Walt Lounsbery

FEASIBILITY STUDIES OF A LOW-COST, EFFICIENT MULTI-MISSION RPV WITH AN EMPHASIS TOWARD THE MILITARY ENVIRONMENT

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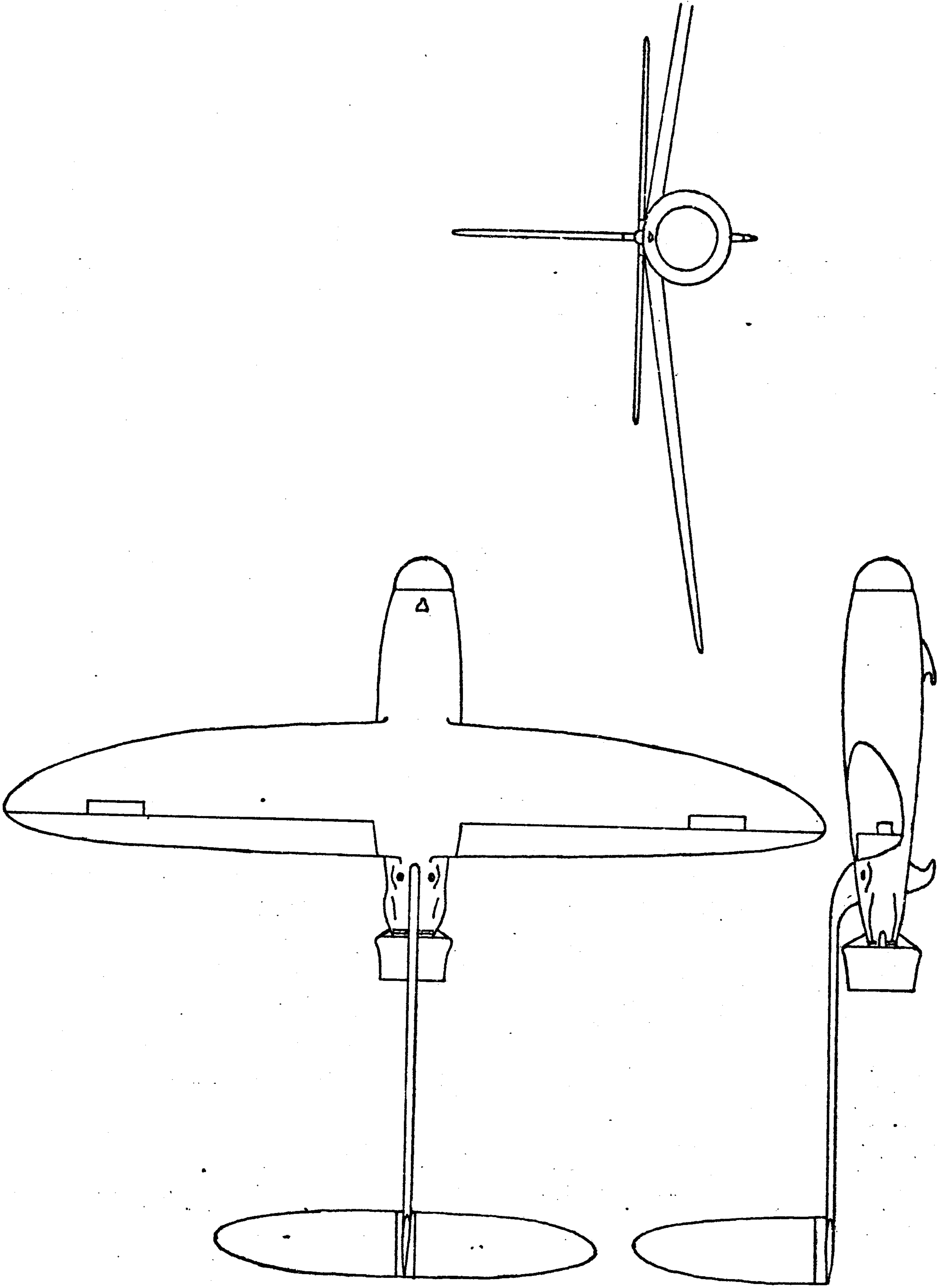
INTRODUCTION REMARKS



GENERAL REQUIREMENTS

Specification	Goal	Expected Performance
Max. Gross Weight	200 lbm	200 lbm $\pm$ 15%
Remote Control Range	50 miles @ 5000 ft. 100 miles @ 10000 ft.	limited by the line of sight
Basic Cost	$25,000	$30,000
Noise	20 db @ 5000 ft.	inaudible above 4500 ft.
Size	15'L x 15'W	12.75'L x 15'W
Cruise Speed	150 mph	125^+ mph
Endurance	20 hr. @ 80 mph	20^+ hr. @ 75 mph

Contrary to popular opinion, the idea of remotely piloted vehicles (RPVs) is not a new one. Back in 1916-1917 a British scientist, A.M. Low, converted a small single-seat airplane into an "aerial target". In the early stages of RPV development, these vehicles were mainly used as target drones and flying bombs. It was not until the 1960's that RPVs started to branch out into other roles where it might be too dangerous or unwise to send a manned craft, such as surveillance over enemy territory. Now RPVs fill a variety of roles: surveillance, reconnaissance, harassment, target designation and observation, navigational aid, and communication relay.

RPVs came into their own in the 1960's, when the military asked why should they send up expensive, easily tracked, manned aircraft into areas where a less expensive, small, long endurance RPV can do the same if not better job. At this time the only thing that is limiting RPVs is its electronics and propulsion. As they grow, RPVs will perform their present roles with increased efficiency plus expand into new roles.

This RPV is based on the idea of a quiet, long endurance, small, multi-mission craft that can be used close to the front with minimum launch facilities. The missions envisioned for this RPV are surveillance-reconnaissance, target designation-damage reporting, electronic counter-measures (ECM), harassment, and navigational and/or communication aid. The surveillance-reconnaissance mission will be close proximity (real time) television surveillance with the ability for high resolution photography for later review. The target designation-damage reporting mission is where the laser target designator aboard the craft is used to locate a target with the option of still photography to assess the damage. As an ECM, navigational or communications carrier, the system can carry a payload that will accomplish any of the above tasks. A harassment mission is possible whereby the RPV carries explosives to either harass enemy positions and/or disable radar antennas behind the lines.

Due to the nature of the above missions, this RPV has to fulfill certain aerodynamic expectations among which efficiency and maneuverability can be mentioned. The aerodynamic configuration of the vehicle has to be such to produce as low a drag as possible and maintain its characteristics over a relatively large range of weights (fully loaded to dry). It has to have acceptable glide and load factor capabilities, and yet be operable through a large range of velocities. The above requirements lead to the choice of elliptic planforms for the wing and the tail, a revolved symetric NACA airfoil section for the fuselage, and a boom mounted tail.

In this remotely piloted vehicle, there are several areas critical to the desired performance. High technology and a close scrutiny of trade-offs must be applied to every area, and the area of propulsion and power generation is no exception. The propulsive system is intimately affected by the performance requirements. Total propulsion weight for a given mission must be as low as possible, yet the twenty hour endurance goal means a substantial portion of vehicle weight can be fuel. This is unacceptable, since the vehicle is intended to carry a payload. The propulsion weight (engine, propulsor, and fuel) is greatly affected by propulsive efficiency, since the fuel load can be great. The high ratio of maximum speed to endurance speed means that the engine should have excess power capacity, perhaps with substantial weight penalty. Cooling must be provided for the engine, and this should be integrated with electronics cooling. Some of these restrictions apply to the propulsor. It must be light and efficient.

It is difficult to arrive at a quiet system. This goal dictates a reasonably quiet engine to reduce muffler size and weight. Conversely, the muffling must not rob power from the engine. Fortunately, large fuselage volume is provided by the configuration. It is easy to enclose the engine and large mufflers.

The engine must provide electrical power for the payloads over a wide range of missions. Adequate power must be generated for peak demand. A storage battery system is necessary for emergency power if the engine fails.

All the high technology associated with the missions, structure, and propulsion will have an associated high cost. Even though the costs incurred in many areas are high, the cost effectiveness of this system compares favorably with other alternatives.

The RPV, in order to carry out its missions, must be light, strong, inexpensive, easily maintained, allow for easy disassembly and quick re-assembly. The structure was designed around these requirements plus the needs of a pusher-prop system, high visibility for its cameras and the ability to handle large take-off and landing loads as well as maneuvering flight. The structure chosen was a Kevlar fabric-epoxy skin over aluminum ribs and spars. This configuration provides a light base from which all the missions may be performed.

It was possible through this configuration to locate the center of gravity at a favorable location and with an excellent range of travel to provide stability throughout the flight envelope. The use of the laminate skin over metal reinforcement enables contouring required of the airfoil sections of the wing, fuselage and tail. Additionally, the reduced use of metal lowers the radar cross-section of the RPV dramatically. This also allows the communications antenna to be laid up in the skin of the fuselage thus affording protection from the rigors of flight and ground handling. All this combines to form a system that is reliable, hard to detect, inexpensive but with high performance.

CHAPTER II

MISSIONS

Chapter II

2.1. MISSIONS

2.1.1 Why a RPV?

Originally, the military saw a remotely piloted vehicle (RPV) as a target drone or as a flying bomb. Today, the military is beginning to see the real advantages of RPV's over manned aircraft. The RPV, unlike manned planes, is not bothered by fatigue due to long flights, and doesn't mind being shot at (too much). These characteristics are important especially in target-spotting aircraft. Now in high ground fire areas, the military sends in a mini-RPV for artillery adjustment and for real-time surveillance. An important by-product of using an RPV for real-time surveillance, is that information can reach to front line areas faster then with a manned aircraft. The underlining reason for this, is that a RPV can be used on a lower operational level (closer to front line command) than manned vehicles, cutting down the time lag caused by chain-of-command communication. Quick response time to the front lines brought with it the mission of target designating. With target deisgnating, troops at the front can have a target located and laser designated before a manned aircraft with laser guided weapon reaches the scene. Again, this is important in high ground fire areas, for the manned aircraft does not leave itself exposed for any great length of time and yet weapon accuracy is improved. These are not the only areas where a RPV can be used instead of, or along with manned aircraft. In the areas of electronic counter measures (ECM), a RPV can do a comparable job as a manned plane, but a RPV can do it with a lower cost because it does not have to waste the time to a man or a very expensive airplane. Other tasks that make similar type claims are communications relays and navigation aids.

2.1.2 Mission Descriptions

The system developed here gives a RPV that can handle all of the aforementioned missions with the added missions of harassment and reconnaissance. The actual RPV was designed for six different missions to cover all of the above tasks. They are 1) surveillance-reconnaissance, 2) target designator - damage reporting, 3) ECM, 4) navigation aid, 5) communication aid, and 6) harassment. Surveillance - reconnaissance mission is designed to provide two different options. The first option is that it will operate as a quiet, close-proximity, real-time, television surveillance craft of long endurance with ability to take high resolution photographs of important or questionable items, for later review. The second variation by pre-programming the autopilot is as a pure long range reconnaissance craft taking photographs behind enemy lines. The target designator-damage reporting configuration will be able to locate and mark a target for an air strike but also can be used for artillery adjustment. An added feature that makes this mission even more useful, is its ability after the strike to take still photographs so the damage can be correctly estimated. ECM covers a wide range of submissions from ground communication jamming to sensor disruption (radar and infrared). The RPV was designed to handle general range of ECM equipment along with an extended duration time to enhance confusion among the enemy before, during, or after an attack. Communications mission RPV will carry communication relay equipment to extend the range of surface-to-surface or surface-to-air communication. The navigation form of the RPV will be able to carry a navigational beacon to aid aircraft in finding targets or to find their way back from a strike. Lastly, the harassment will be able, at first estimation, to carry missiles and/or bombs to damage enemy's equipment or moral.

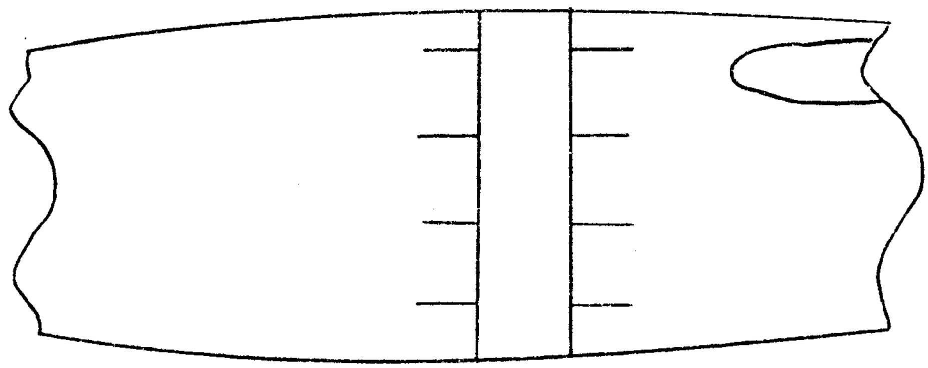
2.1.3 Design Configurations

This RPV will handle all of the previous missions with low down time and with close-to-the-front operation capability for good communications with the front lines. To best perform all these functions, the following items were taken into consideration: 1) launcher and recovery system must operate from as small a clearing as possible, 2) all systems must be mobile so the system can relocate quickly to where it is needed, 3) the RPV must have a system for easy transportation and quick replacement for damaged units, and 4) RPV must have high manueverability to avoid ground fire. The landing and take-off requirement was fulfilled by launching the RPV using a compressed air catapult with recovery done by a net and arresting hooks, (both systems will be gone over in later sections). These systems are very well suited for mobility, for the launcher can fit on a M36 truck and the net can be fitted and stored in a trailer. Ground transportation for the RPV was accomplished by designing the craft to break down into seven major components: 1) payload bay, 2) fuselage, 3) wing, 4) tail boom, 5) vertical tail, and 6) and 7) the two horizontal stabilizers. This breakdown also keeps the down time minimal by allowing malfunctioning components to be quickly replaced for repair at a later date. Missions are likewise augmented by this breakdown, for a change in mission is as simple and quick as changing payload bays.

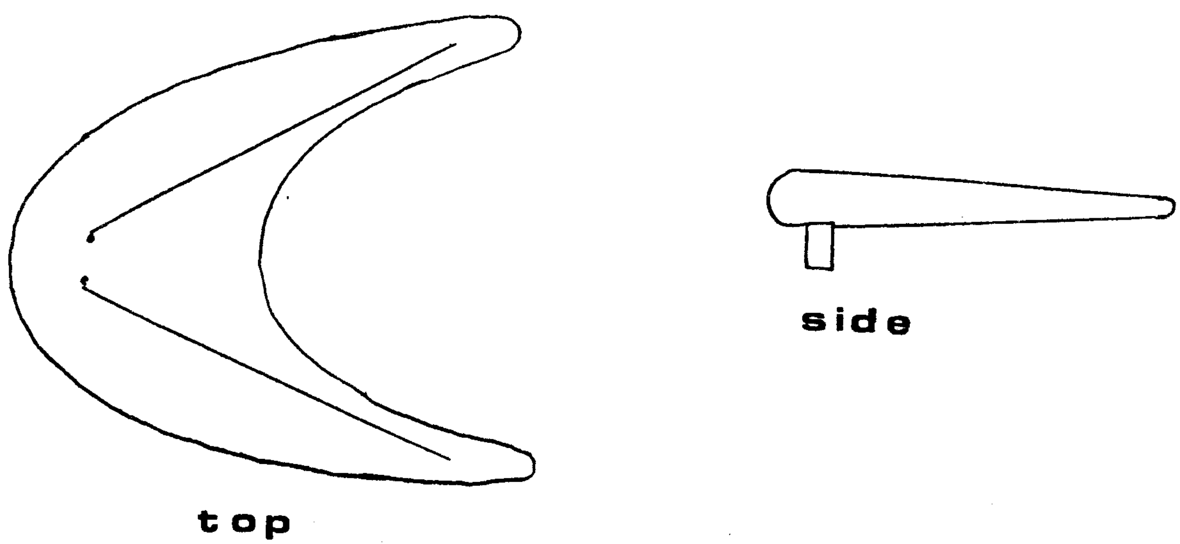
2.1.4 Payloads

Mission payload limitations are greatly dependent on two things, 1) payload bay and 2) guidance and telemetry. Guidance and telemetry has four major demands. They must 1) be able to supply flight information, 2) provide short and long range control capabilities, 3) provide bandwidth capable of transmitting television signals, and 4) keep antennas to a reasonable size. The final system chosen to cover the first two demands was a Melpar Modular Autopilot. The reasons behind the decision were that the unit is small in size, it can handle all flight monitoring, and it can be either have a pre-programmed course or be controlled from the ground. The telemetry system is to be a FM transceiver with two antennas, one dipole mounted on the wing and one band antenna laid-up in the fuselage just forward of the wing (Figure 2.1-1). Using FM frequencies, though, limits the television reception and radio control of the RPV to line of sight. This limitation becomes acceptable when other frequency ranges are looked at. Higher frequency requires a directional antenna which would add weight to the vehicle in form of an antenna-orientation system. For a lower frequency range, the antenna would have to be very large to handle the wavelength necessary.

Mission limitations, with respect to the payload bay, are due to the individual limitations of the components in each mission bay. The components of the bay are also what gives each mission its unique characteristics. For surveillance-reconnaissance, the bay contains one gimbaled Honeywell television camera or Honeywell forward-looking infrared sensor (FLIR) and three TA-8M2 aerial cameras. The television camera and FLIR both weigh, gimbaled, about



a. Band Antenna



b. Dipole Antenna

fig. 2.1-1

15 lbs and require 31 watts of 28 vdc. The TA-8M2 cameras each weigh 13.2 lbs loaded with 100 feet of film and require less than 150 watts. Each camera has independent and automatic exposure control along with variable film rate. The target designator-damage recording operation requires one Honeywell laser target designator fitted with FLIR or television camera and one TA-8M2 aerial camera. The laser target designator is about 42 lb in weight and has a targeting range of more than 2 miles. The actual laser-designator once targeted will stay on target by use of a video-tracker. The ECM, navigation, and communications missions all contain one Honeywell television camera for control purposes but all of the other equipment is either classified, as in the case of ECM, or has to be designed and built for a perticular task which makes further definition impossible. Yet, the RPV can handle a wide range of payloads with its six cubic foot capacity and is able to carry 110 lbs including fuel, so there is no problem anticipated for these missions. The possible harassment vehicle, based only on weight, will carry one television camera and an assortment of missiles (a suggested list of missiles is found in Table 2.1-1) and bombs mounted to four hard points on the wing.

2.1.5 Further Possibilities

Yet, these are not the only possible configurations and uses of the RPV. The civilian sector of the population also has uses for this vehicle. With its surveillance capability, it can be used to look for forest fires, survey traffic, follow wild animal herd migrations, hunting for disaster victims, and many other similar type jobs. The RPV also has application as a communication relay

Name	Weight (lb)	Length (in)	Diameter (in)	Use
LAAW	5.5	35	-	Ground Attack
Redeye	18	47.5	2.7	Anti-Aircraft
Stinger	29.5	60	2.7	Anti-Aircraft
with IFF*	31.5	60	2.7	Anti-Aircraft
Dragon	32.2	48.4	-	Anti-Tank

Suggested Missiles

* Identification Friend or Foe

Table 2.1-1

for mountain and disaster rescue teams. Even these applications have not scratched the surface of the possibilities of the craft. Because of its inherent payload capacity and, given sufficient power and manueverability, the only limit on applications is the users imagination.

2.1.6 Comparison with Existing Systems

The flexibility of this RPV is what sets it apart from other RPV's. At present time, no other system can claim an endurance of 20 + hrs and a range of over 1500 miles.

Equally important, no RPV has the capability to handle such a wide range of tasks and have ability to upgrade electronics without any or little change in the air frame. In light of these facts, this system is felt to be superior to anything presently available.

2.2. TAKE-OFF

The major considerations for the take-off system were that it must be used in a minimal clearing and that it must be mobile. Consideration was given to three different systems, 1) standard runway take-off, 2) rocket assisted, and 3) compressed air catapult. Standard runway take-off was immediately excluded because it required a long clearing and the equipment needed to perform this take-off would add excessive weight and drag to the RPV. The rocket-assisted take-off also had its drawbacks, the largest being the shift in center of gravity (c.g.) location. The propellant necessary for take-off on a 20 ft ramp (so that it will fit on a truck) causes such a c.g. position shift that it makes the control surfaces larger than what is required for normal operations. The final selection was the compressed air launcher used by the Lockheed Aquila. This system was chosen for several reasons: it's an already proven system, it provides the needed amount of acceleration, and it is mobile as it fits on a M36 Army truck. From ref. 1 and simple force calculations, shown in appendix A 2.2-1 it is shown that the Aquila launcher produces 876 lbs of thrust. The force will accelerate the 230 lb RPV to a final velocity of 70 ft/sec in 20 ft. This is well above the stall speed of 60 ft/sec with flaps and nearly the 72 ft/sec stall speed without flaps. This indicates that the Aquila launching system will work for this RPV system with only minor modifications, and those are mainly to the launcher-RPV interface.

2.3. LANDING

As with take-off, the recovery system must be mobile and require a minimal clearing. Four systems were reviewed: normal aircraft, inflatable foam skin, parachute, and net. Again the normal aircraft system was rejected due to weight and clearing requirements. The inflatable foam skin system was also dismissed due to weight and system complexity. The parachute system also had a weight and complexity problem, but added to that was a degree of uncertainty on the actual landing point. The net system was chosen because it can be performed in a limited area and that it adds little weight to the RPV. The net system consists of a 25 ft by 60 ft horizontal net with hydraulic energy dissipators and two arresting hooks placed on the RPV. The hooks are placed on the RPV in such a manner that the hook mounted on the aft section of the fuselage dissipates the energy into the nets. The front hook's purpose is keep the RPV from being thrown from the net on the backlash. To check the feasibility of this system, the Aquila net recovery system was inspected. It was shown in the calculations in appendix A 2.2-1, that the Aquila system dissipates 21773 ft-lbs of kinetic energy. It is also shown that assuming a landing speed of 70 ft/sec at a weight of 230 lbs, that this RPV has to dissipate 17500 ft-lbs of kinetic energy. This indicated that the net recovery system will work for this RPV, but final certainty can only be achieved by full scale testing.

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CHAPTER III

AERODYNAMICS

.1. AERODYNAMICS

3.1.1 Introduction

In general, the missions described in the foregone chapter required a special aerodynamic design being operable through wide ranges of angles of attack and weight with acceptable efficiencies. This required a unique approach to the problem. In the following sections, brief discussions and justifications of the existing system are presented.

3.1.2 Fuselage

The important factors in the design of the fuselage were drag, visibility, internal volume, and simplicity of manufacturing. Table 3.1-1 presents some possible fuselage configurations which were considered in this study, along with advantages and disadvantages associated with each. It was obvious that none of these shapes could be the optimum design. Therefore, the next approach taken was the use of a hybrid structure which could contain the advantages of the considered systems with the least possible defficiencies. To accomplish this task, the fuselage was divided into three main sections: the nose portion which had to accommodate a television camera and possibly a laser target designator, the mid-body which had to have enough internal volume for the electronic packages and would be long and wide enough to fit a wing of reasonable root chord size, and finally the aft section which had to be suitable for containing a power plant of the required size. On the basis of these requirements, a revolved NACA 0018 airfoil was chosen as the basic configuration. This choice would provide one of the lowest drag coefficients for the internal volume available through a wide range of angles of

Shape	Advantages	Disadvantages
Conventional Fuselage	I. Low Drag II. Reasonable Volume Efficiency III. Readily Available Aerodynamic Characteristics	I. Extra Drag due to Canopy II. Extra Skin Friction Drag III. Low Visibility IV. Complex Manufacturing V. High Side Radar Profile
Sphere	I. Maximum Volume Efficiency II. Constant Aerodynamic Characteristics III. Ease of Manufacturing IV. Low Radar Profile V. High Visibility	I. High Drag due to Low Reynold Number II. Low Manueverability in Combination with Wing III. Large Frontal Area
Ellipsoid	I. Low Radar Profile II. High Volume Efficiency III. Ease of Manufacturing IV. Predictable Aerodynamic Characteristics	I. Too Wide at the Tail Causing Excessive Drag II. High Side Radar Profile III. High Drag of Angles of Attack > 0
Flying Wing	I. Predictable Aerodynamic Characteristics II. Low Drag III. Low Radar Profile	I. Complex Manufacturing II. Extremely Low Manueverability III. Low Volume Efficiency

Volume Efficiency = (Total Volume)/(Total Surface Area)

Table 3.1-1

attack. It would also be relatively simple to manufacture. Then the nose portion was modified to accommodate the smallest surveillance and target designating package available, along with maximum visibility. The modified shape was chosen to be a total hemisphere made of an optical quality transparent material to allow the least distortion in the laser beam and the video communication. The aft portion of the basic shape was also slightly modified to fit the power plant requirements. The shortest length of the basic airfoil shape was determined to be 100 inches. This length would allow an internal volume in the mid section large enough to fit the payload, a width near the aft portion to accommodate the power plant, and yet it would be short enough not to locate the center of gravity too far from the nose or generate a high radar profile. This choice was also made on the basis of the fact that the experimental data had shown a height to length ratio of 15% to 20% would give near optimum drag characteristics. Figure 3.1-1 shows the final fuselage configuration.

3.1.3 Wing Planform

The basic requirements for the wing planform were ease of manufacturing along with low induced drag. The latter requirement depicted use of as high an aspect ratio as possible along with an elliptic lift distribution.

To achieve a high aspect ratio, the wing span had to be chosen as long as possible with the smallest wing area. Other determining factors for wing area were the stall speed and the loiter speed. To determine the best wing area for the system, a family of curves were generated which related the wing area, the lift coefficient,

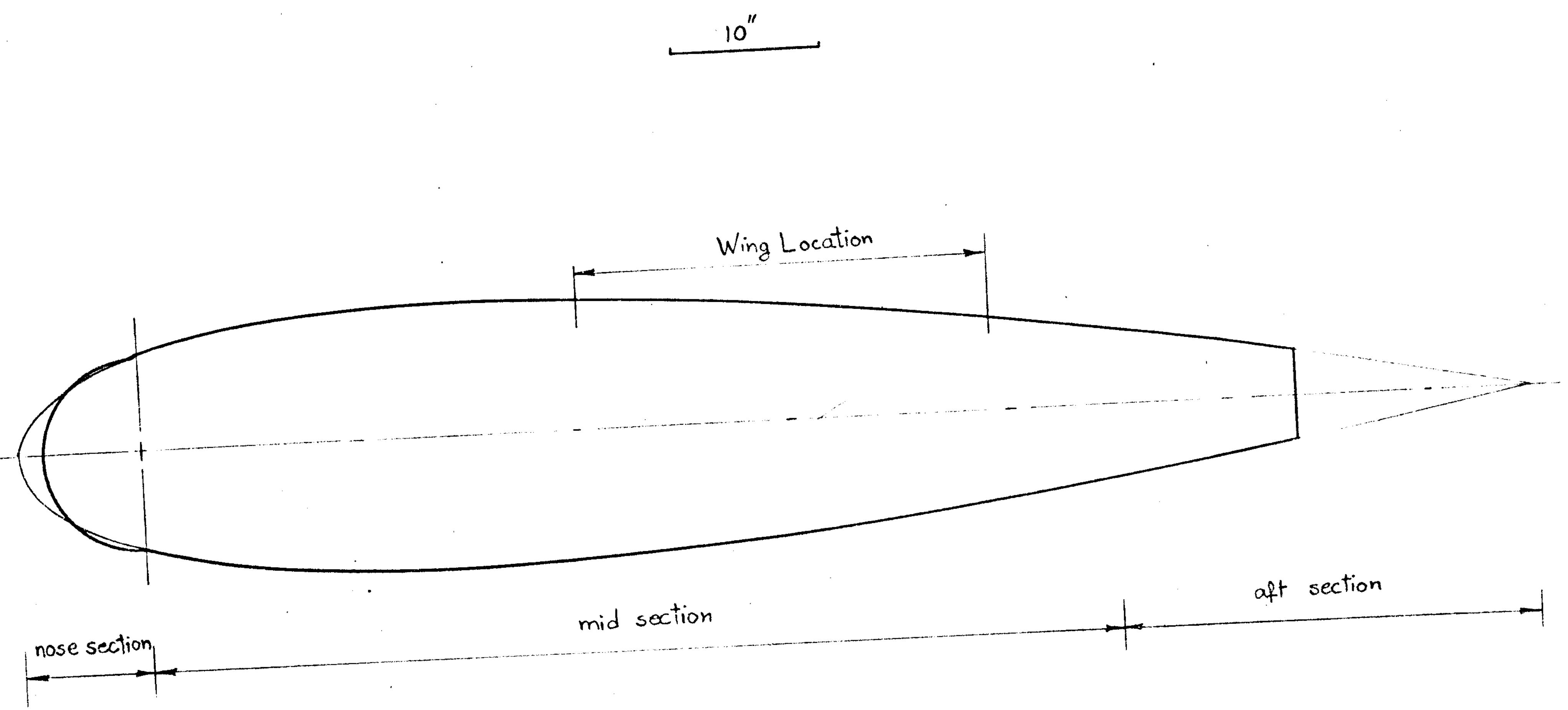
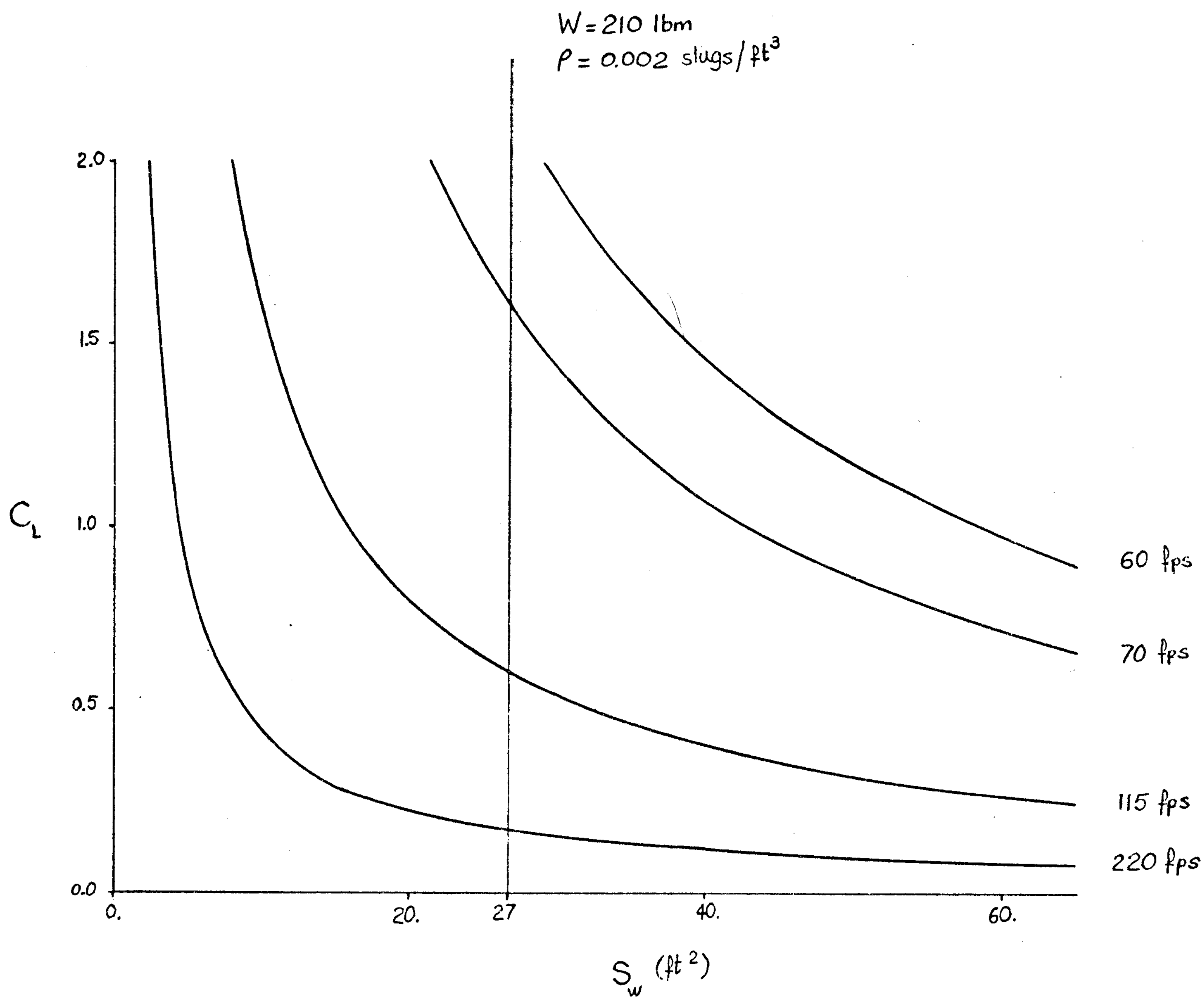


Fig 3.1-1

and the flight speed for a gross weight of 210 lb. at sea level. These curves are shown in Figure 3.1-2. As is shown in this figure, without using exotic high lift devices to achieve a reasonably small stall speed, the wing area had to be around 27 ft^2. At this point, to stay within the designated wing span size and yet to achieve the highest possible aspect ratio, the wing span was set at 15 ft. This would produce an aspect ratio of greater than 8. Later after the choice of the wing airfoil was made, the wing area was further reduced to 25.9 ft^2 to increase the aerodynamic efficiency of the vehicle. This fact will be elaborated further in the next section.

The elliptic lift distribution could be achieved through use of an elliptic planform or through span-wise twist of the wing. The latter idea was rejected due to the fact that a twisted wing could be optimized for only one flight mode. However the vehicle's missions required aerodynamic efficiency from the wing through a wide range of angles of attack. A twisted planform would also provide a smoother stall which could be achieved through the right choice of the airfoil for a straight wing. Also, since the skin was going to be molded out of a fiber reinforced epoxy, the production cost of an elliptic planform would be comparable to that of a straight edged planform with twist. After all, an elliptic planform would have lower profile drag, produce elliptic lift distribution at all angles of attack, and also be operable through a greater range of angles of attack without partial stall. Therefore, it was determined that an elliptic planform would be the most suitable and efficient configuration for this vehicle.



W = 210 lbm
ρ = 0.002 slugs/ft³
C_L
2.0
1.5
1.0
0.5
0.0
0.
20.
27
40.
60.
S_w (ft²)
60 fps
70 fps
115 fps
220 fps

Fig 3.1-2

To maintain the elliptic lift distribution in conjunction with full span flaps, a mathematical scheme was drawn which is presented in Appendix 3.1.

It must also be mentioned that the sweep of the planform was not a critical parameter for this vehicle due to the relatively low maximum operational speeds.

3.1.4 Wing Airfoil

To find a suitable airfoil for this system, several shapes were studied. The basic requirements set for the wing airfoil were low profile drag, high maximum lift coefficient, smooth stall, high values of $C_\ell^{3/2}/C_d$ and C_ℓ/C_d at low lift coefficients, and low values of

$$\left| \frac{d}{d\alpha} \left(\frac{C_\ell^{3/2}}{C_d} \right) \right|$$

beyond the peak point.

Since the ease of maintainability and ruggedness were two of the basic design requirements, the smoothness of the surfaces in actual operational conditions could not be guaranteed. Therefore all the airfoil data were considered for the worst possible case which would be standard surface roughness. This would guarantee that the system would always operate more efficiently than the design case.

A high maximum lift coefficient could imply use of simpler high lift devices during landing and take-off or even if high enough could totally eliminate the need for such devices. Considering that simplicity of the system which would result in low production costs and high reliability was one of the basic design goals, this factor would be one of the more critical ones in making the final decision.

Having an untwisted wing and controlling the vehicle through radio and television systems, depicted use of an airfoil with smooth and predictable stall characteristics.

High values of C_ℓ/C_d and $C_\ell^{3/2}/C_d$ at low lift coefficients would imply relatively higher values of C_L/C_D and $C_L^{3/2}/C_D$, which directly affected range and endurance of the vehicle and consequently the fuel efficiency of the system.

Finally the last requirement which was low values of

$$\left| \frac{\partial}{\partial\alpha} \left(C_\ell^{3/2}/C_d \right) \right|$$

would allow a greater range of C_ℓ through which the vehicle could operate efficiently. Figures 3.1-3 through 3.1-8 show the aerodynamic characteristics of six airfoils which were considered the best.

A more detailed analysis of these airfoils revealed that GA(W)-2 would be the best suitable section. Unfortunately due to lack of any available data in regard to the behavior of this section in conjunction with any simple flap system, the choice was made on NACA 4412 which presented the second best characteristics. This choice was made through loss of part of the maximum lift coefficient. However other characteristics of this section matched those of GA(W)-2 very closely.

At this point a program was set up which would accept the airfoil characteristics data and the planform geometry and would calculate the finite wing characteristics. A listing of this program along with the description of it are presented in Appendix A.3.2. Figures 3.1-9 and 3.1-10 show plots of these characteristics.

NACA 65-210

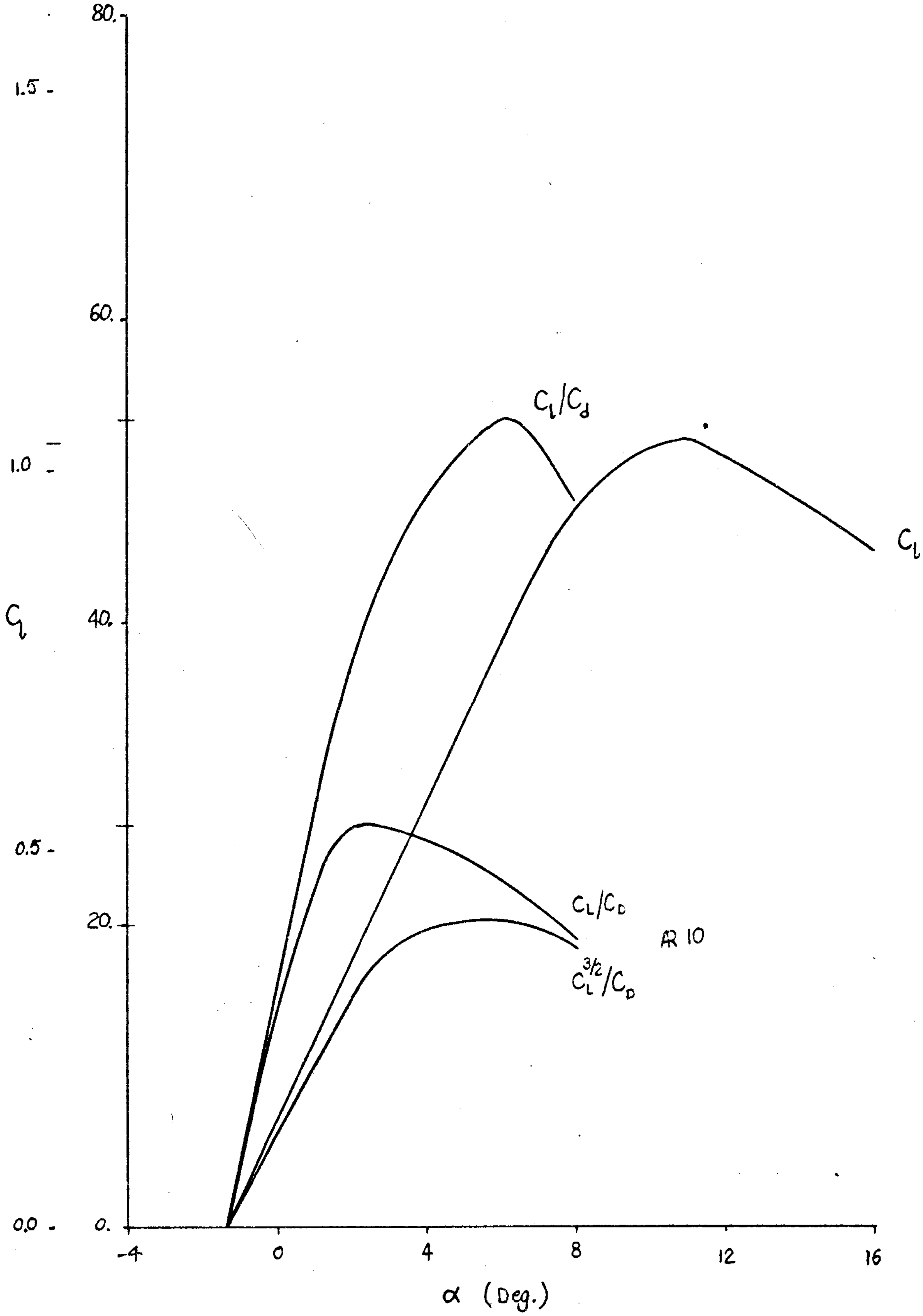


Fig 3.1-3

NACA 23012

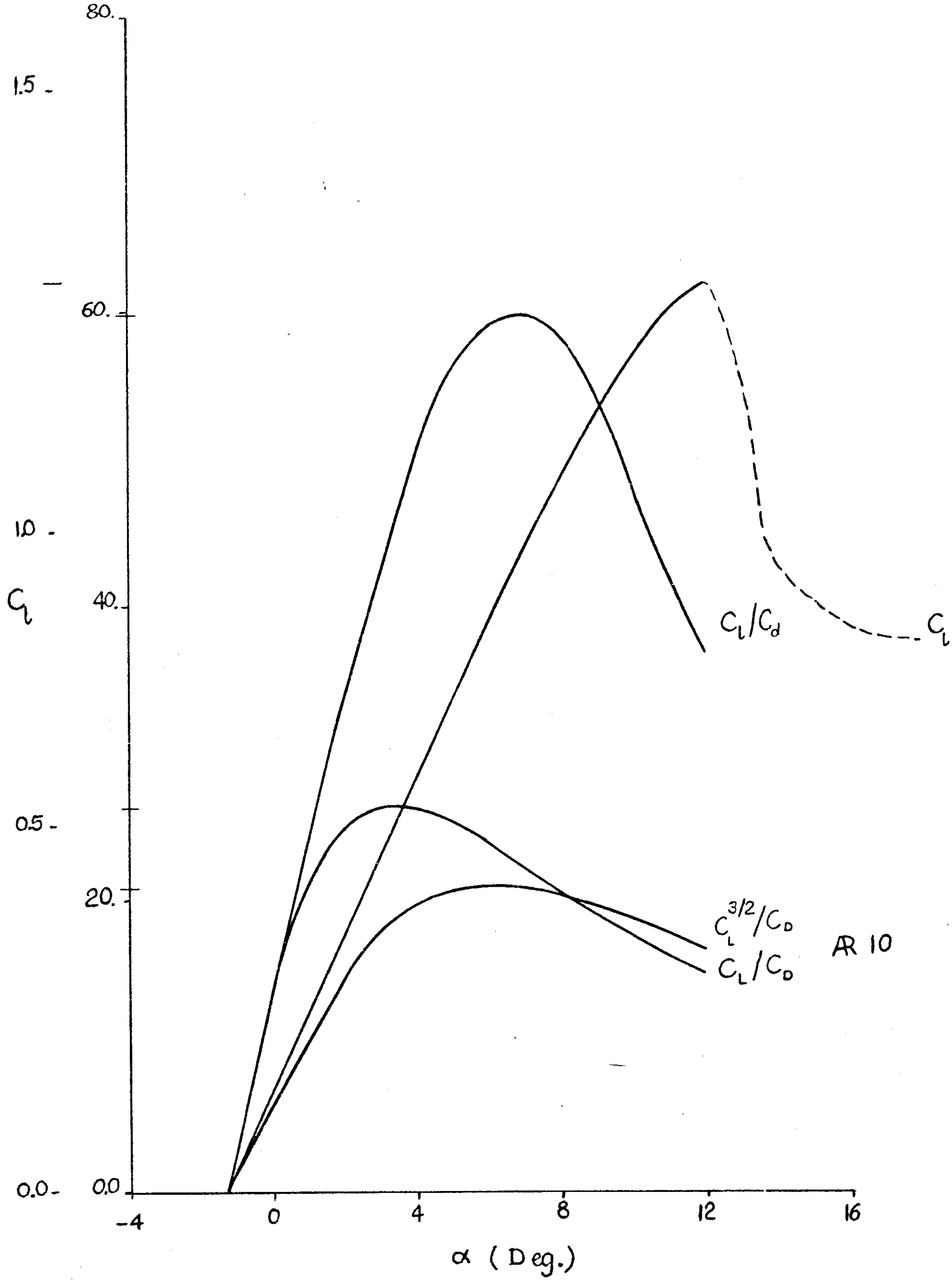


Fig 3.1-4

NACA 65_3-418

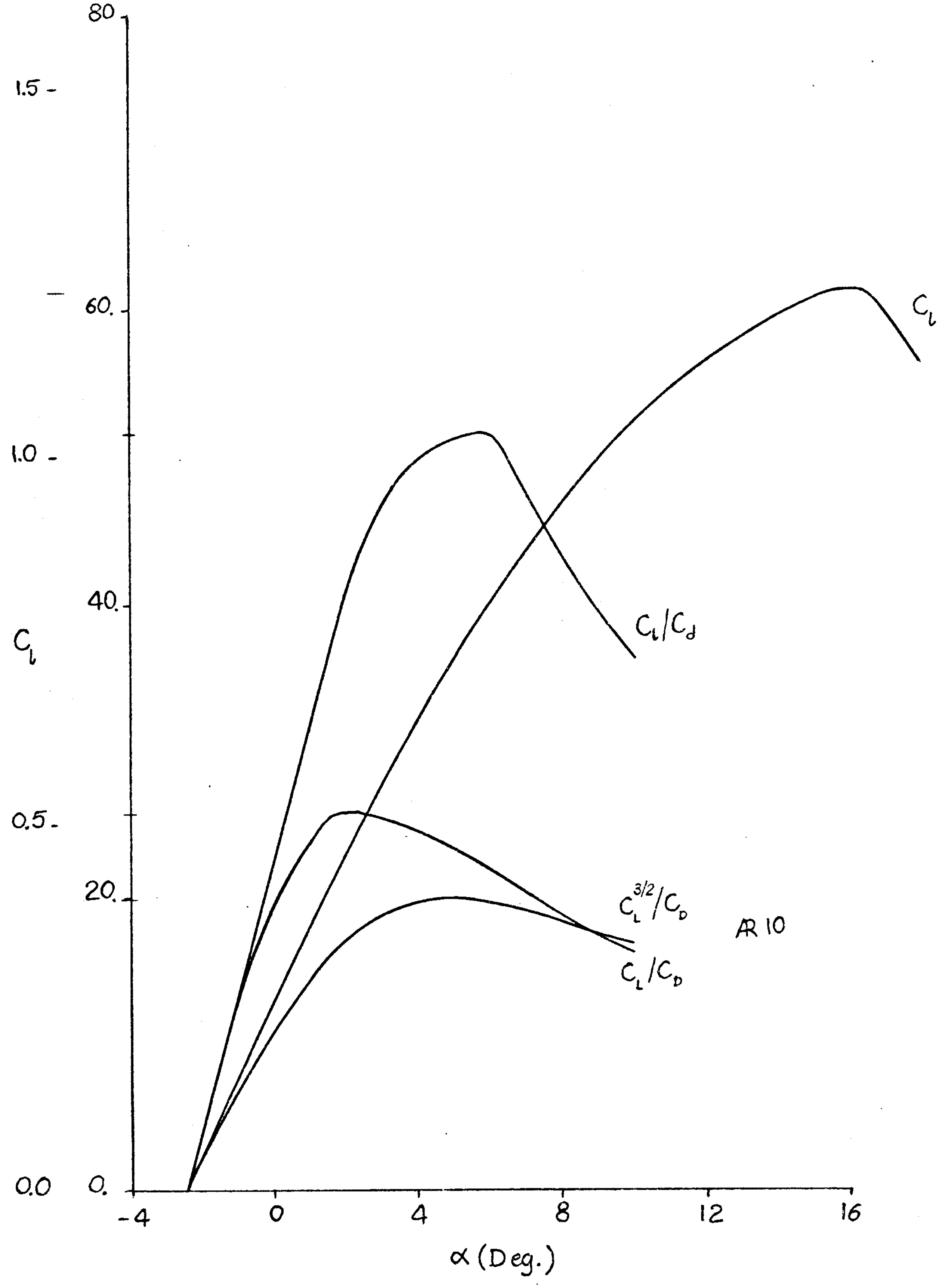


Fig. 3.1-5

NACA 2412

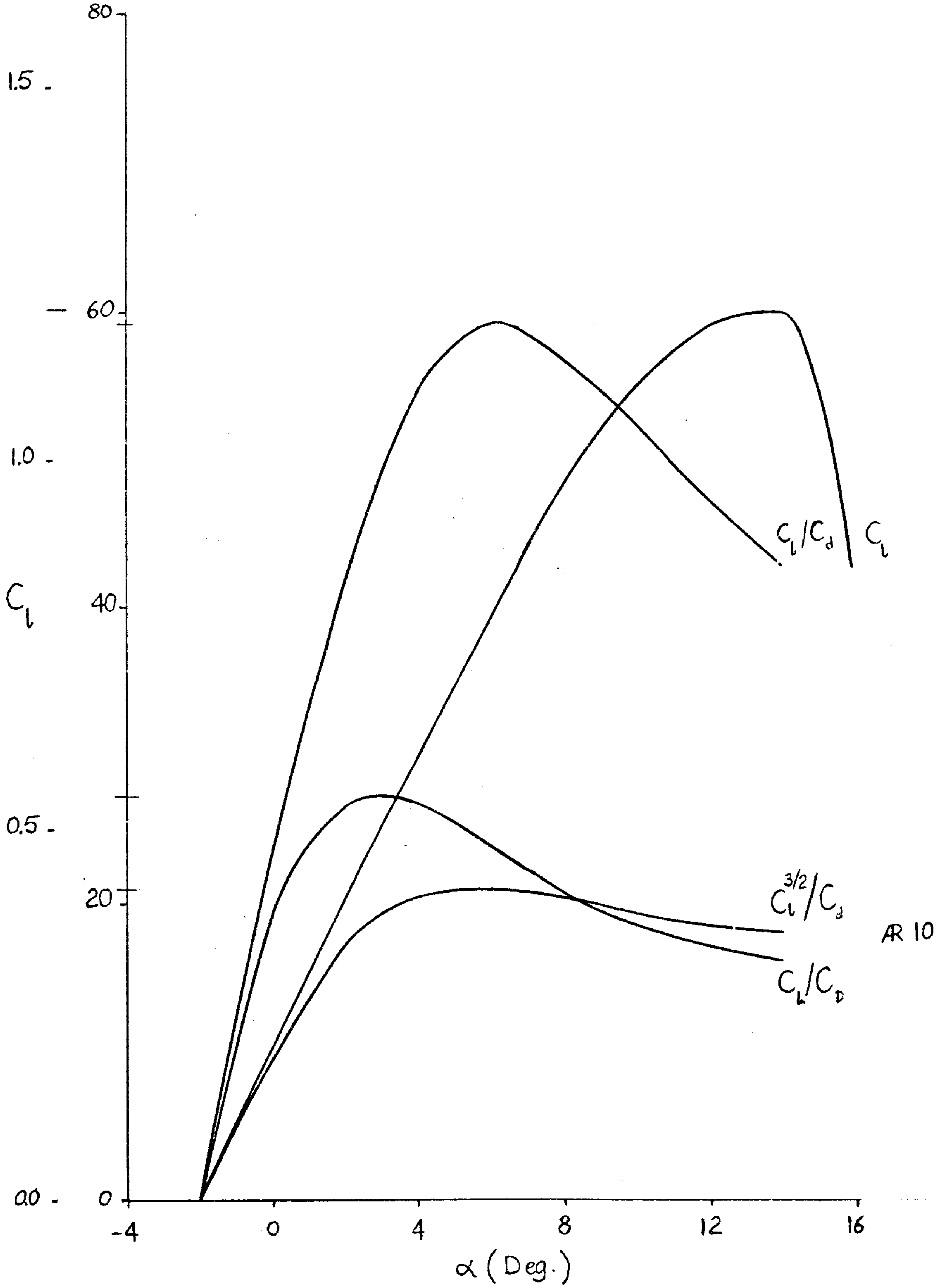


Fig 3.1-6

NACA 4412

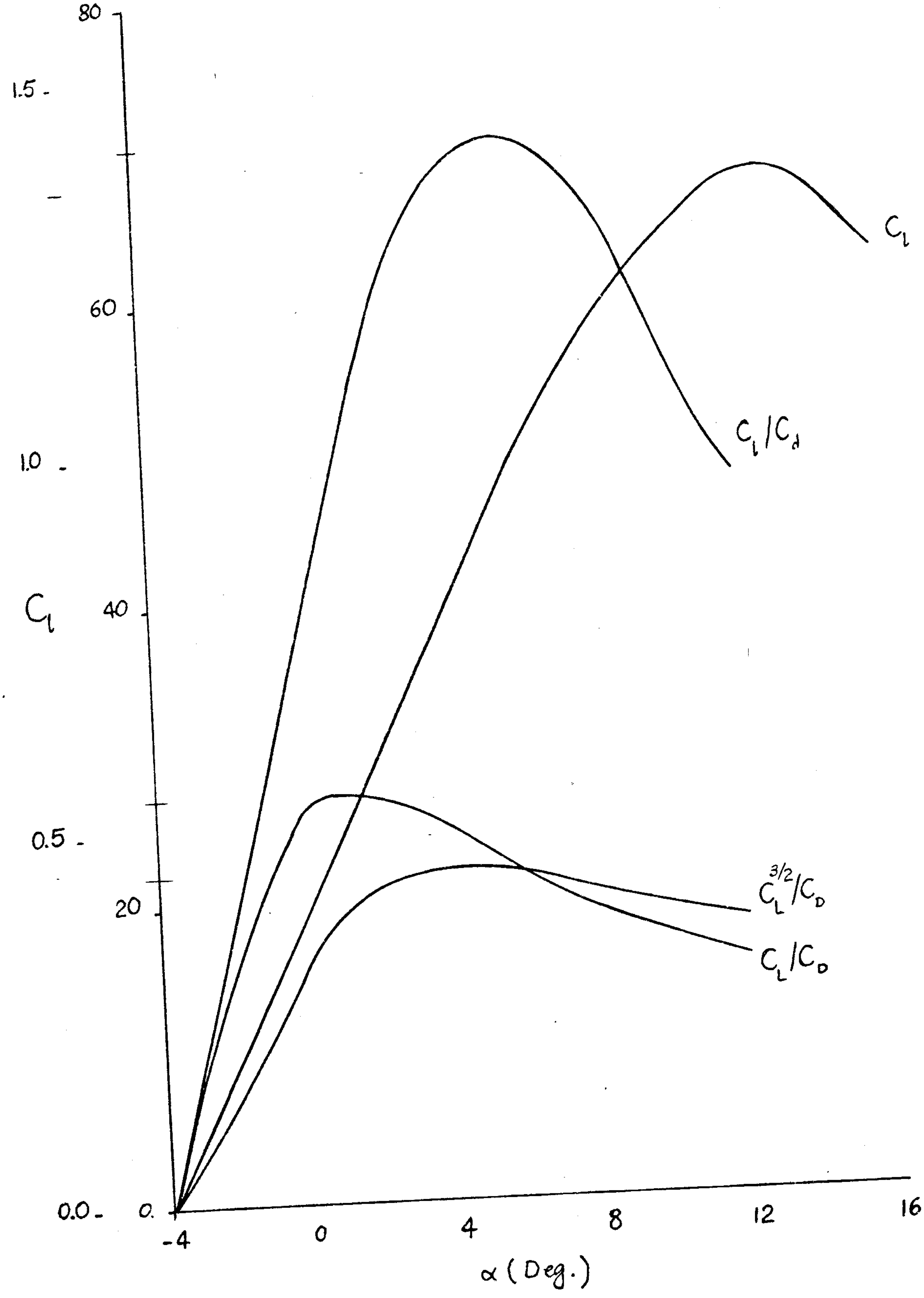
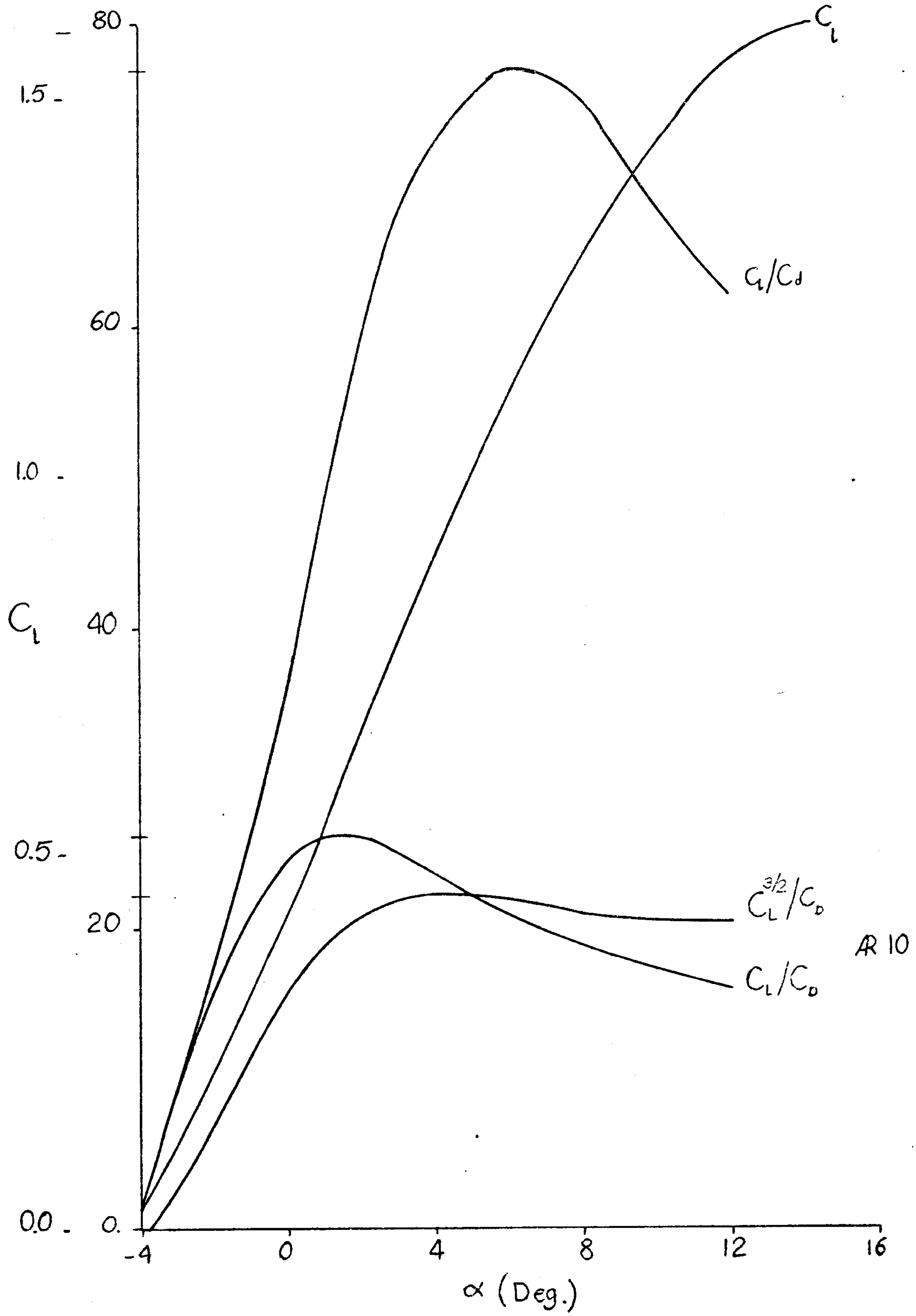


Fig 3.1-7

GA(W)-2



80
1.5
60
1.0
C_l
40
0.5
20
0.0
0.
-4
0
4
8
12
16
α (Deg.)
C_l
C_l/C_d
$C_L^{3/2}/C_D$
C_L/C_D
Æ 10

Fig 3.1.8

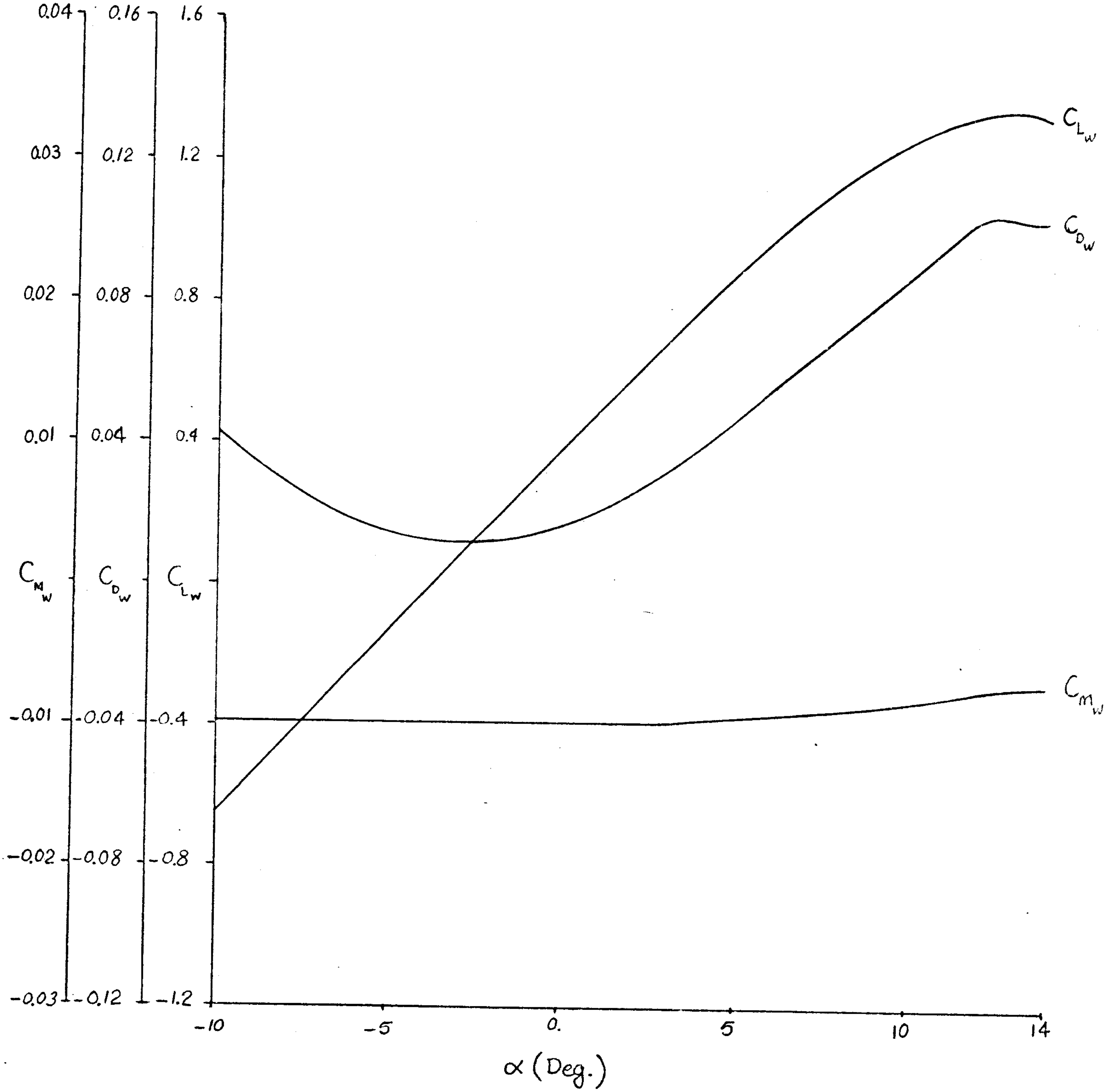

0.04
0.03
0.02
0.01
-0.01
-0.02
-0.03
0.16
0.12
0.08
0.04
-0.04
-0.08
-0.12
1.6
1.2
0.8
0.4
-0.4
-0.8
-1.2
C_{M_w}
C_{D_w}
C_{L_w}
C_{L_w}
C_{D_w}
C_{m_w}
-10
-5
0.
5
10
14
α (Deg.)

Fig 3.1_9

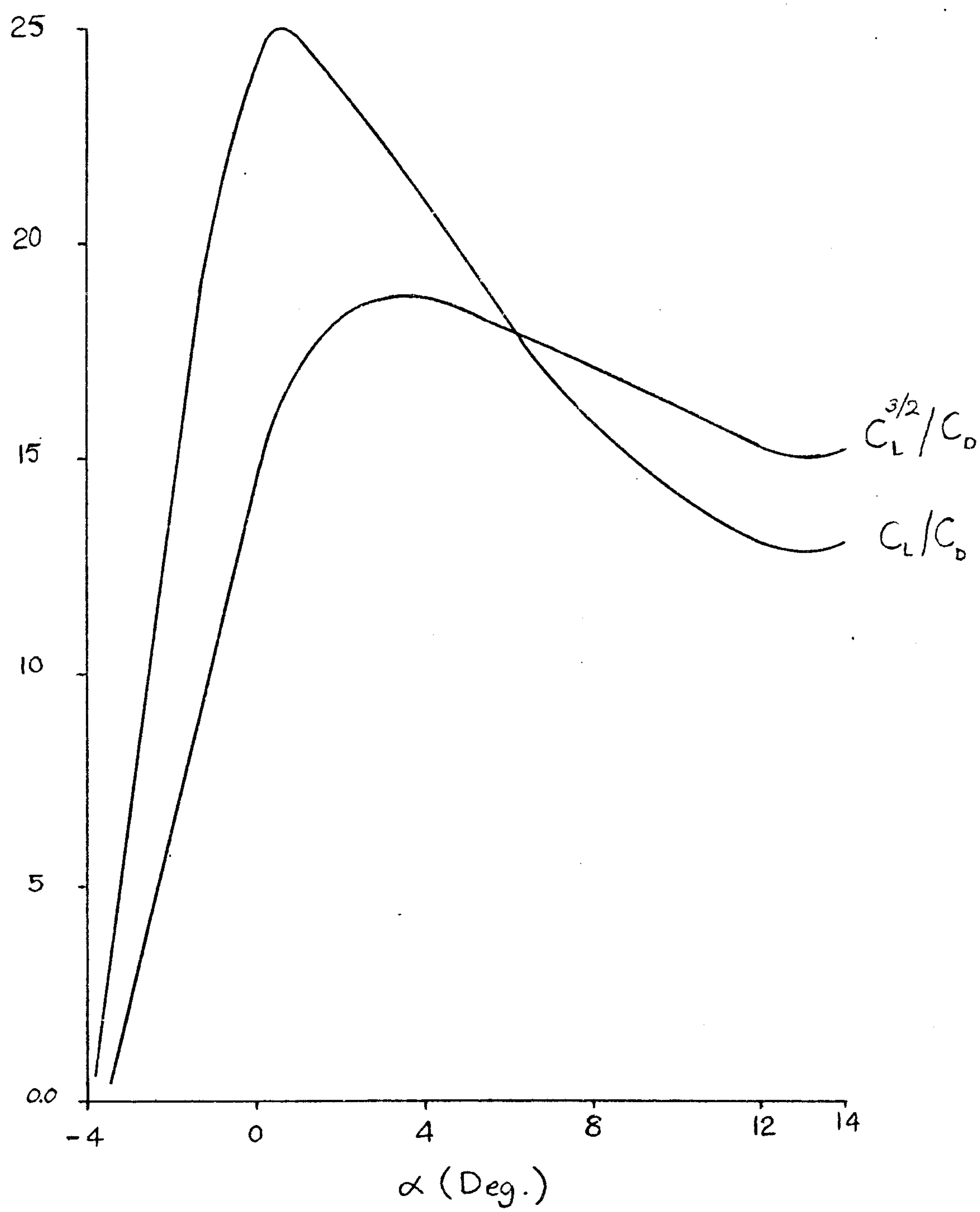


Fig 3.1-10

Since the best values of $C_\ell^{3/2}/C_D$ for this airfoil correspond to lift coefficients of above 0.6, the wing area was reduced to approximately 26 ft^2 in order to allow endurance speeds of approximately 110 ft/sec at 5000 ft altitude.

3.1.5 Flaps

Since the stall speed of the vehicle was one of the most critical parameters, it was decided to use full span flaps in conjunction with spoilers on the wings. Knowing the type of airfoil, some flap estimations were made to find the optimum flap configuration. Through these estimations, it was determined that plain flaps of reasonable size could increase the maximum lift coefficient sufficiently to guarantee very low landing and take-off speeds. They would also require very simple structure and actuation mechanisms which would lead into low production cost and high reliability.

At this point, calculations were performed to find the optimum flap size. Figure 3.1-11 shows the change in the section lift coefficient for different flap deflections and sizes. One the basis of this information, the 25% chord flap was determined to have the best characteristics. Therefore the value of a_2 in Appendix A.3.1 was set equal to 0.25. Figure 3.1-12 shows the semi-span wing planform obtained as the result. Figure 3.1-13 shows also the changes in the airfoil characteristics as the result of flap deflection. The calculation and formulae used for this part are presented in Appendix 3.3.

3.1.6 Vertical and Horizontal Tail Planform

To achieve high manueverability, it was decided to employ all moving horizontal and vertical tails in this design. The very basic

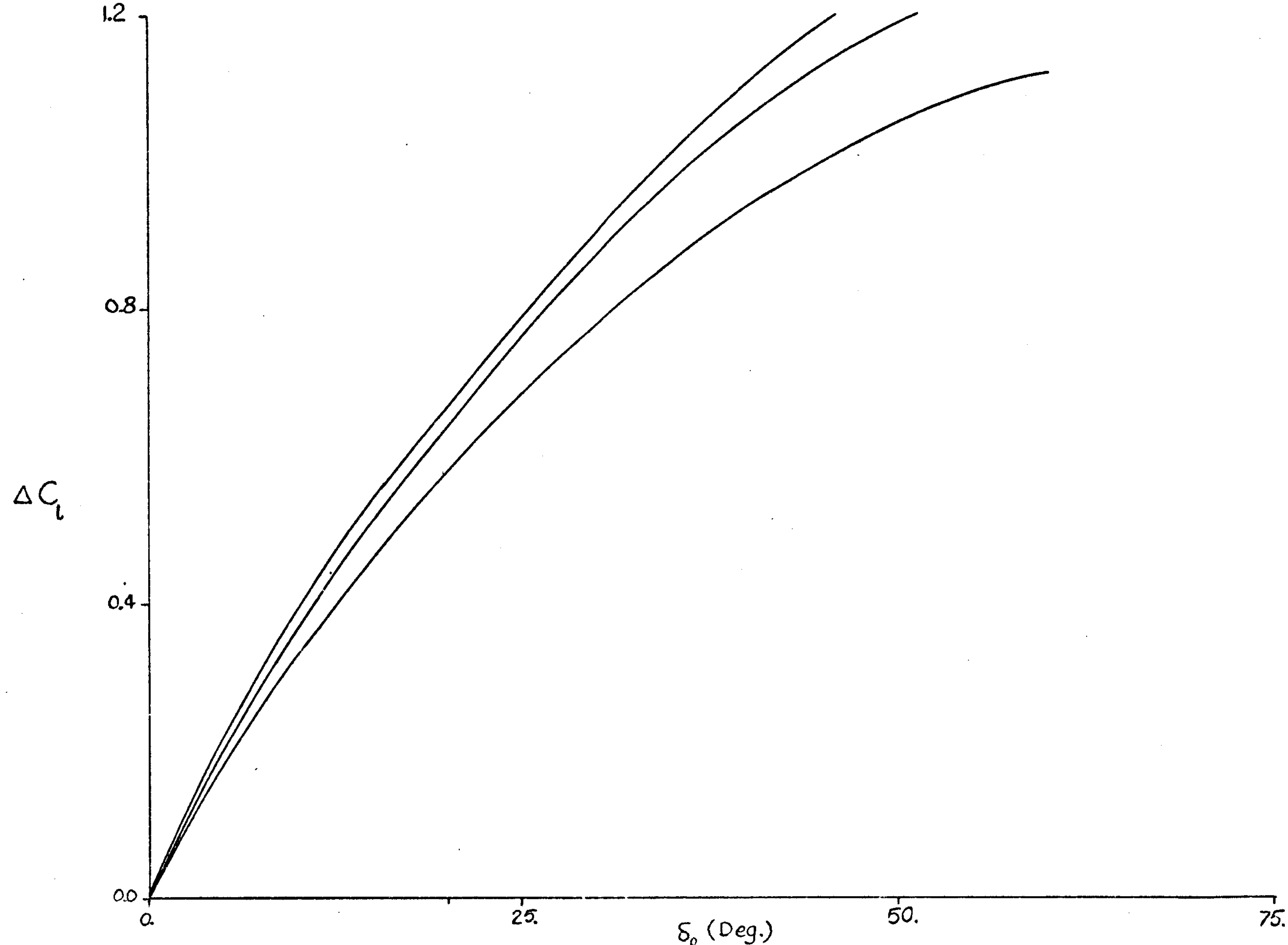


Fig. 3.1-11

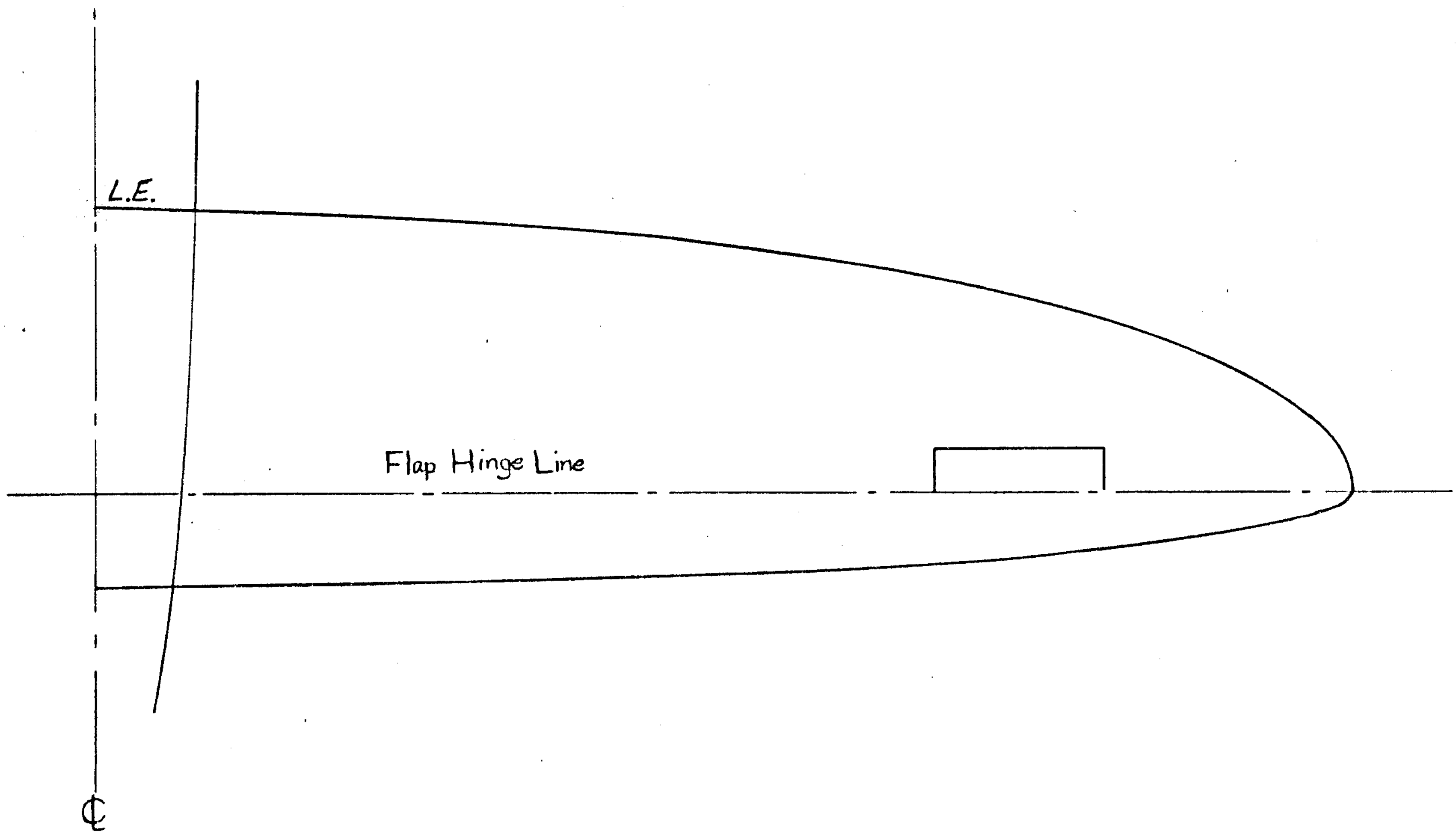


Fig 3.1-12

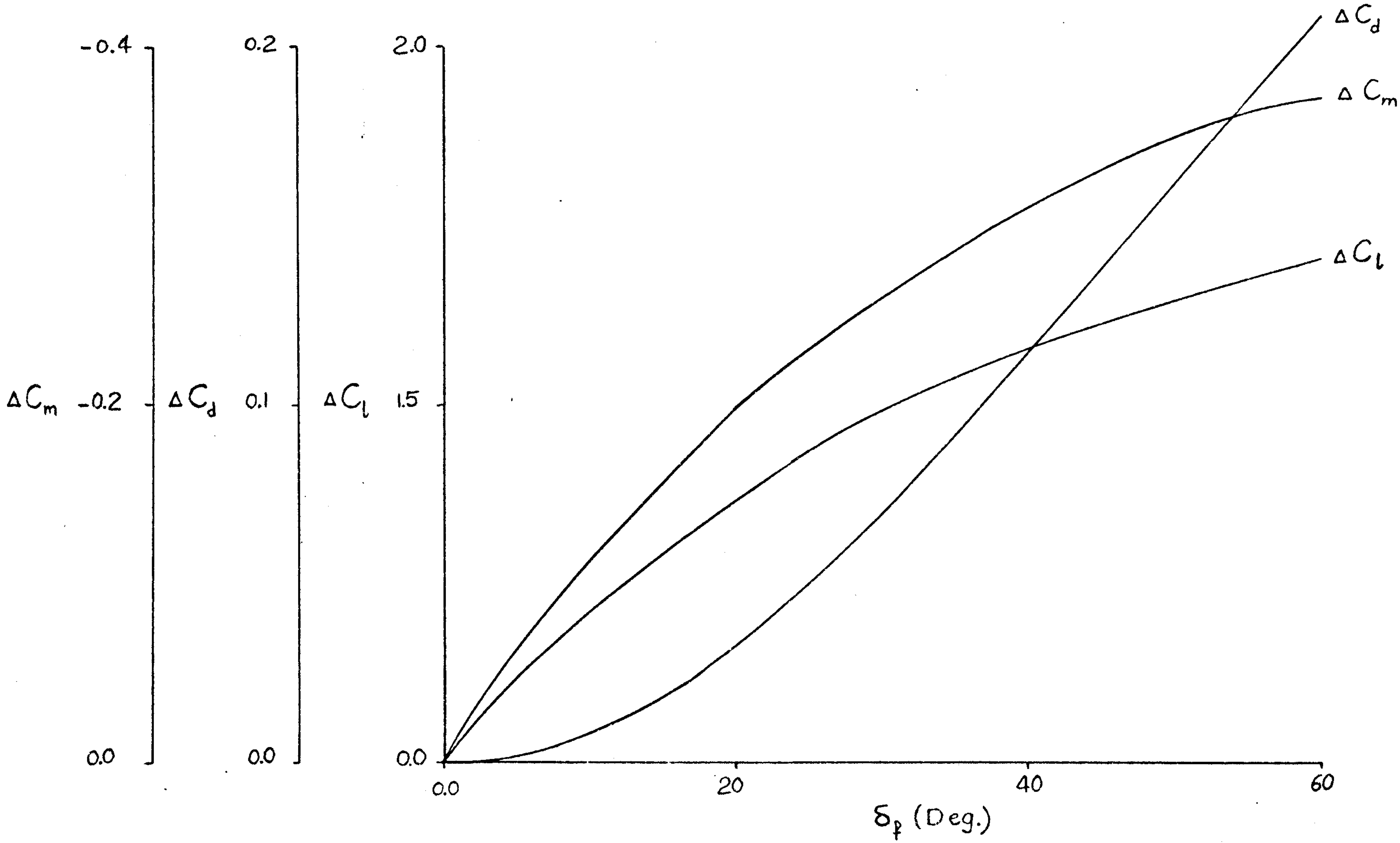

ΔC_m
-0.4
-0.2
0.0
ΔC_d
0.2
0.1
0.0
ΔC_l
2.0
1.5
0.0
0.0
20
40
60
δ_f (Deg.)
ΔC_d
ΔC_m
ΔC_l

Fig.3.1.13

arguments regarding the induced drag of the wing planform also hold true for the vertical and the horizontal tail planforms.

The major trade off which had to be made in this case was between the location of the horizontal tail and the size. Manueverability and minimum down lift of the horizontal tail required a small tail surface located at a far-distance from the nose. However, on the other hand, the length of the tail boom had to be kept as short as possible due to vibrational problems which could arise from a long boom. At this point, as the first approximation, the tail was located at 15 ft from the nose with horizontal tail area of 7.5 ft^2. However, later calculations that took the load factor and down lift of the horizontal tail into account proved that a planform area of 6.5 ft^2 located at approximately 12 ft from the nose would be the most suitable configuration. Details of these calculations are presented in Appendix 3.4.

For the vertical tail, since there was not any spin analysis done, half of the horizontal tail area was chosen for this purpose. This would probably be too large an area, however, it would guarantee spin stability. It would also present the advantage of interchangability of the vertical tail and semi-span horizontal tail. The aspect ratio in this case was determined so that it could be manufactured lightly without loss of much performance. Figure 3.1-14 shows the horizontal tail planform.

3.1.7 Tail Airfoil

In search of an airfoil to fit the tail, the emphasis was put mainly on symmetrical airfoils. Since the tail has to operate at both positive and negative angles of attack, a symmetrical airfoil

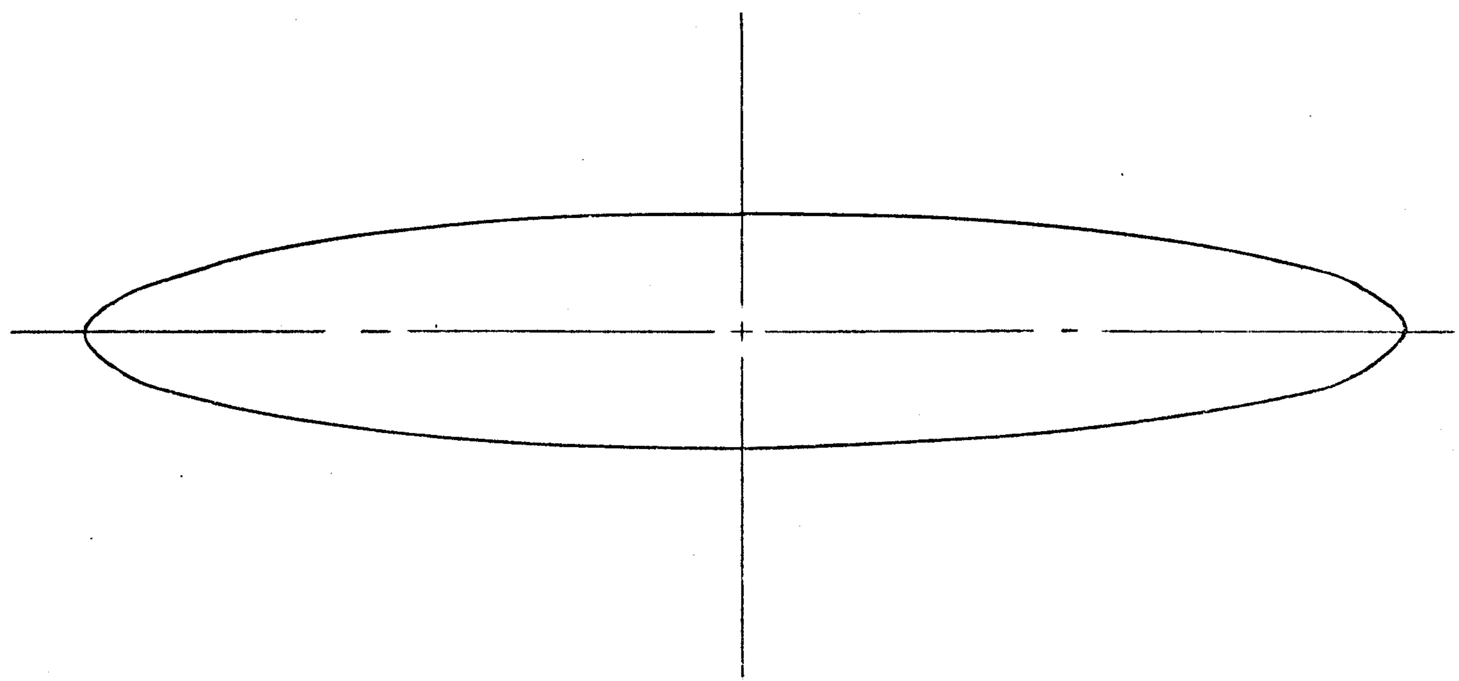
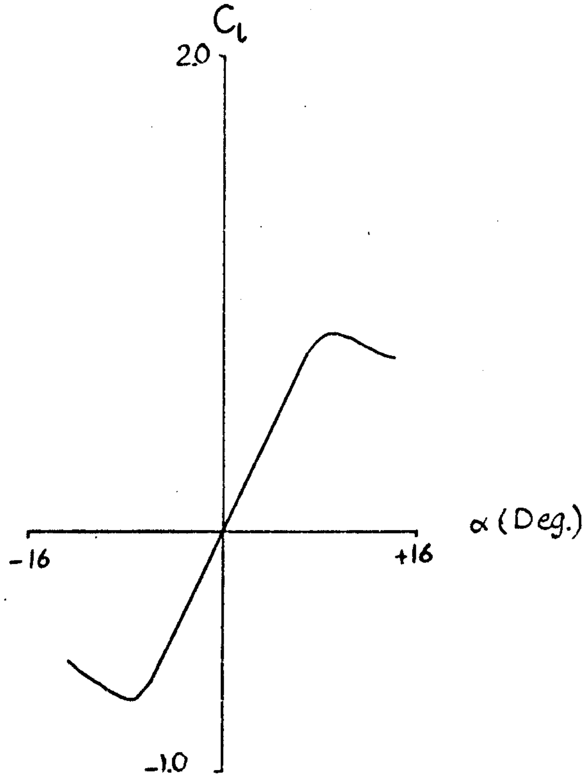


Fig.3.1.14

seemed to be the most suitable. Another requirement set for such airfoil was the low value of $\frac{\partial C_\ell}{\partial C_d}$. This would provide low profile drag for a wider range of lift coefficients which could become marginal during manuevers, and landing and take-off. Figures 3.1-15 through 3.1-18 show four possible candidates. Out of these, NACA 0009 proved to have the highest value of maximum lift coefficient with the lowest value of $\partial C_\ell/\partial C_d$. Therefore this section was used for the tail.

NACA 0006



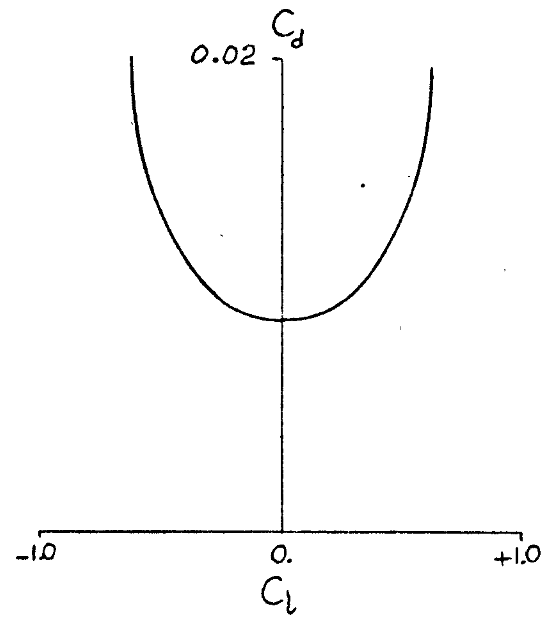
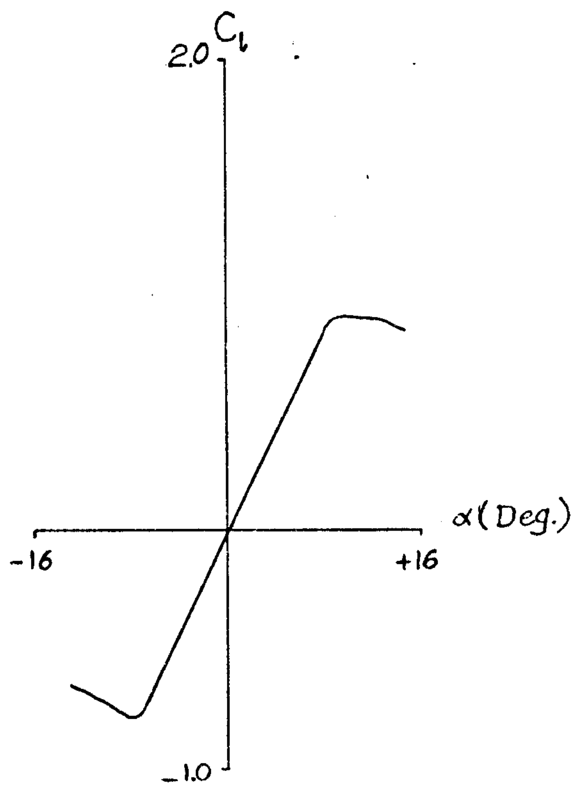


Fig 3.1.15

NACA 0009



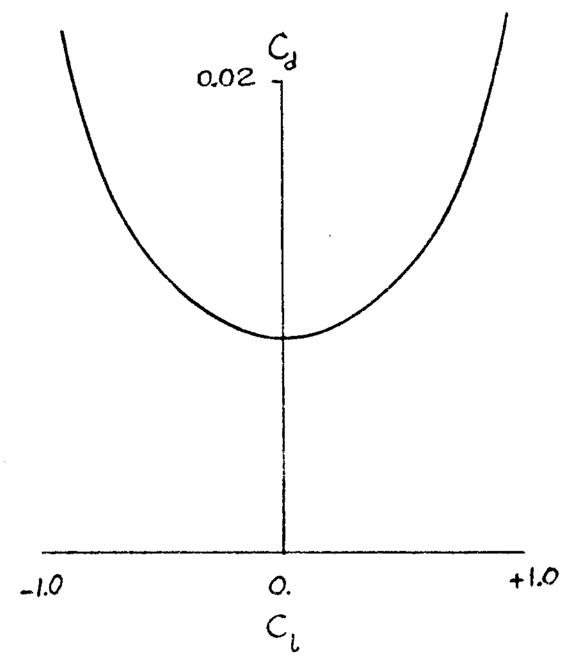


Fig 3.1_16

NACA 65-006

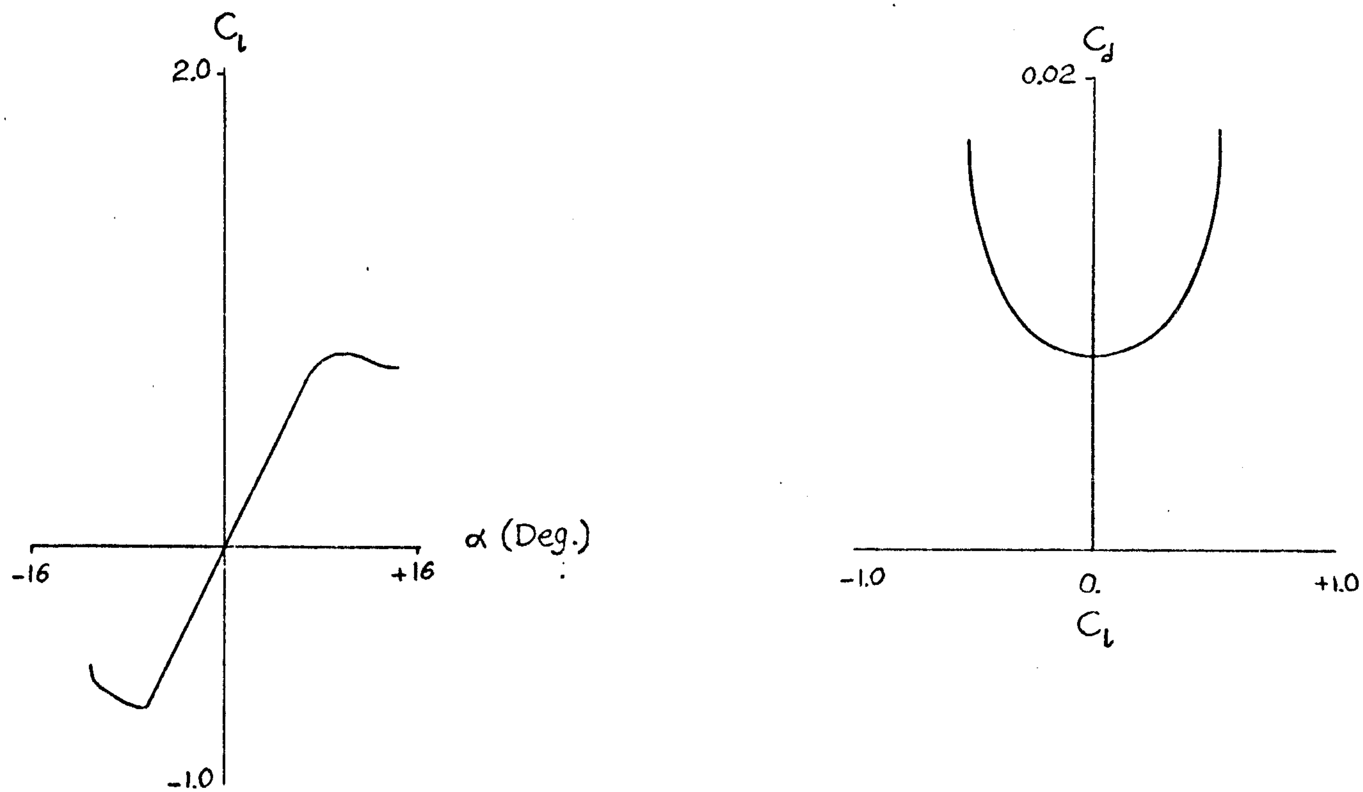


Fig 3.1.17

NACA 65-009

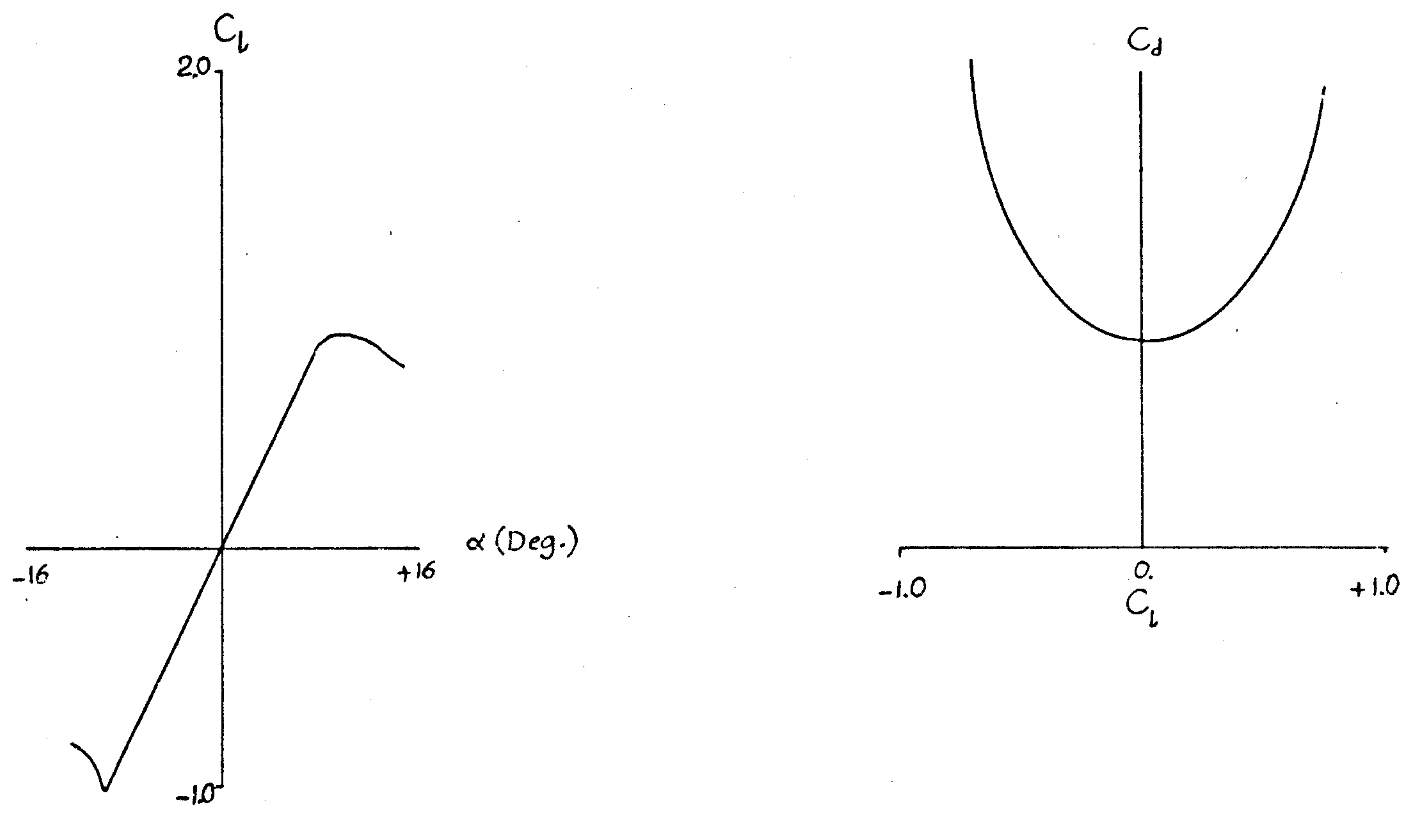


Fig.3.1-18

3.2. STABILITY & CONTROL

3.2.1 Introduction

Due to the long endurance times required to perform every mission, the vehicle would be subjected to great weight changes. These changes at times could be as high as 40% of the maximum gross take-off weight due to the weight of the fuel required to perform each mission. Also the weight of the payload during the different missions, excluding the fuel weight, could be as high as 30% of the design gross take-off weight. This clearly called for a highly manueverable design that could not only be operational for different weights, but would also be efficient under most flight configurations. This goal is believed to have been achieved. In the following sections, a description of the system from the viewpoint of stability and control are presented. It also needs to be mentioned that for a preliminary design of this scale only six static degrees of freedom were considered to be sufficient. However, the analysis of the dynamic behavior of the system can be the subject of further studies.

When all the governing equations in this part were determined, a computer program was set up to correlate the information. The listing of this program is shown in Appendix A.3.5.

3.2.2 Fuselage Pitching Moment

The method used for calculation of the fuselage pitching moment was the one suggested by Roskam[1]. For this purpose, the fuselage and the tail boom were broken down into small sections and then the procedure suggested by Roskam[1] was followed precisely. Figure 3.2-1 shows the sections which were considered along the fuselage. It

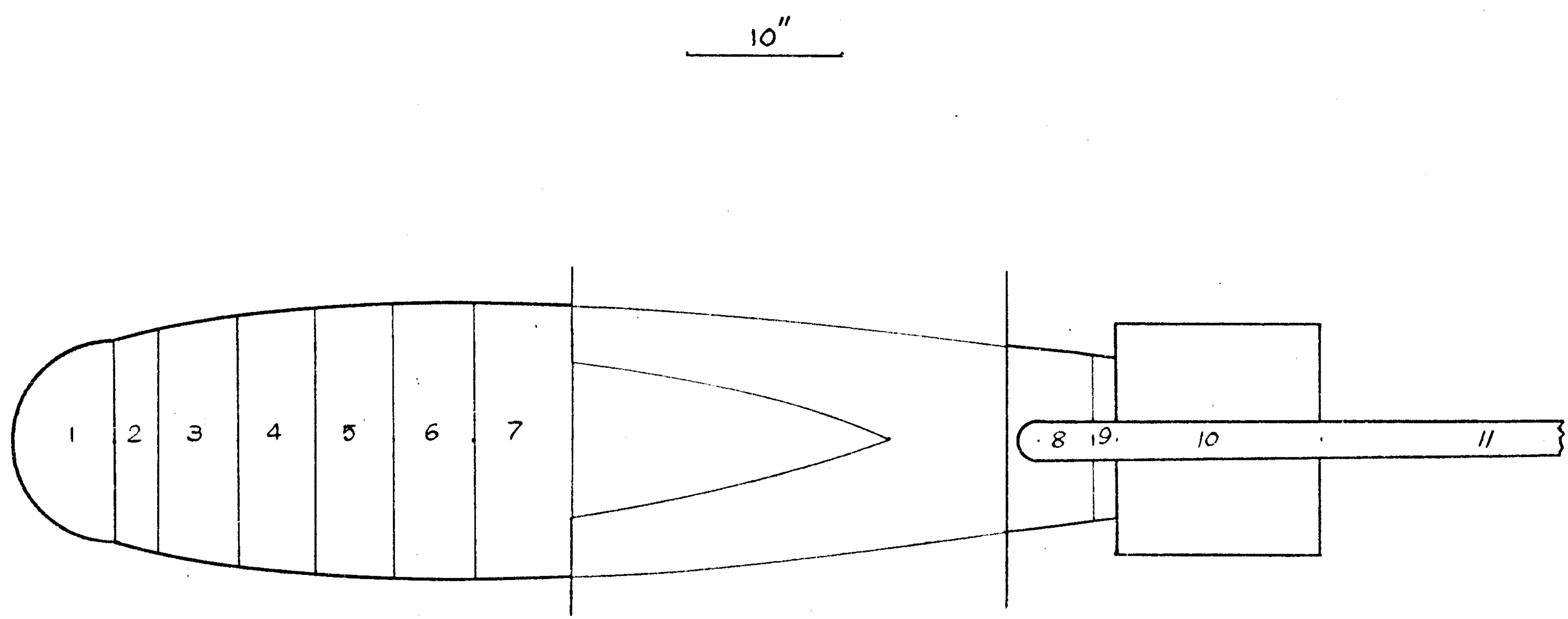


Fig 3.2-1

needs to be mentioned that the upwash ahead of the wing given by this reference was corrected so to fit the vehicle under consideration. The downwash behind the wing was calculated using an imperical functions given by Domash[4]. The validity of these equations were checked using the methods suggested by Roskam[1]. The results obtained from the two methods disagreed by a maximum of 2%. A more detailed discussion of these calculations is presented in Appendix A.3.6. Figure 3.2-1.a shows the calculated values of the downwash at the tail and $d\varepsilon/d\alpha$ for different values of the lift coefficient.

3.2.3 Total Lift Coefficient, C_L

For calculation of this coefficient, it was assumed that the lifting contribution of fuselage was negligible. Therefore the only lifting surfaces considered were the wing and the horizontal tail. The equations used for this part and the details of calculations are presented in Appendix A.3.7. Figure 3.2-2 shows the total airplane C_L for different angles of attack and locations of center of gravity.

3.2.4 Total Drag Coefficient, C_D

As it was mentioned earlier, the surfaces of wing, tail and fuselage were considered to have standard roughness. Due to this fact, the effect of interference drag could be neglected quite safely. Therefore, the total drag coefficient in this part was assumed to be the summation of the drag coefficients of the different parts normalized with respect to the wing area. Also the ratio of the dynamic pressure at the tail to the free stream was assumed to stay at a constant value of 0.95. To predict the fuselage drag

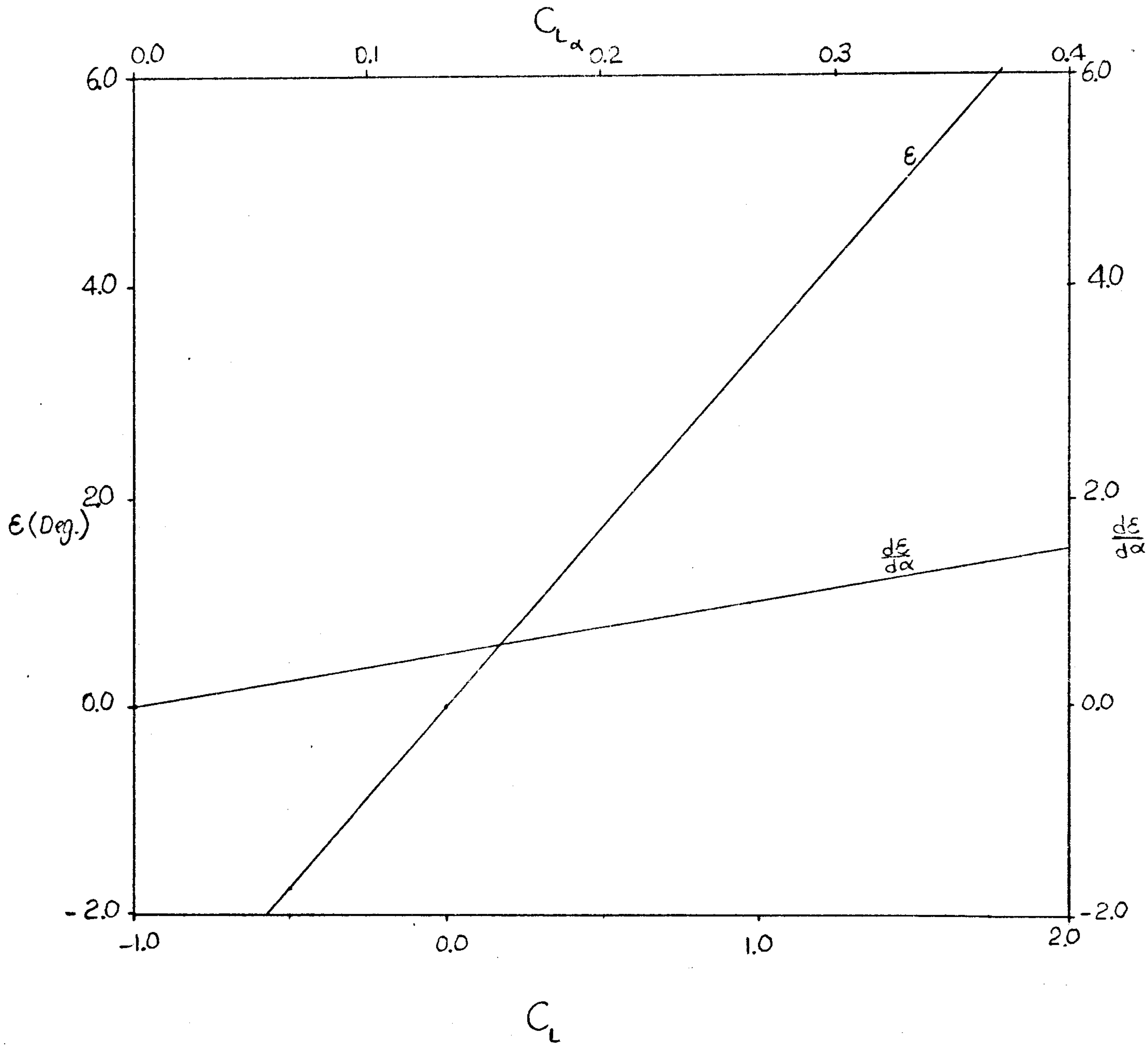

C_{L_α}
0.0
0.1
0.2
0.3
0.4
6.0
4.0
20
0.0
-2.0
ε (Deg.)
ε
$\frac{d\varepsilon}{d\alpha}$
6.0
4.0
2.0
0.0
-2.0
-1.0
0.0
1.0
2.0
C_L

Fig 3.2.1.a

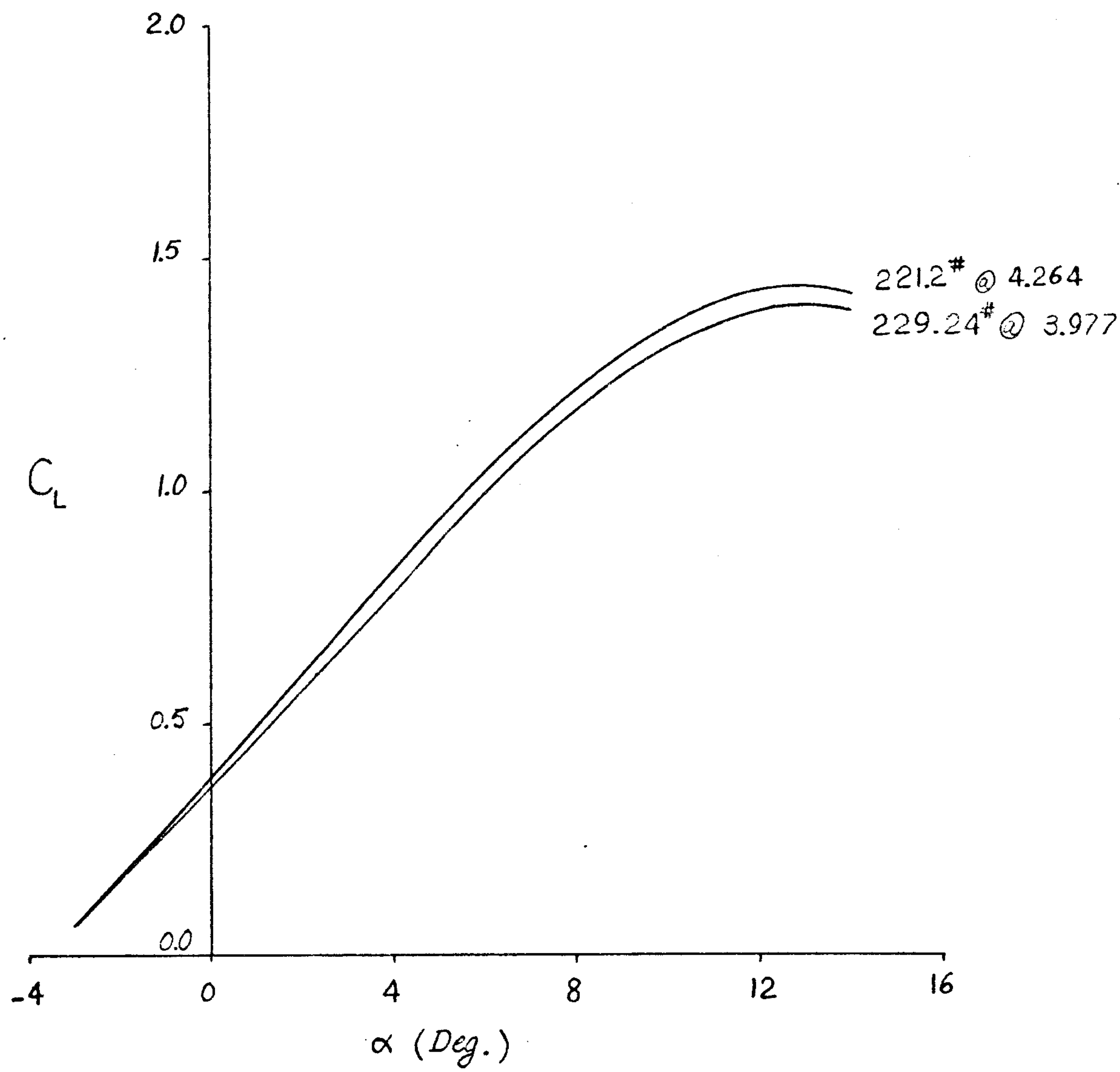


Fig 3.2-2

coefficient at angles of attack, a method was devised which is presented in Appendix A.3.8. This method proved excellent agreement with test data at zero angle of attack.

The detailed calculations and procedures used in this section are presented in Appendix A.3.9. Figures 3.2-3 and 3.2-4 show fuselage drag coefficient and total plane drag coefficient for different C.G. locations, respectively.

3.2.5 Total Pitching Moment Coefficient, C_M.

For this section, the procedure taken was the one suggested by Roskam[1]. The details of this section are shown in Appendix A.3.10.

3.2.6 Airplane Aerodynamic Center, $\overline{X}_{a.c.}$

The airplane location of aerodynamic center was calculated according to the method suggested by Roskam[1]. The location of the wing leading edge at the root was considered as the reference location. The details concerning these calculations are given in Appendix A.3.11.

3.2.7 Airplane Center of Gravity, $\overline{X}_{C.G.}$

The locations of center of gravity during the different missions were calculated and furnished for this part by the Weight Group. A basic coordinate transformation was then performed to find these locations with respect to the wing leading edge at the root. Later all these values were normalized with respect to the mean aerodynamic chord. The results of these calculations are presented in Table 3.2-1.

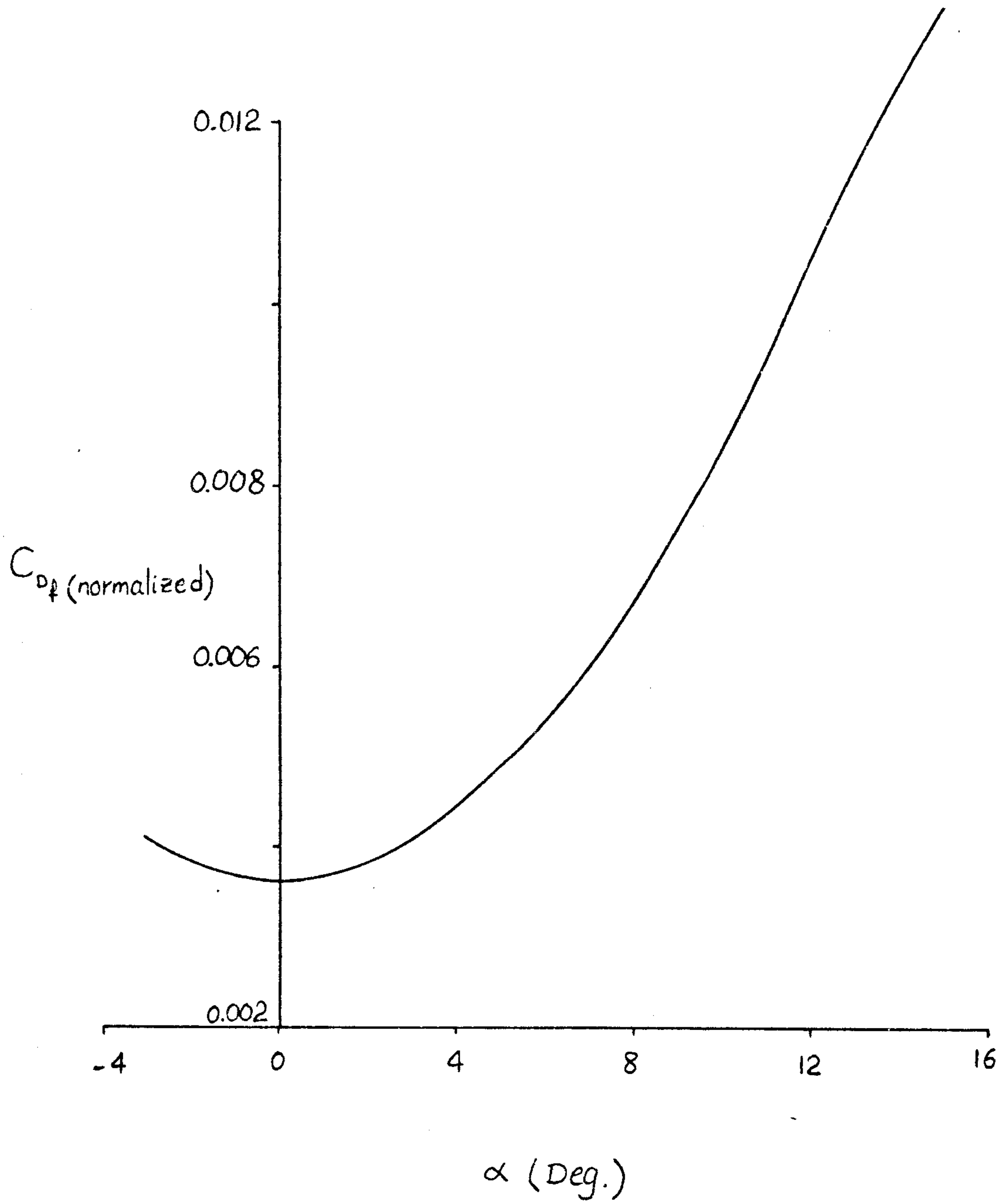


Fig 3.2-3

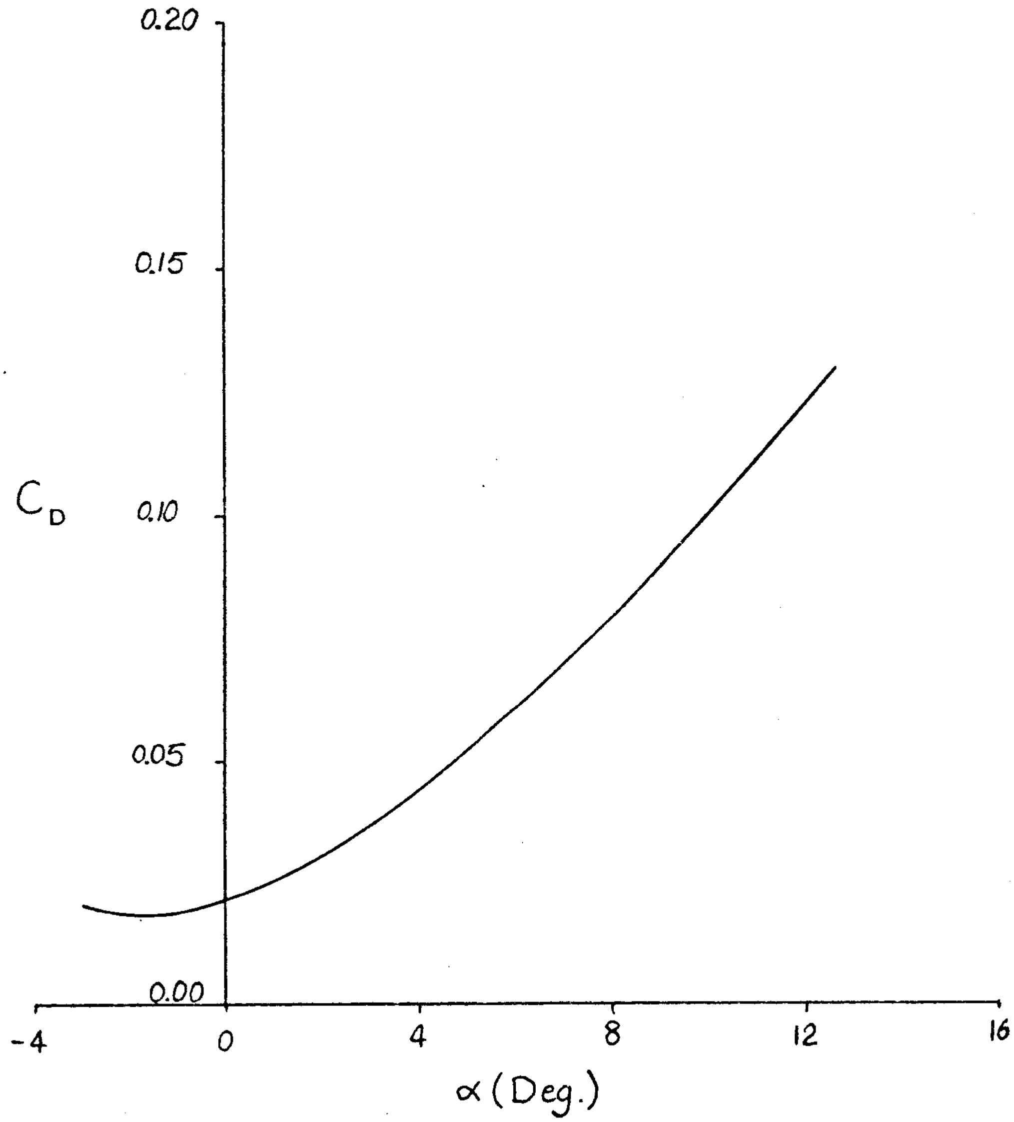


Fig 3.2_4

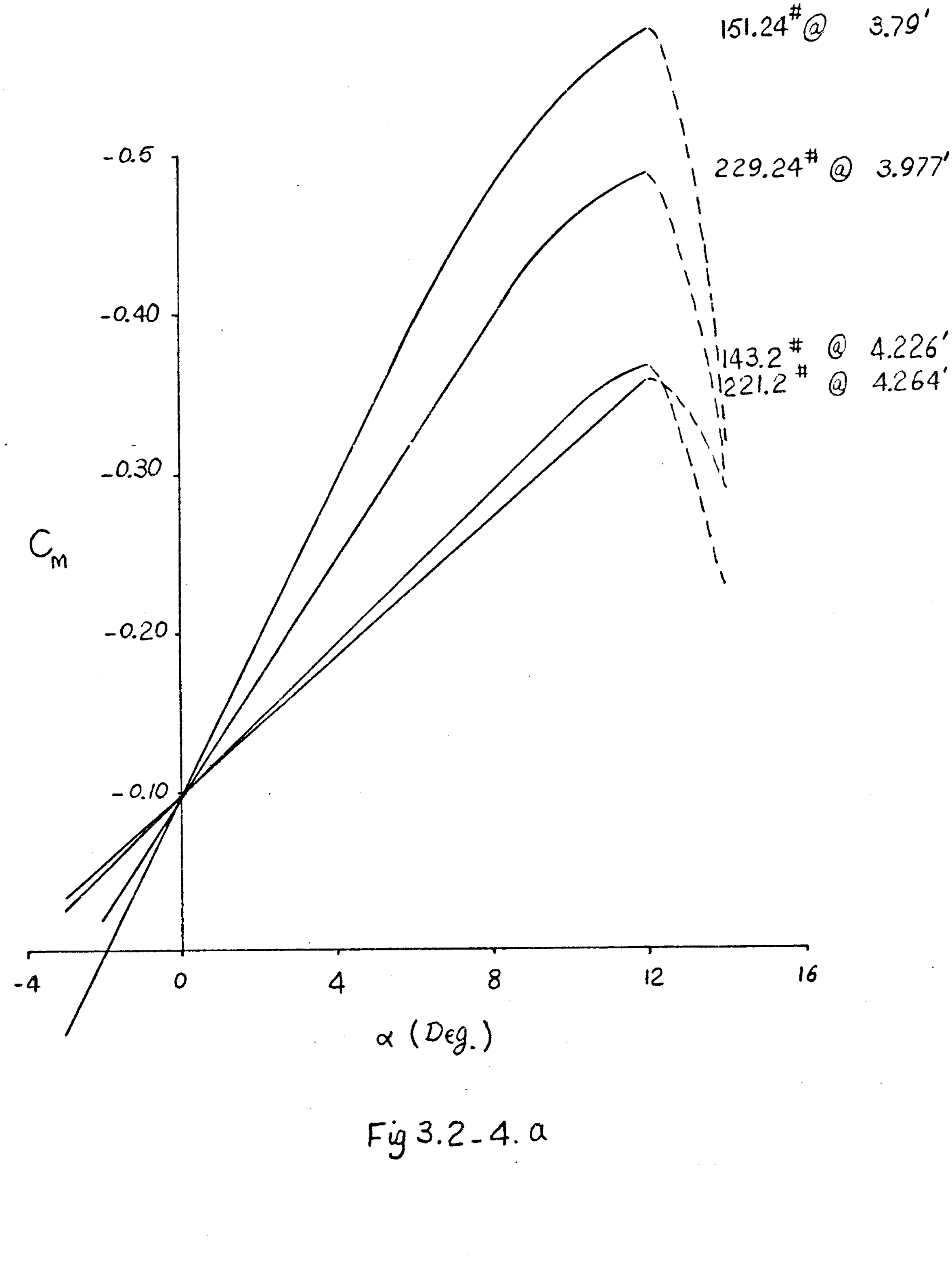

151.24# @ 3.79'
229.24# @ 3.977'
143.2 # @ 4.226'
221.2 # @ 4.264'
-0.5
-0.40
-0.30
-0.20
-0.10
C_M
-4
0
4
8
12
16
α (Deg.)

Fig 3.2-4. a

Mission	Objective	Gross Weight (lbm)	$\bar{X}_{c.g.}$ (ft) (from the nose)	$\bar{X}_{c.g.}$ (normalized)
A	Surveillance & Reconnaissance	143.2*	4.226	0.616
		221.2†	4.264	0.637
B	Target Designation & Damage Reporting	151.24*	3.790	0.373
		229.24†	3.977	0.476

* without any fuel (dry)

† with 13 gallons of fuel (wet)

Table 3.2-1

3.2.8 Wing and Tail Sweep Angles, Λ_n

To find the average sweep angle along the lines of critical fractions of the wing and the tail, the method suggested by Roskam[1] was employed. To use this method, however, the taper ratio of these planforms had to be known. Domash[4] suggests use of 2/3 as taper ratio for elliptic planforms. Appendix A.3.12 shows the governing equation for calculating Λ along with the obtained values using a taper ratio of 2/3.

3.2.9 Total Rolling Moment Coefficient, C_ℓ

This coefficient according to Roskam[1] is composed of four parts as follow:

I. C_{ℓ_o}, representing the basic tendency of the airplane to roll. For an aerodynamically symmetric body this derivatives is equal to zero.

II. C_{ℓ_β}, expressing the rolling tendency due to side-slip. The value of this derivative was calculated using the methods suggested in DATCOM[2] and by Roskam[1]. Figure 3.2-5 shows the behavior of this derivative at different angles of attack.

III. $C_{\ell_{\delta_S}}$, showing the rolling tendency due to spoiler deflection. This derivative was calculated using the data given in DATCOM[2]. Figure 3.2-6 presents the behavior of this derivative with respect to angle of attack.

IV. $C_{\ell_{\delta_R}}$, representing the rolling tendency due to rudder deflection. For calculation of this derivative the method suggested by Roskam[1] was used along with some analytic work. Figure 3.2-7 shows values of $C_{\ell_{\delta_R}}$ for different angles of attack.

The details of calculation along with comments on the obtained values of C_ℓ are presented in Appendix A.3.13.

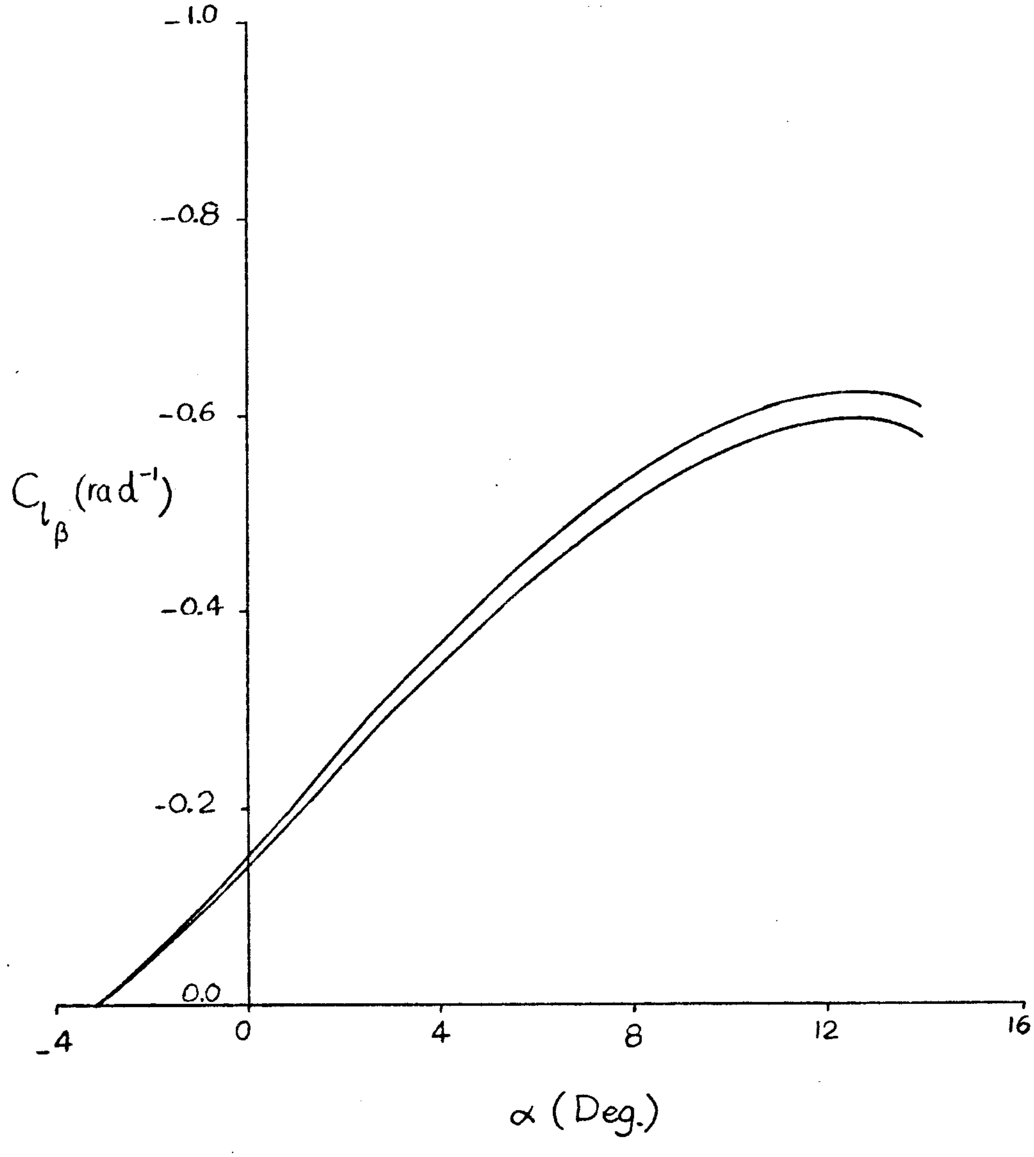


Fig 3.2-5

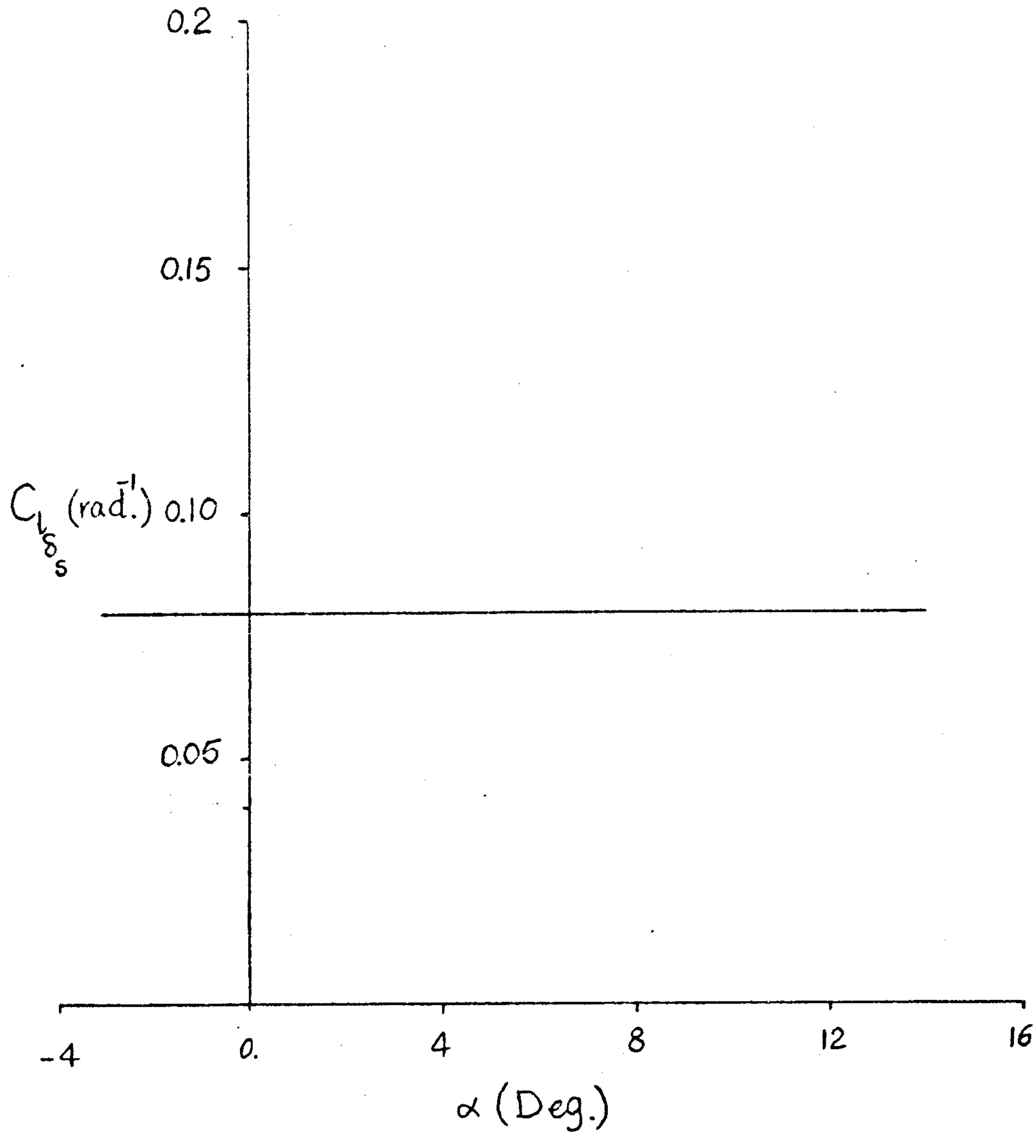


Fig 3.2-6

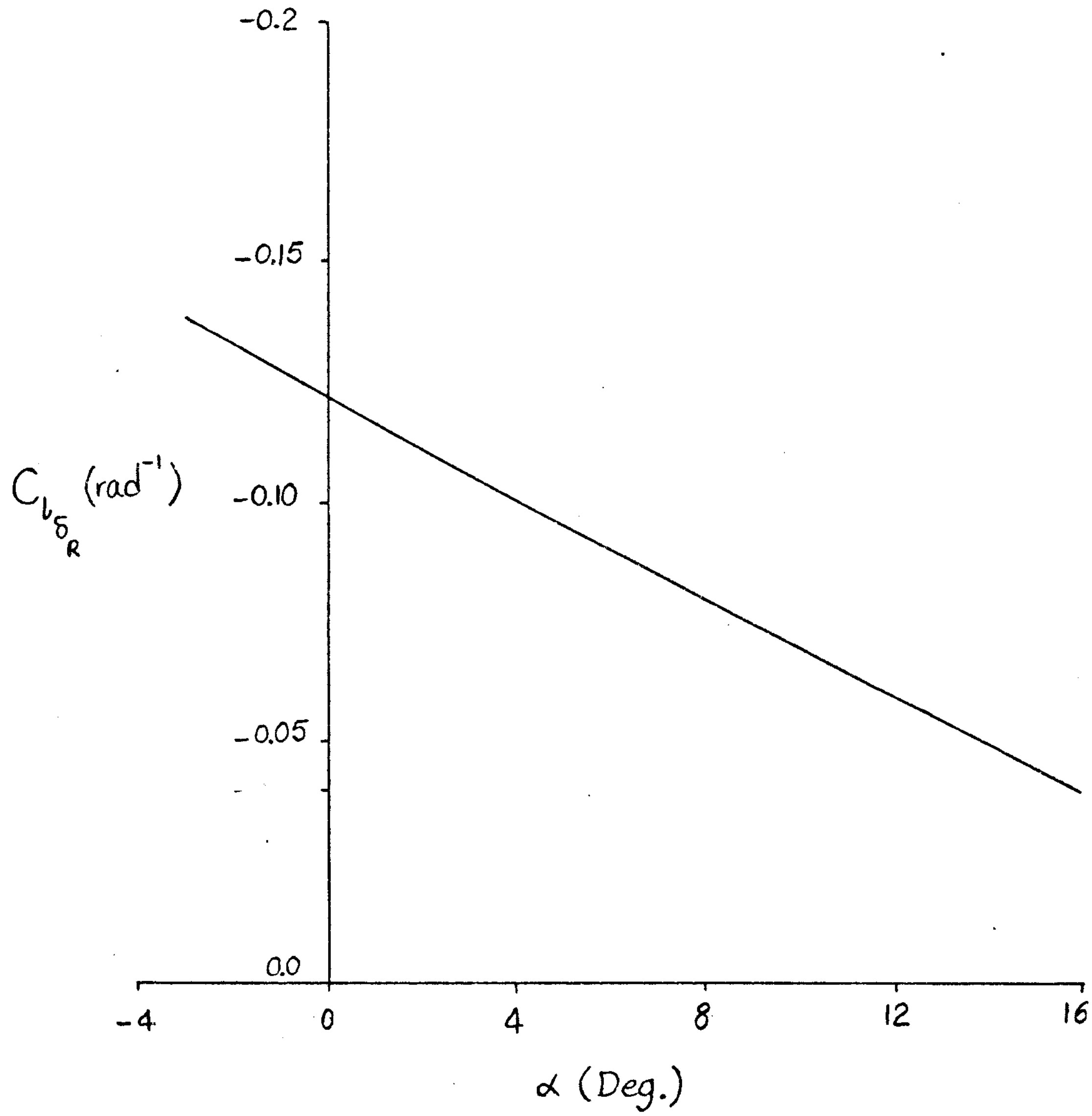


Fig 3.2.7

3.2.10 Total Side Force Coefficient, C_y

According to Roskam[1], this coefficient is composed of four derivatives. These derivatives are as follows:

I. C_{y_0}, inherent side force coefficient which should be zero for a well balanced aerodynamic shape.

II. C_{y_β}, side force coefficient due to a unit side-slip angle. Figure 3.2-8 shows the values of this derivative at different angles of attack.

III. $C_{y_{\delta_S}}$, side force derivative as the result of spoiler deflection. This derivative according to Roskam[1] and DATCOM[2] is negligible in calculating C_y.

IV. $C_{y_{\delta_R}}$, side force derivative due to rudder deflection. This derivative was calculated using methods suggested by Roskam[1] and DATCOM[2]. Figure 3.2-9 presents the behavior of this derivative at different angles of attack.

The details concerning the calculation of C_y along with comments on the obtained data are given in Appendix A.3.14.

3.2.11 Total Yawing Moment Coefficient, C_n

This coefficient resembles C_y very closely. For calculating C_n, the procedures used by Roskam[1] and DATCOM[2] were employed. Figures 3.2-10 and 3.2-11 show the derivatives C_{n_β} and $C_{n_{\delta_R}}$ respectively at different angles of attack. A discussion of the obtained data along with the calculation methods employed for this section are presented in Appendix A.3.15.

3.2.12 Total Side Force Coefficient due to Yaw-Rate, C_{y_r}

Figure 3.2-12 shows the values of this derivative at different angles of attack. Calculations and discussions concerning C_{y_r} are given in Appendix A.3.16.

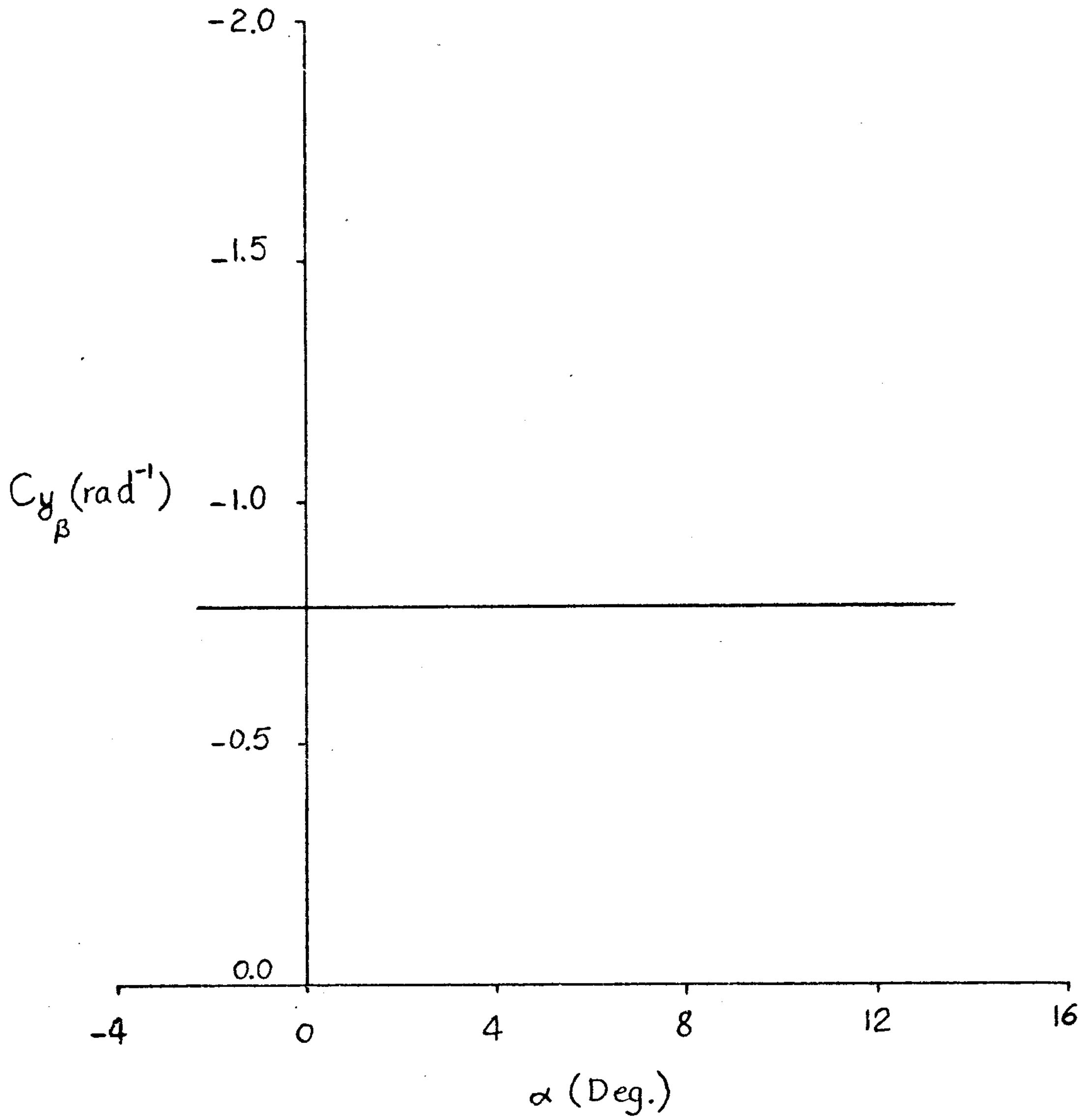


Fig. 3.2-8

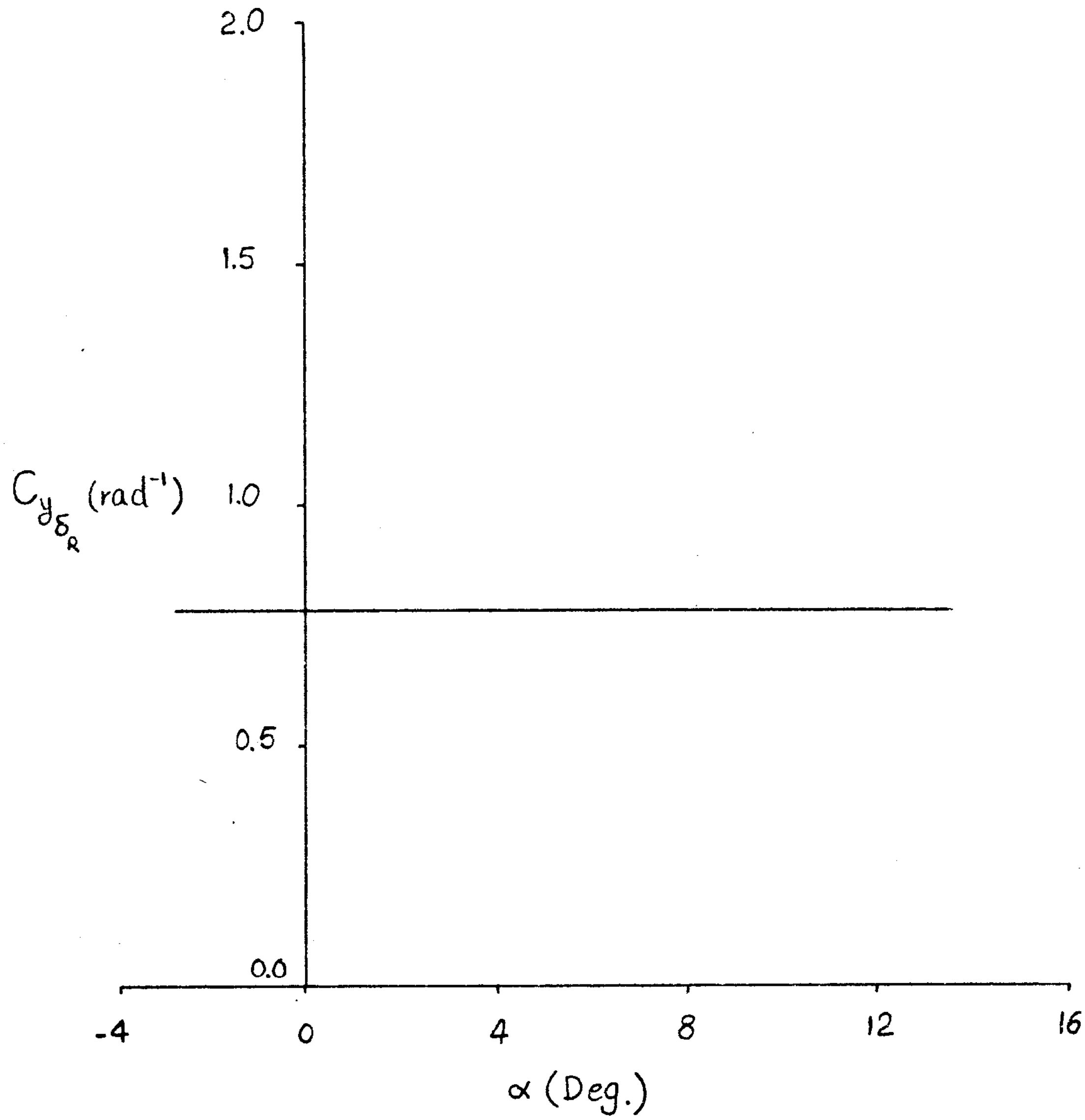


Fig 3.2-9

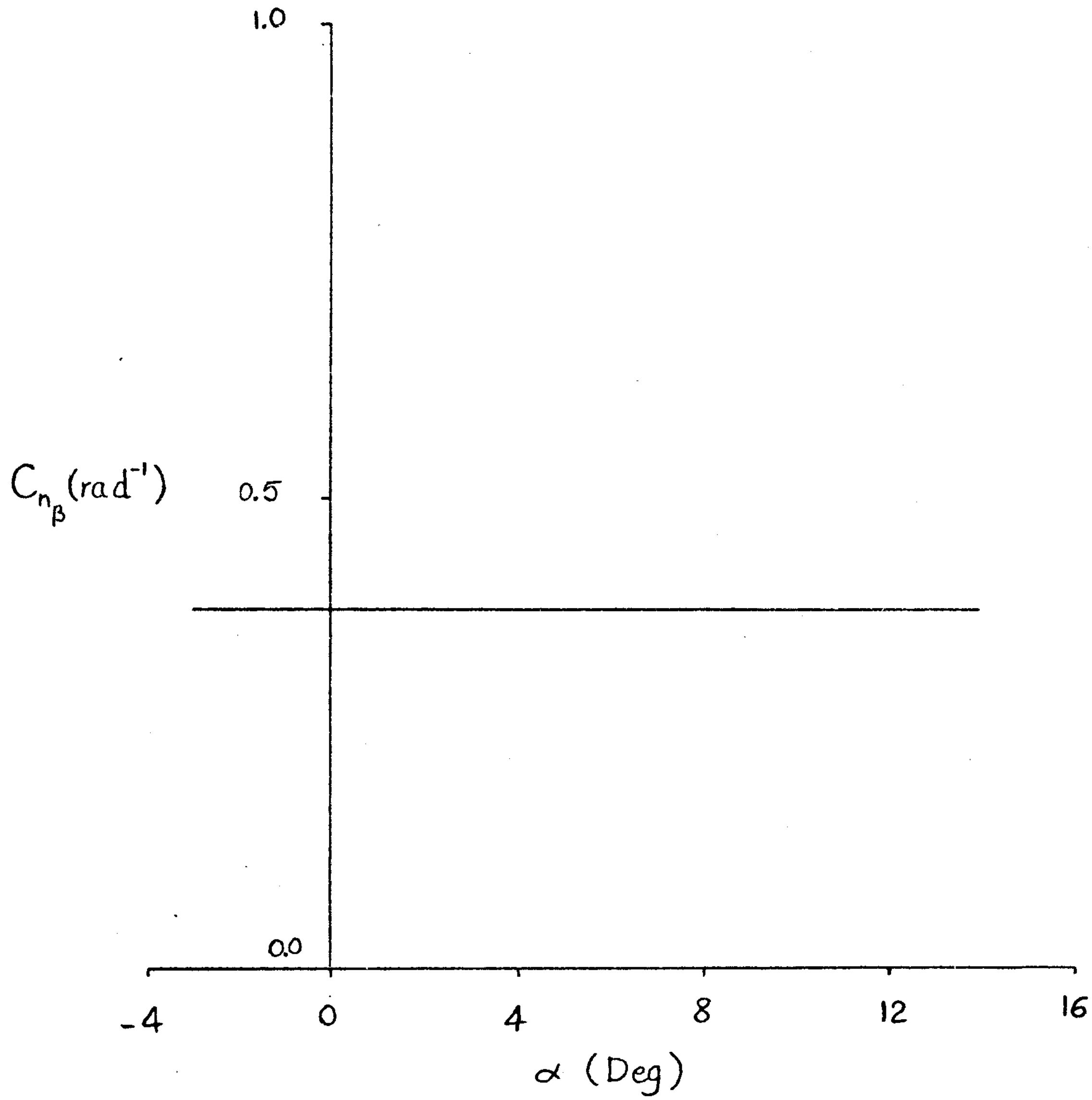


Fig. 3.2-10

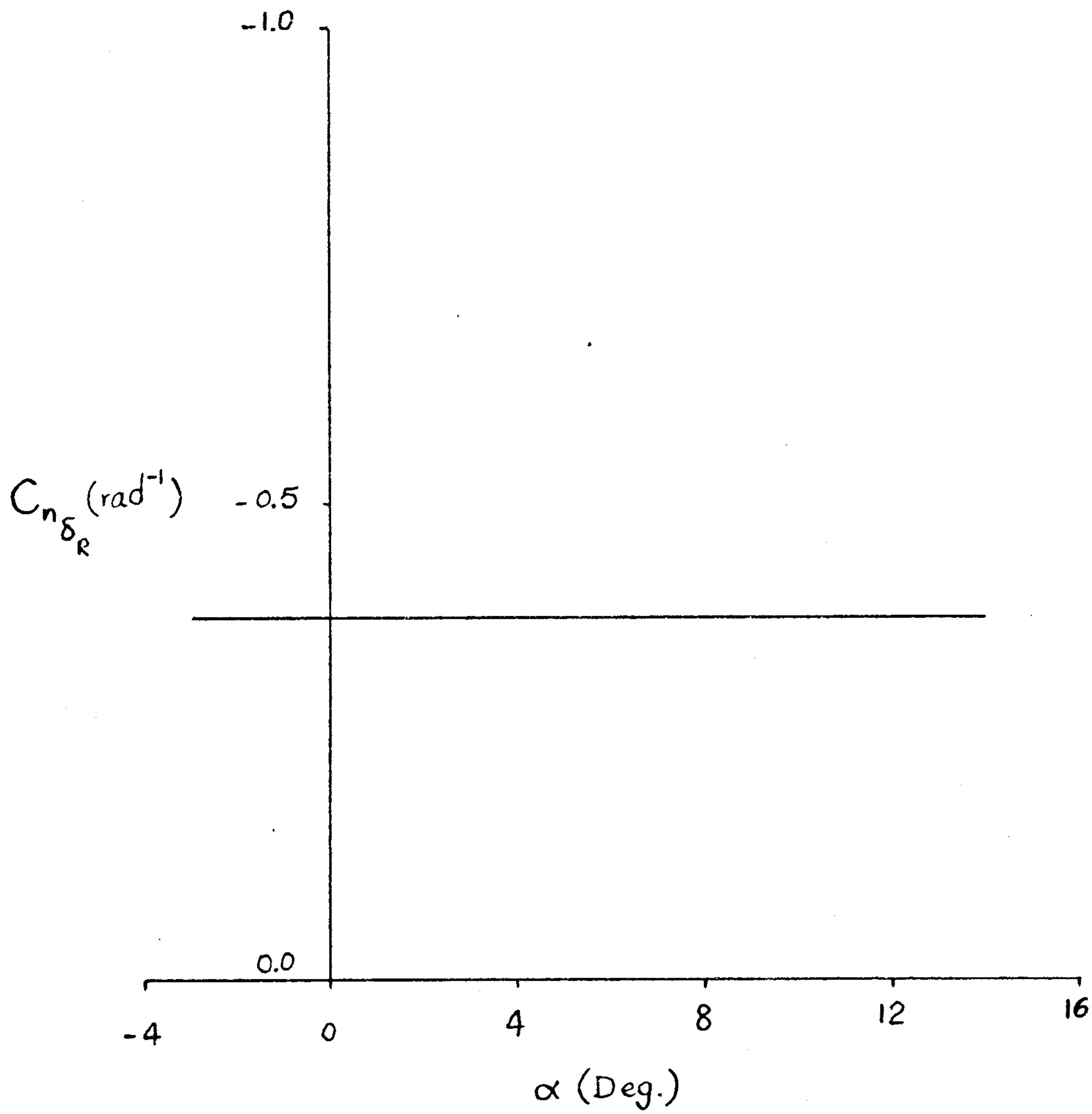


Fig 3.2-11

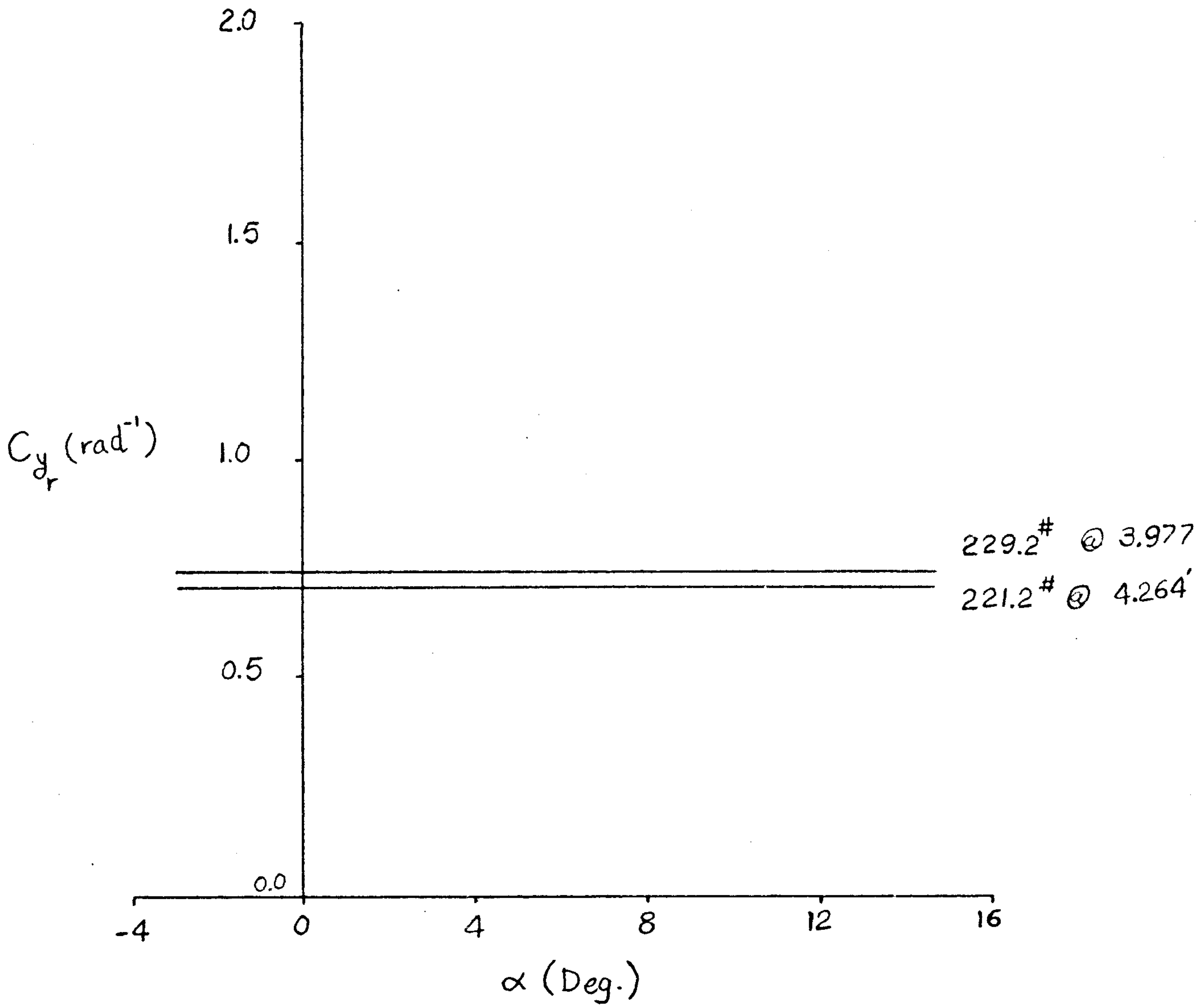


Fig 3.2-12

3.2.13 Total Rolling Moment Coefficient due to Yaw-Rate, C_{ℓ_r}

This coefficient is very closely related to C_{y_r}. Figure 3.2-13 presents the behavior of this coefficient at different angles of attack. The method of calculation for this derivative is given in Appendix A.3.17.

3.2.14 Total Yawing Moment Coefficient due to Yaw-Rate, C_{n_r}

Figure 3.2-14 presents the values of this coefficient at different angles of attack. A discussion of these values along with the methods of calculation are presented in Appendix A.3.18.

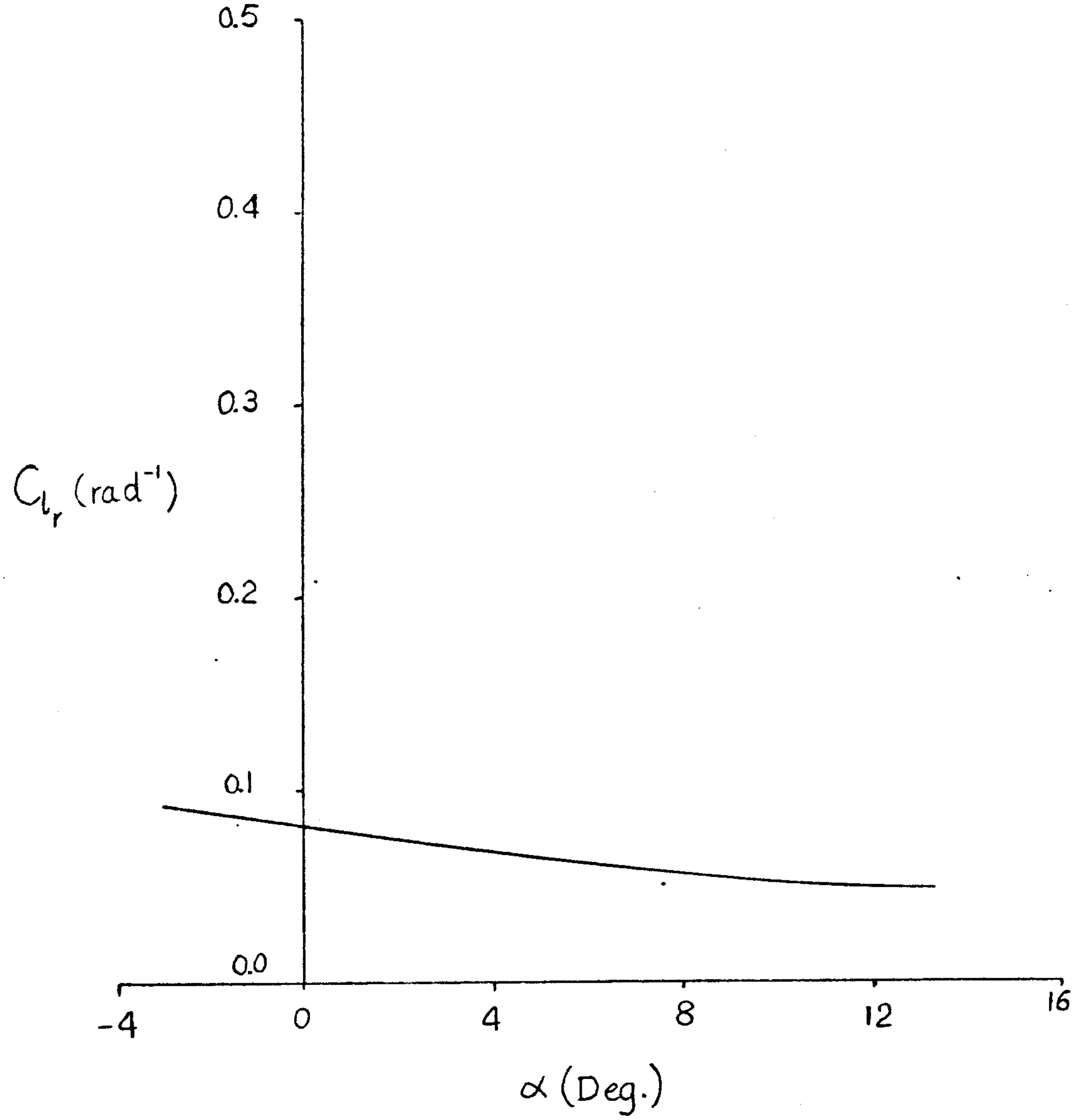


Fig 3.2-13

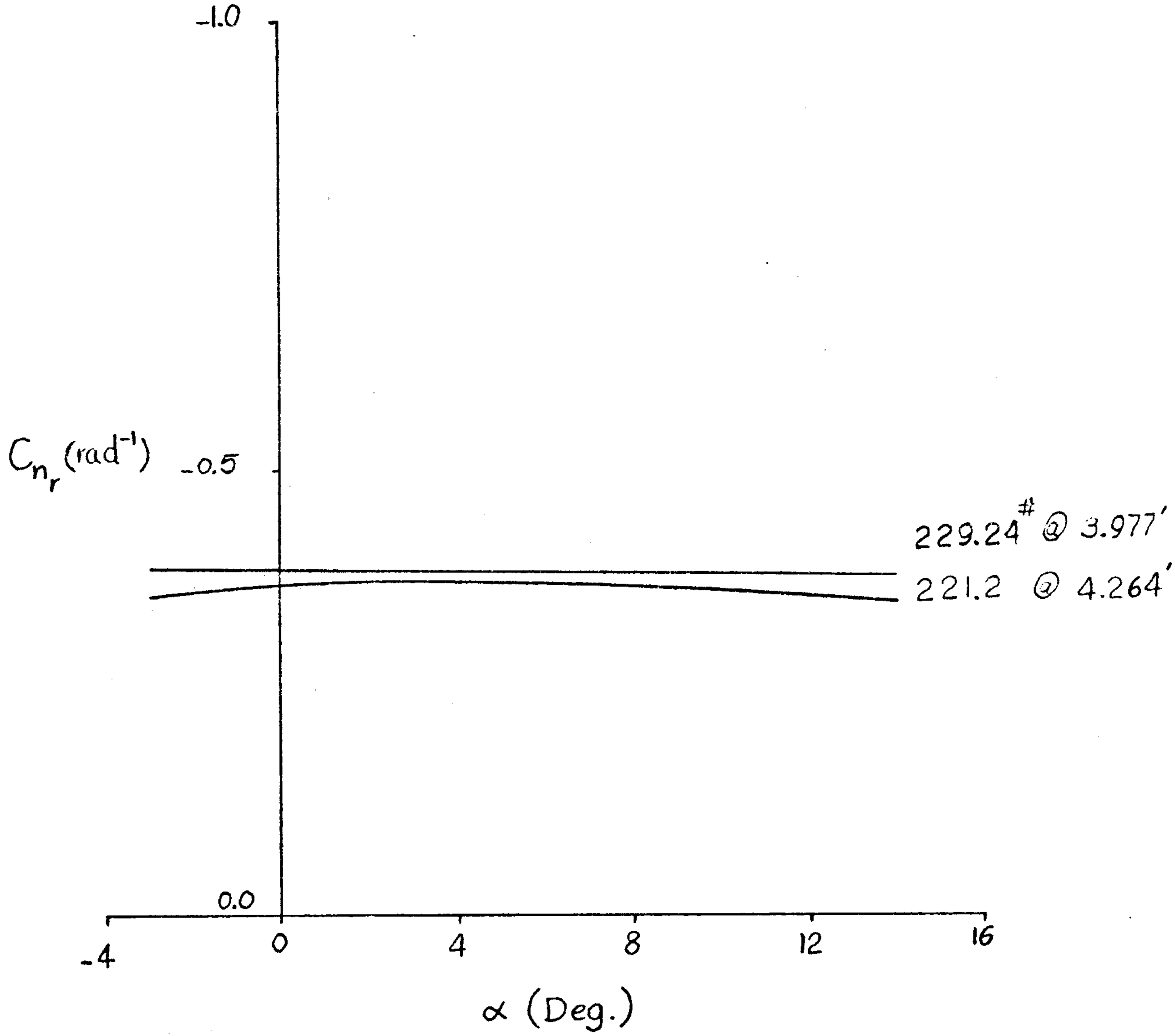


Fig 3.2-14

3.3. PERFORMANCE

3.3.1 Introduction

The missions described in the previous sections imply the requirement for excellent performance characteristics. The short field landing and take-off requirements, which would allow operations of this nature to be performed in geographically tight areas, naturally asked for high rates of climb. This could be satisfied using a slightly over-sized power plant. On the other hand, the maximum endurance times called for by the two primary missions did not allow much larger power plant than was needed for loiter, due to fuel efficiency requirements. Fortunately, this dilemma was resolved through excellent weight distribution and planning furnished by the Weight Group and thorough studies performed by the Propulsion Group.

The results of all trade-off studies performed by the Missions, Aerodynamics, Propulsion, and Structures Groups were combined to devise the best possible configuration. Therefore the performance presented in this section will reveal not only the capabilities of the entire system, but will also signify the extent to which all the aforementioned areas coalesce.

3.3.2 Steady State Level Flight Power Requirements

The total required power in every flight mode was very closely approximated by the product of the forward velocity and the total airplane drag. However, a more exact method was employed, i.e.

$$P_{req.} = \frac{V \cdot W \cdot C_D}{C_L \cos \alpha_T + C_D \sin \alpha_T}$$

which yields

$$P_{req.} = f(V_\infty^{\,3})$$

This expression clearly indicates the direct relation between required power and C_D. The computer program presented in Appendix A.3.5 has the capability of solving the equations of motion for the dynamic pressure required during steady state level flight for different weights, C.G. locations, and air densities. Figures 3.3-1, 3.3-2, and 3.3-3 show power available and power required for different velocities at altitudes of sea level, 5000 feet, and 15000 feet respectively. In Figure 3.3-2, it is note worthy that the minimum power required occurs at a speed of approximately 110 ft/sec. which was one of the design objectives.

3.3.3 Rate of Climb vs. Forward Velocity

Figures 3.3-1 through 3.3-3 not only present the available and the required powers, but they also present the amount of power at any speed, up to the maximum forward speed that can be allocated for climb, if full throttle power is employed. Domash[4] gives the following relation for rate of climb and excess power:

$$\text{Rate of Climb} = \frac{\text{Excess Power}}{\text{Total Weight}}$$

Figures 3.3-4 through 3.3-6 present the rate of climb at different speeds for altitudes of sea level, 5000 ft. and 15000 ft. respectively. Also Figure 3.3-7 shows the ceiling performance of the vehicle on the basis of the calculated rates of climb, through direct extrapolation. Figure 3.3-8 also presents the velocity hodograph for this plane. The terminal velocities expressed in this figure signify the aerodynamic cleanliness of the entire vehicle.

3.3.4 Steady State Level Flight Endurance

For calculating the maximum endurance of the vehicle, the equation suggested by Domash[4] was used, i.e.

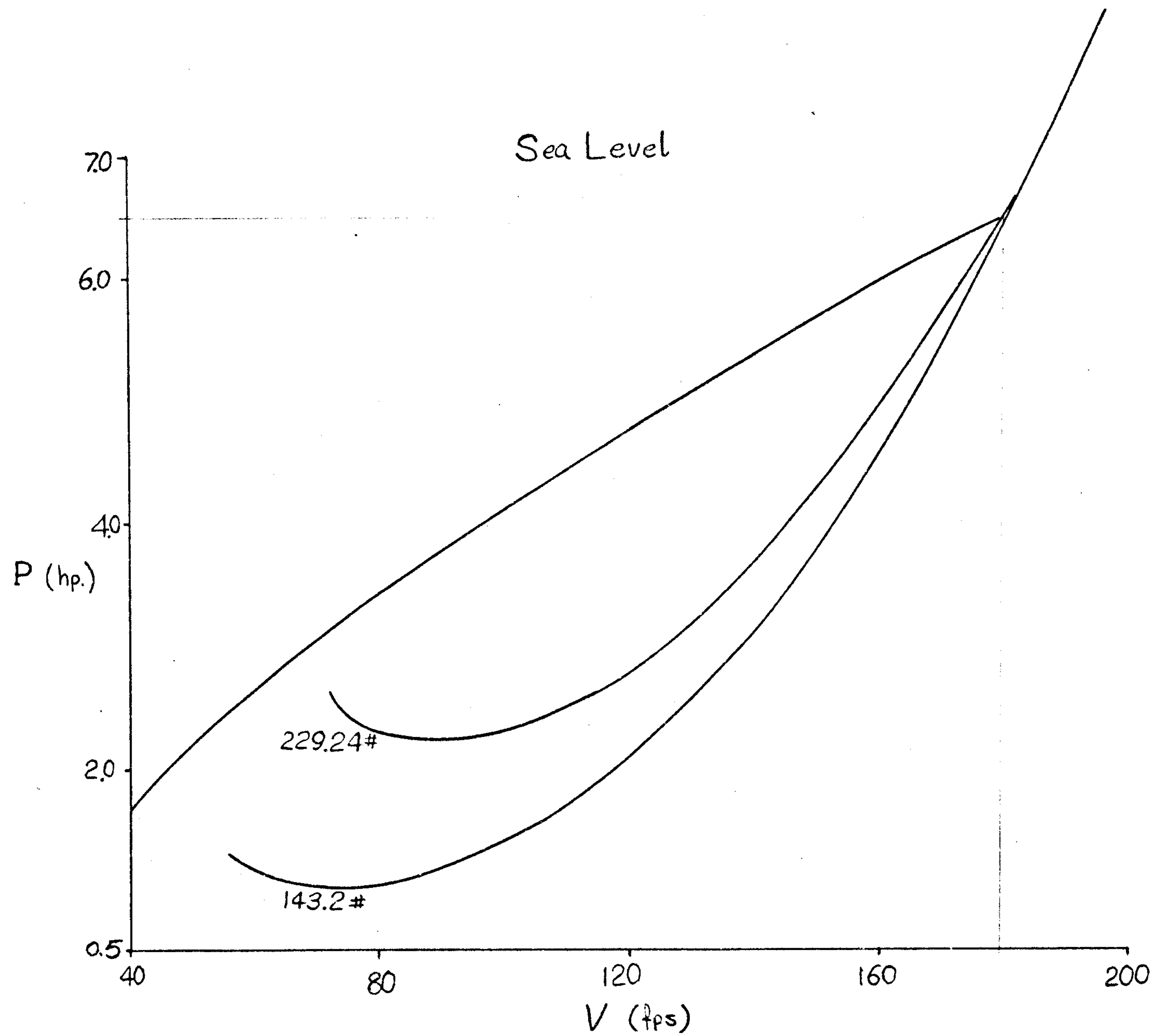


Fig 3.3-1

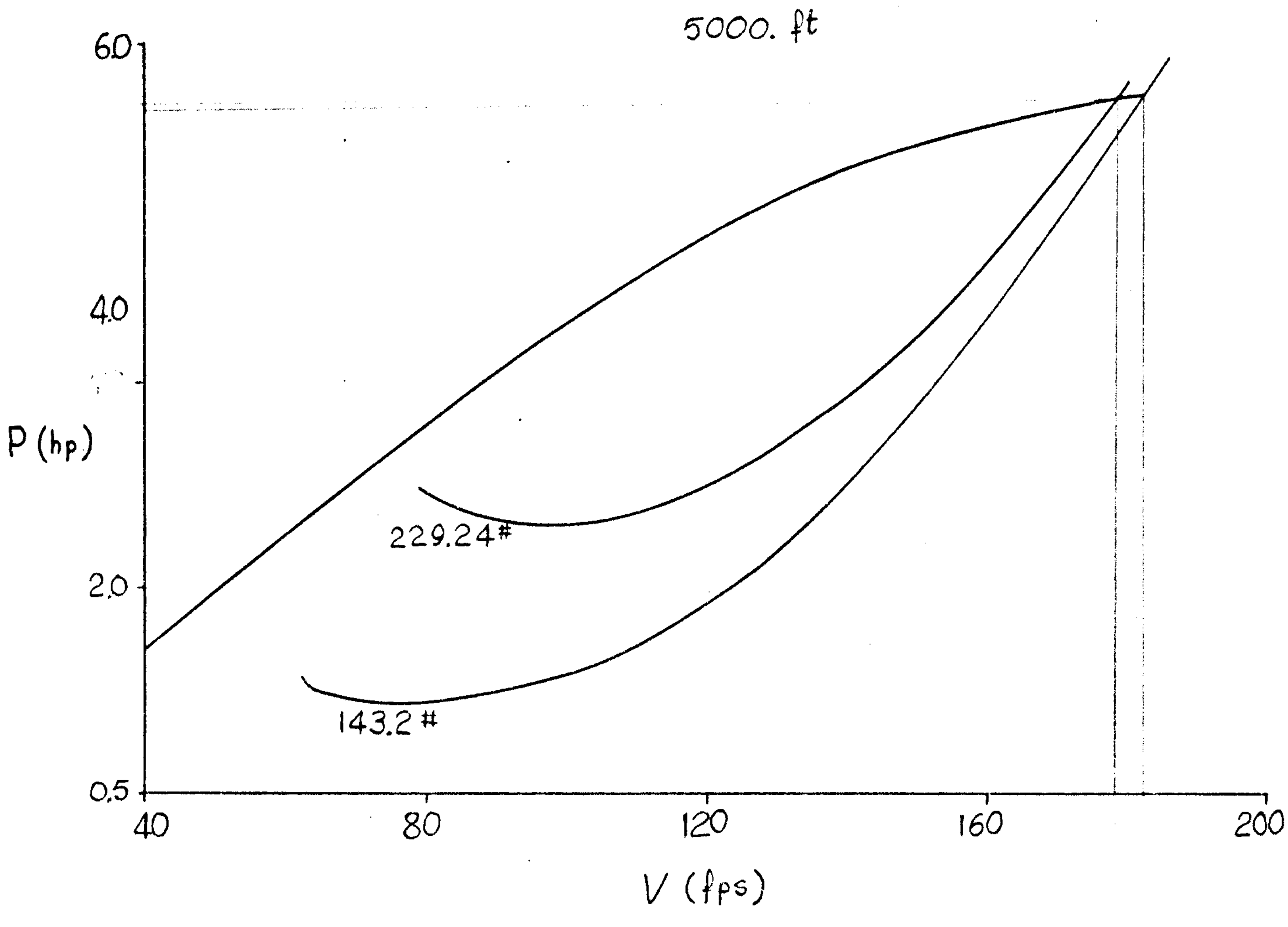


Fig 3.3-2

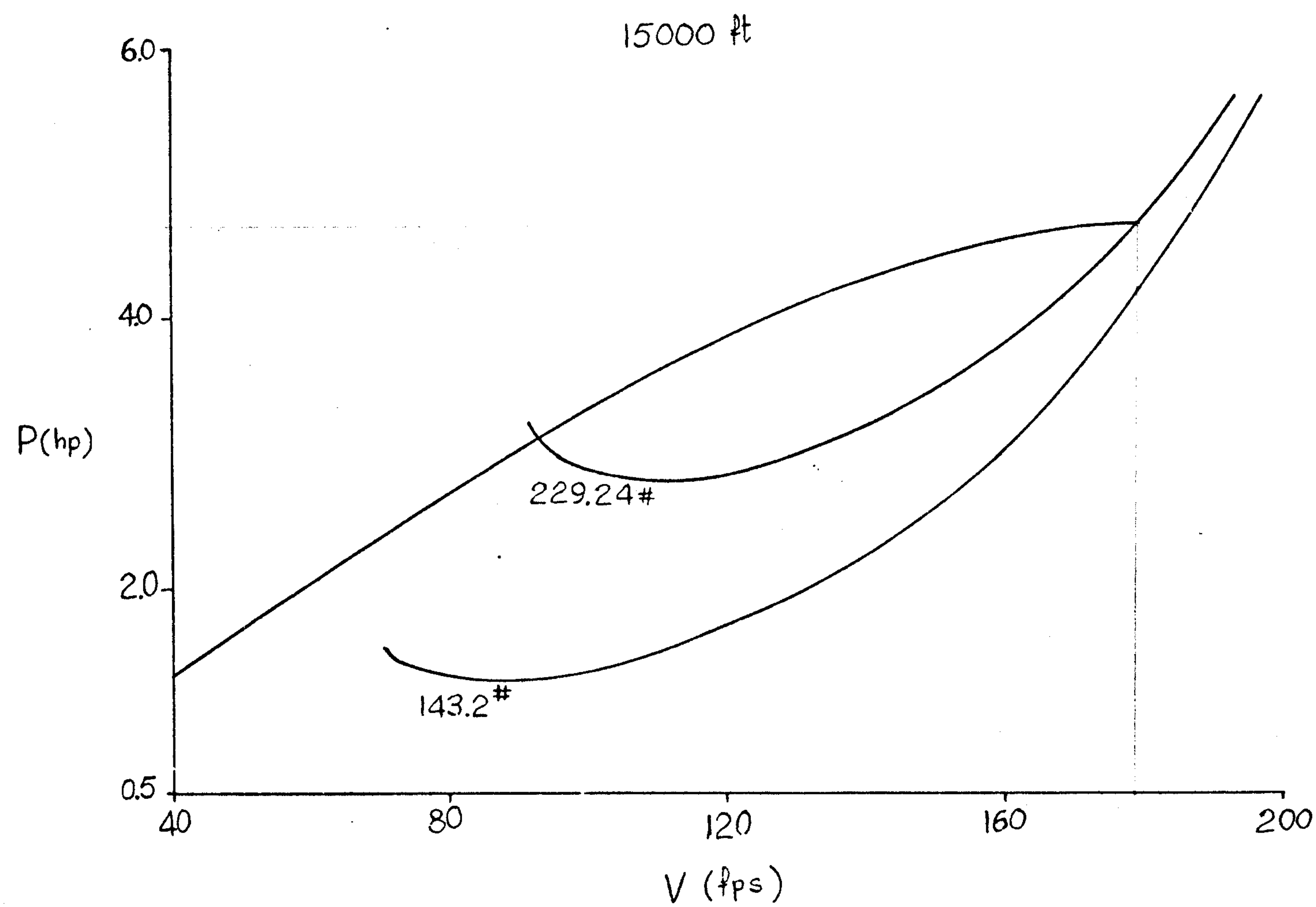


Fig 3.3-3

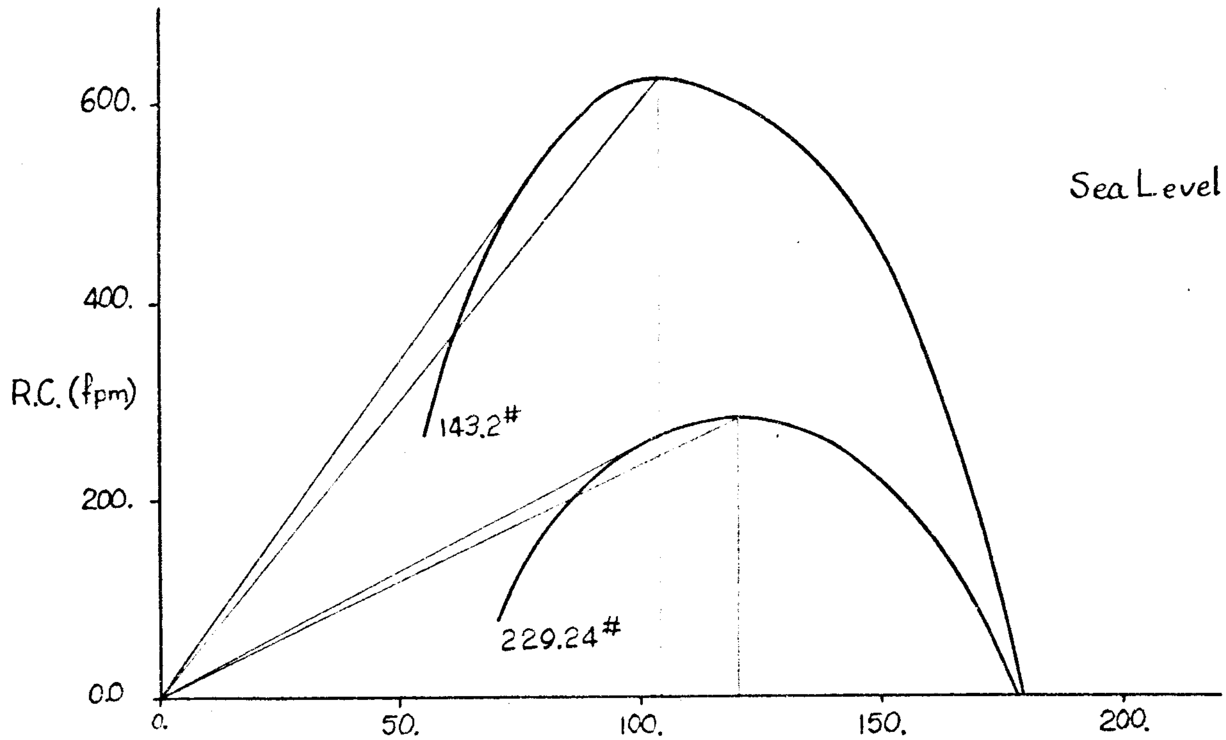


Fig 3.3_4

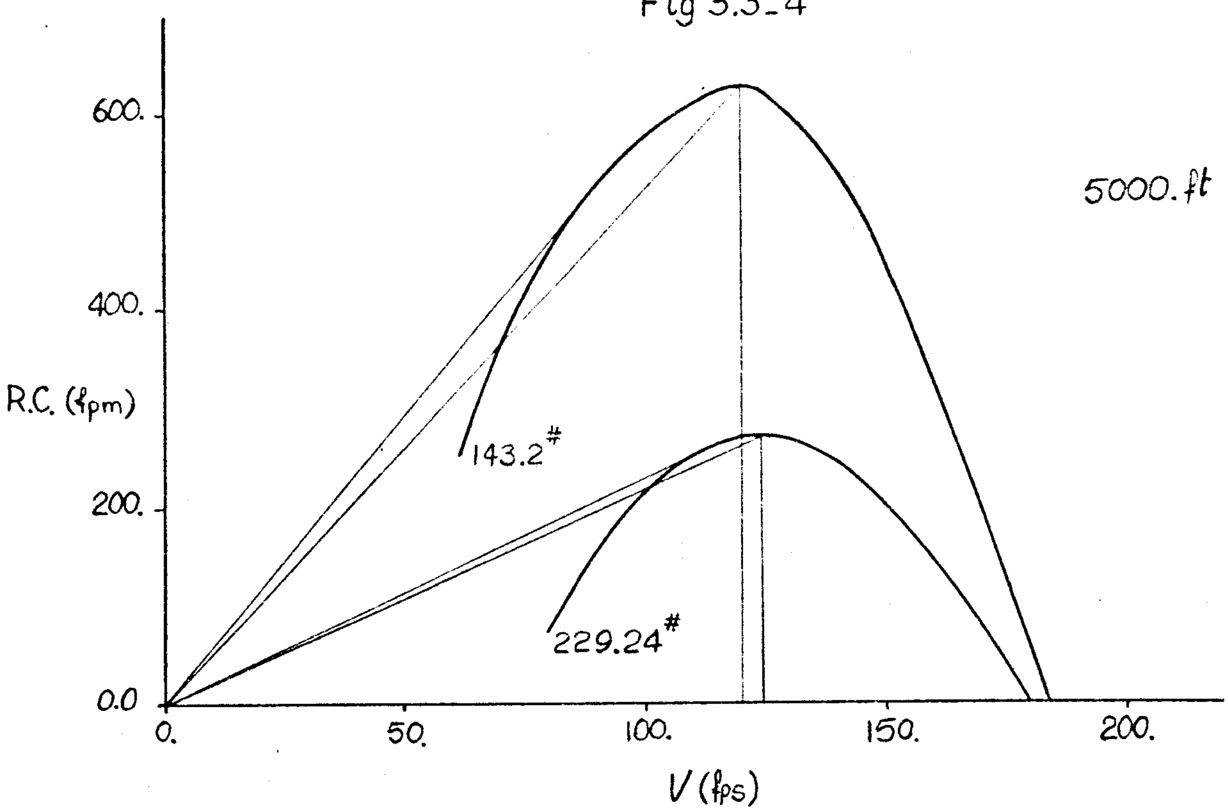


Fig 3.3_5

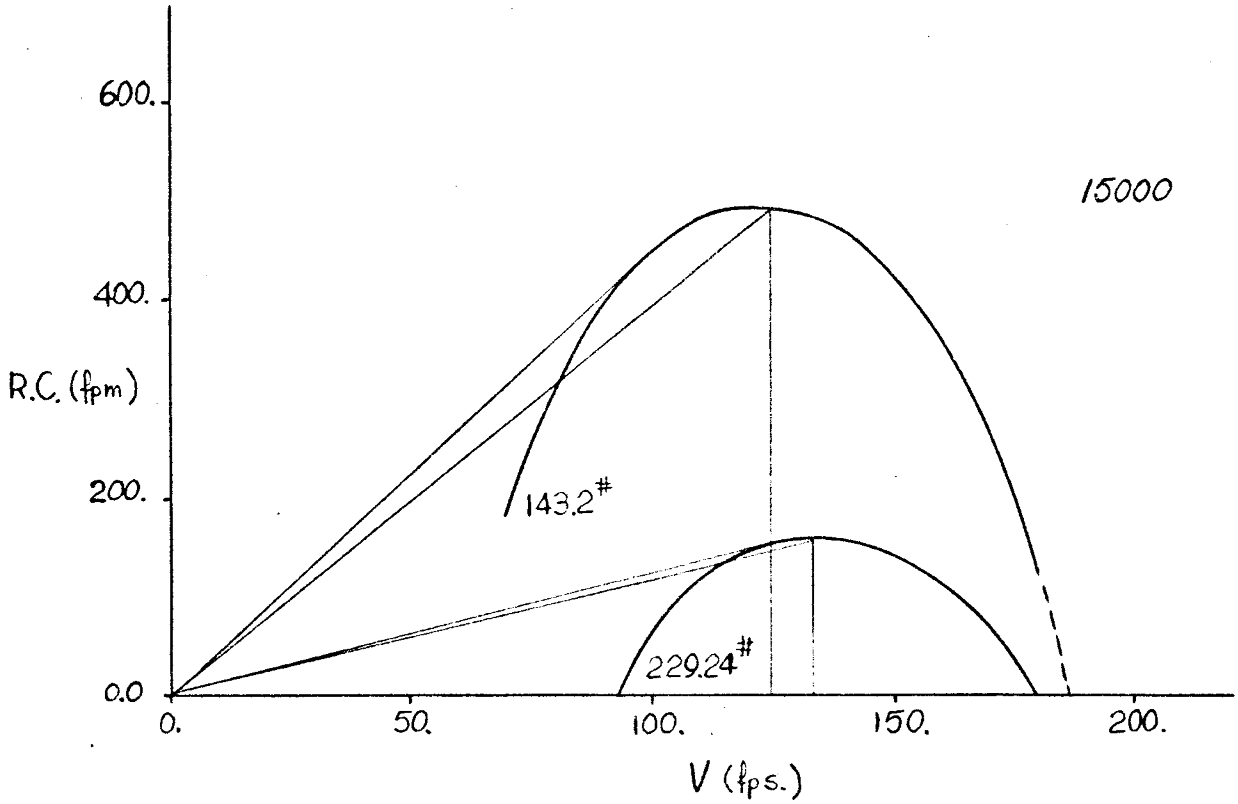


Fig 3.3-6

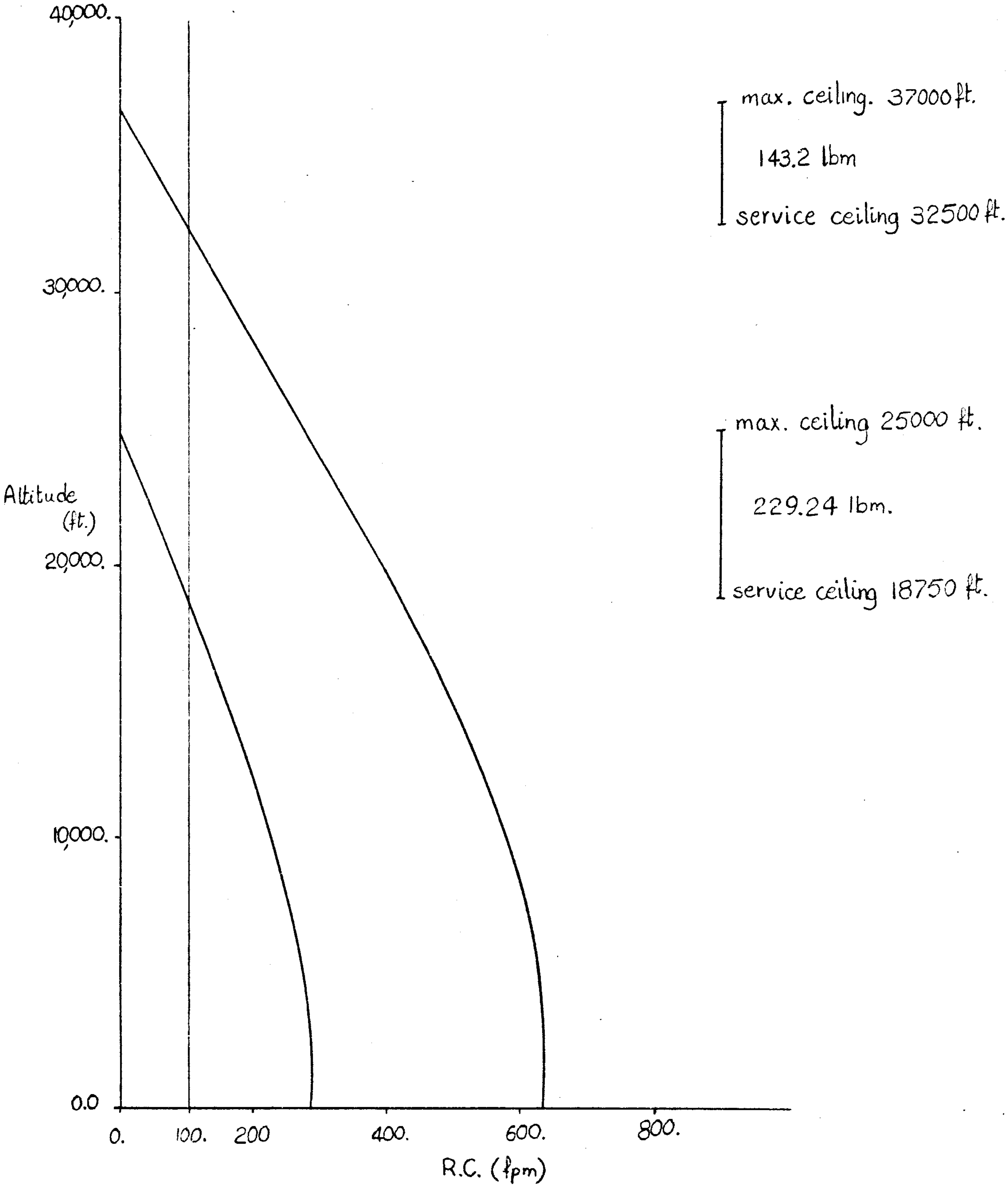
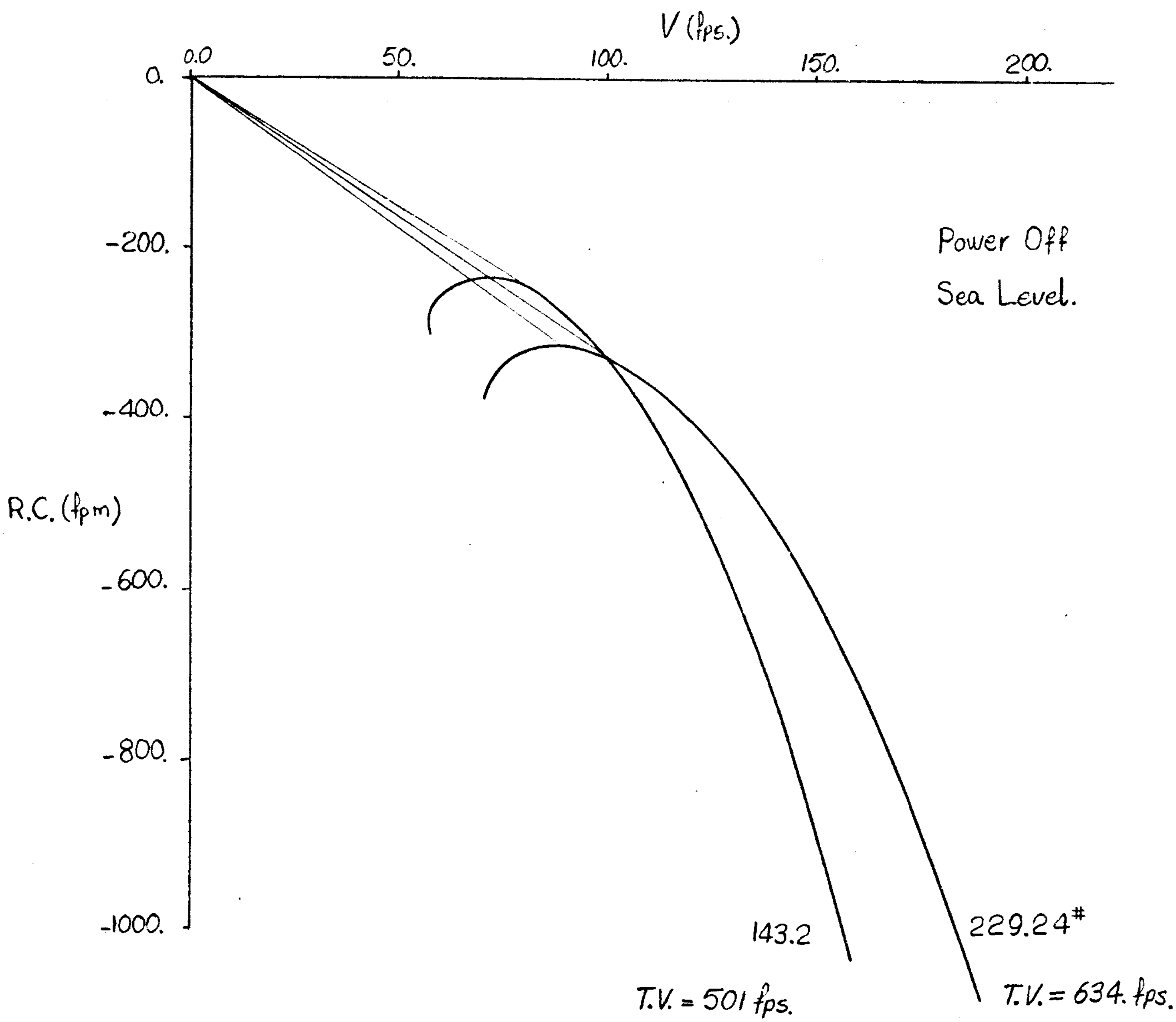

40,000.
30,000.
20,000.
10,000.
0.0
Altitude (ft.)
0.
100.
200
400.
600.
800.
R.C. (fpm)
max. ceiling. 37000 ft.
143.2 lbm
service ceiling 32500 ft.
max. ceiling 25000 ft.
229.24 lbm.
service ceiling 18750 ft.

Fig 3.3-7



V (fps.)
0.0
50.
100.
150.
200.
0.
-200.
-400.
R.C. (fpm)
-600.
-800.
-1000.
Power Off
Sea Level.
143.2
229.24#
T.V. = 501 fps.
T.V. = 634. fps.

Fig 3.3-8

$$E = 778 \frac{\eta}{c} \frac{C_L^{3/2}}{C_D} \sqrt{PS} \left(\frac{1}{\sqrt{W_1}} - \frac{1}{\sqrt{W_o}} \right)$$

where E = endurance time (hr.)

C = specific fuel consumption ($\frac{lbm}{hp} \cdot hr.$)

η = propulsor efficiency

W_1 = final total airplane weight (lbm)

W_o = initial total airplane weight (lbm)

Table 3.3-1 presents the endurance times calculated for different missions according to this equation.

3.3.5 Steady State Level Flight Range

For determining the maximum range of the plane the Breguet formula, suggested by Domash[4] was used which is:

$$R = 375 \frac{\eta}{c} \frac{C_L}{C_D} \ln\left(\frac{W_o}{W_1}\right)$$

where range is expressed in miles. The maximum ranges of the vehicle calculated through this relation are given in Table 3.3-1.

3.3.6 Load Factor Analysis

As it has been mentioned throughout, the manueverability of this vehicle was one of the major design objectives. In fact, this would be one of the determining factors in survivability of the entire system. As a direct result, major emphasis was placed on load factor capability of the system along with its efficiency.

It was proven by the Structure Group that the airframe could withstand a maximum of +12 g's and -8 g's loads. Later, the maximum operational load factors were constrained to +8 g's and -5 g's, allowing for 50% margin of error.

Aerodynamically, the vehicle was designed to allow up to +7.5 g's at an estimated maximum level flight speed of 220. fps. Roskam[1]

Mission	total take-off weight	total-fuel weight	useful fuel† weight	Endurance 100%	Endurance* 85%	Range (mile) 100%	Range (mile)* 85%
A	221.2	78.0	69.0	25.72 hr.	21.86 hr.	1714.	1457.
B	229.24	78.0	69.0	24.28 hr.	20.64 hr.	1636.	1391.

Comments: $(C_L/C_D)_A = 18.49$; $(C_L^{3/2}/C_D)_A = 15.695$

$(C_L/C_D)_B = 18.445$; $(C_L^{3/2}/C_D)_B = 15.835$

altitude 5000 ft.; $\eta = 0.4513$; SFC = $0.7 \frac{\text{lbm}}{\text{hp}}$ hr.;

† fuel allocated for mission = 11.5 gallons

* assuming 85% efficient flight

Table 3.3-1

suggested that during a level turn:

$$\phi = \cos^{-1}\frac{1}{n}$$

where ϕ = bank angle

n = load factor.

Therefore, a maximum of +7.5 g's during a steady state level turn could imply bank angles as high as 82^{o}. Such bank angles during a level turn would allow a radius of turn as low as 200 ft. Table 3.3-2 shows the relation between load factors and radii of turn during different flight configurations.

3.3.7 Take-Off

The take-off hydrolic systems could guarantee a take-off speed of 73 fps for a take-off gross weight of 210 lbm. Therefore, to stay above 1.2 times the stall speed, a flap deflection of only 20^{o} would be required. Such a flap deflection would insure that the vehicle would leave the take-off ramp out of the region of reverse control. It would also allow a rate of climb in the order of approximately 60 fpm. Table 3.3-3 shows the suggested degrees of flap deflection for different take-off gross weights.

3.3.8 Landing

Due to very low landing speed capabilities of the plane, different landing procedures had to be employed during strong gusts and calm weather. The major difference between the two procedures would be the flap setting in order to insure go around capability in turbulent weather. The detailed analysis of the two procedures are shown in Appendix A.3-19.

Mission	Weight (lbm)	Load Factor (g's)	Bank Angle (degrees)	Lift Coeff.	radius of turn (ft.)
	max. 221.2	3	70.53	0.529	532.
A		8	82.82	1.411*	190.
	min. 143.2	3	70.53	0.342	532.
		8	82.82	0.913	190.
	max. 229.24	3	70.53	0.548	532.
B		8	82.82	1.462*	190.
	min. 151.24	3	70.53	0.362	532.
		8	82.82	0.964	190.
C	210. at 312 fps	12	85.2	1.0	252.5

Altitude = 5000 ft; $\phi = \cos^{-1}\frac{1}{n}$; $R = \frac{V^2}{g\tan\phi}$; $C_L = \frac{nW}{qS}$; V = 220 fps

* 20° flap angle required.

Table 3.3-2

Mission	Max. T.O.W.	1.2 V_{stall}	$V_{T.O.}$	Best Flap	R.C.
A	221.2(lbm)	68.4(fps)	69.3(fps)	20^o	67.1(fpm)
B	229.2(lbm)	70.92(fps)	66.9(fps)	20^o	21.6(fpm)

At Standard Sea Level Altitude only.

Table 3.3.3

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CHAPTER IV

PROPULSION AND COSTS

Table of Contents

4.1 Introduction

Before delving into the analysis leading to the final propulsion and power generation system, as well as the costs for the entire system, a word about RPV production and marketing is in order. As in any industry created by military needs, it is often difficult to ascertain exactly what the need is (mission scenerio), who the contractors are and what their relationship is to the final product. The multitude of missions and performance requirements for RPVs and drones extremely complicate matters.

It is felt that without untangling some of the knotty problems presented by contractors, subcontractors, military funded subsystem development, and basic research, the judgements leading to the final propulsion system may seem ethereal. These decisions are best viewed in light of real world demands. However, where present technology or state of the art appears inadequate, theoretical analysis of a radical system or device may be the only way to the desired performance. Conservative estimates and projections of experimental work or theoretical development will be used whenever possible.

This chapter is divided such that a succinct analysis of each area may be presented. Interrelated areas are sequenced so that the determining factor is first. For example, propulsive efficiency and noise are considered the prime factors in propulsion. However, the propulsion consists of a powerplant which provides power to some sort of propulsor and the electrical system as well. This means that the powerplant may be considered foremost, since it consumes the fuel and produces a substantial amount of noise. But seperating powerplant and propulsor is not always possible, as in the case of turbofan engines. Consideration of all powerplants must be included in a unified analysis, and this must precede propulsor and power generation analysis. For these reasons, also, the propulsion system performance is summed up at the end of the propulsor analysis.

4.2 Propulsion

4.2.1 Powerplant

4.2.1.1 Introduction

The powerplant is one of the most critical areas of this RPV system. A quick review of the performance goals confirms this. This RPV should have a range of greater than 100 miles under some conditions, noise of less than 20 dB at 5000 feet altitude (inaudible), maximum speed of about 150 miles per hour, and a flight endurance of at least 20 hours. All of these things must be achieved in a vehicle massing about 200 pounds mass, less than 15 feet long, and 15 feet wide. Of particular interest in powerplant design is the high weight and the wingspan restrictions, which imply a low aerodynamic efficiency (moderate to high drag). Whatever the powerplant uses as fuel must be a reasonable amount of the overall weight, and that means a high efficiency powerplant.

Performance of the vehicle places very important restrictions on the powerplant, but system integration and configuration also place requirements and restrictions on this subsystem. A summary of requirements are:

1. Must supply adequate power for on-board electrical systems and dash speed requirement.

2. Weight must not be prohibitive (including propulsor).

3. Must be small and fit in fuselage with minimum modification to fuselage.

4. Must be reliable.

5. Fuel weight must be minimal.

6. Must be easy to quiet.

7. Must fit in rear of fuselage (Missions requirements: sensor pod is in forward fuselage. Aerodynamics requirement: minimum drag).

8. Fuel must be located near the center of gravity (Stability and control requirement).

9. Vibration should be low (Missions requirement: stability of optical payloads. Structures requirement: minimum fatique).

10. Performance must be insensitive to variation in operating conditions.

4.2.2.2 Choice of Powerplant - General

<u>Systems</u>

A. Rocket

B. Turbojet

C. Turbofan

D. Turboprop

E. Propeller and i.c. engine

F. Ducted propeller and i.c. engine

G. Pulsejet

<u>RPV System Considerations</u>

V = 110 feet/sec

Wing Area = 25.92 square feet

Thrust = 15 pounds foree

Required Power = 3 horsepower

Gross Weight = 210 pounds mass

Endurance = 20 hours

<u>Approach</u>

Optimum powerplant selection will be on the process of elimination. Locating primary goals and considering only those systems which meet the goals should quickly accomplish the selection.

On this basis the endurance goal of twenty hours at loiter is considered first. If the required weight of pwerplant, propulsor, and fuel is too high, then that powerplant and/or propulsion is unacceptable.

A. Rocket

According to Shepard[1] the ideal propulsive efficiency for rockets is

$$\eta_p = \frac{2r}{1+r^2}, \quad r = \frac{U}{V_j}$$

where U is the forward speed and V_j is the jet velocity from the rocket nozzle. Fuel consumption was found for the ideal case (V_j of 17500 fps), the typical case (V_j of 10000 fps), and for a low speed optimized rocket (V_j of 1000 fps). Fuel used for the analysis is a hydrogen and oxygen stochiometric mixture. It must be noted at this point that the low speed rocket suffers from large losses unaccounted for in this analysis. Some of these losses are a large drag penalty associated with a large nozzle and flow losses in the nozzle. The resulting fuel required for 20 hour endurance at loiter condition is:

V_j (fps)	Efficiency	Fuel (lb. mass)
17500	0.0126	1995.6
10000	0.022	1133.74
1000	0.2174	115.67

B. Turbojet

According to Shepard[1] the typical turbojet thrust specific fuel consumption (TSFC) is about 0.6 to 0.8 pounds mass fuel per pound force - hour. At the best TSFC the mass of fuel required is 180 pounds mass.

1-64

C. Turbofan

According to Corning[2] the turbofan can be described in terms of an equivalent turbojet. For a reasonable bypass ratio of 4, the equivalent thrust increase (turbofan over turbojet) is about 2. On this basis and the analysis of part B, the fuel mass required is 90 pounds mass.

D. Turboprop

According to data by Oetting[3] for a turboprop of these reasonable parameters: $\eta_C = 0.85 \quad \eta_T = 0.90 \quad \eta_B = 0.96$

$$\eta_{Prop} = 0.85 \quad \eta_N = 0.97 \quad h^* = 18900\ \frac{BTU}{lbm}$$

Turbine inlet temperature - - 2400° R

specific fuel consumption is 0.48 pounds mass fuel per horsepower-hour. Over 20 hours a fuel consumption of 28.8 pounds mass is incurred.

E. Propeller and Internal Combustion Engine

Assuming a reasonable propeller efficiency of 85%, 3.53 shaft horsepower is required from the ingine. Specific fuel consumption for typical internal combustion engines is 0.6 to 0.8 pounds mass fuel per horsepower - hour. Therefore at the best SFC 42.36 pounds mass of fuel is required.

F. Ducted Propeller and Internal Combustion Engine

Ducted propellers suffer from the same sort of high mass flow rate but low cross sectional area efficiency problems that turbofans do. The duct adds thrust, but it does so while accellerating the flow through the duct. The fan in the duct may operate at 100% efficiency in the duct flow (in actuality less), but when the fan performance is referenced to the conditions outside the duct and added to the shroud performance, total efficiency may actually be less than an open propeller. Taking a propulsor efficiency of 50%, the required mass of fuel is 72 pounds mass.

G. Pulsejet

Pulsejets produce thrust at about 2 to 4 pounds mass thrust per pound force - hour[4], requiring at least 600 pounds mass of fuel for 20 hours at loiter.

This basic analysis is summarized in Figure 2.1 . All of these analysis are very subject to scaling limitations. That is, almost all of the figures used are for full - scale powerplants and propulsors which are not nearly as efficient on a smaller scale. Therefore, since none of the systems would perform better on a smaller scale, the four systems requiring about 50% of the gross weight or more may be thrown out. The systems eliminated by this basic analysis are rockets, turbojets, turbofans, and pulsejets.

In the remaining systems, turboprops and internal combustion engine with propeller or ducted fan, only the turboprop suffers greatly from scaling limitations. This is effectively shown in Figure 2.2 from a paper by Oprecht[5] concerning small gas turbine engines.

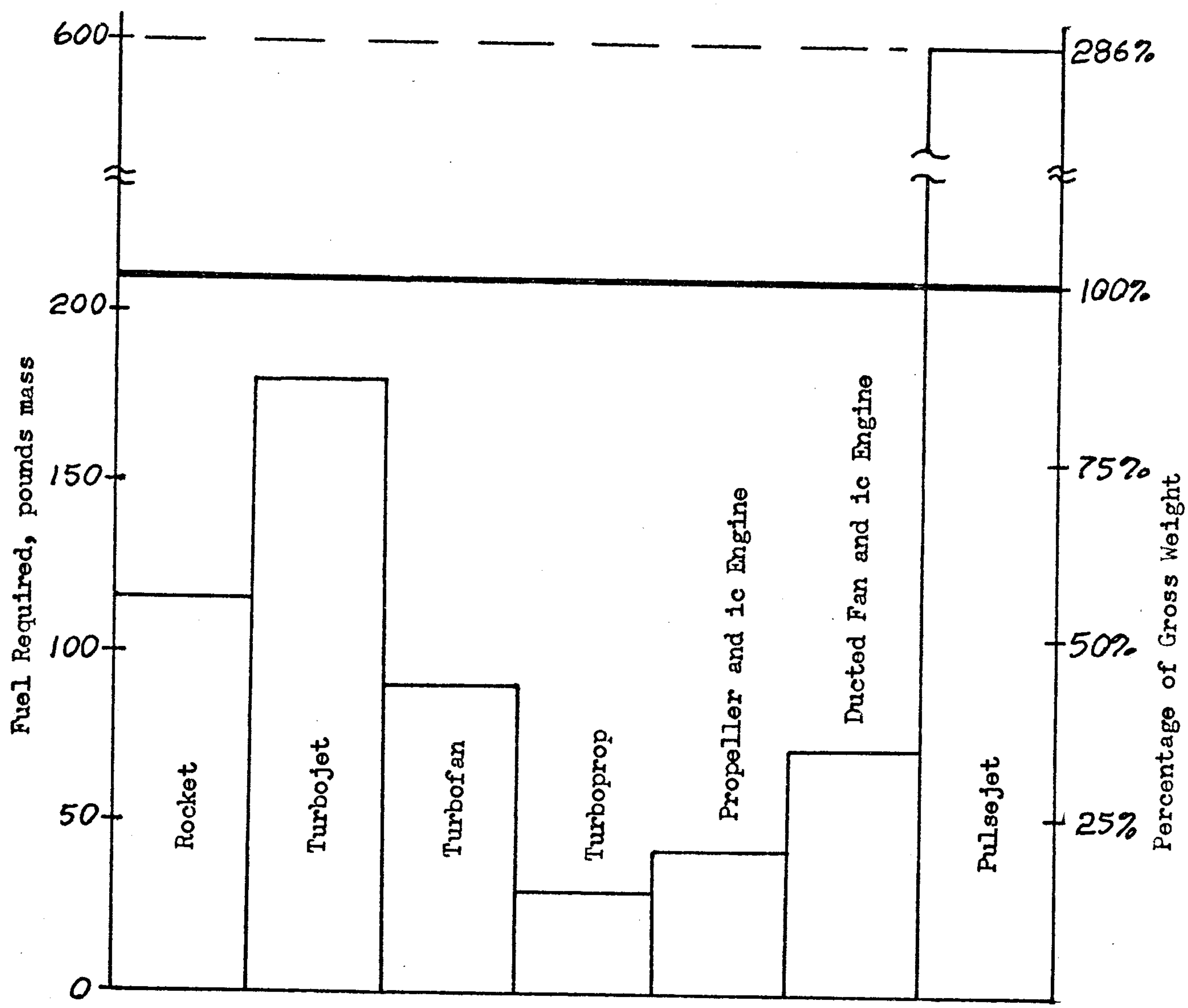


Figure 2.1 Comparison of Idealized Powerplant/Propulsor Systems

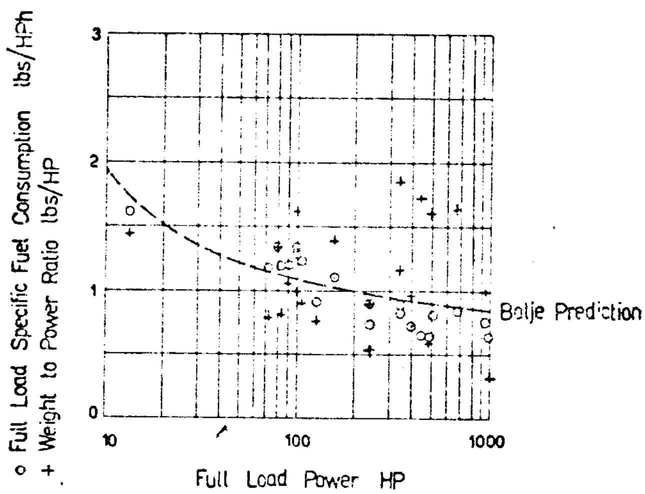


Figure 2.2 Effect of Scaling on Turboshaft Engines

A more recent paper by Heitmann[6] reveals that while the performance goal of a 100 horsepower gas turbine design was 0.7 pounds mass fuel per horsepower - hour, the actual SFC was much higher. Therefore it is unlikely that a turboprop would provide the necessary fuel efficency.

The internal combustion engine is left as the best powerplant to use. In conjunction with a propeller or shrouded fan, it can provide the necessary endurance without an excessive or impossible weight penalty.

Acoustic considerations are also a part of powerplant choice. Fortunately, of all of the powerplants and propulsors considered, the internal combustion engine is the easiest to quiet. This will be shown in Section 3.

4.2.2.3 Choice of Engine

The first necessity in choosing an internal combustion engine (hereinafter referred to as an engine) is to decide what sort of maximum horsepower is required. Preliminary estimates of aerodynamic performance indicated that about 17 shaft horsepower would be required at 150 mile per hour dash. Since the dash speed requirement was considered a goal little related to the missions, engines for RPVs in the ten to twenty horsepower range were considered. Additionally, smaller engines were explored for multiple - engine installations (tandem shaft). Power curves for the eight engines studied are shown in Figure 2.3 . Actual performance is best illustrated in Figure 2.4 .

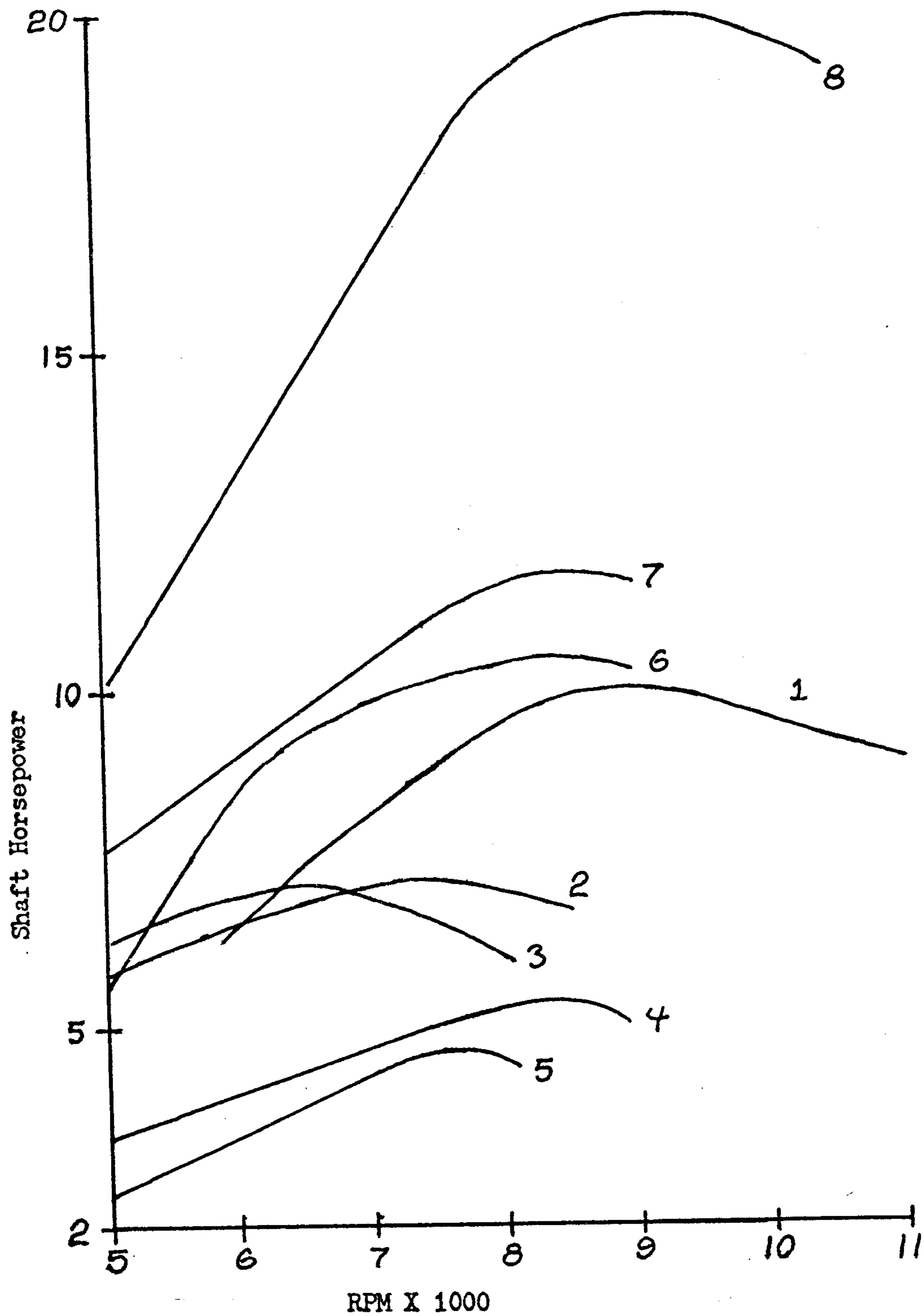


Figure 2.3 Engine Performance

Figure 2.4 Study Engines

Engine #	Model	Performance
1	McCulloch 91 B/1 stock	10 hp at 9000 RPM SFC unavailable
2	McCulloch 91 B/1 RPV mod., gas	7.2 hp at 7500 RPM 0.87 lb/hp-hr
3	McCulloch 91 B/1 RPV mod., glow	7.2 hp at 6500 RPM SFC unavailable
4	Kolbo Korp. RPV engine	5.2 hp at 8300 RPM 3.7 lb/hp-hr
5	Sakert-Riggs RPV engine	4.6 hp at 8000 RPM 3.7 lb/hp-hr
6	McCulloch 101 A/A RPV mod., gas	11.6 hp at 8500 RPM 0.74 lb/hp-hr
7	DH Dyad 160 RPV engine	10.8 hp at 8500 RPM 0.78 lb/hp-hr
8	Teledyne Conti-nental Moters RPV engine	20+ hp 0.7 lb/hp-hr

Performance of engines 1 through 5 were obtained from a paper by T. R. Small[7]. Data for engines 6 and 8 were obtained from Mr. Russ Stanton at the U.S. Army AVRADCOM (Avionics Research And Development Command)[8], as well as from other sources[7,9,10]. The D H Dyad 160 (engine 7) data was from Mr. John Hunton[11], Senior Mechanical Engineer for the Special Projects Department of the Melpar Division of E - Systems, Inc.

At this point it is best to explain how RPV engines are produced and what has governed their specifications. RPVs are designed to be small and light, yet they have a substantial power requirement. At this state of the art in small engines, only 2 cycle powerplants are light and powerful enough to use. Often the RPV payload consists of delicate electronic and radio equipment, which means that the ignition must be shielded if electrical (not diesel or glow plug). The payload is often an optical device (tv camera or photocamera) with a zoom lens sensitive to vibration. Therefore the engine must be low in vibration, particularly in the low frequencies. This usually means a twin opposed cylinder configuration.

Production of RPV engines involves many subcontractors. Cost of the basic engines to the RPV company is often so low that they perform a good deal of remanufacturing on the engine[12]. The base engine, in turn, consists of cylinders, pistons, and other parts obtained from a chain saw manufacturer or a go - cart engine manufacturer.

These are modified and installed in a crankcase designed by the base engine supplier. Carburators are a seperate part supplied by a seperate company. Further complicating the matter is the fact that the RPV manufacturers, after developing their final engine system from the base engine plus carburator(s), will provide this refined engine to other RPV companies.

From the basic analysis, it was seen that required fuel weight for the loiter mission was a primary factor in powerplant and propulsor selection. A consideration of specific fuel consumption is also necessary for the engine analysis. In the basic analysis it was assumed that the engine could operate at 0.6 pounds mass fuel per horsepower - hour. None of the engines considered provide that sort of SFC, but four of them can operate below 0.9 pounds mass fuel per horsepower - hour. These four are models 2, 6, 7, and 8. The others operate at higher SFC, as high as 3.7 pounds mass fuel per horsepower - hour.

Some discussions of the engines themselves are in order. Engine 2 is a single - cylinder, 2 cycle McCulloch go-cart engine supplied stock from the manufacturer until recently. During the summer of 1977, McCulloch stopped production and distribution of their large go - cart engines, fearing litigation from hang glider pilots who had gotten injured while flying gliders powered by converted McCullochs. The discontinued engines were the 91 B/1 and the 101 A/A. However, Horstmann Manufacturing Company, Inc. bought distribution rights for these engines and presently contracts for limited production (about

6000 units) on a sporadic basis[13,11]. The 91 B/1 weighs 7.5 pounds stripped of cooling and ignition, and only provides low SFC with carburator modifications. These modifications (as performed by APL) also decrease the maximum horsepower.

Engine 6 is the McCulloch 101 A/A as re-engineered by Lockheed for the Army Aquila RPV[8,9]. Weight of this powerplant is only 11 ounces more than the 91 B/1, and in the stock configuration it will provide over 15 shp. This is also a single-cylinder powerplant, of almost the exact size as the 91 B/1. Total weight is 8.2 pounds.

Engine 7 is an engine with a long lineage. The basic engine supplier is DH Enterprises. This engine has been tested and refined by the Melpar Division of E - Systems, Inc. and they supply the complete engine system. The system includes mounting and muffler. Power curves for this engine is with the muffler, as well, while all the rest are unmuffled. The total engine system weighs 13.1 pounds and is of the twin opposed cylinder configuration.

Engine 8 is an engine currently under development by Teledyne Continental Motors. Essentially 2 Homelite Model 270 chain saw engines joined about an integral 900 watt alternator, this engine has two seperate crankshafts geared together with the alternator. Performance of Figure 2.3 is only projected. Weight of the 20+ horsepower engine is 21.5 pounds, including the alternator. This engine is technically a twin opposed cylinder configuration.

Obviously, for the dash speed requirement a high maximum horsepower is desirable. At the low end of the scale, engine 2 must be eliminated since engine 6 provides more than four more horsepower with less than a pound of extra weight. Volume of the fuselage is not sacrificed as the engines are basically the same size.

At the top end, a twenty horsepower engine is greatly desirable, especially with the SFC available and the fairly low weight penalty. The high weight penalty and great width could be reduced by using a common crankshaft and eliminating the alternator. However the engine is still in development, so engine 8 is unacceptable at the present.

This leaves the two engines of nearly identical performance. Their performance is presented in Figure 2.5 and 2.6 . Engine 7 is currently in production with proven performance at most moderate altitudes (15000 feet or less). Performance of engine 6 suffers with increasing altitude. A reasonable weight comparison should take into account that engine 6 weighs 13.1 pounds with all accessories. Figure 2.7 illustrates the engine and the weights of the system components. Basically, engines 6 and 7 weigh the same. Model 7 is also a twin opposed cylinder configuration while engine 6 is single cylinder. Figure 2.8 illustrates the basic ~~xx~~ 101 A/A as produced by McCulloch and the part throttle performance as the engine was configured for the Lockheed Aquila. This shows the main advantage of the number 6 engine,

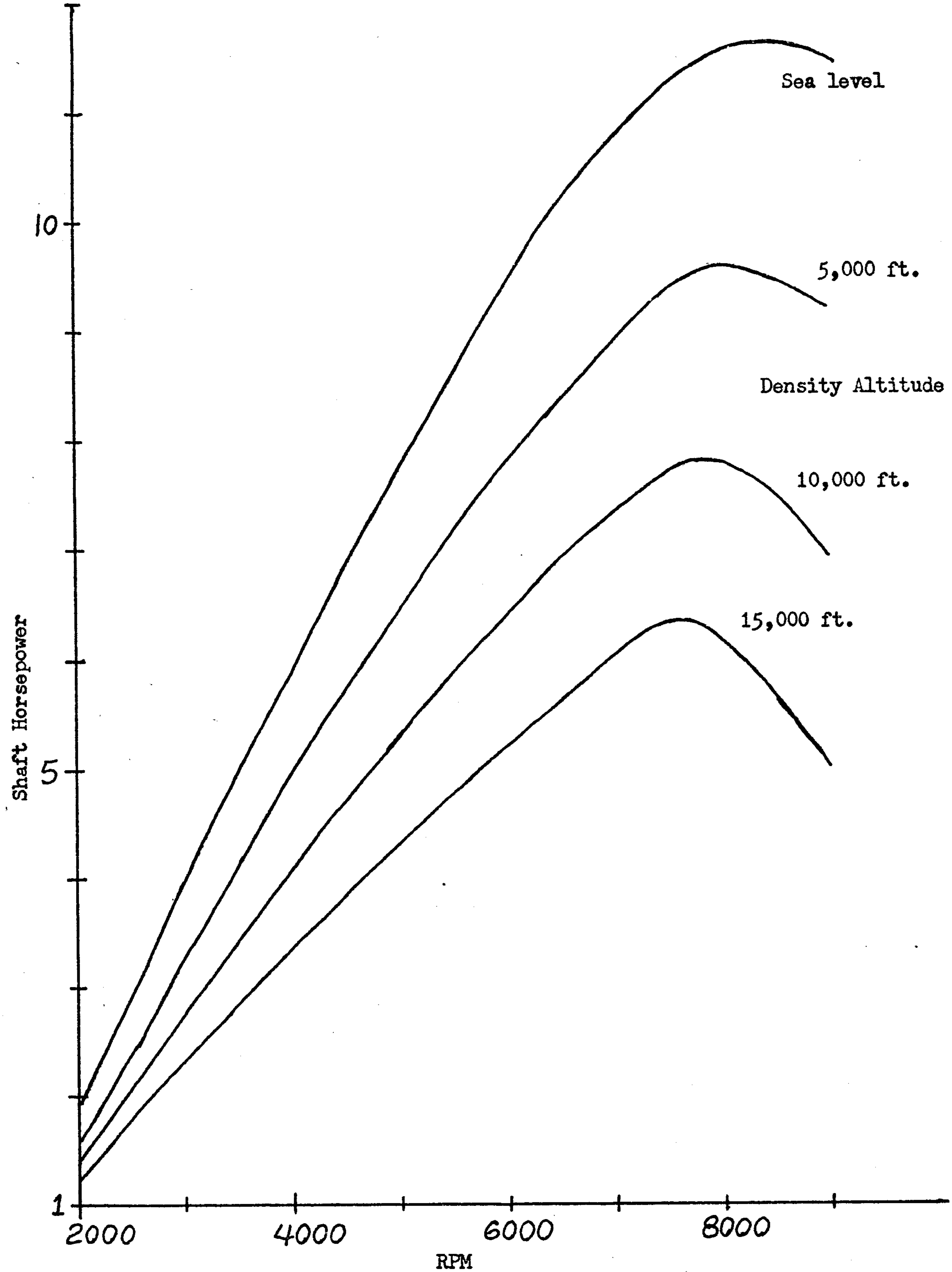


Figure 2.5 Performance of the MC 101 for the Army Aquila

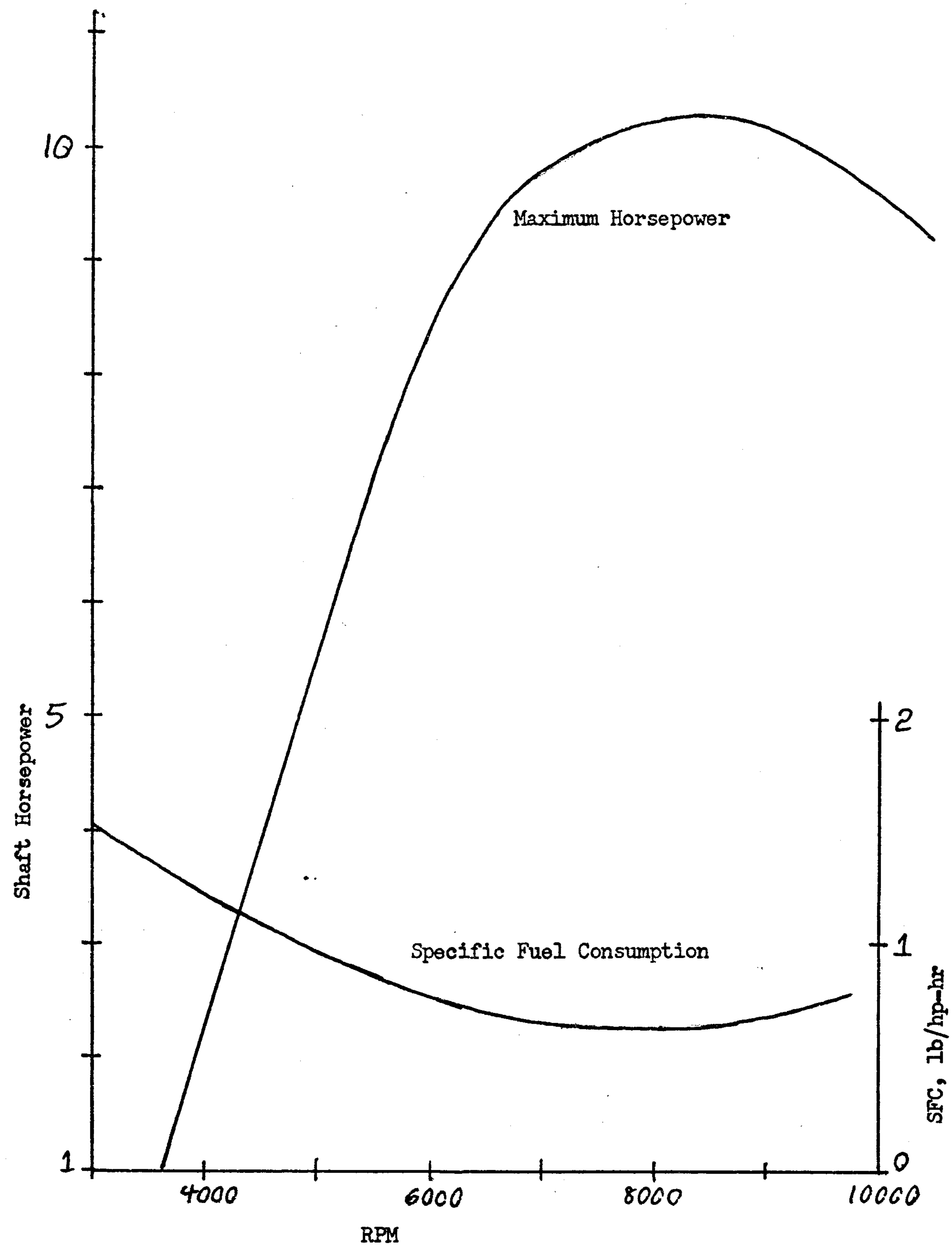


Figure 2.6 Performance of the DH Dyad 160 from E - Systems

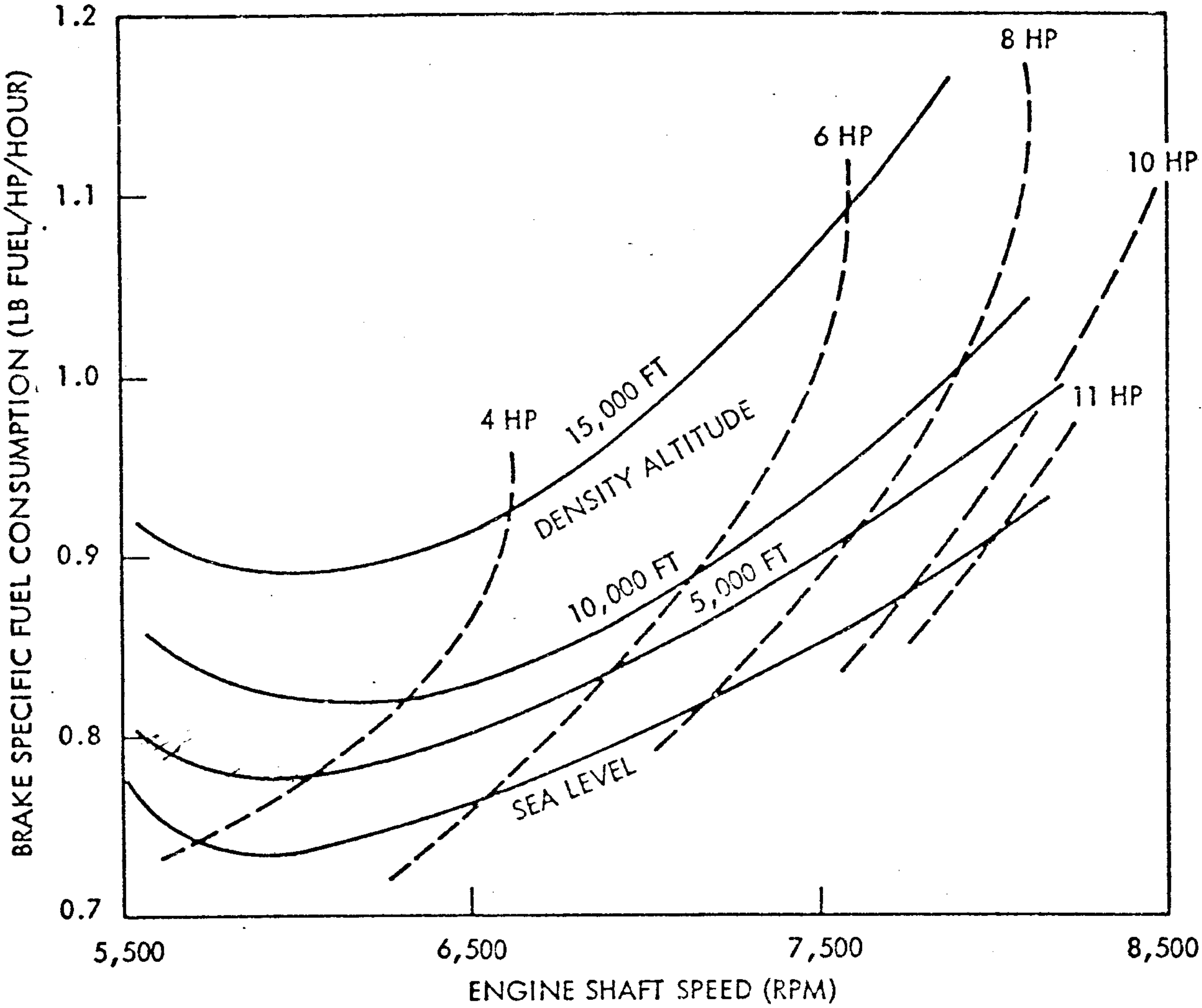
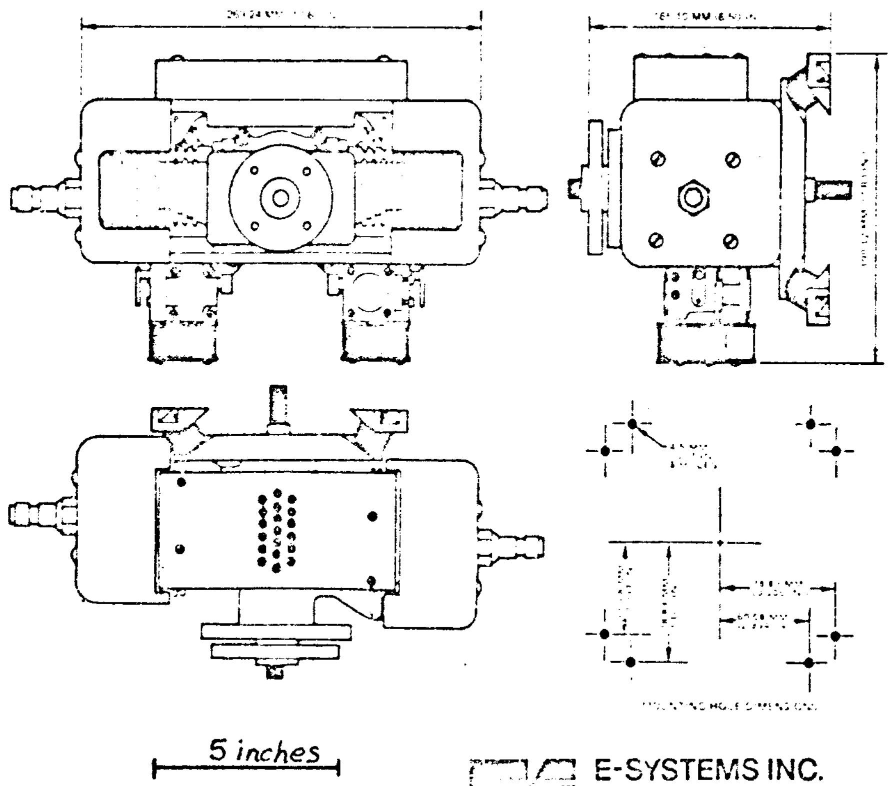


Figure 2.8 Part Throttle Performance of the MC 101 in the Army Aquila



E-SYSTEMS INC.
Melpar Division

TWIN E-160cc ENGINE WEIGHT	LBS.	KG
BASIC ENGINE (with carburetor mount, ignition pick-up, prop. nut and bolts)	8.2	3.72
SPARK PLUGS (shielded, resistor type)	.4	0.18
MUFFLER ASSEMBLY	1.1	0.50
COOLING BAFFLES	.5	0.23
CARBURETOR AND AIR CLEANER (Dual)	.4	0.18
SHIELDED IGNITION SYSTEM (with leads)	1.6	0.72
MOTOR MOUNT	.9	0.41
TOTAL SYSTEM WEIGHT	13.1	5.94

Figure 2.9 Dimensions and Weights of the DH Dyad 160 Engine System as Supplied by E - Systems

a lower SFC. Finally, in the area of cost, the number 6 engine cost Lockheed about $1000 per unit due to high quality control costs. The MC 101 A/A costs about $160 through Horstmann, but Lockheed found that out of 70 engines only 36 usable powerplants could be assembled that would meet performance specifications. The complete engine 7 system, on the other hand, costs $1200.

On this basis the Model 7 engine, the DH Dyad 160 as supplied by E - Systems, was chosen.

4.2.2.4 Engine Subsystems

There are several subsystems for the powerplant which must be considered. These are:

A. Vibration isolating engine mount

B. Muffler

C. Cooling

D. Fuel Tanks

E. Ignition

A. The engine mount is a standard item supplied with the engine. It is illustrated in Figure 2.9 .

B. The muffler will be discussed in Section 4.2.3 . The E - Systems muffler will not be used, rather an in - house muffler will be designed. Performance of the muffler will be estimated in the Acoustics section, with the horsepower loss taken as equal to the E - Systems muffler. The muffler used on this RPV will be mostly of fiberglas construction to keep weight down to a pound or less.

C. Cooling of the engine must be adequate for ground run-up, yet not overcool the engine in flight. Cooling of the electronics is also required. Calculations have shown (see Appendix A.4.2) that 27.5 square inches of intake area provides sufficient air for cooling all systems. The intake area must be distributed properly in several intakes along the airframe, with about fifteen square inches near the nose and 12.5 square inches near the engine.

The intakes will be in the form of NACA airscoops for minimum drag. Pressure losses through the airframe are removed by a blower coaxial with the fan. Forced air cooling is required due to acoustic requirements (enclosed engine). The power loss due to the blower is negligable (on the order of 0.1 horsepower or less). A total system schematic is presented in Figure 2.10 . Thermostatic control of engine compartment temperature through the the flow control helps provide adequate cooling under all loads. The thermal control of the engine compartment also prevents formation of carburator ice.

D. Fuel tank size is not considerable for the amount of fuel capacity required. Preliminary estimations of fuel requirements set the capacity to 78 pounds or 13 gallons. Due to stability and control considerations, the tanks are placed near the center of gravity of the aircraft. The tanks are sized as shown in Figure 2.11 . The four wing tanks and the fuselage tank are molded out of plastic and are intended to support the internal plastic fuel bags, which contain the fuel. Fuel is taken up by a plastic pipe extending into the tank. These pipes have drilled walls so that, as fuel is drained, the flow will not be blocked by the fuel bag. This system was picked for lightness, simplicity, and performance. Since the fuel must flow to the engine when the aircraft is in any attitude, there

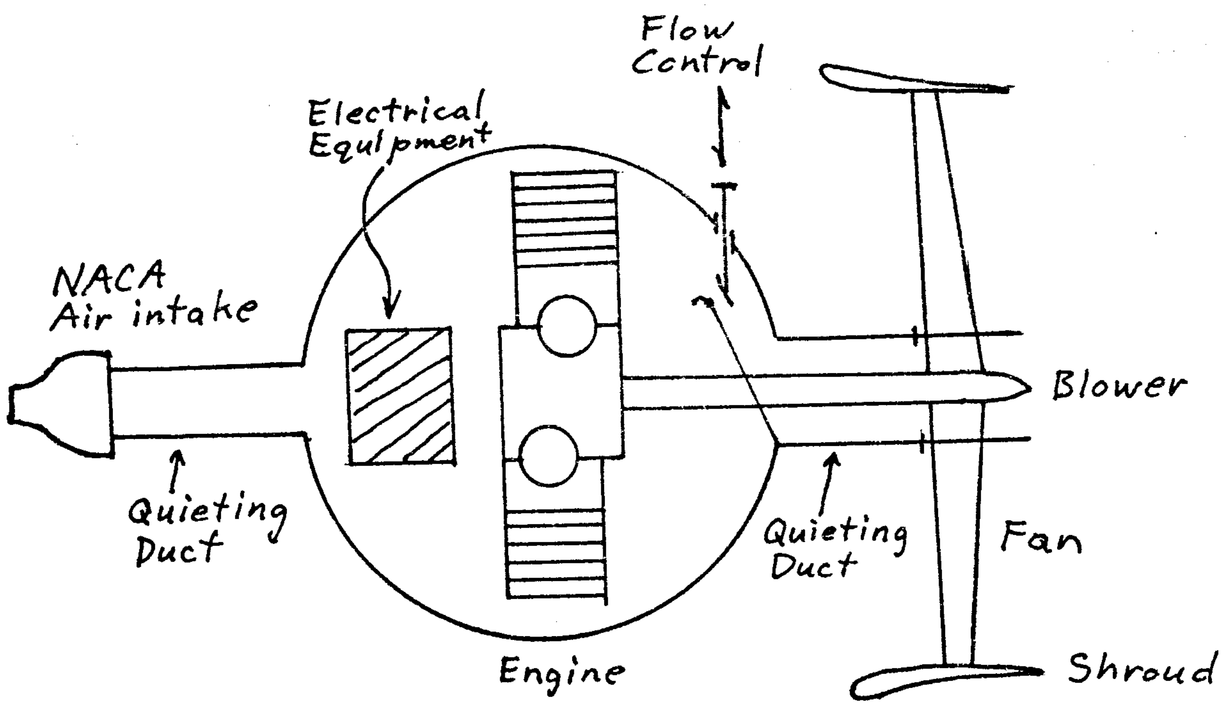


Figure 2.10 Engine Cooling and Propulsor Schematic

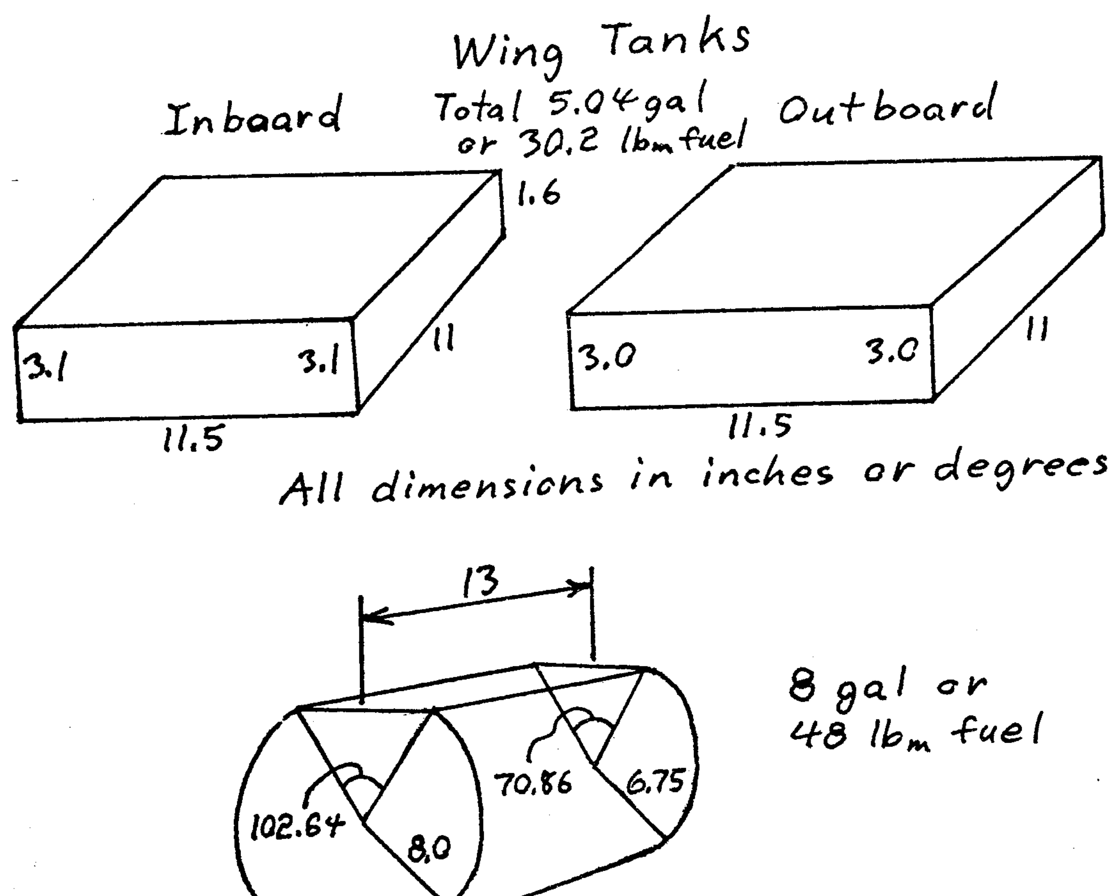


Figure 2.11 Fuel Tank Specifications

should be no air in the tank or possibility that the fuel will not get to the pick-up.

Fuel is introduced to the fuel system through a valve near the engine compartment. It travels up the engine fuel line to the tanks, filling the bags. Removal of air from the bags is simply done during the initial filling. The RPV is tipped nose down and filled. The excess air is bled off with a small hand pump and the system is purged. This initial purging should last for many filling operations, ie. the fuel system should not need purging again.

E. Ignition is cpacitative discharge type, located in front of the firewall. It is a standard item supplied by E - Systems.

The total engine installation is illustrated in Figure 2.12, together with the systems discussed in the following sections.

4.2.2 Propulsor

4.2.2.1 Introduction

The propulsor for this RPV was affected by many considerations which had little to do with performance. The location was fixed at the rear of the fuselage due to payload requirements, yet due to stability and control considerations the propulsor had to be placed on the fuselage centerline. This means that it must operate in an essentially turbulent environment behind the wing and in the boundary layer of the fuselage.

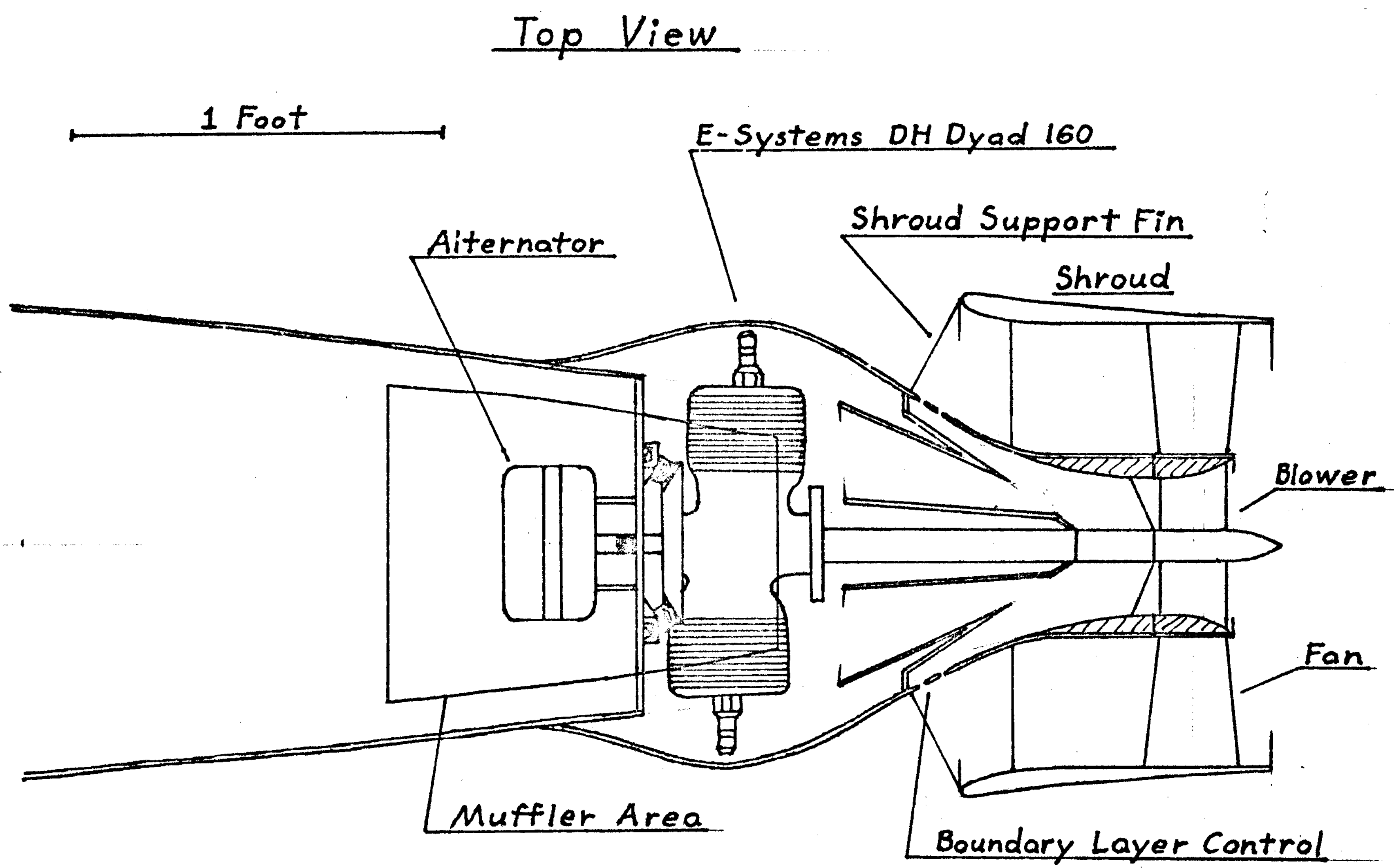


Figure 2.12 Engine and Propulsor Installation

These two factors can have a serious effect on the acoustic and aerodynamic performance of the propulsor. The propulsor must be of small diameter to fit under the tailboom, also not a good restriction from the performance standpoint. However, few propulsors operate in ideal conditions, so that the requirements in summary are:

1. Quiet.
2. Minimum drag.
3. Efficient at loiter.
4. Reasonably efficient at off-design conditions.
5. Low drag when no power is applied.
6. Must be located on fuselage centerline (Stability and control).
7. High thrust at low speed (Landing and take-off).
8. Close to powerplant.
9. Located at rear of fuselage (Missions).

4.2.2.2 Choice of Propulsor

As was seen in the previous section, the choice of propulsor systems were part of the powerplant selection as well. The final two propulsor systems were internal combustion engines with either a propeller or a ducted fan. Full scale trade - offs for ducted fans against propellers are well shown[13] in Figure 2.12 . In the case of the propulsor, since both ducted fans and propellers require a reasonable fuel weight, the determining factors are minimum size and noise.

	Today	1980 Time Period		
	Propulsor/ Reciprocating	Q-FAN/ Reciprocating	Q-FAN/ Rotary Combustion	Q-FAN/ Rotary Combustion
Fan diameter	6.5	3.0	3.0	2.5
Shp	285	387	387	445
Tip speed/rpm	915/2700	640/4060	640/4060	626/4800
Weight	537	680	466	550
Cost, $	6879	10290	7130	7920
Cruise thrust at 0.33 Mach No.	316	329	329	364
Cruise thrust specific fuel consumption	0.439	0.573	0.573	0.595
PNdB/a-c	103	85	85	85

NOTE: Performance does not include engine cowl or installation losses. Q-FAN performance includes duct and external losses.

Figure 2.12 A Performance Comparison of Ducted Fans and Propellers for a General Aviation Aircraft

In both of these areas, the full scale ducted fan is clearly superior. Therefore the ducted fan propulsor was chosen.

There are two parts of the ducted fan system, the duct and the fan. While rather obvious, the relations between the two and how they should be optimized for high performance are not. In the analysis, it was decided to find the optimum shroud designs[14] for different radii and then chose the shroud based on its performance. The fan was designed for maximum efficiency[9,15,16] at the loiter condition.

Initial sizing of the shroud was based on 12.1 pounds thrust required at loiter. It was found that for a reasonable pressure ratio of 1.02 at 5000 feet altitude, a small fan of 5 inch radius is required with a 3 inch radius hub.

However subsequent increase in estimated drag also increased the required radius. A study was performed for shroud chords of 5.5 to 10 inches, radii from 5.52 to 7.2 inches, and two different coning geometries to determine the optimum shroud. The results of the analysis are shown in Figures 2.13 and 2.14 . Obviously the straight duct offers the best performance despite some tuning problems. Performance of the straight duct at loiter is best shown by Figure 2.15 .

Since the fan must operate with the shroud, and is a crucial portion of the efficiency, the fan must be considered at this point. Due to the small radius and low rotative speed the tip radius is an important parameter in fan performance. Fan efficiency increases with radius and rotative speed. However, efficiency decreases with increased forward

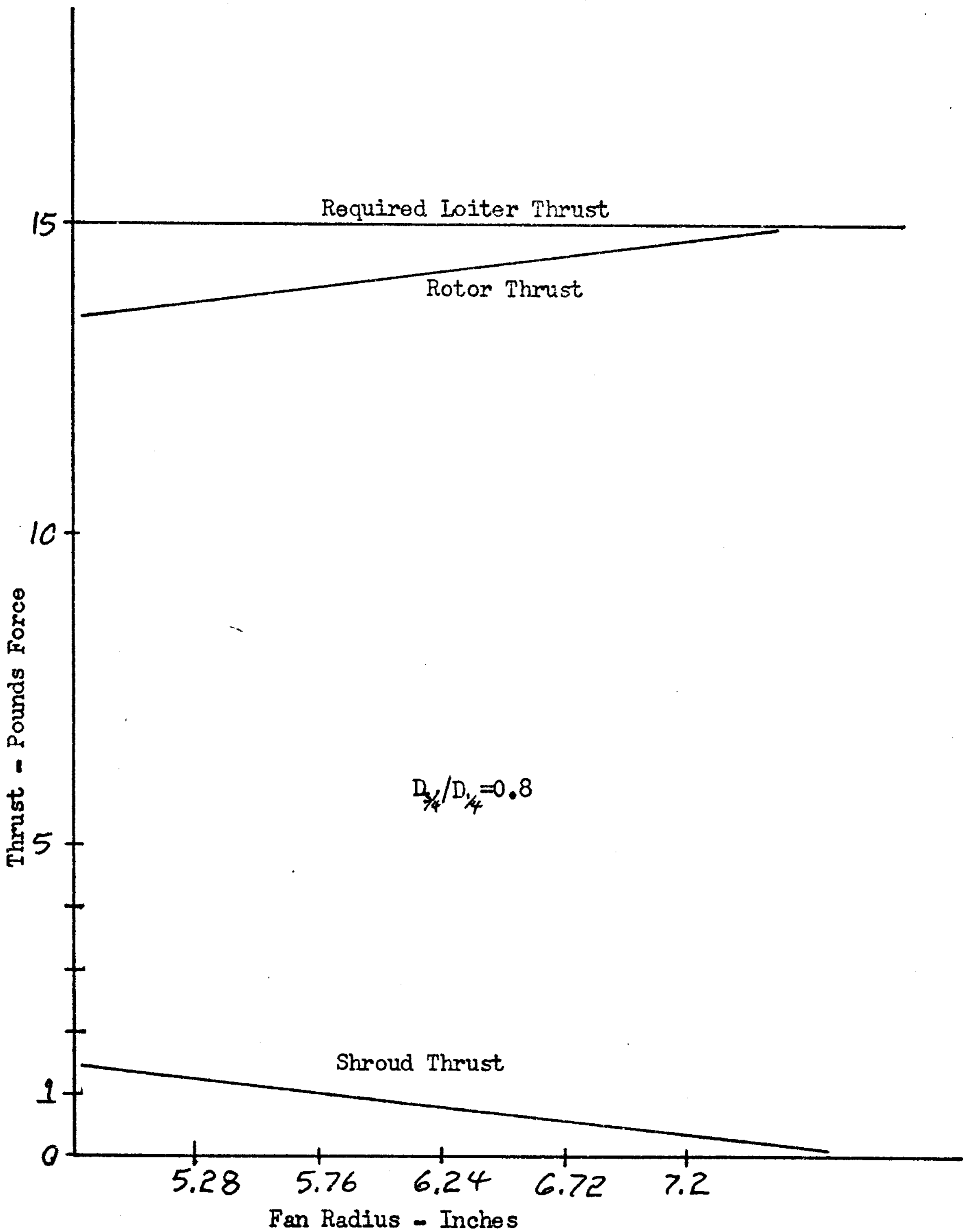


Figure 2.13 Coned Shroud Optimum Performance and Radius

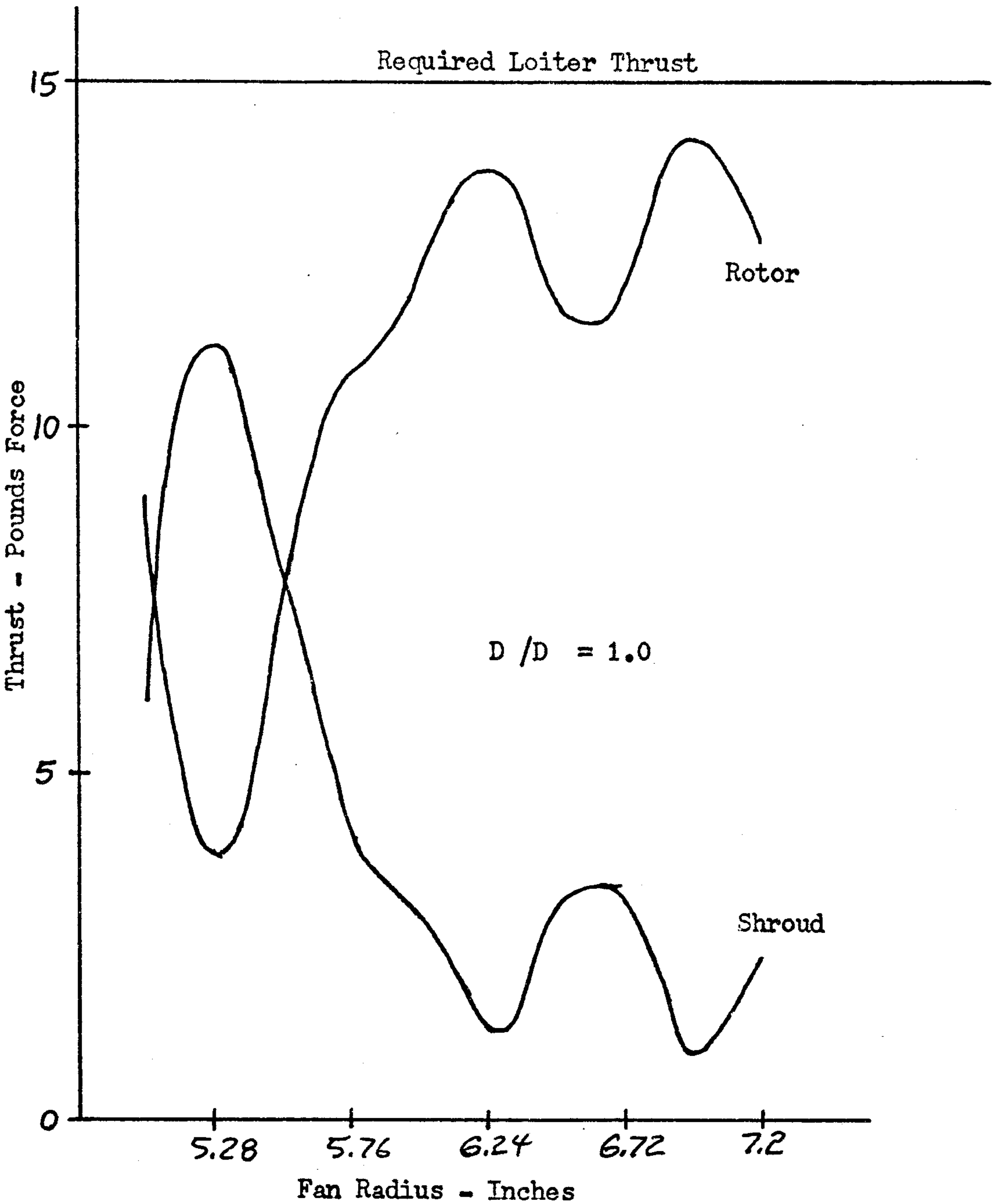


Figure 2.14 Straight Shroud Optimum Performance and Radius

Figure 2.15 Straight Shroud Performance, Loiter Condition

Radius (in.)	Fan Thrust (lb.)	Shroud Induced Velocity (fps)
5.04	9.0	141.871
5.28	5.0	251.403
5.52	7.2	264.894
5.76	10.7	125.337
6.0	12.1	118.298
6.24	13.7	119.575
6.48	12.0	119.836
6.72	11.9	121.619
6.96	14.2	112.568
7.2	12.8	116.617

Figure 2.16 Fan Performance Relative to Shroud Internal Conditions

Pitch (ft.)	$C_{T'_R}$	$C_{T'_Q}$	η
.2	51.10495	18.27524	.1900533
.2399999	46.08577	13.98929	.224706
.2799999	36.14958	11.07721	.2596948
.3199999	28.98628	8.970955	.293857
.3599998	23.81247	7.442678	.3273481
.3999998	19.84402	6.25802	.3604832
.4399998	16.73039	5.313311	.3937546
.4799997	14.09462	4.513007	.4260502
.5199997	11.96348	3.879035	.4557948
.5599996	10.18231	3.34893	.4839051
.5999996	8.67236	2.8983	.5102428
.6399996	7.438142	2.519732	.5369351
.6799995	6.382459	2.196349	.5616004
.7199995	5.45036	1.904989	.5854595
.7599995	4.656245	1.654106	.6080198
.7999994	3.996002	1.44345	.6294274
.8399994	3.425898	1.257425	.6504341
.8799993	2.957793	1.101151	.6717932
.9199993	2.576008	.9708675	.693756
.9599993	2.26325	.8615105	.7167622
.9999992	1.996099	.7653927	.7411896
1.039999	1.755439	.6775855	.7657498
1.079999	1.548331	.6019362	.7895309
1.119999	1.378546	.5391544	.8138754
1.159999	1.234445	.485532	.838192
1.199999	1.103832	.436841	.8617736
1.239999	1.083022	.4285519	.8906109
1.279999	1.036334	.4106002	.9181696
1.319999	.9343053	.3722734	.941527
1.359999	.8355578	.3349558	.9641835
1.399999	.7434327	.2998739	.9864223

velocity with thrust, rotative speed, and radius constant. Therefore a large radius and moderate shroud induced velocity is desirable. On this basis the 7.2 inch radius optimum shroud and fan were chosen as the largest and most efficient that could fit under the tailboom.

4.2.2.3 Propulsor Performance

Before designing the fan a blade airfoil had to be chosen which would give the most efficient design. This small of a fan operates in a very critical Reynold's Number regime which makes choice of the proper airfoil very important, but proper data on the airfoil at the desired condition very sparse. The analysis for airfoil selection is presented in the Appendix, and resulted in choice of the NACA 2412 for the blade section.

A fan was designed for the loiter condition which performs as shown in Figure 2.16 . At the loiter condition, allowing 0.35 horsepower for electrical generation and drag at full gross weight (15 pounds force), the required shaft horsepower is 5.0 hp at an SFC of about 0.7 .

Total performance is entirely adequate except in the landing and take - off condition, as shown in Figure 2.17 and 2.18 . Thrust of the ducted fan is high, but not enough to offset the high drag and provide good climb capabilty. In engineering development of this system, it is recommended that one extra inch of fan radius will provide the necessary low speed performance with no loiter or dash speed

Figure 2.17 Overall Propulsor System (Shroud and Fan) Performance

C_T	C_Q	ρ_V	C_{T_R}'	V_R/V	C_Q/C_Q'	C_P
1.277	0.3427	1.3156	0.756	1.060	1.125	0.2605
2.063	0.5519	1.0797	1.176	1.091	1.190	0.5112
3.095	0.8312	0.934	1.695	1.126	1.268	0.997
4.029	1.0883	0.8474	2.138	1.155	1.333	1.2843
5.007	1.3568	0.7781	2.579	1.182	1.396	1.7437
7.038	1.9087	0.6821	3.431	1.231	1.516	2.7983
9.505	2.5744	0.6055	4.377	1.283	1.645	4.2517
11.738	3.1794	0.555	5.168	1.323	1.755	5.7286
13.049	3.5358	0.5303	5.609	1.345	1.809	6.6675
15.055	4.084	0.4982	6.255	1.376	1.894	8.1975
16.984	4.612	0.4718	6.847	—	—	9.7753
19.013	5.1659	0.4471	7.443	—	—	11.5542
21.027	5.7198	0.4265	8.011	—	—	13.411
23.029	6.274	0.408	8.554	—	—	15.377
25.416	6.9285	0.3892	9.179	1.507	2.272	17.802
27.508	7.5007	0.3745	9.706	1.529	2.339	20.029
29.029	7.9177	0.3642	10.080	1.545	2.386	21.74
30.826	8.4094	0.3538	10.51	1.562	2.44	23.769
34.235	9.3512	0.3356	11.3	1.594	2.54	27.86
37.404	10.246	0.3204	12.0	1.621	2.628	31.98
39.73	11.234	0.3109	12.505	1.64	2.691	36.134
42.105	12.241	0.3017	13.003	1.659	2.753	40.574
43.097	11.809	0.2979	13.207	1.667	2.779	39.636
45.023	12.345	0.2909	13.598	1.682	2.828	42.437
46.506	12.755	0.2859	13.895	1.692	2.864	44.613
48.535	13.326	0.2794	14.293	1.707	2.914	47.695
50.013	13.743	0.2753	14.579	1.717	2.949	49.92
52.035	14.317	0.2687	14.964	1.731	2.997	53.085
53.018	14.596	0.267	15.149	1.738	3.02	54.667
57.647	15.909	0.2551	15.998	1.768	3.125	62.365

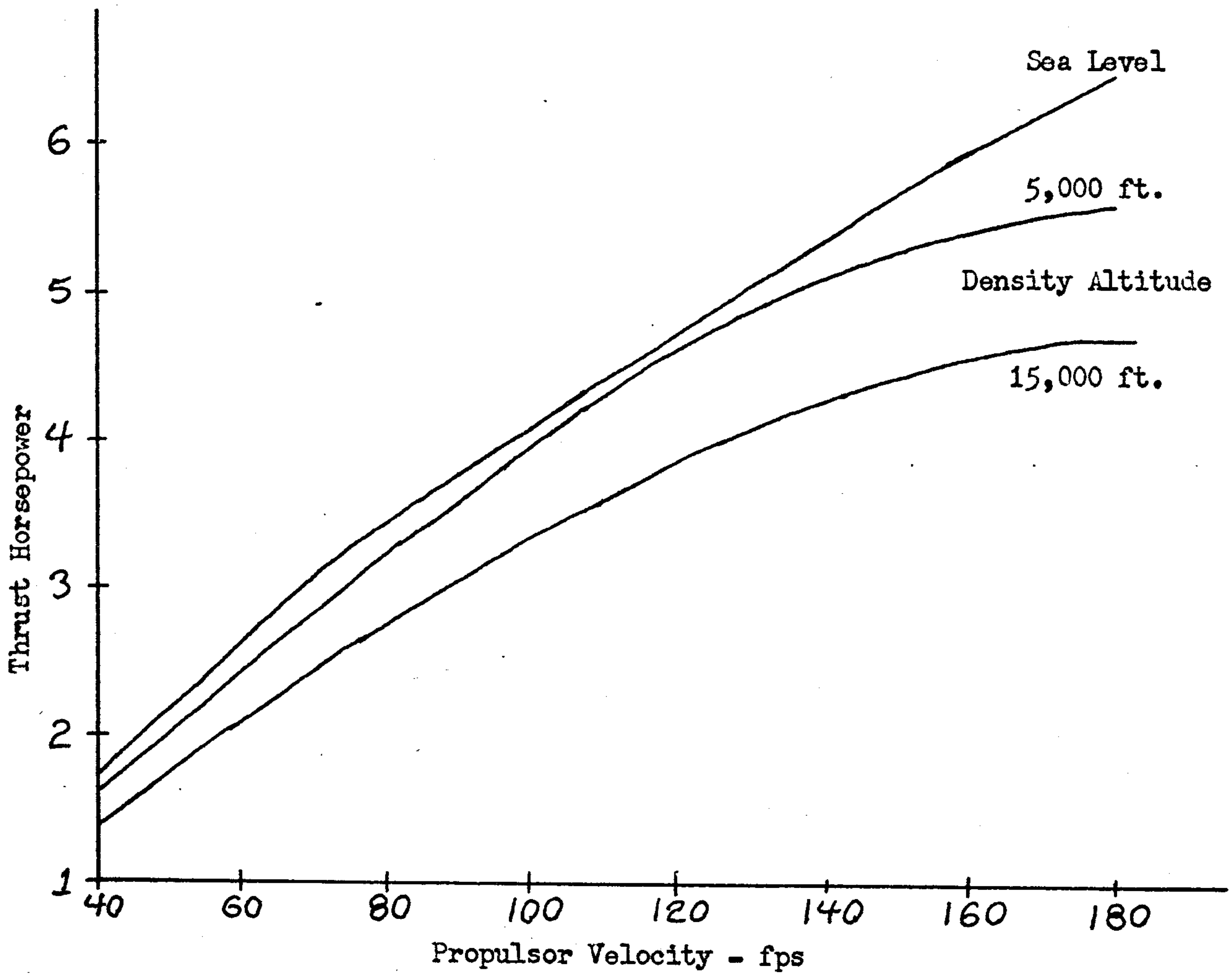


Figure 2.18 Horsepower Available from Propulsor System

performance losses.

4.2.3 Acoustics

4.2.3.1 Introduction

Acoustic performance of this RPV is every bit as important as the other performance characteristics of the aircraft. There are only a few requirements of any acoustic modification to the system:

1. Light weight.
2. Reasonable volume.
3. Low power loss.
4. Effective.
5. Aircraft undetectable (acoustically) at 5000 feet above the ground level.

There are three noise mechanisms involved in the system: the noise produced by the aircraft, the transmission path of the noise, the sensitivity of the listener[17]. The subsections will first concern the sources of noise and then take into account the other two noise factors.

It is interesting to digress at this point and note that Northrop is presently developing a drone which carries audio detection equipment[18]. This equipment uses two microphones to detect range and bearing, as well as the identity, of selected targets. The drone then homes in on the target, be it a tank, troop carrier, or whatever, and destroys it. RPVs, it seems, can be quiet but also can take advantage of noisy ground troops.

4.2.3.2 Engine Noise

Projections of engine noise are based on experimental work by Shimovetz and Smith[19]. Of particular interest is the piston engine test, described in Figure 3.1 and shown in Figure 3.2. The engine was statically tested in an anechoic chamber with a propeller load and an acoustically treated ring cowl. It must be noted that the cowl was open front and back for proper cooling airflow and not very long (about as long as the engine in the axis of rotation of the prop). The unmuffled tests effectively include engine exhaust noise, most casing noise, and propeller noise. Since the propeller was the dominant noise source and results were uncorrected, any projections from this experimental work will be extremely pessimistic. Tests were performed at 4500, 5500 and 6500 RPM, the last corresponding to the loiter condition of this RPV. Therefore results for the unmuffled piston engine at 6500 RPM were used as basic engine noise for the projections.

Even though casing noise (including carburator noise) was effectively included in the analysis, in the actual vehicle steps are taken to eliminate that noise. The entire engine is enclosed and cooling air is channeled through at least a foot of acoustically treated duct, which should attenuate the noise below consideration when compared to other noise sources.

Noise in the tests was found to be dependent on power output. For the engine tested, the horsepower output was about 3 horsepower

ENGINE	RC ENGINE	PISTON ENGINE
Type	charge cooled rotary combustion	2 cylinder opposed 2 stroke cycle
Displacement	2.48 cubic inch	3.8 cubic inch
BHP	5.0 @ 12,000 RPM	5 @ 9,000 RPM
BSFC	0.65 #/BHP HR @ 8000 RPM	
TORQUE	28 inch pounds @ 9000 RPM	
BMEP	71 psi @ 9000 RPM	
Ignition	CD - Spark plug	Glow plug
Ignition timing	15° BTDC	
Compression ratio	11:1	
Weight (lb)	7.34	4.8
Dimensions	6-3/4 inch diameter x 5-3/4 inch	
Fuel	40:1 white gasoline/SAE30	10/10/80 nitro-methane caster-oil/methanol
Carburetor	Tillotson	Tillotson
Manufacturer	unknown	Kolbe Korp Model D-238

Figure 3.1 Specifications of Test Engine

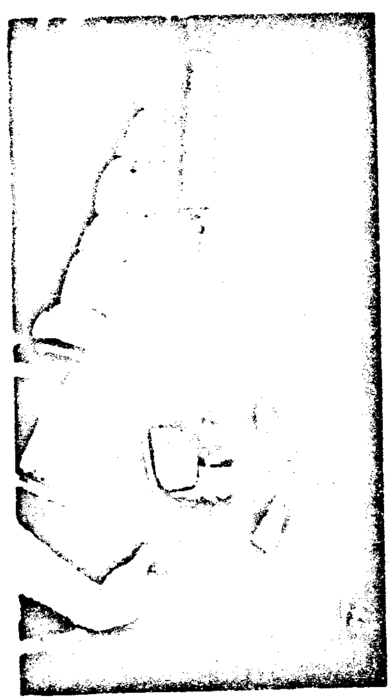


Figure 3.2 Test Engine Configuration

as shown in Figure 3.3 . Increase in spectral sound power is projected at 6 dB for a doubling for horsepower, however it must be noted that during loiter the RPV engine will actually be operating at 5 horsepower, so the increase is pessimistic. The test spectral sound power is shown in Figure 3.4, along with the RPV projected noise spectrum from its engine.

In the tests directivity of the exhaust in both muffled and unmuffled configurations was found to produce a 6 dB decrease in sound power if the exhaust was directed away from the microphone. On the RPV, the exhaust will be directed skyward to produce the same projected attentuation. Therefore the untreated engine (or unmuffled) installation in the RPV corresponds to the sound power of the test engine without muffler, operating at 6500 RPM.

Muffling of the exhaust is an important factor in quieting the engine. In the tests, several mufflers were tried (Figure 3.5). It is projected that a muffler of over twice the size of muffler D will produce comparable quieting of 27 dB. This is pessimistic since volume required for a certain attentuation is roughly dependent on power output, and doubling the size actually quadruples the volume. The final corrected sound power spectrum for the engine with the projected muffler design is shown in Figure 3.6 .

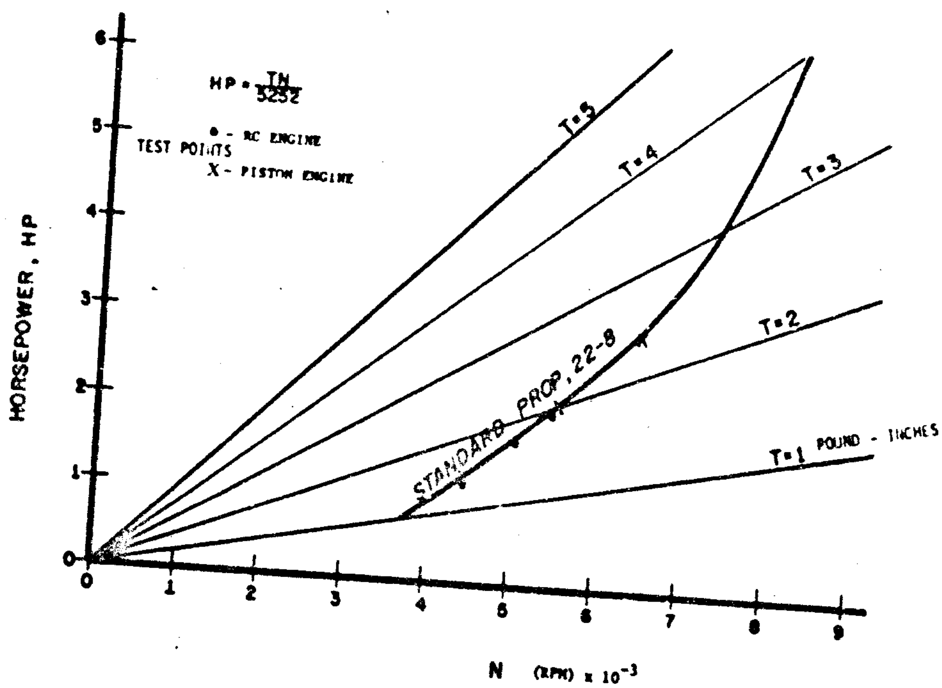


Figure 3.3 Test Engine Propeller Loads

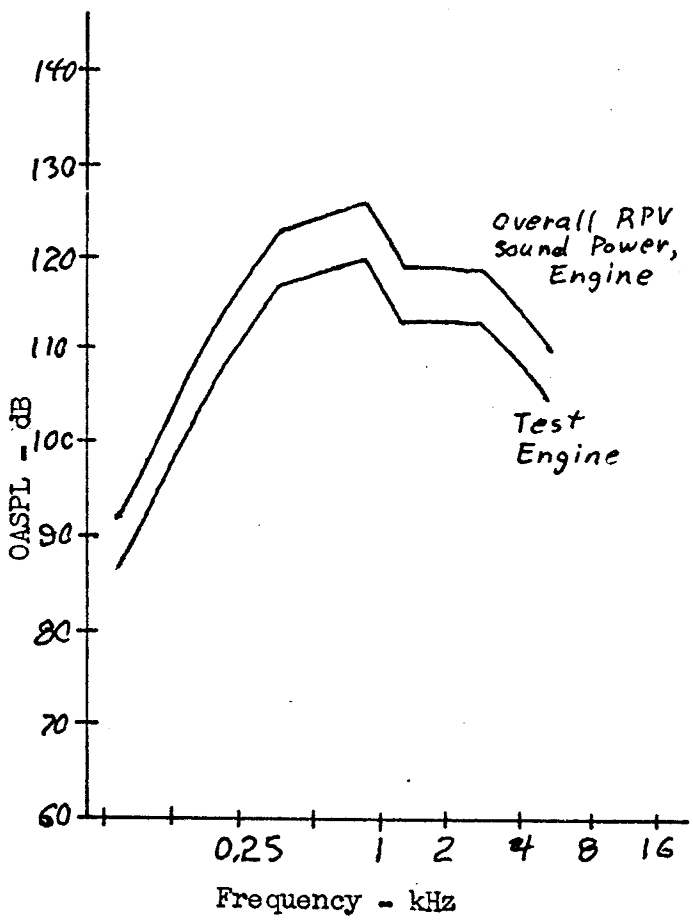


Figure 3.4 Test Engine and RPV Engine Noise Spectrum

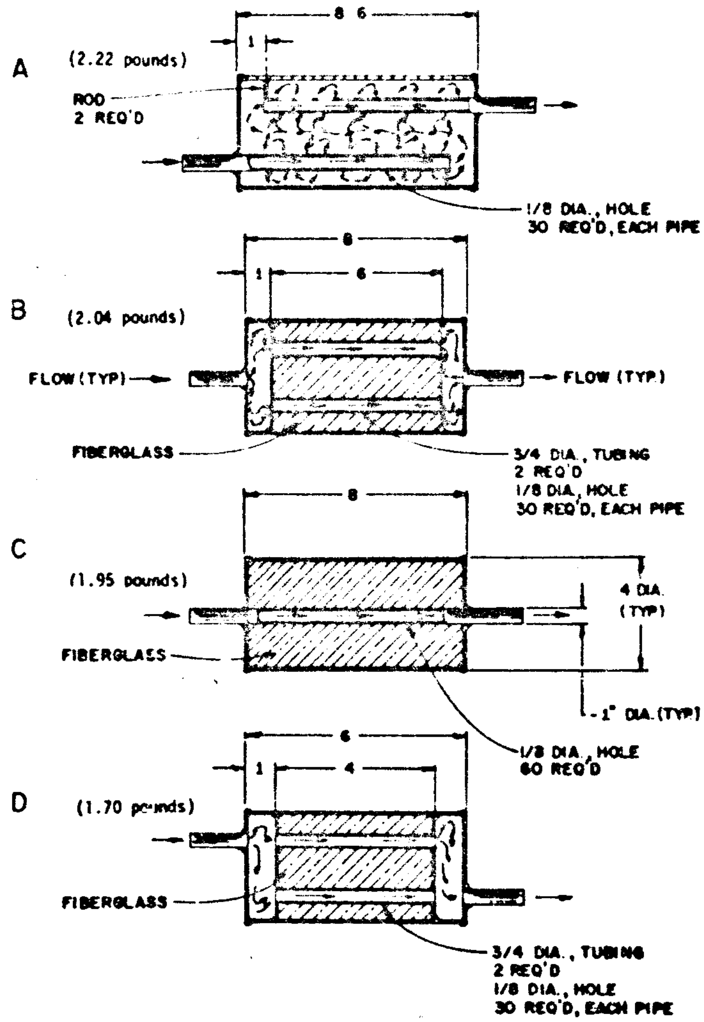


Figure 3.5   Test Engine Mufflers

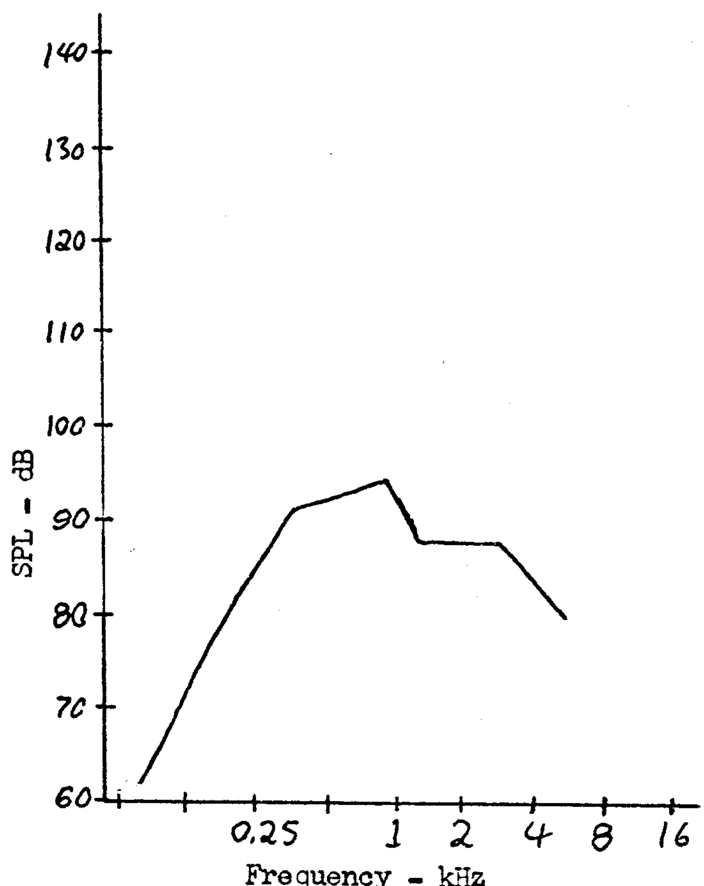


Figure 3.6m Projected Sound Power Spectrum Below RPV, Corrected for All Attenuation Mechanisms on the Vehicle

4.2.3.3 Fan Noise

Ducted fans in the large scale offer many acoustic advantages over open propellers. They are inherently more quiet. However in this system the fan is operating in a turbulent environment and is of a very small size. Only one set of tests for a fan of this size was found[21]. These were static tests of a seven inch diameter fan with a hub-to-tip ratio of 0.482 . Since the operating C_L of the blades was typically above 0.5, the test fan was considered similar to the RPV fan, even though it was probably inherently noisier. At 5440 RPM and with a grid installed, maximum spectral sound pressure level was 80 dB in the far field (4.35 rotor diameters upstream). The grid had a mesh size of 1.126 inches and a mesh rod diameter of 0.22 inches, and was placed 3.5 fan diameters upstream of the fan. This more than adequately simulates the flow conditions of general turbulance caused by the wing and other fuselage protrusions. Experiments were conducted on turbulent boundary layers on the hub as well, which showed that the boundary layer has a marked effect on the fan noise. Therefore the blower fan design was adapted to provide suction for boundary layer control. The required specifications for the blower are given in the Appendix.

Due to the fact that the test rotor was 10 bladed and results were not presented in conventional octave bands, it is difficult to make a straight - forward projection of the RPV fan noise. However, a uniform SPL of 65 dB can be pessimistically assumed across the

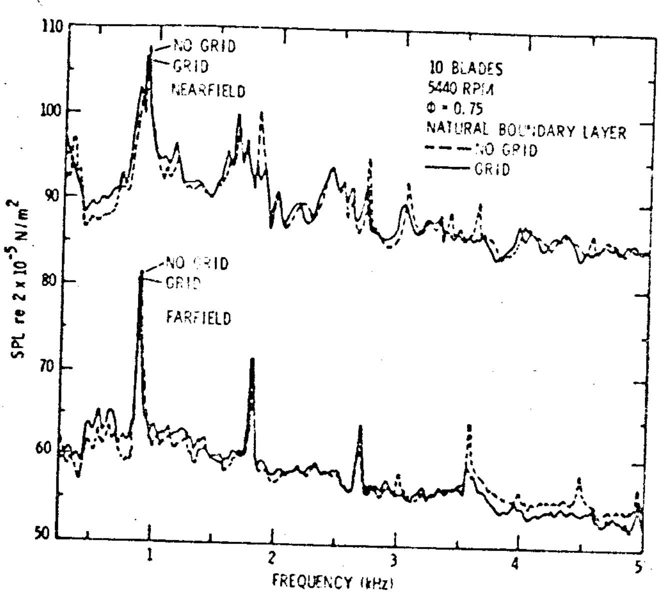


Figure 3.7 Noise Spectrum of Test Rotor

spectrum at the far field 3 feet from the fan. It must also be noted that the test results shown are for noise directly in front of the fan, which is very pessimistic when considering directivity and sideline noise. An SPL of 65 dB at 3 feet corresponds to a sound power level of 75 dB.

4.2.3.4 Aircraft Aerodynamic Noise

Tests of full-scale aircraft have produced a method[22] for predicting aircraft noise. These methods were applied to the RPV and resulted in a prediction of aerodynamic noise for this RPV of 93 dB sound power level. Using the method's non-dimensional plot for the aerodynamic noise frequency spectrum, Figure 3.8, the RPV aerodynamic noise prediction is then as shown in Figure 3.9, after the maximum amplitude frequency of 779.86 Hz is found.

4.2.3.5 Detectability

The detectability of the RPV can be found once the total noise generation is summed up, Figure 3.10 . Using the methodology of Shimovetz[20] the detectability at any distance can be related to the noise of the vehicle, the transmission loss, and the listener's sensitivity. The first two estimations are fairly fixed, but the listener's sensitivity is extremely variable. Detectability of sounds varies in different environments, as can be seen from the experimentally derived data in Figure 3.11 . The curves represent the level at which

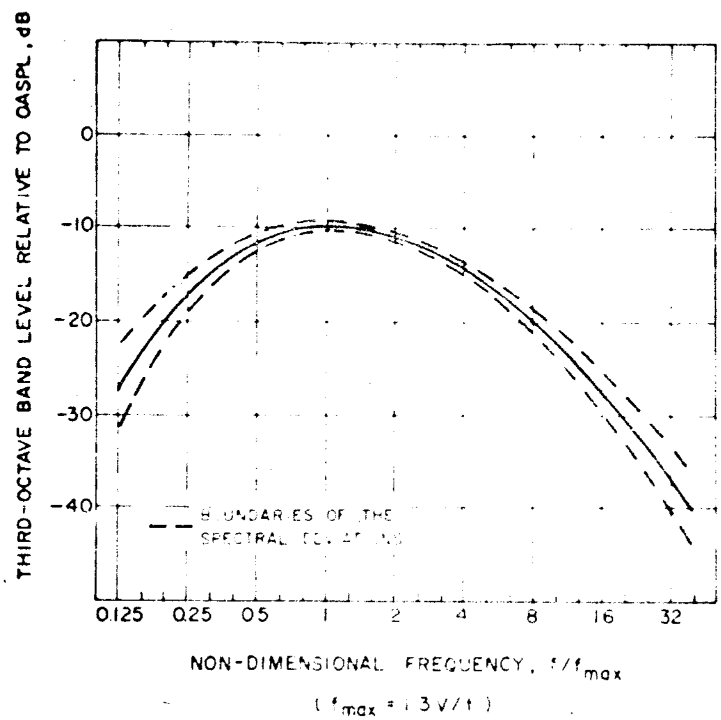


Figure 3.8 Non-Dimensionalized Aircraft Aerodynamic Noise Spectrum

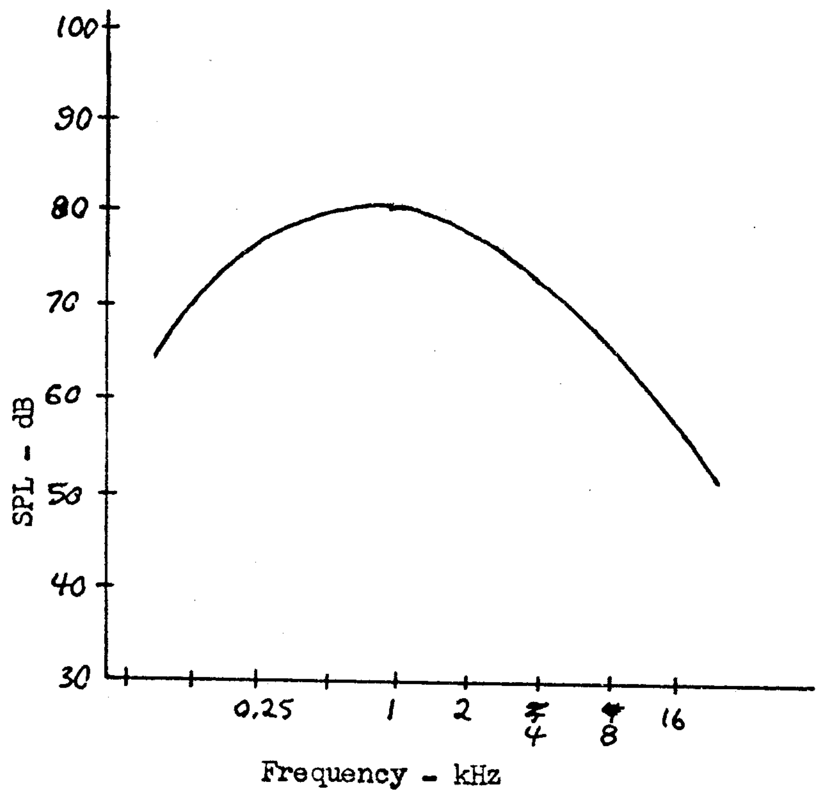


Figure 3.9 Projected Aerodynamic Noise of RPV

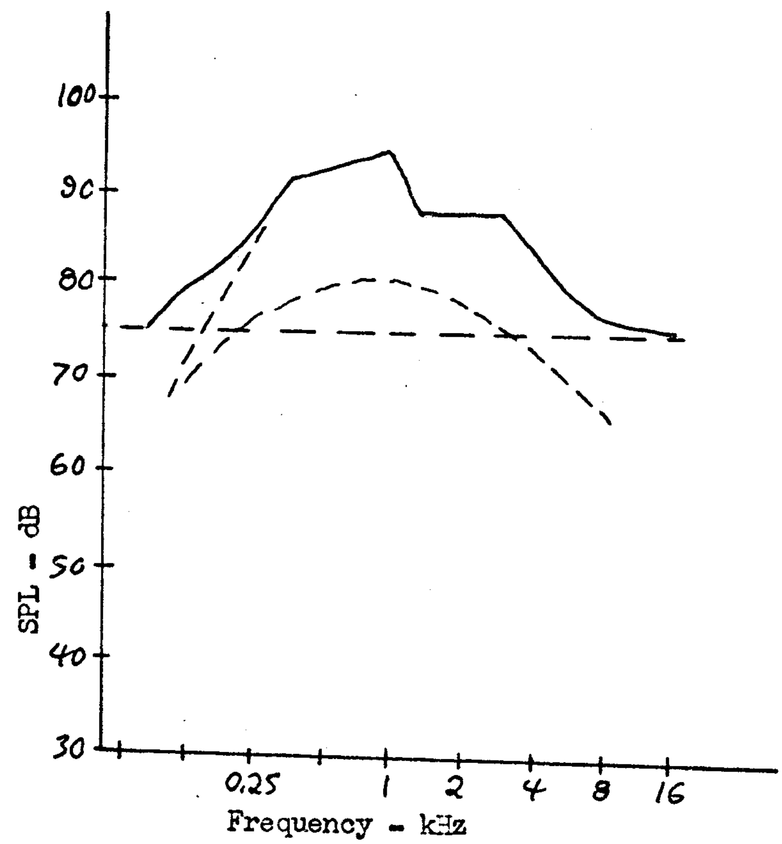


Figure 3.10 Total Noise of RPV Projected Towards the Ground

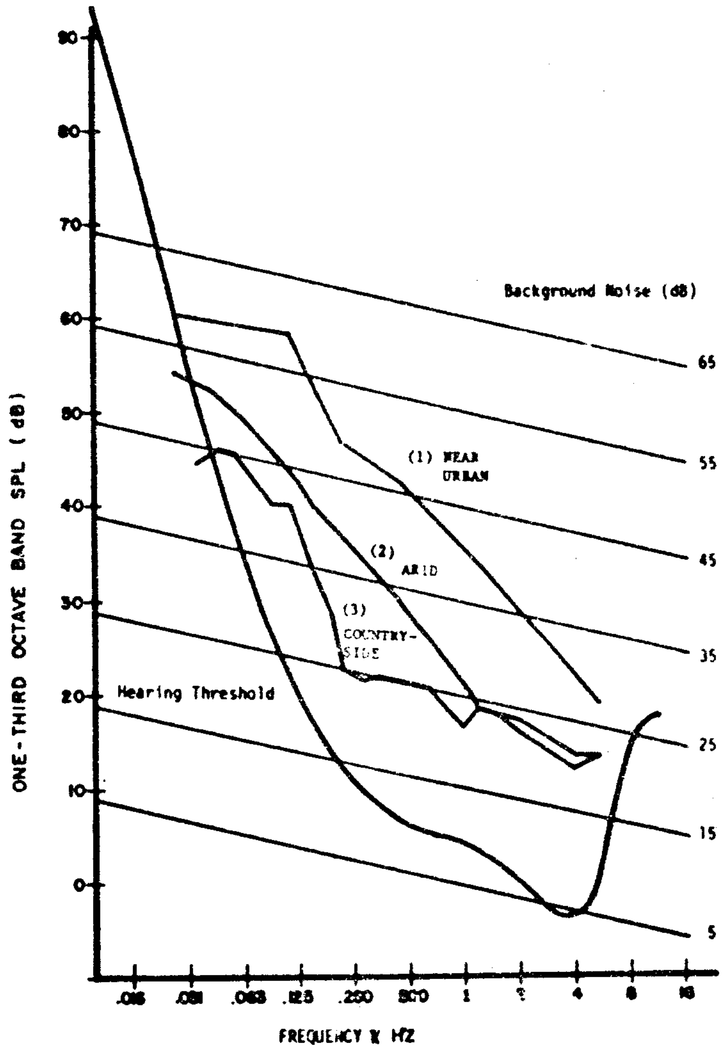


Figure 3.11 Detectability of Noises

a noise will have a 50% probability of detection (but not recognition) with a 1% alarm rate. This data was compiled for a sample group composed of young adults, presumably the portion of the general population with the best hearing. Note that the hearing threshold usually extends well below detectability in any realistic environment, including the desert. If any tone produced by the RPV equals or exceeds the levels shown, there is a 50% possibility it will be detected (but not necessarily recognized).

Using accepted methods for transmission loss in air and distance attenuation, then the noise of this RPV at any altitude may be found. The noise spectrum and overall noise level of the RPV is presented in Figure 3.12 and minimum 50% detection altitudes are compiled in Figure 3.13 .

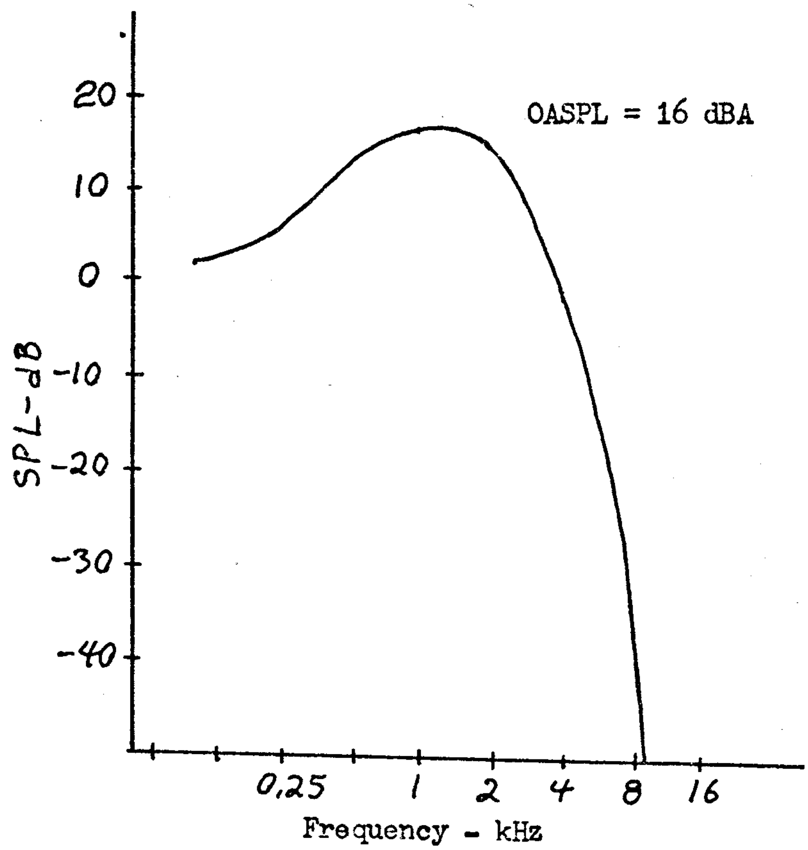


Figure 3.12 Sound Pressure Level at Ground Directly Below RPV. RPV altitude 5000 feet

Figure 3.13 Detectability of RPV

Altitudes at which there is a 50% probability of detection of RPV noise (but not necessarily recognition) at a 1% alarm rate.

Environment	Altitude
Country-Side	3900 feet
Arid	2900 feet
Near Urban	650 feet

4.3 Power Generation and Storage

4.3.1 Introduction

There are only a few requirements for electrical power generation and storage. These are:

Power Generation

1. Adequate power for average power requirement plus a margin.
2. Light weight.
3. Small volume.
4. Close to powerplant.
5. Adequate cooling.

Power Storage

1. Adequate power for emergency engine shut-down and safe landing.
2. Small volume.
3. Light weight.

4.3.2 Choice of Power Generator

Electrical power for all the systems of the RPV is provided by the engine. To best estimate the power required, a similar RPV system was studied, the Lockheed Aquila. Payloads, engine, and control power requirements are alike[8,9]. The Aquila, in fact, requires much more electrical power in the telemetry portion of the system since microwave transmission is used. This RPV uses vhf-uhf transmission which is more efficient. Therefore it would be reasonable, perhaps

Figure 4.1 Batteries Included in Emergency Power Supply Study

#	Voltage	Weight (lb)	Energy-to-Weight Joule/lb	Power-to-Weight Watt/lb
1	6	0.6	6.48×10^5	9
2	12	1.9	6.82	9.47
3	6	1.3	8.64	12
4	8	1.9	7.88	10.95
5	6	2.4	8.1	11.25
6	12	4.5	8.64	12
7	6	2.6	9.97	13.85
8	6	3.3	9.82	13.64
9	12	16.7	10.35	14.37
10	1.25	0.05	3.6	12.5
11	1.25	0.05	3.6	12.5
12	1.25	0.144	3.75	13.02
13	1.25	0.3375	4.27	14.8
14	1.25	0.3375	4.27	14.8
15	1.25	0.3375	1.28	4.44
16	1.25	0.144	3.0	10.42
17	1.25	0.05	3.6	12.5

even a bit pessimistic, to include the Aquila's 600 watt alternator design in this RPV. Shaft horsepower allowance for electrical power generation in the Aquila was 0.35 horsepower[9], and this is also a pessimistic figure. The alternator is located in front of the firewall in the intake air plenum. Size of the alternator is about 6 inches in diameter and 6 inches in length, and it weighs about 6 pounds.

4.3.3 Choice of Power Storage

Two battery types and three brands of batteries were studied for this application. These types are listed in Figure 4.1 with their relevant statistics. Batteries 1 through 9 are Gel/Cell type B (recharging optimized) rechargable cells. Batteries 10 through 14 are Eveready nickel-cadmium cells and 15 through 17 are Ray-O-Vac nickel-cadmium cells. The Gel/Cells are the best type of battery of the group for energy storage and energy delivery (work) per unit weight. Of this group several conbinations were studied for the best overall weight providing the necessary power storage. The optimum configuration had a weight of 8.9 pounds, and a volume of 117.7 cubic inches. Performance was 30 volts at up to 9 ampere/hours peak load with cooling. The batteries are distributed in the air plenum in front of the firewall to best provide adequate forced convective cooling.

One of the good benefits of providing an emergency power pack that is rechargable is that, should peak power demand exceed the alternator rating, the batteries can take up a substantial portion of the load. The batteries can also provide voltage stabilization in the power supply circuitry by acting as capacitors. In addition, for peak climb performance or any other critical flight regime, the alternator can be switched out of the circuit so that full power can be applied to the propulsor.

4.4 Costs

4.4.1 Introduction

When thecapabilities and low cost of RPV systems are seriously considered, it is obvious that the RPV is the only real way to accomplish many tasks. Whether carrying out tactical, intelligence, communications, or navigation functions for the military, or performing any number of civilian tasks, the RPV is proven to be most cost effective. There are many important missions which only the RPV can perform.

The fairly slow growth of modern RPVs is definitely not caused by lack of need. Rather, the modern RPV, equipped with sophisticated electronics, has been restrained in capability until very recently. Lowered cost and the necessary capability of the electronics subsystems have been acheived in only the past few years.

Most RPV development has been for military application, which is unfortunate. There is a large untapped market for civilian - type RPVs which will be discussed later. Costs of all RPV subsystems are dropping rapidly, and civilian acquisition of RPVs is possible by more and more businesses, government agencies, and people. It is still too early to talk about a chiken in every pot and an RPV in every garage, but in a very short time the friendly neighborhood policeman may be checking the citizen's car speed (and taking his license number for his ticket) by RPV.

In the following sections the cost of this particular RPV system will be estimated. Cost - effectiveness for several missions will then be discussed relative to this system, where the cost advantages of some performance traits of this RPV will also be indicated.

4.4.2 Cost of System

The RPV system proposed is a sophisticated system. Many expensive electronics packages are used to provide adequate sensing, control, and payload capability for a multitude of missions. This can best be seen by listing the costs of materials for this RPV:

Engine	$ 1200
Production Airframe	3000
Autopilot	10000
Servos	260
Flap Servo	15
Telemetry	10000
Transponder	500
Alternator	25
Power Supply	70
Batteries	100
Min. TV Camera	2500
Total	$ 27670

Many of these costs are projections based on similar units. For instance the production airframe is estimated from the cost of the

Aquila airframe[8], which is also of Kevlar over a metal structure. Whenever costs were estimated, they were taken at the highest reasonable value.

This last is essentially a listing of subsystem costs and production airframe cost. Since research and development costs on RPVs are a substantial part of the overall cost, merely assuming that they are "hidden" in the difference between this price and the price due to production quantity discounts is unrealistic. While the price for the telemetry and autopilot is well above the most inexpensive subsystems of this type available, it is best to retain the original figures as materials costs instead, and combine the total with engineering, manufacturing, development, and tooling costs. This is a very inflated, double - counting way of predicting production costs, but in view of cost overruns on some aircraft, the effects of inflation (although the cost of high-technology items is coming down), and allowing for inclusion of more sophisticated and expensive subsystems during development, overestimation seems the best strategy.

There are many methods of estimating cost for RPVs. These are usually based on a particular statistical analysis of costs and the characteristics parameters of the airframe and production. One of these methods is the method of Cost Estimating Relationships (CERs) which have been applied to RPVs[23]. Using this method, six cost areas can be predicted with fair accuracy.

A. Engineering Hours

Includes:

1. Design studies and integration.

2. Engineering for aerodynamic models and mock-ups, engine test.

3. Test engineering, laboratory work on subsystems and static test items, and development testing.

4. Release and maintenance of drawings and specifications.

5. Shop and vendor liason.

6. Analysis and incorporation of changes.

7. Material and process specifications.

8. Reliability.

B. Tooling

Hours for tool design and planning, production test equipment, check-out of tools, maintenence of tooling, normal changes, and production planning.

C. Manufacturing Labor

Those hours necessary to machine, process, fabricate, and assemble the major structure of the RPV, and to install purchased parts, government-furnished aeronautical equipment, and off-site manufactured assemblies. Also quality control functions.

D. Materials

Manufacturing materials cost is defined as the cost of the raw material, hardware, and purchased parts necessary for the fabrication and assembly of the RPV airframes.

E. Development Support

Defined as the nonrecurring manufacturing effort undertaken in support of engineering during development. Includes cost of manufacturing labor and material for mock-ups, test parts, static test items, and other items of hardware that are needed for airframe design and development work.

F. Flight Test

Costs incurred by the contractor to carry out flight tests, except for the cost of the RPV itself. Includes engineering planning, data reduction, manufacturing support, instrumentation, spares, fuel, oil, facilities rental, and insurance. Costs incurred by the military are not included.

Usually flight testing is conducted in 3 phases:

(I) Subsystem development test and evaluation.

(II) System development test and evaluation.

(III) System operational test and evaluation.

For the most part only phase I is performed by the contractor. Most of phases II and III is borne by the military.

If this particular RPV system were only to be developed for the civilian sector, it is likely that flight test cost and per unit cost would be much higher, even though the cost of subsystems in the aircraft might be less. On the other hand, experience gained in developing a multi-mission RPV for the military would be directly applicable to civilian versions.

Applying the method of CERs to this particualr RPV system, the costs would break down as follows for a production run of 1000 aircraft:

Engineering	$ 2,845,997
Tooling	519,551
Manufacturing Labor	6,723,495
Materials	5,772,994
Devel. Support	300,497
Flight Test	116,881
Total	$ 16,279,419

This includes cost of 32 flight test vehicles and 8 developmental articles. These quantities are comparable to the Aquila program's 36 total airframes for a program that has nearly completed the system operational test and evaluation flight test phase. Obviously, the state of the art has advanced since the Aquila was developed, and the numbers are pessimistic.

The quantities developed by the CERs used are for 1973 dollars. Since then, a period of rampant inflation has drastically effected wages and costs. The Bureau of the Census figures[24] show that the factor of inflation from 1973 to 1976 was 1.273 . Accounting for 6% inflation in the last two years increases the correction factor to 1.43 . The current projected costs are now:

Engineering	$ 4,069,776
Tooling	742,958
Manufacturing Labor	9,614,598
Materials	8,255,381
Development Support	429,710
Flight Test	167,140
Total	$ 23,279,563

This figure is entirely too optimistic, as can be seen from the low unit aircraft cost (for the 1000 units produced) of only $23,279 . This is due to the fact that the CERs resulted from a survey of 1960's drones and RPVs which had become operational by 1973. These were of current technology (approximately) in everything but the electronics and payloads areas. Since Materials cost is well estimated in the first tabualtion of total parts cost, that may be substituted into the CER analysis.

This is much more realistic for a program of this magnitude. However, the modified total cost of $ 43,800,982 is not quite the total cost, since ground handling equipment (AGE) and spare parts are not included in the CER analysis. In the CER survey of production systems, it was found that this cost was about equal to 25% of the combined costs of engineering, tooling, manufacturing labor, and material. For this RPV the additional cost amounts to $ 10,801,033 . Total program cost is then estimated at $ 54,602,015, a relatively small total cost for 1000 aircraft. Unit cost is $54,602 . A breakdown of costs follows:

	% Total Cost
Engineering	7.45%
Tooling	1.36
Manufacturing Labor	17.61
Materials	52.7
Development Support	7.87
Flight Test	0.31
AGE and Spare Parts	19.78
Engine	2.2%
Airframe	5.5
Autopilot	18.3
Servos	0.5
Flap Servo	0.03
Telemetry	18.3
Transponder	9.16
Alternator	0.046
Power Supply	0.13
Batteries	0.18
Min. TV Camera	0.46

This breakdown shows that electronics account for 46.58% of the total program cost. A modest reduction in electronics cost would have a major effect on program and unit cost. For instance, if a relatively simple ground based computer and direction finder, together with fairly inexpensive telemetry and on-board instruments were used, the cost would drastically drop. Given a reasonable cost for this sort of system of about $ 1500, the program cost would drop to about $ 36,102,000 and unit cost would drop to about $ 36,102 . This is a decrease of 34%. In-house development of electrical systems would easily be cheaper than outside purchase.

Even given the high electronics cost for this system, overall cost is small compared to other RPVs with lower performance. The Aquila RPV, for instance, had a total program cost of $ 15 million through the operational flight test phase for 32 vehicles. Projected cost for a production vehicle were on the order of $ 100,000 . Even allowing for ground equipment costs (Control van about $500 k, launcher or recovery net about $100 k), the $ 55 million figure for production of 1000 vehicles is a very good, inexpensive program cost. This RPV is much more flexible than any other current or projected RPV, and is inherently more cost-effective in any applicable role when compared to other RPVs, thanks to its low cost.

4.4.3 Cost Effectiveness of System

It is best to discuss the cost-effectiveness of RPVs in general before presenting the pros and cons of this system in detail. However, during the general portion of the discussion it would be appropriate to compare the costs of this RPV system to the assumptions for the studies.

Since this RPV is intended for military applications, the cost-effectiveness in those roles will be presented first.

A. Military Cost Effectiveness

It is necessary to establish that this RPV is cost-effective in military roles. Presently all missions except target practice are flown entirely with manned aircraft. One of the missions which RPVs are well suited for is reconnaissance and surveillance.

A comparison of reconnaissance aircraft[25], Figure 4.1, shows the relative performance of operational manned aircraft and the now-defunct Compass Cope RPV. Note that none of these craft cost less than a million dollars per unit, and the greatest on-station time is 12 hours. Figure 4.2 shows how costs of these systems compare under continuous use in areas of tension.

The strike role of the RPV as a highly accurate laser target designator is an extremely effective and important one. As can be seen in Figure 4.3, the RPV in conjunction with the GBU-15 glide bomb or the SUU-54 guided bomb provides excellent standoff range for the manned launching aircraft, with a near-guarenteed strike under fair conditions.

Aircraft	O&M Cost ($/FH)	Total Mission end (Hrs)	End on station (Hrs)	t TR (Hrs)	Attrition rate ** (per 10^5 FH)	Unit cost per aircraft ($M)	Average loiter altitude (10^3 ft)
Compass Cope	475	29.6	27.6	2.0	7.9	3.9*	60.9
U-2	1836	12.0	10.86	1.14	2.5	6.0	
T-43A	848	8.0	7.0	1.0	0.50	6.5	35.0
T-39A	486	3.32	2.40	0.92	0.21	1.8	43.0
RF-4C	1749	2.85	2.05	0.80	4.51	3.14	35.0
A-10	975	9.4	8.05	1.35	2.72	2.2	30.0

* Includes FSD & 100 RPV Buy

** Adjusted for longer sorties

Figure 4.1 Cost and Performance of Reconaissance Aircraft

Aircraft	Annual FH/Year (Hrs)	UE	Attrition A/C (10 Yrs)	Total Buy	10-Yr. O&M Cost ($M)	PME Cost ($M)	Aircraft Cost ($M)	Total Cost ($M)
Compass Cope	3,284	2	3	5	15.6	5.0	19.5	40.1
U-2	3,393	2	1	3	62.3	3.0	18.0	83.3
T-43A	3,554	3	1	4	30.1	4.0	26.0	60.1
T-39A	5,108	7	1	8	24.8	8.0	14.4	47.2
RF-4C	5,460	9	3	12	95.5	12.0	57.6	165.1
A-10	3,633	3	1	4	35.4	4.0	8.8	48.2

Aircraft Comparison,
Squadron Size and Cost

o 200 n.mi. radius o 24 hours on station

Figure 4.2 Cost Effectiveness of Reconnaissance Aircraft

Strike Alternates -- Estimated Costs

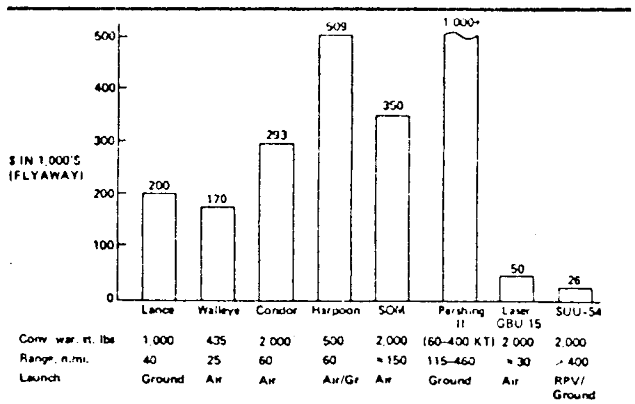


Figure 4.3

Cost Effectiveness –
RPV Tank Army Defense Suppression

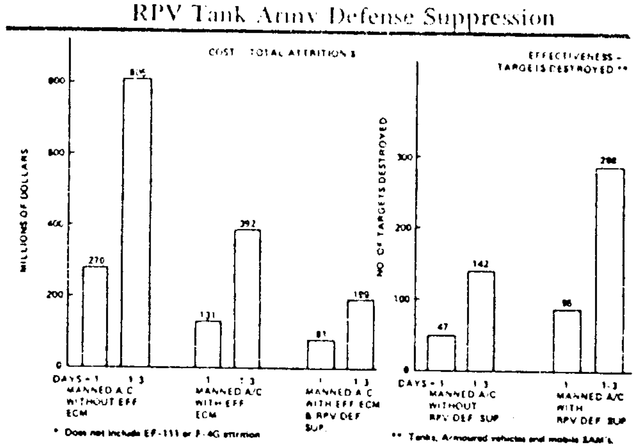


Figure 4.4

Another strike role incolves "harassment" of enemy missle radar installations by long-endurance RPVs fitted with explosives. These types of systems are currently under vigorous develppment, and a key factor is the cheap "throwaway" RPV concept. Northrop is currently developing such a harassment drone[19], with a projected cost of $ 10,000 per unit. One of the main scenerios in the European area is an attack consisting mainly of tanks provided with air cover by radar-guided ground-to-air missles. In this scenerio ECM and harrassment drones have been studied for the first three days of engagement. Figure 4.4 dramatically shows the cost effectiveness of a mixed tank strike force as the RPVs free aircraft for tank strike and disable missle sites.

Another scenerio relavent to the European area is the total airfield disruption task, where airfields are knocked out to effectively ground the enemy air force. An example area was picked for this study (Figure 4.5) which was representative of most possible combat situations. The present tactical situation is shown in Figure 4.6, and comparison of manned only and manned plus strike RPVs is shown. Again, there is a dramatic difference in attack capability, and the RPV becomes a highly effective way to accomplish the task and reduce expensive attrition.

Attrition of RPVs themselves have been considered in the RPV strike role. Although the modeled RPV was sized to deliver 500 pound bombs and be reusable, it is highly possible that smaller RPVs could

Total Airfield Disruption Task

- Twelve airbase Strike zones representative of total E. German capability.
- Northern sector Strike equivalent to ≈ 33% of total conflict.
- Airbases will be struck in sector groups to take maximum advantage of defense surpression.
- Simultaneous Strike of all (45) airbases desirable
- Total airfield disruption requires ≈ 236 equivalent F-4 ordinance loads.

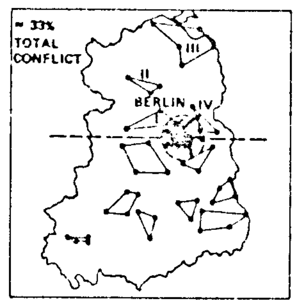


Figure 4.5

Tactical Forces

- The USAF Five Year Plan building to 26 wings, primarily through addition of F-15, -16, and A-10
  - Five active wings air superiority ≈ 360 A/C
  - Six active wings close support ≈ 432
  - Four active wings deep interdiction ≈ 288

  Each mission augmented by active and reserve multi-purpose aircraft.

- Eight TAC fighter wings deployed to Europe (Five U.S. Army Div.) twenty three day reinforcement capability + 16 TAC Air Wings + 10 U.S. Divisions

APPROXIMATELY 300 USAF FIGHTERS AVAILABLE FOR TANK STAGING AREA/AIRFIELD HARRASSEMENT ATTACKS

Airbase Strike Comparison

- Mix of ARPV and manned aircraft
  - 480 ARPV's
  - 236 F-4 sorties
  - 240 Cover aircraft (cap, WW, Flak surpressors).
- ARPV Strike alone
  - 720 ARPV's
- Use of ARPV's alone release the following manned aircraft sorties for combat:
  - 236 F-4
  - 240 Cover aircraft

TOTAL 476 EQUIVALENT AIRCRAFT AVAILABLE

Figure 4.6

deliver smaller bombs more accurately, resulting in the same sort of kill rate per delivery. Characteristics of the weapons systems studied are shown in Figure 4.7 . Studies of total 10 year costs and resource costs are presented in Figures 4.8 and 4.9, where the areas show where the most cost-effective vehicle stands in relation to area and local attrition rates. It must be noted that both the glide bomb and stand off missle usually require some sort of target designation. The RPV is clearly more cost effective than manned systems, and would be much more desirable for laser target designation assuming that both the RPV and the manned aircraft are exposed to the same area attrition rates. Considering the detectability, it is more likely that the RPV would survive.

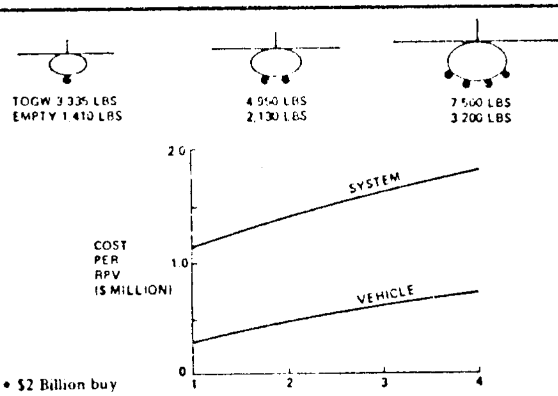
Last, but not least, is there really a need for RPV systems? Figure 4.10 shows the tactical roles that RPVs can assume[27]. It has already been shown that RPVs are more cost effective than non-RPV weapon and delivery systems, but if enough aircraft to perform the mission are already in the inventory, there is really no need for some other system in addition to the current inventory. The effectiveness of the manned air fleet in Europe has already been indicated. However, on an overall basis by task, it can be seen in Figures 4.11 through 4.15 that something is necessary to augment the manned force. RPVs in the various military roles which they accomplish most effectively are one of the best solutions.

The RPV designed, of course, does not fill all the roles that

Analytical Assumptions

Vehicle	System unit $	Unit vehicle $	Relative vulnerability
RPV's			
1 Tgt/sortie	1.149M	0.330M	1
2 Tgt/sortie	1.418M	0.481M	1
4 Tgt/sortie	1.835M	0.723M	1
Glide bomb	0.120M	0.100M	0.75
Stand off missile	0.225M	0.190M	0.75
Supersonic SOM	0.573M	0.550M	0.05
Manned Strike (6 tgt/sortie)	8.0M	8.0M	1
Manned carrier (6 tgt/sortie)	2.0M	2.0M	0.05-0.1 (X ½ area P_K)

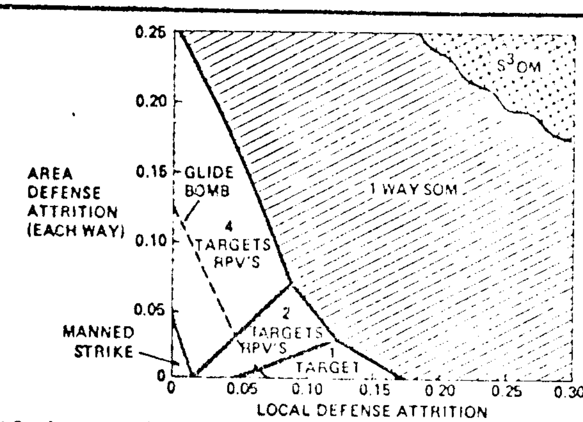
How Big Should A RPV Be?



• $2 Billion buy
• 500 Lbs P/L/target

Figure 4.7

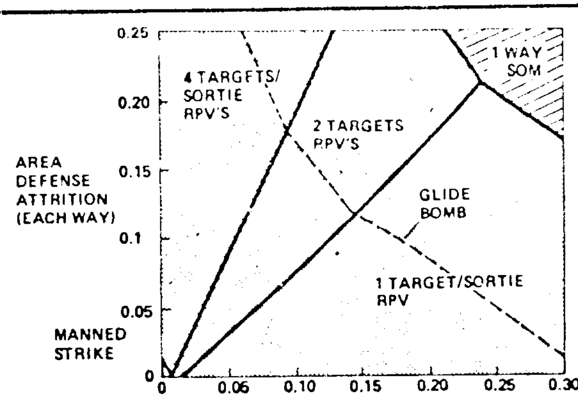
Strike Weapon Options



• Based upon total 10 year costs

Figure 4.8

Strike Weapon Options



• Based upon resource costs

Figure 4.9

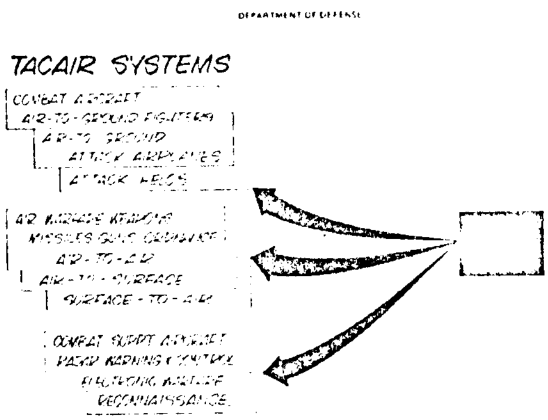


Figure 4.10

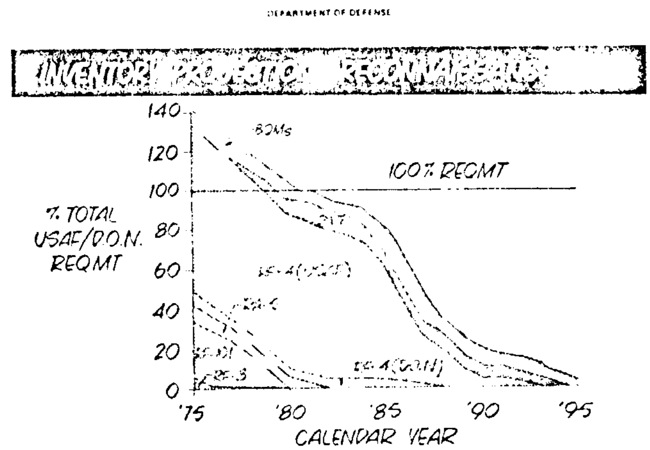


Figure 4.11

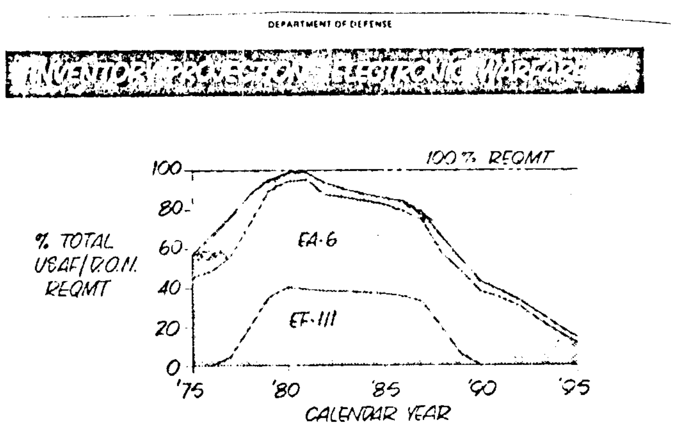


Figure 4.12

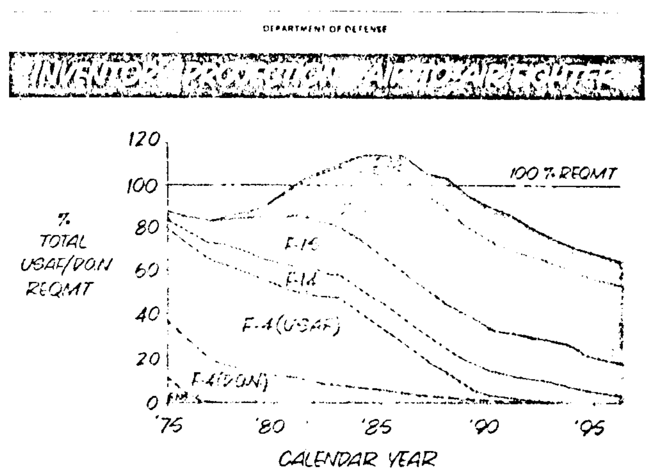


Figure 4.13

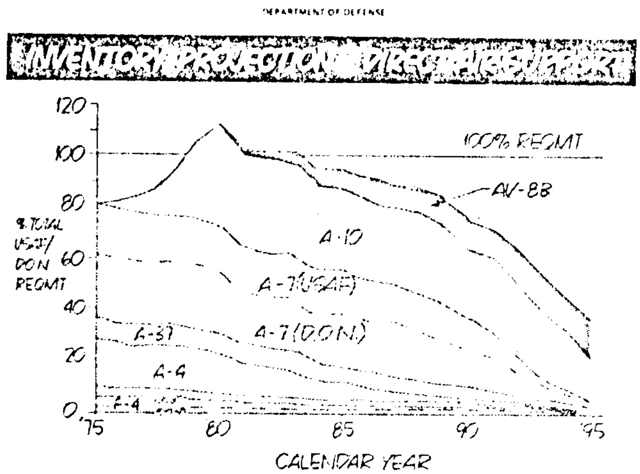


Figure 4.14

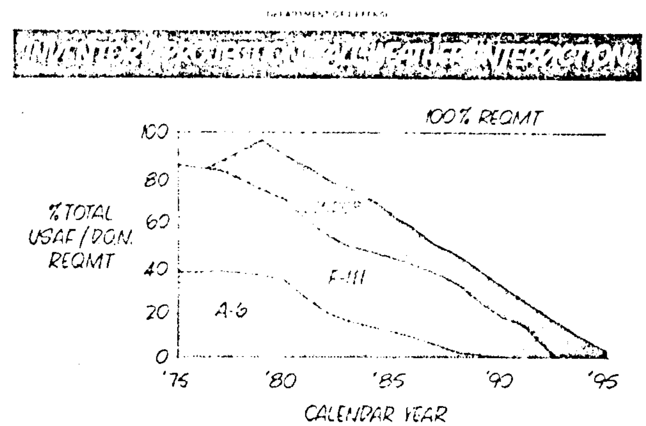


Figure 4.15

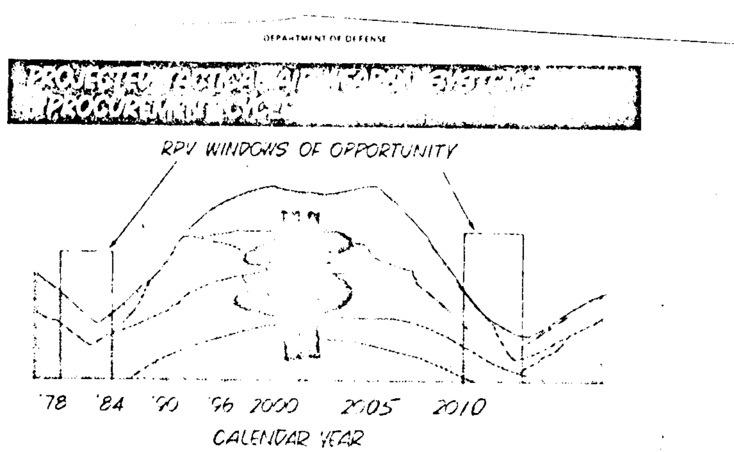


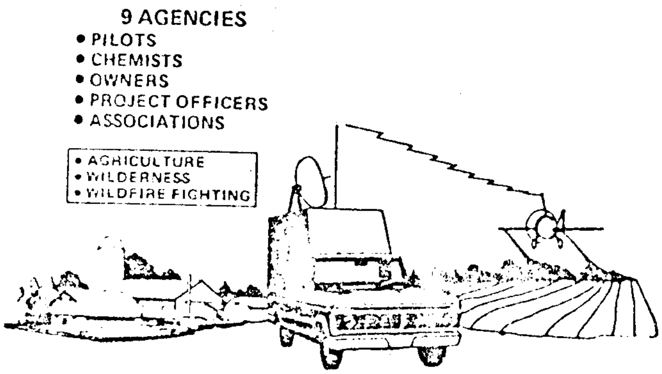
Figure 4.16

RPVs can support. However, it has broader capability and better performance than any other RPV and as such has the best chance of participating in the next TACAIR procurement cycle (Figure 4.16).

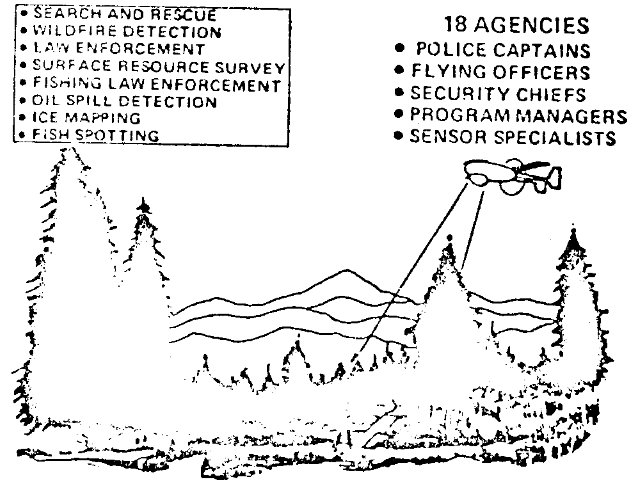
B. Civilian Cost Effectiveness

There are a multitude of missions in the civilian sector which are currently performed by manned aircraft, that can also be accomplished with suitable RPVs. Some of these applications are shown in Figure 4.17 . A study was conducted[28] of some of these missions. Study of cost effectiveness was conducted in a very thorough manner, as the example of security for high-value property mission shows, Figure 4.18 . A few of the many possible applications have been reviewed in this on-going study, and these are presented in Figure 4.19 . Many missions show a cost decrease of 30% to 80% for the RPV over the present systems. During the course of the study, most of the potential users were interviewed, and the desirable RPV characteristics were found, Figure 4.20 .

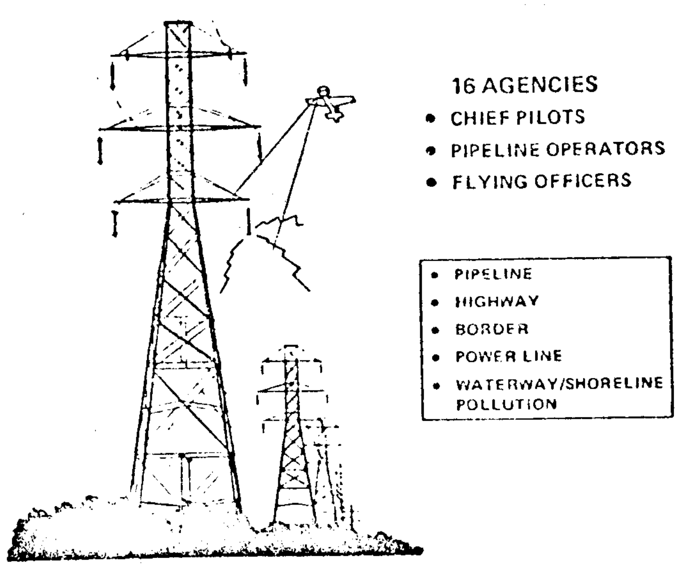
It can be seen that this multi-mission RPV, with its low cost, is well suited to many of these civilian roles. It is quiet, with a long endurance and large payload capacity. Speed is quite sufficient for any mission, with an emergency power system to assure safe landing if the engine fails.



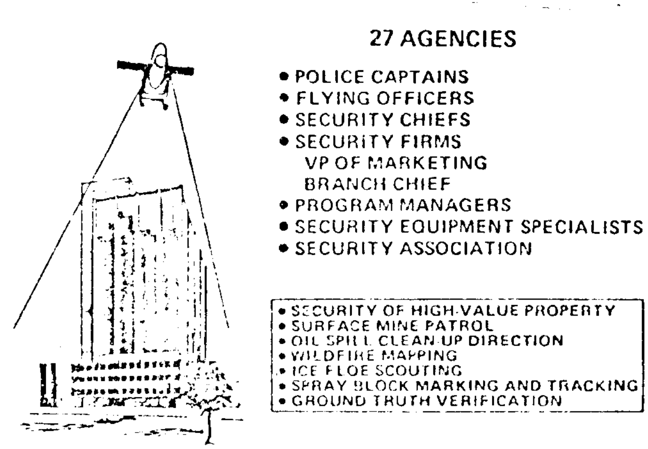
Aerial spraying



Large-area surveillance



Linear patrol



Small-area surveillance

Figure 4.17

Summary of system requirements for security of high-value property

Mission Description
1. Patrol remote highways for accidents, stranded motorists, wanted vehicles, or unsafe road conditions. 2. Follow wanted vehicles. 3. Direct ground units to the scene of a problem.
System Capabilities
1. Patrol anywhere within a 241 km (150 mi) radius of the control station. 2. Operate 8 h/day, 365 days/y, covering 1127 km (700 mi) of highway per day. 3. Fly preplanned daily patrols, with in-flight redirection as desired. 4. Provide real-time visual imagery. 5. Operate at or below 244 m (800 ft) AGL. 6. Distinguish motor vehicles by type, style, make (resolution ≈ 0.5 - 1.0 mrad). 7. Follow a particular vehicle. 8. Locate ground objects well enough to direct ground units. 9. Keep any point under continuous surveillance. 10. Communicate to people on the ground. 11. Keep the highway in the sensor field of view with a minimum of operator attention. 12. Provide a permanent record of imagery. 13. Be operable by a single operator with minimum training. 14. Operation at night and/or in bad weather is desirable. 15. Be less costly than a manned-aircraft system.

System comparisons for security of high-value property mission

SYSTEM FACTORS	SYSTEM REQUIREMENTS		
	HELICOPTER	FIXED WING	RPV
AIRCRAFT	2	1	2
GROUND CONTROL	VOICE	VOICE	GSC IN GUARDHOUSE
PERSONNEL* FLIGHT GROUND	PILOT AND PILOT OBSERVER NONE	PILOT AND OBSERVER NONE	NONE OPERATOR
ALERT LOCATION	LOCAL AIRPORT		PAD NEAR GUARDHOUSE
ENDURANCE (20-MIN RESERVE)	3.3 HR	9.7 HR	1.0 HR
MAINTENANCE	CONTRACTED		

*TWO CREWS PER DAY

Total annual cost comparisons for security of high-value property mission

COST FACTORS	HELICOPTER	FIXED WING	RPV
ANNUAL FIXED COSTS DEPRECIATION INSURANCE HANGAR ANNUAL INSPECTION PERSONNEL	$155,000	$118,000	$74,000
ANNUAL DIRECT OPERATING COSTS FUEL AND OIL PERIODIC INSPECTION MAINTENANCE	$81,000	$54,000	$51,000
TOTAL	$236,000	$172,000	$125,000

Figure 4.18

Weight summaries for example missions

WEIGHT GROUP	CIVIL RPV APPLICATIONS					
	SECURITY OF HIGH VALUE PROPERTY		HIGHWAY PATROL kg (lb)			
			MISSION RPV		RELAY RPV	
	kg	(lb)				
AIRFRAME	13.6	(30)	19.5	(43)	34.0	(75)
PROPULSION	29.9	(66)	7.7	(17)	9.1	(20)
ELECTRICAL	5.9	(13)	5.0	(11)	5.0	(11)
FLIGHT CONTROLS	5.4	(12)	4.5	(10)	5.0	(11)
DATA LINK	2.7	(6)	2.7	(6)	15.9	(35)
PAYLOAD						
ATC TRANSPONDER	–	–	–	–	0.9	(2)
NAVIGATION	–	–	1.4	(3)	1.4	(3)
SENSORS	6.3	(14)	3.2	(7)	1.4	(3)
OTHER	3.6	(8)	4.5	(10)	–	–
EMPTY WEIGHT	67.5	(149)	48.5	(107)	72.6	(160)
FUEL	7.3	(16)	26.4	(58)	31.8	(70)
TAKEOFF GROSS WEIGHT	74.8	(165)	75.0	(165)	104.3	(230)

Summary of civil RPV weights, relative costs, and market size

MISSION	RPV CONFIGURATION	GROSS WEIGHT kg	(lb)	TOTAL ANNUAL* COST DIFFERENCE (PERCENT)	NUMBER OF POTENTIAL RPV SYSTEMS
SECURITY OF HIGH VALUE PROPERTY	ROTARY WING	75	(165)	–25	7500
WILDFIRE MAPPING	ROTARY WING	76	(168)	0	30
WILDFIRE DETECTION				–30	680
MISSION RPV	WING CANARD	445	(980)		
RELAY RPV	WING CANARD	445	(980)		
FISHING LAW ENFORCEMENT	FIXED WING	66	(146)	+200	–
HIGHWAY PATROL				–35	1500
MISSION RPV	FIXED WING	75	(165)		
RELAY RPV	FIXED WING	105	(230)		
PIPELINE PATROL				+130	–
MISSION RPV	FIXED WING	58	(129)		
RELAY RPV	FIXED WING	98	(215)		
AGRICULTURAL SPRAYING	LOW WING MONOPLANE	114	(250)	–25	800
SEVERE STORM RESEARCH	FIXED WING	52	(114)	–40	40
TOTAL					10550

* RELATIVE TO LOCALLY MANNED AIRCRAFT (NEGATIVE SIGN INDICATES RPV ADVANTAGE)

Figure 4.19

Examples of civil RPV technology challenge

PROPULSION
- 5 - 60 hp ENGINES
- 500 hr MTBO
- MORE DURABLE SMALL ENGINES
- LIGHTER BIG ENGINES
- MORE EFFICIENT COOLING SYSTEMS
- THRUST VECTOR CONTROL

AERODYNAMICS
- STABILITY AND CONTROL OF UNUSUAL CONFIGURATIONS
- LOW REYNOLDS NUMBER DATA BASE AND METHODS ($0.2 - 1.0 \times 10^6$)
  - AIRFOIL
  - HIGH LIFT
- EFFICIENCY OF SMALL PROPELLERS, DIAMETER = .3 - .8 m (1.0 - 2.6 ft)
- SHROUDED PROPELLERS
- ELECTRONICALLY STABILIZED VEHICLES
- LOW DRAG COOLING AND VENTILATION SYSTEMS

NAVIGATION AND IN-FLIGHT CONTROL
- TIE-IN WITH EXISTING AIDS
- LINKS TO ATC
- COLLISION AVOIDANCE
- POSITIVE CONTROL COMMAND LINKS
- LOW-COST MULTI-RPV CONTROL
- MIXED VIDEO AND TELEMETRY

LANDING AND TAKEOFF
- V/STOL STABILITY AND CONTROL
- NOVEL METHODS
  - STOWED AUTOGYRO
  - SACRIFICIAL EXTREMITIES
  - SELF-POWERED LAUNCH SYSTEMS

MINIMUM-DAMAGE ABORT
- PARACHUTE BACKUP
- STOWED AUTOGYRO BACKUP
- MULTI ENGINE vs SINGLE ENGINE
- PREPROGRAMMED EMERGENCY LANDING LOCATIONS

ALL SYSTEMS
- LOW COST (ESPECIALLY IMAGING SENSORS)
- FLIGHT QUALITY AT LOW END OF PERFORMANCE SPECTRUM
- ESTABLISH DESIGN PRINCIPLES AND CRITERIA

Figure 4.20

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