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THE DESIGN AND CONSTRUCTION OF A
SHIPLAUNCHED VTOL UNMANNED AIR VEHICLE

by

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The Design and Construction of a Shiplaunched VTOL Unmanned Air Vehicle

by

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ABSTRACT

A Vertical Takeoff and Landing (VTOL) Unmanned Air Vehicle (UAV) was designed to serve as a shiplaunched reconnaissance and over the horizon targeting aircraft. Modeled after the U.S. Army's Aquila, the aircraft features a unique tilting ducted fan propulsion unit. The duct contains the engine, propeller, and control vanes used to provide the VTOL capability and is designed to be rotated as a unit for transition into horizontal flight. The duct also provides a measure of shipboard safety by eliminating the potential propeller blade and other hazards associated with the launch and recovery cycle currently experienced by topside personnel. The advantage of using tilting ducted fan technology is it allows the vehicle to operate off of any ship and will have the dash speed to arrive on station in a timely manner. A 1/2 scale model was built using composite wet lay-up techniques as a technology demonstrator and flight test vehicle. The engine system was tested but failed to produce enough static thrust for vertical takeoff. Research is continuing in the development of a propeller that will provide the necessary thrust.

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I. INTRODUCTION

Unmanned Air Vehicles (UAVs) were once considered expensive toys with little tactical value [REF.1:p.38]. Today they are beginning to emerge as functional military vehicles, capable of replacing or augmenting manned aircraft on routine or hazardous assignments.

> Never send a man where you can send a bullet, said Sam Colt, 19th century inventor and firearms expert. A 20th century variation of this might be Never send a man where you can send a remotely piloted vehicle. [REF.2:p.12]

A. UAV BACKGROUND

The United States involvement with Remotely Piloted Vehicles (RPVs), or pilotless vehicles as they were called, began about 1917, just before the end of WWI. The first attempt was a pilotless biplane used as an aerial torpedo. [REF.2:p.12] In 1919 Elmer Sperry used one to sink a surplus German battleship [REF.3:p.49]. For the most part the initial vehicles were to be used as aerial targets or to carry explosives. Economic restraints following WWI curtailed American RPV research and development. The British, however, continued pilotless aircraft research, and in 1927 the automatic pilot was developed into a practical instrument and was incorporated into the unmanned aircraft to provide for guidance and control. In July of that year, a pilotless aircraft was launched, and successfully flew its 300 mile course. [REF.4:p.260]

During WWII the United States used unmanned aircraft as target drones. By the end of the war, the U.S. military had taken delivery of approximately 14,000 drones. But as the war ended in 1945, so did RPV development. The late 1950's to early 1960's witnessed a resurgence of RPV use and development to fulfill intelligence gathering needs. By 1964 the United States was routinely using RPVs over Southeast Asia for photoreconnaissance, electronic intelligence gathering, bomb damage assessment, psychological warfare (propaganda leaflet dropping), and electronic warfare. Between 1964 and 1975 a total of 3,435 sorties were flown with a survival rate of 84 percent. From 1972 to 1975 the survival rate increased to 90 percent as more sophisticated models were used. After the Viet-Nam conflict the U.S. use of RPVs stopped, and five years later not one operational RPV was left in the inventory. [REF.2:pp.12-13] [REF.5:p.7]

One country that has continued to research and develop RPV technology is Israel. Israel recognized the advantages of RPVs, during the Yom Kippur war of 1973, when it was able to reduce its manned aircraft losses by using inexpensive decoys to confuse Egyptian Surface to Air Missile (SAM) batteries along the Suez Canal. Israel's investment paid off in June 1982, during the "Peace for Galilee" offensive against Syrian forces in Lebanon. Radar reflectors were installed in some of their Scout and Mastiff RPVs to simulate full size aircraft. Syrian radar illuminated the decoys, thinking they were attacking aircraft, and engaged their anti-aircraft guns and missiles on the targets. Another group of Scout and Mastiff RPVs were loaded with explosives and equipped with radar homing equipment. They flew into the area undetected and homed in on the radar emissions, destroying the radar sites and leaving the Syrians blind. Now without radar, the Syrians were vulnerable to attack; Israel sent in manned aircraft to completely destroy the anti-aircraft gun and missile sites. As a result of this combined use of manned and unmanned aircraft, 29 Soviet-made SAM sites were destroyed and not one single Israeli pilot was lost. [REF.1:p.40] [REF.5:p.8]

In 1983 new U.S. interest was generated in the potential use of RPVs as a result of Israel's 1982 success in the Beka'a Valley against the Syrians. In 1985 Secretary of the Navy John Lehman directed NavAirSysCom to implement an RPV program using off the shelf technology so that a unit could be deployed to the fleet as quickly as possible. To accomplish this goal, a technology demonstration or fly-off was conducted from October through December 1985. The conclusion of the demonstration was that the Israeli built Pioneer best satisfied the Navy's needs. [REF.6:pp.15-16]

The Pioneer, based on the Scout, has a wingspan of 16.9 feet and a length of 14 feet. It is equipped with a 26 hp engine, and will reach a maximum speed of 115 mph. The Pioneer has an endurance of 8 hours at an altitude of 15,000 feet, a cruising speed of 92 mph, and a payload of 100 pounds. [REF.6:p.16] In April 1986 the Pioneer system was installed on board the USS Iowa (BB-61). A rocket-assisted takeoff, as opposed to a catapult, is required for takeoff, and a net is used for shipboard recovery. After a brief period of failures, the Pioneer successfully demonstrated its capabilities and has been flying ashore and afloat ever since. [REF.6:pp.15-17]

B. ADVANTAGES AND USES OF UAVS

The key to the success of a UAV rests in the diversity of its payload. These are the sensors and other electronic equipment installed for the various missions. Examples of payloads are cameras, forward looking infrared radar (FLIR) for night vision, communications equipment for aircraft to ground data links, and radar. Given the sophistication of today's hardware and software, equipment can be kept small to allow for small airframes which can evade radar detection, and can carry out a broad spectrum of tasks.

UAVs can be used as reconnaissance aircraft; this use can provide field commanders with real time information. With the advent of high altitude/long endurance UAVs, one can remain on station for up to 38 hours or more, providing continuous reconnaissance information. Adding a radar to the UAV can provide an all-weather reconnaissance capability. UAVs can be used as spotters for artillery, naval gun fire support or to provide laser designation of targets for laser-guided projectiles, bombs, or missiles. This use can alleviate the need for forward observers on the ground, or manned spotter planes. UAVS can be used to measure radiation, chemical, or biological contamination. They can be sent out to gather weather information. They can be used as ASW aircraft by monitoring and relaying sonobuoy information. UAVs can be used to relay friendly communications or to jam enemy communications and radars. They can be used to gather enemy signal intelligence, or they can be used as decoys to protect friendly aircraft. The central theme among all these uses is the advantage of assigning tasks or missions to an unmanned aircraft that would have exposed human pilots to considerable risk. This not only may save the life of a pilot but also the cost of an aircraft. If designed properly the UAV can be considerably cheaper than its manned counterpart. Problems arise when sensors are installed in the UAV. The cost can escalate and the potential for a costly failure greatly increases. For example, the U.S. Army's Aquila moved from $30,000 to over $1.5 million per unit by continual increase of mission requirements [REF.3:49-50].

> One Army officer even proposed painting the UAVs day-glow orange to invite the enemy to expend resources. He's absolutely right. It costs a lot more to try to shoot down a UAV than the UAV is worth, if you design the UAV properly in the first place. [REF.3:p.50]

The best use for UAVs is to augment manned fighter aircraft, each complimenting the other with the unmanned aircraft <u>taking enough heat</u> to reduce the threat to the pilot.

C. ARCHYTAS CONCEPT

The Pioneer has successfully demonstrated its capabilities on board the USS Iowa. It has also revealed a major shortcoming of a conventional take-off and landing UAV. Due to lack of room topside on the Battleship, the normal pneumatic launch system has been replaced by a rocket-assisted takeoff, and because there is no room for a runway a tripod recovery net is used. This system has worked well on the Battleship. However on a Frigate or a Destroyer, neither has the room for the launcher nor the recovery net. Therefore in order to take greater advantage of purposed future UAV systems, these systems will have to incorporate a Vertical Takeoff and Landing capability. This requirement is the basis of the Archytas Tilting Ducted Fan (TDF).

The Archytas TDF was designed to take advantage of two existing systems. The basic airframe is based on the Army's Aquila. The engine and duct assembly is based on the Marine Corps Airborne Remotely Operated Device (AROD). One major problem with the AROD was that the aircraft was designed with the engine hard mounted to the airframe, leading to vibration-induced failure. To alleviate this problem the Archytas' engine is shock mounted to its engine mount with heavy duty rubber mounts. One major advantage of the AROD was its controller which has earned the praise of all who test flew the aircraft. The first construction model of the Archytas TDF is a half scale technology demonstrator and does not currently contain a controller. Follow-on research will include the construction of a full-scale Archytas TDF which will use the controller.

Today's aircraft design buzz word is stealth. The Israelis were able to send UAVs into the Beka'a Valley undetected. The object of this program is to provide the Navy with an aircraft that also can penetrate the enemy's defense undetected and conduct reconnaissance. The Aquila airframe provides that low radar cross section needed and for this reason was picked as the basic design for the Archytas airframe. The Naval Postgraduate School UAV Flight Research program has obtained a full-scale Aquila which will be used for transitioning to the full-scale Archytas TDF.

One problem noted with an early version of the Aquila was its stability. To counter this problem the Archytas was designed with a tail to correct the stability limitation. Finally, besides stealth and stability, the Archytas was designed with one additional concern in mind, that of safety. The ducted fan was picked over a tilt rotor because in a tilt rotor the blades are exposed; trying to handle a UAV with exposed rotors topside with a pitching deck or moderate winds presents an extreme hazard during takeoff and landing cycles. With the propeller contained inside the shroud, this hazard is eliminated.

A ducted fan or ducted propeller is basically a propeller inside a shroud. The usefulness of a ducted fan aircraft depends on its speed. It is a better thrust producer at low speeds, but is not efficient at higher speeds. A ducted fan is quieter than a standard propeller because the fan is about .7 times the diameter of an equivalent propeller and can be run at a higher rpm without exceeding the critical Mach number of its tips. A major disadvantage of a ducted fan is its duct drag. The duct drag is based on the projected area of the duct S_d.

$$S_d = 2\pi r_d c_d \tag{1}$$

Where r_d and c_d are the duct radius and duct chord respectively. The projected duct area S_d is then used in the standard drag equation, with a scaling term to take into account the interference drag from the supports. A standard drag coefficient of C_d = 0.01 is used with a scaling factor of 1.5. The duct drag D_d is then calculated by

$$D_d = 0.015\frac{q}{S_d} \tag{2}$$

where

$$q = \frac{1}{2}\rho V^2 \tag{3}$$

Figure 1 illustrates the advantages of standard and ducted propellers.

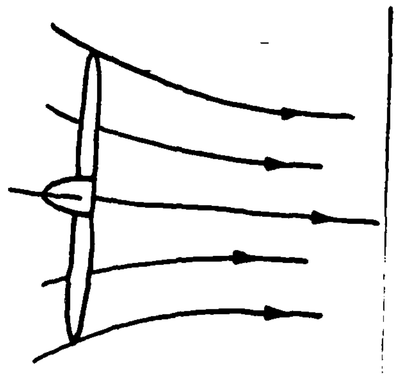
[REF.7:pp.309-311]

D. NAVAL POSTGRADUATE SCHOOL UAV FLIGHT RESEARCH PROGRAM

The Naval Postgraduate School Flight Research program is in support of the UAV Joint Project Office of NAVAIR and began its UAV related research in 1987. The goal is to establish a testbed of radio controlled unmanned air vehicles, capable of demonstrating new ideas and concepts. But with any new idea or concept there are high risks or potential hazards involved. By using UAVs for flight research, the potential for loss of life is eliminated, and the potential financial loss due to some mishap is significantly reduced.

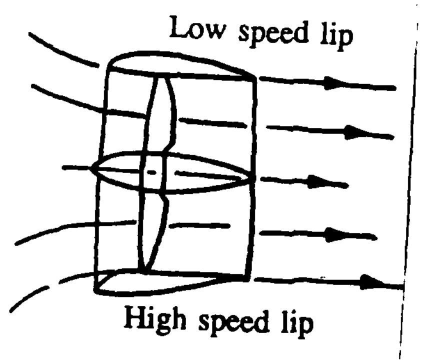
The NPS UAV Flight Research program has established a wide variety of UAVs for research and development. The program currently has an F-16 model instrumented for high angle of attack, agile fighter, research. Currently under construction is an F-18 model that will be equipped and instrumented for dynamic parachute recovery. The flight research program maintains two model helicopters for research of higher

A. STANDARD PROPELLER



- Better at speeds greater than 80 to 100 knots
- Lighter
- Less efficient at high rpm

B. DUCTED FAN (showing separate high and low speed lip profiles)



- Requires odd number of blades to prevent resonance
- Best at speeds less than 80 knots
- Heavier than a propeller
- More efficient at high rpm
- Quieter than a propeller
- Suffers duct drag

Figure 1 Comparison between propeller and ducted fan

harmonic control and vibration reduction. The program just acquired a Mini-Sniffer from NASA for high altitude/long endurance UAV research. The flight research program also maintains a Marine Corps Exdrone, an Army Aquila, and finally a half scale Pioneer. The half scale Pioneer is currently the most advanced flight test vehicle in the inventory. It is currently instrumented with alpha and beta vanes, a pitot static system, a seven channel on-board recorder, and a telemetry unit. The Aquila described earlier will be transformed into a full-scale Archytas TDF complete with a controller, tracks and motor for rotating the duct for transition from vertical to horizontal flight, and will be fully instrumented for gathering flight test data.

This thesis encompasses the design and construction of the Vertical Takeoff and Landing (VTOL) portion of the Archytas. A concurrent thesis titled "Design and Construction of a Composite Air Frame for UAV Research" by Ellwood, discusses the design and construction of the horizontal flight mode components of the Archytas [REF.11]. This includes making the airframe, tail and landing gear. As a follow-on project the Archytas will be tested in three configurations: tailless (thrust vectoring only), long tail, and short tail.

II. ARCHYTAS DESIGN

Before an aircraft can be designed, the designer must know the established requirements. These requirements are called the mission requirements and they include the aircraft's purpose, payload, speed, range, endurance, etc. These requirements are important because they drive the design and are the means of determining the success of the design. [Ref.8:pp.1-3]

The mission that dictated the design of this aircraft is that it is to be a technology demonstrator and a transition vehicle for a full-scale Archytas to be built as a follow-on project. The design was tailored after the U.S. Army's Aquila which was initially designed in the mid 1970's. The Aquila was chosen because of its low radar cross section and the availability of an airframe at the Postgraduate School for the follow-on conversion to a full-scale Archytas.

To successfully perform its mission the Archytas must be light weight, be stable in both vertical and horizontal flight, be able to hover, and have very forgiving flight characteristics for initial flights. The Archytas must also be similar to the Aquila in general appearance in order to facilitate the conversion to the full-scale vehicle. Finally the vehicle design must be simple for ease of construction.

Designing a VTOL aircraft brings a number of unique problems. Two fundamental problems stand out the most because they have the greatest impact on the design of the aircraft. They are balance and thrust matching. Most military aircraft today have the engines in the rear, the avionics in the front, and the fuel and payload (stores) near the center where the center of gravity is located. This design minimizes the effect of weight change on the movement of the center of gravity. As long as the thrust to weight ratio is greater than 1.0 the aircraft will accelerate vertically; however, this design does not lend itself to vertical flight. [REF.9:p.528]

There are two approaches to solving this design problem. The first is to move the thrust to the center of gravity. The second is for the thrust to come from two locations equally matched to provide a balanced force. In this design two engines would be required. One would be located in the rear to provide the standard thrust for normal horizontal flight and would be diverted downward for lift off. A second engine in or near the front would be used to provide the balanced force for lift off, and then would be shut off during forward flight (Figure 2).

Thrust matching was the other fundamental problem. If the engines used for horizontal flight are the same engines used for vertical flight, then the engines required may not be efficient during horizontal flight. This is caused by having an engine large enough to produce the necessary thrust for lift off but may be forced to operate at a thrust setting that is not optimal for cruise efficiency. [REF.9:p.529]

Other problems associated with VTOL design are transition, control, ground effect, foreign object damage (FOD), and weight. Several designs are available for VTOL; some include nozzle-vectoring thrust, tilt nacelle at the center of gravity,

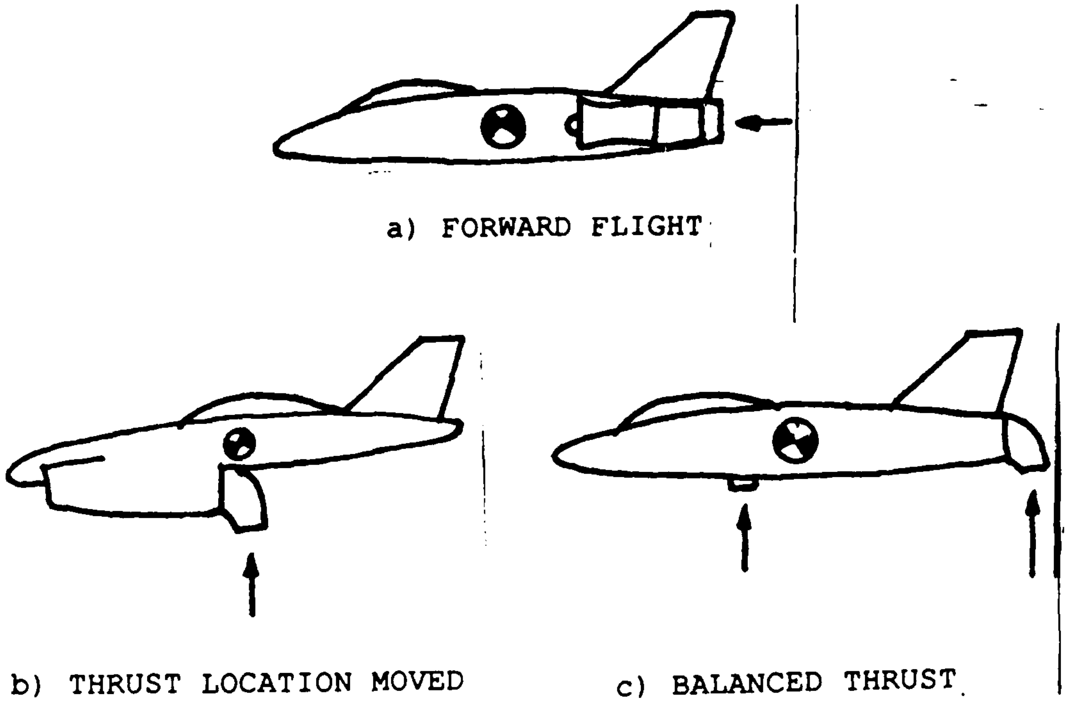


Figure 2 The balance problem

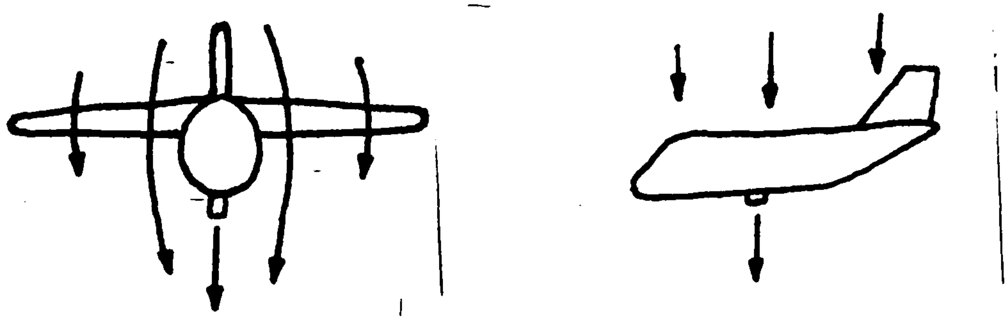
multiple tilt nacelles, tilt wing, fan in wing, and thrust augmented wing. [REF.9:p.531] [REF.10:p.13:11]

The problem of transition will not be covered here, but is intended to be the subject of follow-on study. The problem of control will be discussed later.

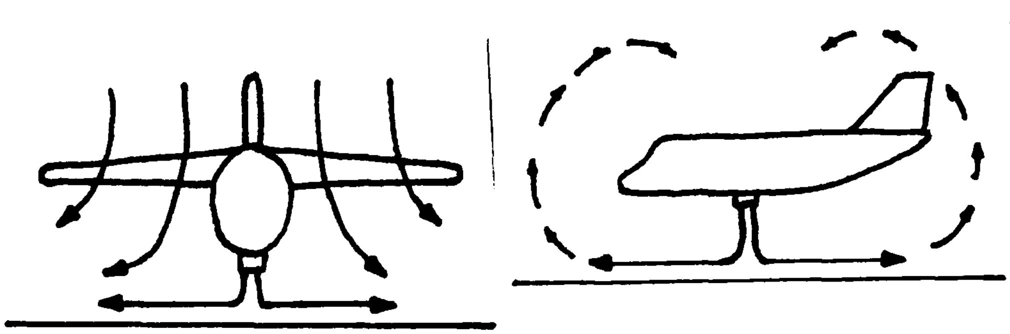
Ground effect problems are based on the location of the engine and duct (nozzle). As the VTOL aircraft is hovering the exhaust that supports the aircraft is also accelerating the airmass around it downward. This creates a flowfield which pushes down on the aircraft (Figure 3a). When a VTOL aircraft with a single nozzle (duct), located at the center of gravity, is near the ground the exhaust striking the ground spreads out along it, thus creating a cyclic effect which increases the mixing and therefore increases the downward force (Figure 3b). This downward force, also called a suckdown force, increases as the aircraft nears the ground. When a VTOL aircraft with separated nozzles (ducts) is near the ground, the exhaust striking the ground produces the same cyclic reaction with the air flow. However in this case instead of the airflow only creating the suckdown force, it also creates an upward force. This upward force, also called fountain lift, pushes the VTOL aircraft upward often countering the suckdown force (Figure 3c). This fountain lift force increases as the aircraft approaches the ground. [REF.9:pp.540-541]

Weight plays a major part in the design of a VTOL aircraft. The aircraft must have a static thrust to weight ratio greater then 1.0; otherwise the aircraft will not get off the ground. However, the design requirements of the VTOL aircraft increase the weight. This weight increase is due to the need of a larger propulsion system to provide the necessary static thrust. This system includes the ducting and nozzles needed to divert the thrust and solve the problem of balance and thrust matching. The control system also increases the weight because of the equipment needed for transitioning between vertical take-off and horizontal flight. The VTOL design does allow for some weight reduction, such as in lighter landing gear. [REF.9:pp.545-547]

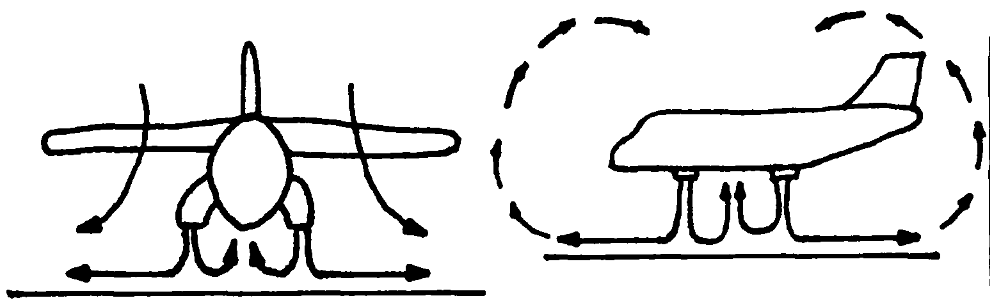
The first step in designing the Archytas was to take measurements of the Aquila and begin down sizing them to half scale to obtain general Archytas dimensions. Next scale drawings were made of the wings, duct, and fuselage. At this point the design process was divided into five categories: the fuselage, wing, tail, landing gear and



a) Flowfield



b) DUCT AT C.G. GROUND EFFECTS



c) MULTIPLE DUCT GROUND
EFFECTS - FOUNTAIN LIFT

Figure 3 Suckdown and Fountain Lift

engine/duct system. This thesis will only cover the part of these categories dealing with vertical flight. The categories contained in the following sections are the fuselage, wing, and engine/duct system. Additional information on the design and construction of the fuselage, tail, and landing gear can be found in reference 11.

A. FUSELAGE DESIGN

The VTOL design that was chosen was the tilt nacelle (duct) at the center of gravity. The fuselage was then designed with two ideas in mind. The first was to keep the basic shape similar to the Aquila, and the second was to design the fuselage around a duct to be centered at the center of gravity. Other important driving points were duct mounting, avionics, fuel, eventual flight test equipment, and airflow through the duct. To aid airflow into the duct, the nose was flattened and the top portion of the fuselage tapered into the duct area. Initial required avionics are the receiver, rate gyros, and battery packs. These items do not require much room; however, a follow on thesis will involve the design of a suitable controller and the size required at this point is unknown.

From the initial design process it was determined that the fuselage will be about 34 in. long, and about 18 in. wide. The center of gravity was designed to be about 24 in. from the nose. This location of the center of gravity was chosen to conform to the center of the duct and allows the thrust line to be at the center of gravity during vertical flight.

Four areas or bays were left open in the fuselage. The nose bay is used for the battery packs and eventual flight test equipment, the center or avionics bay is used for avionics, and the left and right side bays are used for the fuel tanks and for mounting hardware for the duct and wings. These bays are also used to shift ballast as necessary to maintain the center of gravity at the thrust line.

To prevent the center of gravity from shifting as fuel is burned, two interconnected fuel tanks are used. Both fuel tanks are located at the center of gravity, one in each side bay, and are connected by a fuel pump which draws fuel equally. It was estimated that each test flight would not exceed 30 minutes and that 24 ounces of fuel should be sufficient. This would require two 12 ounce fuel tanks. Space is available to accommodate larger fuel tanks if necessary. The spar box was also part of the fuselage design and will be discussed in the next section with the spar design.

B. WING AND SPAR DESIGN

A span of 6 feet was selected as it is roughly half the wing span of the Aquila. The aspect ratio was calculated using equation 4 [REF.10:p.193], and is defined as:

$$AR = \frac{b^2}{S} \tag{4}$$

where b is the wingspan (Figure 4) and S is the reference wing area. This area includes the fuselage area on either side of the duct. The aspect ratio of 4.5 is rather low compared to other propeller aircraft as shown in Table 1 [REF.9:p.51]. The smaller aspect ratio means a smaller wing. This is necessary to minimize the

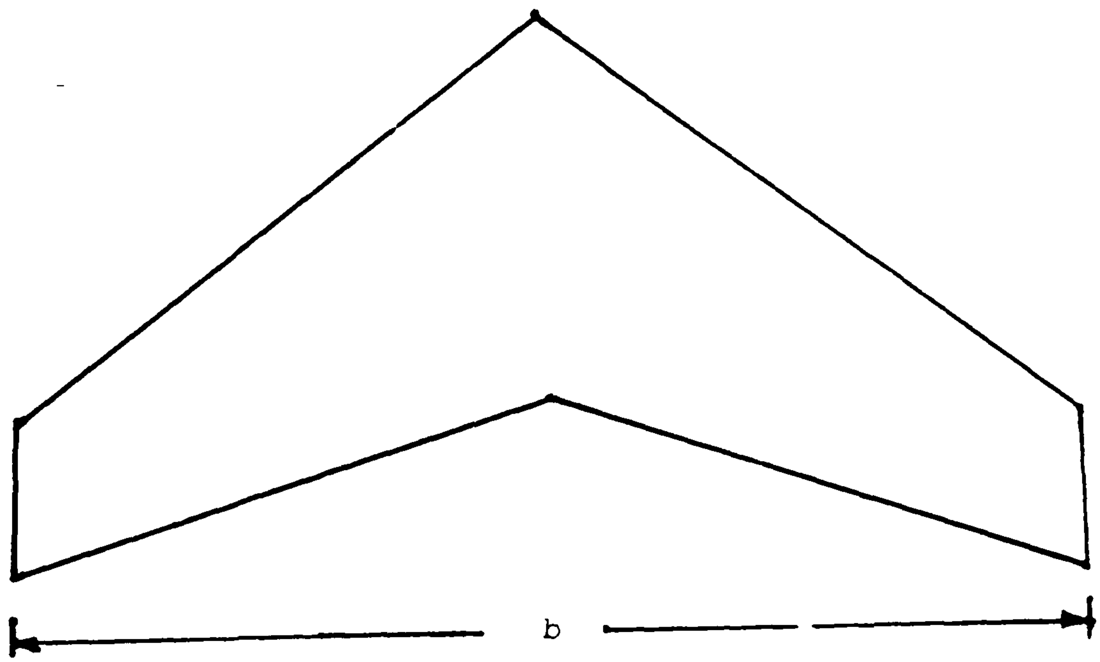


Figure 4 Wingspan Of A Finite Wing, Plan View (Top)

deckspace used, a requirement for small ships. The smaller aspect ratio also allows for stalling at a higher angle of attack which makes the aircraft more maneuverable and since there is no pilot onboard the aircraft was designed for 25 g's.

The wing has a wing area of 1157 in^2, a root chord of 18.38 in and a tip chord of 11.69 in. The wing has a 28 degree leading edge sweep to cut down on the radar cross section and for stability, and 2 degrees of dihedral. This dihedral combined with the sweep of the wing produces a stabilizing rolling moment due to the sideslip, which is important in the lateral stability and control of the aircraft. [REF.10:pp.4-14]

Wing loading is the weight of the aircraft divided by the area of the wing. Wing

TABLE 1 ASPECT RATIO

PROPELLER AIRCRAFT	ASPECT RATIO
Homebuilt	6.0
General aviation single engine	7.6
General aviation multiple engine	7.8
Agricultural aircraft	7.5
Twin turboprop	9.2
Flying boat	8.0

loading affects stall speed, climb rate, turn performance and takeoff and landing distance. The wing loading determines the lift coefficient at a given speed and affects drag through the wing size. Lower wing loading improves take off performance and turning ability but increases drag and weight due to the larger wing area. The designer, to ensure that the wing provides enough lift, should select the lowest possible wing loading but must also balance this with the required thrust to weight ratio. Table 2 provides some sample wing loadings. [REF.9:p.84]

Based on a gross weight of 30 lbs, the calculated wing loading for the Archytas is 3.75 lb/ft^2. As compared to the values in Table 2 this calculated wing loading is rather low, but it is high compared to the 1/2 to 1 lb/ft^2 of most models. To keep this in perspective, the Archytas is a 1/2 scale model. To size the Archytas to full-scale requires increasing the weight by a factor of 8 and the wing area by a factor of 4. This means the 30 lb 1/2 scale Archytas is actually equal to a 240 lb full-scale Archytas. This would provide a wing loading of about 7.5 lb/ft^2.

The length of the mean aerodynamic chord (m.a.c.) and the distance from the root chord leading edge to the leading edge of the m.a.c. are calculated from equations 5 and 6 respectively [REF.12:p.90].

TABLE 2 WING LOADING

AIRCRAFT HISTORICAL TRENDS	TYPICAL WING LOADING (lb/ft^2)
Sailplane	6.0
Homebuilt	11.0
General aviation single engine	17.0
General aviation twin engine	26.0
Twin turboprop	40.0

$$m.a.c. = \frac{2}{3}\left(a+b-\frac{ab}{a+b}\right) \tag{5}$$

$$m = \frac{s(a+2b)}{3(a+b)} \tag{6}$$

Where a is the root chord length and b is the tip chord length. The graphical solution for the mean aerodynamic chord, assuming no twist, is contained in Figure 5 [REF.13:p.92].

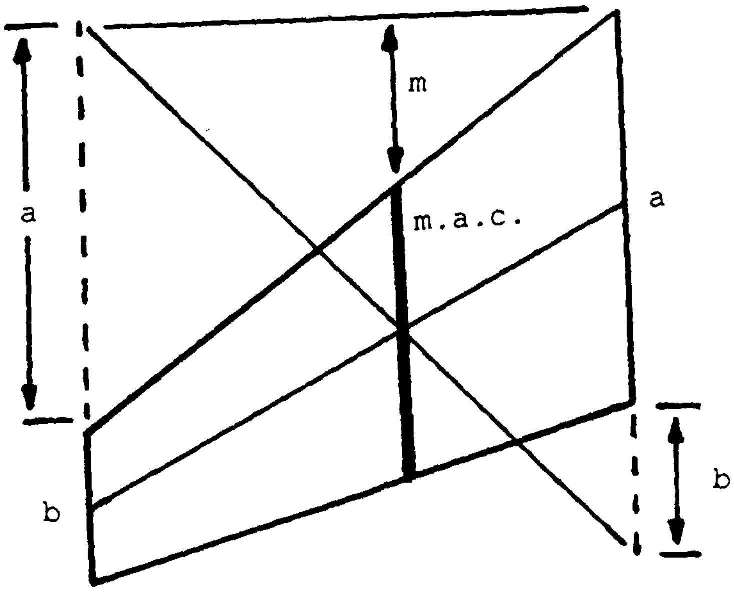


Figure 5 Graphical solution for mean aerodynamic chord

The wing aerodynamic center was estimated at the 0.25 m.a.c. [REF.9:p.49], and the aircraft center of gravity was set at the 0.30 to 0.32 m.a.c.based on the location of the center of the duct.

For roll control purposes, the wing is equipped with a pair of ailerons located 13.4 in. outboard of the wing root. They are 12 in. wide, and have an average chord of 3.65 in. Each aileron has 43.8 in^2 of area. This provides an aileron to wing surface area ratio of 0.076. To lower the approach speed and attain better glide path control, a pair of flaps were installed 2.4 in. outboard of the wing root. They are 9 in. wide, and have an average chord of 4.2 in. Each flap has 37.8 in^2 of area.

For ease of storing the Archytas, the wings and fuselage were designed and constructed in separate parts. The wing is joined to the fuselage through the spar and spar box. The two spar boxes transfer the load by means of a structural support through the fuselage. This support will be shown in the section on fuselage construction. The wing was constructed of composite materials and will be described in the section on wing/spar construction.

The wing spar is one of the most important parts of the wing construction. The wing spar must be able to transfer wing bending moments and shear loads along its length to where it attaches to the fuselage [REF.14:p.67]. The most common type of spar is a box spar, but to save on excess weight the Archytas was designed with an "I" spar.

To begin sizing the spar, the shear load and bending moment must be calculated in order to determine the spar cap and shear web thicknesses. First a constant airload force is assumed along the length of the wing. This will provide the shear load and bending moment as a function of the wing station[1]. The airload is defined as:

[1]The wing stations are positions along the wing. The wing is divided into 10 sections where section 0 is the center of the fuselage and section 10 is the wing tip.

$$L_a = \frac{2Wn\left(C_r+2C_t\left(\frac{X}{B}\right)-2C_r\left(\frac{X}{B}\right)\right)}{B(C_r+C_t)} \tag{7}$$

Where:

W = gross weight of the aircraft less wing weight (lbs)
n = limit flight load factor
B = wing span (ft)
C_r= wing root chord (ft)
C_t= wing tip chord (ft)
X = wing station distance (ft)

The airload is also defined as:

$$L_a = \frac{dL_s}{dX} \tag{8}$$

Where:

L_s = shear load
X = wing station distance (ft)

Rearranging the equation to get,

$$dL_s = L_a dX \tag{9}$$

Now integrating both sides,

$$\int dL_s = \int L_a dX \tag{10}$$

Produces the shear load,

$$L_s = \frac{2Wn\left(C_r X - C_r\left(\frac{X^2}{B}\right) + C_t\left(\frac{X^2}{B}\right)\right)}{B(C_r + C_t)} - W\left(\frac{n}{2}\right) \tag{11}$$

Now integrating equation 11 with respect to X defines the bending moment[2]:

$$M = \frac{2L_d n\left(C_r\left(\frac{X^2}{2}\right) - C_r\left(\frac{X^3}{3B}\right) + C_t\left(\frac{X^3}{3B}\right)\right)}{B(C_r + C_t)} - Wn\left(\frac{X}{2}\right) - \frac{WnB(2C_r + C_t)}{12(C_r + C_t)} + Wn\left(\frac{B}{4}\right) \tag{12}$$

Knowing the wing bending moment, the upper and lower spar cap thicknesses can be calculated using equations 13 and 14 respectively.

$$T_1 = \frac{2M}{yHF_c} \tag{13}$$

$$T_2 = \frac{2M}{yHF_t} \tag{14}$$

Where:

- T_1= upper spar cap thickness (in)
- T_2= lower spar cap thickness (in)
- M = wing bending moment (in*lbs)
- F_c= ultimate compressive strength of the cap material
- F_t= ultimate tensile strength of the cap material
- y = width of the spar cap (in)
- H = height of the spar cap (in)

---

[2]The area being integrated is the area from the wing tip to the wing station. Therefore the limits of integration are from the wing tip (0) to the wing station (X in ft).

Since most composite materials have a higher compressive strength then tensile strength, the upper cap which is in compression is normally thicker then the lower cap which is in tension.

Finally, since the wing shear load is known from equation 11 the wing shear web thickness can be calculated from equation 15.

$$T_3 = \frac{2L_s}{F_s H} \tag{15}$$

Where:
T_3= shear web thickness (in)
L_s = wing shear load (lbs)
F_s= shear strength of the material (psi)
H = height of the spar cap (in)

The previous structural analysis and sizing of the wing spar is combined in a computer program called SPAR which is contained in reference 14.

[REF.14:pp.44-176]

The SPAR program assumes a constant force distribution along the wing and calculates the airload, wing shear load, and bending moment, and sizes the spar cap and shear web thicknesses. The required inputs are the gross vehicle weight less the wing and fuel, the flight limit load factor, the wing span, and the root and tip chords. The SPAR program then sizes the spar by calculating the spar height, cap thickness, and web thickness, from the inputs of ultimate compressive strength, ultimate tensile strength, the shear strength of the material, the spar width, and the percent chord thickness. Since the upper spar cap is in compression and the lower spar cap is in tension separate computer runs are required for each spar cap. The program uses a safety factor of 2, which is common for composite materials. The spar that is determined by the SPAR program is a box spar with two shear webs; thus the thickness given is for each. Therefore if only one shear web is to be used, the thickness must be multiplied by 2.

The SPAR program was run for the Archytas. For the initial inputs, 24.5 lbs was used for a gross vehicle weight less wing (total wing weight is 5.5 lbs). The limit load factor was 25 g's. Since there is no pilot onboard, the aircraft is not limited to 8 g's, and the load factor selected provides for a highly maneuverable aircraft. The wingspan was 6 feet, root chord was 1.53 ft (18.38 in.), and the tip chord was .974 ft (11.69 in.). The fiberglass used for the upper and lower spar caps was an 8 ounce unidirectional cloth which possesses an ultimate tensile strength of 70,000 psi, and an ultimate compressive strength of 43,500 psi. The fiberglass used for the shear web was an 8 ounce bidirectional cloth which possesses a shear strength of 7,300 psi. The final inputs were the spar width of 1.5 inches and the chord thickness in percent chord of 14.13. The computer results of this calculation are tabulated in Appendix A.

The wing spar box was designed in a similar way. Since this is the major connection between the wing and the fuselage, an additional margin of safety was included in the design. The spar box incorporates two shear webs, but instead of the thickness calculated by the SPAR program a greater thickness was used. The spar boxes were fit into two areas forward in the two side bays. The wing spar slides into the spar box and is secured in place by a connecting pin. The pin enters the spar box, passes through the wing spar and then exits the spar box. The spar box is secured to the fuselage by means of epoxy and bidirectional fiberglass. Figures 6, 7, and 8, show the details of the spar and spar box design.

C. ENGINE MOUNT AND DUCT DESIGN

The VTOL aircraft in hover and transition must be controlled by adjusting and redirecting the thrust and airflow. In addition to the standard three axis control, (yaw, pitch, and roll), the VTOL aircraft needs vertical velocity control. This is accomplished by varying the engine throttle. [REF.9:pp.543-544] In the case of the Archytas there are two additional concerns. These are maintaining the vehicle perpendicular while ascending, and countering the gyroscopic effect of the propeller. These are accomplished by the use of control vanes.

The basic design of the Archytas is based around the engine and duct. To accommodate the desired engine, the duct was designed to have a maximum outer diameter of 12 in. The duct is 10 in. long and has a 0.25 in. draft angle from the top to the bottom, to release it from the mold. The duct was made of composite materials, comprising 3 layers of laminated 1/16 in. balsa wood and four layers of fiberglass cloth (two 3 ounce and two 8 ounce). The duct was fastened to the engine mount assembly.

The engine mount assembly was made from aluminum. The main support for the engine mount assembly is an aluminum cylinder (called the support cylinder) (Figure 9). This cylinder is 6 in. long and has a diameter of 2.5 in. On top of the support cylinder is the motor mount. The motor mount (Figure 9), another machined

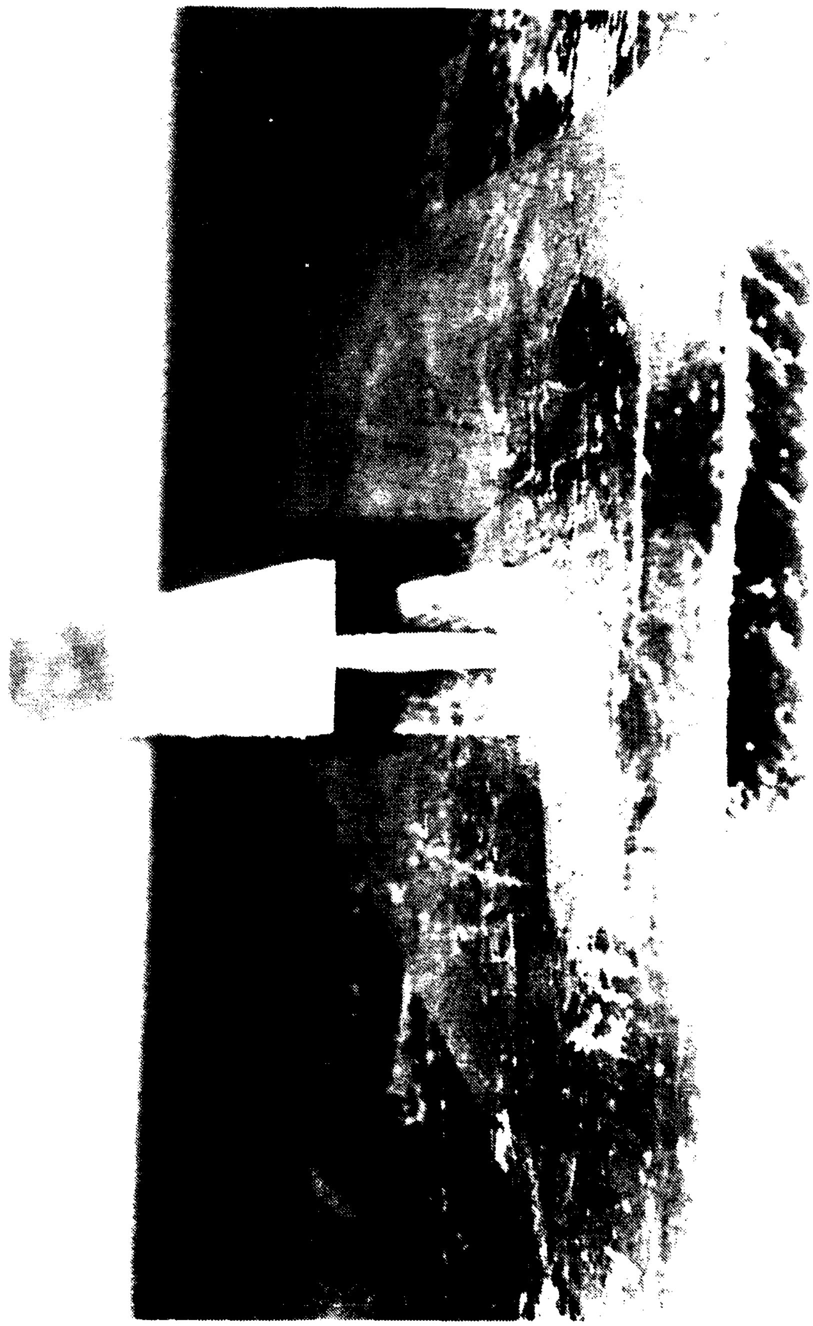


Figure 6 End view of spar

Figure 7 Side view of Spar

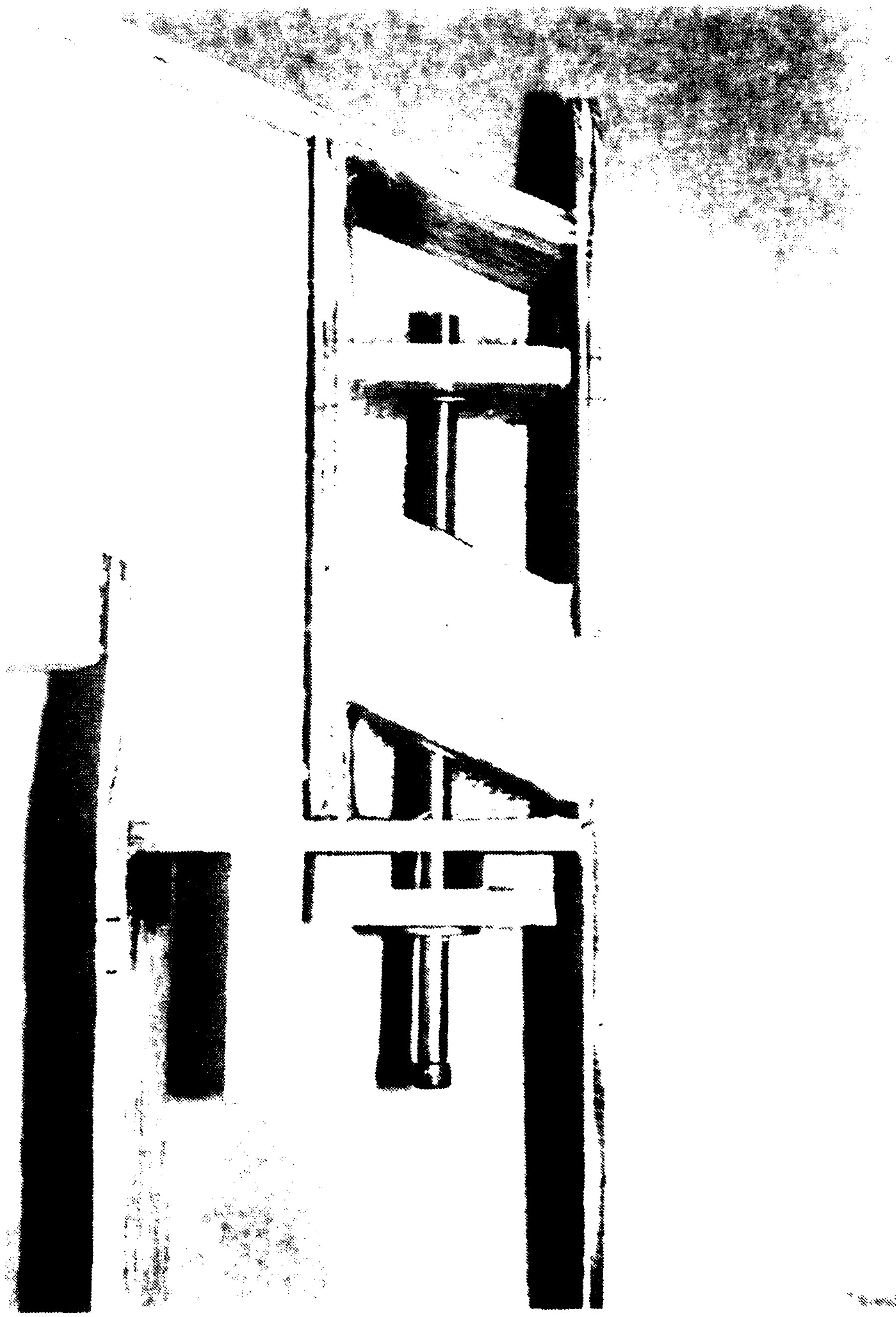


Figure 8 Spar box

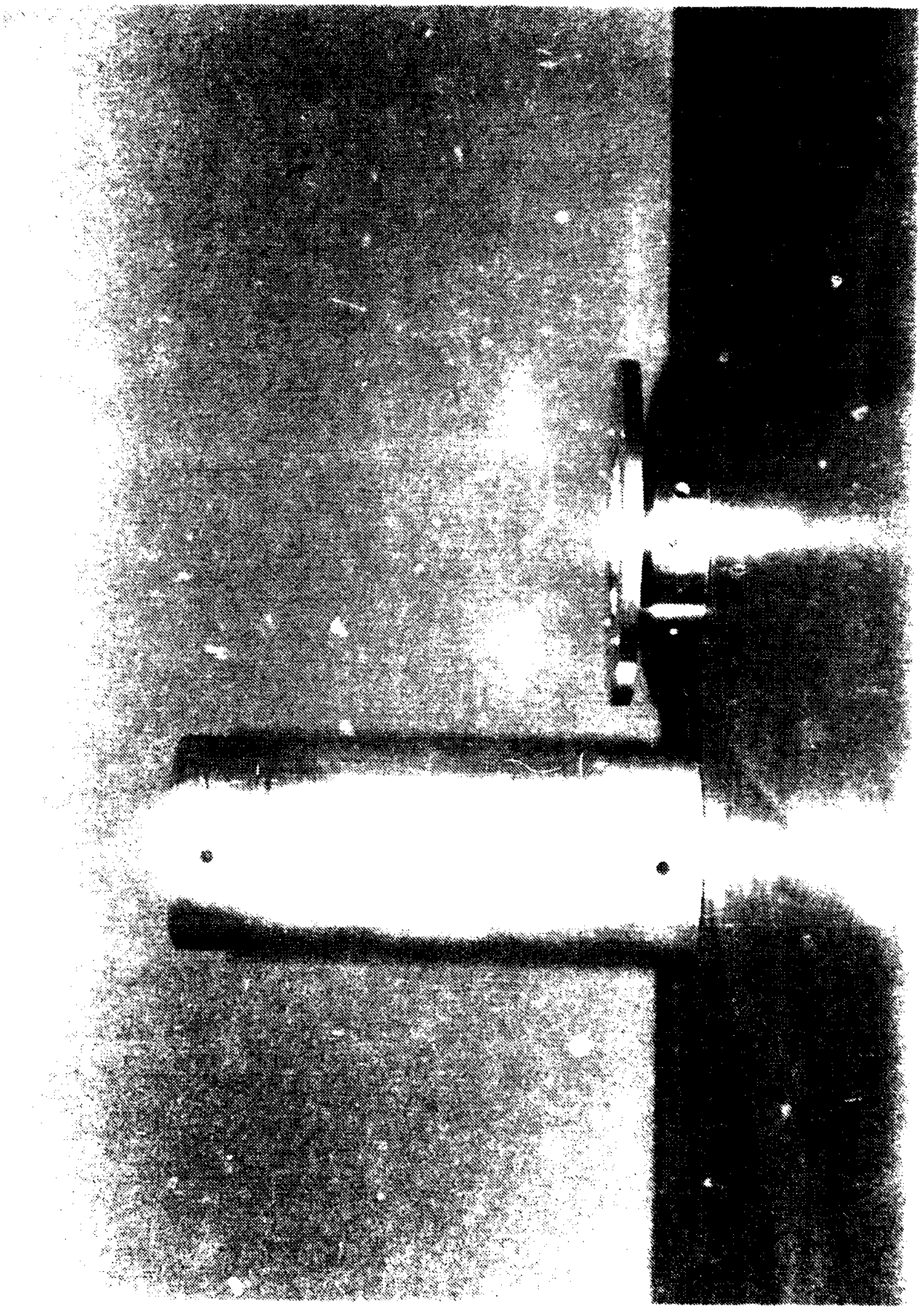


Figure 9 Support Cylinder (Left) and Motor Mount (Right)

aluminum piece, is made up of two parts. The upper part is an aluminum disk with a diameter of 4 in. and is 0.5 in. tall. The lower part is a cylinder which has a diameter of 2.5 in. and is 2 in. tall. The engine is mounted to the upper part, and the lower part fits into the support cylinder.

The duct is connected to the support cylinder by four 0.25 in. diameter aluminum rods (cross-members) each 4.5 in. long. To support the engine mount assembly, a support ring is fastened to the bottom of the support cylinder.

The support ring is 12 in. in diameter and 2 in. tall. It is made of the same material and connected to the support cylinder in the same way as the duct. To stiffen up the engine mount assembly four aluminum rods were sized to fit between the upper and lower cross members and welded together (Figure 10).

To counter the gyroscopic effect of the engine and to provide yaw, pitch and roll control, four control vanes are used. These control vanes are directly driven by servos located in the support cylinder. The airfoil selected for the control vanes was the NACA 0012 because of its symmetric shape. The control vane has a chord of 3 in. and a span of 3.5 in. To produce a maximum moment, the vane is placed as low as possible on the support cylinder. The servo is installed between the cross member pairs rather then between an upper and lower cross member so that the vane will see as much "clean air" as possible (Figure 10). The vane is attached to the servo at the 0.25 chord. The opposite side of the vane is supported by a bearing placed inside a strip of plywood that is connected to the duct.

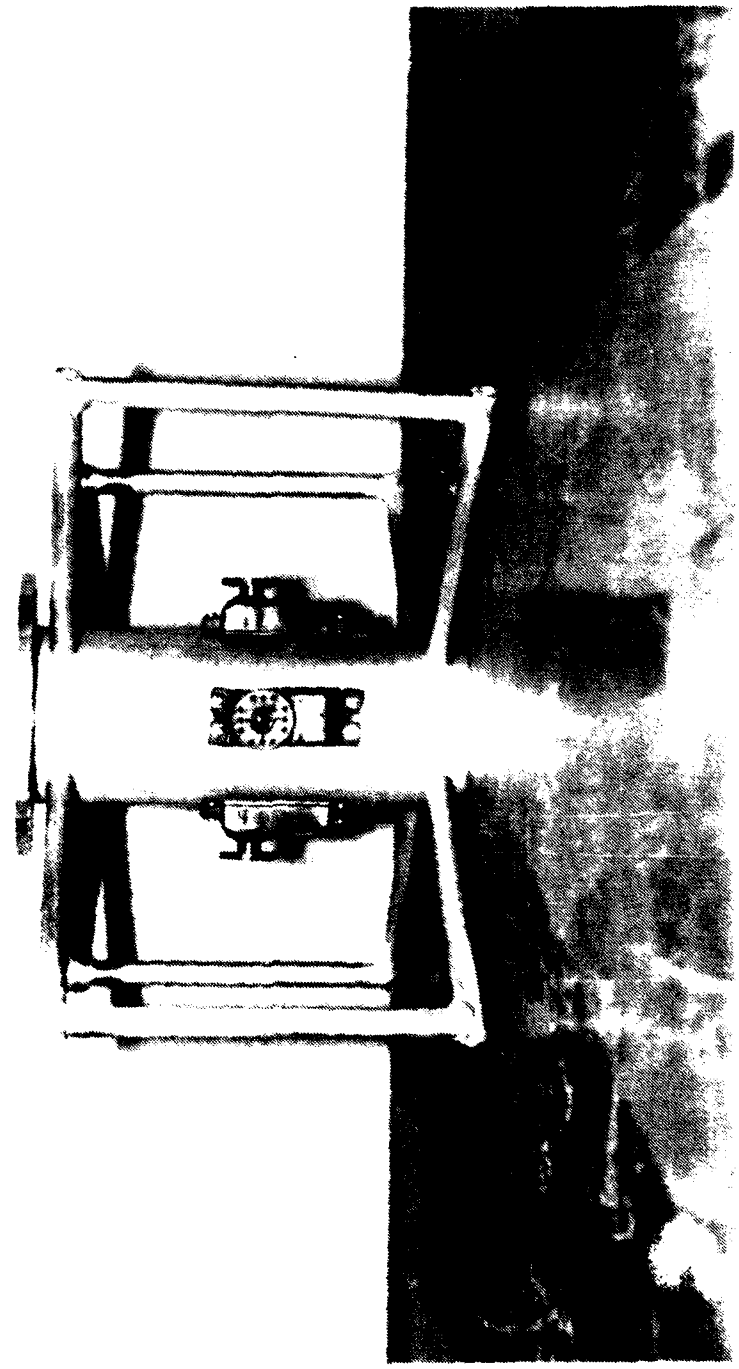


Figure 10 Engine Mount Assembly

A twin cylinder engine was selected to cut down on the vibration that has been a problem in past VTOL UAVs. To further reduce the vibration transferred from the engine, rubber mounts are used to isolate the engine from the duct and fuselage. The rubber mounts connect the engine to the motor mount of the engine mount assembly. The duct and engine mount assembly are connected to the fuselage in two different ways.

In vertical flight mode the duct and engine mount assembly are connected to the fuselage in three places. One screw bolts from each side bay into the duct and one screw bolts from the duct into a threaded insert located in the main support member forward of the duct area. All bolts entering the duct are screwed into threaded inserts attached to the duct. Threaded inserts were used to facilitate mounting the duct to the fuselage.

For horizontal flight mode the duct and engine mount assembly are connected to the fuselage by two bolts on either side. The forward bolts are installed first for starting the engine. To start the engine, the duct needs to pivot on the forward bolts to allow the starter access to the safety spinner; the duct is then rotated to its horizontal position, the rear bolts are installed, and the aircraft is ready to fly.

III. ARCHYTAS CONSTRUCTION

A. FUSELAGE

The fuselage was designed primarily around the duct. Since this is a single engine VTOL aircraft, it is necessary for the thrust line to be at the center of gravity for vertical takeoff and landing. To accomplish this the fuselage was designed in a horseshoe shape. Composite materials were used to facilitate the construction of the aircraft into an Aquila like shape.

The frame of the fuselage was made out of 1/4 in. plywood. Consideration was made for the locations for the avionics, wing spar boxes, fuel tanks, mounting hardware, and eventual flight test equipment. The main structural support is from a 1 in. by 1 in. plywood cross-member which also ties together the spar boxes and is located just forward of the engine. To maintain the low radar cross section and for airflow into the engine an effort was made to minimize the fuselage thickness. The maximum thickness of the fuselage is set at 3 in. to accommodate the largest piece of electronics. As illustrated by Figure 11, the maximum height of 3 in. is located at the front of the fuselage frame, and on either side of the inboard fuselage bulkhead. Connecting the inboard fuselage bulkheads is the 1 in. by 1 in. cross-member. Between the forward bulkhead of the fuselage frame and the cross-member is a block of urethane foam used to house the avionics (avionics bay). This foam block is tapered from the 3 in. of the forward bulkhead to the 1 in. cross-member.

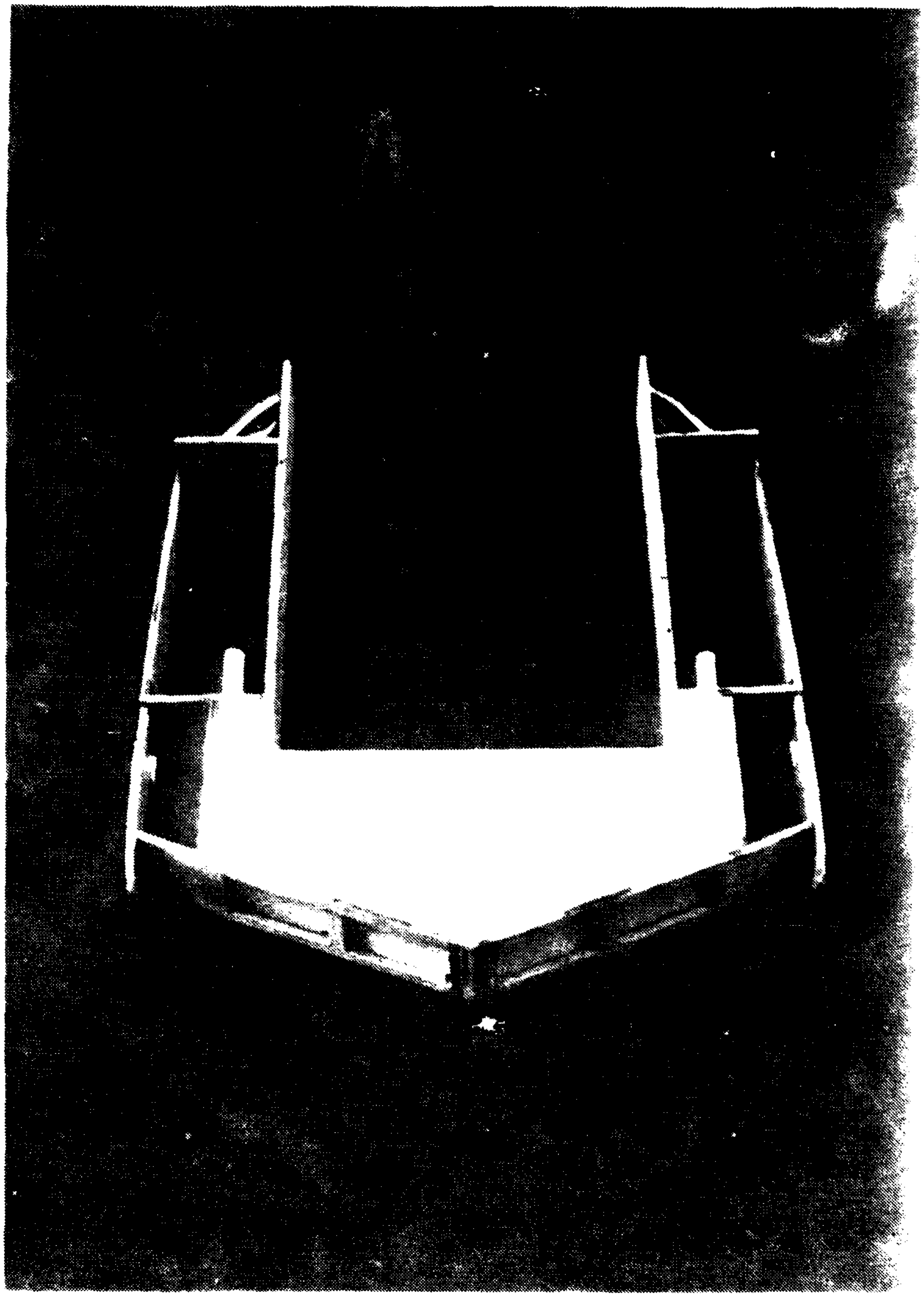


Figure 11 Fuselage Frame

The cross-member is centered so that the taper will provide even air flow into the engine. Wing root chord templates (outboard fuselage bulkhead) and spacers are used to form the two spar box access areas and the two side bays, with the spacers tapered from the inboard bulkhead to the outboard bulkhead. Wiring conduits on both sides of the foam block are used to route wires from the avionics bay to each side bay. Panels were cut out of the forward bulkhead to route wires from the nose bay to the avionics bay and to cut away excess weight (lightening holes).

Once the basic fuselage frame was completed, the spar box was installed. The spar box was secured to the spar box area by fiberglass and epoxy.

The nose (Figure 12) was shaped from the same urethane foam as the center foam block and was epoxied to the front of the frame. Balsa sheets were epoxied to the foam to provide a solid base for cutting out the nose and avionics bay's access plates. The rest of the fuselage was then covered with balsa sheets to completely enclose the structure (Figure 13). The fuselage was then fiberglassed. Once fiberglassed the access panels were cut out and where necessary the foam was removed to create the bays. A balsa sheet was epoxied to the foam inside the nose and avionics bays to provide a solid base for mounting the avionics and necessary hardware. The hardware and fuel tanks installed in the side bays are directly connected to the 1/4 in. plywood bulkheads. The duct and wings are connected to the fuselage from inside the side bays. As described earlier the duct is mounted to the inboard fuselage bulkhead, and the wing is secured by two screws to the outboard fuselage bulkhead as well as pinned through the spar box.

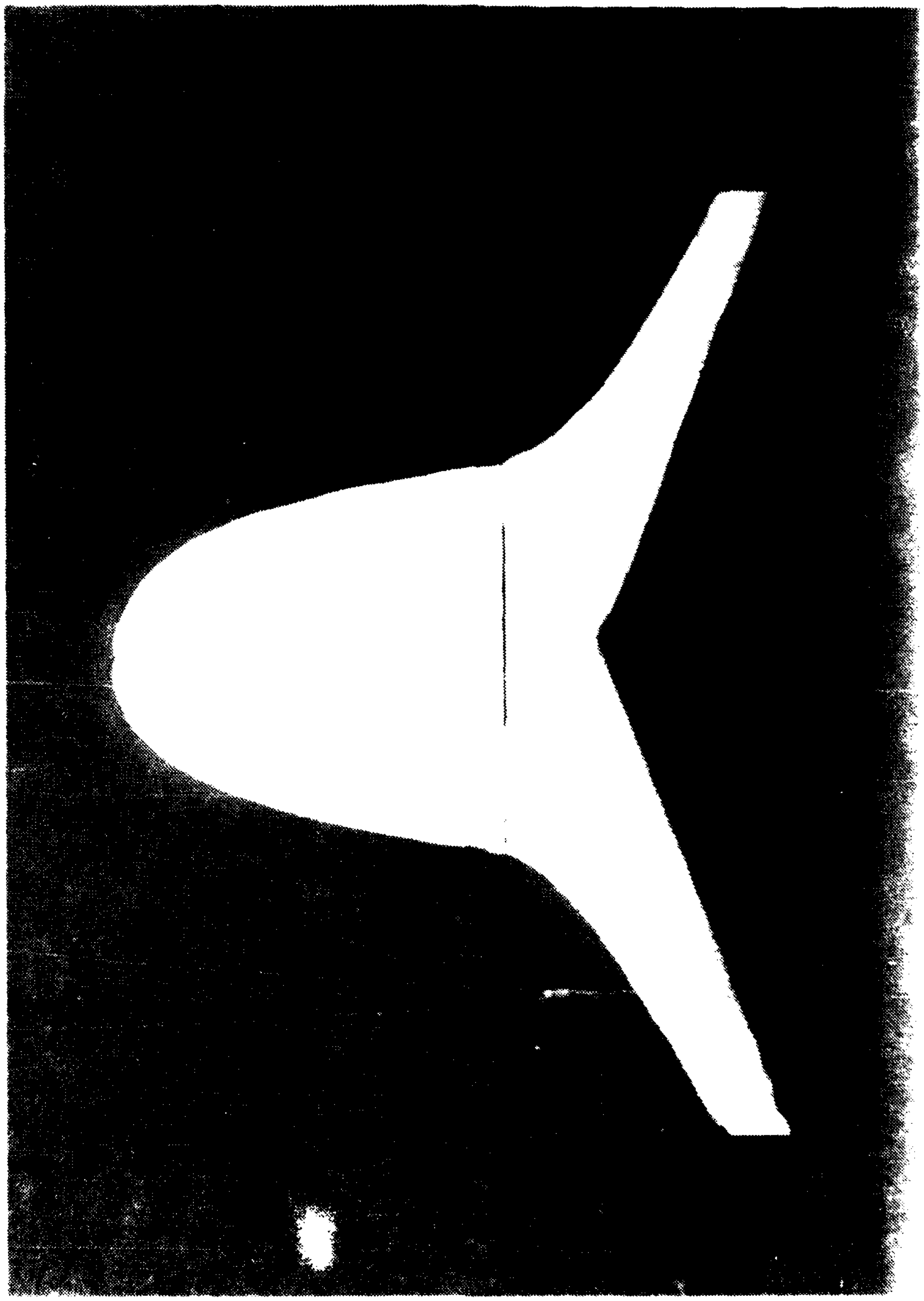


Figure 12 Nose Structure

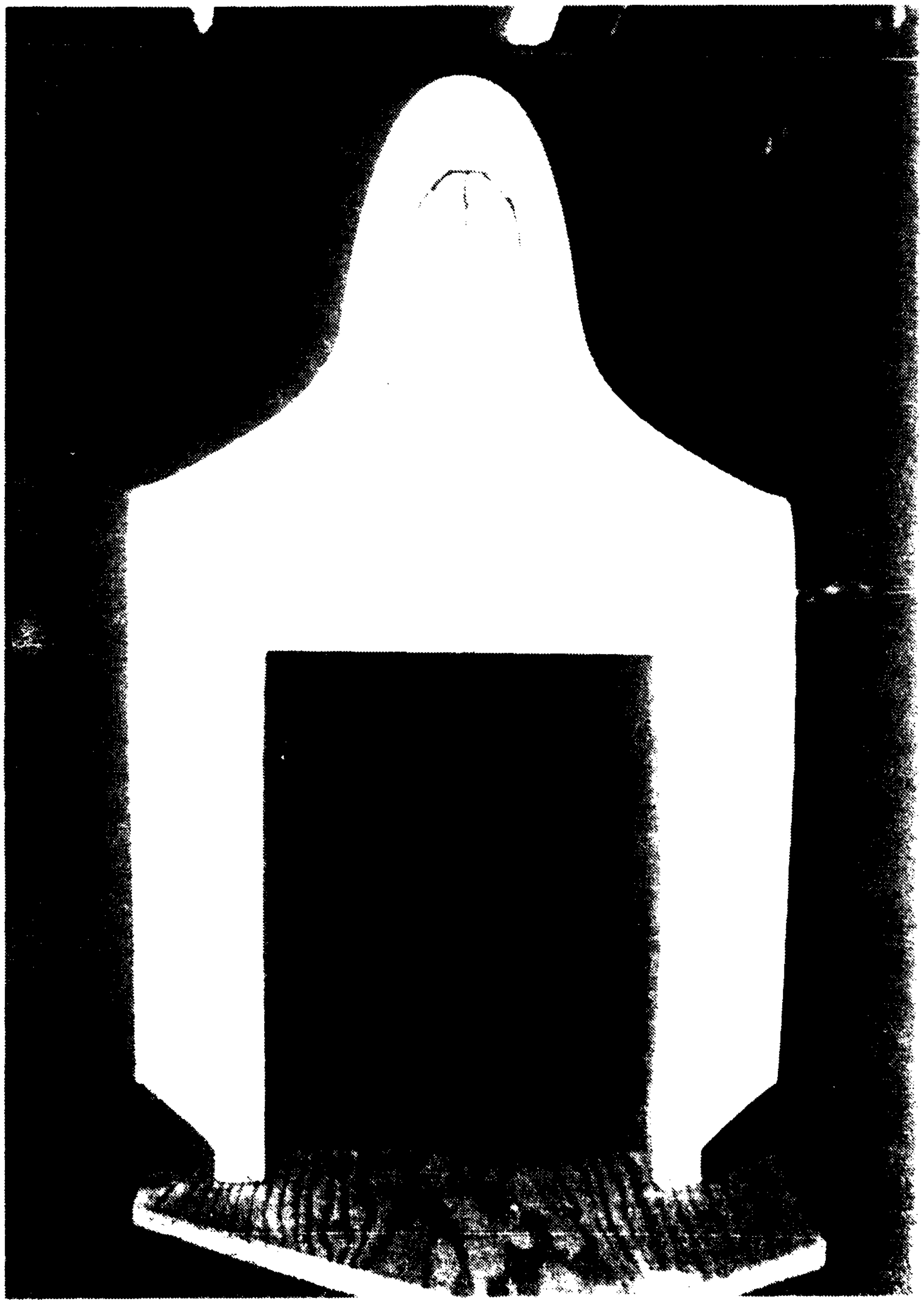


Figure 13 Fuselage Completely Covered With Balsa

B. WING AND SPAR BOX

The wing was constructed using a lightweight composite construction technique which is a trademark of aircraft designer and builder Burt Rutan whose experimental aircraft Voyager holds the record for continuous unrefueled flight. This technique, known as wet lay-up, uses styrofoam or urethane foam, fiberglass, and an epoxy resin. The foam provides a stiff core and the fiberglass-resin matrix provides the necessary strength. [REF.15:p.30]

To begin construction the billet of foam was cut to the desired dimensions. To cut the foam a hotwire was used. The hotwire, shown in Figure 14 consists of a 0.032 in. wire attached to two steel rods separated by a wood 2 by 4. A variable resistor transformer powered by normal household current is connected to the steel rods. This transformer produces the heat that is carried to the wire. The heated wire then slices through the foam.

First the billet of foam was cut to the desired dimensions (Figure 15). For the hotwire to core the billet of foam, plywood templates were made representing the root and tip chord. These templates were proportionally marked so that each end of the hotwire would pass over the templates evenly. Next the templates were attached to either end of the billet and the hotwire passed over the templates producing the wing planform (Figure 16).

The spar was constructed in two parts. The first part was the shear web, which consisted of an 20 in. piece of 1/4 balsa wood sandwiched between two layers of 8 ounce bidirectional fiberglass on each side for a total web thickness of 0.05 in. of

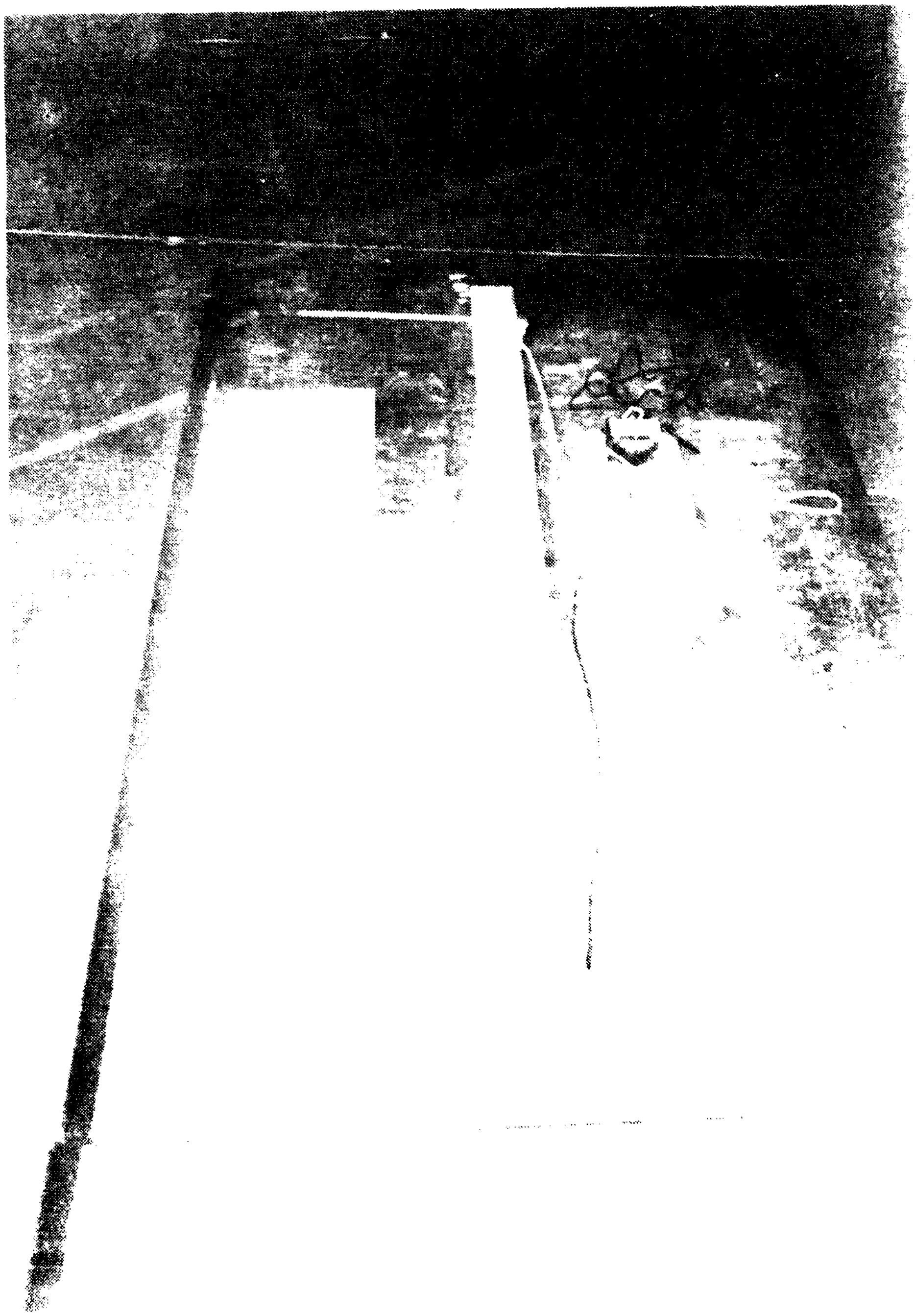


Figure 14 Hotwire equipment

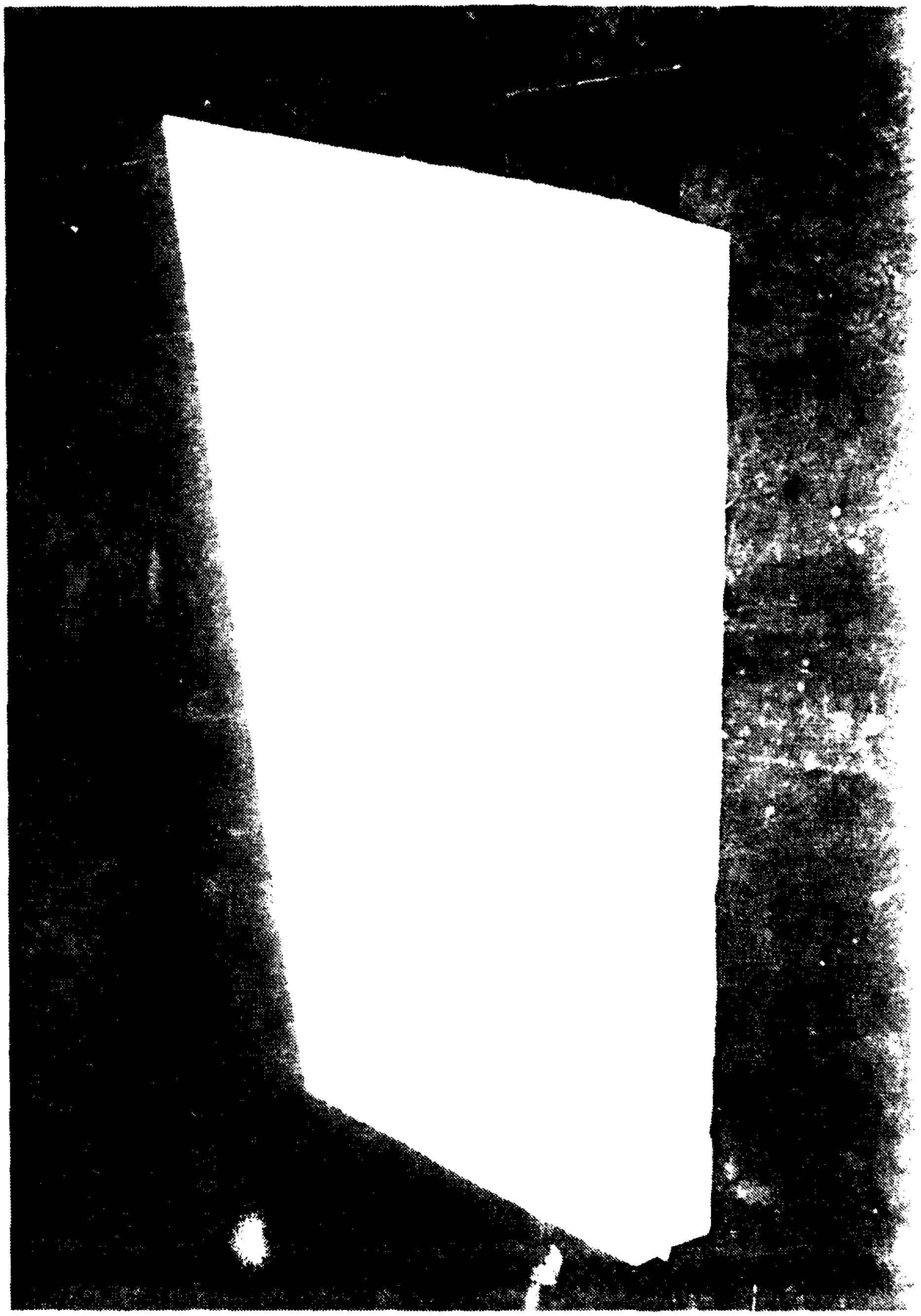


Figure 15 Billet of Foam

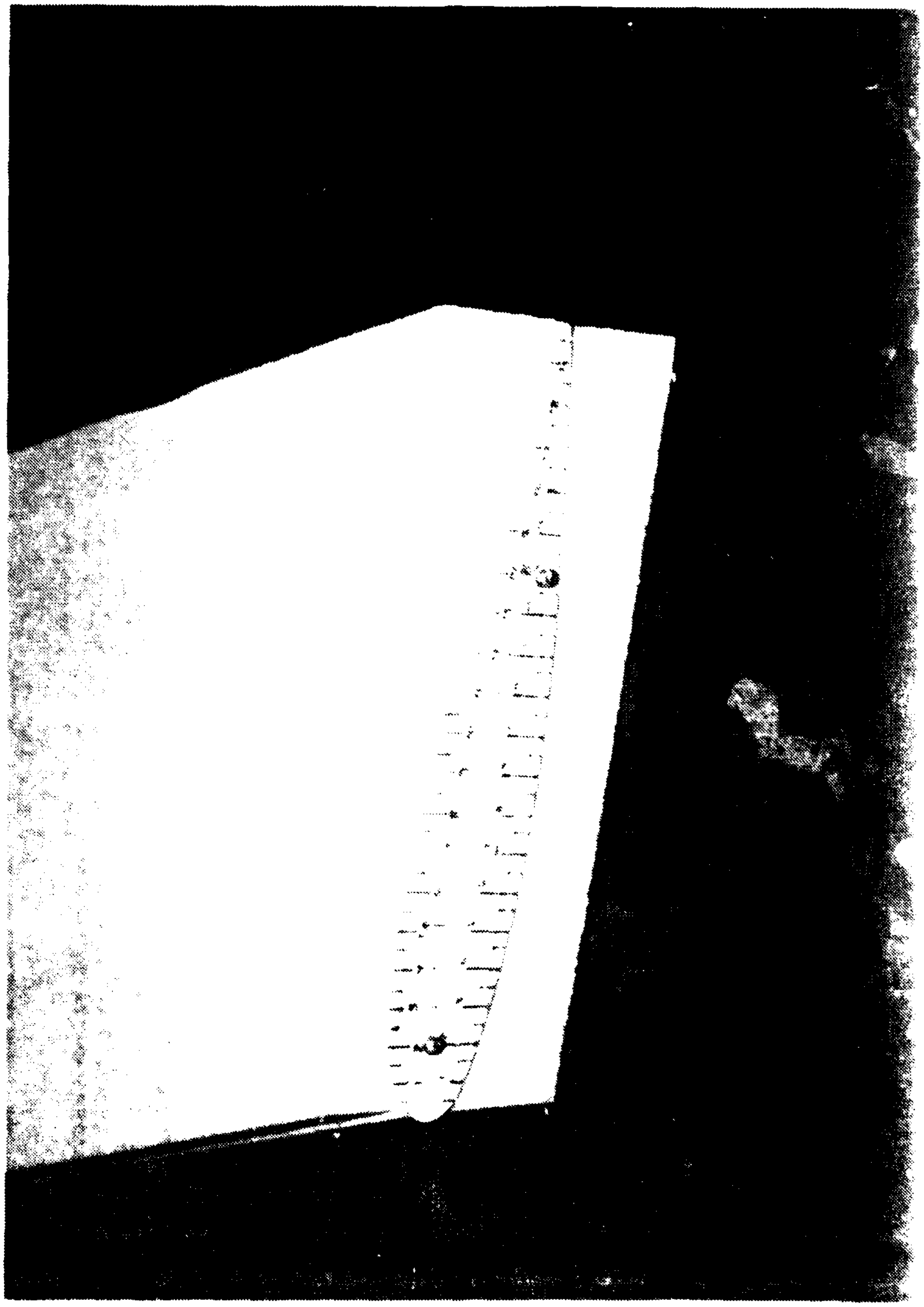


Figure 16 Wing Chord Template Attached To Billet Of Foam

fiberglass.

The second part consisted of the upper and lower spar caps. The spar caps are made from 1.5 in. wide by 18 in. long 1/4 in. balsa wood, sandwiched between two layers of 8 ounce unidirectional fiberglass for a total spar cap thickness of 0.05 in. of fiberglass. The shear web was then cut down to fit inside the wing. The shear web is tapered from 2.2 in. starting at 2 in. from the root to 1.5 in. at the tip (a span of 18 in.). The first 2 in. were cut down to 1.5 in. to conform to the spar box. Added strength was given to these first 2 inches by epoxying a 3 in. piece of fiberglassed balsa wood to either side, as can be seen in Figure 7.

The wing was then cut out along the 0.25 chord location to fit the spar (Figure 17). To give the wing more strength and to provide for a structural attachment for the control surfaces, an additional spar was installed along the 0.725 chord position (Figure 18). This additional spar was constructed similarly to the main spar's shear web and runs the entire length of the wing. The foam wing surface was then glassed with 3 ounce fiberglass cloth and epoxy resin (Figure 19).

The next parts of the wing constructed were the control surfaces. The Archytas, as described in Chapter IIB, is equipped with both flaps and ailerons. The control surfaces were cut out of the wing and can be seen in Figure 20. Both sides of each control surface and the matching sides in the wing were faced with 1/16 in. plywood. The leading edge of each control surface was rounded to reduce the gap while still allowing unbinding motion. The leading edge of each control surface and its matching surface in the wing were glassed with 3 ounce fiberglass. Figure 21 shows the rod

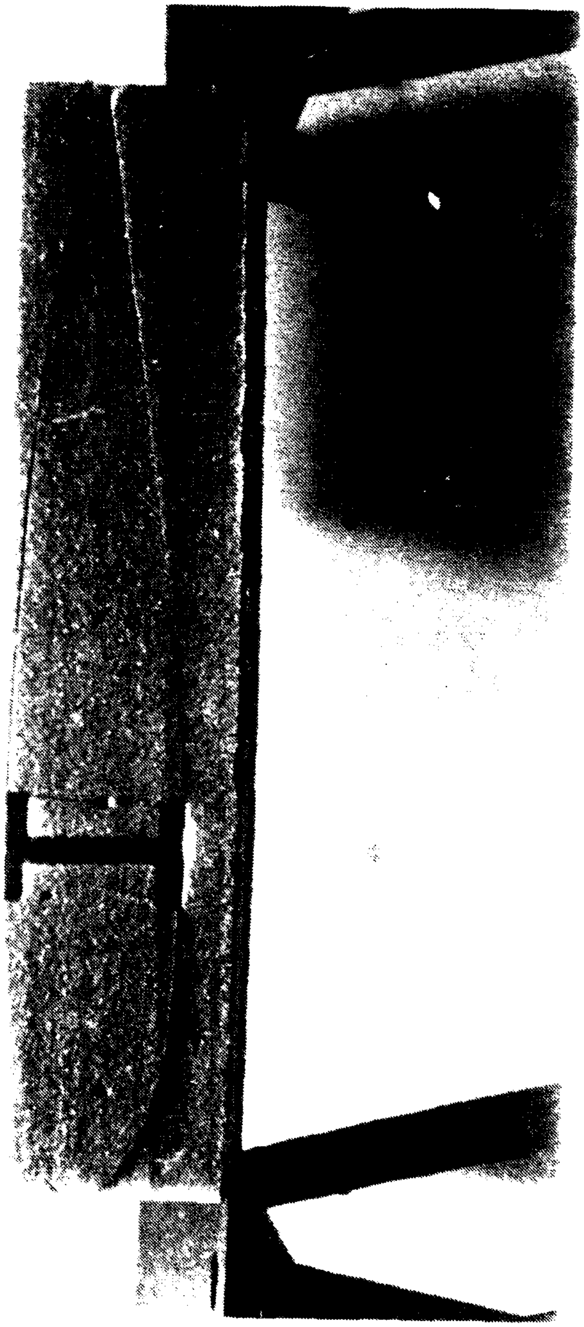


Figure 17 Spar Location On Wing

Figure 18 Wing With Main and Additional Spar Installed

Figure 19 Fiberglass Covered Wing

Figure 20 Control Surfaces Cut Out Of Wing

that was used as a bearing surface inside the control surface. Each control surface servo is located inside the wing to reduce drag.

The wing is connected to the fuselage via the spar box. The spar slides into the spar box through the outboard fuselage bulkhead (matching wing root chord template) as shown in Figure 22. To secure the spar to the spar box a pin is used. As shown in Figure 8 a hole is drilled through the side bay/spar box area bulkhead into the spar box, through the spar, and into an additional support. This support and the spar box are made out of the same material as the spar shear web. To prevent the pin from enlarging the holes during flight or during insertion and removal, steel washers were glued to the spar box and support, and an aluminum insert was installed through the spar. Figure 23 shows the spar box installation on both sides of the fuselage. For added structural support the spar box area was completely sealed by two fiberglassed balsa pieces constructed similarly to the spar's shear web (Figure 24).

C. ENGINE MOUNT AND DUCTING

The duct was made out of composite materials similar to the wing. To construct the basic shape of the duct a solid wood mold was made. The mold had an 11 3/4 in. upper diameter, 11 1/2 in. lower diameter and was 12 in. long. The difference in upper diameter and lower diameter provided a 1/4 in. draft angle to assist in releasing the duct from the mold. The first item placed on the mold was the mold release. Plastic wrap was used and taped to the mold. Next a layer of 3 ounce fiberglass was placed around the mold for a nice smooth inner surface. A layer of 8

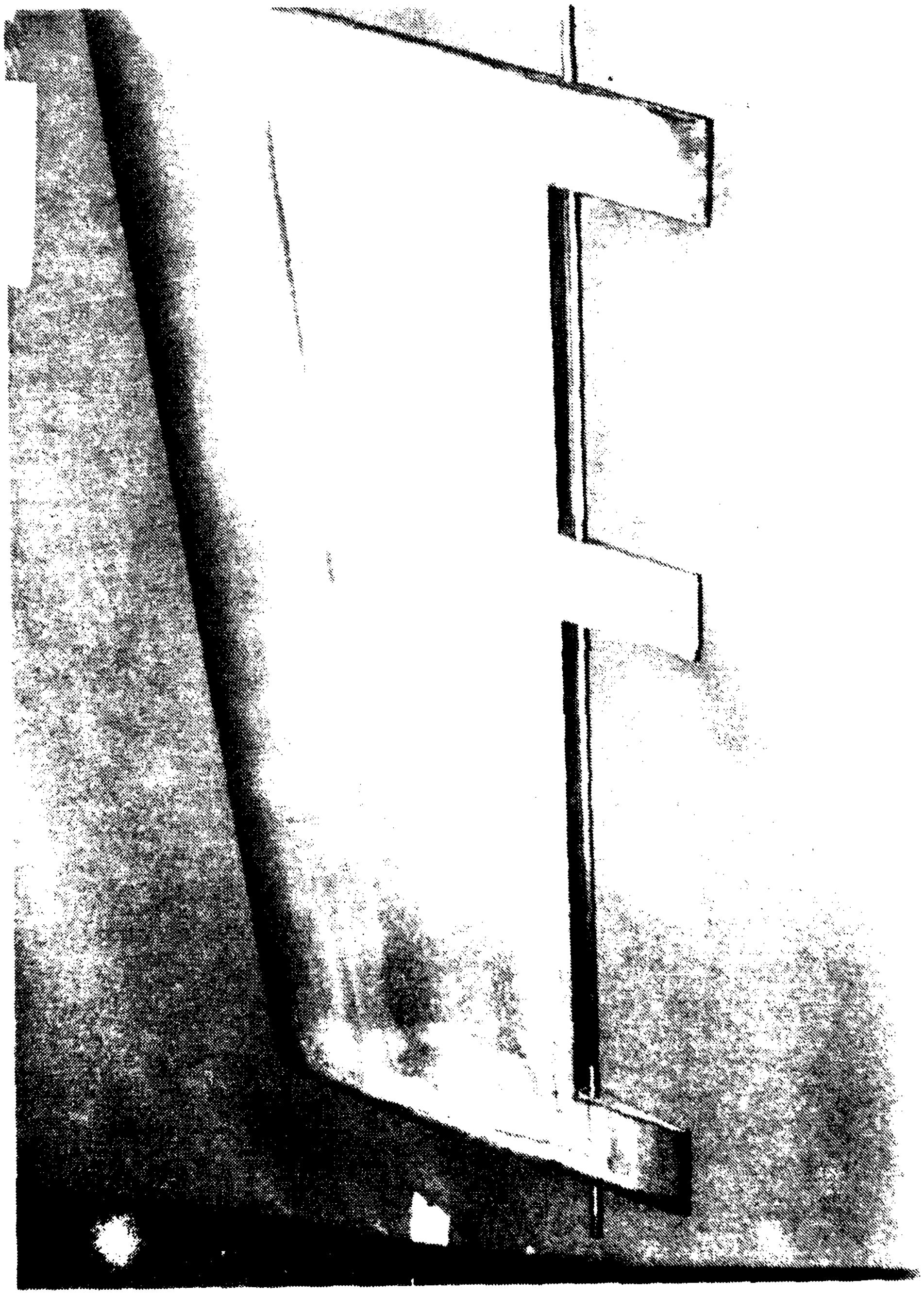


Figure 21 Rod Used As A Bearing For Control Surfaces

Figure 22 Spar Box And Wing

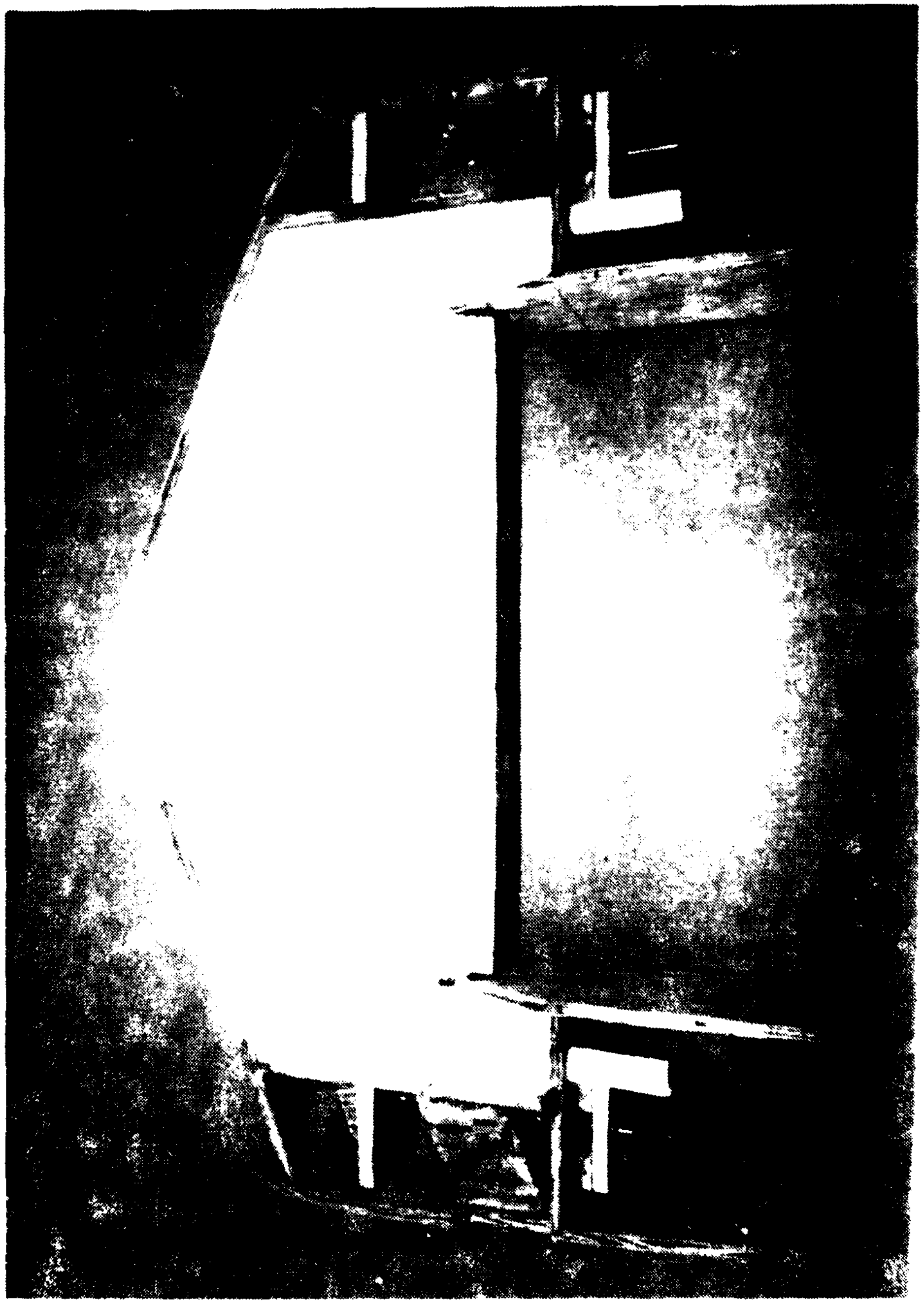


Figure 23 Both Completed Spar Boxes

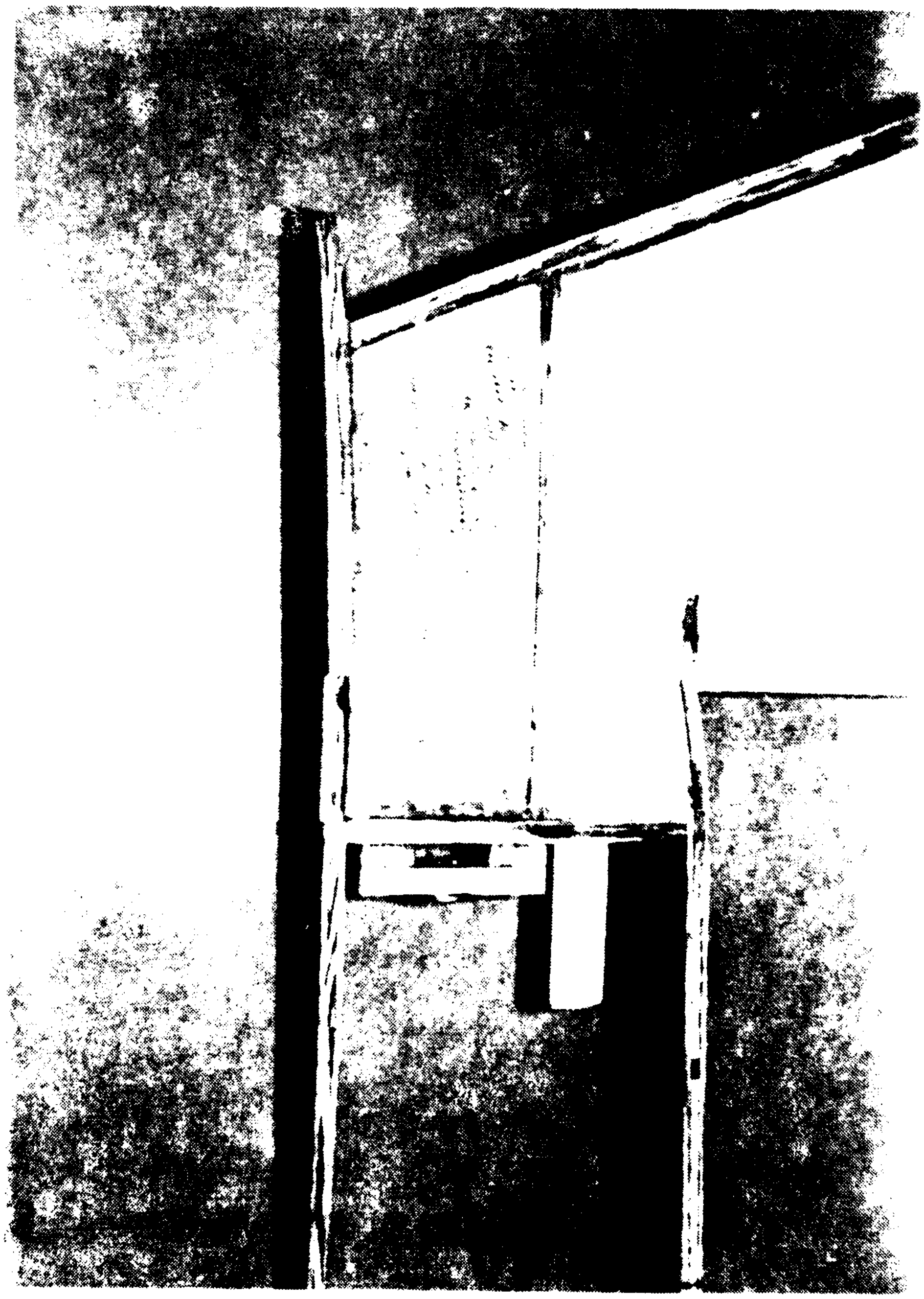


Figure 24 Covered Spar Box

ounce bidirectional fiberglass cloth was then applied. Six strips of 6 in. wide by 12 in. long 1/16 in. balsa wood were layered over the fiberglass. By running the grain of the balsa wood along the length of the mold, the balsa wood conformed to the mold contour. Two additional layers of balsa wood were epoxied onto the first. The seams on each layers were staggered to avoid weak spots. The three layers of balsa wood were followed by a layer of 8 ounce bidirectional fiberglass. Finally a layer of 3 ounce fiberglass was applied to provide for a smooth outer surface. The bottom 2 in. of the duct were cut off to be used for the support ring, leaving the required 10 in. long duct (Figure 25). To add structural support to the duct where it will be mounted to the fuselage, ten 3 in. strips of 8 ounce bidirectional fiberglass were placed in four length wise columns evenly spaced inside the duct. These ten strips were interwoven with an additional ten 3 in. strips placed along the inside circumference of the bottom of the duct. Two 2 in. wide by 3 in. long pieces of 1/4 inch plywood were fiberglassed to opposite inside walls of the duct. These two blocks were drilled and inserts installed for mounting the duct to the fuselage.

The next item to construct was the engine mount assembly. As described earlier, the main support for the engine mount assembly was the support cylinder. Connected to the top of the support cylinder was the motor mount. To fasten the duct and the support ring to the support cylinder, four cross-members were used for each. The support cylinder has eight clearance holes to accommodate fastening the cross-members, four on top and bottom. The motor mount has four matching clearance holes used to fasten the motor mount to the upper cross-members.

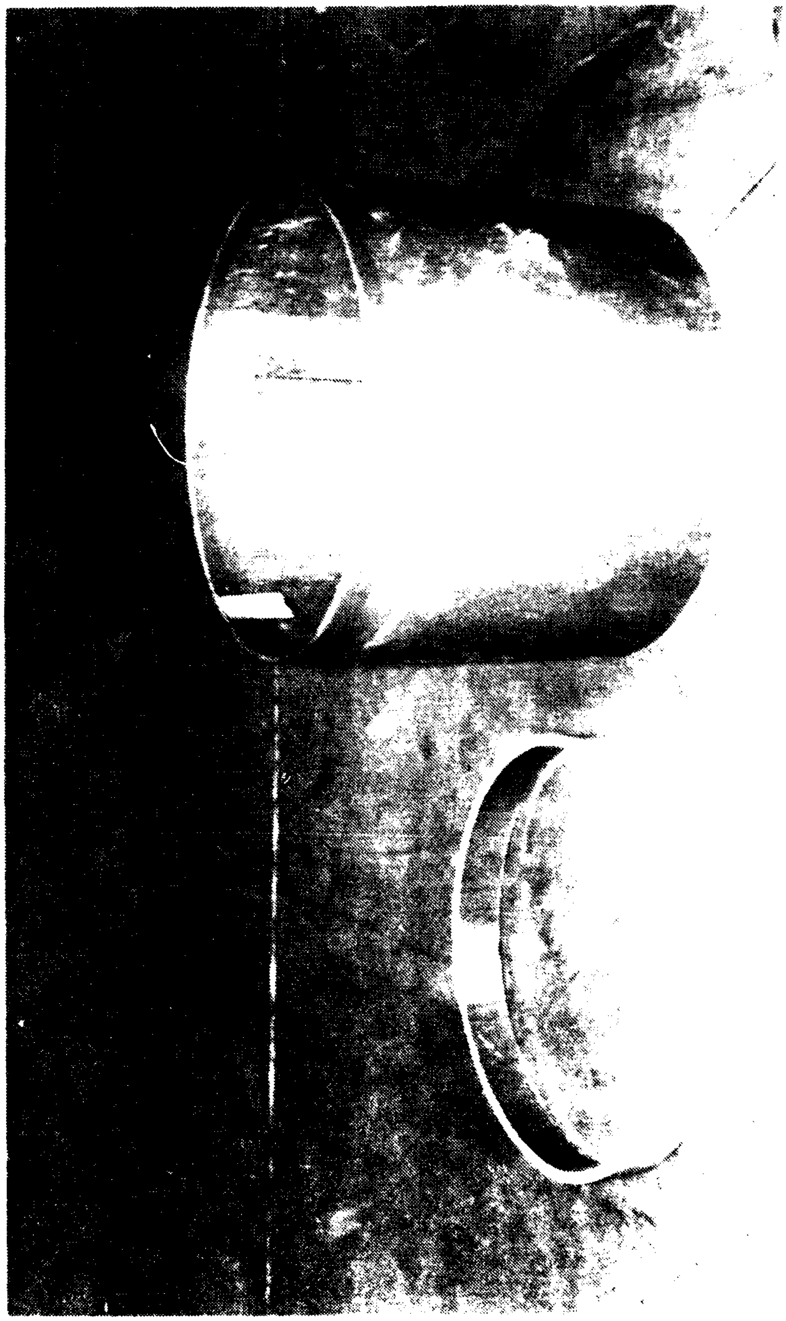


Figure 25 Duct And Support Ring

Figure 26 shows the motor mount and cross-members connected to the support cylinder. To strengthen the engine mount assembly, four aluminum rods were sized to fit between the upper and lower cross-members and then welded in place (Figure 10). The cylinder was then cut to accommodate the four control vane servos and the one throttle servo (Figure 10).

To isolate the engine from the rest of the aircraft to reduce vibration, the engine was mounted to the motor mount with rubber mounts. One problem developed while preparing the engine for mounting. Since the engine is made in Europe, all screws are metric. The rubber mounts available were standard threads. An adapter was required to mount the engine to the rubber mounts and then to the motor mount. The adapter was a 1/2 inch plexiglass ring, tapped for the rubber mounts and attached to the engine (Figures 27 and 28). Figure 29 shows the complete engine and engine mount assembly.

To provide yaw, pitch and roll control as well as to counter the gyroscopic effect during vertical takeoff, four control vanes are used. As described earlier these control vanes are NACA 0012 shape airfoils. These airfoils were made in a similar fashion to the wings. Templates were generated by computer, which were cut out, the edges were sanded to a smooth finish and then were hotwired from the same type foam billet. To add strength and to make it easier to handle the control vanes the templates were left on. After sanding the vanes they were fiberglassed with 4 ounce cloth.

The vanes were connected directly to controlling servos at the 0.25 chord location. The opposite sides of the vanes were supported by plastic bearing surfaces

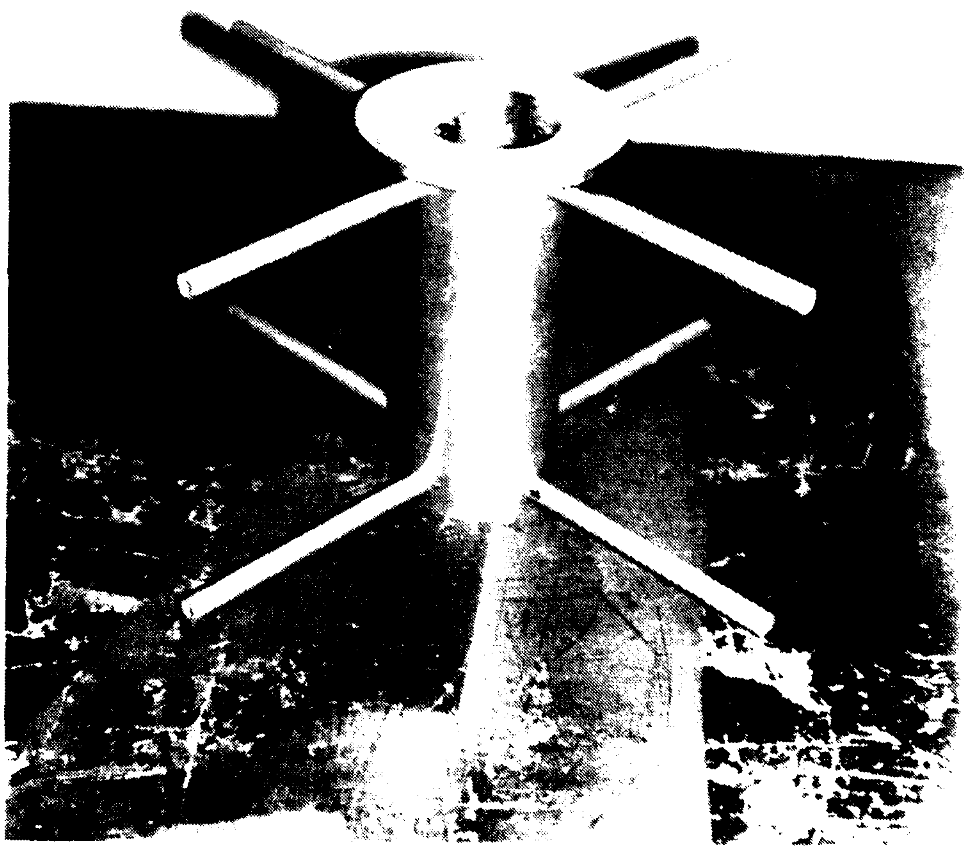


Figure 26 Support Cylinder With Cross-Members

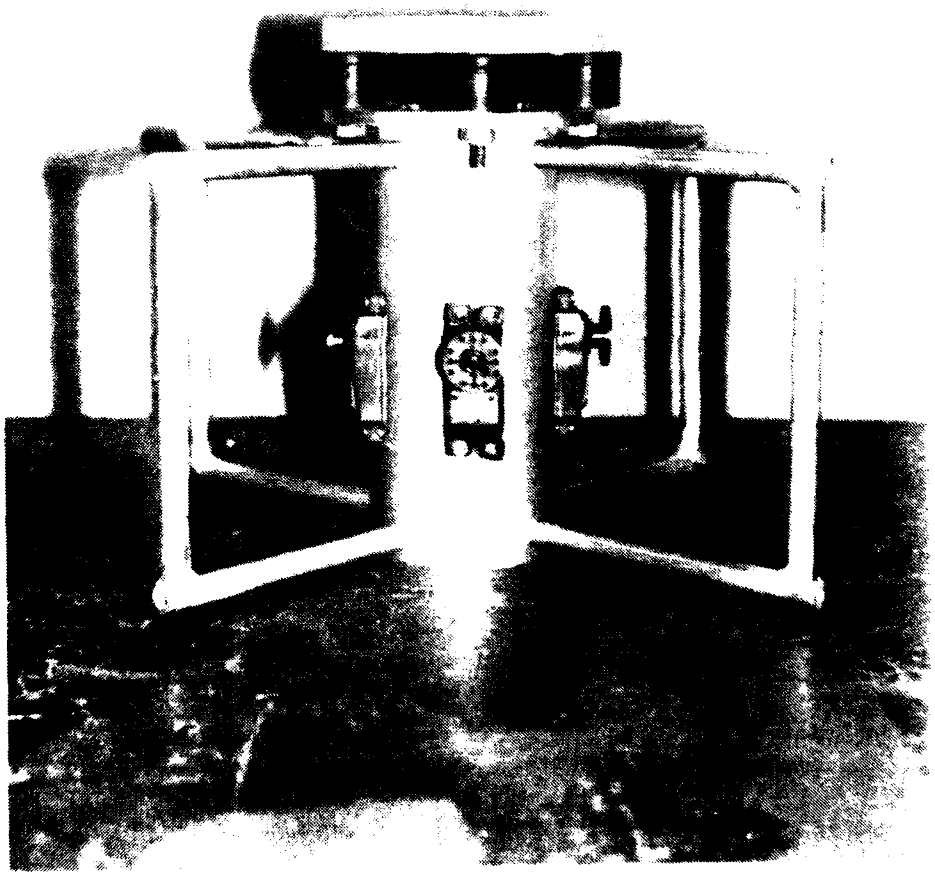


Figure 27 Support Cylinder With Rubber Mounted Adapter

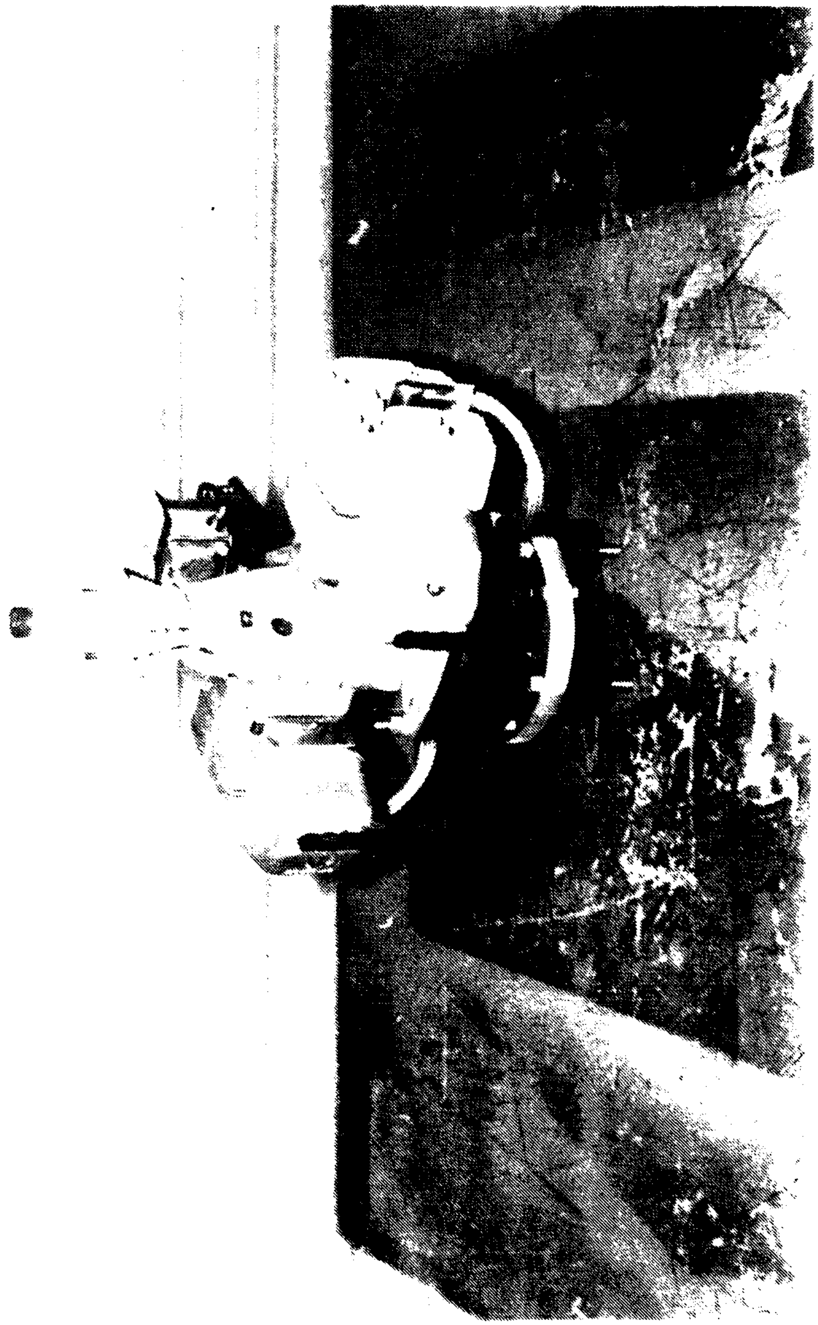


Figure 28 Adapter Attached to Engine

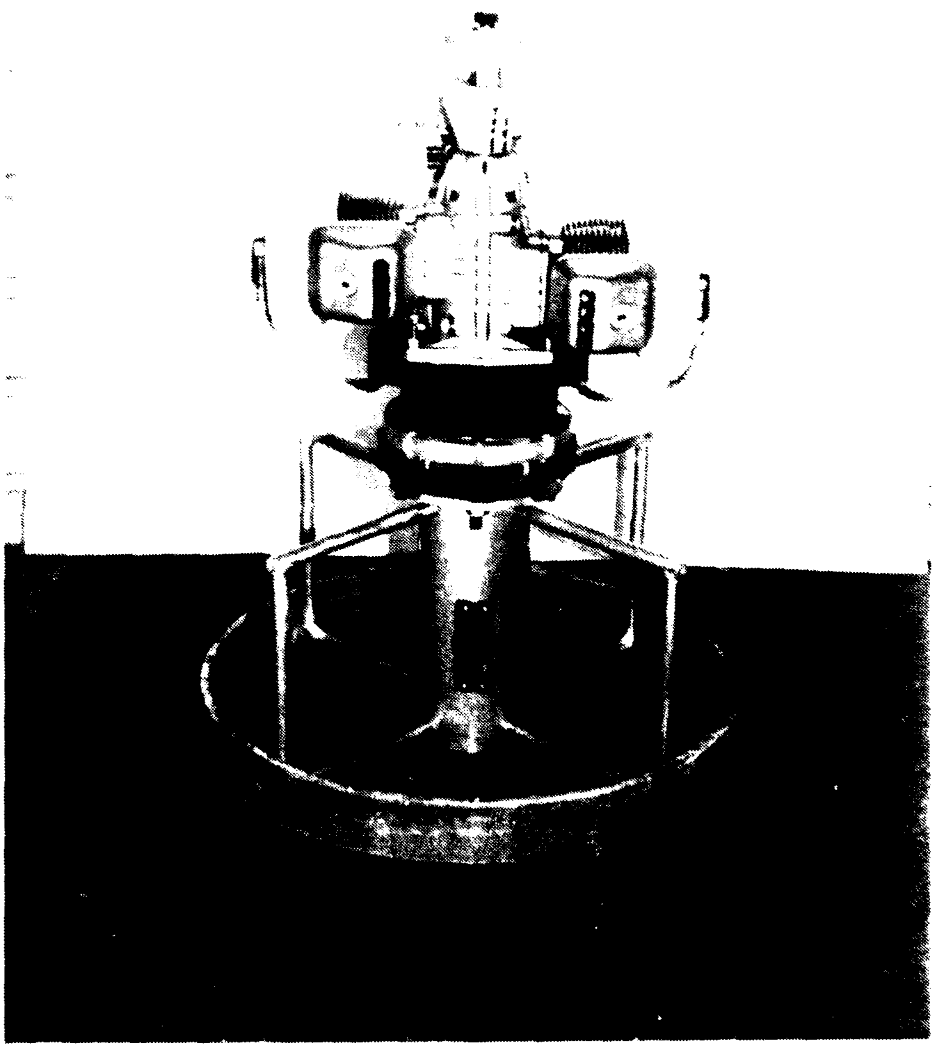


Figure 29 Engine Mounted To Engine Mount Assembly

held in place by plywood attachments to the inner surface of the duct. Figure 30 shows the components of the control vane assembly. Figure 31 shows a completed control vane assembly. Finally, Figures 32, 33, and 34 show the entire duct and control vane construction complete with two and four bladed propellers.

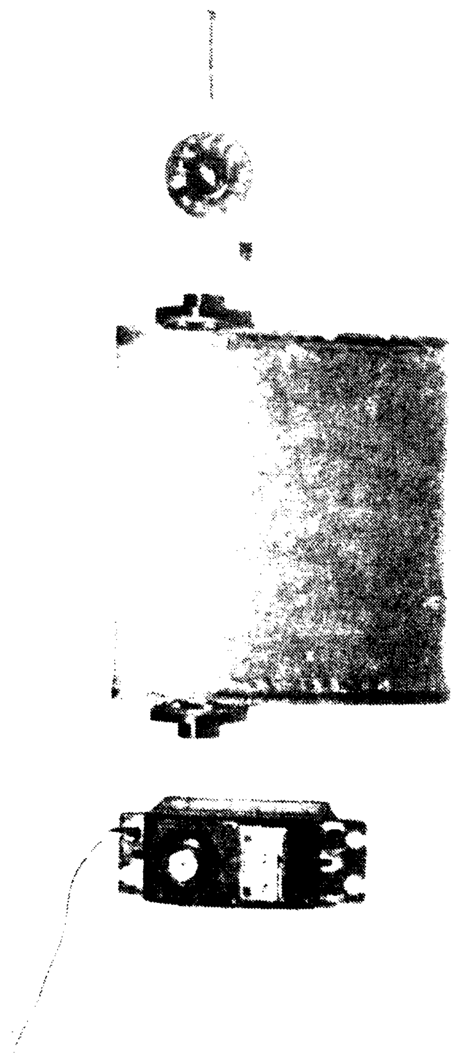


Figure 30 Control Vane Assembly Components

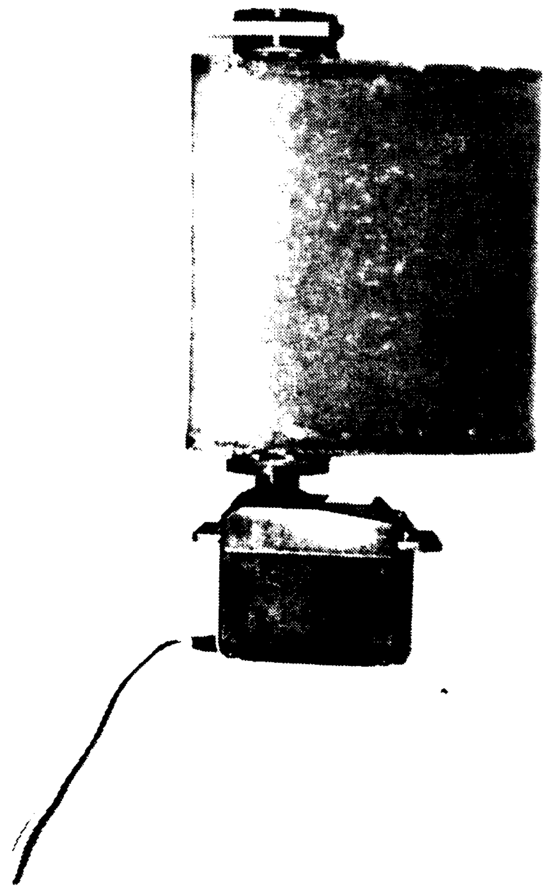


Figure 31 Control Vane Assembly

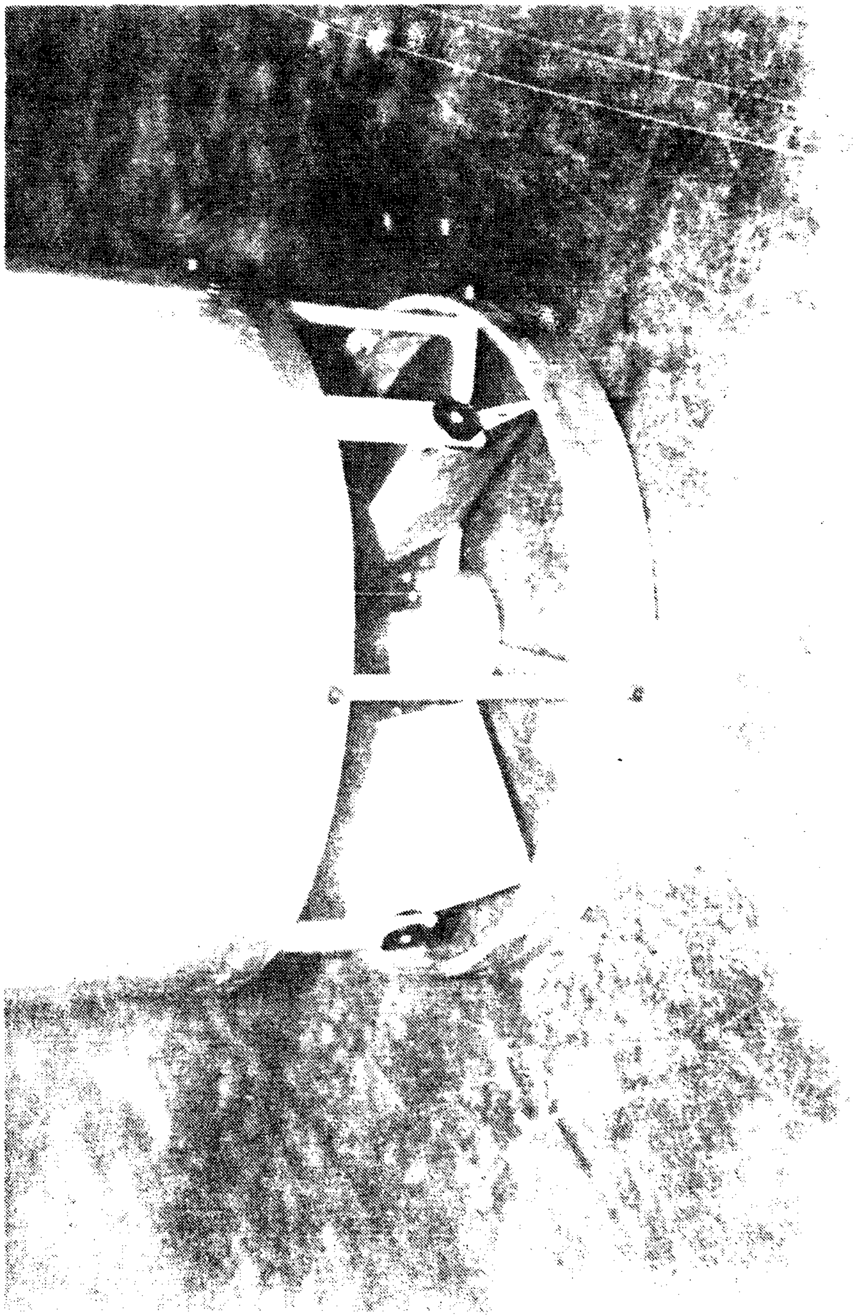


Figure 32 Duct, Support Ring, And Engine Mount Assembly

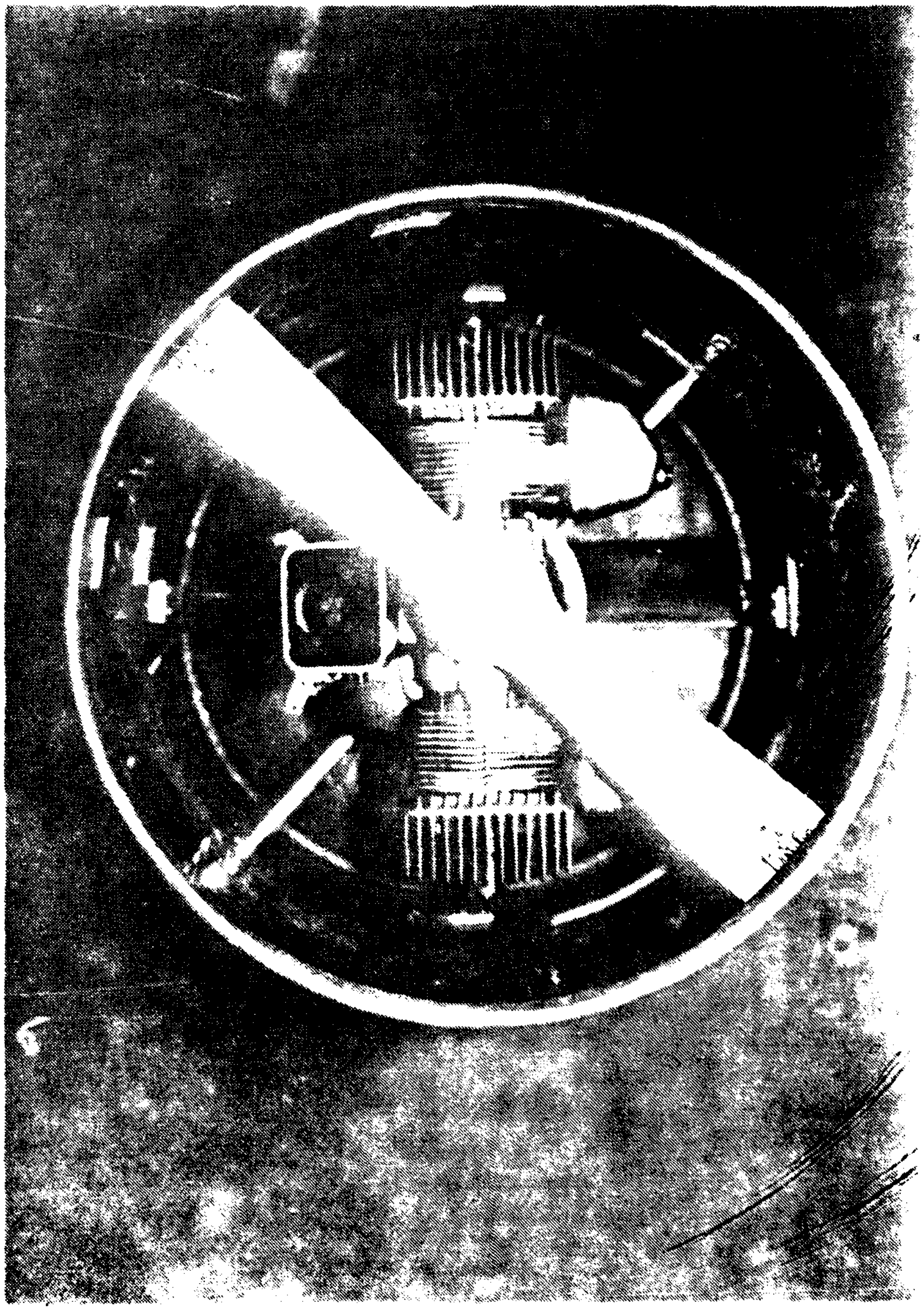


Figure 33 Top View Of Engine System With 2 Bladed Propeller

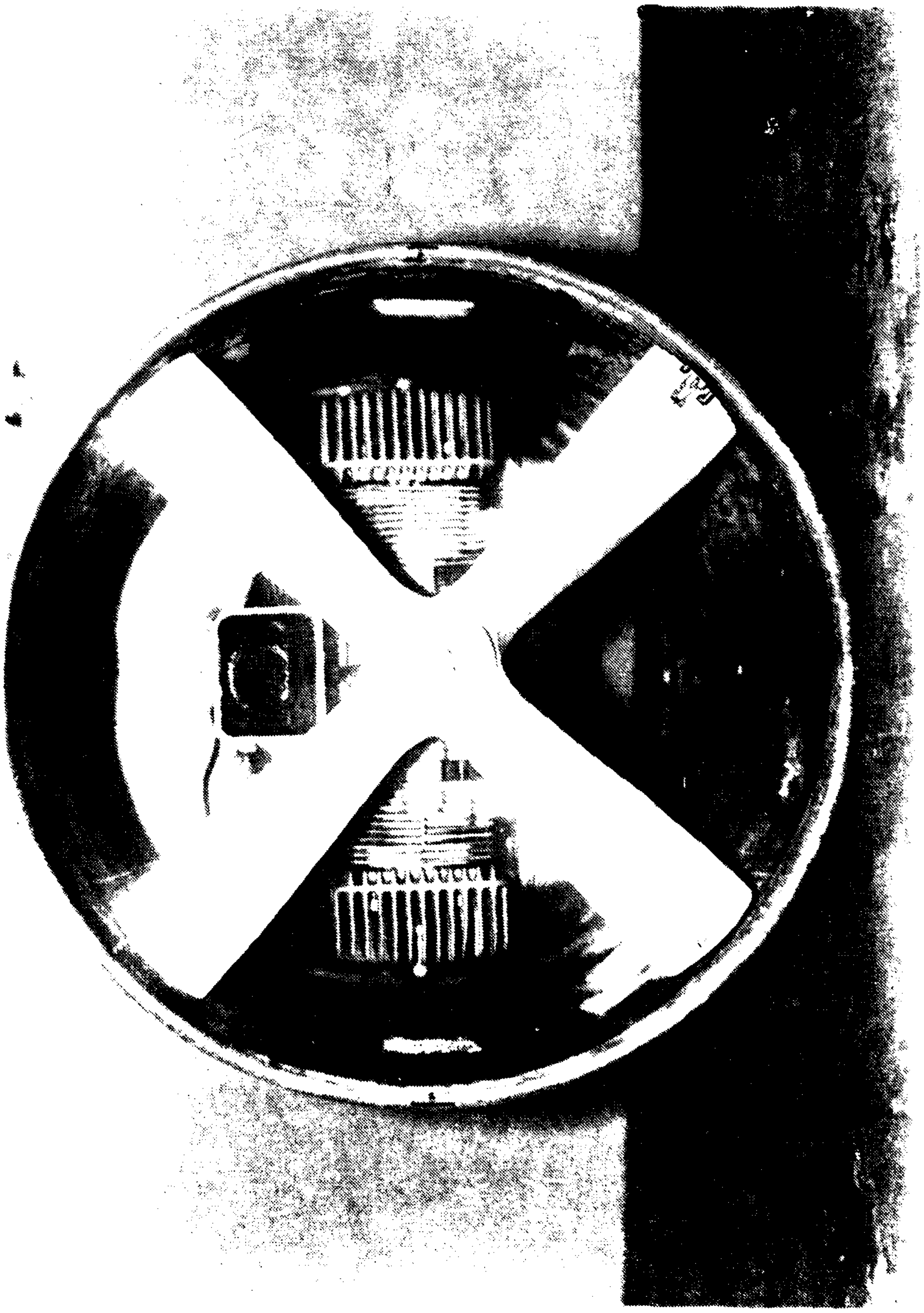


Figure 34 Top View Of Engine System With 4 Bladed Propeller

IV. INITIAL ENGINE AND FLIGHT TESTING

The engine chosen to operate the Archytas was a twin-cylinder, two stroke, glow plug, air-cooled engine of 43.75 cc displacement rated at 2.95 horsepower at 8,800 rpm (Figures 35 and 36). A twin cylinder engine was chosen for reduced vibration over the more popular hobby single cylinder engines. The engine will develop 25 lbs of static thrust with an 18 inch propeller.

Full size propellers could not be used, due to the limited duct diameter of 11.5 in. A series of tests were performed with propellers of reduced diameter to determine actual thrust performance losses with the smaller diameter propellers. It was hoped that the desired thrust could be achieved at higher engine speeds with the use of four bladed propellers. Large propellers were cut down, rather than smaller diameter propellers being used, to maintain the load capability of wider blades.

Before conducting the thrust tests the engine was broken in. The engine was mounted to the engine test stand for the two hour break-in period was required by the engine manufacturer (Figure 37). For the break-in period a 20-8 propeller[3], one of the propellers recommended by the factory, was used. During the break-in period minor adjustments were required to proportion the fuel air mixture. After the initial

[3]When referring to R/C model propellers the first number refers to the length and the second number refers to the pitch. For example the 20-8 propeller mentioned is 20 inches in diameter and has 8 inches of pitch.

Figure 35 Super Tartan Engine (Side View)

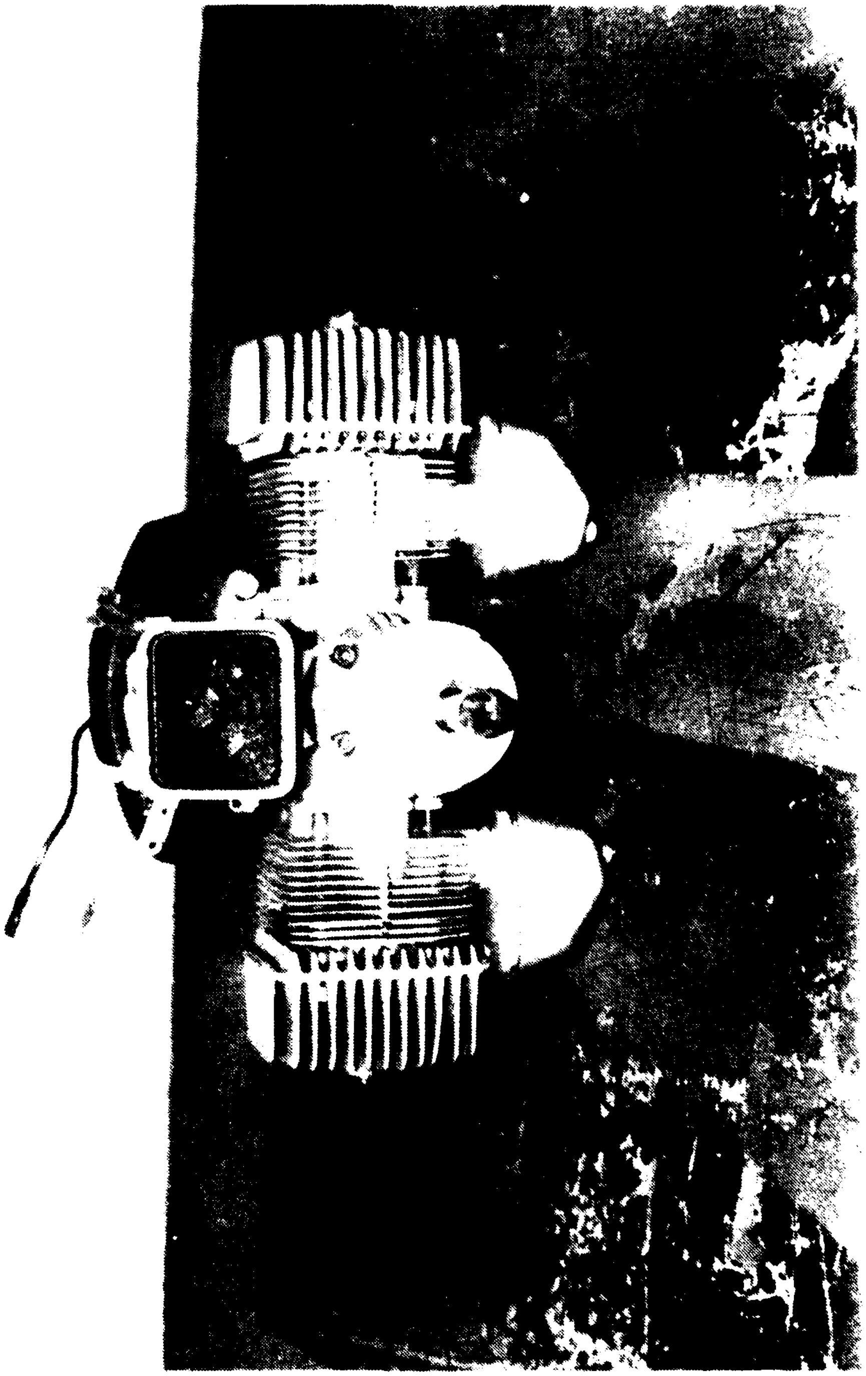


Figure 36 Super Tartan Engine (Front View)

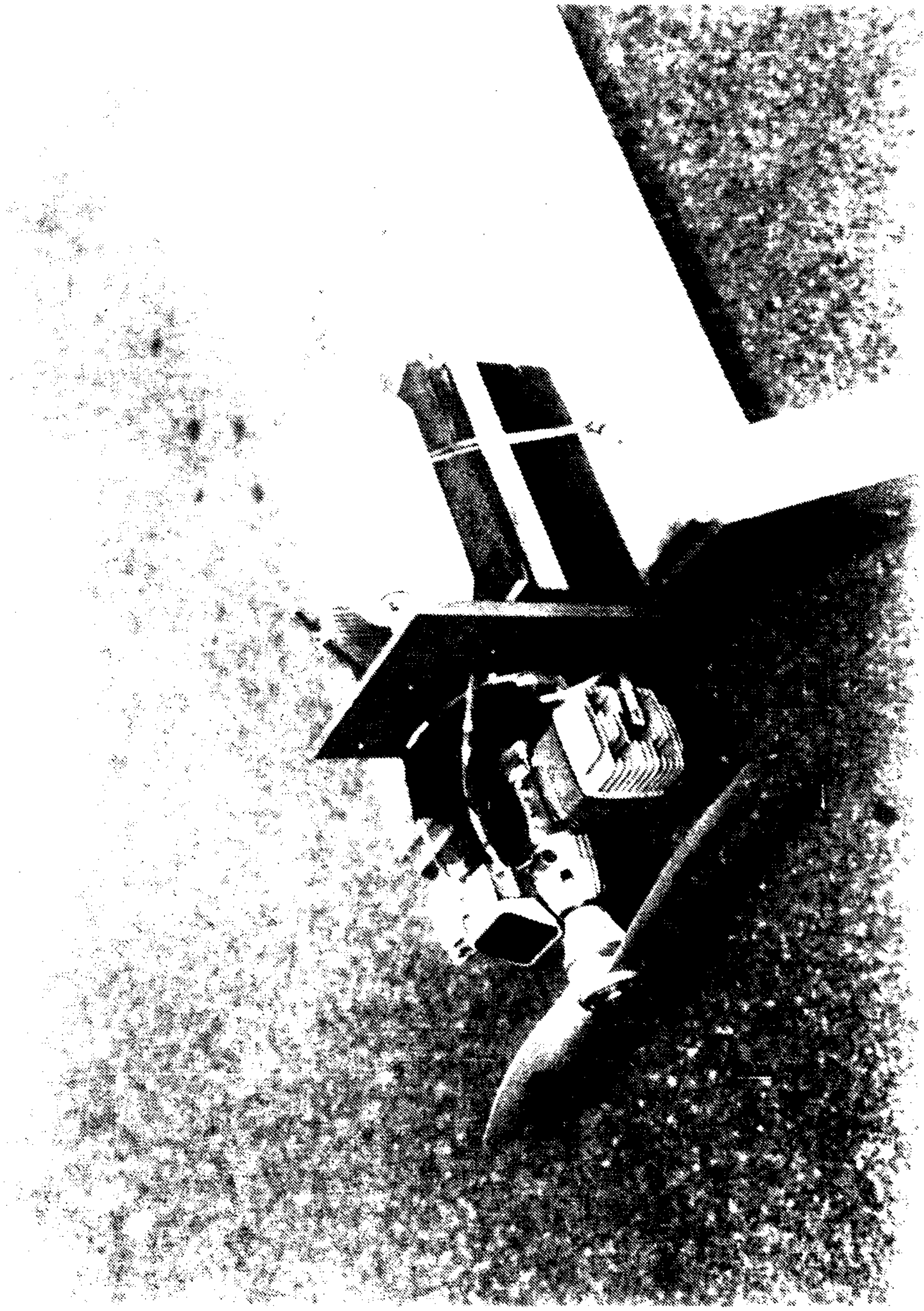


Figure 37 Engine Mounted To The Test Stand

two hour period was over, the engine was run to check for vibration. At low rpm the engine shook from side to side but the propeller seemed to remain steady. A cut down propeller was put on the engine. The initial indication from running this propeller was that there was a significant loss of thrust. The cut down propeller seemed to vibrate more at low rpm but at higher rpm it steadied out.

Initial engine testing indicated a significant loss of thrust with a large increase in rpm. A quantitative indication was now needed to see if enough thrust was developed by the engine for vertical takeoff. A thrust stand was built to accomplish this. The thrust stand was basically a platform on bearings with an engine stand and a scale. The engine was then mounted to the thrust stand and tested (Figure 38). The Futaba PCM 1024A transmitter has a built in tachometer sensor and was used to monitor the engine rpm (Figure 39).

Engine testing consisted of three parts: throttle setting, thrust measurement, and rpm monitoring. Three propeller sizes were chosen: 18-6, 20-6, and 20-8. Three propeller versions were also chosen: full size, two bladed, and four bladed. The full size was used as a data base, and the outcome of the cut version would determine which would be used in the actual flight test. The propellers were cut down to 11 1/8 inch length and the tips were rounded to a radius of 5 9/16 to reduce tip losses.

Each propeller size and version was tested at three engine throttle settings: 1/2 throttle, 3/4 throttle and full throttle. The procedure was to set the throttle and monitor the rpm. Once the rpm settled down to a relatively steady amount, values for the thrust and rpm were recorded. The test was then repeated for each throttle setting

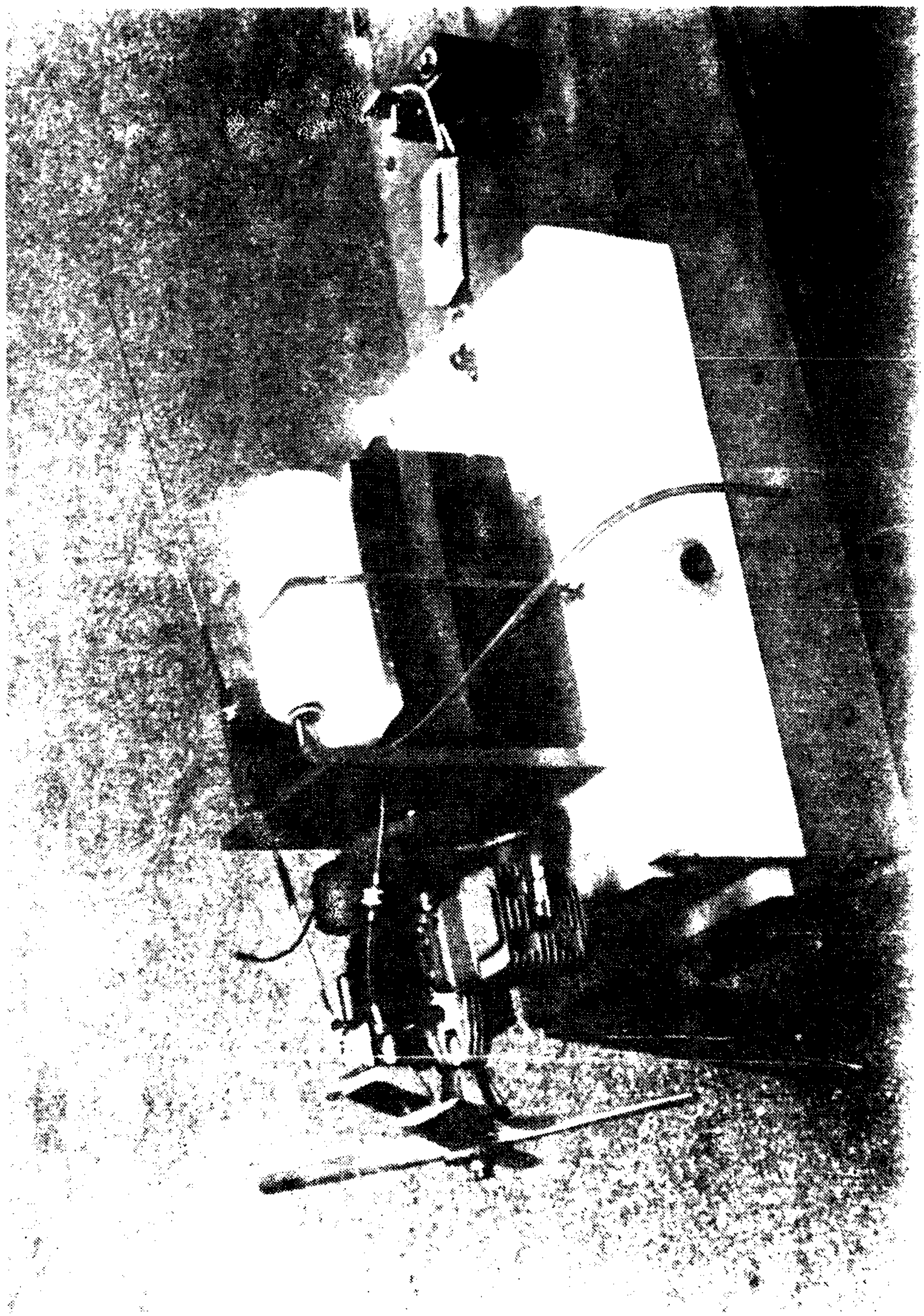


Figure 38 Engine Mounted To Thrust Stand

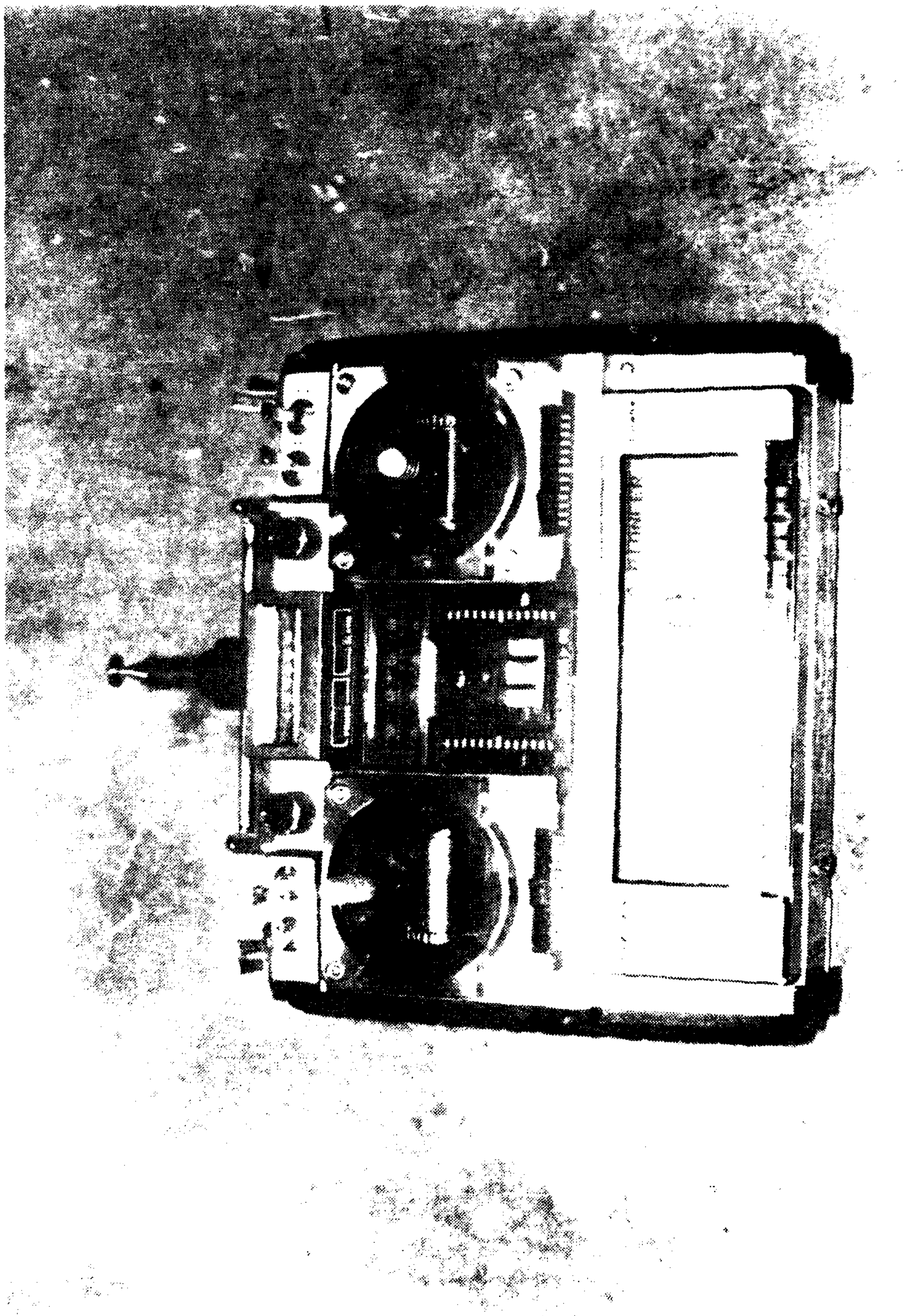


Figure 39 Futaba Transmitter

and for each propeller size and version.

The results of the test were not promising. The full size propellers provided the expected amount of thrust of about 20 pounds. The cut versions did not meet expectations. The two bladed version's range of maximum thrusts were from 5 to 6 lbs which indicated about a 75 % loss in available thrust. The four bladed propellers' thrust was identical for the 18-6 and 20-6. The anticipated increase in thrust by using the four bladed propellers did not materialize. There was a moderate gain of about 8 %, but it was still not enough thrust to provide the ability to take off vertically. The 20-8 four bladed propeller was not available for testing. The results from the previous tests would tend to indicate that the 20-8 four bladed propeller probably would not provide enough thrust either, and therefore will not be tested.

The thrust and rpm for each setting were collected and are contained in Tables 3 through 10.

TABLE 3 18-6 FULL SIZE PROPELLER

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	12.5	7,100
3/4	18.0	7,300
FULL	19.5	8,100

TABLE 4 18-6 PROPELLER 2 BLADE CUT VERSION

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	4.0	10,600
3/4	4.5	11,300
FULL	5.0	11,500

TABLE 5 18-6 PROPELLER 4 BLADE CUT VERSION

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	5.0	9,000
3/4	6.0	10,000
FULL	6.5	10,100

TABLE 6 20-6 FULL SIZE PROPELLER

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	14.0	7,500
3/4	19.0	8,500
FULL	20.0	8,600

TABLE 7 20-6 PROPELLER 2 BLADE CUT VERSION

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	4.0	10,300
3/4	5.0	11,500
FULL	5.0	11,500

TABLE 8 20-6 PROPELLER 4 BLADE CUT VERSION

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	5.0	9,200
3/4	6.0	10,100
FULL	6.5	10,400

TABLE 9 20-8 FULL SIZE PROPELLER

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	12.0	6,200
3/4	18.0	7,100
FULL	20.5	7,400

TABLE 10 20-8 PROPELLER 2 BLADE CUT VERSION

THROTTLE SETTING	THRUST (LBS)	RPM
1/2	2.0	8,100
3/4	4.5	10,100
FULL	6.0	11,100

The thrust loss caused by cutting down the propellers was much greater than expected. The four bladed propeller did not regain as much thrust as expected. The problem and solution lies in the pitch of the propeller. The propellers selected were long with small pitch. This length was to provide a larger surface area. However when these longer propellers are cut, the section left did not have much pitch. Most of the pitch lies near the tip. Since the propellers were essentially flat, they did not load up and therefore did not provide the necessary thrust. This lack of loading the propeller created another problem. The maximum rpm suggested by the manufacturer is 8,800 rpm. With the cut propellers, the rpm was exceeding this value, at times by as much as 3000 rpm. A propeller with more pitch, as with a full size propeller, would load the engine and decrease the rpm to a safer value.

The conclusion reached from these tests indicate a new propeller is needed. This propeller must have a greater amount of pitch and this pitch must be included in the critical first 5 inches of the propeller. The propeller tip must be rounded to cut down any tip losses, and the propeller needs to be tested in both a 2 bladed and 4 bladed version to determine maximum thrust. Finally the engine rpm must be monitored to avoid excessive rpm that could damage the engine and possibly lead to the loss of the vehicle.

V. EXISTING VTOL UNMANNED AIR VEHICLE TECHNOLOGY

Vertical takeoff and landing unmanned air vehicle technology is not a new concept. For years research and development have been carried out on several different concepts. The following is a brief review of some of the research and vehicles recently developed.

The U.S. Marine Corps Airborne Remotely Operated Device (AROD) was developed by Sandia Laboratories for the Naval Ocean System Center. The AROD was powered by a small piston engine driving a two bladed vertically mounted ducted propeller. The vehicle weighed 85 lbs and had an endurance of about an hour [REF.17:p.8]. It was capable of hovering or flying at a top speed of 30 knots [REF.18:p.896]. The AROD was controlled by radio or by tethered fiber optic cable and used four control vanes for flight control [REF.17:p.8]. An onboard controller was used for stability augmentation by decoupling the control axes, which allowed the AROD to be flown by minimally trained personnel [Ref.18:p.896]. The AROD program was canceled due to its inability to dash at a reasonable speed. The controller, however, was considered its best feature.

The controller consisted of four control loops: yaw, pitch, roll rate, and altitude rate. Sensors that provided the necessary feed back to the controller included a vertical gyro, rate sensors in all three axes, an altimeter, and a vertical accelerometer.

The roll rate and altitude rate control loops were both independent of all the other loops. Due to the gyroscopic behavior of the propeller design, the yaw and pitch control loops are dynamically coupled, and must be decoupled in order to provide the proper control signals. A linear quadratic regulator (LQR) synthesis technique was used. This LQR method was used with the multiple input/ multiple output (MIMO) type requirements of the gyroscopic coupling of the AROD's yaw and pitch controller and provided the necessary loop interconnections so that the proper control signals reach the pitch vane and yaw vane servos. A complete description of the LQR technique used is contained in reference 18.

At the Naval Postgraduate School three theses have been written related to the AROD. The first, entitled "An Autopilot Design For The United States Marine Corps' Airborne Remotely Operated Device" used optimal control theory for designing the automatic flight control system for the AROD. Optimal control theory uses feedback control gains to solve the state system. The advantage of using optimal control theory is that it provides solutions for high order, nonlinear, time varying, MIMO systems. A program called OPTCON which allows a state space system to be input and uses matrix calculations solves for the optimal feedback gains, is included in reference 19.

The second, entitled "A Dynamic Simulation and Feedback Control Scheme For The U.S. Marine Corps' Airborne Remotely Operated Device (AROD)" uses optimal control theory for designing the automatic flight control system for the AROD. In this case a linear approach vice nonlinear was used for analysis. Three reasons drove this approach. The first was that linear development is well documented and easier to implement. The second was that linear approximations have provided good results for nonlinear systems. Finally the third reason was that the optimal control solution can often be reduced to a constant gain, which means that a minimum amount of memory storage of the onboard computer will be devoted to control implementation. Reference 20 also contains a computer model of the AROD and was developed by considering it as both a gyroscope and an air vehicle. [REF.20]

The third thesis, entitled "An Inexpensive Real-Time Flight Simulator For The United States Marine Corps' Airborne Remotely Operated Device" is a flight training simulator program. This program provides simulation of a flight over terrain generated from digital data compiled by the Defense Mapping Agency. The AROD portion of the simulation is provided by the linear model developed in reference 20. [REF.21]

Other UAVs similar in design to the AROD are the Moller Aerobots P115M and R124M. The P115M is 1 ft 6 in. tall, has an overall diameter of 1 ft 8 in., a duct diameter of 1 ft 3 in. and uses a two bladed propeller powered by a 5 hp engine. It is capable of carrying a 10 lb payload. It is estimated to have a maximum speed of 56 knots and a maximum range of about 30 nautical miles (nm). The R124M is 2 ft 4 in. tall, has an overall diameter of 2 ft 6 in., a duct diameter of 2 ft and uses a seven bladed propeller powered by a 50 hp engine. It is capable of carrying a 45 lb payload. The R124M is estimated to have a maximum speed of 152 knots and a maximum range of 278 nm. Both vehicles are reported to have low radar cross sections and low noise signatures. Thrust vectoring within the ducts is used for stability and control. [REF.5:p.115-116]

Counter rotating propeller UAVs eliminate the control problem of gyroscopic coupling created by the use of a single propeller [REF.18:p.896]. Two of this type of UAV are the Canadair CL-227 Sentinel and the ML Aviation Sprite. The Canadair CL-227 Sentinel, often called the "Peanut" because of its shape, is 5 ft 4.5 in. tall, has a body diameter of 2 ft 1 in. and a blade diameter of 9 ft 2.25 in. Its maximum payload is 99.2 lb and its maximum speed is 70 knots; a typical mission endurance at an altitude of 1,640 ft is 3 to 4 hours [REF.5:P.28]. The Sentinel is capable of carrying a wide range of payload packages, including TV camera, FLIR, mini-sonars, radio-relay equipment and laser designators. The Sentinel's extensive use of composite materials has resulted in a low radar cross section (about 0.1 m^2), which includes the blade contribution. The Sentinel incorporates automatic and manual recovery methods. It is capable of lowering a Kevlar line which is connected to a pulley system and can be brought down either manually or by an electric winch. [REF.22:pp.787-788] One major disadvantage of this system is the exposed blades. As mentioned above the blade is over 9 feet wide, creating a rather large disc area where a potential problem could occur when trying to recover this UAV on board during heavy seas.

The next counter rotating type UAV is the ML Sprite. The Sprite is 2 ft 11.5 in. tall, has a body diameter of 2 ft 1.5 in. and a blade diameter of 5 ft 3 in.

[REF.5:pp.97-98]. It is powered by two 6.5 hp engines which provide for a maximum speed of 70 knots and a maximum endurance of 2 hours. The Sprite is equipped with a laser altimeter which gives it an automatic landing capability. [REF.23:p. 28]

The Sprite currently has a radar cross-section of 0.5 m^2 which will be reduced to 0.3 m^2 once in production by the installation of a cowling to smooth out its shape. The Sprite is capable of carrying a maximum payload of 13 lbs which include a thermal imager and a TV camera. Other potential payloads include jammers, chemical agents and radiation monitors/detectors, and communications relay equipment.

The major disadvantage of this system is that the rotors are exposed which create potential hazards to personnel handling the UAV. [REF.24:PP.450-452]

Bell-Boeing has designed a tilt rotor demonstrator called the Pointer. The Pointer is based on the V-22 Osprey which they developed for the Marine Corps. The first test flight was in November of 1988, but due to funding disagreements the partnership split in September 1989. Boeing took the fuselage and Bell the tilt rotor dynamics. Both companies then continued the work independently; Boeing has developed the Tracer and Bell the Eagle Eye. Initial estimates for the Pointer included a maximum speed of 160 knots, maximum range of 400 nm and a maximum endurance of 5 hours. Initial reports indicate that both UAVs will be similar to the Pointer. There are several advantages to these designs. The main is the ability to dash out and then hover. The major disadvantage is the safety concern. Once again the blades are exposed and therefore create a hazard to handling personnel. [REF.23:pp.28-29]

Three companies are currently investigating saucer shaped UAVs as a means for VTOL. The first company is Moller which has two. The first is the E410M which has a two bladed propeller in each of four ducts. The E410M is 10 in. tall, has a diameter of 2 ft 10 in., and each duct has a diameter of 10 in. It is capable of carrying a 10 lb payload. No other information was available. The second is the 200-XR which has a seven bladed fan in each of eight ducts. The 200-XR is 3 ft tall, has a diameter of 9 ft 4 in., and each duct is 1 ft 8 in. in diameter. It is capable of carrying a 400 lb payload. It is estimated to have a maximum speed of 78 knots and a range of 104 nm. [REF.5:pp.115-116]

The next company is Sikorsky. Their UAV, called the Cypher, is a 5 ft diameter proof of concept vehicle. The Cypher uses a shrouded coaxial rotor system. The vehicle was designed to eliminate the need for launching and recovery systems and would be able to land in an area of 16 m^2, which would accommodate a Frigate or Destroyer. The final design will be powered by a 65 hp rotary engine and have an endurance of 3 to 4 hours. Top speed is estimated at 70 knots. The Cypher is estimated to be able to carry a 150 lb payload. [REF.25:p.24]

The third company is Cordray. The system under development is called SHADOW, for Subsonic Hovering Armament Direction and Observation Window. It is a 24 in. diameter saucer shaped UAV. The Shadow has undergone wind tunnel testing; however no other information is currently available. [REF.26:p.13]

The saucer shaped UAVs provide the shrouding around the propeller for safety but a major draw back is that the saucer-like UAVs do not possess the necessary dash speeds that will be necessary for over the horizon targeting; it will take the vehicles a considerable length of time to get on station.

The final VTOL UAV to be discussed is the Grumman Design 754. It is a 9800 lb aircraft powered by two jet engines. The first is a 9,000 lb thrust lift engine. The second is a 1,300 lb thrust cruise engine. The 754 will be able to cruise at 210 knots for 14 hours at altitudes of 27,000 to 37,000 ft. with a 150 lb payload. The UAV will have a wing span of 51 feet and with folding wings, would be able to operate off of a destroyer or a frigate. [REF.26:p. 117]

This UAV will provide a dash speed not available with any other UAV discussed. The problem lies in the two separate engines. When one engine is not in use, the aircraft is carrying dead weight which reduces the available payload.

VI. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

The goal of this thesis was to design and build a Tilting Ducted Fan UAV to take advantage of its VTOL capability. The VTOL capability eliminates the launch and recovery equipment and deck space required by conventional UAVs. It reduces damage to the airframe or electronics by not needing to fly into a net or land in the salt water. The ducted fan provides a measure of safety for topside personnel handling the aircraft's pre or post launch cycle. The tilting capability allows the aircraft to have the best of both worlds. It will have the ability to takeoff and land vertically, and the ability to transition into horizontal flight to dash to its assigned station.

The airframe's shape was selected to provide a low radar cross-section. Composite construction techniques were used to aid in low radar reflectivity as well as to build a light weight aircraft which is able to withstand a 25 g loading.

Finally, the major advantage of building a UAV is cost savings. With today's shrinking defense dollar using UAVs to augment manned aircraft missions or to replace manned aircraft for routine or risky missions saves the lives of pilots, as well as reduces the cost of flight operations or possibly the cost of an aircraft. In flight research experimental aircraft are expensive to build and maintain. The risk of losing a pilot or aircraft may be high. The use of UAVs as technology demonstrators allows for concepts to be continually developed, and tested at a reduced cost.

The Archytas TDF was designed and built (Figure 40) but not completely tested. The aircraft was constructed to design weight; however during engine testing, it was determined that not enough thrust was being developed for liftoff. Follow up research will concentrate on a propeller design to overcome this problem.

B. RECOMMENDATIONS

Several recommendations are being made for possible follow on projects. The recommendations are contained in three categories.

The first deals with the 1/2 scale Archytas TDF. Follow up research needs to be conducted to design a propeller with sufficient pitch to provide the necessary thrust to continue with the VTOL research. With the proper propeller the Archytas TDF can be instrumented and test flown for a proof-of-concept evaluation.

The second recommendation deals with the controller. As a follow on thesis a controller needs to be developed to provide stability augmentation for vertical flight. The controller is also needed for use in controlling the transition from vertical to horizontal flight and for thrust vectoring in the horizontal flight mode. The controller will also provide some autonomy for future flight testing. The development of the controller could possibly be carried out in conjunction with the Department of Electrical and Computer Engineering.

It is finally recommended that development of the full- scale Archytas continue. Knowledge gained from flight testing the 1/2 scale vehicle will be applied to completing an air vehicle capable of vertical takeoff and landing and transitioning to and from horizontal flight.

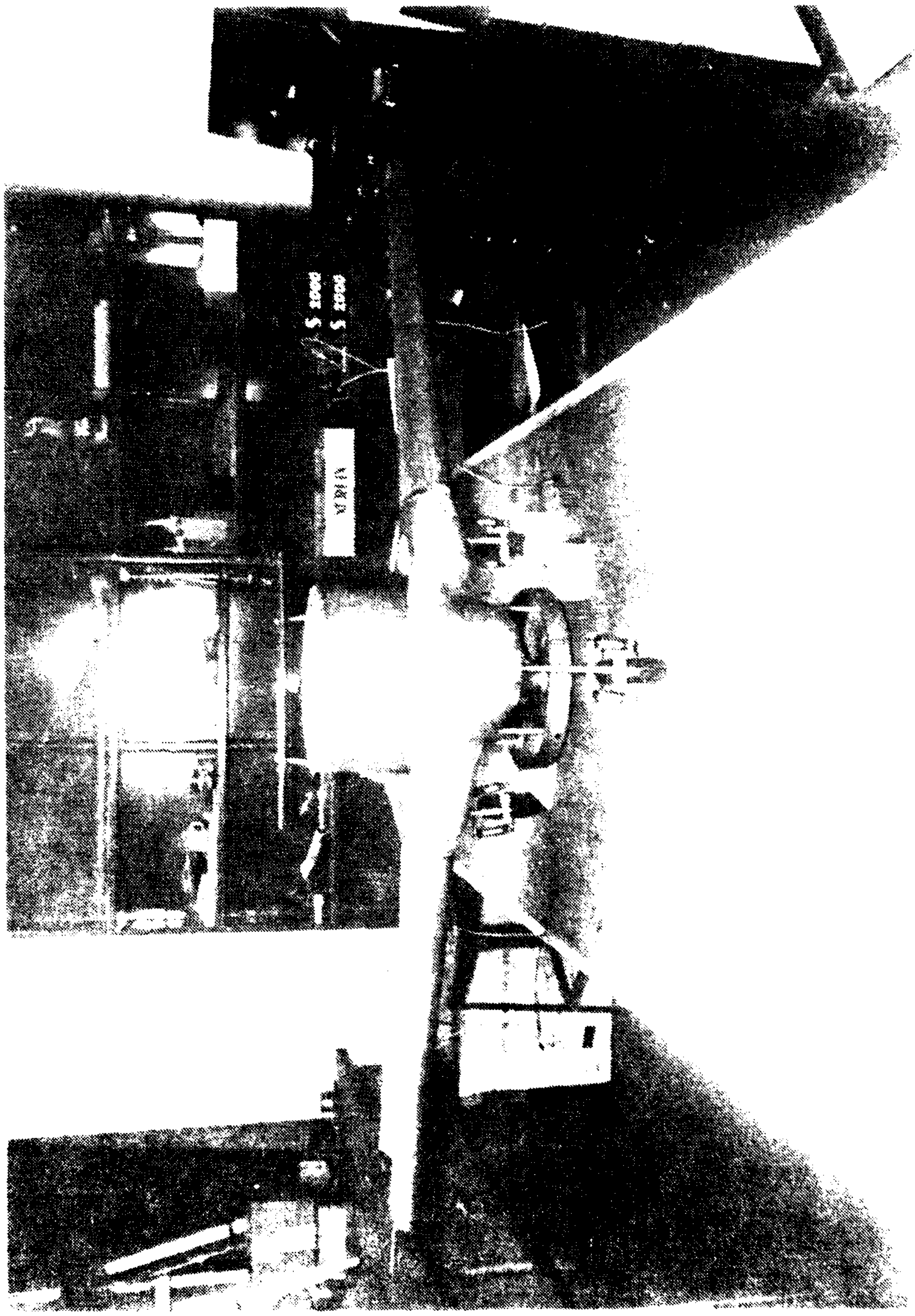


Figure 40 Completed Archytas TDF

APPENDIX A: WING STRUCTURAL CALCULATIONS

The following dimensions were used in the spar program to determine the upper and lower spar cap thicknesses, and the shear web thickness:

- Load Factor = 25.0 g
- Wing Span = 6.0 ft
- Root Chord = 1.53 ft
- Tip Chord = .974 ft
- Spar Cap Material (compressive load)
    - F_s = 43,500 psi
- Spar Cap Material (tensile load)
    - F_t = 70,000 psi
- Shear Web Material = 7,300 psi
- Spar Width = 1.5 in
- Chord Thickness = 14.13 %
- Total Wing Weight = 5.5 lbs

Table 11 shows the first computer run with upper spar cap, and shear web thickness based on a gross vehicle weight minus wings of 24.5 lbs.

Table 12 shows the second computer run with the lower spar cap, and shear web thicknesses based on a gross vehicle weight minus wings of 24.5 lbs.

TABLE 11 THICKNESS CALCULATIONS FOR W = 24.5 LBS
(NOTE: ALL MEASUREMENTS IN INCHES)

WING STATION	SPAR HEIGHT	CAP THICKNESS	WEB THICKNESS
0.0	2.08	7.54E-2	2.02E-2
3.6	2.00	6.24E-2	1.84E-2
7.2	1.92	5.04E-2	1.67E-2
10.8	1.85	3.95E-2	1.48E-2
14.4	1.77	2.97E-2	1.29E-2
18.0	1.70	2.12E-2	1.10E-2
21.6	1.62	1.39E-2	8.96E-3
25.2	1.54	8.08E-3	6.87E-3
28.8	1.47	3.71E-3	4.69E-3
32.4	1.39	9.59E-4	2.40E-3
36.0	1.32	0.0	0.0

TABLE 12 THICKNESS CALCULATIONS FOR W = 24.5 LBS
(NOTE: ALL MEASUREMENTS IN INCHES)

WING STATION	SPAR HEIGHT	CAP THICKNESS	WEB THICKNESS
0.0	2.08	4.68E-2	2.02E-2
3.6	2.00	3.87E-2	1.85E-2
7.2	1.92	3.13E-2	1.67E-2
10.8	1.85	2.45E-2	1.48E-2
14.4	1.77	1.85E-2	1.29E-2
18.0	1.70	1.32E-2	1.10E-2
21.6	1.62	8.67E-3	8.96E-3
25.2	1.54	5.02E-3	6.87E-3
28.8	1.47	2.30E-3	4.68E-3
32.4	1.39	5.96E-4	2.40E-3
36.0	1.32	0.0	0.0

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