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FINAL DESIGN PROPOSAL

THE PENGUIN

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

May 1990

Department of Aerospace and Mechanical Engineering
University of Notre Dame
Notre Dame, IN 46556

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The Penguin

A low Reynolds number powered glider for station keeping missions

Designed and produced exclusively by the Aerospace Design Group "C"

J.K. Costello, D.W. Greene, T.T. Lee, P.T. Matier,
T.R. McCarthy, R.J. McGuire, and M.J. Schuette.

May 1990

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Executive Summary

The Penguin is a low Reynolds number (approximately 100,000) remotely piloted vehicle (RPV.) It has been designed to fly three laps indoors around two pylons in a figure-eight course while maximizing loiter time. The Penguin's low Reynold's number mission is an important one currently being studied for possible future flights in the atmospheres of other planets and for specialized military missions.

Although the Penguin's mission seemed quite simple at first, the challenges of such low Reynolds number flight have proven to be quite unique. In addition to the constraint of low Reynolds number flight, the aircraft had to be robust in its control, highly durable, and it had to carry a small instrument package.

The Penguin's flight plan begins with takeoff on a runway of 150 feet. It will actually lift off in 51 feet, and the remaining runway distance will be used to climb to the cruise altitude of 15 feet. The aircraft will then begin it's three laps around the pylons. After completing the last lap, the Penguin will land on the same runway and come to a stop in approximately 30 feet.

Aerodynamically, the Penguin is similar to standard taildragger-type sailplane designs. The 7 foot span rectangular wing is mounted on the top of the fuselage and is canted at a 3° dihedral. It uses the Wortmann FX63-137 airfoil. The long fuselage is rectangular and is highly tapered aft of the wing. The empennage has standard horizontal and vertical tail surfaces.

Supporting the structure of the Penguin are two box beams for the fuselage and wing, and two simple beams in each of the horizontal and vertical tails. The box beam in the wing is located at the maximum thickness of the wing, while the simple beams in the empennage are located at the leading edge and the trailing edge (just prior to the control surfaces). The fuselage box beam runs the entire length of the aircraft. The forward section of the fuselage is much stronger than the aft since it supports the engine and the avionics as well as the load from the wings.

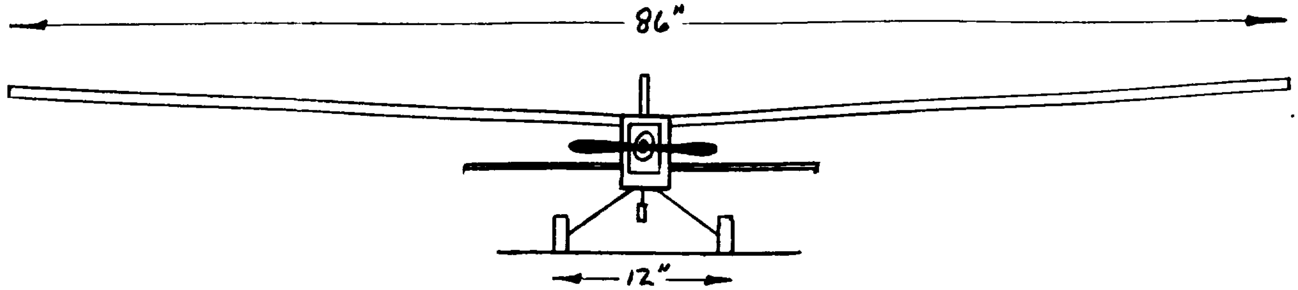
The Penguin is driven by an ASTRO 15 electric motor that provides more power than the RPV will need. The excess power may prove to be useful in a stall situation that may arise since the Penguin will cruise at a velocity close to the stall velocity ($V_{cruise} = 1.3\ V_{stall}$.) A two-blade, 10 inch diameter propeller provides the thrust.

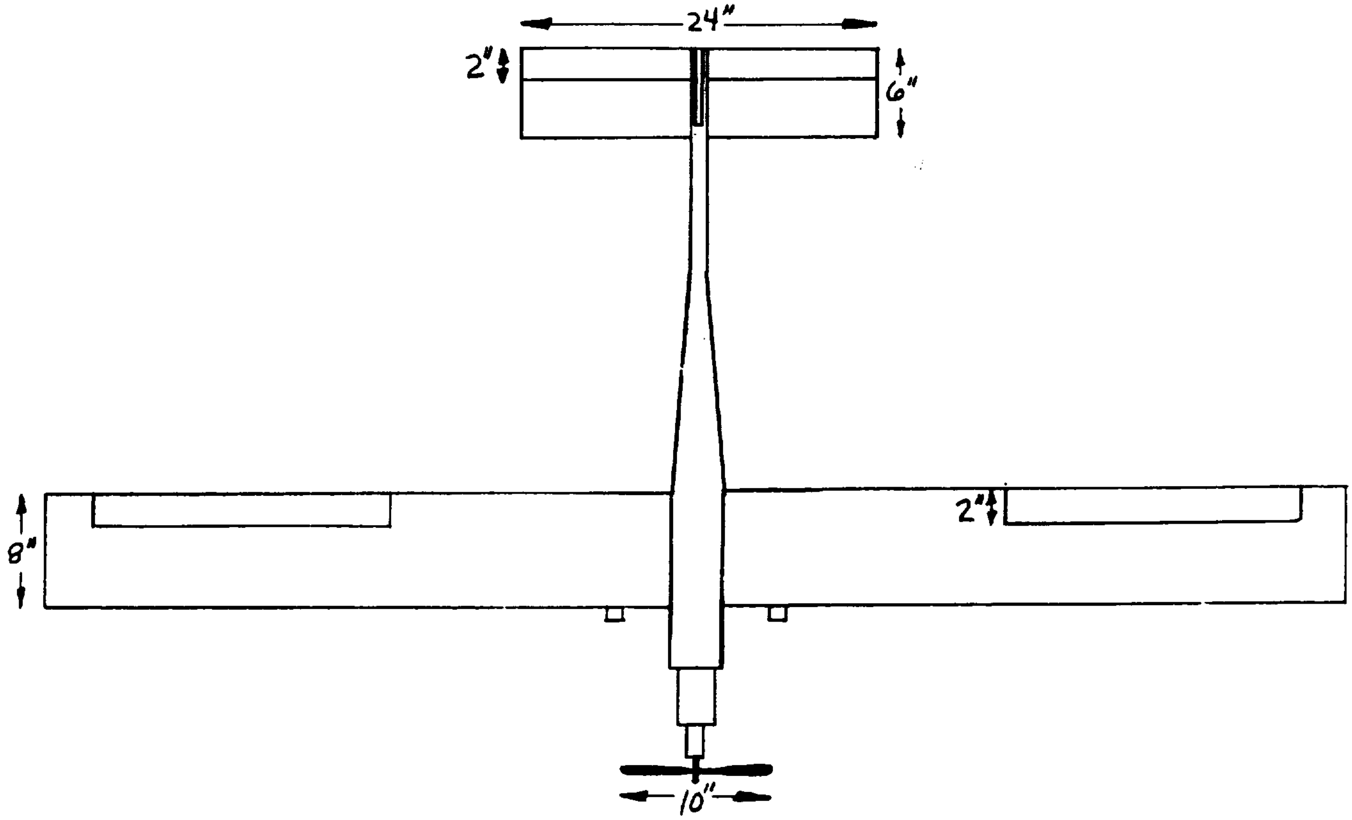
Since the RPV had to be highly maneuverable, it makes use of large rudder, aileron, and elevator surfaces. It's large horizontal and vertical tail surfaces are located far aft of the wing in order to provide static stability and are placed in the wash of the propeller for added effectiveness. The dihedral of the wing provides roll static stability.

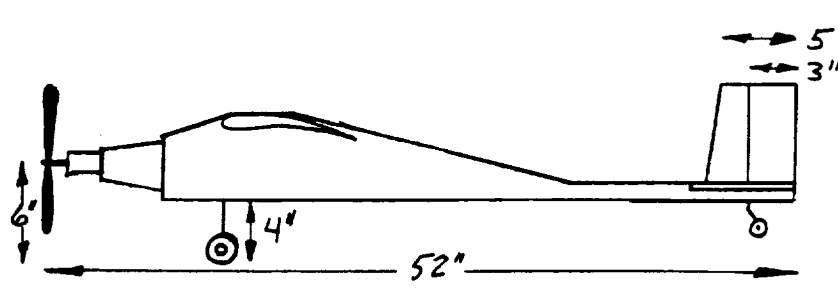
Some problems that may arise include the possible early drain of the batteries due to added power needed to maintain altitude in the turns, the interference effects of the propeller and fuselage wakes, and the possibility of structural failure due to the inexperience of the manufacturers.

The Penguin meets the challenges of the project and provides a test specimen for future experiments in the low Reynolds number regime. The design provides for a highly-maneuverable RPV capable of sustained flight at low speeds.

3-View Drawing of the Penguin







SPECIFICATIONS SUMMARY

RPV DESIGN CHARACTERISTICS:

Weight = 3.125 (lbs)
Wing Chord = 8.0 (in)
Wing Span = 7.0 (ft)
Wing Area = 4.67 (ft^2)
Aspect Ratio = 10.5
E = .78
Dihedral = 3 (deg)
Wing Mount Angle = 6 (deg)
Cl max = 1.1
Cl takeoff = 1.0
Cl cruise = 0.9
Cdo = .016
Engine = Astro 15
Propeller = Zinger 10-4
Propeller Efficiency = .72
Fuselage Length = 3.5 (ft)
Fuselage Width = .208 (ft)
Max. Load Factor = 2.0
Vertical Tail Area = .42 (ft^2)
Horizontal Tail Area = 1.04 (ft^2)
Battery Pack Voltage = 15.6 (v)
Battery Pack Capacity = 270 (mah)
Airfoil = Wortmann FX-63-137
Horiz. & Vert. Stab. Airfoil = Flat Plate

RPV PERFORMANCE DATA:

(Environment -- Standard Sea-level Conditions.):

Stall Speed = 22.6 (ft/s)
Max. Speed = 56.1 (ft/s)
Cruise Speed = 25 (ft/s)
Cruise Reynolds Number = 106000
Cruise Altitude = 15 (ft)
Takeoff Distance = 51.2 (ft)
Landing Distance = 96 (ft)
Range = 2609 (ft)
Endurance @ 25 (ft/s) = 105.3 (s)

Review of Design Requirements
Mission Scoping Study

The following request for proposals provided Group C with the design specifications for a remotely piloted vehicle (RPV).

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 number 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 than 10^5. Particular applications are low speed flight at very high altitudes, low altitude flight of very small aircraft and flight in other planets' 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 is 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 conditions. 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 $2x10^5$ and as close to $1x10^5$ as possible. The greatest measure of merit is associated with achieving the lowest mean 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 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.
5. Evaluate the feasibility of the extension of the aircraft developed under this project to high altitude station keeping application for atmospheric sampling.

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 can 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 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. The complete aircraft must be able to be disassembled for transportation and storage and fit within a storage container no larger than 2'x2'x4'.
f. Safety considerations for systems operations are critical. A complete safety assessment for the system is required.

Low Reynolds number, station keeping flight is the mission objective for this design project. According to the "Request for Proposals", an RPV was to be designed to maintain controlled and level flight around a predetermined course at low Reynolds numbers. The highest priority mission objective for this flight was to obtain Reynolds numbers between 100,000 and 200,000 while maximizing the flight endurance.

In order to successfully approach the mission objective, primary design requirements were established. Principally, the constraints imposed by the confined environment of the Loftus Center, by minimum endurance and range requirements, and by the necessity for ease of installation and assembly had to be addressed.

Evaluation of these mission requirements enabled the group to categorize the primary constraints. The ability to takeoff and land in a 150 ft. strip, to establish effective stability and control for all flight speeds, and to execute low speed figure eights while maintaining altitude, were of extreme importance to satisfy the confined environment constraints. The ability to climb to cruising altitude in reasonable time and to complete three figure eight patterns around two pylons placed 150 ft. apart were main considerations to satisfy the endurance requirements. Ease of installation of the instrument package and compactness for transportation were necessary to satisfy assembly constraints.

General guidelines enabled minimum performance limits for the RPV's capabilities to be determined. The group assembled concrete design objectives and a concrete mission flight plan to insure that these limits were met. The mission of the Penguin was to simulate low speed flight at high altitudes, low altitude flight of very small aircraft, or flight in another planet's atmosphere. In order to approximate these conditions, the Penguin needed to fly at low Reynolds numbers between the ranges of 100,000, and 200,000.

Target objectives for the Penguin have been established from the Reynolds number requirements. The Penguin will attempt to fly as close to the Reynolds number of 100,000 as possible. A realistic cruising velocity of between 25 ft/s to 30 ft/s requires the chord to be 8-10 inches. The cruising speed between 25 ft/s and 30 ft/s ideally represents the lowest speed that can be maintained to successfully negotiate the course. The primary goal of the Penguin's mission is to maintain focus on these objectives.

The group feels that the mission requirements have been adequately addressed and the mission is possible. Engine and propeller analyses have demonstrated the power capabilities to meet the requirements associated with takeoff and climb constraints. Aerodynamics, stability, and control analyses have insured Cl characteristics and sufficient rudder and aileron control to maintain altitude in the turns and provide maneuverability over the closed course. Structural and weight analyses have enabled fabrication of a compact and easily disassembled RPV. And performance analysis has led to adaptation of lightweight, durable batteries that will allow for the Penguin to complete the three lap mission given. None of the requirements have been sacrificed or altered by the group and the Penguin will fly the mission as was stated by the Request for Proposals.

A summary table of the design requirements and objectives is provided.

Table DR1: Summary of the Requirements and Objectives

	Requirement	**Objective**
Re	100000-200000	100000
Takeoff Distance	150 ft.	75 ft
Propulsion System	Non air breathing/ Does not emit mass	Electric
Controls System	Maneuverable/ Controllable	Aileron, Rudder, and Elevator control
Endurance	Maximize loiter time.	110 s
Flight Path	Fly closed course	50 ft turns. Figure 8's consisting of 942 ft of turns and 1081 ft of straight away
Storage	Fit into compact box.	Easily disassembled.

Detailed Flight Plan

Our target takeoff distance is 75 ft. and is based upon another design objective which is the desire to initiate the first turn at cruising speed and altitude. The takeoff strip is 150 ft. long and, therefore, with the 75 ft. desired takeoff strip, a factor of safety of two is produced. The extra distance provided by the target takeoff will allow the RPV to climb to cruising altitude (15 ft.) and attain cruising speed before executing the turn. An additional margin of safety for the turn would be inherent in the cruise speed and altitude in case of accidental altitude loss or stall.

In order to achieve the optimistic takeoff distance our power plant must be capable of accelerating the RPV at 2.67 ft/s^2. Achieving cruising altitude at the first turn will require a rate of climb of approximately 4.7 ft/s. Based on the acceleration (2.67 ft/s^2), takeoff distance (75 ft.), takeoff velocity (20 ft/s), and cruising altitude (15 ft.), the first maneuver should be encountered within 3 seconds after takeoff. Pilot reaction will dictate much of the success of the mission.

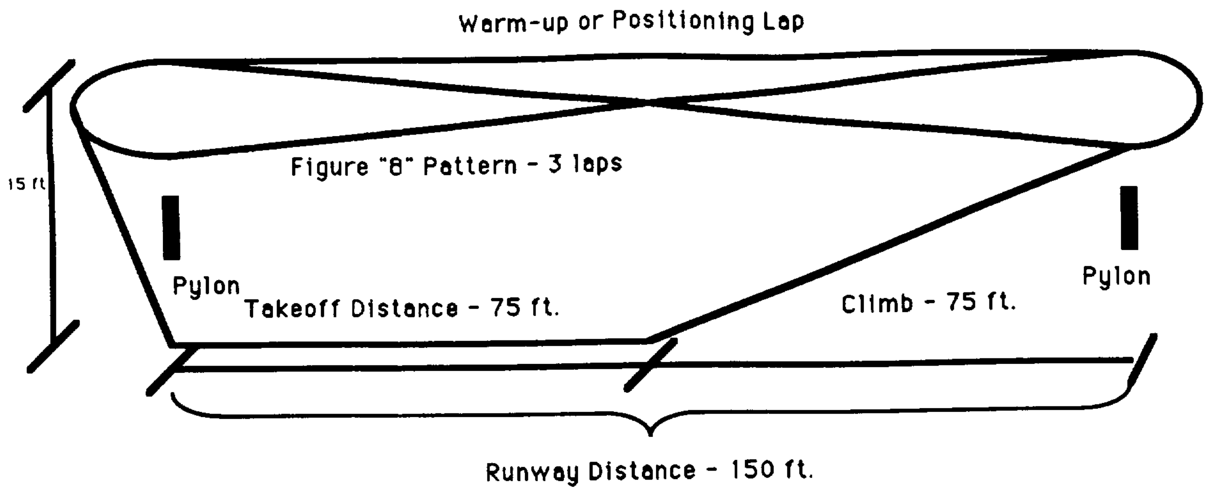
Group C hopes to achieve a total time from initiation of the ground roll phase to the first turn of less than 10 seconds. Such a time would be ample for the pilot to gain control and plan for the necessary maneuvers. The RPV, upon reaching cruising, will initiate a 50 ft. radius turn. The Penguin will navigate three figure eights, proceed through a positioning lap to set it up for landing, and land. The "figure eights" consist of 942 ft of turning flight and 1081 ft of straight flight. Therefore, static and dynamic stability will be necessary so the pilot doesn't have to struggle to keep the RPV airborne.

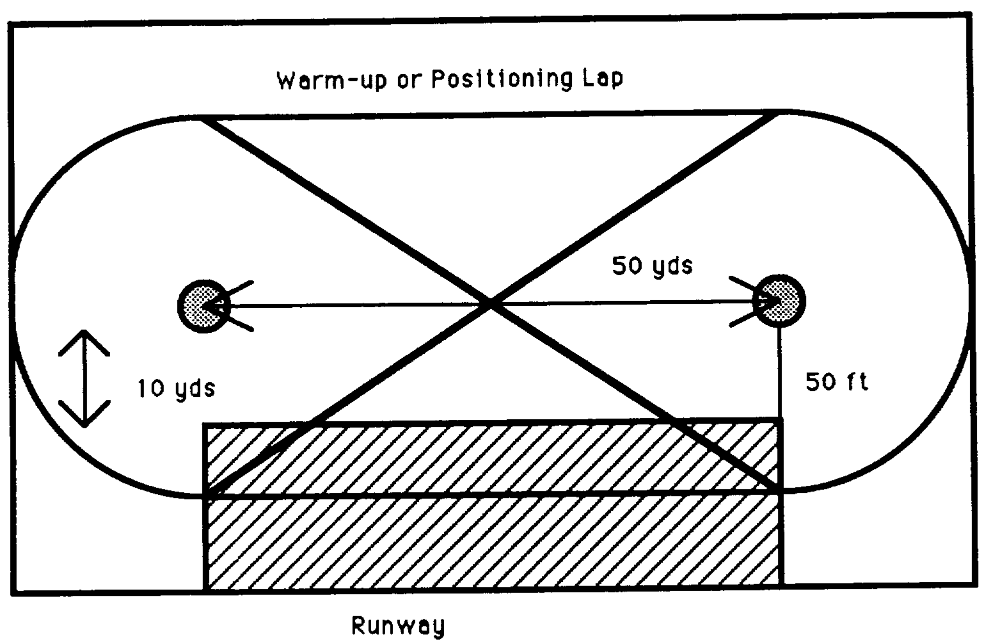
Of particular importance in this mission is turning flight. Turning flight represents almost half of the mission (about 46%) and, therefore, is extremely important. Critical to the success of the mission is avoiding stall in the turns. Indeed, attention must be taken to see that the velocity over the inside wing remains above stall speed.

Endurance and range requirements are approximated at 110 seconds and 2720 ft. respectively. These values enable us to fly the particular mission and include the warm-up or positioning lap to set us up for landing.

Ground handling at landing and takeoff is achieved by traditionally placed, tail dragger landing gear. In addition, the tail wheel will be connected to the rudder for better ground handling qualities. A comprehensive flight plan is seen in the following figures.

Within less than half of the allotted runway the Penguin will takeoff. Immediately after takeoff the penguin will climb to an altitude of 15 ft. This will allow the Penguin to address the first turn at the appropriate altitude for turn. After reaching the 15 ft altitude, the airplane will be trimmed in order to initiate the 50 ft radius turn at steady level flight. The pilot then can control the airplane in such a manner as to negotiate a successful turn. After turning the Penguin will be trimmed once again for the steady portion of the flight. The Penguin will initiate a second 50 ft radius turn after which it will complete the figure eight in steady level flight. The Penguin will perform three figure eights in succession. The flight path taken by the Penguin requires it to complete a positioning lap for approach and landing. The positioning lap is simply an oval which will begin immediately following the final figure eight. About 3/4 of a lap will be required to position the Penguin at the start of the runway. The Penguin will begin its descent after coming out of the second turn and will try to land as close to the beginning of the runway as possible. It will use ground friction and its rudder/wheel steering capabilities to eventually come to a stop.


Warm-up or Positioning Lap
Figure "8" Pattern - 3 laps
15 ft
Pylon
Takeoff Distance - 75 ft.
Climb - 75 ft.
Pylon
Runway Distance - 150 ft.


Warm-up or Positioning Lap
50 yds
10 yds
50 ft
Runway

Concept Selection

Introduction

Following the mission scoping study and the formulation of the design requirements and objectives (DRO), development proceeded with a definition of group concensus design objectives (see table CS-1). Articulation of these ideas is central to the cocept selestion phase.

Table CS-1: Group Design Objectives

Factors of Critical Importance	Of Little or No Concern
1) Simplicity of Analysis and Construction	1) Size
2) Reynolds Number of $1x10^5$ to $2x10^5$	2) Noise
3) Cost	3) Aesthetics
4) Weight	4) Innovation/Originality

Although many of these goals are specific in focus, some are all-inclusive in nature (simplicity, for example), allowing special application to the choice of concepts; but before one can select, a suitable pool of views must be collected.

In accordance with the DRO and the mission evaluation, a vague picture of the final aircraft may be arrived at. Initially, and at most, the aircraft may be described as "a lightweight, slow moving one, possibly with large wings." With this in mind, the Delphi method of participation was employed for this configurational phase: each group member was charged with the conception of one individual design, completely free from interpersonal bias (the most famous product of the Delphi technique, the fastest known airplane to date, was retired this year--the SR-71). Seven separate designs fell into two broad categories--the biplane, and the powered glider. From these, a final concept was arrived at. What follows are the design descriptions, evaluations, and the evolution of the Penguin aircraft.

The Biplane

The biplane designs invariably involved two moderate aspect ratio wings situated over various fuselage locations and supported by several schemes.

Two aspects of the biplane are attractive. The first relates strongly and directly to the very novelty of reviving the lost glory of the "barnstorming" era. Having a biplane would be a great "gimmick." The second reason is an alleged reduction in the induced drag. That is, having two wings would lower the Cl load on each wing, thus mitigating the induced drag term in the drag polar. The obvious tradeoffs are a possible increase in wetted area and wing weight.

The extra rigging and external structure promises to add some parasite drag. Also, it is true that a reduced Cl per wing will destroy some induced drag; but recall that there are now two wings. The sum of the induced drag terms from each wing might well equal or exceed the amount of induced drag spared through the biplane configuration. However, the primary cause of disqualification for the biplane relates to ease of analysis. Quite simply, two many additional design variables had to be manipulated to justify a particular design: fuselage orientation, external support, and aerodynamic interference, above the usual qualities attributed to the single wing--surface area, aspect ratio, dihedral, washout, taper ratio, and angle of incidence. Recognizing the lack of experience with biplane analysis and time constraints, the biplane alternative, though glamorous and exciting, was summarily rejected.

The Powered Glider

A more realistic option was realized through a conventional, high-aspect ratio, sailplane or powered glider. The original idea employed a complex wing geometry which enjoyed almost all techniques of lift optimization (taper and twist), save the use of high-lift devices, and drag reduction (aspect ratio and winglets).

A high aspect ratio (AR>7) will reduce the induced drag, as will winglets. The disadvantage of a high aspect ratio, is of course, the increased size and weight of the wing. The same may be said about the use of winglets. Both recourses will result in a small increase in form drag. Foremost among advantages however, is the absence of any analytical difficulties. The mono-wing has been well studied by group personnel and is free of any of the stated biplane effects.

The chief source of discontent with this particular version of the powered glider resides with the technical difficulties encountered in manufacturing the wing. Specifically, the varying size of the wing sections together with a slight linear twist angle of 3 to 5 degrees is extremely tedious and challenging to engineer, much less without blueprints. In a similar vein, no group member possessed the requisite amount of experience warranted by a design of this variety. Citing simplicity again, this version of the powered glider was rejected as well.

The Penguin

A modification of the powered glider wing to zero taper and twist (keeping a simple dihedral) finally satisfied group design objectives. The Penguin still retains many sailplane characteristics, preserving the high aspect ratio sailplane-like flavor. The Penguin is hence a simpler, conservative derivative of the previous design. Several other smaller design concepts were also assessed for the penguin:

A T-tail empennage pattern was submitted but resolved against because of severe connection problems with servomotor control rods and actuation points, drastically limiting the range of elevator travel. A twin-empennage boom with a centrally mounted pusher propulsion system was also vetoed on the grounds of simplicity.

Choice of landing gear orientation was dominated by takeoff discussions, but was surpassed by the necessity for positive steering control during taxi maneuvers. The Penguin's landing gear was thus arrayed in a tail-dragger orientation.

Maneuverability and flight path requirements (recall the required figure-eight flight path) in the indoor Loftus Sports Center dictated the presence of a full set of control surfaces--elevator, rudder, and ailerons.

Finally, structural weight may be minimized by ridding the fuselage of any unwanted space. This calls for a close fit of avionics and propulsion equipment in the forward position (a narrow forward fuselage), and the gutting of the aft fuselage section towards the empennage (essentially reducing it to a connecting beam). Such a decision also entails the added benefit of reducing the parasite drag.

Summary

It is recognizable that the major criteria in the configuration conception phase were simplicity of analysis and construction. Adjudication of various parameters was handled and verified by rules of thumb and qualitative discussion. Little in-depth quantitative study was carried out; such concentrated study is exclusively the province of the parametric trade studies. Based on the factors as stated in table CS-1, the biplane, powered glider, and the Penguin measure up as follows:

Table CS-2: Suitability of Studied Concepts

Criterion	Biplane	Powered Glider	The Penguin
Simplicity of Analysis	No	Yes	Yes
Ease of Construction	No	No	Yes
Innovation/Originality	No	No	No
Aesthetics	Yes	No	No

The Penguin was eventually selected as the mission concept, embodying the time-honored yet oft neglected principle of engineering known as simplicity.

The Penguin then, is a high aspect ratio powered glider mounting a high wing with dihedral but with no twist or taper. The highly conventional design features standard placement of propulsion system, horizontal and vertical stabilizers. It will operate with aileron, rudder and elevator, and has a tail-dragging landing gear installation.

Performance

Introduction

The entire design of the Penguin RPV was centered around performance goals which were based upon the requirements of the mission. These goals and requirements are discussed in the "Design Requirements and Mission Definition" sections. Some of the important goals are:

* Cruise velocity of 25 ft/s
* Reynolds number of 100,000
* Takeoff distance of 75 ft
* Rate of Climb of 4.9 ft/s
* Turn Radius of 50 ft
* Endurance of 83 s

Table PF#1 gives the expected values for these and other performance characteristics of the Penguin. The following sections will discuss the Penguin's design and performance characteristics in important areas, especially the ones given above. Particular qualitative attention will be given to the Penguin's takeoff, cruise, and landing phases as such design abilities are most important for successful prototype flying. Rate of climb, turn radius, and endurance are less important with regards to attaining flight, yet are of paramount importance for successful completion of the mission. These three characteristics will be examined from a more quantitative point of view so that the feasibility of successful mission completion can be studied. The discussions will also involve a comparison of the expected performance with the performance goals.

Cruise

Since the majority of the flight mission is to be performed in the cruise condition, cruising performance was the highest design priority. The flight mission itself is to fly a figure eight course around two poles which are 150 ft. apart (at opposite ends of a runway). The two requirements which

TABLE PF#1

RPV PERFORMANCE:

(Takeoff & Climb @ 8500 RPM; Cruise @ 4500 RPM):

(Standard Sea Level Conditions) (RPV Weight = 3.125 (lbs))

Takeoff Distance = 51.2 (ft)

Takeoff Time = 2.98 (s)

Takeoff Velocity = 23.7 (ft/s)

Landing Distance = 96 (ft)

Max. Current Draw at Takeoff (amps) = 13.7

Rate of Climb = 10.4 (ft/s)

Cruise Altitude = 15 (ft)

Cruise Velocity = 25.0 (ft/s)

Cruise Reynold's Number = 106000

Current Draw at Cruise (amps) = 9.1

Max. Bank Angle @ Cruise Vel. (deg) = 35.1

Turn Radius @ Cruise Vel. (ft) = 27.7

Max. Velocity = 56.1 (ft/s)

Stall Speed = 22.6 (ft/s)

Maximum Bank Angle (deg) = 60.0

Min. Turn Radius (ft) = 18.33

Endurance = 105.3 (s)

Range = 2609 (ft)

were the driving force behind the design are listed in order of importance as follows:

* to fly at a Reynolds number as close to 100,000 as possible.
* to maximize flying time for three laps of the course given above.

Other practical considerations, such as the problems associated with handling large wing span RPV's and problems with storage and shipping of the RPV components, also played a role in the design process.

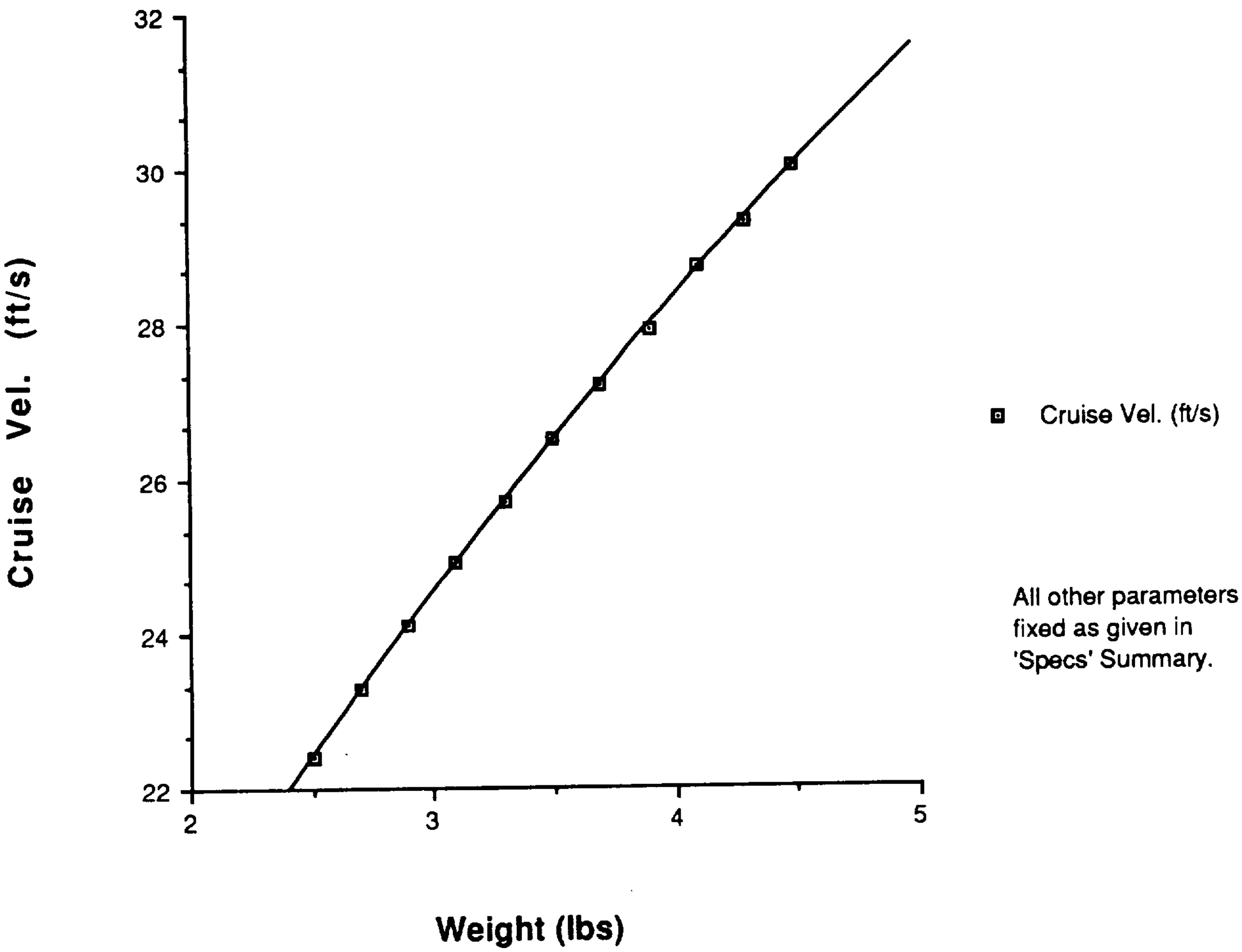
The problems posed by the design requirements arise from the fact that many of the design variables depend on each other. For instance, to achieve a smaller Reynolds number, a shorter wing chord must be used. Using a shorter chord for an RPV of fixed weight and wing span necessitates the use of a higher flight velocity to maintain lift. Higher flight speed in turn increases the Reynolds number and decreases the loiter time around a course of fixed distance. The above analysis becomes circular. For the Penguin, trial and error produced the following design characteristics which deal with the problems of low Reynolds number and high loiter time flying:

* .667 ft. chord -- small chord reduces Reynolds number.
* 7.0 ft. wing span -- large wing span increases wing area, thus producing more lift at a given speed, which allows for an RPV of fixed weight to fly slower. Flying slower decreases Reynolds number and increases loiter time.
* FX-63-137 airfoil -- High CL max (1.1) produces higher lift at lower speeds with results similar to those above (decreased Re; increased loiter).
* 3.125 lbs. weight -- lower weight means less lift is needed to fly at a given speed. This keeps wing area (span) down to a managable size.

The wing airfoil section, large wing area, and light weight of the Penguin combine to create high lift capability which allows low flight speeds at low Reynolds numbers.

Particular attention was given to see that the Penguin attained the Reynolds number and cruise velocity goals. Table PF#1 shows the cruise velocity and cruise Reynolds number for the Penguin. The Reynolds number of 106000 slightly overshot the Reynolds number goal of 100,000. The

FIGURE PF#1


32
30
28
26
24
22
2
3
4
5
Cruise Vel. (ft/s)
Weight (lbs)
Cruise Vel. (ft/s)
All other parameters fixed as given in 'Specs' Summary.

cruise velocity of 25 ft/s perfectly matches the 25 ft/s design goal. Since the weight estimation of 3.125 lbs. may not exactly match the actual prototype weight, Figure PF#1 gives an idea of the effect of excess weight upon cruise velocity. Note that in the case of a 45% weight underestimation (W=4.5 lbs.), the RPV could still fly at 30 ft/s. Such a speed would harm Reynolds number and loiter time performance somewhat, but would still allow for a successful mission.

Takeoff:

Takeoff was an important consideration in the design process of the Penguin RPV. The flight mission requirements state that a runway of 150 ft. distance must accommodate the RPV. The takeoff characteristics of the Penguin were chosen with this mission requirement in mind.

The mission poses two related problems for takeoff. First, the runway space is limited. Takeoff must be accomplished within 150 ft. Secondly, the figure eight flying mission necessitates that the RPV be able to execute a safe turn after reaching the first pole (150 ft distance). In this case, the second problem somewhat magnifies the first. The design must not only lift off the ground before 150 ft., it must achieve a safe cruising speed and altitude before reaching 150 ft. so that the first turn can be successfully accomplished without stalling the RPV.

Certain design characteristics of the Penguin deal with the takeoff problem presented above. The important items are:

* the Astro-15 engine -- This engine is overpowered for an RPV of this size. It produces approximately .3 HP at the static condition.
* the Zinger 10-4 propeller -- Matched with the Astro-15 engine, this propeller provides a thrust to weight ratio of approximately .65 at the static condition.
* Wing Area of 4.67 ft^2 -- A large wing area for a 3.125 lbs. RPV; this will give the Penguin the ability to generate the necessary takeoff lift at low speeds.
* 10° Wing angle of attack at takeoff -- Utilizes the airfoil's high CL capabilities by giving a CL of approximately 1.0 at takeoff.

The high power and static thrust of the propulsion system coupled with the high lifting power of the wing should give the Penguin the necessary capabilities to takeoff from the Loftus Center 150 ft. runway and safely enter the first turn.

The actual takeoff performance of the Penguin is given in Table PF#1. The takeoff distance of 51.2 ft. is considerably shorter than the design objective of 75 ft. Such a takeoff roll will leave nearly 100 feet of excess runway to compensate for variable pilot technique or pilot error. For the RPV to perform as expected, its actual weight must equal the predicted weight of 3.125 lbs. Figure PF#2 shows the effect upon takeoff distance should the Penguin's weight deviate from the predicted value. Note the high power, high lift characteristics of the Penguin's design allow for successful takeoff from the 150 ft runway even at weights approaching 5 lbs. This plot can be thought of as showing a degree of safety for takeoff versus weight.

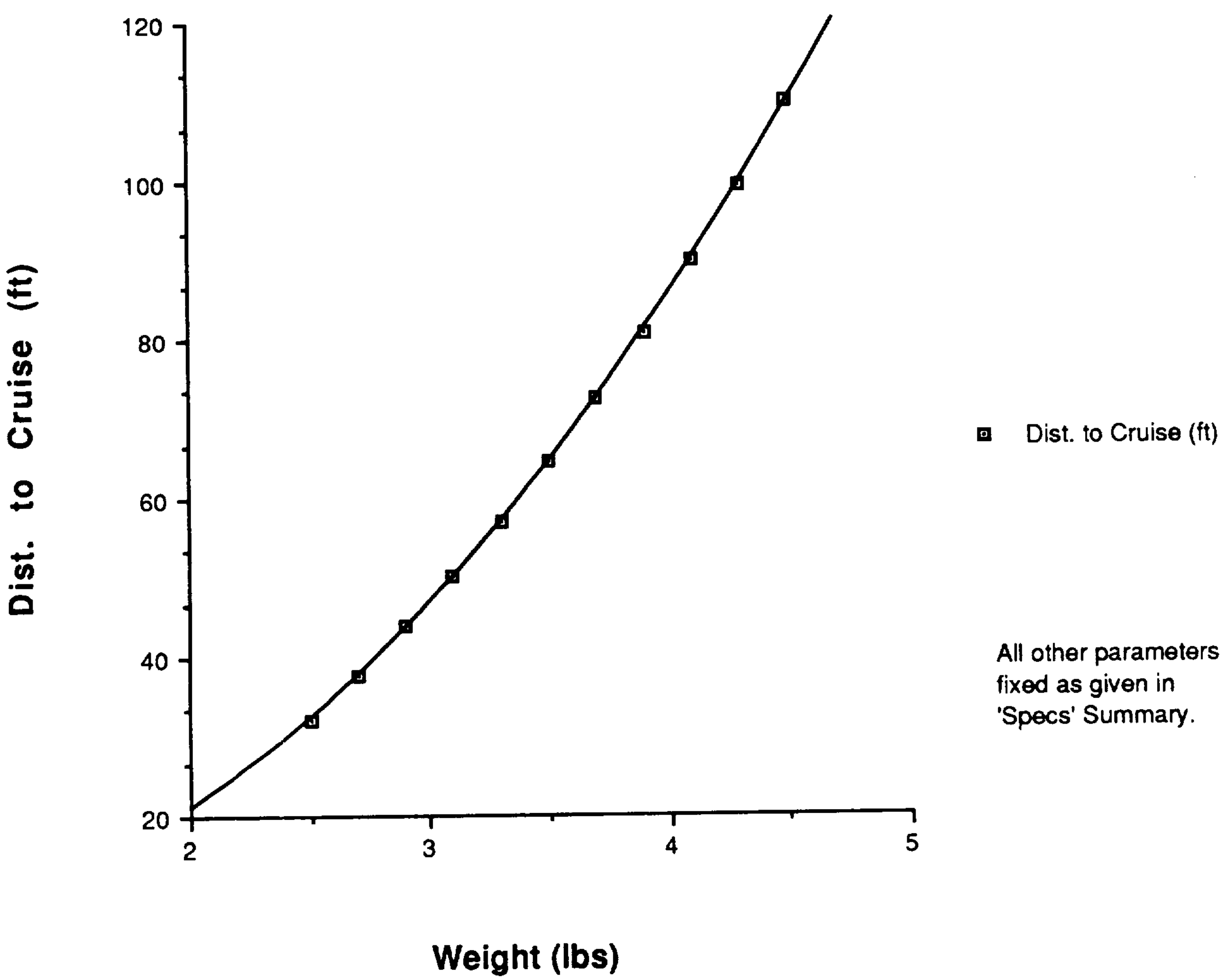
Landing

The requirements placed upon the RPV landing system by the mission also factored into the design choices. As with takeoff, the RPV must restrict its operations during landing to a 150 ft. runway. Another consideration for the RPV's landing system was the safety of the RPV during the landing procedure.

Exotic landing assistances such as a parachute were not considered as the necessary extra weight for the system could not be sacrificed and the problems with manufacture and maintenance promised to be taxing. A belly-flop with no landing gear was ruled out because it would severely damage the RPV's propeller and would also jeopardize other parts of the RPV. Two landing gear configurations were considered. The first was a tricycle gear arrangement. The second was a conventional tail dragger setup. The tail dragger was chosen because such a configuration is easier for a pilot to handle while landing. The tail dragger also minimized weight as the tail wheel did not need to be full size.

The most important design characteristics that deal with the problems of landing distance and safety are the Penguin's large wing area and high lift airfoil. These two characteristics combine to create a low stall speed which translates into low landing speed. Low landing speed is the

FIGURE PF#2



120
100
80
60
40
20
2
3
4
5
Dist. to Cruise (ft)
Weight (lbs)
Dist. to Cruise (ft)
All other parameters fixed as given in 'Specs' Summary.

primary method used to control the landing distance as the Penguin has no brakes. Since the Penguin relies solely on drag and ground friction to bring it to a stop, a lower landing speed will give shorter distances. Low landing speed also makes the pilot's task of maneuvering the RPV for landing a lot easier. Reduced strain on the pilot reduces the chance for error and the chance for damage to the RPV or the Loftus Center environment.

Table PF#1 shows the landing distance for the Penguin. The 96 ft. distance is calculated from equations of motion using only drag and ground friction as retarding forces. This calculated value corresponds to a landing factor of safety of 1.56 for a 150 foot runway.

Rate of Climb

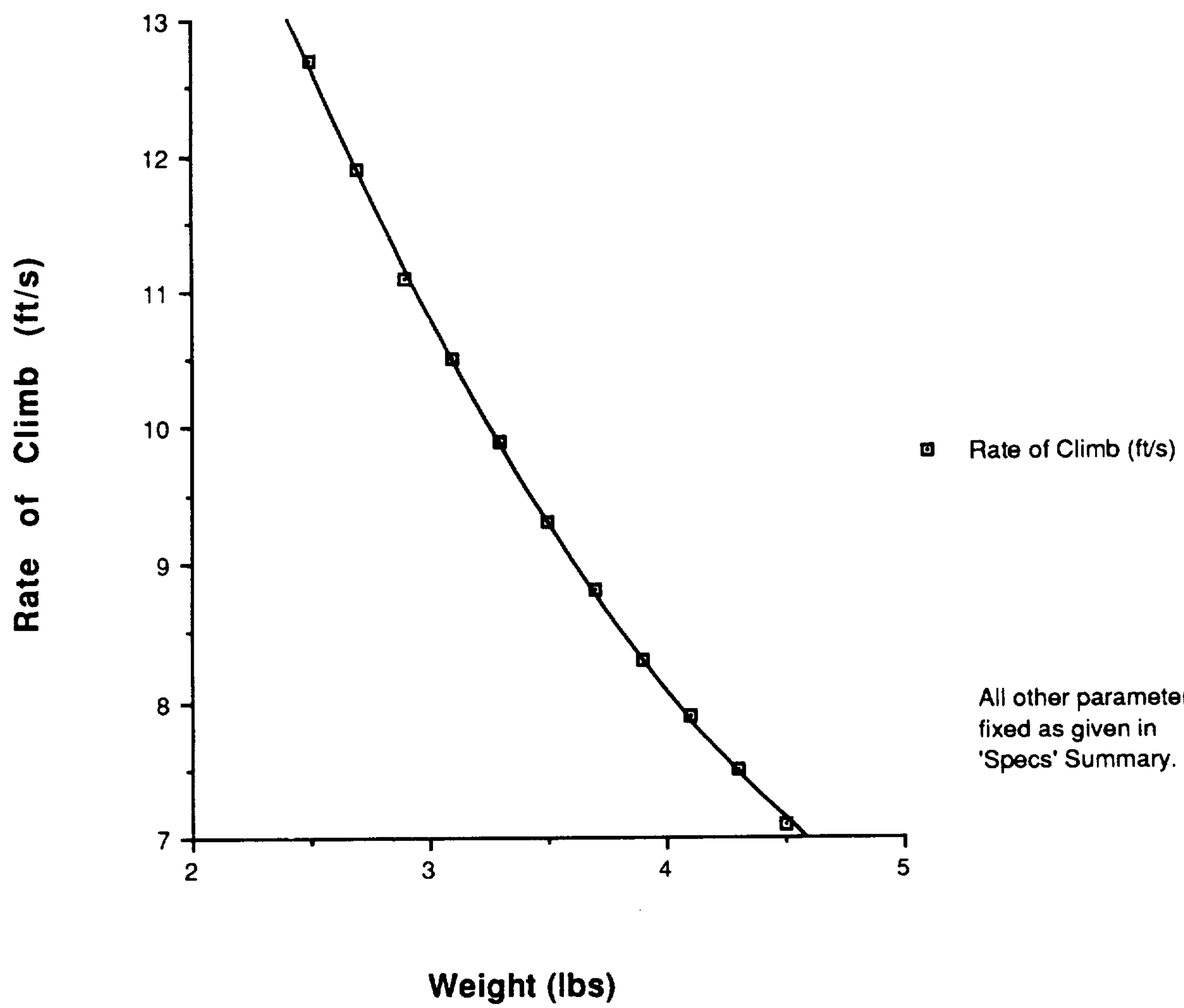
The primary importance of rate of climb lies in the ability of the Penguin to achieve cruising altitude quickly and efficiently for the sake of successful mission completion. A design goal of 4.9 ft/s was based upon the desire to have the Penguin at cruising altitude before entering the first turn of the Loftus course. Table PF#1 gives a value of 10.4 ft/s for the Penguin's rate of climb. This value is 112% better than the design goal. Thus, the Penguin has ample climb ability to achieve the desired altitude goal before the distance requirement becomes a problem.

The excellent climbing characteristic of the Penguin is mainly the result of the high power to weight ratio provided by the Astro-15 engine, Zinger 10-4 propeller propulsion system. The light weight of the Penguin also contributes to the rate of climb performance. Figure PF#3 shows how the rate of climb performance would be affected if the weight of the Penguin was to change. Note that the Penguin has enough power to maintain a rate of climb well above the design goal for weights up to 4.5 lbs. Such ability ensures the Penguin of more than adequate performance in this area.

Turn Radius

As the flight mission is to successfully navigate a figure eight course with the Penguin, turning performance will be of high importance. The main problem with turning flight at low speeds is that the inside wing senses a decreased velocity in the turn. If the cruise speed is already low (near stall speed), the decrease in velocity over the inside wing may result in the stall of that wing. Careful study of the maximum allowable bank angle

FIGURE PF#3



All other parameters fixed as given in 'Specs' Summary.

and corresponding minimum turn radius for particular cruise velocities becomes important.

Table PF#1 gives the maximum bank angle and minimum turning radius possible for a cruise velocity of 25 ft/s, while avoiding stall of the inside wing. The turn radius of 27.7 ft. is nearly twice as good as the design goal of 50 ft. As a 50 ft. turn radius should be able to successfully navigate the figure eight course, the 27.7 ft. turn radius easily allows for the flying of the course with the corresponding cruise speed of approximately 25 ft/s. Such speed also allows the pilot to make the turn at some radius between 27.7 and 50 ft., thus allowing for some degree of safety in avoiding the stall of the inside wing.

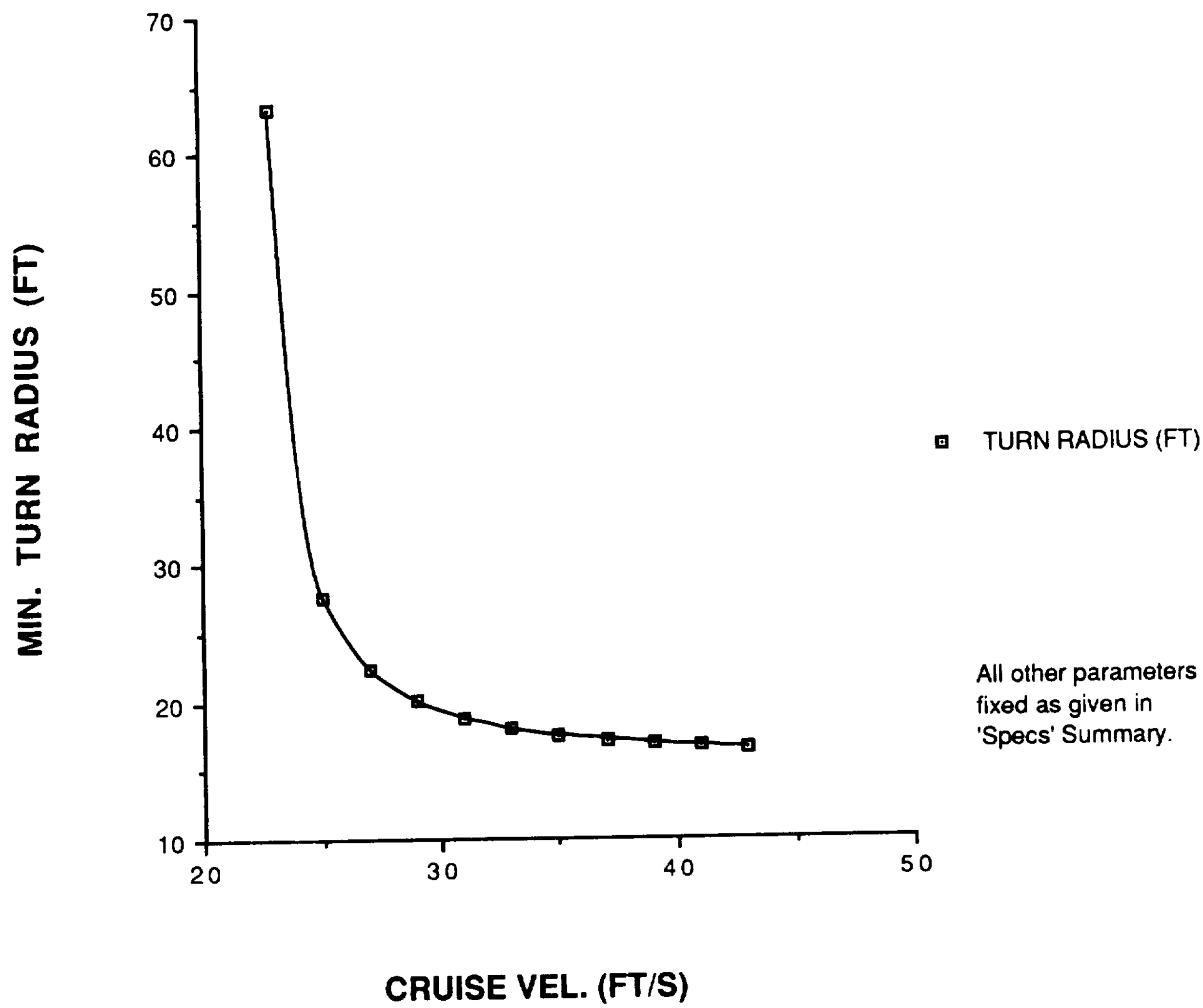
Increasing the cruise velocity before entering a turn can also decrease the minimum turn radius or increase the degree of safety for a fixed turn radius. A greater difference between the RPV's speed entering the turn and the RPV's stall speed will allow for a greater velocity decrease over the inside wing before stall occurs. Figure PF#4 shows the relationship between minimum turn radius and cruise velocity. Note the minimum turn radius asymptotically approaches 18.33 ft. as the speed increases toward its maximum value of 56.1 ft/s.

Endurance

Perhaps the most important performance characteristic for the Penguin is its endurance. As the Astro-15 motor draws a large amount of current from its batteries, the battery capacity must be maximized or endurance will be short. The Astro-15 also operates with high voltage requirements, which necessitates the use of a large number of batteries connected in series. The possibility of connecting some batteries in parallel to assist in increasing battery pack endurance is not possible because of the voltage requirement of connecting the batteries in series. Hooking up a whole second set of batteries in parallel would add a lot of undesired weight. Larger batteries are the best way to increase capacity, however these also seriously add to the RPV's weight.

This problem proved to be the most difficult to solve in the design of the Penguin. The final solution came in the form of batteries which were lighter and had greater capacity than common RPV batteries for which the Penguin was originally designed. The batteries (actually made for

FIGURE PF#4



MIN. TURN RADIUS (FT)
70
60
50
40
30
20
10
20
30
40
50
CRUISE VEL. (FT/S)
TURN RADIUS (FT)
All other parameters
fixed as given in
'Specs' Summary.

telephones) decrease battery pack weight by 10%, while increasing battery pack capacity by 8%.

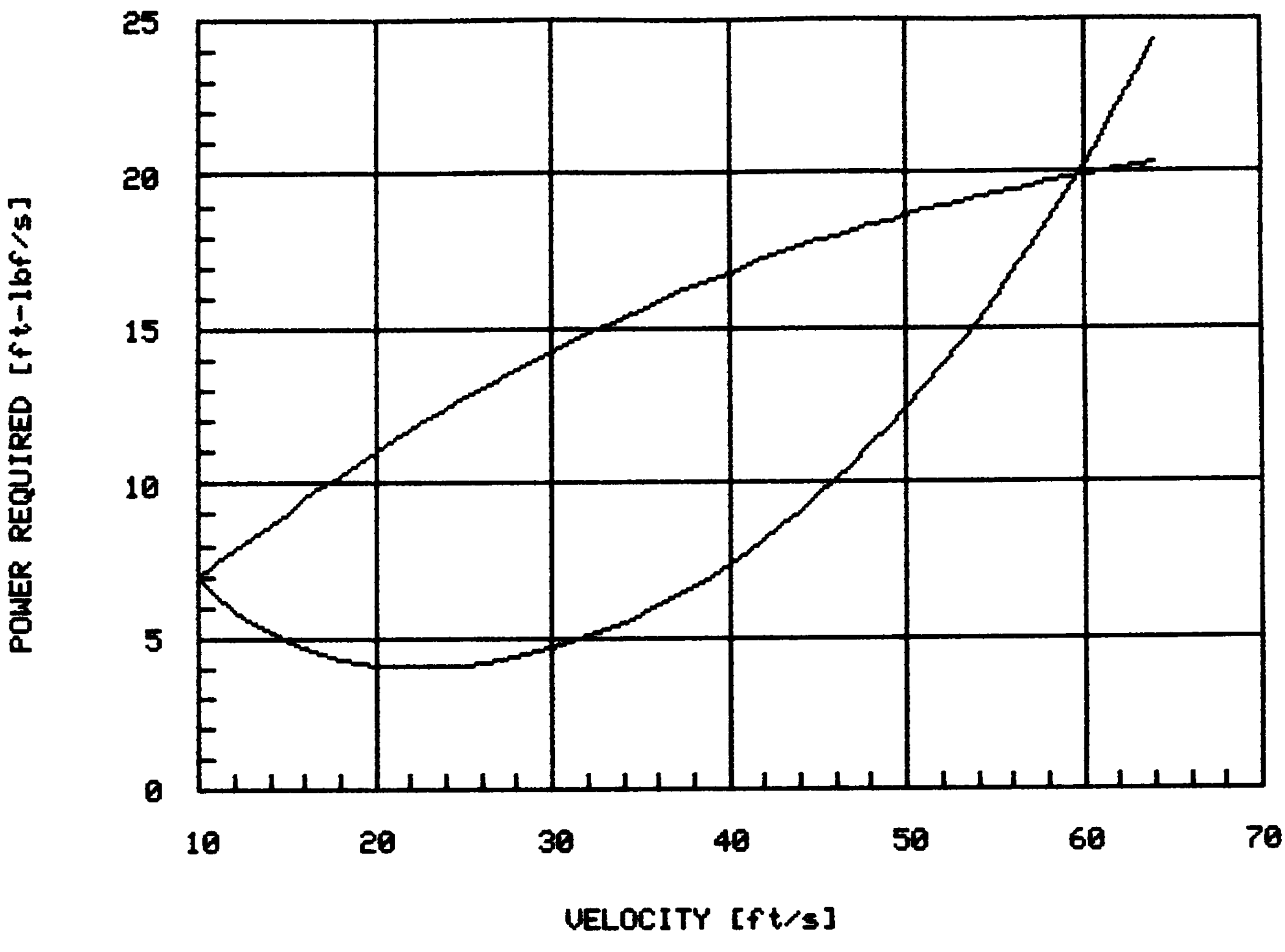
Table PF#1 gives an endurance for the Penguin of 105.3 seconds. This exceeds the design minimum goal of 83 seconds which will allow for a small amount of extra flying after the three figure eight laps of the test course are completed. Considering that the endurance of the Penguin is barely above the minimum, all possible efforts must be made to ensure endurance maximization. Figure PF#5 shows the power required for level flight of the Penguin. Note the minimum power required is approximately at 25 ft/s. Thus, endurance will be at or near its maximum at this cruise velocity, which is where the Penguin will fly.

The endurance, limited though it is, should be enough to complete the mission and land safely provided that energy is not wasted in unnecessary extra laps or pre-takeoff engine idling. If further tests determine that endurance is still lacking, the possibility of using larger batteries will have to be examined.

Conclusion

The Penguin design seems capable of meeting the performance requirements of its mission. One advantage of the design is that the engine has a large capacity for producing power. If future studies or tests determine that certain power related performance characteristics (rate of climb, cruise speed, etc.) must be improved, the engine power is available provided that the battery pack capacity can be adjusted as needed. This lends the design some flexibility should unexpected problems arise. Continual improvements in the performance analysis and testing of the Penguin will be most important during prototype construction and subsequent flying. Such analysis and testing will hopefully identify problems and give solutions as the Penguin design progresses.

Figure PF#5



25
20
15
10
5
0
POWER REQUIRED [ft-lbf/s]
10
20
30
40
50
60
70
VELOCITY [ft/s]

Aerodynamic Design

Airfoil Selection

The airfoil section chosen for The Penguin was the Wortmann FX63-137. This airfoil was chosen primarily because of it's high lift capabilities at low Reynolds Numbers. The FX63-137 also has a relatively high stall angle, and low zero lift angle. The FX63-137 has a thickness of 13.59%, and a high camber of 5.94% which creates a large 'nose down' moment coefficient of -0.08.

The lift curve slope of the Wortmann can be seen in figure AE1. It has a slope/degree value of .071/degree, a zero lift angle of -7.0, and a Cl max of 1.2 occurring at an angle of approximately 11-12 degrees. For the specified mission, The Penguin will be flying at a cruise angle of 6 degrees which corresponds to a Cl of approximately 0.98.

Manufacturing errors are expected in the airfoil. Due to these virtually unavoidable errors, a Cl of only 0.90 is expected. Hopefully, this can be achieved without too much difficulty. However, if this Cl can not be achieved, the Penguin will have to fly her mission with a higher velocity. On the other hand, if a Cl of near 0.98 can be achieved, the Penguin will be capable of flight speeds lower than the design speed.

Wing Planform Design

The wing of The Penguin has an area of 4.67 ft^2, this corresponds to a constant cord of .667 ft and a span of 7.0 ft. The wing area was chosen based on equation AE1, at sea level conditions, an estimated weight of 3.125lb, a flight velocity of 25ft/s, and a approximate Cl of 1.0:

$$Cl = \frac{W}{\frac{1}{2}\rho V^2 S} \qquad \text{AE1}$$

After the area was chosen, studies were performed on the effects of geometric twist and taper ratio, as they pertain to lift coefficient, and drag coefficient. Geometric twist, which did decrease the drag slightly, is not used on the Penguin's wing because it caused too much of a loss in lift. This loss in lift is due to the angling down of the wing tips. Taper, which would

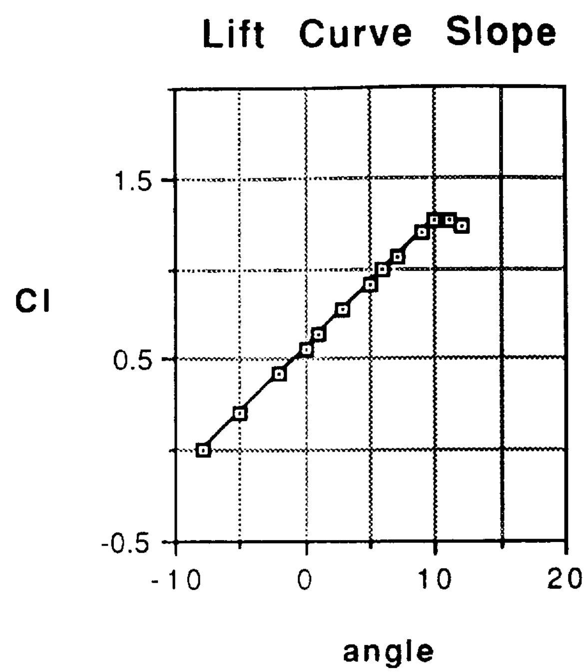


Figure AE1

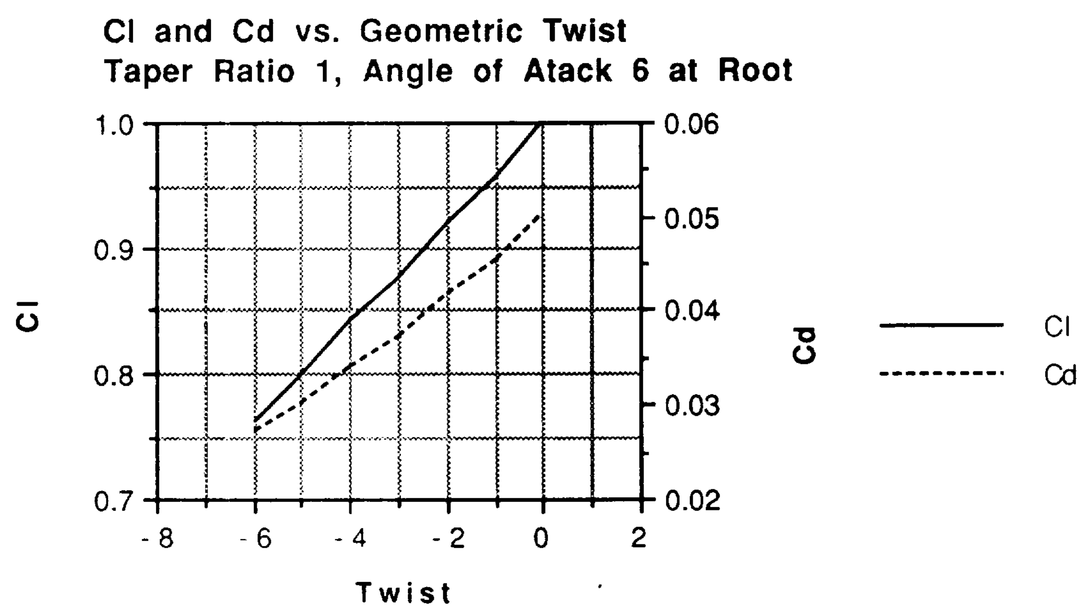


FIGURE: AE2

slightly increase the lift, and slightly increase the drag, is also not used. Taper is not used because the manufacturing headaches involved with taper out-way the aerodynamic benefits. In other words, the group would rather except the slight loss in lift than have to construct different size ribs, and angle the spars. Graphs on the effects of twist and taper can be seen in figures AE2-3.

Empennage

The vertical and horizontal tails of The Penguin will be flat plates. The tail will produce an upward lift which will be useful since weight is of such great concern. Both the vertical and the horizontal tail sections are designed with control surfaces. The empennage will be discussed in more detail in the Controls section of this report.

Fuselage Design

The fuselage of the Penguin was designed for simplicity. The major factor in the fuselage design was to keep a rectangular shape, while minimizing the size, thereby reducing the drag. The size of the fuselage was made just large enough in volume, to hold the design payload, the instruments (servos, batteries etc.), and the propulsion system.

The final size of the fuselage turned out to be 4.5ft in length, and 0.25ft in width. The height of the fuselage varies with the length of the plane. It is 0.5ft in height at the forward end of the plane, this is to accommodate for the propulsive system and a majority of the control systems. As we move back along the plane the height tapers down to 0.25ft. The tapering reduces both the weight of the aircraft(due to less material), and the drag on the airplane. The height of the aft end of the aircraft reduced to 0.23ft which is the thinnest it could be to ensure structural stability, with any degree of reliability.

Drag Prediction

The drag on the Penguin was estimated by a Drag prediction method written by Dan Jensen. This drag prediction method assumes that the complete aircraft can be broken down into into two primary component, these are the induced drag and the parasite drag. Thus the drag can be represented by equation AE2, where the first term on the right hand side is the parasite drag, and the second term on the right hand side represents the induced drag.

$$CD = CD_0 + CD_i \qquad \text{AE2}$$

The induced drag varies with Cl, thus equation AE2 can be expressed as:

$$CD = CD_0 + CL^2 / \pi \, e \, AR \qquad \text{AE3}$$

The term 'e' is the efficiency factor of the entire aircraft and can be calculated, for a strait wing aircraft (such as the Penguin), by equation AE4.

$$e = 1.78\ (\ 1.00 - 0.045\ AR0.68) - 0.64 \qquad \text{AE4}$$

The parasite coefficient is determined by equation AE5:

$$CDo = Cf\ S_{wet} / S_{ref} \qquad \text{AE5}$$

where Cf is the skin friction coefficient, S_{wet} is the total aircraft wetted area, and S_{ref} is the aircraft reference area, usually the wing area. The skin friction coefficient for an RPV such as the Penguin, can be estimated to be approximately .004.

Upon calculating equation AE3, a drag polar was obtained. The drag polar of the Penguin can be seen in figure AE4. It has a maximum lift to drag ratio of 21.4, occurring at an angle of attack of 0.0 degrees.

At the mission cruise angle of 6 degrees, The Penguin has a total drag coefficient of 0.05 which corresponds to an induced drag of .037 and a pressure drag of 0.013.

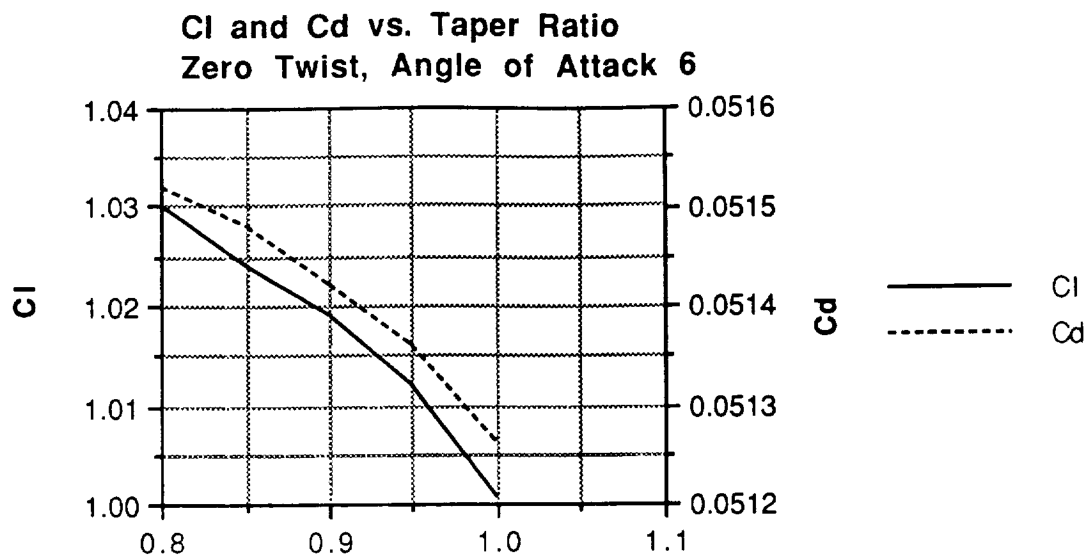


Figure AE3

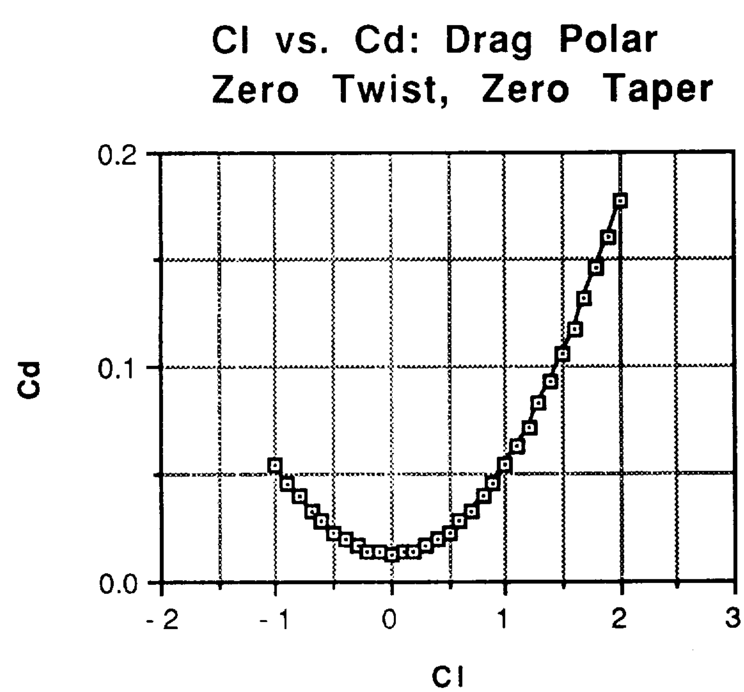


Figure AE4

Weight Estimation

Preliminary Estimates

The preliminary target design weight of the Penguin was based upon two factors: the available lift during cruise and the combined weight fraction of the propulsion system, avionics, and payload. During steady level flight at a velocity of 25 ft/sec, the Penguin can produce approximately 50 oz of lifting force assuming a lift coefficient of 0.9 and a wing area of 4.7 ft^2 . The combined weight of the propulsion system, avionics, and payload was determined to be 27.1 oz. Assuming these components account for 55% of the total plane weight results in a total plane weight of 49.3 oz. The final result of these two calculations was an initial plane weight estimate of 50 oz. The preliminary structural weight of 20 oz was based upon a weight fraction of 40% of the total plane weight. And similarly, the preliminary landing gear weight of 2.5 oz was based upon a weight fraction of 5%.

Final Estimates

As indicated in Table WE-1, the prefabrication design weights can be separated into the categories of known and estimated weights. The known weights consist of the propulsion system, avionics, and payload. As mentioned previously, their combined weight is 27.1 oz which is 53% of the plane's final weight. The estimated weights consist of the plane's structural components and the landing gear. The final estimate of the total structural weight is 19.6 oz which is 38.6% of the Penguin's total weight. This estimate is based upon the material density and dimensions of each structural component including the skin. The final estimate of the landing gear weight is 4.0 oz which is 8% of the total weight. The gear's steel wire struts weigh 2 oz while the remaining 2 ounces is reserved for wheels and mounting brackets.

The Penguin's final weight was estimated to be 50.7 ounces which is only 0.7 oz above the target design weight. This estimate is expected to be lower than the plane's final weight because of unaccounted for components. These components include the push rods and linkages necessary for connecting the servos to the control surfaces and additional reinforcing structural members. However because the plane is overpowered, this is not

FIGURE WE-1

Weight Estimation

Known Weights

electric motor	10.3 oz
motor batteries	5.0 oz
receiver	1.0 oz
receiver batteries	2.0 oz
speed controller	3.0 oz
servo (3)	0.6 oz
payload	2.0 oz
propellor	2.0 oz

Total avionics and propulsion weight=27.1 oz

Estimated Weights

fuselage	9.6 oz
wing	8.5 oz
vertical tail	0.5 oz
horizontal tail	1.0 oz
front gear	3.5 oz
rear gear	0.5 oz

Total structural weight=19.6 oz

Total estimated plane weight=50.7 oz

a major concern. For example, assuming steady-level flight conditions and total lift coefficient of 0.9, a 10% increase in the Penguin's weight requires an increased velocity of only 1 ft/sec to provide adequate lift.

Center of Gravity Location and Travel

In order to achieve proper static stability, an x-axis center of gravity located at 30% of the wing chord was desired. Initial calculations based upon the preliminary component weights indicated the center of gravity was located slightly aft of the 30% chord. Therefore, the nose of the plane was extended approximately 1 inch to insure the center of gravity could be placed at the desired position. After final estimates of the plane's structural weights and dimensions were made, the internal configuration was laid out according to the following constraints:

1) The speed controller must be within 9 inches of the receiver and 8.5 inches of the engine.
2) The receiver battery pack must be within 4 inches of the receiver.
3) The motor batteries must be within 5 inches of the speed controller.
4) The rudder and elevatror servos must be positioned such that their control linkages readily connect to the control surfaces. This most likely implies being located aft of the other internal components.
5) The aileron servo must be positioned such that its control linkage connects both ailerons.
6) All internal components must be arranged such that the center of gravity is located at 30% of the wing's chord.

As indicated in the top of Figure CG-1, positioning the motor batteries and speed controller directly behind the motor and the payload, receiver, and receiver batteries aft of the 30% chord, does result in a center of gravity located at the 30% chord (13.5" on our reference system).

Due to fluctuations in the actual center of gravity position, a means of relocating the center of gravity at the 30% chord position was desired. The error associated with the values used for the center of gravity calculation and unaccounted for structural components are expected to be the two major factors necessitating a relocated center of gravity. The most questionable values contributing to error are the individual centers of gravity of the plane's structural components (most notably the fuselage and wing). Although the estimated weights of these structures are believed to be accurate, their centers of gravity were simple estimates. Also, the position of the center of gravity is extremely sensitive to additional weight in the empennage and rear fuselage. However, as indicated in the middle drawing of Figure CG-1, placing the receiver, receiver batteries, and payload forward of the 30% chord, results in the center of gravity moving forward 0.4 inches to the 25% chord position (13.1" on our reference system). As indicated in the bottom drawing of Figure CG-1, returning the previously mentioned components to their original positions along with repositioning the motor batteries and speed controller closer to the 30% chord, results in the center of gravity moving aft 0.4 inches to the 35% chord position (13.9" on our reference system). This provides a total center of gravity travel range of ±0.4 inches depending upon the internal configuration of the plane.

X_{CG} TRAVEL

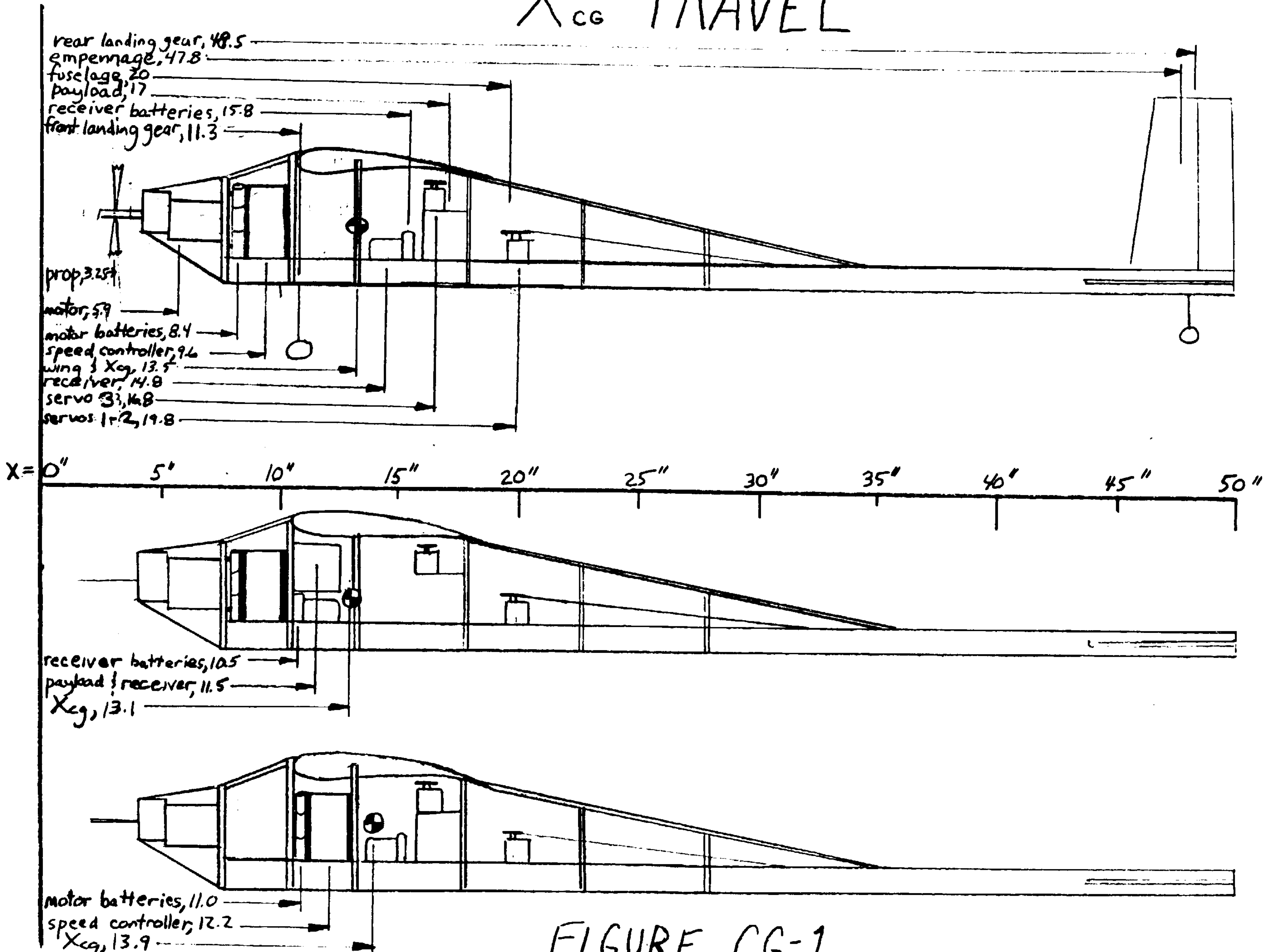


FIGURE CG-1

Structural Design

V-N Diagram

The V-n diagram (velocity vs. the load factor) for the Penguin can be seen in figure St.1. The maximum velocity was defined to be 50 ft/s because the nature of the aircraft mission is to fly indoors. At velocities greater than 50 ft/s, it was judged that it would be too difficult to avoid obstacles (namely the walls) inside the Loftus facility. The loading factor of 4 is twice the value judged to reasonable for indoor flight. The loading factor of 2 corresponds to 60° banked turn, and a turning radius of 17.6 feet at the level flight cruising velocity of 31 ft/s. This radius is sufficiently below the design radius of 50 feet, and was therefore determined to be an adequate upper limit. A factor of safety of 2 was chosen due to the inaccuracy of the calculations in the beams due to assumptions, and the fact that actual manufactured beams would be slightly different from the designed beams.

Flight Loads

The maximum flight loads for the Penguin can be estimated from the V-n diagram. Using the limit load factor of 4, and a weight of 50 ounces, a lift of 12.5 lbs. is generated. This load is shared equally by each side of the wing. The equivalent lifting load is therefore 6.25 lbs. per wing located at the mid-span, 21 inches from the root. This is equivalent to a moment of M_z = 136.5 lb-in at the root. The drag of the wing was found using an estimated coefficient of drag of 0.082, and the maximum velocity of 50 ft/s. This in turn yielded a drag force of 0.57 lbs. per wing also located at the mid-span, which in turn created a moment of M_y = 12 lb-in at the root. These were the forces used to calculate the stress in the wing spar.

Since the wing forces were already known, it was then necessary to determine the tail forces. Because of the relatively low aspect ratio of the horizontal tail (4.8), and the flat plate configuration, it was very difficult to calculate a value of C_{Lmax} for the tail. For this reason the maximum coefficient of lift with full deployment of the elevators was estimated to be

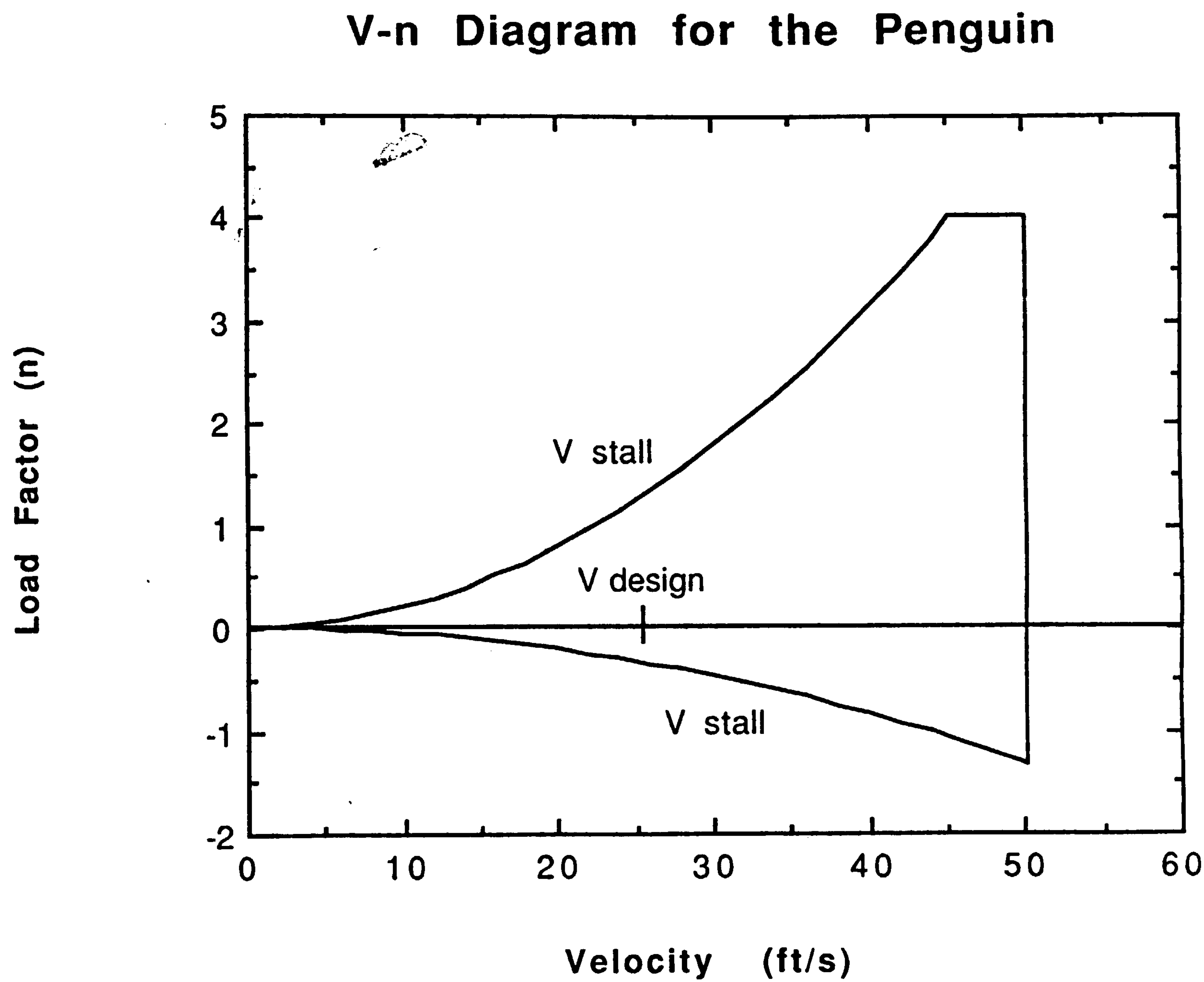

V-n Diagram for the Penguin
Load Factor (n)
5
4
3
2
1
0
-1
-2
0
10
20
30
40
50
60
V stall
V design
V stall
Velocity (ft/s)

Figure St.1

1.0. This then was multiplied by a factor of safety of 1.3 yielding a Cl_{max} of 1.3. Using the definition of the lift coefficient

$$C_L = \frac{L}{.5rV^2S} \quad \text{(St.1)}$$

where L is the lift, and S is the surface area of the wing, the maximum lift for the horizontal tail was found to be 4 lbs. Using a moment arm of 6 inches, or half the span, the moment M_z can be found to be 12 lb-in. The drag forces were neglected.

Similarly for the vertical tail, the maximum coefficient of lift with the factor of safety of 1.3 was found to 1.8. This corresponds to a lifting force of 2 lbs. for the vertical tail. For simplicities sake the moment arm was estimated to be 4 inches, slightly larger than mid-span. This corresponds to a bending moment of M_z = 8 lb-in. The drag force was again neglected. A summary of all internal loads can be seen in table St. 1

Component	**F_{normal} max**	**F_{drag} max**
wing	6.25 lbs/wing	.57 lbs/ wing
horizontal tail	4 lbs.	neglected
vertical tail	2 lbs.	neglected

Table St. 1

Internal Layout

The wing was designed to have only one load carrying spar located at the maximum thickness of the wing. The location of the spar at the maximum thickness was the result of two factors. First, because this location is the thickest, for any given beam configuration, the moment of inertia about the z-axis will be the largest, which in turn reduces the direct stress in the beam. Also, because the location of the maximum thickness is very near 30% chord, it also is very near the center of pressure for the wing, which can be estimated to be at 25% chord. Since the beam is located so close to the center of pressure, moments due to the lift and drag forces about the 30% chord will be negligible.

The load carrying member of the fuselage is a single box beam that runs the entire length of the fuselage. It should be noted that originally the fuselage was designed as a box beam because it was felt that this type of configuration would be lighter than a truss design. After some preliminary calculations it was determined that both designs would be comparable in weight; however, it was felt that a box beam would be easier to manufacture. Therefore, the Penguin's fuselage was designed as simple box beam capable of withstanding maximum flight and ground loads.

The design of the empennage is different from that of the fuselage and wing in that the structural loads are shared by two simple beams in both the horizontal and vertical tail. The beams are located at the leading edge, and at the trailing edge just prior to the control surfaces. These beams share the tail loads based upon an area weighted percentage. The structural layout of the entire plane can be seen in figure St.2

Materials Selection

After the loads, and corresponding bending moments had been determined it was necessary to choose the type of material that would be used to carry these loads. The following materials were considered: wood (namely balsa, spruce, and plywood), metals, composites. These materials were weighted in the following areas on a scale of 1 to 3 (1 being the best): strength, weight, machinability, availability, and cost. These rankings are as follows:

material	strength	weight	machine	available	cost
wood	3	2	1	1	1
metal	2	3	2	2	2
composite	1	1	3	3	3

Table St.2

From the sum of the material's ranking, it was determined that wood had the lowest score, and hence the highest ranking. Thus, wood was chosen as the material for the entire structure of the airplane (excluding the monokote film, which was not considered in any calculations.) The properties of spruce are $E = 1.3 \times 10^6$ psi, and $\rho = 0.016$ lb/in.^{3}, with a

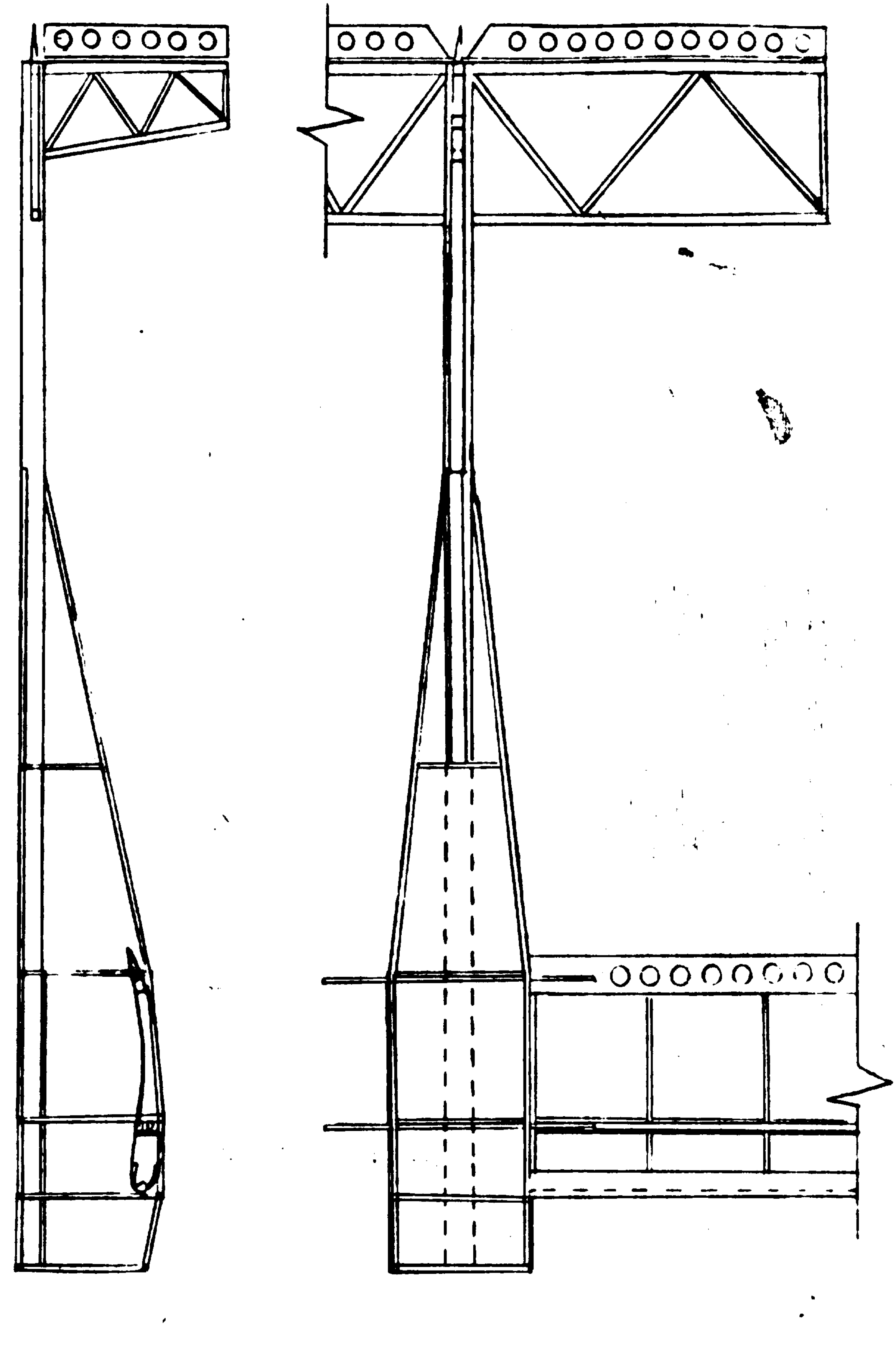


FIGURE St.2

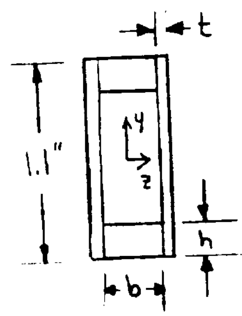
maximum direct stress σ_{xx} = 6200 psi. The properties of balsa wood are E = 65000 psi, and ρ = 0.0058 lb/in.^{3}, with $\sigma_{xx_{max}}$ = 400 psi. [1] It is important to note here that torsional effects were neglected during the analysis. The reason for this is that from consultation with experienced RPV pilots, there was a general consensus that for the slow speed mission, torsional effects would be small for a box beam design of the fuselage.

Finally, in order to verify that wood would be capable of withstanding the maximum loads, computer spreadsheets were written to study the characteristics of various beam configurations. The spreadsheets calculated the direct stress in the beams. For all beams capable of withstanding maximum loads, weight was used as the measure of merit to determine which configuration should be used for the Penguin. The direct stress was calculated using advanced beam theory for heterogeneous, constant temperature beams:

$$\sigma_{xx} = -\frac{E\,M_z\,y}{E_1\,I_{zz}^*} + \frac{E\,M_y\,z}{E_1\,I_{yy}^*} \quad \text{(St.2)} \quad [2]$$

Using this formula and the properties of balsa and spruce wood, a simple spreadsheet algorithm was written to determine the maximum stresses that would occur for any given box beam made out of balsa, spruce, or a combination of the two woods. The results from this program can be seen in table St.3. The type of wood used in the different beam designs can be determined by looking at the modulus of elasticity, E. If E = 1300000, the wood is spruce, and if it is 65000 the wood is balsa. The maximum stresses found in the caps and webs are labeled as xx - caps and xx - webs, respectively. The units of these stresses are psi. So long as these stresses are below the maximum allowable stress (400 psi for balsa, 6200 psi for spruce) the beams is considered. From the data, it is clear to see that beam 3, with 1/8 inch by 1/8 inch spar caps, and a 1/32 inch thick spar web, made entirely of spruce, and weighing 1.1 ounces, is the lightest beam. (It should be noted that beams 1 and 6 exceed $\sigma_{xx_{max}}$, and therefore were not considered.) The stress due to shear in the spar was calculated to be less than 100 psi, and thus more than an order of magnitude below the spar

	beam 1	beam 2	beam 3	beam 4	beam 5	beam 6
b	0.1875	0.25	0.125	0.1875	0.1875	0.375
h	0.1875	0.25	0.125	0.1875	0.1875	0.375
t	0.0625	0.03125	0.03125	0.0625	0.09375	0.125
E - caps	1300000	1300000	1300000	65000	1300000	65000
E - webs	65000	65000	1300000	1300000	65000	65000
rho -caps	0.016	0.016	0.016	0.0058	0.016	0.0058
rho - webs	0.0058	0.0058	0.016	0.016	0.0058	0.0058
Izz	0.31071517	0.47151563	0.01439974	0.01460671	0.31764746	0.06798307
Iyy	0.01074422	0.01438599	0.00140584	0.00663465	0.01560059	0.05629069
xx - webs	406.839329	283.431901	5813.34304	5224.68534	371.482009	1077.22297
xx - caps	6740.68735	5147.29734	5546.59919	255.582121	5987.38666	1071.07186
weight (ozs.)	1.29192	1.61196	1.0752	1.75245	1.55988	2.16804



$$I_{zz} = 2\left[\frac{E}{E_1}\left(\left(\frac{1}{12}bh^3\right) + (bh)\left(.55 - \frac{h}{2}\right)^2\right) + \frac{E}{E_1}\left(\frac{1}{12}t(1.1)^3\right)\right]$$

$$I_{yy} = 2\left[\frac{E}{E_1}\left(\frac{1}{12}b^3h\right) + \frac{E}{E_1}\left(\left(\frac{1}{12}(1.1)t^3\right) + (1.1t)\left(\frac{b+t}{2}\right)^2\right)\right]$$

TABLE St.3

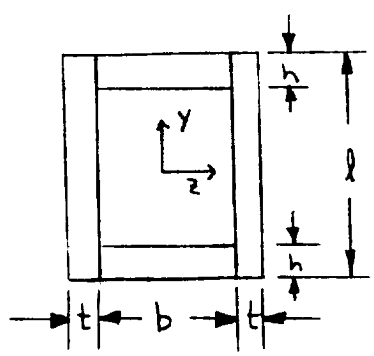
maximum stress. It was therefore determined that for all beams, σ_{xx} would be the limiting factor.

In order to calculate the stress on the fuselage it was assumed that the fuselage could be modeled as a cantilevered beam (at the center of gravity of the airplane), where the tail forces and the wing forces could be considered to be independent of each other. That is to say that the fuselage was modeled in two separate ways, the first model being a cantilevered beam from the c.g. to the tail section, and the second being a cantilever from the c.g. to the wing. M_z for the cantilevered tail was found to (4 lbs.)x(34 inches) or 136 lb-in. M_y was found to be (2 lbs.)x(34 inches) or 68 lb-in. For the cantilevered wing model, the lift and drag forces previously determined were multiplied by a 2 inch moment arm to calculate M_z and M_y. The direct stress was then calculated as before using equation (St.2). The results from these calculations can be seen in table St.4. This table reads in the same manner as St. 3. From this data it was determined that Beam 4, the 1/8 inch thick beam with external dimensions of 3/4 inch by 1/2 inch was the lightest beam (2.88 oz.) that had maximum stresses below the maximum allowable stresses. It should be noted that beams 5 and 6 failed in the cantilevered tail condition, but no beam failed in the cantilevered wing model. In fact the stresses due to the cantilevered wing were approximately an order of magnitude lower than allowable stresses for all of the beams except for beam 6, which was still less than 50% of the allowable stress. Clearly from these results it can be seen that the cantilevered tail is the limiting condition.

The beam configuration model for the vertical and horizontal tail is primarily the same, except for one small fact. Because the vertical tail is tapered, the leading edge beam is 8 inches long, whereas the trailing edge beam is 7.5 inches long. The horizontal tail has no taper, so both the leading and trailing edges are 12 inches long (per side). Since the drag was neglected for the tail, the stress can be found by neglecting the contribution of the M_y term in equation (St.2). This equation then reduces to

$$\sigma_{xx} = -\frac{E\,M_z\,y}{E_1\,I_{zz}^*} \quad \text{(St.3)} \quad [2]$$

	beam 1	beam 2	beam 3	beam 4	beam 5	beam 6
b	0.75	0.5	0.5	0.25	0.125	0.75
h	0.125	0.125	0.125	0.125	0.125	0.5
t	0.125	0.125	0.125	0.125	0.125	0.125
l	1	1	0.75	0.75	0.75	1
E - caps	65000	1300000	1300000	1300000	1300000	65000
E - webs	65000	1300000	1300000	1300000	1300000	65000
rho -caps	0.0058	0.016	0.016	0.016	0.016	0.0058
rho - webs	0.0058	0.016	0.016	0.016	0.016	0.0058
Izz	0.05696615	0.04492188	0.02115885	0.01497396	0.01188151	0.08333333
Iyy	0.05696615	0.02734375	0.02115885	0.00716146	0.00321452	0.08333333
From tail						
xx - webs	1790.53714	2446.31056	3615.50769	5779.73123	8258.76331	1224
xx - caps	1641.32571	2135.45342	3213.78462	4592.82213	5614.51014	1122
From wing						
xx - webs	228.205714	291.975155	460.8	660.996047	847.37021	156
xx - caps	226.011429	287.403727	454.892308	643.541502	808.484134	154.5
weight (ozs.)	1.827	4.32	3.6	2.88	2.52	4.176



$$I_{zz} = 2\left[\frac{E}{E_1}\left(\left(\frac{1}{12}bh^3\right) + (bh)\left(\frac{l-h}{2}\right)^2\right) + \frac{E}{E_1}\left(\frac{1}{12}tl^3\right)\right]$$

$$I_{yy} = 2\left[\frac{E}{E_1}\left(\frac{1}{12}b^3h\right) + \frac{E}{E_1}\left(\left(\frac{1}{12}lt^3\right) + (lt)\left(\frac{b+t}{2}\right)^2\right)\right]$$

TABLE St.4

The percentage of the load that is carried by each equation was assumed to be an area weighted percentage

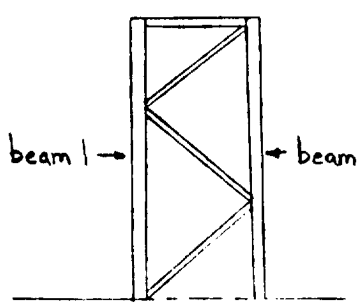
$$\%M_z = \frac{A_i E_i}{(A_1 E_1) + (A_2 E_2)} \times M_z \quad \text{(St.4)}$$

Using equations (St.3) and (St.4) and the previously determined moments, a spreadsheet was written to calculate the direct stress in the empennage. The results for the horizontal tail can be seen in table St.5, and the results of the vertical tail can be seen in table St.6. These tables read much like St.3 & 4, except that the percentage of the loads carried by each beam is designated as % - 1 and % - 2 for beams 1 & 2 respectively. The stresses in each beam are labeled as xx - 1 and xx -2 with units of psi. From the data of table St.5, it can be seen that case 4, with a 1/4 inch by 1/4 inch square beam at the leading edge made of spruce, and a balsa beam of 3/16 inches by 3/16 inches at the trailing edge, is the lightest configuration in which the maximum allowable stresses are not exceeded. From the data of table St.6 it is easy to see that for the vertical tail case 5, a configuration of an 1/8 inch by 1/8 inch square balsa beam for the leading edge, and a 1/4 inch by 1/4 inch trailing edge spruce beam has the lightest weight for any configuration that does not exceed the maximum allowable stresses.

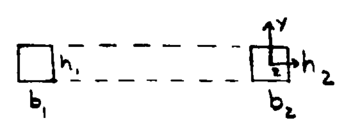
Landing Gear

The Penguin's landing gear will consist of a standard, three wheeled, tail dragger configuration. This basic configuration was chosen over a tricycle gear because of its weight and ground steering advantages. The landing gear system was designed such that the Penguin is capable of executing straight and turning ground maneuvers, a stable takeoff, and a stable landing. With this tail dragger configuration, the plane will be capable of taking off with the wheel still dragging or with the tail raised. Because of the wing stall possibilities and minimal difference in takeoff distance, the latter scenario will most likely be employed. As seen in Figure LG-1, the front gear consists of two, commercially available, air filled wheels connected to steel wire struts which are mounted to the bottom of the horizontal tail

horizontal	case 1	case 2	case 3	case 4	case 5	case 6
beam 1:						
b	0.25	0.25	0.25	0.25	0.25	0.375
h	0.25	0.25	0.25	0.25	0.25	0.375
E	1300000	1300000	1300000	1300000	1300000	65000
density	0.016	0.016	0.016	0.016	0.016	0.0058
Izz	0.00032552	0.00651042	0.00032552	0.00651042	0.00651042	0.00164795
Iyy	0.00032552	0.00651042	0.00032552	0.00651042	0.00651042	0.00164795
beam 2:						
b	0.25	0.25	0.1875	0.1875	0.125	0.375
h	0.25	0.25	0.1875	0.1875	0.125	0.375
E	1300000	65000	1300000	65000	65000	65000
density	0.016	0.0058	0.016	0.0058	0.0058	0.0058
Izz	0.00032552	0.00032552	0.000103	0.000103	2.0345E-05	0.00164795
Iyy	0.00032552	0.00032552	0.000103	0.000103	2.0345E-05	0.00164795
% - 1	0.5	0.95238095	0.64	0.97264438	0.98765432	0.5
% - 2	0.5	0.04761905	0.36	0.02735562	0.01234568	0.5
xx - 1	2304	4388.57143	2949.12	4481.94529	4551.11111	682.666667
xx - 2	2304	219.428571	3932.16	298.796353	455.111111	682.666667
weight oz.	0.768	0.5232	0.6	0.4623	0.4188	0.6264



TOP



SIDE

$$I_{zz} = \frac{1}{12} b h^3 \left(\frac{E}{E_1}\right)$$

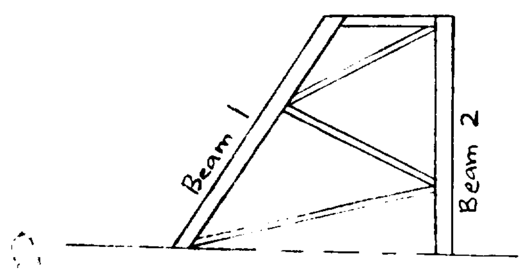
$$I_{yy} = \frac{1}{12} b^3 h \left(\frac{E}{E_1}\right)$$

NOTE: DRAWINGS NOT TO SCALE

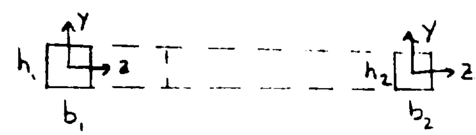
TABLE ST. 5

vertical tail

vertical	case 1	case 2	case 3	case 4	case 5	case 6
beam 1:						
b	0.25	0.125	0.25	0.25	0.125	0.375
h	0.25	0.125	0.25	0.25	0.125	0.375
E	1300000	1300000	1300000	1300000	65000	65000
density	0.016	0.016	0.016	0.016	0.0058	0.0058
Izz	0.00032552	0.0004069	0.00651042	0.00651042	1.0173E-06	0.00164795
Iyy	0.00032552	0.0004069	0.00651042	0.00651042	1.0173E-06	0.00164795
beam 2:						
b	0.25	0.25	0.25	0.125	0.25	0.375
h	0.25	0.25	0.25	0.125	0.25	0.375
E	1300000	65000	65000	65000	1300000	65000
density	0.016	0.0058	0.0058	0.0058	0.016	0.0058
Izz	0.00032552	0.00032552	0.00032552	2.0345E-05	0.00032552	0.00164795
Iyy	0.00032552	0.00032552	0.00032552	2.0345E-05	0.00032552	0.00164795
% - 1	0.5	0.83333333	0.95238095	0.98765432	0.01234568	0.5
% - 2	0.5	0.16666667	0.04761905	0.01234568	0.98765432	0.5
xx - 1	1920	25600	3657.14286	3792.59259	379.259259	568.888889
xx - 2	1920	640	182.857143	379.259259	3792.59259	568.888889
weight oz.	0.352	0.106	0.25	0.2065	0.1774	0.2871



SIDE



BOTTOM

$$I_{zz} = \left(\frac{E}{E_1}\right)\frac{1}{12} b h^3$$

$$I_{yy} = \left(\frac{E}{E_1}\right)\frac{1}{12} b^3 h$$

NOTE: DRAWINGS NOT TO SCALE

TABLE ST. 6

fuselage. Similarly, the rear gear consists of a single, commercially available wheel attached to a steel strut. This strut is attached to the rudder hinge and rotates with it, thereby providing more ground steering capability for the Penguin.

The primary parameters to be determined for this design were the position and sizing of both the front and rear gear. The front gear will have a 12 inch tread and be mounted in a position 6.5 inches below and 2.2 inches forward of the plane's center of gravity. In addition to providing the desired 1.5 inch propeller clearance, this positioning also yields adequate stability during ground maneuvers, takeoff, and landing. Both the tread and the height of the front landing gear were determined based upon the "turnover angle method" found in Andy Lennon's R/C Model Airplane Design The rear gear is attached to the rudder and extends 1.7 inches below the fuselage. This positioning results in a tail angle of 4 degrees. This coupled with the wing's steady, level flight attack angle of 6 degrees results in a total wing attack angle of 10 degrees when the plane is taxiing with all wheels on the ground. As mentioned previously, the plane can takeoff at this angle, however this attack angle is probably too close to the wing's predicted stall angle of approximately 12 degrees. Therefore, the most likely takeoff scenario is one in which the Penguin's elevator control is used to lift the plane's tail (thus reducing the wing's attack angle) before the plane takes off.

The sizing of the landing gear was governed by weight and product availability constraints. Of the suitable, commercially available wheels (3", 2.25", 1", and 0.5" diameter), the 2.25 inch diameter and 1.0 inch diameter wheels seemed to be the most logical choices for the front and rear wheels, respectively. The "roughness" of the turf may require opting for larger sizes, but this will have to be determined after a ground roll test of the completed plane. Because of its strength, simplicity, and low weight 0.078 inch and 0.055 inch diameter steel piano wire will be used to make the front and rear landing gear struts. The strut will experience 9.3 lbs. reaction load during landing assuming that the Penguin lands on a single gear at an accleration of 2 g's. The stress for such a landing is 2 ksi, which is well below the 36 ksi yield strength [2]. The combined total weight of the struts was calculated to be less than 2 oz which leaves 2 oz of additional weight in reserve for the wheels and mounting brackets. Unfortunately, the weights of the wheels could not be determined prior to purchase, but they are not expected to exceed their maximum design weight.

References

1. - Aerospace Design Lab handout
2. - Allen, David H., and Haisler, Walter E., Introduction to Aerospace Structural Analysis, John Wiley & Sons (New York: 1985)

PROPULSION SYSTEM SELECTION

Introduction

The aim of the propulsion section is plain--to select an engine and propeller which fulfills the design requirements and objectives, to wit:

Table ES-1: Propulsion System Goals:

Performance Measures of Merit	The chosen system must be able to: 1) sustain flight at a speed of 25 to 30 ft/s, 2) provide excess power for a steady R/C of 5 ft/s, and 3) produce sufficient thrust for a takeoff velocity of no less than 20 ft/s over a 75 ft runway.
Design Standards of Merit	The following design concepts are desirable: 1) Given the experience level of the group, the lack of readily available data, and time constraints, the system must be easily analyzable. 2) The system should be relatively lightweight. 3) The system should be relatively low in cost.

Of critical importance is the cruise condition. To fly at a mean chord Reynolds Number of 100,000 requires rigid control over aircraft velocity. Note that many of the propulsion section objectives are collateral in nature, relating strongly to performance (c.f.). Details not covered in Section III may be found in that section. The Propulsion System Selection is divided into two parts. The first, engine selection (ES), concerns itself with the major criteria: emission, available power, wherein cruise and climbing are discussed; current draw, static thrust, and cost. Part two deals with propeller selection and takeoff performance.

Part I: Engine Selection (ES)

Initial Engine Screening

In accordance with the non-emissive directive as stated in the RFP, the type of system is immediately narrowed to two categories--carbon dioxide and electric engines. The CO_2 system incorporates compressed gas cartridges which generate high pressure flow across a turbine-like device connected to a propeller shaft. Although capable of meeting performance standards, the major disadvantage lies with the cost, which lies well above the $90.00 budget ceiling. Electrics, however, offer a lightweight, clean, and more economical alternative.

Many small electric motors are available on the market, but very limited performance information, often none at all, is supplied. Without a working knowledge, caution is necessary in procurement--one cannot decide merely by picking a motor that "looks the right size," or "sounds powerful." Furthermore, experimentation is precluded by procurement and research time. In short, such detailed work is beyond the scope of this study; Readily available data must be used, such as that of the *Astro 05* and the *Astro 15*.

Previous design teams have successfully used *AstroFlight Inc.* brand cobalt geared systems for similar RPV missions, and have already compiled performance data. *AstroFlight* classifies electric engines according to an equivalent cubic inch displacement of a 2-cycle glow plug engine. In regards to capacity and power then, the *Astro 15* ranks higher, having 0.10 cubic inches more cylinder volume than the *Astro 05*.

Power Available and Required

Given the manufacturer's performance data, power available and power required curves may be generated. For the available power, one multiples the available gear power by the propeller efficiency at a given flight condition (velocity and shaft RPM). An aircraft weight, wing surface area, and propeller diameter must be assumed to calculate the lift coefficient and advance ratio (these were 3.125 lbs, 4.67 ft^2, and 10 in, respectfully). These values, in addition to the drag polar, will be used to calculate the required power. For the limiting drag requirements, two drag polar extremes were arrived at by substituting best and worst case aerodynamic and geometric data. The scenarios are:

Table ES-2: Drag Polar Extremes

Scenario	Cdo	e	AR
"Worst"	0.027	0.7	6
"Best"	0.010	0.9	9

Both power curves are each plotted for the Astro 05 and 15 (figures ES-1 and ES-2). At cruise conditions, limiting drag polars indicate a minimum required power of 6-11 watts. Both engines appear to operate at the same RPM's for steady level flight between 25 to 30 ft/s. The 05 needs an RPM range between 8790 to 8910, while the 15 needs a band of 8790 to 8900. Both RPM limits are easily attainable by their respective engines.

The power curves over the full range of RPM's are displayed in figures ES-4 and ES-5. The Astro 05 plot indicates a maximum power available of about 80 ft-lbf/s at about 45 ft/s. The maximum speed is in excess of 50 mph. The Astro 15 produces a maximum of 120 ft-lbf/s at 40 ft/s, and will probably achieve speeds close to 55 mph. At cruising speed, both engines are also capable of supplying the extra 16 ft-lbf/s needed for climbing. Given power requirements for cruise and climbing alone, either of the two engines would suffice. However, the target design weight of 50 ounces is twice that recommended for the Astro 05 (see table ES-3). Furthermore, the 18% increase in motor weight (1.5 oz) is outmatched by a 60% gain in available power.

Table ES-3: Selected Motor Characteristics
* does not include battery weight.
** manufacturer suggested value.

Motor Type	Weight [oz]*	Max. Power [W]	Aircraft Weight [oz]**
Astro 05	8.36	125	22
Astro 15	9.86	200	50

It is of great significance to understand why the RPM decreases while the gear power increases--the manufacturer tests his engines by artificially increasing the load torque (essentially by "pinching" on the shaft); this effects an increase in the armature current as the engine attempts to maintain the RPM. The actual operational mode of the real RPV involves a speed controller, which pulses a maximum voltage at a controllable frequency. The engine hence receives an rms voltage, which determines, together with the applied aerodynamics loads, the propeller RPM and power. There exists a discrepancy then, between the manufacturer's correlation between RPM and power and the actual correlation. In the installed system, power does increase with increasing RPM.

Therefore, the RPM information in figures ES-1 to ES-4 are nominal at best. True RPM-power relationships may be arrived at analytically through the propeller torque method. Rather than only employing the manufacturer's data, the propeller torque method relies on simple blade element theory to produce thrust, torque, and efficiency curves (see part two for more details).

Note that power deliverable as a function of RPM changes significantly (see figure ES-5) when using the propeller torque method. This technique suggests a more accurate cruising RPM of 4250 to 5350 for the Astro 15; a maximum static R/C ≈ 12 ft/s at 8800 RPM; and a maximum velocity of about 40 mph. It is the opinion of the propulsion team that these numbers represent the most realistic propulsion performance results.

Power Draw

The Astro systems are powered by nickel-cadmium rechargeable batteries in series. The Astro 05 ideally requires 14 V to operate, whereas the Astro 15 requires 16 V. It is possible to run at lower potentials (such as using a speed controller, in which an rms voltage will be seen), but performance will vary beyond the given data. The greater the load torque, the greater the armature current I, reducing duration and increasing battery heat. *For a given RPM and/or gear power,* the Astro 15 requires less current than the 05 to operate (see figure ES-6). The difference amounts to as small as 1 amp at high RPM's (low torque loadings) to as much as 5 amps at low RPM's (high torque). For a more detailed analysis on battery duration and flight endurance, see the Performance section.

Static Thrust

Without propeller analysis (c.f.), prediction of static thrust is approximate at best; a *tentative* number may be calculated through empirical relations such as that proposed by Karl H. Falk [Aircraft Propeller Handbook,

Figure ES-1: Power Balance, Astro 05

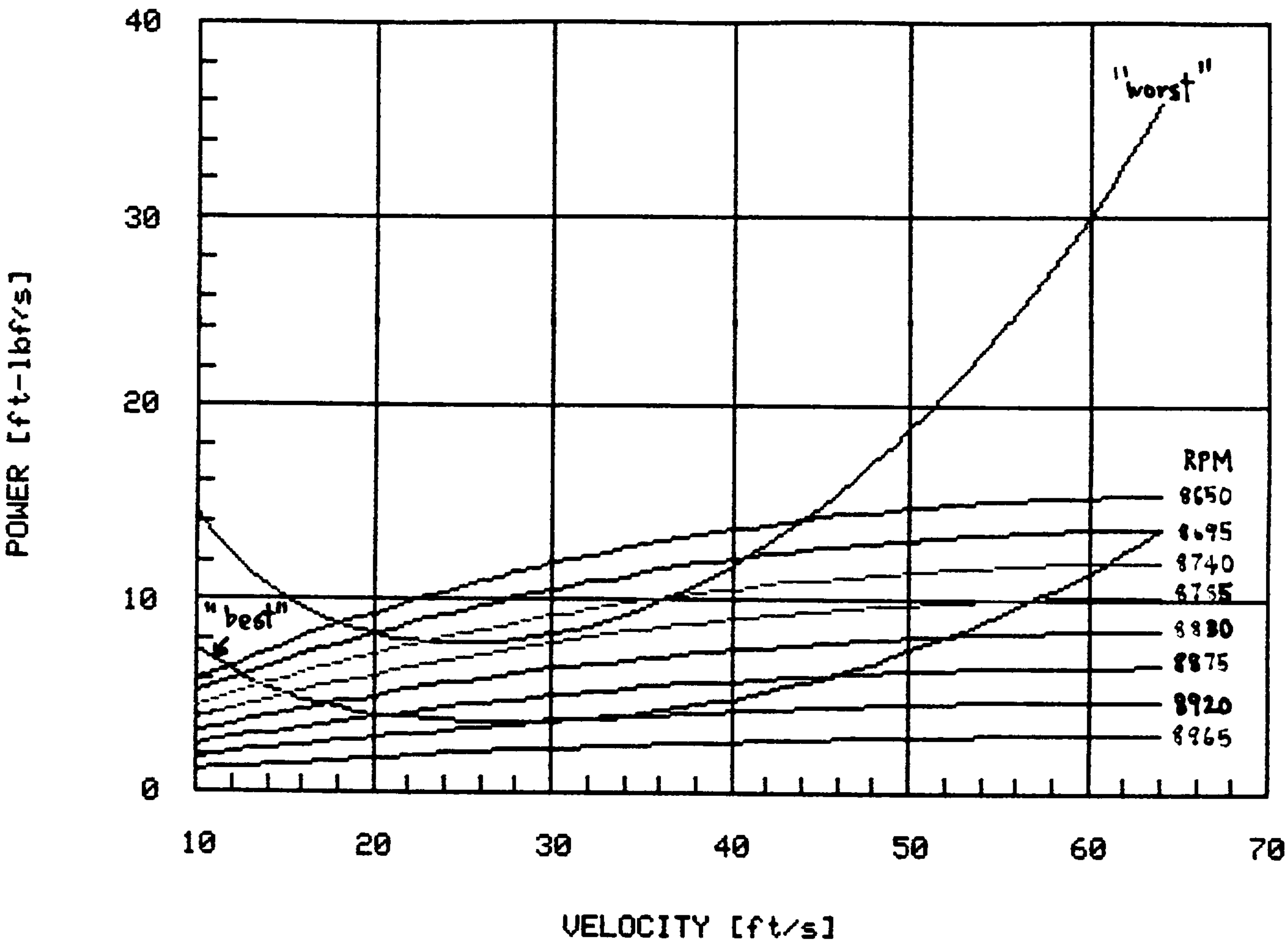


Figure ES-2: Power Balance, Astro 15

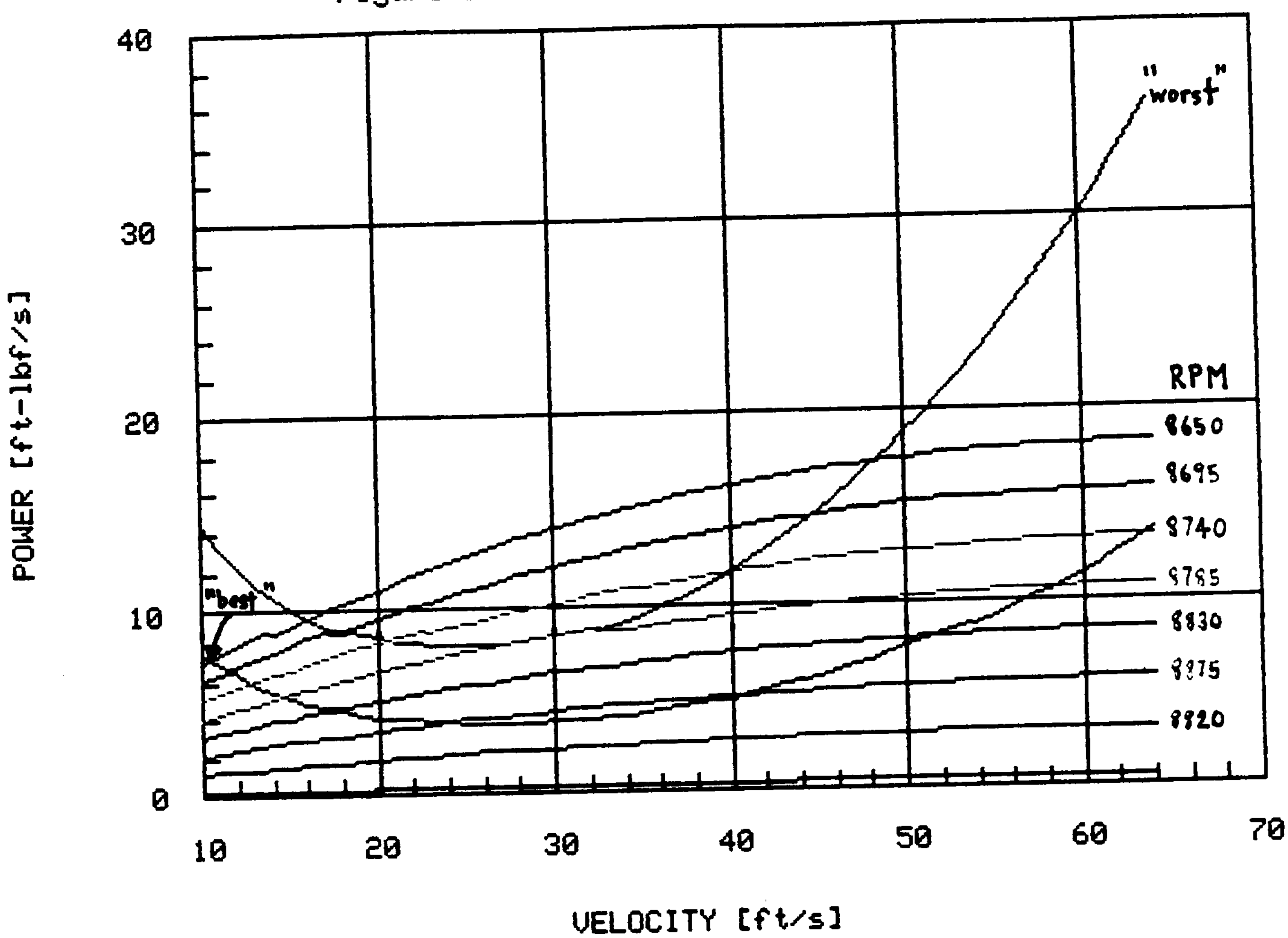


Figure ES-3: Power Balance, Astro 05

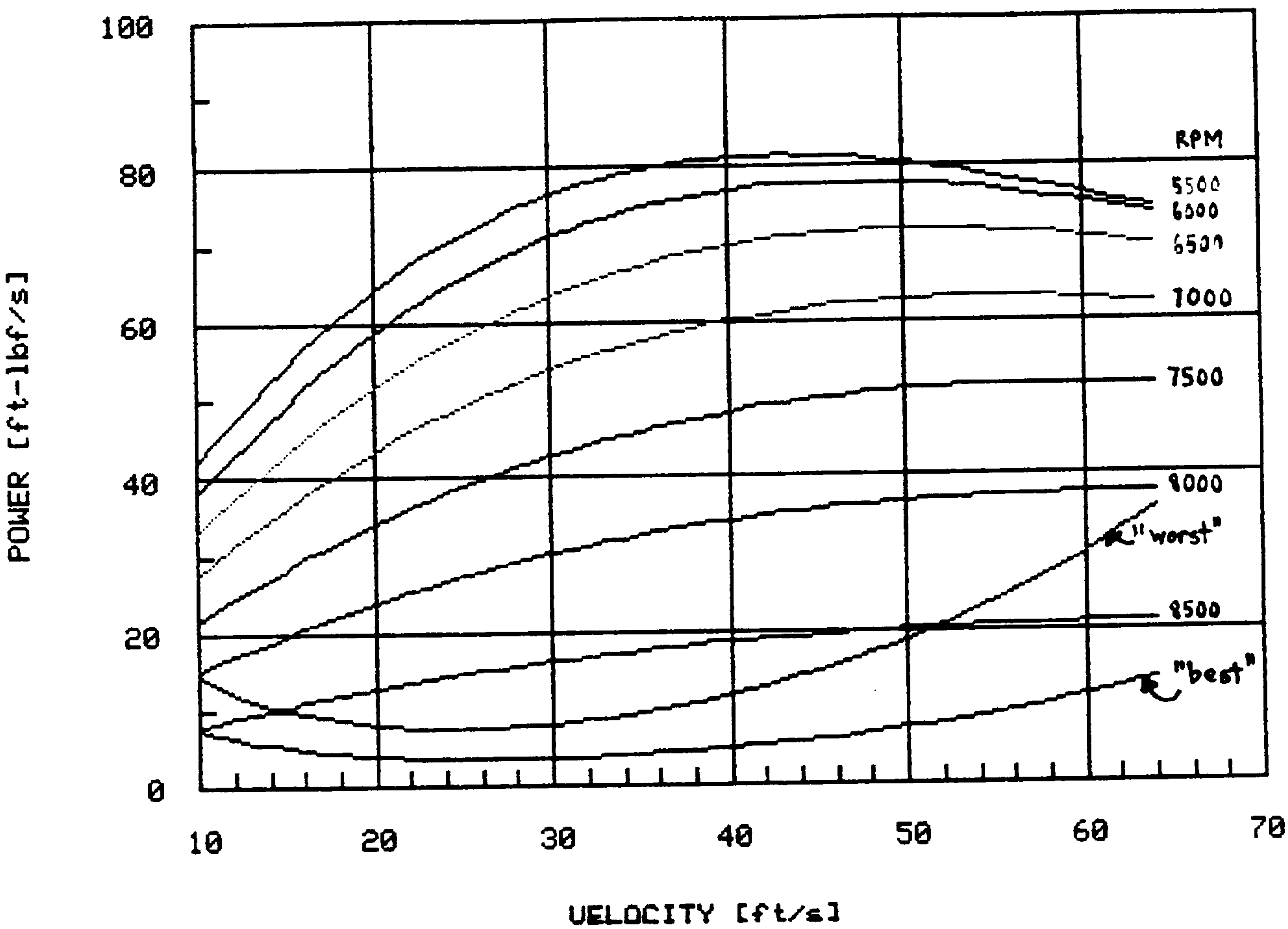


Figure E5-4: Power Balance, Astro 15

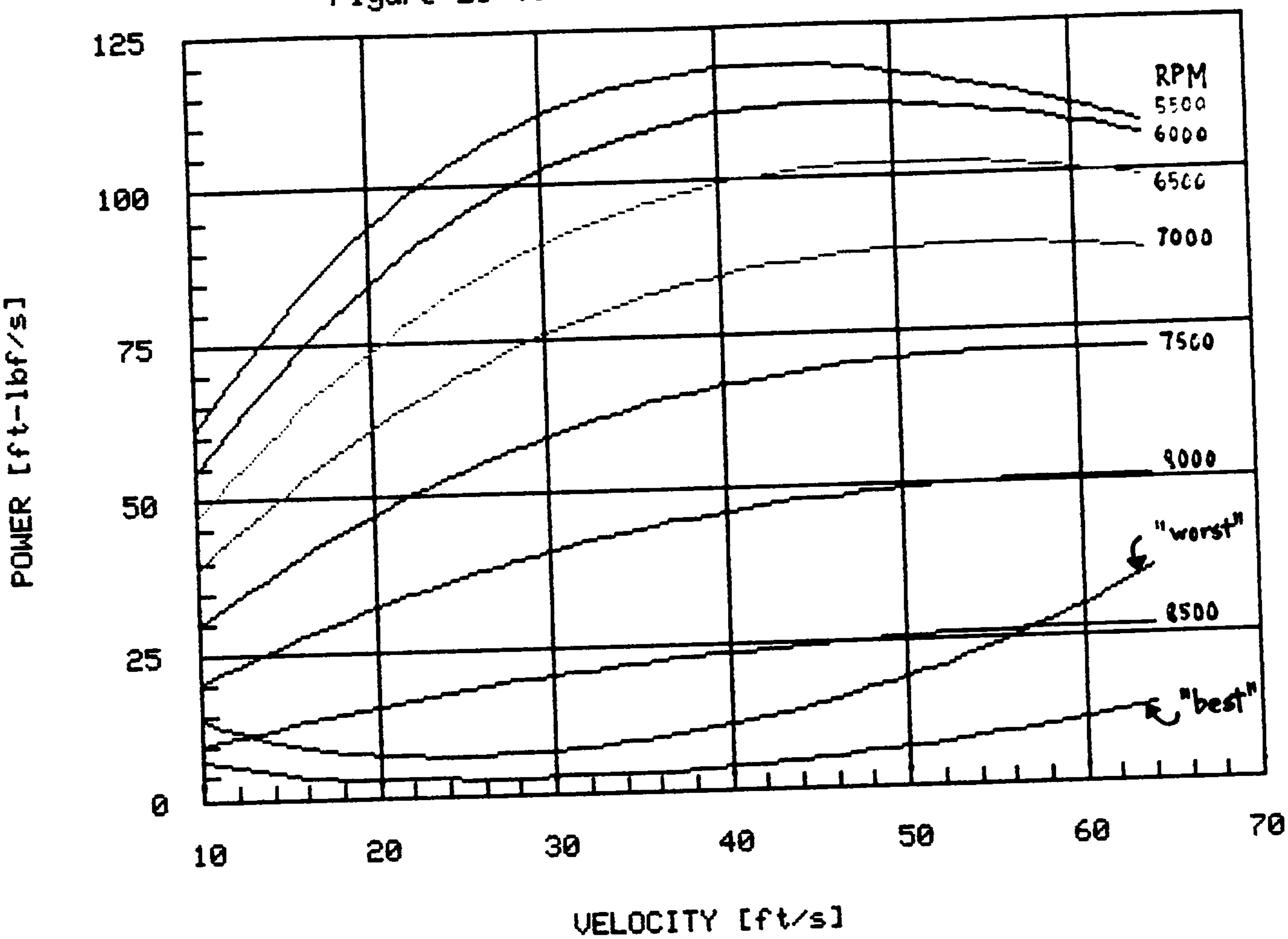


Figure ES-5: Power Balance for Steady Level Flight Propeller Method Astro 15

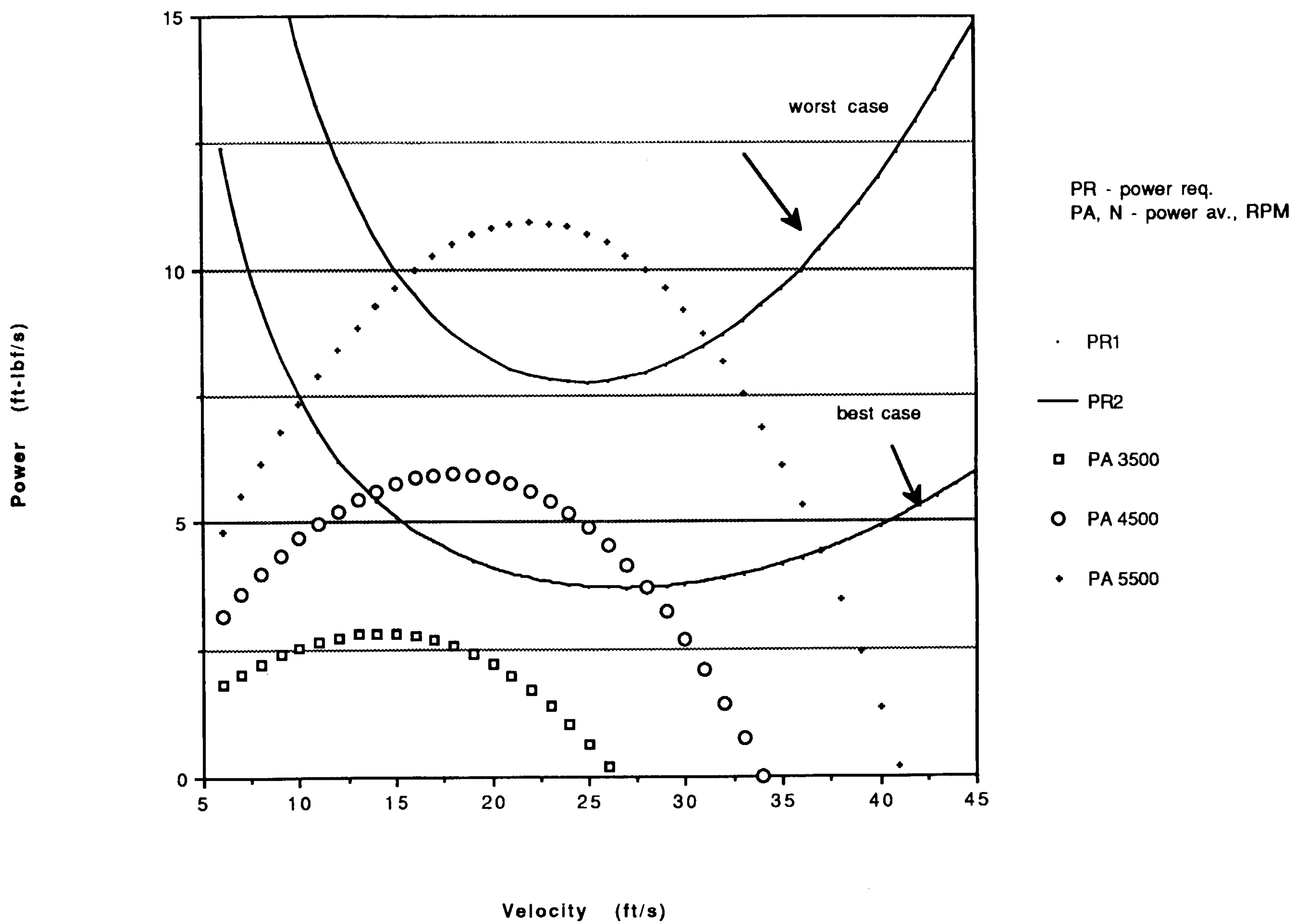
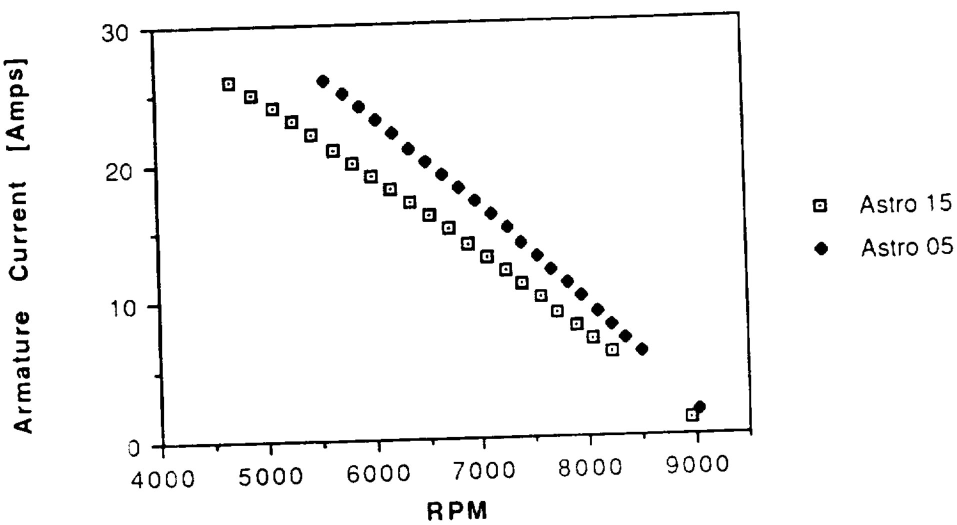


Figure ES-6 : Current Draw Comparison



New York: Ronald, 1937]: T [lbf] = (29000•SbHp) / (RPM•D), where SbHp is the gear power in horsepower, and D is the propeller diameter in feet. Using this relation, the maximum static thrust for the Astro 05 and 15 (8" prop) are, respectively, 1.4 lbf and 2.5 lbf. The actual installed static thrust will likely be about as much as 20% lower. As acceleration of a 3.3 lb aircraft to 20 ft/s after 75 ft requires a minimum force of about 0.3 lbf (neglecting ground roll friction and drag), both engine systems are more than sufficient. See part two for a more detailed coverage.

Cost of Propulsion System

A nominal system comprises only the motor and gearing. A complete propulsive unit involves a speed controller, batteries, wiring, propeller and spinner. A new *Astro Challenger Cobalt* 05 (geared) motor costs $90.24 with postage and handling, fully 100% of the $90 dollar funding (students were, however, allowed to spend an additional $90 of their own money). With batteries, the system will cost even more. On the other hand, the Aerospace Department already owns two Astro Challenger Cobalt 15's (geared with engine mount), and will allow their use at no cost. The deal is valued at $100.24 [all values quoted from Tower Hobbies, Tower Talk Catalog, 28 February 1990, and include postage and handling]. Cost efficiency was determined to be an important group objective (see the concept selection section).

Part I Summary

So far, the Astro 05 and 15 electric engines were evaluated on the basis of emission, weight, performance at cruising speeds, climbing, power draw, static thrust, and system cost. On the basis of these criteria, both electric engines possessed advantages and disadvantages. A collection of the results is listed in the following table:

Table ES-4: Summary of Trade Study: Engine Selection

Performance Criterion:	Best System for Specified Performance: Astro 05	Astro 15	Equal
Zero Emission			√
Weight	√		
Cruise Capability			√
Climb Capability		√	
Power Draw		√	
Static Thrust		√	
Cost of System		√	

The Astro 15 motor was eventually selected as the engine of choice, easily meeting all the performance-based measures of merit (see table ES-1). In keeping with the DRO target weight of approximately 3 pounds, it was decided that a sacrifice in weight was justified when compared to higher power capacity and more importantly, system cost.

Presently, the design incorporates the Astro 15 engine in a conventional nose mounting. The power required will tend towards the "best" case (black) line on the Power Balance curves, due to an increase in the aircraft's aspect ratio to 10.5. The aircraft will cruise with an RPM in the range 4250 to 5350, producing sufficient excess power for a maximum R/C = 12 ft/s, and can fly at speeds ranging from 17 to 40 mph.

Part two of Propulsion System Selection follows with a discussion of propeller selection based on takeoff performance, among other factors.

Part II: Propeller Selection (PS)

Historical Background

The Wright Brothers, forefathers of aeronautics left a profound impact on propeller development. The difficulty of propeller development is best summed up by Orville's foreboding remarks,

> "It is hard to find even a point from which to start, for nothing about a propeller, or the medium in which it acts stands still for a moment. The thrust depends upon the speed and the angle at which the blade strikes the air; the angle at which the blade strikes the air depends upon the speed at which the propeller is turning, the speed the machine is traveling forward, and the speed at which the air is slipping backward; the slip of the air backward depends upon the thrust exerted by the propeller and the amount of air acted upon. When any of these changes it changes all the rest, as they are all interdependent upon one another. But these are only a few of the factors that must be considered...."
>
> Orville Wright†

Introduction

Fortunately, with the advent of high speed computers the difficulty the Wright Brothers encountered with variability is simplified. Complex computer programs can model both the propeller and the flow surrounding the propeller. Although there is still a great deal of variability involved, tedious experimentation is reduced to broad looping calculations. A series of experiments can be run in considerably less time than the Wright Brothers ran theirs.

Method

The computer played a major role in our propeller analysis. A Simple Blade Element Theory program with modifications represented the means

† John D. Anderson, *Introduction to Flight*, McGraw Hill Book Company, NY., 1985 pg 492.

by which we were to solve the propeller problem. Values obtained through the simple blade element program were utilized in a takeoff program which yielded the answers to our propeller problem.

A database established with various commercial propellers was used to extrapolate such information as the thrust coefficient, power coefficient and efficiency versus advance ratio. Some of this data was then inserted into a takeoff program which resulted in determining our static thrust, takeoff distance, battery drain at takeoff, and cruising propulsive efficiency. After comparing the propellers we were able to select the one that best fulfilled our mission and had the best qualities in our flight regime.

Propeller Algorithms

Simple Blade Element Theory Program†

The simple blade element program had the capabilities of analyzing a variety of commercially available propellers. These included the Zinger J, Master Airscrew, TopFlight, and Tornado Series propellers. Previous measurements of the blade pitch distribution, blade chord length distribution, and blade thickness distribution for the commercial propellers were placed in a database.

Analysis was done on the four groups of propellers. This analysis included varying the blade diameter from eight inches to twelve inches varying the number of blades between two and three, and varying the RPM ranges to include our entire flight regime.

The program was able to be modified in its analysis. To simulate real flight conditions we included effects of tip losses and induced velocity. In addition, we chose the NACA 44XX LOWRE to model our propeller cross section. The NACA 44XX LOWRE is an excellent model for low Re flight. One final modification allowed us to make Mach and Reynolds number adjustments in the analysis. The resulting C_P and C_T data were implemented in the takeoff program

† The Simple Blade Element Theory Program was written by Barry Young as a graduate student at the University of Notre Dame.

Takeoff Program††

The takeoff program enables us to utilize the data obtained in the simple blade element program to estimate those "parameters of importance" (i.e. static thrust, battery drain, and takeoff distance) for the propellers previously mentioned. This program takes certain critical pieces of data such as weight of the airplane, Cl cruise, and ground friction coefficient, m, to determine those "parameters of importance". Comparison of the propellers yields the desired propeller.

Results

The Zinger 10-4 was selected as our propeller for the "Station Keeping Mission". The Zinger 10-4 is a ten inch, two blade propeller with a four inch pitch. Pitch of the propeller was determined from Figure PS-5 taken from pg. 24 of A.G. Lennon's R/C Model Airplane Design. This figure graphs flight velocity, cruise RPM and blade pitch on separate vertical axes. By simply drawing a straight line through our desired cruising RPM and flight velocity, a specific pitch results. In this case the cruising RPM is approximately 4500 RPMs, the velocity is 17mph and, therefore the pitch is 4".

Figures PS-1 and PS-2 on the following two pages show the results of the propeller analysis. The Zinger 10-4 has excellent qualities in our flight regime and was eventually chosen as the propeller for the Penguin based on the following results. The Zinger 10-4 has more than the minimum required static thrust to overcome ground friction and achieve takeoff. This analysis only served to dispose of one of the propellers out of the group of eight, the Zinger 8-4. It was noticed, however, that the Zinger 10-4 was among the four best in this category. This did not necessarily rule out the others that were not among the top four. Further analysis was needed.

†† The Takeoff Program was written by Dr. Stephen Batill of the University of Notre Dame.

FIGURE PS-5

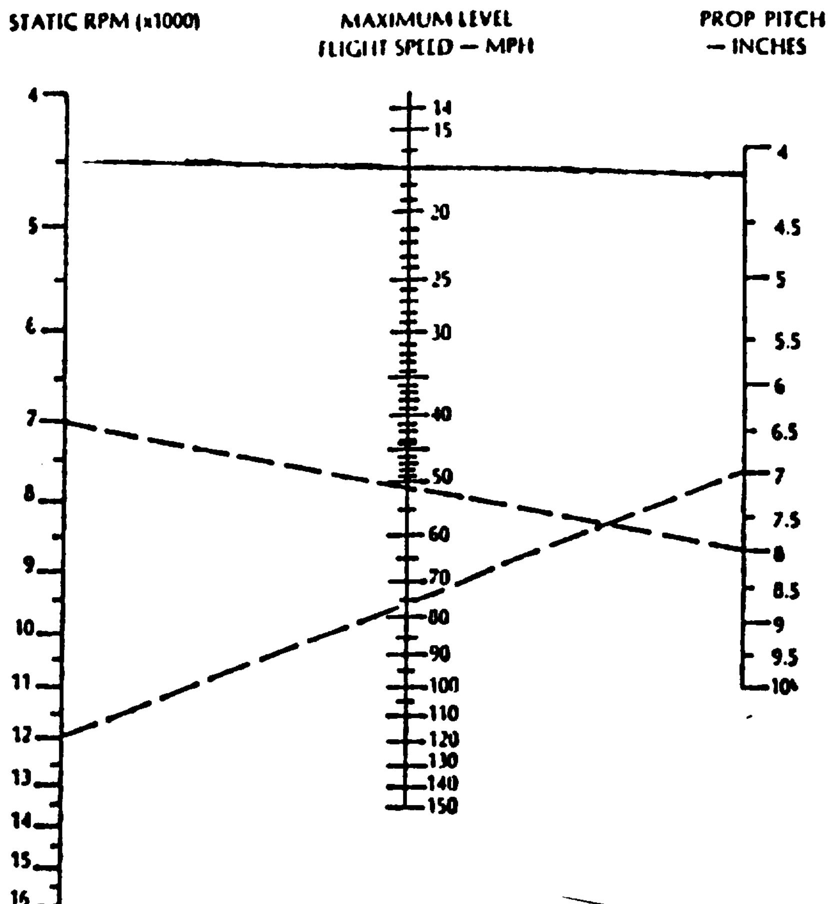
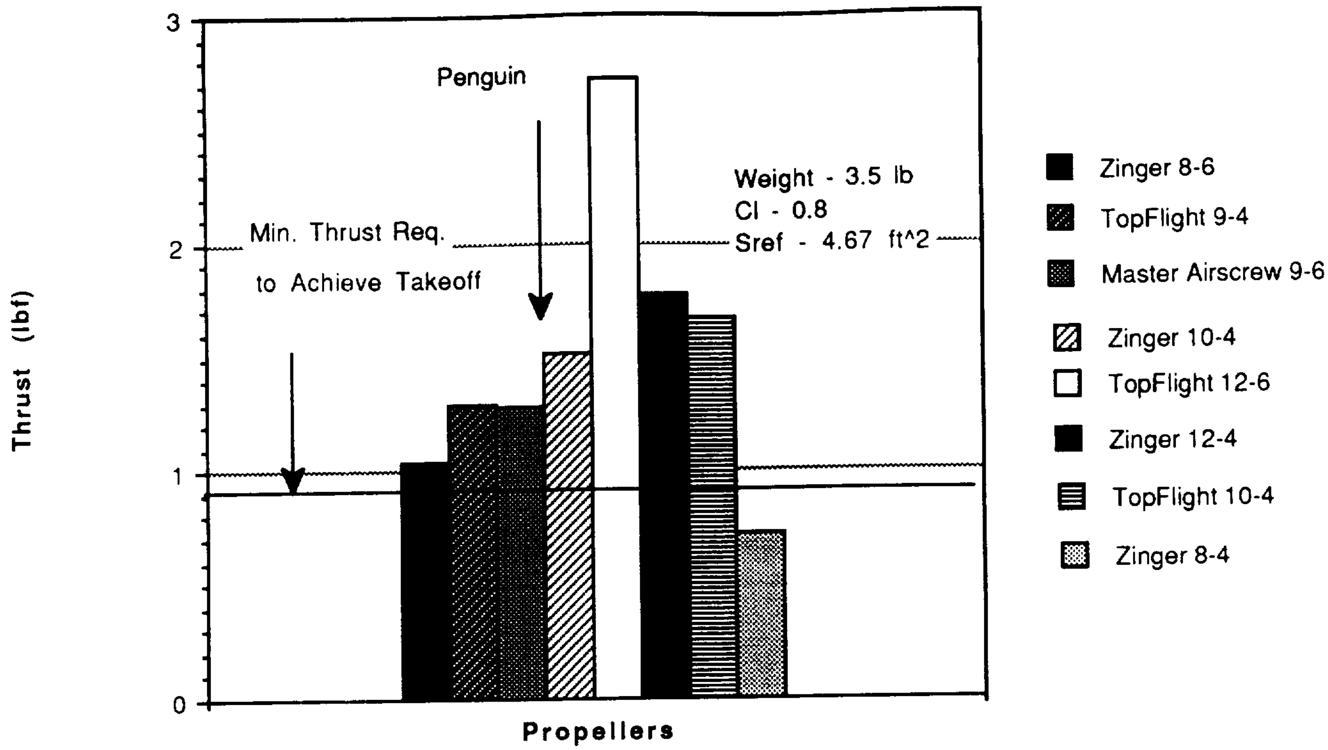


Fig. 1-15. Nomograph for quick determination of rpm, pitch, and speed.

Figure PS-1:
Maximum Available Thrust For Various Props



Takeoff Distance for Various Propellers

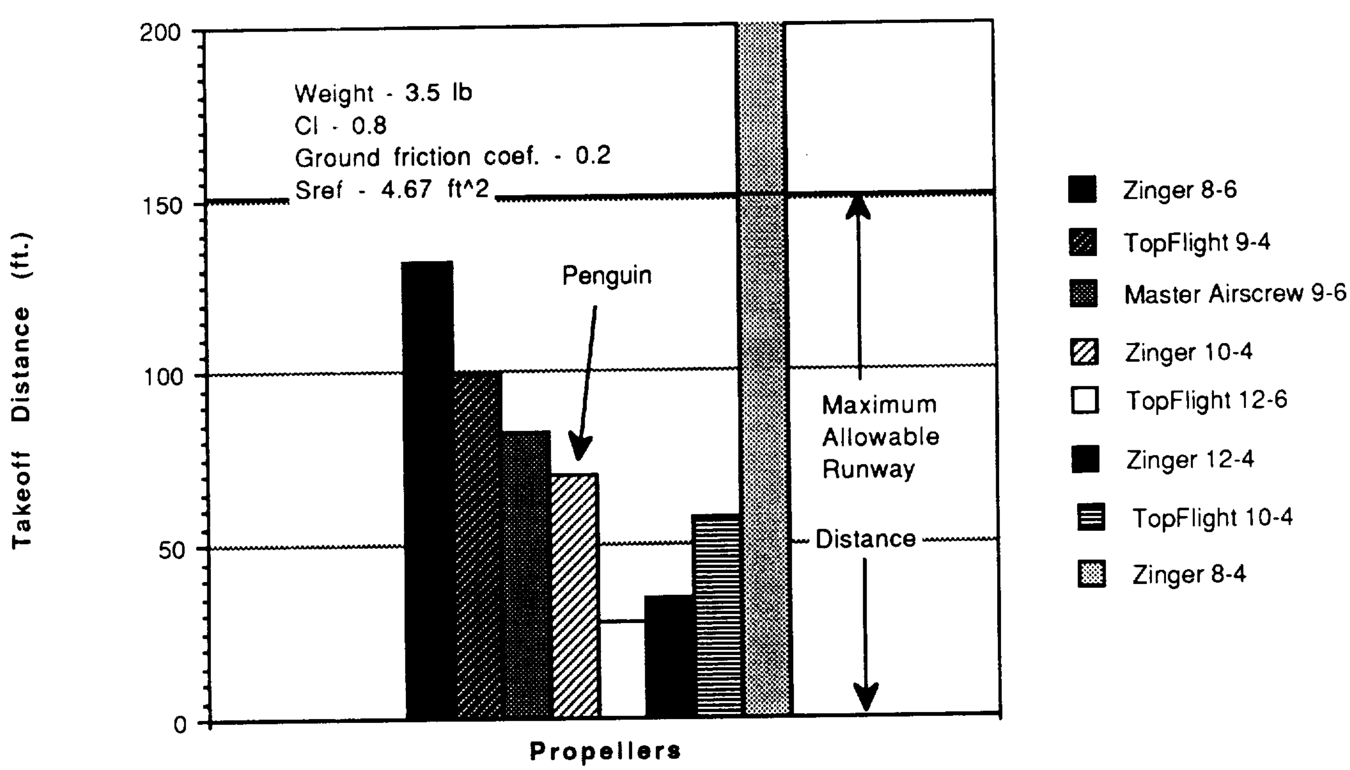
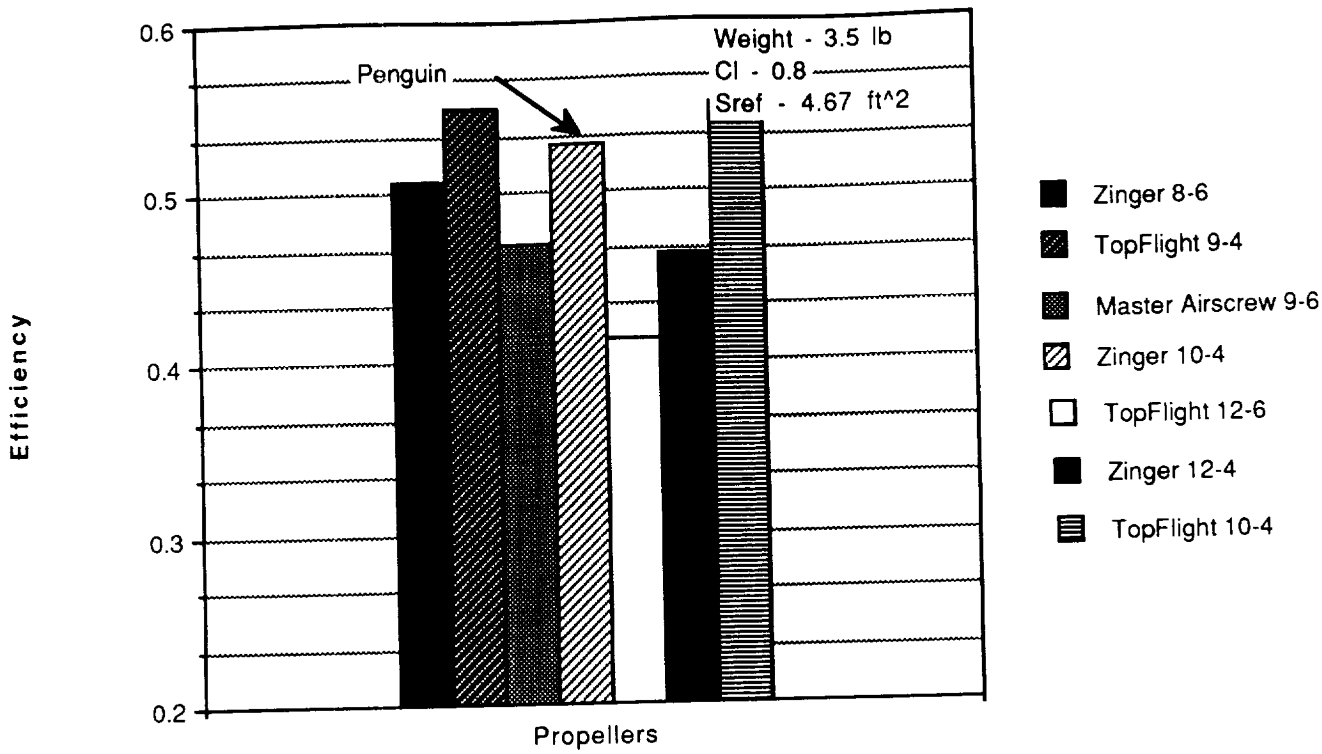
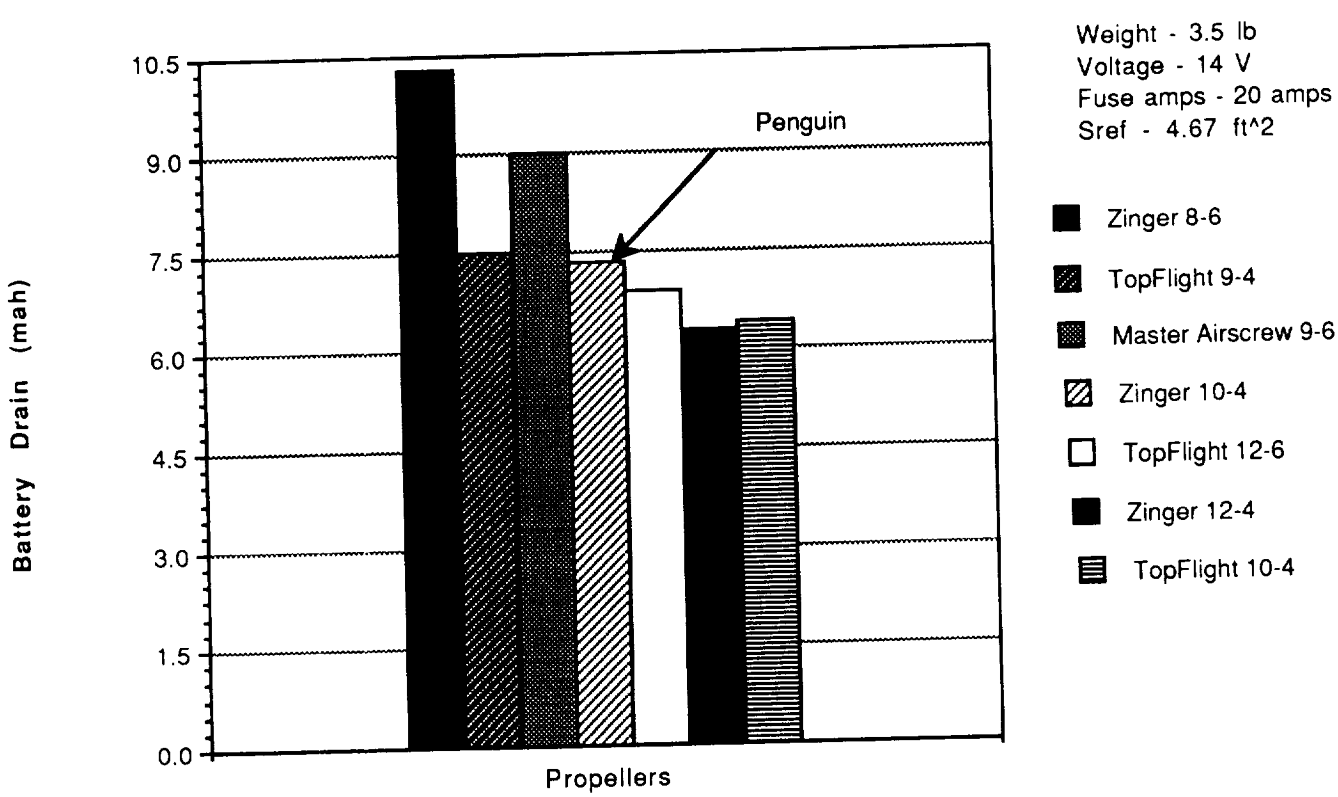


Figure PS-2:

Efficiency for Various Propellers at Cruise V∞



Battery Drain for Various Propellers at Takeoff



By using the Zinger 10-4, the Penguin is able to become airborne well under the maximum runway distance of 150 ft. The objective of the mission was to get the Penguin airborne as soon as possible to achieve steady level flight at altitude before the first turn. The design objective chosen was 75 ft as the runway distance limit to achieve takeoff. This ruled out the Zinger 8-6, TopFlight 9-4, and Master Airscrew 9-6.

The efficiency was then examined. The efficiency at cruise is relatively high for the TopFlight 10-4 and the Zinger 10-4. These two propellers were the only ones left to be compared. The battery drain at takeoff is moderate for the Zinger 10-4 but is slightly greater than that of the TopFlight 10-4.

The final question that had to be examined was availability. It was a bit disheartening to find that the TopFlight 10-4 was not commercially available anymore so the Zinger 10-4 seemed to be a logical replacement with its fine performance in the comparisons. The Zinger 10-4 was therefore, chosen as the propeller for the Penguin and the station keeping mission.

Part II: Summary

Structural constraints, as well as availability, had an impact on our propeller selection. Although the TopFlight 10-4 had the best overall characteristics, we were not able to locate one and, therefore, decided on the Zinger 10-4. The twelve inch propellers can be seen to have similar capabilities as the Zinger 10-4. The structural benefits including shorter landing gear length have justified the use of the ten inch propeller over the twelve inch propeller. A collection of the results is listed in the following table:

Table PS-3: Summary of Trade Study: Propeller Selection

Performance Criterion: (Diameter)	Best System for Specified Performance: 8"	9"	10"	12"
Static Thrust	4	3	2	1
Power Required to Turn Propeller	1	2	3	4
Structural Considerations	1	2	3	4
Battery Drain at Takeoff	4	3	2	1
Efficiency at Cruise	3	1	2	4
Current Draw at Takeoff	1	3	2	4
Takeoff Distance	4	3	2	1
Totals:	**18**	**17**	**16**	**19**

From the preceding results the 10" propeller has the lowest score and, thus, the greatest potential advantages inherent in its use. In addition an analysis was taken into account to determine the advantages and disadvantages associated with two or three blade propellers. These results are summarized in the following table PS-4.

Table PS-4: Summary of Two and Three Blade Propellers

Advantages	Two Blade	Three Blade
Thrust Available		√
Power Required to Turn Prop.	√	
Availability	√	
Torque on Propeller	√	

Therefore, as can be seen the two blade propeller has greater advantages than the three blade propeller and can be justified in its use. In summary, we have used two computer programs which have enabled us to determine the best propeller for our mission, the Zinger J 10-4, two blade propeller.

Stability & Control

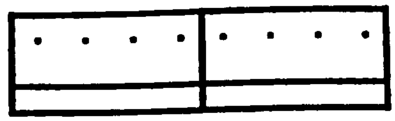
This chapter will discuss the stability and control of the Penguin. Since the it is typical of most sailplane designs, it employs aft horizontal and vertical tails for pitch and yaw stability. Roll stability is provided by a wing dihedral. Control of the vehicle comes from the three primary control surfaces: elevator, rudder and ailerons. We have access to a four-channel receiver, and therefore decided that a coordinated turn using both ailerons and rudder would be wise for this difficult mission.

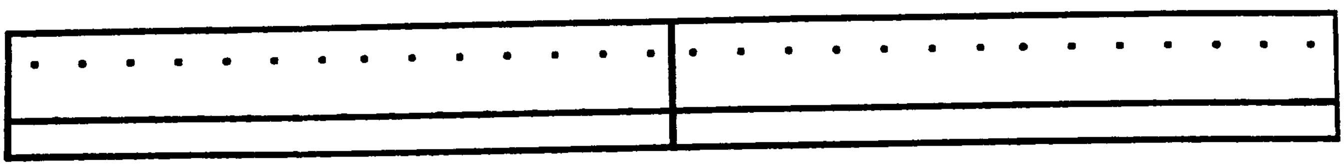
Static stability was the first concern under the topic of stability and control. After preliminary estimates were made using empirical formulas [1], they were then fine-tuned using more exact methods such as a vortex simulation computer program [2] and analytical expressions [3]. After the values were finalized, they were checked against the limits given by a large database of RPV's of similar type [4].

Longitudinal static stability was considered first. It was decided that the center of gravity (CG) would be placed aft of the aerodynamic center and as close to the neutral point as possible while providing a comfortable static margin of 20% of the wing chord. This was done to insure that the horizontal tail would generate as small a lifting force as possible. Since the horizontal tail is an inefficient lifting surface that creates a large amount of induced drag, it's lift was kept to a minimum. The CG was placed at 30% chord aft of the leading edge of the wing, a value that is standard for most vehicles of this type. A simple moment balance determined that the tail would actually have a small upward lifting force.

Horizontal Tail Sizing

The finite element program, Lin Air™, allowed us to model the wing and horizontal tail surfaces in the flow along with their interference effects through a vortex simulation routine. This program requires that the geometry and performance characteristics of the surfaces be entered through a data file (see SB-1.) Once the parameters for the wing (which were fixed by the aerodynamics group) and the horizontal tail (which were to be varied) were entered, Lin Air provided useful data about the lift and





Element #1 (Wing):

Span 7 ft.
Area 4.67 ft^2
#panels 14

Element #2 (Tail): Span 2 ft.
Area 1.04 ft^2
#panels 4

moment coefficients of the aircraft. Knowing the desired performance characteristics of the Penguin, we were able to fix the horizontal tail area at 150 in^2, its distance from the CG at 34.4 in, its span at 24 in and its pitch at positive 4° with respect to the fuselage reference line. These values were decided upon because they provided trim at cruise, they yielded a negative moment curve slope, and they provided for a horizontal tail volume ratio of 0.96 which is within the range specified in [4]. Uing the program fcilitated the development of parametric sweeps. The results of the sweep used to determine the appropriate tail ange are presented in SB-2 and show that a tail angle of 4° will provide trim at the cruise condition.

The Lin Air™ program, however, does not take several effects into account. First, it cannot simulate the contribution to the pitching moment of either the fuselage or the wing. Though it was later determined that the pitching moment slope contribution by the fuselage was small enough to be considered negligible (-.0003 /deg), the moment caused by the wing could not be ignored. Also, since the moment coefficient of the wing was negative (-0.08) the horizontal tail did not have to provide as much lift as the Lin Air program calls for, and the actual angle of attack for the horizontal tail will only be 1°. The computer program also does not account for the effect of propwash over the horizontal tail which will serve to increase the dynamic pressure that the tail sees due to higher velocities over the surface. Having a smaller tail angle will also account for this by reducing the possibility of a strong nose-down pitching moment due to increased lift on the tail from propwash.

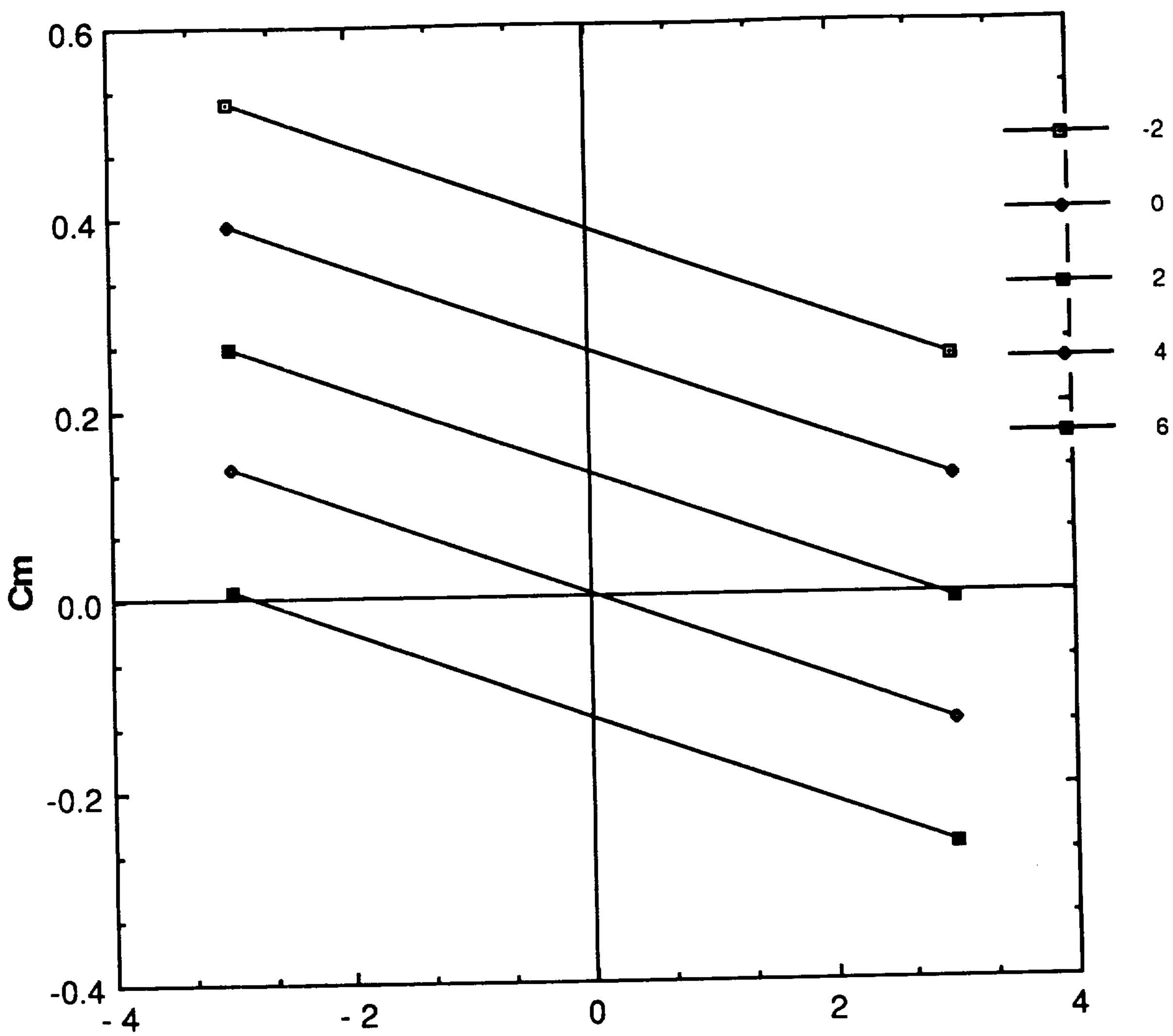
The slope of the pitching moment curve is -0.044/° for the above mentioned configuration.

Vertical Tail Sizing

When determining the vertical tail size, it was found that there are no hard analytical expressions for producing exact values of area, height, and taper. There do exist, however, formulas based upon previous designs that may be utilized to size the vertical tail. These were used in our design.

From [1] it was determined that the vertical tail area would be 60 in^2. When this value was compared against the ranges given in the database [4] it appeared to be too low. Based on the expressions given in the database, a final value of 53 in^2 was chosen. This yields a yaw angle coefficient slope of

Effect of Tail Angle



Alpha

SB-2

1.15/rad which is computed in SB-3. A positive value is desired for this parameter and since the fuselage is very thin, it will not provide much negative yaw stability. The vertical tail with a volume ratio of 0.066 is large enough to overcome the negative stability effect of the fuselage.

The height of the vertical tail was also determined through the range given in [4]. It stands at a height of 7.5 inches up from the base of the fuselage.

Elevator Sizing

After sizing the empennage, the next step was to determine the size and travel of the control surfaces. The critical condition which will require maximum elevator power are the takeoff and landing phases of the flight. Although it is likely that the aircraft will lift off the ground from its three point position, it may be necessary to lift the tailwheel from the ground during the takeoff roll. Using a suggested [1] elevator area of 52.5 in^2 (35% of the horizontal tail area), the elevator effectiveness was set at .0093/°. This is sufficient to lift the tail off the ground during roll as well as provide trim at the expected flight attitudes. The elevator will have a travel of +30°/-20° which, for the above mentioned elevator effectiveness, is ample for the entire flight regime.

Rudder Sizing

As with the elevator, the rudder has to be able to provide enough moment to overcome any undesirable yaw attitudes that would cause the RPV to sideslip. After consulting the references and examining previous designs, it was found that most aircraft of this type have rudders that are approximately 25%-50% of the vertical tail area. Aircraft that relied on wing dihedral and not ailerons to cause banking in turns had larger rudders than those with ailerons, as would be expected. Since the Penguin will be flying at extremely low speeds and hence will have very little dynamic pressure acting on the empennage, a rudder size of 40% of the vertical tail area, or 42 in^2 was chosen. Although this is oversized for an aircraft that makes use of ailerons, extra control power is desireable when flying at low speeds that are near the stall speed. It is also desirable to have extra control power when taxiing and while flying in an indoor facility. The rudder will have a 20° travel in either side of the vertical tail.

C_{n_β} calculation:

Wing + fuselage contribution

$$C_{n_{\beta wf}} = -K_n K_{R1} \frac{S_{fs}}{S_w} \frac{l_f}{b}$$

$$k_n = .001 \qquad [3]$$

$$K_{R1} = 1.0 \qquad [3]$$

$$C_{n_{\beta wf}} = (-.001)(1.0)\frac{105\text{ in}^2}{672\text{ in}^2}\frac{42\text{ in}}{84\text{ in}} = -7.8\times10^{-5}/\text{deg} = -4.47\times10^{-3}/\text{rad}$$

Vertical tail contribution

$$C_{n_{\beta v}} = V_v \eta_v C_{L\alpha v}\left(1+\frac{d\sigma}{d\beta}\right)$$

$$V_v = 0.38 \qquad C_{L\alpha v} = \frac{2\pi}{\pi(.96)} = 2.084$$

$$\eta_v\left(1+\frac{d\sigma}{d\beta}\right) = .724 + .306\,\frac{60.48/672}{1+\cos 0^\circ} + .4\,\frac{2.5}{5} + .009(10.5) = 1.156$$

$$C_{n_{\beta v}} = (.38)(2.084)(1.156) = 1.16/\text{rad}$$

$$C_{n_\beta} = C_{n_{\beta wf}} + C_{n_{\beta v}} = -4.47\times10^{-3}/\text{rad} + 1.16/\text{rad}$$

$$\underline{C_{n_\beta} = 1.15/\text{rad}}$$

Aileron Sizing

The ailerons will be placed on the outbaord sections of the trailing edge of the wing and will allow the RPV to make a coordinated turn with a minimal loss of altitude. This is crucial to this particular mission because the aircraft will be flying so close to the ground. It was observed in previous designs that the when relatively short (spanwise) ailerons were placed on the outboard sections of the wing that the section where the ailerons were located would twist when they were deflected, thus severely decreasing their effectiveness. The ailerons on the Penguin comprise 12% of the total wing area and extend across four ribs of the wing. This will distribute the increased wing load encountered during deflection and minimize the possibility of any one wing section experiencing twist. The ailerons do not extend out to the wing tips but are located four inches inboard of the tip. This was done for structural purposes, in order to maintain the strenght of the wing tips.

Summary

Penguin has been designed to be both stable and highly maneuverable. It makes use of a long fuselage, large empennage, and large control surfaces with adequate travel to accomplish this. The combination of ailerons, rudder, and elevator will allow it to perform coordinated turns around the pylons, while the inherent stability of the RPV will insure that it will be easy to fly.

References

1. Lennon, A. G., R/C Model Airplane Design, 1986 by Markowski

2. Lin Air, finite element software

3. Nelson, R. C., Flight Stability and Automatic Control, 1989 by McGraw-Hill

4. Stability and Control, handbook of ranges for RPV surfaces design

Cost Estimate

The following is a cost estimate for the various parts of The Penguin:

Structures	
mylar skin	$25
wood	$80
balsa	$30
ply-wood	$20
spruce	$30
landing gear	
struts	$12
wheels	$8
structure total	$125
Propulsion	
batteries	$45
electric engine	$90
propeller	$ 5
propulsion total	$140
Controls	
control rods and hinges	$15
radio with 4 servos	$300
Controls total	$315
Man Hours	
Design (@ $10/hr)	$3000
Fabrication (@ $10/hr)	$2500
Man Hours total	$5500
TOTAL	**$6080**

As can be seen by this chart, the major contributor to the cost of The Penguin is the man hours necessary for the design and the fabrication. It should be noted that the above cost does not include the overhead costs, which equipment costs and work-place costs.

Technology Demonstrator

The Technology demonstrator is full scale prototype of the design proposal. During the construction phase of the Penguin, however, certain adjustments to the design had to be made due to lack of available materials and lack of construction expertise. Because of these adjustments the Penguin prototype weighed 12% more than the proposed weight. Explanations for the added weight are:

1) Unavailability of the proposed spar caps forced the selection of a spar cap which was twice as large as the design,
2) The necessity of extra balsa wood near the rear of the fuselage to support the mylar Monokote™,
3) The weights of control rods and hinges were not included in initial weight estimations

Although a 2 oz spinner did have to be added to the plane for static stability purposes, this should not have effected the weight values since a 2 oz payload (which was never put in the plane) was accounted for in the original weight estimate. A final component weight breakdown can be found in Table TD-1. Since the Penguin is designed to be overpowered, the 12% increase in weight, while increasing the necessary velocity (and thus the Reynold's number), will still allow for flight within the target Reynold's number regime of 100,000 to 200,000. The required flight velocity increase due to the increased weight is only 2 ft/s (see figure PF.1), and allows for flight at a Reynold's number of 108,000.

Upon completion of the Penguin, it was found the C.G. was 0.75 inches aft of the desired position (30% of the mean chord). This was a result of the additional weight of the control rods and horizontal tail. This additional weight was in the rear of the plane where the center of gravity was highly sensitive to small weight changes. In order to remedy this situation the engine was extended 1.5 inches within the engine mount and a 2 ounce, solid brass spinner was used instead of a lighter weight plastic spinner. Although this did move the C.G. forward, it was determined that an

Table TD-1: Penguin Component Weight Breakdown

Component	Estimated Weight	Actual weight	Percent Difference
avionics	4.8 oz	4.8 oz	0 %
propulsion system	20.3 oz	20.3 oz	0 %
payload	2.0 oz	0.0 oz	NA
spinner	0.0 oz	2.0 oz	NA
fuselage	9.6 oz	10.0 oz	4.2 %
left wing	4.3 oz	6.4 oz	48.8 %
right wing	4.3 oz	6.1 oz	41.9 %
vertical tail	0.5 oz	0.5 oz	0 %
horizontal tail	1.0 oz	1.2 oz	20.0 %
front gear	3.5 oz	2.4 oz	-31.4 %
rear gear	0.5 oz	0.5 oz	0 %
control linkages	0.0 oz	2.0 oz	NA
total weight	50.7 oz	56.2 oz	10.8 %

additional ounce of lead ballast needed to be secured to engine cowling to fix the C.G. at the proper location of 30% chord.

The final version of the Penguin did employ the three wheeled, tail dragger configuration with the positions of the gear as indicated in Figure TD-3. The front gear consisted of a 0.125 inch diameter steel strut and a 2.25 inch diameter Air-Lite™ wheels. The strut diameter was twice as large as the design because of its availability at no cost. The front gear's total weight including mounting plate was 2.4 oz. The rear gear was attached to the rudder and constrained by a sleeve attached to the fuselage. This sleeve (rather than the rudder) absorbs the loads placed upon the rear wheel yet still allows the gear to rotate freely. The final weight of the rear gear including the sleeve was 0.5 oz. The total landing gear weight of the fabricated Penguin was 2.9 oz which is 1.1 oz less than the predicted value. A complete table of the final configuration of the Penguin compared to the design configuration can be seen in table TD-2.

Summary of Dimensions of the Technology Demonstrator

	estimated	actual
WINGS		
Wing Area	4.67 ft^2	4.67 ft^2
Wing Loading	10.7 oz/ft^2	12.0 oz/ft^2
Wing Span	7 ft.	7 ft.
Wing Chord	8 in.	8 in.
Wing Planform	rectangular	rectangular
Aspect Ratio	10.5	10.5
Wing Location	High	High
Dihedral	3°	3°
HORIZONTAL STABILIZIER & ELEVATOR		
S_s/S_w %	22.3%	20.4%
Horizontal Stabilizer Area	150 in^2	132 in^2
Horizontal Stabilizer Chord	6.25 in.	6.00 in.
Horizontal Stabilizer Span	24 in.	22 in.
S_e/S_s %	35%	33%
Elevator Chord	2.2 in.	2 in.
Elevator Span	24 in.	22 in.
Tail Length (C.G. to tail A.C.)	34.4 in.	34.4 in.
Horizontal Tail Vol. Ratio	.96	.84

VERTICAL STABILIZER AND RUDDER		
S_v/S_w %	8.0%	7.7%
Vertical Tail Area	53.8 in^2	52 in^2
Vertical Tail Vol. Ratio	.066	.064
Vertical Stabilizer Mean Chord	7.2 in.	6.9 in.
Vertical Stabilzer Height	7.5 in.	7.5 in.
S_r/S_s %	50%	35%
Rudder Chord	3.6 in.	3.0 in.

HARDWARE	
Motor Size	Astro 15 Cobalt
Battery Pack	16 V, 250 mahs
Landing Gear	Steerable Tail Dragger
Wheel Material	Plastic
Tire Material	Rubber
Front Gear Tire Diameter	2.25 inches
Tail Gear Tire Diameter	1 inch
Front Landing Gear	1/8 in. piano wire
Tail Landing Gear	1/16 in. piano wire
Control Funtions	Elevator, Rudder, Aileron, and Motor
Aileron Actuation	Control Rods
Elevator Actuation	Control Rod
Rudder Actuation	Control Rod
Motor Actuation	Speed Controller

Table TD-2

Flight and Ground Test Results

The preliminary ground test plan to test the Penguin's maneuverability, and to make sure it could be adequately controlled during take-off and landing procedures. To accomplish this, low speed taxi tests were performed by the Penguin. As a result of these tests, it was determined that the Penguin could be adequately controlled during the ground phases of its mission.

The next step in checking the Penguin's performance were take-off and landing tests. The goal of these tests was run the Penguin at full power, lift it off the ground were it would sustain steady-level flight for a couple of seconds, and then land it. The results of these tests are as follows:

1. From the initial take-off and landing tests it was found that the left wing was providing more lift than the right wing, causing the plane

Landing Gear Positioning

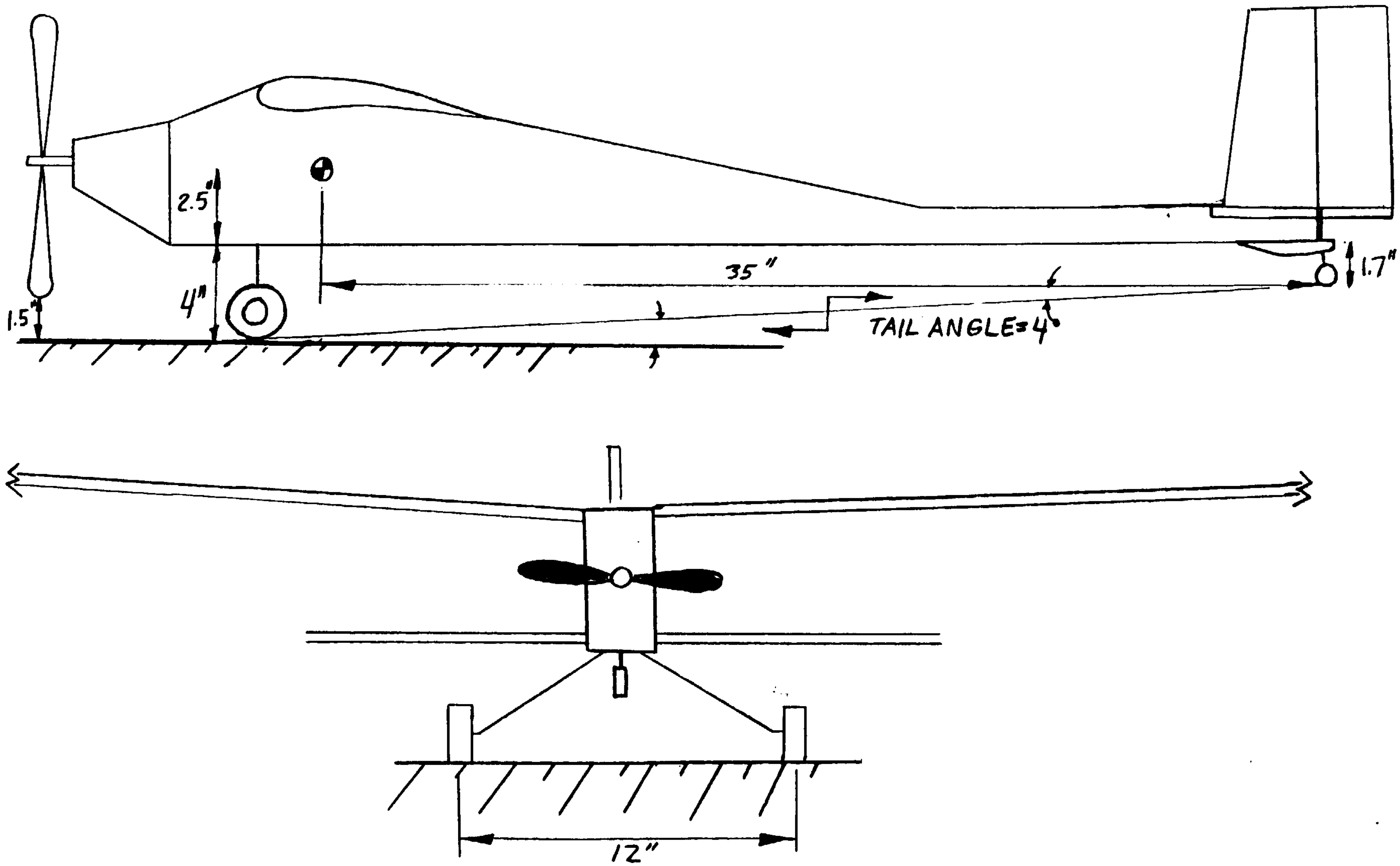


Figure TD-3

the ground loop. Initially this was felt to be result of asymmetric aileron deflection.

2. After the ailerons were adjusted to yield a considerable increase of lift in the right wing, there was no noticeable decrease in ground looping.
3. Through a simple weight balance, it was found that the right wing was heavier causing the plane to rotate about its axis. This was corrected by the addition of ballast (pocket change) to the left wing. This did allow for a short period of wing's level flight, but was still difficult to trim.
4. Upon inspection of the wings, it was thought that the right wing might be stalled due to wash-in off the wing, which radically decreased the lift of the wing. This was corrected by the addition of wash-out of the right wing by simply reheating the mylar skin while twisting the wing to a more suitable angle. The results of his change are best summed up by pilot Joe Mergen, "This is an entirely different plane. If I had the space to properly trim, I could fly this plane hands-off!"
5. The results of these flight tests were felt to be sufficient to warrant execution of the planned mission.

Mission Flight Results

Unfortunately, despite the fact that it was felt the Penguin could successfully complete the mission, a faulty battery pack prevented it actually doing so. On the night of the mission flight, the battery pack was unable to take a proper charge, and therefore, the Penguin did not have adequate power to take-off. Several different attempts to charge the battery pack failed, much to the chagrin of the designers. An alternative power source was borrowed from another group, but because it had already ben used, it too was not at full capacity, and again an attempt to take-off failed. Finally, with a few minutes left to attempt flight, a battery pack, speed controller, and motor (Astro 15, with different leads) were borrowed from a plane that was unable to utilize them. After installation into the Penguin, a successful oval course completed. Because of time constraints, the pilot was unable to attempt to fly the mission.

Conclusion

The Penguin original design has fulfilled or exceeded all expectations placed upon it. With an adequate power source, the Penguin lifted-off in only 40 ft, which took only 2.9 seconds to achieve. This is well below the estimated 51 feet. The Penguin cruised in steady-level flight at an estimated velocity of 24 ft/s. This is compares favorably to the designed cruise velocity of 25 ft/s, and yields a Reynold's number of 102,000. The pilot was able to maneuver the plane in a coordinated turn of radius 30 feet, well within the initial constraint of 50 feet. With an adequate battery pack, the Penguin has proven to be capable of comfortably satisfying mission requirements.

Discussion of High Altitude Flight Feasibility

Real world applications for Remotely Piloted Vehicles often involve high altitude, station keeping missions. Such missions require the ability to fly at low Reynolds numbers for long periods of time. The development of the Penguin RPV resulted in concentrated study of the problems associated with low Reynolds number, long duration flight. Thus, the experience gained during the development of the Penguin RPV, can be of some assistance in the prediction of the problems associated with high altitude RPV flight.

Perhaps the most apparent problem with RPV high altitude flight is the difficulty associated with climbing to cruise altitude. A capable RPV would need to possess a large amount of fuel and a high rate of climb in order to gain the high state of potential energy associated with high altitude. Another problem stemming from the one mentioned above deals with the RPV's weight. Low weight is necessary for a high rate of climb, but the large fuel requirement will inevitably increase RPV weight. No easy solution to this problem exists. Although research into low weight fuel or climb assistance methods (rocket assisted climb, etc.) may help the RPV design, some engineering compromise between fuel and weight must be attained.

Once an RPV achieves high altitude cruise, other problems arise from the long duration and station-keeping mission requirements. Long duration flight requires a maximization of RPV endurance. Again, a large amount of fuel is needed. Also of major importance in maximizing the endurance is the minimization of drag. Station-keeping flight requires low cruising speed which, coupled with low air density at high altitude, gives low flight Reynolds numbers. Low Reynolds number, low speed flight causes two major problems for an RPV which must minimize drag. Low Reynolds number flow results in laminar boundary layers occurring over the RPV's lifting surfaces. Laminar flow is extremely susceptible to the formation of separation bubbles on the upper surface of an airfoil. If separation bubbles cover an appreciable area on a wing, the wing's performance is severely decreased as separation drag becomes dominant. The low RPV cruising speed causes another drag problem. Low speed flight requires high airfoil lift coefficients to generate the necessary lift to fly. A three dimensional induced drag. Both the induced drag and the separation drag have to be minimized for the benefit of RPV endurance.

Any successful high altitude RPV will have to be able to deal with both the separation drag and the induced drag problems. Separation drag must be eliminated by careful airfoil selection and study of methods used to avoid flow separation (boundary layer tripping, etc.). Induced drag can be reduced by designs incorporating devices such as high aspect ratio wings and winglets.