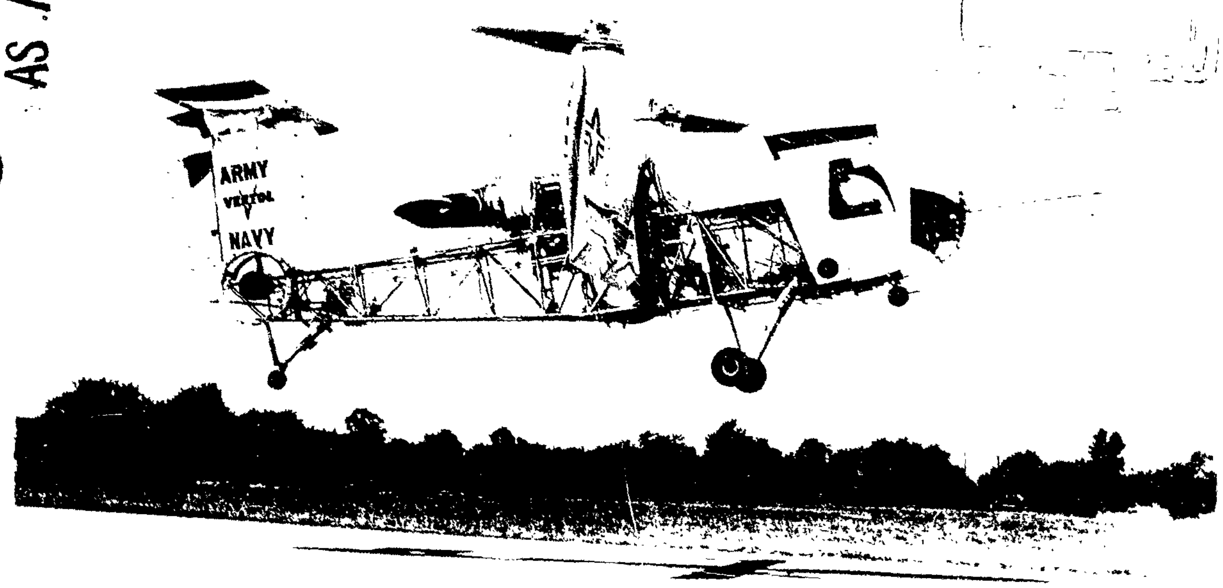
Development of the
U.S. ARMY VZ-2
(Boeing-Vertol 76)
Research Aircraft



Sponsored by the
U.S. ARMY
under the direction of the
OFFICE OF NAVAL RESEARCH

BOEING
VERTOL DIVISION

CHRONOLOGY

15	APRIL	1956	VERTOL DIVISION AWARDED CONTRACT NOnr 2136(00) TO DESIGN, CONSTRUCT, AND FLIGHT TEST THE U. S. ARMY VZ-2 TILT WING AIRCRAFT
1	APRIL	1957	ROLL OUT OF VZ-2
30	APRIL	1957	FIRST RUN-UP OF VZ-2
25	JULY	1957	COMPLETED 10 HOUR TIEDOWN TEST
13	AUGUST	1957	FIRST HOVER OF VZ-2
7	JANUARY	1958	FIRST AIRPLANE FLIGHT OF VZ-2
28	MARCH	1958	COMPLETED HOVER AND AIRPLANE FLIGHT PROGRAM
10	APRIL	1958	STARTED 50 HOUR TIEDOWN TEST
29	APRIL	1958	COMPLETED 50 HOUR TIEDOWN TEST
16	MAY	1958	STARTED FLIGHT BUILD-UP FOR CONVERSION
15	JULY	1958	FIRST FULL CONVERSION OF VZ-2 FROM HOVER TO FORWARD FLIGHT AND BACK TO HOVER
14	APRIL	1959	COMPLETED FLIGHT PROGRAM
15	APRIL	1959	EJECTION SEAT TESTED AT PHILADELPHIA NAVAL BASE
24	APRIL	1959	ARRIVED AT EDWARDS AIR FORCE BASE FOR ALTITUDE FLIGHT PROGRAM

(continued on inside back cover)

CONTRACT NOnr 2136(00)

Technical Report
R-219

DEVELOPMENT OF THE U.S. ARMY VZ 2 (BOEING VERTOL-76) RESEARCH AIRCRAFT

August 1963

Prepared by
Vertol Division
The Boeing Company
for

The U. S. Army
under the direction of the
Office of Naval Research

PREFACE

This Final Report contains a summary of Vertol Division's Development Program of the Boeing-Vertol 76, hereinafter referred to as the Army VZ-2, Tilt Wing Research Aircraft (Figure 1), sponsored by the U. S. Army and procured under the Office of Naval Research Contract NOnr 2136(00). Vertol Division's effort, from 15 April 1956 to 28 February 1963, to design, construct and flight test a tilt wing research vehicle is described.

For Vertol Division, the program was under the direction of Mr. W. Z. Stepniewski, presently Assistant Director of Engineering Research. Under Mr. Stepniewski's supervision, J. Mallen, presently Chief of Aerodynamics, conducted parametric studies which led to the conclusion that the tilt wing is one of the most promising VTOL configurations. P. Dancik was Project Engineer for the program from April 1956 until October 1962, when he was succeeded by J. Cline who is presently Project Engineer for the program. K. Gillmore and D. Richardson contributed to the development program, and J. Cline, W. Reichard and T. Tarczynski were responsible for major aircraft components. P. Sheridan, presently Assistant Chief of Aerodynamics, R. Loewy, presently Consultant for Vertol Division Engineering Department, and W. Peck, presently Chief Structures and Dynamics Engineer represented analytical groups contributing to the project. Flight Testing was under the direction of P. Dancik and J. Cline; and L. Lavassar was Chief Test Pilot for the program. Mr. L. L. Douglas, Assistant General Manager, represented the interest of Vertol Division top management for the program and personally made numerous technical contributions to the development of the VZ-2.

Vertol Division is indebted to Lt. Col. D. L. Ritter, Lt. Col. L. Robertson, and T. Wilson of the Office of Naval Research who managed the program for the Army; to J. Reeder who directed the flight research program at NASA; to F. Gustafson who, with the assistance of Vertol Division engineers, coordinated technical aspects of the program at NASA's Langley Research Center; to L. Tosti of Langley Research Center for his work on free flight and static tunnel model testing; and to P. Curtis of Princeton University for his work on the dynamic model at the Princeton Track Facility.

Acknowledgement is made for the liberal use of material from reports and memos published by NASA and Princeton University, in the preparation of this report. This material is listed in the Reference Section of the report.

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I. SUMMARY

On 15 April 1956 Vertol Division was awarded Contract NOnr 2136(00) to design, construct, and flight test a tilt wing VTOL research vehicle. This contract, which was sponsored by the Army and technically directed by the Office of Naval Research, resulted in the roll out of the VZ-2 (Figure 2) eleven and one-half months after receipt of the contract. Vertol Division's design philosophy throughout the program was guided by the Army's desire to explore the tilt-wing VTOL principle within the shortest possible time and at minimum cost. Consequently, every effort was made to simplify the program and to reduce cost without compromising the requirements for a flight article which would demonstrate the fundamental engineering principle of the tilt wing concept of VTOL. In an attempt to reduce the cost and time of development, it was decided that a minimum size aircraft should be realized. Further, pursuing this design philosophy and trying to keep technical unknowns to a minimum, Vertol Division used standard and existing components, whenever possible. This permitted more efficient use of development time on the main areas, such as the control system, the tilt wing system, and the dynamic and aerodynamic aspects.

In parallel with the design phase, model force and free flight tests were conducted at NASA, Langley Field, and a dynamically similar model was tested at the Forrestal Research Center of Princeton University. During the earlier phases of development, full scale propeller tests were performed in the 40 foot by 80 foot wind tunnel at NASA's Ames Research Center, Moffet Field, California. Prior to the first hover on 13 August 1957, ground instability tests, preliminary 10 hour tiedown tests, and taxi tests were accomplished. Additional hover and taxi tests indicated various problem areas. However, no modifications were required before airplane flights which were started on 7 January 1958.

All modifications resulting from the ground and flight tests were installed before the first 50 hour tiedown test to completely qualify the aircraft dynamic system. This test was accomplished prior to conducting build-up transition tests which were started on 6 June 1958.

During the build-up conversion testing, STOL takeoffs and landings were accomplished to evaluate the flight characteristics with the wing at intermediate tilt positions. The first full conversion was made on 15 July 1958 (Figures 3, 4, and 5). Instrumented data of stress levels, control positions and performance were recorded throughout the ground and flight tests. In addition,

still photographs were taken to record airflow over the tufted right wing in intermediate positions, and all flights were monitored with movie cameras.

The overall flight test program of the Army VZ-2 consisted of three stages:

The first stage (July 1958) represented the basic program to demonstrate conversion capabilities.

The second stage (April 1959) was started with an extended flight test program to obtain recorded data on the full flight envelope. This stage was conducted by Vertol Division at Philadelphia and at Edwards Air Force Base in California. Prior to the altitude tests at Edwards Air Force Base, the nose of the fuselage was modified to accommodate an ejection seat. At Edwards, a directional stability deficiency and a longitudinal control oversensitivity was uncovered in forward flight. These conditions were corrected and the flight program was completed at Edwards. The aircraft (Figure 6) was then delivered to NASA at Langley Field, Virginia where, with Vertol Division support, the flight research program was initiated in December 1959. The second stage of the program was further supplemented by evaluation flights performed by pilots from NASA-Ames, the Air Force, the Naval Air Test Center, and Vertol Division.

During the second stage, deficiencies associated with partial power descent were discovered, but were remedied through the incorporation of a droop snoot on the wing's leading edge.

Also during the second stage of the flight research program, the VZ-2 was tested in the full scale wind tunnel at Langley Field by NASA. A comparison was made of the aerodynamic characteristics of the aircraft with and without the droop snoot leading edge attached to the wing. In addition to this comparison, full scale wind tunnel tests of leading edge slats were included. Subsequently, an attempt was made to correlate pilots' opinions on flying qualities in forward flight, particularly during partial power descent, with wind tunnel measurements.

To evaluate the possibilities of control simplification, the aircraft was modified to provide yaw control in hover without the yaw fan. This was accomplished by means of differential ailerons outboard of the nacelles and by enabling the pilot to phase out the yaw fan. On 28 December 1960, flight tests

performed by a NASA test pilot indicated the feasibility of this concept. When the initial flights and evaluations were completed by NASA and Armed Forces pilots, full scale wind tunnel tests were performed at NASA, Langley Field.

The third stage (November 1961) of the VZ-2 flight research program resulted from the above tests. The aircraft was modified by the addition of full span flaps and ailerons suitable for yaw control in hover. Also, to compensate for the increased weight due to the various modifications to the aircraft during the overall flight research program, the transmission was upgraded from 630 horsepower to 700 horsepower. A second 50 hour tiedown test and a two hour flight check of the VZ-2 was then performed by Vertol Division. Finally, in the latter part of 1962 and early 1963, NASA initiated inspection and acceptance tests at Langley Field, and scheduled their new program of evaluation of the VZ-2 in its present configuration. Table I lists the physical characteristics of the VZ-2 in its present configuration.

TABLE I

PHYSICAL CHARACTERISTICS OF THE VZ-2 AIRCRAFT	
ROTORS	
Diameter (before flap modification), ft.	9.5
Diameter (after flap modification), ft.	9.67
Blade Chord, in.	13
Blade twist (linear, root to tip), deg.	19.2
Airfoil Section	NACA 0009 with 0.5 in. cusp.
Blade Taper Ratio	1
Solidity $\left(\frac{nc}{\pi R}\right)$ (before flap modification)	0.218
Solidity $\left(\frac{nc}{\pi R}\right)$ (after flap modification)	0.215
Distance between Propeller Axes, ft.	14.67
Operational Speed, rpm	1,416
Differential Pitch, deg.	$\pm$ 2
BASIC WING	
Span (Excluding tips), ft.	24.88
Chord, ft.	4.75
Airfoil Section	NACA 4415
Taper Ratio	1
Sweep, deg.	0
Dihedral, deg.	0
Pivot, percent Chord	37.6

TABLE I (continued)

Ailerons -	
Chord, ft.	1.25
Span, ft.	5
Tilt Range (referenced to upper longeron), deg.	9 to 85
MODIFIED WING	
Span (excluding tips), ft.	24.88
Chord, ft.	5.25
Airfoil Section	Modified NACA 4415
Taper Ratio	1
Sweep, deg.	0
Dihedral, deg.	0
Pivot, percentage chord	34.0
Ailerons -	
Chord, ft.	1.05
Span, ft.	10.3
Flaps -	
Chord, ft.	1.76
Span, ft.	10.3
Tilt Range (referenced to upper longeron), deg.	9 to 85
VERTICAL TAIL	
Height, ft.	5.43
Approximate Mean Geometric Chord, ft.	5.90

TABLE I (continued)

Sweep at leading edge, deg.	28
Basic Airfoil Section	NACA 0012
Rudder -	
Chord, in.	21.5
Span, in.	58.0
HORIZONTAL TAIL	
Span (less tips), ft.	9.90
Chord, ft.	3.00
Sweep, deg.	0
Taper Ratio	1
Airfoil Section	NACA 0012
Dihedral, deg.	0
Length (distance from wing pivot to leading edge of tail), ft.	10.475
Hinge Point (distance from leading edge), in.	8.3
CONTROL FANS	
Diameter (both fans), ft.	2.00
Moment arm about Wing Pivot (both fans), ft.	12.35
Number of Blades	4
Rotor Speed, rpm	5,850
Fuselage Length, ft.	26.4
Engine	Lycoming T53

TABLE I (continued)

Weight as Flown with Ejection Seat, lb.	3,500
Center of Gravity (for 9° wing incidence), percent M.A.C.	33.5
Center of Gravity (for 85° wing incidence), ft. forward of pivot point (measured along longitudinal axis)	0.135
Weight after Flap Modification, without Ejection Seat, lb.	3,653
MOMENTS OF INERTIA	
Aircraft Weight = 3,432 lb -	
I_X, Slug-ft^2	1,634
I_Y, Slug-ft^2	2,937
I_Z, Slug-ft^2	3,988
Aircraft Weight = 3,204 lb -	
I_X, Slug-ft^2	1.560
I_Y, Slug-ft^2	2,899
I_Z, Slug-ft^2	3,985
TOTAL CONTROL TRAVELS	
Lateral Stick, in.	$9\frac{1}{8}$
Longitudinal Stick, in.	$11\frac{1}{8}$
Pedal, in.	6

II. CONCLUSIONS

During the six year development of the U.S. Army VZ-2 research aircraft, Vertol Division gained extensive experience in the VTOL field. The following conclusions are based upon this experience:

1. The technical feasibility of the tilt wing concept has been proven, and a sound basis has been established for the development, design, and construction of a tilt wing aircraft for specific missions.

2. Although new problems were exposed, methods of solving them were indicated. The concentration of hovering controls exclusively within the propeller-wing assembly, by using full span ailerons for yaw control, differential collective pitch for roll control, and monocyclic pitch for pitch control, will eliminate the need for yaw and pitch fans in future models.

3. A more complete comparison and correlation of test results (NASA flight test, Contractor flight test, NASA wind tunnel and Princeton University Long Track data) will provide comprehensive technical data for future designs.

III. RECOMMENDATIONS

Based upon the insight gained into the tilt wing concept during the development of the U. S. Army VZ-2 research aircraft, Vertol Division recommends the following:

1. In order to extract the maximum amount of information that can be of use in future tilt wing aircraft design, establish a program to analyze and correlate all available data relating to the VZ-2 development.

2. Broaden the technological base of the tilt wing concept by paving the way for further simplifications of this concept through extended aerodynamic and structural research and development in the following areas:

    a. Establish realistic control and stability requirements, and determine the simplest possible systems for meeting these requirements. Use of monocyclic pitch control and propeller slipstream for yaw control are examples of this approach.

    b. Investigate methods of improving flight characteristics in conversion and partial power descent of tilt wing aircraft.

    c. Study ground effect in both hovering and STOL operations of tilt wing aircraft.

    d. Investigate the most suitable means of propeller controls and of governing the propeller-powerplant system, including horizontal gust aspects at high forward speeds.

    e. Develop technologies to establish the simplest possible propeller blade design.

    f. Conduct studies of fail-safe dynamic systems for use by tilt wing aircraft.

IV. DESCRIPTION OF AIRCRAFT

INTRODUCTION

Early in 1956, Vertol Division was awarded Contract 1681(00), sponsored by the U. S. Army under the direction of the Office of Naval Research, to make a comprehensive parametric study of various VTOL configurations. This study, Reference 12, indicated that six systems (Figure 7) appeared to be most promising for meeting the Army's requirements for a transport mission requiring 300 miles per hour or more cruising speed, a mission radius of 425 statue miles, and limited hovering capability. Of the various VTOL configurations studied, Vertol Division considered the tilt wing concept to be the most promising. One of the primary advantages of the tilt wing configuration is its exceptional payload to gross weight ratio, as shown in Figure 8. On 15 April 1956, shortly after completion of the parametric study, Vertol Division was awarded Contract NOnr 2136(00) to develop the tilt wing flight research aircraft, and preliminary design analyses were initiated immediately.

The U. S. Army VZ-2 is a tilt wing aircraft combining rotary wing and fixed wing capabilities (Figure 9). It is powered by a Lycoming T-53 engine which drives the two main rotors and two tail fans. The main rotors are used for vertical lift and roll control in hover and for propulsion in forward flight. The tail fans provide pitch and roll control in hover and supplement the tail surfaces in forward flight, since the fans and tail surfaces operate simultaneously through all regimes of flight. Modifications, incorporated in 1961, provided additional yaw control in hover through slipstream submerged full span ailerons. In addition, a pilot operated mechanism is provided to permit limited or complete phaseout of the yaw fan. Major components of the VZ-2 are described below:

A. AIRFRAME, EMPENNAGE AND LANDING GEAR

The fuselage is a welded 4130 steel tube truss (Figures 10 and 11). The cockpit area (Figure 12), which is enclosed by a large plexiglass bubble, houses an ejection seat, the flight and engine controls, and instrumentation. The tail is essentially a "T" configuration with a pitch fan embedded in the horizontal tail and a yaw fan located at the base of the vertical tail. The horizontal tail is

mechanically counterbalanced and is the all-flying type (Figure 13).

The engine is mounted on top of the fuselage, in the area of the wing trailing edge, and utilizes a bifurcated exhaust duct to prevent thrust asymmetry. The engine air inlet is faired with a bellmouth. The fuel tank is located in the fuselage under the wing hinge (Figure 14).

A conventional tail wheel landing gear is used with an auxiliary bumper wheel under the cockpit.

B. WING

Conventional aluminum alloy stressed skin construction is used in the wing (Figure 15). The main spar is approximately at the quarter chord point, with the hinge and fuselage cutout immediately behind the spar. Nacelles are located midway on each half span of the wing, so that the wing is completely immersed in the slipstream outboard of the fuselage. Originally the wing section was an NACA 4415 airfoil. In 1961 it was modified with the addition of a full span trailing edge flap as shown in Figure 16. The flap was added to improve the rate of descent capability of the aircraft. To make it easier for the pilot to fly through transition, flap deflection is programmed with wing tilt. The aft segment of the double slotted Fowler flap also serves as an aileron for lateral control in conventional airplane flight and as a yaw control device during hover and conversion. An aileron deflection of 20 degrees "down" and 30 degrees "up" is provided throughout all flight regimes (Figure 17).

C. PROPELLERS

The aircraft had two, three-bladed wooden propellers which were 9 feet, 6 inches in diameter; later, the diameter was increased to 9 feet 8 inches when the wing was modified with trailing edge flaps. For manufacturing simplicity, a constant chord blade with linear twist was selected. A twist of -24 degrees was used to provide an acceptable compromise between good static thrust in hovering and sufficient propulsive efficiency in forward flight. Flap hinges are provided for the blades to relieve the large moments of a rigid propeller operating at high angles of attack in the presence of a wing. The conventional

helicopter-type lag hinge was omitted to avoid the possibility of mechanical instability.

D. DRIVE SYSTEM

Power from the engine is transmitted through a main gear box, immediately ahead of the engine, transversely through the wing hinge to gear boxes, and forward to the propellers (Figure 18). Total reduction in this system is from 18,735 engine turbine rpm to 1,410 rpm, (or from an engine shaft rpm of 5,860 to a propeller rpm of 1,410). The auxiliaries and tail fans are driven from the central gear box through a lower central gear box. Tail fan shafting is led inside the fuselage to gear boxes at the fans.

E. CONTROL SYSTEM

The cockpit controls, which consist of stick, rudder pedals, and collective pitch lever, are typical of those found in helicopters. An "Up-Down" switch to position the wing is provided on top of the stick. The only engine controls are a speed selector and a power lever. These are mounted on the left hand side of the cockpit just forward of the collective pitch lever. This arrangement of the controls permits the pilot to fly the aircraft without removing his hands from the control stick or collective pitch lever. Propeller speed is selected manually prior to takeoff and is automatically maintained during flight by a governor on the engine. Figure 19 is a schematic of the VZ-2 Control System.

Longitudinal control in hovering and transition is accomplished by means of a horizontal tail fan. This fan is sized to develop adequate control over the available center of gravity range. Control of the fan is achieved by changing blade pitch. Since the fan is powered by the main drive system, it operates as long as the main rotor propellers are turning. The primary longitudinal control in forward flight is the elevator. Authority of the horizontal fan and of the elevator is partially reduced with decreasing wing tilt. In this way, oversensitivity to pitch control in forward flight is eliminated.

Lateral control in the airplane regime is achieved by use of conventional ailerons. As the wing tilts up, a phasing system transfers lateral stick motions from the ailerons to differential collective pitch of the laterally disposed

propellers which provide control in hovering and transition. Yaw control in hovering and transition is accomplished by means of differential ailerons and flaps and/or a yaw fan. Authority of the yaw fan remains unchanged regardless of wing position. However, the pilot can, if he wishes, either partially or completely phase out the yaw fan. Primary yaw control in forward flight is provided by a rudder.

All of the control systems incorporate hydraulic boost. A Stability Augmentation System (SAS) is provided for the roll and pitch axes (Figure 20). This sytem produces control inputs proportional to angular velocities in the extensible links which move the appropriate controls hydraulically to stabilize the aircraft. Differential extensible links are used so that there will be no feedback to the cockpit controls. SAS is used in hover and in conversion to a wing tilt of approximately 45 degrees, after which it can be turned off to avoid pitch oscillations in forward flight.

F. INSTRUMENTATION

Instrumentation (Figure 21) was installed to gather information on performance, control, stability and strain data. The original instrumentation installed in the VZ-2 aircraft included an 18-channel oscillograph. This was replaced at a later date with a light-weight, 36-channel recorder which was loaned to Vertol Division by the NASA, Langley Field facility. The addition of this NASA recorder greatly contributed to the program by reducing the number of flights needed to obtain accurate data for performance and control analyses.

G. STRESS CRITERIA

Stress criteria was established for 2,700 pounds gross weight and was based upon a 4g symmetrical load factor in the aircraft configuration. However, because of many modifications (ejection seat, full span flaps and ailerons, empennage modifications, etc.) incorporated during the program, gross weight increased to 3,700 pounds. As a result, g levels of 2.9 and 2.2 now represent symmetrical and unsymmetrical flight load factors for the forward flight configurations. Stress levels for high speed yaw and roll were investigated and jump takeoffs were calculated to a 2.0 g limit vertical flight load for hover. Fatigue analysis of the drive system was based upon continuous hovering at the rated power of the engine. The propeller system was stressed, under the original

design criteria, for a high speed flight condition as a helicopter rotor and for maximum loads during the conversion cycle. Although this was performed at 2,700 pounds gross weight, operating allowables on the propeller assembly were determined by bench tests, and these allowables have not been exceeded in the present gross weight configuration. Unlimited life of the dynamic components was provided for the above fatigue conditions.

V. TECHNICAL PROGRAMS

INTRODUCTION

The technical program is discussed in three phases: the first phase covers the basic development and flight test programs; the second, the extended flight test program; the third, the final development program which covers evaluation of the full span flaps and ailerons. The entire program demonstrated the feasibility of the tilt wing concept.

A. INITIAL DEVELOPMENT PROGRAM

1. DEVELOPMENT TESTING

Development testing to obtain fundamental data on performance, control, and stability was conducted at Vertol Division and at other installations, such as NASA-Langley Field, NASA-Ames, and Princeton University. This testing included the following:

a. Propeller Dynamic Study - Early in the program Vertol Division conducted a propeller dynamic study to predict the steady and vibratory flap and chordwise bending stresses of the propeller blades as they are affected by variations in propeller thrust, wing-tilt angle, and forward speed. This study predicted the general stress levels of the blades prior to the first conversion flight, thereby assuring structural integrity of the blades in the most critical flight regimes. Later, the accuracy of these predictions was confirmed during the instrumented flight program.

b. Smoke Tunnel Tests at Princeton University - Flow phenomena, associated with the propeller-wing combination, were studied through flow visualization in the smoke wind tunnel at the Forrestal Research Center of Princeton University. This study indicated that the propeller slipstream helps reduce the actual wing angle of attack with respect to the resultant slipstream (Figure 22); and that, due to the rotation of the slipstream, the degree of stall alleviation is slightly different on either side of the propeller axis.

c. <u>One-Quarter Scale Model Tests at Princeton University</u> - A one-quarter scale model of the VZ-2 (Figure 23) was constructed and tested at the Forrestal Research Center of Princeton University. With the aid of this model, insight was gained into the control, stability, and general aerodynamic phenomena of transition. Tests were conducted under simulated free flight conditions to obtain aerodynamic and stability derivatives of the tilt wing. From tests of the model in the tail-off configuration, a relationship was obtained between wing-tilt angle and tail force required to balance the pitching moment and between the wing tilt and flight speeds corresponding to steady state, accelerating, or decelerating flight conditions. The results of these tests, conducted at Princeton, are shown in Figure 24.

d. <u>NASA One-Quarter Scale Model Tests</u> - A one-quarter scale powered model (Figure 25) was constructed at NASA-Langley Field to study control and stability characteristics (Reference 35). This model was designed to be dynamically and geometrically similar to the VZ-2 (Figure 26), so that data could be obtained on propeller differential collective pitch, ailerons, horizontal tail, and rudder. However, there were several differences between the model and the research aircraft: the ailerons, horizontal tail, and rudder motions of the model were either neutral or in full deflection; hovering pitch and yaw air jets replaced the tail fans of the full-scale aircraft; and propeller thrust was varied by controlling the speed of the model's drive motor, rather than by changing collective pitch. However, rolling motions in hover and transition could be controlled by varying the differential collective pitch, as in the full-scale aircraft.

Four tests were conducted to determine stability and control characteristics: two of these consisted of remotely controlled free flights in the full-scale wind tunnel and the Langley Control Line Facility; and two consisted of force test investigations in the wind tunnel.

Both free flight tests were flown and evaluated by pilots who controlled each axis of the model from remote stations on the ground. Flight regimes included hovering in still air, vertical takeoffs and landings, and slow and accel-

erated transitions at constant altitude from hovering to forward flight. In the first series of free flight tests, which were conducted in a full-scale wind tunnel, stability and control characteristics were generally satisfactory, except for a controllable long period pitching oscillation in hover and a large pitch moment at about 70 degrees wing incidence. In the second series of free flight tests, which were conducted at the Langley Control Line Facility, rapid transitions from hovering to forward flight could be performed easily. However, the transitions from forward flight to hovering were occompanied by a strong nose-up pitching moment which occasionally was uncontrollable because of the limited pitch control. Also, the model was more difficult to control during slow decelerations and during any given wing-tilt for aft center of gravity conditions.

The force test investigations (Reference 17), conducted by NASA, were designed to obtain data for the measurement of longitudinal and lateral stability and control characteristics in forward flight and in the transition ranges. The first series of tests provided preliminary data; the second, more advanced data.

e. <u>Tail Fan Test Under CH-21 Whirl Tower</u> - Ducted fan tests were conducted at Vertol Division to arrive at the most efficient design for the VZ-2 tail control fans. A VZ-2 tail assembly was mounted beneath the Vertol Division CH-21 rotor blade whirl test rig (Figure 27), and tests were performed to evaluate the conditions that would be encountered by the proposed design within the flight envelope in forward flight. In addition general behavior of the fan was observed and, with the aid of tufts, flow was studied. No operational difficulties were indicated in forward flight and firm design of the tail control fans was established.

f. <u>NASA Slipstream Flow Tests</u> - A number of brief investigations were conducted by NASA to determine the character of the slipstream flow along the ground (Reference 18). These tests were made with three different models: a small scale general research model having four 3-bladed propellers; a one-eighth scale model of the Hiller X-18; and a one-quarter scale model of the VZ-2. In general, the tests involved tuft surveys and slipstream dynamic-pressure measurements. A more extensive series of tests were conducted with the VZ-2 model hovering near the ground to obtain erosion measurements of

gravel near the slipstream and to investigate the unsteady rolling, yawing and pitching moments in this regime.

The slipstream flow tests led to the following conclusions:

1. The flow studies indicated the presence of a stronger and deeper slipstream flow along the center line of the aircraft in the spanwise direction. This effect is caused by an intensification of the individual slipstreams as they meet at the planes of flow symmetry. The intensified flow along the center line of the aircraft is amplified by the presence of the fuselage which causes the dynamic pressure to be greater in front of the aircraft than one would expect, considering the slipstream of the individual propellers.

2. Gravel, if sufficiently small, was rapidly eroded by the slipstream and could be thrown high into the air if it struck even very small fixed obstacles on the ground: i.e., obstacles with a height of less than the diameter of the gravel.

3. The propellers, reacting to an erratic inflow from the recirculating slipstream, are apparently a primary source of erratic moments experienced by tilt wing VTOL aircraft operating near the ground.

g. Ejection Seat - This particular modification was initiated under the Initial Development Program, prior to the extended flight test program at Edwards Air Force Base, California. The final ejection seat installed in the VZ-2 research aircraft was adapted, with assistance from the Navy Air Crew Equipment Laboratory, from basic units previously flight tested at the Philadelphia Naval Base. Vertol Division testing was performed in an ejection rig (Figure 28). To compensate for the directional instability caused by these modifications, the tail pipe was redesigned and a dorsal fin was added to the aircraft.

2. SUPPORTING ANALYTICAL STUDIES

Analytical study programs were performed by Vertol Division throughout the Initial Development Program to provide sufficient analytical methods and data for predicting the qualities needed to develop a tilt wing research aircraft. These programs consisted of investigations of performance, controllability, and

stability -- with and without SAS -- as well as vibration, and surveys of strain and noise levels.

a. <u>Performance</u> - Performance analyses were made for hover, conversion, airplane flight and short takeoff. The data was obtained from a combination of experimental and theoretical data. The experimental data was derived from wind tunnel model tests and full scale flight tests of the VZ-2 during the Initial Development Program. Theoretical data, obtained by this study, was based upon conventional aircraft theory in airplane flight and theoretical force and moment equations adapted to an IBM 650 computer program (Reference 6) for the conversion, or transition phase. The following major trends were indicated for each flight regime by this performance study:

   1. <u>Hover</u> - In the latter stages of the Initial Development Program, ample data was obtained from numerous flight tests to define hovering performance.

      Agreement, as indicated by Figure 29, was achieved between theoretical rotor analysis and flight test data. In the region where the VZ-2 aircraft operates during hover (wing incidence of 85 degrees), a difference of 17.5 horsepower exists between the power required as predicted by NASA in TN D-318 Wind Tunnel Data (Reference 39) and that shown by theory and flight test. The reason for this discrepancy may be due to the difference between the configuration of the wind tunnel model and the research aircraft. The wind tunnel model has a drive motor of relatively large diameter in the rotor slipstream. This motor could produce a larger slipstream momentum loss than that experienced by the VZ-2 aircraft with its wing located in the rotor slipstream.

      As shown in Figure 30, a difference exists between the thrust versus collective pitch plot, predicted in NASA TN D-318, and flight test and theoretical data. This discrepancy may also be due to the difference between the wind tunnel model and the VZ-2 research aircraft.

      Figure 31, obtained from C_T versus C_p plots, presented in Reference 8, for the VZ-2 rotor, shows that the maximum Figure of Merit (.7) occurs at a value of C_T/σ which corresponds to the design hover weight of the VZ-2 research aircraft.

stability -- with and without SAS -- as well as vibration, and surveys of strain and noise levels.

a. Performance - Performance analyses were made for hover, conversion, airplane flight and short takeoff. The data was obtained from a combination of experimental and theoretical data. The experimental data was derived from wind tunnel model tests and full scale flight tests of the VZ-2 during the Initial Development Program. Theoretical data, obtained by this study, was based upon conventional aircraft theory in airplane flight and theoretical force and moment equations adapted to an IBM 650 computer program (Reference 6) for the conversion, or transition phase. The following major trends were indicated for each flight regime by this performance study:

   1. Hover - In the latter stages of the Initial Development Program, ample data was obtained from numerous flight tests to define hovering performance.

      Agreement, as indicated by Figure 29, was achieved between theoretical rotor analysis and flight test data. In the region where the VZ-2 aircraft operates during hover (wing incidence of 85 degrees), a difference of 17.5 horsepower exists between the power required as predicted by NASA in TN D-318 Wind Tunnel Data (Reference 39) and that shown by theory and flight test. The reason for this discrepancy may be due to the difference between the configuration of the wind tunnel model and the research aircraft. The wind tunnel model has a drive motor of relatively large diameter in the rotor slipstream. This motor could produce a larger slipstream momentum loss than that experienced by the VZ-2 aircraft with its wing located in the rotor slipstream.

      As shown in Figure 30, a difference exists between the thrust versus collective pitch plot, predicted in NASA TN D-318, and flight test and theoretical data. This discrepancy may also be due to the difference between the wind tunnel model and the VZ-2 research aircraft.

      Figure 31, obtained from C_T versus C_p plots, presented in Reference 8, for the VZ-2 rotor, shows that the maximum Figure of Merit (.7) occurs at a value of C_T/σ which corresponds to the design hover weight of the VZ-2 research aircraft.

2. <u>Conversion</u> - The purpose of the VZ-2 program was primarily to prove the feasibility of conversion from hover to forward flight and back to hover, using a tilt wing configuration. Consequently, a major portion of the performance study was spent investigating the transition phase.

The VZ-2 research aircraft, in the latter stages of the Initial Development Program, successfully completed numerous conversions under various conditions. As shown in Figure 32, the rotor horsepower variation with velocity during conversion, predicted by the Vertol Division's IBM Program 1442, compares favorably with that obtained from flight test. The engine power required, measured in numerous flight tests, has been corrected for transmission efficiency to obtain rotor horsepower.

The horsepower required by the pitch and yaw fans to trim the aircraft is presented in Figure 33. It is of interest to note that the control power required for trim increases from hover to about 35 knots and then decreases as the airplane control surfaces become more effective.

Wing incidence versus forward velocity is shown in Figure 34. As indicated by this plot, flight test data and Vertol Division's IBM program are substantially in agreement. At a given wing incidence, there exists a maximum difference of only five knots between the flight test data and the NASA wind tunnel data.

In addition, Figure 35 illustrates the excellent agreement of collective pitch obtained from flight test as compared to the predicted data from Vertol Division's IBM program.

3. <u>Airplane Flight</u> - Theoretical and actual power required variations with forward speed are illustrated in Figure 36. Because of the transmission structural limitations, the power available shown in Figure 36 is limited to 630 horsepower. The predicted power required is based upon a rotor rpm of 1420 to agree with flight

test data. The maximum velocity, as shown in Figure 36, is 128 knots. By operating at a lower rotor rpm, a higher propeller efficiency can be obtained and thus the maximum velocity can be increased 4 to 7 knots. This difference in efficiency is shown in Figure 37. Power required for equilibrium flight was calculated from zero thrust drag (NASA Memo 11-3-58L), and 30 horsepower was added for transmission losses, control requirements, and accessories. Except at the higher velocities, the agreement between Vertol Division's IBM program and the NASA data is acceptable. Figure 38 shows Vertol Division's data to be about one percent higher at the lower velocities and three percent higher at the higher velocities. This difference at higher velocities can be accounted for by the disagreement in propeller efficiency between the NASA Wind Tunnel Data (TN D-318) and Vertol Division's propeller analysis.

Rates of climb and descent are shown in Figure 39. Rate of climb is calculated from the excess power obtained by using the power available and analytical power required from Figure 36. Rate of descent is calculated from Figure 40, which presents C_L and C_D based upon unpowered wind tunnel model data. At approximately 100 knots, the aircraft can climb at 1800 feet per minute and descend at 3000 feet per minute.

b. Controllability - During design of this test aircraft, the only guidance regarding control response requirements in hovering was that which could be obtained from helicopter experience, and an attempt was made to meet these standards (Table II).

TABLE II

INITIAL ACCELERATION PER INCH OF CONTROL DISPLACEMENT (deg/sec^2/in.)

Axis	Helicopters	Design	V-76 Estimated from Tail Force Measurements
Pitch	8	8	7.9
Roll	25-40	25	-
Yaw	6-20	7	5.2

Flight experience seems to indicate that response in roll, corresponding to 4 degrees of differential collective pitch, was too high. When reduced to 2.5 degrees, a more balanced control system was obtained with the result that no SAS about the roll axis is required in hovering. Pitch and yaw were found to be marginally adequate. The pitch control response was brought to its present limit by increasing the total thrust and improving the response gradient in the neutral stick area (Figure 41 and References 4 and 31).

Analysis of the yaw fan installation disclosed that the fuselage tubes, lubrication lines, and the transmission case created considerable drag, making the yaw fan only 40 percent as efficient as the pitch fan. To correct this, collective pitch was increased from 20 degrees to 35 degrees; the yaw fan, which was located 4 inches from the fuselage at the tail section, was relocated laterally to 7 inches from the fuselage; the round safety guard was replaced by a ring which is elliptical in cross-section, and clearance between the ring and the fan was reduced from 1 inch to 1/2 inch; and the area was cleaned up. These changes resulted in over a 100 percent increase of effective thrust of the yaw fan for full rudder input (Figures 42 and 43).

In 1961, full span flaps and ailerons were added to improve the handling characteristics of the aircraft in various regimes of flight.

The aileron modification was made to determine the effectiveness of ailerons in the propeller slipstream in providing yaw control in hover and at low speeds. The aileron control is phased with wing incidence between rudder pedals and lateral stick (Figure 44). The effectiveness of the ailerons is discussed later in this report.

Control effectiveness in pitch and yaw rapidly improved with forward speed as action of the control fans is supplemented by aerodynamic surfaces. No difficulties were encountered about these axes, either in conversion or in steady state flights at a fixed wing tilt angle.

The problem of roll control in conversion required more attention. Since it is required that the aircraft be fully controllable in steady state flights at any fixed wing position (between hovering and airplane wing incidences),

it was necessary to provide an adequate roll control even if wing stall resulted in the development of large assymetric moments about the longitudinal axis. Through a proper scheduling of ailerons and differential collective pitch at all wing tilt angles, enough rolling moment was provided to deal even with the case of a complete stall of one-half of the wing, while the other half experiences its maximum lift coefficient.

Instrumented tests and pilot evaluation indicate that ample control for a research-type aircraft is available and that longitudinal stick trim positions are not excessive in any flight regime. Maintaining altitude under all flight conditions, even close to the ground (10 to 15 feet), does not present any problem.

c. Stability, With and Without Stability Augmentation System (SAS) - Dynamic stability studies conducted on an analog are summarized in Figure 45. They indicate that in hovering with a wing incidence of 82.5 degrees and down to a wing incidence of about 60 degrees, the aircraft is divergent in pitch motion. However, at lower wing incidences (see i_w = 50 degrees and lower in Figure 45), the aircraft becomes dynamically stable.

Figure 45 also indicates (see broken lines in Figure 45) that, with a Stability Augmentation System consisting of a pitch rate damper with a .10 second lag, stability can be obtained in hovering and at high wing incidences. Similar improvements may be expected for the roll axis.

The opinion of pilots seems to confirm the analog findings regarding the case of controlling the aircraft in hovering and at high wing incidence with the SAS.

The Stability Augmentation System, adapted from Vertol Division's helicopter SAS system, was installed in the VZ-2 to assist the pilot through transition, but subsequent tests showed that stability augmentation is not needed in all regimes of flight. Referring to the longitudinal stick position curve (Figure 46), the solid line gives a steady trim change through conversion. The broken lines represent the extremes of control motions used with the pitch rate damper turned off. It is interesting to note that the pitch rate system is helpful in providing stability augmen-

tation for hover and for approximately 25 percent of the initial wing tilt. As the wing is further tilted, longitudinal stability increases rapidly; and, during the remainder of the conversion cycle, the pilot notices negligible difference in the longitudinal stability characteristics regardless of whether the SAS is "ON" or "OFF".

d. Vibration - Vibration level of the VZ-2 in hovering, conversion, and forward flight compares favorably with that of light planes in the same weight class.

Lack of any noticeable low frequency vibrations (buffeting) in conversion and at high incidences can be attributed chiefly to the slipstream effect maintaining the wing below the stall. However, theoretical studies, as well as tuft examination, seem to indicate that the wing stall angle may be exceeded in some flight conditions and that the flow separates over some portions of the wing. The fact that, even in this latter case, buffeting can hardly be noticed can be attributed to the favorable characteristics of the NACA 4415 airfoil section.

Absence of the higher frequency vibrations, induced by the propellers, may be credited to a careful analysis of the problem and an incorporation of structural changes required for favorable dynamic characteristics of the wing-propeller combination.

The VZ-2 blades are semi-articulated, having flapping hinges but no lag hinges. The flapping hinges are used to reduce the blade root bending moment. This serves to keep the change in propeller pitching moments through transition at a reduced value.

Flapping propellers can generate large Coriolis forces in the chordwise direction of the blades. These forces are ordinarily relieved by the use of lag hinges. However, the presence of lag hinges, in combination with the many wing elastic motions, would, almost certainly, lead to some form of mechanical instability. Prevention of this instability, either through a very high natural frequency of the wing (extreme rigidity) or introduction of a complicated system for damping of wing elastic motions, would necessitate a high penalty in weight and cost. The problem was solved by taking advantage of the wing flexibility in the normal

to the chord direction, accompanied by considerable damping, which provided some attenuation of propeller induced forces in the plane perpendicular to the span. Furthermore, a deliberate attempt was made to remove natural frequencies of the wing propeller system from the propeller harmonics.

e. Strain Survey - Selected strain gauge channels were monitored throughout the ground and flight test program to confirm the structural integrity of critical areas and to assist in the analysis of failures. Reference 26 summarizes the effort required to gather, measure, and compare strain data with the design loads and the bench test endurance limits.

f. Noise Survey - Since the VZ-2 is a research vehicle, no attempt was made to control noise level, either in the cockpit or externally. However, under TRECOM contract DA 44-177-TC-562, "Study to Establish Realistic Acoustic Design Criteria for Future Army Aircraft," some noise data was collected on this aircraft. Generally, this data indicated that internally the sound levels of the VZ-2 were comparable to a fixed wing utility aircraft and externally to a light army helicopter (Reference 33).

3. BASIC TEST PROGRAM

A buildup test program was initiated for the VZ-2 research aircraft to investigate all critical areas of the design prior to performing the ultimate VTOL, STOL and conversion flights. Structural tests of critical components were performed on the bench. When the research aircraft was completed, a series of ground tests was conducted to check out the dynamic system functionally to obtain preliminary structural, as well as operational, information before performing the flight tests.

a. Structural Tests - Numerous structural tests were conducted on the components of the VZ-2 research aircraft to insure its structural integrity. The tests performed consisted of bench fatigue tests, ultimate strength tests, and tests of a functional nature. The following criteria were used to determine which components required testing:

1. The most critical components.
2. The difficulty of analysis for structural and aerodynamic loads.

3. The components showing any history of failure during operation. The extent and history of the bench fatigue tests are presented in Reference 26.

b. Ground Tests - Before conducting flight tests, a ground test was performed consisting of an operational checkout, a mechanical instability test, a ten hour tiedown test, and fan tests.

1. Operational Checkout - Prior to any operational tests, static calibration and proof load of the control system were conducted in both the airplane and helicopter configurations. Only minor adjustments were required to obtain satisfactory control surface motions, and the proof load test indicated complete control system integrity for anticipated flight loads. The initial ground operation of the aircraft involved depreservation of the engine and wing transmissions, and functional checkout of the wing tilting mechanism and control system. This was accomplished without the tail fans or propellers installed. No major discrepancies were noted. Next, the complete dynamic system was installed, with the exception of the propellers, and a no-load run was conducted in this configuration to obtain transmission gear pattern checks. Inspection of the gears at the completion of the run indicated satisfactory patterns, and the aircraft was completely assembled in preparation for the ground tests.

   Functional checks of the hydraulic and brake systems were performed to demonstrate proper operation of all components and tightness of the systems. The electrical system and communications equipment were given a thorough functional checkout and proved to be satisfactory.

2. Mechanical Instability Test - To provide reasonable assurance that no dangerous mechanical instability existed, attempts were made at various rotor speeds from ground idle to normal operating rpm to excite the instability of the aircraft: first, by pilot applied control motions; and later, by externally applied shaking forces. These external forces were applied by shaking one wing with a variable frequency electronic shaker with force applied in a direction which would

excite coupled flap bending torsion and chord bending wing modes, simultaneously. Also, attempts were made to excite the system with impact loading by striking the wing with a heavy rubber mallet.

During the initial mechanical instability test, the aircraft was ballasted to approximately 5000 pounds gross weight in addition to being tied to the ground with slack safety cables. Later the ballast was removed and the instability test performed at normal gross weight. A snubbing rig (Figure 47), consisting of four cables connected to each nacelle, was provided. The lines of force of these cables intersected the center of rotation of the rotor disc. Each of these cables was interconnected through a whiffle tree system. This cable system was secured to a lever arrangement held charged by a bomb shackle which, when released, permitted a large dead weight to be dropped, thereby quickly snubbing the aircraft to the ground. Four separate electrical bomb shackle release buttons were provided: one for the pilot, and three for observers. In addition, a mechanical release control was provided for the observer who was monitoring the aircraft motions. This quick action snubbing device was chosen because of the relatively short time period during which unstable motion of the aircraft could develop.

During the mechanical instability tests, wing motions at the nacelle were continuously monitored on a direct writing recorder. Main rotor-propeller blade stresses were also monitored. No evidence of mechanical instability was exhibited by the aircraft during any phase of this test program.

3. <u>Ten Hour Tiedown Test</u> - A 10 hour tiedown test was conducted on the aircraft prior to flight test (Figure 47). The 10 hour tiedown and ground instability tests were completed within a period of two weeks -- the last six hours of the tiedown in two days. The tiedown test consisted of operation of the aircraft at scheduled powers with the wing in the vertical, intermediate, and full-down (airplane) positions. The power schedule called for 30 minutes at hover power, 5 minutes at the takeoff power, and 25 minutes at cruise power for every hour of test. During these tests, the wing was cycled at various

tilt rates and the controls held in a prescribed position to record power and transmission temperature.

Only minor discrepancies developed during the 10 hour tiedown. Parts of the trailing edge of several power turbine blades broke loose. This failure was attributed to hot engine starts, which were corrected by use of a higher capacity external power unit for starting. Fluctuation of fuel pressure developed during the endurance test. This was due to clogging of the fuel-system manifold harness by corrosion of the fuel-level indicator. The tank was cleaned with hot water, a new fuel indicator was installed and the nozzle was cleaned. The yaw fan bearing housing overheated due to high friction between a U-cup seal and a bearing shaft. The U-cup seal was replaced with a labryinth seal. Since then this assembly has been operating without trouble.

The tiedown was followed by a complete disassembly and inspection of the entire dynamic system. The flight control system, landing gear installation, and all critical welded joints received a thorough inspection. This inspection revealed no deficiencies.

4. Tail Fan Tests - Preliminary vibration characteristics and thrust and power requirements of the horizontal tail-fan assembly were recorded under both hover and simulated forward flight condition at various control settings. This was accomplished by mounting the tail assembly beneath a rotor blade whirl tower (Figure 27) and by using the downwash of a helicopter rotor system to simulate forward flight. Stick forces were also recorded, but under hover conditions only.

There did not appear to be any strong vibrations while the tail fan assembly was operated under static or simulated flight conditions. However stick forces, as measured, were considered too high for manual operation. The addition of counterweights to the fan blades reduced the operating forces. A hydraulic boost system was later incorporated during the flight program.

Tail-fan thrust and power measurements indicated a deficiency in thrust compared to theoretical calculations. These calculations were based on a free rotor,

plus an estimated 15 percent increase in thrust due to the shroud. Approximations were made for interference and inflow effects, but these approximations later proved to be too optimistic.

Tuft studies indicated that the hub region of the fan had the same flow pattern whether the controls were in positive-pitch or negative-pitch positions. This was attributed to the centrifugal pumping action of the exposed control spider and the rectangular box structure supporting the fan bearing mounted on top of the vertical fin. The control spider was covered with a shallow spinner and the rectangular box was faired. Static test results of the faired assembly indicated an improvement of thrust in both directions.

c. Taxi and Flight Tests

1. Taxi Tests - Taxi tests were conducted to evaluate ground handling characteristics. These tests disclosed that a larger rudder area, a steerable tail wheel, and replacement of the original spring strut tail gear by an oleostrut assembly were required.

With the incorporation of the above modifications, the aircraft was capable of performing turns from standstill and maintaining directional heading during takeoff and landing. The new tail gear strut also eliminated a pitching motion of the fuselage during vertical takeoffs and landings, and during high speed taxi tests (Figure 48)

Taxi tests up to takeoff speeds were accomplished with satisfactory results. The aircraft could be taxied at reasonable speeds with the wing tilted as high as 60 degrees from the horizontal. Normally, the wing angle was held below 45 degrees, since dirt could be blown into the cockpit at higher wing angles.

Taxi tests also disclosed that the wing, when tilted at a high angle, made a very effective air brake for deceleration. Thus a procedure for shortening the ground run after an airplane landing was developed. This consisted of reducing the propeller pitch to the minimum pitch setting of 3 degrees when the wing was down, and then tilting the wing up. Using this procedure, full stops from 90 knot runs were accomplished without the use of wheel brakes. The main wheel brakes would have worn excessively because of their small size. However, the incorporation of a steerable tail wheel and the use of the wing as a brake during roll-out alleviated this problem.

2. Hover Tests - The initial hover flight attempt showed certain control deficiencies. It was desired to hover a few feet off the ground during the initial flight. However, owing to a sensitive collective pitch system, the aircraft rose rapidly to an altitude of approximately 10 feet. Difficulty in controlling the aircraft about the pitch axis was encountered. This was due to the low sensitivity of the longitudinal control system near the neutral position. The pilot immediately landed the aircraft.

The collective pitch sensitivity was reduced approximately 40 percent. In addition, the longitudinal control system was modified to provide for a more sensitive stick gradient near neutral and an overall increase in control. The final longitudinal control provided a maximum pitching acceleration of approximately .6 radian per second per second in hovering. The directional control was also modified in a manner similar to the pitch control.

Conversion tests later disclosed that more directional control was required at high wing angles. Lateral control provided a maximum rolling acceleration of approximately 2.5 radians per second per second in hovering and was considered satisfactory by the Vertol Division pilot. However, NASA pilots preferred less sensitive roll controls; consequently, during 1961, modifications to response in roll were reduced by 40 percent. From the viewpoint of a research vehicle, the systems modified in 1957 have provided adequate controllability in all regimes of flight.

The increase in total pitch-fan and yaw-fan control imposed additional power requirements on the tail-fan drive shafts and gears. This required an increase in strength of these components. In addition to the high steady torques caused by increased power requirements, fluctuating torques of up to $\pm$ 100 percent of the steady values existed in the tail-fan drive shafts and led to tail shaft failures. These large cyclic torques resulted from rapid control displacements and the corresponding change in power required. Increased rate of control displacement created a corresponding increase in the amount of cyclic torque which caused the fan shaft to fail.

Analog studies were conducted to evaluate possible remedies. Among those investigated were alteration of the natural frequency by incorporating stiffer shafts, damping the system by use of an inertial damper, and incorporation of a fluid coupling in the drive system. Stiffening the drive shafting up to 10 times the original stiffness had little effect on the magnitude of cyclic torques. Inertia damping showed an improvement, but the weight of the dampers was prohibitive. Finally, the fluid drive coupling was chosen as the best solution. The analog studies showed that the amount of dynamic overshoot was reduced to less than 5 percent with this installation Subsequent flight test data confirmed this result.

3. Airplane Flights - Following taxi and hover tests, airplane flights were conducted. These flights consisted of takeoff, straight and level flight over the runway at altitudes of approximately 10 feet, and landings. Airplane control and stability characteristics were pilot evaluated during these tests. Results of the airplane evaluation were satisfactory and no modifications to the aircraft were required. Takeoffs have been accomplished in approximately 400 to 500 feet. The deceleration procedure developed during taxi tests was successfully employed.

d. Fifty Hour Tiedown Test - Prior to proceeding with conversion tests, an additional 50 hours of tiedown testing was conducted to fully qualify the dynamic system and all modifications which were incorporated into the aircraft during its previous flight program. This program was changed from the original 10 hour tiedown; i.e., it was made more severe. The control system was cycled continuously throughout 50 hours to simulate flight operations conservatively. The only discrepancies that occurred were a leaky fuel tank and failure of a control quadrant shaft. These parts were replaced with strengthened units which operated satisfactorily.

e. Conversion Tests - Conversion tests were initiated on 6 June 1958. At this time, in-flight wing tilts down to 74 degrees from hover were accomplished. Further tests disclosed marginal yaw control at approximately 75 degrees to 65 degrees wing incidence in addition to an unsymmetrical yaw response. Efforts to increase the yaw effectiveness were described earlier in this report. Nevertheless, owing to the aerodynamic inefficiency of the yaw fan, it still attained only 60 percent of the thrust of the pitch fan in low speed flight.

A program was initiated to determine at what wing angle directional controllability improved. This consisted of a series of STOL takeoffs, stabilized flights, and landings with the wing fixed at various wing angles from 65 degrees down to 40 degrees in 5 degrees increments. It was learned that directional stability was adequate from 60 degrees down to the airplane configuration. These STOL tests isolated the marginal yaw-control condition to

approximately a 10 degree range from the hover position.

To improve the handling characteristics prior to further conversion tests, stability augmentation was provided in the form of pitch and roll rate damping. These proved to ease the control burdens on the pilot.

Conversion tests were resumed for the prime purpose of determining whether the pilot would experience any difficulty in flying through this marginal yaw condition. Results disclosed a slight yawing of the aircraft in the 75 degree to 65 degree wing incidence range. However, as the wing approached the 60 degree position, directional control became more effective and below that the pilot experienced no difficulty.

On 15 July 1958 the first complete conversion was attempted. The original intent was to convert from hover to the airplane configuration, and then to land. However, since no buffeting nor vibration problems were encountered, and since the aircraft exhibited good control and stability characteristics when converting from hover to forward flight, the pilot tilted the wing up and reconverted to the helicopter configuration to complete the conversion cycle.

Since the initial conversion, a total of 15 complete conversions were accomplished by 10 October 1958. Altitude control was maintained without any effort, and stick trim positions were not excessive during the conversion cycle. A typical time history is shown in Figure 46. The aircraft has been hovered and flown in the airplane and STOL configurations in winds up to 17-20 knots without any automatic stabilization system. Subsequent conversions have been accomplished with the damper systems turned off with satisfactory results.

f. <u>Availability and Maintenance</u> - After qualification of the aircraft by a 50 hour tiedown endurance test, an aircraft availability rating as high as 91 percent was achieved with days airborne per month as high as 81 percent. The average availability rate since tiedown was 70 percent (Figure 49). This average falls within the availability

range of military fixed wing aircraft and well above military rotary wing aircraft. No special maintenance attention was received by this research aircraft during its test phase; i.e. other than what standard operational aircraft normally require. However, the above statistics are presented only as a "feel" for the availability rate. They seem to indicate that the tilt wing configuration even with its capabilities as a rotary wing and as a conventional aircraft, need not possess compound maintenance problems; but, on the contrary, ranks with operational aircraft in this area.

B. EXTENDED FLIGHT TEST PROGRAM

1. TEST PROGRAMS BY VERTOL DIVISION AND NASA

After conversion was demonstrated under the basic flight test program, extended flight test work was started on the VZ-2. This program consisted primarily of a strain survey conducted by Vertol Division to confirm the structural integrity of the aircraft and of tests to evaluate the handling characteristics of the VZ-2. The extended flight test program covers the period up to modification of the aircraft with full span flaps and ailerons.

a. Strain Survey - As part of this program, parallel static and fatigue test programs were conducted to augment the flight strain data. Figures 50 and 51 show the main rotor blade and tail-fan blade retention systems undergoing test. Recorded strain data indicated no unusual or dangerous stress levels.

Before delivery to NASA, Langley Field, Vertol Division conducted high altitude flights at Edwards Air Force Base to extend the flight envelope in the airplane regime of flight, prior to evaluation by NASA pilots.

b. Short Takeoff - In order to study the short takeoff capabilities of the VZ-2 research aircraft, a number of short takeoffs were performed while the aircraft flight path was recorded using a multi-exposure camera. For each preselected wing incidence, runs were performed at a power setting lower than was required to hover.

An example of an actual short takeoff at a power setting lower than required to hover is shown in Figure 52.

Initially, an attempt was made to set the required take-off power prior to the ground roll. However, a pronounced nose-over tendency was encountered due to the rotor thrust vector being above the aircraft center of gravity. Attempts to expedite the takeoff, using large power increases prior to attaining the level flight trim speed for a particular wing angle, also produced an appreciable nose-down moment. Nevertheless, when the trim speed had been reached, abrupt power changes merely influenced the climb angle without perceptible stick trim changes. A tricycle landing gear configuration would alleviate this nose-over tendency while the aircraft was on the ground roll.

Figure 53, obtained from flight test, shows the takeoff distance versus ratio of average rotor horsepower at take-off, to rotor horsepower required to hover.

Calculations, based on several model tests, indicate that the tilt wing, when allowed to make a short run along the ground before taking off, has an inherent overload capability. The wind tunnel models utilized both leading and trailing edge high lift devices on the wing in the takeoff configuration. As shown in Figure 54, an aircraft with a hover gross weight installed power ratio corresponding to the VZ-2 has an overload capability of 25 percent when it takes off from a 500 foot field over a 50 foot obstacle. The takeoff distance required increases as the hover gross weight-installed power ratio decreases. This is due to the fact that, as the weight-power ratio decreases, the propeller disc loading and wing loading increase; consequently, the required takeoff speed is higher.

c. <u>Evaluation of Flying Qualities</u> - The Flight Research Program by NASA consisted of hover tests, STOL tests, partial power descents at various wing positions (Reference 22), and evaluation of differential ailerons (Reference 21) for yaw in hover, as well as investigation of droop snoot and wing fences.

The wing fences shown in Figure 55 proved to be of little value in improving the descent capability and handling characteristics of the VZ-2. The droop snoot, the geometry of which is also defined in Figure 55, greatly improved the transition of the aircraft. The allowable rate of descent increased over 1000 feet per minute at the transition speed where the aircraft previously experienced stall even at a rate of climb of 500 feet per minute. As reported in

Reference 24,, the droop snoot so greatly improved the characteristics of the aircraft that serious stall limitations in the descent and level flight decelerations were essentially eliminated from the range of practical flight operation. With the droop snoot, the aircraft became, by comparison with the original configuration, a pleasure to fly.

These initial NASA flight tests were further supplemented by evaluation flights performed by the representatives of NASA, the Air Force, the Naval Air Test Center and Vertol Division. These included the following: On 13 August 1957, the VZ-2 was flown by Leonard J. Lavassar of Vertol Division; on 14 March 1959, by John P. Reeder of NASA; on 2 December 1959, by James B. Whitten of NASA; on 22 June 1960, by Fred J. Drinkwater, 3rd., of NASA, and by Major Donald R. Segner of the U. S. Marine Corps; on 6 July 1960, by Robert A. Champine of NASA; on 11 July 1960, by Major Walter J. Hodgson of the U. S. Air Force; on 19 August 1960, by Joseph J. Algranti of NASA, and by Donald L. Mallick of NASA; on 23 August 1960, by Harold E. Ream of NASA; and on 1 September 1960 by Robert W. Sommer of NASA. In all, the aircraft was flown by eleven pilots.

To evaluate the possibility of control simplification through elimination of the yaw fan, the aircraft was modified by providing yaw control in hovering through the original ailerons (outboard of the nacelles only), and by permitting the yaw fan to be phased out by the pilot. Flight tests in hovering and partial conversion by a NASA test pilot at Langley Field indicated the feasibility of this concept.

d. <u>NASA Wind Tunnel Test on Full-Scale Propellers</u> - Between 1960 and 1963 three wind tunnel tests were conducted at the NASA-Ames 40 foot by 80 foot wind tunnel (Figure 56) to obtain force and moment data for performance, and analyses. These tests included the following:

1. An investigation of propeller performance, including in-plane forces and out-of-plane moments for three propellers operating through a range of thrust-axis angles of attack from 0 degrees (horizontal) to 85 degrees (Reference 39). The operating conditions were selected to simulate those anticipated for VTOL/STOL aircraft in the takeoff, landing and transi-

tion regimes. The propellers differed widely in planform: one was a conventional propeller; one had an exceptionally high solidity; and one was a VZ-2 propeller.

The results of the investigations revealed that, for all three propellers, similar variations in the forces and moments with thrust-axis angle of attack and advance ratio were present. Further, thrust and power were nearly constant, and in-plane forces and out-of-plane moments increased approximately linearly over large ranges of thrust-axis angle of attack for constant blade angles and axial inflow ratios.

2. A second test, which included an investigation of both rigid and flapping propellers of the VZ-2 type, was made to verify and expand the results obtained from the test described above (Reference 40).

   In addition, a comparative investigation of the use of cyclic control on the VZ-2 rigid and flapping propeller was made to determine the feasibility of the system for aircraft pitch control. The results of this investigation indicated that the rigid propeller is 2 to 2.5 times as effective per degree of cyclic as the flapping propeller in producing a control moment on the aircraft. The pitching moment due to cyclic appears to be insensitive to speed and angle of attack change, and there is no appreciable change on the propeller control moment when tested in the presence of the aircraft wing.

3. Tests of a rigid and a flapping propeller were made at various rates of descent from 0 to 6000 feet per minute to determine the effects of operation in the vortex-ring state upon the thrust force produced by the propellers (Reference 38). The test covered a disc loading range from 0 to 36 pounds per square foot. The results of these tests indicated that thrust oscillations developed and increased in magnitude with the rate of descent, as the propeller moves into the vortex-ring state. However, the oscillations generally diminished as disc loading was increased. For example, at a disc loading of 30 pounds per square foot the thrust oscillations started at approximately 1000 feet per minute rate of descent and increased to about 25% of the steady thrust value at 5000 to

6000 feet per minute rate of descent. With further increases in the rate of descent, the oscillations decreased as the propeller moved toward the windmill-brake state.

e. Full Scale Aircraft Wind Tunnel Test by NASA - Wind tunnel tests of the full scale aircraft (Figure 57) were initiated by NASA at Langley Field, upon completion of their flight research program. These tests were conducted to make a comparison between the plain wing and the wing equipped with either the droop snoot or leading edge slats. An attempt was also made to correlate pilots' opinions on flying qualities in forward flight with wind tunnel measurements. In 1963, a report will be published by NASA covering the full scale wind tunnel test of the VZ-2 VTOL airplane and its correlation with flying qualities obtained from NASA flight tests.

On the basis of the full scale aircraft wind tunnel test and as a result of the extended flight test program, it was decided to modify the original VZ-2 with full span flaps and ailerons to increase its overall performance.

C. FINAL DEVELOPMENT PROGRAM

1. MODIFICATIONS TO BASIC CONFIGURATION

The major modifications made to the VZ-2 Research Aircraft consisted of changes dealing mainly with the development of an ejection seat, improvement to the control system, the addition of flaps, upgrading of the transmission system, addition of a new type of landing gear, and physical modification of the rotor-propeller.

a. Control System and Flaps - The following major modifications were made to the control system during the final development program:

1. Pitch control of the horizontal fan and authority of the elevator were partially reduced with wing-tilt (Figure 58). In this way, oversensitivity to pitch control in high speed forward flight was eliminated.

2. A special pilot-operated mechanism was provided to select the amount of yaw fan authority (Figure 59). This permitted a gradual phasing out of yaw fan inputs, during evaluation of the effectiveness of the differential ailerons, as a means of yaw control in hovering and transition.

3. The differential collective pitch control system was revised to reduce the lateral control sensitivity in hover and low speed conversion (Figure 60).

4. Full span flaps as described in Figure 17 were added to improve the stall characteristics of the wing in various regimes of flight. Consequently, the wing chord was increased from 4.75 feet to 5.25 feet. Flap position is scheduled with wing-tilt. It remains in neutral position when the wing is at the full-up or full-down position. For intermediate wing tilts, it assumes positions as shown in Figure 61. However, instead of automatic flap position with wing-tilt, the pilot can select any position between 0 degree and 30 degrees at any wing position through a special trim selector. The flap consists of two segments: one being the proper flap, and the other formed by the full span ailerons which deflect down as the flap is extended and deflected. In the up-wing position, the ailerons are actuated by the pedals to provide yaw control. As the wing goes down, the pedal action is gradually phased out, and response to the lateral stick displacement is gradually phased in (Figure 44). In addition, the original tail pipe was modified and a dorsal tail fin was added to improve directional stability (Figure 9).

b. Transmission System - Addition of the full span ailerons and flaps to the VZ-2 increased the weight of the basic configuration from approximately 3400 pounds to 3700 pounds. In addition, the main transmission gears were shot peened and the overall transmission was upgraded

from 630 to 700 horsepower. Because of this modification the transmission system received an additional 50 hour tiedown test to confirm its structural integrity.

c. Landing Gear - The original VZ-2 configuration, prior to the addition of full span ailerons and flaps, used a Cessna type tail landing gear. Later, a Vertol designed gear was installed to improve taxiing of the aircraft. Additional stress analysis showed that the new gear was capable of meeting the design requirements of the VZ-2 research aircraft at higher gross weights.

d. Propeller - To improve hover gross weight, power available was increased from 630 to 700 horsepower and the propeller blade diameter was increased two inches to maintain the same disc loading.

2. FINAL FLIGHT TEST PROGRAM

Upon completion of these modifications, the VZ-2 was evaluated in a two hour flight test at Vertol Division prior to delivery to NASA at Langley Field, Virginia. Flight acceptance tests were initiated in the latter part of 1962 and were completed early in 1963. Since then, the aircraft has been undergoing extensive flight tests to evaluate the use of differential ailerons for low speed control and to evaluate flaps for improvement of transition performance. The most recent VZ-2 flight test data is analyzed in the following paragraphs. This data was obtained from NASA, at Langley Field and is, as yet, unpublished.

Hover control data for pitch, roll and yaw is presented in Figures 62, 63 and 64 respectively. In these figures hover control power and damping characteristics of various VZ-2 configurations are compared with the requirements of MIL Specification 8501A and AGARD Report 408. These comparisons show that roll control and damping meet Military Specifications, but that yaw and pitch characteristics are below the required levels. These findings are confirmed by pilots' comments.

A rate of descent envelope has been defined, based on flights with the flap deflection programmed with wing angle, as shown in Figure 61. As Figures 65 and 66 illustrate, the rate of descent capability of the VZ-2 aircraft has been improved con-

siderably, compared to the basic configuration, by the use of flaps.

The boundary should be interpreted as extreme limits; operationally the rate of descent capability would be less. There is a major difference between the boundaries established for the plain wing and the "droop snoot" configurations and the programmed flap case. The former versions experienced an abrupt stall boundary, while the latter version exhibited a gradual onset of buffet and instability with increasing rate of descent. The limits established at speeds higher than 40 knots were due to buffet, some loss of lateral control, wing drop, and other stall phenomena. Below 40 knots the limits are due to the violent yaw motions caused by the directional static instability of the VZ-2. As Figure 67 illustrates, the aircraft is directionally unstable over a considerable yaw angle range. This is attributed to the poor flow at the vertical tail caused by the wake of the wing center section, fuselage and engine. The yawing motions of the aircraft below 40 knots when descending are described by the pilots as intolerable, but these motions are not due to the flaps since the pilots experienced similar characteristics at moderate rates of descent before the aircraft was modified with flaps.

It should be noted that free flight tests on a 1/4 scale model of the VZ-2 in the NASA Langley 30 foot by 60 foot tunnel showed that a great improvement in flying qualities could be obtained if both leading edge droop and trailing edge flaps were used. Vertol Division Wind Tunnel tests, on aerodynamic configurations similar to the VZ-2, indicate that a considerably higher rate of descent capability can be obtained with leading and trailing edge devices used simultaneously than with either device used separately.

Some of the pilots' comments on flying the VZ-2 in ground effect are of interest. According to the pilots there is a tendency for the aircraft to "float" as it approaches the ground at wing angles of 30 degrees and lower. Between 40 degrees and 45 degrees the aircraft tends to settle in, and pilots have seen evidence of the wing-propeller wake striking the ground and running upstream ahead of the aircraft.

The ratio of power required for level flight to hover power required has been determined from flight test with and without

flaps. This data is presented in Figure 68 as a function of speed together with the optimum power ratio. This optimum power variation with speed was obtained assuming that the effective span was equal to the wing span in calculating the induced power. The parasite and profile power were also included. It is evident that flaps give a considerable reduction in power required, and in fact, below 30 knots give a power required which approaches the optimum figure.

Besides demonstrating the feasibility of the tilt-wing concept, the VZ-2 program resulted in some significant contributions to tilt-wing technology. Among these are the use of differential ailerons for directional control and automatic phasing of mechanical controls from hover to forward flight. It has laid the foundation for future tilt-wing progress -- the Tri-Service Transport is a case in point -- and it has supplied engineering personnel with valuable experience in the field.

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PHYSICAL DATA

Component	Measurement
Wing Span	24' - 11"
Length	26' - 5"
Height	10' - 0"
Horizontal Tail Span	9' - 11"
Wing Area	130.5 sq ft
Wing Airfoil Section	NASA 4415
Propeller Diameter	9' - 8"
Tail Fan Diameter	2' - 0"
Wheel Tread	7' - 0"
Weight Empty	
Standard Pilots Seat	3259 lb
Ejection Seat	3329 lb
Gross Weight	
Standard Pilots Seat	3653 lb
Ejection Seat	3723 lb

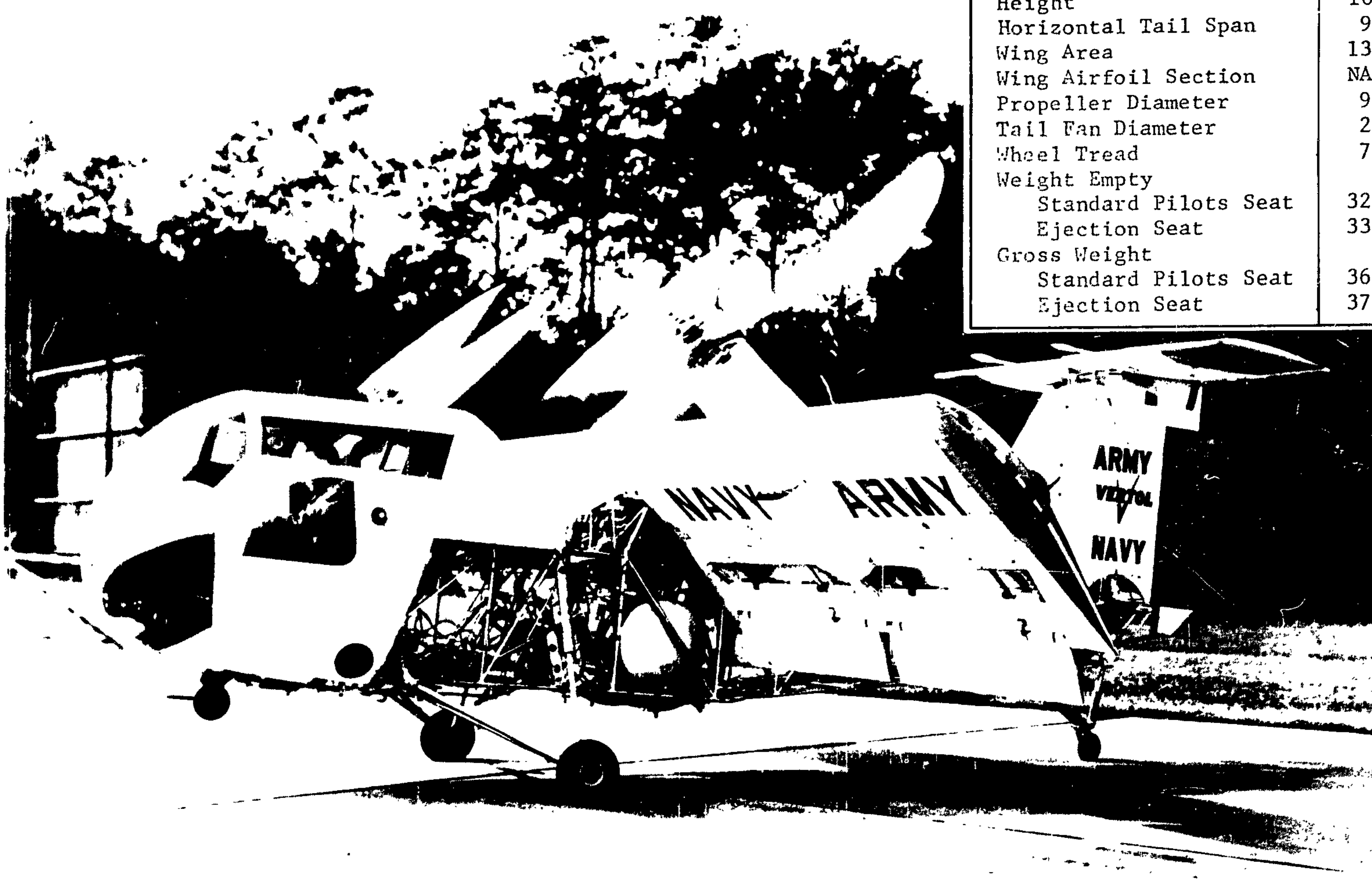


FIGURE 1. U.S. ARMY VZ-2 (BOEING-VERTOL 76) RESEARCH AIRCRAFT WITH FULL SPAN FLAPS AND AILERONS

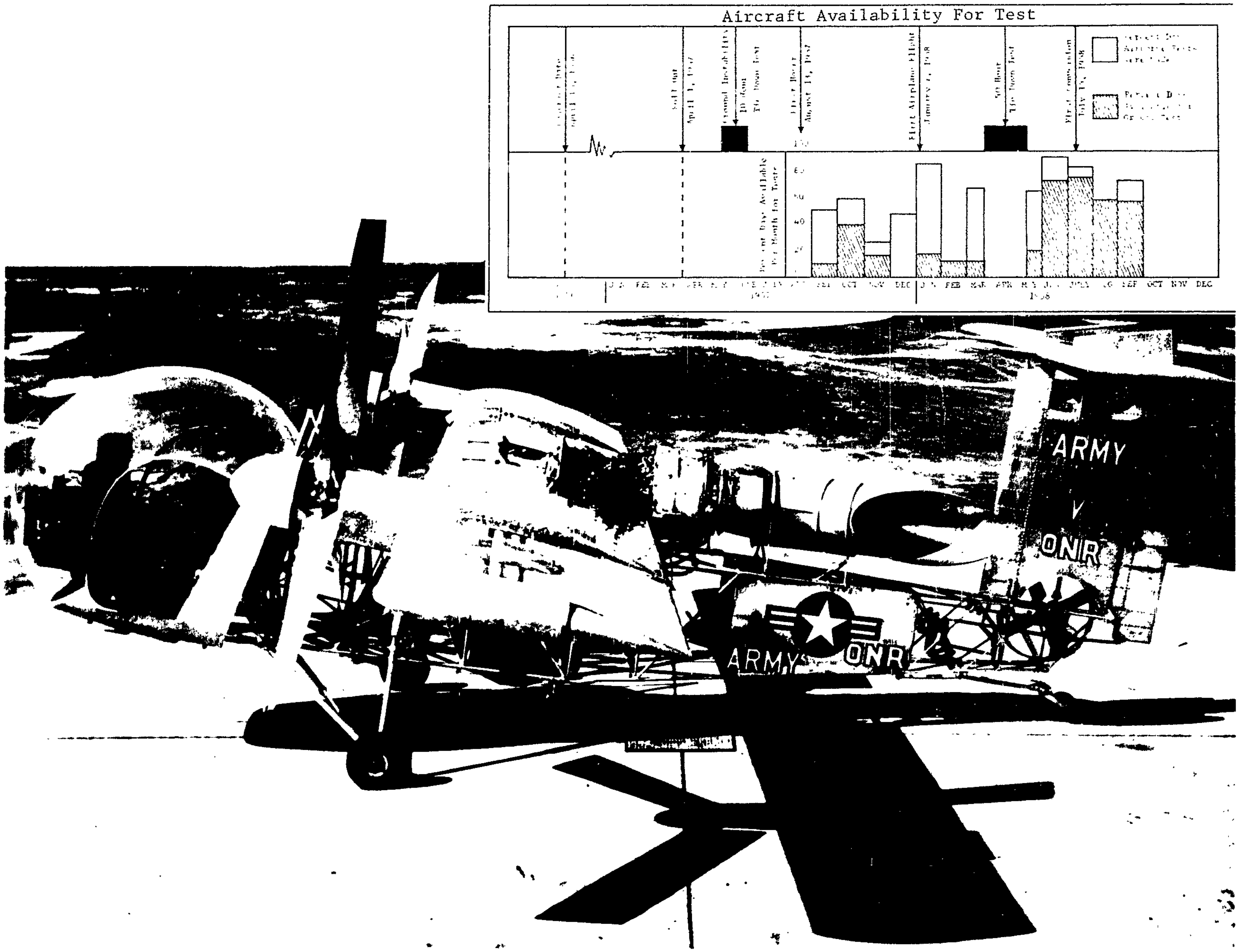


FIGURE 2. ORIGINAL VZ-2 CONFIGURATION AT ROLLCUT

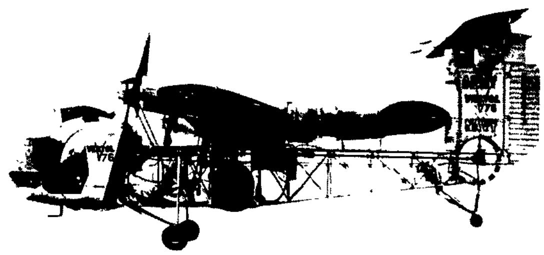


FIGURE 3 VZ-2 RESEARCH AIRCRAFT IN 100 PERCENT CONVERTED CONFIGURATION

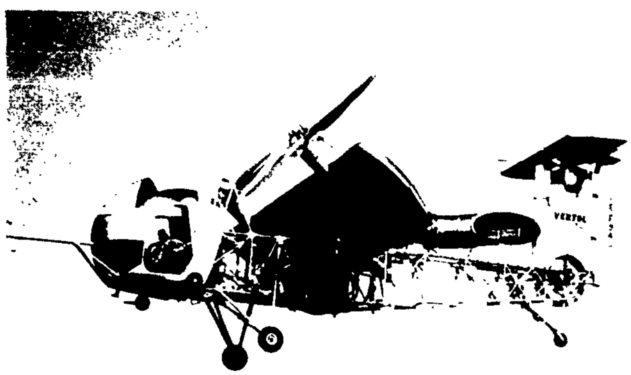


FIGURE 4 VZ-2 RESEARCH AIRCRAFT DURING CONVERSION

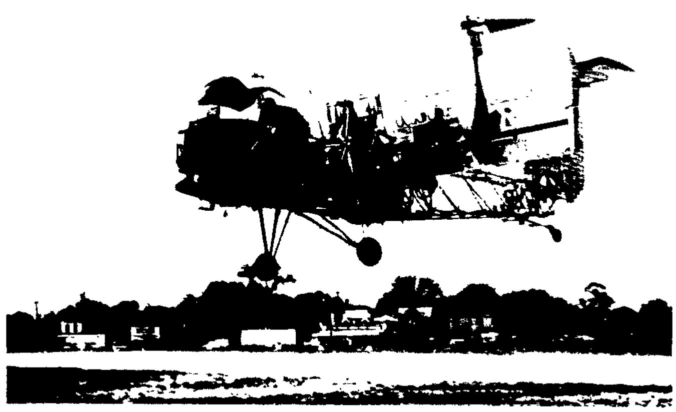


FIGURE 5 VZ-2 RESEARCH AIRCRAFT IN HOVER

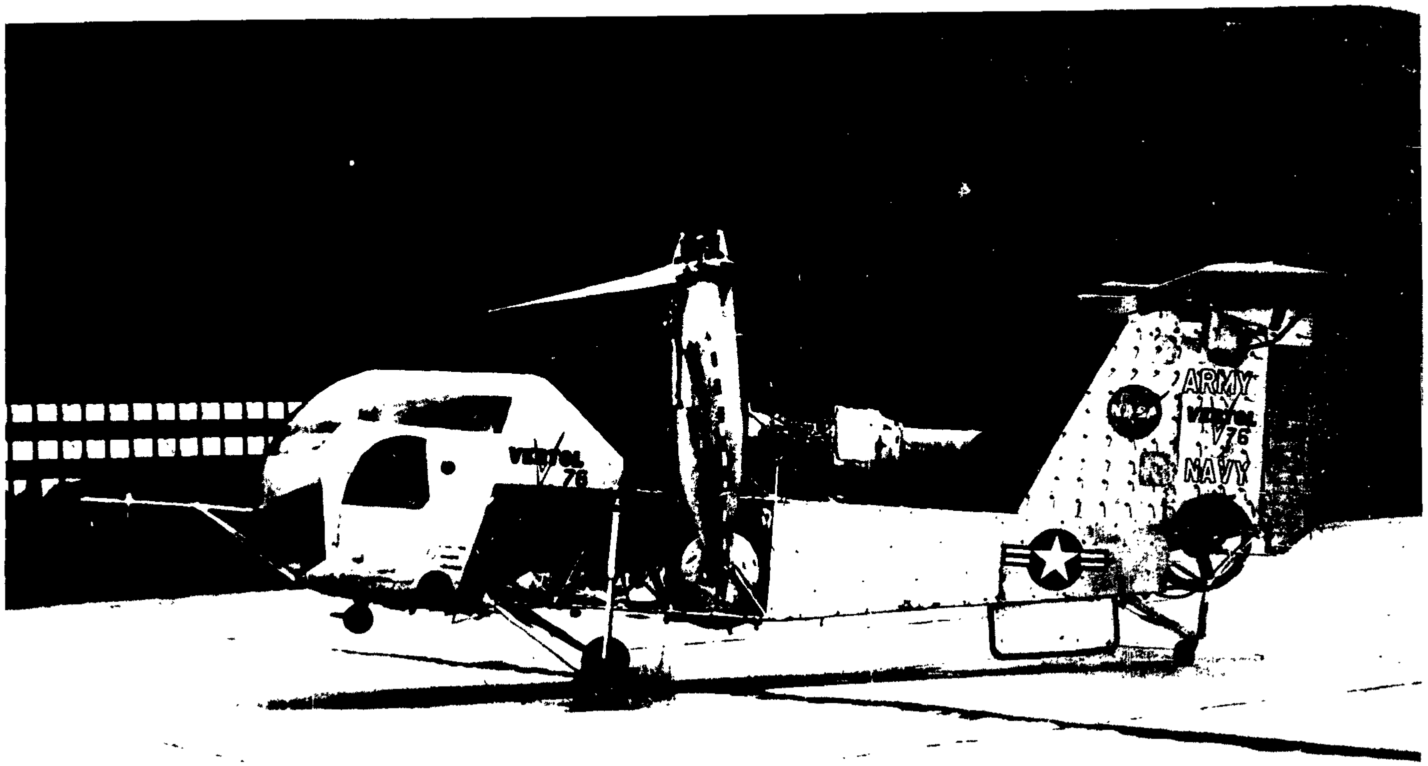
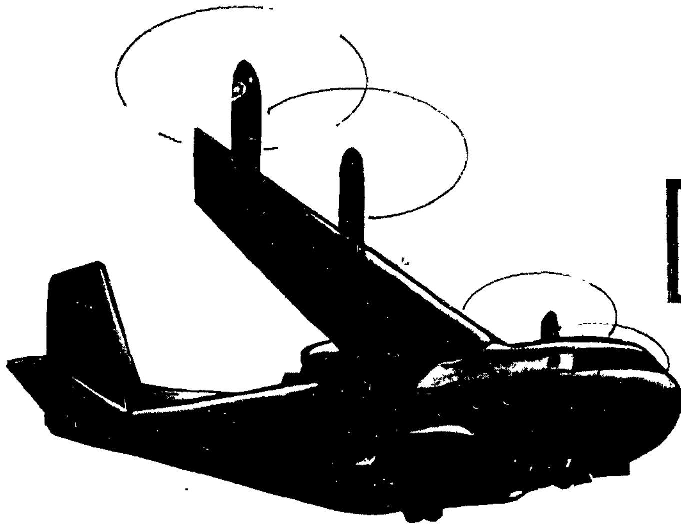
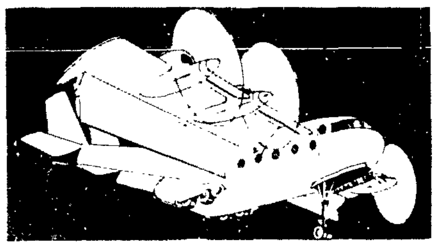


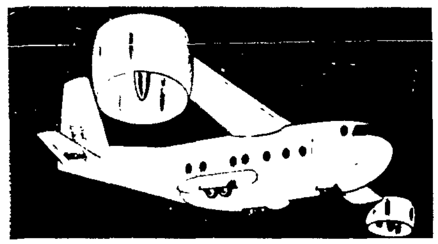
FIGURE 6. VZ-2 RESEARCH AIRCRAFT AT NASA, LANGLEY FIELD, PRIOR TO THE INSTALLATION
OF FULL SPAN FLAPS AND AILERONS



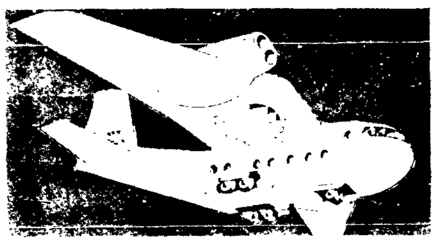
TILT WING
PROPELLER



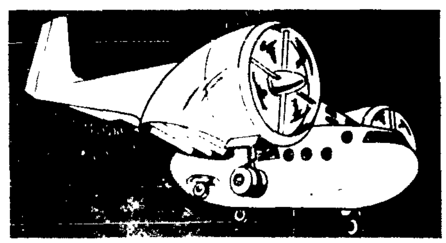
VECTORED
LIFT



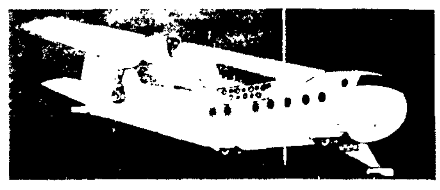
TILTING DUCTED
PROPELLER



VERTODYNE



AERODYNE



HOVER-JET

FIGURE 7. SIX VTOL CONFIGURATIONS

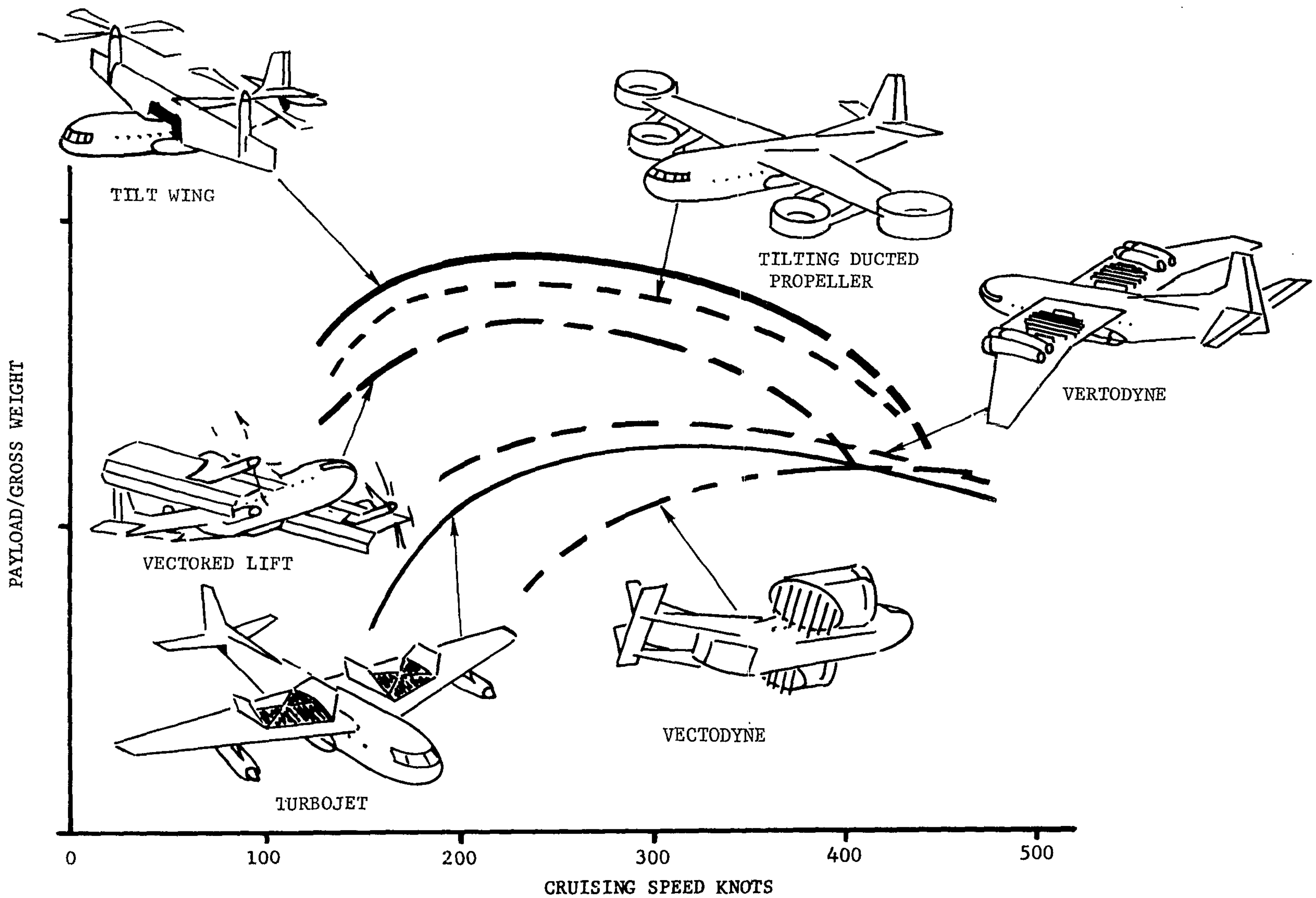


FIGURE 8. PAYLOAD/GROSS WEIGHT VS CRUISING SPEED FOR THE SIX MOST PROMISING VTOL TRANSPORT SYSTEMS

1

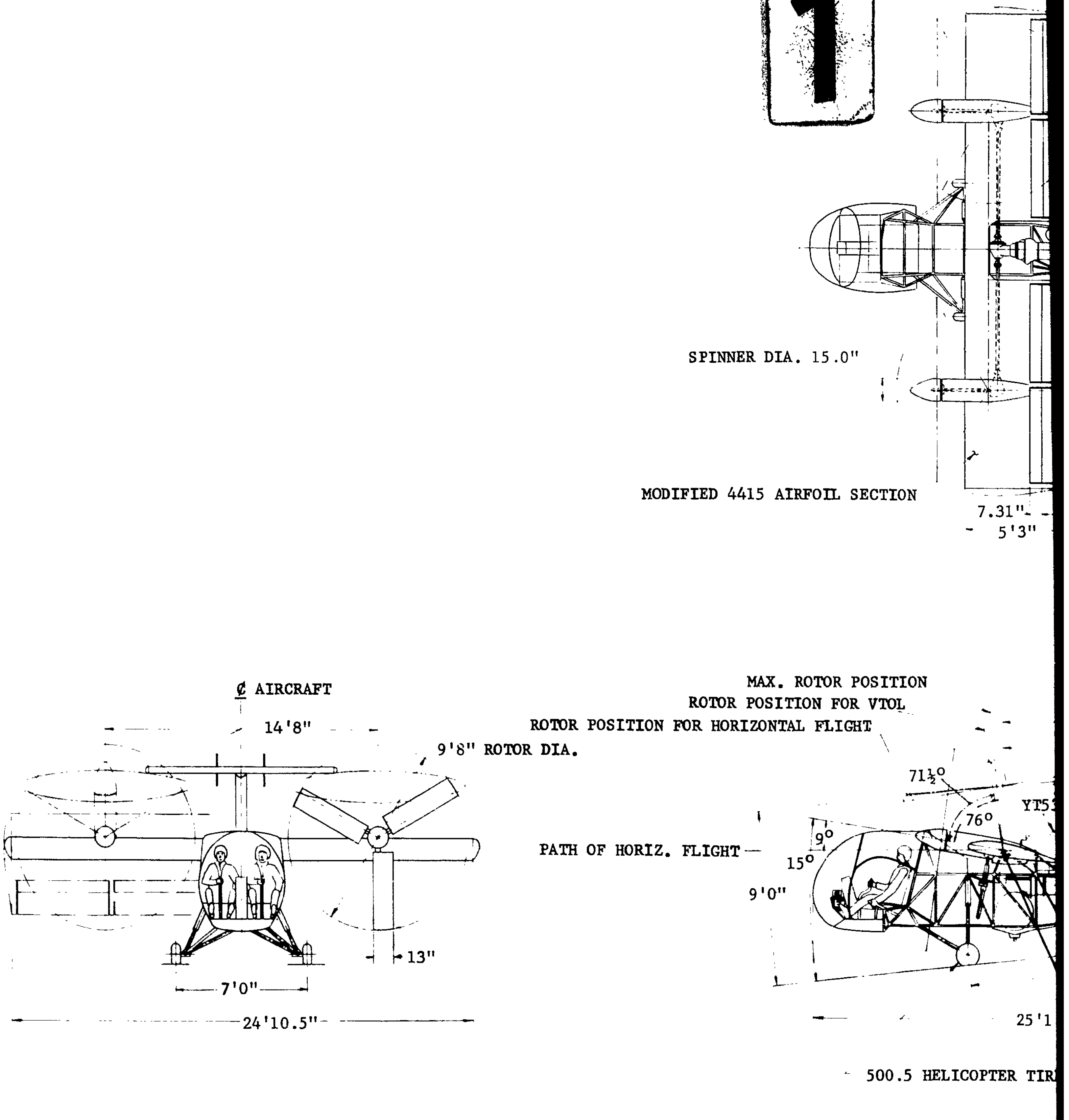


FIGURE 9. VZ-2 GENERAL ARRANGEMENT

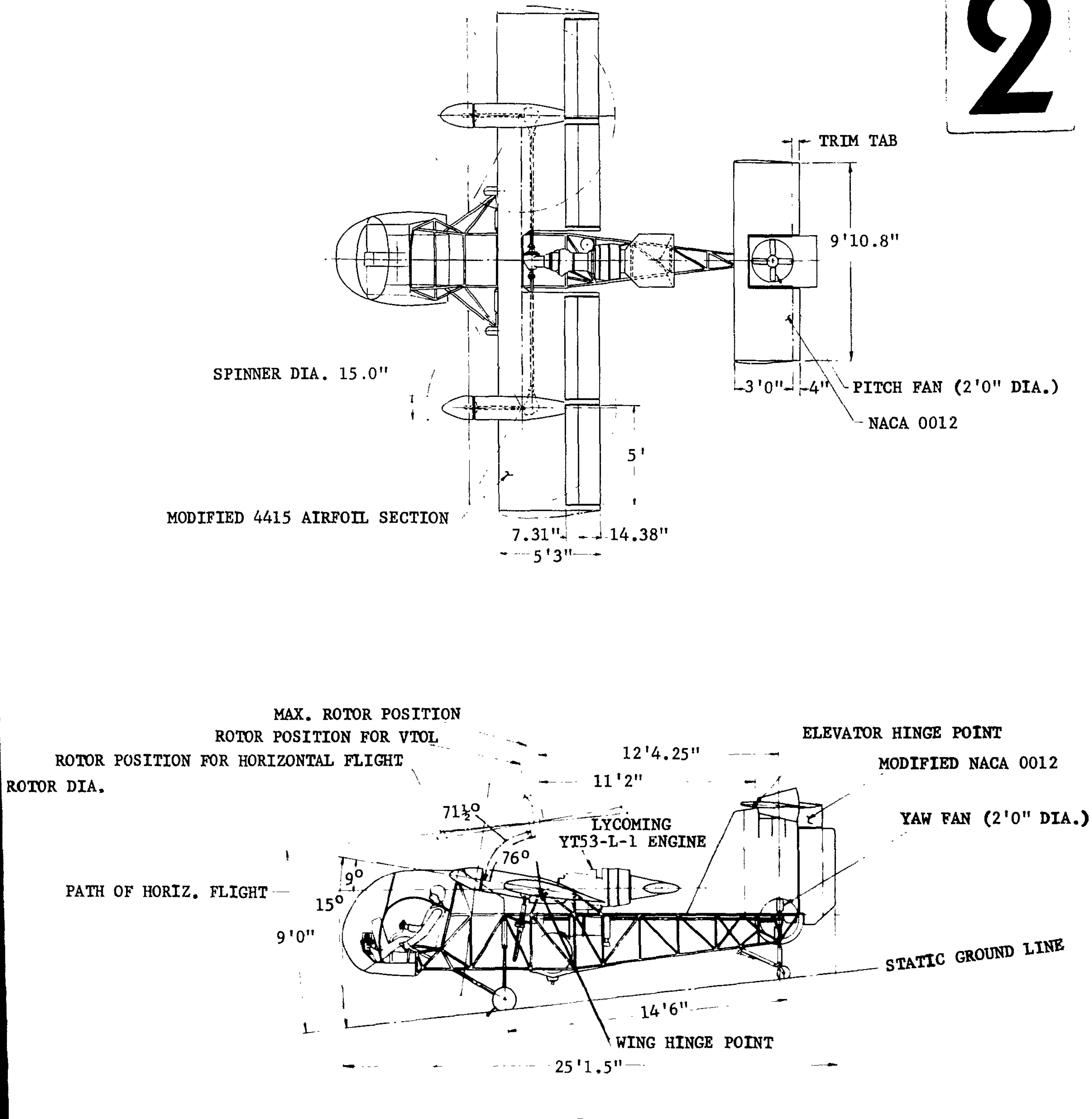


FIGURE 9. VZ-2 GENERAL ARRANGEMENT

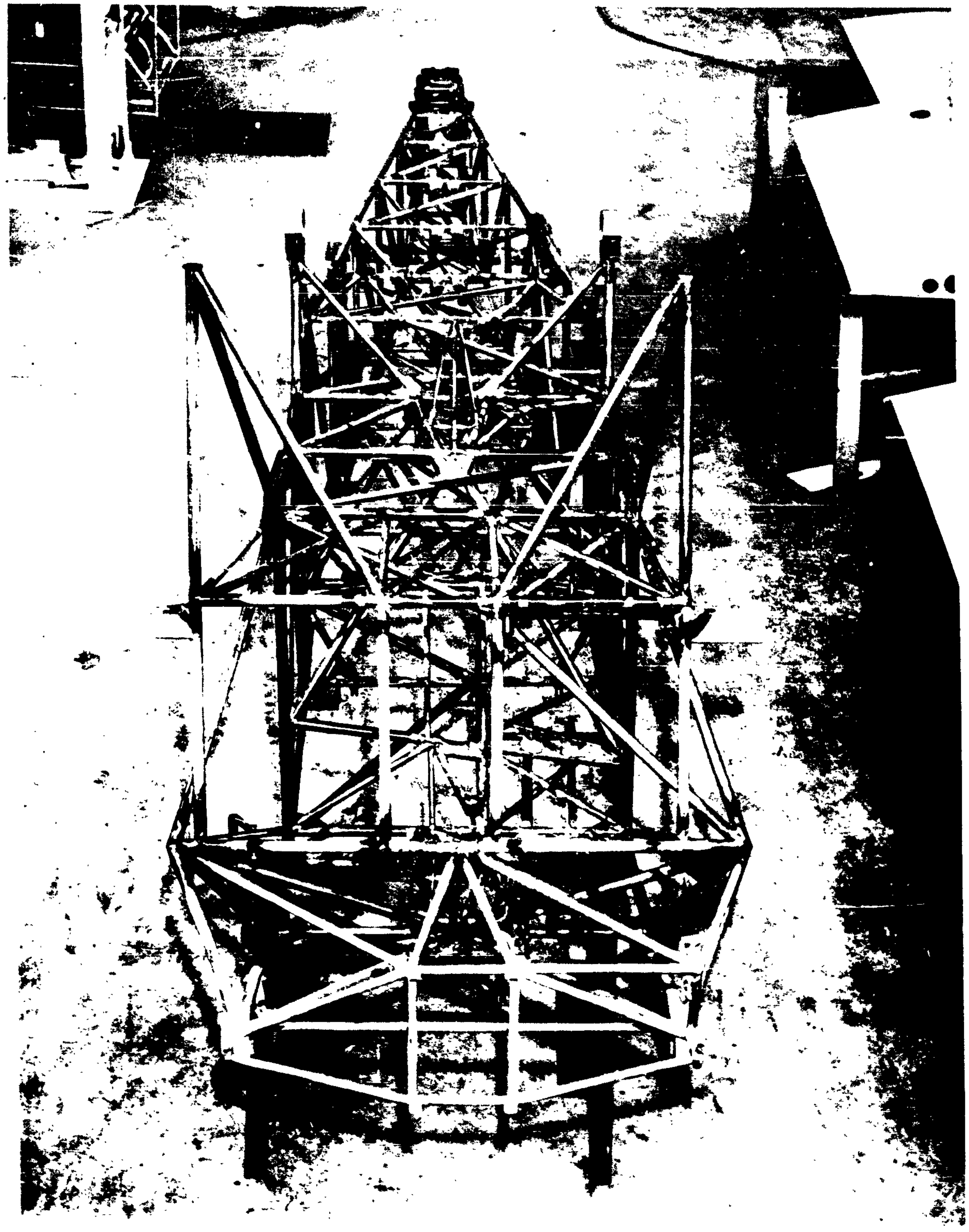


FIGURE [illegible]. BASIC FUSELAGE STRUCTURE

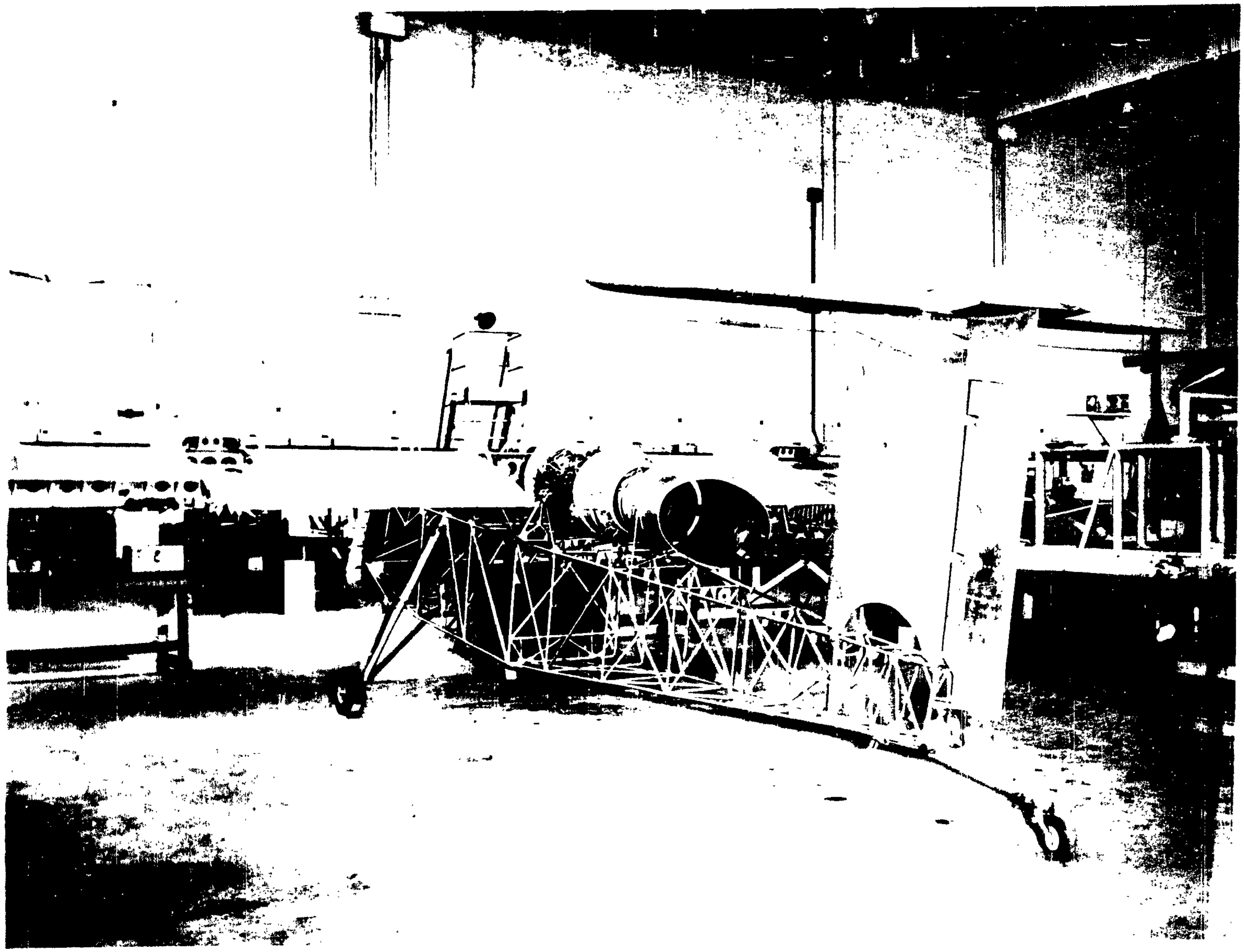


FIGURE 11. ASSEMBLY OF BASIC COMPONENTS - FUSELAGE, LANDING GEAR, TAIL AND WINGS

FIGURE 12 U.S. ARMY VZ-2 COCKPIT AREA

FIGURE 13. BASIC TAIL STRUCTURE WITHOUT FANS

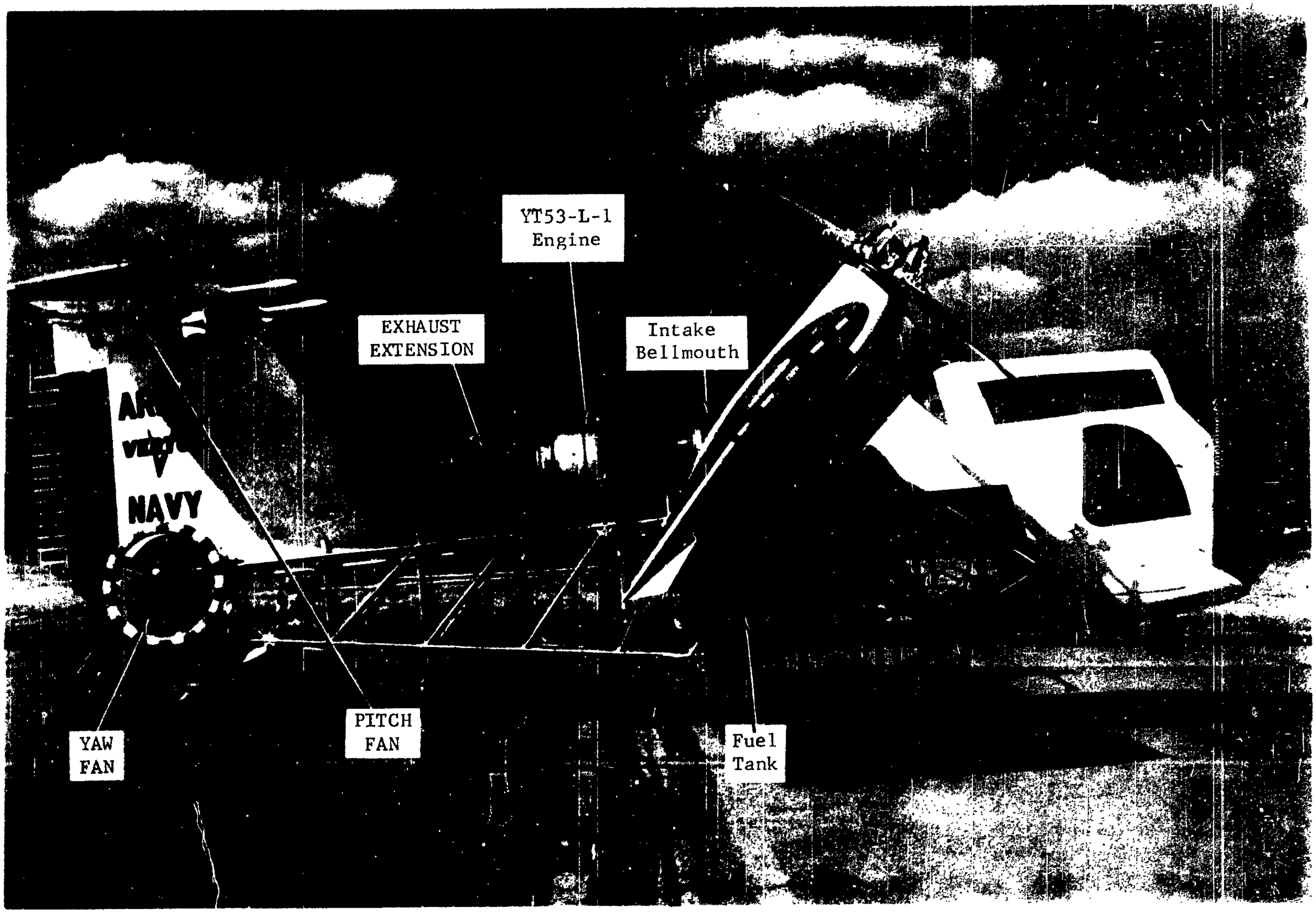


FIGURE 14. POWER PLANT INSTALLATION OF THE VZ-2 RESEARCH AIRCRAFT WITH THE MODIFIED TAIL PIPE

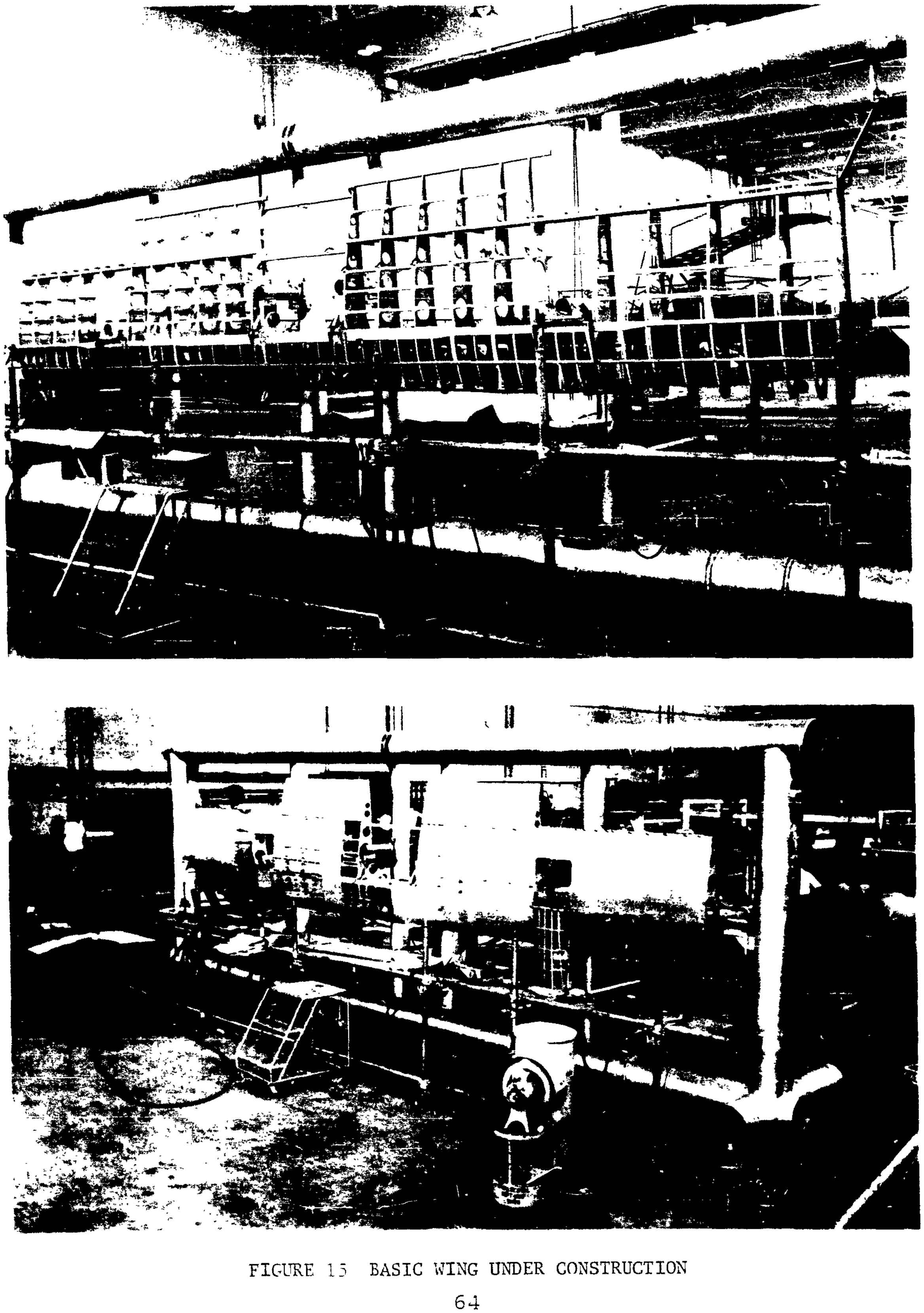
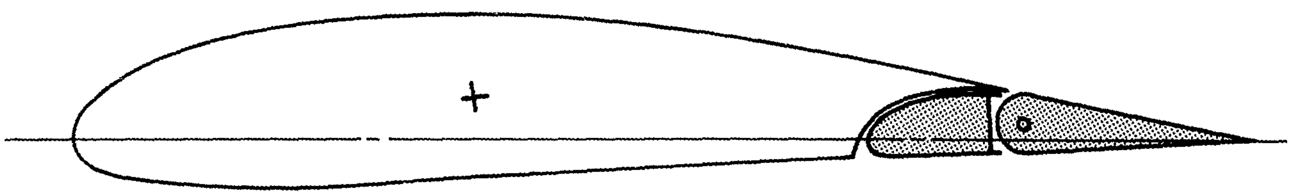


FIGURE 15 BASIC WING UNDER CONSTRUCTION



MODIFIED* NACA 4415

Flap Chord	33.7%	
Aileron Hinge Point	80%	
Aileron Deflection - down	20°	
- up	30°	
Fully deflected flap	30°	(extends wing chord 11.5%)

*Modification includes extending the chord 10.5%, and extending the upper and lower surfaces straight back to the trailing edge from the original 70% chord station.

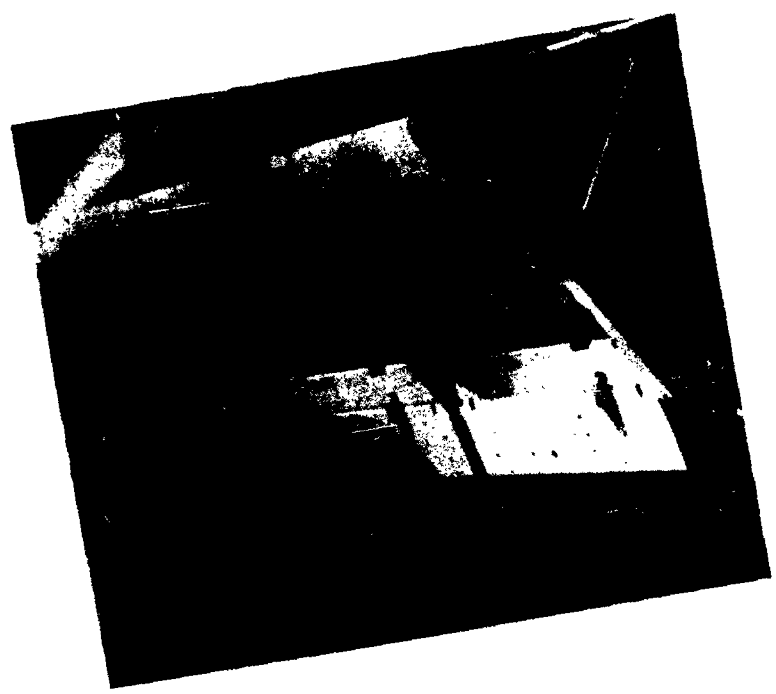


FIGURE 16.

VZ-2 FULL SPAN FLAPS AND AILERONS

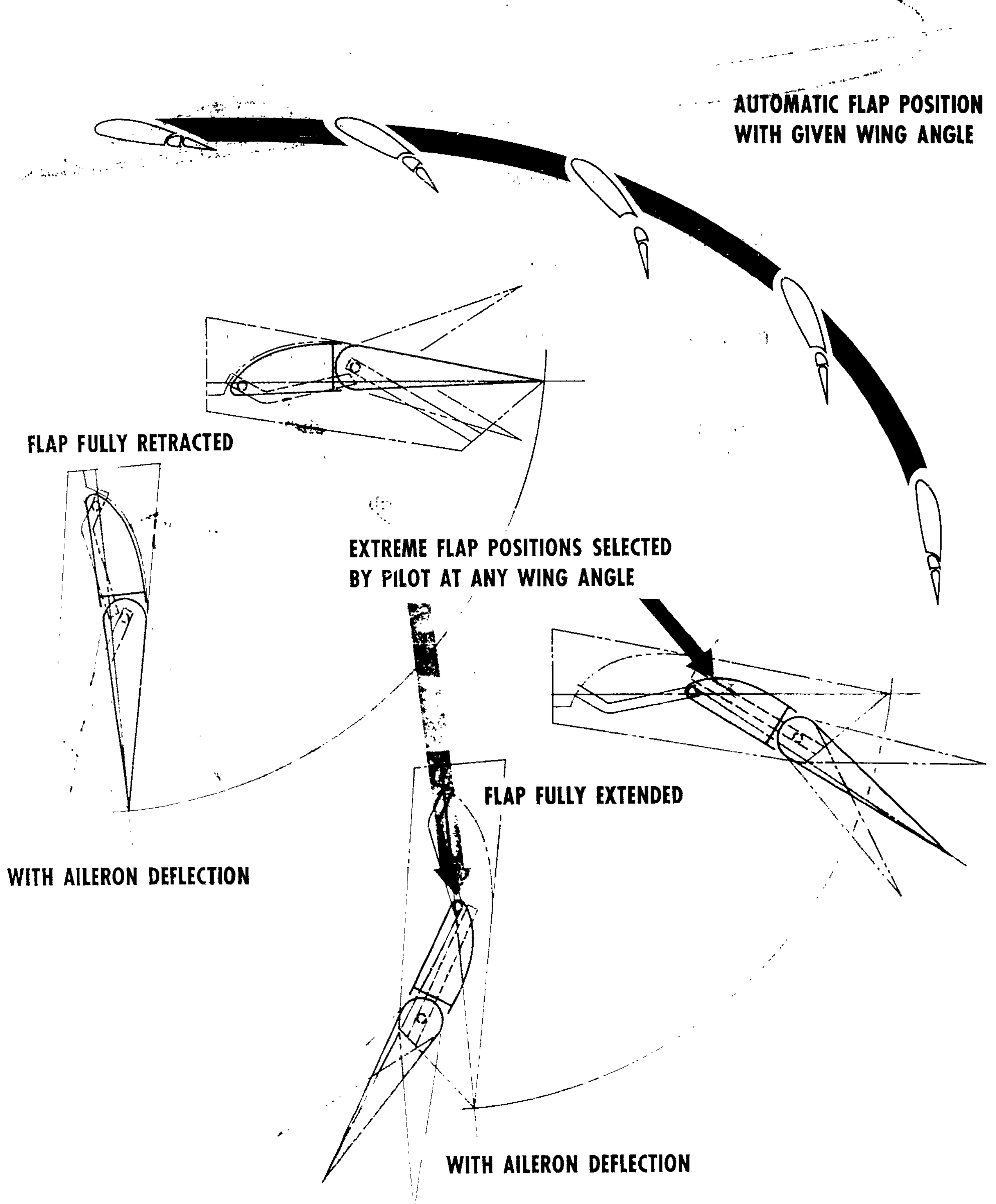


FIGURE 17.

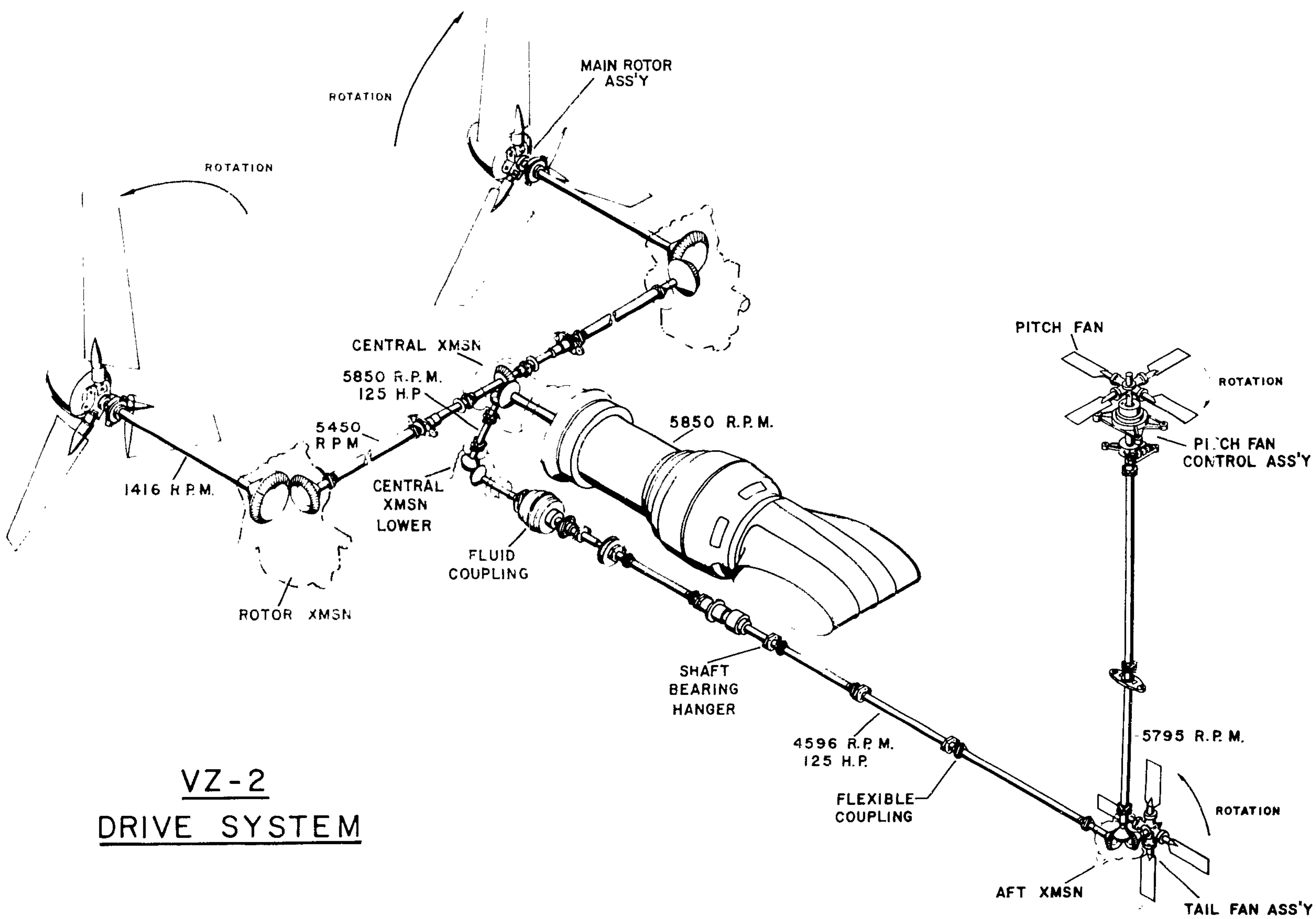

MAIN ROTOR ASS'Y
ROTATION
ROTATION
CENTRAL XMSN
5850 R.P.M.
125 H.P
5450 R P M
1416 R.P.M.
CENTRAL XMSN LOWER
FLUID COUPLING
ROTOR XMSN
5850 R.P.M.
PITCH FAN
ROTATION
PI.'CH FAN CONTROL ASS'Y
SHAFT BEARING HANGER
4596 R.P.M.
125 H.P.
FLEXIBLE COUPLING
5795 R.P.M.
ROTATION
AFT XMSN
TAIL FAN ASS'Y
VZ-2
DRIVE SYSTEM

FIGURE 18.

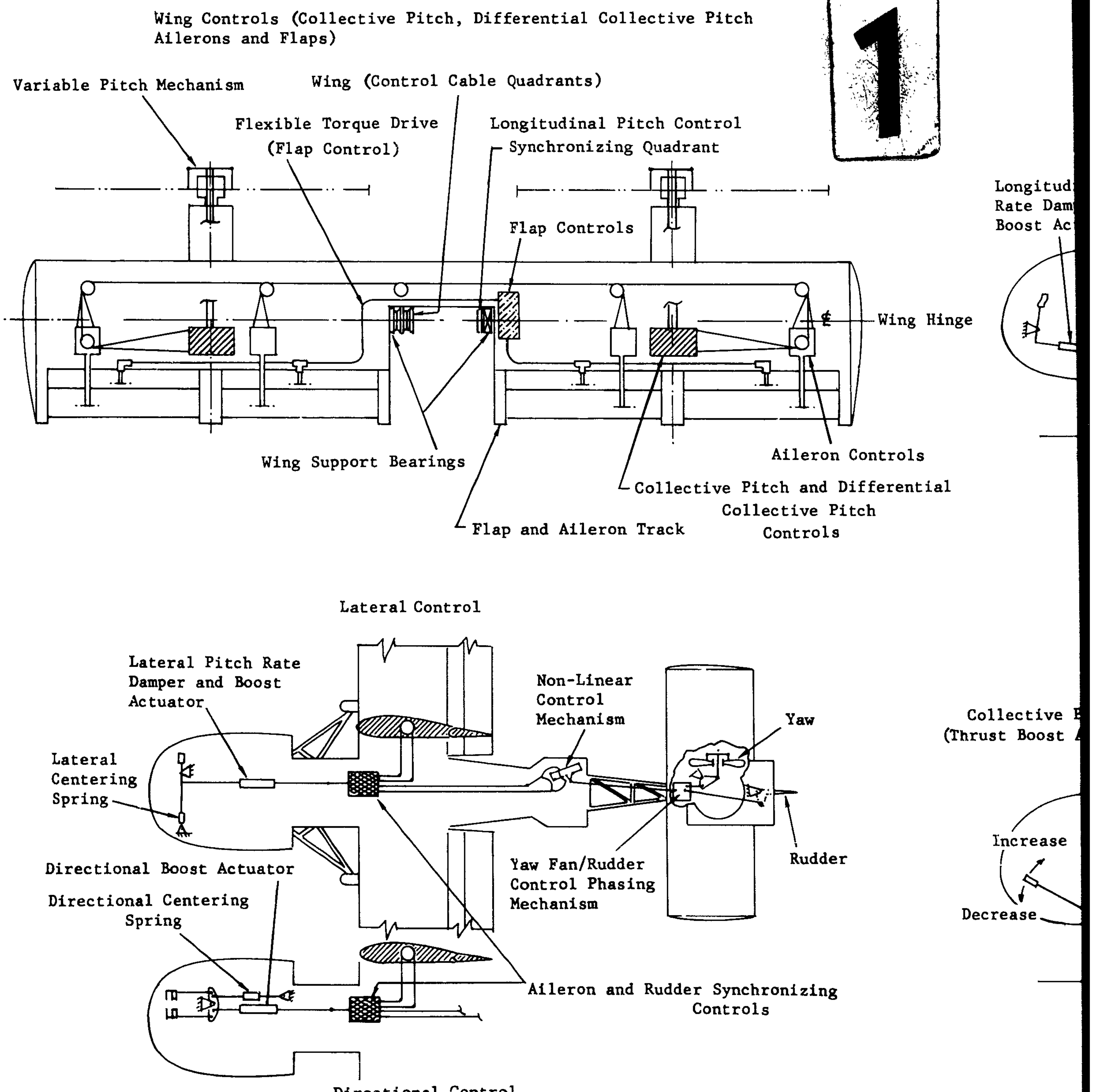


FIGURE 19. SCH

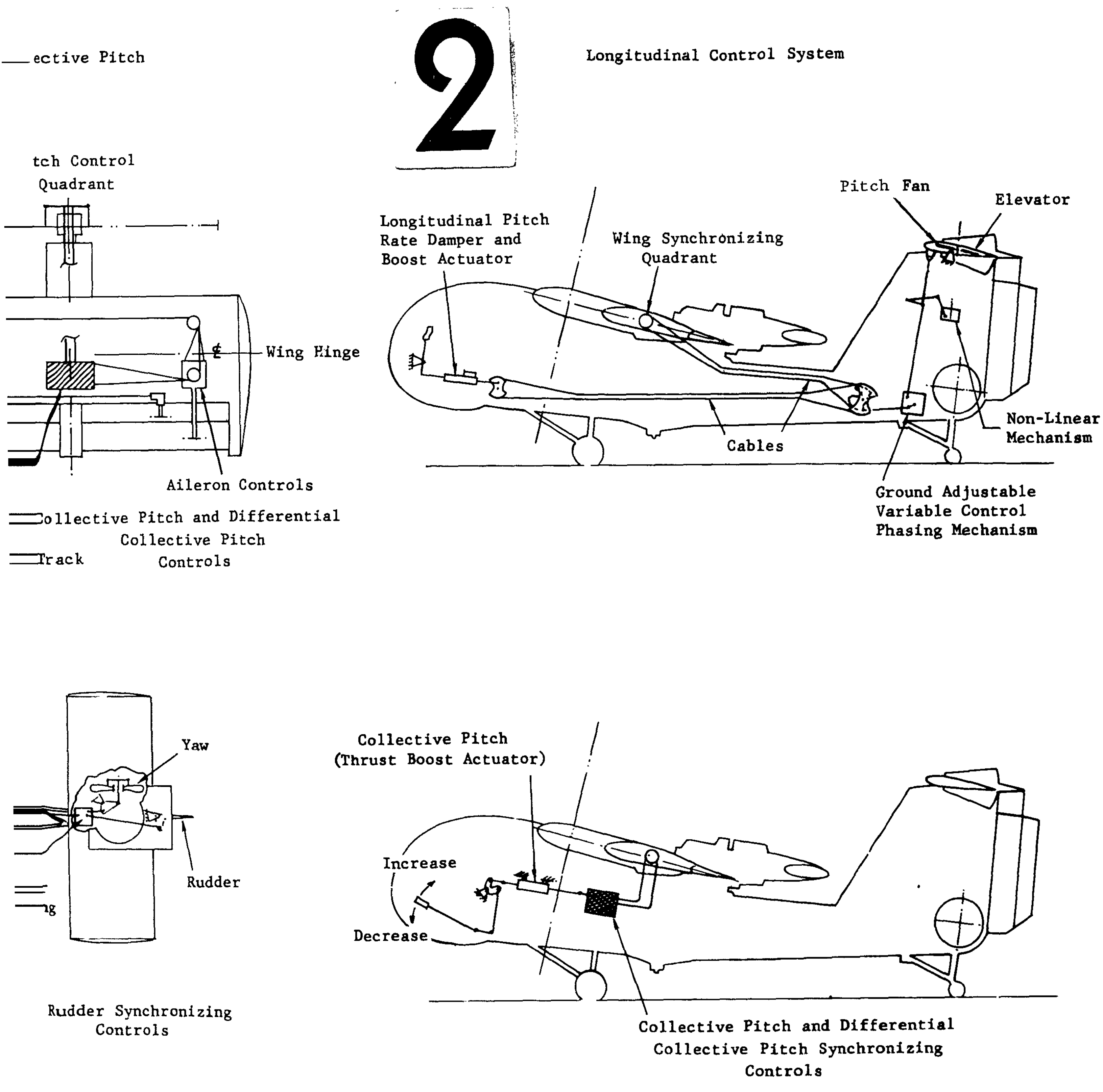


FIGURE 19. SCHEMATIC OF THE MODIFIED CONTROL SYSTEM OF VZ-2 RESEARCH AIRCRAFT

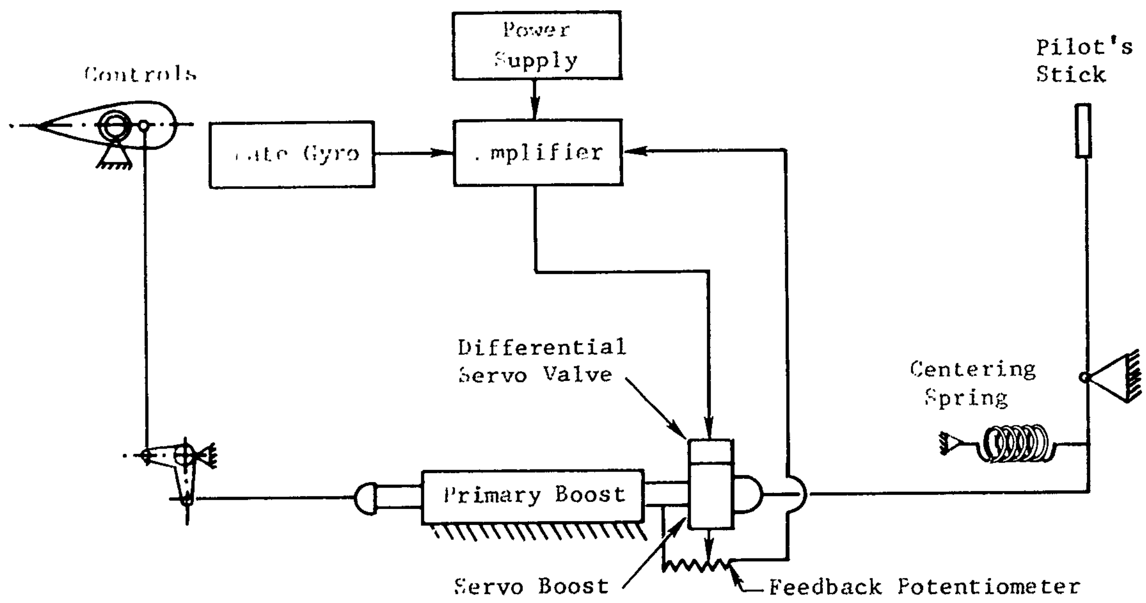
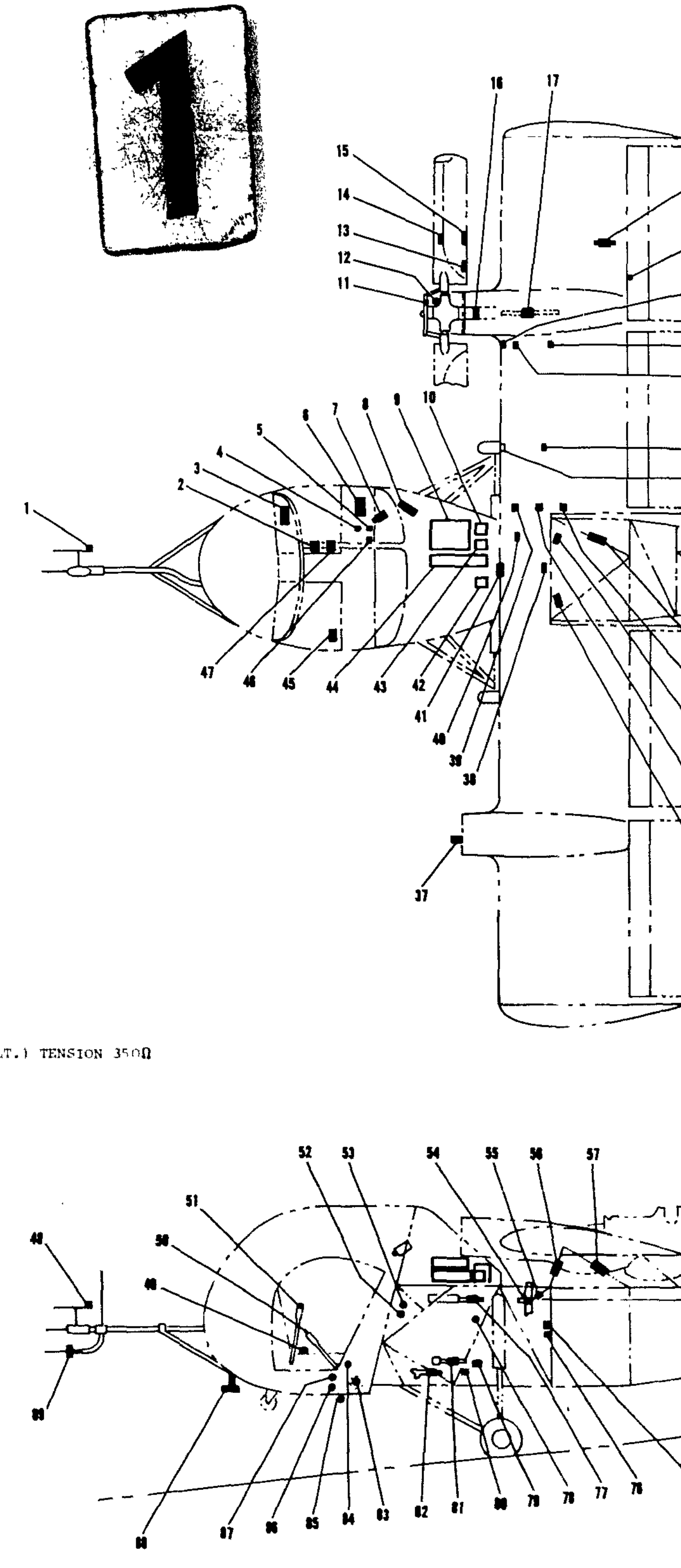


FIGURE 20. SCHEMATIC OF STABILITY AUGMENTATION SYSTEM (SAS) USED IN PITCH AND ROLL

1 ANGLE OF ATTACK
2 A.V. PITCH
3 4F4H 6 VOLT BATTERY (2)
4 LOW AIRSPEED TRANSDUCER
5 HIGH AIRSPEED TRANSDUCER
6 LOW AIR SPEED OSCILLOGRAPH
7 COCKPIT CAMERA
8 BALANCE PANEL (4)
9 OSCILLOGRAPH
10 TIMER
11 MAIN ROTOR SPIDER Q76R3041-23 BENDING 120Ω
12 C.P.T. BLADE FLAP
13 ROTOR BLADE Q76R1002-101 "E" TENSION 350Ω
14 ROTOR BLADE Q76R1002-101 "A" BENDING 350Ω
15 ROTOR SLADES Q76R1002-101 "D" TENSION 350Ω
16 MAIN ROTOR SHAFT Q76R3042-1 BENDING 120Ω
17 C.P. SHAFT SK9029-11 BENDING 120Ω
18 AILERON ROD Q76C1073-20 TENSION 350Ω
19 AILERON POSITION POT. 10K
20 WING STA. 80 F_1 & F_2 ABSOLUTE 120Ω
21 WING STA. 80 C_1 & C_2 ABSOLUTE 120Ω
22 WING STA. 80 H_1 & H_2 TORSION 120Ω
23 WING STA. 40 B_1 & B_2 ABSOLUTE 120Ω
24 WING STA. 40 E_1 & E_2 ABSOLUTE 120Ω
25 ENGINE TORQUE TRANSMITTER 5K
26 HORIZ. STAB. SPAR 350Ω BENDING
27 PITCH FAN BLADE Q76R8003-1 BENDING 350Ω
28 PITCH FAN SPIDER Q76R8044-1 BENDING 120Ω
29 PITCH FAN SHAFT Q76R8050-11 BENDING 120Ω
30 FUSE. HORIZONTAL TUBE Q76S0001-126 BENDING 350Ω
31 FUSE. DIAG. TUBE Q76S0001-106 BENDING 350Ω
32 FUSE. DIAG. TUBE Q76S0001-111 BENDING 350Ω
33 WING STA. 22 Vc
34 FUSE. WING HINGE TUBE Q76S0001-154 TENSION 350Ω
35 WING STA. 22 BMc
36 FUSE. WING HINGE TUBE Q76S0001-153 TENSION 350Ω
37 ROTOR 1/REV.
38 WING STA. 0 A_1 & A_2 ABSOLUTE 120Ω
39 WING STA. 22 BM_f
40 WING STA. 10 G_1 & G_2 TORSION 120Ω
41 WING STA. 0 D_1 & D_2 ABSOLUTE 120Ω
42 C.P.T. CONTROL BOX
43 POWER SUPPLY
44 BALANCE PANEL
45 A.V. YAW
46 ALTITUDE TRANSDUCER
47 A.V. ROLL
48 ANGLE OF YAW
49 RUDDER POSITION POT. 10K
50 CAMERA SWITCH
51 INSTRUMENTATION SWITCH
52 C.P. POSITION POT. 10K (OSC)
53 C.P. POSITION POT. 10K (IND)
54 GIMBAL SUPPORT Q76E7042-4 TENSION 350Ω
55 FUSE. GIMBAL SUPPORT TUBE Q76S0001-142 (T-7) TENSION 120Ω
56 FUSE. WING HINGE TUBE Q76S0001-151 L.H. - 152 R.H. (T-2 RT.) & T-5 LT.) TENSION 350Ω
57 FUSE. WING HINGE TUBE Q76S0001-150 (T-3 R.S.) (T-6L.S.) TENSION 350Ω
58 HORIZ. SHAFT TORQUE 120Ω SK9064-6 OR -15
59 YAW FAN SPIDER Q76R8044-1 BENDING 120Ω
60 LONG. CONTROL ROD SK8099-12 TENSION 350Ω
61 TAIL CAMERA
62 ELEVATOR POSITION POT. 10K
63 LONG. CONTROL ROD SK8099-110 TENSION 350Ω
64 (UPPER) VERT. TAIL FWD. SPAR BENDING 120Ω
65 RECORD LIGHT
66 LONG. CONTROL ROD SK8099-211 TENSION 350Ω
67 (LOWER) VERT. TAIL FWD SPAR BENDING 120Ω
68 YAW FAN BLADE Q76R8003-1 BENDING 350Ω
69 YAW FAN SHAFT SK9216-1 BENDING 350Ω
70 FUSE. TUBE (T-8) Q76S0001-406 TENSION 120Ω
71 AFT DIR. CONTROL ROD SK8092-24 TENSION 350Ω
72 RUDDER POSITION POT. 10K
73 FUSE. LONG. TUBE Q76S0001-431 TENSION 350Ω
74 FUSE. DIAG. TUBE Q76S0001-428 TENSION 350Ω
75 LAT. ACCELEROMETER
76 LONG. ACCELEROMETER
77 C.P. ROD Q76C1064-9 TENSION 350Ω
78 WING POSITION POT. 10K
79 DIFF. COLL. LINK Q76C1065-7 TENSION 350Ω
80 LAT. STICK POSITION POT. 10K
81 LAT. ROD SK9177-20 TENSION 350Ω
82 LONG. ROD SK9078-3 TENSION 350Ω
83 RECORD LIGHT
84 LAT. STICK POSITION (OSC.) POT. 10K
85 RUDDER POSITION (PEDAL) POT. 10K
86 LONG, STICK POSITION POT. 10K(IND.)
87 LONG. STICK POSITION POT. 10K(OSC.)
88 F.A.T. (NASA)
89 SWIVEL STATIC PROBE



VZ-2 INSTRUMENTATION

FIGURE 21.

PH

41-23 BENDING 120Ω

"E" TENSION 350Ω
"A" BENDING 350Ω
1 "D" TENSION 350Ω
2-1 BENDING 120Ω
DING 120Ω
TENSION 350Ω
K
OLUTE 120Ω
OLUTE 120Ω
SION 120Ω
OLUTE 120Ω
OLUTE 120Ω
R 5K
BENDING
-1 BENDING 350Ω
4-1 BENDING 120Ω
-11 BENDING 120Ω
6S0001-126 BENDING 350Ω
1-106 BENDING 350Ω
1-111 BENDING 350Ω

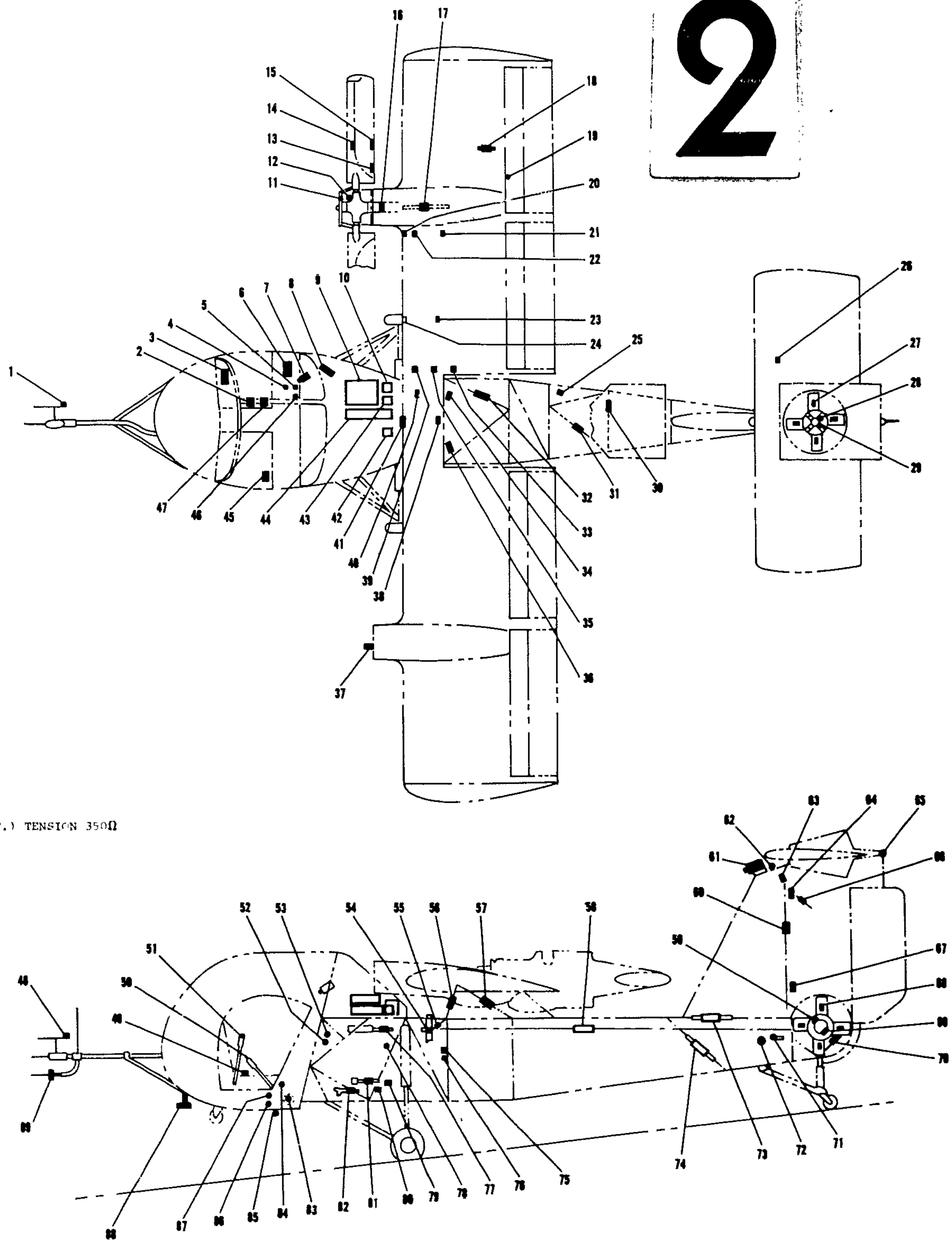
6S0001-154 TENSION 350Ω

6S0001-153 TENSION 350Ω

LUTE 120Ω

SION 120Ω
LUTE 120Ω

2



VZ-2 INSTRUMENTATION

FIGURE 21.

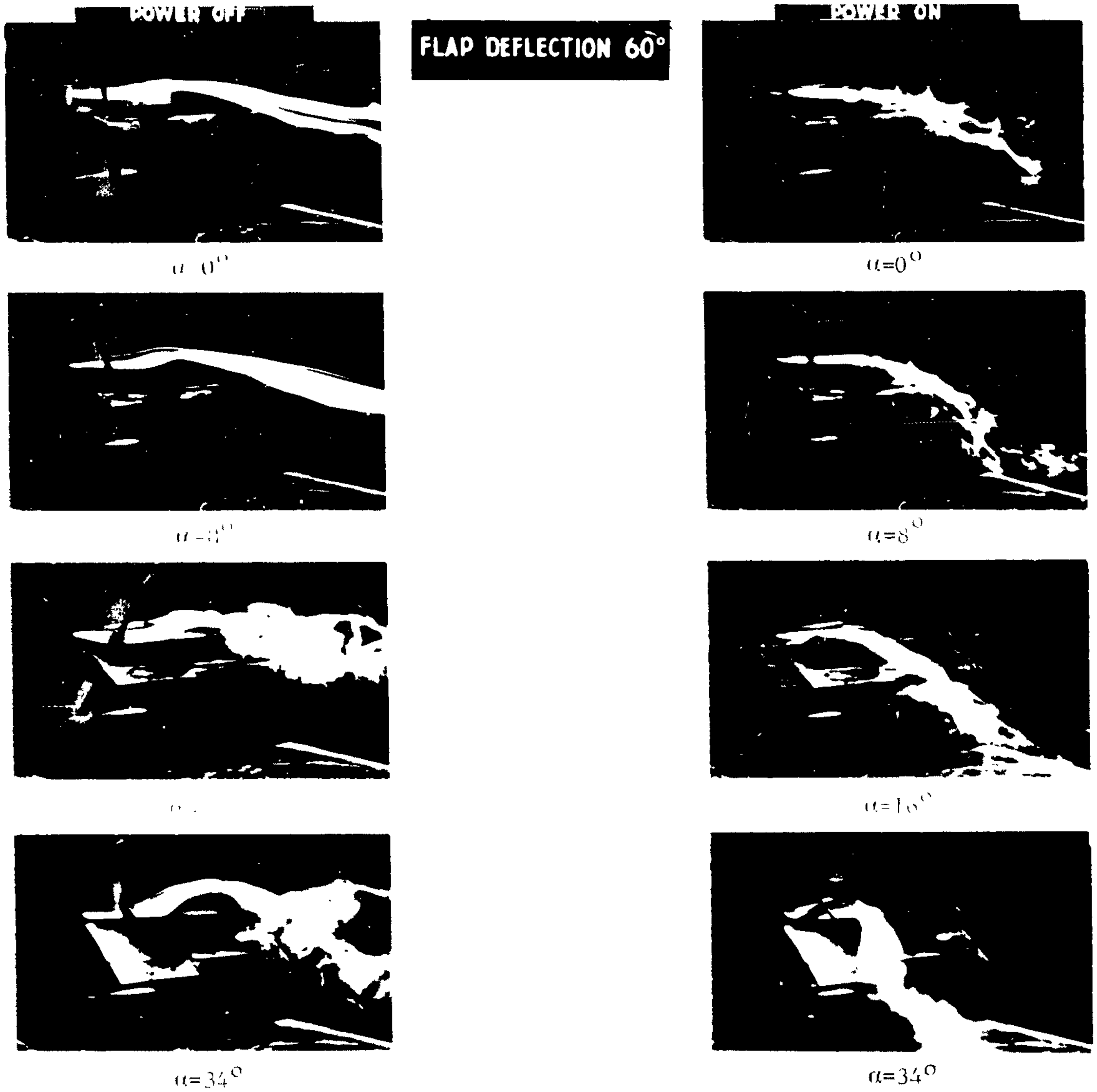


FIGURE 22. WING-PROPELLER MODEL IN PRINCETON SMOKE TUNNEL

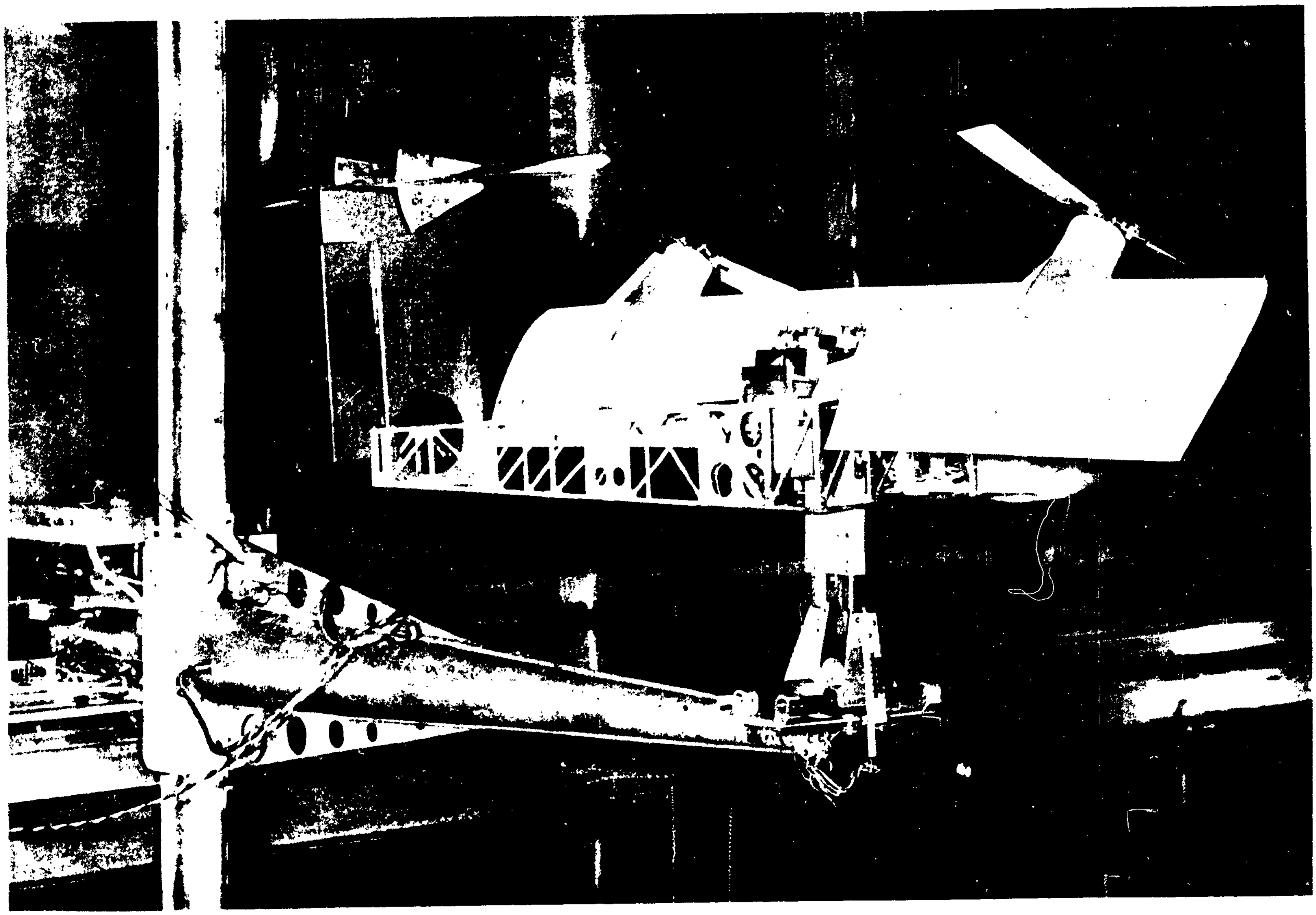


FIGURE 23. DYNAMIC MODEL OF THE VZ-2 RESEARCH AIRCRAFT ON THE PRINCETON UNIVERSITY TEST TRACK

TAIL FORCE REQUIRED TO BALANCE PITCHING MOMENT IN STEADY FLIGHT & AT .2g ACCELERATION AT VARIOUS TILT ANGLES & CORRESPONDING FLIGHT SPEED

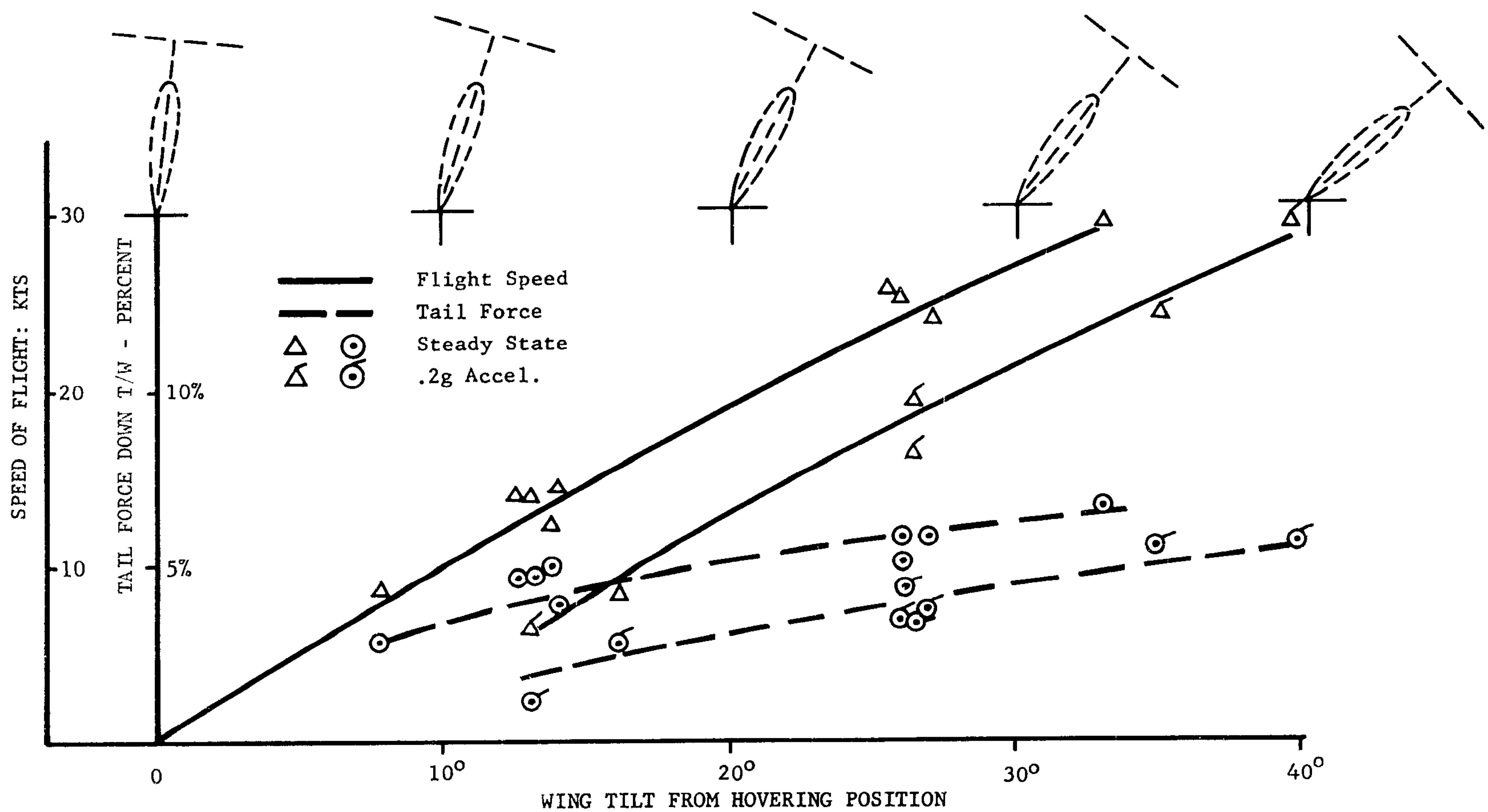


FIGURE 24. EXAMPLE OF RESULTS OBTAINED FROM THE PRINCETON UNIVERSITY TRACK

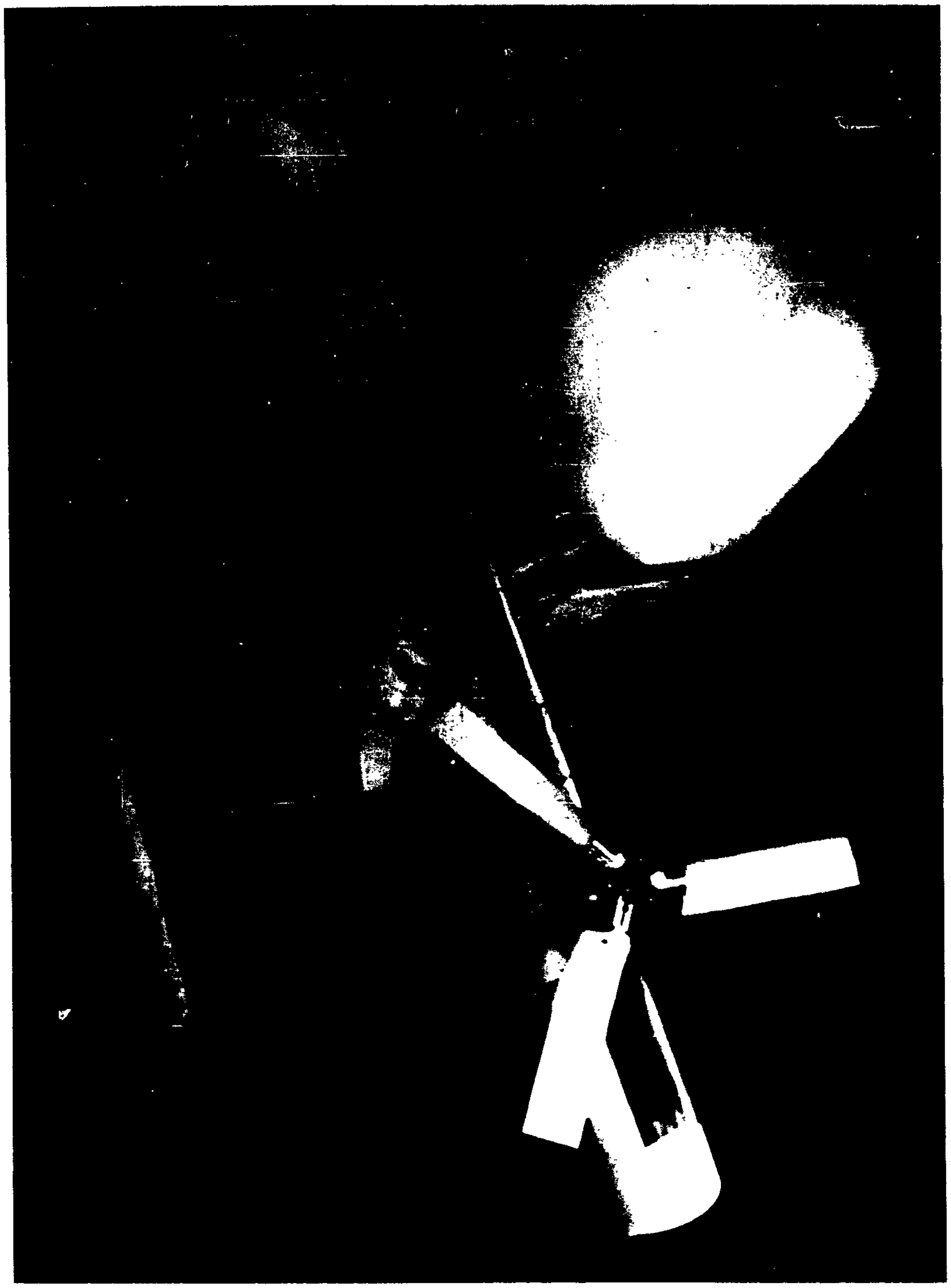


FIGURE 25. NASA 1/4 SCALE POWERED MODEL OF THE U.S. ARMY VZ-2 AIRCRAFT

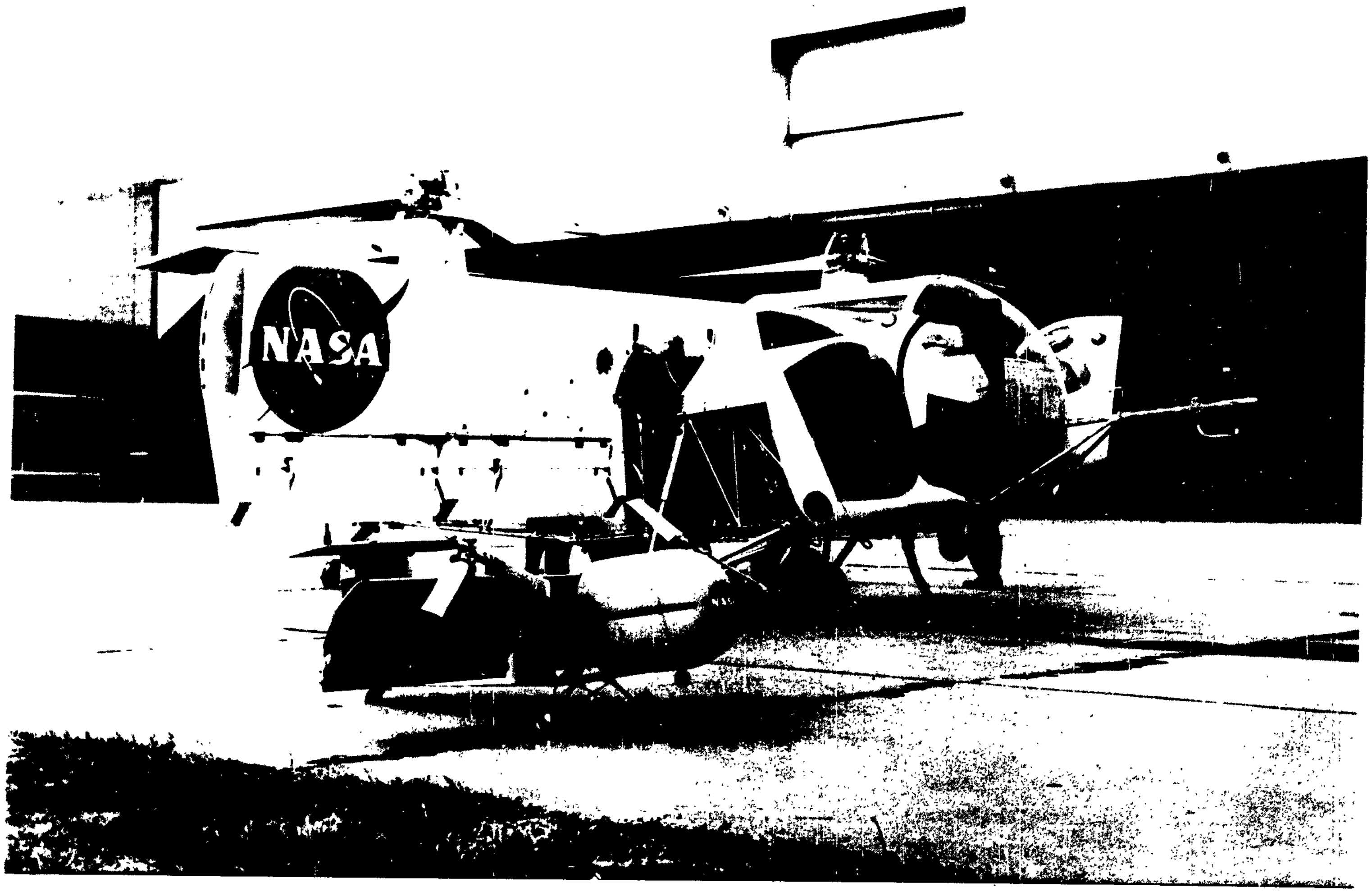


FIGURE 26. VZ-2 AND ONE-QUARTER SCALE POWERED MODEL AT NASA - LANGLEY FIELD

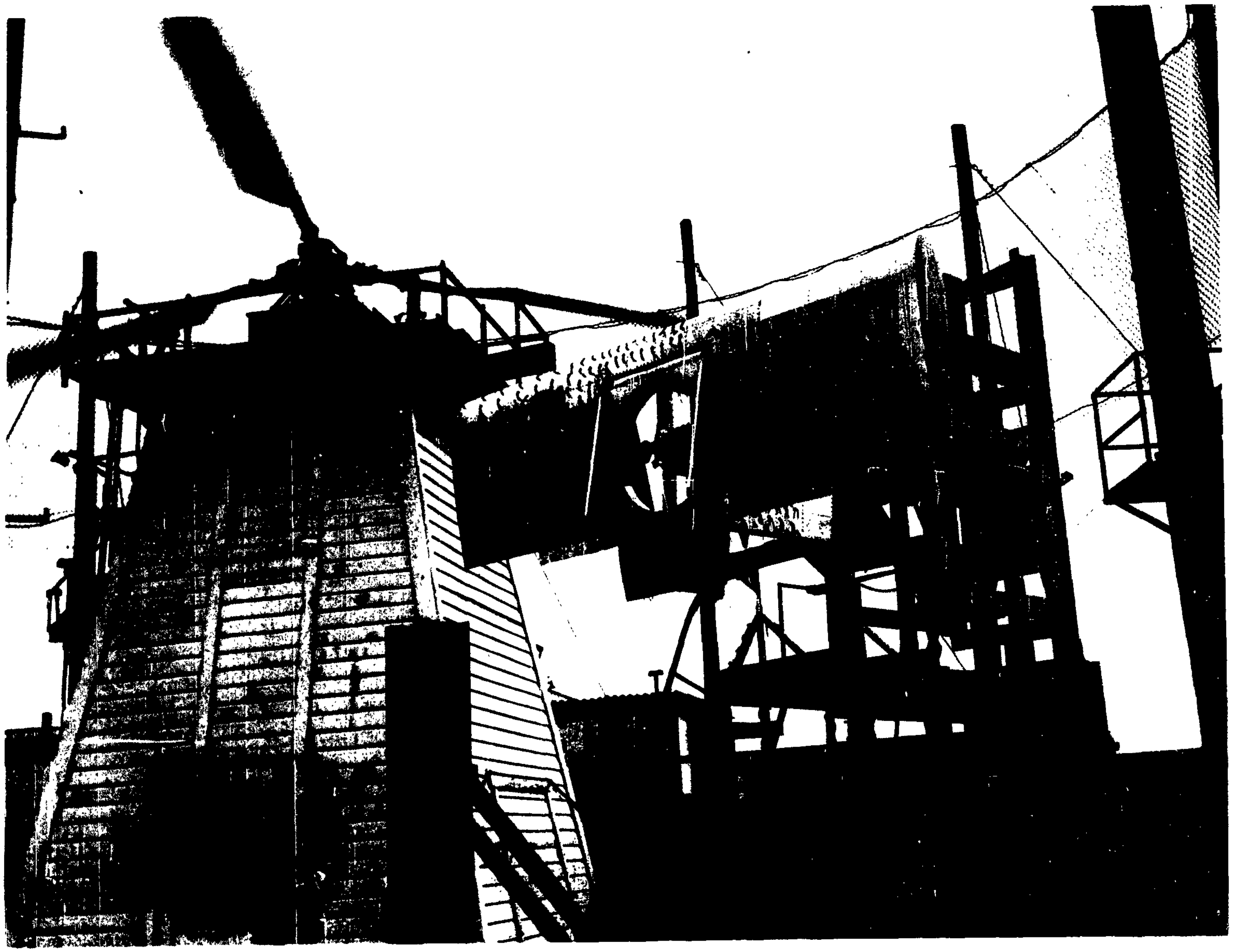


FIGURE 27. U.S. ARMY VZ-2 VTOL TAIL ASSEMBLY GROUND TEST

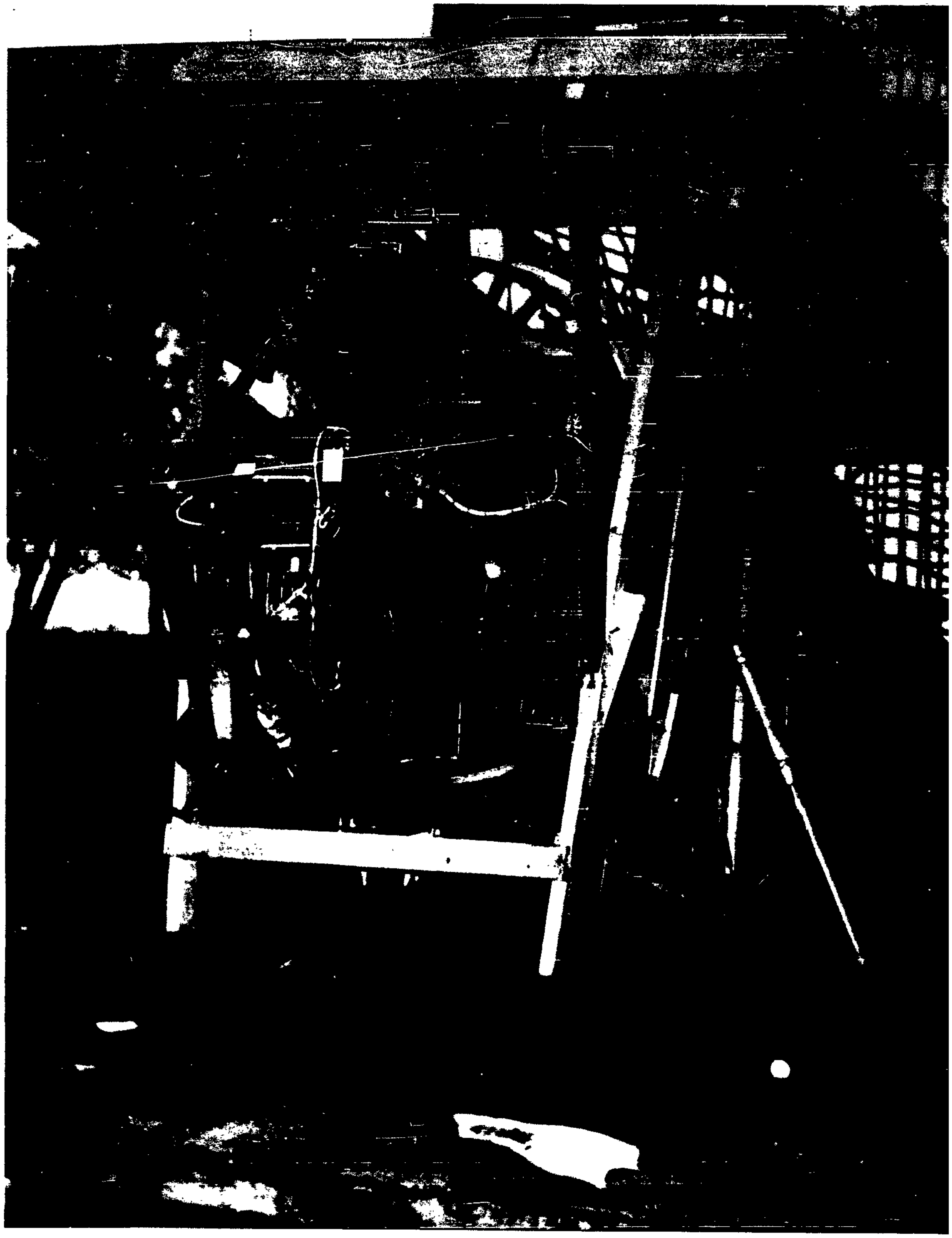


FIGURE 28. EJECTION SEAT PROOF TESTING RIG

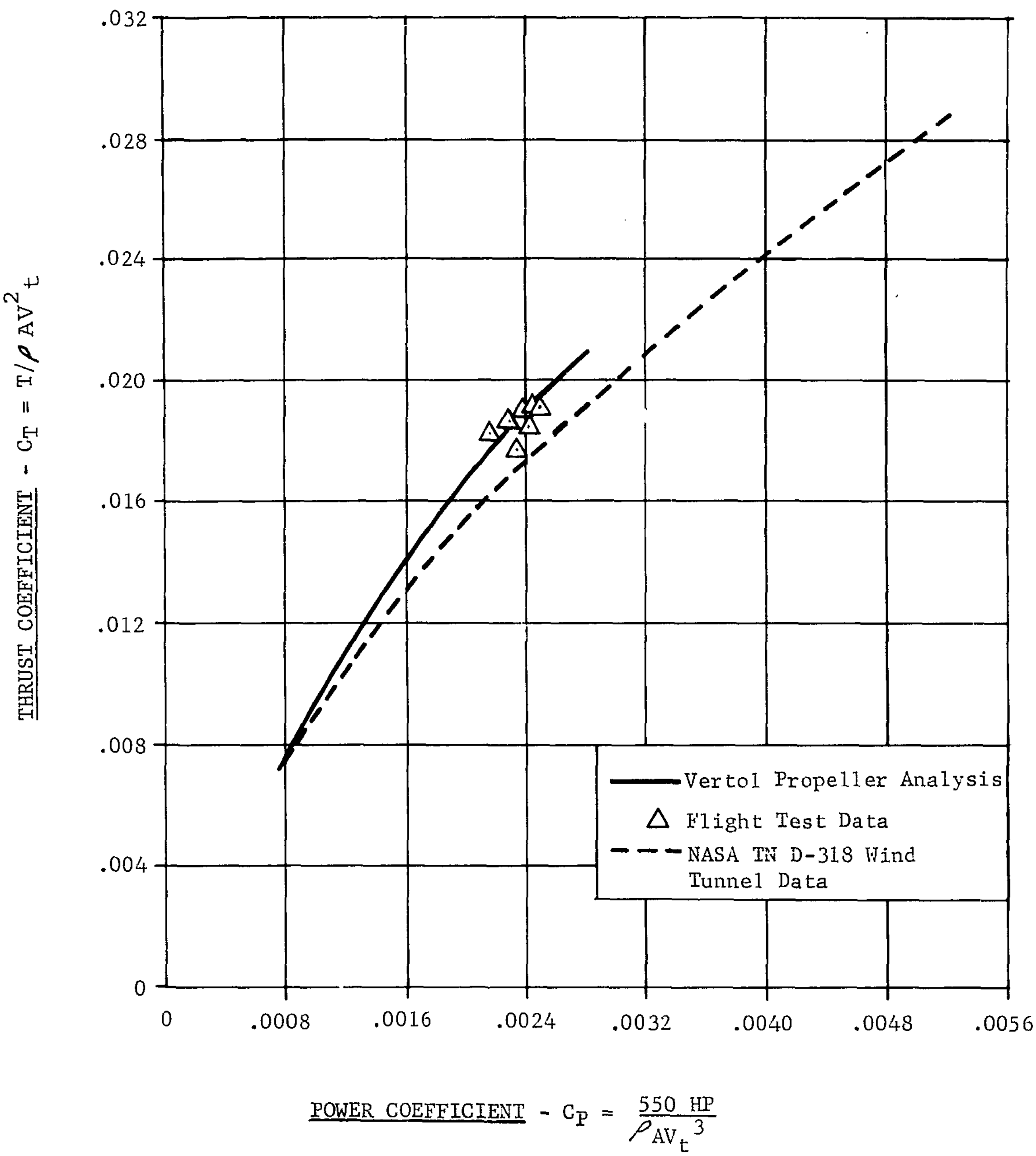


FIGURE 29. THRUST COEFFICIENT VS POWER COEFFICIENT (HOVER CONFIGURATION)

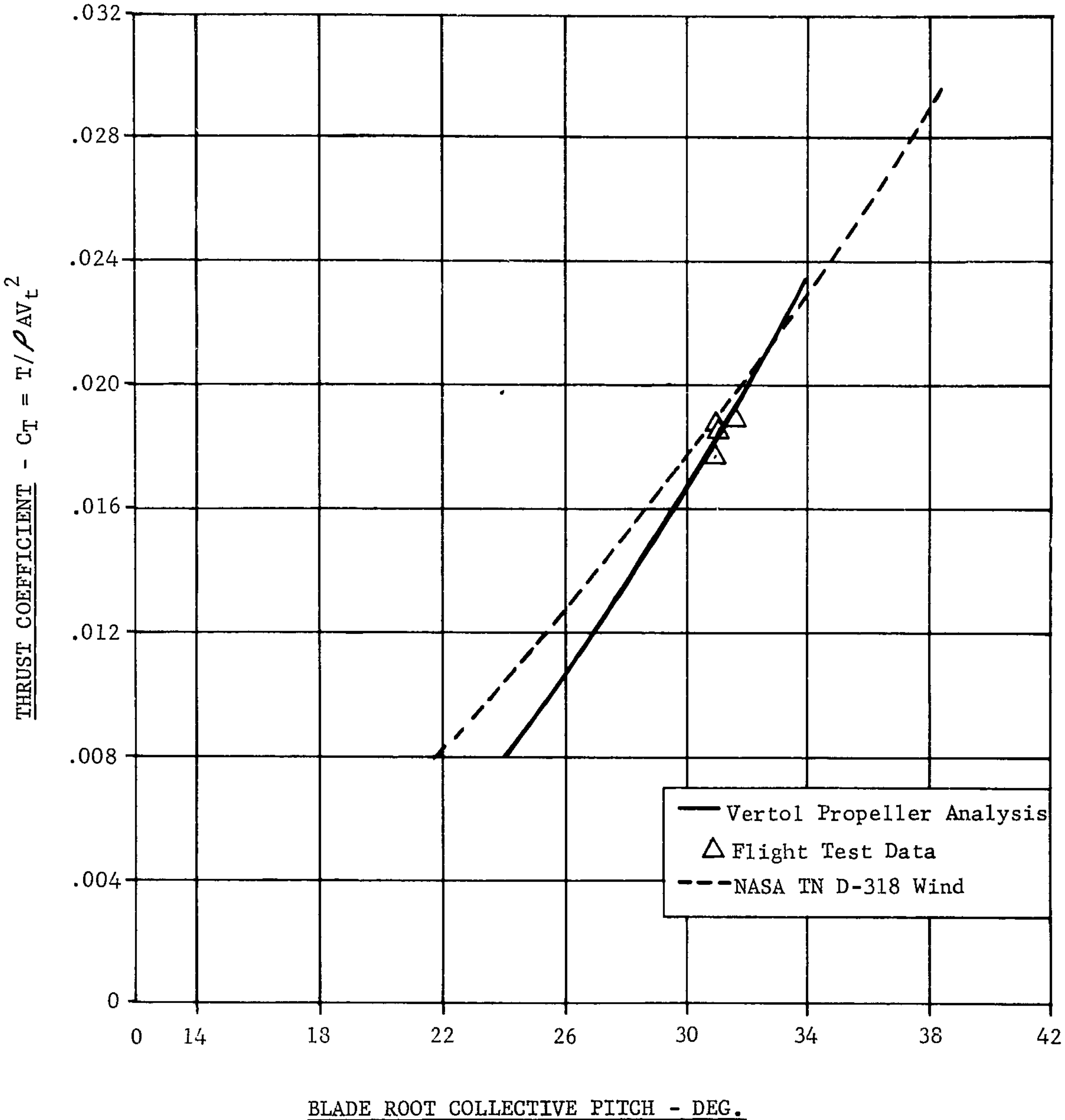
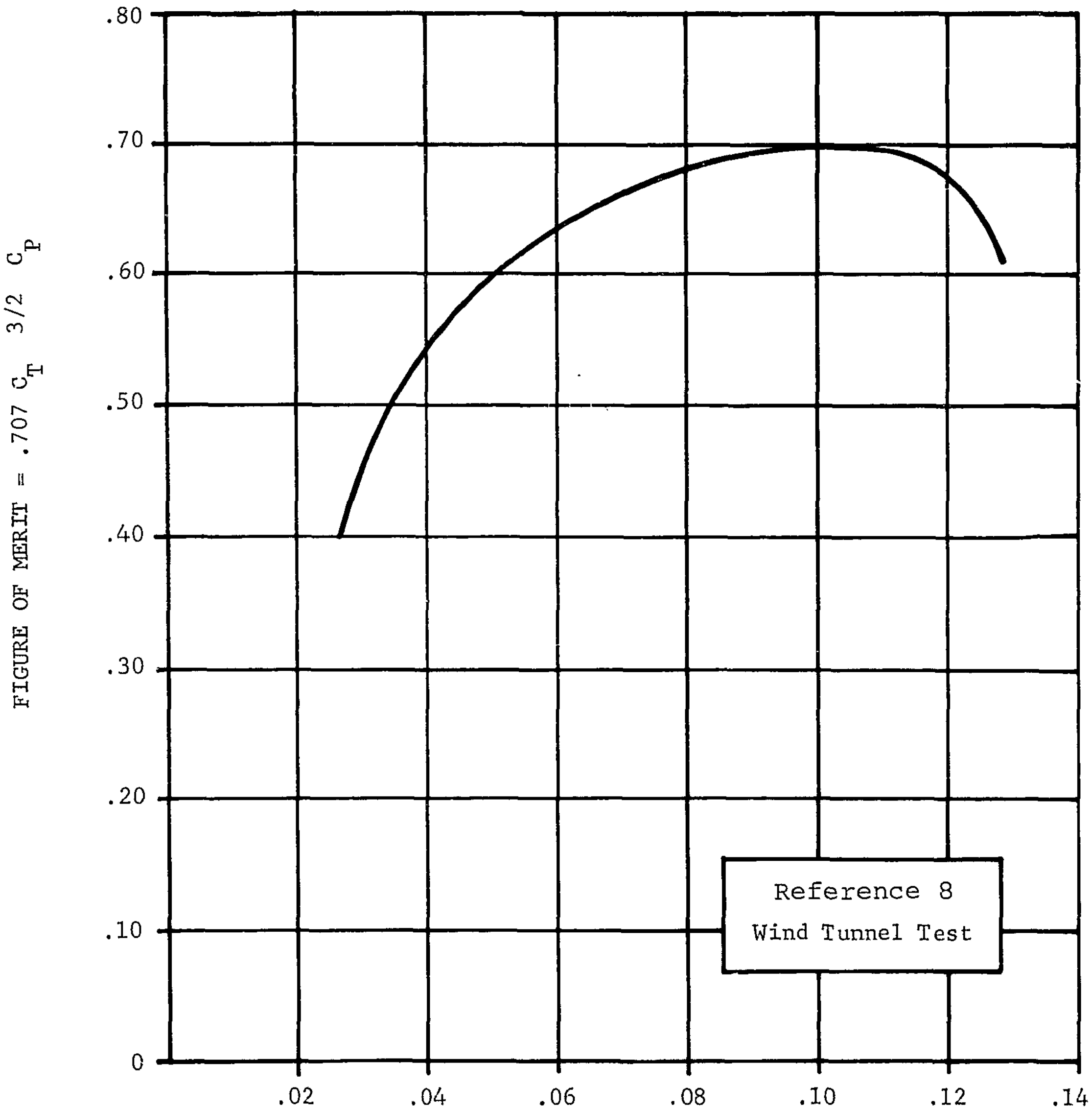


FIGURE 30. THRUST COEFFICIENT VS COLLECTIVE PITCH (HOVER CONFIGURATION)



$$\frac{C_T}{\sigma} = \frac{T}{\sigma \rho A V_t^2}$$

FIGURE 31. FIGURE OF MERIT VS C_T/σ (HOVER CONFIGURATION)

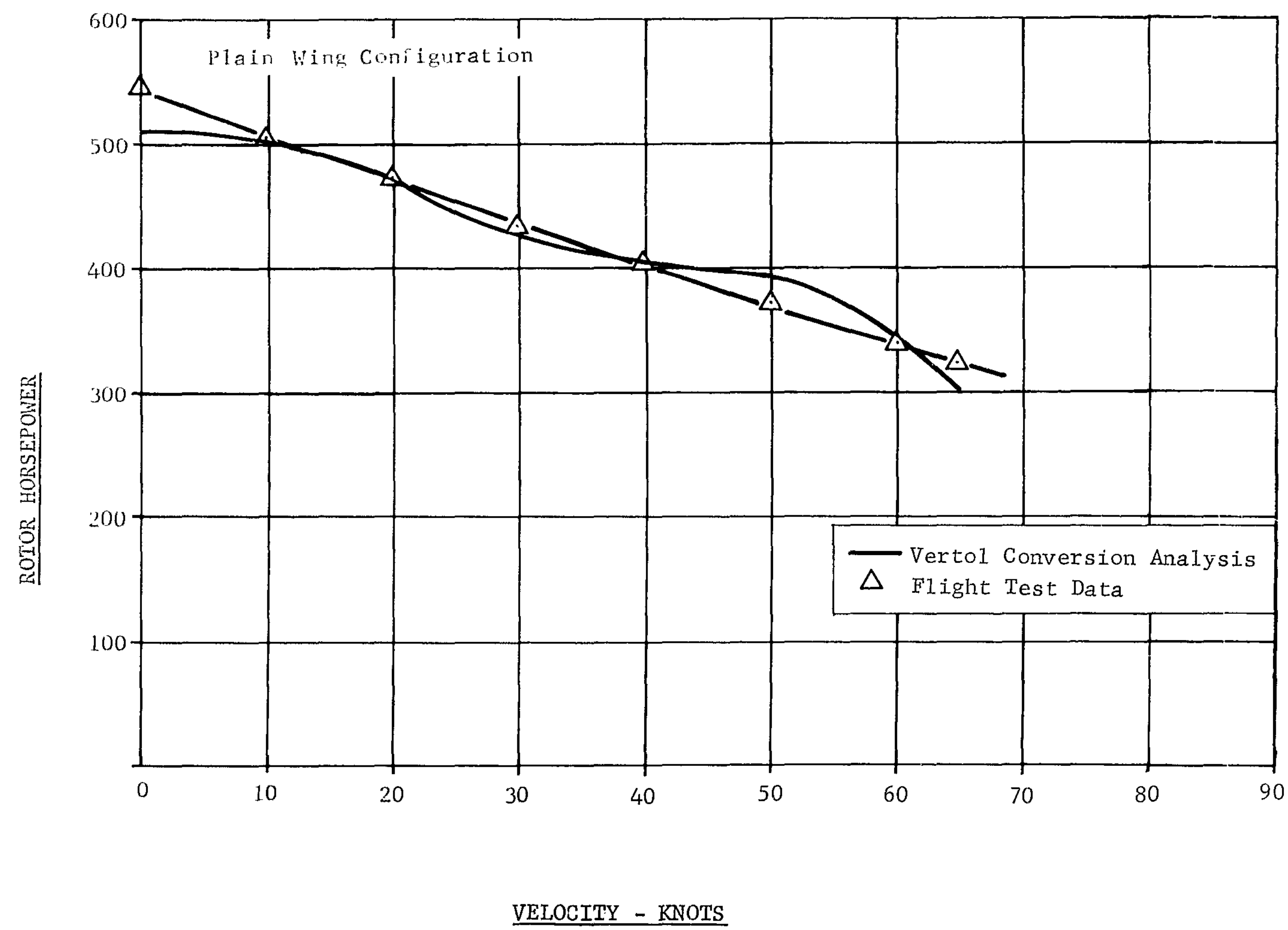


FIGURE 32. ROTOR HORSEPOWER VERSUS VELOCITY FOR THE VZ-2

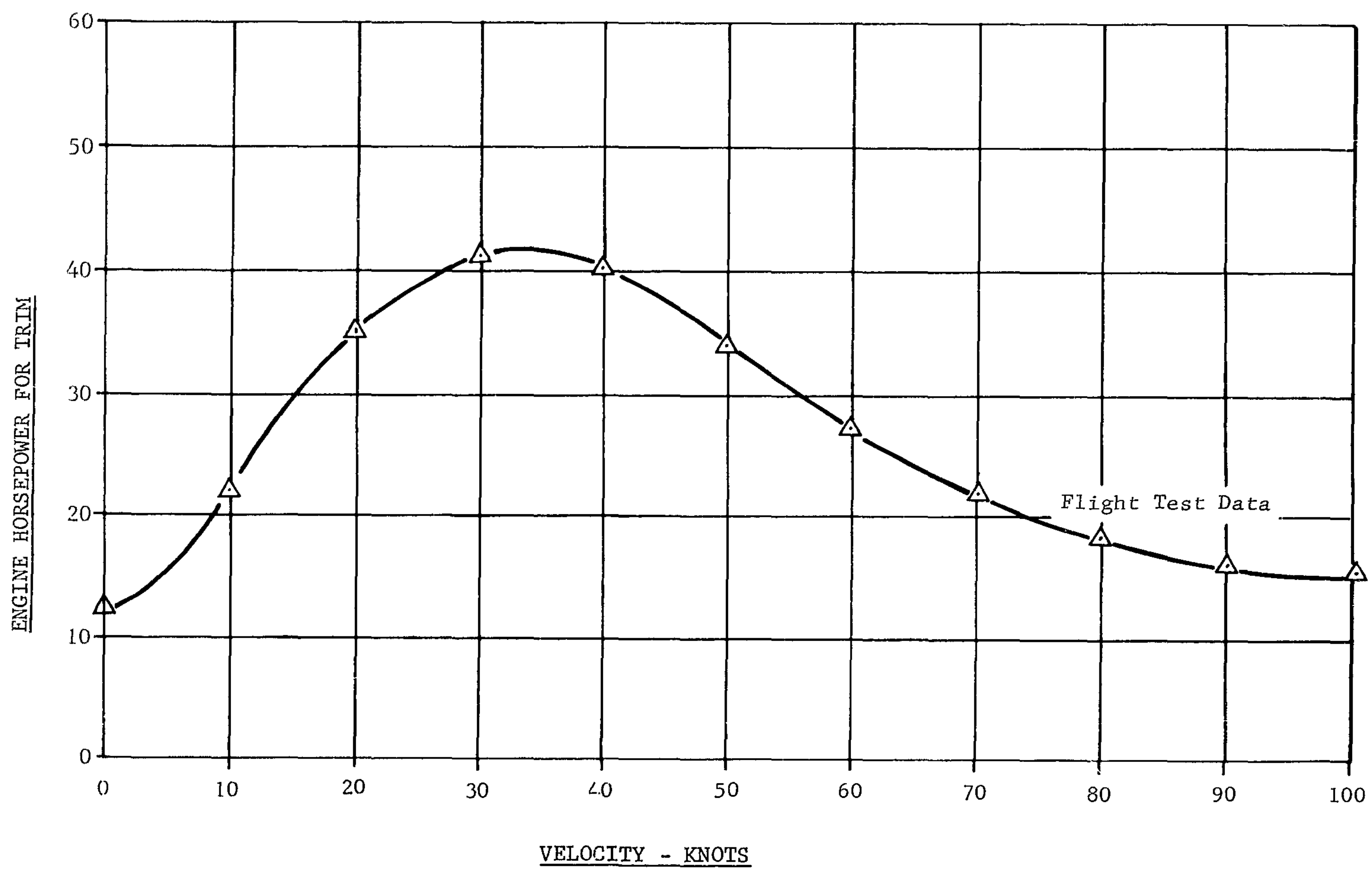


FIGURE 33. TAIL FAN POWER REQUIRED FOR TRIM VS. VELOCITY

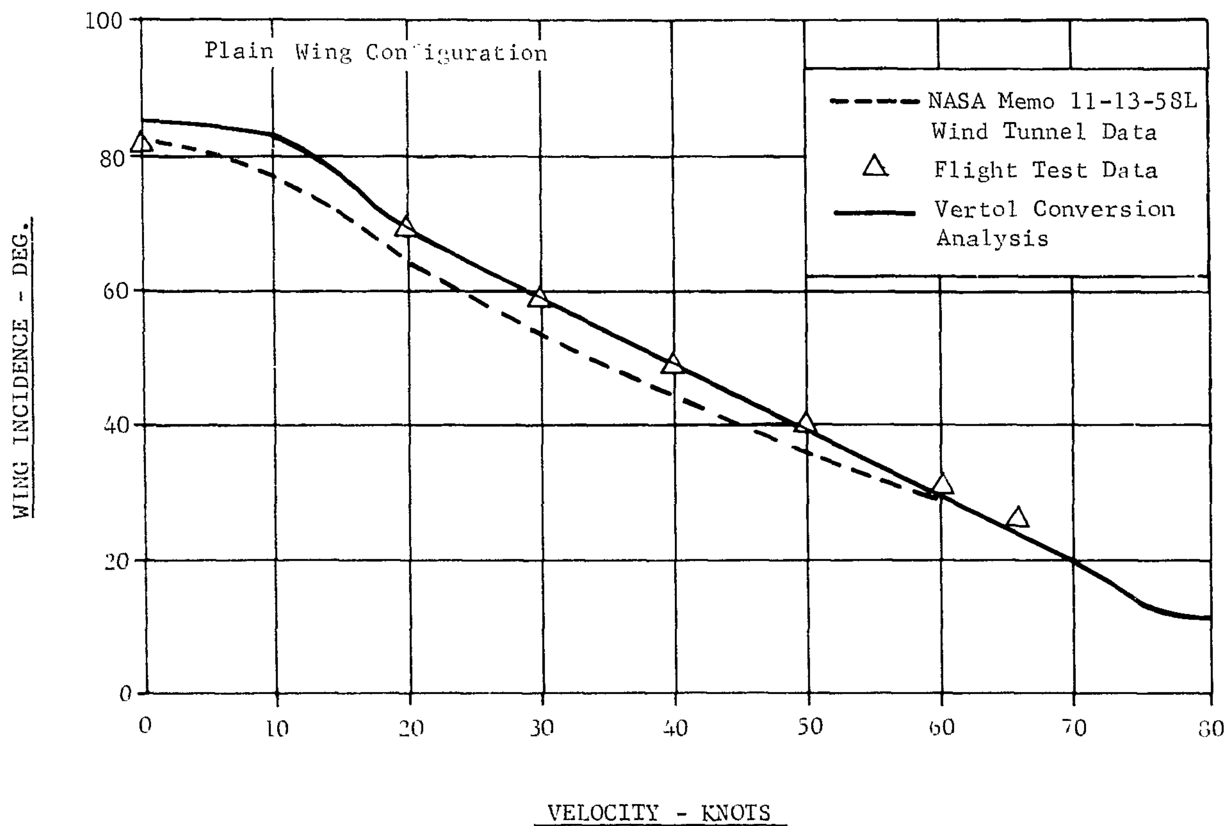


FIGURE 34. WING INCIDENCE VS. VELOCITY FOR THE VZ-2

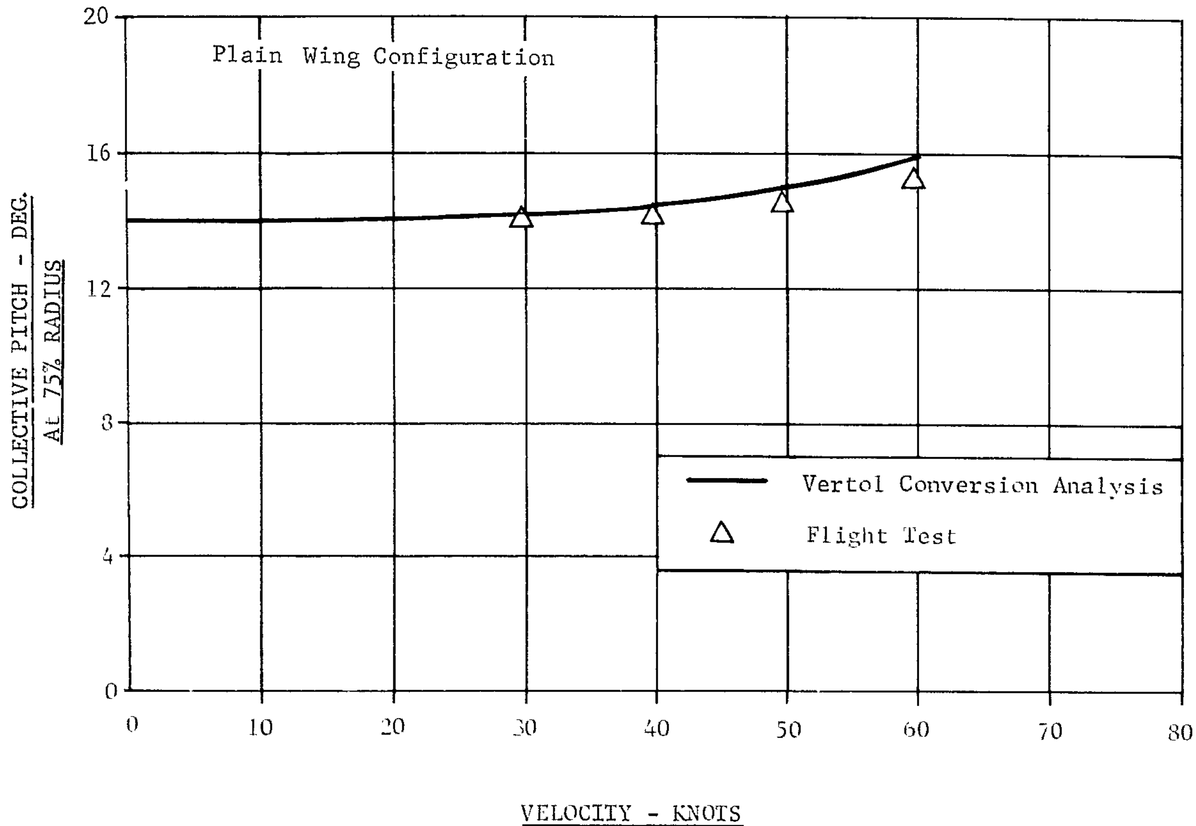


FIGURE 35. COLLECTIVE PITCH VS VELOCITY (CONVERSION CONFIGURATION)

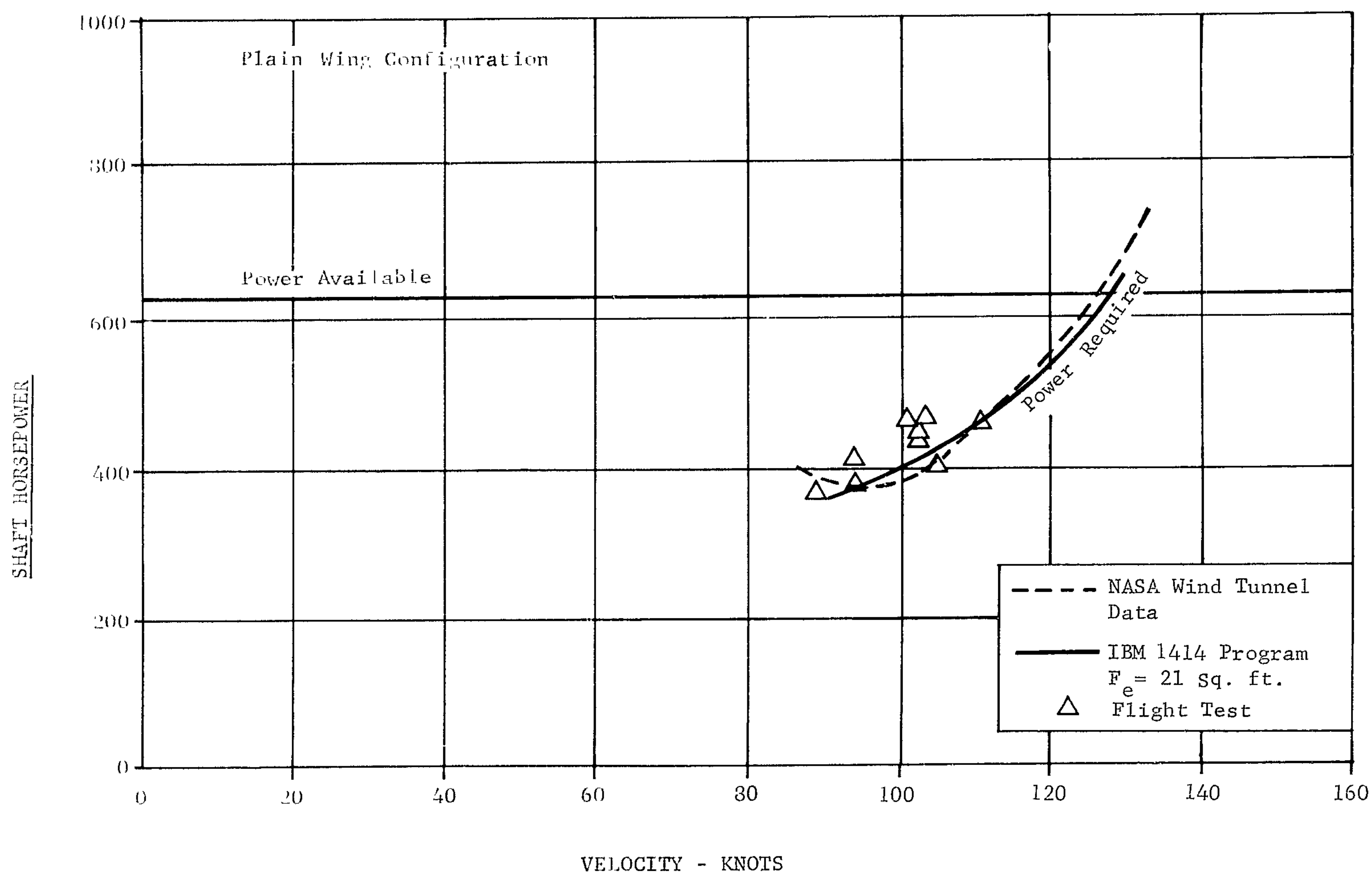


FIGURE 36 SHAFT HORSEPOWER VS VELOCITY (AIRPLANE CONFIGURATION)

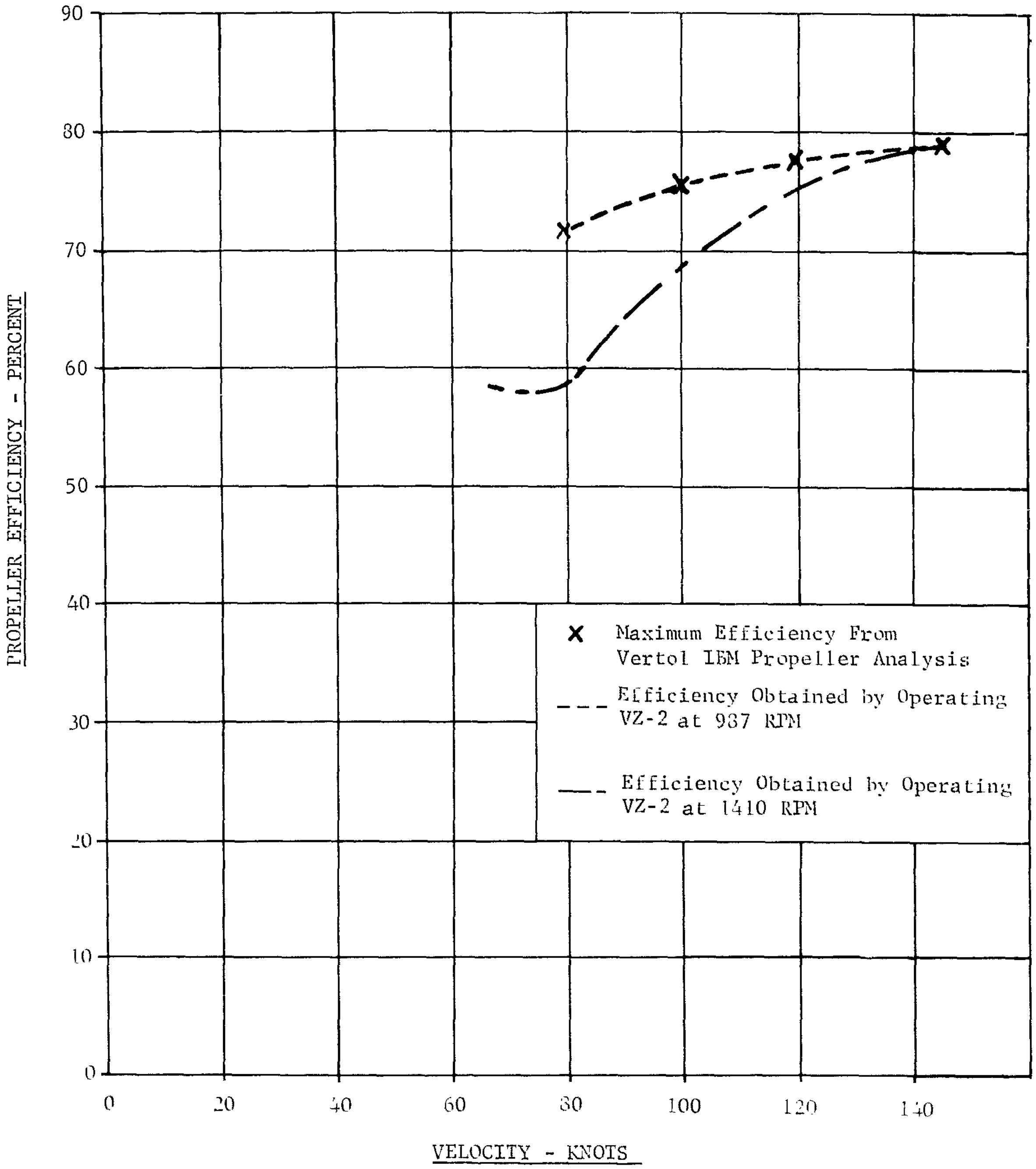


FIGURE 37. PROPELLER EFFICIENCY VS VELOCITY (AIRPLANE CONFIGURATION)

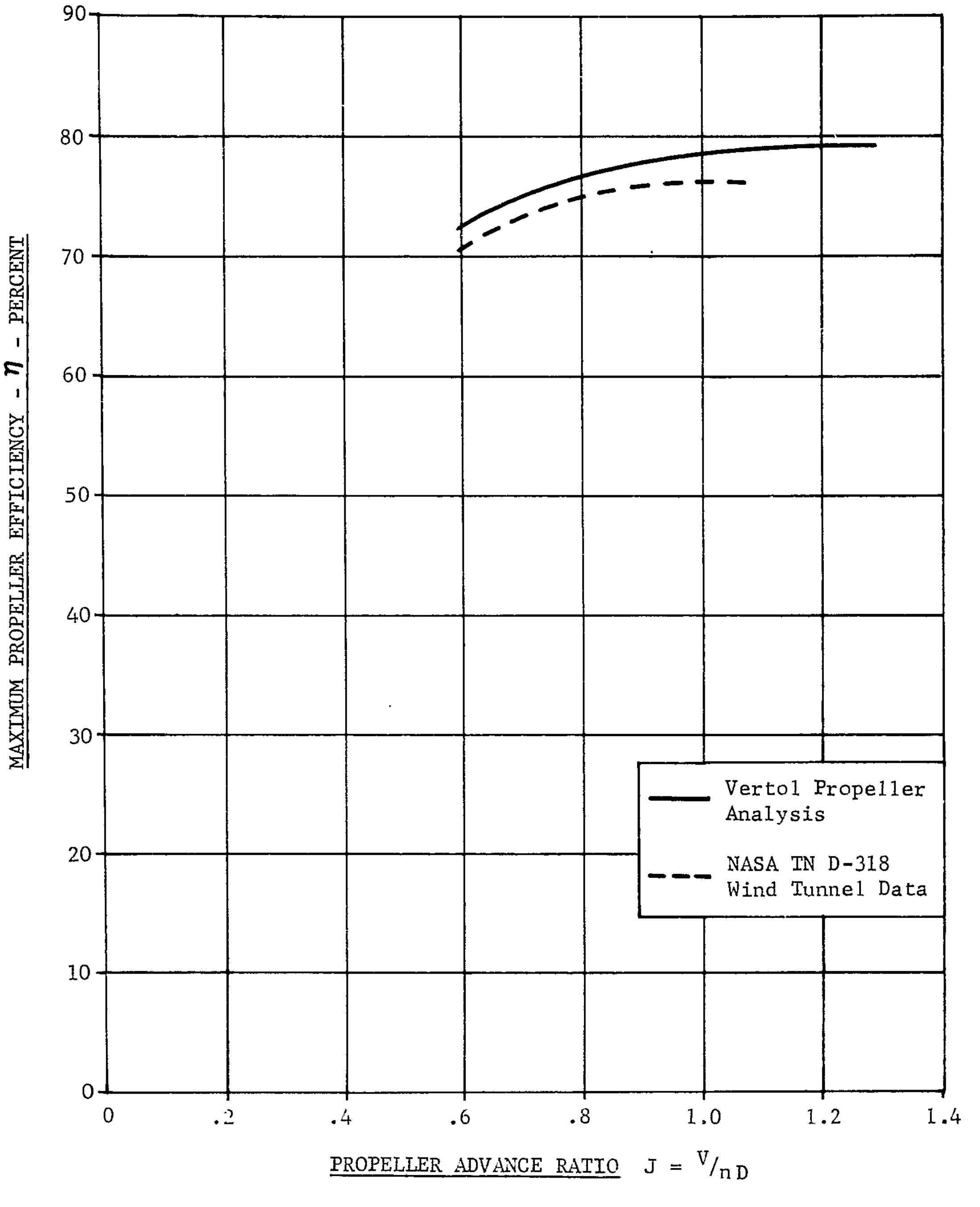


FIGURE 38. PROPELLER EFFICIENCY VS PROPELLER ADVANCE RATIO

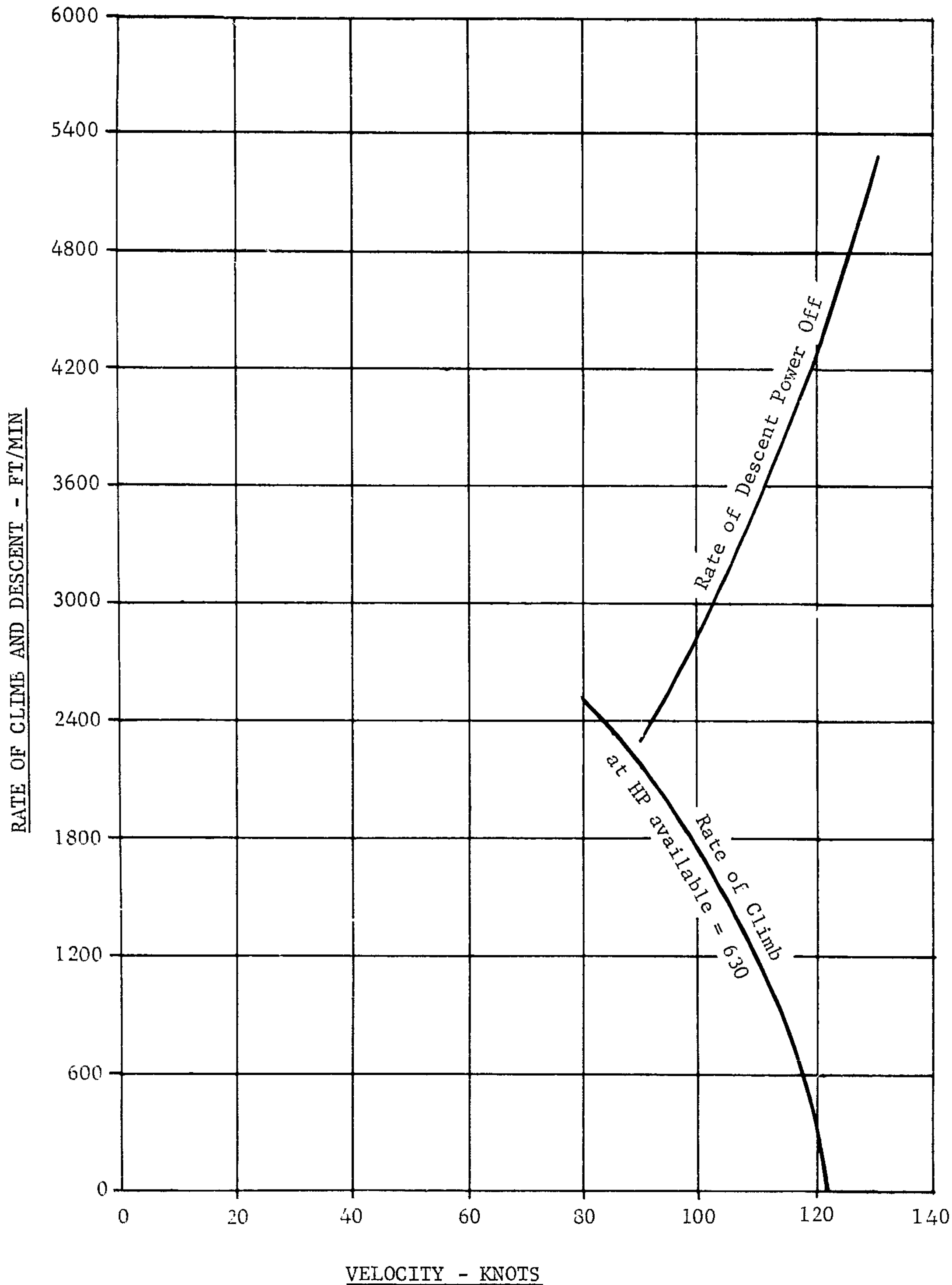


FIGURE 39. RATE OF CLIMB AND DESCENT VS VELOCITY (AIRPLANE CONFIGURATION)

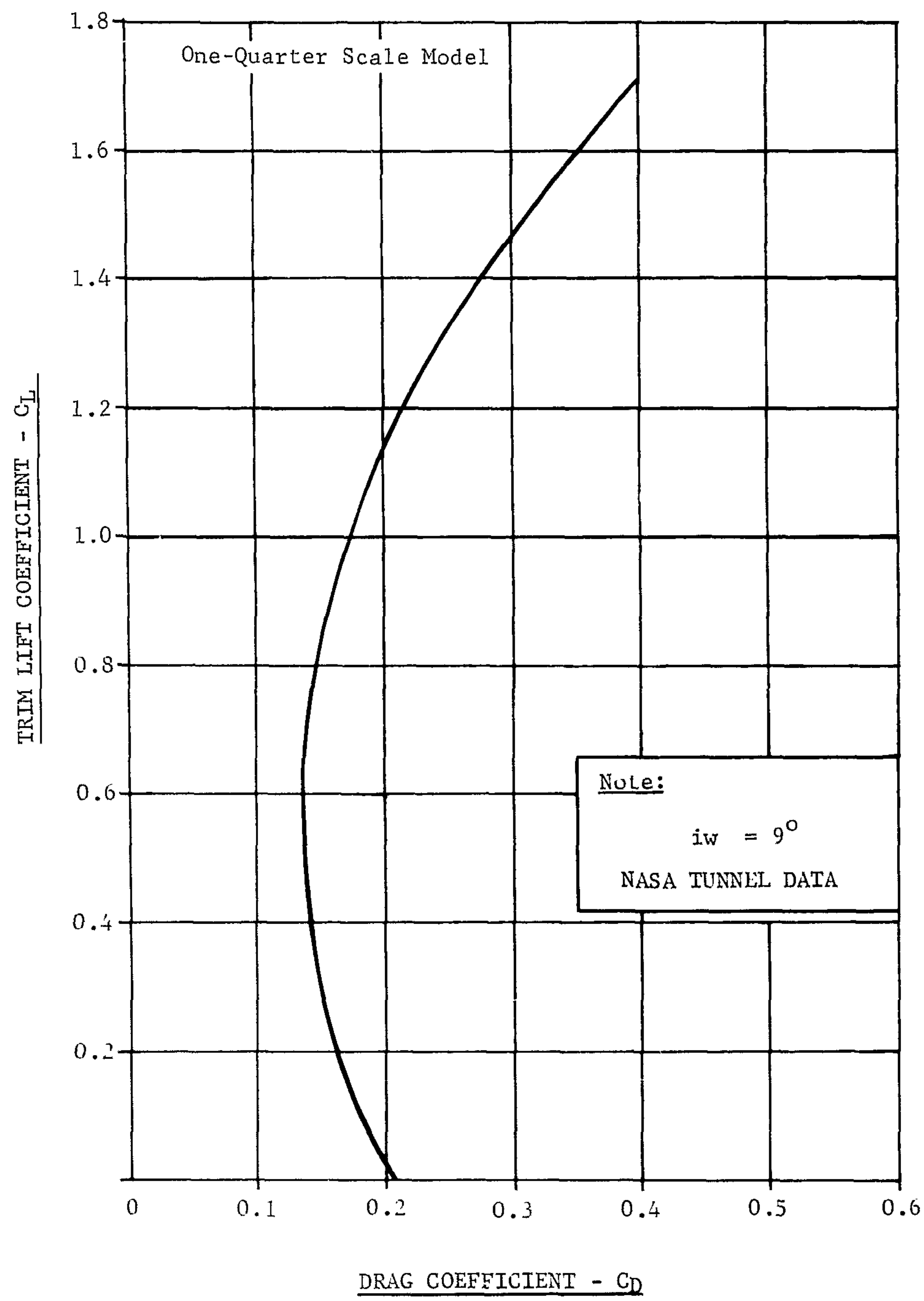
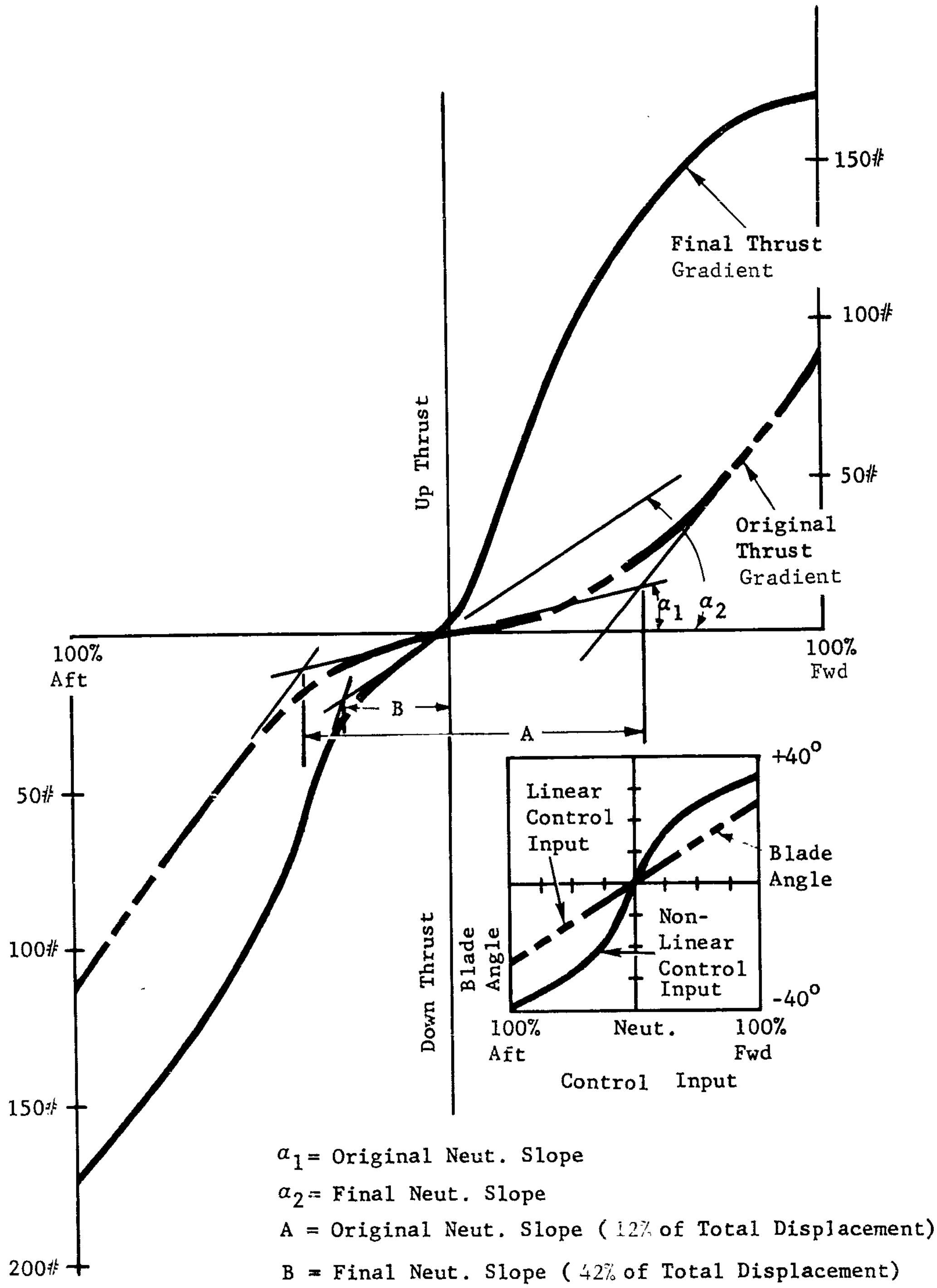


FIGURE 40. LIFT COEFFICIENT VS DRAG COEFFICIENT (POWER OFF)

VERTOL 76 PITCH CONTROL
COMPARISON OF CONTROL INPUT VS
THRUST OUTPUT (ORIG. & FINAL SYSTEMS)



α_1 = Original Neut. Slope

α_2 = Final Neut. Slope

A = Original Neut. Slope (12% of Total Displacement)

B = Final Neut. Slope (42% of Total Displacement)

FIGURE 41. THRUST VS CONTROL DISPLACEMENT FOR PITCH FAN

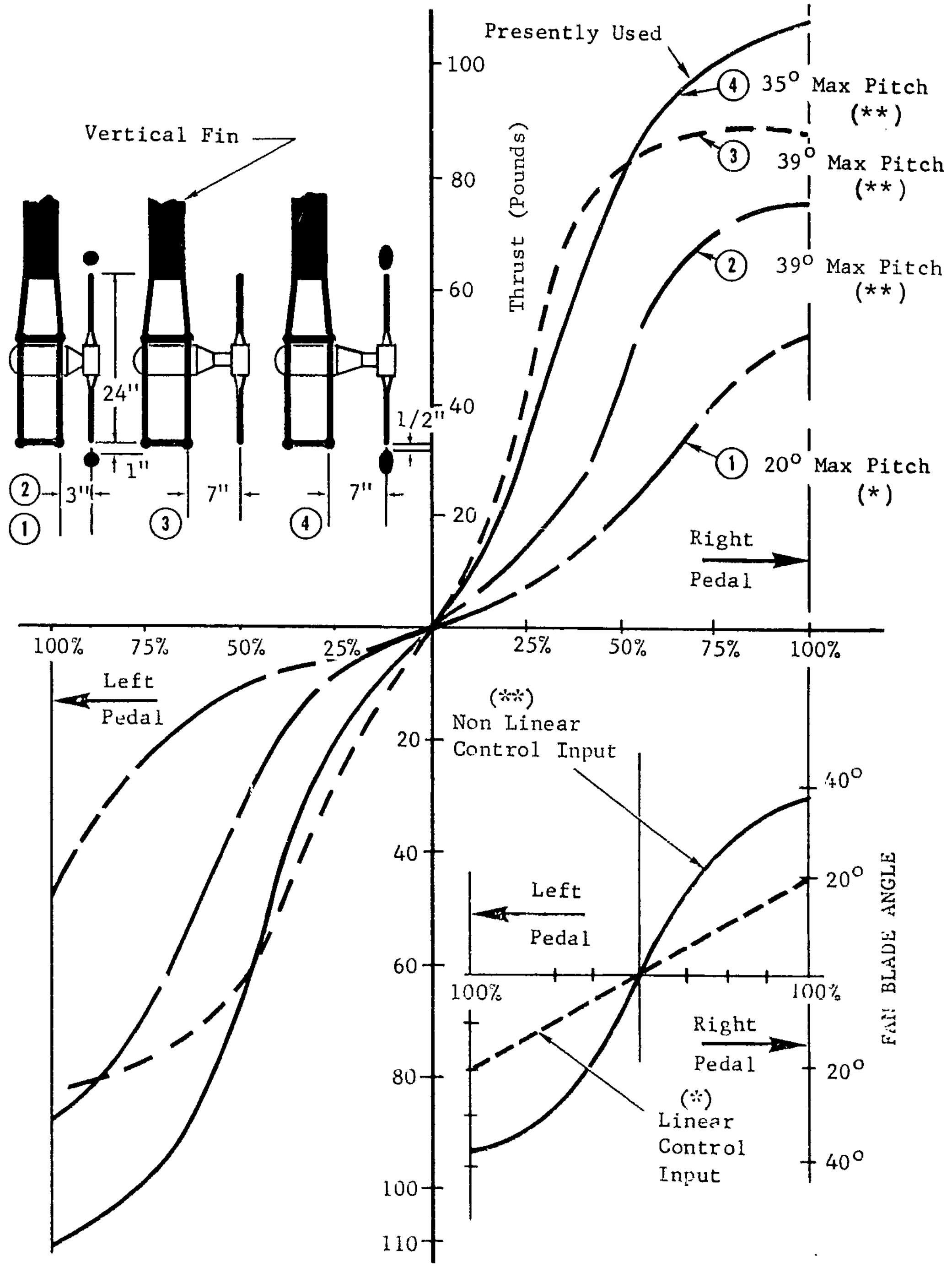


FIGURE 42. THRUST VS PEDAL DISPLACEMENT FOR VARIOUS YAW FAN CONFIGURATIONS

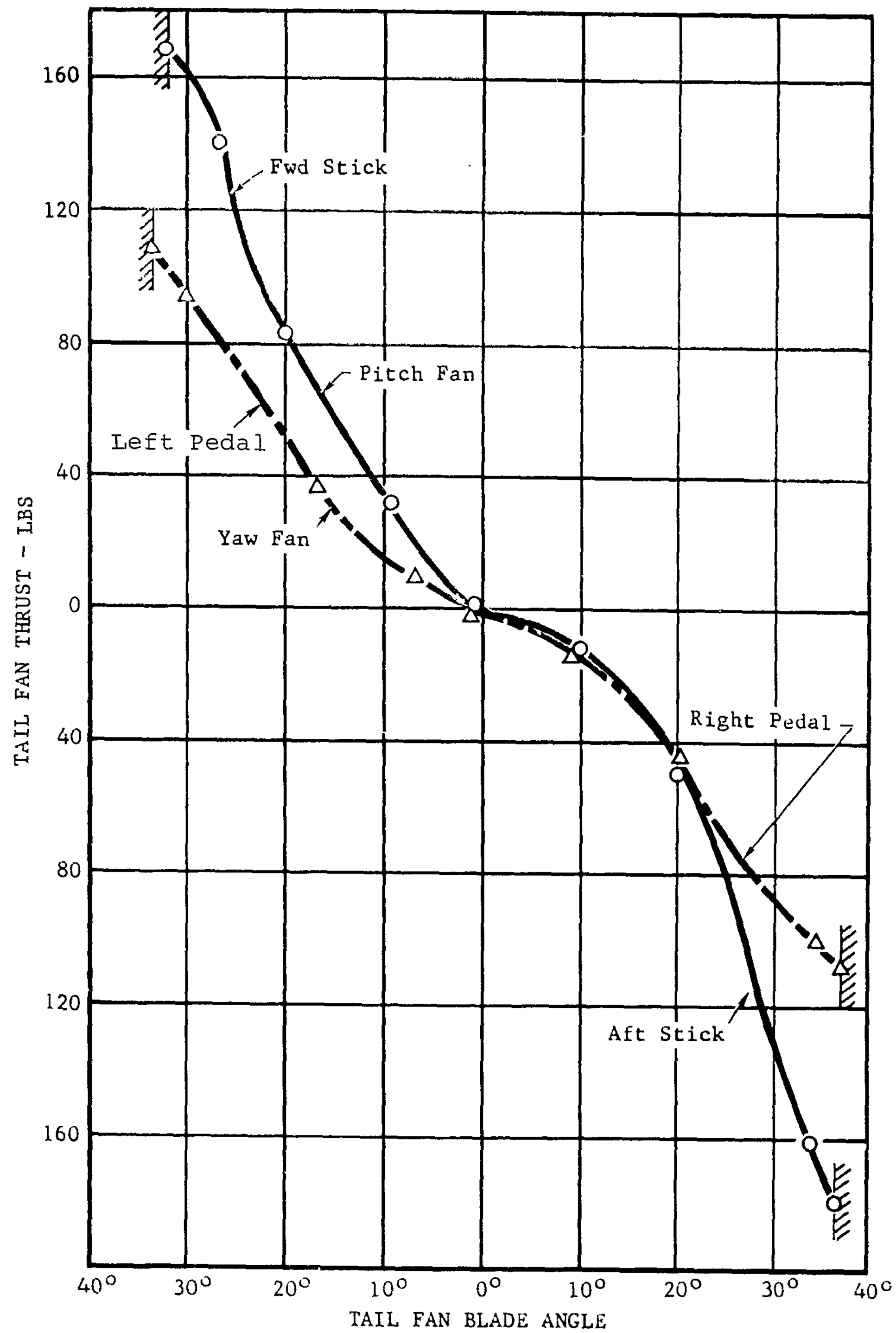


FIGURE 43. A COMPARISON OF THRUST VS BLADE ANGLE FOR PITCH AND YAW FANS

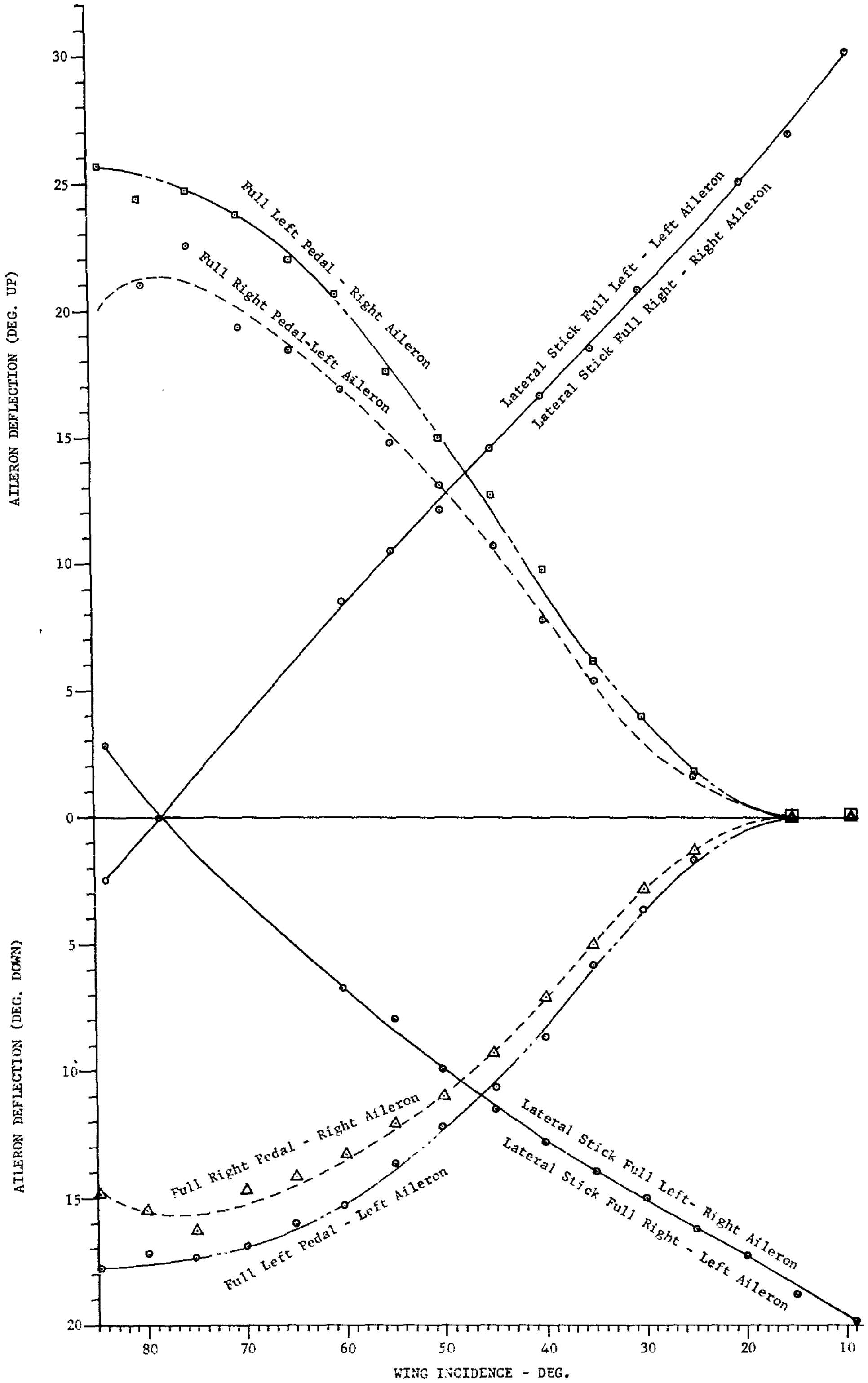


FIGURE 44. INPUTS TO AILERONS FROM LATERAL STICK AND PITCH VS WING INCIDENCE (FULL SPAN FLAPS AND AILERONS CONFIGURATION)

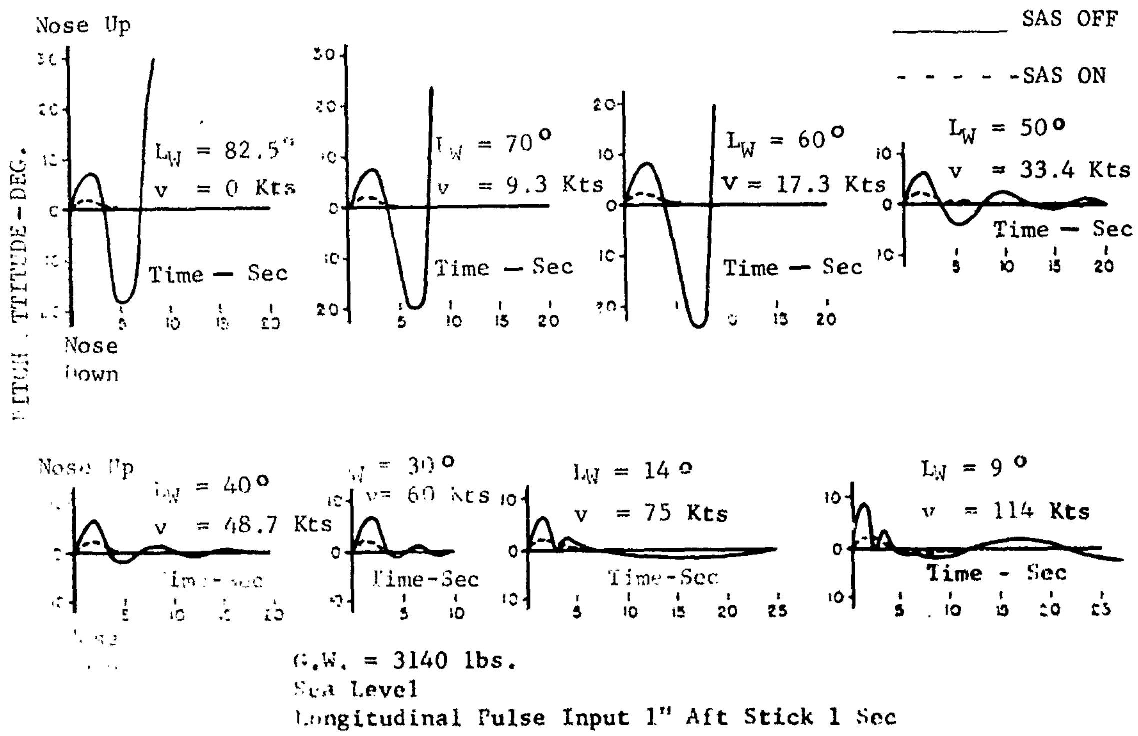


FIGURE 45. ANALOG STUDIES OF LONGITUDINAL DYNAMIC STABILITY IN HOVERING AND CONVERSION WITH AND WITHOUT THE STABILITY AUGMENTATION SYSTEM

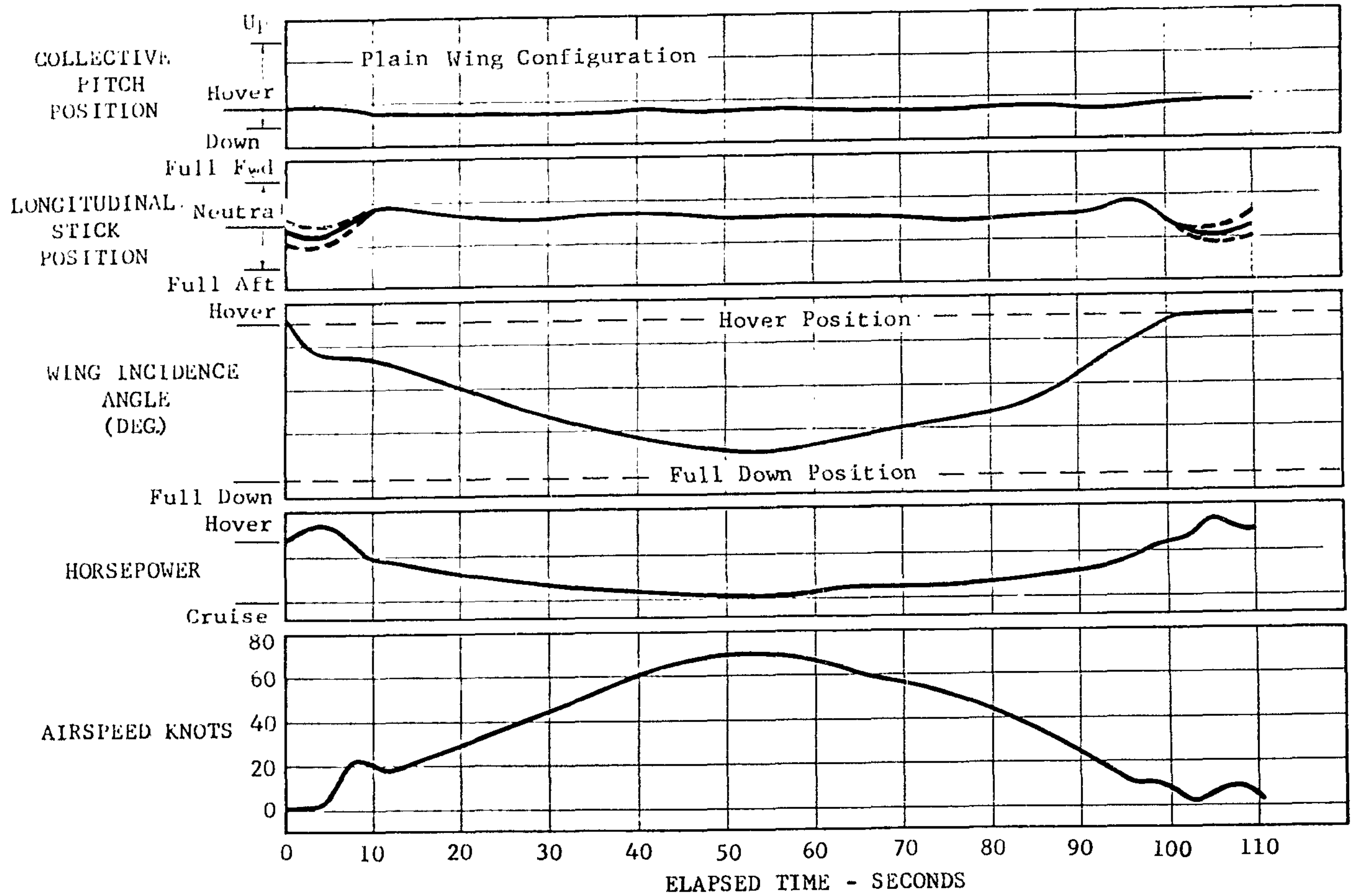


FIGURE 46. TYPICAL TIME HISTORY OF CONVERSION FROM HOVER TO AIRPLANE TO HOVER

FIGURE 47. VZ-2 IN TIEDOWN TEST RIG.

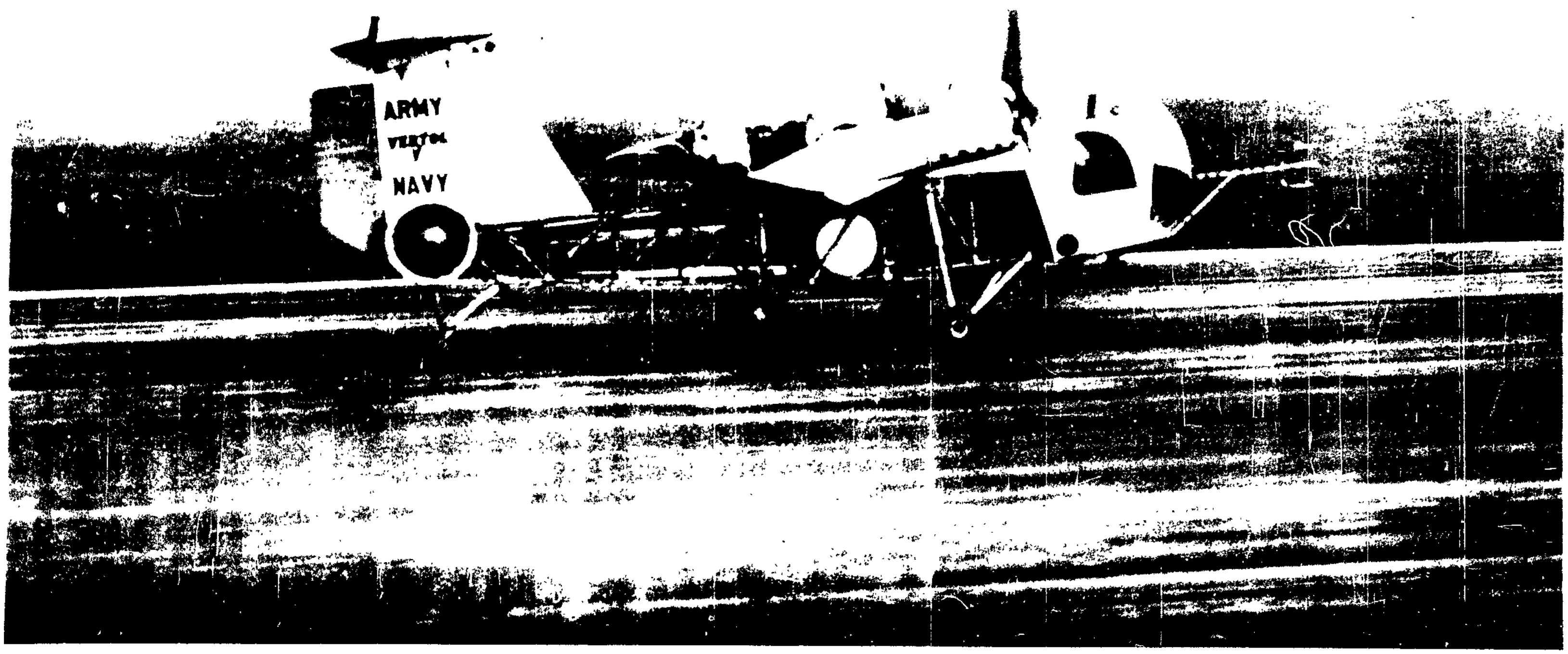


FIGURE 48. VZ-2 IN HIGH SPEED TAXI

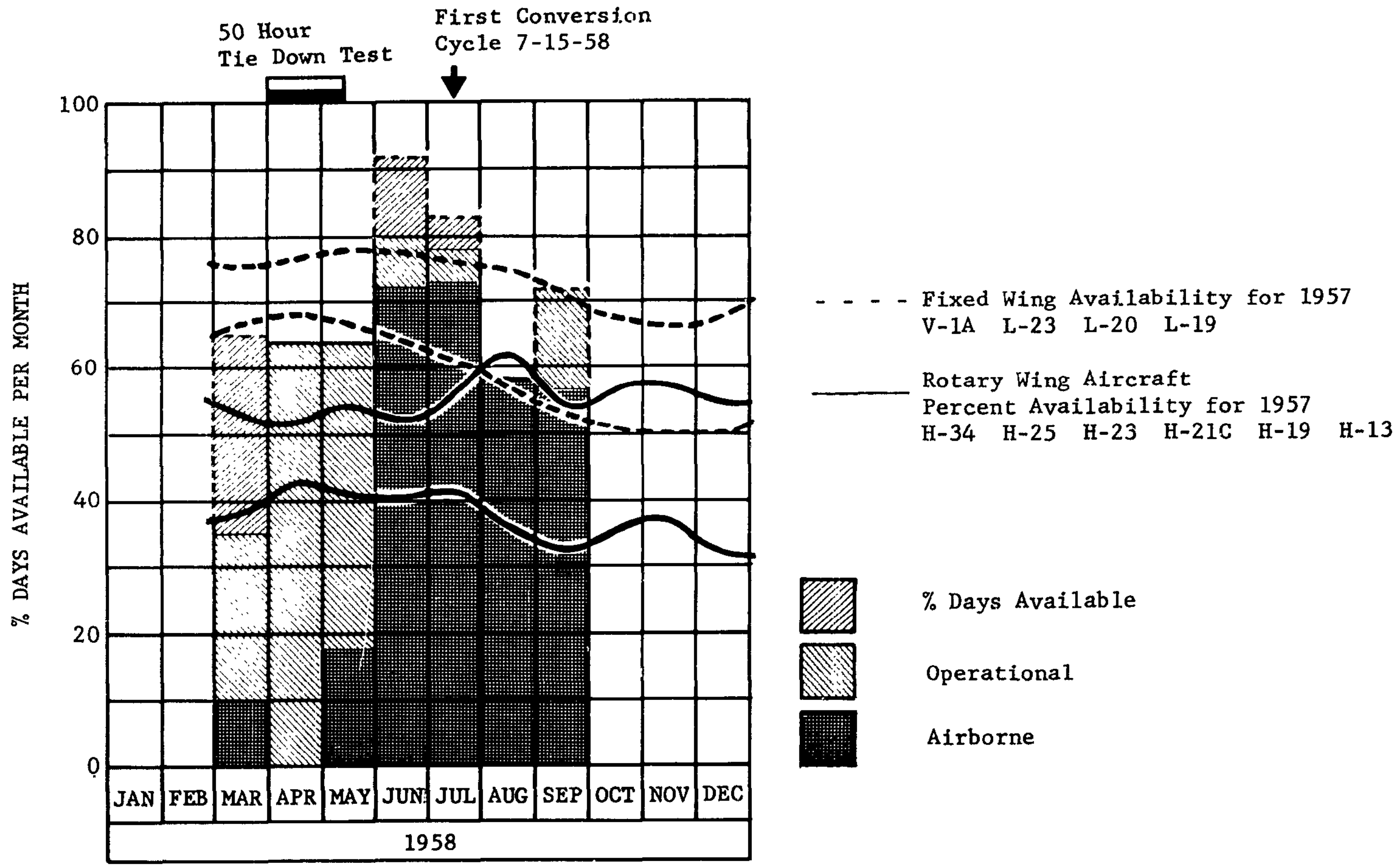


FIGURE 49. VZ-2 VTOL/STOL PERCENT AVAILABILITY

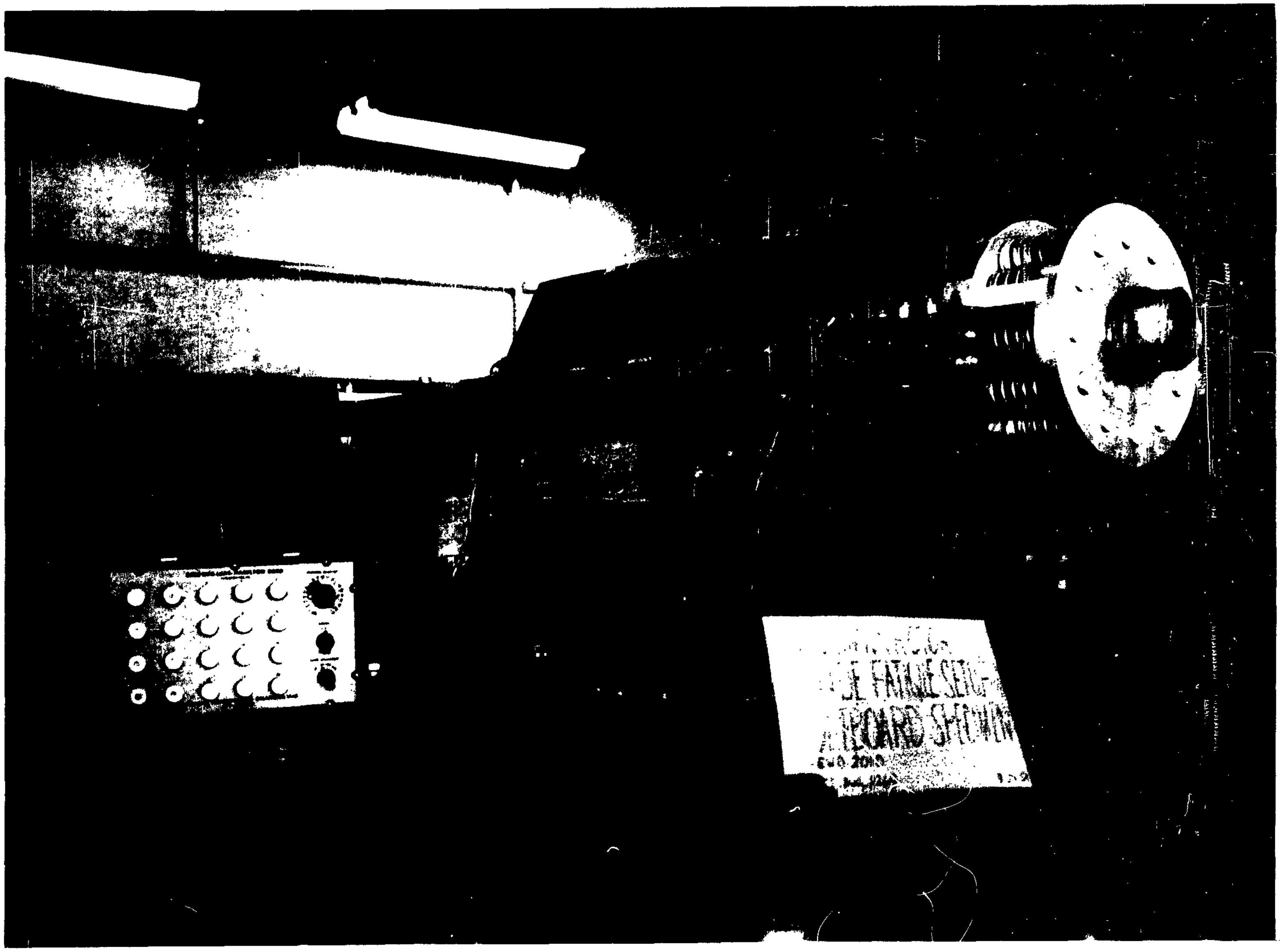


FIGURE 50. FATIGUE TEST SET-UP OF MAIN ROTOR BLADE TEST SPECIMEN

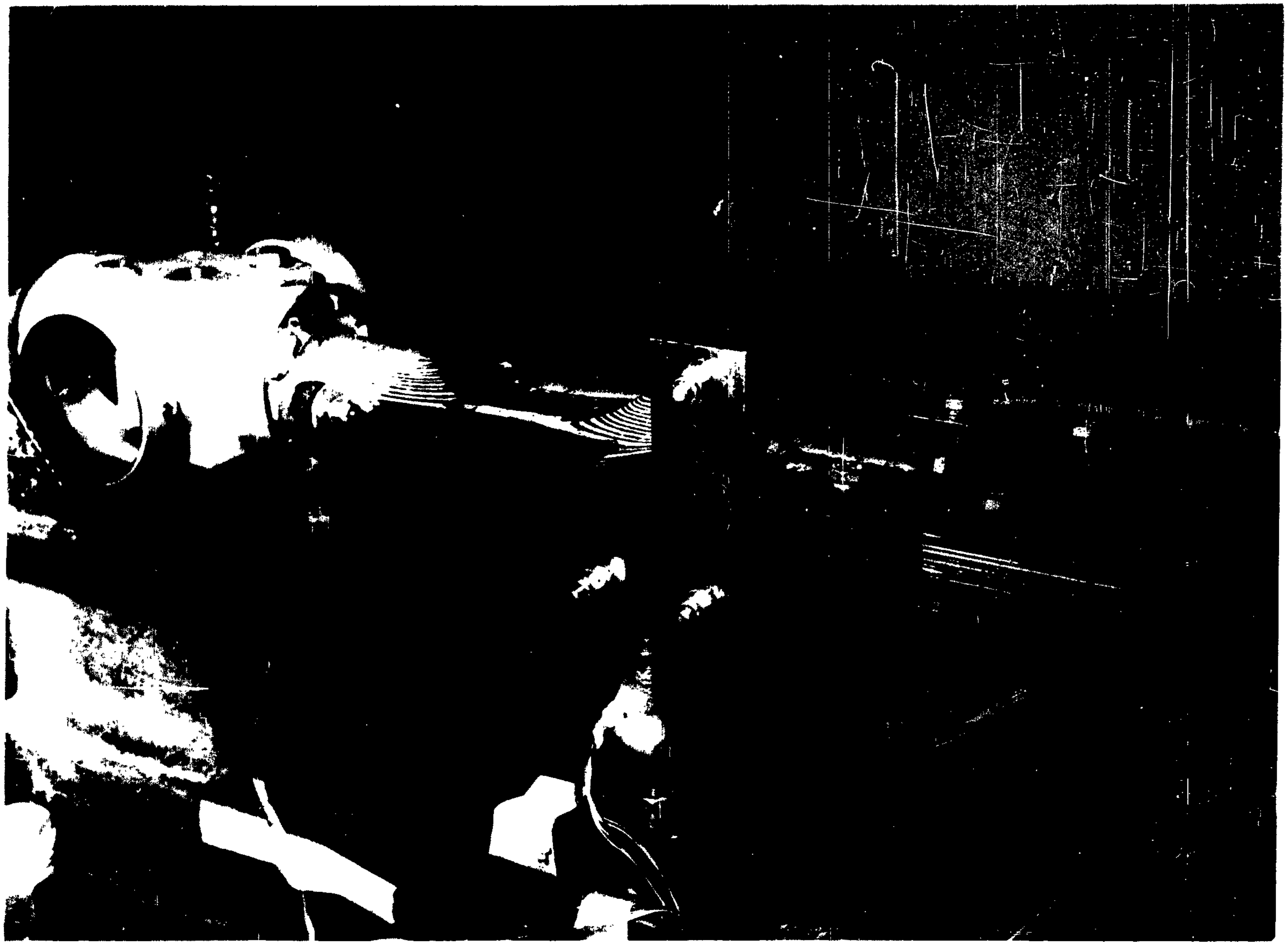


FIGURE 51. FATIGUE TEST SET-UP OF TAIL FAN-BLADE TEST SPECIMEN

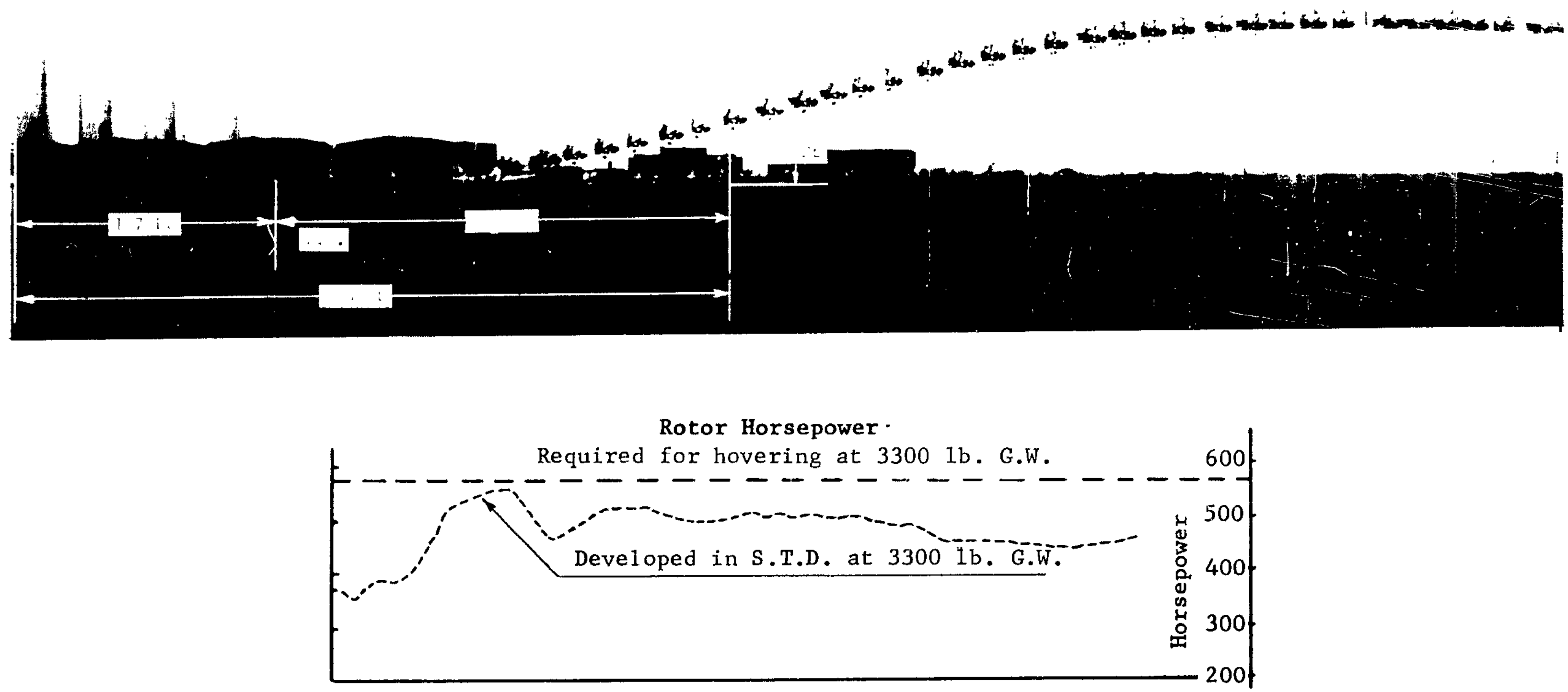


FIGURE 52. EXAMPLE OF AN ACTUAL SHORT TAKEOFF AT A POWER SETTING LOWER THAN IN HOVERING.

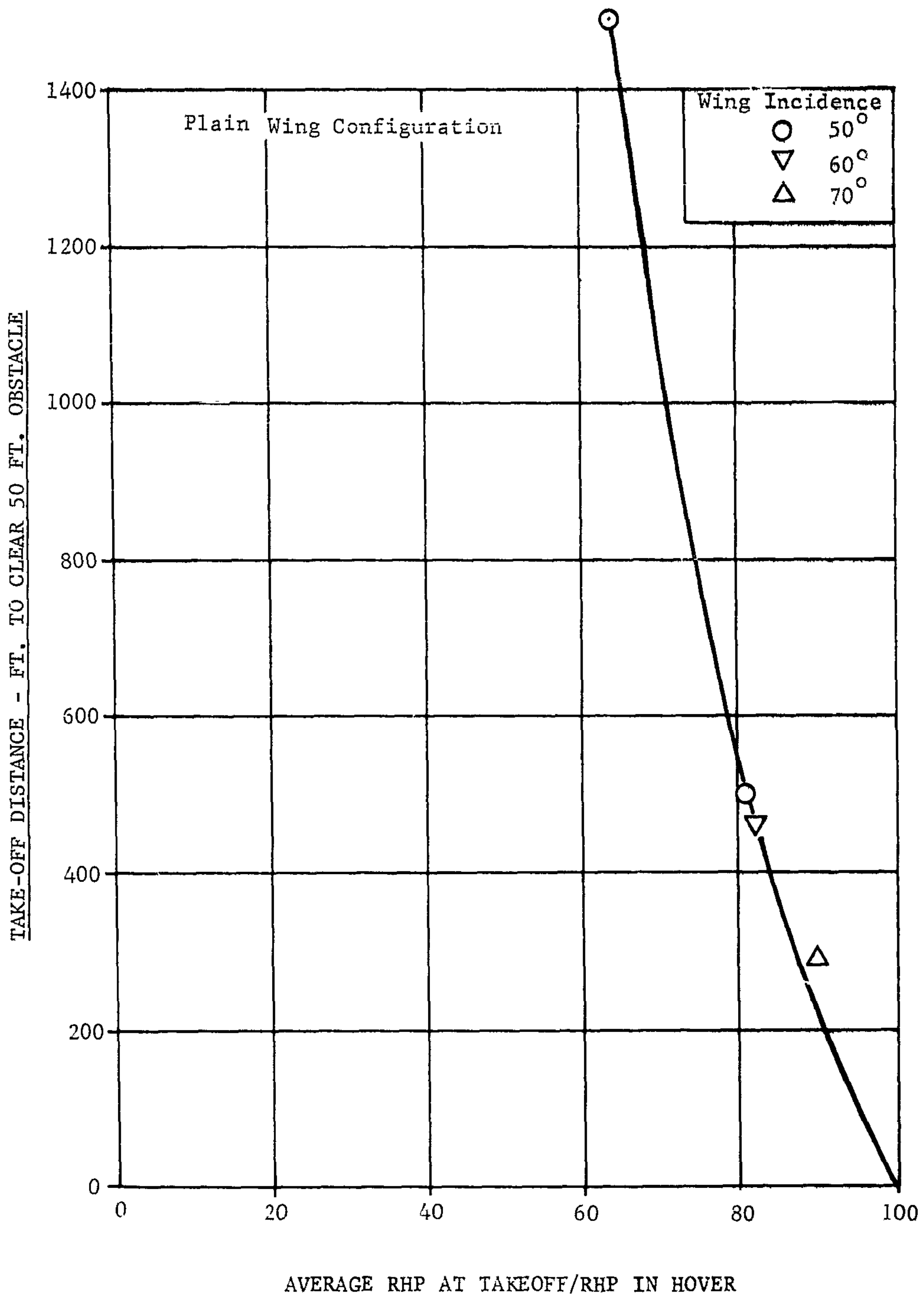


FIGURE 53. TAKEOFF DISTANCE OVER 50 FT OBSTACLE VS PERCENT OF HOVERING-ROTOR HORSEPOWER (FLIGHT TEST DATA)

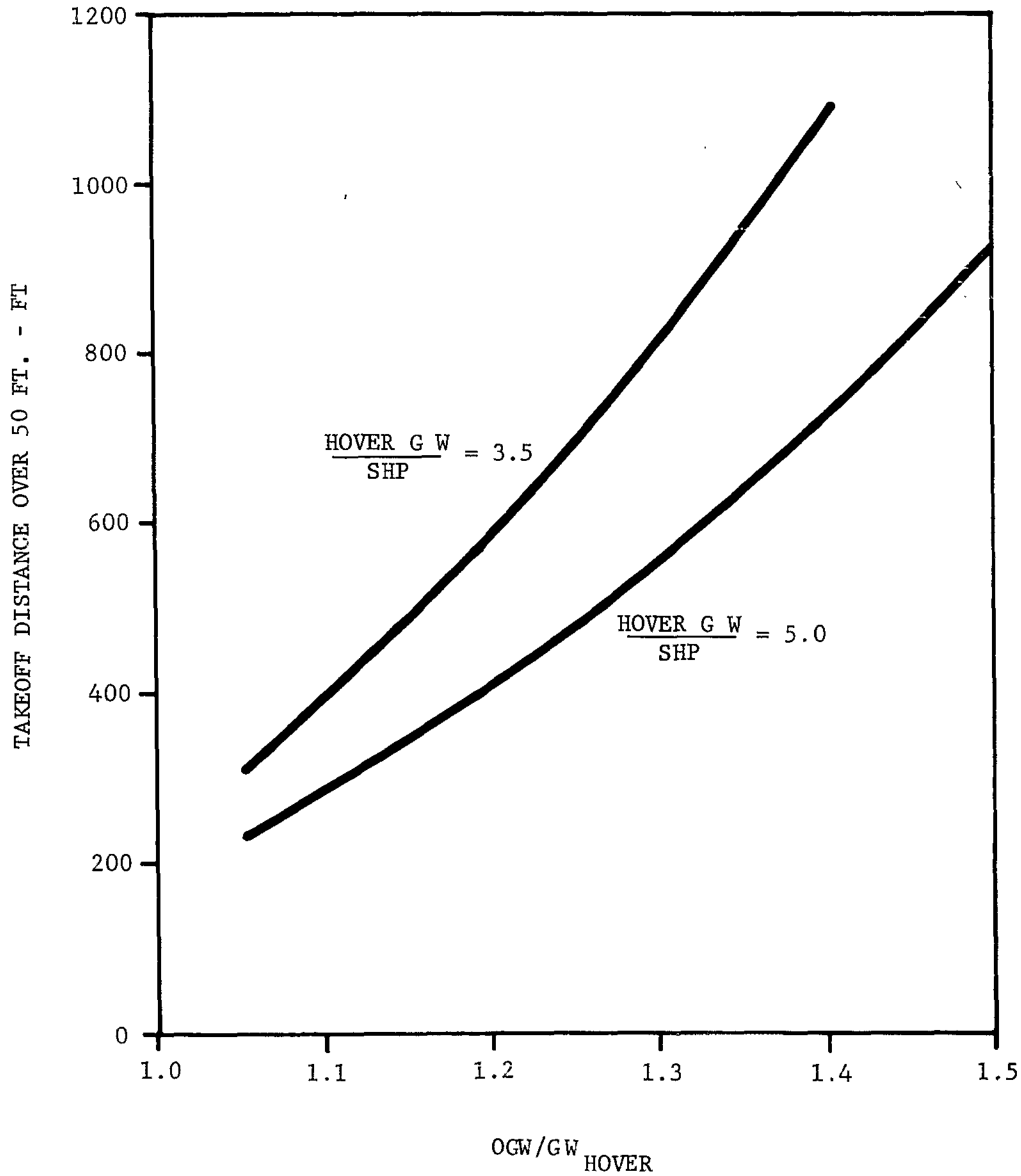
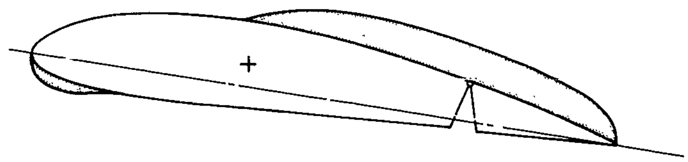


FIGURE 54. STOL TAKEOFF DISTANCE OVER 50 FT. VS. OVERLOAD GROSS WEIGHT RATIO

Ordinates of Droop Snoot - % Chord

X	Y
1.25	-4.0
2.5	-4.895
5.0	-5.526
7.5	-5.570
10	-5.421
12.5	-5.123
15	-4.763
17.5	-4.474
20	-4.263
22.5	-4.105
25	-3.979



BASIC AIRFOIL SECTION NACA 4415

Ordinates of Fence - % Chord

X	Y
35	11.34
40	15.11
50	18.42
60	18.95
70	17.89
80	16.08
90	13.34
95	10.79
97.5	8.82
100	.16

l.e. radius = 4.961%
center @ x = 4.721%, Y = - .456%
aileron hinge point = 76%

FIGURE 55. WING FENCE AND DROOP SNOOT

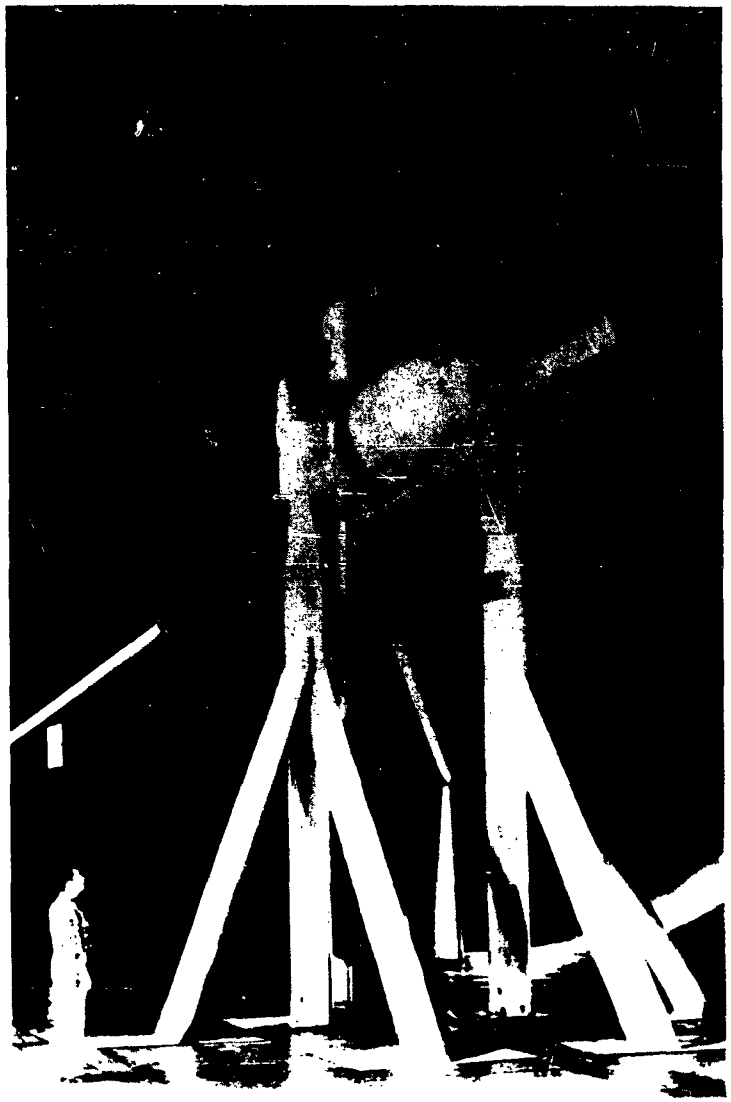


FIGURE 56. PROPELLER AND TEST STAND INSTALLED IN AMES 40 FT. BY 80 FT. WIND TUNNEL

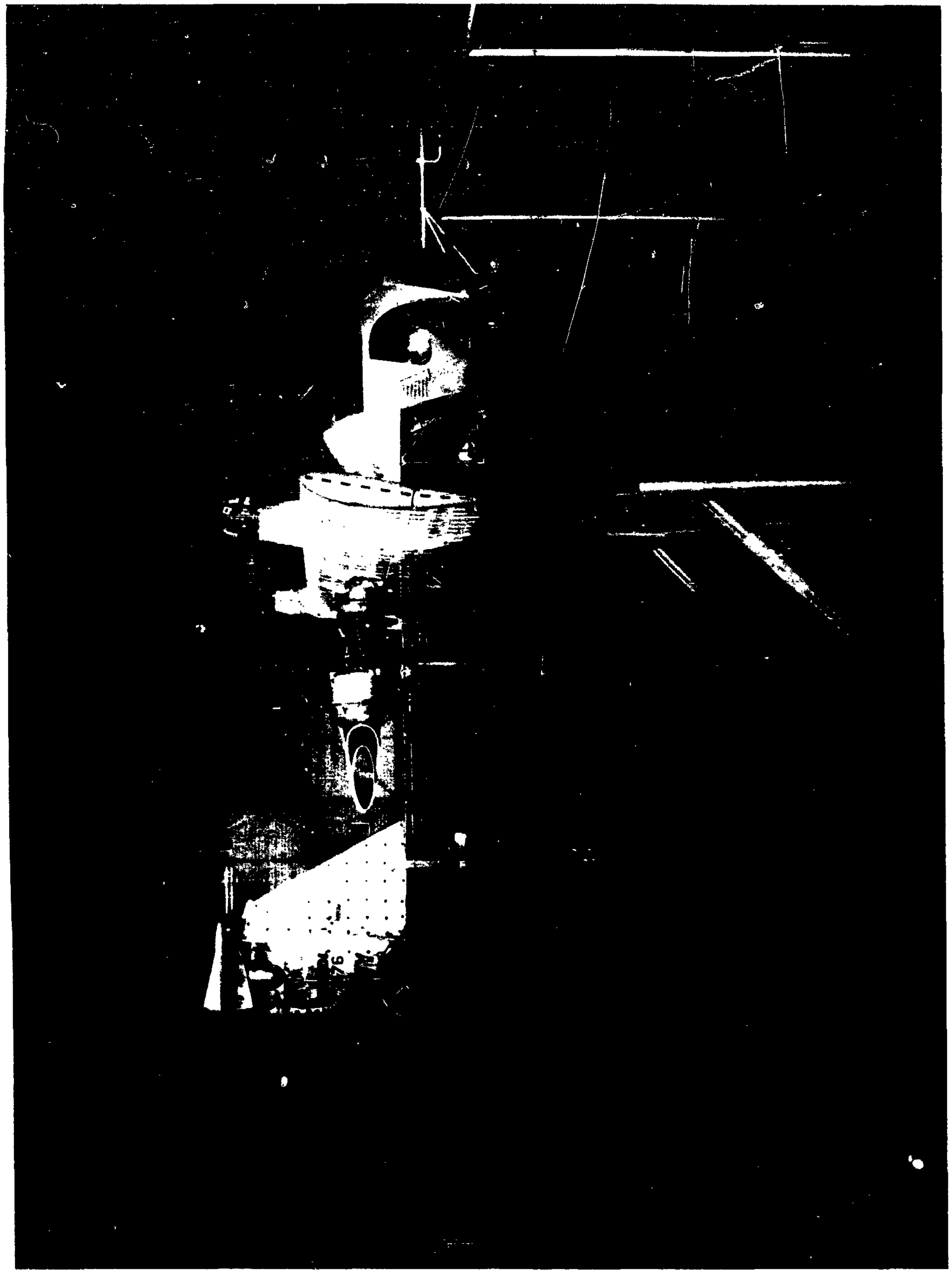


FIGURE 57. FULL SCALE WIND TUNNEL TEST OF THE VZ-2 RESEARCH AIRCRAFT

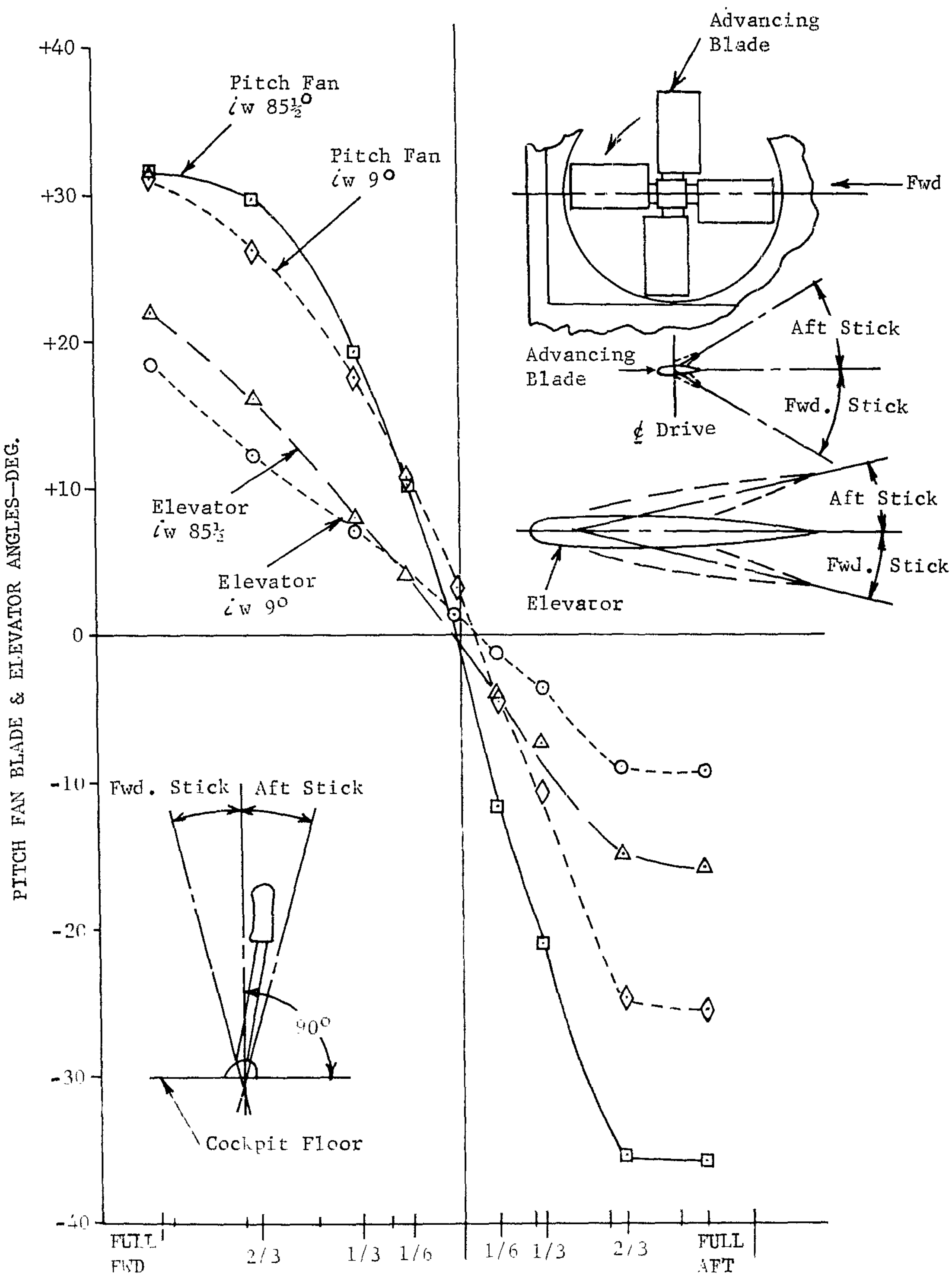


FIGURE 58. ELEVATOR DEFLECTION AND HORIZONTAL FAN PITCH VS CONTROL DISPLACEMENT FOR WING UP AND WING DOWN POSITIONS (PRESENT CONFIGURATION)

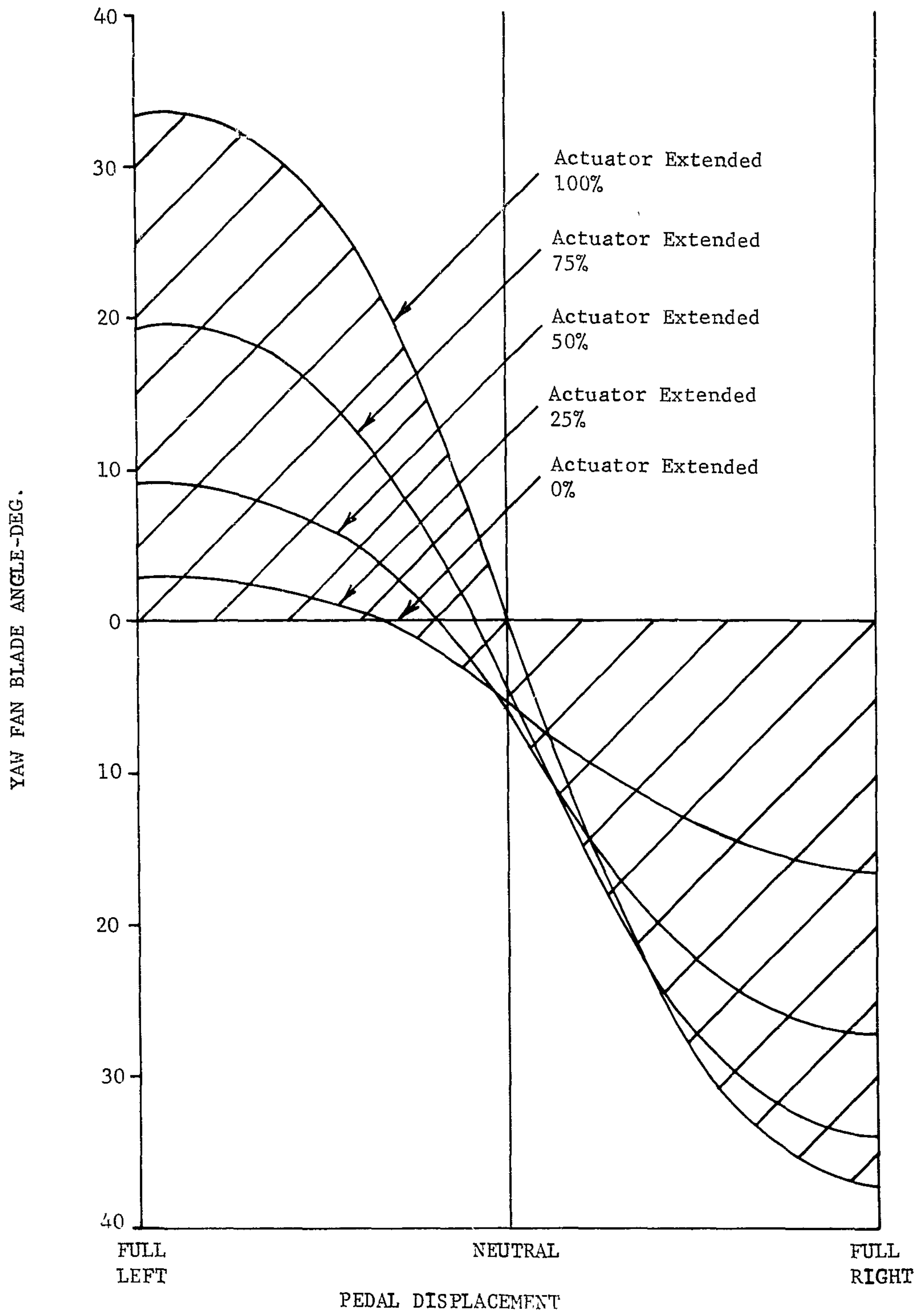


FIGURE 59. YAW FAN BLADE ANGLE ENVELOPE (YAW FAN DEPHASING MECHANISM)

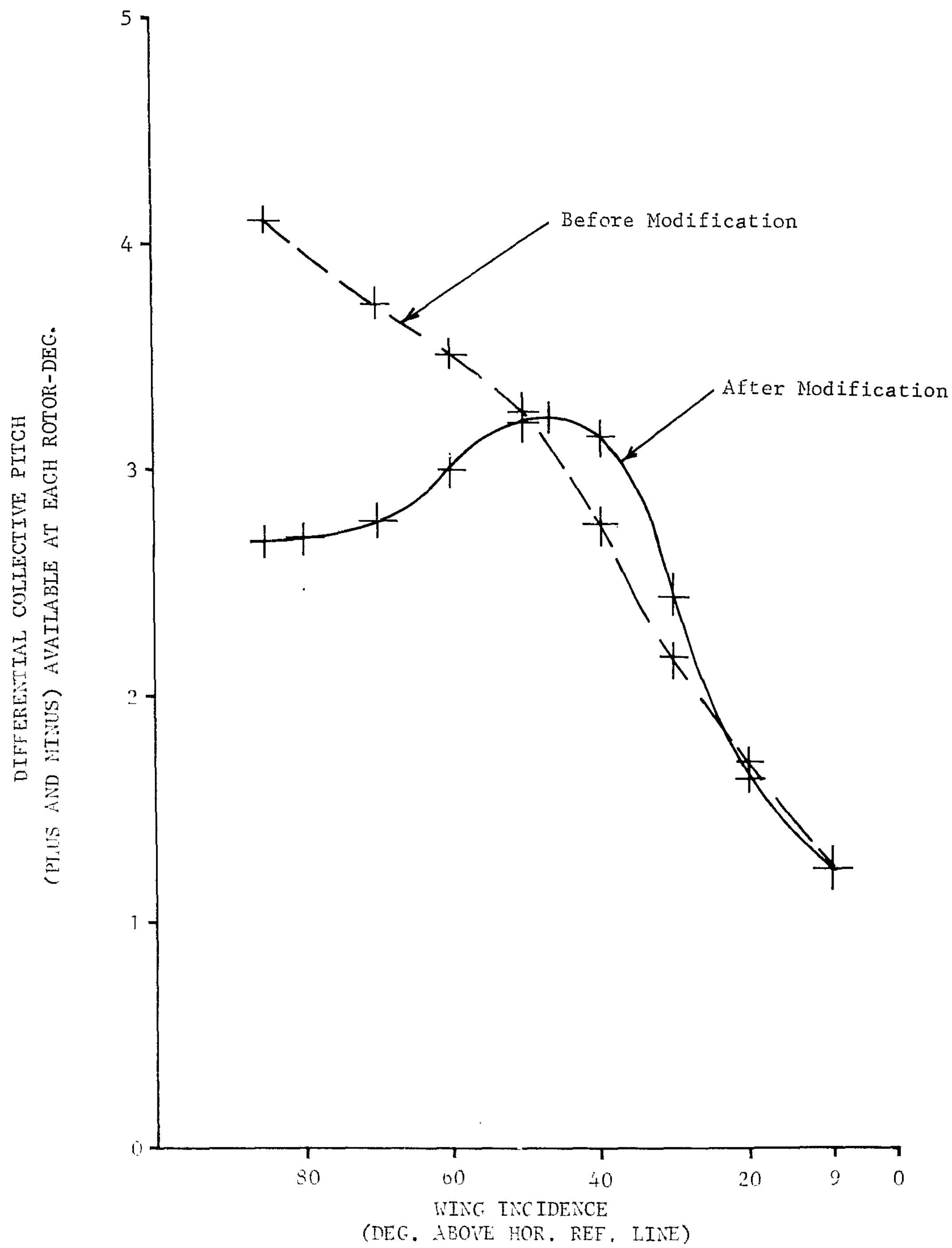


FIGURE 60. DIFFERENTIAL COLLECTIVE PITCH VS WING INCIDENCE (CONVERSION)

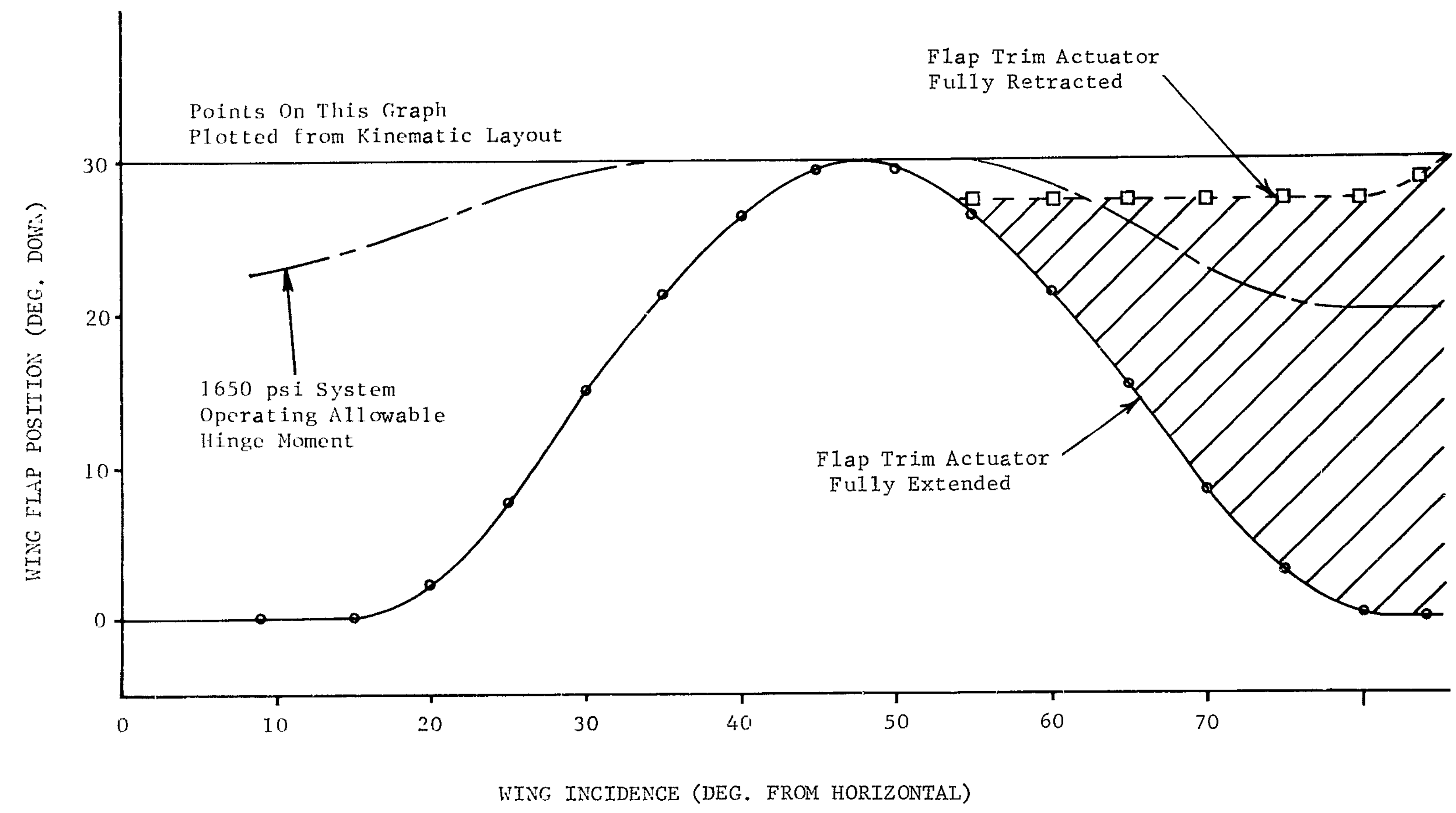


FIGURE 61. WING FLAP POSITION VS WING INCIDENCE

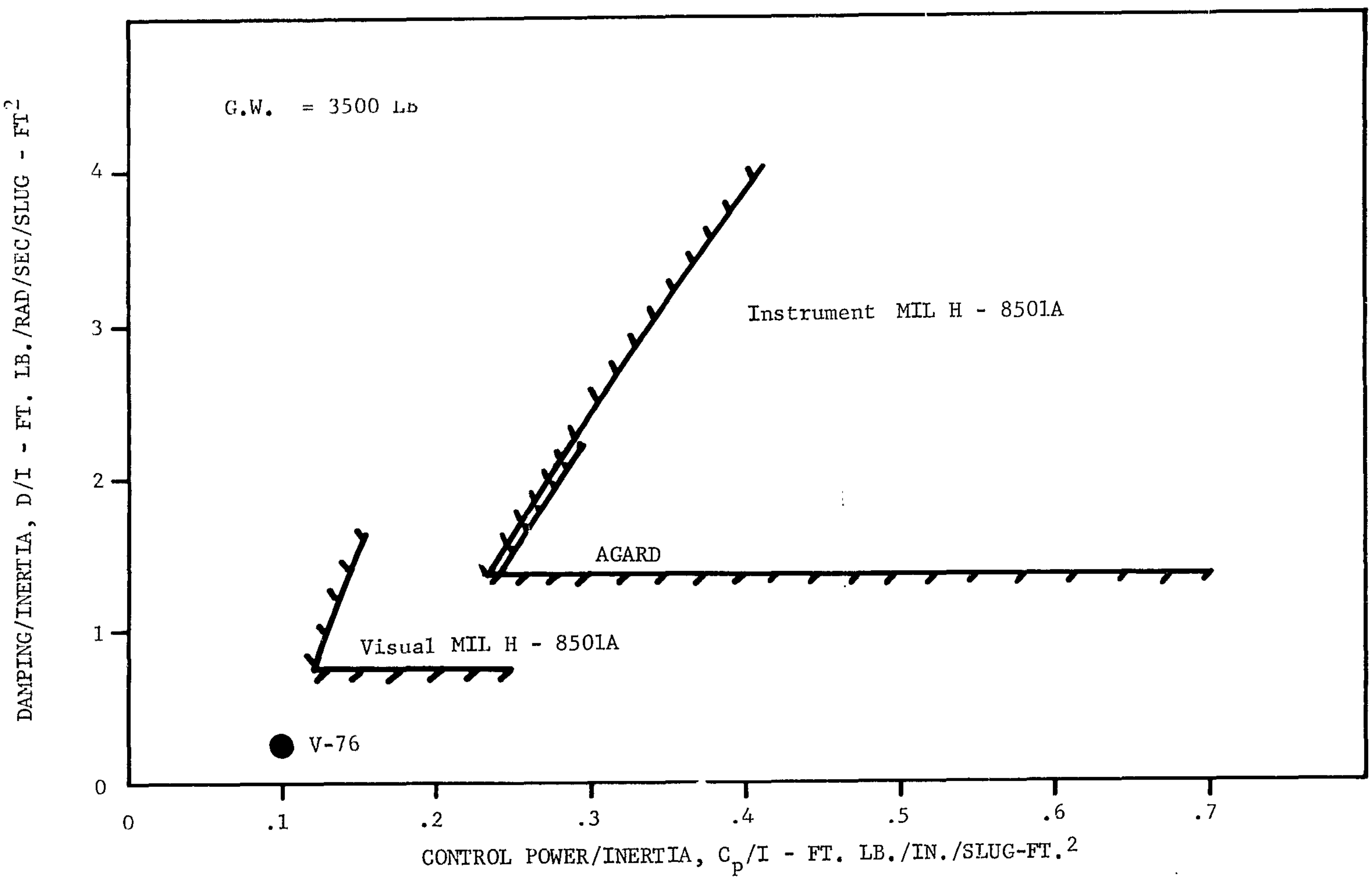


FIGURE 62. PITCH CONTROL AND DAMPING REQUIREMENTS-HOVER

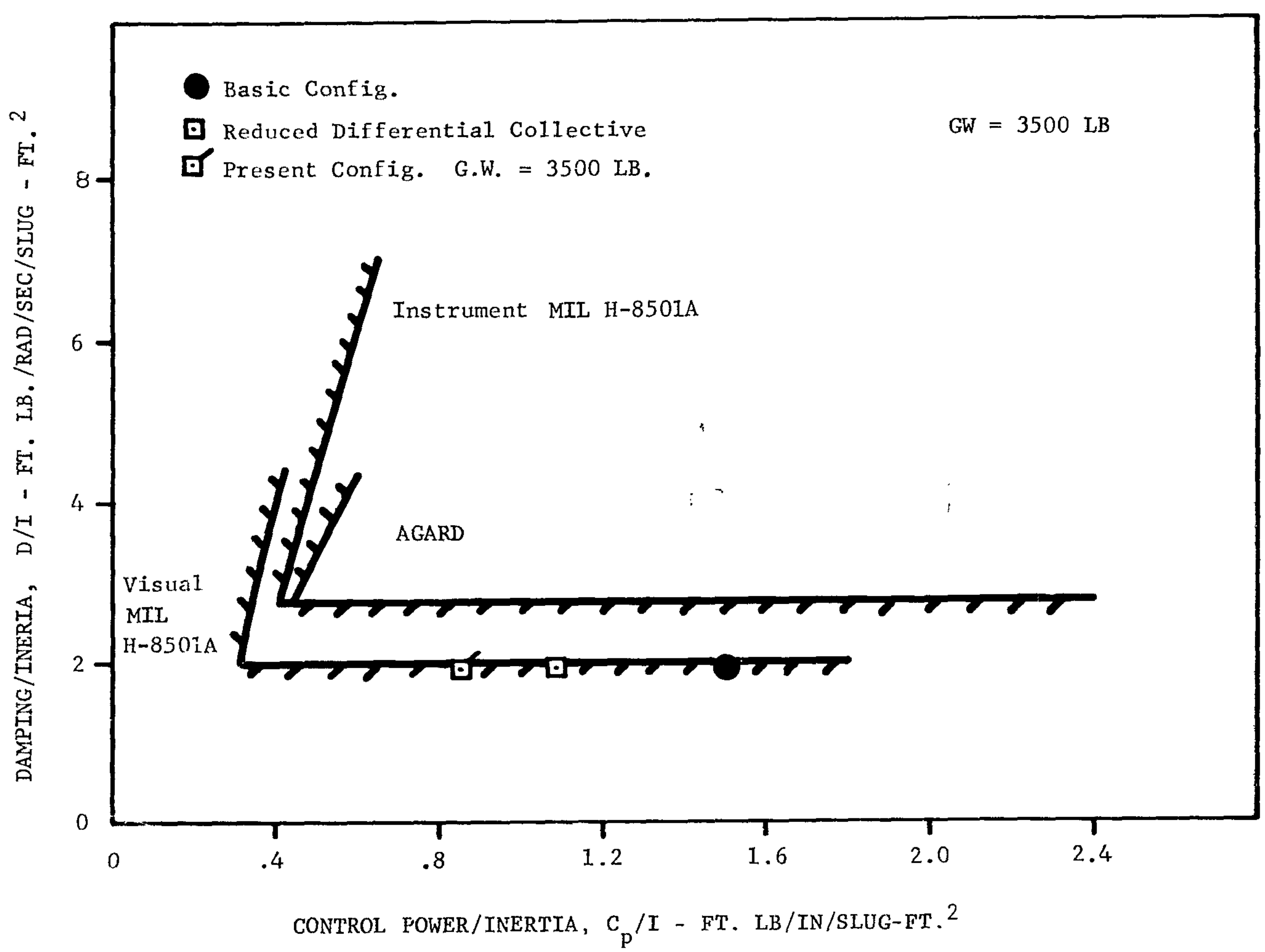


FIGURE 63. ROLL CONTROL AND DAMPING REQUIREMENTS - HOVER

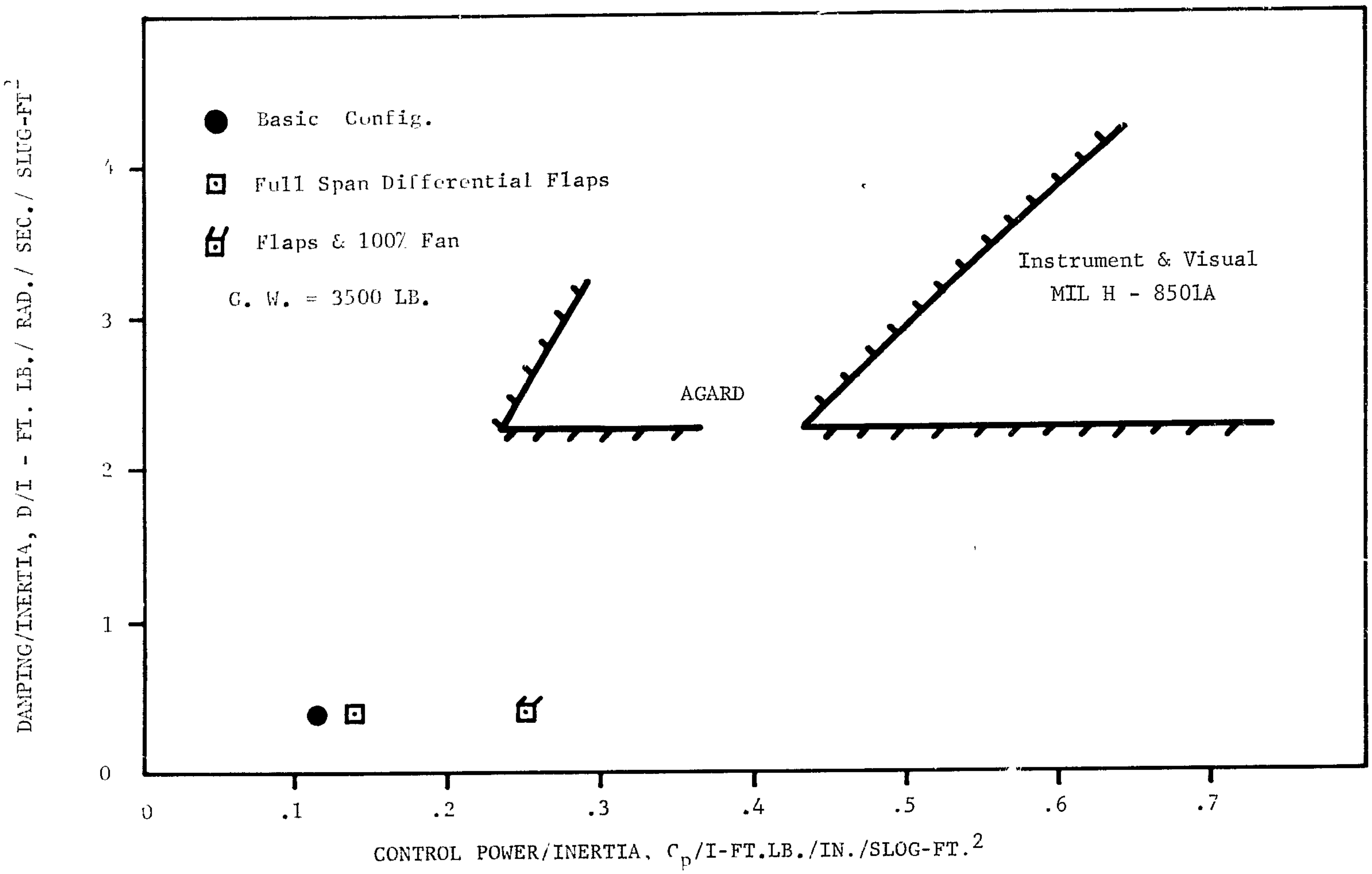


FIGURE 64. YAW CONTROL AND DAMPING REQUIREMENTS - HOVER

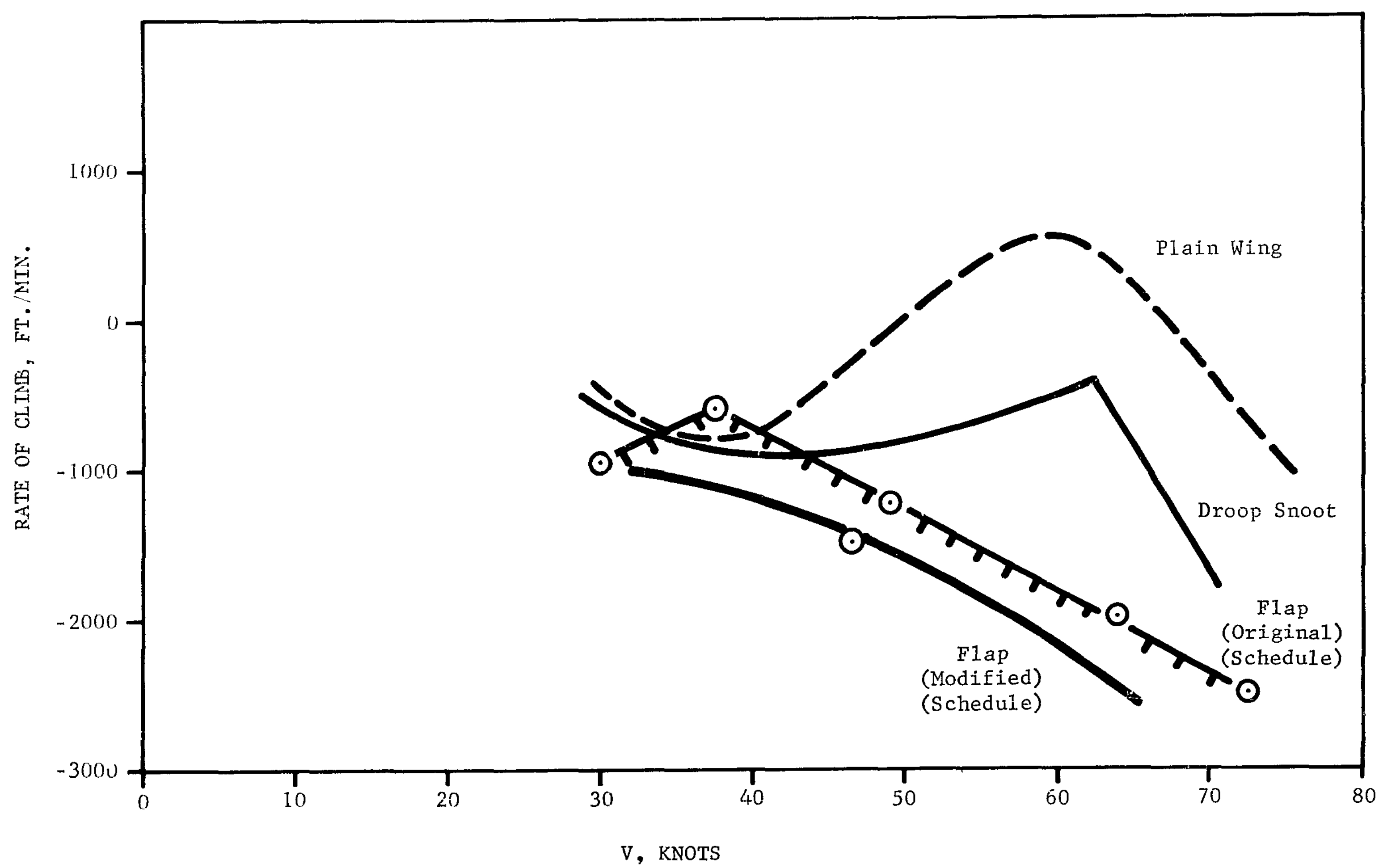


FIGURE 65. RATE OF DESCENT LIMITS VERSUS SPEED

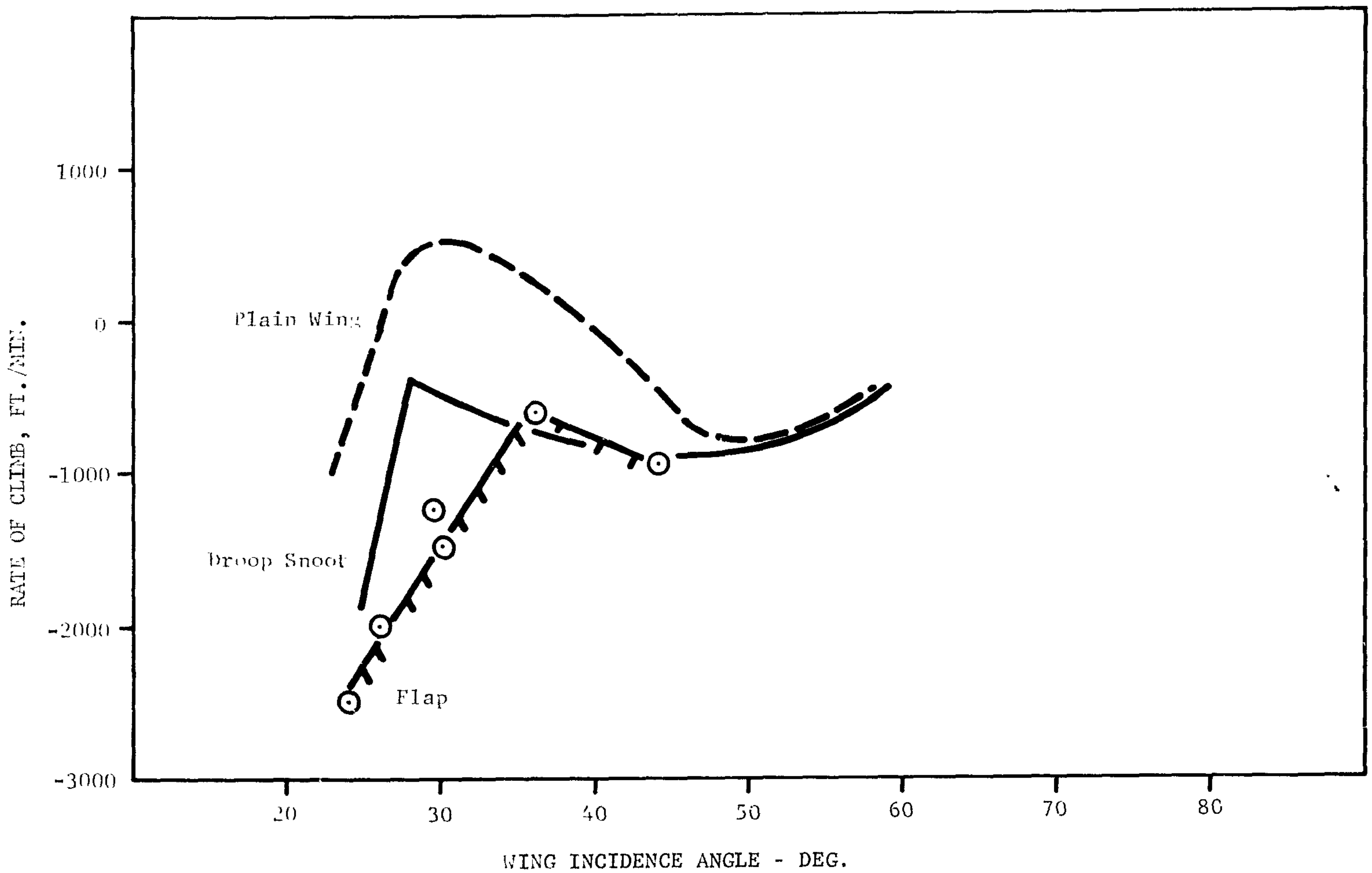


FIGURE 66. RATE OF DESCENT LIMITS VERSUS WING INCIDENCE

REF. NASA CONFERENCE V/STOL, NOV. 1960, P. 165

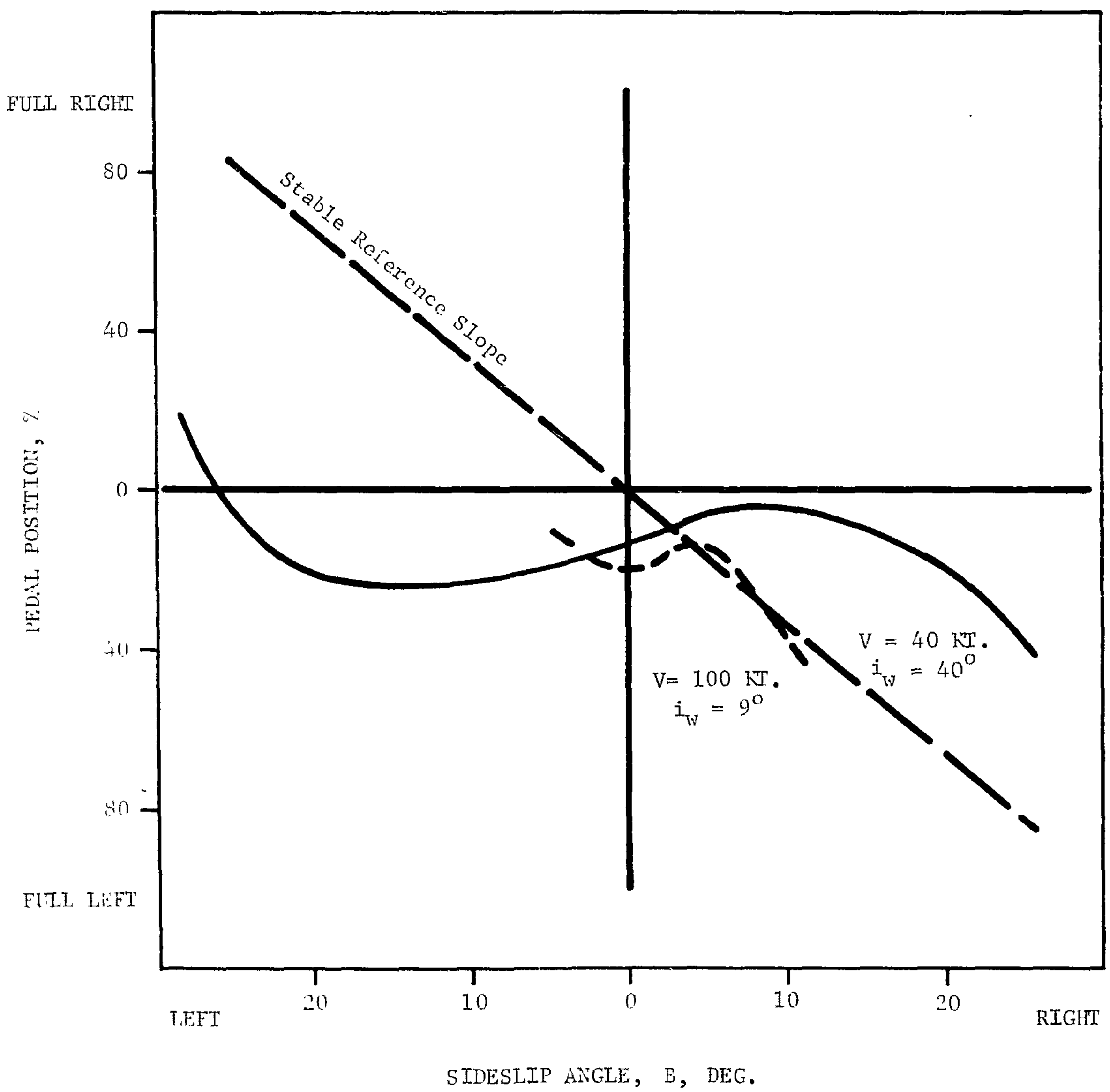


FIGURE 67. STATIC DIRECTIONAL STABILITY

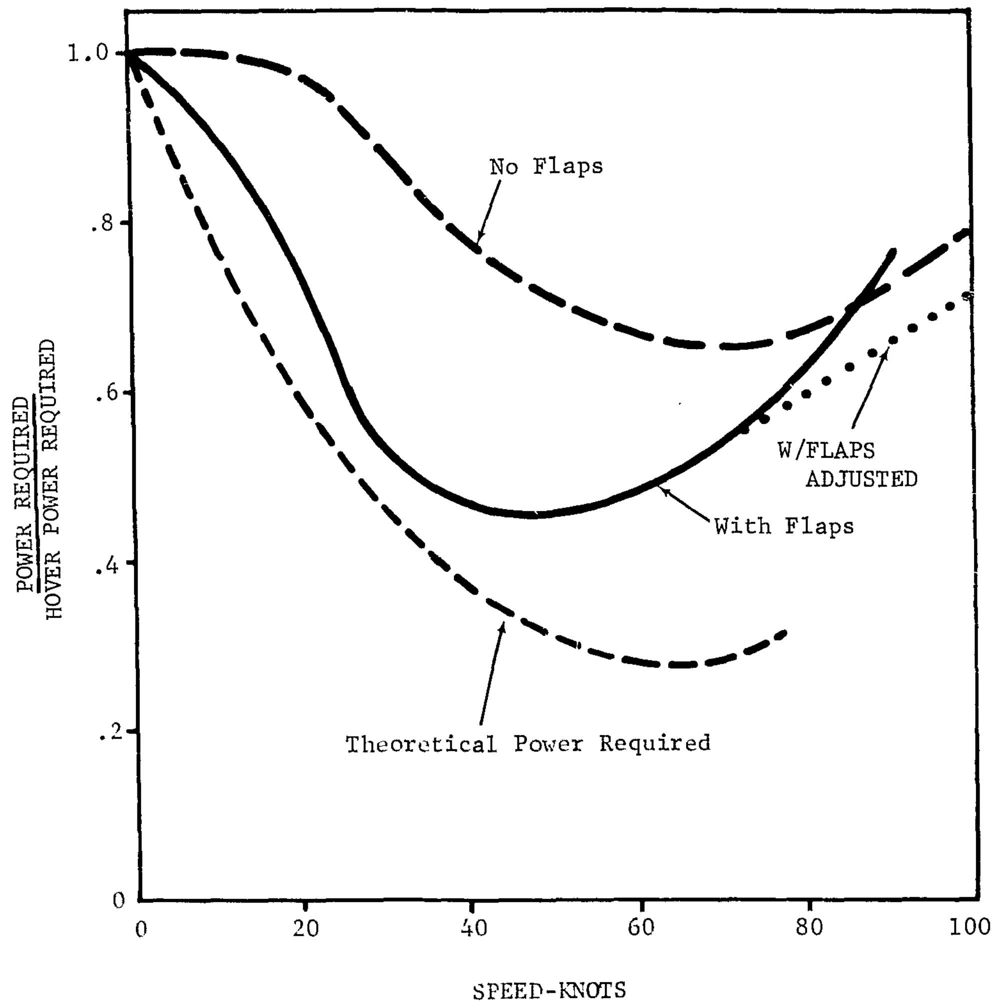


FIGURE 68. POWER RATIO VERSUS SPEED

(Continued from inside front cover)

8	OCTOBER	1959	COMPLETED ALTITUDE FLIGHT PROGRAM AT EDWARDS
20	NOVEMBER	1959	FLIGHT PROGRAM STARTED BY NASA AT LANGLEY FIELD
18	JULY	1960	DROOP SNOOT INSTALLED AND TESTED ON WING AT LANGLEY FIELD
5	JANUARY	1961	COMPLETED FLIGHT PROGRAM AT LANGLEY FIELD
9	FEBRUARY	1961	FULL-SCALE WIND TUNNEL TEST OF VZ-2 BY NASA AT LANGLEY FIELD
22	MARCH	1961	VZ-2 RETURNED TO VERTOL DIVISION FOR INSTALLATION OF FULL SPAN FLAP AND AILERONS AND UP-GRADING OF TRANSMISSION
7	NOVEMBER	1961	STARTED 50 HOUR TIEDOWN TEST OF MODIFIED CONFIGURATION
16	NOVEMBER	1961	COMPLETED 50 HOUR TIEDOWN TEST
20	AUGUST	1962	STARTED FLIGHT PROGRAM AT VERTOL DIVISION
7	SEPTEMBER	1962	COMPLETED FLIGHT PROGRAM AT VERTOL DIVISION
18	SEPTEMBER	1962	EXTENDED FLIGHT PROGRAM STARTED BY NASA AT LANGLEY FIELD
17	JANUARY	1963	COMPLETED HOVER FLIGHTS AT LANGLEY FIELD
26	AUGUST	1963	NASA CONTINUING VZ-2 FLIGHT