NASW-4435

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UNIVERSITY of
NOTRE DAME

NASA/USRA UNIVERSITY ADVANCED DESIGN PROGRAM 1991-1992

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FINAL DESIGN PROPOSAL

The S.T.o.R.M.

Air Transport System Design Simulation

May 1992

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

(NASA-CR-192070) THE S.T.O.R.M. (TM): AIR TRANSPORT SYSTEM DESIGN SIMULATION Final Design Proposal (Notre Dame Univ.) 115 p

N93-18349

Unclas

G3/05 0141685

The S.T.o.R.M.™ Design Document

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The S.T.o.R.M.™ Design Document

Written in the Spring of 1992 by Team Asylum, Inc.
Printed in the United States of America, and on Macintoshes everywhere on campus.

This document was created in fulfillment of the requirements for graduation in May. Credit is due to most of the Aerospace faculty for their instruction and guidance and to the Graduate Assistants who lent their time to us over the past three months. A very special thanks to Professor Batill who had faith in us.

Table of Contents

Section A

Executive Summary

1) Design Summary

The members of Team Asylum have proposed a helicopter design concept, called the S.T.o.R.M., in order to meet the market demands for an aircraft to perform overnight package delivery services in Aeroworld. The helicopter concept was chosen over a fixed wing aircraft design to fulfill the mission requirements for a variety of reasons, all of which will be discussed later. However, many critical design areas needed to be investigated as part of the helicopter concept's selection.

One of the most significant design factors was the weight of the aircraft. This determined the selection of the propulsion system necessary to get the S.T.o.R.M. off the ground, and maintain flight once airborne. Through a lengthy analysis of helicopter flight principles, it became apparent to Team Asylum that if the S.T.o.R.M. could be provided with the necessary power to hover, it would be able to sustain forward flight at a cruise velocity of 25 ft/sec. This is due to the fact that a helicopter requires more power to hover than to maintain forward flight. Therefore, the design team realized that the weight of the aircraft and the selection of the propulsion system necessary for flight were very interdependent. Using the provided data bases along with researched weight estimates, the S.T.o.R.M. was determined to weigh within the range of 4.77 lbs. and 7.33 lbs., depending upon the weight of the payload being transported. In an attempt to fulfill the mission requirement mandating possible delivery of the .04 oz/cubic inch cargo, a propulsion system which enabled the S.T.o.R.M. to carry 2.56 lbs. of cargo within the 1024 cubic inch payload bay would be required. The Astro 25 motor was selected because of its ability to deliver the necessary power required, while at the same time trying to keep the battery-package and motor weights to a minimum.

Another significant factor that went hand in hand with the motor selection was the choice of the main rotor. Since the main rotor is the primary source of lift for the helicopter, its proper selection became increasingly important. In order to stay within the bounds of the power available limits of the Astro 25 motor, a rotor diameter would need to be selected to provide the necessary lift, yet, at the same time not be so large that it would suffer severe drooping at the rotor tips or be in danger of clipping the tail rotor during rotation. A main rotor diameter of 50 inches was chosen in order to best fulfill the aforementioned constraints.

The advantages that a helicopter concept provides for the required mission were deemed many upon first analysis and consideration. The S.T.o.R.M.'s ability to eliminate takeoff and landing distance constraints and even loiter time due to its vertical takeoff and landing capabilities was viewed as a major advantage in time and fuel savings. The S.T.o.R.M.'s ability to fly at slow speeds and thus stay under the Aeroworld sound barrier of 30 ft/sec was also a desirable design aspect. The S.T.o.R.M.'s nonexistent turn radius would enable the aircraft to maneuver (i.e. to avoid unforeseen obstacles) better than a conventional airplane design.

However, some disadvantages for this concept exist as well. The excessive weight of S.T.o.R.M.'s design along with the tremendous power requirements necessary for its flight hinder the helicopter's range and endurance capabilities. Thus, it became necessary to decrease the market that could be served. Instead of servicing all of Aeroworld, only the large island could be serviced for the concept to remain economically feasible. Economically, the technological complexity of the S.T.o.R.M.'s development became a hindrance because of its exorbitant cost. Although the smaller market (the large island) would provide a 48% return on the original investment, it seems that the helicopter concept falls somewhat short of its originally conceived efforts to fulfill all of the mission requirements. However, the technological advancements made by Team Asylum were bold, exciting, and innovative and should provide new generations with the valuable information necessary to successfully complete future missions.

The final design characteristics of the S.T.o.R.M. incorporated an Astro 25 motor, powered by 14 Panasonic 140SCRC batteries, thus allowing the helicopter to fly at a cruise velocity of 25 ft/sec. With a payload volume of 1024 cubic inches and a full payload of 2.56 lbs., the S.T.o.R.M. would require 255 Watts of power to hover and 237 Watts of power to fly at the aforementioned cruise velocity. The lift for the aircraft will be provided by a Clark-Y 50-inch diameter main rotor, which in turn will be stabilized by an 8-inch diameter, symmetric tail rotor. An overall length of 31 inches, a height of 16 inches, a fuselage width of 8.25 inches, and a landing gear base width of 20 inches round out the critical dimensions for the S.T.o.R.M., thus making it compact enough to fit in the 2 ft x 2 ft x 5 ft storage container area. The helicopter exhibits an empty weight of 4.77 lbs. and a full-cargo weight of 7.33 lbs., with a maximum range capability of 5875 feet. The S.T.o.R.M., despite its technological complexities, was an invaluable source of information and enjoyment for Team Asylum and hopefully will benefit many design teams in the future.

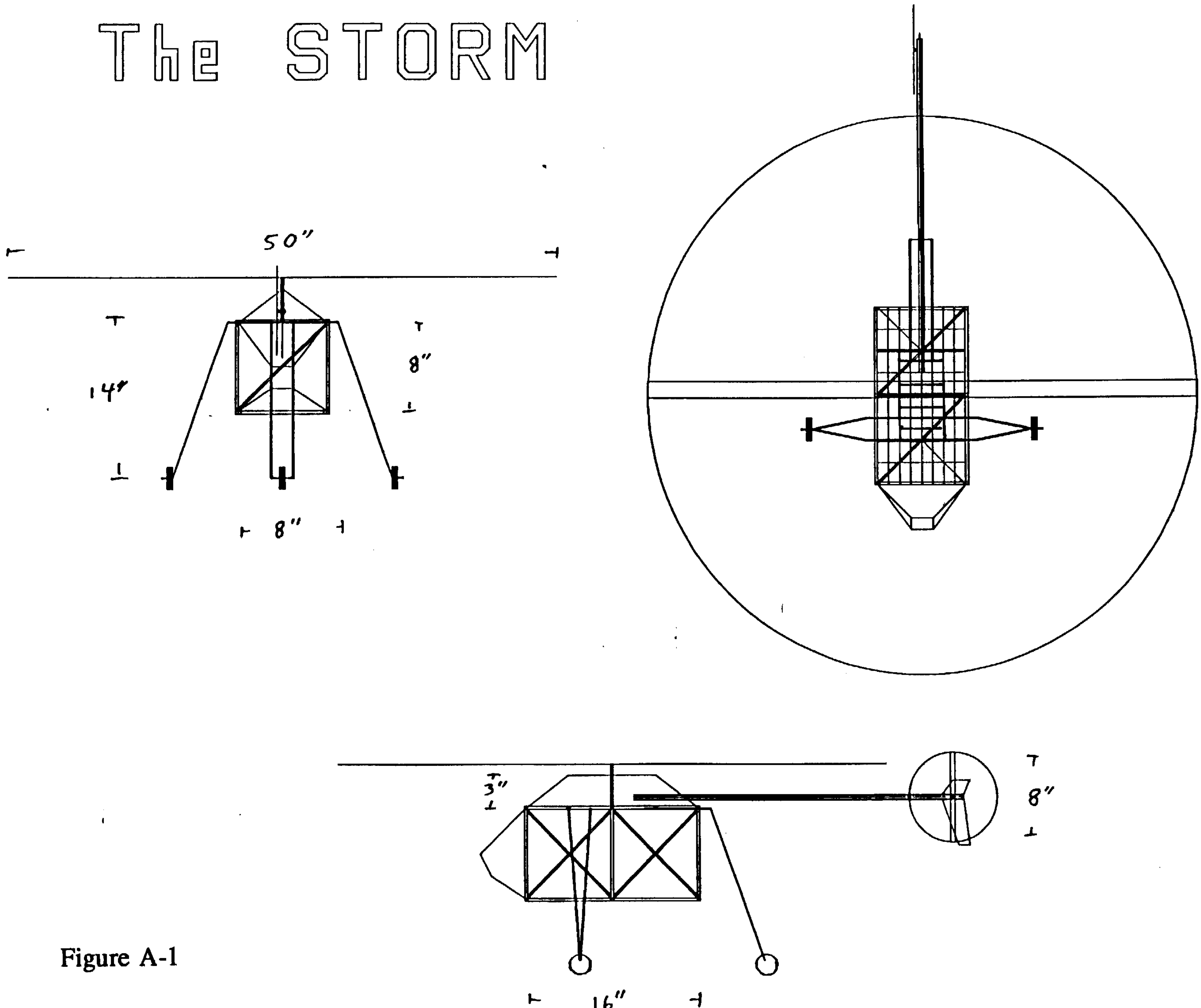


Figure A-1

3) *Critical Data Summary*

Parameter	Initials of RI:	as of PDR	FINAL
*[all distances relative			
to common reference			
and in common units]*			
DESIGN GOALS:			
V cruise [ft/s]	all	25	25
Altitude cruise [ft]	requirement	25	25
Turn radius [ft]	bonus	0	0
Endurance [min]	phil	4.25	2.46
Max Payload Volume [in^3]	all	1024	351
Range-max payload [ft]	phil	5875	
Payload at Max R (wgt) [oz]	phil	40	
Range-min payload [ft]	phil	8500	
Weight (MTO) [oz]	chad and ken	120	48.24
Design life cycles	all	600	600
Aircraft sales price [$]	chad	368000	420000
Target cost per in3 payload	chad	1.50	1.50
Target cost per oz payload	chad	28.40	28.40
BASIC CONFIG.			
Disk Area [in^2]	frank	1810.1	967.6
Weight(no payload) [oz]	ken	56.5	48.22
Weight(maximum) [oz]	chad & ken	96.5	52.15
length [in]	ken	31	28
rotor span [ft]	frank	4	2.9
height [in]	ken	16	18
width (fuselage) [in]	ken	8.25	5.6
ROTOR:			
Aspect Ratio	frank	16	11.7
number of blades	convention	2	2
Span [in]	frank	50	35.1
Chord [in]	frank	1.5	1.5
taper Ratio	frank	1	1
gear ratio	frank	15:1	8:1
Airfoil section	frank	Clark-Y	symmetric
Design Reynolds number	frank	184000	98200
t/c	frank	15	15
Incidence angle (root) [º]	frank	5	5
CDo -rotor	frank	0.0053	

CLo - rotor	frank	0.35	
CLalpha -rotor	frank	0.0812	
FUSELAGE			
Length [in]	ken	31	17
Diameter - max [in]	ken	8.25	5.6
Diameter - min [in]	ken	4	5.6
Diameter - avg [in]	ken	7.67	5.6
Payload volume [in^3]	ken	1024	351
Total volume	ken	1340	387
Planform area [in^2]	ken	140	68
Frontal area [in^2]	ken	68.06	33.9
TAIL ROTOR:			
number of blades	doug	2	2
Disk area [in^2]	doug	50.26	38.48
span [in]	doug	8	7
aspect ratio	doug	10.67	9.33
chord [in]	doug	0.75	.75
gear ratio	doug	3.33	
taper ratio	doug	1	1
incidence angle [º]	doug	3	3
Airfoil section	doug	NACA 0006	symmetric
Tail moment arm [in]	doug	32	20
WEIGHTS			
•all weights are in oz.			
Weight total (empty)	chad	79	48.22
Avionics	doug	3.5	3
Payload (max)	chad	40	0
Engine & Engine Controls	all	38	8.62
Rotor	frank	6.32	1.76
Fuel (battery)	phil	23.8	13.45
Structure			
Rotor	frank	6.32	1.76
Fuselage/emp.	ken	3.25	1.72
Landing gear	ken	5.13	5.46
Icg - max weight [in]	doug	12	6.69
Icg - empty [in]	doug	12	6.69
PROPULSION			
Type	frank	Astro 25	Astro 05
number	frank	1	1
placement	ken	top	top

Pavil max @engine [W]	chad	272	149.4
Preq cruise [W]	frank	237	134
max. current draw [A]	phil	15.18	
cruise current draw [A]	phil	14.11	
Rotor diameter [in]	frank	50	35.1
Pitch range [º]	frank	-4 to 10	-3 to 9
Number of blades	convention	2	2
max. rotor rpm	frank	1000	1000
cruise rotor rpm	frank	1000	1000
max. thrust [lbs]	frank	7.58	
cruise thrust	frank	6.72	
battery type	phil	Panasonic	Panasonic
number	phil	14	8
individual capacity [mAhr]	phil	1400	1400
individual voltage	phil	1.2	1.2
pack capacity	phil	1400	1400
pack voltage	phil	16.8	9.6
PERFORMANCE			
Vmin [ft/s]	all	0	0
Vmax [ft/s]	all	30	30
Vstall [ft/s]	chad	78.36	78.36
Range max - Rmax [ft]	phil	5875	
Endurance @ Rmax [min]	phil	4.25	
ROC max [ft/s]	frank	3.31	
Autorotation	chad	yes	yes
SYSTEMS			
Landing gear type	ken	tricycle	tricycle
Main gear position [in]	ken	8	3.86
Main gear length [in]	ken	14	11
Main gear tire size [in]	ken	2.3	1.5
tail gear position [in]	ken	20	16.54
tail gear length [in]	ken	14	11
tail gear tire size [in]	ken	2.3	1.5
TECH DEMO			
Payload volume	all		351
Payload Weight	chad		0
Gross Take-Off Weight	chad and ken		48.22
Empty Operating Weight	chad and ken		48.22
Zero Fuel Weight	chad and ken		35.27
Disk Area	frank		967.6

Tail Rotor Area	doug		38.48
C.G. position	doug		6.67
Range max	phil		3690
Endurance max	phil		2.46
V cruise	all		25
Turn radius	all		0
Airframe struct. weight	ken		26.65
Propulsion sys. weight	all		8.62
Avionics weight	chad		3.1
Landing gear weight	ken		5.46
ECONOMICS:			
unit materials cost	chad	299.00	351
unit propulsion system cost	chad	225.00	225
unit control system cost	chad	339.00	339
unit total cost	chad	863.00	915.11
scaled unit total cost	chad	345200.00	366044
unit production manhours	chad	100.00	47
scaled production costs	chad	100000.00	47000
total unit cost	chad	445200.00	413044
cargo cost ($/in3)	chad	10.50	8.33
single flight gross income	chad	10752.00	9234.87
single flight op. costs	chad	8868.00	6234.00
single flight profit	chad	1884.16	1294.23
#flights for break even	chad	6860	4659

Introduction

The following report and analysis proposes the development of a unique concept in air cargo transportation. In the creation of the S.T.o.R.M., this design group has broken new ground in a technical area not previously covered by AE441, Inc. This discussion seeks not only to examine and predict the performance characteristics of the final design, but, it also explores those analytical procedures, techniques and criterion which direct and influence the design process. The ultimate goal, then, is not so much the actual concept created, but rather to gain an awareness of the total design process and the real world parameters which govern technical development.

Section B

Mission Evaluation and Requirements

1) Request For Proposals

The following request for proposals provided Team Asylum with the design specifications for a remotely piloted vehicle.

Air Transport System Design

OPPORTUNITY

The project goal will be to design a commercial transport which will provide the greatest potential return on investment. Maximizing the profit that you aircraft will make for an "overnight" package delivery network can be accomplished by minimizing the cost per "package." G-Dome Enterprises has conducted an extensive market survey for an airborne package delivery service and is now in the market for an aircraft which will allow them to operate at a maximum profit. AE441, Inc. has agreed to work with them to establish a delivery system. This includes a market analysis, the establishment of a distribution concept and the development of a number of aircraft concepts to help meet this market need. This will be done by careful consideration and balancing of the variables such as the payload, range, fuel efficiency, production costs, as well as maintenance, operation and disposal costs. The service may operate in any number of markets provided that they use only one airplane design and any potential derivatives (your company does not have the engineering manpower to develop two different designs). Consider derivative aircraft as a possible cost-effective way of expanding the market.

OBJECTIVES

1. Develop a proposal for an aircraft and associated flight control system which must:
    a) Be capable of carrying the two standard parcel packing containers, a 2" cube and a 4" cube whose weights can vary form 0.01 to 0.04 oz/in^3.
    b) Not travel at speeds beyond the sonic barrier of 30 ft/s.
    c) Be able to land and take-off on runways that are, at best, 75 feet long.
    d) Use one or a number of electric propulsion systems from a family of motors currently available.
    e) Be able to perform a sustained, level 60' radius turn.
    f) Be able fly to the closest alternate airport and maintain a loiter for one minute.
2. All possible considerations must be taken to avoid damage to surroundings or personal injury in case of system malfunction.
3. 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.
4. Evaluate the feasibility of the extension of the aircraft developed under this project to cover a wider or possibly expanded market.

SYSTEM REQUIREMENTS AND CONSTRAINTS

The system 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 under its own power.
c) The complete aircraft must be able to be disassembled for transportation and storage and must fit within a storage container no larger than 2'x2'x5'.

d) Safety considerations for systems operations are critical. A complete safety assessment for the system is required.
e) The radio control system and the instrumentation package must be removable and a complete system installation should be able to be accomplished in 30 minutes.
f) All FAA and FCC regulations for operation of remotely piloted vehicles and others imposed by the course instructor must be complied with.

2) *Mission Evaluation*

The primary consideration for the design of the S.T.o.R.M. was its status as a cargo transport for Aeroworld. The need to service all the cities presented not only technical design challenges, but also required consideration of routing possibilities and economic feasibility. As with all projects in the real world, there are technically (aerodynamically) optimal and economically optimal solutions. These two optimal solutions, however, are very rarely met with the same design. It is therefore necessary to perform a trade-off in many areas to arrive at the best compromise of the two.

Economically, it is desirable to develop an aircraft that is as durable as possible which is measured in terms of the working stress reduction factor versus flight cycles (take-off/landing cycles). Optimally then, it is desirable to have the aircraft travel the longest possible routes to cut down on the number of flight cycles per day. This goal would also yield optimum maintenance costs. Over the longer the routes, however, the higher operating and fuel costs are economically prohibitive. The best compromising solution is to establish a distribution network with a hub in one of the central-most cities that are subject to the most daily freight. As City J and City K "distribute" and "receive" the most overnight mail of any of the cities in AeroWorld, they are the most likely candidates for the hub. Using several aircraft to fly "spoke" patterns to and from the hub each day from the feeder cities, it would be possible for G-Dome Enterprises to establish a distribution network that would provide overnight service to the greatest number of areas while realizing the greatest return on investment. The routing decisions were finalized while performing economic feasibility studies that will be discussed later in this document. What resulted was a delivery system that relied mainly on flights to the hub and back. It also required some intra-regional flights to take care of local cargo traffic. This routing scheme is presented in Figure B-1:

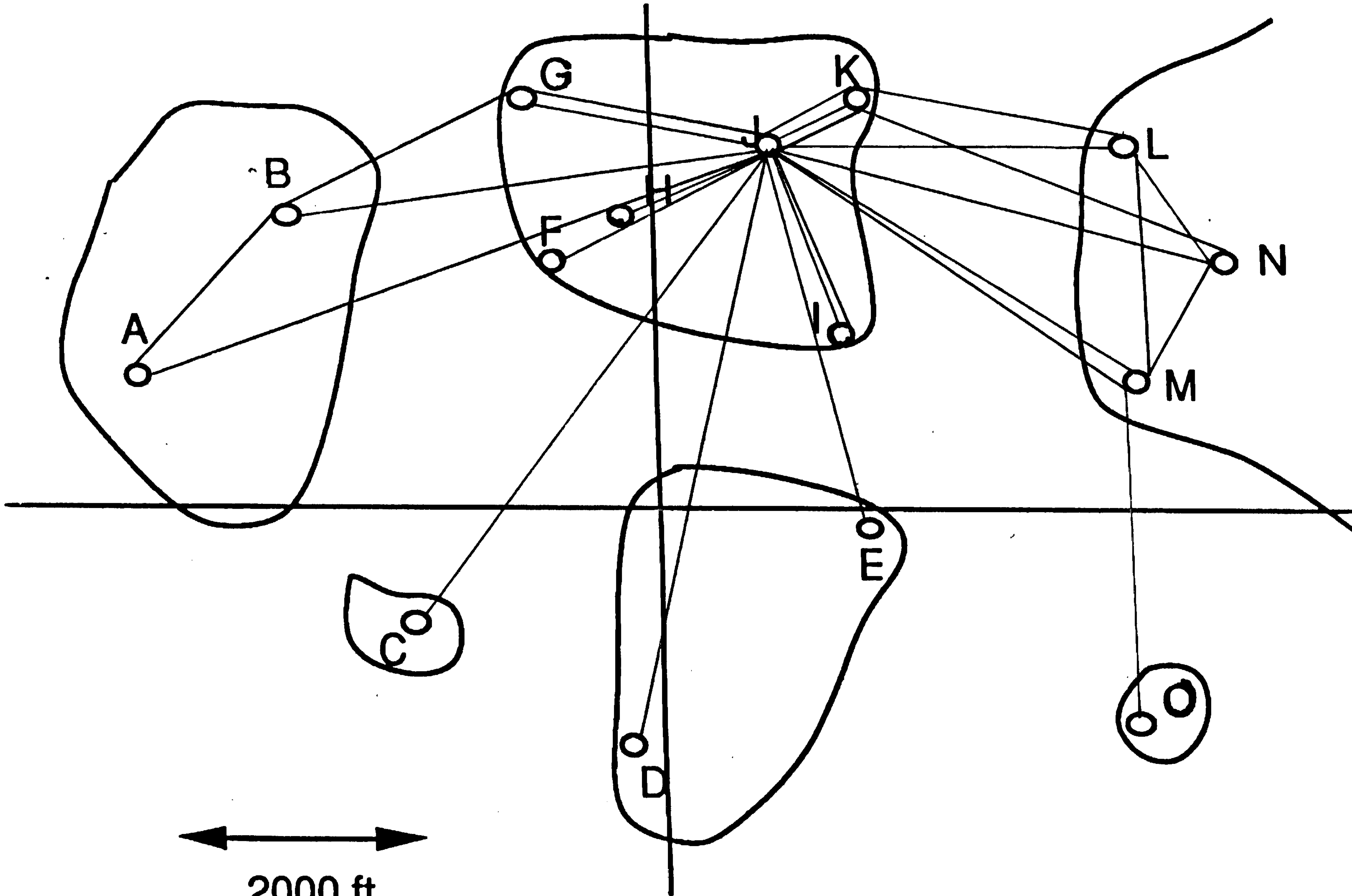


Figure B-1 Proposed Delivery Network

The routes in green represent aircraft flights that simply operate from the appropriate city to the hub and back again in the morning. Those routes in blue represent the "intra-regional flights" which, in many cases, do not even land at the hub. The combination of these flights services the entire Aeroworld market and allows each aircraft to make multiple trips per day while still allowing for adequate ground handling times and safety assurance.

The economics of the operation also plays a very significant role in the design of the aircraft itself. Obviously, since this is a cargo aircraft which must carry cargo "cubes", the most efficient cigar shaped fuselage is not extremely feasible and a rectangular fuselage seems appropriate in order to obtain an optimum amount of interior cargo space. The length of the aircraft should be such that it can hold enough of the containers to meet customer demand and routing. The maximum amount of cargo space should be around 1000 in^3 (to allow for realistic loads from city to city) which would correspond to a "maximum" cargo weight of 40 oz assuming the maximum cargo density of 0.04 oz/in^3. Derivatives of the aircraft could be developed to include stretch models for shorter hauls with more cargo, and longe-range, shortened models which may be more appropriate should further exploration of Aeroworld determine a need for such expansion.

The design and construction of the vehicle should be kept as simple as possible in order to keep the production costs to a minimum. These production costs are perhaps the most visible. Elaborate designs or construction costs which eke out a 5% improvement in range or endurance, for example, are seldom welcome in the face of a 20% higher price tag. The operational costs, being a function of the number of servos required, puts additional emphasis on the control aspects of the craft.

One of the most important technical aspects of the design are the restrictions on take-off/landing distances. This is especially significant in City B whose runway is 40% shorter than most. This influences virtually every portion of the design especially the selection of the propulsion system and airfoil (for $C_{L_{max}}$ characteristics) and in the determination of allowable weight.

Due to the "hub" system that is recommended and the relatively short distances involved, range is not seen as an important parameter for this particular concept and geography. If the operating geography was to change, derivatives would be relatively easily developed. Fuel economy, on the other hand, is critical. Over the life of the aircraft, perhaps the most expensive operating cost is the fuel, and any seemingly minor improvements in this area often result in very substantial decreases in the overall costs of flying the plane (hence a lower cost per in^3 or oz.).

Other requirements which are not as subject to trade-offs are the handling qualities which dictate that the plane must be able to perform a sustained, level 60' radius turn; the loiter capabilities which say that the aircraft must be able to fly to the closest alternate airport and loiter for one minute, and that the complete aircraft must be able to be disassembled for transportation and storage within a container no larger than 2'x2'x5'. It must also contain a radio control system and instrumentation package that is removable. Complete system installation must be able to be accomplished in 30 minutes.

From this mission evaluation, it was possible to set down some definitive design requirements and objectives. With the tremendous number of specifications desired, it was obviously important to identify some critical areas on which to focus. Table B-1 outlines those critical areas as perceived by Team Asylum and indicates their target values. Also listed are the actual values obtained by our concept design. Many of the causes for the discrepancies, shortfalls, and bonuses came as a result of the design team keeping the formulation of these goals independent of aircraft configuration.

The range and endurance targets outlined below are based on requirements posed by our initial distribution scheme for the cargo. With the largest air distance from City J being 5900 ft (City A), and allowing for loiter time and distance, 6500 feet was chosen to be a design goal. The endurance reflects the time to travel the maximum range at the anticipated cruise velocity of 29 ft/s, chosen such that the aircraft would not exceed the speed of sound during normal operation, in addition to the one minute loiter time. The cost estimates were based on approximated component costs from previous years' designs and anticipated increases in certain areas due to the cargo carrying requirements.

	DR&O	ACTUAL
Cost per cubic inch	\$1.00 - \$1.50	\$8.66 Original Network \$3.09 Revised Network
Cost per aircraft	\$368,000	\$445,200
Maximum Cargo Volume	1000 in^3	1024 in^3
Maximum Cargo Weight	40 oz.	40 oz.
Cruise Velocity	29 ft/s	25 ft/s
Turn Radius	60 ft	0 ft
Takeoff Distance	75 ft	0 ft
Range	6500 ft	5875 ft
Endurance	5 minutes	4.25 minutes

Table B-1: Design Requirements and Objectives

One of the most obvious goals that was not met was the delivery cost per cubic inch. This was primarily due to our initial prediction that fuel costs would be minimum compared to the production, maintenance, and operation costs for the aircraft. This certainly did not prove to be the case as the fuel costs typically accounted for about 85% of the total costs. This inflated cost was also due to the fact that the production costs were slightly more than expected due to the high cost of machining the blades and buying all the necessary systems.

While our final concept selection contributed to our downfall in economics, it proved to be a boon with respect to maneuverability and landing site accessibility. The very nature of helicopter operation meant that the concept was capable of outperforming its initial goals dramatically in these areas.

It may also be noted from Table B-1 that the range and endurance predictions fall rather short of initial hopes. This was due to the dedication to maintaining the promised cargo volume and weight. Through careful planning of the distribution network, however, it is believed that the range and endurance will be sufficient for the mission.

Section C

Concept Selection Studies

Once the design requirements and objectives had been identified, it was possible to examine the feasibility of various aircraft configurations in fulfilling the mission expectations. Early factors in the design process that figured into the concept selection were

- low-speed flight ability
- takeoff distance
- ease of construction

In light of the first two criterion, ideas were bounced around as to the viability of a rotary wing concept as compared to the standard fixed wing configuration. As discussions progressed, each concept was thoroughly evaluated.

1) Fixed Wing - Aft Tail/Empennage

Like other design teams, the natural choice for the cargo transport was the conventional fixed wing configuration with the aft tail. This was the configuration that was the most familiar to the engineers and this seemed to be one of the greatest advantages. Additionally, existing delivery networks use similarly configured aircraft, so pilot availability would not be a problem. Four of the six original individual concepts were of this form and the team seemed set on its development.

As the design team considered the concept presented in Figure C-1, however, several questions lingered unanswered in the backs of each engineer's mind. Preliminary weight versus wing surface area analyses indicated that in order to fly at such low speeds, below 30 ft/s, the wing size would simply be enormous with spans approaching ten feet. Given the size constraints on the Technology Demonstrator, this was seen to be a critical issue. The space constraints would dictate that the wing be "separated" in places which runs counter to safety requirements. Additionally, with the exceptionally large weights associated with the cargo mission, questions as to the aircraft's ability to use existing 75 foot runways were called to mind. These problems seemed to add strength to the argument for a rotary wing aircraft. The numerous advantages of the fixed wing configuration, however, were undeniable; with such a large data base, it would be possible to spend more time concerned with achieving the low speed/high weight capabilities than with basic design complication. Perhaps there was a way to combine the advantages of fixed-wing aircraft to a slightly different configuration.

Advantages	Disadvantages
Large Data Base	Large Wing Area
Relatively Fuel Efficient	Takeoff Distance
Tried and True	Low-Speed Capability

Table C-1. Considering the Aft Tail Configuration

2) *Fixed Wing - Canard Configuration*

In attempting to stay with the fixed-wing concept, the design team endeavored to find a way to decrease the wing area while still maintaining lift and low-speed capability. The solution seemed to present itself in the form of a canard configuration, shown in Figure C-2, in which the canard acts as a second lifting surface. This *would* allow the main wing span to be markedly decreased. Caught up in the exhilaration of finding a slightly more revolutionary concept that seemed to eliminate or minimize many of the previous concerns, it was easy to overlook some of the canard's disadvantages as shown in Table C-2.

Advantages	Disadvantages
Large Lifting Surface => less span	Stability and Control Problems
Innovative and "Unconventional"	Takeoff Distance
	Construction Difficulty
	Small Data Base

Table C-2. Considering the Canard Configuration

While the main-wing size could decrease, there would still be some potential problems with attaining the desired takeoff distance goals. Additionally, in talking with some of the senior engineers and upper management, it was discovered that the canard poses some very difficult stability and control problems which had proved very challenging to past design groups. These stability concerns deal with the relative inability to foresee the effects of interference of the canard on the main wing and the difficulty of center of gravity placement. Construction and operations complications could result if the canard is to be used as a control surface. As a result of these disadvantages, fueled by the desire to develop something that was a little more innovative, the design team was once again willing to reevaluate the feasibility of a rotary-wing aircraft.

3) *Final Concept Selection*

The final concept that the group decided to pursue was that of a helicopter. Although the initial concept took the form of the internal cargo carrying helicopter depicted in Figure C-3, evolution saw it develop into the crane variety, meaning that a cargo hold is attached to the main helicopter structure. This concept was explored because it represented a new technology that possessed certain advantages that would enable it to easily meet certain mission requirements. Due to the fact that rotary wing aircraft can hover and fly at low airspeeds, the helicopter concept could easily accomplish the requirement that the normal operating speed should be below 30ft/s. Also, the short take-off and landing distances provided by current runways are more than sufficient room to accommodate a helicopter. In fact, the runways are large enough for several helicopters to be safely serviced simultaneously. This will reduce loiter time and help to insure that the packages will be delivered on schedule. The ability to sustain a level turn with a 60ft turn radius is easily accomplished by a helicopter which can effectively rotate about a point. Another advantage of the helicopter concept is ease of construction. The helicopter airframe is small compared to that of a fixed wing aircraft thereby decreasing assembly time.

The disadvantages that were discovered include the large cost of acquiring the helicopter propulsion system. This will work to off-set the advantage of short production time to result in a prototype production cost similar to that of a fixed wing aircraft. The largest disadvantage of the helicopter concept is the large power requirement. This results in a heavier, more expensive propulsion system which has a very high fuel consumption. The high fuel consumption, in turn, results in low endurance. In spite of these disadvantages, the helicopter could be competitive in a smaller market. The advantages and disadvantages of the final concept selection are given in Table C-3.

Advantages	Disadvantages
Low Speed Capability	Expensive Parts
Short Takeoff and Landing Distances	Large Power Requirement
Small Turn Radius	High Fuel Cost
Multiple Landings Possible	Low Endurance
Simple Construction	Weight Penalty

Table C-3. Considering the Helicopter

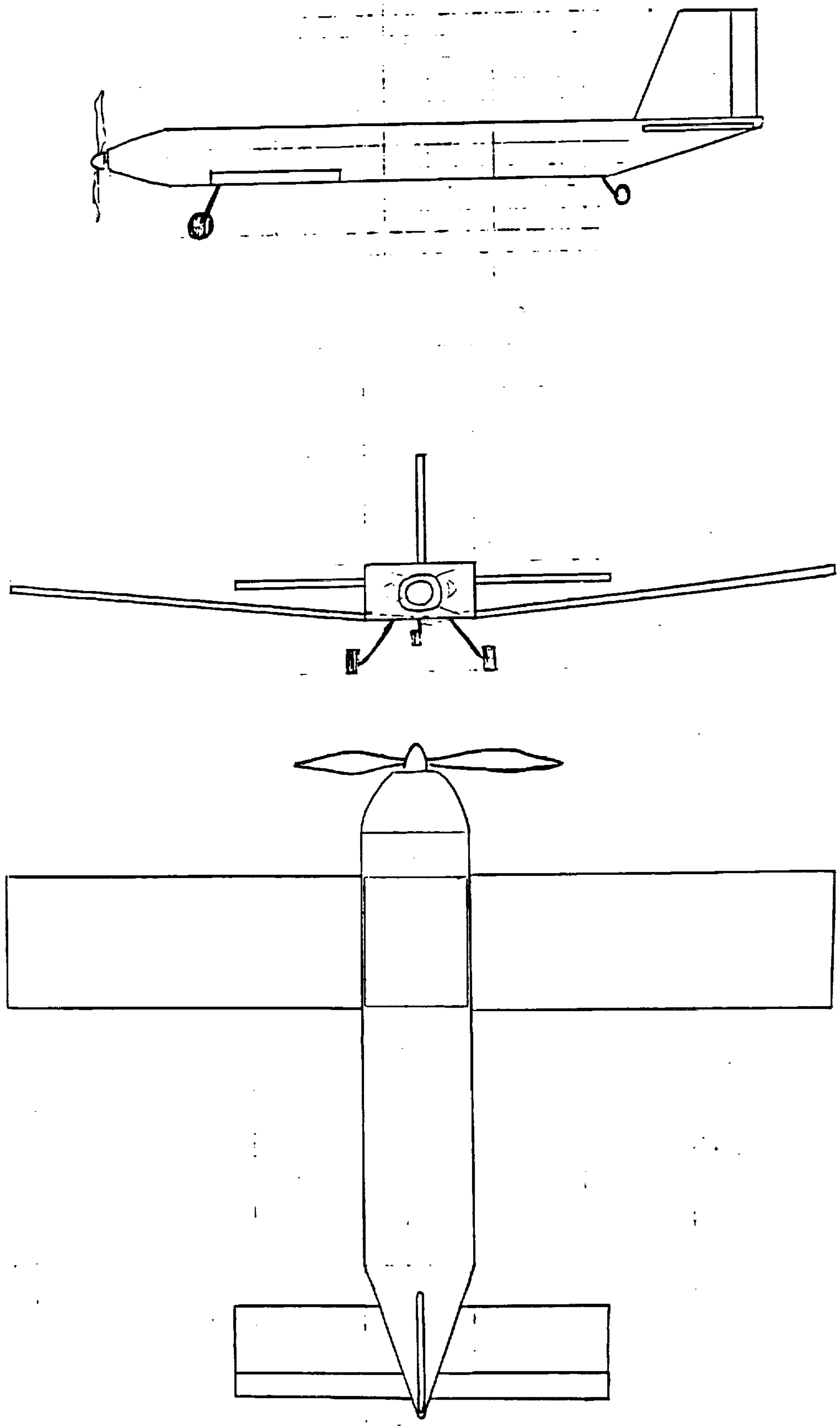


Figure C-1

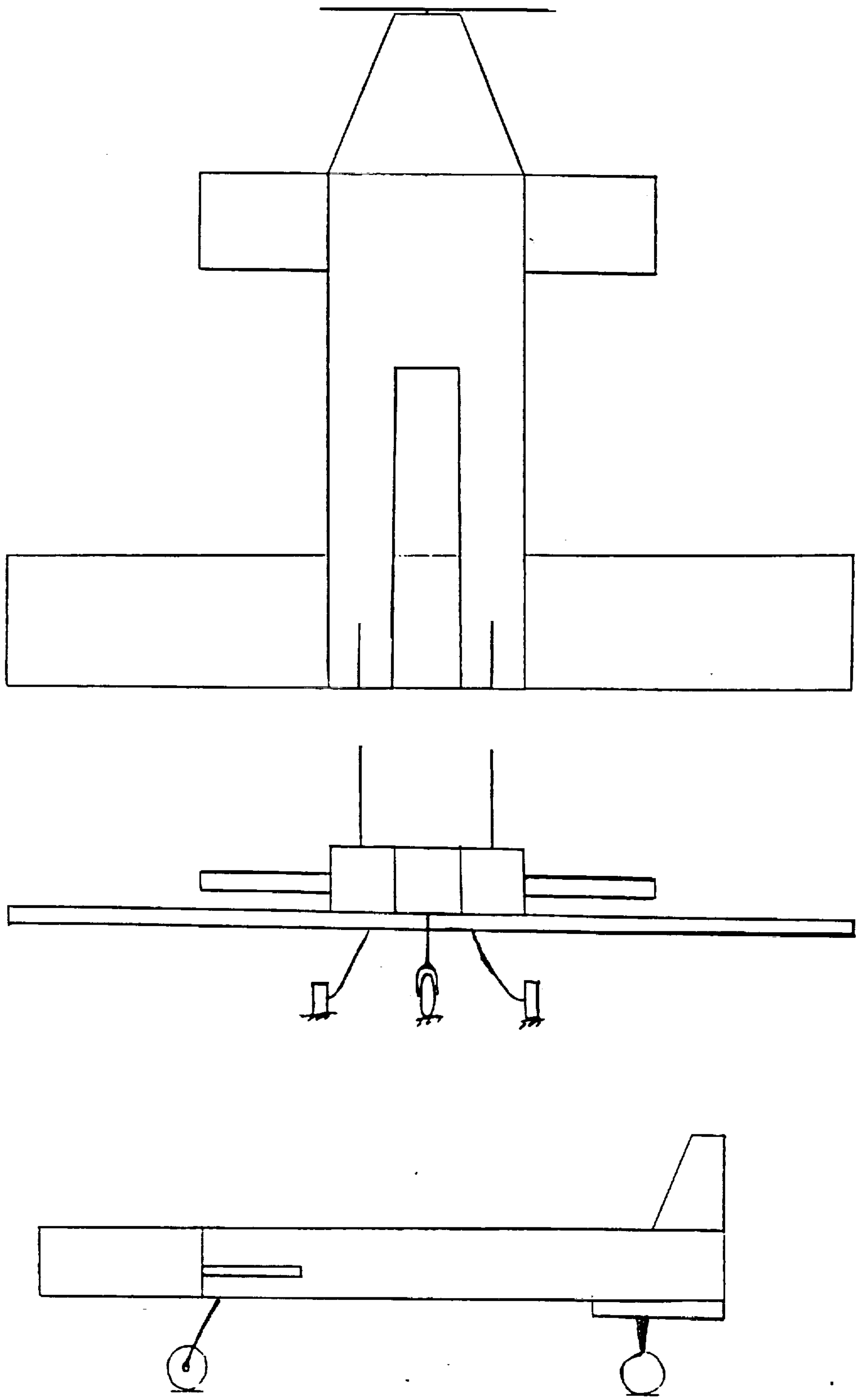


Figure C-2

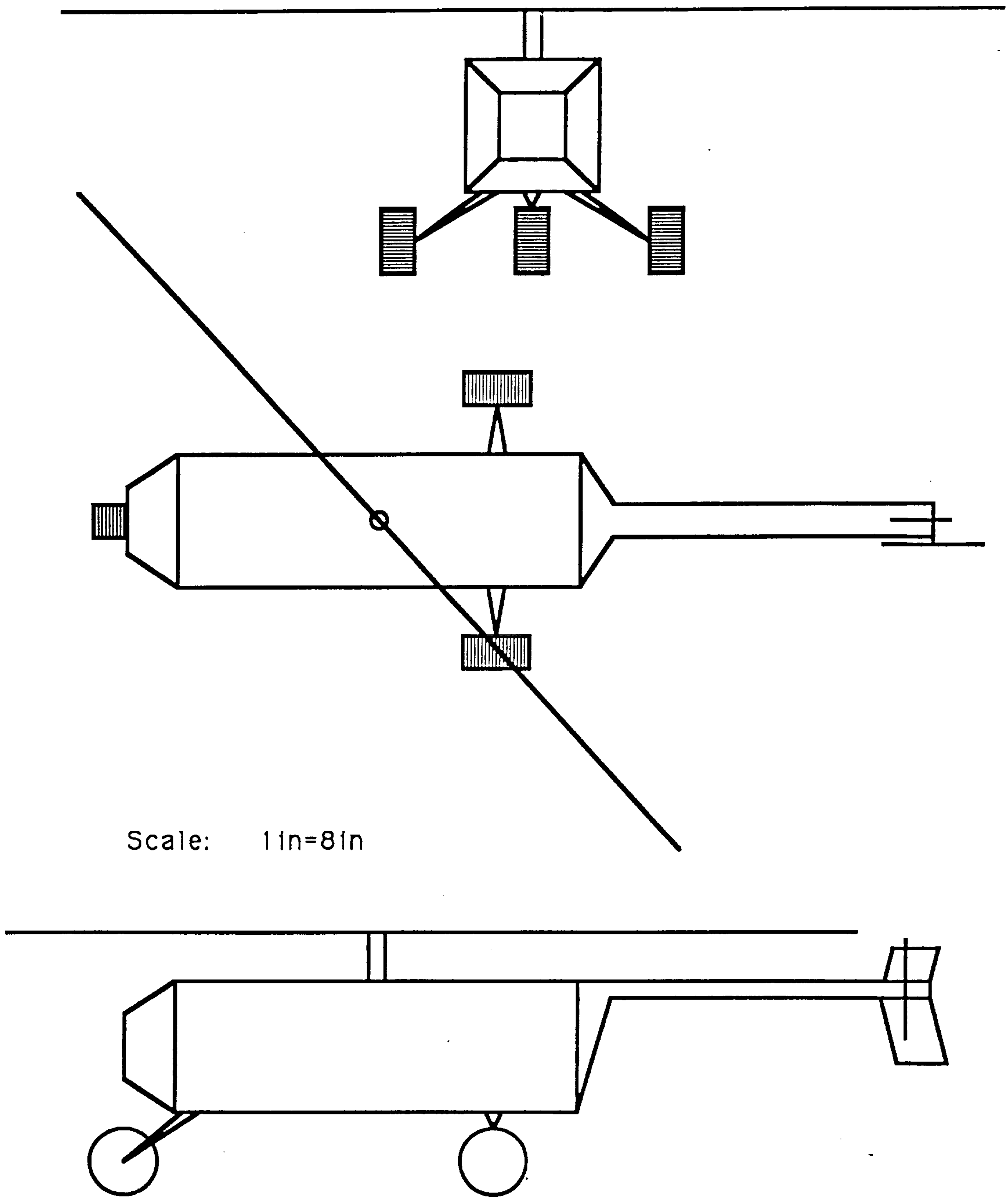


Figure C-3

Section D

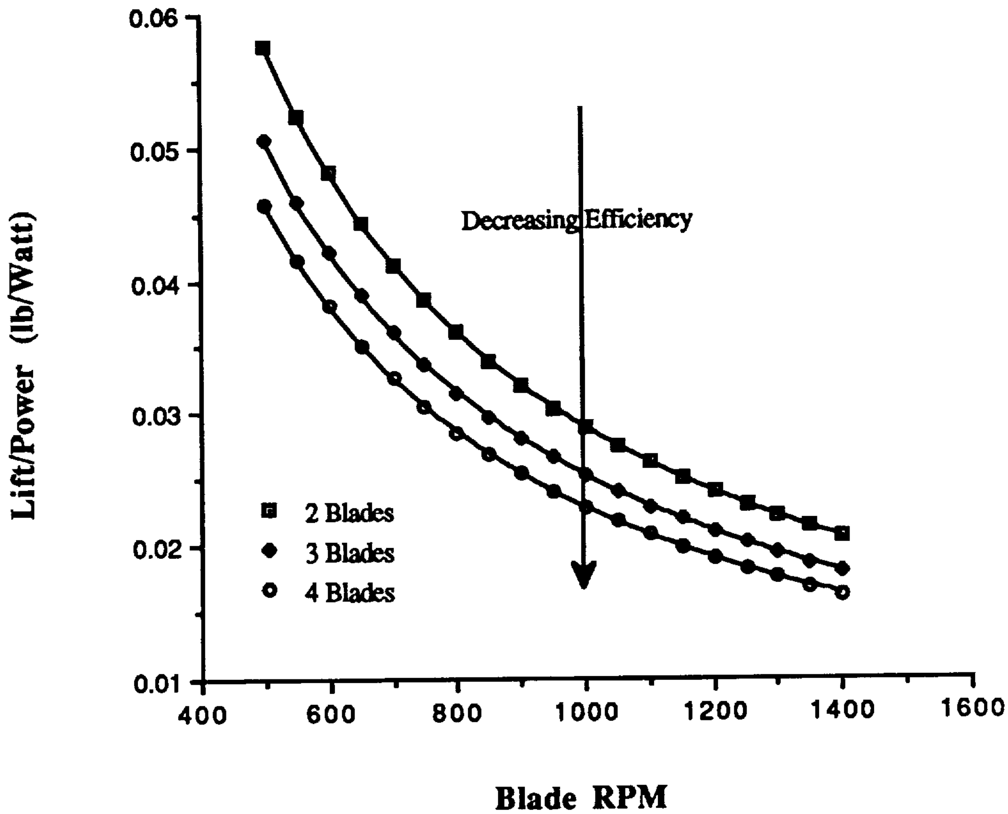
Aerodynamic Design Detail

1.) Airfoil Selection and Rotor Dimensions

The most important aspect of any aircraft design is the design of the lifting surface since this is what eventually causes the aircraft to fly. For the S.T.o.R.M., this lifting surface took the form of the main rotor. The main rotor of a helicopter provides both the lift and the thrust for the craft. As with all craft the lift produced by the lifting surface, in this case the main rotor, had to equal or exceed the weight of the helicopter in order for it to fly. For the 7.33 lb. S.T.o.R.M., the main rotor had a Clark-Y cross-section, 1.5 inch chord, and a 50 inch diameter.

For the S.T.o.R.M. it was decided that two blades would be utilized. This was due to the fact that a two-bladed rotor is more efficient than any other multiple bladed design. This study was done using the lift analysis and total power required relationships for the Clark-Y cross-section (a detailed description of this power analysis is provided in Section E, Propulsion System Design). Figure D-1 shows the lower efficiencies, characterized by a lower lift/power ratio, for three and four-blade rotors compared to a two-blade rotor.

Figure D-1
Decrease in Rotor Efficiency with Increasing Number of Blades



After deciding that two blades were the most efficient, the airfoil cross-section that would be utilized for the main rotor had to be decided upon. Initial research unveiled that helicopters traditionally used two types of airfoil cross-sections for their main rotors. They all used either symmetrical or Clark-Y cross-sections. Initial studies explored the aspects of using symmetric cross-sections because it was thought that symmetric cross-sections would be easier to manufacture and would not create any large aerodynamic pitching moments that would hinder the rotor performance. Initial studies were conducted with an NACA 0012 cross-section. This cross-section was chosen because of its low drag characteristics. Mounting this airfoil section at a 10 degree angle of attack resulted in an effective lift coefficient, C_L, of 0.5 and a drag coefficient, C_D, of 0.02. Due to packaging size constraints it had been decided to limit our main rotor size to approximately 48-54 inches in diameter.

The first aspect considered was how much lift could a rotor of the given dimensions and a NACA 0012 cross-section produce. The analysis of this area was quite simple. An analysis was developed to determine the maximum lift that could be produced by the different rotor sizes using a two inch chord. This chord was decided upon because it produced lift values near our weight of 7.33 lbs. at relatively acceptable RPM's. RPM values between 800 and 1200 RPM's were considered acceptable because the torque needed to spin the rotor was a function of the rotor RPM cubed and became quite large at higher RPM's. A spreadsheet determined the tip velocity of each rotor size at different RPM values using the relationship: $V_{tip} = 2p(RPM/60)r$ where r was the radius of the rotor in feet and the answer was given in feet/second. Once the tip velocity was known, the lift per blade was determined through the relationship:

$$L_{blade} = (1/6)\rho C_L S V_{tip}^2$$

where: ρ = density of air (0.00237 slugs/ft^3)

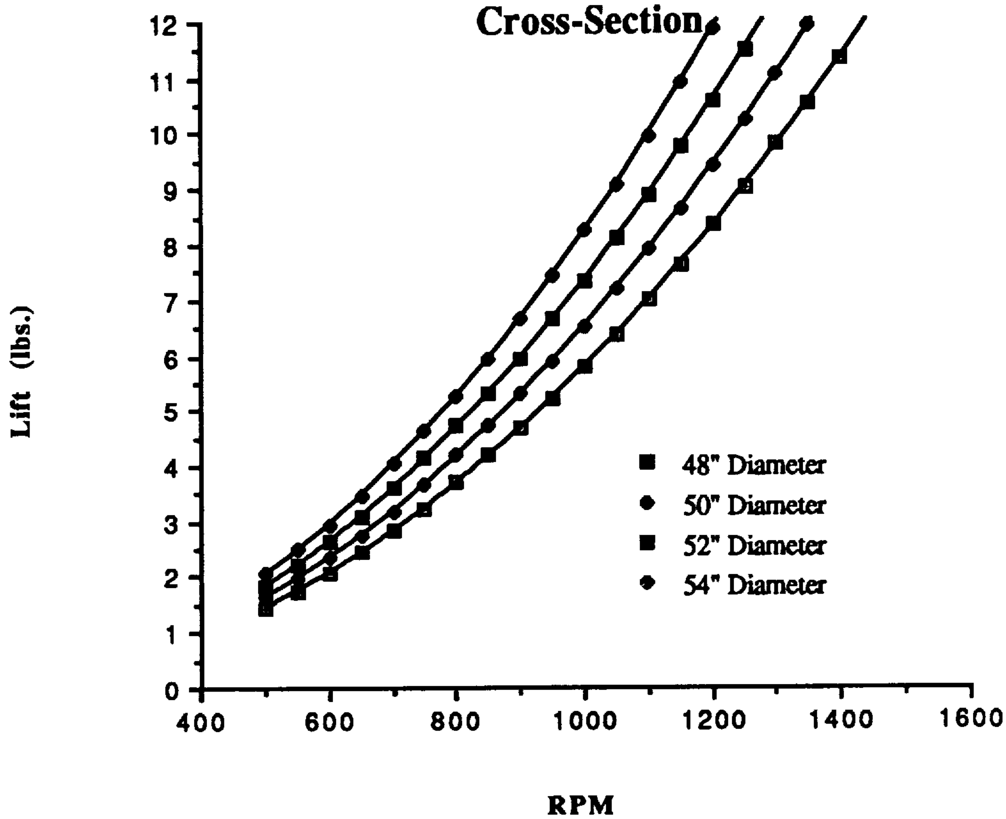
C_L = lift co-efficient for the airfoil section (0.5)

S = area of single rotor blade in sq. ft.

1/6 = reduction value for rotating blade including tip losses and inflow losses (Drake p. 13)

The total lift for the rotor was then found by simply multiplying the lift per blade by two, the number of blades. Figure D-2 was constructed to show the effect of varying the main rotor diameter and RPM on the lift of a rotor with a NACA 0012 cross-section.

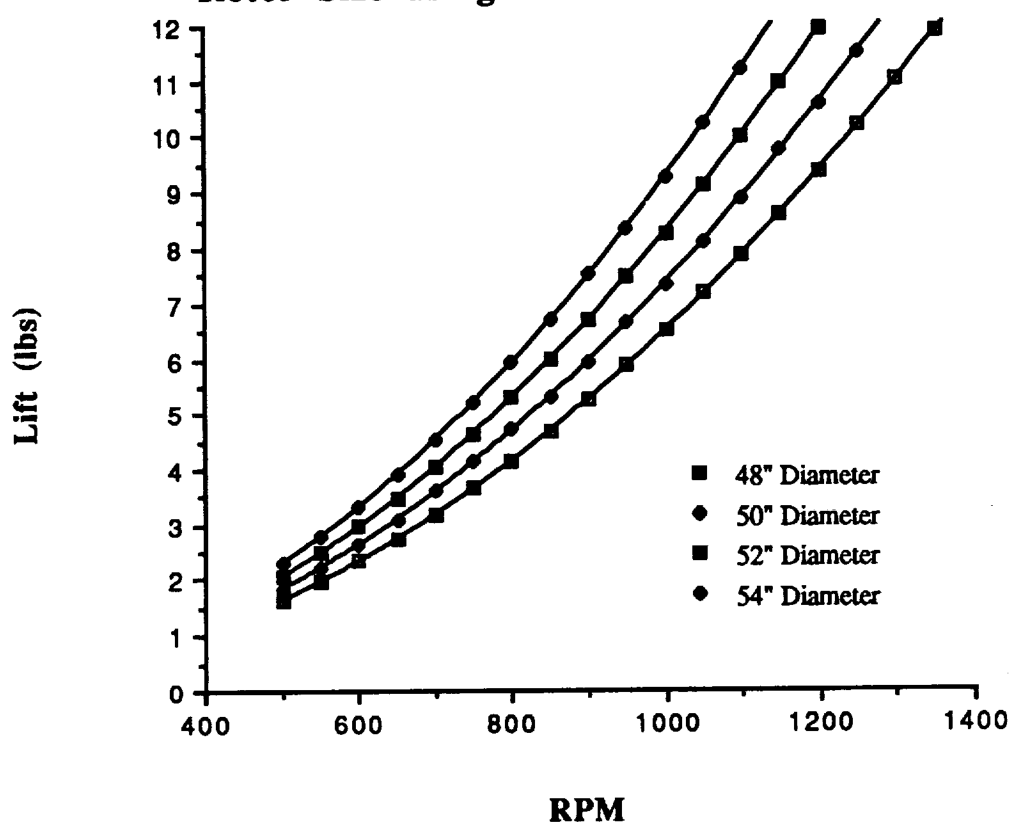
Figure D-2
Increase in Lift for Increasing RPM for Different Rotor Diameters and NACA 0012 Cross-Section



Note that as the blades become larger they produce more lift at the same RPM as the smaller blades. This, of course, was as expected. In order to get the required lift for the S.T.o.R.M., rotor RPM's between 900 and 1100 would have to be considered. To achieve greater lift at the same rotor size and RPM's, a cross-section with a higher C_L was deemed advantageous.

Team Asylum next considered a Clark-Y cross-section for the rotor. This was found to be a much better option for the rotor design because it produced higher C_L values at relatively the same C_D value and a lower angle of attack. The lift curve for the Clark-Y airfoil and rotor was placed in Appendix A. This last point was critical because if the rotor blade was mounted at a lower angle of attack, it would have less chance of stalling during maneuvers. Mounting a Clark-Y cross-section at only 5 degrees angle of attack produced a C_L of 0.75 while only increasing the C_D value to 0.0219, compared to 0.02 for the NACA 0012. Since the lift produced by this cross-section was so much higher than the NACA 0012, a chord length of only 1.5 inches was needed to produce the desired lift values in the target RPM range. The smaller blade area was deemed favorable because it again would require less torquing force to turn the blade. The same analysis done on the NACA 0012 section was done on the Clark-Y section and it led to the results presented in Figure D-3.

Figure D-3
Increase in Lift for Increasing RPM and Rotor Size using Clark-Y Airfoil Section



Notice here that the desired lift of 7.33 lbs. can be achieved by three of the four rotor sizes at or below 1000 RPM's. Based on this enhanced lift performance, Team Asylum has decided that the main rotor of the S.T.o.R.M. will employ a Clark-Y cross-section, a 1.5 inch chord, 50 inch diameter, and will be mounted at a 5 degree angle of attack.

2.) ***Rotor Design***

With the rotor geometry now defined, it is important to consider the rotor structure itself. As wing design is crucial for fixed wing craft, rotor construction is essential to a successful helicopter. Recalling the requirements set forth in the DR&O and some basic aerodynamics, one can easily deduce the need to adapt specialized blades to the standard hub in order that the stock assembly have lift equal to the weight of the vehicle and the added weight of the company parcels. In considering the modification of the blades, it was necessary to consider the ability of the hub and its various linkages to withstand the additional weight and RPM induced by the larger blades. Detailed studies of the blade adaptors and material standard to the *EP Concept* assembly was necessary to ensure the success of the Technology Demonstrator. Various considerations were made in order to complete this study. Geometry is the main factor. Material and commercial availability are also crucial.

Using the cross section of the blade, two are presented here for comparison -- the NACA 0018 and the NACA 23012, the mass of the blades was estimated. (Simply estimate a section area and multiply this times the radius of the blade to get the volume and then by the density of the material to get the mass.) Since wood blades are heavier than plastic, and more easily obtained, wood density was used to estimate the mass of the blades. Blades are typically made with a hard wood leading edge, oak or pine, and a soft trailing edge, balsa. The division is about fifty per cent for each material. Using the appropriate densities and three different radii (46in, 50in, 54in), the mass per blade was obtained and used to get the radial force exerted at the center of the hub. Since the force varies with the rotational velocity of the head, graphs were made to show the relationship. The parabolic figures shown in Figures D-4 and D-5 illustrate that the force increases with revolutions per minute of the blades. An increase in blade radius causes a substantial increase in the radial forces at the center of hub. For example, at 1000RPM, the design RPM for S.T.o.R.M., the force jumps from 4000 to more than 5000ftlb/sec^2 as the radius increases from 46in to 54in.

Notice also that the symmetric blade exerts more force on the hub for each radius than the NACA 23012 -- this is of course because of the additional weight of the blade because of an increased area.

Figure D-4
Variation of Radial Force with RPM for a Symmetric Airfoil (NACA 0018)

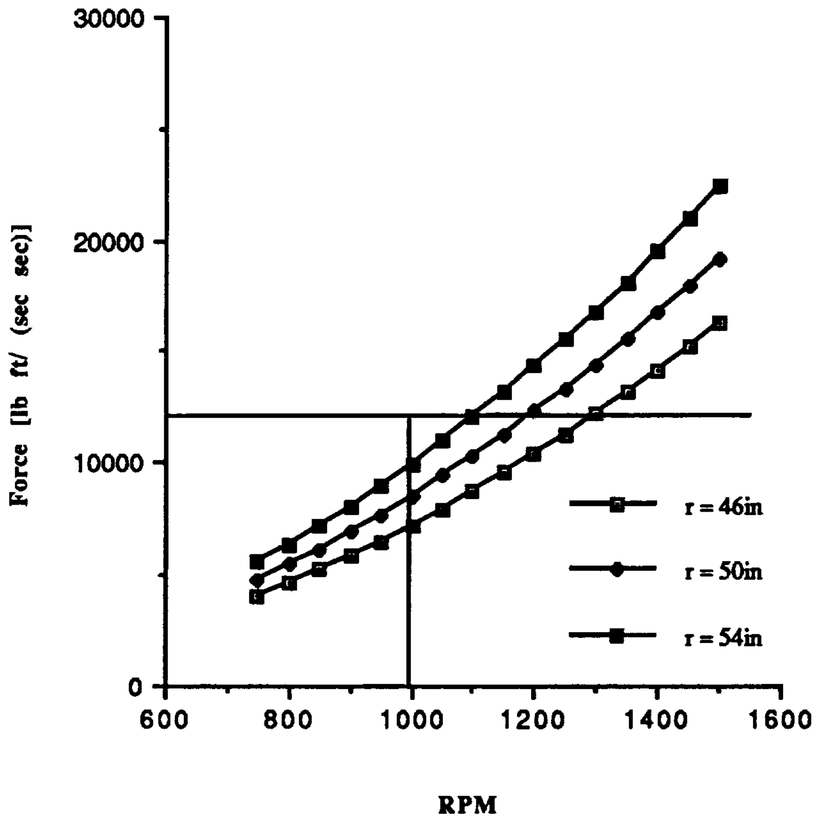
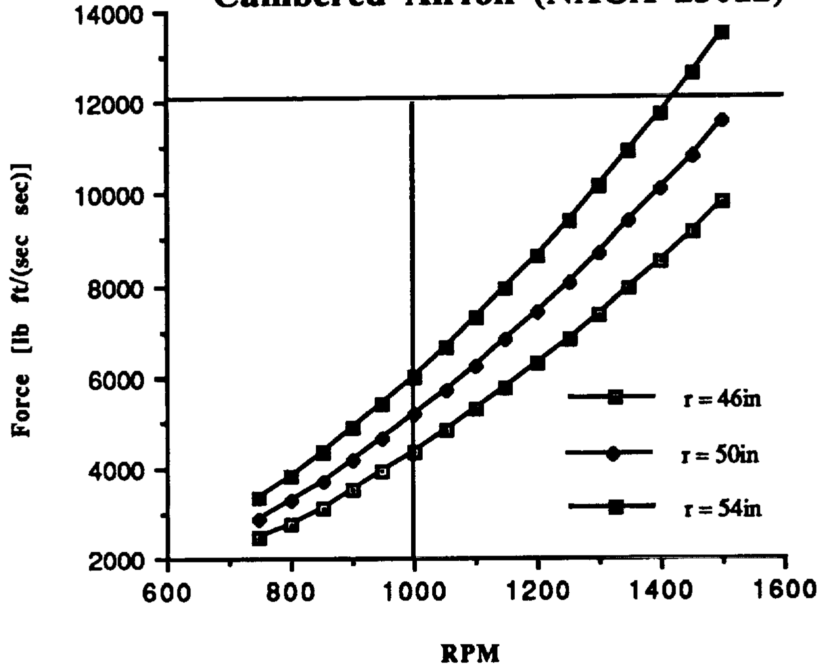


Figure D-5
Variation of Radial Force with RPM for a Cambered Airfoil (NACA 23012)



Using the design values to find the radial force at the hub that the assembly handles due to the stock parts gives the maximum acceptable value for force at the hub. This value was calculated to be just less than 12000ftlb/sec^2. The figures presented in this section show that the concept stays well below that limit. Also, knowing that the design RPM is 1000, the curves show that either airfoil section will clearly satisfy the mission, while providing the safety that is important to everyone involved.

3.) ***Drag Prediction***

There were two types of drag that needed to be considered for the S.T.o.R.M. First of all there was the skin-friction drag of the fuselage. This was determined using the relationship:

$$\text{Drag} = (1/2)\,\rho V^2 C_D A$$

where: V = freestream forward velocity

C_D = drag coefficient for fuselage (0.5 for spherical section)

A = frontal area of fuselage

It can be seen that in hover, ie. when $V = 0$, the skin-friction drag from the fuselage has no effect on the performance of the S.T.o.R.M.

The second major drag component was the downwash drag created on the fuselage due to the flow through the blades. In most cases, the "vertical drag is ignored" because the additional thrust required to overcome this vertical drag was found to be small in comparison with the total rotor thrust to lift the helicopter itself (Payne, p.57). Using the relationship found in Payne, p. 59:

$$\text{vertical drag/total rotor thrust} = \Delta T/(T+\Delta T) = (C_d n^2/4e)A/\pi R^2$$

where: ΔT = additional thrust to overcome vertical drag (by definition, equal to drag)

T = total thrust to overcome weight of aircraft (by definition, equal to weight of 7.33 lb)

C_d = drag coefficient of top of the fuselage (square = 1.2, taken from Payne, p. 58)

n = velocity ratio between induced flow with and without vertical drag

e = Oswald efficiency of the fuselage

A = Area of fuselage subjected to vertical flow (=.79 ft^2 = total top surface area)

R = Main rotor radius

From experiments conducted by Fail and Eyre with a rectangular wing under a rotor, it was found that the value for $(C_d n^2/4e) = 0.7$ (Payne, p. 59). Therefore, the value for $\Delta T/(T+\Delta T)$ for the S.T.o.R.M. was found to be 0.04. Therefore, the vertical drag produced by the fuselage was equal to 4% or 0.3 lbs while in hover.

Section E

Propulsion System Design Detail

1. Maximum Power Required

The maximum power required for the S.T.o.R.M. to operate, like that for all helicopters was the power to hover. The power to hover breaks down into two areas: the torquing power needed to rotate the rotor, and the induced power needed to pull the air down through the rotor.

The torquing power needed to rotate the main rotor was a direct function of the drag on the rotor blades themselves. The drag on the blades determines the torque needed to turn the blades through the relationship: Torque = 0.8R X Drag X # of blades

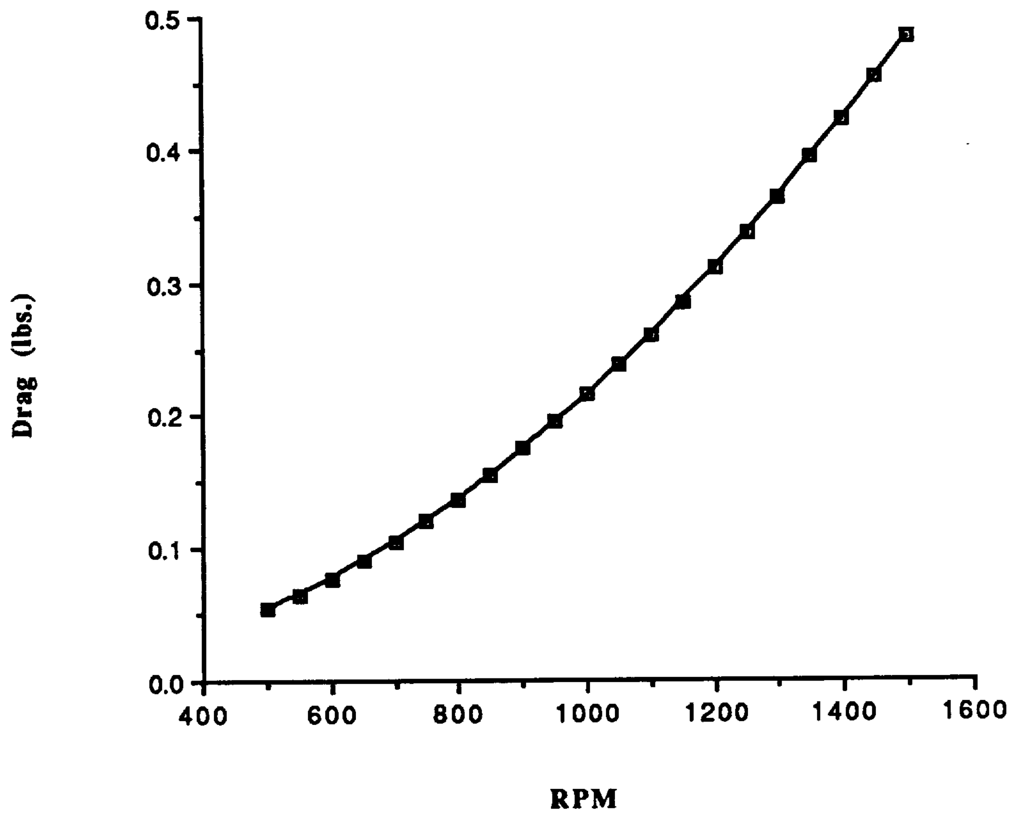
where: 0.8R = 80% radius of blade which is where the total drag on the blade acts (the proof for this can be found in Appendix A of Drake's book)

The drag on the rotor blade was determined analogous to the method used to determine the the lift produced by the blade. This relationship turned out to be: Drag= $(1/6)\rho S V_{tip}^2 C_D$

where: $C_D = 0.0219$ for the Clark-Y cross-section.

The drag of the main rotor at different RPM values is illustrated by Figure E-1.

Figure E-1
Drag on Main Rotor at Different RPM



Note that at the S.T.o.R.M.'s operating RPM of 1000, there was only slightly greater than 0.2 lbs of drag created by the rotor blades and thus created a torque about the main rotor hub of .536 ft-lbs (103 in-oz.). The power needed to overcome this torque to turn the main rotor blades was found through the relationship:

$$P = 2\pi N(\text{Torque})/(550\text{ft-lb/HP})(747\text{ Watts/HP})$$

where: N = Rotor revolutions/second

It was found that S.T.o.R.M. needed 76.2 Watts of power to overcome the torque produced by the rotor blades.

The induced power to hover was a function of the induced velocity of the air through the blade and the lift desired from the blade. The induced velocity to hover, U, was found through the relationship:

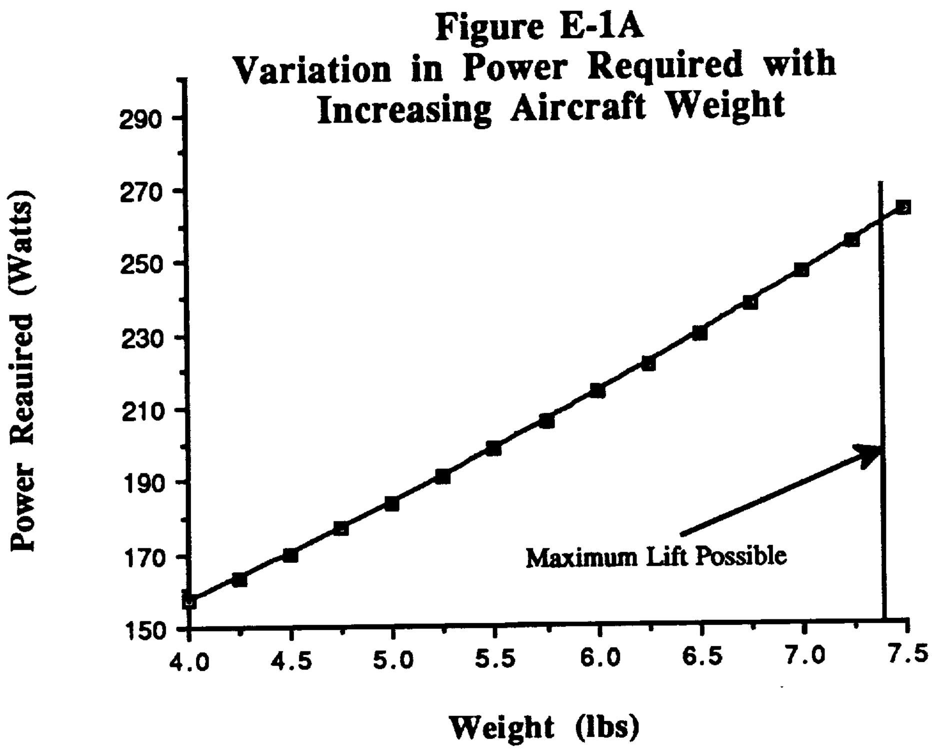
$$U = (1/.8)\mathrm{sqrt}(L/2A\rho)$$

where: 1/.8 = constant accounting for 80% power loss due to tip losses of the rotor

L = lift desired

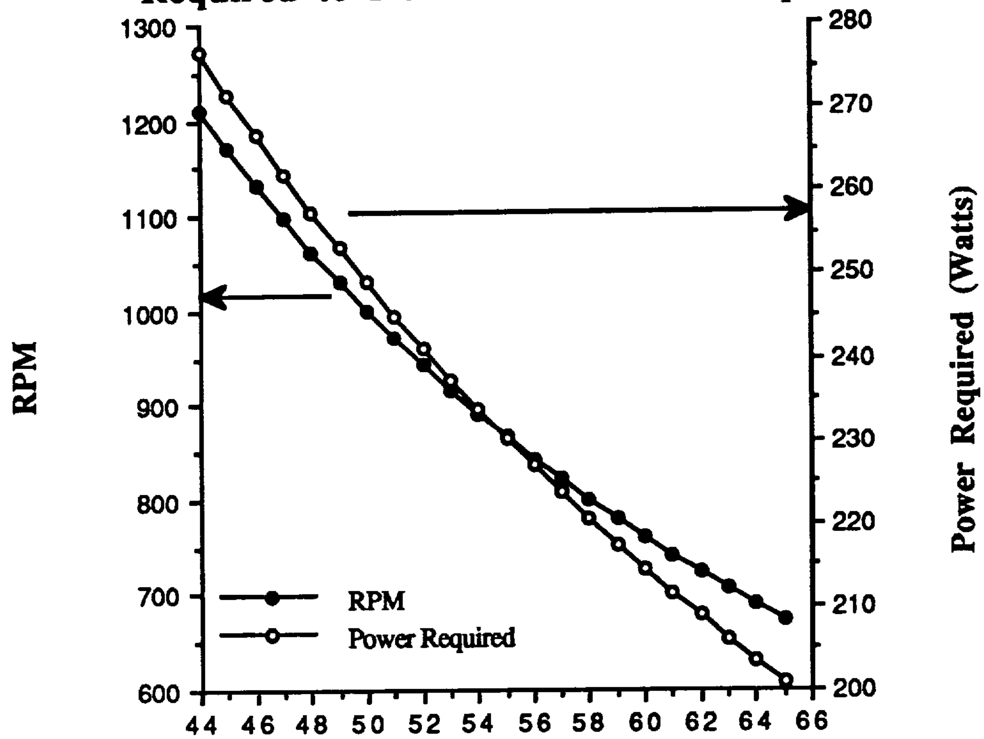
A = swept area of main rotor blades (Drake p. 12)

For the 7.33 lb S.T.o.R.M., the induced velocity was found to be 13.3 ft/sec. The induced power required was then found through the simple relationship that the power required, P = LU/(550 ft-lb/HP)(747 Watts/HP) (Drake p. 13). The S.T.o.R.M. consumed 132.5 Watts due to induced power to hover. It should be noted here that the additional 0.3 lbs of thrust needed to overcome hover drag (as determined in Section D, subsection 3) would only require 5.4 Watts of power. Therefore, since this ends up being only 2% of the total power required to hover S.T.o.R.M., hover drag was assumed to be negligible in the power required equations. Figure E-1A illustrates the effect of the weight (which must be equal to the lift desired in order to hover) of the S.T.o.R.M. on the power required to hover. The maximum lift line was determined by determining the greatest amount of lift that the 50 inch diameter blades could produce at 1000 RPM, which was 7.4 lbs.



It was evident that the maximum power needed to hover was directly related to the rotor size and the RPM that was used. Figure E-2 shows the relationship for total power required as a function of changing rotor diameter and rotor RPM.

Figure E-2
Correlation between RPM and Power Required to Hover 7.334 LB Helicopter



Rotor Diameter (in)

Notice that large blades spinning at lower RPM's utilize less power to produce the desired lift than small blades spinning at higher RPM's. It was decided that the S.T.o.R.M. would have a 50 inch diameter rotor because of the size constraint that the entire aircraft should be able to fit inside a 2' X 2' X 5' box. For the desired 50 inch rotor diameter, approximately 255 Watts of power was needed to hover. This value included 10% power loss in the mechanical gearing and the fact that the tail rotor consumed 10% of the main motor power.

2) *Propulsion System Selection*

Once the total power required for the S.T.o.R.M. was found, a motor had to be selected that could provide the necessary power to hover the helicopter. Since Glow-Plug gas engines have considerably greater thrust-to-weight characteristics than electric motors, the ideal engine for the storm would be a 0.5 Horsepower Glow-Plug engine, but due to the environmental restrictions of

Aeroworld and the Loftus Flight Test Center, Team Asylum was required to power the S.T.o.R.M. with an electric motor.

The first motor considered for the S.T.o.R.M. was the Astro Cobalt 40 motor, which would produce 450 Watts of power. Running at 75% efficiency, this motor would still provide 335 Watts of power, more than enough to hover. This motor would weigh a total of 94 oz. including the 18-1200mA cells. Since the Astro Cobalt 40 motor produced more power than needed, smaller motors were considered that would weigh less and allow for more cargo carrying capability.

The second motor considered was the Astro Cobalt 25 motor, which produces a maximum power output of 270 Watts, which was slightly greater than the 255 Watts required for the S.T.o.R.M. to hover. This motor, including the 14-1200mA cells would weigh 76 oz. Comparing to the Astro Cobalt 40 motor it can be seen that an increase in weight of 18 oz. only produces an additional 180 Watts of power. In fact, just to lift that additional 18 oz. requires approximately 110 Watts of power. (This was found using the given rotor dimensions and finding the power required to hover 18 oz.) With the expected increase in structure that would be required to support the heavier propulsion system, the total power required for the larger engine could easily approach the 180 Watts of additional power it would provide. Since weight savings was a major concern in this aircraft design as it is in all aircraft design, the Astro Cobalt 40 motor was not chosen because its increase in available power did not justify its additional weight.

The Astro-Cobalt 25 motor produced a maximum output of 270 Watts at 12,000 RPM. This maximum power available was extremely close to our power required to hover of 255 Watts. Keep in mind that the power to hover included an estimated 10% loss in gearing, but that only an additional 6% loss would mean that the S.T.o.R.M. could not hover.

An ideal propulsion system would have been something in between the Astro Cobalt 25 and the Astro Cobalt 40. A power output of approximately 300-325 Watts would provide the S.T.o.R.M. with an additional 18-21% margin for error with respect to total power required calculations. However, since current manufacturers provided only the Cobalt 25 and Cobalt 40 motors, team Asylum chose the Astro Cobalt 25 motor to power the S.T.o.R.M. and will concentrate on eliminating as much mechanical loss as possible.

The S.T.o.R.M.'s propulsion system was made up of both a gear driven, and a belt driven system. The Cobalt 25 motor had to be geared down from 12,000 RPM to 1000 RPM to power the main rotor. This was done with two gears attached to the drive train of the motor. Great care must be taken in aligning and lubricating the 6:1 and 2:1 gears in order to eliminate mechanical losses in the gears. Two gears were needed because the rotors have been designed to spin counterclockwise, the same direction as the motor spins.

Besides powering the main rotor, the motor also had to provide power to the stabilizing tail rotor. This was accomplished by using a belt drive system that attaches to the 6:1 gear and then runs through the tail boom to the tail rotor where it is to be geared down to the desired speed of 3600 RPM.

In order to provide the necessary power to get the S.T.o.R.M. off the ground, much less to maintain forward flight, proper battery selection became increasingly important. Because increased weight, a very undesirable characteristic, ensues with the addition of each battery cell, it became necessary to choose a battery package that would provide the required motor power for flight, yet still keep the overall aircraft weight to a minimum. The Panasonic 140SCRC battery was chosen and packaged in a fourteen battery-pack configuration, thus providing 16.8 total volts (1.2 volts/battery) for the Astro 25 motor to draw from at one amp•hr. Although the battery capacity is rated at 1.4 amp•hr, a conservative estimate of 1.0 amp•hr was used in all of the range and endurance calculations. This protective measure was taken in case the battery pack suffered any losses due to efficiency. The total weight of the fourteen batteries is 25.2 ounces, which amounts to about 23% of the aircraft's total weight.

3. Engine Control

The engine control device chosen for the S.T.o.R.M. was the Futaba FP-MC114H power controller. The advantage of this power controller was that it was capable of powering itself and all of the servos using the main battery pack. Other controllers require a separate set of batteries to power themselves and the servos. Thus the FP-MC114H helps Team Asylum keep the weight of the craft down because an extra battery pack was not needed.

The power controller did two things to change the power output of the S.T.o.R.M. First of all, as the throttle was adjusted, the RPM of the rotor increased to 1000 RPM at approximately the 80 percent throttle setting. As the RPM's increased with increasing throttle, the collective pitch of the main rotor blades also increased. This increased the angle of attack of the blades and thus created more lift as the throttle was increased. Above the 80% throttle setting, only the collective pitch of the main rotor changed. The advantage of changing the collective pitch was that a change in the collective pitch of the blades provides an immediate change in lift and thus makes vertical control much easier. Strictly changing the RPM does change the lift, but it takes a longer time to change the speed of the blades. (Schlüter, pp. 30-34). The S.T.o.R.M. was designed to hover at 1000 RPM with a blade angle of attack of 5 degrees.

Section F

Preliminary Weight Estimation Detail

1) Component Weights

The component weights of the S.T.o.R.M. are given in Table F-1. These weights were estimated from material weights, construction plans, and available parts. In some cases, weights were extrapolated from the information provided by a limited data base.

Component	Weight (oz.)	Distance from C.G. (in.)	Moment about C.G. (in.-oz.)
Fuselage	3.25	1.8	5.85
Engine	13	3.	39
Mounting Plate	2.5	2.1	5.25
Servomotor	.67	.9	.6
Servomotor	.67	-1.4	-.94
Servomotor	.67	-1.3	-.87
Servomotor	.67	1.4	.94
Landing Gear (Front)	4.0	4.	16.
Landing Gear (Tail)	2.0	-14.	-28.
Batteries	25.2	.315	7.94
Main Rotor Blades	6.32	0.	0.
Receiver	1.4	.8	1.12
FET (Speed Controller)	1.6	3.	.48
Main Rotor Head Assembly	2.82	0.	0.
Tail Rotor Head Assembly and Tail Rotor Blades	.42	-32.	-13.44
Tail Fin	.25	-31.	-7.75
Rubber bands	.07	.9	.06
Glue	1	0.	0.
Tail Boom	.8	-16.	-12.8
Gearing for Tail Boom	.42	-32.	-13.44

Rotor Shaft and Gearing for Main Rotor	8.29	0.	0.
Totals	**76.02**	-	**0.**
Cargo	0 to 40	0.	0.

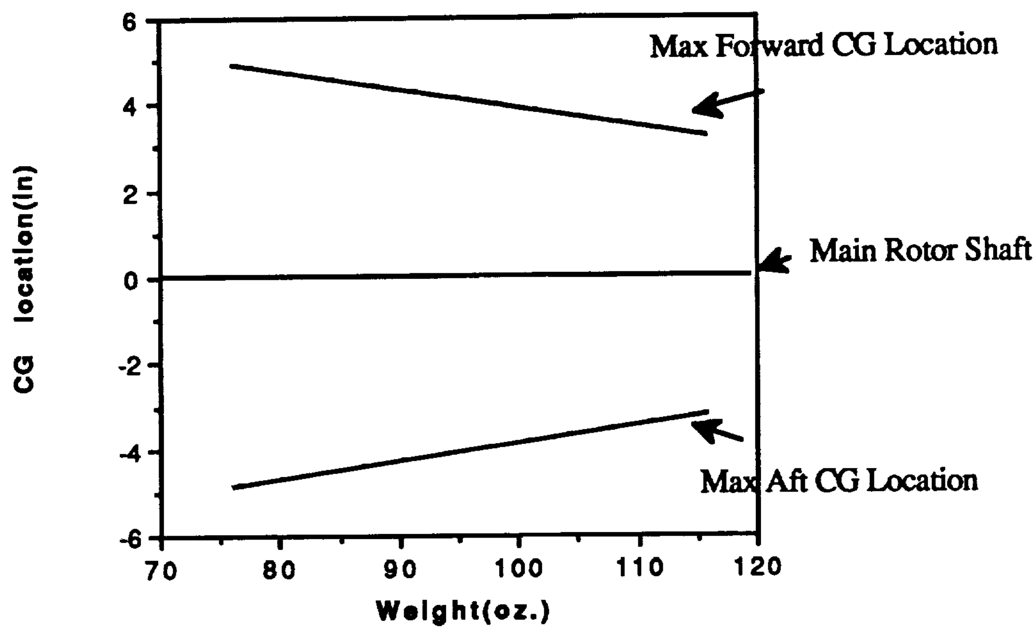
Table F-1. Component Weight Breakdown

2) *Center of Gravity Location and Travel*

For helicopter control, it is imperative that the center of gravity be located directly beneath the main rotor shaft. This is due to the fact that "a helicopter is at best 'neutrally stable', and more likely to be 'unstable'." (Schlüter, p.57) Positioning the c.g. anywhere other than beneath the main rotor creates an inherent instability. To overcome this instability, cyclic pitch can be used to create localized increases in lift. However, this significantly hinders the aircraft's maneuverability. Depending upon the amount of cyclic pitch available, the helicopter can become unmaneuverable, and even uncontrollable, if the c.g. is too far from the main rotor shaft. A more detailed discussion of how cyclic pitch works can be found in Section G. Since the loss of maneuverability is highly undesirable, all efforts should be made to insure that the c.g. is located beneath the main rotor shaft. Using the equation for the lift on a rotor blade(Drake, p.137), the cyclic pitch of the S.T.o.R.M. can overcome about 370 in.-oz. before the helicopter is no longer maneuverable. Figure F-1 shows the maximum c.g. displacement fore and aft of the rotor shaft as a function of weight. At these maximum locations, the helicopter can no longer maneuver.

The distances used in Table 5 were taken from the location of the main rotor. Since the moments due to the weights sum to zero about this point, it is the location of the center of gravity. Note that moments causing a pitch down were taken as negative for this table. The batteries are attached with Velcro and can be moved. By changing the location of the batteries, the c.g. can be placed beneath the rotor shaft for various cargo configurations. The center of gravity lies along the centerline of the aircraft due to symmetry. This symmetry should be maintained when cargo is loaded if at all possible. If not, the batteries can also be moved perpendicularly to this line to stabilize the aircraft.

Figure F-1



Section G

Stability And Control System Design Detail

1) ***Helicopter Control***

Due to the main rotor torque drive, the helicopter's fuselage will want to rotate in the opposite direction of the main rotor. To prevent this from occurring, a torque compensation mechanism is needed. This directional control of the fuselage about the aircraft's vertical axis can be done using one of the following methods: a propeller mounted on the side of the fuselage, a jet of air blown out sideways from the tail of the fuselage, a tail propeller with adjustable vanes, or a variable pitch propeller mounted crosswise at the tail of the fuselage. In addition to these mechanism for balancing the main rotor torque, the use of two main rotors was investigated. By operating the rotors in opposite direction, almost all of the torque effect can be eliminated. However, due to the complexity in the power system and controllability of such an aircraft, designing a helicopter with two main rotors was eliminated. With all of these considerations, a single shaft-driven main rotor with a tail rotor to counteract the torque is used in the design of the S.T.o.R.M. due to it's wide usage and availability.

2) ***Tail Rotor Sizing***

To determine the specific sizing of the tail rotor, an accurate calculation of the torque produced by the main rotor is needed. Knowing the amount of torque which the fuselage will encounter during flight provides the basis to size the tail rotor and the tail boom. The maximum torque will occur when the helicopter is hovering. To maintain this flight condition requires the most power output by the engine and, thus, results in the greatest torque produced to turn the main rotors. An equal and opposite torque felt by the fuselage will result. To counter this torque and maintain directional control, a sufficiently large tail boom arm needs to be employed. Likewise, an appropriately sized tail rotor needs to be designed which will be capable of producing the necessary "pull" force to counter the torque.

In deciding upon these parameters, a trade off in involved. A longer tail boom arm will require a smaller tail rotor diameter to produce the required pull force to counter the torque. Similarly, a short boom arm will need a larger rotor diameter to produce the same countertorque.

However, a longer tail boom arm adds weight to the overall design and would lead to the need for a more powerful engine to compensate for this added weight. So a smaller arm is desired to reduce weight. Another constraint which comes into play is the size of the main rotor blades. If the tail boom arm is shorter or around the same length as a main rotor blade, interference between the main and tail rotors will occur. For obvious reasons, this is undesirable. Since the S.T.o.R.M. will have a main rotor radius of 25 inches, it was determined that a tail boom length of 32 inches would be used to eliminate physical interference between the blades and prevent highly disturbed flow over the tail rotor blades.

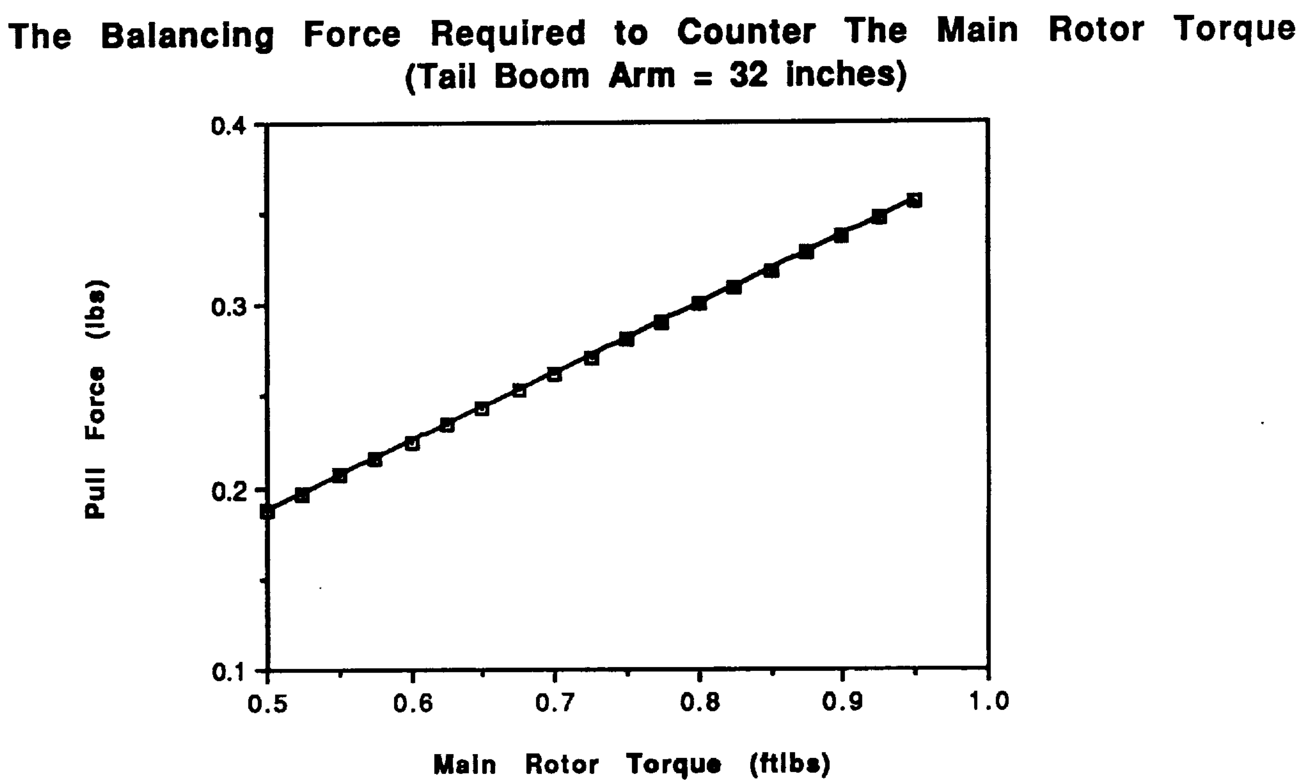


Figure G-1

Once the tail boom length is determined, a proper diameter for the tail rotor needs to be sized. To accomplish this, the tail rotor must be able to produce a sufficiently large pull force. In other words, the diameter needs to be large enough to produce a horizontal lifting force to balance the torque on the helicopter. This required pull force has a linear relationship to the torque required to balance the aircraft and is presented above in Figure G-1.

The tail rotor diameter could be determined by knowing the maximum pull force it must produce in order to result in a proper countertorque on the helicopter. From previous calculations, it was determined that the maximum torque that the motor will produce will be 0.53 ft-lbs.. Thus, the tail rotor must produce a minimum force of 0.2 lbs.. The tail rotor diameter was calculated from knowing the horizontal lift and using the same lift equation that was used for the main rotor:

$$L_{blade} = (1/6)\rho C_L S V_{tip}^2$$

where, again: ρ = density of air (0.00237 slugs/ft^3)

C_L = lift co-efficient for the airfoil section (0.5)

S = area of single rotor blade in sq. ft.

1/6 = reduction value for rotating blade including tip losses and inflow losses (Drake p. 13)

and $V_{tip}^2 = \pi$ D*RPM/60 (D = tail rotor diameter)

Figure G-2 illustrates the tail rotor diameters that are necessary for a range of pull forces and rotation speeds.

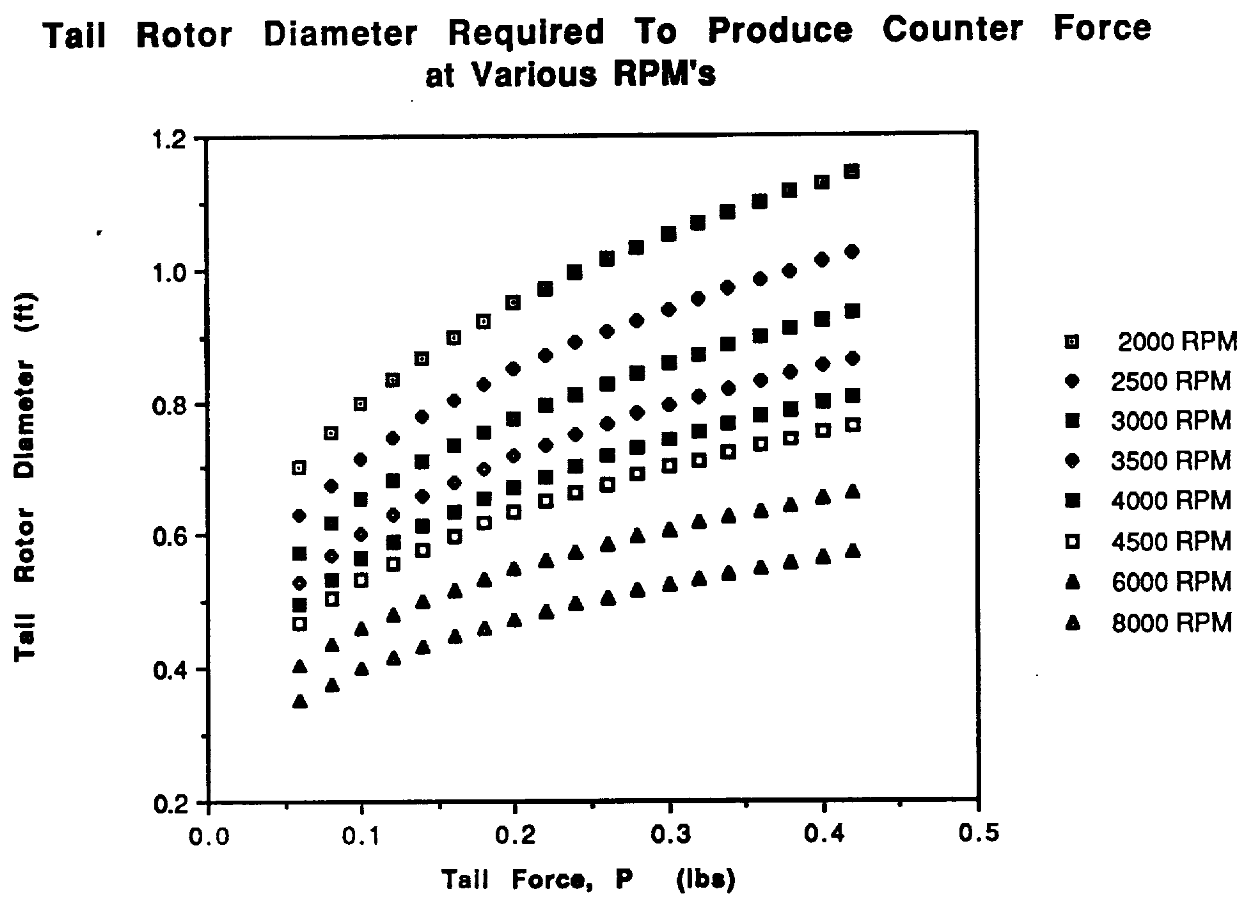


Figure G-2

From the previous graph, it seems that there are several different tail rotor diameters which will produce the required horizontal tail force. This is correct, yet a critical factor in deciding upon a tail rotor diameter is the rotational speed at which the blades will have to run in order to produce the same force. Figure G-2 shows that as the diameter of the rotor increases, the speed of the blades will decrease. If a smaller tail rotor diameter is chosen, the rotational speed will be extremely high. Large rotational speeds, however, are unfavorable due to the large resulting centrifugal forces produced on the tail rotor. The relationship between the rotor diameter and the rotational speed with centrifugal force comes from John Drake's text. The following equation is used:

$$F = (W/g)\omega^2 k$$

where $\omega = 2\pi RPM/60$
$k = (D^2/12)^{1/2}$
W = weight of rotor blade (2 oz.)

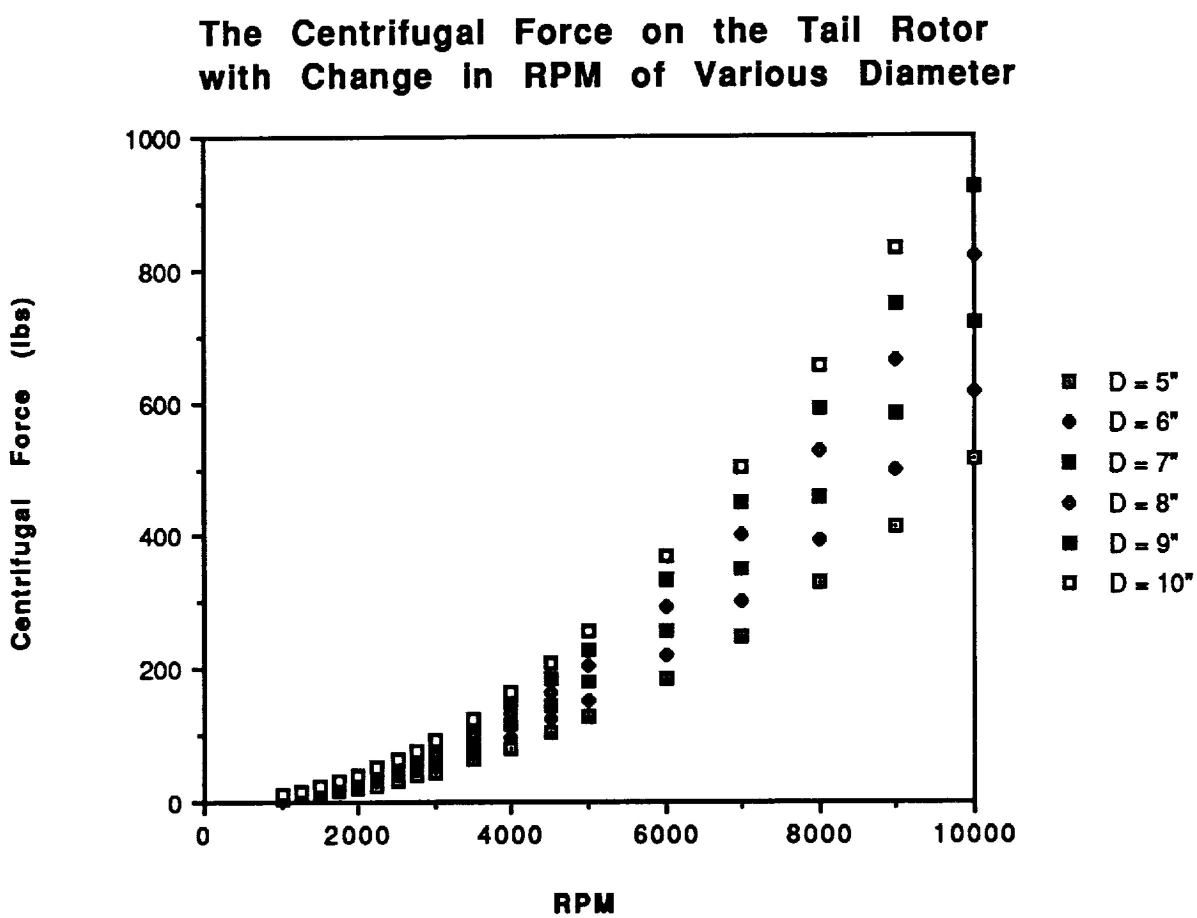


Figure G-3

Using the previous equation, a graph of the sensitivity of the centrifugal force due to the rotor diameter and the rotational speed can be determined.

Figure G-3 provides a visual explanation of how both an increase in the rotor diameter and an increase in the RPMs will produce a larger centrifugal force. The centrifugal force, however, increases linearly with rotor diameter and exponentially with RPM. The sensitivity of this force, therefore, is greatly reduced by limiting the speed of the blade rotation. Designing a helicopter with a tail rotor diameter which produces a centrifuagl force greater than 150 lbs. is a safety risk. For this reason, the rotational speed of the tail rotor blades should not exceed 4000 RPMs. Using this maximum rotational speed as a constraint, an appropriate tail rotor diameter to produce the required horizontal tail force can be designed. Refering back to Figure G-2, it can be seen that in order to produce a force of 0.2 lbs, a rotor diameter greater than 7.8 inches (0.65 ft) needs to be used. This diameter size will produce the necessary force while maintaining a rotational speed of 4000 RPMs. Although a larger rotor diameter would produce the required force at a slower rotational speed, the increase of the diameter adds undesirable weight to the overall aircraft. Since one of the main design concerns with the S.T.o.R.M. is the lack of power, a smaller tail diameter is ideal. A tail rotor diameter of 8 inches is designed for the S.T.o.R.M. since it was determined that the tail rotor would run at 3600 RPMs.

The tail boom length of 32 inches and the tail rotor diameter of 8 inches represent the minimum values required to maintain the S.T.o.R.M.'s equilibrium flight in the directional attitude. To change the aircraft's directional attitude, a tail rotor with collective pitch is used. This type of system is the most commonly used in helicopter design and allows the pitch of the tail rotor blades to be adjusted, providing additional thrust to the tail rotor. This control of the tail rotor thrust has a great advantage because it is very responsive and requires no change in the rotor's rotational speed. As there is only one motor driving both the tail rotor and the main rotor, changing the rotation speed of the tail rotor would also require a change in the rotation speed of the main rotor which would obviously affect the vertical flight of the helicopter.

3) *Vertical Control*

While the tail rotor will control the direction in which the helicopter's fuselage points, it will not control the direction in which the helicopter will *fly*. This is solely a function of the main rotor. The main rotor has two primary functions: to produce the thrust required in order to lift the aircraft off the ground (vertical flight) and to produce the thrust required to move the aircraft forwards or backwards (horizontal flight). There are two ways to control the vertical flight. The first is by changing the rotation speed of the main rotor. This is probably the most straightforward system mechanically. However, the reaction time between when the pilot inputs the control and when the helicopter produces the output is longer and requires a great deal of practice on the pilot's behalf.

The second way to control vertical flight is by means of collective pitch variation. Here, the rotor blades are not rigidly fixed to the rotor head and have the possibility to alter angle of incidence during flight. A collective change in the incidence angle for the blades is made and a corresponding change in the thrust is the result. The main reason for such a system is safety. Collective pitch also allows the blades to autorotate, enabling the helicopter to glide if the motor should quit. Additionally, with this system immediate control is provided compared to the delayed control of rotor speed controlling. This form of control is the one chose for the S.T.o.R.M. due to its safety potential and manufacturing availability.

4) *Horizontal Control*

A couple of mechanisms for horizontal control of the S.T.o.R.M. were studied. The first was by shifting the center of gravity of the helicopter in the fuselage. This would require that a mass be moved along a system of tracks to allow for such a center of gravity shift. This shift would produce a tilt of the rotor disc and thereby create a horizontal thrust. The major drawback to such a system is that it is very unresponsive and complicated. For this reason, cyclic pitch control will be used to create horizontal flight movement. This system involves changing the pitch of the rotor blades independently from the rotor head through use of a swash plate. The pitch of each blade varies with one rotation according to the angle of incidence of the squash plate. As an individual blade rotates, it constantly changes pitch to follow the inclination of the swash plate, which results in a horizontal movement in the desired direction of the swash plate. In addition, cyclic pitch provides the helicopter with the ability to overcome improper center of gravity location. If the c.g. location is not located directly under the main rotor shaft, cyclic pitch alters the rotor head to compensate for the weight imbalance. This means of accomodating a faulty c.g. placement is not recommended as the controllability in the direction of the shift will be decreased while controllability in the direction opposite will be extremely sensitive.

Section H

Performance Estimation

1) Take-off and Landing Predictions

Since the S.T.o.R.M. is capable of vertical take-offs and landings, the effective take-off and landing distances are zero.

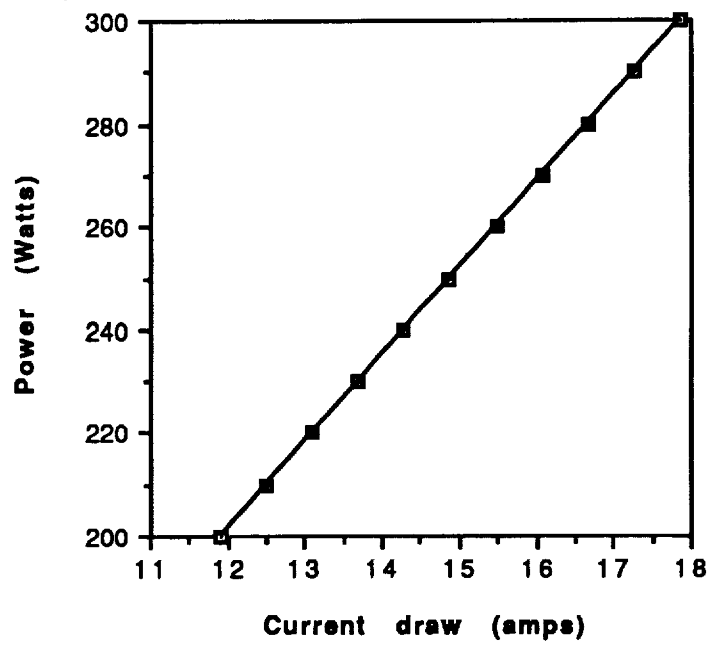
2) Range and Endurance

The range and endurance characteristics of the S.T.o.R.M. relied heavily upon the battery selection for the design. An important functional as well as economic aspect of this battery selection process was the current draw necessary to provide the motor with the power required to permit hover and forward flight. The current draw, i, was calculated in the following manner:

$$i = \text{Motor Power} / \text{Battery Voltage}$$

and its relation to the power required for the S.T.o.R.M.'s flight can be examined in Figure H-1.

Figure H-1
Battery Current Draw Necessary to Power Motor



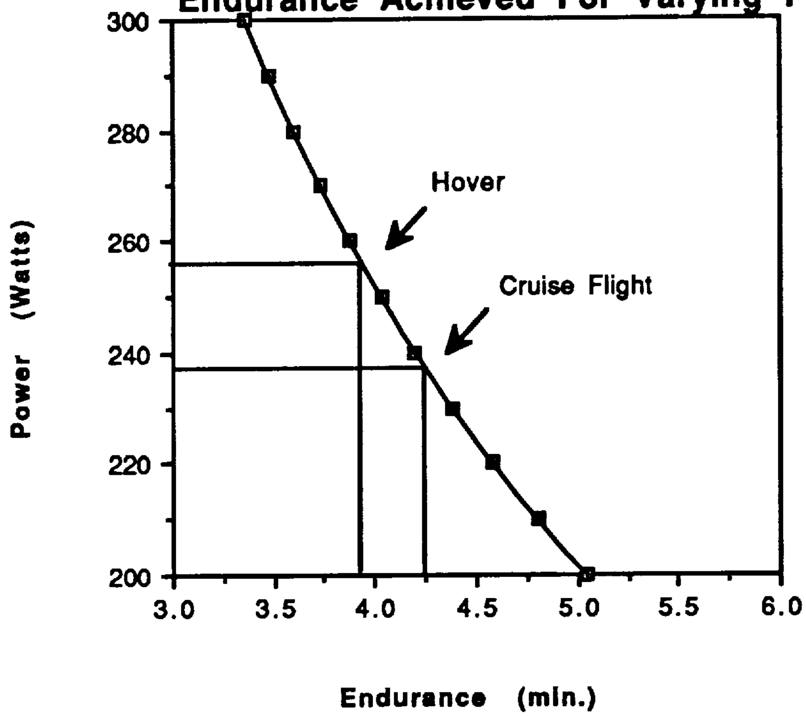
As can be viewed, a current draw of 15.18 amps is necessary to provide the power required to hover (255 Watts), and 14.11 amps is necessary to provide the power required (237 Watts) for flight at the cruise velocity.

With the current draw calculated, the battery duration could then be determined, thus providing the S.T.o.R.M.'s endurance capabilities. The endurance, E, was calculated using the following equation:

$$E = 1 / [(i / 1.0 \text{ amp} \cdot \text{hr}) \cdot (1 \text{ hr} / 60 \text{ min})]$$

Figure H-2 shows the effects that power required has on the S.T.o.R.M.'s endurance performance capabilities.

Figure H-2
Endurance Achieved For Varying Power



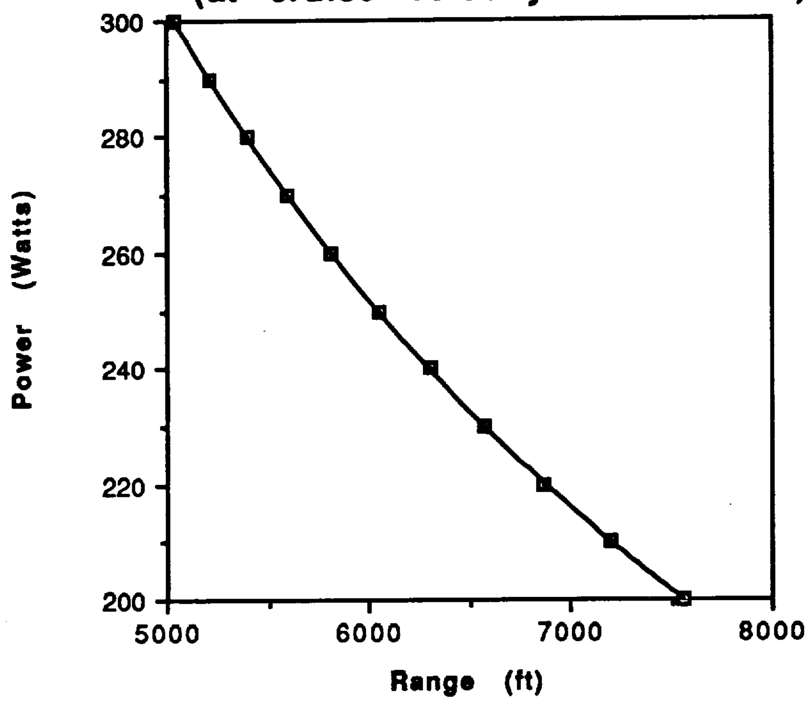
At the aforementioned hover power of 255 Watts, flight could ensue for 3.95 minutes, whereas flight could last for 4.25 minutes at the cruise power of 237 Watts. Both of these values fall somewhat short of Team Asylum's original design goal of 5 minutes.

As a direct result of the S.T.o.R.M.'s endurance capabilities, the aircraft's range possibilities can be determined. The helicopter's range was calculated using the following formula:

$$R = E \cdot 60 \text{ sec/min} \cdot \text{velocity}$$

where R stands for the range of the aircraft. Since the range of the S.T.o.R.M. is nonexistent while it is hovering, the range was calculated for flight while at the cruise velocity of 25 ft/sec. The effect that the power required has on the S.T.o.R.M.'s range can be observed in Figure H-3.

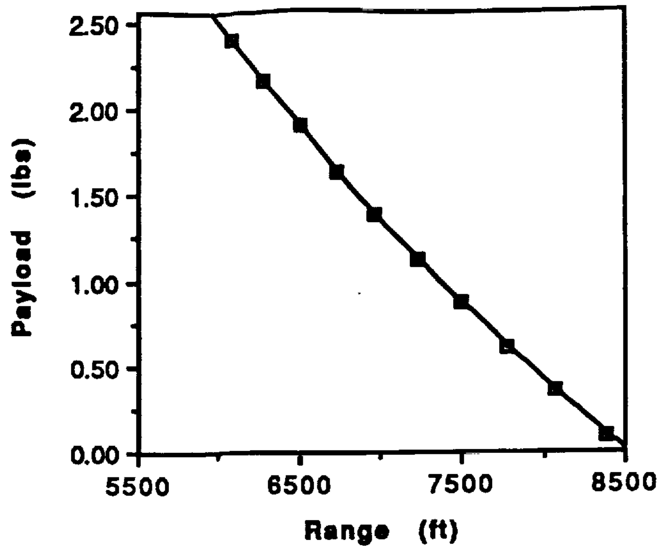
Figure H-3
Range Achieved For Varying Power
(at cruise velocity = 25 ft/sec)



Allowing for a ten second hover period for takeoff and landing purposes respectively, and using the remaining 3.92 minutes for flight at the cruise velocity, a maximum range for the aircraft of 5875 ft can be achieved. Once again, this falls short of Team Asylum's original design goal of 6500 ft.

It should be noted that, due to Team Asylum's commitment to transport the greatest density cargo of 0.04 oz/cubic inch in order to fulfill the original mission requirements, the endurance and range characteristics of the S.T.o.R.M. have been greatly hindered. If the payload density was reduced, the S.T.o.R.M.'s endurance and range capabilities would be greatly improved. In fact, this logical summation can be observed in the payload-range diagram shown in Figure H-4.

Figure H-4
Payload-Range Diagram for the SToRM



One can see that for lesser payload weights, the helicopter's range capabilities greatly improve. The S.T.o.R.M.'s range improves so much so that it would be capable of a range of just over 8500 feet if it was required to carry no payload whatsoever.

One advantage of the helicopter over a fixed-wing aircraft is that the loiter time of one minute should be all but negligible. Irregardless of adverse weather conditions, aircraft traffic, or runway length, the S.T.o.R.M. will be able to takeoff or land at a moment's notice. This advantage will protect greatly against the waste of valuable fuel and will allow the package delivery system to more easily remain on schedule. If necessary, the lack of a loiter time will also allow the S.T.o.R.M. to fly up to its maximum range. This could be a very beneficial aspect that would allow one of the helicopter transports to continue on to further destinations, especially when carrying lighter payloads.

3) *Power Reductions due to Ground Effect and Translational Flight*

i) *Ground Effect*

As shown earlier, the power required to hover the S.T.o.R.M., 255 Watts, was rather close to the maximum power available from the Astro Cobalt 25 motor, 270 Watts. Team Asylum was concerned that the S.T.o.R.M. would not be able to fly if the motor did not perform at optimum levels. These concerns were put to rest when it was found that the S.T.o.R.M. would not need 255 Watts of power to hover within ground effect. Ground effect was a phenomenon that occurred within approximately 1 rotor diameter of the ground. When the S.T.o.R.M. operates within ground effect, "the downward air stream from the rotor blows out all round (the helicopter) and forms a cushion of air slightly above the pressure of the surrounding air. This air cushion is the same principle that a hovercraft relies upon. The effect of this air cushion is to enable the rotor to support a much greater weight for a given horsepower at heights less than one rotor diameter." (Drake p. 18). This reduction in power required to hover due decreased the induced velocity through the rotor by a factor, G. John Drake recorded experimental data that related the reduction factor, G, to the value h/r, the rotor's altitude divided by its radius (Figure H-5).

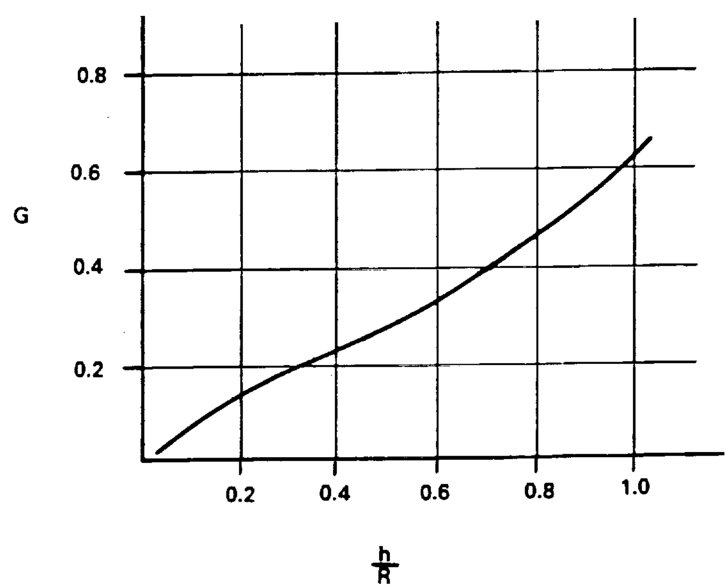
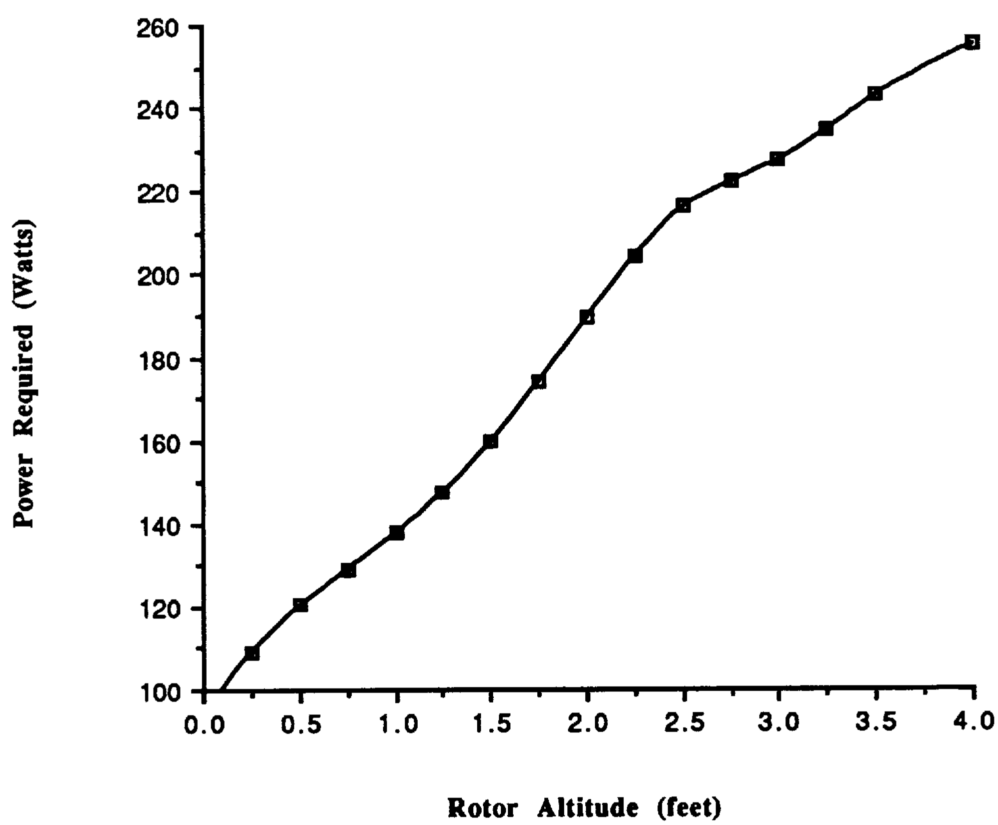


Figure H-5

Mr. Drake's data was reproduced on CricketGraph software and a fourth-order equation was found that best fit the data. Using this best fit approximation, and the fact that the induced power to hover within ground effect was the power to hover outside of ground effect multiplied by G, Figure H-6 was constructed to illustrate the reduction in the power required to hover S.T.o.R.M. at different altitudes. It should be noted that the horizontal axis, rotor altitude, represents the actual height of the rotor above the ground, not the actual altitude of S.T.o.R.M. Although the graph in Figure H-6 appears to increase linearly to an altitude of four feet, theorectical predictions indicate that the power required would hold constant at 255 Watts at higher altitudes. The error was due to the fact that the best-fit curve through Mr. Drake's data was not 100% accurate. The curve should asymptotically approach a limiting value. Ground effect did show that it was possible for S.T.o.R.M. to become airborne even if the full power of the Astro Cobalt-25 motor was not available.

Figure H-6
Power Reduction due to Ground Effect



ii) Power Required for Translational Flight

Besides flight in ground effect, it was also found that S.T.o.R.M. required less power to fly forward than it did to hover. This revelation came from the fact that helicopters operate primarily at very low flight speeds, which corresponds to the "back side" of the power curve. This is the portion of the power required vs. flight speed that most conventional airplanes dread venturing into. On the "back side" of the power curve, it actually takes more power to fly slower, and thus, less power to fly faster. When a helicopter flies forward, the main rotor is tilted forward an angle β to provide the forward thrust. As the helicopter flies forward, the new free-stream velocity increases the induced velocity through the rotor and thus reduces the airflow the rotor must pull through itself to lift the helicopter (Drake p. 21). The additional induced velocity, U' can be found trigonometrically through the relationship, $U' = V \sin \beta$,

where: V = forward flight speed (ft/sec)

β = angle of tilt of rotor plane (assumed to be 15 degrees).

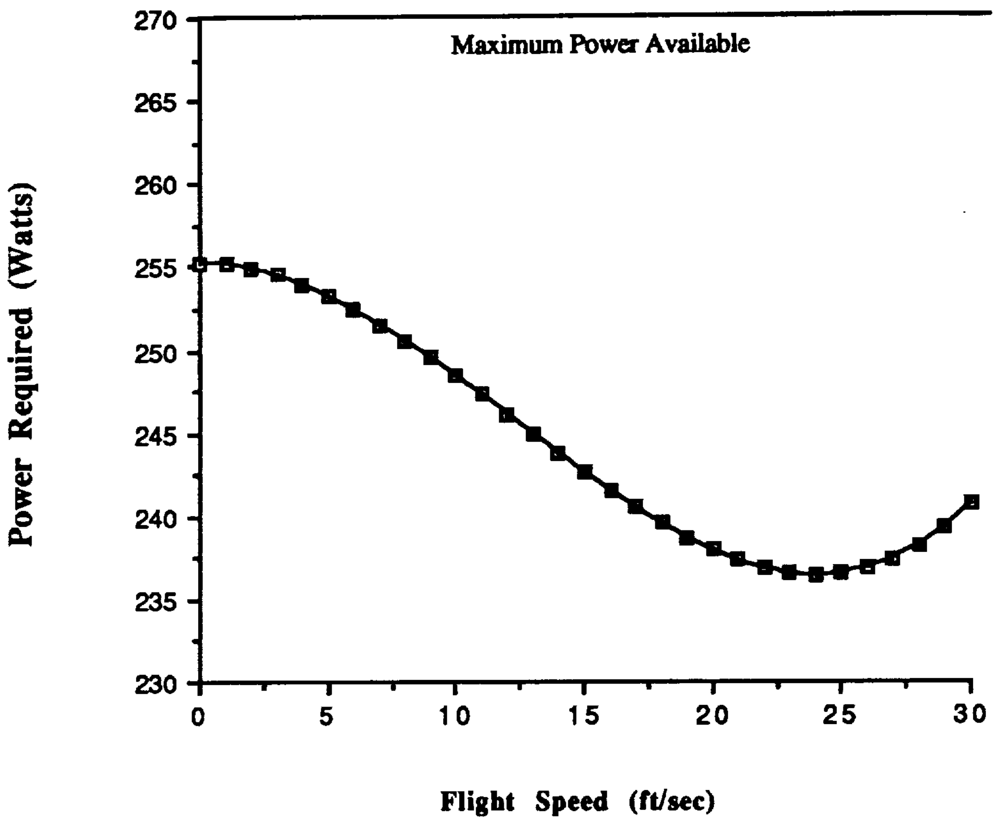
Substituting this value of U' back into the equation:

$$U' = (1/.8)\mathrm{sqrt}(L'/(2A\rho))$$

the value for the translational lift (or the lift produced by forward flight), L' can be found. This translational lift reduced the direct lift that needed to be produced by the main rotor and thus, decreased the power required to keep S.T.o.R.M. airborne.

In translational flight, the drag due to the fuselage also had to be overcome by the power of the motor. However, since the S.T.o.R.M. was designed to operate at low flight speeds, the effects of the fuselage drag was almost negligible. The power curve for the S.T.o.R.M. follows as Figure H-7. Note that the power required reaches its minimum value of 237 Watts at a forward flight speed of 25 ft/sec. This flight speed was thus chosen as the cruising speed for S.T.o.R.M. Above this value, the fuselage drag finally became a factor and that was why the power required began to rise above the forward speed of 25 ft/sec.

Figure H-7
Power Curve for 7.33 LB STORM



Between ground effect and the effect of translational flight on the power required to fly the S.T.o.R.M., it is conceivable that if the Astro Cobalt-25 motor does not produce the 255 Watts needed to hover outside of ground effect, then the S.T.o.R.M. should be able to leave the ground utilizing ground effect and approximately 200 Watts of power and then immediately begin an accelerated forward flight that would allow the S.T.o.R.M. to climb out of ground effect due to the reduction in power required to fly at forward speeds up to 25 ft/sec.

4) Rate of Climb

Since S.T.o.R.M.'s power required was always less than the power available from the motor, S.T.o.R.M. should always be capable of a positive rate of climb. Since rate of climb is defined as:

(Power Available - Power Required)/Aircraft weight

S.T.o.R.M. has a rate of climb from hover = (270 Watts - 255 Watts)/7.33 lbs

= 1.5 ft/sec

and at cruise speed of 25 ft/sec a max. rate of climb

= (270 Watts - 237 Watts)/7.33 lbs

= 3.31 ft/sec.

Section I

Structural Design Detail

1) Flight and Ground Load Estimation

Due to the fact that the mechanical properties of the main rotor blades were unavailable, it was not possible to accurately create a V-n diagram for this aircraft. However, the maximum flight loads during hover would be 7.3 lbs. The maximum load that the system is capable of generating is 7.4 lbs. The ground load on the blades is 6.32 oz. Each landing gear leg supports 2.5 lbs during ground operations. It should be noted that the helicopter blades stall at high velocities, so the shape of a V-n diagram would be rectangular. The maximum forward speed of the helicopter is 39 ft/s. Due to the way a helicopter flies, the loads at higher velocities are similar to the hover loads.

2) Basic Structural Components

The airframe consists of the engine platform, the tail boom, and the cargo bay. The engine platform is a flat plate to which the rest of the structures are attached, and thus it is subjected to fairly large loads which are transmitted to it through the other components. The tail boom will be a hollow cylinder. The cargo area is simple truss design. The two side walls and the floor use the configuration shown in Figure I-1. The floor will have a sheet of wood laid over it to prevent the cargo from falling through. The back wall will be constructed as shown in Figure I-2. The shaded area is removable to enable easy cargo access. A top view is shown in Figure I-3 with the shaded area representing where the engine platform will be. The flaps that fold over the propulsion system are simple 4 member trapezoidal trusses. The entire cargo area will be covered with a thin protective coating, Monokote™.

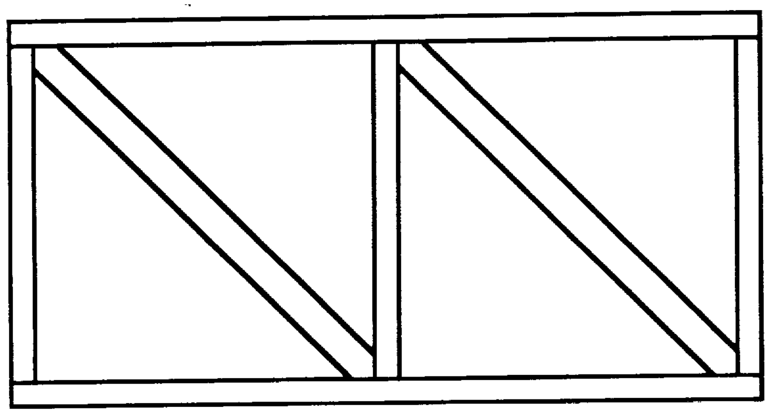


Figure I-1

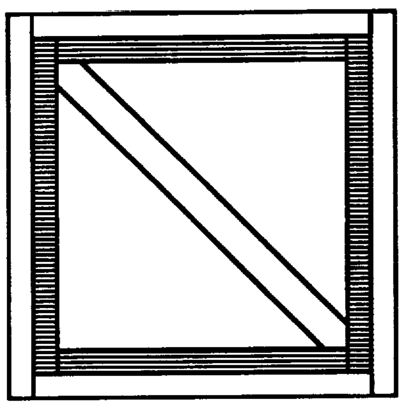


Figure I-2

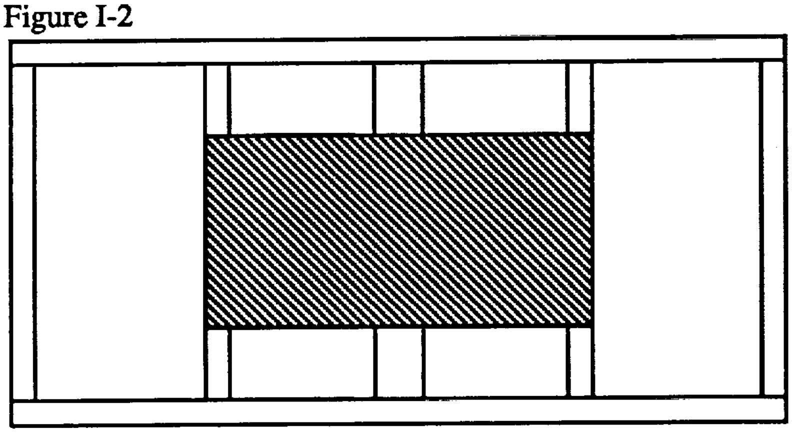


Figure I-3

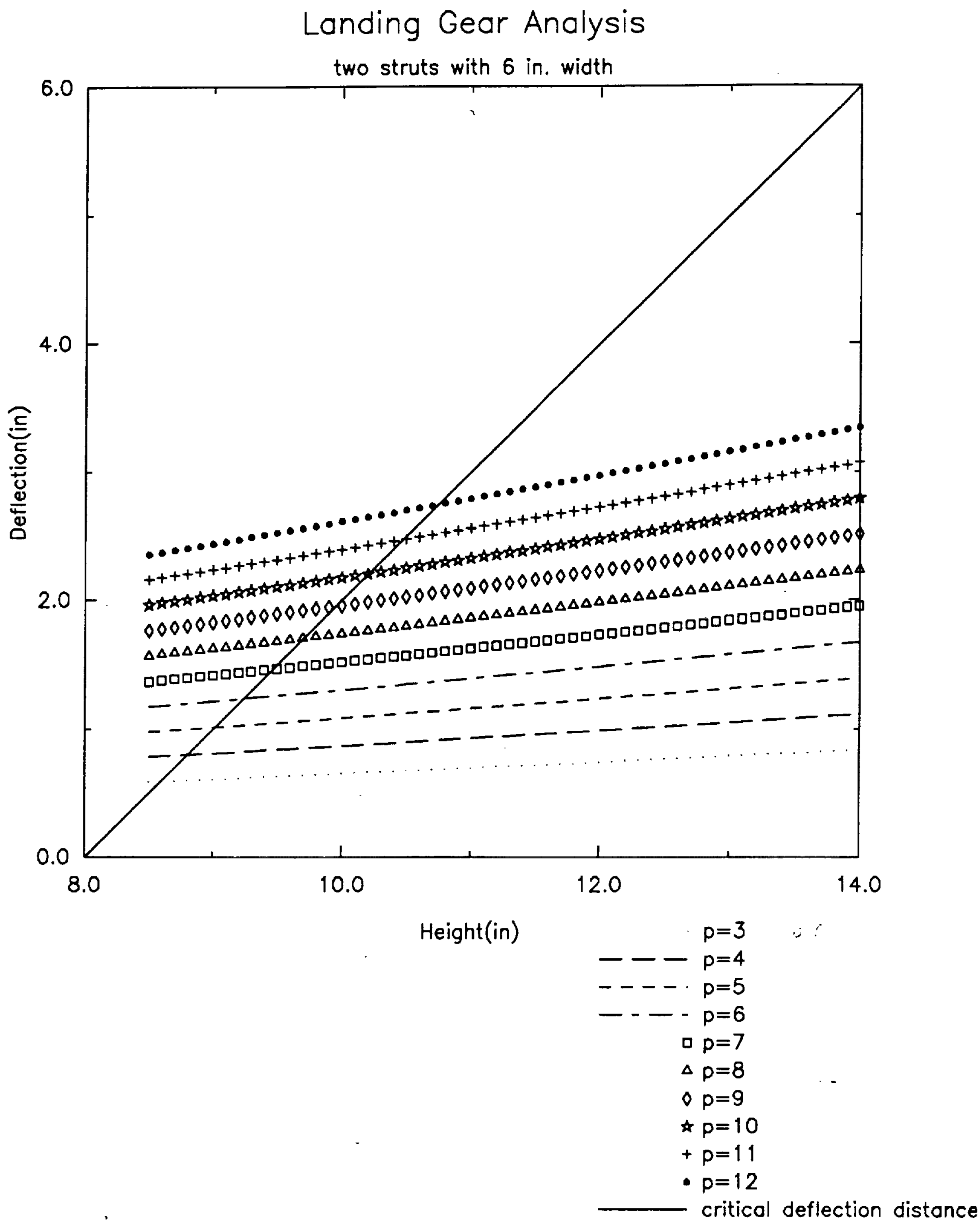
3) *Materials Selection*

The engine platform will be constructed of sturdy plywood to insure that the loads it encounters can be withstood. Engine torques of .53 ft-lbs., landing loads of 5 to 15 lbs., and propulsion system lift forces of 5 to 7 lbs. will be directed to the platform through various components. With a yield stress of 4000 psi and ultimate stress of 5000 psi, the plywood platform will be capable of handling several times the loads of normal operating conditions. This insures protection form fatigue. The tail boom is an aluminum alloy, chosen because of its light weight, .8 oz, and sufficiently high strength. The cargo bay trusses will be made from light weight balsa wood. The loads to which this structure will be subjected are relatively low, 0-40 oz. for normal operation, and the structure can easily accommodate these loads. The protective covering will be black Monokote™ because it is light, smooth, and easy to work with.

4) Landing Gear

Using simple beam theory analysis, the landing gear was designed to sustain excessive loads. A tricycle gear will be used, and each of the three tires will be supported by two 1/8 in. diameter steel wires. Steel was chosen because of its excellent strength characteristics. The length and angles of the landing gear were arrived at using a FORTRAN program that is contained in Appendix B. This program found the deflection distance as a function of height, width, number of struts, and load. Figure I-4 shows the deflection distance of the final landing gear configuration for various loads. The two tires of the main gear will be 20 in. apart and 14 in. below the engine platform to which they are connected. The rear tire will be 14 in. below the engine platform and 14 in. behind the center of gravity. A 1g load for a tire would be about 2.5 lbs. As the landing gear is configured, the craft can sustain a 5g landing without the cargo bay striking the ground. This is important because the cargo bay was not designed to sustain excess loads. For a 5g landing, the steel wires are not subjected to loads that are sufficient to cause plastic deformation or buckling.

FIGURE I: 4



Landing Gear Analysis
two struts with 6 in. width
6.0
4.0
2.0
0.0
Deflection(in)
8.0
10.0
12.0
14.0
Height(in)
p=3
p=4
p=5
p=6
p=7
p=8
p=9
p=10
p=11
p=12
critical deflection distance

Section J

Construction Plans

1) *Major Assemblies*

Engineering modifications to the prefabricated assembly were considered by the engineering design team and in the future, these variations may be integrated into the craft. At the present time, however, they will only be discussed and not built into the technology demonstrator.

Because of the intense detail and engineering technology involved in the mechanics of a rotary-wing aircraft, the base unit is a commercial assembly purchased through the sub-contractor Helicopter World. The assembly is part of a package including mechanics, radio, engine, and servos. The stock assembly will be used,without modifications, to ensure a successful demonstration of the technology. Also, use of the standard assembly, along with a capable pilot, will help guarantee the safety of the spectators and those involved in the test flights.

A fuselage was added to the initial assembly allow for cargo carrying capabilities. The fuselage was constructed to be light weight and aerodynamic. The large surface area of the fuselage allows an additional money-making avenue through sponsorship. The success of this tactic has been shown by the novelty of the Goodyear blimp.

Other modifications of the main assemblies of the helicopter are an extended tail boom and improved landing gear. The boom contains, internally, the drive and control mechanisms to the tail rotors. Hence, the extension of this anti-torque arm is very sensitive and crucial. The tri-cycle landing gear, obviously essential to a flying mission, was designed according to section I-4 to withstand appropriate loads. The re-inforced wire legs are connected to the main load-carrying platform under the mechanics and fit around the fuselage.

2) *Complete Parts Count*

The parts list for this design is extensive with a list of more than 120 items. This list is presented in APPENDIX C part *ii*.. To this list was added the materials required for the construction of the fuselage, mounting plate, and landing gear. The fuselage consists of a balsa frame encapsulated with black Monokote™.

The forward landing gear is a bent wire construction (1/8 in. diameter steel) with a length of 11 inches from both sides of the mounting plate. The back gear employs a double thickness wire of the same diameter whose length is, again, 11 inches. The light-weight foam wheels are 1.5 inches in diameter.

The additions to the base assembly will has a parts list including the following items:

1. fuselage construction materials
2. additional tail boom with appropriate belt and control devices
3. longer blades appropriate length, section, chord
4. fly bar extension
5. landing gear materials

3) *Assembly Sequence*

Assembly began with the base assembly. Abridged instructions for its construction are presented here.

1. Battery mount installation
2. Landing skid installation
3. Tail rotor assembly
4. Fin installation
5. Rotor head installation
6. Servo selection/mounting
7. Cyclic control rod installation
8. Elevator control rod installation
9. Tail servo installation/Linkage assembly
10. Collective pitch linkage installation
11. Radio gear installation

Upon completion of this segment of construction, the fuselage work began by laying out a full-scale draft copy of the design. Using this measurement and placement tool, the team cut the 1/8 inch balsa rod into the appropriate lengths for each of the areas needed. The simple box configuration was selected because it was optimizes cargo space and is relatively straightforward to manufacture. After the basic frame was glued together and a small nose cap was added in front, the body was carefully Monokoted. The batteries, in the case of the technology demonstrator, were secured with velcro to the bottom of the fuselage which had been reinforced with vertical members to transfer some of the weight to the mounting plate.

The landing gear was constructed by measuring and bending the wire rods to the design lengths and attaching them to the load-carrying platform. To secure them to the plate, special slots were added to the plate to stop the sideways sliding motion of the legs. (because of the weight of the S.T.o.R.M., the legs of the landing gear had a tendency to separate, causing the fuselage to reach the ground.) Wheels were added to the free ends of the legs and secured with alloy lugs.

Derivatives of the S.T.o.R.M. will have the capability to handle the volume cargo prescribed in the DR&O. In order to accomplish this, however, the aforementioned size modifications -- extended tail, blades, and fly bar -- must be included. The blades and fly bar must be constructed by a contractor as the workshop in Hessert does not contain the necessary equipment, nor does the design team or company have the necessary technology to do this themselves. Construction of the extended tail is just as complex; an additional aluminum pipe must be spliced into the standard boom. An additional problem arises from the method in which the tail rotors are driven. The tail blades are driven via a toothed belt that is housed on the interior of the boom. Due to the large forces being exerted on the belt during normal operation, the vulcanized rubber belt cannot simply be spliced with another to increase its length. To do so would compromise the safety of the Aeroworld inhabitants as well as jeopardize the flightworthiness of the S.T.o.R.M. For this reason, the extended belt must be provided by a subcontractor as well.

Section K

Environmental Impact and Safety Issues

1) *Disposal Costs for Each Component*

The state of the ecology today dicates that the engineer consider the effect on the environment of any given design. After the life of this project has come to an end, the S.T.o.R.M. will remain. As conscious individuals, engineers must consider the future. Disposal of the aircraft is important, yet this concept presents a rather special case. Due to the high initial cost of the original components, the vehicle will remain intact to be handed down as a learning resource for others who may be interested in this form of technology. The prototype and its documented development will form a resource pool from which future generations of innovative thinkers can draw data, hands-on experience, and spare parts.

Undoubtedly, the first several derivatives and improved models will be used in the same manner. Ideally, all crafts should be handed down and renovated -- changed into something new. Realistically, however, the best that can be hoped for is that the parts may be recycled and used in future projects or for other purposes. If this is not possible, there exist various agencies today that recycle different metals and plastics. The helicopters must not, and will not, remain whole only to stock landfills.

2) *Noise*

Some concern over the noise issue was raised at the beginning of this project. Helicopters are typically loud aircraft. Possibly because of the extremely low mission speed, however, the S.T.o.R.M. is really rather quiet. In fact, testing in the same environment with fixed-wing aircraft demonstrated to the Aeroworld inhabitants that this craft is no louder than any of the other competing designs. The concerns for the noise effects of the S.T.o.R.M. are aided by the fact that the propulsion system is electric. Unfortunately this is perhaps the only advantage of using an electric engine for a rotary-wing aircraft. In any case, since this will be primarily a night flyer, surrounding residential areas will be content and less apt to complain about airport noise after a peaceful night's rest.

3) ***Waste and Toxic Materials***

Waste and toxic materials must be disposed of properly. Main contributors here are battery packs and exhaust. The battery packs can be recycled through any number of already existing organizations. Care must be taken in consideration of exhaust, as well as any toxic materials that may result from this operation. Spills and accidents will be handled quickly and quietly, according to defined regulations. Exposure to engine exhaust even on a daily basis is not a health risk to any of the inhabitants of Aeroworld or to the pilots who will fly the aircraft. Company regulations prohibit indoor operation of any hardware and outside ventilation is adequate to avoid any problems.

Section L

Economic Analysis

The ultimate design objective for the economic feasibility of the concept is to produce an aircraft that will deliver similar amounts of cargo in similar lengths of time for a substantially less price than other aircraft currently in the marketplace. This is translated into focusing on minimizing the cost per cubic inch of cargo.

The economic costs of the aircraft, and therefore the required charges per in^3 or ounce, are affected by a whole host of variables as described in the costing equations described in the course material. These variables include prototype cost, prototype construction hours, number of servos, the distance travelled, cruise velocity, battery exchange time, battery size, fuel costs, design life, maximum payload, design weight and design range. This list is obviously very long but by performing parametric sweeps with several of the aforementioned variables, it becomes abundantly clear which variables contribute most significantly to the overall cost of the delivery system.

1) *Production Costs*

The main variables influencing the production costs were the construction manhours and the total cost of the prototype. By fixing the values for the fuel consumption, 13.81 amps, the fuel cost, $10 /mAhr, and the maintenance and operations costs, the effect of varying these design variables could be examined. The results are seen in Figures L-1 and L-2 below:

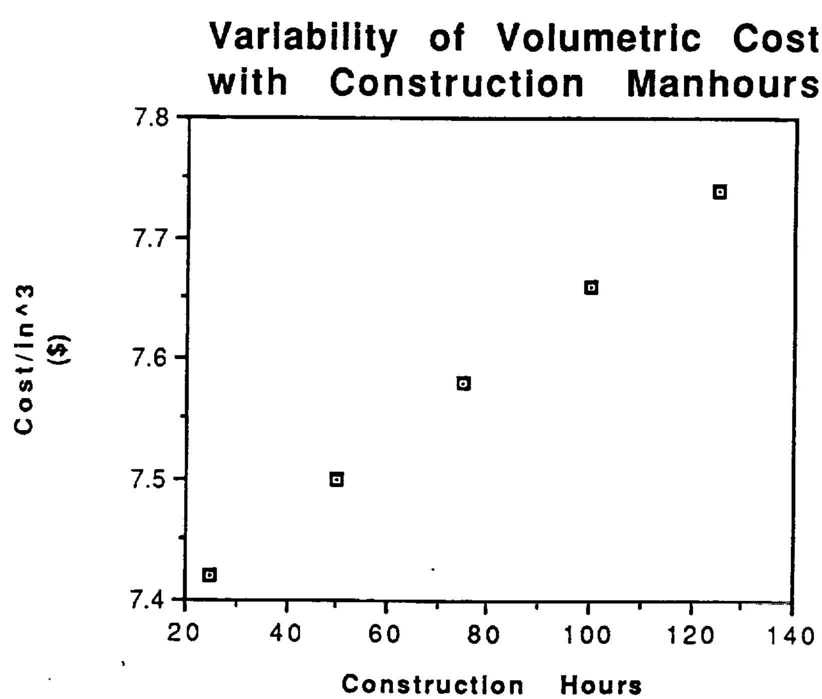


Figure L-1

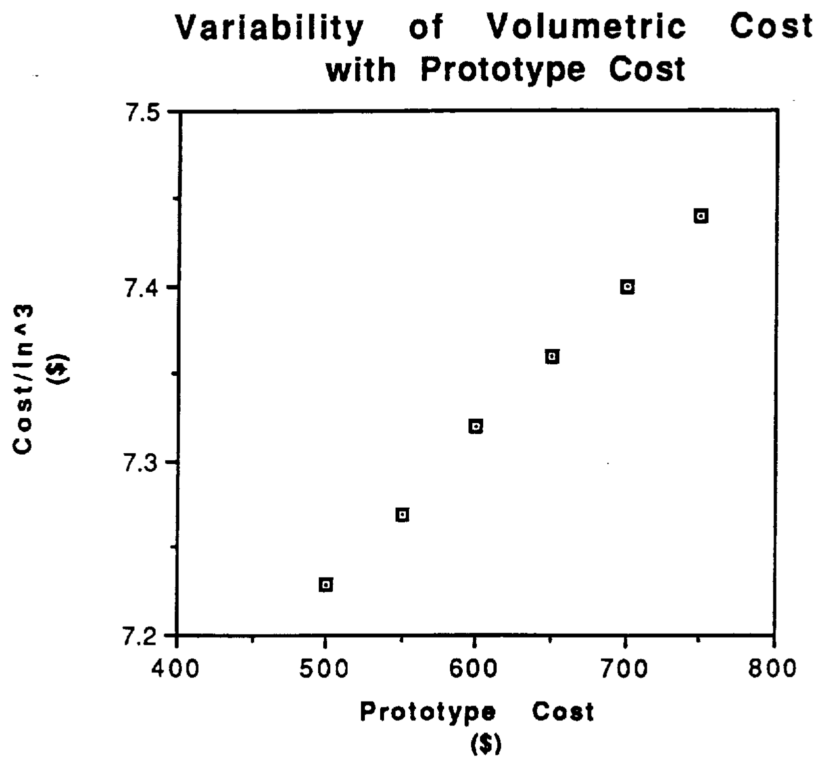


Figure L-2

It can be seen from both figures that even as much as a 600 percent increase in one of the design variables (construction manhours), the delivery cost/in^3 is increased by only four percent. This is despite the fact that the production costs account for 14.6 percent of the overall delivery costs for a package. These figures were obtained using the spreadsheet in Appendix D. This spreadsheet computes the unit costs per flight and total delivery costs based on the actual flights flown as shown in the distribution network of Figure B-1. This distribution system requires 28 helicopters to deliver the 34690 in^3 of cargo through a total flight distance per day of 160133 feet. This spreadsheet also assumes a total volume of 1024 in^3.

The overall production costs for the S.T.o.R.M. were slightly higher than those originally anticipated. As mentioned previously, this was due to the higher than expected cost of purchasing all of the technology and assembling it. An attempt was made to defray this cost somewhat by predicting a slightly reduced assembly time of 100 hours, but the cost per aircraft still weights in at a hefty $445,200. This represents an actual cost to the manufacturer, not the customer, and it is based on prototype construction costs of $863.00. With the construction of the technology demonstrator it became clear that the final costs would be even more. The final cost of the technology demonstrator was $915.11 with a construction time of 47 manhours. The reduction in manhours slightly offset the additional costs of the materials, but the S.T.o.R.M. remains significantly more expensive than competing designs.

2) *Maintenance Costs*

The maintenance costs, being based only on the time required for a complete battery exchange, did not factor into either of the distribution network costing schemes significantly. This can be seen from Figure L-3 below:

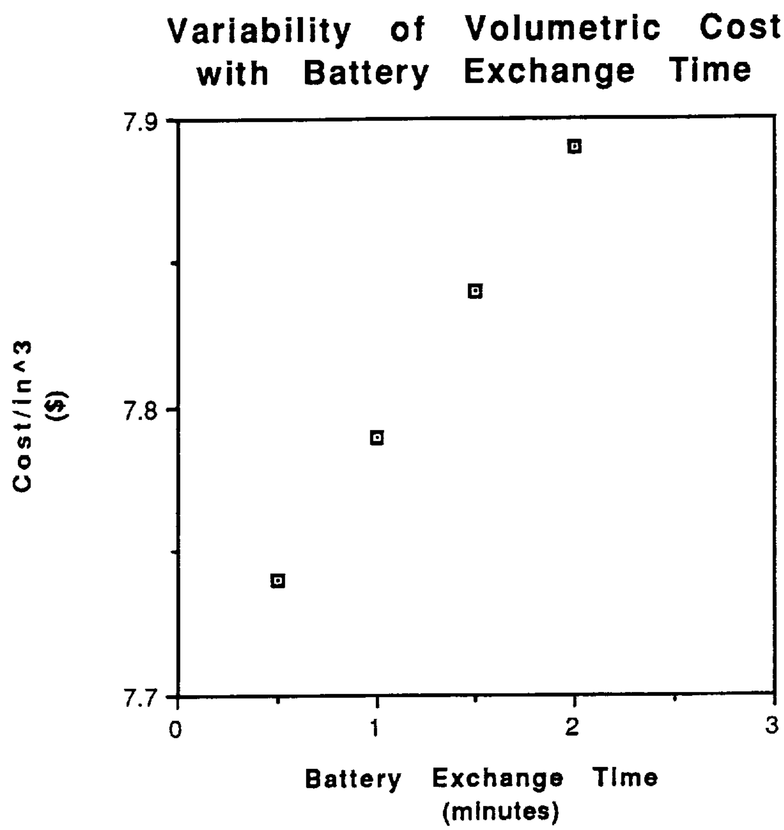


Figure L-3

The maintenance costs accounted for 0.59 percent of the total delivery costs and for this reason, battery accessibility was not made a priority in the design process. Using a predicted battery exchange time of 2 minutes, the maintenance costs per flight were only $50.00.

3) *Operation Costs*

It was initially thought that the operations costs, based on the flight times and the number of servos, might be quite a significant portion of the total flight cost. This influenced the distribution network design and it was one of the reasons the helicopter appeared to be disadvantaged due to the large number of servos required. As it turned out, the operation costs were the least significant of any of the total cost components with a cost per flight of only $23.00 which amounts to .37 percent of the total delivery costs. The almost complete independence from the delivery costs of the number of servos used in the design is obvious from Figure L-4:

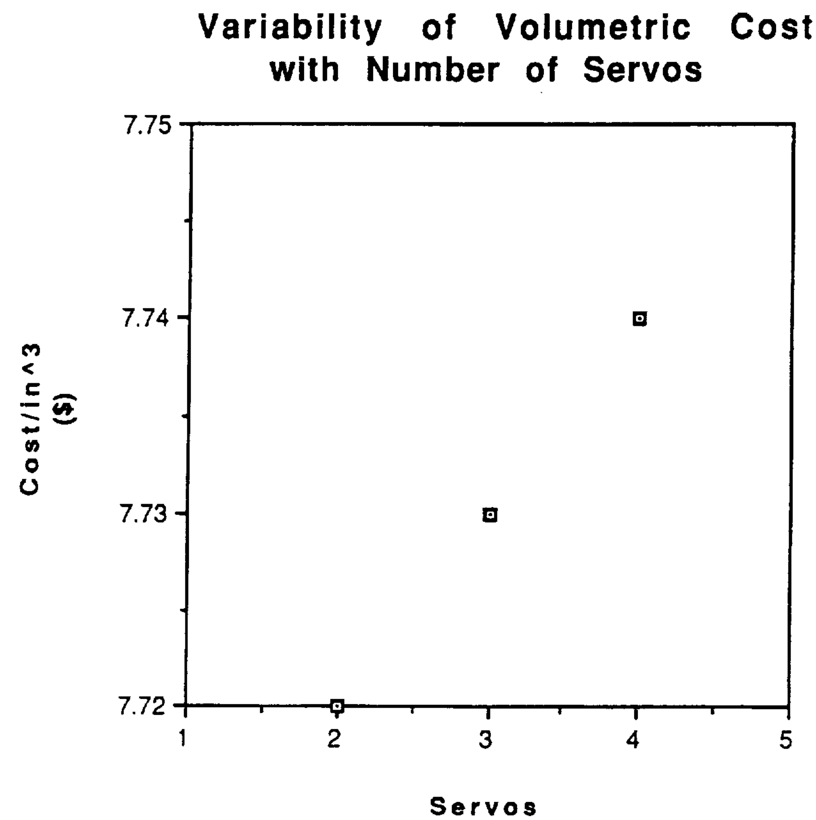


Figure L-4

4) *Fuel Costs*

What does prove to be the driving factor in the economic viability of the helicopter design is the fuel costs which are directly related to the fuel cost (per milli-amp) and the average flight distance, and its respective flight time. Perhaps the most significant and fuel saving element in the design process is the distribution network itself. Anything that can be done to minimize the number of planes used and the average flight times can shave the fleet costs and delivery costs by as much as 28%. Figure L-5 shows the overall importance of each of the cost variables as explained earlier.

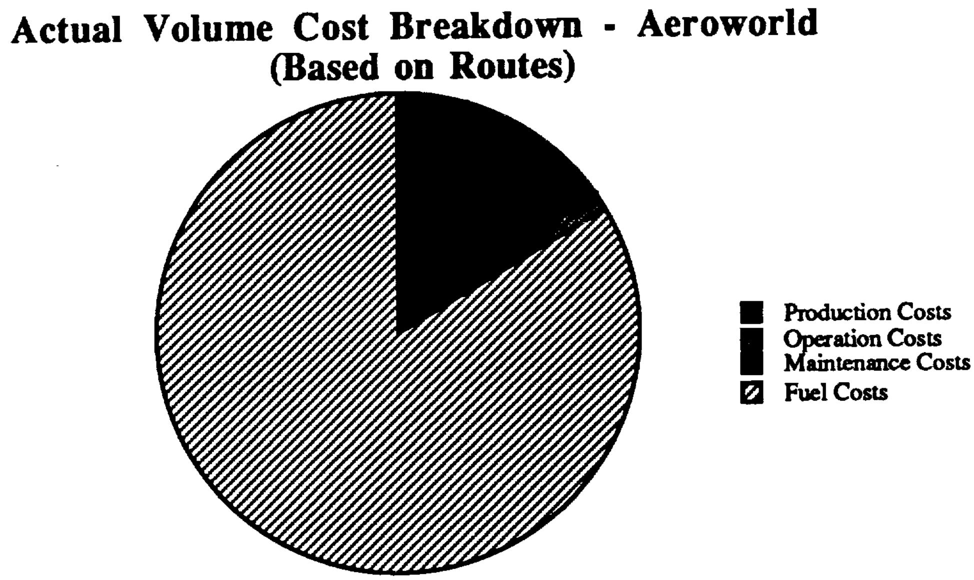


Figure L-5.

It is clear that the most potential for savings and cost reduction is through fuel savings. This is a direct result of the distribution network established which influences flight times and, therefore, the amount of fuel burned. In order to determine the ultimate distribution network, it was first attempted to write a computer program that would evaluate the delivery costs for networks with hubs in each of the cities in Aeroworld. The computer would divide the amount of deliverable material from any given city by the cargo volume of each aircraft to arrive at a required fleet size. The flight distances were simply a summation of the distances of each city from the hub multiplied by the number of aircraft servicing that city. Economic analysis was performed using the same design parameters and values as mentioned earlier. With this program, however, it was difficult to take into account such intangibles as relative locations of one town to the other and possible combination of routes. Using the computer program, therefore, it was only possible to determine delivery costs based on having aircraft doing nothing more than flying from one town to the hub and back again at various fill levels. This yielded very high delivery costs due to the relatively large amount of wasted cargo space and extremely high mileage per day. Using this system, with a hub at City J and City K, the delivery costs per in^3 were $13.91 and $14.97, respectively. Using the incredible power of the human mind and its ability to put together some more meaningful route combinations and networks, it was possible to achieve a delivery cost per in^3 of $8.66. This network was designed such that the aircraft always flew the shortest possible distances to deliver the parcels in aircraft that operated at or near the volume capacity and that did not fly any more than four flights per day. This network employed a single hub with primarily a spoke concept, but it combines routes that would otherwise fly at less than 75% capacity and it also uses supplemental routes to deliver cargo intra-region on city-exclusive routes, that is, routes that don't even go to the hub at any time. Using this distribution network of 28 planes with the same number of flights per day as recommended by the computer simulation, the aircraft fly at an average fill of 96%.

Due to the low range predictions and short flight times, it is proposed that the optimum mission for the S.T.o.R.M. may be as a short hop, intra-region cargo carrier servicing local markets only as shown in Figure L-6. With this distribution scheme, the fuel costs become less and less important allowing significant price reduction measures by making significant those things over which the designers have some measure of control. The cost breakdown for the regional markets is shown in Figure L-7.

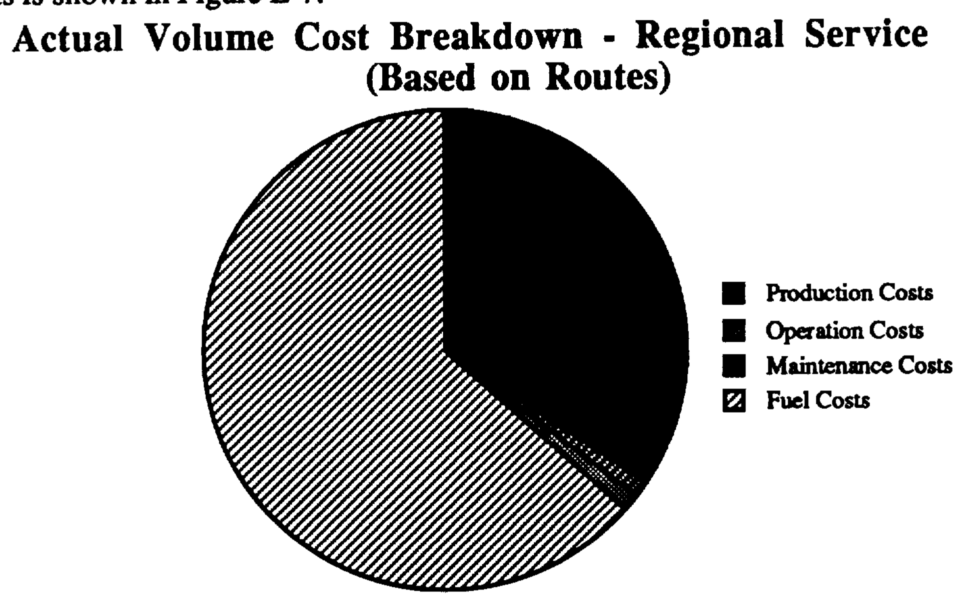
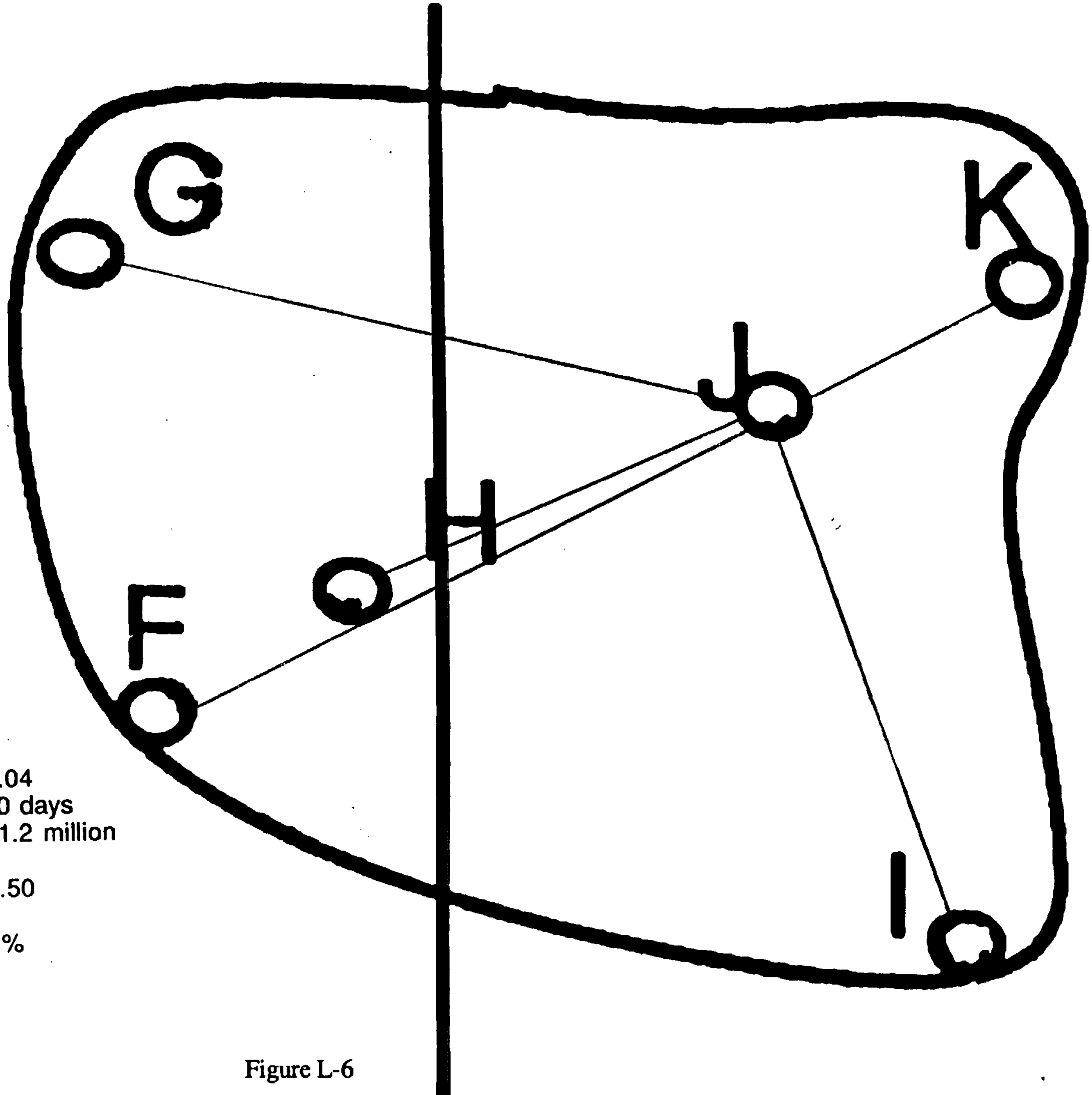


Figure L-7.



Revised Network

Number of Helicopters	10
Flights per Day	20
Cost/in^3	\$3.04
Fleet Life	300 days
Fleet Life Cost	\$11.2 million
Proposed Charged Cost	\$4.50
Return on Investment	48%

Figure L-6

4) *Fleet Life Cost*

A final consideration is the fleet life cost. As the fleet needs replaced on the order of once every 250 days, it is obviously very important to reduce the replacement cost to a bare minimum. Feet life cost is reduced by reducing the number of flight cycles designed for as well as decreasing the number of aircraft required to suitably distribute the cargo. This can be seen in Figure L-8 below:

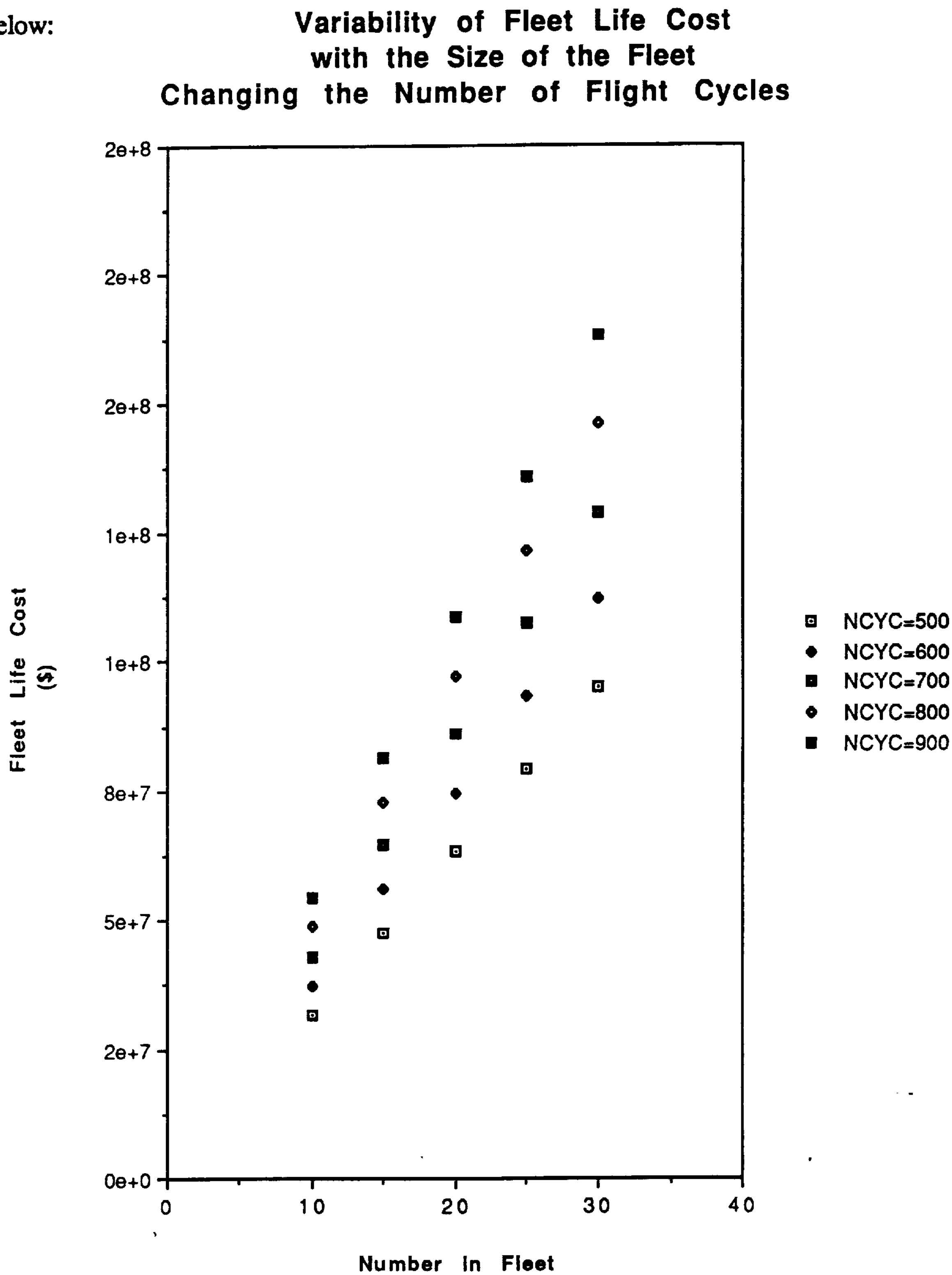


Figure L-8

These two things were taken into consideration when planning the distribution network and the low design flight cycles is very good for the structures group as well. For our network of 28 planes, flying 68 flights/day, designed for 600 flight cycles, the fleet life cost should be right around $73,000,000. This seems very high but to start an entire design network, it is really quite reasonable.

Section M

Technology Demonstrator Development

1) Complete Configurational Data, Geometry, Weights and Center of Gravity

Table M-1 gives the final configuration for the prototype that was created. The center of gravity corresponds to the location of the main rotor shaft. The positive direction for the distances was taken to be the direction from the rotor shaft towards the nose of the fuselage. Due to a small round-off error, the sum of the moments about the c.g. is not exactly equal to zero.

A comparison of Tables F-1 and M-1 quickly reveals a large discrepancy between the initial design and the prototype which emerged. The reasons for this stem from problems which occurred that resemble those found in "the real world". The high costs associated with implementing a new technology put a strain on the resources that management was willing to appropriate towards the development of the prototype. The design team was encouraged to attempt to keep costs down without sacrificing safety considerations. Since the purpose of the prototype was to be a "technology demonstrator", it was decided that the best way to safely demonstrate the feasibility and capabilities of this technology was with a scaled down model. The prototype created is approximately 70% the size of the original design. This size was determined by the fact that the main rotor blades used in the prototype are approximately 70% the size of the blades used in the original design.

By creating a scaled down prototype, the following problems were eliminated, enabling the prototype to be constructed with fewer resources. The purchase of expensive, custom made main rotor blades was no longer necessary. The addition of a much larger engine and the creation and installation of the appropriate gearing was similarly avoided. The lengthening of the tail boom and the acquisition of a longer belt to drive the tail rotor was not necessary. The most difficult problem that was avoided concerned the speed controller. The larger engine would have required a larger voltage supply. However, this increased voltage far exceeded the allowable voltage range of the speed controller. Speed controllers with sufficiently high voltage ranges are not currently available. Fortunately, the required voltage of the scaled prototype fell within the speed controller's allowable range.

The scaled fuselage volume was 351 in^3 housed within a prototype boasting a total length of 28 in. and height was 18 in. The wheel base of the main landing gear was 19 inches. Further details concerning the final prototype geometry can be found in the critical data summary.

Component	Weight (oz.)	Distance from C.G. (in.)	Moment about C.G. (in.-oz.)
Fuselage	1.73	1.26	2.18
Engine	5.96	2.6	15.5
Mounting Plate	1.13	1.93	2.18
Servomotor	.67	.87	.58
Servomotor	.67	-1.38	-.92
Servomotor	.67	-1.26	-.84
Servomotor	.67	1.42	.95
Landing Gear (Front)	3.17	2.83	8.97
Landing Gear (Tail)	2.87	-9.84	-28.24
Batteries	15.66	1.35	21.14
Main Rotor Blades	1.76	0.	0.
Receiver	1.41	.83	1.17
FET (Speed Controller)	1.59	3.03	4.82
Main Rotor Head Assembly	2.82	0.	0.
Tail Rotor Head Assembly and Tail Rotor Blades	.42	-21.26	-8.93
Tail Fin	.25	-20.87	-5.22
Rubber bands	.07	.79	.06
Glue	.25	0.	0.
Tail Boom	.38	-11.81	-4.49
Gearing for Tail Boom	.42	-21.26	-8.93
Rotor Shaft and Gearing for Main Rotor	8.29	0.	0.
Totals	**50.86**	-	**-.02**
Cargo	0.	-	0.

Table 6. Component Weight Breakdown

2) ***Flight Test Plan and Test Safety Considerations***

The Flight Test plan for the S.T.o.R.M. was quite complex. The first step was to balance the main rotors. This was done by inserting a screw through the holes used to affix the rotors to the hub, taping the rotors together so they wouldn't rotate and then balancing them about their intersection on a human finger. If the blades did not balance, then tape could be added near the tips to bring them into balance. Team Asylum found that the rotor blades for the S.T.o.R.M. prototype were perfectly balanced. Once the blades were determined to be in balance, they were attached to the hub, the batteries were connected and the battery pack was turned on.

The avionics checks were made with the engine turned off. This was due to the fact that the blades could cause serious bodily injury to any Team Asylum member and thus, they were not allowed to spin. The first check made was made on the tail rotor. When the yaw stick was moved to the left the tail rotors decreased in angle of attack, and when the yaw stick was moved to the right, the tail rotors increased in angle of attack. In the neutral yaw position the tail rotors were set at a zero angle of attack. The second avionics check was made on the swash plate control device. It was checked to make sure that the main rotor plane tilted in the direction desired. When the input was given to roll left or right, the swash plate was to move left and right. When the pitch control was deflected upward, the swash plate was to rotate forward, and when the pitch control was deflected downward, the swash plate was to rotate backward. The only problem encountered with the swash plate was that the pitch control was connected backward. This was easily solved by reversing the servo control on the radio transmitter. The final avionics check was the collective pitch of the blades. According to our pilot, Kane Kinyon, the rotor blades for the prototype should be set at approximately 3 degrees angle of attack at hover power. The collective pitch of the blades did change with changing throttle settings and the collective pitch control arms were shortened to place the blades at 3 degrees angle of attack at 80% power.

Once all of the controls were set, the blades had to be trimmed. This was accomplished by turning on the motor and allowing the blades to rotate. The throttle was increased to approximately

75% power and the rotors were observed directly from the side. If the swept disk of the rotors seemed to oscillate, it meant that one tip was higher than the other, which decreases the total power of the blades. The engine was turned off and the lower blade was adjusted by shortening the control arm and the test was repeated until the blades were trimmed. During this phase of testing, the tail rotor was removed and the prototype was held down by members of Team Asylum holding onto the end of the tail boom and each of the front wheels.

Full power-up tests served to elevate safety concerns to a major issue. Unlike airplanes where a 10-12 inch diameter propellor was spinning fast and had to be avoided, the S.T.o.R.M. prototype had a 35.5 inch diameter rotor and a 7 inch diameter tail rotor that had to be avoided. The tips of the main rotors were moving at approximately 105 feet/second and could cause serious harm. Whenever the prototype's rotors were spinning, no one other then members of team Asylum were allowed within 10 feet of the aircraft. Also, when trim tests were being performed, all team members were required to wear eye protection.

The flight test plan in the Loftus flight test center was to show the full range of maneuverability of the S.T.o.R.M. prototype. The flight test plan was to illustrate the vertical take-off and landing capability of the prototype, the zero turn radius capability, and the ability to fly at 25 feet/second.

3. Flight Test Results

The first flight test performed on the S.T.o.R.M. prototype on April 23, 1992 was a preliminary hovering test. The prototype's landing gear were affixed via nylon rope to three weighted stands, in an equilateral triangle configuration, approximately 5 feet apart. The pilot for this first test was Doug Murray. As Mr. Murray increased throttle, the prototype became airborne. Once airborne, the prototype flew straight backwards and struck one of the supports, damaging the tail rotor. It was determined that this accident occurred because the rear landing gear was too weak and the prototype sat on the ground in a tail down attitude. This meant that the main rotors were also tipped back, which directed the lift force backward and caused the prototype to fly in that direction upon breaking contact with the ground. For future tests, the landing gear was reinforced so that the prototype rested in a level configuration.

The second test performed on April 28, 1992 was a test to trim the blades at full throttle. For this test the pilot was Chad Kerlin. Team members Phil Holloway, Doug Murray, and Frank Augustyniewicz manually restrained the prototype by holding onto each front landing gear and the tail assembly, respectively. During this test, the prototype was taken to full throttle and back to idle. Once in idle, the radio transmitter was turned off. For some reason, this created a power surge that caused a loose horizontal stabilizer fin to swing upwards which extended it into the rotor disk, hitting the main rotors. This contact damaged the stabilizer, the main rotor blades and bent the tail boom. This clearly illustrated the force transmitted by the blades and enhanced the team's awareness of safety. After this flight test it was decided that any other powered flight attempts would be completed only with an experienced helicopter pilot.

The third flight test was conducted on April 30, 1992 in the basement of the Hessert Center with team pilot Kane Kinyon from the Michiana R/C Helicopter Club. Before this test, Mr. Kinyon attached a yaw correcting gyro to the prototype. This gyro changed the pitch of the tail rotor to compensate for any nose movement of the prototype. According to Mr. Kinyon, this type of stabilizing device is a must for any helicopter. This test was an untethered hovering test. Upon the first test, the prototype did not lift off from the ground. Mr. Kinyon then raised the pitch of the main rotor blades approximately 3 degrees by shortening the collective pitch control levers and the second attempt resulted in successful hovering approximately 3 feet off of the ground. During this test it was determined that the prototype was neutrally stable and that all control devices were functioning properly. During this test, Mr. Kinyon flew the prototype forward, backward, sideways, and pirouetted in both directions. This test ended after approximately two minutes when the battery charge was depleted.

From the results of the third flight test it was determined that since the power drain from the batteries for our prototype was so great, two battery packs would be needed for the final flight test at the Loftus center. One pack would be used for final trimming and one pack would be used for the actual flight demonstration. The final flight test took place in the Loftus flight test center on May 1, 1992. The pilot for this test was again Kane Kinyon. During the final flight test, the prototype successfully took off, hovered and flew throughout the range of its capabilities. The prototype flew at a maximum forward speed of approximately 15 feet/second and achieved a maximum altitude of approximately six feet. The edges of the prototype's flight envelope were not approached due to the pilot's fear of damaging the equipment.

4) ***Manufacturing and Cost Details***

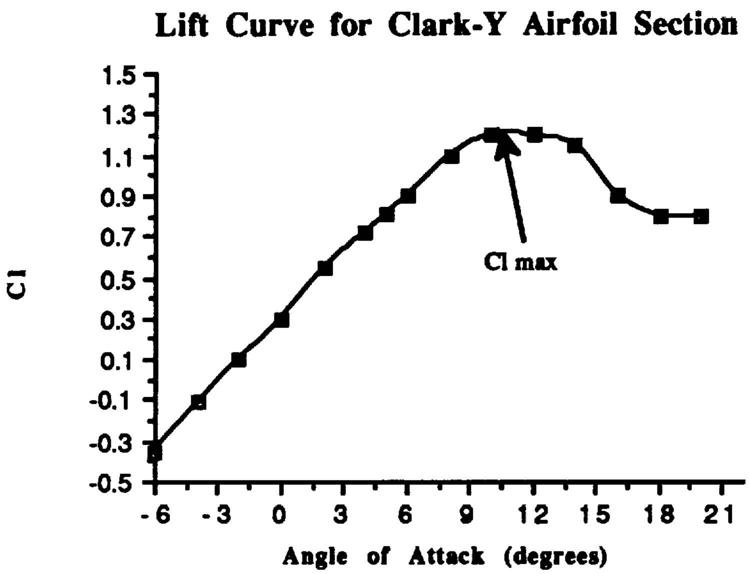
Component	Labor Hours	Cost ($)
Base unit assembly	22	311
Propulsion System	1	225
Avionics	1	339
Cargo Fuselage	10	26.40
Landing Gear	4	5.42
Mounting Plate	3	8.29
Testing	3	
Testing Repairs	3	48.83
Totals	47	963.94

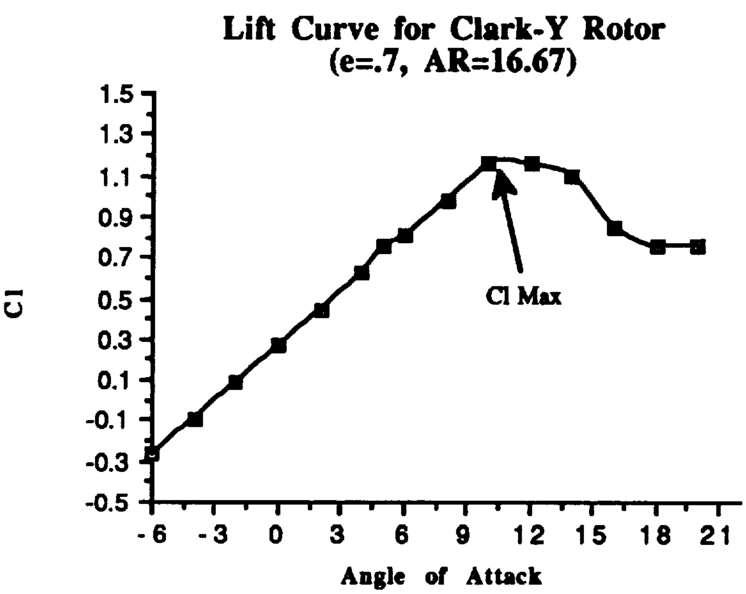
Table M-2

Construction of the technology demonstrator took the form of major mechanical systems installations rather than the sawing and glueing of other teams. This was expected, yet the time required to complete the assembly process was quite a bit under the 100 manhours initially projected. This is due to the fact that the base unit was somewhat more pre-assembled than was previously thought. This allowed the design team to spend more time actually learning about the various systems that make the helicopter operate and also allowed the group to teach others as to its use.

Appendix A

Airfoil and Rotor Data


Lift Curve for Clark-Y Airfoil Section
1.5
1.3
1.1
0.9
0.7
0.5
0.3
0.1
-0.1
-0.3
-0.5
Cl
Cl max
-6
-3
0
3
6
9
12
15
18
21
Angle of Attack (degrees)


Lift Curve for Clark-Y Rotor
(e=.7, AR=16.67)
1.5
1.3
1.1
0.9
0.7
0.5
0.3
0.1
-0.1
-0.3
-0.5
Cl
Cl Max
-6
-3
0
3
6
9
12
15
18
21
Angle of Attack

Appendix B

Landing Gear Sizing Program Code

land.f

```
      real l,w,d,p,e,i,r,a,b,h,pc,n,pn,pi,del,pcr,wgt,x
      real l1,l2,p1,p2,p3
      open(10,file='pstuff1')
      open(11,file='pchecks1')
      x=0.
      h=8.5
      w=6.
      p=3.
      n=1.
      e=2.8E7
      r=.0625
      pi=4.*atan(1.)
      i=pi*(r**4)/4.
      do 1 j=3,12
      p=real(j)
      write(10,*)'p=',p
      do 2 m=85,140
      h=.1*real(m)
      b=atan(w/h)
      a=atan(h/w)
      l1=sqrt(h*h+w*w+x*x)
c     l2=sqrt(h*h+w*w)
      pc=(p/n)*cos(b)
      pn=(p/n)*sin(b)
c     p2=pn/(1.+(2.*l2*l2*l2/(l1*l1*l1)))
      del=pn*(l1**3)/(3.*e*i)
      d=del*sin(b)
      pcr=pi*pi*e*i/(l1*l1)
      wgt=2.*(pi*(r**2)*l1*4.537037)
      write(11,*)h,p,d,pc,pcr,wgt
      write(10,*)h,d
2     continue
1     continue
      do 3 j=80,140
      h=.1*real(j)
      d=h-8.
3     write(10,*)h,d
      stop
      end
```

Appendix C
Construction Plans

i. MAJOR ASSEMBLIES
Exploded view of the main frame
Exploded view of the rotor head

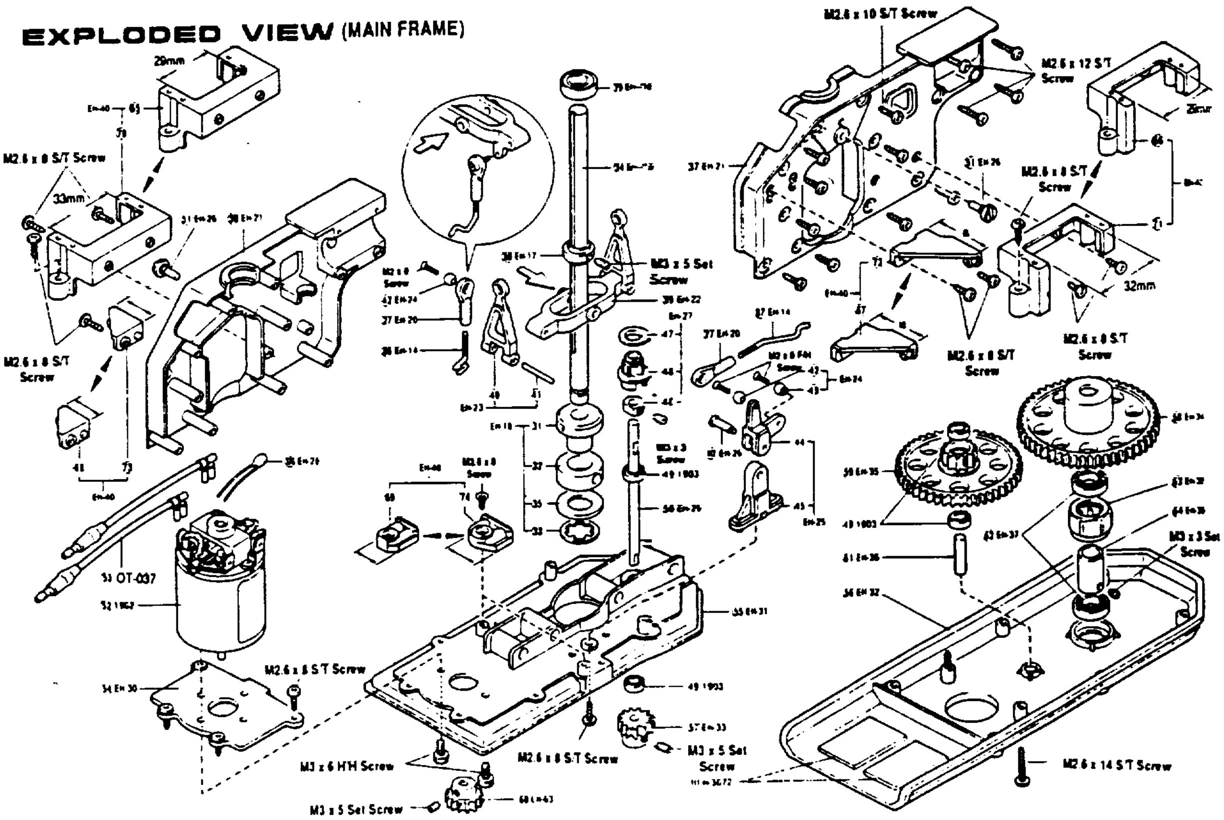
ii. PARTS LIST
Unedited list of *EP Concept* parts

APPENDIX C

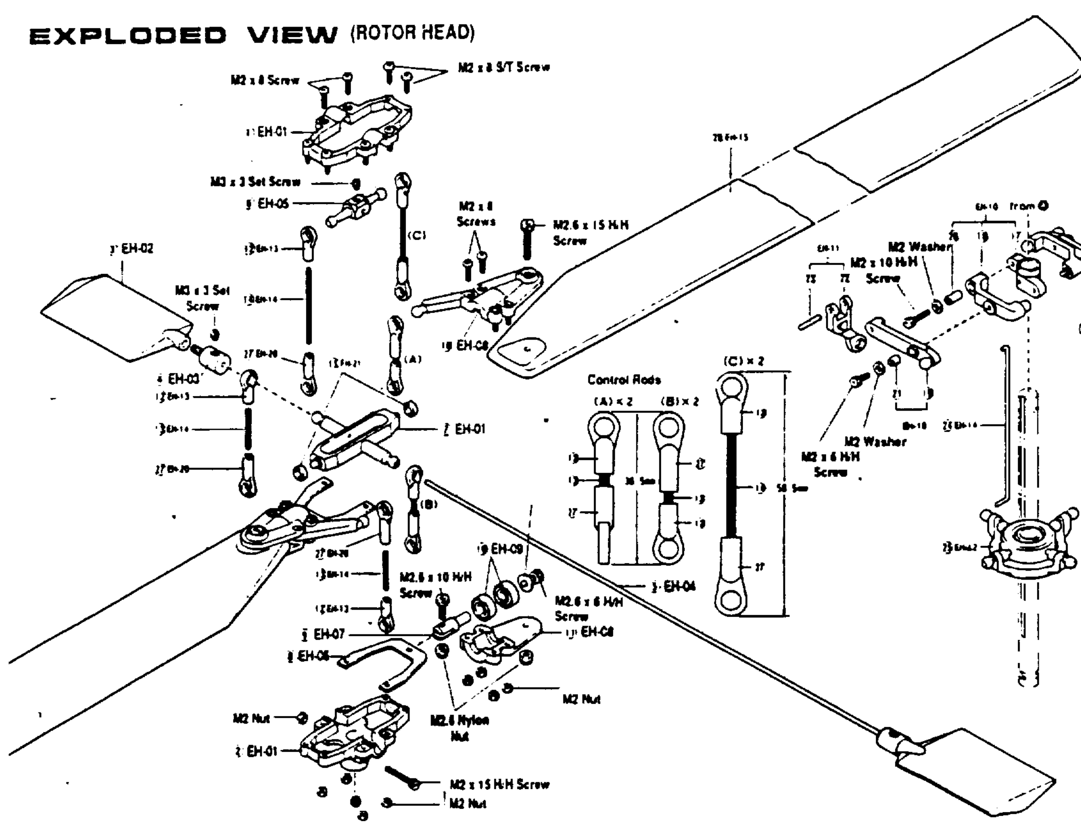
Part *i.*

Exploded Views of the Main Frame of the *EP Concept* and the Rotor Head of the *Concept* as Given in the Construction and Operation Documention.

EXPLODED VIEW (MAIN FRAME)



EXPLODED VIEW (ROTOR HEAD)



APPENDIX C

Part *ii.*

Complete Parts Listing as Given in the Construction and Operation Documention for the *EP Concept*

No.	Description	Qty.
1	rotor head (A)	1
2	rotor head (B)	1
3	stabilizer paddles	2
4	paddle collars	2
5	fly bar	1
6	hiller control lever	1
7	stabilizer seesaw	1
8	flapping hinge	2
9	feathering shaft	2
10	main rotor grip (A)	2
11	main rotor grip (B)	2
12	ball end (S)	6
13	M2 X 17 rod	4
14	M2 X 37 rod	2
15	3 X 6mm bearing	2
16	4 X 10mm bearing	4
17	mixing base	1
18	mixing lever	2
19	cyclic lever	2
20	lever bushing (A)	2
21	lever bushing (B)	2
22	cyclic lever link	2
23	cyclic pin (2 X 10mm)	2
24	pitch rod	1
25	swash plate	1
26	control rod (L)	1
27	ball end (L)	9
28	main blades	2
29	7 X 14mm bearing	1
30	pitch rod guide	1
31	pitch slider guide	1
32	pitch slider	1
33	10mm lock ring	1
34	main mast	1
35	pitch slider spacer	1
36	tie strap	4
37	frame (L)	1
38	frame (R)	1
39	fore/aft lever	1
40	fore/aft link	2
41	2 X 14 mm link pin	2
42	4.8mm ball (A)	2

43	4.8 ball (B)	1
44	pitch lever	1
45	pitch mount	1
46	locking collar	2
47	sprocket collar	2
48	front drive sprocket	1
49	4 X 8mm bearing	6
50	tail drive shaft	1
51	mast guide screws	2
52	motor	1
53	motor lead set	1
54	motor plate	1
55	chassis (A)	1
56	chassis (B)	1
57	counter gear	1
58	main gear	1
59	center gear	1
60	pinion gear	1
61	center gear shaft	1
62	7 X 14mm bearing	2
63	one-way bearing	1
64	one-way shaft	1
65	mount (1)	1
66	mount (2)	1
67	mount (3)	1
68	mount (4)	1
69	mount (5)	2
70	mount (6)	1
71	mount (7)	1
72	mount (8)	1
73	mount (9)	1
74	mount (10)	2
75	battery clips	2
76	front battery mount	1
77	skid brace (fr)	1
78	skid brace (rr)	1
79	rubber band	1
80	skid	2
81	body	1
82	canopy	1
83	grommet	2
84	body mount (rr)	1
85	decal sheet	1
86	control rod (bent)	1
87	pitch control rod	1
88	tail rotor grip (A)	2
89	tail rotor grip (B)	2
90	tail center hub	1
91	3 X 6mm bearings	4

92	tail rotor blades	2
93	ball end (S)	2
94	tail slider rail	1
95	slider ring	1
96	6 X 10mm bearings	1
97	slide bushing	1
98	tail gear box (L)	1
99	tail gear box (R)	1
100	tail drive sprocket	1
101	control lever	1
102	lever bushing	1
103	control rod guide	1
104	tail boom	1
105	belt	1
106	control rod tube	1
107	tail control rod	1
108	horizontal tail	1
109	fin mount	1
110	vertical fin	1
111	double sided tape	1
112	6mm lock ring	1
113	pitch lever pin	1
114	tail output shaft	1
115	adjustable link	1
116	E-clip (E-2)	1
117	tail screw	2
118	double sided tape strips	2
119	skid cap	4
120	capacitor	1

Appendix D

Economics Spreadsheet

	A	B	C	D	E	F	G	H	I
1	TOTAL COST OF PROTOTYPE			665	dollars				
2	PROTOTYPE CONSTRUCTION TIME			75	man-hours				
3	NUMBER OF SERVOS			4					
4	DISTANCE TRAVELLED PER DAY			160133	ft				
5	CRUISE VELOCITY			25	ft/s				
6	AVERAGE FLIGHT TIME			1.57	minutes				
7	TIME REQUIRED FOR BATTERY EXCHANGE =			0.5	minutes				
8	BATTERY SIZE			1400	mA-hr				
9	FUEL COSTS PER MILLIAMP HOUR =			10	dollars				
10	CURRENT DRAW AT CRUISE			13.81	amps				
11	MAXIMUM FLIGHT TIME POSSIBLE =			6.08254888	minutes				
12	MAXIMUM POSSIBLE RANGE			9123.82332	ft				
13									
14				%COST/FLT					
15	UPC =	341000	dollars		UNIT PRODUCTION COSTS				
16	UPCF =	568.33	dollars	13.43	UNIT PRODUCTION COST PER FLIGHT				
17	OCPF =	24.3301955	dollars	0.58	OPERATION COST PER FLIGHT FOR MAX RANGE				
18	MCPF =	25	dollars	0.59	MAINTENANCE COST PER FLIGHT				
19	FCPF =	3613.46	dollars	85.40	FUEL COST PER FLIGHT - MAX RANGE AND PAYLOAD WEIGHT				
20	NFLEET =	28			NUMBER OF AIRCRAFT IN FLEEET				
21	NCYC =	600			DESIGN FLIGHT CYCLES				
22	DESVOL =	1024	in^3		MAXIMUM PAYLOAD VOLUME				
23	DESWGT =	40	oz		MAXIMUM PAYLOAD WEIGHT				
24	DRANGE =	6500	ft		MAXIMUM DESIGN RANGE				
25	FLVPD =	34690	in^3		FLEET VOLUME PER DAY				
26	FFPD =	68			FLEET FLIGHTS PER DAY				
27									
28									
29	UNIT VOLUME COST PER FLIGHT			0.00045288	dollars/flt				
30	UNIT WEIGHT COST PER FLIGHT =			0.01159361	dollars/flt				
31	FLEET LIFE COST			71082853.6	dollars				
32	FLEET LIFE VOLUME			17203200	in^3				
33	FLEET LIFE WEIGHT			672000	oz				
34	FLEET COST PER OZ			105.78	dollars/oz				
35	FLEET COST PER IN^3			4.13	dollars/in^3				
36	FLEET LIFE TIME			247.06	days				
37	FLEET COST PER VOLUME MOVED			8.29	dollars/in^3	<= ACTUAL UTILIZATION!!			
38	DISTRIBUTIVE EFFICIENCY =			50.1809513	%				

11.84

Appendix E
References

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