger power ranges than brushed motors.[9] Therefore, a brushless motor and speed controller system was selected, despite its higher cost. The motor selected was a MaxCim 15-13Y, since the technical data showed it was more than capable of producing the required amount of power. At the writing of this report, a this motor has not been delivered for testing. Therefore, a commercially produced software package called ElectriCalc (version 1.0) was used to determine the exact gearing ratio required. This was determined to be 3.5. The final propulsion system components are listed in Table 5.3.

5.5 Airfoil Selection

An exhaustive search of experimental airfoil data was conducted and analyzed for the final configuration using the cruise analysis. The Clark-Y was selected since it produced the best range. The airfoil selection was limited to airfoils with thickness ratios of approximately 12% or larger. Thinner sections, such as R/C sailplane airfoils, would have required heavier structure, resulting in a large weight penalty. High-lift airfoils were eliminated owing to high drag. This limited the remaining airfoil candidates to a few "sport" R/C airfoils and some older NACA designs. The performance plots for Clark-Y were shown in Fig. 4.2 and the handling qualities were discussed in Section 4.3.2. The horizontal and vertical stabilizers use the same, symmetric S8025 airfoil. Its geometry and performance are shown in Fig. 5.11.

5.6 g-Load Capability

A V-n diagram was constructed to decide on the g-load capability of the final aircraft. The aircraft was designed for limit load factors of 3.0 and -1.5, and a gust velocity of 15 ft/sec at the dive speed. Figure 5.12 shows the resulting V-n diagram with the gust lines overlaid. This analysis shows that the gust loads are always equal to or less than the maneuvering loads.

5.7 Stability and Control Analysis

The analyses of longitudinal stability and control and horizontal tail sizing were performed exactly as in section 4.2.4 and the static margin objective for the final design is 10-12%. All final design values are given in Table 5.4. The resulting trim plots are shown in Figures 5.13 and 5.14. They show pitch sufficient control at both cruise and low-speed conditions in ground effect. The relative angle of incidence between the horizontal tail and wing was set such that the aircraft would trim at the cruise lift coefficient with no elevator deflection.

A similar analysis was performed in yaw and roll in order to size the vertical tail and to set the dihedral angle. These calculations for the final design yielded estimates for yaw and rolling moments due to sideslip angle, $C_{n\beta}$ and $C_{l\beta}$, which compare reasonablely well with those suggested by Raymer[4], and those listed for existing aircraft in Roskam.[10] These two derivatives along with $C_{m\alpha}$, the pitching moment change with angle of attack, are tabulated in Table 5.5 along with those for two other aircraft types.

A trim analysis, similar to that for the longitudinal case, was performed to examine rudder effectiveness. Raymer[4] suggests that an aircraft should be able to operate in a cross wind of 20% of take-off speed using less than twenty degrees of rudder deflection. This is equivalent to an 11.5° side-slip angle. This criterion along with the historical data mentioned previously was used to size the rudder. Figure 5.15, indicates that the final design will trim in this side-slip condition. The ability of the ailerons to hold the aircraft level in this condition was also checked using a similar trim analysis and was found to be sufficient.

5.8 Take-off and Climb Analysis

In the final design stages, a climb analysis was added to the take-off calculations discussed in section 4.2.6. The climb performance was estimated by integrating the equations of motion after the takeoff. Similar to the take-off, the climb analysis also accounted for the effects of headwind but did not account for ground effects. The results for the final design are shown in Figs. 5.16 and 5.17. In the no headwind case, the aircraft barely clears the 10 foot obstacle. This is directly attributable to the thrust profile. The thrust data used for these calculations is for a maximum motor voltage of 16 volts (see Fig 5.18), which was limited by the power supply used during the propulsion tests. The final battery pack will output nearly 23 volts and will result in greater take-off thrust. Thus, the final design should easily satisfy the take-off requirement.

5.9 Final Range, Endurance Performance and Payload Fraction

All of the final design parameters that have not already been tabulated for the final design appear in Tables 5.6 to 5.8. Incorporating all of the final design parameters into the cruise model yielded a maximum range of 14.5 miles as shown in Fig. 5.19. The endurance at this condition is 31.5 minutes (see Fig. 5.20) and the corresponding drag profiles are shown in Fig. 5.21. The payload fraction is 0.44. The "energy budget" of the final design mission is given in Table 5.9. The data show that the take-off and climb use minimal amounts of energy and that the turns use a significant amount. The number of complete laps predicted for the RPR-1 final design is 37, or 12.5 miles. Note that this differs from the maximum cruise distance of 14.5 miles. This difference results from the additional energy consumption due to the turns and take-off and climb. Since the turns analysis resulted in an optimistic prediction (see Section 4.3.8), the actual number of laps will be lower than 37. The actual number of laps is also likely to be lower than predicted because of uncertainties in system modeling and off-design flight conditions.

5.10 Conclusions

This section has described all analyses and results used to satisfy the design requirements established in the rules. In addition to meeting these requirements, the aircraft design has been optimized for the range competitive performance parameter. The design also contains many improvements over the YRPR-1 prototype. Specifically, advances in fuselage, landing gear design and propulsion system have increased the accuracy of the performance predictions.

Table 5.1 Weight and Balance Data

Component	Weight (lbf)	Distance from Propeller (in)	Weight*Distance (lb-in)
Propeller	0.13	0.00	0.00
Engine	0.59	3.00	1.78
Radio Receiver	0.14	26.50	3.64
Speed Controller	0.13	6.50	0.85
Main Battery	2.44	11.70	28.52
Servo Battery	0.28	16.00	4.40
Payload	7.50	20.60	154.65
Tail Servos	0.14	29.00	3.99
Main Gear	0.68	18.25	12.32
Fuselage	0.71	20.00	14.26
Tail Boom	0.29	48.00	13.87
Tail	0.55	64.00	35.01
Tail Wheel	0.11	64.00	7.23
Spinner	0.08	0.00	0.00
Control Rods	0.09	48.00	4.51
Wing	3.34	20.75	69.26
Total	17.18		354.31
		C.G. (in. from prop.)	20.62
		Payload Fraction	0.44

Table 5.2 Comparison of Maximum Range for Wing Planform Optimization

Area (ft^2)	Span (ft)	Range (mi)	Difference
14	14	15.5	--
12	12	14.5	6%

Table 5.3 Propulsion System Data for Final Design

Motor	Max-Cim 13-15 Y Brushless
Gear Ratio	3.5 : 1
Propeller	Zinger 15-12
Cells	Sanyo RC 2000 2200mAh
Battery Pack	19 Cells in Series
Speed Controller	MAXμ35-25NB

Table 5.4 RPR-1 Stability and Control Results for Final Design

Tail Attributes	Horizontal	Vertical
Area	264 sq. in.	96 sq. in.
Aspect Ratio	4.13	1.5
Taper Ratio	1	1
Airfoil	S8025	S8025
Control Surfaces		
Elevator	30% chord	
Rudder	30% chord	
Ailerons	17% chord, 40% span	
Locations	(measured from propeller line - no spinner)	
Wing MAC	19.00 in.	
Vertical Tail MAC	62.38 in.	
Horizontal Tail MAC	62.38 in.	
Center of Gravity	20.63 in.	
Neutral Point	21.84 in.	
Miscellaneous		
dihedral	2° on outboard panels	
static margin	10.13%	

Table 5.5 Comparison of Lateral Directional Stability Derivatives

	RPR-1 Jack	4-place General Aviation*	Medium Size Business Jet*
$C_{n\beta}$	0.245	0.065	.127
$C_{l\beta}$	-0.148	-.089	-.11

*Data from Ref. 10

Table 5.6 Wing Data for Final Design

span	144 inches
mean aerodynamic chord	12 inches
area	1728 square inches
aspect ratio	12
taper ratio	1
sweep	0
airfoil	Clark-Y
estimated weight	3.4 lbf with aileron servos

Table 5.7 Drag of Miscellaneous Components

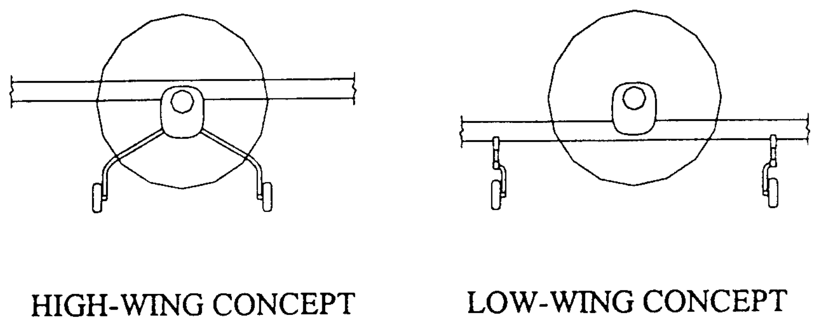
Component	Estimated Drag (C_DA/S_{ref})
Fuselage	0.00313
Interference	0.00301
Landing Gear	0.00407
Total	0.01021

Table 5.8 Miscellaneous Aircraft Components

Component	Brand	Model	Comments
Radio	Airtronics	IN660	6 channels
Servos	Airtronics	94102	Standard size
Landing Struts	Robart	RoboStrut 671	0.5" diameter
Wheels	Robart	Smooth Tread Scale Wheel	2.25" diameter
Spinner	TruTurn	TT-2002B	2" diameter

Table 5.9 Distrubution of Battery Energy Among Flight Mission Segments

Mission Segment	Energy Consumed	Percent of Total Energy
Take-off and Climb	7436 J	4.2%
360 deg Turns × 2	2620 J	1.4%
Cruise: straight flight	122,174 J	67.6%
Cruise: turning flight	48,470 J	26.8%
Totals	180,700 J	100%



Advantages of the Low-Wing Concept

- Reduced landing gear strut length and hence reduction in the associated weight and drag.
- Increase in landing gear stance, resulting in greater stability during taxi, takeoff and landing.
- Movement of the gear and gear attachment out of the propeller slipstream - resulting in further drag reduction.
- More efficient wing-fuselage structural connection.

Figure 5.1 Comparison of low and high-wing configurations.

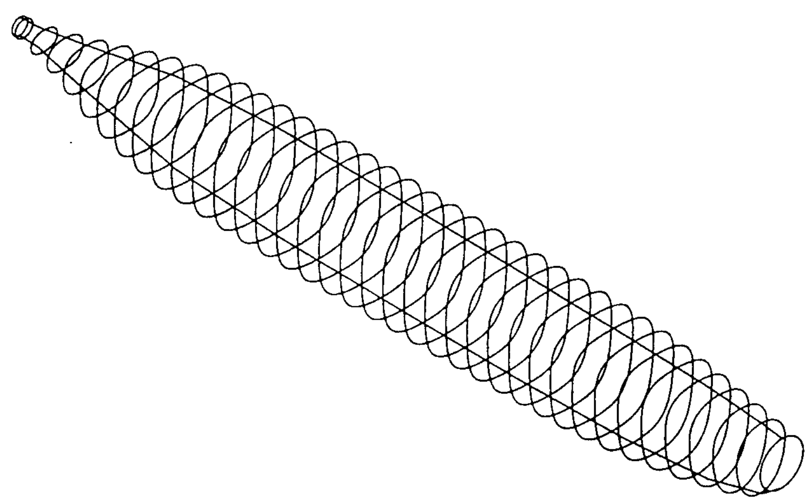


Figure 5.2 Fuselage cross sections from LOFTSMAN.[8]

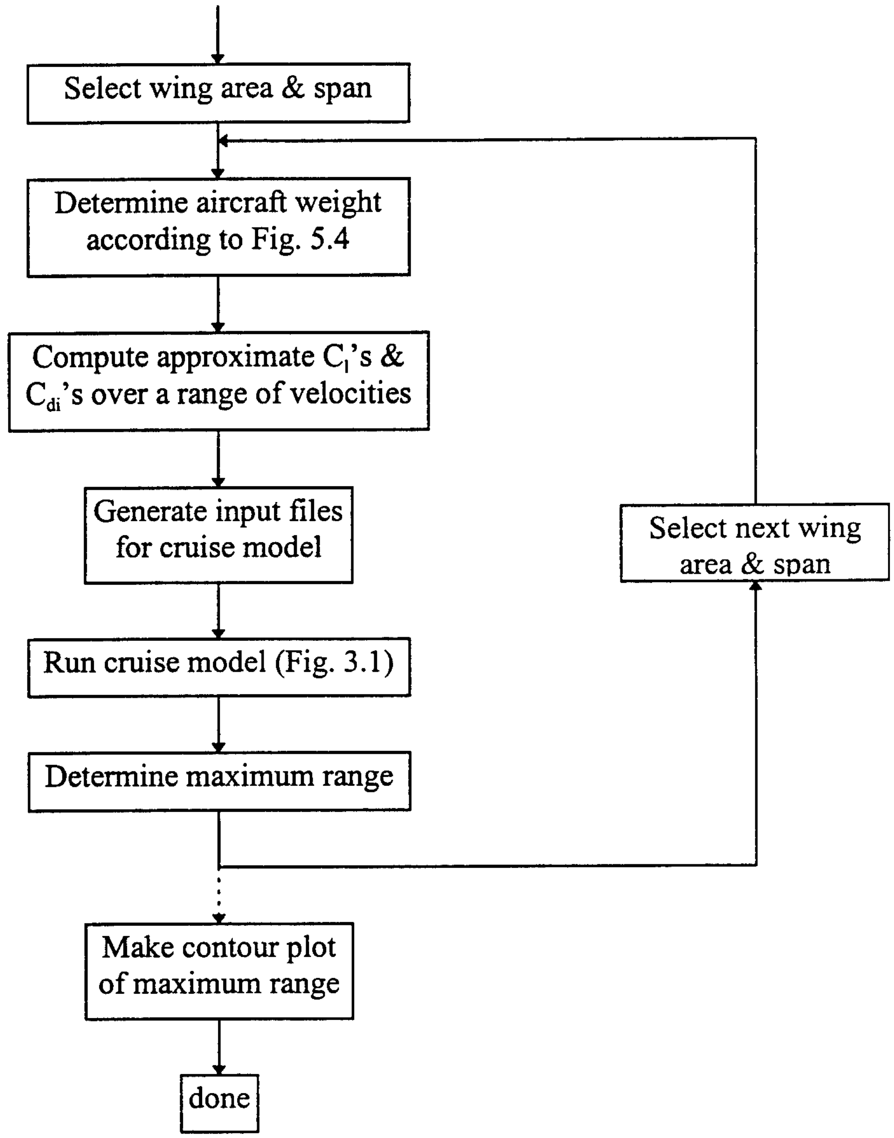
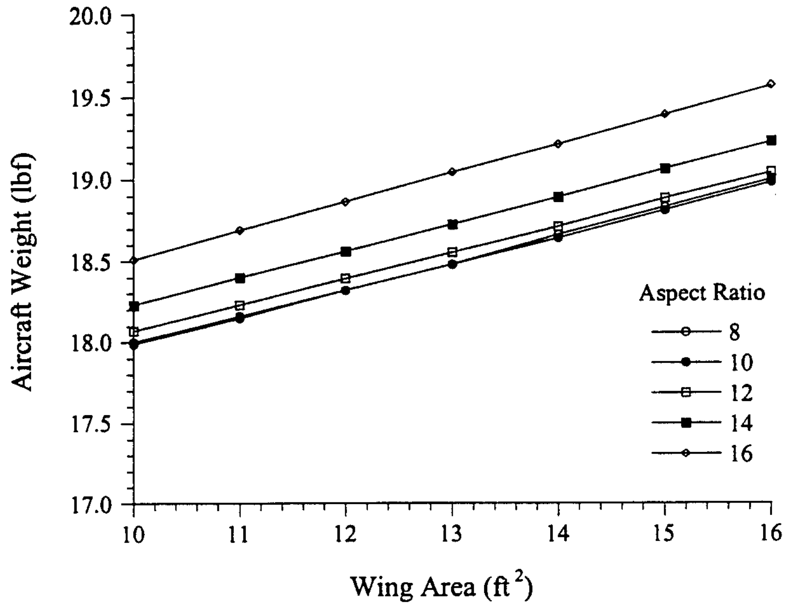


Figure 5.3 Wing optimization program flow chart.



Modeling Equations

S = Area (sq ft)
b = Span (ft)
AR = Aspect Ratio = b^2/S
W = Weight (lbf)

$$W_{total} = W_{fixed} + W_{sheet} + W_{ribs} + W_{spar}$$

W_{fixed} = aircraft gross weight - wing weight

$$W_{sheet} = 0.0793 \times S$$

$$W_{ribs} = 0.0420 \times S^2/b$$

$$W_{spar} = [\ 0.0221 + 0.2085\times10^{-2}\times AR - 0.4796\times10^{-3}\times AR^2 + 0.3360\times10^{-4}\times AR^3\ ] \times b$$

Figure 5.4 Aircraft weight as a function of span and area.

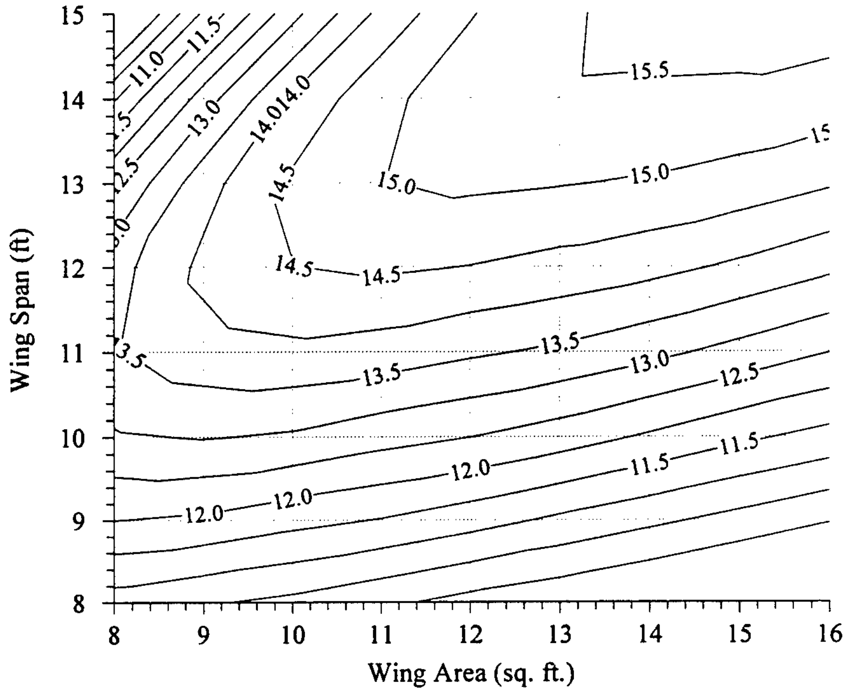


Figure 5.5 Range contours (in miles) as a function of wing span and area.

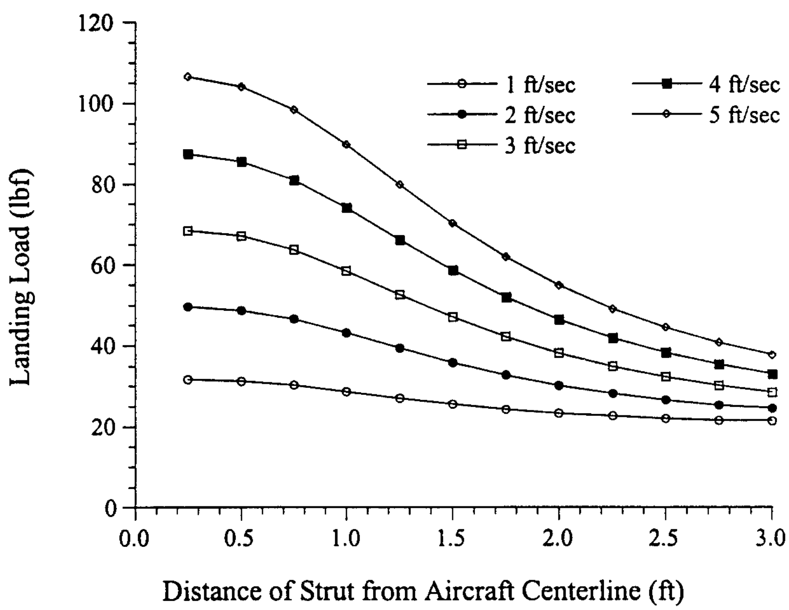


Figure 5.6 Landing gear strut loading for several vertical landing speeds.

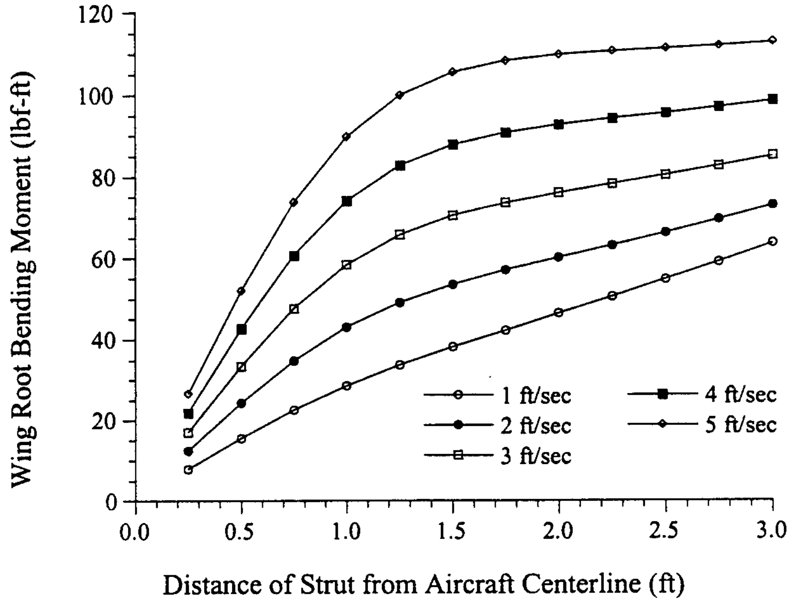


Figure 5.7 Wing root bending moment as a function of strut distance from centerline for several vertical landing speeds.

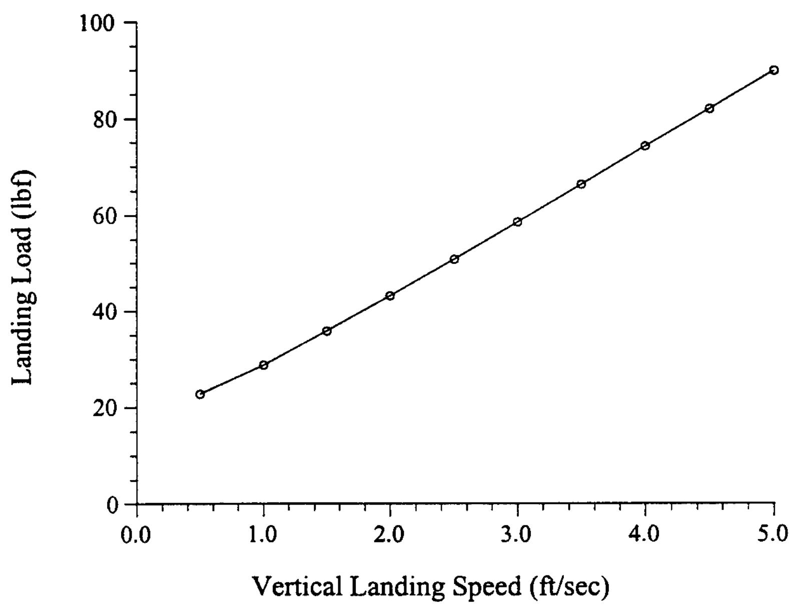


Figure 5.8 Strut loading for final design gear stance.

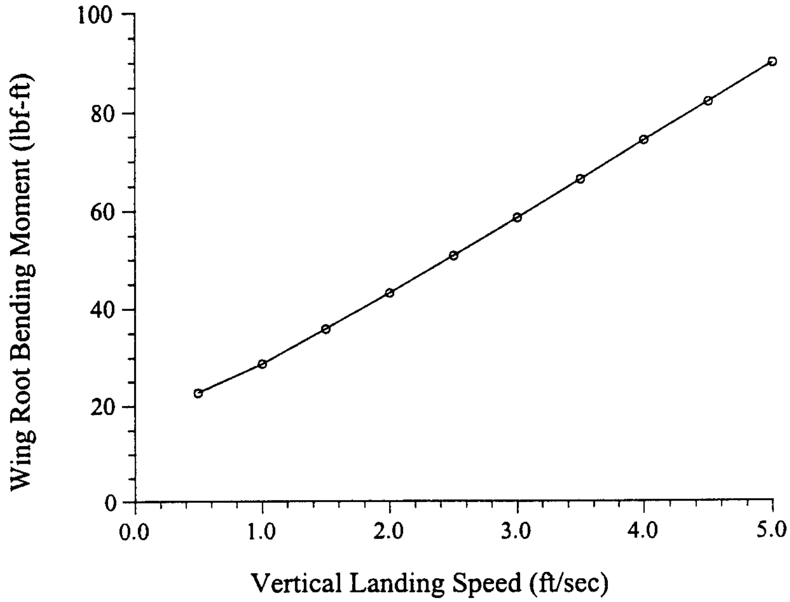


Figure 5.9 Wing root bending moment for final design gear stance.

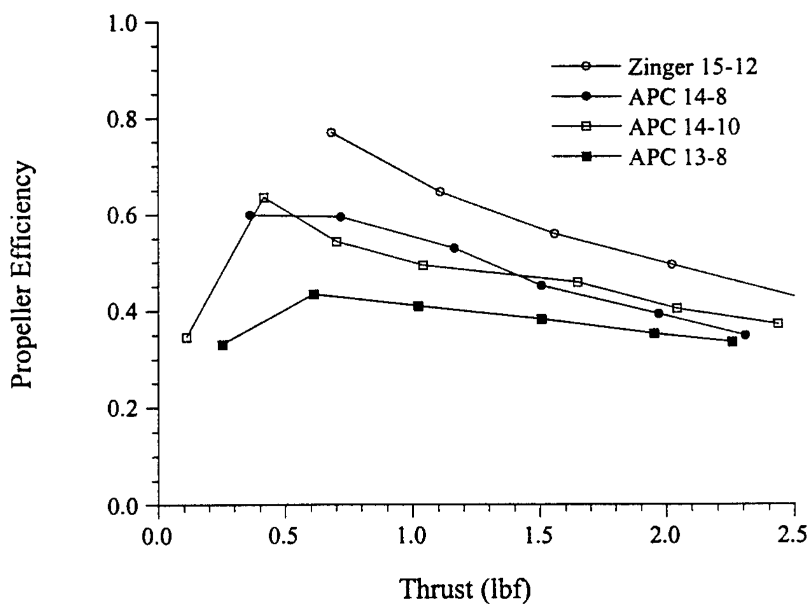


Figure 5.10 Propeller efficiencies at 40 ft/s airspeed for three propellers tested in the final propulsion wind-tunnel tests. (APC 13-8 is from initial tests, shown here for comparison)

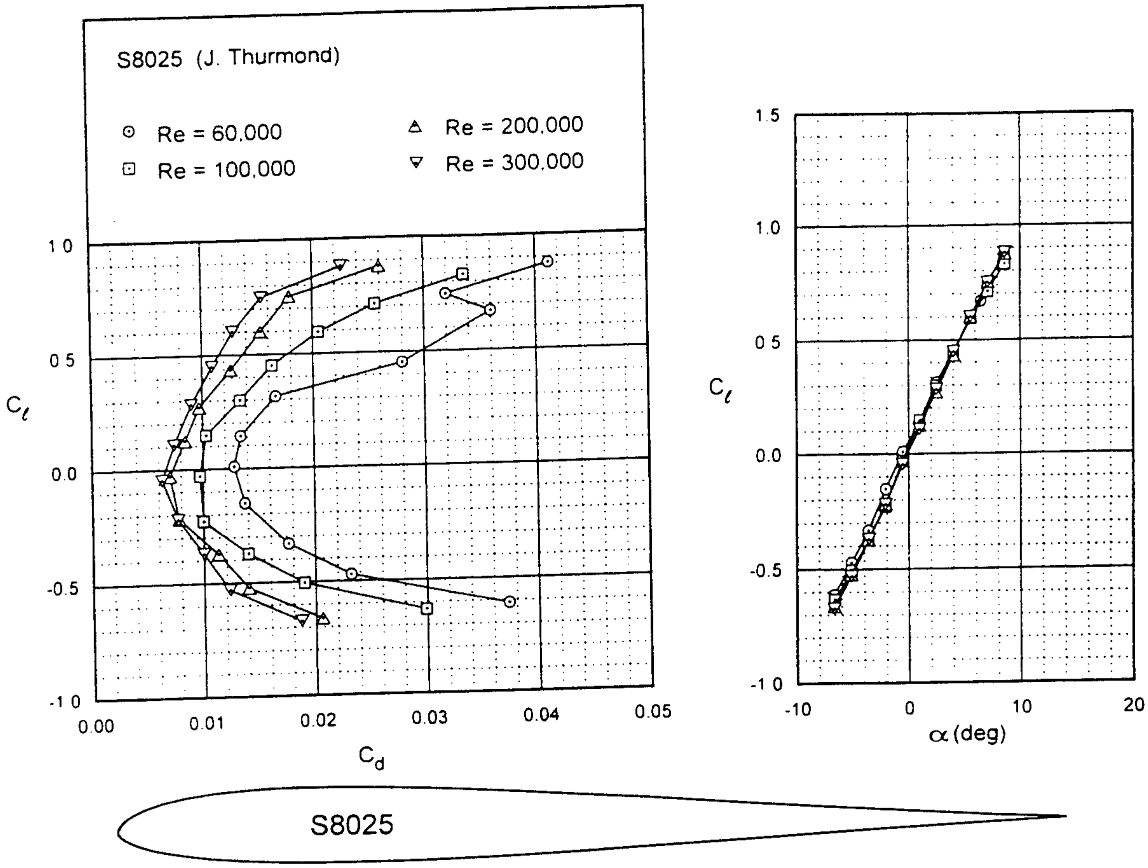


Figure 5.11 Performance data and airfoil profile for S8025 (after Ref. 11)

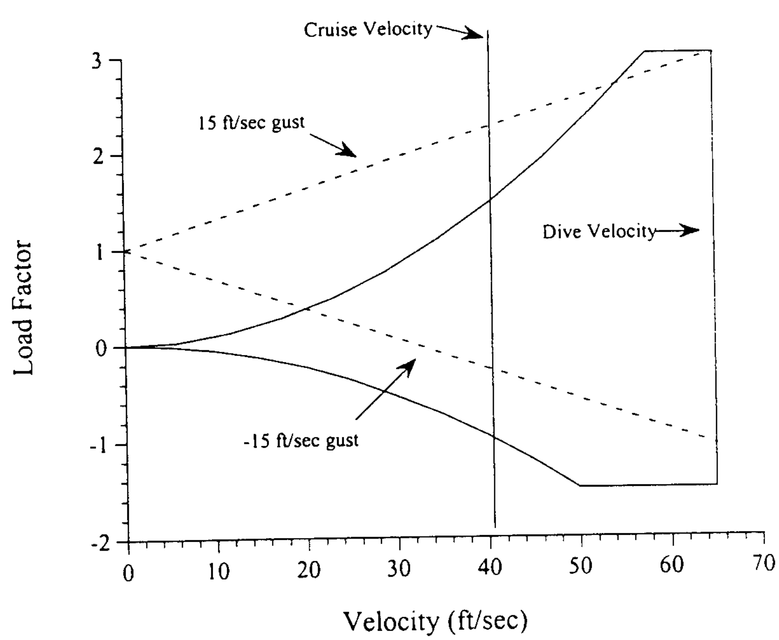


Figure 5.12 V-n diagram

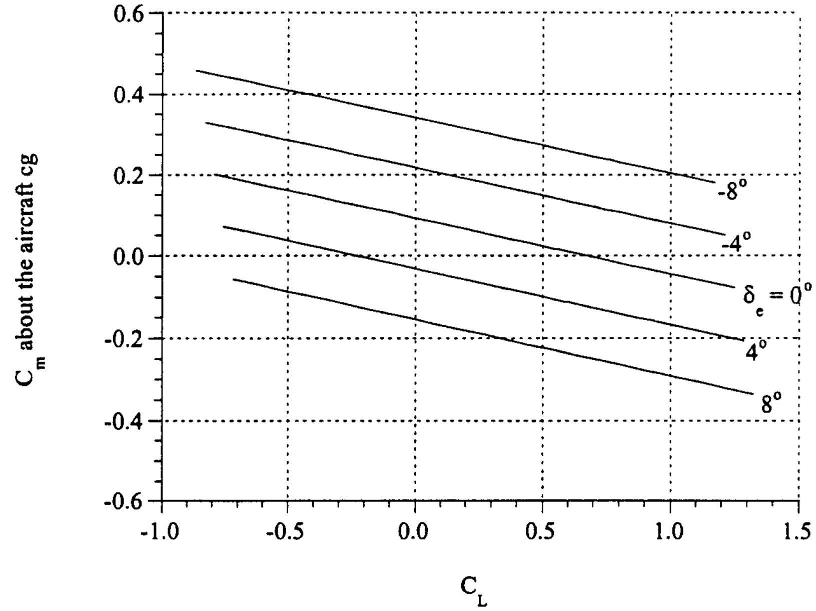


Figure 5.13 Longitudinal trim plot at cruise condition.

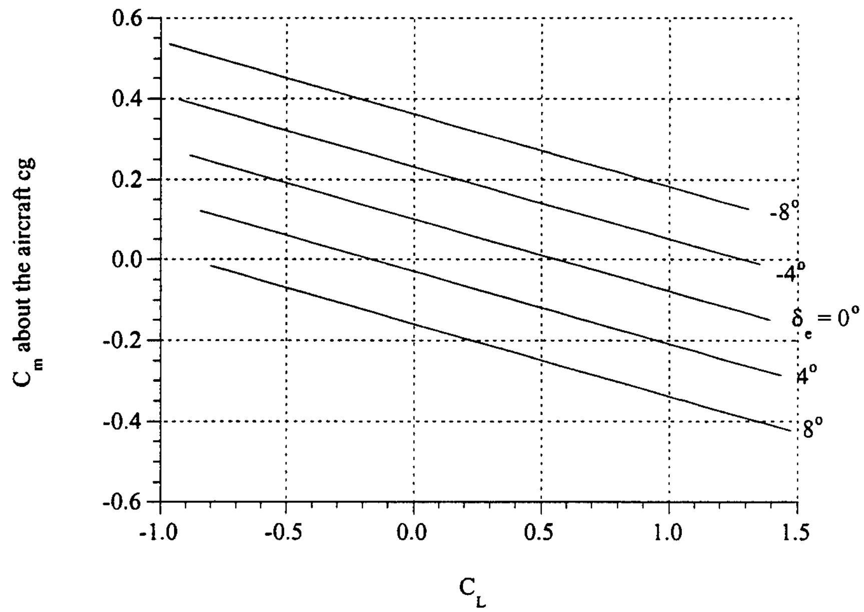


Figure 5.14 Longitudinal trim plot at takeoff/landing conditions, i.e. low-speed-in ground effect.

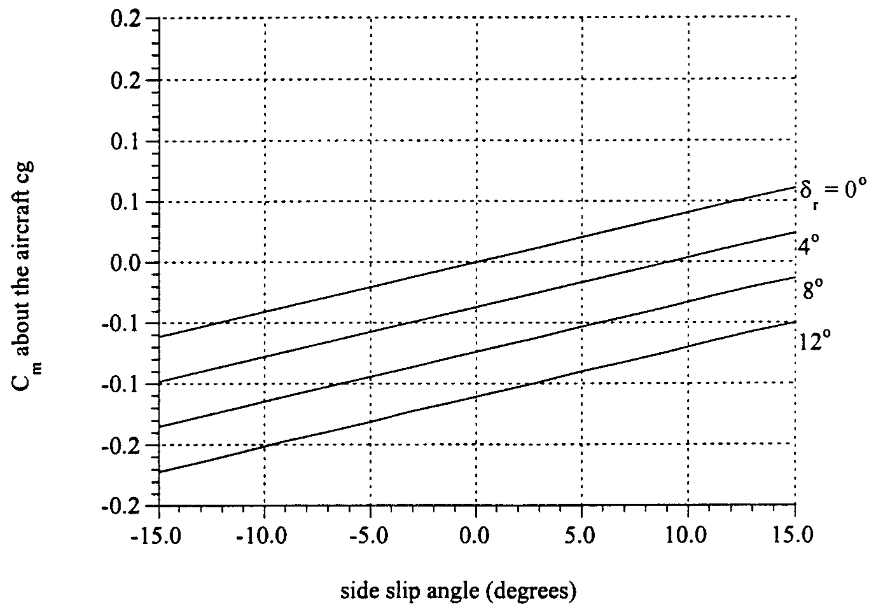


Figure 5.15 Rudder trim plot demonstrating ability to trim in sideslip conditions.

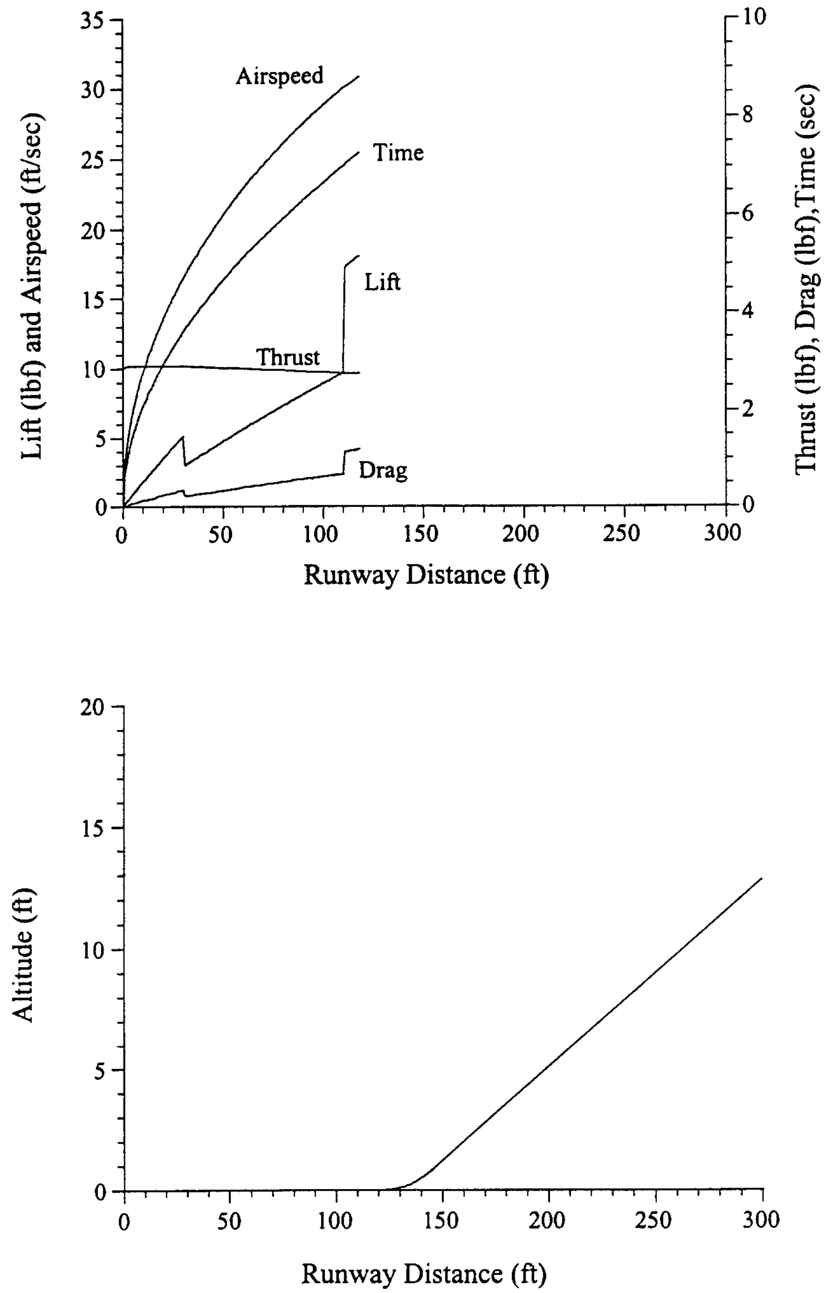
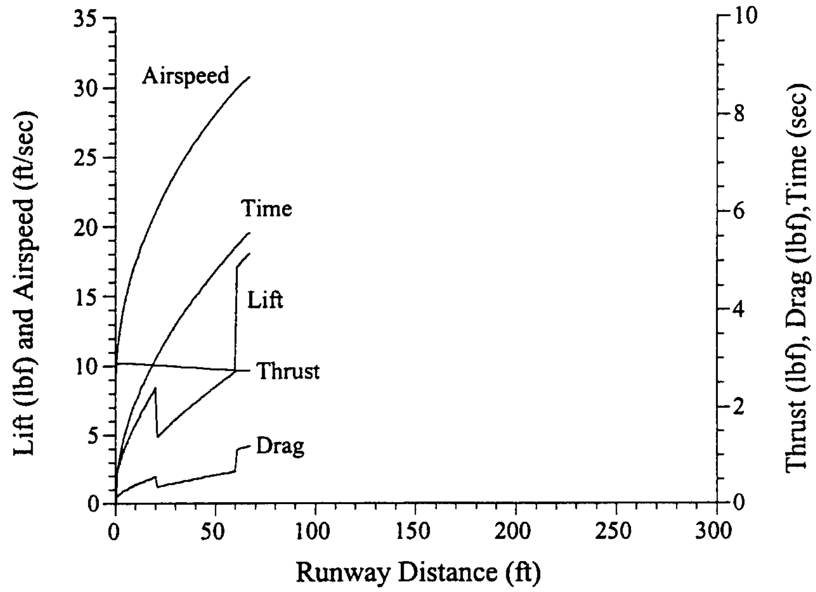


Figure 5.16 Results of take-off and climb analysis - no headwind.



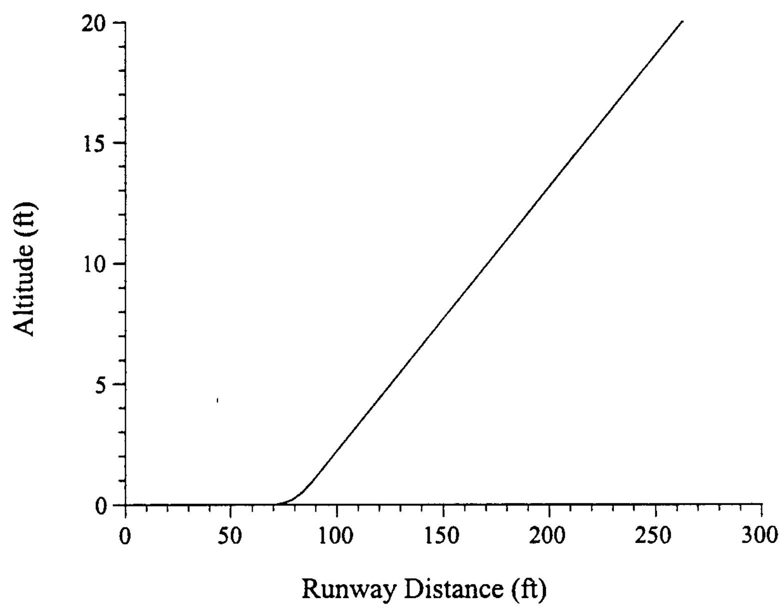


Figure 5.17 Results of takeoff and climb analysis for a 5 mph headwind.

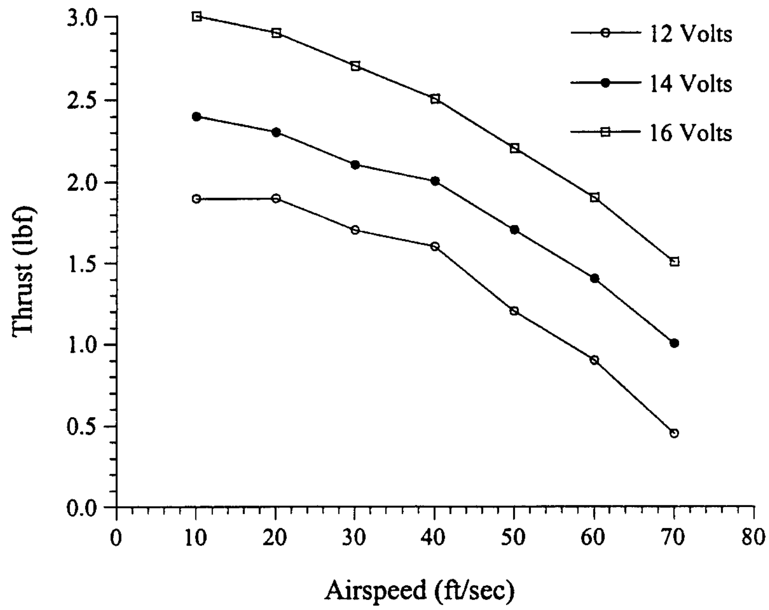


Figure 5.18 Thrust profiles for various motor voltages, Zinger 15-12 propeller.

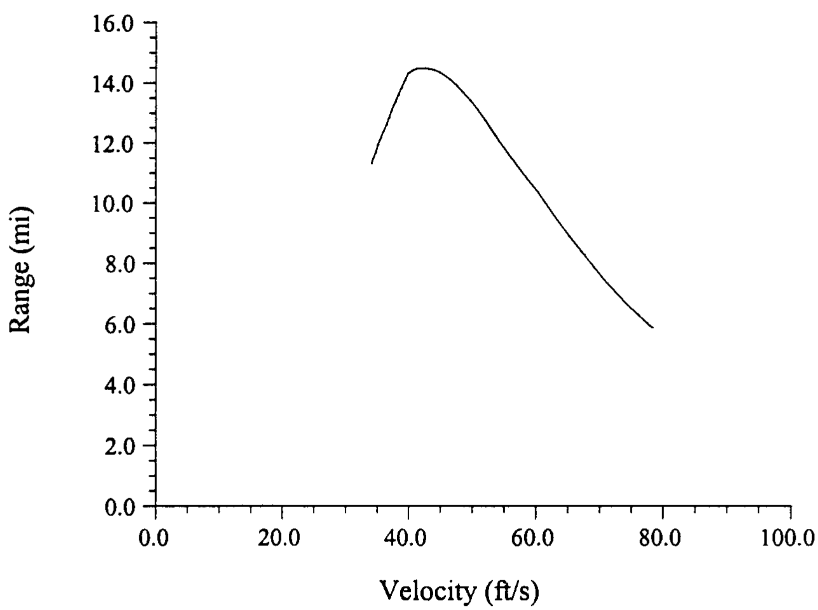


Figure 5.19 Results of cruise analysis - range.

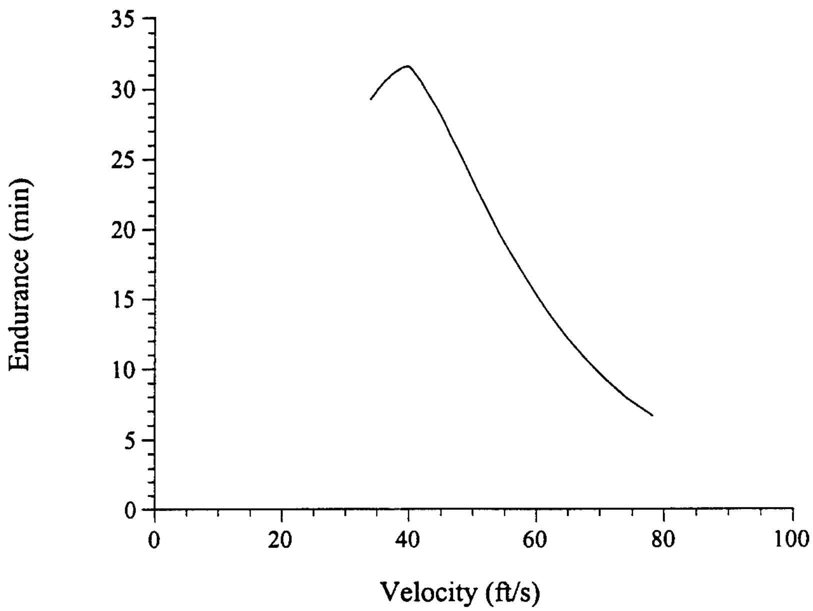


Figure 5.20 Results of cruise analysis - endurance.

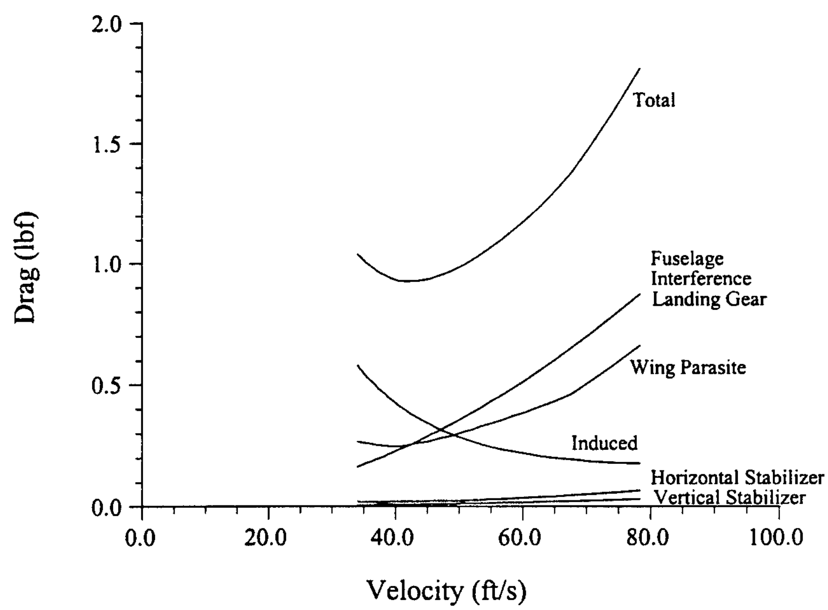


Figure. 5.21 Results of cruise analysis - drag profiles.

6.0 Manufacturing Plan

6.1 Component Requirements and Design Drivers

The overall component requirements and design drivers used to select the manufacturing processes for the final design are:

- The components must satisfy the structural requirements with an adequate factor of safety.
- The components must be built as lightweight as possible, utilizing common manufacturing techniques while maintaining good accuracy.
- The components should be manufactured from readily available, low cost materials.

In addition to these general requirements, there are special requirements unique to particular components. For example, the fuselage requires easy access to internal components and easy installation/removal of the payload. The component-specific requirements are discussed below, in connection with the manufacturing processes investigated.

6.2 Manufacturing Processes Investigated

Several possible construction techniques were investigated for each of the major aircraft components. These methods are very common in model aircraft construction and helped to form the basis of the analysis tools discussed in Section 6.2. "Major aircraft components" refers to the wing, fuselage, empenage, and tailboom.

6.2.1 Wing

The wing construction was the most critical, and relied upon experience with the prototype and the manufacturing techniques described in Section 4.2.2. The available techniques involved combinations of foam core or built-up sections sheeted with balsa or thin plywood, containing basswood or composite spars. Additional methods consisted of foam cores with composite skins. The possibility of constructing the wing in multiple sections using different methods was also considered.

6.2.2 Fuselage

Specific design requirements for the fuselage were specified as: (1) smooth, streamlined shape (see Section 5.1), (2) structural integrity, and (3) easy access to internal components and payload. To meet the first requirement, it was decided that the fuselage external skin, or "shell", must be molded from composite materials. The shell could be a structural component or simply a covering for aerodynamic fairing only. There were several manufacturing techniques investigated for constructing this shell. The most viable techniques included laying up composite layers over a foam plug which could later be removed, or using a similar plug to construct a female mold in which the shell could be formed. The non-structural skin fuselage would require a lightweight (wooden) structural frame, contained within the shell.

6.2.3 Empenage

The manufacturing processes associated with the tail surfaces were essentially the same as those used for the wing. However, the structural requirements were not nearly as severe, which made some techniques more appropriate.

6.2.4 Tailboom

The final design features a long cantilevered tailboom protruding from the fuselage. This proved to be a difficult design problem, and several manufacturing methods where investigated including a custom built composite structure consisting of a foam core, carbon strengthening members, and a fiberglass shell. A carbon fiber composite tube; either pre-made or custom manufactured using a carbon fiber sleeve was also considered.

6.3 Figures of Merit

Figures of merit (FOMs) for screening competing manufacturing processes were established based upon the component requirements and design drivers listed in Section 6.1. The quantitative value judgments for each of these FOMs are summarized in Table 6.1.

- Structural Adequacy: The ability of the resulting component to satisfy the structural requirements.
- Weight: The overall weight of the resulting component.
- Skill Required: The required skill level of the construction team necessary to execute a particular manufacturing process with high accuracy.
- Materials Availability: The ease and speed of acquiring the building materials.

- Time Required: The amount of time required to complete a particular component for a given manufacturing process.
- Cost: The cost of the materials required to use a given manufacturing process. This also accounts for the cost of "specialized" tools or equipment associated with a given manufacturing process.

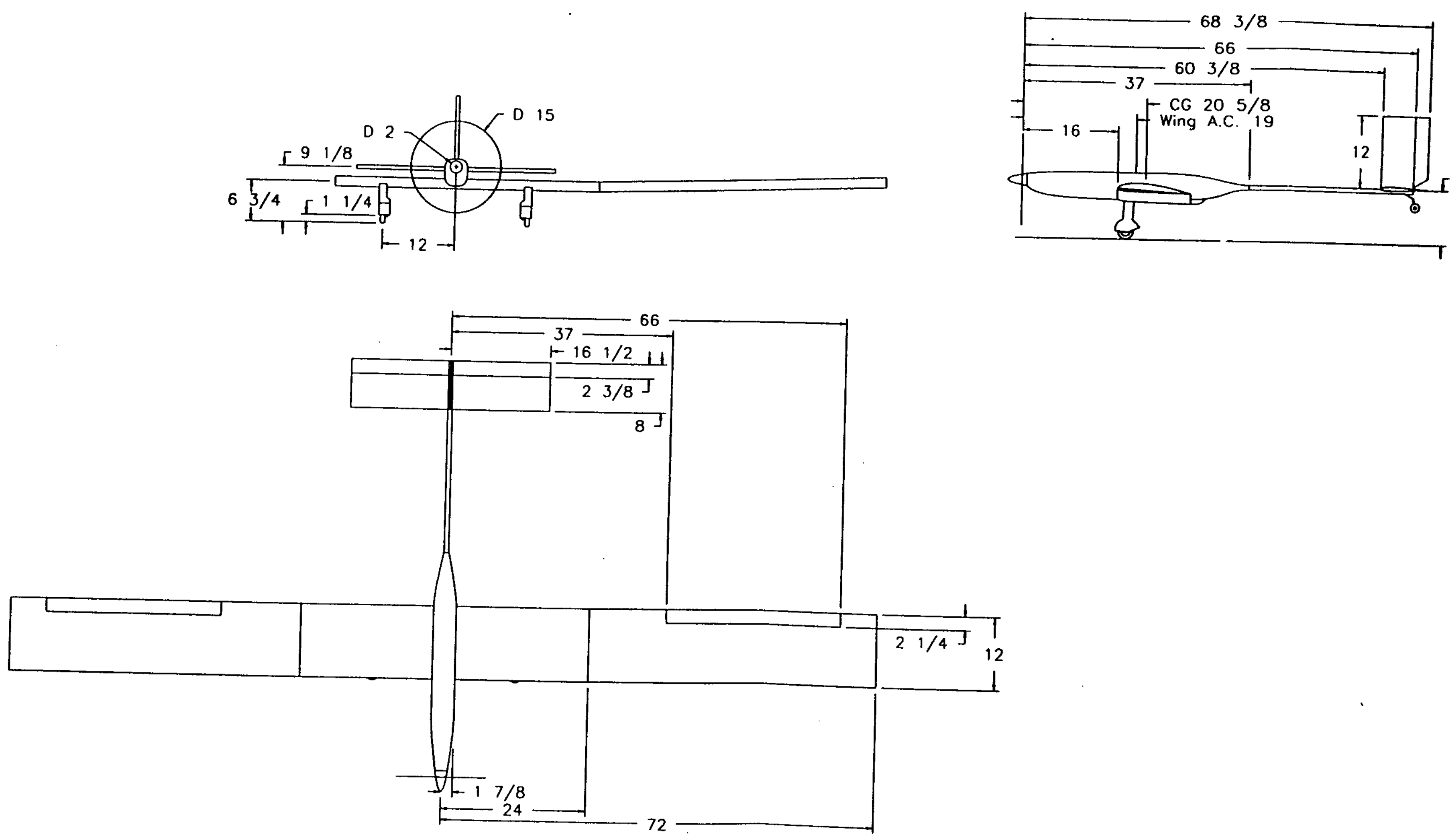
6.4 Methods of Analysis Used to Screen Manufacturing Processes

The analysis method for screening the manufacturing processes was evaluation based on past experience, instead of using elaborate manufacturing models. For example, the experience base built from the test-wing sections was utilized. This provided excellent design data in terms of evaluating strength, weight, time to build, cost, etc. The same is true for the prototype aircraft. Most of the manufacturing decisions associated with the wing, fuselage, tail and tailboom were made on the basis of previous experience with the prototype and of the R/C modelers on the project team. The prototype construction experience offered information regarding: (1) an evaluation of the skill of the construction team, (2) how quickly materials could be obtained from certain suppliers and revealed new suppliers of materials, (3) construction times for various components, and (4) how well students with busy class schedules could work together to get the job done. All of this experience was drawn upon during the evaluation of the figures of merit.

6.5 Results: Manufacturing Processes for the Final Design

6.5.1 Wing

The wing manufacturing process for the final design wing is a combination of composite and built-up construction. The final ranking of the competing processes is shown in Table 6.2. In the prototype aircraft, built-up construction was chosen for the wing surfaces. However, it was decided that such a wing strength would not have a high enough factor of safety. This is reflected in the assigned figures of merit for the final design. The final design wing has a four foot center wing section built from an extruded foam core, covered with thin balsa and two-four foot outboard sections built-up from basswood spars and balsa. The choice of a stronger center section was also motivated by the additional loads due to landing gear impacts.



University of Illinois at Urbana-Champaign	
Aeronautical & Astronautical Engineering Department	
AIAA Student Design/Build/Fly Competition :996/1997	
RPR-1 Jack	External 3 View (All dimensions in inches)

RPR-1 *Jack* Project Team

Faculty Advisor

- Dr. Kenneth R. Sivier

Upperclassmen

- Andy Broeren
- Ashok Gopalarathnam
- Chad Henze
- Benjamin Keen
- Sam Lee
- Christopher Lyon
- Joshua Minks

Underclassmen

- Patrick Boyssmith
- Martin Klipp
- Chong Hin Koh ← non member
- Pong Lee
- Shalim Mody
- Vijay Ram ← non member
- Patrick Schuett ←

6.5.2 Fuselage

The manufacturing process selected for the final design fuselage is a non-structural external shell, with an internal structure built from aircraft spruce. The final ranking of the competing processes is shown in Table 6.3 and shows that the choice of using a thin composite shell (non-structural) was made based on the estimated inability of a structural shell to support the required loads. It was decided that the fuselage shell would be constructed by laying up composite layers over a foam plug and later removing the plug. This decision was made based on the techniques simplicity and relatively short construction time.

6.5.3 Empenage

The manufacturing process selected for the final design tail surfaces consists of a foam core and fiberglass covering. The final ranking of the competing processes is shown in Table 6.4. Owing to their relatively small size and small aerodynamics loads, the weight of the tail surfaces was less important than that of the wing, therefore ease and quickness of construction was more important.

6.5.4 Tailboom

The selected "manufacturing process" for the final design tailboom is to buy a commercially available carbon fiber tube. The final ranking of the competing processes is shown in Table 6.5. The pre-made tube was selected because it was a simple and easy solution to a difficult problem. The time saved in not custom manufacturing a tailboom far outweighed the additional cost, and this time was devoted to custom manufacturing the other components.

6.6 Construction Details

The wing center section core was cut from housing insulation foam using a hot wire as were the cores for the empenage. A carbon fiber tube similar to that chosen for the tailboom was installed in the top surface of the wing as a spar. This entire structure was then sheeted with 1/32 inch balsa, and a balsa leading edge was installed. The outboard panels were built up of balsa ribs and shear webs, along with ¼ inch square basswood spars. The details of this construction are shown in the drawing package. These panels were then sheeted using 1/16 inch balsa. The tail surfaces were covered with two layers of light fiber glass cloth (2 and 0.5 oz./sq yd) and epoxy resin. The horizontal stabilizer's long span required that a ¼" basswood spar be installed for added stiffness. As shown in the drawing package, the fuselage internal structure consists of two ¼" by ¾" spruce beams connected with cross braces. Formers where attached to these beams to support the outer shell. This outer shell was constructed by laminating several layers of fiberglass over a plug carved from the same insulation foam used in the wing and tail cores. The foam was then removed by dissolving with acetone. The shell was cut along the circumference near the middle of the wing and each end slides off of the frame for easy access to the internal components and payload.

Figure 6.1 is a proposed and actual timeline for construction of the competition aircraft. At the time of the writing of this report, construction is proceeding on schedule. If everything goes as planned, the first test flight of the final aircraft, RPR-1, will be on March 22, 1997.

6.7 Cost Reduction Methods

The choice of such a simple configuration was perhaps the most important cost reduction method. This was manifested in the manufacturing processes and material selection. First, all of the construction materials used are standard equipment in the R/C model community and all of these materials are commercially available from many different suppliers and can be obtained in a matter of a few days to one week. A lot of the building materials can be obtained directly from the local hardware store or building center. For example, the extruded polystyrene foam core was cut from a 4 x 8 foot sheet of home insulation purchased in town on the same day. The manufacturing processes require no special tools or machinery. In some cases, such as the landing gear or tailboom, the costs were reduced by buying completed products directly off the shelf of suppliers. While these parts may have a large initial cost, they come complete. They could have been custom made for less cost in terms of materials, but, they would have ultimately been more expensive in terms of material cost, plus labor, and quality of finished product.

Table 6.1 Figures of Merit Used in Manufacturing Plan Formulation

Figure Of Merit	Ranking		
	5	3	1
Structural Adequacy	high	average	low
Weight	light	average	heavy
Skill Required	easy to construct	average difficulty	difficult to construct
Materials Availability	easily acquired	average availability	difficult to acquire
Time Required	short	average	long
Cost	low	average	high

Table 6.2 Final Ranking of Figures of Merit for Wing

	Ranking		
Figure Of Merit	wood built up	sheeted foam	combination
Structural Adequacy	1	4	4
Weight	5	1	4
Skill Required	3	4	4
Materials Availability	3	3	3
Time Required	3	5	4
Cost	4	3	3
Total	19	20	22

Table 6.3 Final Ranking of Figures of Merit for Fuselage

Fuselage Structure	Ranking	
Figure Of Merit	structural shell	internal structure
Structural Adequacy	3	5
Weight	4	3
Skill Required	3	4
Materials Availability	3	3
Time Required	4	4
Cost	4	4
Total	21	23

Table 6.3 Continued

Fuselage Shell	Ranking	
Figure Of Merit	plug construction	mold construction
Structural Adequacy	3	3
Weight	3	3
Skill Required	5	3
Materials Availability	5	4
Time Required	5	3
Cost	3	3
Total	24	19

Table 6.4 Final Ranking of Figures of Merit for Tail Surfaces

	Ranking		
Figure Of Merit	wood built up	foam-wood	foam-glass
Structural Adequacy	2	5	4
Weight	5	2	3
Skill Required	2	4	5
Materials Availability	3	3	3
Time Required	1	1	4
Cost	4	3	3
Total	17	21	22

Table 6.5 Final Ranking of Figures of Merit for Tailboom

	Ranking		
Figure Of Merit	composite structure	custom built tube	pre-made tube
Structural Adequacy	3	3	5
Weight	4	3	2
Skill Required	1	3	5
Materials Availability	3	3	3
Time Required	1	3	5
Cost	3	3	1
Total	15	18	21

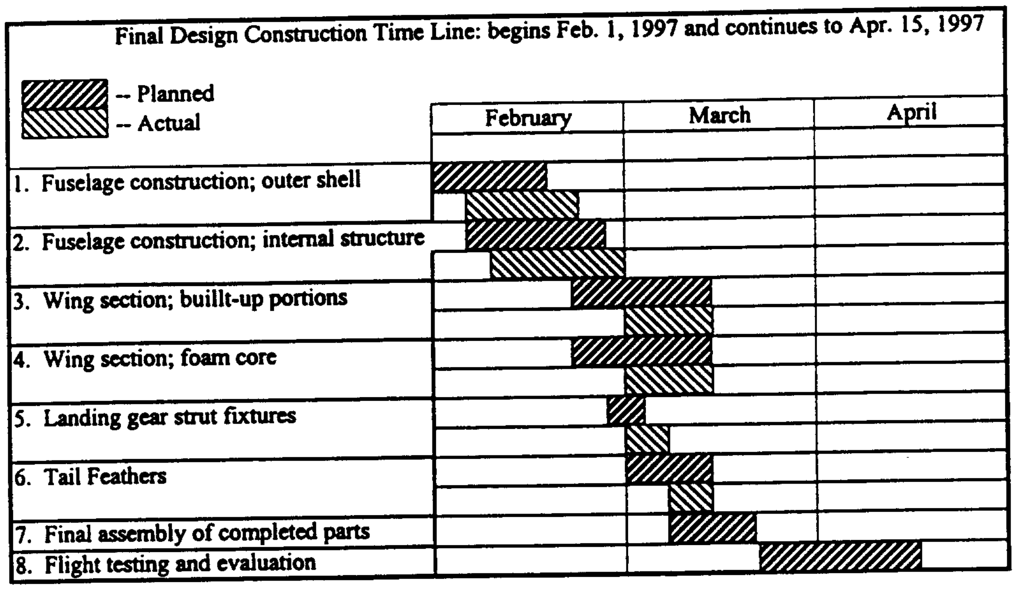


Figure 6.1 Project timeline for construction of RPR-1.

REFERENCES

1. Roth, M.R., "Static and Dynamic Performance Analysis of Low-Reynolds Number Propellers," University of Notre Dame internal report, 1994.

2. Nickel, K. and Wohlfarhrt, M., *Tailless Aircraft in Theory and Practice,* translated by E.M. Brown, 1 st ed., AIAA Education Series, Washington, D.C., 1994.

3. Boucher, R.J., *Electric Motor Handbook*, AstroFlight Inc., 1994

4. Raymer, D.P., *Aircraft Design: A Conceptual Approach*, 2nd ed., AIAA Education Series, Washington, D.C., 1994.

5. Kern, R.A., Private Communication, Sept. 1996.

6. Selig, M.S., Donovon, J.F., and Fraser, D.B., *Airfoils at Low Speeds*, SoarTech 8, SoarTech Publications, Virginia Beach, VA, 1989.

7 Lyon, C.A., Broeren, A.P., Gopalarathnam, A., Giguere, P., and Selig, M.S., *Summary of Low-Speed Airfoil Data—Vol. 3*, SoarTech Publications, Virginia Beach, VA, to be published, June 1997.

8. Garrison, P., LOFTSMAN version 3.3 users manual, Aerologic, Los Angeles, CA, 1996

9. Cawley, B., "Brush up on Brushless Motors," *Model Airplance News*, Aug. 1996, pp. 90-93.

10. Roskam, J., *Airplane Flight Dynamics and Automatic Flight Controls*, Roskam Aviation and Engineering Corporation, Ottawa, KS, 66067, 1979.

11. Selig, M.S., Lyon,C.A., Gigure,P.,Ninham,C., and Guglielmo,J.J., *Summary of Low-Speed Airfoil Data—Vol. 2*, SoarTech Publications, Virginia Beach, VA, 1996.

ADDENDUM

AIAA branch- Syracuse University

ADDENDUM -Aircraft Design

Since the original proposal, there have not been any design changes to the aircraft. Using a Clark-Y airfoil, the wing span is 84 inches. The fuselage is 50 inches in length. A Cobalt 640 G engine and Astroflight speed controller will power the aircraft.

The battery pack contains twenty 1700 SCRC cells. Weighing just under 2.5 lbs, the cells are assembled in two packs of ten cells each. Located directly behind the speed control, the two packs will lie side-by-side. This battery lay-out was selected since batteries heat up faster when stacked on top of each other. The batteries, the speed controller, and servos will be attached to a fiberglass panel inside the fuselage. Fiberglass was chosen because it is lighter than ply wood and stronger than balsa. This panel is located three inches from the bottom of the fuselage. The steel payload is located below the panel.

The payload consists of two 3.75 LB blocks of steel. Each block will sit vertically on either side of the fuselage. Placed inside, these blocks are centered around the CG of the aircraft. With this placement, the aircraft is expected to remain stable during flight. To accomodate the 7.5 lbs, that section of the fuselage is being reinforced with 1/4 inch plywood. The payload will be inserted and removed through a "doorway" section built into the fuselage.

One change was made in the manufacturing process. A wing jig was used for building the wing. The jig consists of two metal rods which mount to blocks of wood at either end. Two holes are drilled in each rib, and the ribs attached to the rods. The rest of the wing is then attached to the ribs and glued together. The purpose of using the jig was to aide in properly aligning all of the parts andto hold them in position while the glue dried.

To help those team members without experience in building Radio Control aircraft, a team member donated an instruction manual from another aircraft. By following the basic steps in the manual, our experienced builder was not required to always be present during the manufacturing process. This allowed for the plane to be built more quickly and accurately. The aircraft will be covered with Top Flight Monokote.

The majority of the materials and parts were purchased through Tower Hobbies and Superior Balsa and Hobby Supply. The aircraft will be flown using a Futaba 4 Channel radio donated by a team member. A private hobby store, Hobby City (located in Burtonsville, MD) is supplying the two battery packs at cost. Our electrical engineering department will build the battery charger. Miscellaneous parts were purchased at the local hobby store, Walt's Hobby Shop. Presented in Table A is the original budget proposal for the aircraft. Table B provides the actual expenses. Included in Appendix A are some additional performance calculations.

ADDENDUM -Lessons Learned

The biggest mistake made, was to design and build the aircraft in one semester. The first half of the year should be dedicated to the design of the aircraft, and the second semester for building and testing. To ensure that this happens next year, strict deadlines have to be set in September. They must also be enforced.

In addition, a more technical aspect of the design process should be followed. We went about the process by listening to suggestions from experienced builders and using an article from "RC Modeler" magazine. This article provided "basic" dimensions and proportions which should be used when designing the various sections of an aircraft. What should have been done was to first determine a "mission" profile for our aircraft. The profile consists of determining what flight characteristics the aircraft must possess. For instance, how well it should glide, the amount of lift needed, and the type of maneuvers it must perform. Five or six airfoils which fit these characteristics should then be selected for analysis. The analysis includes selecting a desired velocity, wing loading, and estimating the aircraft's weight. Using these constraints, the lift coefficients, drag coefficients, and moment coefficients are then calculated for each airfoil. The airfoil which best fits the desired profile is then selected for the design. From these results, the aircraft's aspect ratio and required wing span can be determined and the rest of the plane designed from there.

Table A: AIAA Design Build Fly Budget Proposal

- Four Channel radio and 3 servos	$150.00 *
-Engine Speed Control	$160.00 *
-Geared Engine	$185.00 *
-4 Battery packs	$80.00
-Battery Charger	$160.00 *
-Heat Gun and Iron	$40.00 *
-Monokote Covering	$10.00 per roll (atleats 2 rolls)
-Glue	$30.00
-Wood	$50.00
-Miscellaneous	$70.00

(miscellaneous includes such items as control rods, hinges, push rods, screws, etc.)

TOTAL	**$ 945.00**

Everything marked with an asterick is a one time expense, these items will be able to be used again, ensuring proper care. The batteries may also be reuseable for atleast next year also. This expenses total up to $ 775.00 of the $ 945.00.

Table B: Actual AIAA Design Build Fly Expenses

-Astroflight Engine	$185.00
-Speed Controller	$89.95
-Balsa, Ply Wood, and Glue	$140.00
-Two Battery Packs	$300.00
-Landing Gear	$40.00
-Miscellanous	$165.00
TOTAL	**$920.00**

Miscellanous items include propellars, control rods, hinges, push rods, monokote, etc.

Appendix

performance calculations

Weight with payload is 15 lbs

$C_L = .432 \quad C_{D_0} = .0114 \quad S = 7 ft^2 = 1008 in^2$

$V_{max} = 65 ft/sec \quad V_{stall} = 34 ft/sec \quad V_{TO} = 40.8 ft/sec$

$= 44 mph \quad = 23 mph \quad$ Chord = 12 in

$K = 780 \quad AR = \frac{Span^2}{wingarea} = \frac{84^2}{1008} = 7$

$Re = speed (mph) \times chord (in) \times K$

$= 44 \times 12 \times 780 = 411840$

$C_{L\alpha=0} = 1.5 \quad \alpha_0 = 18°$

$\alpha = \alpha_0 + \frac{(18.24 \times C_L) \times (1+T)}{AR}$

T = planform adjustment factor = .16

$= 18 + \frac{(18 \times 1.5) \times 1.16}{7} = 22.53$ degrees

$W/S = 2.85$

$C_m = -.0796$

$\sigma = 1$ density ratio of air

pitching moment (in-oz) $= \frac{C_m \times \sigma \times V^2 \times S \times C}{3519}$

$= \frac{-.0796 \times 1 \times 44^2 \times 1008 \times 12}{3519} = 529.7$ in-oz

$\frac{C_L}{\alpha} = \frac{1.5}{22.5} = .067$ per degree α

C_L needed during flight $V = 30 mph$

$C_L = \frac{lift \times 3519}{\sigma \times V^2 (mph) \times S (in^2)}$

lift = weight

w/ payload $L = 15 Lbs = 240 oz$

$C_{L needed} = \frac{240 \times 3519}{30^2 \times 1008} = .930$

$\frac{.930}{.067} = 13.88$ degrees

For Clark Y $\alpha_{L=0} = -5^{\circ}$

$13.88 - 5 = 8.88^{\circ}$ wing Set at in Fuse

w/out payload $L = 120$ oz

$C_L = \frac{120 \times 3519}{1 \times 30^2 \times 1008} = .465$ $\frac{.465}{.067} = 6.94$

$6.94 - 5 = 1.94^{\circ}$ $\sim 2^{\circ}$ wing Set at 2° from ~~center~~ line of Fuse

Stall angle is 18° or 23° from zero lift

$a = 23 + \frac{(18.24 \times 1.5 \times 1.16)}{7} = 27.53^{\circ}$ from zero lift

or 22.53° AoA at altitude

ground effect height of wing from ground $= 6''$ Fuse ht $+ 14''$ prop $= 20''$

$\frac{20 \text{ in}}{84 \text{ in wing span}} = .24$ Stall is reduced to .93 of value at altitude at .24 of wing span or at $.24 \times 84 = 20$ inches

α gnd effect $= .93 \times 22.53 = 20.95^{\circ}$

Load Factors $N = 1 + \frac{(1.466 \times mph)^2}{R \times G}$ $R \equiv$ turn radius

$G \equiv 32.2\,ft/s^2$

for 50ft radius at 30mph

$$N = 1 + \frac{(1.466 \times 30)^2}{50 \times 32.2} = 2.20$$

w/out payload $2.20 \times 120 = 264$ oz → wing lifts during Flight (turn)

$2.20 \times .465 = 1.023$ → C_L during turn

w/ payload $2.20 \times 240 = 528$ oz

$.930 \times 2.2 = 2.04$

must increase turn radius w/ payload
accelerate during turn and use up elevator

REPORT DOCUMENTATION PAGE

Form Approved
OMB No. 0704-0188

1. AGENCY USE ONLY *(Leaveblank)*	2. REPORT DATE	3. REPORT TYPE AND DATES COVERED
	1 August 1997	Final Report, 1 April - 31 December 1997

4. TITLE AND SUBTITLE
D/B/F 97: FINAL REPORT OF THE AIAA STUDENT AIRCRAFT DESIGN, BUILD & FLY COMPETITION

5. FUNDING NUMBERS
G: N00014-97-1-0421
PR: 97PR04795-00

6. AUTHORS
By Gregory Page, Chris Bovias, Michael Selig and the student participants of D/B/F 97.
Compiled by Wil Vargas, AIAA

7. PERFORMING ORGANIZATION NAME(S) AND ADDRESS(ES)

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

8. PERFORMING ORGANIZATION REPORT NUMBER

97DBF7501

9. SPONSORING/MONITORING AGENCY NAME(S) AND ADDRESS(ES)

Office of Naval Research
800 North Quincy St (ONR 351)
Arlington, VA 22217-5660

10. SPONSORING/MONITORING AGENCY REPORT NUMBER

11. SUPPLEMENTARY NOTES

12a DISTRIBUTION/AVAILABILITY STATEMENT

APPROVED FOR PUBLIC RELEASE

12b. DISTRIBUTION CODE

13. ABSTRACT *(Maximum 200 words)*

This report is made up of the combined reports of eight separate teams of students who entered the 1997 Design, Build & Fly Competition. The objectives of the Design, Build & Fly Competition were to have students teams design, build and fly unmanned remote control electric aircraft designed for maximum range on a limited battery. A "fly-off" took place on a private airstrip at Ragged Island, Md., in April 1997. Winners of the contest: 1st place, University of Illinois at Urbana-Champaign; 2nd, Virginia Polytechnic Institute and University; 3rd, Texas A&M University. The Design, Build & Fly Competition was supported by Cessna, the Office of Naval Research and the AIAA Foundation.

14. SUBJECT TERMS

Unmanned / Remote / Control / RC / Student / Design / Build / Fly / AIAA

15. NUMBER OF PAGES

468

16. PRICE CODE

17. SECURITY CLASSIFICATION OF REPORT	18. SECURITY CLASSIFICATION OF THIS PAGE	19. SECURITY CLASSIFICATION OF ABSTRACT	20. LIMITATION OF ABSTRACT
			SAR

INSTRUCTIONS FOR COMPLETING SF 298

The Report Documentation (RDP) is used in announcing and cataloging reports. It is important that this information be consistent with the rest of the report, particularly the cover and title page. Instructions for filling each block of the form follow. It is important to ***stay within the lines to meet optical scanning requirements.***

Block 1. Agency Use Only *(Leave blank)*.

Block 2. Report Date. Full publication date including day, month, and year, if available (e.g., 1 Jan 88). Must cite at least the year.

Block 3. Type of Report and Dates Covered. State whether report is interim, final, etc. If applicable, enter inclusive report dates (e.g., 10 Jul 87 - 30 Jun 88).

Block 4. Title and Subtitle. A title is taken from the part of the report that provides the most meaningful and complete information. When a report is prepared in more than one volume, repeat the primary title, add volume number, and include subtitle for the specific volume. On classified documents enter the title classification in parentheses.

Block 5. Funding Numbers. To include contract and grant numbers; may include program element number(s), project number(s), task number(s), and work unit number(s). Use the following labels:

C -	Contract	**PR** -	Project
G -	Grant	**TA** -	Task
PE -	Program Element	**WU** -	Work Unit Accession No.

Block 6. Author(s). Name(s) of person(s) responsible for writing the report, performing the research, or credited with the content of the report. If editor or compiler, this should follow the name(s).

Block 7. Performing Organization Name(s) and Address(es). Self-explanatory.

Block 8. Performing Organization Report Number. Enter the unique alphanumeric report number(s) assigned by the organization performing the report.

Block 9. Sponsoring/Monitoring Agency Name(s) and Address(es). Self-explanatory.

Block 10. Sponsoring/Monitoring Agency Report Number. *(If known)*

Block 11. Supplementary Notes. Enter information not included elsewhere such as: Prepared in cooperation with . . . ; Trans. of . . . ; To be published in . . . . When a report is revised, include a statement whether the new report supersedes or supplements the older report.

Block 12a. Distribution/Availability Statement. Denotes public availability or limitations. Cite any availability to the public. Enter additional limitations or special markings in all capitals (e.g., NOFORN, REL, ITAR).

DOD - See DoDD 5230, "Distribution Statements on Technical Documents"
DOE – See authorities.
NASA – See Handbook NHB 2200.2.
NTIS – Leave blank.

Block 12b. Distribution Code.

DOD - Leave blank.
DOE – Enter DOE distribution categories from the Standard Distribution for Unclassified Scientific and Technical Reports.
NASA – Leave blank.
NTIS – Leave blank.

Block 13. Abstract. Include a brief *(Maximum 200 words)* factual summary of the most significant information contained in the report.

Block 14. Subject Terms. Keywords or phrases identifying major subjects in the report.

Block 15. Number of Pages. Enter the total number of pages.

Block 16. Price Code. Enter appropriate price code *(NTIS only)*.

Blocks 17. - 19. Security Classifications. Self-explanatory. Enter U.S. Security Classification in accordance with U.S. Security Regulations (i.e., UNCLASSIFIED). If form contains classified information, stamp classification on the top and bottom of the page.

Block 20. Limitation of Abstract. This block must be completed to assign a limitation to the abstract. Enter either UL (unlimited) or SAR (same as report). An entry in this block is necessary if the abstract is to be limited. If blank, the abstract is assumed to be unlimited.