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FINAL DESIGN PROPOSAL

THE FX/90

A Proposal in Response to a Low Reynolds Number Station Keeping Mission

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Department of Aerospace and Mechanical Engineering
University of Notre Dame
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The FX/90 - A DESIGN PROPOSAL

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Executive Summary

1 - Introduction

The FX/90 is a remotely piloted vehicle designed to fly at Reynolds numbers below 2×10^5. Several applications exist for this type of flight, such as low altitude flight of very small aircraft. The design presented here allows investigation into the unique problems involved in low Reynolds number flight, which will, in turn, further understanding of this flight regime. A three view drawing and specifications summary follow this executive summary.

2 - Flight Plan

The aircraft will operate in a steady flight environment, free from significant atmospheric turbulence and weather effects. The aircraft will take off within 75 ft., and will climb to an altitude of 20 ft. within an additional 90 ft. of ground distance. The aircraft will then commence its flight plan, which consists of three figure 8 loops around two pylons spaced 150 ft. apart. Upon completion of the three laps, the aircraft will travel around the flight envelope and return to the pit area for landing. It can do so under powered flight, or it can travel an additional 60 ft. and then glide the remaining distance.

3 - Structure

The F-90 has a 39 in. fuselage which is constructed of balsa and plywood. The fuselage consists of two sections. The forward section is a 3.5 in. x 3.5 in. x 17 in rectangular structure in which the propulsion and flight control systems are located. The rear section is a 22 in. boom with a truss structure and a square cross-section which tapers to a point. The boom provides a moment arm for the tail surfaces. The length of 22 in. is a compromise between the advantages of a longer moment arm and the disadvantages of the associated increase in weight. The truss construction was chosen for its high strength and torsional stiffness with a minimal weight.

4 - Landing Gear

The landing gear for the aircraft is a detachable carriage on which the aircraft rests. The aircraft accelerates for take-off while on the carriage. At take-off, the aircraft lifts off of the carriage, and completes its flight plan without landing gear. Landing is accomplished by setting down on the smooth lower surface of the fuselage. The propulsion system uses a foldable propeller to prevent damage during landing. Care must be taken to prevent the carriage from impacting with anything after take-off.

5 - Propulsion

The aerodynamic planform is a rectangular wing (no taper or sweep) with a chord of 9 in., a wingspan of 72 in., and is constructed entirely out of styrofoam. Styrofoam was chosen for its low weight and relative ease of construction. "Aircraft quality" styrofoam was chosen for its high strength and hardness and its smooth surface, which eliminates the need for a coating material. Special care must be taken when handling the wings, particularly the thin trailing edges.

The propulsion system is a puller configuration mounted on the front of the fuselage. It consists of an Astro 05 engine and a 10-6 two bladed propeller. The

Astro 05 engine was chosen for its light weight and adequate available power. The 10-6 propeller was chosen for its efficiency in conjunction with the 05 engine and for its moderate diameter. The maximum velocity and rate of climb, as well as the maximum range and endurance, all exceed the design requirements due to an excess of available power and battery energy storage.

6 - Stability and Control

Control of the aircraft is accomplished through the use of two movable control surfaces: elevators for pitch control, and a rudder for yaw control. In addition, a large dihedral angle was used to couple the yaw and roll axis. This allows for roll maneuvers to be accomplished through the use of the rudder, as well as providing adequate spiral stability. Ample rudder was provided in order to allow a high maneuverability as required by the flight plan.

7 - Areas of Concern

There are several areas of concern. At take-off, the landing gear will detach while travelling at approximately 24 ft/s, which is a safety concern. The aircraft flies at a high angle of attack, giving the aircraft a low tolerance to gusts, and brings points along the wing close to stall during maneuvers. The impact of a landing without landing gear, as well as its effects on components of the aircraft, is relatively uncertain. Finally, the performance of the foldable propeller is not well documented, and consequentially are not thoroughly understood.

8 - Conclusion

Overall, we are quite satisfied with our design. The aircraft is soundly constructed, highly maneuverable, and adequately powered; and we are confident in its ability to fulfill the mission objectives. Furthermore, the investigation into alternative technologies, most notably the styrofoam wing and the detachable landing gear, holds promise to improve the performance of the aircraft. Finally, we feel that all of the areas of concern are relatively minor. Therefore, we feel the FX/90 design deserves further development, and eventually production.

Design Group F

Specifications Summary

Specification	Range	Symbol
Performance		
Endurance	7.99 min @ R_{max}	E
E_{max}	8.48 min @ cruise	
Range	12,210 ft @ cruise	R
R_{max}	14,389 ft @ min drag	
Take-off Distance	30 ft	$D_{t/o}$
Landing Distance	10 ft	D_l
Landing Gear	carriage	
Weight Fully Loaded	2.75 lb	W
Maximum Load Factor	1.08	n_{max}
Reynolds Number	114,480	Re
Speed	24 ft/sec @ take-off	$V_{t/o}$
	24 ft/sec @ cruise	V_{cruise}
	20.8 ft/sec @ stall	V_{stall}
Rate of Climb	360 ft/min	R/C
Propulsion		
Engine Type	Astro 05 geared	
Engine Power	115 W	P_{avail}
Static Thrust	0.67 lb	T
Propeller Type	10-6	
Propeller Efficiency	0.5 - 0.55	$\varepsilon\,\bullet_{prop}$
Dimensions		
Fuselage		
Height	4 in	h
Width	3.5 in	w
Length	17 in	l
Wing		
Surface Area	4.38 sq ft	S
Mean Chord	9 in	c
Span	5.84 ft	b
Aspect Ratio	7.79	AR
Wing Loading	.616 lb/in^2	
Angle of Incidence	7 degrees	i_{wing}
Dihedral	13 degrees	Γ
Taper Ratio	1.0	
Horizontal Tail		
Chord	5. in	c_h
Area	69.4 sq in	S_h
Aspect Ratio	2.78	AR_h

Incidence Angle	-0.37 degrees	i_t
Elevator		
Area	5.97 sq in	S_e
Max Deflection	10 degrees	δ
Vertical Tail		
Chord	6 in	c_v
Area	50.46 sq in	S_v
Aspect Ratio	1.4	AR_v
Rudder		
Area	29.78 sq in	S_r
Max Deflection	20 degrees	β

<u>Stability and Control</u>

Static margin	7%	
Horizontal volume ratio	.30	V_h
Vertical volume ratio	.028	V_v

<u>Aerodynamics</u>

Airfoil Section	FX-63-137B-PT	
Max. Lift Coefficient	1.23	C_{Lmax}
Drag Coefficient	0.16	C_{do}
Efficiency	0.79	e
Max Lift to Drag ratio	17.7	L/D_{max}

3 VIEW DRAWING OF FX/90

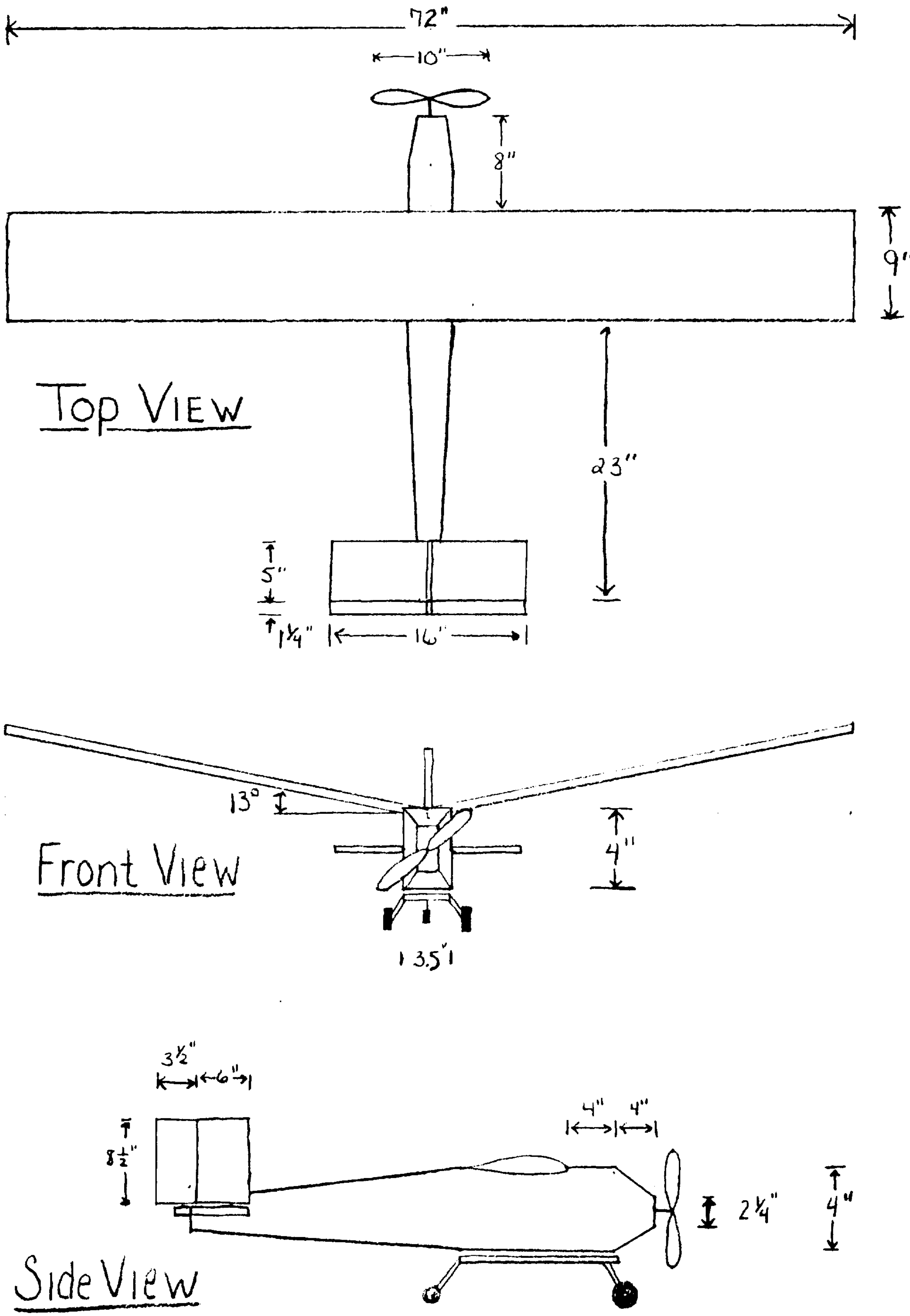


Table of Contents

1- Mission Study

Group F received the following request for proposals:

FLIGHT AT VERY LOW REYNOLDS NUMBERS - A STATION KEEPING MISSION

OPPORTUNITY

Most conventional flight vehicles are designed to operate in a flight regime such that the Reynolds numbers based on mean wing chord are in excess of 10^6 and some currently are approaching 10^8. Recently there has been interest expressed in vehicles which would operate at much lower Reynolds numbers, less that 10^5. Particular applications are low speed flight at very high altitudes, low altitude flight of very small aircraft and flight in other planet's atmospheres such as Mars. There are many unique problems associated with low speed flight which pose challenges to the aircraft designer and which must be addressed in order to understand how to exploit this low Reynolds number flight regime. Since many of the anticipated missions for this type of aircraft are unmanned, it necessary to couple developments in unmanned aircraft development with our knowledge of low Reynolds number aerodynamics in order to develop an aircraft which can fly as slow as possible at sea level condition. This study will help to better understand the problems associated with flight at these very low Reynolds numbers. Considering the potential applications, the aircraft must also be very robust in its control and be highly durable.

OBJECTIVES

1. Develop a proposal for an aircraft and associated flight control system which must be able to:

a. Maintain level controlled flight and fly a closed course at flight speeds corresponding to Reynolds numbers less than 2×10^5 and as close to 1×10^5 as possible. The greatest measure of merit is associated with achieving the lowest chord Reynolds number possible and maximizing the loiter time on a closed course.

b. Be maneuverable and controllable so that it can fly a closed pattern and remain within a limited volume of airspace.

c. Use a propulsion system which is non-airbreathing and does not emit any mass, (i.e. rocket, etc.).

d. Be able to be remotely controlled by a pilot with minimal flying experience or an autonomous onboard control system.

e. Carry an instrument package payload which weighs 2.0 oz and is 2"x2"x2" in size.

2. Take full advantage of the latest technologies associated with lightweight, low cost radio controlled aircraft and unconventional propulsion systems.

3. All possible considerations must be taken to avoid damage to surroundings or personal injury in case of system malfunction.

4. Develop a flying prototype for the system defined above. The prototype must be capable of demonstrating the flight worthiness of the basic vehicle and flight control system. The prototype will be required to fly a closed figure "8" course within a highly constrained envelope. A basic test program for the prototype must be developed and demonstrated with flight tests.

SYSTEM REQUIREMENTS AND CONSTRAINTS

The system design shall satisfy the following.

a. All basic operation will be line-of-sight with a fixed ground based pilot, although automatic control or other systems will be considered.

b. The aircraft must be able to take-off from the ground and land on the ground.

c. The aircraft must be able to maximize the loiter time within a restricted altitude range on a figure "8" course with a spacing of 150 ft. between the two pylons which define the course.

d. Ground handling and system operation must be able to be accomplished by two people.

e. The complete system must be disassembled for transportation and storage and fit within a storage container no larger than 2' x 2' x 4".

f. Safety considerations for systems operations are critical. A complete safety assessment for the system is required.

SPECIAL CONSIDERATIONS FOR THE TECHNOLOGY DEMONSTRATOR

a. The Technology Demonstrator will be a full sized prototype of the actual design.

b. The flight tests for the Technology Demonstrator will be conducted in the Loftus Center on a closed course similar to that described above. The Demonstrator will be required to complete 3 laps on the course. The altitude must not exceed 25 ft. an any point on the course.

c. Takeoff must be accomplished within the 150 ft. takeoff region shown in fig. 1-1.

d. Loiter time will be based on the time needed to complete the 3 complete laps in the air.

e. The design team must make provisions for measuring altitude and flight speed during the tests.

f. The propulsion system for the technology demonstrator must not contain any chemicals or any other substance which could prove harmful to the Loftus Center or the aircraft operators.

g. The radio control system and the instrumentation package must be removable and a complete system installation should be able to be accomplished in 30 min.

h. System control for the flight demonstrator will be a Futaba 6FG radio system with up to 4 S28 servos.

i. All FAA and FCC regulations for operation of remotely piloted vehicles must be complied with.

The request for proposals provided three areas which the design group was able to develop into more specific design requirements and objectives.

1.1 Reynolds Number

The first area is the range of Reynolds numbers. Group F choose an Re of 1×10^5 as their objective. This would allow us to maximize one of the two greatest measures of merit, to be maximized while also allowing the design to remain well within the constraint of $Re = 2 \times 10^5$. It was then decided that the minimum chord length would be 9 in. This was due to the difficulty of constructing very small wings and to the excessive wing length which results from very low chord lengths. From the knowledge of the Reynolds number and chord length, the design flight speed was then estimated to be between 22 ft/s and 24 ft/s, and the maximum weight was estimated to be approximately four lbs.

$$Re = 1 \times 10^5$$
$$V = 22 - 24 \text{ ft/s}$$
$$W = 4 \text{ lbs.}$$

1.2 Airspace Restrictions

The second area which provided specifications was the limited airspace. The group decided that the aircraft must be able to take-off and climb to cruise altitude prior to commencing the first turn. This led to several specific requirements. The aircraft was required to take-off within 75 ft. With an estimated take-off speed of 25 ft/s, this meant that the propulsion system would be required to provide an acceleration of approximately 4.2 ft/s^2. The aircraft was further required to climb to the cruise altitude of 20 ft. within a ground distance of 90 ft. This lead to a required climb angle of 12.5°, a required rate of climb of 300 ft/min, and the ability to withstand a load factor of 1.024.

While in flight, the aircraft must complete turns around the pylons while remaining within the airspace limits. To do this, group F required the aircraft to turn with a radius of 45 to 60 ft. This requires a bank angle of 18.5^{o} to 21.7^{o}, and a load factor of 1.054. Furthermore, the control system must be able to control the aircraft about all three axis, and to perform turning maneuvers in both directions.

Finally, upon completing the final lap, the aircraft would be at the end of the take-off area. The group decided that the aircraft should be able to glide around the perimeter of the airspace to the beginning of the take-off area for landing. This required the plane to have a minimum glide angle of 3.07^{o}, and to have the ability to descend 20 ft. in a ground distance of 377 ft. These requirements are summarized below. The flight plan is presented in figure 1.1.

Altitude = 20 ft.

<u>Take-off</u>

Distance = 75 ft

V_{t-o} = 25 ft/s

a = 4.17 ft/s^2

<u>Climb</u>

Ground Distance = 90 ft

R/C = 300 ft/min

Climb angle = 12.5^{o}

n = 1.024

<u>Turn</u>

Radius = 45 - 60 ft.

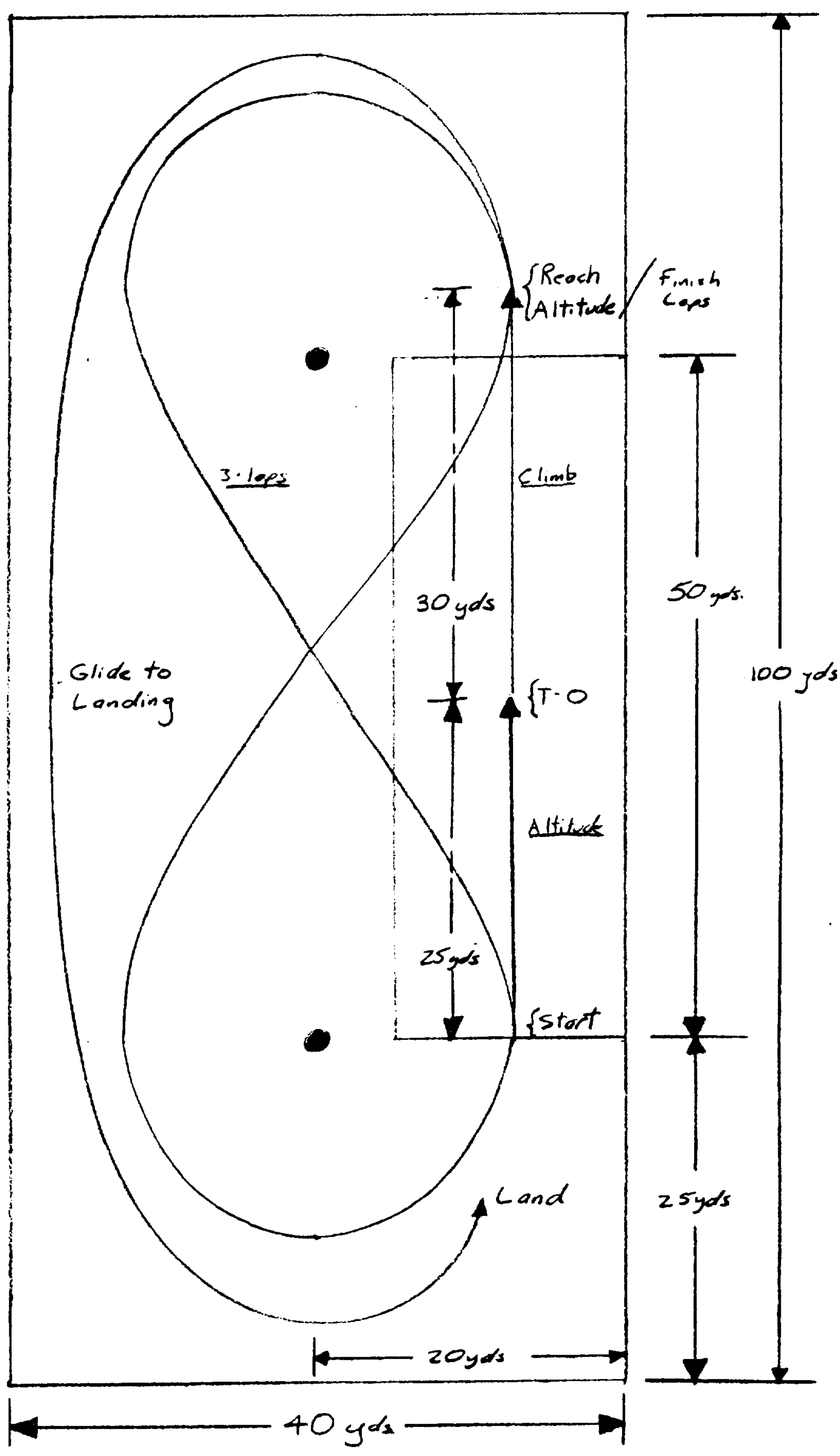
Bank Angle = 18.5 - 21.7^{o}

n = 1.054

<u>Landing</u>

Distance = 75 ft

Fig 1-1 Airspace and Flight Plan



Min. Glide Angle = 3.07^{o}

Descend 20 ft. in 377 ft. ground distance

1.3 Required Number of Laps

The third area is the requirement that the aircraft complete three laps. From knowledge of the course and the proposed flight plan, the required range was computed to be 1900 to 2280 ft based on three laps. Using this value with the design speed range, the required endurance was computed to be 1.38 to 1.67 minutes.

Range = 1900 - 2280 ft

Endurance = 1.38 -1.67 min.

1.4 Requirements Specified in request for proposals

The request for proposals also gave several specific requirements for the mission. The payload capacity was specified to be 2 oz in weight and have dimensions of 2 in x 2 in x 2 in. The aircraft is required to take-off and land from the ground. Therefore, some type of landing gear is required. In addition, the group chose to require a battery powered electric propulsion system in order to meet the conditions that the propulsion system be non-airbreathing and not emit any mass.

1.5 Ground Handling and Safety

Ground handling imposed additional requirements. In order to meet the transportation size constraints, the length of the fuselage was limited to less than 48 in., and the wings were required to be detachable from the fuselage and to separate in the middle. By separating in the middle, the wings will be able to fit into the storage container. To meet the 30 min. radio system installation requirement, the fuselage is required to provide access to the radio system, possibly through the use of an access panel. In order to withstand ground handling loads, the aircraft must have strength in all directions, not simply the directions in which flight loads are applied, and the aircraft must be able to withstand a 2 to 3 ft. drop.

1.6 Safety

To ensure safety, three requirements were developed. First, the aircraft must be constructed from collapsible materials in order to avoid damage to surroundings, or people, in the event of a crash. Second, the pilot must be able to stop the engine, so that the propeller can be prevented from spinning if impact is imminent. Finally, the aircraft must use a visible color scheme, which will allow the pilot and bystanders to avoid collisions.

2 - Concept Selection

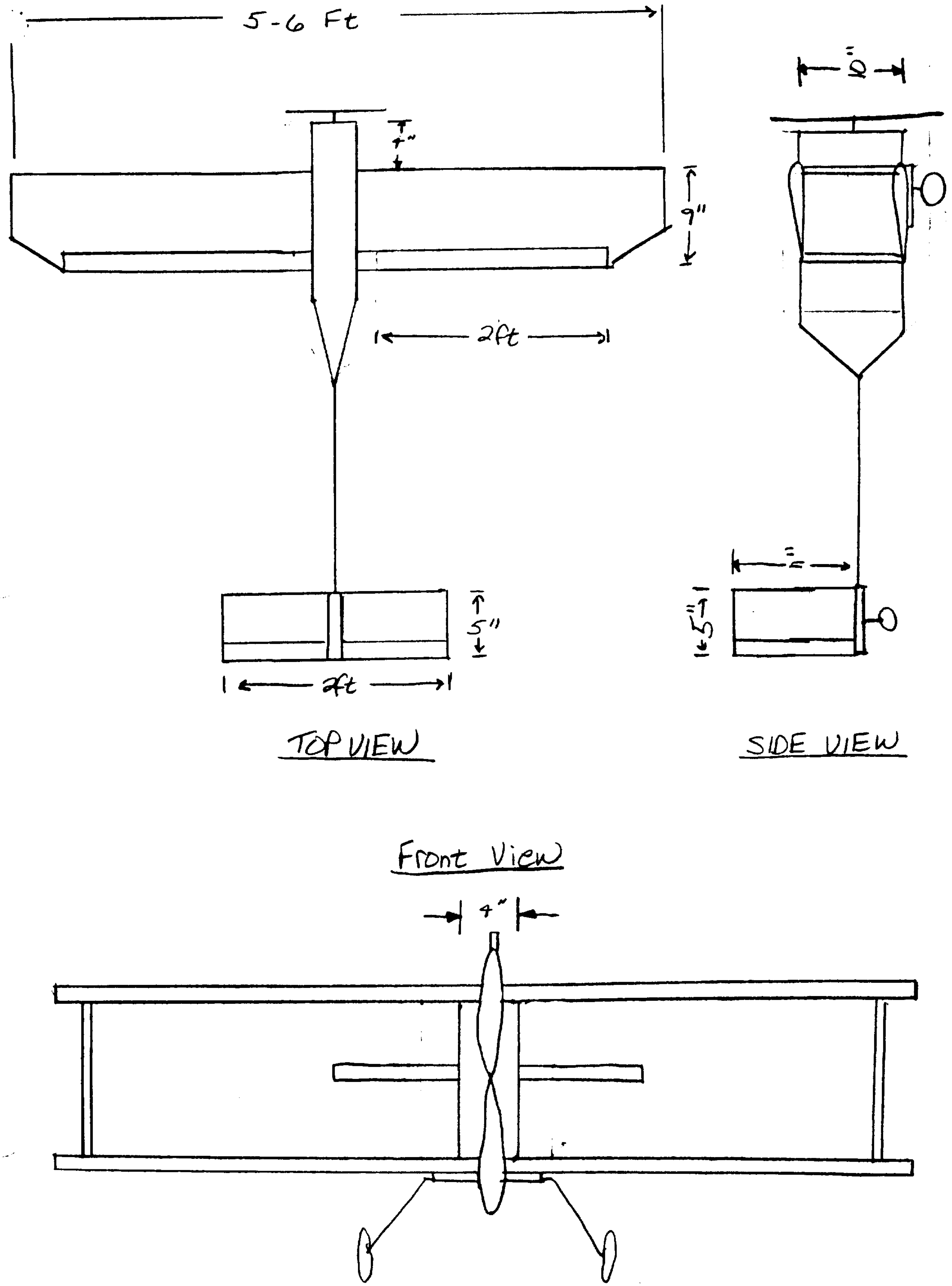
Once the design requirements and objectives were determined, Group F began the concept selection phase. Initially, each member submitted an initial concept. This allowed each member to individually consider different options for the aircraft, and to further consider the advantages and disadvantages of each. Once these were completed, two sub-groups of three people each were formed. Each group discussed the various options, and decided upon a sub-group concept. These concepts are shown in figs. 2-1 and 2-2, and are discussed briefly below.

2.1 Sub-group One Concept

Sub-group one developed a concept of a single engine biplane. The mean chord was approximately five to seven inches in order to provide a low Reynolds number. Flaps were included to provide additional lift in the low design velocity range. The wings were constructed of styrofoam, which it was thought would provide a low weight and simplify construction. Roll control was provided by ailerons which ran the width of either the upper or lower pair of wings. Pitch and yaw stability and control were provided by a single horizontal stabilizer and a single upward vertical stabilizer located at the rear of the aircraft. The stabilizers were to be constructed of a wood framework due to their small size.

The fuselage consisted of a rectangular forward section, which would contain the flight systems, and a solid boom which would provide an attachment for the tail section. The propulsion system was a single electric engine with a two-bladed propeller. The engine was a tractor type configuration, and was located in the fuselage. The landing gear would be detachable in order to decrease the inflight weight of the aircraft. Thus, the aircraft would utilize a belly landing.

FIG. 2-1: Sub-Group I Concept



2.2 Sub-group Two Concept

Sub-group two developed a concept which had several significant differences from sub-group one's concept. This concept was of a single engine monoplane. The aircraft would have high aspect ratio wings mounted above the fuselage, with a mean chord of approximately eight to ten inches. Note that this was not as low as the mean chord in sub-group one's concept, as this would have resulted in an overly long wing length. The wings had no flaps, and were constructed of a wood framework covered with heat-shrink plastic. Stability and control was provided by ailerons on the wings, an upward vertical tail and a single horizontal stabilizer positioned at the rear of the aircraft.

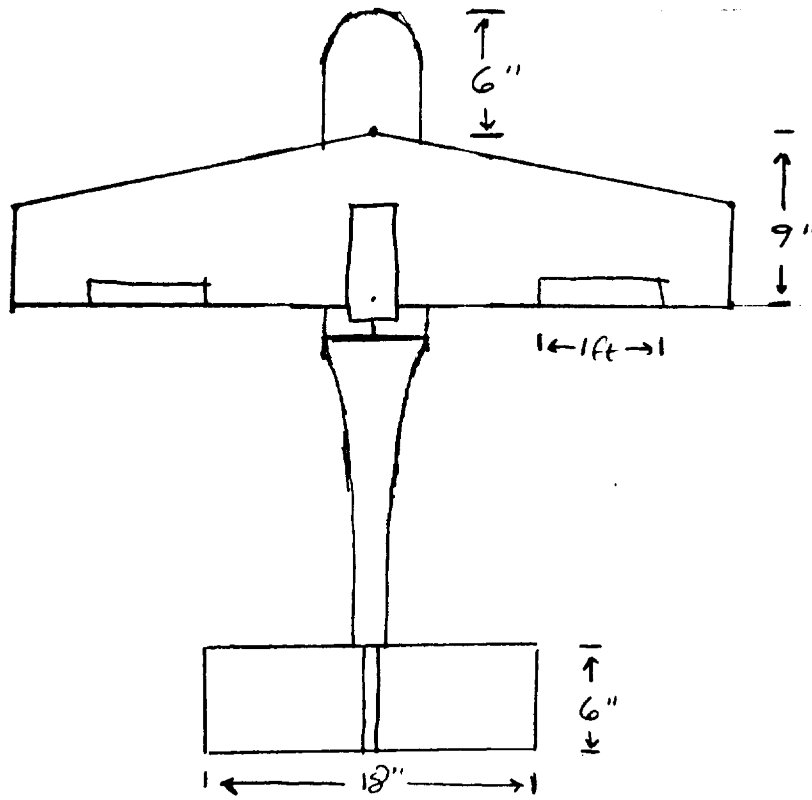
The propulsion system was a pusher configuration with a two bladed propeller, which was mounted above the wings and over the fuselage. This fuselage was conceptualized to be very aerodynamically shaped. It featured a cylindrical configuration with a rounded nose. The forward section contained all the flight systems with the exception of the engine. The rearward section tapered down to smaller diameter boom, which was positioned as low as possible in order to provide clearance for the propeller. The landing gear was fixed to the fuselage, with a single forward wheel and two rear wheels.

2.3 Group F Concept

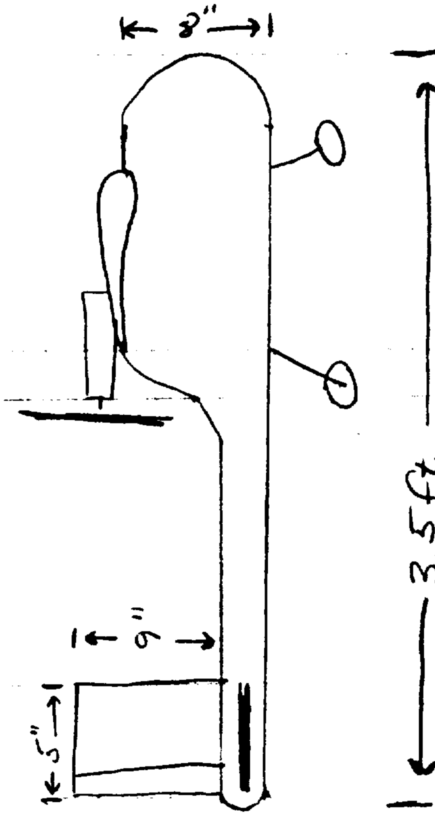
2.3.1 Comparison of Sub-Group Concepts

A comparison of the two sub-group concepts provided the following considerations:

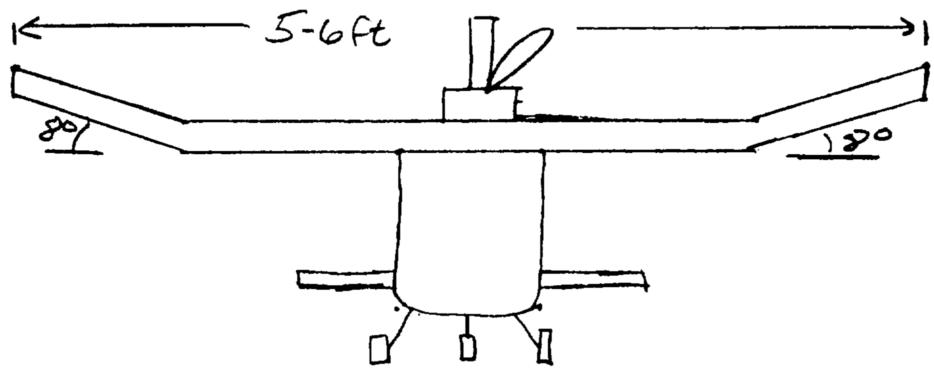
FIG 2-2: Sub-Group II Concept



TOP VIEW



SIDE VIEW



Front View

1) Monoplane or Biplane

2) Stryofoam or Wood Frame wings

3) Include or Do Not Include flaps

4) Tractor or Pusher propulsion system

5) Rectangular or Cylindrical fuselage

6) Solid or Tapered Frame boom

7) Permanent or Detachable landing gear

A discussion of each of these considerations follows.

2.3.2 Discussion of Concept Considerations

The group decided on a monoplane configuration. The advantage of the biplane would have been to limit the wingspan. The wingspan had the potential to become large due to the low design velocity, low chord length, and need for a large surface area. However, the group felt that it would be able to keep the wingspan to a reasonable size even with the monoplane. In addition, the group perceived a lack of data on biplane design, as well as on its effects on the other aircraft systems. For example, no information was available on what the lift distribution would be between the two wings, how the wings would affect each other's pressure distribution, and what sort of effect a biplane would have on stability and control. In contrast, there was a large database on the monoplane configuration.

Flaps were deemed unnecessary. Due to the low Reynolds number range, the take-off speed was very close to the cruise velocity. As a result, the wings would provide comparable lift in both cases, contrary to aircraft which cruise at higher speeds. As a result, during take-off the wings did not need the higher lift coefficient that the flaps provide. Furthermore, flaps add weight through the hinges and servo, they add drag, and they increase the difficulty of construction, all of which was considered undesirable. After discussion with experienced modelers, it was also decided not to use ailerons. Ailerons suffer from the same disadvantages as flaps, and it was felt that roll control could be provided with ample yaw control and ample dihedral.

The group also decided on a tractor propulsion system. The tractor system was much easier to integrate with the fuselage, and it did not place any constraints on the fuselage design. Furthermore, the engine could be located within the fuselage, which would eliminate drag on the propulsion system. Finally, no clear advantage to the pusher system was seen.

Next, the group decided on a rectangular fuselage. Although a circular fuselage would have less drag, these savings would be minimal due to the low design velocity. Furthermore, a circular fuselage would have complicated the fuselage design, the placement of the system components within the fuselage, and the construction. In fact, it was suspected that such a design was beyond our capabilities to construct given our lack of experience and lack of advanced tools.

The group also settled upon detachable landing gear. It was felt that this would provide a significant saving in weight, which was considered a critical design area. The detachable gear would also provide a large savings in drag, since it would not have been possible to have retracted the gear. The group did not feel that the problems associated with detaching the gear were prohibitive. Specifically, it felt that landing could be accomplished with a belly landing, with the rectangular fuselage preventing the aircraft from rolling over, and that the rudder would provide adequate ground control. It was also felt that designing such a gear system would be relatively straight forward.

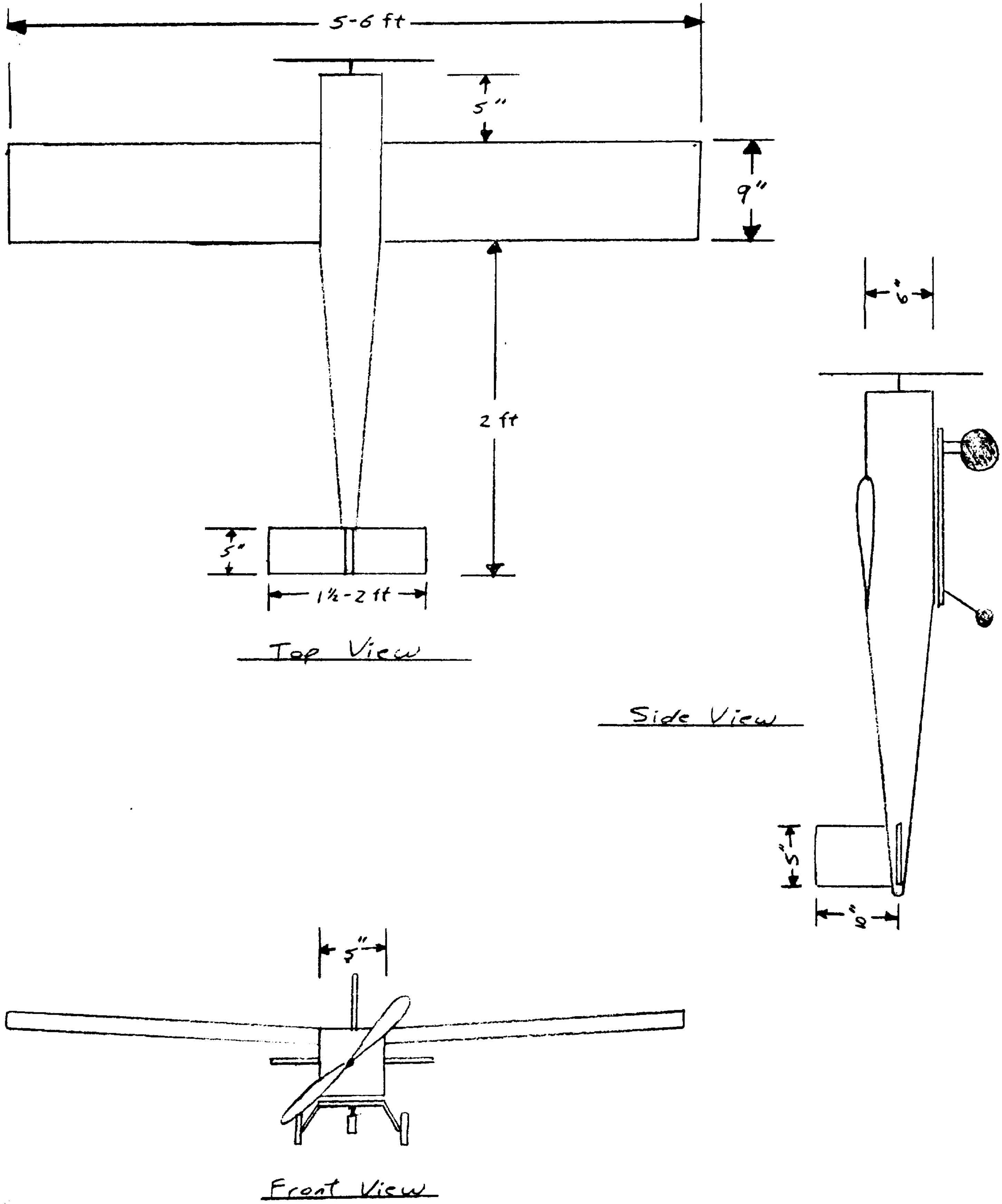
Finally, the group postponed a final decision on two items until they could be investigated further. These items were the styrofoam versus wood wings, and the frame versus solid boom. More information was needed on the wings concerning the weight of each type, the strength of styrofoam, and the construction methods. More information was needed on the boom concerning the relative drag, strength, weight, and construction methods. The styrofoam and boom subjects are discussed further in sections 7.3.4 and 7.3.2 respectively.

2.3.3 Final Concept Configuration

The final concept was then a monoplane with high aspect ratio wings, no flaps, and no ailerons. The concept further had a puller propulsion system, a rectangular fuselage, and detachable landing gear. Stability and control were provided by an upright single vertical stabilizer with a rudder, and a rear single horizontal stabilizer with an elevator. The concept is show in a 3 - view drawing in figure 2-3.

As a final note, the low Reynolds number mission had a significant impact on the concept. The low velocity and chord length reduced the lift per unit length provided by wings, which made weight a critical design area. Thus a biplane was considered in order to obtain more wing surface area and hence more lift, detachable landing gear was adopted, and weight was made a strong consideration in wing material selection. The low velocity also reduced the impact of drag, which lead to the selection of a rectangular fuselage. Note that drag was still important, and was a consideration in choosing no flaps and a puller propulsion system.

Fig 2-3: Group F Concept



3 - Aerodynamics

3.1 Airfoil Selection

The selection of an airfoil was very crucial in the design of the FX/90. Only those airfoils which perform well in low speed flight were considered, since the FX/90 is flying at a low Reynold's number. In order to select a specific airfoil, a list of criteria was made which took into account the C_{lmax}, the stall angle, the C_D, and the separation bubble characteristics. Those airfoils which met these conditions for low speed flight were the Epler 214, Wortmann FX-63-137, Aquilla and SD-7032A. These airfoils were then evaluated by looking at their lift curve slopes and drag polars at Reynold's numbers in the range specified by our mission (100,000 < Re <200,000).

The maximum sectional lift coefficient was an important factor in the selection of the airfoil. It was imperative to select an airfoil which could produce enough lift across the wing to enable the RPV to become airborne and remain in steady level flight during the turn. Therefore, only airfoils with a sectional C_{lmax} of 1.3 or better were considered.

A low drag was also a desired characteristic of the airfoil. Along with this, though, it was important to keep stall behavior in mind. In exchange for low drag, it is common for airfoils to have bad stall characteristics, which result in an abrupt stall. Since the RPV will be flying at slow speeds and near to stall, it was concluded that a trade-off would be made favoring gentle stall characteristics rather than an extremely low drag coefficient.

Along with this, a high stall angle was desired. A larger and angle of incidence is available with a high stall angle, while still allowing for an adequate margin of safety. Note that this margin of safety consists of fixing the cruise angle of attack several degrees below stall in order to allow fór changing flight conditions. This is a desirable feature, since a higher α_{wing} increases the amount of lift generated. Airfoils with an α_{stall} less than 10 were not considered in this design.

The drag polar provided a means for examining the separation bubbles of the airfoil. It was important to keep the flow attached to the wing during flight, thus only those airfoils with minimal separation bubble characteristics were examined.

Upon examination of the four airfoils, it was apparent that the E214 and the FX-63-137 had superior Cl, Cd and stall characteristics to the Aquilla and the SD-7032A. Thus, the final selection of the airfoil was between the E214 and the Wortmann FX-63-137. Table 3.1 compares the characteristics of each of these airfoils. The E214 overall showed obvious separation bubbles, an adequate C_l=1.33, and an average C_D. However, the high C_{lmax} =1.6, high stall angle =11.5 degrees and the gentle stall characteristics of the FX-63 outweighed the advantages of the E214. The one major disadvantage of the FX-63 is its thin trailing edge. A strong design of the wing, though, should not pose a problem in its fabrication. The Wortmann was also chosen for its known favorable separation bubble characteristics in the low Reynold's number regime.

TABLE 3.1 Airfoils and Their Respective Characteristics

Airfoil	C_{Lmax}	C_D	Stall Angle
E 214	1.3	0.0205	10
FX-63-137	1.6	0.0200	11.5

The Wortmann airfoil has a camber of 5.94% of the chord and a thickness of 13.59% of the chord. Its C_D vs. alpha slope is 0.0967 per degree, which was corrected for aspect ratio (see fig 3.1). From this value the distributed lift coefficient for the wing was calculated using the Finite Wing Theory[1] (see fig 3.2). From the estimated weight and speed and the desired approximate surface area, it was determined that a

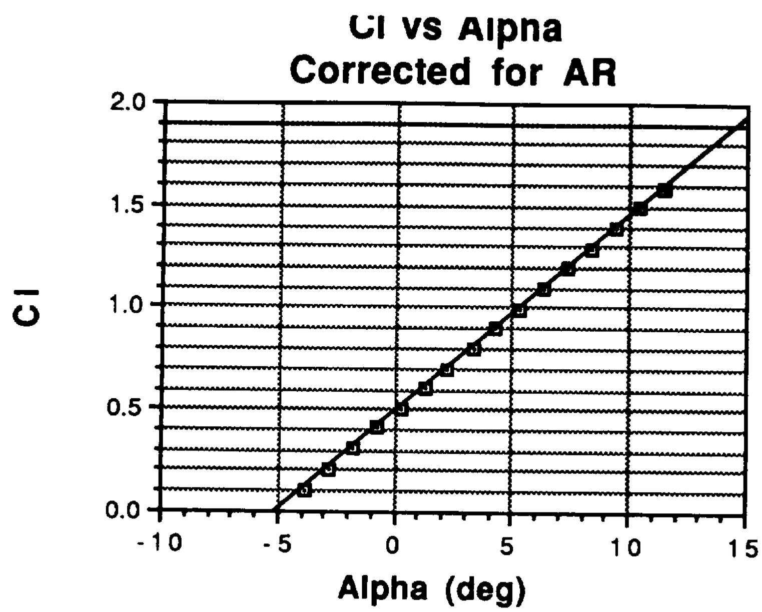

Cl vs Alpha
Corrected for AR
Cl
2.0
1.5
1.0
0.5
0.0
-10
-5
0
5
10
15
Alpha (deg)

Fig 3.1

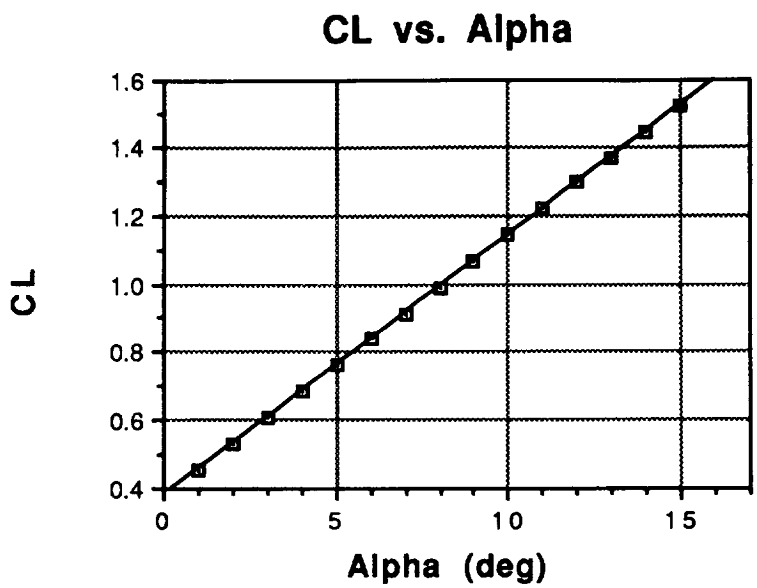


Fig 3.2

C_L = 0.9 was required for steady level flight. Looking at figure 3.2, a C_L = 0.9 corresponded to a 7.0 degree geometric angle of attack.

Taking this C_L = 0.9 and multiplying it by the maximum load factor of 1.08, which occurs during the turn, a C_{Lmax} = 0.973 is required. This produces 2.916 lbs of lift, and corresponds to 8 degrees on figure 3.2. During the turn, the effects of dihedral increase the angle of attack by a maximum of 1.8 degrees. This brings the angle of attack up to almost 10 degrees. It is at 10 degrees that the design maximum for alpha is set. The geometric angle of attack is set at 7.0 degrees, which allows a large enough range for an increase in angle of attack to avoid stall during the turn.

3.2 Wing Design

The wing characteristics were some of the first preliminary parameters defined for the FX/90. Initially, the surface area, span, and aspect ratio were estimated, based on an estimated aircraft weight and the necessary lift required to support this weight at both cruise and while turning. A high aspect ratio was desired to obtain the necessary lifting surface area while minimizing the mean chord, which in turn minimizes the chord Reynold's number. The preliminary design parameters are located in Table 3.2.

The airfoil selected for the FX/90 was the Wortmann FX-63-137. Once this was selected, the preliminary estimates of the wing characteristics were recalculated. An updated weight estimate provided by the weights team was used in these calculations, along with data provided by the Finite Wing Theory analysis. The actual C_{Lmax} which could be obtained was 14% less than that approximated in the preliminary design. Thus the wingspan, surface area and aspect ratio increased by 33%, 33% and 27% respectively, in order to obtain enough lift to become airborne. The final wing design parameters are compared to the preliminary parameters in Table 3.2.

TABLE 3.2 Preliminary and Final Wing Dimensions

Parameter	Preliminary	Final	% Difference
AR	5.86	7.79	33
S (ft^2)	3.29	4.38	33
b (ft)	4.39	5.58	27
C_{Lmax}	1.4	1.23	14

From the start of the design, several parameters remained constant. The size of the mean chord is based on the design requirement of a low mean chord Reynold's number. A wing mean chord of 9 inches was set in order to keep the Reynold's number small, as well as to keep the wing span at a reasonable length, due to weight restrictions which were always a factor.

Twist was not considered for the wing for two reasons. First, this addition brings about construction difficulties. Second, twist introduces a change of angle of attack along the wing. This factor was not desirable for the FX/90, since the wing design had a very narrow range of angles of attack, and was flying very close to stall. Sweep was also excluded from the design since the aircraft was flying at a negligible Mach number, this is not necessary. The addition of sweep decreases the effective velocity along the wing. Furthermore, since the FX/90 is flying at such a low velocity, to lower the velocity anymore would lower the Reynold's number and thus have adverse affects on the performance of the wing.

Taper was also not incorporated into the design of the wing. One main reason supporting this decision was the simplicity of designing a wing without taper. Also, figure 3.3 shows that as the taper ratio is decreased, there is an insignificant increase in C_L. For example, at 8 degrees alpha, there is an increase in C_L of approximately 0.03 as the taper ratio decreases from 1.0 to 0.6.

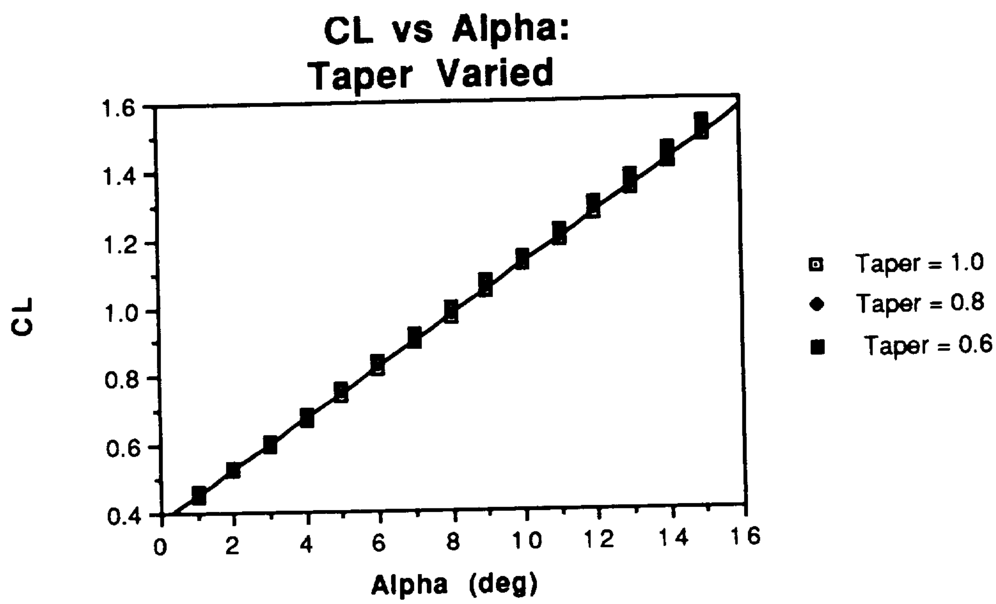

CL vs Alpha:
Taper Varied
1.6
1.4
1.2
1.0
0.8
0.6
0.4
CL
0
2
4
6
8
10
12
14
16
Alpha (deg)
Taper = 1.0
Taper = 0.8
Taper = 0.6

Fig 3.3

Due to the decrease in Reynold's number at the tip of the wing with a decrease in taper ratio, adverse effects begin to occur in the performance of the wing and airfoil. It was calculated that the minimum taper ratio which the FX/90 could effectively fly at is 0.8. At this taper ratio, though, a minimal increase in C_L is present.

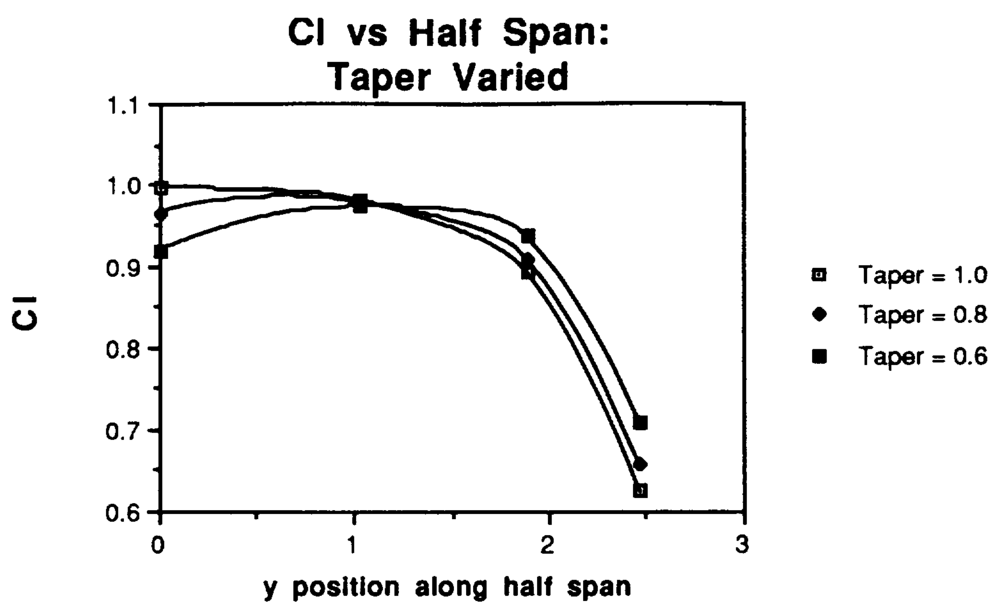
The sectional lift distribution over the span of the wing is also affected by taper. Figure 3.4 represents the sectional lift distribution over the half-span at tapers of 1.0, 0.8 and 0.6. For taper = 1.0, an even lift distribution is obtained over the entire wing. As the taper ratio decreases, as shown for taper = 0.6, a loss in lift over the center of the wing is produced. Thus, with a taper = 1.0, there is more lift generated, which is optimal for this design.

3.3 Drag Prediction

The drag forces on the aircraft are calculated from a drag estimation technique[2]. This drag estimation is broken down into two areas: parasite drag and induced drag.

The parasite drag value is the airplane drag value at zero lift. This value is estimated by a parasite coefficient, multiplied by an effective area, which is then divided by the reference area of the RPV. The final parasite drag obtained for the FX/90 was $C_{Do} = 0.016$, which was a realistic value compared to similar RPV's. This component of drag can be located on the drag polar at the bottom of the graph's parabola (see fig 3.5).

Before the induced drag can be calculated, the efficiency must be found. This factor was determined by a breakdown method based on three main efficiency contributions: the wing, the fuselage and other extending surfaces of the aircraft. The wing efficiency factor, e_{wing}, is based on contributions from both the airfoil section and the planform size. The body efficiency factor, e_{body}, based on the value of the fuselage cross-sectional area compared to the wing area, and a prescribed body efficiency parameter. The other efficiency factor is also a prescribed value, obtained



Cl vs Half Span:
Taper Varied
Cl
1.1
1.0
0.9
0.8
0.7
0.6
0
1
2
3
y position along half span
Taper = 1.0
Taper = 0.8
Taper = 0.6

Fig 3.4

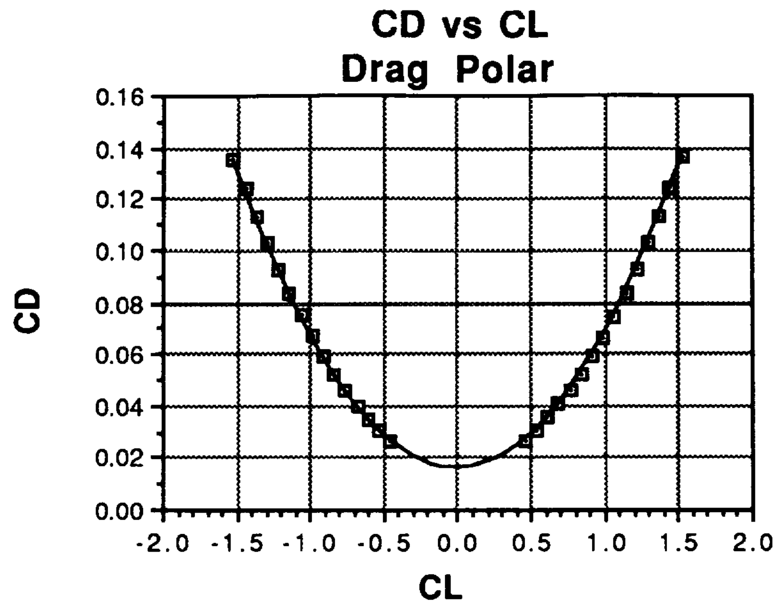

CD vs CL
Drag Polar
CD
0.00
0.02
0.04
0.06
0.08
0.10
0.12
0.14
0.16
CL
-2.0
-1.5
-1.0
-0.5
0.0
0.5
1.0
1.5
2.0

Fig 3.5

from the drag prediction method[2]. Based on these contributions, the FX/90's efficiency is: e=0.79.

The induced drag was then calculated, using the formula $C_{Di} = C_L^2 /(\pi e AR)$. Lastly, the coefficient of drag was be calculated by adding the parasite drag and induced drag together. It was from this relationship between C_D and C_L that the drag polar graph is obtained.

4 - Propulsion

4.1 The System

The FX/90 was designed in a tractor configuration. This was chosen to eliminate the structural difficulties incurred when using a "pusher" configuration. After power available and power required comparisons were examined for several different configurations of motor, propeller, and batteries, a geared Astro Flight 05 electric motor in combination with a 10-6 Zinger folding propeller, powered by seven, rechargeable, 500 mah Nickel-Cadmium batteries, and regulated by a speed controller was selected to drive the FX/90. A light, but powerful system, it met the desired performance goals.

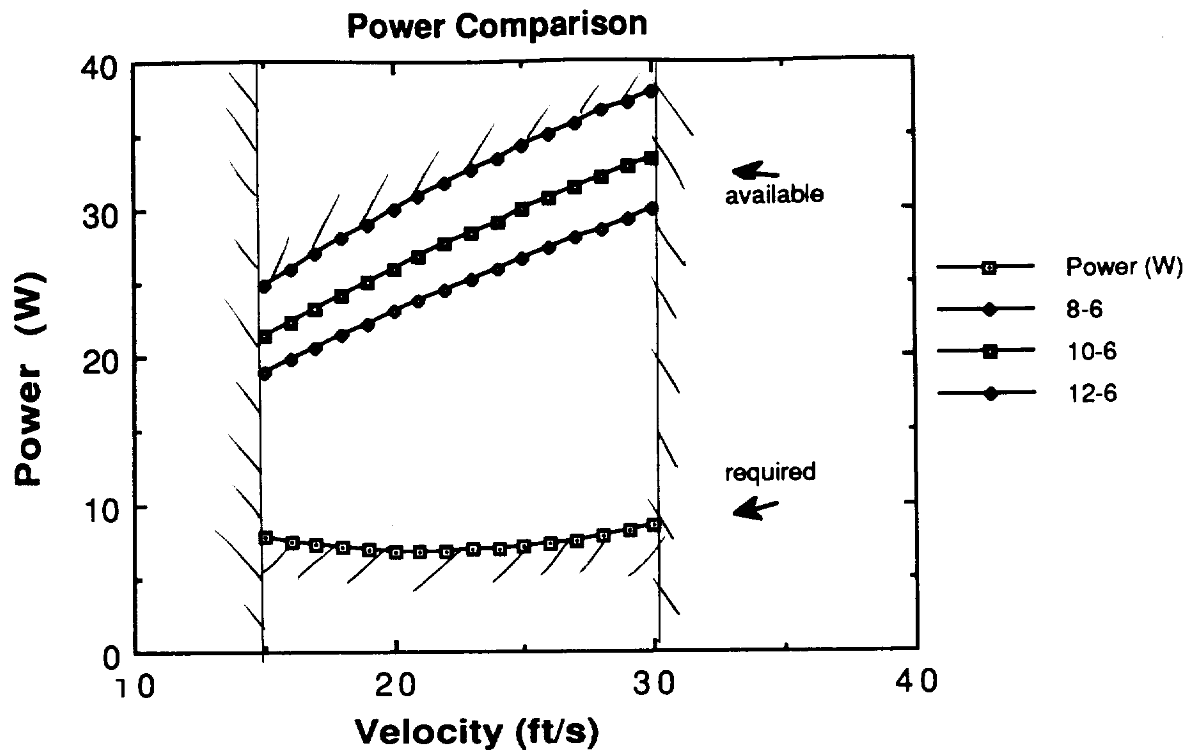
4.2 System Selection

4.2.1 The Motor

As specified in the Request for Proposals, only a propulsion system which was non-airbreathing and did not emit any mass could be investigated for use. While the possibility of using a rubber band to power a propeller was initially considered, an electric motor which could efficiently provide the necessary performance was decided upon. The Astro Flight line of cobalt motors was then investigated.

As mentioned, a power comparison was made using acquired motor and propeller characteristics. The power required versus the power available for the target velocity range of 10 - 30 ft/s is shown in Figure 4.1. This figure represents the aircraft at maximum RPM, which is approximately 8500, with each of the proposed propellers. The excess power evident from this graph (R/C =318 ft/min at 24 ft/s, 10 inch propeller) substantiates the fact that the geared Astro 05 will enable the aircraft to take off and climb as desired.

Figure 4.1



This amount of excess power is necessary to overcome any underestimation of the total drag at the low speeds that will be encountered, and to provide for a factor of safety (F.S. = 1.5); this will ensure that enough power will be available to take off and climb despite a wide drag variance. This excess power will also enable the pilot to escape a stall situation by increasing the velocity and reducing the angle of attack. Additionally, in comparison with the Astro 15, for instance, 10 oz. of system weight can be saved by using the Astro 05. The geared 05 also has the ability to create more power at lower speeds with a propeller of larger diameter than a direct drive motor. The Astro Flight company estimated that the " Astro gear boxes allow the motor to turn a larger propeller and therefore produce about 1.5 times the thrust at about 2/3 the speed "(Astro Flight Inc. brochure and price list, 1990). This combination of power available, speed reduction and weight minimization led to the selection of the geared Astro 05.

4.2.2 The Propeller

While it was shown in Figure 4.1 that any of the three propellers that were examined would provide the necessary excess power to take off and climb, there were specific reasons for selecting the Zinger 10-6. The 8-6 performed well, but an eight inch propeller can be used with a direct drive motor, and therefore, using an eight inch diameter propeller would not really take advantage of the previously mentioned capabilities of a geared motor. The small height of the main fuselage created some sizing problems for our "launching" gear configuration when trying to allow for adequate ground clearance for a 12 inch diameter propeller. As well as this sizing problem, the added drag of the 12-6 ruled that propeller out. In the end, the Zinger 10-6 met the design goals for power available, had a reasonable efficiency range, and minimized the structural complications.

The decision to utilize a folding propeller came as a result of the concern over the possibility of breaking the propeller upon landing. A folding propeller allows the two blades to fold back around the cowling as the motor is throttled down to a stop. A metal "z bar" attaches to the shaft, and two hinges allow the prop to fold. Hopefully this will minimize the chance of breakage upon impact. A wooden propeller was chosen due to the ability of the manufacturer to construct the desired diameter and pitch propeller out of this material less expensively than a plastic, fiberglass or metal propeller

4.3 System Performance

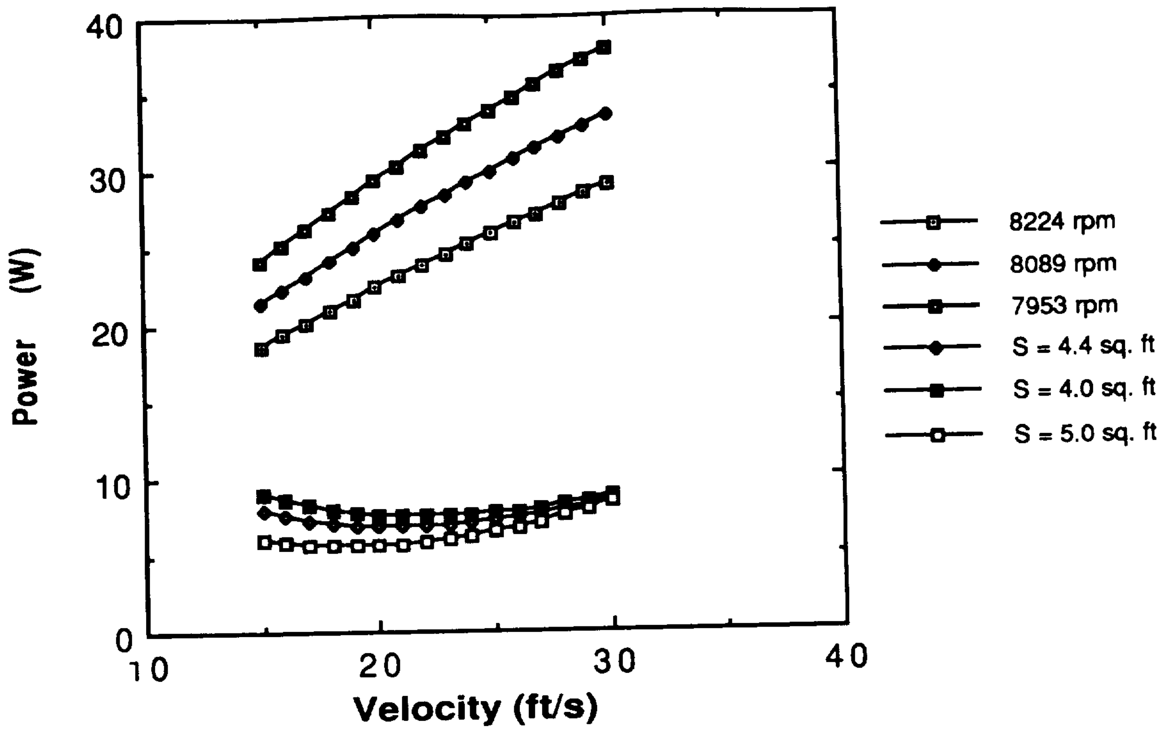
Performance estimates for this propulsion system were made through the manipulation of a simple spreadsheet that is described in Appendix A. Basically, the data gathered to perform the power comparisons was extended to find such values as thrust, takeoff distance, and time to altitude using equations noted in Appendix A. From the power comparisons, the power available at take off is 30 W, and at cruise is 29 W. The geared Astro 05 (rated at 115 W= .154 hp) provides .67 lbs of static thrust (3 N), according to the equation found in Falk's Aircraft Propeller Handbook:

$$\text{Tst \{lb\}} = \frac{(29000) * \text{SbHp}}{n * D} = \frac{(29000) * .154}{8000 * .833} = .67$$

where n is engine speed [RPM] and D is propeller diameter [ft]. This engine- propeller combination produces 4 lbs of thrust at the FX/90's cruise velocity of 24 ft/s and 8089 RPM.

An investigation of the effects of changing the total wing surface area and the total aircraft weight on the power required revealed certain trends and sensitivities. For the Astro 05, Zinger 10-6 configuration and a weight of 2.8 lbs, the power required increased up to 40% for a 25% increase in the surface area. While these were significant changes, the amount of excess power in the suggested maximum RPM range for this configuration still satisfied the power requirements for take off and climb. Figure 4.2 shows this sensitivity. Note that the FX/90 will be operating at the bottom of the power required curve, where the power required is at a minimum.

Figure 4.2
Variation of Power Required with Surface Area



The constraints indicated on the graph point out the range in which the FX/90 should operate to take off and climb: above the stall velocity, below a maximum velocity safe for indoor flight, and between the maximum and minimum rates of climb.

4.4 System Components

First, a speed controller will be utilized to allow the pilot to regulate the motor from idle to full speed. This is a necessity in indoor flight, especially considering the maneuvers to be performed during the flight. Secondly, a vented cowling will allow for the cooling of the motor during flight as well as improve the flow which passes the fuselage. Finally, the batteries being used provide the necessary endurance for flight tests and the mission, while greatly decreasing the weight of the propulsion system. Appendix A shows the calculations which substantiate the fact that only 3% of the battery pack is consumed during take off and climb, making several test runs and the mission feasible.

5 - Performance

5.1 Take Off and Landing

Although fixed landing gear is clearly the most conventional method of ensuring a successful take off and landing, the FX/90 was designed with the idea that the gear would detach upon take off. The weight savings anticipated for the FX/90 without fixed gear was the driving force behind this concept. The launching cart will allow for the natural rotation into a climbing orientation, and allow the FX/90 to take off in a distance considerably less than the specified 75 ft. In fact, it should take off after only 30 ft of ground roll, and the FX/90 has the capability to climb to the desired altitude. However, while confidence in a successful take off is high, the success of landing without gear is another story.

The FX/90 has been constructed in such a way that it should withstand the impact of landing at a glide slope of up to 5° with the pilot bringing the nose up before touchdown. The wing can withstand impact originating at its tip, because it is styrofoam and it will not shatter. All fragile internal components will be supported and padded to ensure that they will not be damaged upon impact. While take off will occur over a "normal" distance, landing will be rather abrupt, as the turf will tend to grab whatever element of the FX/90 contacts it in any way.

5.2 Flight Performance

The FX/90 is designed for low altitude flight (< 25 ft). The low speeds that will be used make excess power an important quantity, in order to ensure flexibility within that small range. Also, because of the closed loop nature of the mission, the range and the endurance of the FX/90 are also pertinent parameters. It has been estimated that the maximum range of the FX/90 configuration is 14,389 ft, with an associated endurance of 7.99 minutes in a minimum drag configuration. Additionally, the maximum endurance was found to be 8.48 minutes, with an associated range of 12,210 ft at the cruise condition of V=24 ft/s. These calculations were made as follows.

To calculate the maximum endurance, the condition for minimum power had to be examined. This occurred at our cruise velocity of 24 ft/s. A linear interpolation of the Motor Amps listed in the spreadsheet to the point where power required equalled the power available, led to the amperage to be drawn at cruise, 3.43 Amps. Knowing that just 3% of the 500 mah capacity of the batteries had been used to reach cruise, it is known that the remaining 97%, or 485 mah, would decide the endurance and the associated range at that condition. Accordingly,

Max Endurance = (.485 Ah / 3.43 A) * (3600 sec / 1 hour) = 508.8 sec = 8.48 min

Range = 508.8 sec * 24 ft/s = 12,211 ft

The maximum range occurs at the FX/90's maximum speed for indoor flight. Again, a linear interpolation was executed to find the amperage draw at a cruise velocity of 30 ft/s. For this amperage of 3.64 Amps, and with the remaining 485 mah,

Endurance = (.485 Ah / 3.64 A) * (3600 sec / 1 hour) = 479.4 sec = 7.99 min

Max Range = 479.4 sec * 30 ft/s = 14,389 ft

6 - WEIGHT ESTIMATION

6.1 Component Weights

Weight is, of course, a very important parameter in the design of any aircraft. It's value has a significant effect on many of the other vital parameters of the aircraft. As can be expected in a mission of this type, it is desirable to have the lightest possible aircraft in order to meet the major requirements of a low Reynolds number and high endurance. Therefore, compromises had to be made in many areas in order to obtain the light weight vehicle which is desired.

The weight of the vehicle was estimated by taking the estimates of each different component in the the three areas of materials: propulsion, equipment, and structures. The individual pieces in each of these three areas and their weights can be seen in Table 6.2. As can be seen, the resulting total weight is a reasonable 45.3 oz. (2.83 lbs.). It is realized, of course that this estimate will, if anything, grow since there may be many small unaccounted pieces and materials which will used during the construction of our aircraft. The weight distribution was organized as shown in table 6.1.

Table 6.1

Material Area	weight (oz.)	percentage of weight
Propulsion	12.96	28.6
Equipment	7.38	19.4
Structure	24.0	53.0
Total	45.34	100

This is as expected considering the type of mission which being run.

In an attempt to save some weight, the aircraft was designed to use detachable landing gear. This is a carriage-type structure which will assist the aircraft during takeoff, but it will stay on the ground while the aircraft lifts off the carriage and into the air. As discussed in section 4.2.2, this required the use of a folding propeller, which is approximately 0.25 ozs. heavier than a fixed propeller. However, the weight savings obtained from the lack of landing gear (approximately 2 oz.) more than made up for this additional weight.

As far as the weight of the propulsion system is concerned, it is fairly straight forward as to what is being looked for. It was desired to obtain the smallest, lightest engine that will fulfill the mission. The propulsion section investigated the Astro 035, Astro 05, and the Astro 15 engines. They selected the Astro 05 in part due to its lighter weight than the Astro 05. On this same note, based on the necessary power supply needed from the energy source, it was decided that the mission could be completed using only 7 AA batteries (5.5 oz. total) down from the initial battery pack of 9.8 (ozs.) a 4.3 oz weight savings.

The weight due to the equipment needed to control the aircraft, such as the receiver, speed controller, etc., did not allow for a lot of room to save weight, since most of the pieces were at set weights. It is believed that almost 2 ozs. of weight was saved, though, when it was decided not to use ailerons on the aircraft. They were found to be unnecessary for the mission, weight was saved by not having the servo, hinges, supporting hard points, and control rods needed for their use.

The wings and fuselage make up a major portion of the total weight of the aircraft. Therefore these are very important points in the design as far as the weight is concerned. Two individual studies were performed in these areas and the results are presented in the structural design portion of this document.

6.2 Center of Gravity

The location of the center of gravity is crucial to the aircraft design. The airplane stability and control are governed by moments about the center of gravity, with the rotational motion of the airplane as a response to these moments. For longitudinal static stability, the position of the center of gravity must always be forward of the neutral point. Since the individual component weights were fixed, it was possible to vary the c.g. position by changing the placement and internal arrangements of individual equipment pieces. Since the center of gravity was movable, it was possible to locate the neutral point at a desired location. Then using the designed static margin of 7%, a center of gravity was desired which was 0.5 inches behind the 1/4 chord of the wing.

Table 6.2 was used to find the center of gravity of the entire aircraft using the 1/4 chord of the wing as a reference point. Using initial rough estimates drawn from our preliminary design scheme the weight was almost 0.5 lbs heavier than the final estimate, and the center of gravity was more than 1.5 inches behind the 1/4 chord of the wing. At this point, the internal arrangement of the equipment, sizing of the fuselage, and materials selections, which all have a significant effect on the center of gravity location, had not yet decided upon. As the design began to take a more definite shape and certain variables began to be set, the center of gravity estimate became more accurate. Along with the decisions to use the lighter Astro 05 and fewer batteries the center of gravity moved back even further. Then, due to changes such as shortening the portion of the fuselage from the wing to the tail, lengthening the main fuselage, and moving much of the equipment forward in the internal arrangement, the center of gravity was placed very near to the desired point. It should be noted that there are many miscellaneous weights that come into play when the construction began taking place. This moved the center of gravity slightly. However, the proper factors of safety were used and enough room was left in order to adjust the equipment accordingly as was necessary.

Weight Estimation, Center of Gravity

ITEM	WEIGHT (oz)	IN FRONT OF 1/4 CHORD (IN.)	Wt. x Pos
PROPULSION			
motor	5.5	9	49.5
battery(motor)	5.46	5	27.3
prop	2	11	22
mount	1	10	
EQUIPMENT			
receiver,antenna	0.95	2.5	2.375
battery (reciever)	2	2.5	5
servos (2)	1.2	-1	-1.2
speed controller	3.23	4.5	14.54
STRUCTURE			
vert tail	1.5	-28.75	-43.13
horiz. tail	2	-28.75	-57.5
wings	10.5	0	0
fuselage	10	-4	-40
C.G. IN FRONT BY (inches)	-0.47		

Table 6.2

It should also be noted that the center of gravity location in the y and z directions were also considered. It was surmised that the cross sections in the corresponding perpendicular planes are near to symmetric. Therefore the c.g. in these directions did not have as much of an effect on the stability of the aircraft as did the c.g. in the x direction. A small moment arm, along with the ability to trim the control surfaces to counteract these moments, led us to neglect rearranging the design to correct for any instabilities.

7 - Structural Design

7.1 V-N Diagram

The velocity versus load (V-N diagram) can be seen in Figure 7.1. The load factor considers all of the forces applied to the aircraft structure except the weight. During the mission, there are certain phases which may give unfavorable loadings which could exceed the allowable loads of the aircraft structure. The loads during take-off, climbing, steady-level flight, and turning were all studied and are discussed in section 7.2. The V-n diagram shows the flight envelope of our aircraft, subject to stall limits, velocity limits, and wind gusts. The large extent of the stall curve indicates how much of a consideration stall was, which further indicates the influence of the low Reynolds number flight in the design. The low stall line only allowed for a small inflight load factor, and a small factor of safety.

7.2 Flight and Ground Load Estimations

During takeoff of our mission, it is desirable to obtain a quick rate of climb. This acceleration in the upward direction cause more loading on the aircraft than during steady level flight. The load factor is given as $n=1+a/g$, which at our desired acceleration gives an n= 1.024. During the turns, the aircraft will experience some increased loading, much like it did in the takeoff phase. The resulting load factor of the 50 ft. radius turn is calculated to be n= 1.11. Also taken into consideration is the possibility of wind gusts. Although the aircraft flys inside Loftus, where the wind is practically nonexistent, small gusts were compensated for. There is still a slight chance that the aircraft will encounter a small disturbance. The last area which was investigated concerning the load factor was the landing. This maneuver will be executed without landing gear so a precise value for the load factor cannot be found. In order to lessen the load during landing, the pilot will be instructed to have as slow of a descent as possible, with a small flare maneuver to give the softest possible landing.

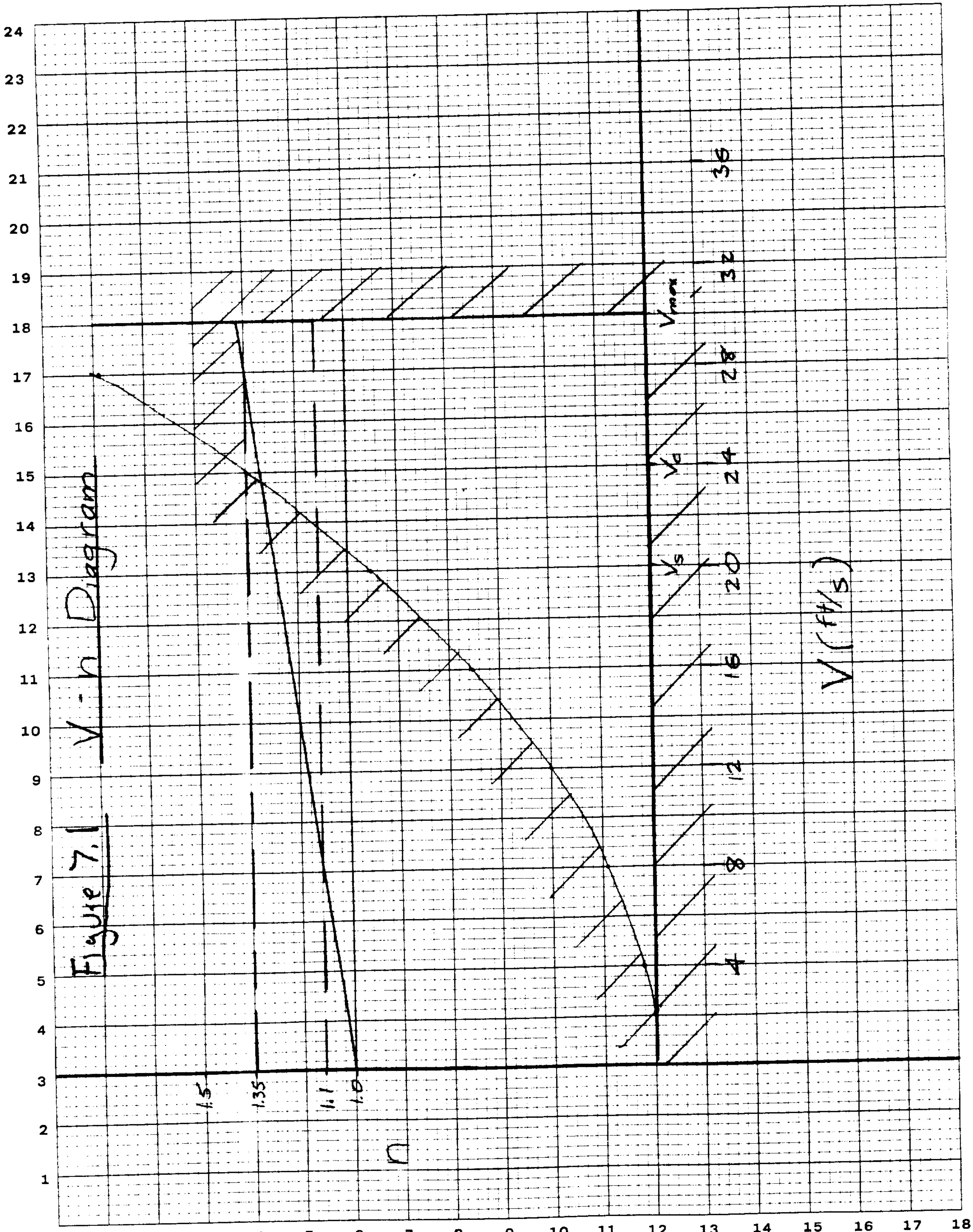


Figure 7.1 V-n Diagram

Although it cannot be sure that the pilot will be able to execute this properly, the aircraft is not designed for a crash landing and withstanding a crash is not a mission design requirement.

7.3 Substructures and Materials Selections

There are many different ways to construct an aircraft. Once a conceptual design for our aircraft was decided upon, it was necessary to decide how to construct it in order to end up with the desired geometry, weight and strength of the structure. This led to studies of the different possibilities which could be used to construct the body of the aircraft.

7.3.1 The Main Fuselage

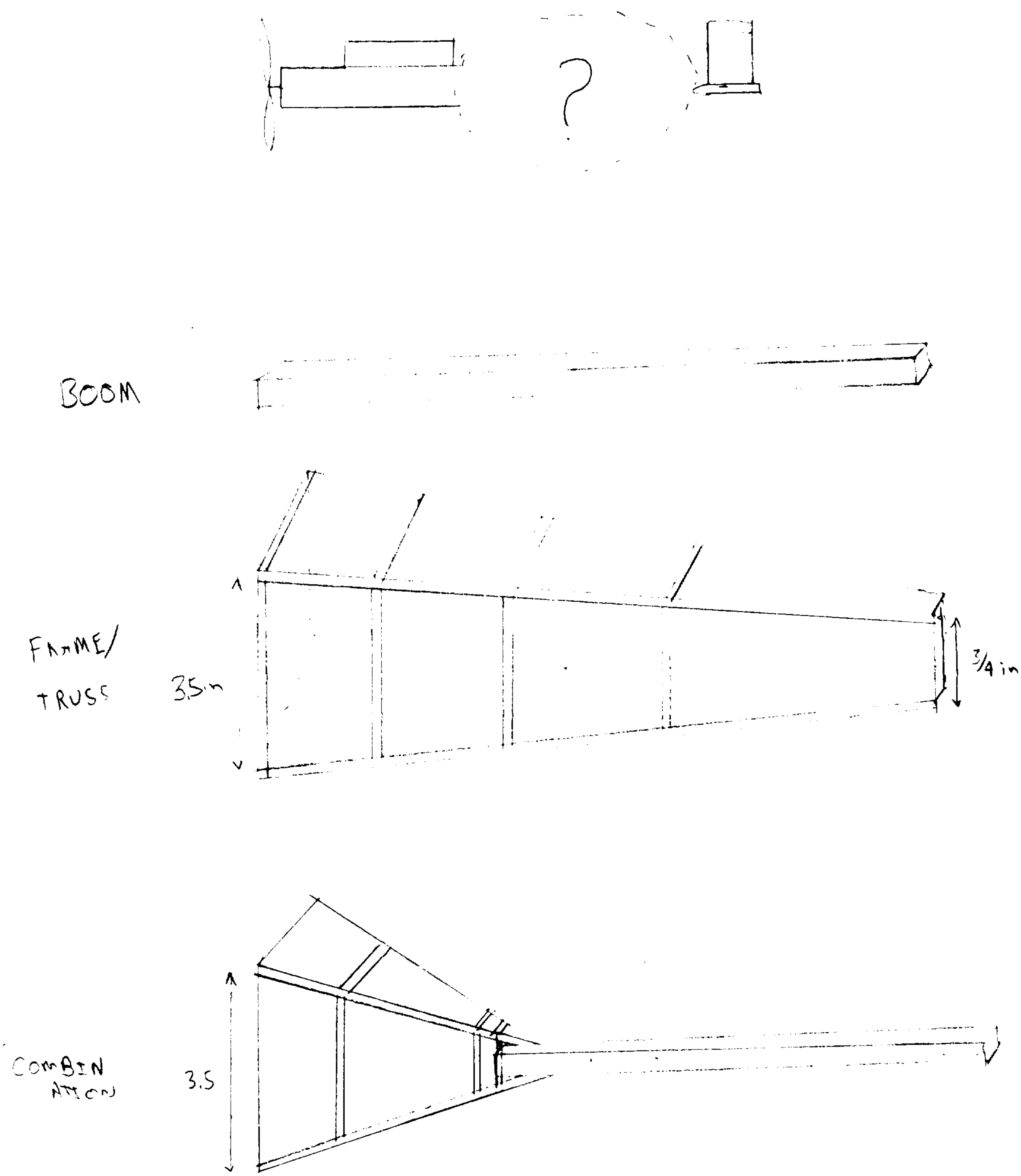
The main fuselage is of great importance to the overall design of our aircraft. Since it does not contribute much lift to the aircraft, it was designed to limit the amount of drag which it produces and the amount of weight used. It's purpose is to hold the equipment used in the aircraft and to support and transmit the loads from the propeller, wing and tail. After a little bit of thought, it was decided that a rectangular cross section would have comparable weight and drag to any other configuration and it would be easy to construct. Therefore this seemed to be the most efficient design for our performance and time requirements. In order to fit all the needed equipment and obtain the necessary center of gravity location, the dimensions of the main fuselage cross section were made 3.5 in. wide, 4.0 in. high, and 17 in. long. It was constructed with all wood. The sides will be made of plywood sheeting, which has a high strength, since this must hold the main loads (wing, propeller, and tail) in the vertical and axial directions. The side pieces of plywood will be cut to form fit the airfoil section of the wing in order to allow the wing to inset into the fuselage. Since it is judged that the side forces on the aircraft will not be as severe as the longitudinal forces, the top and bottom will not need as much support as the side pieces of the fuselage. Only balsa sheeting is needed on the top and bottom of the fuselage, instead of the heavier plywood. The internal supports to hold the cross sectional shape will be hallowed rectangular slabs of plywood. These will include the firewall, a middle support at the leading edge of the wing, and one at the trailing edge of the wing to help support the fuselage/tail section.

7.3.2 Rear Portion of Fuselage

A was conducted which was aimed at deciding how to design the portion of the fuselage which connects the horizontal and vertical tails with the main fuselage. This is a very important piece of the whole aircraft and has a critical affect on the entire mission. The ideas which were considered are (1) a small boom made out of a single beam, (2) a frame/truss configuration, (3) or a combination of the two (see fig. 7.2). The trade-offs which were considered had to do with the weight, strength, length, tip deflection, and tip slope. A smaller structure has less surface area and therefore less skin friction drag, but one must also be aware of the possible pressure drag that can arise from a discontinuous design with sharp corners (boat drag). Of course, the ultimate design would be the strongest, lightest, most rigid structure with no drag. But it is known that the lightest possibility is not the strongest, the most rigid does not have the least amount of drag, etc. Therefore the analysis used found the best combination of the above properties.

There were certain guidelines, as called for by other sections of the aircraft design, which had to be followed in this analysis. There were restrictions put on the design from other sources. It was desirable to keep the entire craft short enough to fit in the packaging box (i.e 4 ft.). Therefore the length of the portion of the craft from the main fuselage to the tail section was constrained to stay under 30 in. long. Also, the stability and control department requested that the length remain longer than 20

Figure 7.2



inches in order to provide the tail moment arm needed for a stable aircraft with the designed tail-sizing. The cross-section of the main fuselage is rectangular in shape and has the fixed dimensions of 4 in. high and 3.5 in. in width. This is where the root of our structure is anchored and must be fit to. On the other end, it was considered that a width of 0.5 in. at the rear section of the structure would greatly help when attaching the vertical and horizontal tail.

The forces on the horizontal and vertical tails, which were used to give stabilizing moments and maneuvering ability to the craft, were estimated during steady flight, turns, and takeoff, etc.. The estimates of these forces were dependent on the stabilizing moments necessary, and the length from the to the tail (the moment arm). Thus, the force was a function of the length of the section and an estimated percentage (10%) of the wing loading (3 lb.) and was estimated at

$$F_t = (30/L)(3\text{ lb.})(10\%) = 9/L$$

The study was limited to those easily available materials and material sizes (aluminum, and wood sized by 1/16 ths).

When considering the boom configuration, three types of materials were investigated: balsa, spruce, and aluminum. Table 7.1 shows the characteristics of these materials.

Table 7.1

Material	E(psi)	ρ (lb/in^3)	σ_{xx} (psi)	σ_{yy} (psi)	τ_{xy} (psi)
balsa	65E3	.0058	400	600	200
spruce	1.3E6	.016	6200	4000	750
aluminum	1E7	1E7	20E3	15E3	12E3

A small beam such as the boom may provide the necessary strength and still have a very small surface area in order to reduce skin friction drag, and weigh less than the other possible configurations. A computer code was designed in which each run explored the different characteristics involved when varying the length, material, and thicknesses of the boom. The maximum stress found in the boom was at the root (connecting to the main fuselage) for all cases, since this was where the largest moment arm was located and the cross-sectional area was constant along the entire length. But, for each run it was found that the allowable stress was not the limiting case, since it was well within limits for each of the materials. The figures of merit were most important were the tip deflection and slope at the tail end of the structure. It is easy to see how any bending of significant magnitude will change the desired angles of attack for the tail section and therefore throw off many of the designed geometries of the empennage.

The only beam thicknesses which were considered are those which were reasonable when considering ground handling (anything smaller than 1/16 was unreasonable). It was seen that in order to get a desirable tip slope (under 1.5 degrees), a considerable penalty was paid in the area of weight. For balsa the lightest configuration which had a sufficiently low slope, had a weight of 1.85 oz.. With spruce, this weight was 1.28 oz., and it was 2.5 oz. for the aluminum. It was desired to limit the weight to 2.0 oz. and limit the maximum tip slope to 1.5 degrees. These parameters effectively defined an operating region. It was also considered that, with a boom structure, it would be more difficult to attach the tail surfaces securely (as requested by the chief engineer), since they were very small. Also the skin friction savings may not be as much as expected since the two control rods which connect to the elevators and rudder will be exposed outside the structure.

Another area of investigation was the frame or truss configuration. This was attached to the main fuselage and sloped back to a smaller cross-section at the tail. It was thought that the frame would have better moment of inertia qualities with its wider area and therefore need less material, leading to a lighter structure. There would be struts within the truss in order to force the four outer beams, which carry the axial load, to keep their shape. The weight increase of the struts was accounted for and seemed to be a small price to pay for the extra rigidity. Tip deflection and tip slope were not a problem with the truss structure. Although it had more surface area than the boom, the gentle angle should eliminate most pressure (boat) drag which, according to the aerodynamics department, will make up for the added skin friction drag. Also, the servo arms would be internal with this configuration and there would be plenty of area at the rear in order to attach the tail section securely to the rear end of our structure.

The moment of inertia changed along the frame since its cross-section was not uniform along its length (wider and taller near the main fuselage than at the tail). Once again it was insured that the allowable stresses of the materials (balsa and spruce) were not exceeded at any point along the frame. Since the frame was always taller than its width and the forces in the horizontal and vertical directions were approximately equal, the moment of inertia in that direction was smaller and therefore the horizontal side force from the tail forms the limiting case. The greatest stress was found at the front section since the high value of the moment of inertia in this region did not make up for the large moment arm at this point.

Both the spruce and balsa stayed well within their values for allowable stress. When finding values for the weight it was estimated that the internal supporting for each would increase the weight by 50%, and the outer covering of the heat-shrink wrapping would add 0.2 oz. As expected, a shorter structure made out of lighter material and thinner cross-section gave the lightest weight. The spruce, although stronger, did not offer any significant advantage over balsa when using the beam sizes which were investigated.

The combination of both a truss and boom was also looked at. The advantage being sought was a lighter structure than the pure frame and a more rigid one than the pure boom. This option was disregarded without a technical/numerical analysis though. After a bit of thought, it was concluded that this configuration would instead offer the disadvantages of the boom (tip deflection, tip sloping and a small area to attach the tail) without losing the savings in weight. The reasoning behind this is that there would have to be considerable support in order to withstand the moments at the connecting joint between the truss and boom. This alone would add enough weight to cancel any other advantages involved with the boom/frame configuration. Along with this, one can reason that the extra effort involved in constructing this joint would also not be worth it.

Therefore, it seems as though the balsa frame structure was most advantageous to our assigned mission. It was lighter than the spruce while still offering enough strength to hold the loads which will be experienced during the mission. It was also lighter than any boom which was able to resist the tip sloping of the tail end. Therefore our aircraft makes use of the balsa frame structure 22 inches in length consisting of beams with an acceptable cross-sectional thickness of 3/16 X 1/4 inches with the internal supports sized with a cross section of 1/8 X1/8 inches.

7.3.3 Rudder and Elevator

A short study of the rudder and elevator structural design resulted in the design of using simple truss structure hinged to the vertical stabilizer and horizontal stabilizer, respectively. The actual dimensions of the control surfaces were designed by the stability and control department. It was discovered that the truss structure would be lighter than the other alternative investigated, a solid flat plate. Even though the solid flat plate had equal or greater strength, this did not make up for the weight savings involved with a truss structure, since both could hold the loads on the control surface.

7.3.4 Wings

With the airfoil section and wing characteristics already chosen by the aerodynamics group, the next step in the design of the wing for the FX/90 was determining its material composition. A "conventional" wing for this type of aircraft was made up of a balsa and spruce wood structure which was then covered with heat shrinking mylar. A wing of this type was composed of many pieces of wood which must be cut out and joined together with glue. Although this construction process can be a simple one, it requires a large amount of time to complete. Not only was it time consuming to cut and join the separate pieces, but the glue used to join them must also be allowed time to cure. Also time consuming was the process of placing the mylar covering over the wooden structure. To avoid the time commitment necessary to construct a "conventional" wooden wing for the FX/90, especially in the need for a replacement wing while the aircraft was being used in the field, a wing composed of another material was considered. This "other" material should reduce the time of construction for the wing while remaining within the constraints of weight, structural integrity, technology required for construction, and availability and cost of the material. The material found to meet all of these constraints was aircraft quality styrofoam.

Although the use of a solid wing simplifies the construction process, it introduces the problem of weight consideration. Therefore, an estimation of the weight of a wing made of solid aircraft quality styrofoam was necessary. This estimation was performed by determining the volume of the wing and also by knowing the density of the aircraft quality styrofoam. With a span of 72 inches and a mean chord of 9 inches, the estimated weight of the solid foam wing was determined to be approximately 9.7 ounces (See Appendix B). In comparison with the weight estimates of wood wings, the weight of the solid foam wing was about the same or less than that of a wooden wing of the same dimensions. Therefore, a solid foam wing meets the weight constraint.

Structurally, a solid styrofoam wing must be able to withstand the compression and tension stresses associated with the lift loadings on it. For this reason, an estimation of the maximum stresses on a solid wing of the FX/90 under the load factor of n=1 were performed. To perform this estimation, the airfoil section of the wing was approximated to be a rectangle with the same area as the airfoil section. This rectangle had the dimensions of 7.02 inches in the chord direction and 0.90 inches in the thickness direction. To simulate the lift loading on the wing, two point loads, each half the estimated weight of the aircraft, were placed at the half-span of each wing. By following this procedure, the maximum stress, tension and compression, was found to be 35.3 $lbs./in.^2$ at the maximum thickness of the wing (See Appendix B). It was known that aircraft quality styrofoam can withstand many times this stress. Therefore, it was a structurally suitable material for the wing of the FX/90.

One other major consideration in the feasibility of a solid styrofoam wing for the FX/90 was the technology which was required to actually construct the wing. The process of constructing the solid foam wing was actually very simple in concept. The wing of the FX/90 had no geometrical twist and was of constant airfoil section and chord length. To produce a wing, very few items were needed. They were:

1. A bow with a piece of nichrome wire stretched across it

The length of the wire must be slightly greater than the half-span length of the wing. The wing will be constructed in two halves to meet packaging constraints.

2. A Variac AC/DC voltage converter/amplifier

a DC current was run through the nichrome wire to heat it up. The hot wire was used to slice through the styrofoam.

3. Two templates of the airfoil section

The templates are placed on a piece of foam with dimensions slightly larger than that of half of the wing. The hot wire was then run along the edges of the templates, effectively cutting the wing out in the shape of the airfoil section.

4. Plywood sheeting and screws

These are used to make a joint which would join the two wing halves at the desired dihedral. The joint must be able to withstand the moment created by the lift of the wings.

The main drawback of this construction system was the fact that electricity was necessary. If a wing were to be constructed in the field, some sort of generator would be necessary to produce the electricity necessary. Overall, the technology required to cut the styrofoam wing was not extremely advanced or difficult to obtain. As for the choice of aircraft quality styrofoam as the type of styrofoam used, it contains a flame-retardent chemical especially for this cutting process.

As for the availability and cost of a solid styrofoam wing, aircraft quality styrofoam was carried by many aircraft part suppliers in different sized blocks. The cost of the foam depends primarily on the size of the airfoil being constructed. It may be necessary to purchase more styrofoam than was really needed, only because the material was sold in a certain size block. The cost of the foam and joint materials for the FX/90 was estimated to be about $35.

8 - Stability and Control

When analyzing the stability and control of the FX/90 there were three principal areas of interest. The longitudinal or pitch control, the lateral or roll control, and the yaw control. If the plane was to fly properly, it must be stable around all three of these axes. Because control about the longitudinal axis can be decoupled from the other two axes, it will be analyzed first.

8.1 Longitudinal Stability

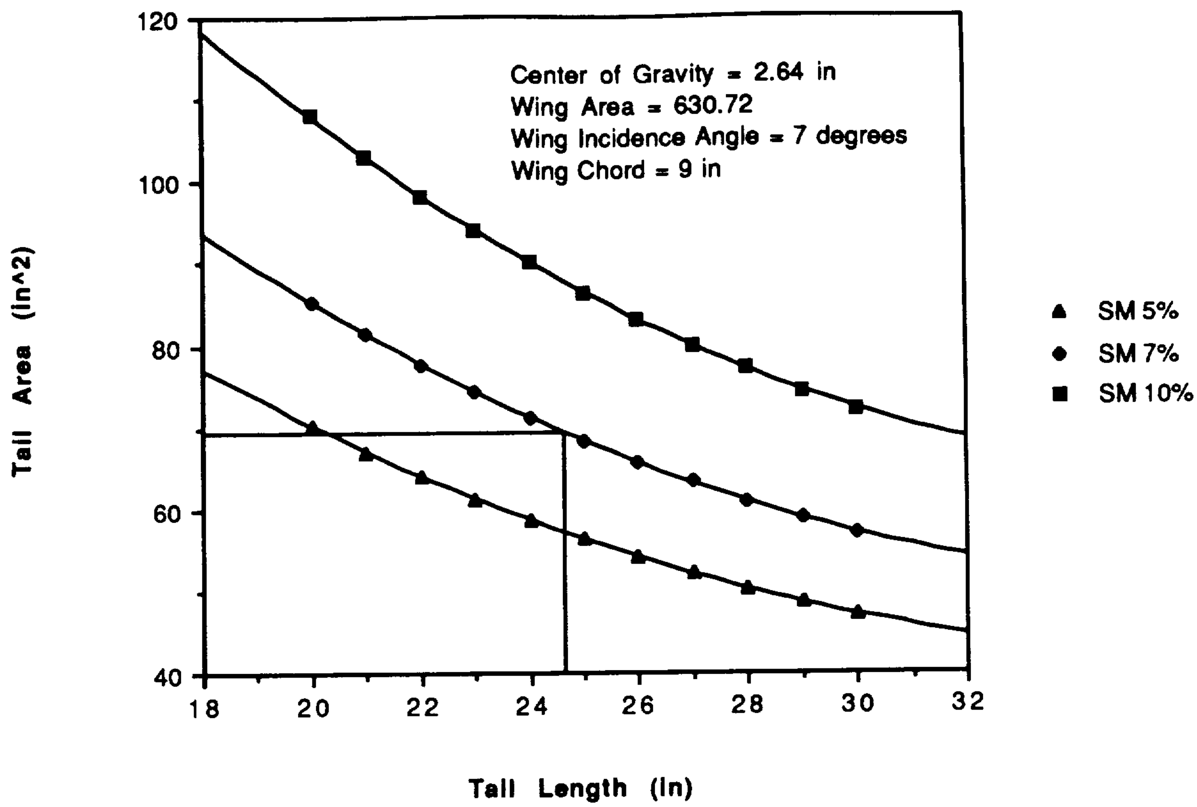
8.1.1 Static Margin

When analyzing the longitudinal characteristics of the FX/90, the parameter of interest was the static margin. The static margin was defined as follows:

$$\text{Static Margin} = \frac{X_{NP}}{c} - \frac{X_{CG}}{c}$$

If the FX/90 was to be statically stable, the neutral point must be located aft of the center of gravity. Therefore the static margin should be some positive value. For remotely piloted vehicles, it was recommended that the static margin lie between 5% and 10%. If the static margin falls below 5%, the plane will begin to become neutrally stable, while a static margin greater than 10% will require an exceptionally large horizontal tail area. In figure 8.1, the tail length was plotted versus horizontal tail area for various values of static margin for the FX/90.

Figure #8.1 Tail Area vs. Tail Length



From this plot it can be seen that for a given tail length, the required tail area increases with increasing static margin. With this restriction in mind, a value for the static margin of 7% was finally decided upon.

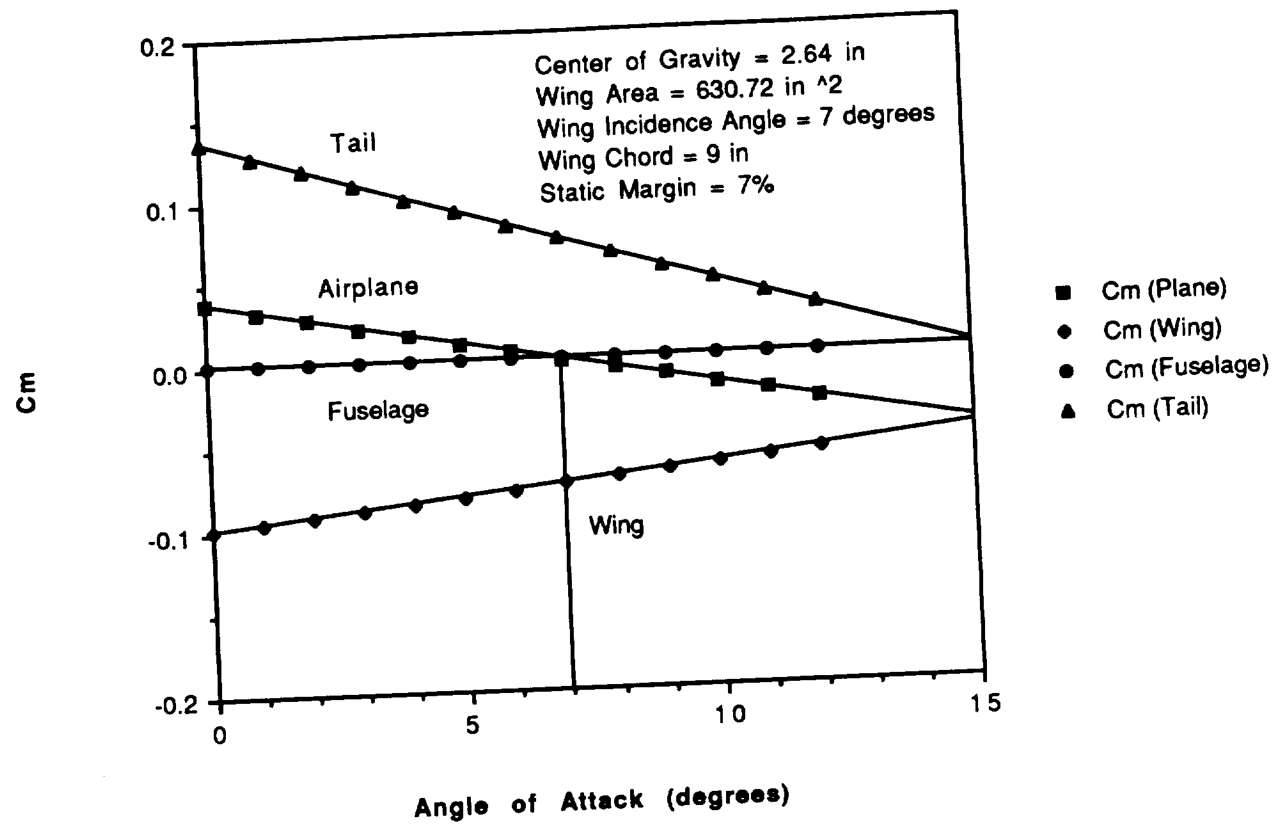
8.1.2 Horizontal Tail Area

From the weights and balance group, the center of gravity for the FX/90 was estimated to be 2.64 inches behind the leading edge of the wing. This was the value for the center of gravity that will be used in determining the required tail area. The effect of movement on the center of gravity location will be investigated later.

From the structures group, it was decided that the optimum tail boom length would be 22 inches. This set the length of the tail at 24.61 inches. From figure 8.1 it can be seen that the area of the horizontal tail was 69.4 in^2.

As a check on the static stability of the FX/90, the moment coefficient, Cm, was plotted versus angle of attack in figure 8.2.

Figure #8.2 Angle of Attack vs. Cm



From this graph it can be seen that the slope, or Cm_{α}, for the FX/90 was negative, which meets the requirement for a statically stable airplane.

8.1.3 Tail Incidence Angle

The next area of interest was the incidence angle of the horizontal tail. Because of the high incidence angle of the wing, only a very small negative tail incidence angle was needed. In order to minimize the drag on the FX/90, the tail incidence angle was picked such that at cruise velocity the fuselage would be at zero degrees angle of attack. The following equation was used in determining the tail incidence angle:

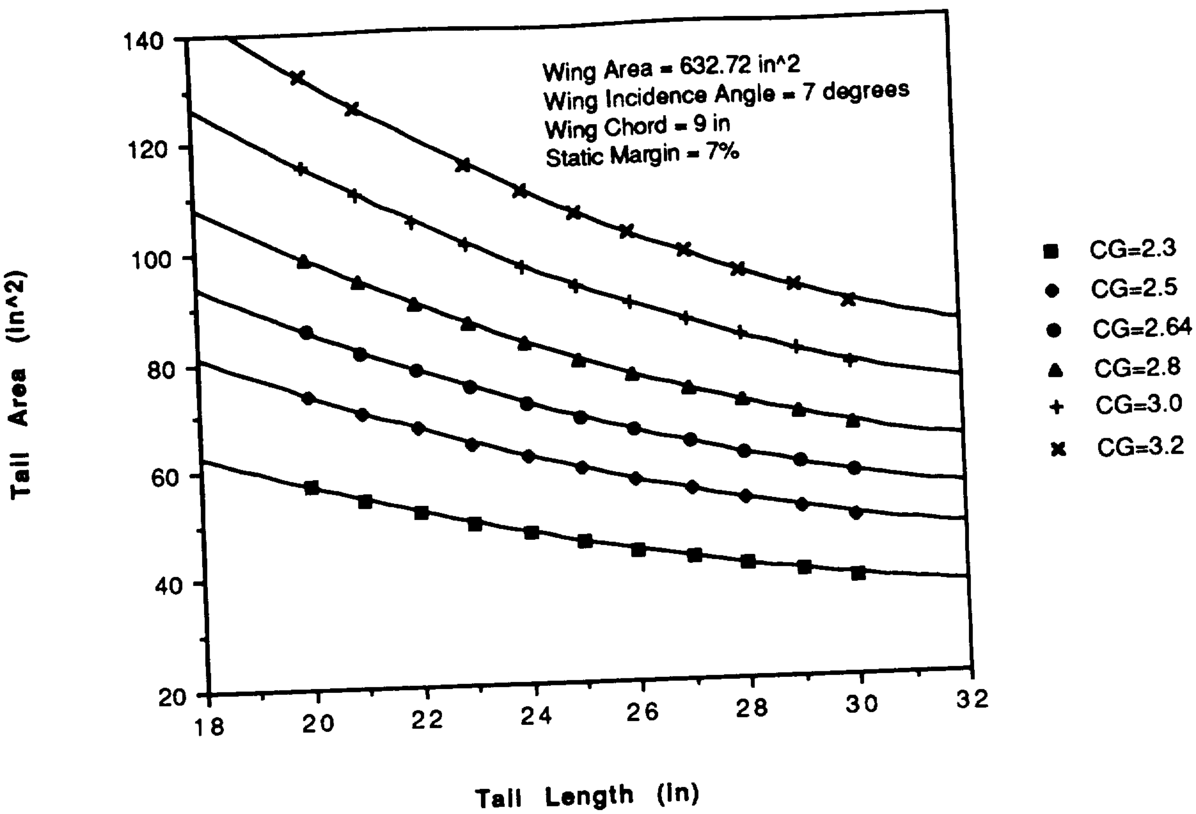
$$Cm_0=Cm_{0_W}+Cm_{0_F}+nV_HCL_{\alpha_t}(\varepsilon_0+i_w-i_t)$$

In this equation, everything was known except for the tail incidence angle and Cm_o. In order to determine Cm_o for the FX/90, the angle of attack of the wing at cruise velocity was needed. This angle was simply the incidence angle of the wing, which was calculated by the aerodynamics group to be 7 degrees. From figure 8.2, it can be seen that Cm was zero at 7 degrees, corresponding to a trimmed condition. Cm_o can also be determined from figure 8.2. With Cm_o known, the tail incidence angle was calculated to be -0.37 degrees.

8.1.4 Effects of Center of Gravity

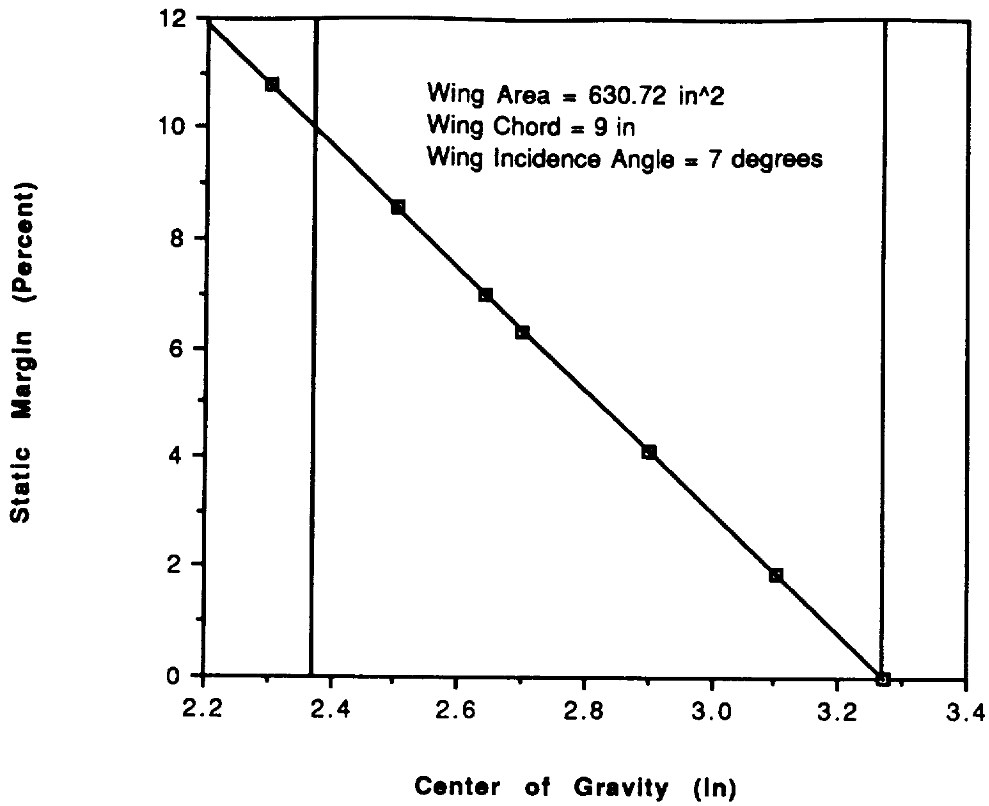
From the weights and balance group, the center of gravity for the FX/90 was estimated to be 2.64 inches behind the leading edge of the wing. In the technology demonstrator, the actual value may vary. Because of this, the effect of center of gravity position on the static stability must be analyzed. In figure 8.3, the tail length versus tail area was plotted for various center of gravity positions for the FX/90.

Figure #8.3 Tail Area vs. Tail Length



Because the static margin was earlier fixed to be 7%, the tail area on the FX/90 must become larger as the center of gravity moves toward the rear of the airplane. Once the horizontal tail was built, the tail area will no longer be able to be changed. This means the static margin of the FX/90 must change. In figure 8.4, the effect of center of gravity location on static margin can be seen.

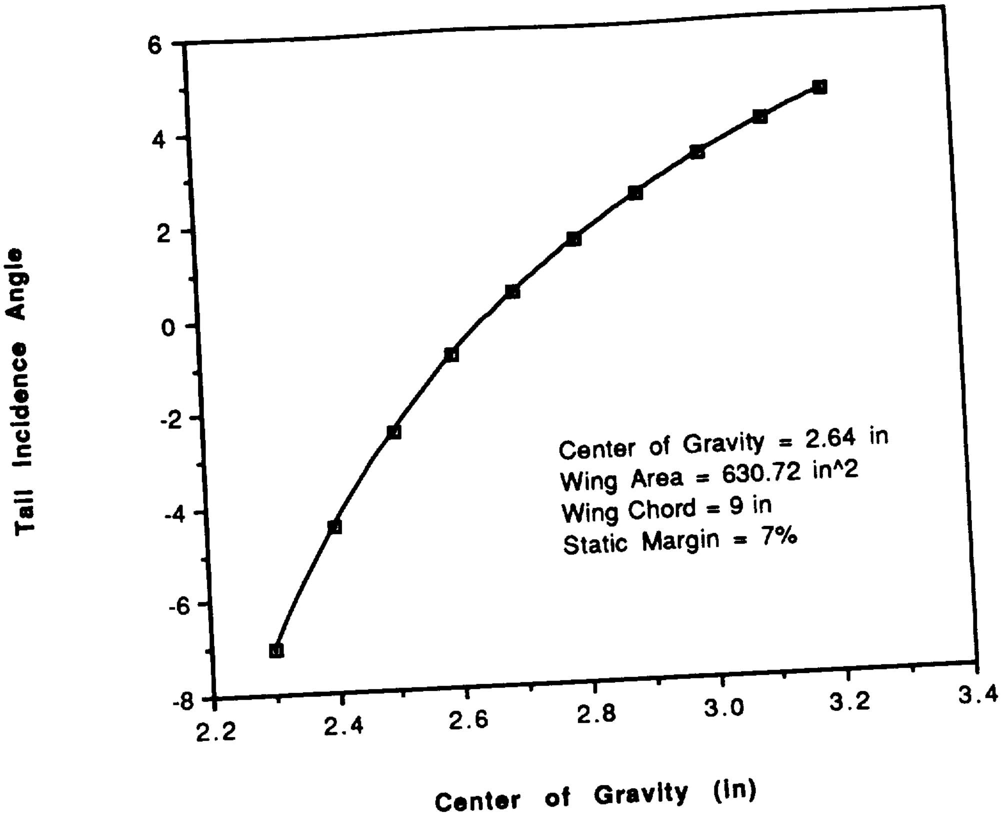
Figure #8.4 CG vs. Stratic Margin



From the graph, it can be seen that the FX/90 will become unstable if the center of gravity moves more than 3.27 inches behind the leading edge of the wing. In addition, if the center of gravity moves too far forward, the static margin will become too large, eventually making the elevator ineffective. Because of this, the maximum forward movement of the center of gravity which will be allowed was 2.37 inches corresponding to a static margin of 10%.

The center of gravity location will also have an effect on the required tail incidence angle. The center of gravity location for the FX/90 was plotted versus required tail incidence angle in figure 8.5.

Figure #8.5 CG vs. Tail Incidence Angle



From this figure, it can be seen that small changes in the center of gravity location would have a large impact on the desired tail incidence angle for the FX/90. Figures 8.4 and 8.5 show the importance of keeping the center of gravity close to the desired position. If the center of gravity should fall aft of the desired position, figure 8.4 shows that the tail area would need to be enlarged if a static margin of 7% was to be maintained. Figure 8.5 shows that a center of gravity position forward of the desired value would require a negative tail incidence angle, which would mean a negative tail lift on the FX/90.

8.1.5 Elevator Sizing

The final area of interest in longitudinal stability was the sizing of the elevator. To do this, it was first decided that the maximum deflection of the elevator would be 10 degrees in either direction. Next, the maximum moment that the elevator would need to overcome was estimated. It was determined that the maximum moment would occur if the airplane was at zero degrees angle of attack. Using the following equation, a value for the required elevator moment coefficient Cm_{de} could then be calculated.

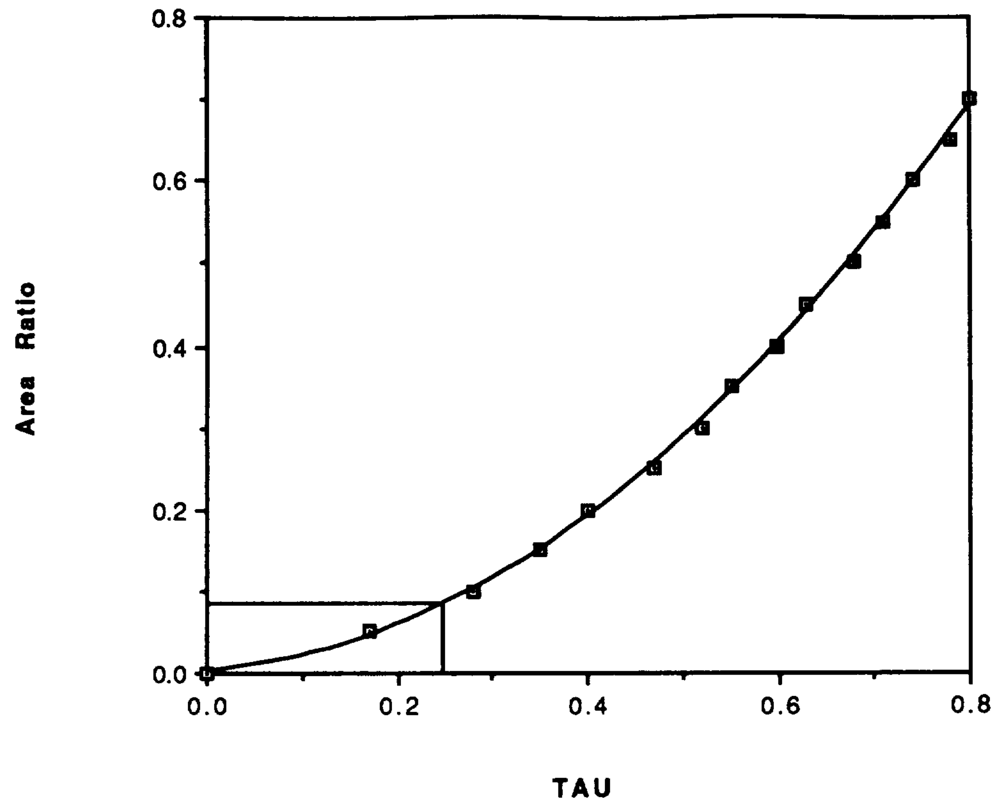
$$Cm_0 = Cm_{\delta_e}\delta_e \qquad \delta_e = 10$$

Once Cm_{de} was known, a value for t can be found using the equation:

$$Cm_{\delta_e} = -nV_H CL_{\alpha_t}\tau$$

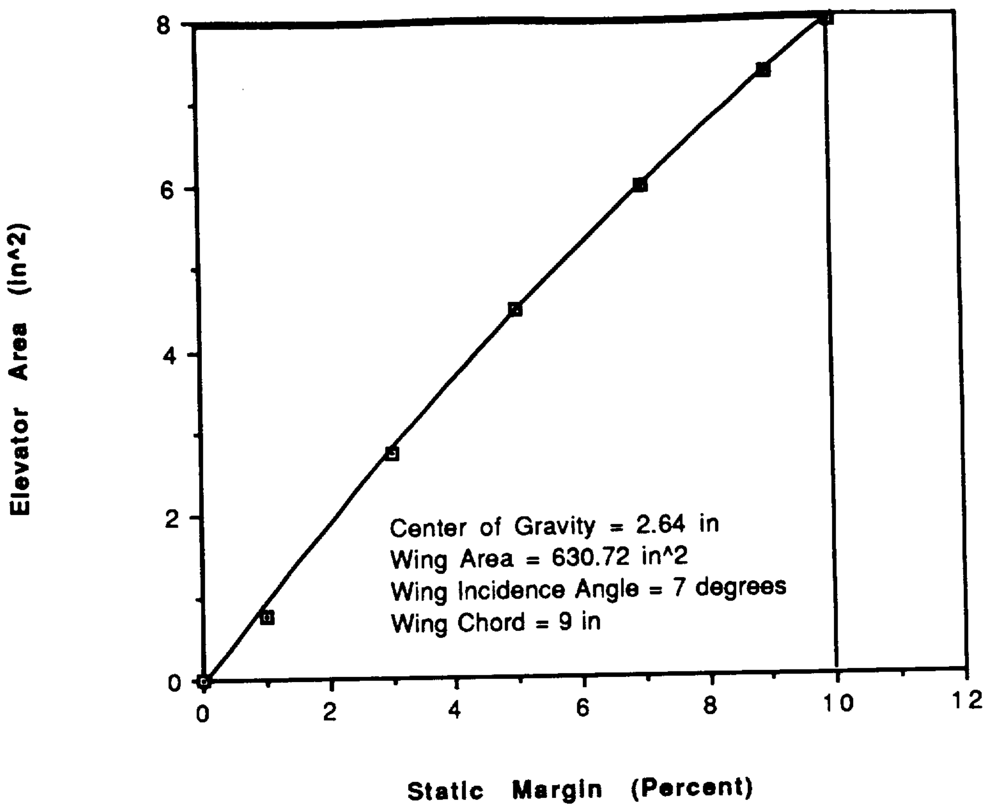
With τ known, the ratio of elevator area to horizontal tail area could found using figure 8.6.

Figure #8.6 Tau vs. Area Ratio



Overall, the static margin was also of importance when sizing the elevator. The static margin determined the maximum C_{mo}, which in turn influenced the required elevator area. From figure 8.7 it can be seen that the required elevator size increased with increasing static margin. From this figure it can be seen that the largest elevator area needed for the FX/90 occured for a static margin of 10%.

Figure #8.7 Static Margin vs. Elevator Area



From figure 8.7 the required elevator area was found to be 7.95 in^2. The parameters of interest for the horizontal tail of the FX/90 are summarized in table 8.1.

Table 8.1

Horizontal Tail Summary	
Area	69.4 in^2
Tail Volume Ratio	0.30
Chord	5 in
Tail Length	24.61 in
Elevator Area	7.95 in^2
Incidence Angle	-0.37 deg

Before flying the technology demonstrator, it was important to check the center of gravity location. If this location was between 2.37 and 3.27 inches behind the leading edge, the horizontal tail summarized above will provide sufficient longitudinal static stability and control.

8.2 Lateral Stability

The important design parameter in the the area of lateral dynamics was the dihedral angle. In this analysis, lateral dynamics describes motion about the roll, or x, axis. Lateral dynamics consists of two sub-areas, lateral, or rolling, response and spiral stability. Since the FX/90 does not have airelons, roll response and spiral stability were chiefly determined by dihedral. Note that there are two aspects of dihedral to be considered, dihedral type and dihedral angle, Γ.

The significant figure of merit in roll response was the maximum steady state roll. The spiral stability was quantified in terms of a yaw angle, β, which would produce an equivalent rolling moment. The design goals were to provide for an adequate roll rate and provide for a small positive spiral stability. A large roll rate increased the aircraft performance by enabling the aircraft to achieve desired bank angles quickly. It was not desired to be too large however, as this would have made the plane overresponsive. A positive spiral stability enhance handling by preventing the aircraft from gradually spiralling into a dive without constant attention. The spiral stability was desired to be small, however, to avoid a large resistance to circling maneuvers.

In addition to these areas, there was a third area to be considered, which was stall behavior. The significant figure of merit in stall behavior was the change in angle of attack, α. A small change in angle of attack minimized the potential for one wing to stall.

The aircraft reached steady state roll at the point where the roll moment and the roll damping balance. Thus the parameters which affected roll rate were a compromise between the parameters which affect roll moment and damping. The result was that the roll rate was inversely proportional to the wingspan, and was proportional to the velocity, yaw angle, and dihedral amount. However, the wingspan and velocity were fixed by other design considerations. Consequentially, yaw angle and dihedral amount were the parameters which can be used to affect the roll rate.

A similar analysis was used on spiral stability. The aircraft, while circling, developed a rolling in moment due to lift generated by an unequal airspeed over each wing. However, the aircraft developed a rolling out moment due to the circle radius and due to the effect of drag generated by the same unequal airspeed over the wings. An aircraft was spirally stable if it naturally rolled out more than it rolled in.

The required yaw, as well as the natural yaw due to drag effects, was inversely proportional to the surface area, circle radius, dihedral, and the velocity squared, and was proportional to the wingspan and weight. The natural angle due to circle radius

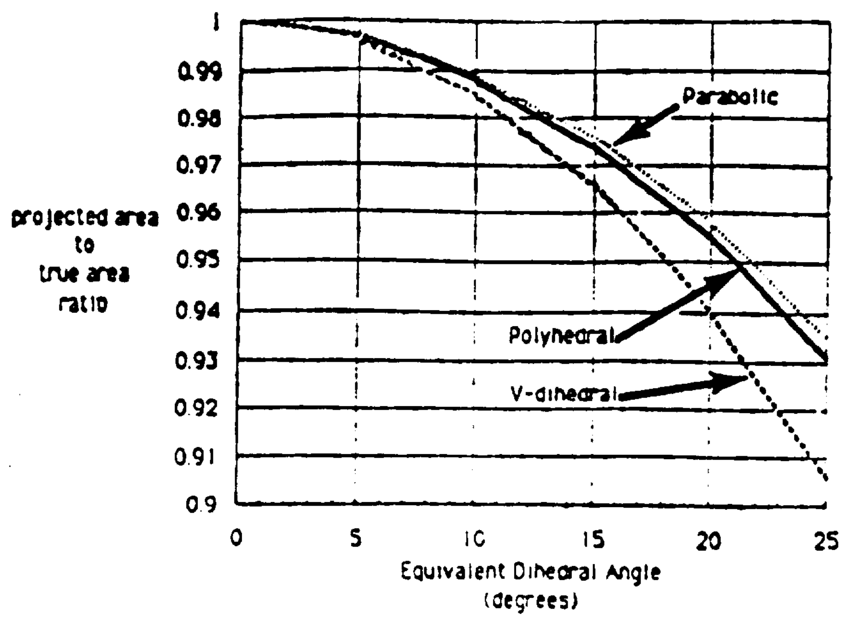


Fig 8.8

(source - see ref. 3)

was dependent on the circle radius and the vertical stabilizer moment arm. Again, all of these parameters are fixed except for the dihedral amount.

Finally, the change in angle of attack, $\Delta\alpha$, was proportional to the dihedral and the yaw angle for small angles. These variables are related through the equation: $\tan(\Delta\alpha) = \sin(\beta) * \tan(\Gamma)$. Delta a was also dependent on the type of maneuver being performed and the type of dihedral.

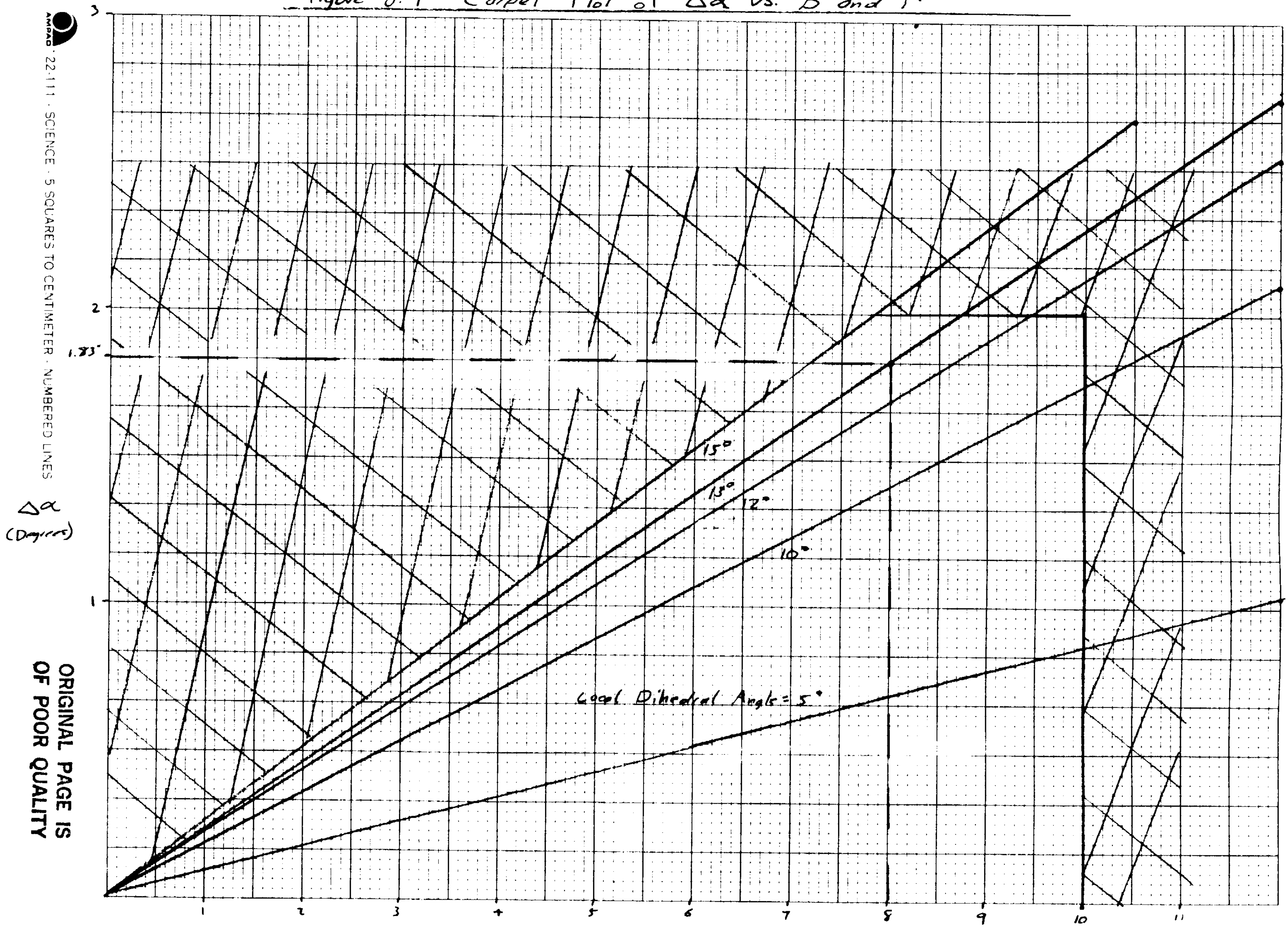
From this, it was clear the only variable parameters were dihedral amount, dihedral type and yaw angle. Increasing the dihedral angle increased the roll rate and the change in angle of attack, while increasing the yaw angle increased the rol rate.

The aerodynamic group provided a constraint on the change in angle of attack of two degrees. The stability and control group provided a constraint on the maximum yaw angle of ten degrees. The dihedral angle itself was constrained by its effect on efficiency. Dihedral tilted the lift vector, thus reducing the effective component of the lift. Figure 8.8 shows the percentage loss in surface area, and hence lift, as a function of dihedral angle. For a dihedral angle of 13 degrees, the percentage loss was only 2.75%. Thus the efficiency was only reduced by 2.75%. As a result, the dihedral angle was not strongly constrained. However, angles greater than about 15 degrees were be avoided, as the percentage loss began to increase rapidly at this point.

Since the $\Delta\alpha$, dihedral angle and yaw angle are linearly related, they are presented on a carpet plot (see fig. 8.9). The constraints on all three variables are shown on the plot. This then clearly indicates the acceptable design range.

DISCUSSION AND RESULTS - DIHEDRAL TYPE: The choice of dihedral type was driven mainly by considering the effect on angle of attack. There were other considerations, such as efficiency, location of the heavy joints relative to the center of gravity, and flutter. However, this aircraft operated at a low velocity and thus close to stall, so the change in angle of attack was critical. Furthermore, the other effects were minor. Thus the other effects were neglected.

Figure 8.9 - Carpet Plot of $\Delta\alpha$ vs. β and Γ



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A polyhedral dihedral wing had a small dihedral angle on the inboard portion of a wing, and a large dihedral angle on the outboard portion of the wing. Thus, a polyhedral wing with the same effective dihedral as a V type wing would have a larger dihedral angle on the outboard portion of the wing. Since delta alpha was proportional to dihedral, the polyhedral wing would have a larger delta alpha on the outboard portion than the V type wing during roll transition. Thus the polyhedral wing had a higher maximum change in angle of attack (see fig. 8.10).

During steady state roll, however, damping reduced the angle of attack increasingly along the wing in the outboard direction. Thus, on the polyhedral wing, the damping was greatest precisely where the maximum change in angle of attack was greatest. The result was that during steady state roll, the V type wing had a greater peak change in angle of attack than the polyhedral wing (see fig 8.11).

When the two cases are compared, it can be seen that the peak in steady state roll was less than all of the peaks in roll transition. Thus roll transition was critical case. Since the V type wing performs better in this case, this type of dihedral was chosen.

DISCUSSION AND RESULTS - DIHEDRAL ANGLE: Ref. 3 provided a method of determining the roll rate and spiral stability as a function of the relevant parameters. The article presented three plots of the roll rate versus one parameter for a specific airplane, and the other relevant parameters were stated. Figure 8.12 shows one of these plots as an example. The roll rate for the FX/90 was found from each plot, and was then scaled by the ratios of the parameters between the given plane and the FX/90. The roll rates found from each of the three plots were then averaged.

The rolling moments used for spiral stability were quantified in terms of a yaw angle which would provide an equivalent rolling moment. The equivalent yaw angle for the rolling in moment was calculated through a procedure similar to the one used to calculate the roll rate. (fig 8.13) The equivalent yaw angle for the rolling out moments were

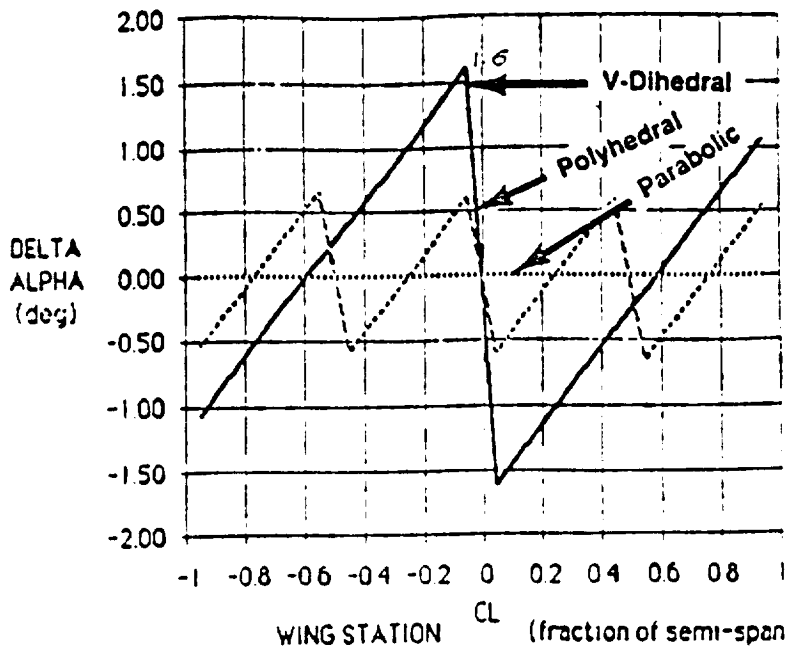


Fig 8.10
(source - see ref. 3)

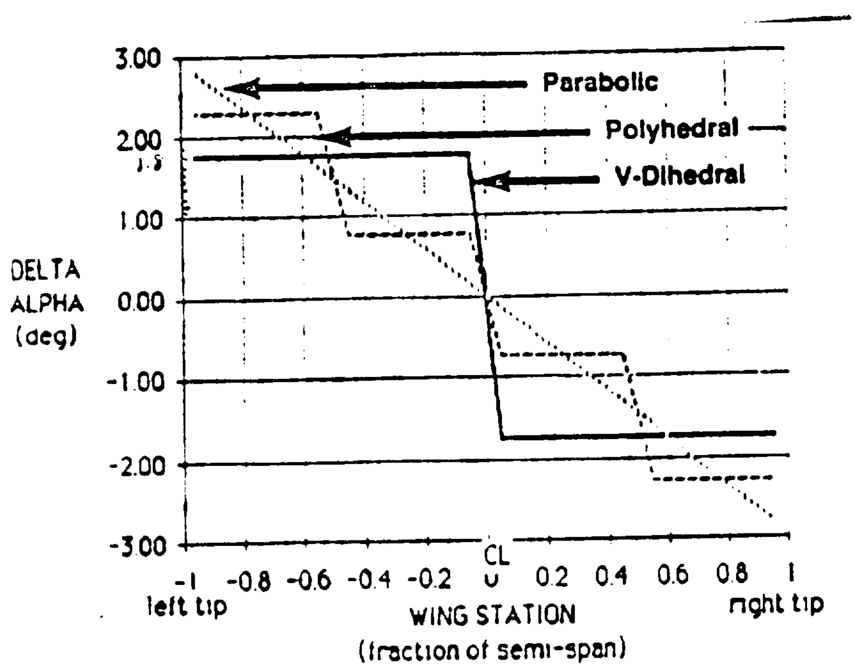


Fig 8.11
(source - see ref. 3)

Fig. 5

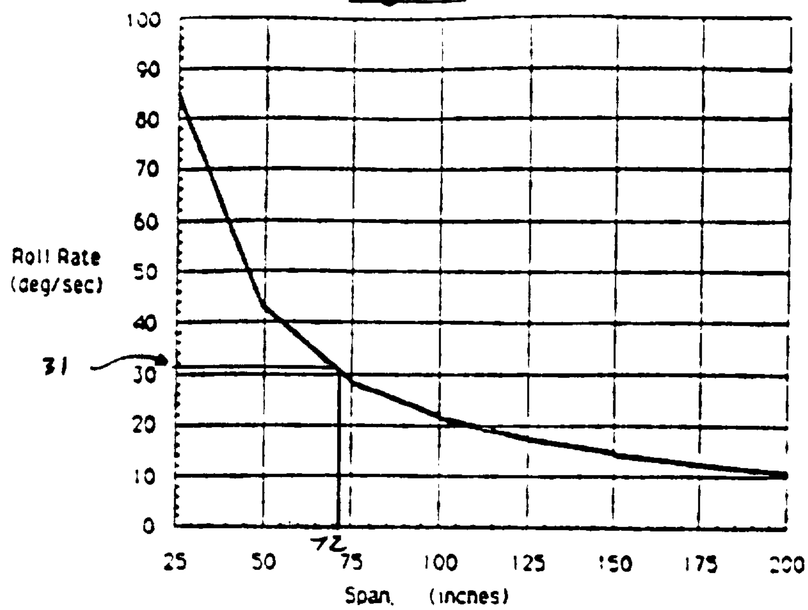


Fig 8.12
(source - see ref. 3)

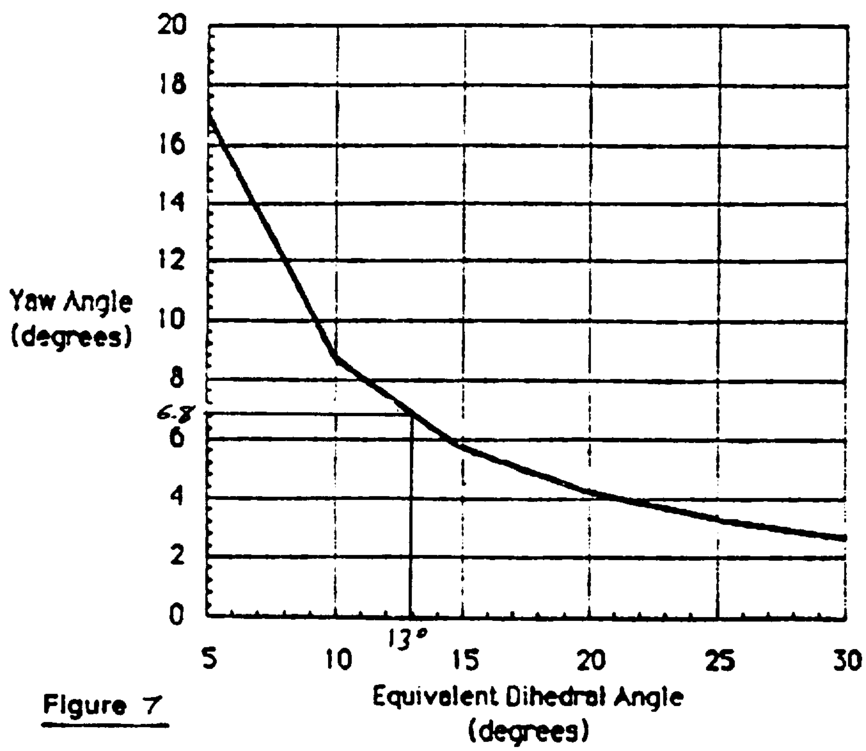


Figure 7

Fig 8.13
(source - see ref. 3)

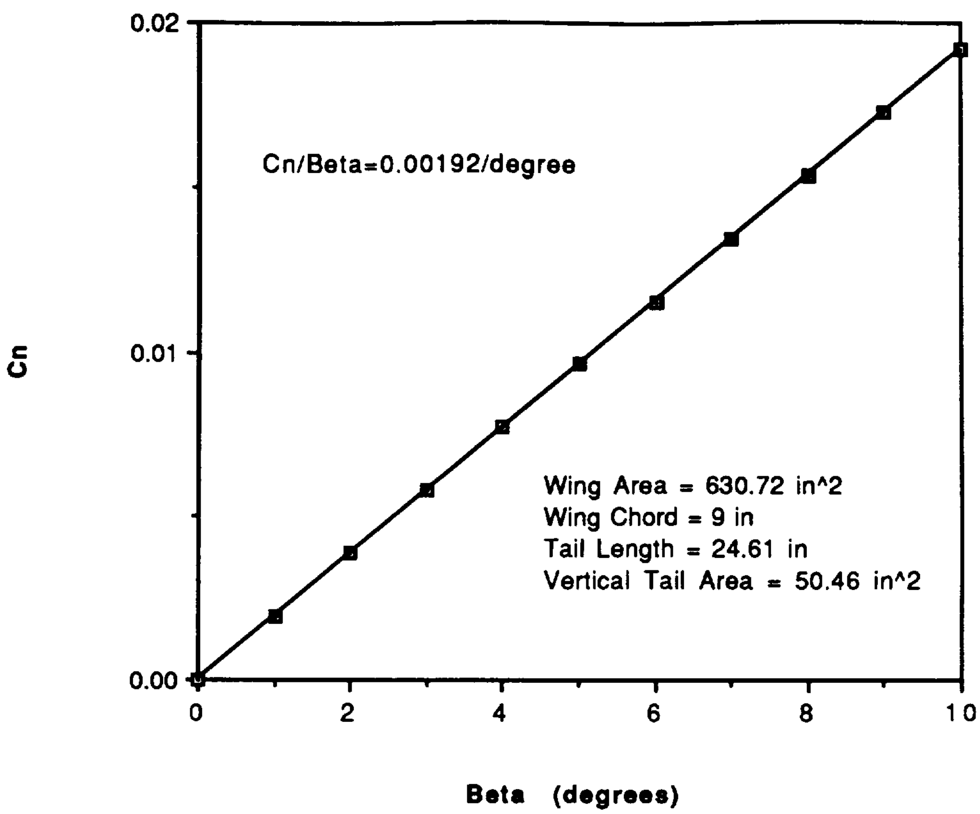
calculated in two ways. The yaw angle due to the circle radius was found by simply finding the ratio of the tail moment arm to the perimeter, and multiplying by 360°. The yaw angle due to drag effects was estimated by dividing the lift yaw angle by the L/D ratio.

The dihedral angle was chosen within its constraint to give a small positive spiral stability. The yaw angle was then chosen to give a large roll rate. Both of these quantities were not chosen larger than necessary in order to avoid obtaining excessive values for the spiral stability and roll rate, and to minimize the change in angle of attack. The dihedral was chosen to be 13°, and the design yaw angle for rolling was chosen at 8°. This gave a maximum change in angle of attack of 1.83°. The roll rate which resulted was 31 $^{\circ}$/s, which meant that the aircraft would require 0.79 seconds to reach the design bank angle of 21° at 23 ft/s, and would cover 18.16 feet during the maneuver. The resulting spiral stability was 0.35°, which was positive but small in magnitude, as desired.

8.3 Yaw Stability

When analyzing the yaw characteristics of the airplane, the first parameter of interest was the area of the vertical stabilizer. For remotely piloted vehicles, it was recommended that the ratio of vertical tail area to wing area be 0.08. Using this ratio, the vertical tail area was calculated to be 50.46 in^2. As a check on the static stability, the yaw moment coefficient Cn was plotted versus sideslip angle β in figure 8.14.

Figure #8.14 Cn vs. Sideslip Angle



From this graph it can be seen that the slope, or Cn_β, was positive. This was the requirement for a statically stable airplane.

The other area of interest was yaw control, or the sizing of the rudder. It was decided that the maximum deflection of the rudder would be 20 degrees in either direction. A maximum β of 10 degrees was chosen, since an excessive β increases drag during turns. From the graph above, the value of Cn required by the rudder was found for a sideslip angle of 10 degrees. Using the following equation, the value for τ can be calculated:

$$Cn=-n_V V_V C L_{\alpha V} \delta_r \tau \qquad \delta_r=20$$

Once τ was known, the rudder area was found to be 29.78 in^2. The parameters of interest for yaw stability and control are summarized in table 8.2.

Table 8.2

Vertical Tail Summary	
Area	50.46 in^2
Tail Volume Ratio	0.028
Chord	6 in
Tail Length	24.61 in
Rudder Area	29.78 in^2

9 - Technology Demonstrator

The technology demonstrator was proof of RPV design. The purpose of the technology demonstrator was to verify that the design proposed was airworthy and that the material selection and construction methods of the craft are appropriate. Much of the design and construction methods presented here are a result of solid advice given by the construction advisor, Mr. Joe Mergan. Mr. Mergan's years of RPV experience were a great asset to this group in this area of our proposal.

9.1 Flight Test Plan

The flight test plan to prove the airworthiness of the FX/90 was comprised of five separate tests. The first test was merely a ground taxi test, and each consecutive test will become increasingly critical. The five individual test are:

1. Slow Taxi Test

   The FX/90 will be run on the ground at low speeds to check ground handling qualities.

2. High Speed Taxi Test

   The FX/90 will be accelerated to near lift-off speed to ensure adequate take-off distance and stopping capabilities in the occurrence of an aborted take-off. The pilot will also be able trim the rudder in the yaw plane during this test.

3. Take-off and Wide Turn Test (to the left)

   The FX/90 will lift off and fly once around the pylon course as an oval, making wide turns around the pylons. This will give the pilot a feel for the roll qualities of the aircraft in shallow turns.

Also during this test, the pilot will check climbing and descending qualities of the aircraft in the straight-away portions of the course.

This test will allow the pilot to find the correct throttle setting for cruise flight. This test will also test the landing capabilities of the aircraft.

4. Take-off and Wide Turn Test (to the right)

   This is the same test as test #3, but in the opposite direction.

5. Mission Flight Test (as prescribed by the mission directive)

   Three laps of the figure-eight course described in the mission directive.

9.2 Flight Test Safety Considerations

There are four main areas of safety considerations for the flight test facility and the spectators viewing the testing of the FX/90. The first area of safety was in the visibility of the aircraft. The FX/90 will incorporate a bright and "catchy" color scheme. The color scheme entails using a bright white color on the fuselage and tail along with international accents on other areas of the FX/90. This color scheme will not only allow the pilot to keep the aircraft in sight, but will also allow him to monitor the relative state of the aircraft. This means that the pilot will be able ascertain the pitch, roll, and yaw attitudes of the aircraft in flight. The bright color scheme of the FX/90 will also allow the spectators to keep the aircraft in sight. In case of an emergency landing or accident, the bright color scheme will help the spectators keep away from the craft.

The second area of safety was in the materials and methods from which the FX/90 was constructed. One of the main features of the FX/90 was its solid styrofoam wing. In the event of a collision with part of the a spectator or Loftus Center , the styrofoam wing will be able to deform or break without causing much damage to the object it was striking. By doing this, the styrofoam wing was absorbing some of the kinetic energy of the aircraft, leaving less energy to do more damage. The rest of the FX/90 will be constructed out of balsa wood. Balsa wood was a soft wood and was able to absorb a fair amount of energy before it fails. Because of these characteristics, the wooden structure of the FX/90 will probably fail before doing considerable damage to a spectator or Loftus Center in the event of an accident.

The third area of safety consideration for the FX/90 was in its propulsion system. The FX/90 electric motor will be throttled by a speed controller. The speed controller will allow the pilot to stop or "cut" the motor in the event that a collision was imminent. By stopping the rotation of the propeller, the damage it can cause was reduced. Another safety precaution in the propulsion system of the FX/90 was its foldable propeller. This will reduce the damage to any object the propeller might strike, including the ground upon a normal landing.

The last area of safety for the FX/90 was in the area of positive control. In order to retain positive control of the aircraft, several precautions are being taken. The first precaution was in the control surface linkages of the aircraft. The control surface hinges and push rods connected to them will be securely fastened with possible redundancy through safety wire attachment. Secondly, Mr. Joe Mergan will test a radio control system in Loftus Center prior to the flight testing of any aircraft to check for any electromagnetic interference or radio signal reflection from the structure of the center.

By following these guidelines it was felt that injuries to spectators and damage to Loftus Center can be reduced. It goes without being said that minimizing injuries to the spectators was much more important than minimizing the damage to the Loftus Center.

July 24, 1990:

Page 62 removed because of

funding informaiton.

Philip N. French
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Document Evaluator

61-A

It must be noted that the costs listed above are for the aircraft only. These costs do not include overhead costs such as labor costs and production facility costs. It must also be noted that at this time these costs are primarily estimates. The actual cost of the finished FX/90 could vary depending on the actual amounts of the materials, such as wood and styrofoam, that are used in its construction.

10 - Feasibility of Mission Extension

As designed, the FX/90 meets the low speed and low altitude criterion effectively, but not without some serious shortcomings. In order to extend its use to a high altitude, station keeping mission, several things would have to be kept in mind. First of all, high altitude, outdoor flight would present the problem of gusts, which the FX/90 has shown a high sensitivity to in flight. Also, high altitude flight would necessitate strengthening the structure, as well as increasing the wing area and all control surface areas due to a lower density, thereby increasing the total weight. Most importantly then, the FX/90 would require much more power. This would entail a different motor if not just an increase in battery capacity. Essentially, as designed the FX/90 would not be suited to high altitude flight, while a redesign of some areas would provide satisfactory results.

Appendix A

This was an explanation of Excel spreadsheet used in propulsion system analysis. For any given velocity and aircraft weight, with certain characteristics of the propeller and motor known, the following can be estimated:

Advance Ratio	Time to Lift Off
Power Available	Rate of Climb
Power Required	Time to Altitude
Excess Power	% Battery Used for Take off
Thrust	% Battery Used for Ascent
Take off Distance	% Battery Used for Cruise
Acceleration	

The following items must be known:

Motor Amps @ RPM

RPM

Propeller Efficiency as Function of Advance Ratio

Gear Power

Equations found in Anderson's Intro to Flight and Nelson's Flight Mechanics were used for the analysis which follows.

Advance ratio = velocity/(RPM * prop diameter)

Efficiency= Function of Advance ratio

Power Available= Gear Power * Efficiency

Power Required = Q*Surface Area *Cd,o*Velocity +

Weight^2/(Q*S*pi*Oswald eff*AR)

Excess Power = Power Available - Power Required

Thrust = [(2/efficiency-1)^2-1]*rho*forward velocity^2*prop area/2

T.O. distance = Xgr = 1.44*W/[g*rho*Clmax*(To/W - friction coeff.)]

Acceleration = 1/2*velocity/Xgr

Time to T.O. = (X gr/ acc)^.5

Rate of Climb = Excess power/weight

Time to Altitude = Desired Flight Altitude/ R of C

% bat T.O = (Time to T.O * RPM/3600)/500 mah

% bat asc = (Time to Alt.*RPM/3600)/500mah

% bat. cruise= 1 - (% bat T.O + % bat. asc)

An example of the data produced by this analysis for the FX/90 at 24 ft/s, 2.8 lbs and with a total surface area of 4.4 sq. ft follows.

ASTRO 05 W/ 7 500 mah   ZINGER 10-6

Ascent Velocity (m/s)	Cruise Velocity (m/s	Motor Amps	RPM	Advance Ratio	Efficiency	Gear Power	Power Avail
7.32	7.01	1.8	9028	0.19153011	0.47691535660188	0	0
		6	8489	0.20369111	0.49628849771845	34.9	17.32047
		7	8357	0.20690844	0.50130388963759	42.5	21.30542
		8	8224	0.2102546	0.50647164337318	49.9	25.27294
		9	8089	0.21376361	0.51183802226481	57	29.17477
weight (N)		10	7953	0.21741907	0.51737103867777	63.8	33.00827
13.34		11	7816	0.22123002	0.52307740415442	70.4	36.82465
		12	7678	0.22520629	0.52896411848497	76.7	40.57155
		13	7538	0.22938894	0.53508271095647	82.7	44.25134
		14	7397	0.23376151	0.54139873898928	88.3	47.80551
		15	7254	0.23836971	0.54796670559896	93.7	51.34448
		16	7111	0.24316325	0.55470306541931	98.9	54.86013
		17	6965	0.24826042	0.56175962959338	103.7	58.25447
		18	6818	0.25361306	0.56905244129254	108.1	61.51457
		19	6670	0.25924046	0.57659076041994	112.3	64.75114
		20	6521	0.26516391	0.5843839985216	116.2	67.90542
		21	6370	0.27144959	0.59249608583726	119.7	70.92178
		22	6217	0.27812994	0.60094094960567	122.9	73.85564
		23	6063	0.28519443	0.60967513593801	125.8	76.69713
		24	5907	0.29272623	0.61876688653573	128.4	79.44967
		25	5750	0.30071893	0.6281689038057	130.6	82.03886
		26	5591	0.30927095	0.63795119269592	132.5	84.52853

ASTRO 05 W/ 7 500 mah ZINGER 10-6

Power req. (W)	Excess Power	Thrust (N)	Take Off Dist.	Acceleration (m/s2	Time to L.O. (sec)	Rate of Climb	Time to Alt.(sec)
6.727588055	-6.727588	19.5267	- - - - - - - - - -	0	- - - - - - - - - - - -	0	- - - - - - - - - - -
6.727588055	10.592881	17.80261	2.31436407	11.57605249282	0.4471318389	0.794069	7.67943335938
6.727588055	14.577827	17.38997	2.36649089	11.32106616791	0.4572026664	1.0927906	5.58020880482
6.727588055	18.545347	16.97816	2.4209088	11.06658789225	0.4677161279	1.3902059	4.3864005469
6.727588055	22.447179	16.56428	2.47818155	10.81083021132	0.4787811423	1.6826971	3.62394398089
6.727588055	26.280684	16.15161	2.53805054	10.55581818788	0.4903477444	1.9700663	3.09532732643
6.727588055	30.097061	15.74031	2.60066877	10.30165788827	0.502445499	2.2561515	2.70283266087
6.727588055	33.84396	15.33057	2.66619978	10.04845930382	0.515105996	2.5370285	2.40359935428
6.727588055	37.523752	14.91966	2.7353217	9.794533482742	0.5284602526	2.812875	2.1678887467
6.727588055	41.077921	14.51074	2.80775862	9.541845859219	0.5424549626	3.0793044	1.98031737772
6.727588055	44.616892	14.10119	2.88425917	9.288763060448	0.5572347582	3.3445946	1.82324038902
6.727588055	48.132545	13.69694	2.96397075	9.038955610759	0.5726349217	3.6081368	1.69006895036
6.727588055	51.526886	13.28978	3.04883524	8.787355796008	0.5890306206	3.8625851	1.57873543409
6.727588055	54.786981	12.88566	3.13801483	8.537626968251	0.6062599898	4.1069701	1.48479289678
6.727588055	58.023554	12.48481	3.23177941	8.289922228581	0.6243751745	4.3495918	1.40197064667
6.727588055	61.177833	12.0875	3.33041524	8.044402289948	0.6434314759	4.5860444	1.32968620461
6.727588055	64.194193	11.69139	3.43493268	7.799628847984	0.6636240953	4.8121584	1.26720682459
6.727588055	67.128055	11.29692	3.54574902	7.55586473269	0.6850336555	5.0320881	1.21182299148
6.727588055	69.969544	10.90704	3.66253226	7.314938977144	0.7075960106	5.2450933	1.16261040585
6.727588055	72.72208	10.51963	3.7864537	7.075538780407	0.7315374559	5.4514303	1.11860551573
6.727588055	75.311271	10.1376	3.91714735	6.839466996703	0.7567872819	5.6455225	1.08014801975
6.727588055	77.800945	9.758999	4.05588744	6.605508750713	0.7835916708	5.8321548	1.04558267286

% Batt. Used for Take off	% Batt. Used for Ascent	% Batt. used for Cruising
0	0	0
0.0014904394629911	0.02559811119794	0.97291144933907
0.0017780103692739	0.02170081201874	0.97652117761198
0.002078738346326	0.01949511354178	0.97842614811189
0.00239390571173	0.01811971990443	0.97948637438384
0.0027241541353248	0.01719626292462	0.98007958294006
0.0030705002717193	0.01651731070529	0.98041218902299
0.0034340399735497	0.01602399569523	0.98054196433122
0.0038166573798128	0.01565697428169	0.9805263683385
0.0042190941532198	0.0154024684934	0.98037843735338
0.004643622985287	0.01519366990852	0.98016270710619
0.0050900881927598	0.01502283511427	0.97988707669297
0.0055630669726879	0.0149102790997	0.97952665392761
0.0060625998975282	0.01484792896782	0.97908947113465
0.0065906268422299	0.01479857904817	0.9786107941096
0.0071492386212923	0.01477429116229	0.97807647021642
0.0077422811123813	0.01478407962017	0.97747363926745
0.0083726335672023	0.01481116989591	0.97681619653689
0.0090415045803705	0.01485557740805	0.97610291801158
0.0097538327448525	0.0149147402097	0.97533142704545
0.010510934471252	0.01500205582989	0.97448700969885
0.011318546355628	0.01510286083019	0.97357859281419

Appendix B

Weight Estimate for Foam Wing

$area_{cross\ section} = 6.318\ in.^2$ span = 72.0 in.

$Weight_{total} = Weight_{wing} + Weight_{joint}$

$Weight_{wing} = Volume_{wing}\ X\ density_{foam}$

$Volume_{wing} = area_{cross\ section}\ X\ span$

$Volume_{wing} = 454.90\ in.^3$

$density_{foam} = .018\ ounces/in.^3$

$Weight_{wing} = 8.19$ ounces

$Weight_{joint} = 1.5$ ounces (estimated)

$Weight_{total} = 9.69$ ounces

Stress Analysis for Foam Wing

$stress_{max} = (M_y{*}Y_{max})/I$

$I = 1/12\ b\ h^3$ b=7.02 in. h=0.9 in.

$I = 1/12\ (7.02\ in.)(0.9 in.)^3$

$I = 0.4265\ in.^4$

$Y_{max} = 1/2 *$ maximum thickness

maximum thickness = (9in.)*(.135) (13.5% of chor $Y_{max} =$

0.608 in.

$M_y = Weight_{aircraft} * 1/2 * (1/4 * span)$

$Weight_{aircraft} = 44$ ounces span = 72 in.

$M_y = 396$ in.-ounces

$Stress_{max} = (396$ in.-ounces$) * (0.608 in.) / (0.4265\ in.^4)$

$Stress_{max} = 35.28\ lbs./in.^2$

Appendix C

The following data was submitted after the completion of the final proposal. It is related to the weights of the completed prototype.

Wings (inc. joint)	10.54 oz
Fuselage, tail, motor mount and control rods	11.79 oz
Battery pack	5.46 oz
Speed controller	3.23 oz
Receiver	0.95 oz
Radio battery	2.0 oz
Servos (2)	1.2 oz
Total	2.74 lbs
Cart weight	4.82 oz

This should be considered preliminary data.

The final c.g. position for the prototype was not provided.

References

1) Nelson, Robert C., Atmospheric Flight Mechanics, University of Notre Dame, IN 1989.

2) Jenson, Daniel T., Drag Prediction Methodology for Low Reynold's Number Flight Vehicles, University of Notre Dame, IN 1990.

3) Beron-Rawdon, Blaine, "Dihedral", Model Aviation, Aug-Nov 1988.