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U. S. ARMY
TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

TRECOM TECHNICAL REPORT 63-42

SUMMARY REPORT
HIGH-PERFORMANCE-HELICOPTER PROGRAM
PHASE I

Task 1D121401A14301
(Formerly Task 9R38-13-014-01)
Contract DA 44-177-TC-711

September 1963

prepared by:

BELL HELICOPTER COMPANY
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HEADQUARTERS
U S ARMY TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

This report has been reviewed by the U. S. Army Transportation Research Command and is considered to be technically sound. The results substantiated that basic helicopter configurations can be designed that will achieve much higher performance than currently demonstrated. This report is published for the exchange of information and the stimulation of ideas.

The Army is currently continuing this program and sponsoring several similar programs to investigate rotor systems in still higher speed regimes.

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SUMMARY REPORT,

HIGH-PERFORMANCE-HELICOPTER

PROGRAM - PHASE I

Bell Report No. 533-099-005

Prepared by

BELL HELICOPTER COMPANY

For

U. S. ARMY TRANSPORTATION RESEARCH COMMAND
Fort Eustis, Virginia

FOREWORD

This report summarizes the results of a flight research program conducted to substantiate that the significant gains in helicopter range, productivity, and speed predicted by theory are possible. The program was conducted by Bell Helicopter Company under USATRECOM Contract DA 44-177-TC-711 (Reference 1). Design of the high-performance helicopter commenced upon receipt of contract, 7 August 1961, and modifications to the test helicopter were begun in January 1962. The initial flight of the modified helicopter was conducted on 10 August 1962. Demonstration and flight-test-evaluation flights were completed in January 1963.

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HIGH PERFORMANCE HELICOPTER IN FLIGHT

SUMMARY

A flight research program was conducted to ascertain whether the gains in range, productivity, and speed predicted by theory can be accomplished with a practical helicopter suitable for Army field use. The test results have proved that these gains can be accomplished without compromise of the utility or basic cargo space of the vehicle. Further, it was shown that structural loads and vibrations can be maintained within acceptable limits, as predicted, and that even higher speeds can be attained within existing technology.

For the program, a YH-40 helicopter with UH-1B dynamic components was modified into a high-performance configuration. These modifications included aerodynamic changes to the fuselage, a new control system, and a tilting pylon system which allowed in-flight adjustment of fuselage attitude. Additionally, a new three-bladed rotor was fabricated which could be mounted to the mast either rigidly or through a gimbal (universal joint).

Ground and flight tests were conducted for evaluation of the test vehicle with the standard two-bladed rotor and the rigid and gimbaled three-bladed rotors. The two-bladed rotor was tested both with and without the stabilizer bar. Flights were also conducted to check airflow about the fuselage and to determine the overload gross weight takeoff capability with special wheels mounted on the skid landing gear. At the conclusion of the evaluation tests, demonstration flights were conducted with Government pilots.

The results of the flight tests showed the high-performance helicopter to have improved performance, as well as a significant reduction in loads and vibrations, as compared to the standard UH-1B. A power-limited, level-flight true airspeed of 157 knots was indicated with the two-bladed rotor, and 156 knots with both three-bladed rotors. Maximum level flight speeds of 155 and 151 knots were attained with the two-bladed and three-bladed rotors, respectively. In a shallow dive, a speed of 162 knots was attained with the two-bladed rotor.

At 130 knots, the high-performance helicopter with the two-bladed rotor required 270 horsepower less than the UH-1B (with the same rotor system). This power is equivalent to a parasite drag area reduction of 11 square feet, and represents over 50 per cent of the total apparent drag of the UH-1B at that speed. Based on the test results, the cruise speed, range, and productivity of the research vehicle were improved by about 20 per cent.

In-plane rotor-hub oscillatory moments of the high-performance helicopter were reduced about 60 per cent as compared to the standard helicopter with the same two-bladed rotor system. For the same vibration level, the high-performance helicopter achieved speeds 20 to 30 knots faster than the production UH-1B helicopter.

For all configurations tested, the controllability was satisfactory. The high-performance helicopter, with stabilizer bar, evidenced improved handling qualities over those of the standard UH-1B helicopter, especially in yaw and pitch. As predicted by theory, pitch and roll stability deteriorated with increased speed. This was especially noticeable in roll since the elevator-horizontal stabilizer and center-of-gravity position provided adequate speed stability and pitch damping even at high speeds.

The stability characteristics of the three-bladed rotors were found to be unsatisfactory, primarily because of a pitch-roll coupling due to nonoptimum controls arrangement and hub restraint. The rigid rotor evidenced significant increases in damping and control power over the nonrigid rotors and shows considerable promise.

It is concluded that significant gains in range, productivity, and speed are possible for utility helicopters.

CONCLUSIONS

Based on the results of this program, it is concluded that:

- Significant gains in helicopter range, productivity, and cruise speed can be accomplished within the present state of the art.

- Current analytical methods can be used to predict structural loads, vibration, and rotor performance within the range of parameters investigated. Additional researches are needed to define airfoil characteristics at higher blade Mach numbers.

- The standard UH-1B two-bladed rotor is capable of higher flight speed than previously assumed.

- Special consideration must be given to the design of high-speed rotors if the control loads are to be kept within acceptable limits. Stiff blades which maintain their relationship with the pitch change axis will prove beneficial.

- The cambered vertical fin provided a significant contribution in counteracting main-rotor torque at the higher speeds and resulted in reduced tail-rotor loads and flapping angles.

- Roll stability deteriorates with increasing speed, requiring constant pilot attention. Therefore, improved roll stability will be desirable for high-speed flight. Although pitch stability also decreases at the higher speeds, satisfactory pitch stability can be achieved by a horizontal stabilizer and forward location of the center of gravity.

- The over-all cabin vibration levels measured were adversely influenced by the fuselage response characteristics of the test vehicle. Lower vibration levels can be expected by changing the fuselage response characteristics.

- With the three-bladed gimbaled rotor system, excitations can be generated by the pilot which couple with pylon motions and result in high oscillatory rotor loads. With the rigid system (during ground operation), the pilot is capable of developing very large control moments with resulting high mast bending stresses.

- The drag reduction accomplished on the test vehicle is in close agreement with the reduction predicted by analytical studies and wind-tunnel tests. The performance gains accomplished with this test vehicle can be realized on production helicopters without significant compromise in utility or maintainability by proper attention to design detail in the initial development stages.

RECOMMENDATIONS

As a result of this program, it is recommended that:

- Greater emphasis be given to clean aerodynamic design during the preparation of new helicopter design and procurement specifications.

- The present research program be continued to investigate speeds beyond the present capability of the existing research vehicle.

INTRODUCTION

Early helicopter research efforts were concerned with the solution of fundamental problems of vertical flight and with the development of improved mechanical components. As these problems were solved, it became possible to direct more attention to higher speed and increased performance capability.

Both industry and Government researches had shown that major increases in the helicopter's range, productivity, and speed were possible; however, it was recognized that these increases, predicted by theory, needed to be substantiated by flight research. Positive flight test results, if achieved, would pave the way for major improvements in the performance of the next-generation helicopter.

In April 1960, the United States Army initiated a study program to investigate and recommend modifications to current helicopters which would allow demonstration of the predicted gains. The results of this study, based on analyses by three major helicopter manufacturers, showed that the performance of the existing machines could be increased significantly.

As a part of its independent research effort, Bell Helicopter Company participated in that study program. The results of the Bell study, which are reported in References 3 through 8, showed that major improvements in the performance of the UH-1B helicopter could be achieved and demonstrated with a suitably modified machine. Subsequent to the study, a proposal was submitted to the Army for a high-performance-helicopter flight research program.

The proposed program was reviewed by the Army, and in August 1961 a contract was awarded to Bell Helicopter Company for the design, manufacture, and flight test of the high-performance flight research helicopter.

The objectives of the program were to:

1) Increase the state of the art of rotary-wing aircraft with respect to drag reduction, high-speed stability, vibration, rotor and control-system fatigue loads, and rotor-blade stall and compressibility effects.

2) Demonstrate by flight test that the significant gains in range, productivity, and cruise speed which have been predicted by theory can be achieved with a practical helicopter.

3) Provide recommendations for increasing the performance capability of all Army helicopters.

This report summarizes the results of the program to date.

DESCRIPTION OF TEST VEHICLE

The aircraft made available for the program was a YH-40 helicopter, reconfigured with YUH-1B dynamic components. This aircraft was first flown in June 1958 and had accumulated about 345 hours of flight time during numerous test programs at the contractor's facility. Figure 1 shows the test vehicle in the YUH-1B configuration prior to the start of modification into the high-performance configuration. Figure 2 shows the same vehicle after completion of its modification into a high-performance research vehicle.

The principal modifications accomplished on the basic helicopter included changes to the pylon mounting, the fixed and rotating controls, and the external fuselage lines. Components were fabricated for a three-bladed rotor which could be configured as either a gimbaled or a rigid rotor system and interchanged with the standard two-bladed rotor. Also, wheel assemblies were fabricated for evaluation of overload take-off capabilities.

As a design philosophy, UH-1 components were selected for use in the modification wherever possible to minimize costs and development problems during the flight test program. Additionally, the structural design criteria for the new rotor components were based on UH-1B load data, extrapolated for the higher airspeeds expected with the test vehicle, rather than the lower loads which had been predicted by the analytical studies. This approach allowed simpler and less costly structural analysis techniques to be utilized and provided additional margins of safety to cover the possibility that the loads encountered during the flight tests might prove to be higher than expected.

AERODYNAMIC CHANGES TO FUSELAGE

Aerodynamic modifications were made to the fuselage to reduce the aerodynamic download and parasite drag area. The modifications included the addition of fairings to the aft fuselage and landing gear, and around the mast, controls, and pylon. Additionally, the vertical stabilizer was modified into a cambered airfoil, and external protuberances such as mirrors, rain gutters, antenna, etc., were removed. A new engine air induction system was installed to provide increased ram pressure recovery and to prevent drag due to separation over the inlet lip at high speeds. Figure 3 shows the difference between the standard UH-1B and the high-performance-helicopter fuselage lines.

The aft fuselage and pylon fairings are generally of fiberglass and aluminum honeycomb sandwich construction, built up on plaster patterns. The fairings are installed over the existing fuselage skins.

Aft Fuselage Fairing

The aft fuselage fairing is provided with a removable panel on each side of the helicopter for access to the battery, electrical, and service compartments. The aft fuselage fairing design is based on wind-tunnel test results which indicated a decrease in fuselage drag and a reduction in turbulence aft of the fuselage. In conjunction with the aft body fairing, the sliding cargo door is replaced by a hinged door. The door is hinged at the forward edge and reduced in width by approximately 50 per cent. The area aft of the new door is covered by a metal skin which extends aft to provide a smooth transition with the aft body fairings.

Pylon Fairings and Engine Inlet System

The pylon fairing is comprised of four major sections: a leading edge, a trailing edge, and a right- and a left-hand center section. The center and trailing edge sections are removable to provide access to the transmission and engine areas. The engine inlet system is incorporated as an integral part of the center sections of the pylon fairing.

The engine induction system is comprised of a right- and a left-hand high-speed inlet and ducting, right- and left-hand "blow in" doors for hovering and low-speed flight, and a plenum chamber. The high-speed inlets are designed to be of minimal area consistent with engine airflow requirements (88 square inches) and to prevent drag due to separation over the inlet lip at high speeds. To provide for hovering and low speeds where ram effects are negligible, the high-speed area is augmented by spring-loaded "blow in" doors which open into the plenum chamber when the inlet static pressure becomes less than ambient. The "blow in" doors increase the inlet area by approximately 100 square inches. Figure 4 shows the pylon fairing and engine air inlet.

Vertical Stabilizer

The vertical stabilizer was modified into a cambered airfoil section by the addition of a new leading and trailing edge. Figure 5 is a comparison of the standard fin and the cambered fin of the test vehicle. The purpose of the cambered section is to reduce tail-rotor flapping motions and loads at higher speeds by utilizing aerodynamic lift to supplement the tail-rotor thrust in counteracting main-rotor torque. The tail-rotor gearbox is covered by a removable fiberglass fairing. Cooling air for the gearbox enters at the leading edge and exits through louvers near the trailing edge.

Skid Landing-Gear and Overload Takeoff Wheels

Airfoil section fiberglass fairings are installed over the landing skid cross tubes. Fairings are also installed at the cross-tube-to-fuselage junction to eliminate interference drag. The fairings are constructed of wood and are slotted vertically to allow normal gear deflection without changing the energy absorption characteristics of the gear. The junction fairings are positioned such that they align with the fairings installed on the cross tube in flight when the landing gear is unloaded. Figure 6 shows the faired landing gear with the ferry mission takeoff wheels installed and also shows the misalignment of the cross tube and junction fairings when the gear is loaded.

TILT PYLON SYSTEM

To minimize the fuselage download and parasite drag, a tilt pylon mechanism was installed to permit in-flight adjustment of fuselage attitude. During the design phase, several concepts were investigated and a system was selected in which the pylon was mounted in a cradle which pivoted about a lateral line through the geometric center of the input driveshaft. This location of the pivot axis resulted in minimum misalignment of the input driveshaft and allowed a fixed engine position for the full range of pylon tilt without exceeding the capability of the input driveshaft couplings. Figure 7 shows the basic geometry of the system.

The mast conversion angle range of 4 to 11 degrees forward of the vertical was established based on the calculated angles for maximum performance within the range of power and expected drag for the high-performance fuselage.

To obtain the minimum input driveshaft misalignment angles over the full conversion range of 7 degrees, the engine was repositioned from the existing 3 degrees nose-down angle to a 7-1/2 degrees nose-down angle. The engine relocation entailed only minor modifications to the engine mount, firewalls, and throttle controls.

Although the tail-rotor driveshaft misalignment did not exceed 5 degrees over the full conversion range, the standard tail-rotor coupling could not be used because of angular and axial alignment limitations. New tail-rotor driveshaft couplings were fabricated using standard main input couplings, modified to allow for increased axial motion.

For installation of the tilt pylon system, it was necessary to remove the existing pylon support structure above the work platform and to install new structure to support the pylon cradle. The pylon cradle provides a tiltable base for the pylon. Attachment of the transmission pylon to the cradle is essentially the same as for the standard configuration. Pylon conversion angle is accomplished by tilting the cradle about the pivots located on the cradle support structure. The cradle is moved by two servo-controlled hydraulic cylinders mounted between the forward edge of the cradle structure and the airframe. The hydraulic servos are interconnected to insure uniform travel and are controlled by an electrical actuator. The electrical actuator is, in turn, controlled by a switch on the collective stick which allows the pilot to vary the pylon-fuselage angle while in flight.

CONTROL SYSTEMS

The control system was modified to accommodate the tilting pylon system and to be adaptable for both the two- and three-bladed rotor configurations which were to be evaluated during the program. To provide a universal system which would allow the maximum utilization of components common to all rotor configurations while maintaining maximum control stiffness, the collective-cyclic mixing functions were transferred from the rotating system (pylon based) to the fixed system (fuselage based). Figure 8 shows the control system for the high-performance configuration. For comparison, the standard control system is shown in Figure 9.

Fixed Controls

The fixed controls remain unchanged for each of the rotor systems except in the case of the three-bladed rigid configuration, where the lateral sensitivity can be reduced by changing the lateral interconnect bellcrank input and output arms. For collective control inputs, the swashplate moves axially on a sleeve attached to the upper transmission case; while for cyclic inputs, the swashplate tilts about a spherical bearing. The swashplate tilt axis is disposed 45 degrees from the direction of cyclic stick travel; i.e., for forward cyclic stick inputs, the swashplate tilts down 45 degrees to the right of forward. Control inputs are boosted by three servo-operated hydraulic cylinders based on the pylon cradle. The cylinders are moved simultaneously for a collective input and differentially for cyclic inputs. For forward and aft cyclic, the diametrically opposed right forward and left aft cylinders move in opposite directions while the left forward cyclic remains fixed. The left forward cylinder moves with a lateral cyclic input while the other two cylinders remain fixed. Collective moves all cylinders together.

Decoupling linkages are installed between the fuselage-based controls and pylon cradle to eliminate unwanted control motions due to pylon tilting.

The cyclic-collective mixing functions are accomplished by two sets of mixing levers. The forward and aft cyclic-collective mixing function utilizes the mixing lever system as originally installed in the aircraft except that for the new machine a collective input replaces the lateral input. Both output tubes move uniformly in the same direction with a collective input and in opposite directions with a cyclic input. The lateral cyclic-collective mixing lever operates in a similar manner. The collective and cyclic control stick installations are the same as for the UH-1B.

Rotating Controls

The control motions for each particular rotor system are obtained by interchanging components of the rotating control systems. The rotating swashplate ring is provided with attachment points for both the two- and three-bladed rotor controls. The full control displacement of swashplate motion and resulting rotor-blade angle change for the various rotor systems are as follows:

Swashplate Travel		2-Bladed Rotor	3-Bladed Rotor	
			With Idler	No Idler
Collective	3.2 in.	16-1/2°	19°	16-1/2°
F/A Cyclic	±12°	±13°	±14.7°	±13°
Lateral	±10°	±11°	-	-
Reduced	±7.6° (25% reduction)	-	± 9.7°	± 8.3°
Lateral	±6.0° (40% reduction)	-	± 8.4°	± 7.8°

Elevator and Tail-Rotor Controls

The tail-rotor controls for the high-performance helicopter are the same as those of the standard UH-1B.

The elevator control system was modified to incorporate hydraulic boost. Synchronization was provided by coupling the elevator with collective, fore-and-aft cyclic, and pylon position. Additionally, to facilitate the test program, it could be trimmed in flight through small angles.

Hydraulic System

During the flight program, it was found desirable to incorporate a dual hydraulic system for the test helicopter. The dual system consists of a primary and a secondary system which share common control cylinders but have independent pumps, supply and return lines, and reservoirs. The systems are arranged such that a broken pressure line would cause automatic switching to the second system without loss of control boost.

ROTOR SYSTEMS

Main-Rotor Systems

The helicopter may be configured with either the standard two-bladed or a three-bladed rotor system. In addition, the two-bladed rotor may be flown with and without the gyro stabilizer bar, and the three-bladed rotor may be configured as either a gimbaled or a rigid system with only minor component changes required. The basic features of each rotor system are described in the following paragraphs.

Two-Bladed System

The two-bladed rotor for the high-performance helicopter is a standard UH-1B rotor system. The rotor is a semirigid, "see-saw", underslung, feathering axis design. By changing pitch horns, the rotor may be configured either with or without the stabilizer bar. Data for this rotor system are given below:

Number of Blades	2
Airfoil Designation	NACA 0012
Chord	21 inches
Diameter	44 feet
Blade Twist	-10 degrees
Blade Area (total)	77 square feet
Disc Area	1521 square feet
Solidity	.0507
Rotor RPM @ 6600 Engine RPM	324
Tip Speed	746 ft/sec
Disc Loading @ 6500 lbs GW	4.3 lb/sq ft

Three-Bladed Rotor System

The three-bladed rotor for the high-performance helicopter may be configured as either a gimbaled or a rigid system with only minor component changes required. Basic data for the rotor are presented below:

Number of Blades	3
Airfoil Designation	NACA 0012
Chord	21 inches
Diameter	42 feet
Twist	-6 degrees
Blade Area (total)	110 square feet
Disc Area	1385 square feet
Solidity	.0795
Rotor RPM @ 6400 Engine RPM	314
Tip Speed	691 ft/sec
Disc Loading @ 6500 lbs GW	4.7 lb/sq ft

Production components are used in the system whenever possible. The rotor blades are essentially UH-1B main-rotor blades except for differences in length, twist, and spanwise mass distribution. The grips and blade retention systems are assembled from stock UH-1B components. The spindles are made from hand-forged steel billets machined to adapt the grip assemblies to the three-bladed yoke. The yoke is a steel machining and is common to both the rigid and the gimbaled configurations. A beamwise flexure is machined in the yoke between the mast and spindle attachment points. Components for either the gimbaled- or rigid-mast configuration are bolted to a mounting flange on the yoke. Figures 10 and 11 show the gimbaled- and rigid-rotor configurations, respectively.

Figure 12 shows the component buildup for the gimbaled configuration. Four trunnion bearing assemblies are spaced 90 degrees apart in a gimbal ring. The bearings are attached in diametrically opposed pairs to a mast-mounted trunnion and the rotor yoke support so as to effect a universal joint between rotor and mast.

To change from a gimbaled to a rigid mount, the gimbal ring and trunnion are replaced by an upper cone, which is splined to the mast, and by a lower cone and split ring, which bear against the mast below the yoke and hold the rotor hub in rigid relationship with the mast. Figure 12b shows the rigid component buildup. By comparing Figures 12a and 12b, it can be seen that the rotor mounting may be readily changed from the gimbaled to the rigid system.

Tail Rotor

The tail rotor is a production UH-1B tail rotor. Basic data for the tail rotor are as follows:

Number of Blades	2
Airfoil Designation	NACA 0015
Chord	8.41 inches
Diameter	8.5 feet
Twist	0 degrees
Blade Area (total)	5.96 square feet
Disc Area	56.8 square feet
Solidity	.105
Rotor RPM @ 314 Main Rotor RPM	1604
Rotor RPM @ 324 Main Rotor RPM	1654
Tip Speed @ 314	714 ft/sec
Tip Speed @ 324	736 ft/sec

INSTRUMENTATION

The helicopter components, such as the hubs, blades, pitch links, etc., were instrumented as they became available during the fabrication phase of the program. As the helicopter neared completion, the instrumentation equipment, i.e., oscillographs, slip rings, position transducers, etc., was installed and checked out.

Instrumentation was installed to record and/or monitor the test helicopter's performance, stability, controllability, rotor and control loads, fuselage vibrations, and other information as desired during the ground and flight test programs. The information was recorded on two oscillographs installed on the cabin bulkhead and beneath the passenger seats. Figure 13 is a photograph of the instrumentation installed in the cabin.

INSTRUMENTED ITEMS

Specific channels of instrumentation were provided for recording the following information:

Airspeed
Rotor azimuth
Gas producer speed
Engine and rotor rpm
Differential torquemeter
Outside air temperature
Pressure altitude
Mast conversion angle
CG accelerometers
Pilot and copilot location accelerometers
Pitch and roll attitude gyros
Cyclic, directional, and collective control positions
Main-rotor flapping and feathering position
Main-rotor mast moments
Main-rotor hub assembly beam and chord moments
Main-rotor blade beam and chord moments at three stations
Main-rotor drag brace loads
Main-rotor pitch link loads
Cyclic and collective control tube loads
Pylon lift link load
Pylon motion (3 pickups)
Tail-rotor hub flapping position
Tail-rotor hub beam and chord moments
Tail-rotor blade beam and chord moments
Horizontal stabilizer moments
Horizontal stabilizer position
Pylon actuator cylinder loads

Wiring and connectors for all channels were routed to the oscillographs. To reduce the possibility of reading errors in data reduction, only the specific channels necessary for a particular test were connected into the oscillographs. If postflight inspection of data indicated an area of particular concern, additional channels were connected to provide a more comprehensive evaluation of the area in question. In general, vibration, fuselage attitude, power, rotor flapping, yoke loads, cyclic and collective control positions, and pylon elevator positions were recorded for each flight. Additional information was recorded as necessary throughout the test program.

CALIBRATION AND REPEATABILITY

All instrumented items were calibrated either in the laboratory, on the ship, or in flight. Pre- and postflight calibrations were made for all oscillograph recorded items. The ship's airspeed system was calibrated against a trailing bomb airspeed sensor. All data given herein are corrected for instrument and other errors. True airspeeds are used throughout the report, with horsepower and gross weights corrected to standard conditions.

Throughout the program, the repeatability of all flight data was good. Power data from flight to flight were repeatable within 25 to 50 horsepower. Strain gage and position data were repeatable within less than 5 per cent of the magnitude of the item being measured. Acceleration levels showed the greatest variation of all items measured during the program (10 to 20 per cent). It is believed that this resulted from the many different center-of-gravity locations, fuselage loading distributions, mass conversion angles, etc., flown during the test program.

GROUND TESTS

During the course of the buildup of the helicopter, many functional, proof, and other tests were conducted in accordance with References 1 and 9 to insure satisfactory operation and to provide data for later phases of the program.

Upon completion of the modification and miscellaneous tests, the helicopter was tied down and ground runs were conducted to establish satisfactory operation of the vehicle. The various proof, functional, and ground run tests are discussed below.

CONTROLS PROOF LOAD

An operational proof load test was performed to assure that the modified control system would not experience excessive deflections or interferences detrimental to safety of flight. The main-rotor and synchronized-elevator controls were loaded through the full range of control stick travels and pylon conversion angles. The tail-rotor system was not included in these tests since it had not been modified.

The main-rotor controls were tested to 80 per cent of the design limit loads as specified in Reference 9. The synchronized elevator controls were loaded by applying a 4000-inch-pound moment about the support bearings and at the same time moving the elevator by means of the cyclic and collective stick and trim actuator.

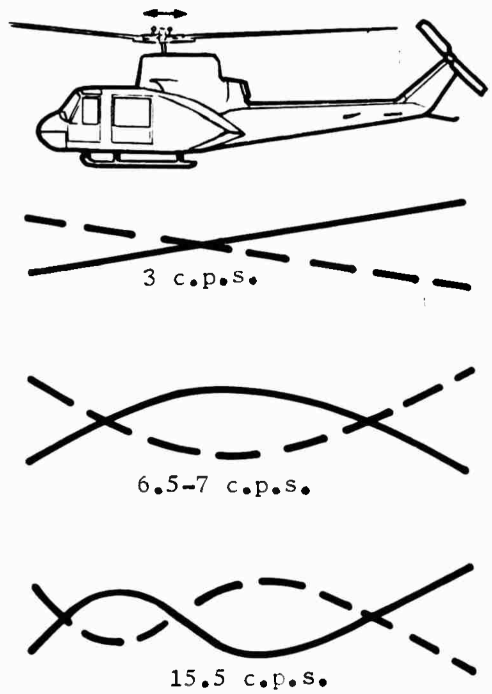
During these tests, a condition of excessive deflection in the collective system and several minor interferences were discovered. After the correction of these items, the system was retested and found to be free of interference, excessive deflection, or binding that would be detrimental to flight safety.

HYDRAULIC SYSTEM TESTS

The hydraulic system was tested in accordance with procedures established by Reference 9. Nominal system pressure was applied from a portable hydraulic test stand connected to the helicopter ground test fittings. The system was thoroughly checked for leakage, and all controls were cycled to the limits of stroke to insure that the components and flexible lines did not interfere with adjacent structure.

FUSELAGE VIBRATION (SHAKE) TESTS

Fuselage vibration tests were conducted to provide base-line data on fuselage response and to establish the effects of the fuselage modifications on its response. The fuselage natural frequencies in the longitudinal and lateral directions were measured for the basic helicopter configuration at 7078 pounds gross weight, center-of-gravity location at Station 128.7, nuetral pylon conversion angle (7-1/2 degrees) forward, 200-pound pilot and copilot, full fuel and oil, and 300 pounds of ballast at Station 4.0. For these tests, the rigid rotor was installed and the blades were replaced by weights. The excitation was provided by a mechanical shaker with a frequency range of 1 to 30 c.p.s. Additionally, the vertical response to two-per-rev and three-per-rev excitations at the pilot and copilot stations was determined for a number of loading conditions other than that defined above.



FUSELAGE LONGITUDINAL NATURAL FREQUENCIES AND MODE SHAPES

The longitudinal natural frequencies determined by these tests are shown in the adjacent sketch. The measured lateral natural frequencies were 4.25 c.p.s. (rigid pylon-fuselage roll) and 7.0 c.p.s. (rigid fuselage-pylon bending).

The vertical response for the various loading conditions is shown in Table 1, page 39. By comparison of the response for the various loading conditions, it can be seen that small weight relocations produce significant changes in response. In general, additional weight in the nose reduces two-per-rev response and increases the three-per-rev response. Added weight in the tail is helpful in reducing three-per-rev response.

TIEDOWN (GROUND) RUNS

Ground runs were conducted in accordance with Reference 9 with the helicopter tied down to establish satisfactory operation in general, and specifically to obtain data for evaluating the new and modified components of the system. The initial ground runs were conducted with the standard two-bladed rotor installed. In addition to the normal complement of instrumentation, the main-rotor input shaft and couplings were instrumented to provide data on coupling alignment and surface temperature.

The initial run was restricted to operation with the pylon cradle in the neutral (7-1/2 degrees mast angle) position, and coupling alignment data were obtained to allow the coupling performance to be evaluated. A review of these data indicated that the coupling alignment was well within the acceptance limits and that it could be expected to operate satisfactorily over the full conversion range of the pylon. This was later confirmed from data taken during operation at the extremes of pylon tilt and by subsequent disassembly inspections of the couplings.

During the evaluation of the couplings, the other helicopter components and systems were subjected to a thorough shakedown. All controls were operated to their maximum displacements within the limits of safe tiedown operation. Control operation with the hydraulic boost was satisfactory; however, without hydraulic boost, the force required to move the cyclic stick in the fore-and-aft direction was excessive. The cause for the high stick force was determined to be due to the action of the irreversible valve of the servo cylinder which caused an effective hydraulic lock when loaded by a collective feedback force.

To move a cylinder manually, the servo valve has to be thrust in the direction of desired motion. To hold a feedback force from the rotor, the valves have to be thrust against the force. Thus, with a steady down collective feedback force, both fore-and-aft cylinder valves are thrust up; and the cyclic stick is hydraulically locked, since for fore-and-aft stick motion, one cylinder must be moved down.

A number of changes were incorporated to improve "boost off" operation, including the modification of the hydraulic irreversible valves. These changes resulted in boost-off control system operation that was deemed satisfactory at the time, and the helicopter was considered ready for flight. Later, during the flight program, it was determined that the boost-off rate limiting was excessive and that the pilot would experience considerable difficulty in maintaining control in the event of an unanticipated boost system failure. Consequently, the secondary hydraulic system described previously was installed at that time.

Additional ground runs were conducted for each of the rotor and other systems evaluated during the flight program. During these ground runs, the machine was operated throughout its RPM, power, and control ranges except as limited by tiedown operation.

FLIGHT TESTS

The modified helicopter was first flown on August 10, 1962 with the standard two-bladed rotor and gyro stabilizer bar. The initial flights were limited to hovering and low speeds and were for the purpose of pilot familiarization and helicopter shakedown. During these flights, pylon-rotor stability was investigated. An engine and transmission oil cooling survey was conducted, and additional data were obtained on the input drive shaft coupling. Cooling was found to be adequate, the drive shaft operation was found to be satisfactory, and the rotor-pylon stability was considered acceptable with and without hydraulic boost.

Following the familiarization and shakedown flights, the helicopter was flown up to a speed of 120 knots to evaluate pylon behavior and to explore controllability. During these flights, it was found that the pylon was inadequately damped and that the rotor and elevator control rigging required some changes. Rigging changes were accomplished and several damper configurations were evaluated. Satisfactory pylon damping was obtained by installing friction dampers between the upper transmission case and the cabin roof structure (reference page 34). This damper installation has since been incorporated as a part of the production UH-1D helicopter.

Upon completion of these preliminary flights, the test vehicle was considered ready for complete evaluation. Exploratory flights were conducted in a conventional buildup manner. After each flight, data were evaluated to determine whether the helicopter could safely enter a higher speed regime. After establishing a basic flight envelope, comprehensive evaluation flights were conducted at various helicopter gross weights, rotor speeds, altitudes, etc. During these flights, quantitative data were obtained on power, stability, rotor loads, and cockpit vibrations. Speeds up to 162 knots were attained.

Similar tests were conducted for evaluation of the test vehicle without the stabilizer bar, and with the three-bladed rotor system in both its gimbaled and a rigid configuration. The exploratory testing required for the latter configurations was less than for the standard rotor, since even though the rotor systems were new, the basic helicopter operation had been established during the earlier tests.

Later in the program, specific flight tests were conducted to evaluate the airflow over the high-performance-helicopter fuselage (tuft survey), to determine the overload (running) takeoff capability of the machine, and to establish a safe maneuver envelope for future demonstration flights.

For the airflow investigation, the helicopter was tufted as shown in Figure 14, and the tuft behavior was observed and recorded on motion picture film throughout the helicopter speed range and for various pylon conversion angles. As was expected at this time, due to the low drag of the ship, the tufts indicated smooth airflow over the fuselage with little turbulence and no fully stalled areas except for the lower portion of the tail boom in the area of the horizontal stabilizer.

During the evaluation of the running takeoffs, it was found that the capability of the machine used in this manner greatly exceeded the allowable landing gear loads. It is estimated that the maximum takeoff gross weight for a running takeoff exceeds 12,000 pounds for the test vehicle. Due to the landing gear restrictions, the overload takeoffs were simulated at a gross weight of about 8,000 pounds and with a power less than that required to lift the skids off the ground. Figure 15 shows the helicopter using the landing gear wheels during a simulated overload running takeoff.

During the tests to establish maneuver limits for the demonstration flights, steady-state turns of approximately 1.5 g (over 45 degrees bank angle) were flown at airspeeds of 100 and 126 knots. No attempt was made to obtain power or structural limits of the machine.

Testing was completed in February 1963, after 48 hours of ground and flight time. At the conclusion of the flight test evaluation, the test vehicle with the two-bladed rotor installed was demonstrated to pilots from USATRECOM, NASA, and the Army Aviation Board of Fort Rucker, Alabama.

FLIGHT TEST RESULTS

Presented in this section are the test results obtained with the high-performance flight research helicopter (HPH). Additionally, standard UH-1B data are given for purposes of comparison. Measured performance, stability and control, rotor and control loads, and fuselage vibrations are presented, discussed, and compared with corresponding values predicted during the study program mentioned earlier.

PERFORMANCE

Hovering Performance

Hovering data for the high-performance helicopter with two- and three-bladed rotor configurations are shown in Figure 16. Also given is the hovering performance for the UH-1B as obtained during the Air Force Category II Flight Tests (Reference 8). All data are corrected to 324 rotor r.p.m. and to sea-level, standard day conditions. As expected, the hovering performance of the test vehicle with the two-bladed rotor is essentially the same as that of the standard UH-1B.

The three-bladed rotor is shown to require more power in hovering as compared to the two-bladed rotor. At a gross weight of 7,000 pounds, the power difference is about 85 horsepower. A simple analysis indicates that the power loss may be broken down as follows:

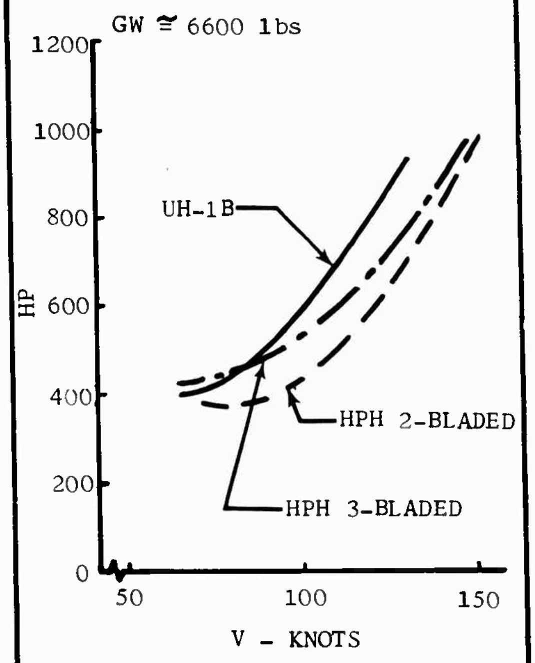
Component	Δ Horsepower
Induced	28
Profile (at 324 r.p.m.)	42
Hub drag	15
TOTAL	85

The induced power is increased directly due to the shorter blade radius and also due to the increased fuselage download resulting from the higher disc loading. Profile power is increased due primarily to the increased solidity; however, the outboard location of the blade root doublers is also significant. Due to the higher solidity of the three-bladed rotor, the rotor was operated at a lower tip speed (314 r.p.m.) to minimize the power difference.

Level-Flight Performance

Level-flight speed power data at two gross weights for each rotor configuration are shown in Figures 17 through 20. The highest speeds were attained with the two-bladed rotor without the stabilizer bar. With this configuration, 155 knots true airspeed was attained in level flight, and 162.5 knots was attained during a shallow 4-degree dive at reduced power. The maximum airspeed with the standard rotor with the stabilizer bar was 150 knots; with the three-bladed rotor, 151 knots.

In each case the maximum level-flight airspeed was limited by the power available for the test conditions. Under standard conditions when the full installed power is available (1100 s.h.p.), a power-limited true airspeed of 157 knots for the two-bladed rotor and 156 knots for the three-bladed rotor is indicated.



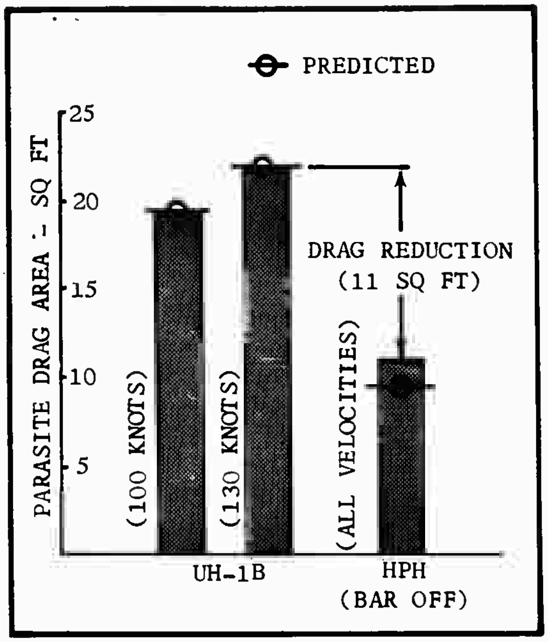
POWER VS VELOCITY

A comparison of the power versus velocity for the UH-1B and the test vehicle with two-bladed and three-bladed rotors is shown in the inset. Predicted values (from Reference 3) are shown on Figures 18 and 19. For the two-bladed helicopter without the stabilizer bar (Figure 18), the agreement is satisfactory, although slightly on the low side. The discrepancy is partially explained by the fact that in Reference 3 a fuselage parasite drag of 9.5 square feet was used, which is about 1.5 square feet less than believed to be the actual value for the research vehicle in the configuration tested.

For the three-bladed rotor helicopter, the predicted power is considerably lower than the measured power. This difference is due to the drag of the unfaired blade grips and hub of the three-bladed rotor and the higher than estimated fuselage drag discussed above. The drag due to the blade grips is more significant on the three-bladed rotor, since the grips are located further from the center of rotation than for the two-bladed rotor. The predicted values were based on faired blade grips. Grip fairings made for the program were removed after ground run since they were found to be unsatisfactory from a structural standpoint; consequently, no test data were obtained with the grips faired.

Drag Reduction

The bar graph on the following page shows the estimated and actual parasite drag areas for the UH-1B and the high-performance helicopter. The estimated drag values for the UH-1B are taken from Reference 3.



DRAG COMPARISON

Actual drag values for the UH-1B were obtained from the fuselage attitude data of the Category II tests (Reference 10) and the full-scale wind-tunnel test drag data of Reference 2. Actual drag values for the high-performance helicopter are based on the flight test results of the program. It is seen that the apparent drag of the UH-1B increases with speed as predicted. This is due to the increased nose-down fuselage attitude and the resulting increase in download. Also indicated on the graph is the over-all drag reduction of 11 square feet achieved with the high-performance helicopter. This value is slightly less than predicted due to the additional drag of the unfaired rotor grips.

The estimated parasite drag values for the test vehicle in its various configurations are given below:

Two-bladed rotor without the stabilizer bar	11 square feet
Two-bladed rotor with the stabilizer bar	13 square feet
Three-bladed rotor	13 square feet

For the configuration with the two-bladed rotor without the stabilizer bar, the predicted drag reduction was realized if allowance is made for the unfaired blade grips ($f = 1$ square foot, Reference 4). The drag of the two-bladed high-performance helicopter with the bar is higher than anticipated. This is due to the drag of the stabilizer bar being about one square foot higher than that measured during the one-half-scale wind-tunnel tests of Reference 4. It is believed that this is the result of stabilizer bar-pylon interference.

An itemized breakdown of drag reductions of the fuselage and a comparison with predicted values is given in Table 2. The tabulated drag reductions are for the test vehicle with the standard UH-1B rotor without stabilizer bar compared with the UH-1B.

Effect of Pylon Tilt

Figure 21 shows the effect of fuselage attitude on the required power at a speed of 107 knots. The fuselage attitude was varied by tilting the pylon and trimming the elevator. It is seen that a .5-degree nose-up fuselage attitude is optimum and that a 4-degree deviation from this optimum causes a considerable power loss. At 107 knots, the optimum pylon tilt angle was found to be about 7 degrees (depending on the elevator setting); while at maximum speed, optimum results were obtained with the pylon at about 10 degrees.

Range and Productivity

Because of the reduced power requirements for the test vehicle as compared with the UH-1B at the same airspeeds, the high-performance helicopter has significantly improved range and productivity, in addition to higher cruise speeds. Table 3 summarizes the increase in cruise speed and range for the test vehicle as compared to the standard UH-1B at its best cruise speed.

Productivity is defined as the product of payload times cruise speed. Consequently, for the same payload, the increase in productivity for the test vehicle is the same as the increase in range and cruise speed as shown in Table 3.

The best performance was obtained with the standard two-bladed rotor (without stabilizer bar) installed on the test vehicle. The measured performance for that configuration forms the basis for the comparison shown in Table 4. This table compares the performance of the UH-1B and the high-performance vehicle for a typical transport mission. The comparison assumes the same basic weight for each vehicle.

From Table 4, it can be seen that the high-performance vehicle can deliver the same payload over a greater range, or deliver the same payload in less time for the same range, than the UH-1B. Assuming the same range, 20 per cent more payload can be delivered in the same time; or the same payload can be delivered in 16 per cent less time with a 15 per cent reduction in fuel required. Since the test vehicle encompasses the same fuselage cargo volume as the standard helicopter, it is evident that significant productivity improvement can be obtained without compromise of the basic utility of the machine.

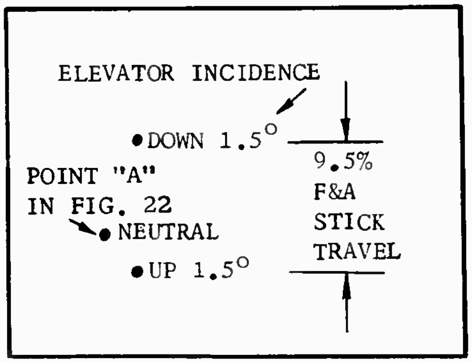
The range and productivity for the three-bladed rotor configuration are considerably less than those shown in Table 4. It is believed that for a production three-bladed rotor design where tip speed, solidity, and engine characteristics are properly matched, the productivity and range figures would be comparable to those shown for the production two-bladed rotor.

STABILITY AND CONTROL

The flying qualities of the high-performance helicopter were evaluated with four different basic rotor and control configurations: the two-bladed rotor with and without the stabilizer bar and the three-bladed rotor with the gimbaled and the rigid mast attachment. The following discussion summarizes the salient flight test results with respect to stability and control for each configuration.

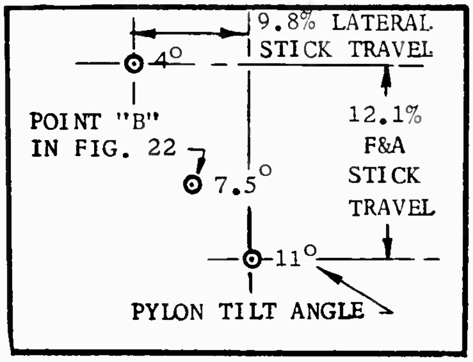
Controllability

Controllability data were obtained for each of the four configurations. Figure 22 is a composite of typical stick plots obtained during the flight testing. It shows longitudinal and lateral cyclic stick position as a function of speed for the two-bladed rotor with stabilizer bar and for the three-bladed rigid rotor. From such plots, the level-flight longitudinal controllability information can be extracted. Data for all four configurations are presented in Figure 23, and the longitudinal gradients in level flight are seen to be very nearly the same and possess positive gradients. Another longitudinal controllability characteristic to be noted is the effect of the synchronized elevator. The synchronized elevator proved to be effective in controlling fuselage attitude during forward flight (see inset at left), and showed a definite improvement over the standard installation with respect to autorotation characteristics.



EFFECT OF ELEVATOR TRIM AT 80 KNOTS - TWO-BLADED ROTOR

A factor affecting both longitudinal and lateral control was the pylon tilt angle. The cyclic stick motion required for trim during pylon "conversion" in hover is shown in the inset at the left. By inspection, it is seen that both aft and right cyclic are required for trim as the pylon is converted forward. The aft stick requirement is due to the apparent forward fuselage center-of-gravity shift with respect to the rotor mast as the pylon tilts forward. Right stick is required to counteract the left rolling moment due to the increase in vertical arm length between the main-rotor hub and the tail rotor as the fuselage attitude becomes more tail-down during forward pylon tilt.



EFFECT OF PYLON TILT IN HOVER
TWO-BLADED ROTOR

For the high-performance helicopter, the directional control required for trim at high speed was significantly reduced as compared to the standard helicopter. This reduction is due to the lower power required and the off-set, cambered vertical fin. For example, at 126 knots, the high-performance helicopter with the two-bladed rotor required 270 horsepower less than the UH-1B. This reduces the thrust required on the tail rotor by about 160 pounds. At that speed, it is calculated from the change in tail-rotor collective that the tail-rotor thrust is further reduced by about 180 pounds. Thus, at 126 knots, the total tail-rotor unloading of the high-performance helicopter is about 50 per cent of that of the unmodified machine. At 150 knots, the vertical fin unloaded the tail rotor of the high-performance helicopter by an estimated 51 per cent.

Static Stability

Constant power stick plots were obtained for all configurations to investigate longitudinal static stability. Up to and including the highest speeds investigated for static stability (125 knots), all configurations were stable. The only indication of difficulty is with the rigid, three-bladed rotor at aft c.g. and at high speeds (see Figure 24). A plot of the slope of the trim curves against speed indicates that for this configuration, neutral static stability occurs at 143 knots. This condition evidenced itself at high speeds in that trim speed was more difficult to hold with the rigid than with the gimbaled rotor configuration. The change in rotor force vector orientation with respect to the helicopter center of gravity is primarily responsible for the deterioration of static stability with the rigid rotor. A change in elevator trim or center-of-gravity shift forward moves the neutral stability point to a higher speed.

The directional static stability characteristics of the test vehicle were similar to those of the unmodified machine. The high-performance helicopter and the UH-1B evidence positive static directional stability.

Dynamic Response During Autorotation Entry

The behavior of the helicopter during an autorotation entry at cruise speed was determined early in the test program during pilot familiarization flights because of its importance to safety. Figure 25 is a time history of the control position and fuselage response with the two-bladed rotor and stabilizer bar during a rapid power reduction from 126 knots airspeed. As shown by the figure, no excessive or abrupt control motions were required to return the helicopter to a stabilized flight condition. The power reduction was accomplished primarily by decreasing main-rotor collective pitch. The fore-and-aft cyclic stick position change that is normally required when entering autorotation has been virtually eliminated by the use of the synchronized elevator. The roll attitude trace shows the left rolling tendency and the oscillatory nature of the roll response.

During autorotation entry, the test helicopter was reported to have more of a left rolling tendency than the standard helicopter. This resulted from the fact that little fore-and-aft stick motion was required to trim (because of the elevator synchronization); however, the standard right lateral stick motion was necessary, and was therefore more apparent to the pilot. When no correction was made, left rolling resulted.

Dynamic Stability

Reduced roll stability at higher speeds was encountered with all configurations. The reason for this is that the rotor's contribution to roll damping decreases with speed and can even become negative in the 140- to 160-knot range. This lack of roll damping causes oscillations induced by lateral-directional coupling, the stabilizer bar, or pilot inputs to require constant pilot attention. Stability augmentation is required as design speeds are increased.

The lateral-directional coupling mentioned above is not the same as the persistent low-amplitude yaw oscillation found in the UH-1B. The high-performance helicopter did not evidence that characteristic due to the larger vertical fin area and reduced turbulence over the fin and tail rotor.

1. Dynamic Stability With Two-Bladed Rotor

- Lateral Control Pulse - Figure 26a shows the high-performance-helicopter roll response with the standard rotor and stabilizer bar at 132 knots in level flight. The left rolling tendency mentioned earlier is not apparent in this figure. The improvement in roll characteristics at this speed is believed to be the result of fuselage and rotor interference effects. A general deterioration of roll stability takes place with increased speed due to the reduction in rotor damping.

  The typical roll response for the high-performance helicopter with the two-bladed rotor without a stabilizer bar is shown in Figure 26b. Both left and right control inputs are shown. Without the stabilizer bar, the roll response becomes unstable at speeds in excess of approximately 120 knots. The roll response at 134 knots following a right cyclic input is unstable, and the period of the oscillation (6 seconds) is twice that recorded with the stabilizer bar in the system. The attitude-sensitive component of the stabilizer bar input is primarily responsible for this change. Although the increase in period is desirable, the large decrease in damping causes this configuration to be somewhat difficult to fly. No directional oscillation was reported; however (through lateral directional coupling), the roll oscillation that was observed has elements of the Dutch roll mode. Recent testing with heading information recorded indicates that this mode is coupled directionally and is lightly damped but not unstable. The response for a left cyclic input is similar to that for a right input. The traces shown record the most severe motions that the helicopter was allowed to execute. The left rolling tendency was apparent even following a right cyclic input.

- Longitudinal Control Pulse - Figure 27a is a time history showing the fuselage attitude response during and following a longitudinal cyclic pulse at 126 knots (for the two-bladed rotor with bar). The purpose of the maneuver was to evaluate the nature of the long-period phugoid that is characteristic of the pitch response of the helicopter. The pitch attitude exhibits both the standard short- and long-period responses. The short-period (2.2 seconds per cycle) oscillation is strongly damped; however, the long-period (38 seconds per cycle) oscillation is lightly damped. It is the latter oscillation which can become unstable when an aft center of gravity or hub restraint is used. Records taken at aft center of gravity to record this effect were not long enough to determine the damping; however, no difficulty in controlling the pitch attitude was reported. The roll oscillation can be seen to be unstable (a lateral cyclic correction was finally made). The left rolling tendency is apparent in the roll attitude trace. This tendency has also been observed at high speeds on a standard UH-1B and is believed to be the result of nonlinear aerodynamic coupling from the fuselage.

2. Dynamic Stability With Three-Bladed Rotors

- Longitudinal Control Pulse - The response of the high-performance helicopter with the three-bladed gimbaled rotor to a fore-and-aft pulse input is shown in Figure 27b. As was the case with the two-bladed system with no stabilizer bar, the roll mode of motion was found to be strongly unstable. This three-bladed configuration is as unstable at 103 knots as the two-bladed at 134 knots (see Figure 26b). This arises from the fact that the roll damping contribution of the rotor decreases as rotor solidity increases. The roll damping term has a strong effect on the nature of the lateral-directional or Dutch roll oscillation. The tendency of the helicopter to continue a left roll is evident from the roll attitude trace. A malfunction of the directional gyro prevented the recording of heading information; however, the pilot reported that only a minimum of yaw motion took place. The high directional damping and low roll damping causes the oscillation to appear mainly in the low-energy roll mode. The pitch attitude trace indicates an initial response following the fore-and-aft cyclic input and then follows a coupling pattern with roll. The helicopter pitches down as it rolls to the right, and up as it rolls to the left. This oscillation appears to have a period on the order of 10 to 15 seconds. The roll divergence to the left made recovery from the 30-degree left roll necessary.

The response of the three-bladed rigid rotor to a fore-and-aft pulse is shown in Figure 27c. The major oscillation of interest involves both pitch and roll. The oscillation, with a period of 15 seconds in both pitch and roll, is primarily the long-period pitch phugoid that is coupling into the roll axis through the mechanism of asymmetric rotor inflow during pitching and rolling, pylon and mast control coupling, and δ_3. That the pitch rather than the roll mode is causing the instability is based on the fact that the over-all stability was improved for the more forward center of gravity flights.

Control Response

1. Moment and Damping Ratios

Figure 28 presents a comparison of control power versus damping in roll for the rotor systems tested. Only the roll characteristics are shown since they represent the major area of interest from a handling qualities standpoint. Although the sharpness and duration of the majority of control inputs were not suitable for the exact extraction of these data, the general area of operation is known and so indicated. For reference, satisfactory, unsatisfactory, and desirable areas for hover from TND 58, Reference 11, are also shown.

The two-bladed rotor with stabilizer bar is essentially identical to the standard UH-1B since the slight increase in lateral control and inertia compensate each other. For the high-speed case, the decrease in roll damping causes the bar mode to become unstable as had been predicted, so that the net damping value becomes difficult to determine. The control power increases only slightly with speed since the rotor force vector increase with speed is the only effect tending to modify it.

Removal of the stabilizer bar decreases the roll damping and eliminates the short-period oscillatory nature of the roll response. The damping shown for the no-bar case is that value which is effective throughout the response. At increased speeds, the roll damping is near zero. Although this is not a desirable situation, the helicopter was easily flyable as had been predicted by simulator tests on the Bell dynamic flight simulator.

The three-bladed gimbaled rotor has slightly decreased damping because of its increased solidity, and decreased control power over inertia because of the increased roll inertia from the increased rotor weight. The high-speed point for the three-bladed rotor appears from flight to be essentially at zero damping.

The three-bladed rigid rotor was found to slightly more than double the control power of the gimbaled rotor. This value is based on the effective reduction in stick position necessary to trim a change in center of gravity in hover, and therefore includes the feathering washout introduced by pylon and mast deflection. Also included are the induced effects caused by the rotation of the wake surrounding the rotor in hover.

The lateral control gearing was reduced by approximately 25 per cent of standard for the first flight and by 40 per cent during the course of the flight testing to maintain satisfactory control sensitivity.

Since the basic control power was increased by a factor of two times the gimbaled configuration, the damping was also doubled. The decrease in roll damping with speed is not affected by hub restraint so that the high-speed point is the same increment below the hover point as was found with the gimbaled system.

2. Control Phasing

The three-bladed rotor system was designed with the rotor pitch horn oriented such that no rotor pitch-cone coupling would take place and the vertical gust response and coning stability would not be affected by feathering. The flexure for coning is outboard of the flapping axis of the blades in the gimbaled configuration; consequently, pitch-flap or δ_3 coupling is introduced. In the initial configuration, the δ_3 angle was 40 degrees.

A control phasing problem was encountered during initial tests with the three-bladed rotor in the gimbaled configuration. When the helicopter became airborne, a pitching and rolling oscillation was excited that appeared to be divergent, and the flight was rapidly terminated. Additional ground runs were conducted with control phasing changes (swashplate pickup retarded relative to the rotor). For further flight test of this rotor configuration, the pitch horn was changed to provide less δ_3 and the swashplate retardation was established at 19 degrees.

Model tests with a low Locke number rotor were conducted to evaluate the transient behavior of the rotor's thrust vector orientation with respect to the direction of a control input. The effects of δ_3, swashplate retardation, and pylon-control coupling were investigated. It was found that certain combinations of the above variables result in uncontrollable oscillations following a disturbance. In general, swashplate retardation was found to be the most important variable affecting the oscillation; and with sufficient retardation, a stable system resulted.

It can be concluded from the flight tests and model work that large values of δ_3 can be used with a low Lock number rotor if adequate swashplate retardation is provided. Although satisfactory control phase characteristics were obtained for the majority of the systems tested, considerable work is still needed in this area. Through the use of swashplate retardation, the relative angle between cyclic inputs and the resulting response of the helicopter can be controlled. Additional control phasing is introduced due to pylon motion, mast bending, pitch flap-coupling, and hub restraint. Many of the above variables were changed during the flight tests, and it was found that a pilot is very sensitive to control phasing.

From an acceleration standpoint, it is obvious that the phasing must be close, since only 15 degrees of improper input phasing will cause angular acceleration phasing to be off by 45 degrees. It is this effect that can very easily cause the helicopter to roll to some extent when only longitudinal cyclic inputs are introduced. The quantity being sensed, together with the time interval of importance in making the phasing "feel" right to the pilot, is not clearly understood. Indications are that the initial acceleration in the first few tenths of a second is most important. This is followed in importance by angular rate and displacement, which are effective after the time period on the order of a second.

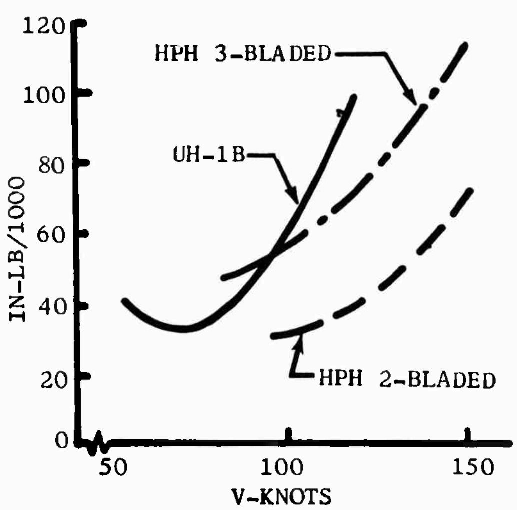
STRUCTURAL LOADS AND VIBRATION

Throughout the test program, data were recorded to determine the structural loading of the major components and also the cabin vibration levels. After each flight, these data were analyzed to determine that the loads had not exceeded the limits for safe operation and that the helicopter could safely enter a higher speed regime. These data also provided a basis for comparison of the analytically derived and measured loads. In general, the loads were found to be in close agreement with the loads predicted by the studies, thereby substantiating the validity of the analytical methods upon which the high-performance helicopter was based.

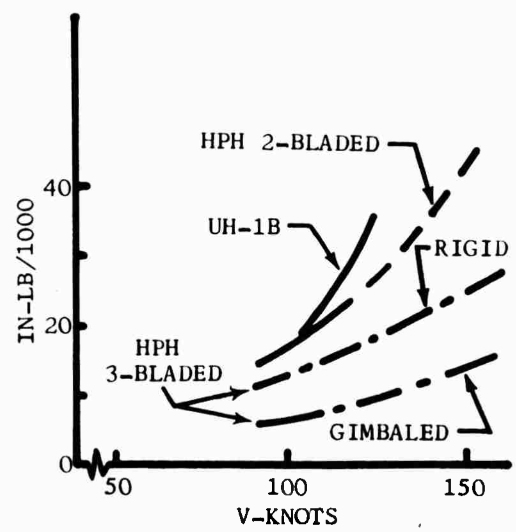
Main-Rotor Loads

The distribution of the loads along the blades followed closely the trends found on the UH-1B, and as a result only the moments at the rotor hub are presented since they are indicative for the entire rotor. Shown below is a comparison of the main rotor oscillatory chord and beam moments for the UH-1B and the high-performance helicopter.

CHORD BENDING MOMENT



BEAM BENDING MOMENT



The oscillatory loads of the two-bladed rotor show the influence of drag and download reduction by comparison with the standard UH-1B test results.

The three-bladed chord loads on the test vehicle are higher than the two-bladed but still lower than the loads on the standard UH-1B helicopter. The oscillatory beam moments of the two-bladed rotor are slightly lower than the UH-1B rotor loads. While the flexure in the three-bladed rotors reduced the moments for the gimbaled configuration considerably, the moments for the rigid configuration are higher than those of the gimbaled rotor because of the steady fuselage moments about the hub.

1. Two-Bladed Rotor

The measured oscillatory beam moments with the two-bladed rotor were in close conformance with those predicted by the analytical studies reported in Reference 3. A fuselage parasite drag of f = 10 square feet was assumed in the studies for two gross weights (6500 pounds and 7500 pounds). From Figure 29 it can be seen that the predicted and measured oscillatory beam moments are essentially the same. These moments occur primarily at a two-per-rev frequency.

Figures 30 and 31 show the measured and predicted oscillatory chord loads for the two-bladed rotor. Also shown on Figure 30 are similar data for the standard UH-1B. The principal frequencies of the oscillatory chord loads are one-per-rev and three-per-rev. The three-per-rev moments constitute about 30 per cent of the total chord moments. The reduction of the in-plane loads of the high-performance helicopter is due to the lower download and drag of the fuselage of that machine. The difference between the predicted and measured loads may be explained by the pylon mounting modification of the high-performance helicopter. It is known that pylon mounting influences rotor load due to the effective hub restraint it introduces in the plane of rotation.

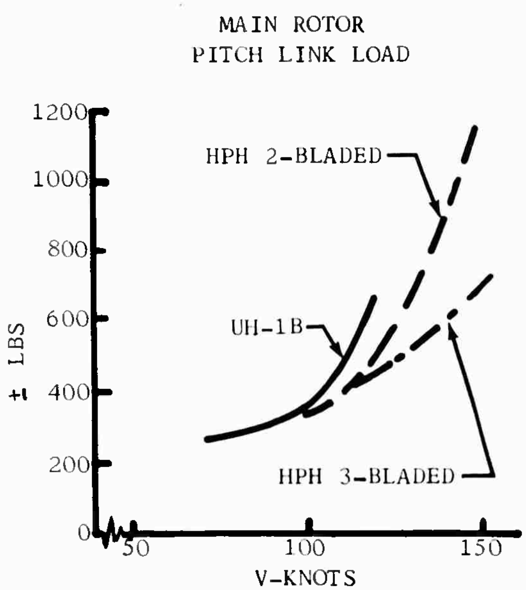
2. Three-Bladed Rotor

The oscillatory beam and chord yoke loads for the three-bladed rotors are shown in Figure 32. The in-plane moments for the rigid and gimbaled rotor are the same. Although the total oscillatory chord load was correctly predicted, it was found that the one-per-rev component was underestimated and the two-per-rev component was overestimated. This also indicated on the figure. The oscillatory beam moments are correctly predicted, and are primarily three-per-rev for the three-bladed gimbaled rotor. For the rigid rotor, a one-per-rev component is added, which is caused by the steady hub moments due to the fuselage center-of-gravity location and the combined fuselage and elevator aerodynamic pitching moments about the hub.

The predicted loads for the three-bladed rotor (Reference 3) did not include the effect of small changes made during the final design stage or of concentrated blade weights. The location of the concentrated weight in the rotor blade has a significant effect on the magnitude of the rotor beamwise moments as discussed in the section on problems (see page 33).

Control Loads

A general comparison of the control system loads is shown below. Flight test data are given by Figure 33. At the time of the high-performance-helicopter design study (Reference 3), no attempts were made to predict control loads. Since then, methods have been developed to calculate these loads. Trends found experimentally during the high-performance-helicopter program and reported in the following discussion were confirmed by theoretical results. However, much work remains to be done before satisfactory correlation between calculated and measured control loads will be obtained.



With the test vehicle, the rise in control loads occurs at higher speeds than for the standard machine. The reason for this is primarily due to the reduced fuselage download on the test vehicle, and consequent delay of blade stall effects. The additional improvement with the three-bladed rotor is due to the beamwise flexibility of the rotor yoke inboard of the pitch change axis. From these test data, it is concluded that control loads for high-speed rotors can be held within acceptable limits by the design of stiff blades which maintain their relationship to the pitch change axis.

Tail-Rotor Loads

The tail-rotor loads and flapping as compared with the UH-1B are shown in Figure 34. At lower speeds, the reduction of loads is primarily the result of the reduced main-rotor power required for the test vehicle. At the higher speeds (130 knots and above), the effect on the cambered vertical fin becomes significant in unloading the tail rotor. It is estimated that the amount of unloading by the fin is about 50 per cent at the maximum forward velocity. At the initiation of the flight tests, the standard tail-rotor yoke was installed on the test vehicle. During the program, it was found that the endurance limit of the standard yoke was exceeded at high speeds. The standard yoke was replaced with a shot-peened yoke which has an endurance limit 27 per cent higher than the standard. With the new yoke, the loads remained within the endurance limit for all flight conditions.

Cabin Vibration

The research vehicle fuselage is a modified YH-40 which had a considerably higher response to two-per-rev excitations than the production UH-1A or

HU-1B aircraft, and consequently a direct comparison of the test vehicle and UH-1B vibration levels can be misleading. Although the test fuselage exhibits a higher response than the UH-1B, particularly at two-per-rev, it was found that the research vehicle could be flown up to 30 knots faster than the production machines before encountering the same vibration levels. Figure 35 shows the vibration level as a function of airspeed at the pilot and copilot stations for the two-bladed and the three-bladed rigid rotor configurations. The sudden reduction of pilot vibration at high speeds with the rigid three-bladed rotor is difficult to explain, but it could be caused by a change in interaction of rotor and control force excitations. Figure 36 shows the vibration characteristics of the three-bladed gimbaled rotor and illustrates the influence of changes in loading on the fuselage response. Note that the higher gross weight condition had lower vibration levels. The vertical cabin response to three-per-rev rotor forces as deduced from the measured shear load in the hub flexures indicates $\pm$ .25 to .3 g per 1000 pounds oscillatory vertical force. This is approximately twice the value as determined by the shake tests conducted earlier, and indicates that other three-per-rev inputs such as oscillatory inplane shear loads, control system feedback, and downwash effects on the fuselage also are sources of cabin vibration.

PROBLEMS

Several problems were encountered during the program which merit further discussion. These are given below.

Hovering Incident

On 29 August 1962, while descending from out-of-ground-effect hovering, the pilot experienced a loss of directional control and an apparent loss of cyclic control. The helicopter yawed right and pitch nose-down, and descended and contacted the ground in a slight nose-down, rolled-right attitude. Other than a slightly yielded landing gear cross tube, no damage was incurred.

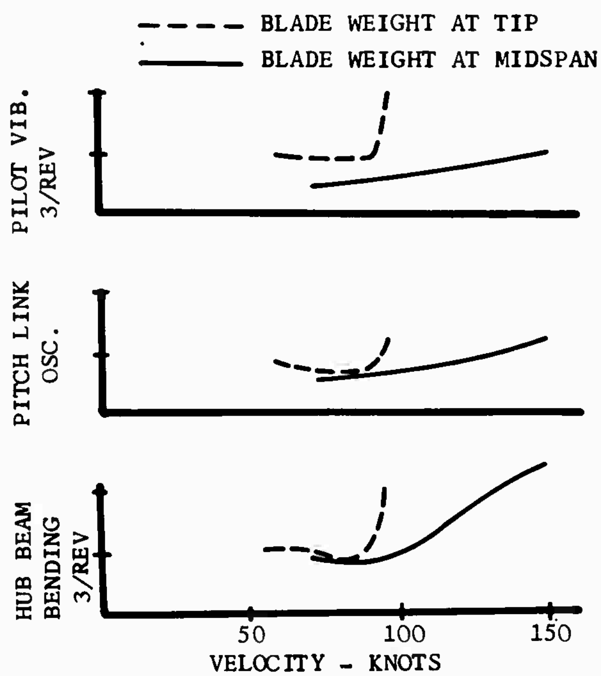
A thorough inspection of the controls and hydraulic boost system was conducted, and it was found that the tail rotor pitch was 2 degrees deficient for full left-pedal input. Disassembly of the tail-rotor pitch change assembly disclosed that the tail-rotor quill bearing was incorrectly installed and had resulted in a bearing separation failure which reduced the available directional control.

Further investigations were conducted to determine the cause for the apparent loss of cyclic control. The controls were proof tested and the hydraulic boost system was functionally tested, including a comprehensive laboratory test of the hydraulic pump. The cyclic system operation was found to be satisfactory; and although a momentary loss of hydraulic boost is believed to be the cause for the reported loss of cyclic control, no malfunctions of this system could be induced. The helicopter was then operated on tiedown for a thorough shakedown and evaluation of the control and hydraulic boost systems. During this period, several difficulties were experienced with the hydraulic boost system. High-frequency (pump one-per-rev) pressure variation resulted in several fatigue failures of the pressure lines. Control system operation with hydraulic boost "off" was marginal. A test was conducted with all hydraulic fluid removed from the system to simulate a hydraulic line failure, and the control system was found to be unsatisfactory due to oscillatory stick feedback forces.

As a result of these tests, it was determined that probable cause of the incident was a hydraulic system malfunction and that boost "off" control system operation was unsatisfactory considering a possibility of a hydraulic line failure. Accordingly, a standby, or secondary, hydraulic system as described below was designed and installed in the test vehicle. After installation of the secondary hydraulic system, no further problems were encountered with control system operation.

Beamwise Resonance - Three-Bladed Rotors

During the initial tests with the three-bladed rotor, it was found that a sudden increase in beamwise oscillatory (three-per-rev) blade loads occurred at a speed of approximately 80 knots, and also during deceleration to hover. As a quick flight test check, half of the tip weight was removed from each blade. This change was beneficial, allowing 100 knots speeds before build-up of the higher loads. A detailed computer study was conducted, and the results indicated that relocating the tip weight to midspan would prove beneficial. This was confirmed by flight tests as shown in the insert below.



EFFECT OF BLADE MASS DISTRIBUTION

Figure 37 is an example of the results of the computer study, which indicated that blade bending near the hub is reduced by adding weight at midspan. The study evaluated both parabolic and rectangular air-load distribution. Although the blade deflection curves change with the air-load distribution, the midspan weight location appears most desirable. For both assumed distributions, the midspan location lowers the second mode beamwise frequency and minimizes the magnitude of the vertical hub shear due to the three-per-rev blade beam bending. The change in blade mass distribution also resulted in corresponding reductions in vibration and control loads.

Chordwise Resonance - Three-Bladed Gimbaled Rotor

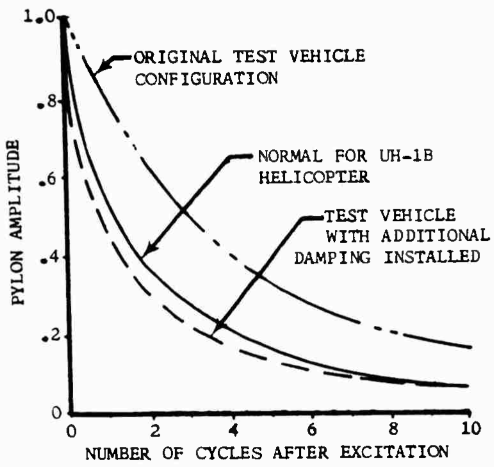
During hovering flights with the three-bladed gimbaled rotor, loads were measured which were considerably above the endurance limits of the various structural components. The high loads were the result of rapid cyclic stick inputs introduced while investigating control system rate limitations and pylon behavior. It was determined from records taken during these flights that the magnitudes and frequency of the cyclic stick excitations were 20 per cent fore-and-aft cyclic motion and 10 per cent lateral cyclic motion applied at 2.19 c.p.s. This excitation was near the natural frequency of the pylon system and coupled with the rotor speed of 5.17 c.p.s. to cause excitations in the rotating system at 7.38 c.p.s. This was very close to the natural (first cantilevered mode) chordwise frequency of the blades, and resulted in high moments and stresses.

The rotor was removed and inspected for structural damage; at the same time, the oscillograph records were reviewed for determination of the amount of fatigue damage incurred and to establish new limitations on component lives. From the damage analysis, it was determined that except for the drag braces, the rotor is satisfactory for additional use. Possible solutions to this problem include pilot technique and control rate limiting.

After this problem was discovered, a careful inspection of records taken during tests with the rigid rotor was conducted to establish if a similar situation existed. It was found that due to the rigid rotor influence on pylon frequency, the problem was peculiar only to the gimbaled configuration. With the rigid configuration, however, large cyclic control inputs while on the ground do cause large control moments and result in high mast stresses.

Pylon Damping

During initial tests, it was found that pylon damping was inadequate. It is believed that the design changes associated with the pylon tilting mechanism, such as the replacement of the aft rubber mount by a metal spring, resulted in a decrease in damping compared to the standard pylon configuration. A comparison of the UH-1B and the test vehicle pylon damping is shown in the insert on the following page. Friction dampers were installed between the upper transmission case and the cabin roof to



PYLON DAMPING

increase the pylon damping. The effect of this change, also shown in the insert, resulted in an acceptable degree of pylon damping. A slight increase in apparent damping was noted during flights with the hydraulic pylon actuators mechanically locked out, indicating that some reduction in damping was due to the small deflections of the pylon actuators under load. It was further found as the program progressed that the damping deteriorated. This slow deterioration in damping is believed to be the result of normal wear and looseness of the dampers and the pylon tilt mechanism.

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APPENDIX
TABLES AND FIGURES

TABLE 1

EFFECT OF FUSELAGE LOADING ON THE VERTICAL RESPONSE OF THE PILOT AND COPILOT STATIONS TO A 1000-POUND OSCILLATORY FORCE EXCITATION

Configuration	2/Rev Pilot $\pm$ g	2/Rev Copilot $\pm$ g	3/Rev Pilot $\pm$ g	3/Rev Copilot $\pm$ g
1. baseline	.15	.18	.09	.03
2. as 1 + 100 lbs. in tail	.13	.17	.08	.05
3. as 2 + 10 lbs. on each skid	.10	.15	.08	.04
4. as 3 + 2 x 100 lbs. under pilots' seats	.10	.14	.08	.05
5. as 3 + 200 lbs. removed from Station 4	.15	.20	.05	.04
6. as 2 + shotbags instead of oscillograph rack, add struts in cabin on inner rails	.15	.18	.08	.04
7. as 6 + remove struts	.15	.20	.10	.05
8. as 6 + move struts in cabin to outer rails	.15	.15	.10	.04
9. as 6 + replace shotbags with oscillograph rack	.15	.15	.10	.05
10. as 6 + remove 100 lbs. from tail and 200 lbs. from nose	.15	.15	.18	.18
11. as 6 + add 200 lbs. in nose and 100 lbs. to tail at Station 101	.15	.15	.15	.09
12. as 1 + struts on outer rails and cowling removed	.12	.13	.15	.10
13. as 12 + rubber washers at tailboom attachments	.12	.15	.13	.07

TABLE 2 HIGH PERFORMANCE RESEARCH VEHICLE PARASITE DRAG REDUCTION OVER UH-1B - SQUARE FEET (TWO-BLADED ROTOR - NO STABILIZER BAR)			
	Estimate (Reference 3)	Test Vehicle	
	130 Knots	100 Knots	130 Knots
Rotating Controls, Hub, Bar	3.4	3.2	3.2
Fairing Skid Gear	1.3	1.0	1.0
Engine Inlet	1.3	1.2	1.2
Protuberances	.2	1.2	1.2
Tail Rotor Gearbox Fairing	.2	.2	.2
Fuselage	.6	1.2	1.2
Download and Induced Drag	5.5	1.4	3.0
TOTAL	12.5	9.4	11.0

TABLE 3 HIGH PERFORMANCE RESEARCH VEHICLE CRUISE SPEED AND RANGE INCREASE OVER UH-1B GROSS WEIGHT - 6500 POUNDS		
Configuration	Cruise Speed Knots	Percentage Increase in Range and Cruise Speed Over UH-1B
UH-1B	109	-
Two-Bladed Rotor With Stabilizer Bar	124.5	14
Two-Bladed Rotor No Stabilizer Bar	130	19
Three-Bladed Rotor	121	11

TABLE 4

RANGE AND PRODUCTIVITY COMPARISON FOR TYPICAL MISSION

		UH-1B Basic	High Performance Helicopter Same Mission Time	High Performance Helicopter Same Mission Radius
LEAVE ORIGIN				
Basic Weight	lbs.	4880	4880	4880
Cargo	lbs.	800	800	800
Fuel (165 Gal.)	lbs.	1073	1073	915
Takeoff Weight	lbs.	6753	6753	6595
AT DESTINATION				
Elapsed Time	min.	66.2	66.2	55.5
Distance	n. mile	113	134.6	113
Landing Weight	lbs.	6246	6246	6163
Discharge Cargo	lbs.	-800	-800	-800
Takeoff Weight	lbs.	5446	5446	5363
RETURN TO ORIGIN				
Landing Weight	lbs.	4989	4989	4972
Reserve Fuel (10% Total)	lbs.	-107	-107	-92
Basic Weight	lbs.	4882	4882	4880
Total Time	min.	132.4	132.4	111.0
Total Distance	n. mile	226.0	269.2	226.0
Average Velocity	knots	106.7	128.0	128.0

FIGURE 1 - YUH-1B HELICOPTER

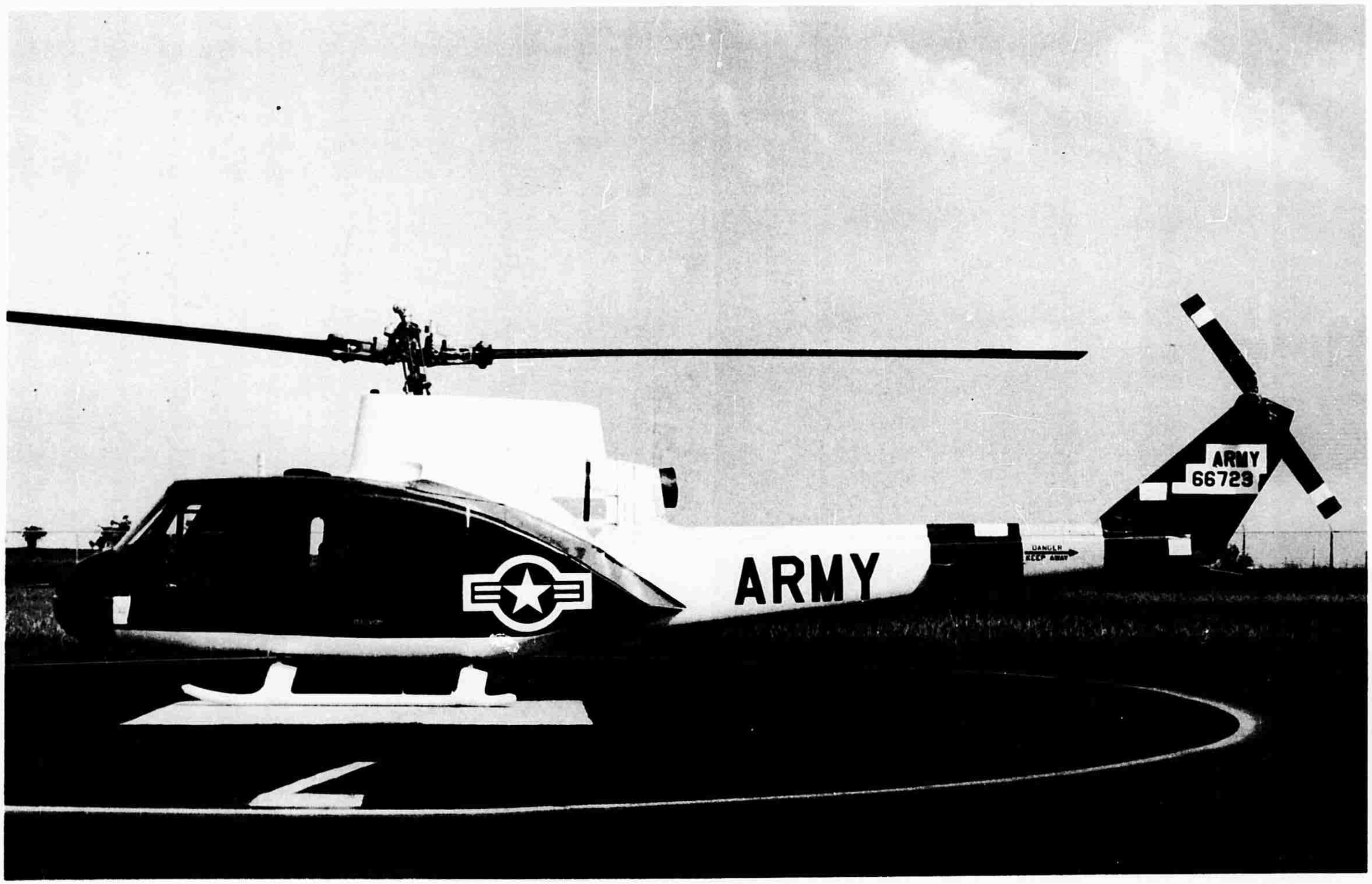


FIGURE 2 - HIGH-PERFORMANCE HELICOPTER

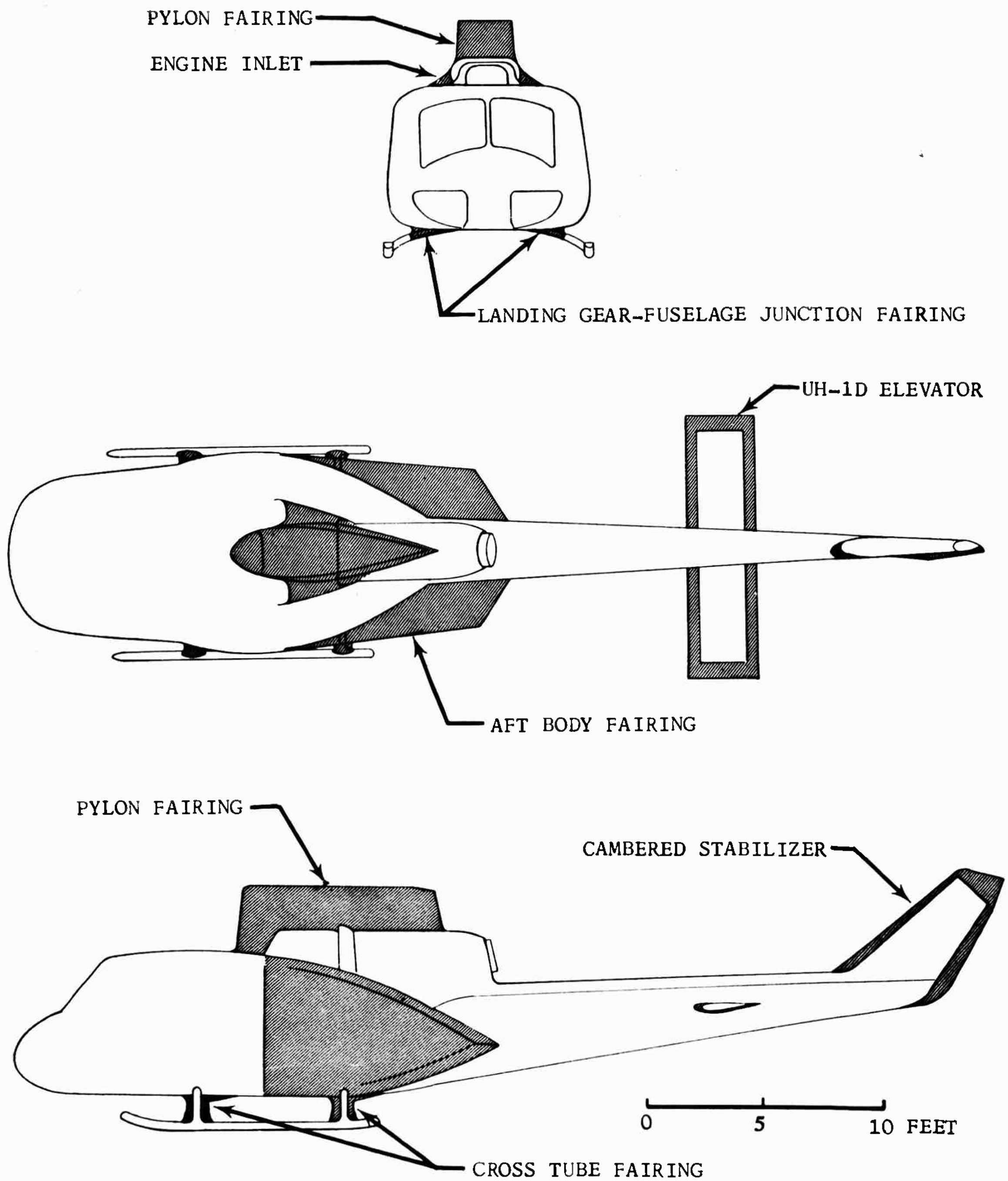


FIGURE 3 - CONFIGURATION COMPARISON OF HIGH PERFORMANCE AND UH-1B HELICOPTERS

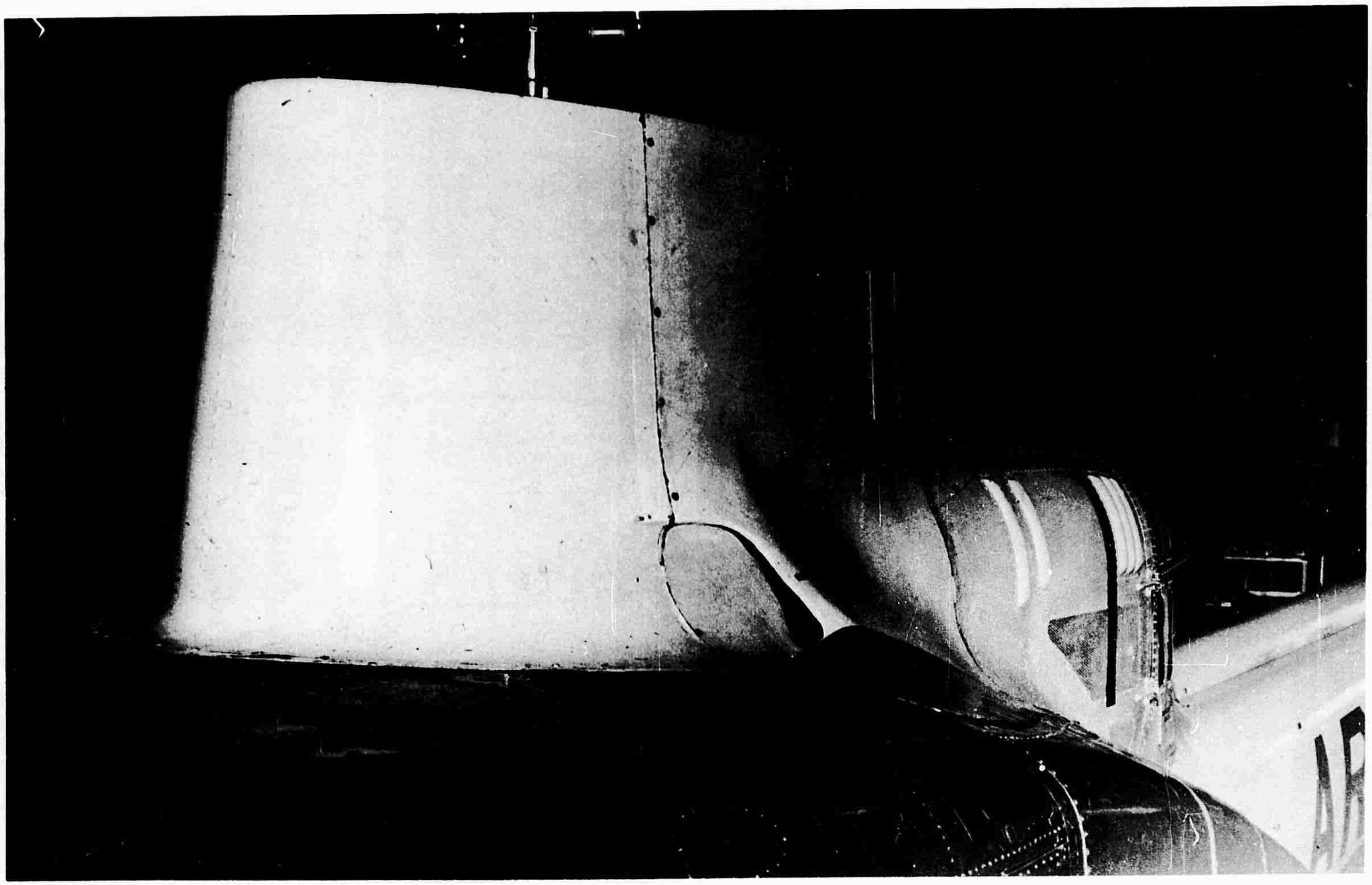


FIGURE 4 - PYLON FAIRING AND AIR INLET INSTALLATION

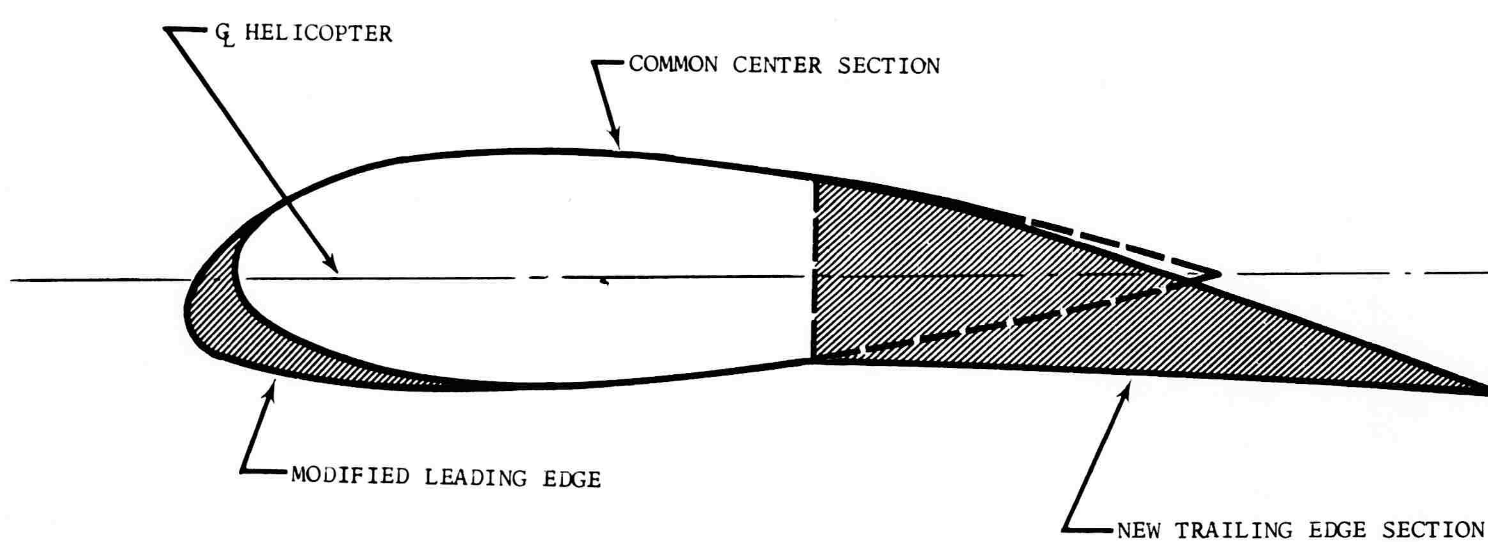


FIGURE 5 - VERTICAL FIN COMPARISON - HIGH-PERFORMANCE HELICOPTER AND UH-1B

FIGURE 6 - FAIRED LANDING GEAR WITH TAKEOFF WHEELS

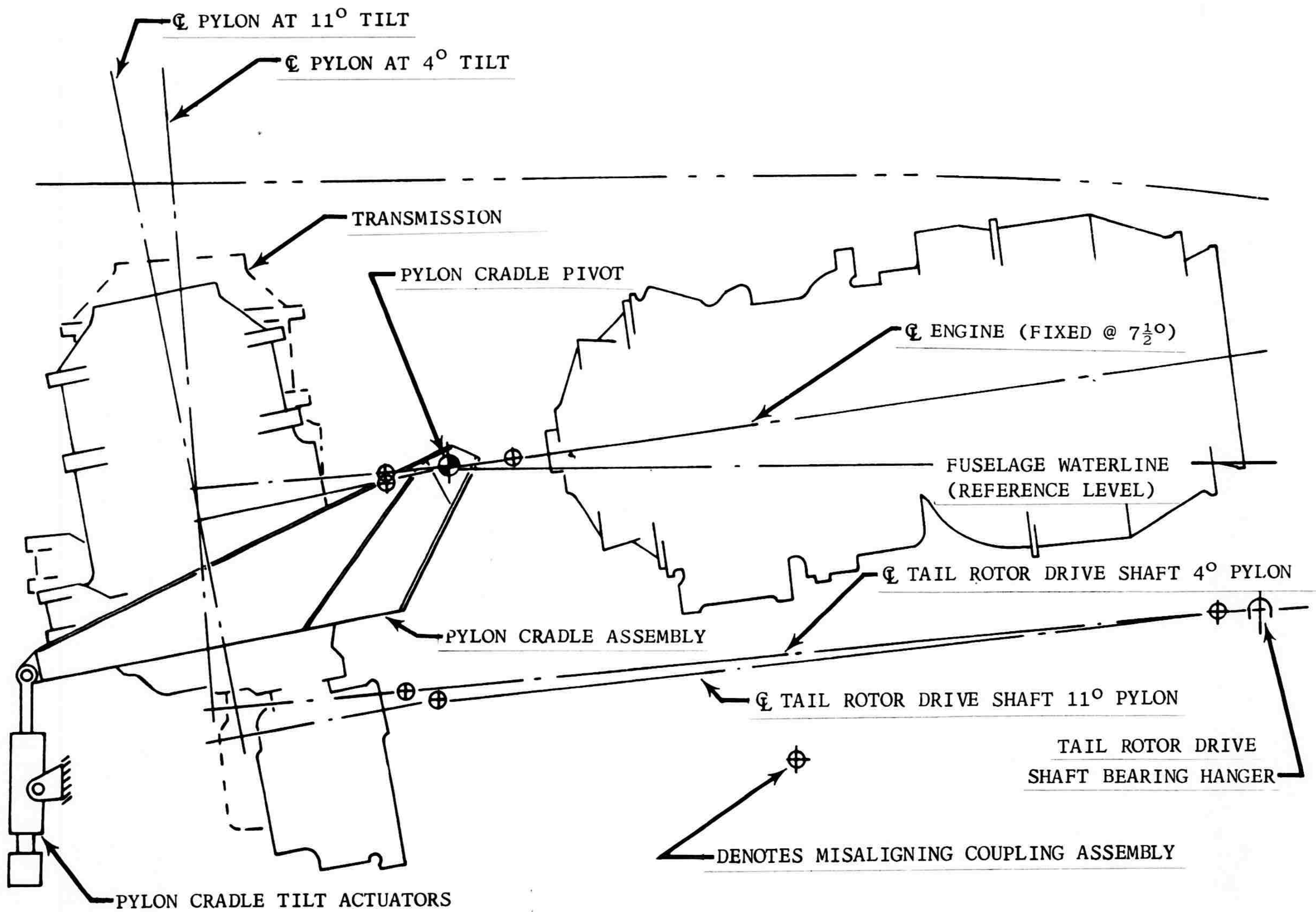


FIGURE 7 - GEOMETRY OF TILT PYLON SYSTEM

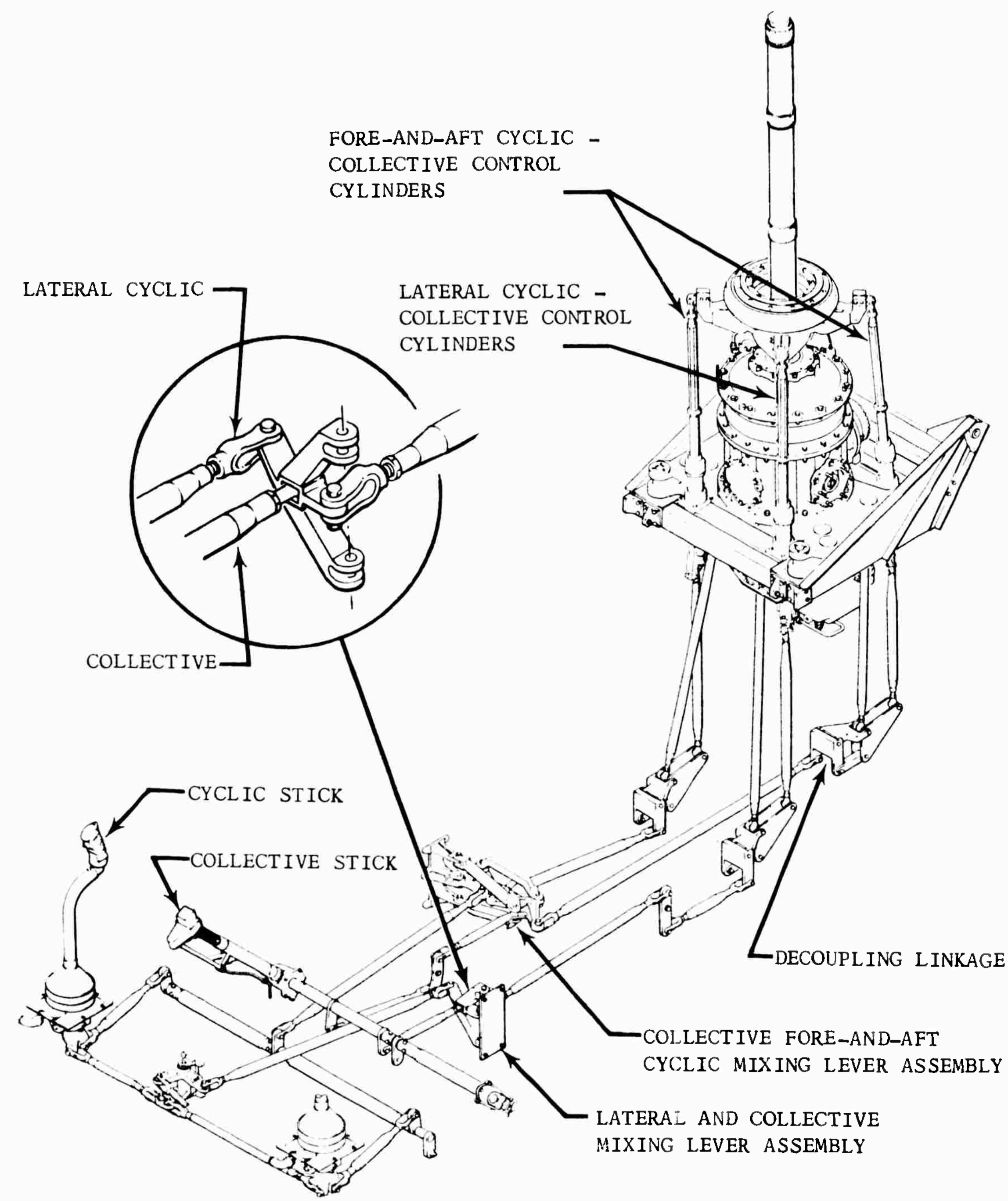


FIGURE 8 - HIGH-PERFORMANCE HELICOPTER CONTROL SYSTEM

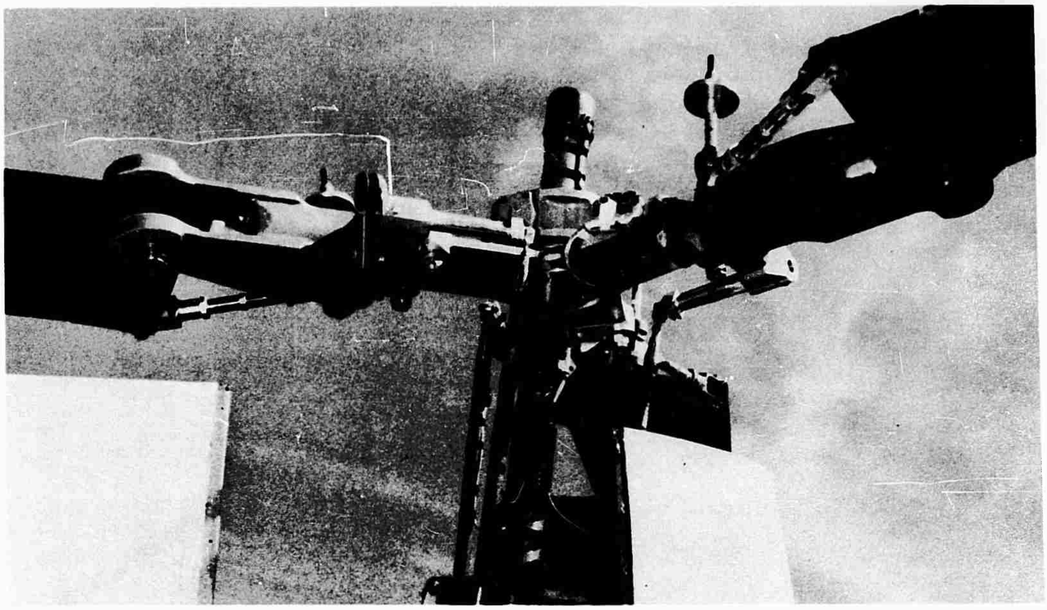


FIGURE 10 - THREE-BLADED GIMBALED MAIN ROTOR

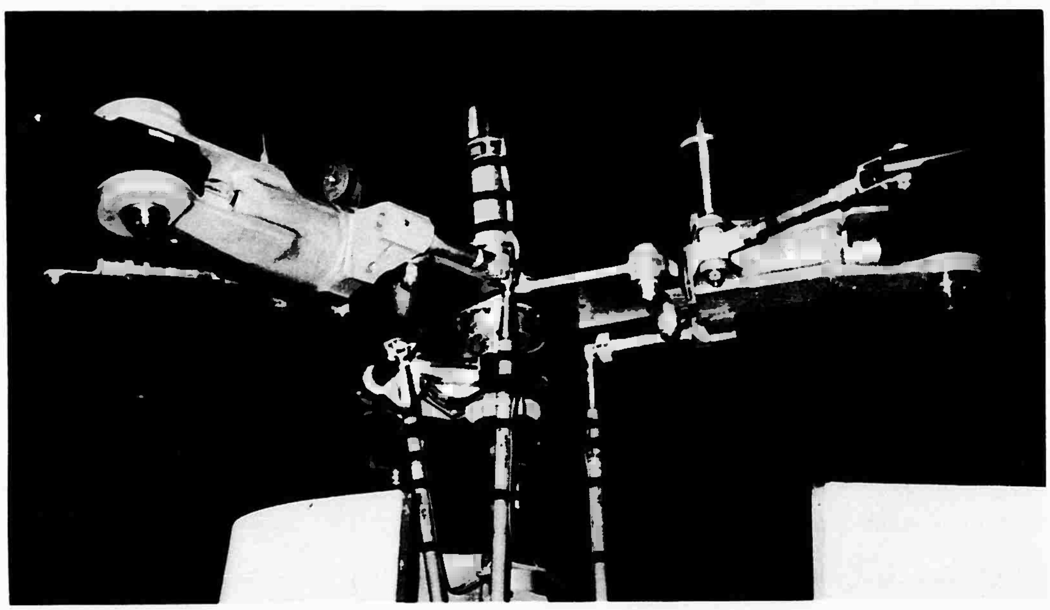
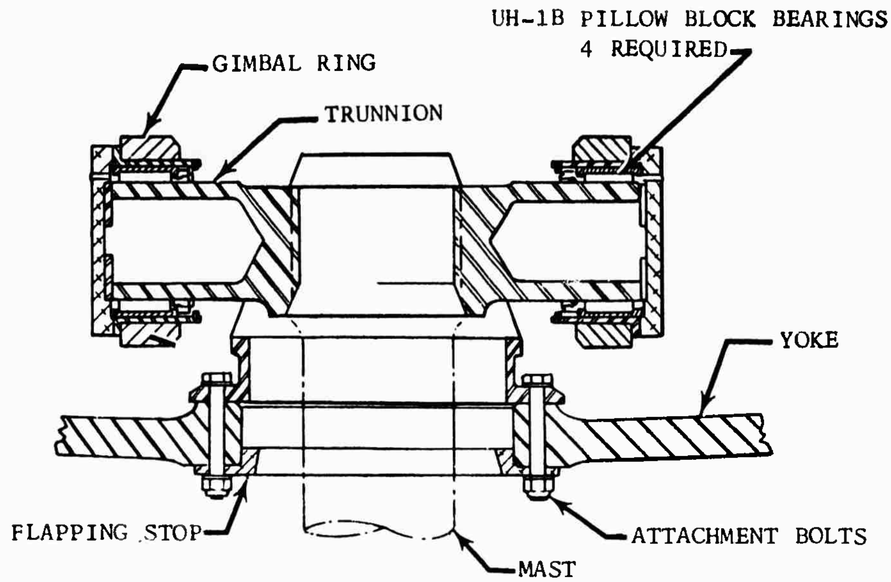
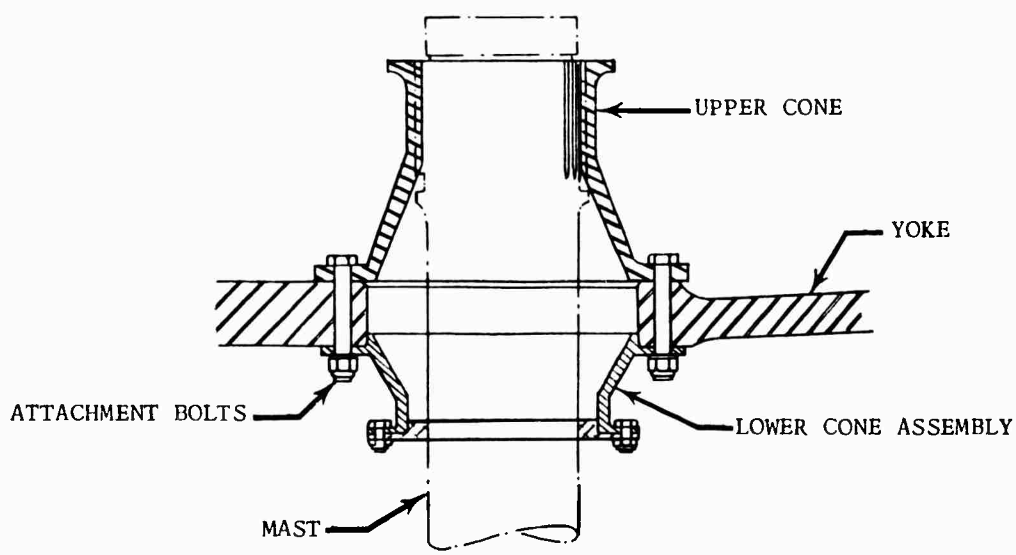


FIGURE 11 - THREE-BLADED RIGID MAIN ROTOR



A. GIMBALED ATTACHMENT



B. RIGID ATTACHMENT

FIGURE 12 - THREE-BLADED ROTOR-MAST ATTACHMENTS

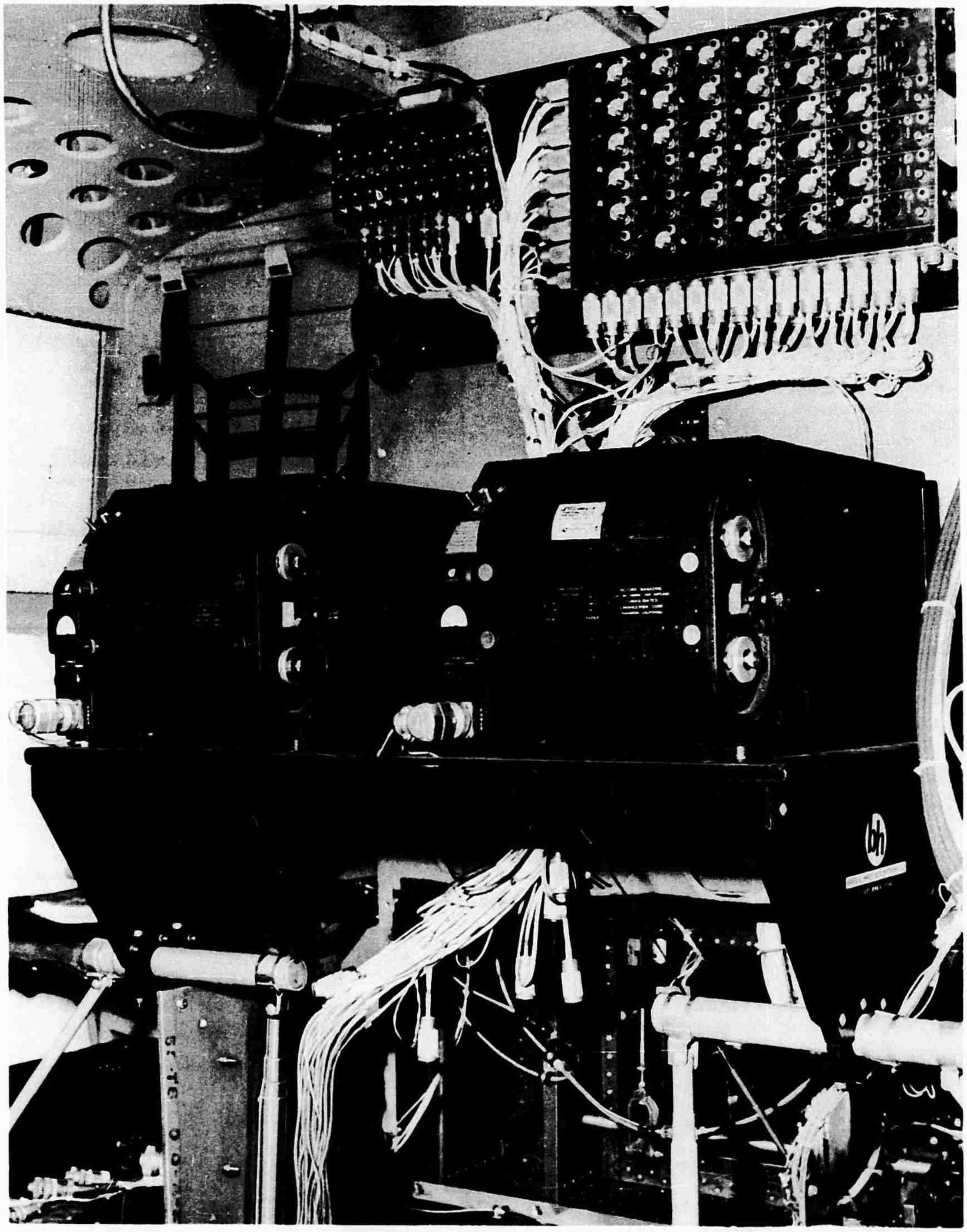


FIGURE 13 - INSTRUMENTATION INSTALLATION

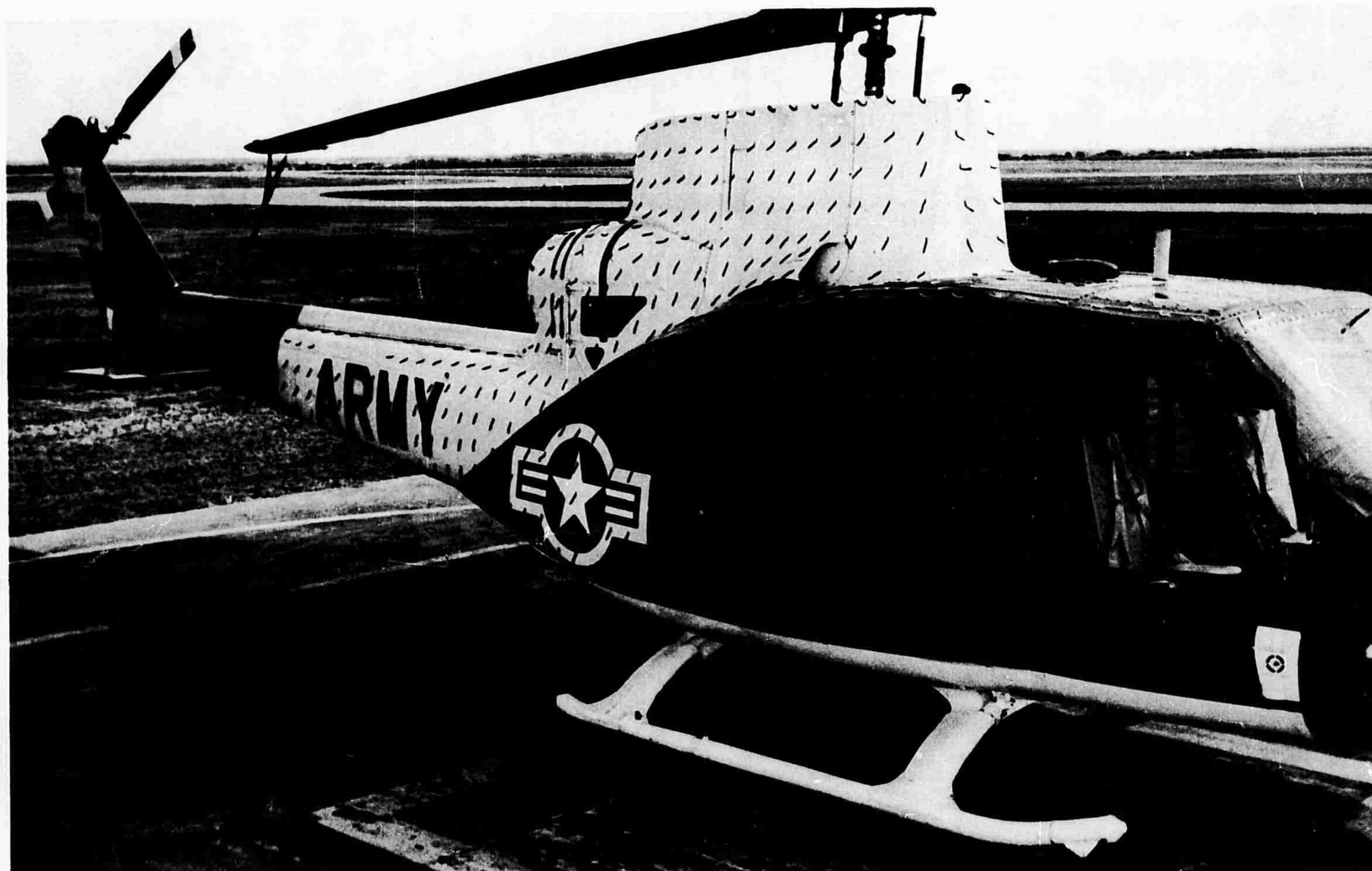


FIGURE 14 - TUFTED FUSELAGE

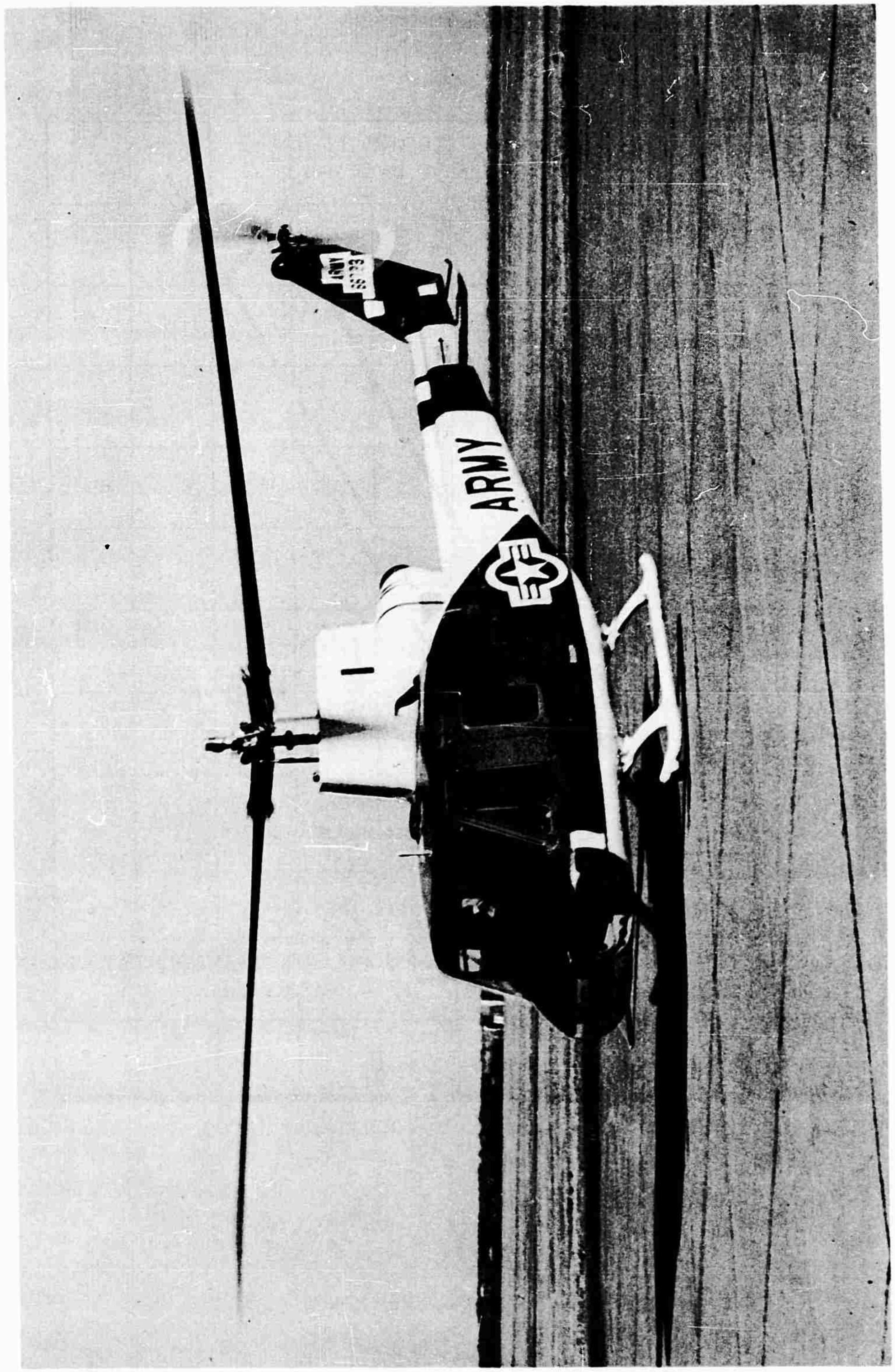


FIGURE 15 - OVERLOAD GROSS WEIGHT TAKEOFF TEST

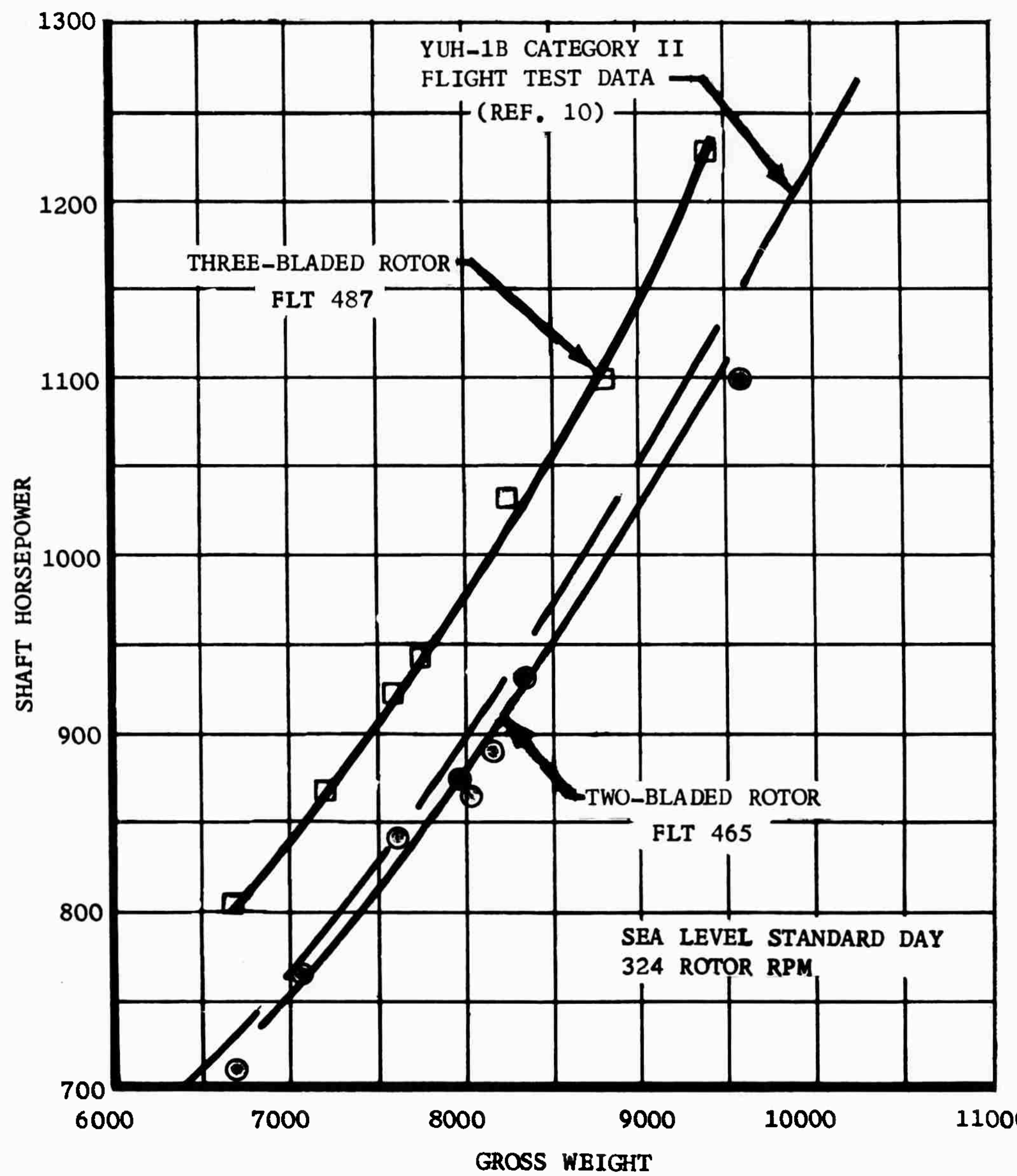


FIGURE 16 - HOVERING PERFORMANCE.

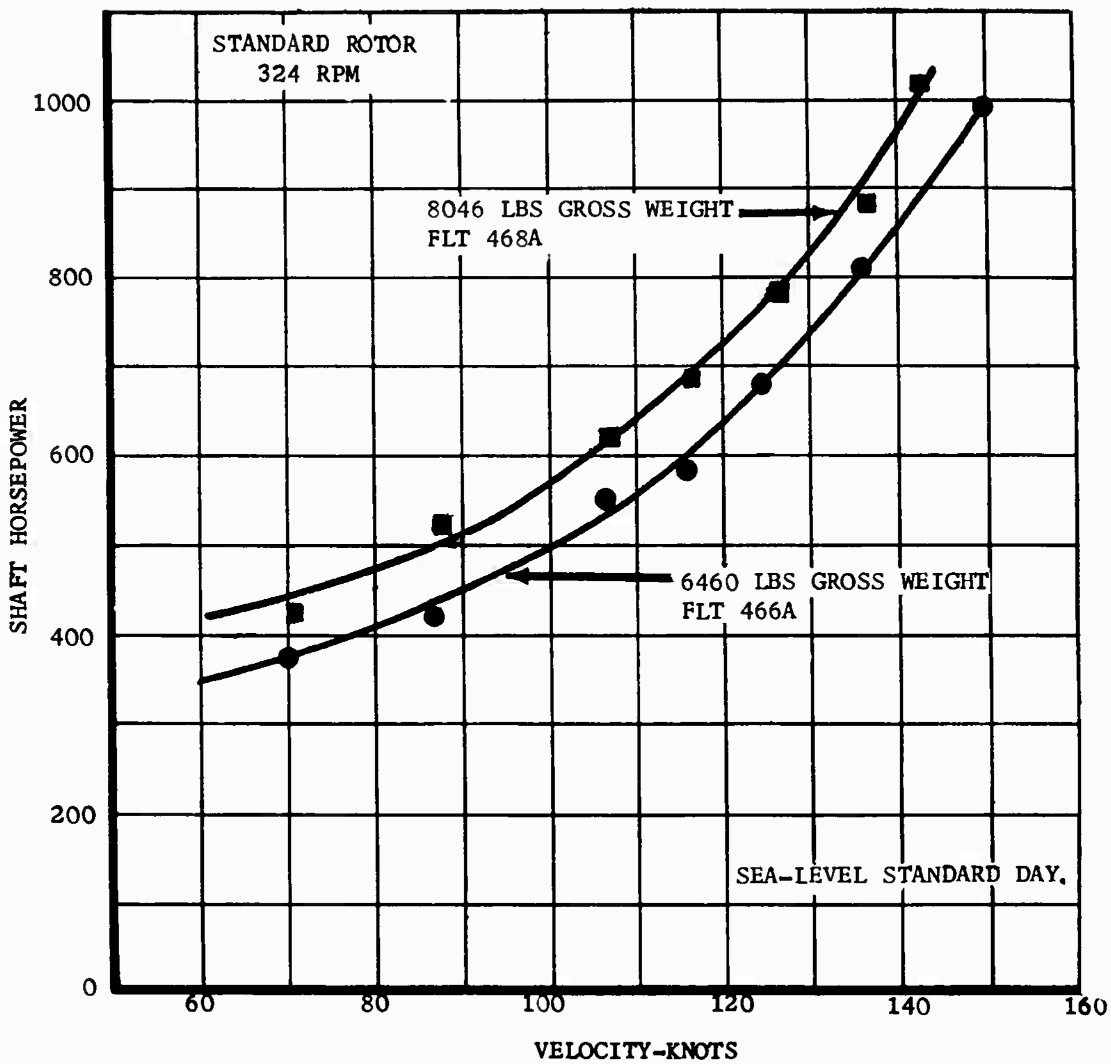


FIGURE 17 - LEVEL FLIGHT PERFORMANCE,
TWO-BLADED ROTOR

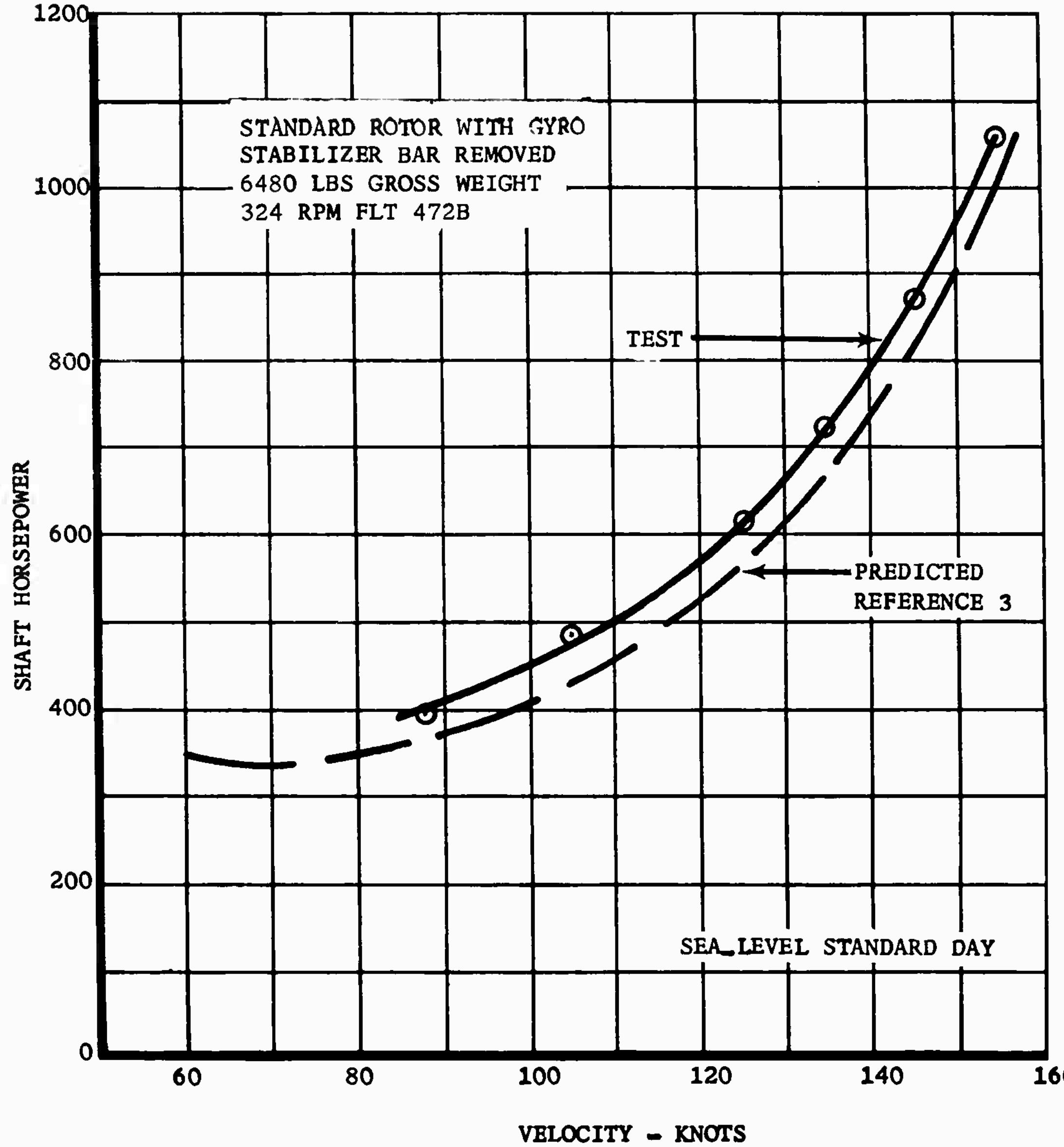


FIGURE 18 - LEVEL FLIGHT PERFORMANCE
TWO-BLADED ROTOR WITHOUT STABILIZER

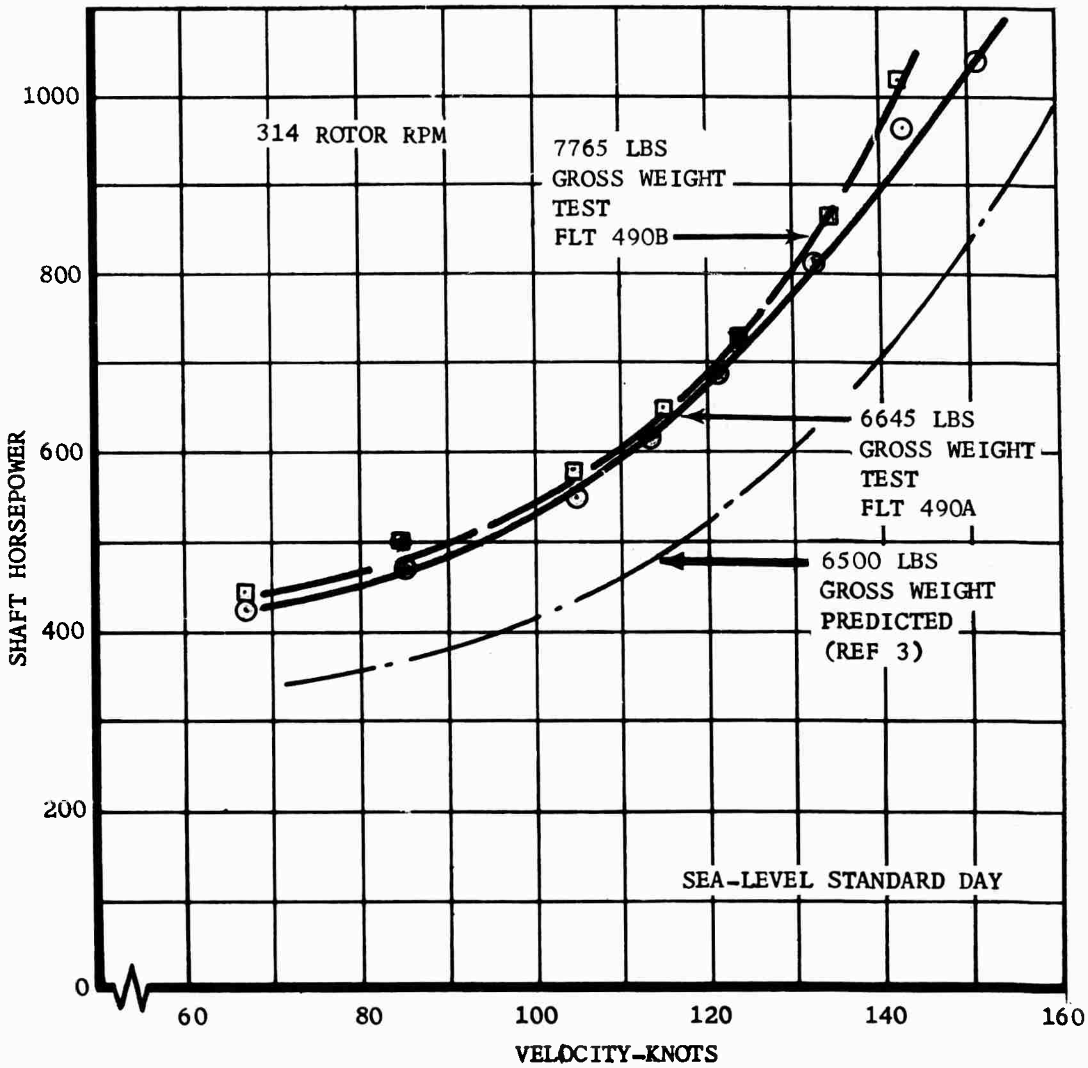


FIGURE 19 - LEVEL-FLIGHT PERFORMANCE, THREE-BLADED GIMBALED ROTOR

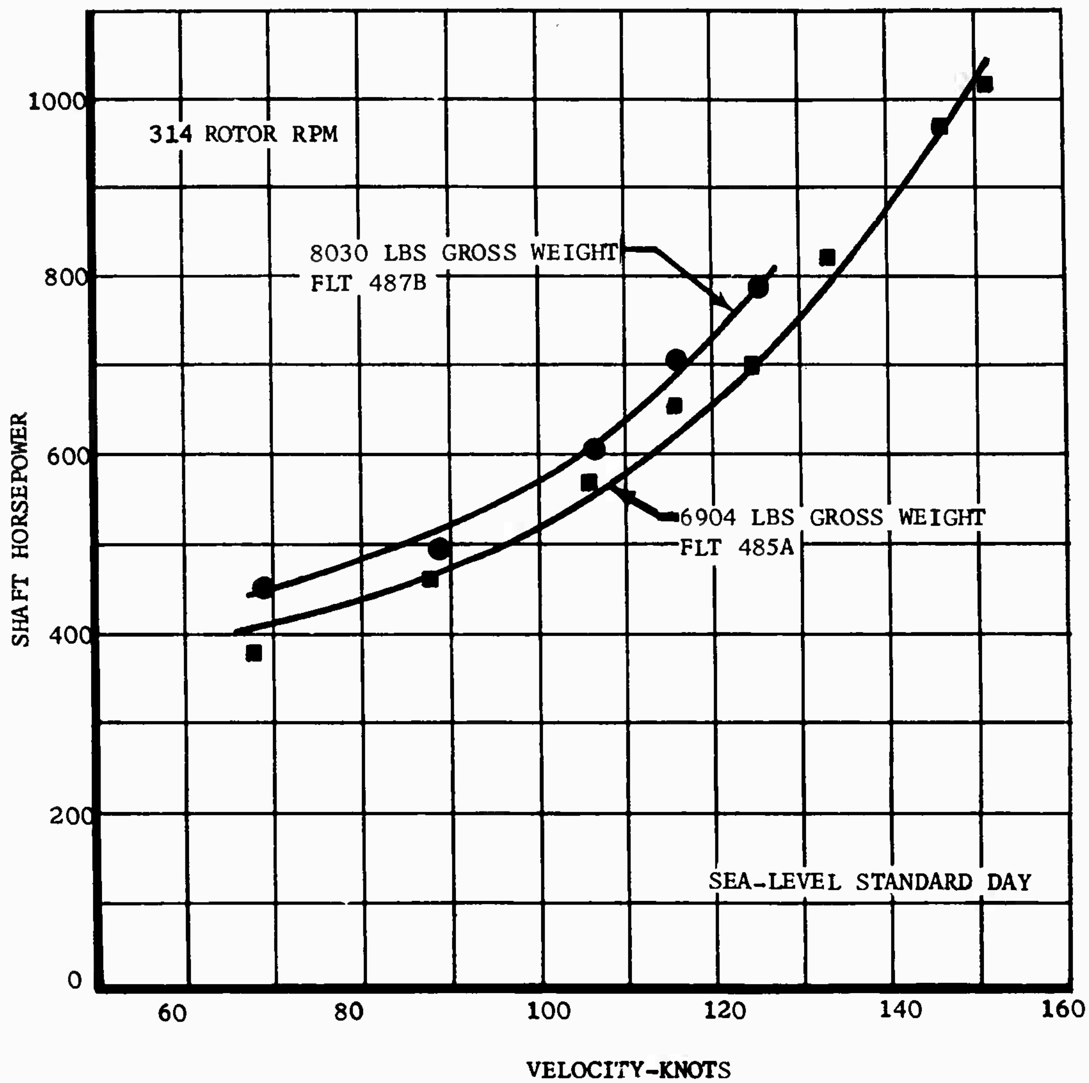


FIGURE 20 - LEVEL-FLIGHT PERFORMANCE, THREE-BLADED RIGID ROTOR

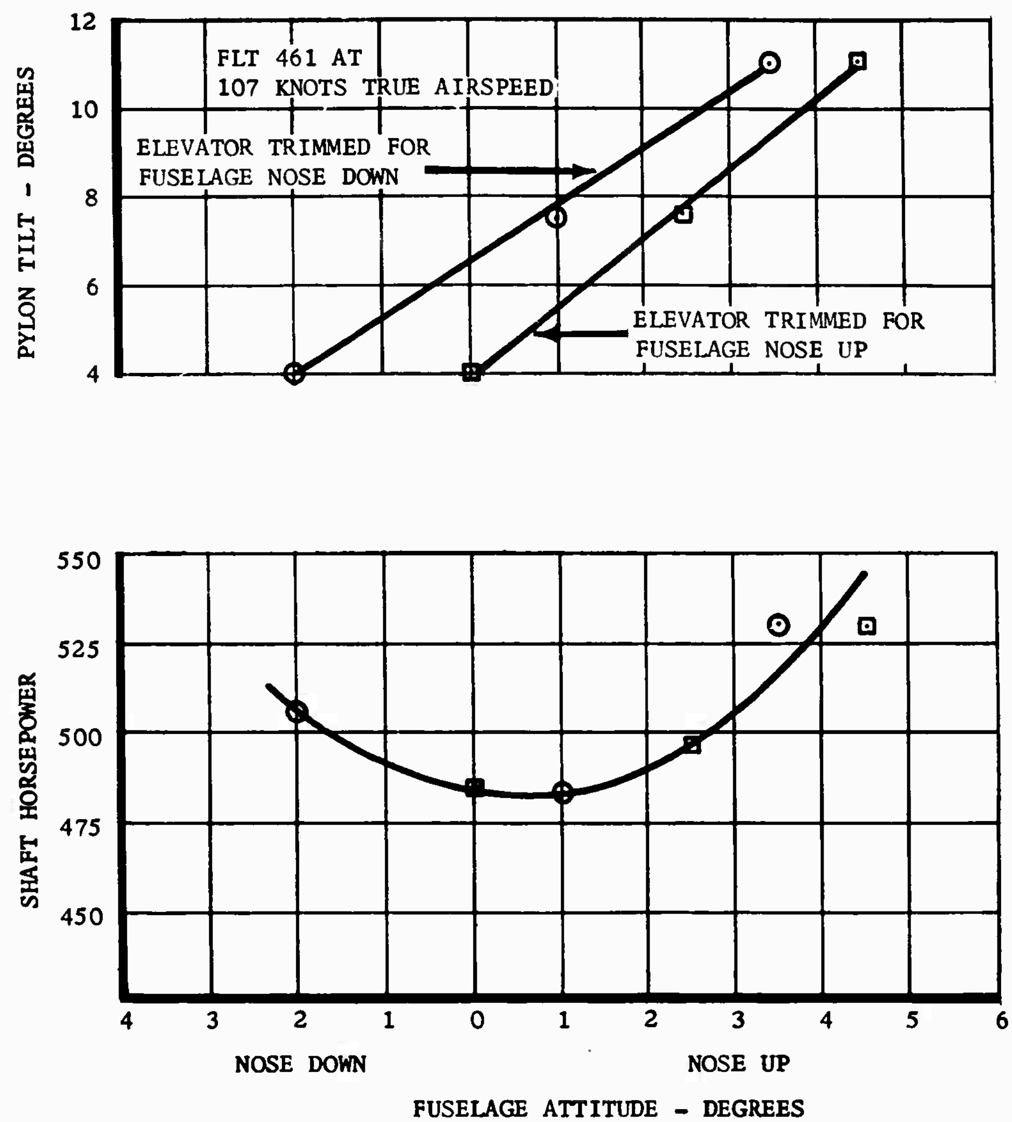


FIGURE 21 - POWER VS FUSELAGE TRIM

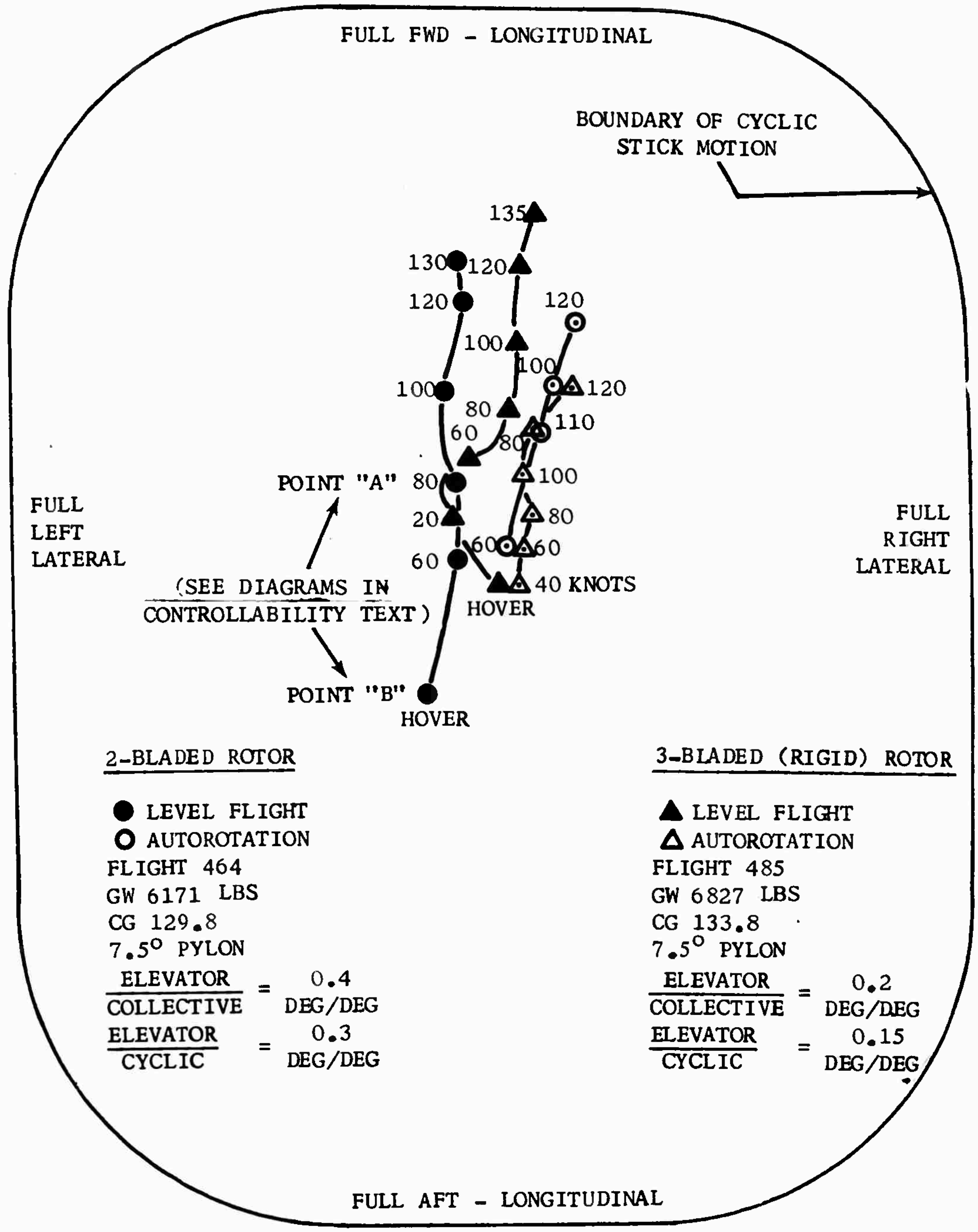


FIGURE 22 - SAMPLE STICK PLOTS

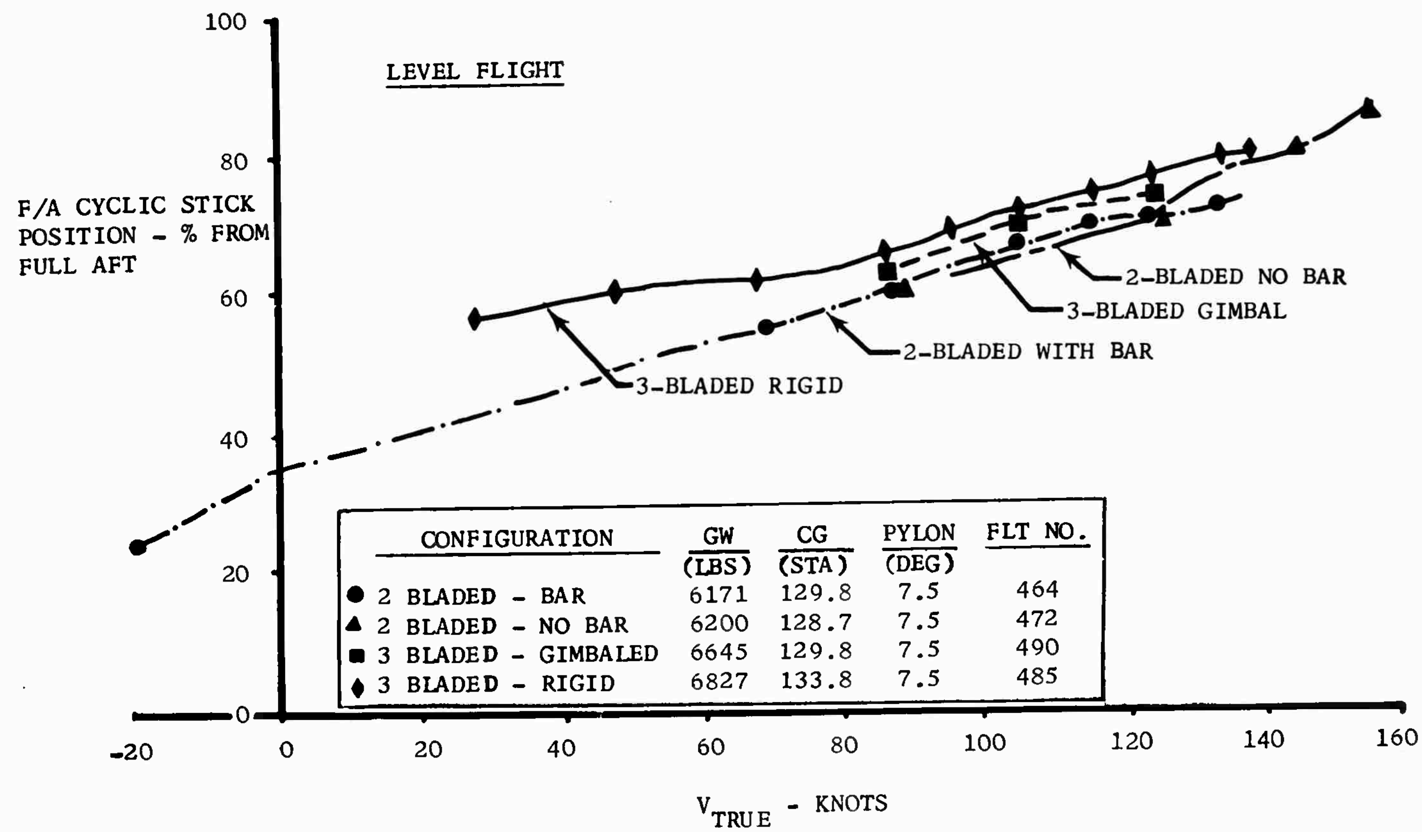


FIGURE 23 - LONGITUDINAL CONTROLLABILITY, LEVEL FLIGHT.

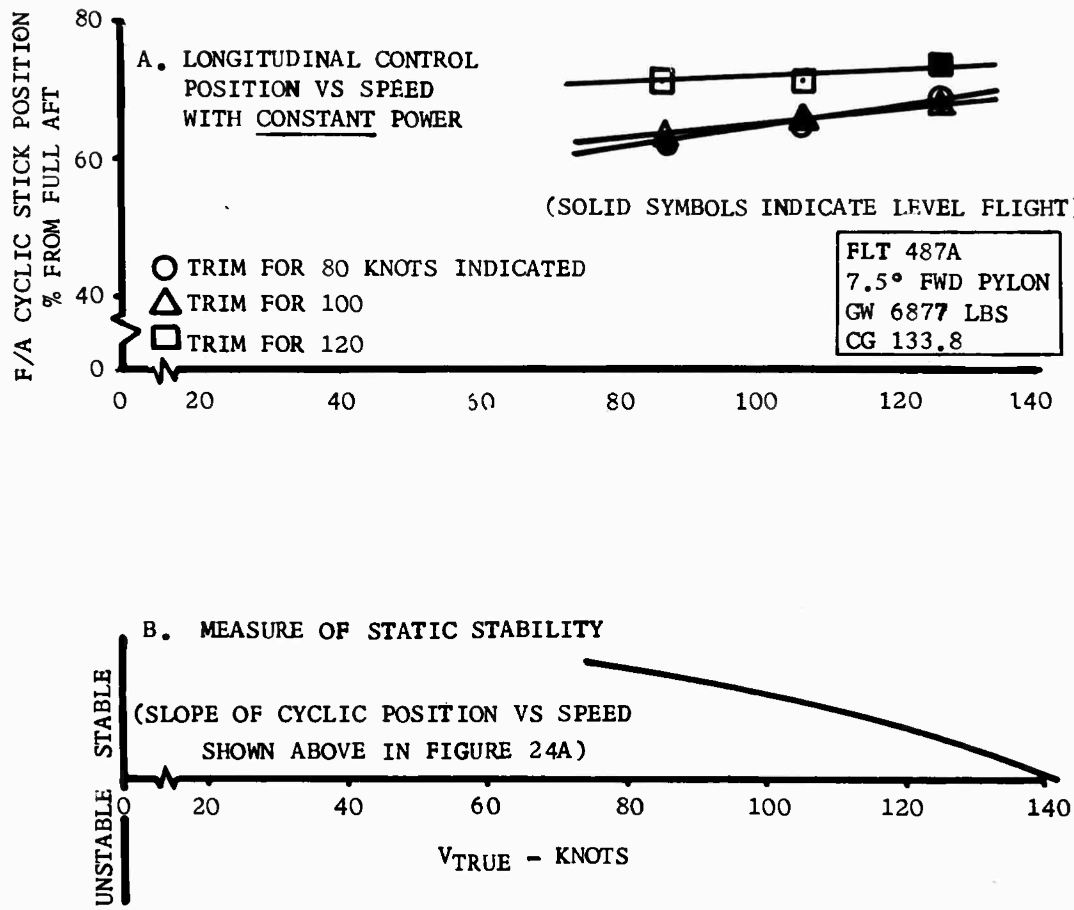


FIGURE 24 - LONGITUDINAL STATIC STABILITY CHARACTERISTICS OF THREE-BLADED RIGID ROTOR

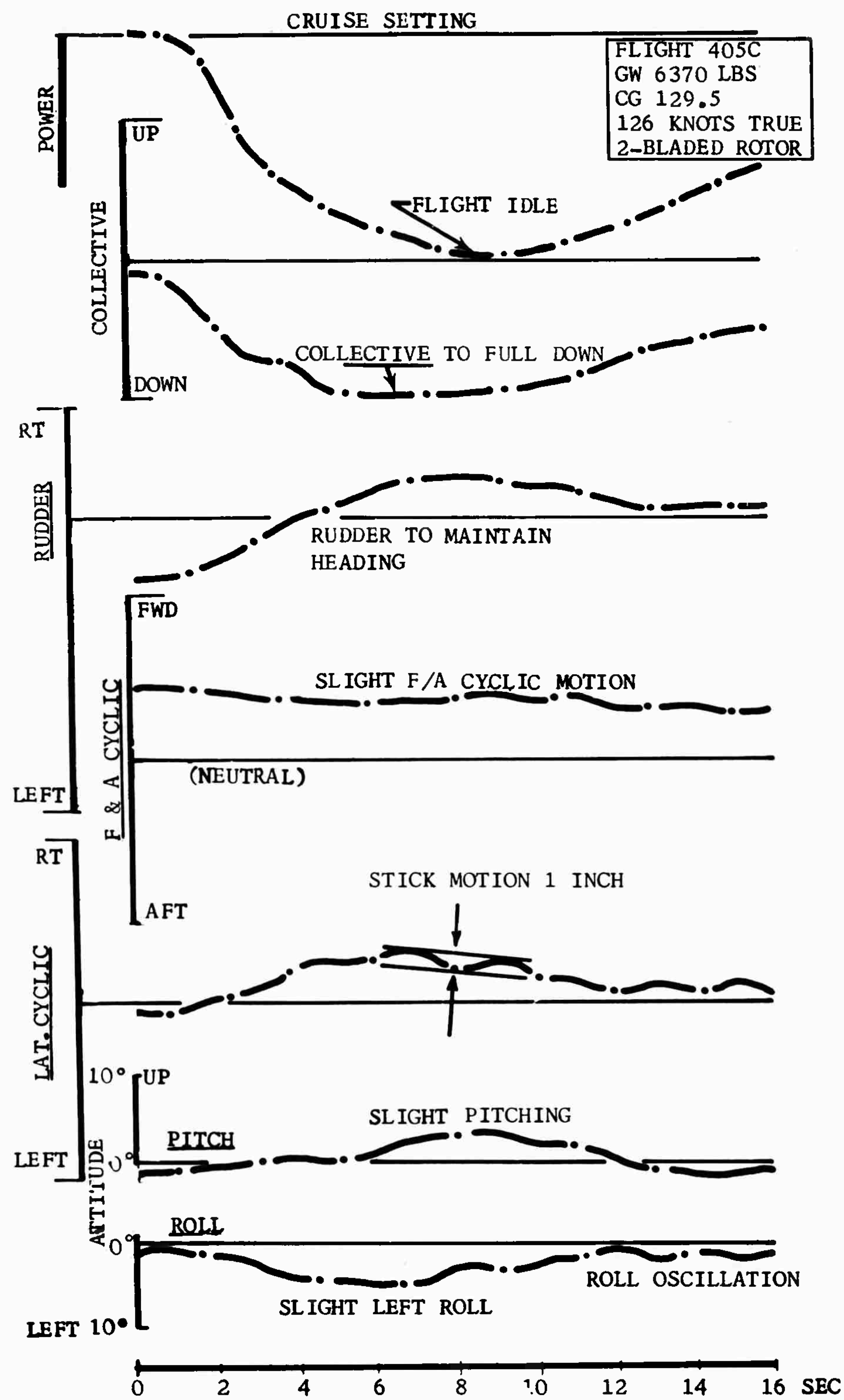
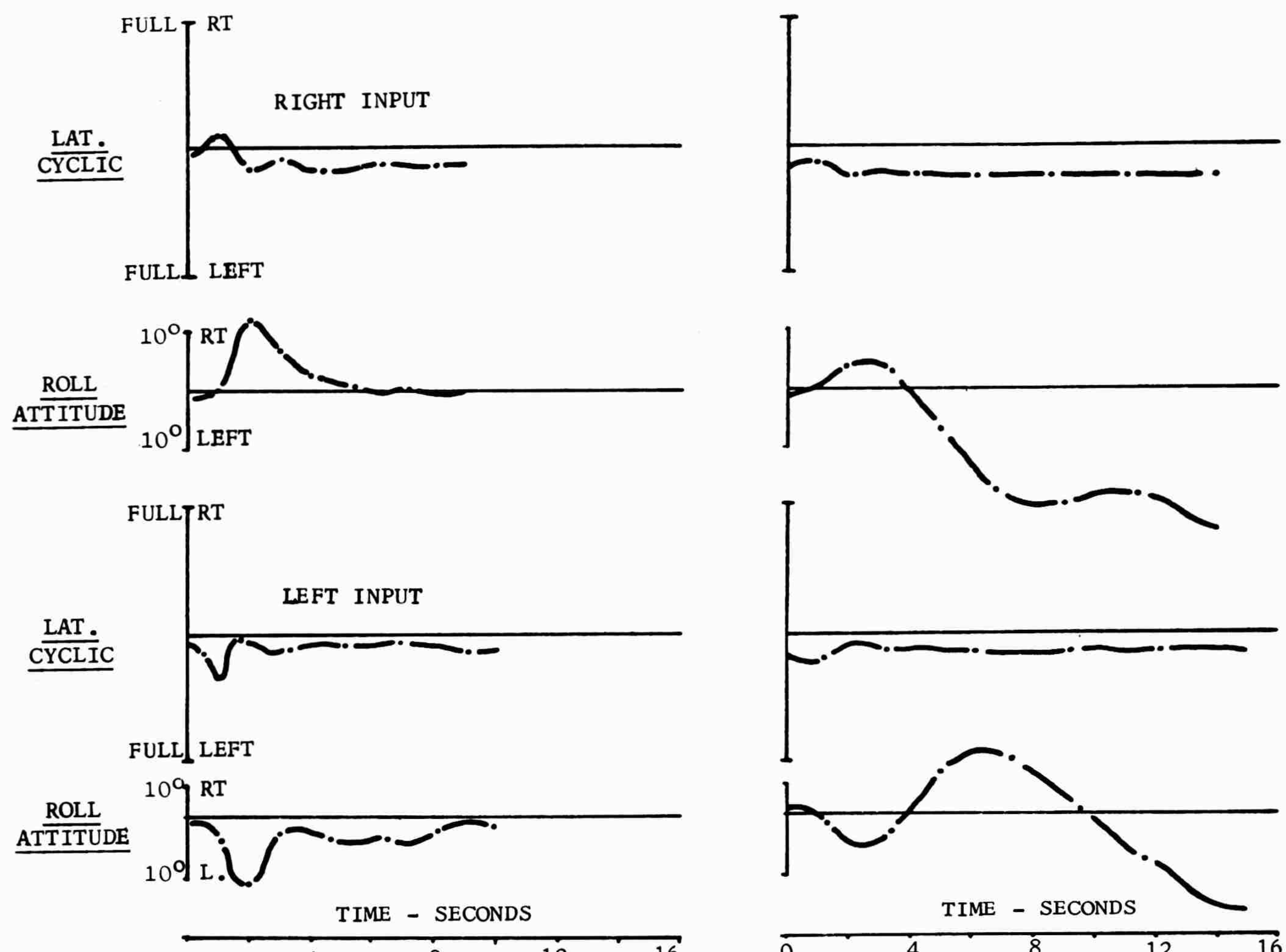


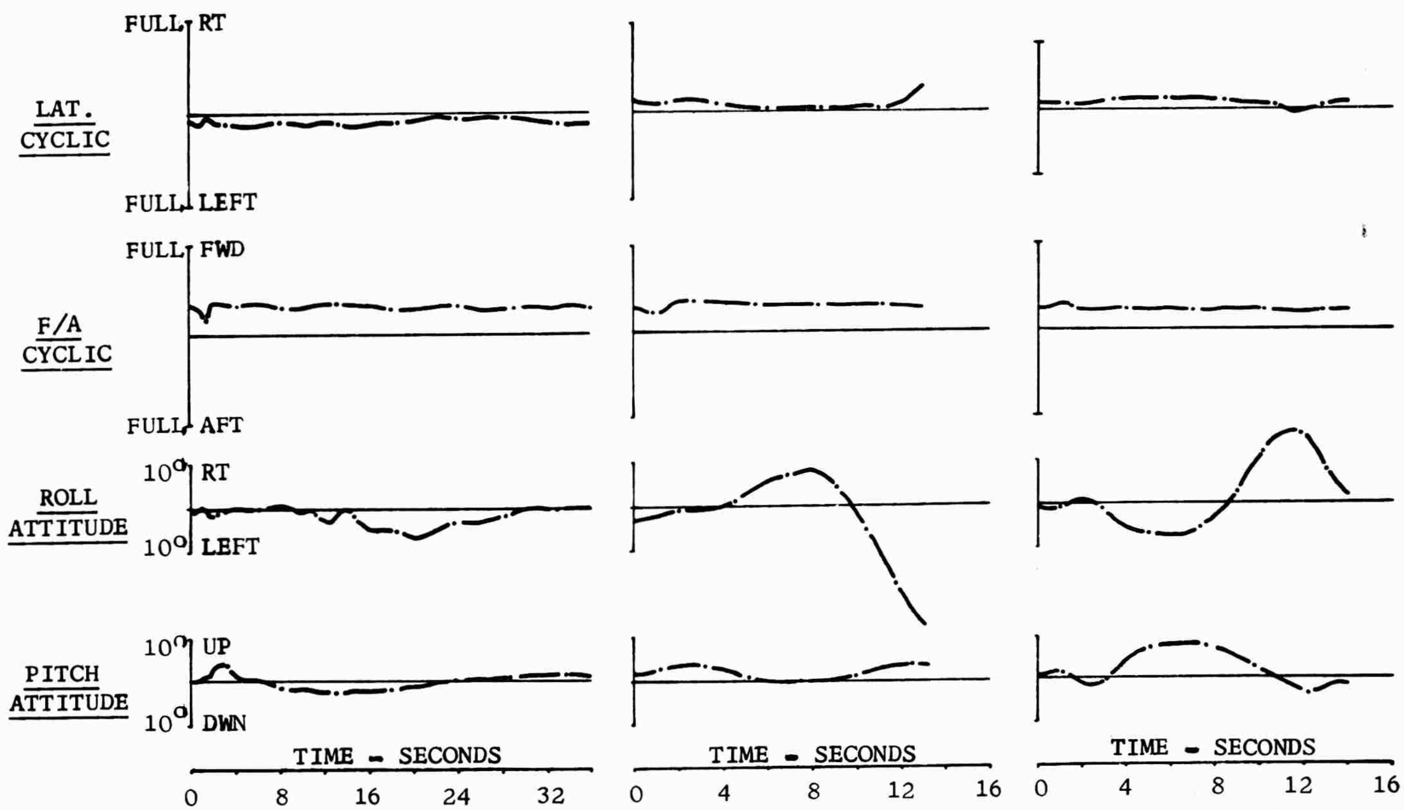
FIGURE 25 - TIME HISTORY OF AUTOROTATION ENTRY



a. 2-BLADED ROTOR - WITH BAR
132 KNOTS

b. 2-BLADED ROTOR - NO BAR
134 KNOTS

FIGURE 26 - DYNAMIC STABILITY - LATERAL CONTROL INPUT.



a. 2-BLADED ROTOR-WITH BAR
126 KNOTS

b. 3-BLADED GIMBALED
103 KNOTS

c. 3-BLADED RIGID
116 KNOTS

FIGURE 27 - DYNAMIC STABILITY - LONGITUDINAL CONTROL INPUT.

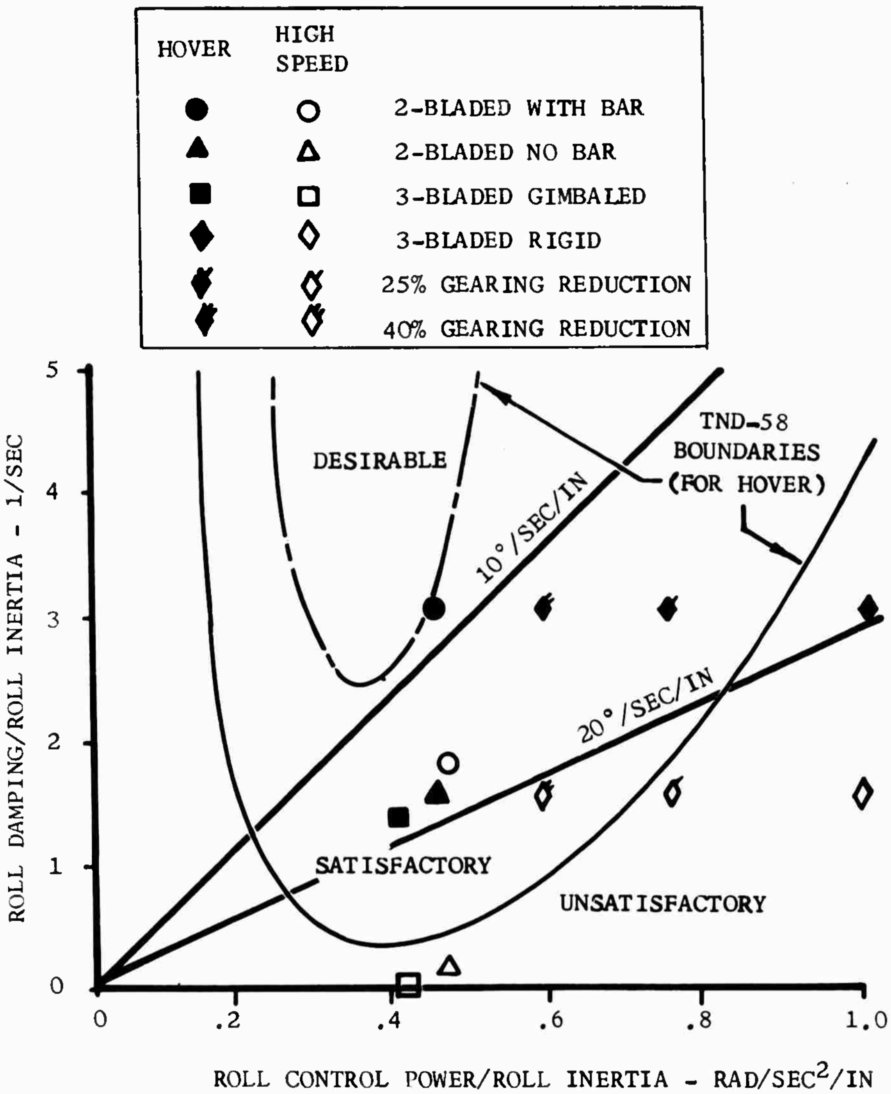


FIGURE 28 - CONTROL RESPONSE MAP

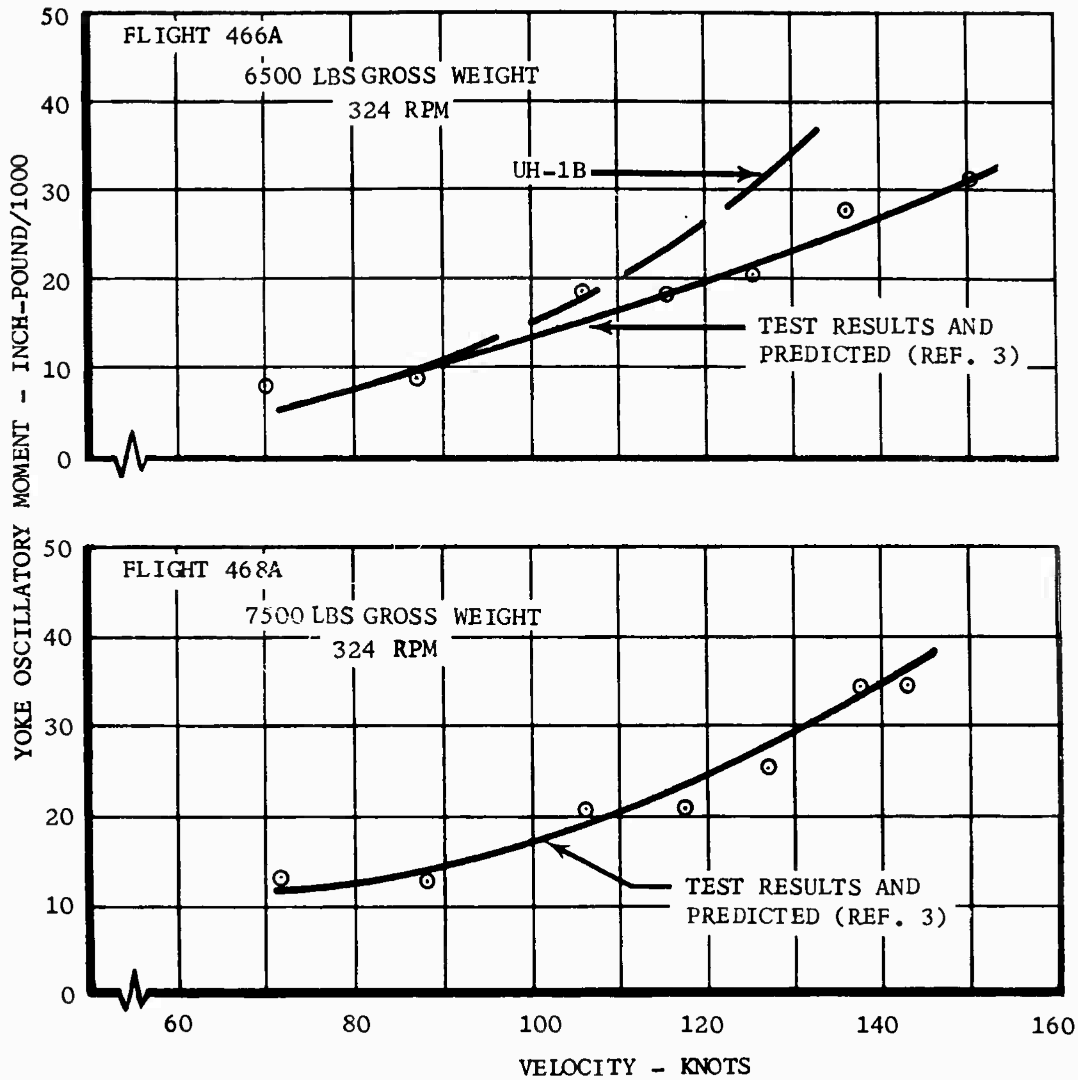


FIGURE 29 - MAIN-ROTOR BEAMWISE LOADS, TWO-BLADED ROTOR

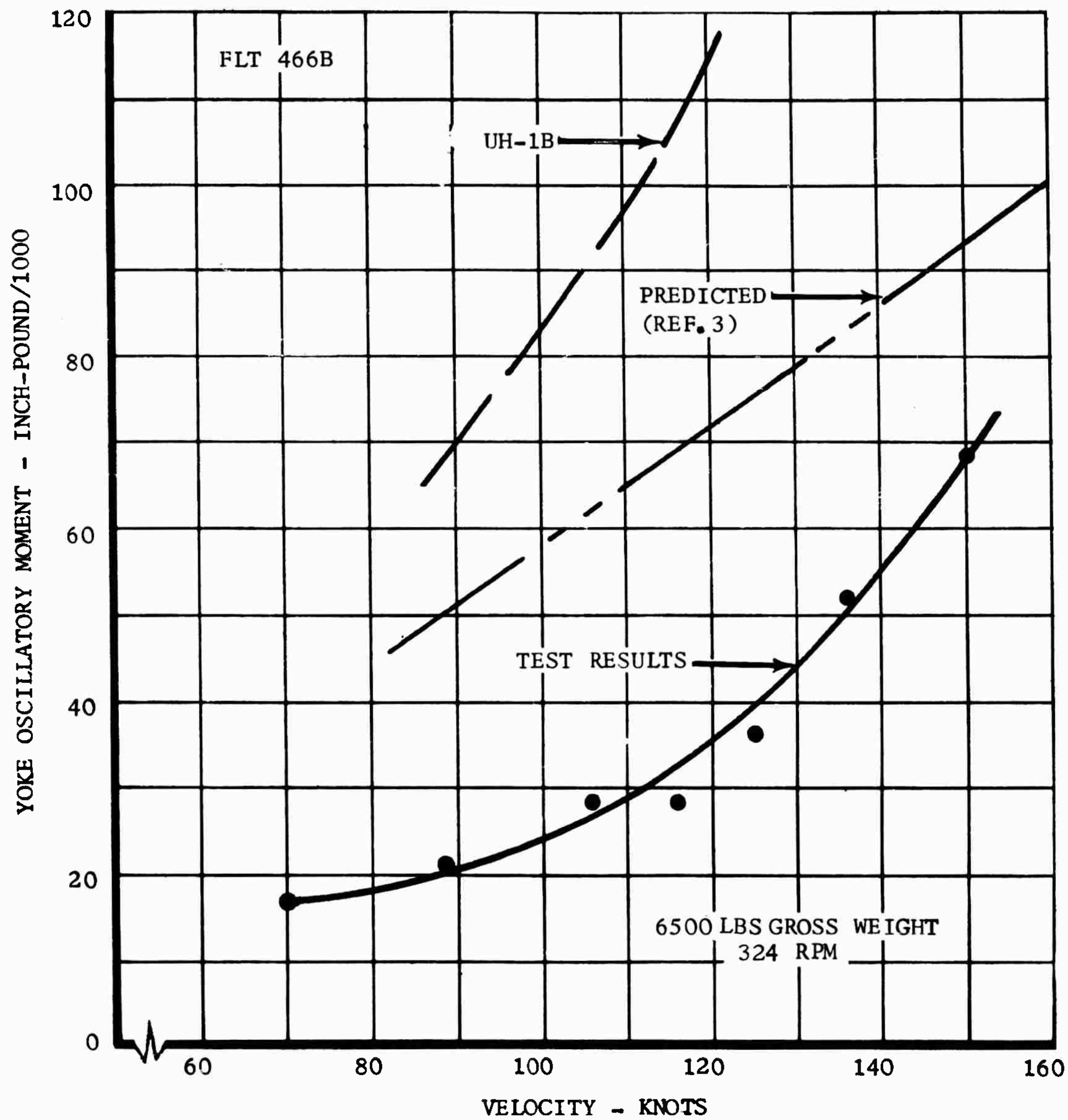


FIGURE 30 - MAIN-ROTOR CHORDWISE LOADS,
TWO-BLADED ROTOR.

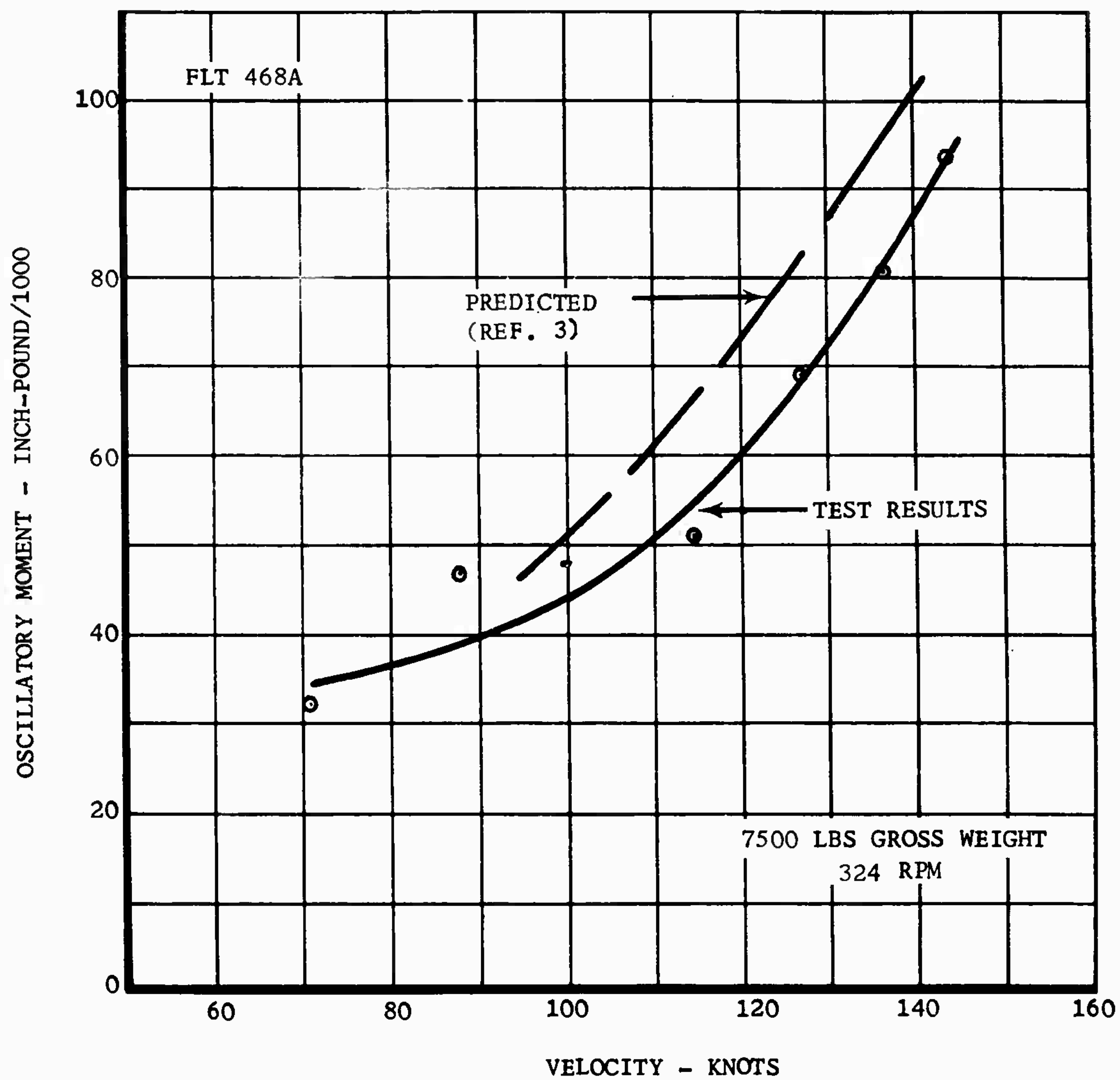


FIGURE 31 - MAIN-ROTOR CHORDWISE LOADS. TWO-BLADED ROTOR.

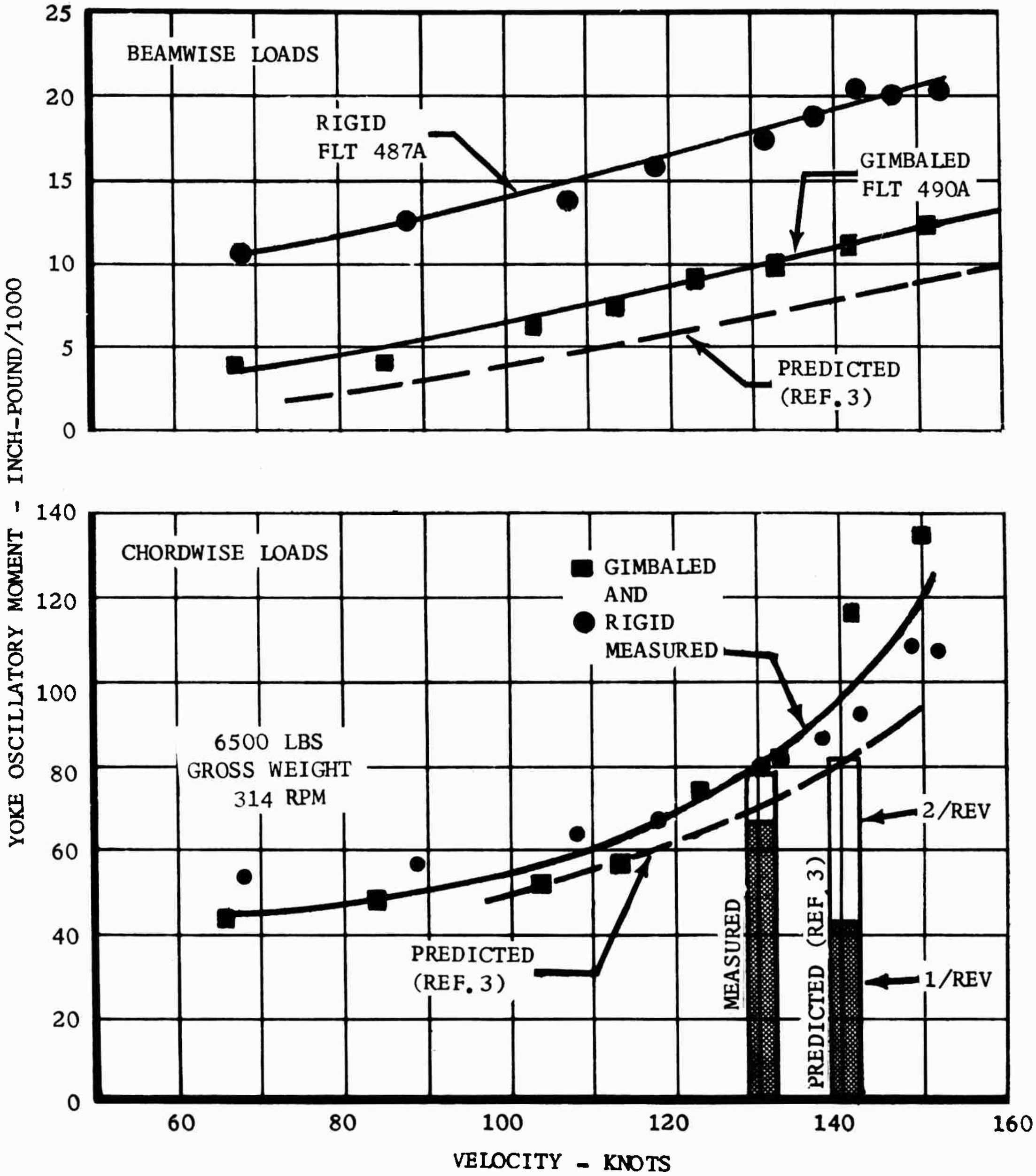


FIGURE 32 - MAIN-ROTOR LOADS, THREE-BLADED ROTOR.

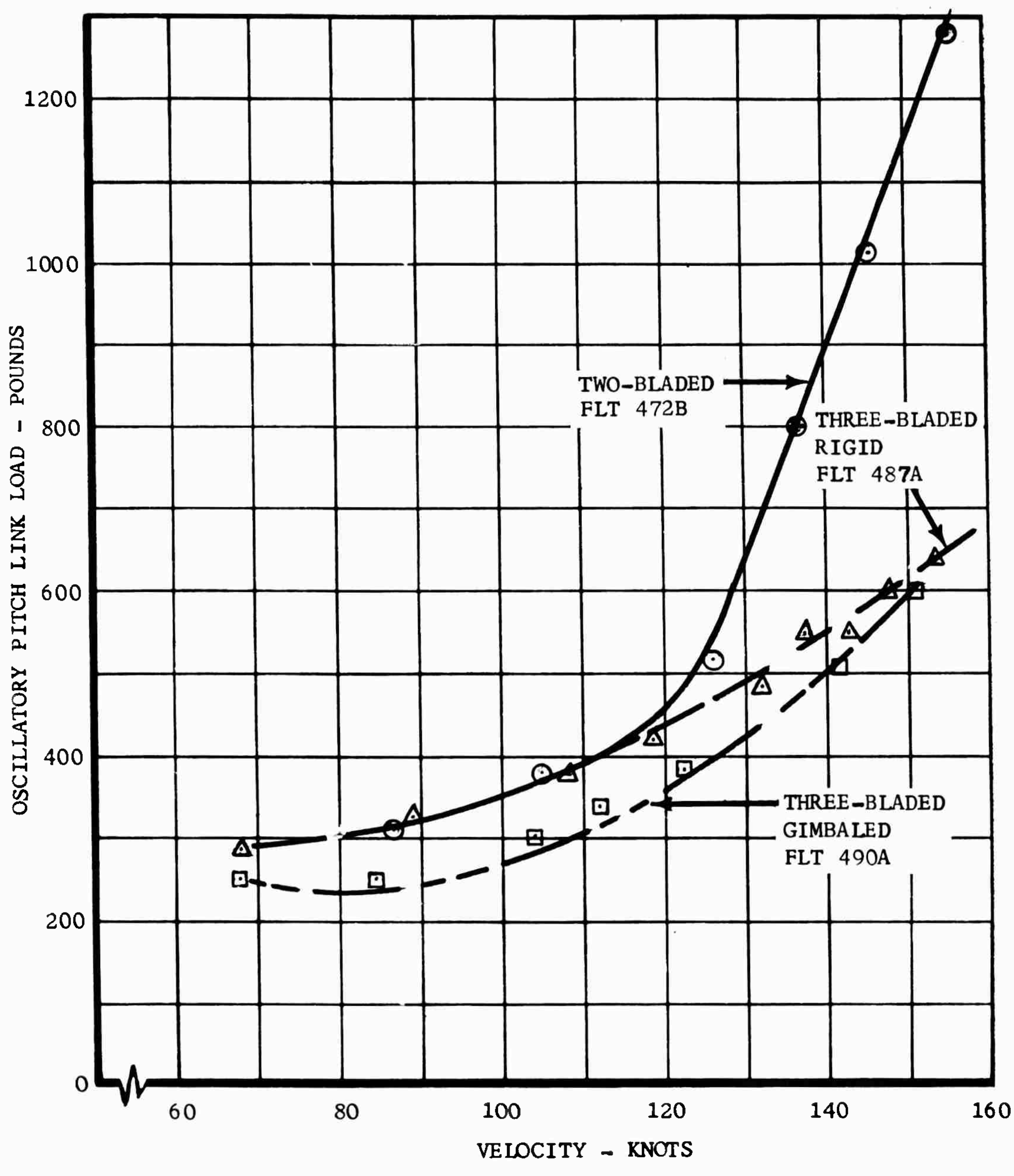


FIGURE 33- MAIN-ROTOR CONTROL LOADS.

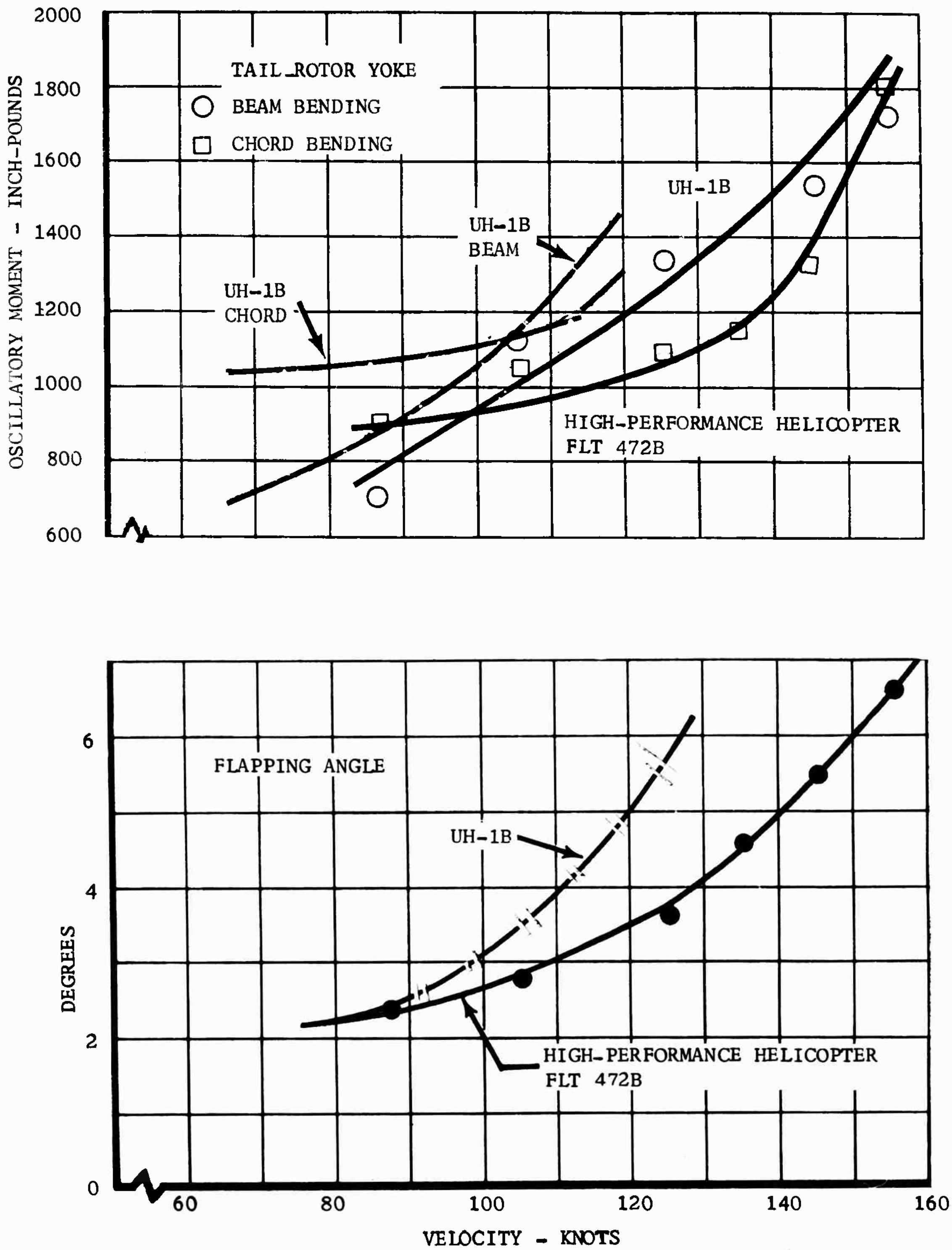


FIGURE 34 - TAIL-ROTOR LOADS AND FLAPPING ANGLES.

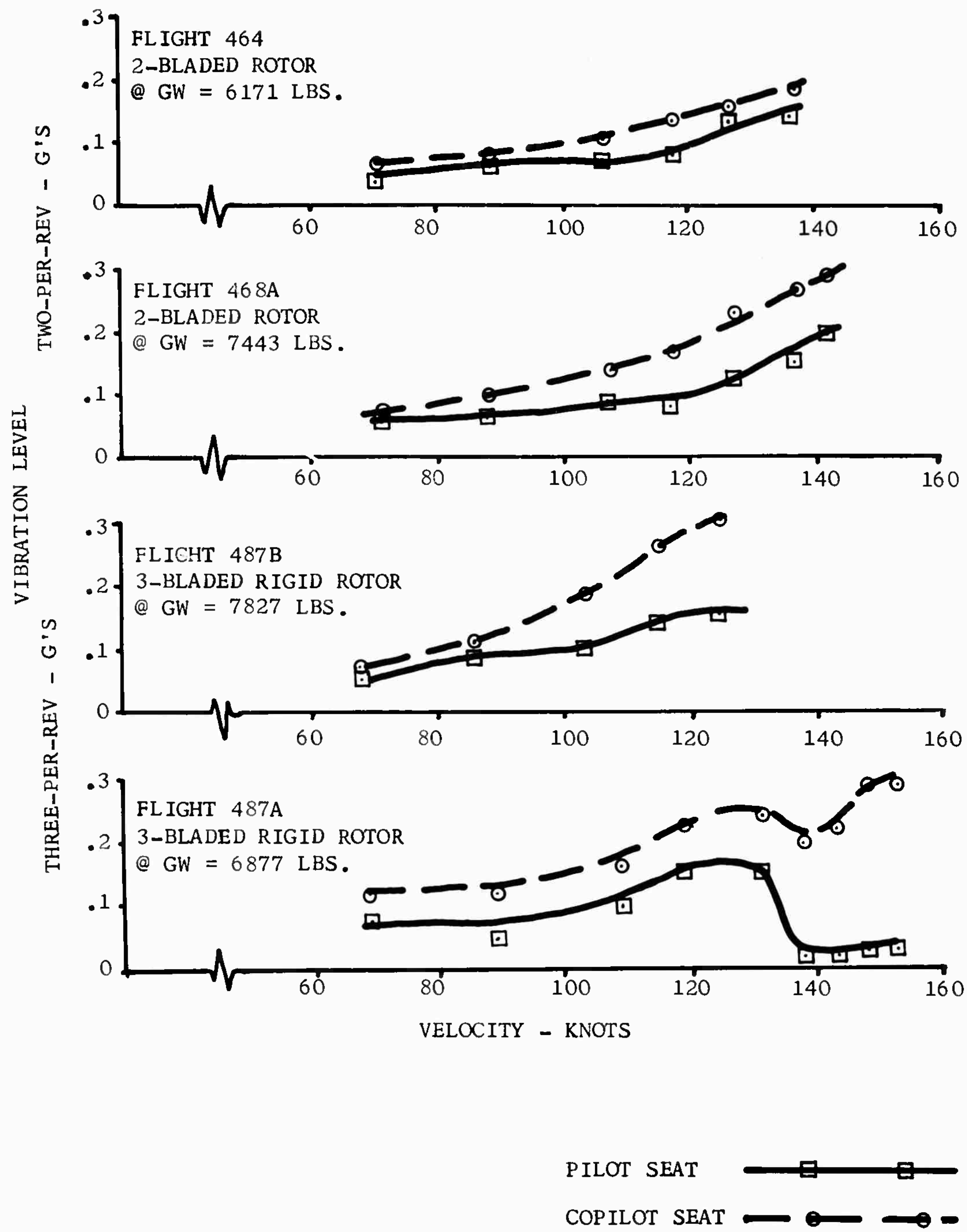


FIGURE 35 - MEASURED VIBRATION LEVELS

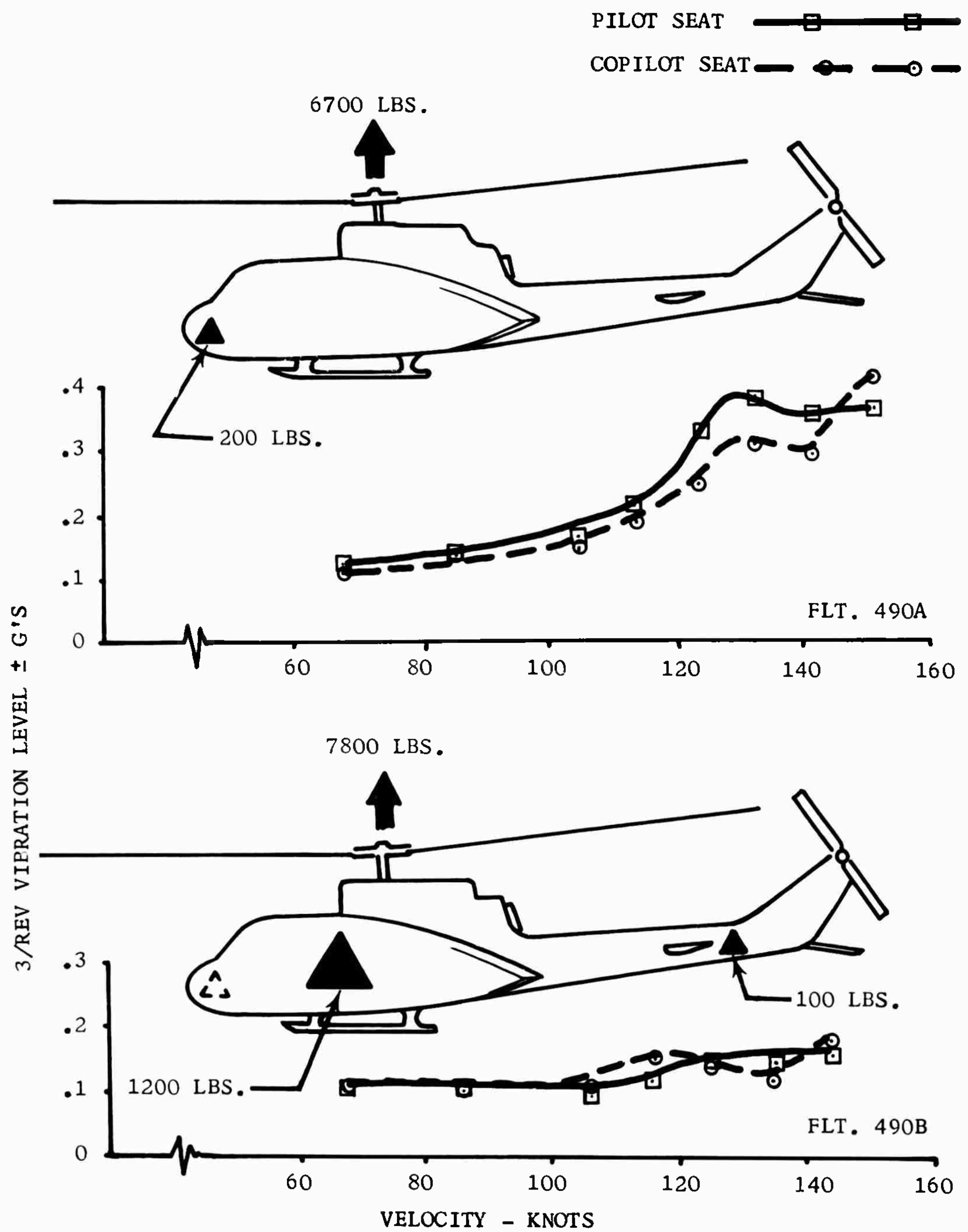


FIGURE 36 - FUSELAGE LOADING INFLUENCE ON VIBRATION LEVELS, THREE-BLADED GIMBALED ROTOR.

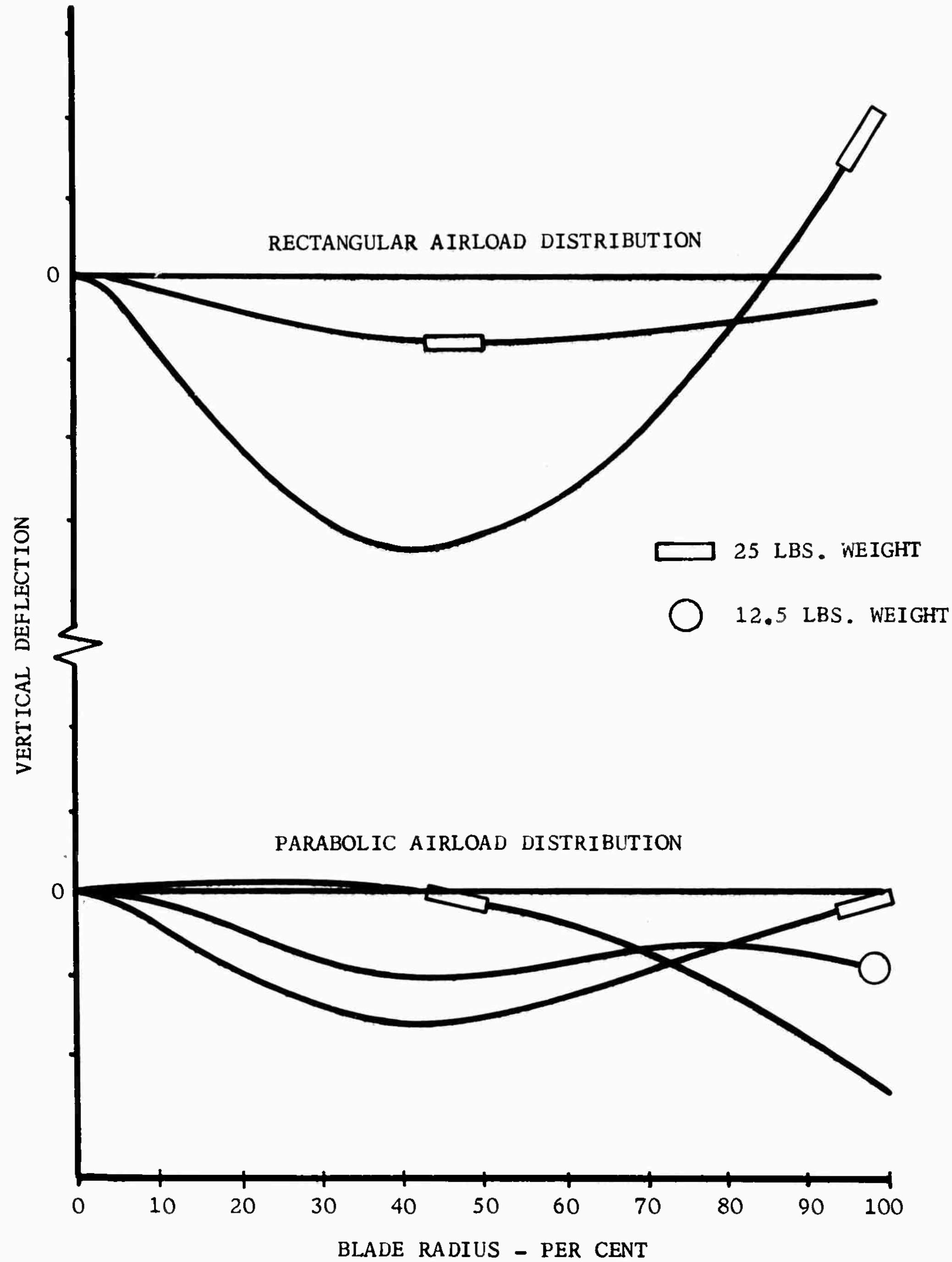


FIGURE 37 - BLADE RESPONSE TO 3/REV AIRLOADS AND INFLUENCE OF MASS DISTRIBUTION.

DISTRIBUTION

U. S. Army Infantry Center	1
U. S. Army Command & General Staff College	1
Army War College	1
U. S. Army Arctic Test Board	1
U. S. Army Aviation Test Board	1
Aviation Test Office, Edwards AFB	1
The Research Analysis Corporation	1
Army Research Office, Durham	2
Office of Chief of R&D	1
Naval Air Test Center	2
Army Research Office, OCRD	1
U. S. Army Aviation School	1
U. S. Army Combat Developments Command Transportation Agency	1
U. S. Army Transportation Board	1
U. S. Army Aviation and Surface Materiel Command	4
U. S. Army Transportation Center and Fort Eustis	1
U. S. Army Transportation Research Command	66
U. S. Army Transportation School	4
U. S. Army Research & Development Group (Europe)	1
TC Liaison Officer, U. S. Army Aviation School	1
Air Proving Ground Center, Eglin AFB	1
Air University Library, Maxwell AFB	1
Air Force Systems Command, Wright-Patterson AFB	1
Bureau of Naval Weapons	1
U. S. Naval Postgraduate School	1
U. S. Army Standardization Group, Canada	1
Canadian Army Liaison Officer, U. S. Army Transportation School	3
British Army Staff, British Embassy	4
U. S. Army Standardization Group, U. K.	1
National Aviation Facilities Experimental Center	1
NASA-LRC, Langley Station	2
Ames Research Center, NASA	2
NASA Representative, Scientific and Technical Information Facility	1
U. S. Government Printing Office	1
Defense Documentation Center	10
U. S. Army Mobility Command	4
U. S. Army Materiel Command	8
U. S. Army Limited War Laboratory	1

Bell Helicopter Company, Fort Worth, Texas, SUMMARY REPORT, HIGH-PERFORMANCE HELICOPTER PROGRAM, PHASE I - J. F. Van Wyckhouse and W. L. Cresap, TRECOM Tech Rept 63-42, Sep 1963, 78p. incl.illus. tables (Contract DA 44-177-TC-711) USATRECOM Proj 9R38-13-014-01

Unclassified report

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