USARTL-TR-79-9

LEVEL II

FLIGHT TEST EVALUATION OF THE HIGH INERTIA ROTOR SYSTEM

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AD A071648

June 1979

Final Report for Period 21 September 1976 - February 1979

Approved for public release;
distribution unlimited.

Prepared for

APPLIED TECHNOLOGY LABORATORY
U. S. ARMY RESEARCH AND TECHNOLOGY LABORATORIES (AVRADCOM)
Fort Eustis, Va. 23604

APPLIED TECHNOLOGY LABORATORY POSITION STATEMENT

In 1977 the High Energy Rotor System (HERS) program investigated (through flight tests) the effect of rotor blade inertia on flight safety, performance, and control. These tests provided quantitative and qualitative data to properly evaluate the operational benefits and limitations of very high rotor inertia. Data were recorded substantiating the reduction and elimination of the low-speed "avoid" region of the height-velocity diagram (deadman's curve). Aircraft maneuverability was evaluated, and the effect of the High Energy Rotor on helicopter handling qualities was determined.

The results of this program indicated that reserved (stored) energy can be utilized in a number of ways to provide operational benefits in the nap-of-the-earth (NOE) environment: The HERS eliminates the height-velocity diagram; the stored energy can be used to execute a safe landing following an engine failure from any airspeed or altitude; the stored energy provides additional power to augment the helicopter's performance; and the additional power can be used for acceleration, pop-up, and climb-out maneuvers. It was determined that the effects of the High Energy Rotor improved the handling qualities of the basic helicopter.

Messrs. William A. Decker and Robert P. Smith of the Aeronautical Technology Division, Aeromechanics Technical Area, served as project engineer and assistant project engineer, respectively, for this effort.

Unclassified

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

REPORT DOCUMENTATION PAGE

READ INSTRUCTIONS BEFORE COMPLETING FORM

1. REPORT NUMBER: USARTL-TR-79-9	2. GOVT ACCESSION NO.
3. RECIPIENT'S CATALOG NUMBER	
4. TITLE (and Subtitle): FLIGHT TEST EVALUATION OF THE HIGH INERTIA ROTOR SYSTEM	5. TYPE OF REPORT & PERIOD COVERED: Final Report, 21 Sep[redacted]976-1 Feb[redacted]79
	6. PERFORMING ORG. REPORT NUMBER
7. AUTHOR(s): L. W. Dooley, R. D. Yeary	8. CONTRACT OR GRANT NUMBER(s): DAAJ02-76-C-0064
9. PERFORMING ORGANIZATION NAME AND ADDRESS: Bell Helicopter Textron, P. O. Box 482, Fort Worth, Texas 76101	10. PROGRAM ELEMENT, PROJECT, TASK AREA & WORK UNIT NUMBERS: 63211A 1F2632 11D157 11 003 EK
11. CONTROLLING OFFICE NAME AND ADDRESS: Applied Technology Laboratory, U.S. Army Research and Technology Lab. (AVRADCOM), Fort Eustis, Virginia 23604	12. REPORT DATE: June 1979
	13. NUMBER OF PAGES: 119
14. MONITORING AGENCY NAME & ADDRESS (if different from Controlling Office)	15. SECURITY CLASS. (of this report): Unclassified
	15a. DECLASSIFICATION/DOWNGRADING SCHEDULE

16. DISTRIBUTION STATEMENT (of this Report)

Approved for public release; distribution unlimited.

17. DISTRIBUTION STATEMENT (of the abstract entered in Block 20, if different from Report)

18. SUPPLEMENTARY NOTES

19. KEY WORDS (Continue on reverse side if necessary and identify by block number)

Helicopters
High Inertia Rotor
Autorotational Landing
Height-Velocity Restrictions
Transient Rotor Power
Flight Test
Lock Number

20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The objectives of this program were to document the reduction in height-velocity restrictions for autorotational landings, the increased transient performance and maneuver capability using modified flight techniques, and the handling qualities of a modified OH-58A helicopter with a High Energy Rotor System (HERS). The rotor incorporated increased chord blades and provision for three different tip weights resulting in Lock number variations of 2.61, 3.14, and 5.43.

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Unclassified

SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered)

20. ABSTRACT (Continued)

The HERS has demonstrated the ability to reduce or eliminate the height-velocity restrictions on autorotational landings. Through the use of a simplified pilot technique, normal autorotational landing procedures were much easier and safer to employ. Low-speed maneuverability was greatly enhanced through significant transient power extraction from stored rotor energy. Longitudinal acceleration from hover was quadrupled and lateral acceleration was doubled with the HERS technique. Vertical climb rate was doubled with the HERS. This transient power is available to offset the weight of the HERS. The effect of the HERS on helicopter handling qualities was evaluated and determined to be improved over the basic helicopter.

Additional tests indicated that the HERS in combination with hub restraint will provide a good combination for the NOE environment.

PREFACE

The work reported herein was performed by Bell Helicopter Textron under Contract DAAJ02-76-C-0064, "Increased Aircraft Agility with High Energy Rotor System," awarded in September 1976 by the Eustis Directorate of the U.S. Army Air Mobility Research and Development Laboratory (USAAMRDL).*

Technical program direction was provided by Mr. W. A. Decker and the Army Flight Evaluation was flown by Major R. K. Merrill. The BHT project engineer was Mr. T. L. Wood. Principal BHT personnel associated with this program were B. M. Cassady, L. W. Dooley, L. W. Hartwig, and R. D. Yeary.

*Redesignated Applied Technology Laboratory, U. S. Army Research and Technology Laboratories (AVRADCOM), Fort Eustis, Virginia, 1 September 1977.

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1. INTRODUCTION

Nap-of-the-earth (NOE) tactics require a helicopter to operate almost continuously in a flight regime (low altitude and airspeed) where, in the event of an engine failure, a safe autorotational landing would be very hazardous for conventional helicopters. In order to provide an additional safety margin, Bell Helicopter Textron (BHT) has designed, by modifying existing hardware, a High Energy Rotor System (HERS) for an OH-58A helicopter which, using rotational energy stored in the rotor, eliminates the normal height-velocity restrictions of that helicopter.

Initial testing of this system was performed under a BHT independent research and development program to prove the concept. This report presents the results of a contracted flight test program with the Applied Technology Laboratory, Fort Eustis, to document the reduction or elimination of height-velocity restrictions, the increased transient performance and maneuver capability using modified flight techniques, and the handling qualities of a modified OH-58A helicopter with the HERS.

The rotor blades were designed such that this rotational inertia could be varied from approximately the standard OH-58A inertia up to over twice this value. This design allowed the effect of a wide variation in Lock number to be investigated with all other parameters being held constant. Lock numbers of 5.43, 3.19, and 2.61 were tested at a disk loading of 3.16 lb/ft^2 for effect on height-velocity restrictions and handling qualities. These Lock numbers correspond to blade inertias of 323, 550, and 672 slug-ft^2, respectively.

1.1 DESCRIPTION OF TEST VEHICLE

The helicopter used for this program is a modified OH-58A as shown in Figure 1. This photograph shows the highest inertia configuration (blade inertia - 672 slug ft^2) and the static droop of the blades due to the additional tip weights is apparent. This helicopter is the prototype OH-58A and because it has been used for several development programs since then, it has several differences from the standard OH-58A. The differences that affect the performance of the contracted tasks are outlined in subsequent paragraphs. A detailed description of the helicopter modifications is also reported in Reference 1.

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[1] T. L. Wood, and H. Bull, SAFETY OF FLIGHT DATA FOR THE HIGH ENERGY ROTOR SAFETY OF FLIGHT VERIFICATION REVIEW (SOFVR) EVALUATION, Bell Helicopter Textron, Report Number 699-099-051, Fort Worth, Texas, March 1977.

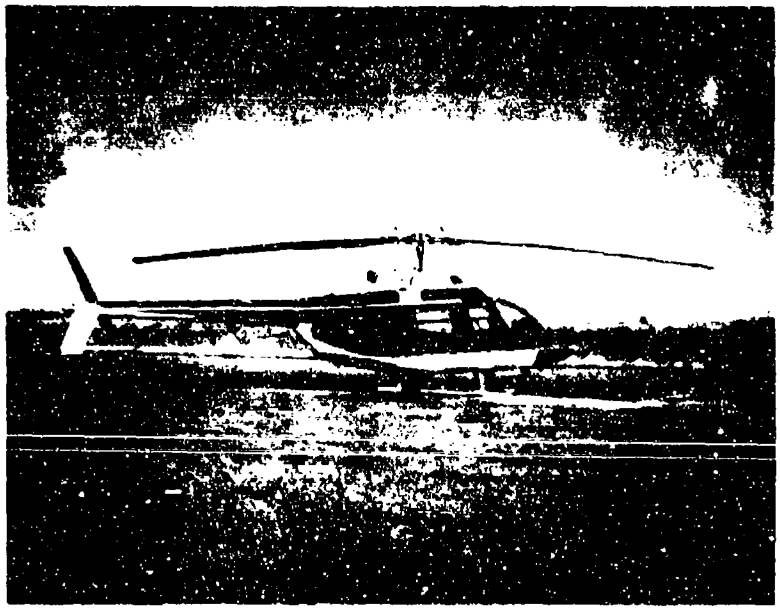


Figure 1. Experimental high energy rotor system(HERS).

1.1.1 Rotor Blades

The High Energy Rotor System was constructed by modifying existing rotor components. The desired blade inertia, chord, and airfoil characteristics were defined through analytical studies but were constrained by the available rotor hardware compatible with an OH-58A. Thus the final blade configuration is a compromise between the modifications possible to existing hardware and the results of the analytical definition.

The rotor blades are modified BHT Model 206 (JetRanger) blades. Four external steel doublers were added to carry the increased centrifugal force. Two of the doublers run full span while two run half-span. The rotor chord was increased to 16 inches by adding a 3-inch full span trailing edge tab to provide chord stiffness for blade tuning. The blade inertia was increased by the addition of four, or three tungsten tip weights per blade in the spar cavity to give 672 and 550 slug-ft^2 rotor inertia as shown in Figure 2. The standard rotor inertia can be obtained when only one weight per blade is used.

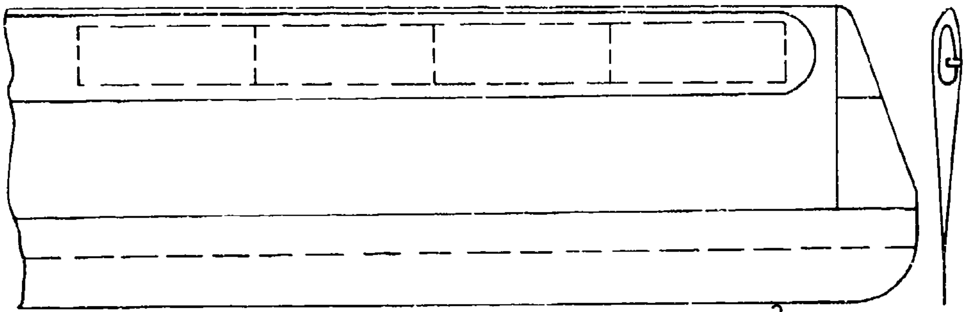
The addition of the external doublers on the upper surface close to the leading edge reduces the maximum lift coefficient capability of the airfoil, which results in some decrease in autorotational performance during the touchdown phase of landing where high lift coefficient is required.

Although the blade inertia is over twice that of the standard OH-58A blade, the lowest inplane frequency is above 1 per revolution, precluding the possibility of ground resonance. The cantilever blade frequency is 0.85 per rev, but coupling with the pylon raises the frequency to 1.3 per rev.

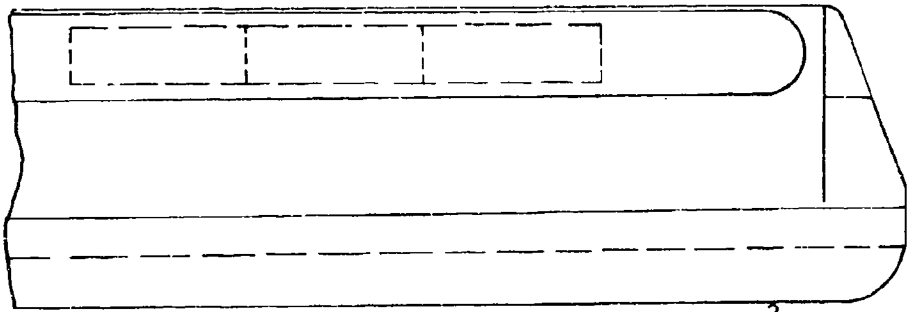
1.1.2 Main Rotor Hub

A BHT Model 640 experimental hub was used for this program. This is a flex-beam type hub similar to the hub used on the AH-1G Cobra helicopter. The test hub employs elastomeric flapping and feathering bearings and is shown in Figure 3. This hub was used primarily because no modifications were required to carry the increased centrifugal loads of the HERS.

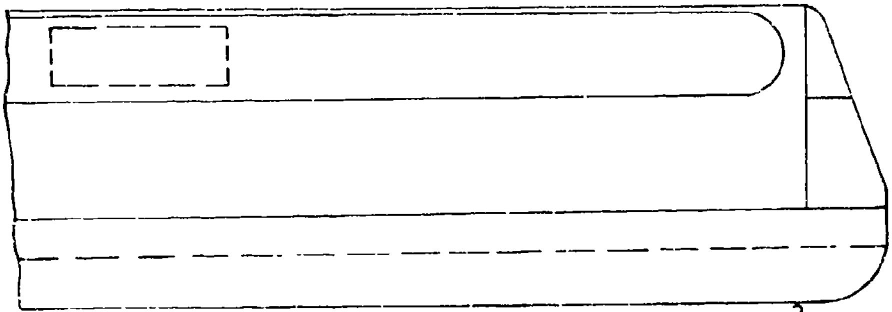
Chinese weights were attached to the hub to help balance the tennis racket moment (inertial moment tending to lower the blade pitch) which was caused by the modified mass distribution of the test blades. These weights, along with adjustments of the blade trailing edge tab, were used to reduce control tube loads, permitting boost-off flight.



High Inertia Configuration - I_b = 672 slug-ft^2/blade
55 lb tip weight



Mid Inertia Configuration - I_b = 550 slug-ft^2/blade
40.5 lb tip weight



Standard Inertia Configuration - I_b = 323 slug-ft^2/blade
11.5 lb tip weight

Figure 2. High energy rotor tip weight configurations.

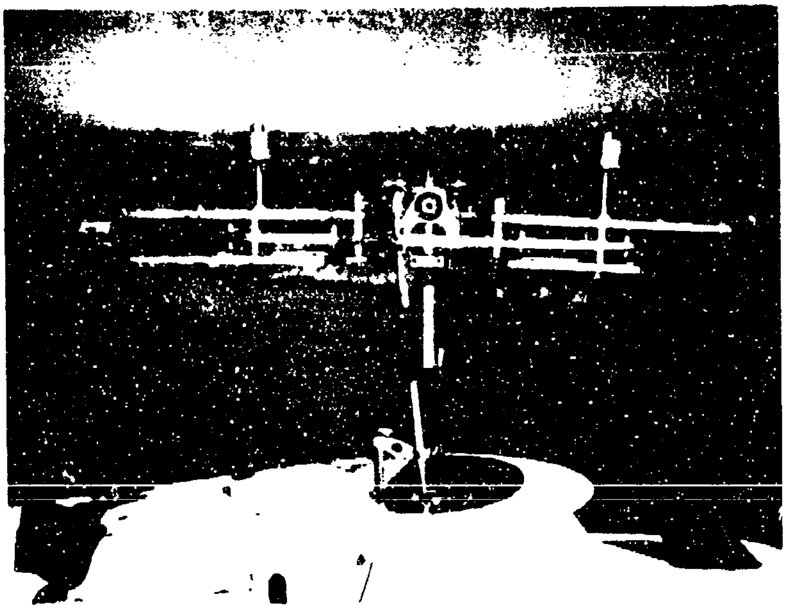


Figure 3. Model 640 rotor hub.

1.1.3 Flight Control System

The control system of the basic helicopter was modified to improve the handling qualities in the pitch and roll axes. No changes were made in the directional control system and the collective system was adjusted only for setting the low blade angle in autorotation to account for performance differences of the main rotor blades.

A two-axis stability and control augmentation system (SCAS) was installed and used to provide control quickening in the pitch and roll axes. All SCAS functions were disconnected except for a feedforward loop which supplied an initial overshoot and washout of cyclic control inputs. This is described in more detail in Section 4.2 of this report.

The cyclic control system gearing was also modified by decreasing the cyclic stick travel required to command full swashplate travel. This increased the control sensitivity and response characteristics for the HERS configurations as described in Section 4.

1.1.4 Engine Controls

To fully explore the potential of the HERS, the ability to bleed main rotor RPM to extract rotational energy from the rotor during power-on maneuvers was required. Since a torque limiter was not available on the test aircraft, the engine power was reduced by limiting the maximum N_I throttle setting. By reducing the maximum power available, the transmission would not be overtorqued at low rotor RPM during power-on operation. A cockpit adjustment was provided to vary the maximum N_I throttle setting as explained in Reference 1. The N_I adjustment could be made only on the ground and a N_I topping check had to be performed to assure limiting did not occur prior to achieving 91 percent N_I.

1.1.5 Hydraulic System

The hydraulic system was modified to increase system pressure from 600 psi to 1000 psi. This modification was incorporated to account for the increase in control load from the spring rate of the elastomeric bearings in the 640 rotor hub. Flight evaluation of the 640 rotor hub with standard blades had identified the required change to the hydraulic system prior to the HERS test program.

1.1.6 Flight Envelope

The experimental high energy rotor was constructed and evaluated as a BHT Independent Research and Development program to establish the flightworthiness of the hardware and determine feasibility of the concept. The flight envelope has been increased to include level flight from hover to 110 knots, low-rotor-rpm level flight from hover to 60 knots, left and right turns to 1.5g at 80 knots, symmetrical pullups and pushovers from 1.5g to 0.5g entered at 80 knots, in-ground-effect acceleration and decelerations, roll reversals, hover throttle chops, and full autorotational landings up to a maximum gross weight of 3062 pounds. The blade, hub, and control system loads were acceptable for all conditions tested, and no adverse dynamic loads were encountered. The above flight envelope was adequate for this evaluation.

1.1.7 Weight of Rotor System

The addition of tip weights to the rotor requires an increase in centrifugal force retention capability in the rotor system. However, no weight increases are required in the transmission drive system or tail rotor as a result of using the transient power that is inherent in the HERS. For the highest inertia rotor tested, the increase in helicopter empty weight was 134 pounds, of which 46 pounds was the increase in tip weight. The remaining 88 pounds included the weight increase of the experimental rotor hardware due to increased centrifugal loads, additional rotor chord, and all other changes required to attain this experimental configuration.

If the HERS is designed into the helicopter initially, much of the additional weight could be eliminated. No estimate of the weight required for a production HERS is made; however, designs with more efficient airfoils and with composite material will result in a point design rotor system with a lower weight penalty.

1.2 INSTRUMENTATION

Data were recorded during this test program with an onboard oscillograph for most parameters. Autorotational landing and NOE performance and maneuverability tests were also filmed with a ground-based grid camera to record flight path time histories. Correlation between the onboard recorder and the grid camera was maintained with a light mounted on the side of the helicopter.

For autorotational landing tests, the external light was switched on automatically by a contact switch when the throttle reached the flight idle position. An oscillograph trace also recorded the condition of the light. The time of the throttle chop was thus recorded on both airborne and ground-based equipment. To indicate the touchdown time, a switch was mounted on the aft end of the skid tube which switched off the light automatically at touchdown.

A listing of installed instrumentation during this test program is provided in Appendix A.

1.3 TEST PROGRAM

The contracted flight test program consisted of evaluating three rotor blade inertias of 672 (γ=2.61), 550 (γ=3.19), and 323 (γ=5.43) slug-ft^2. All the contracted test flights were conducted by Bell Helicopter Textron Chief Pilot, Mr. L. W. Hartwig, at the BHT flight Research Center in Arlington, Texas. The Army Pilot evaluation was conducted by Major Robert K. Merrill, USAAMRDL, Langley Directorate. Both the contractor and Army evaluation of the highest rotor blade inertia were witnessed by Mr. William Decker, USAAMRDL, Eustis Directorate. The testing is summarized in Table 1. The height-velocity restrictions and handling qualities of each configuration were evaluated. Transient maneuvers typical of NOE operations were evaluated at the highest inertia. The total contracted test program consisted of 21.9 flight hours and 0.6 hour ground run.

A log of flights is presented in Appendix B.

TABLE 1. SUMMARY OF TEST PROGRAM FLIGHTS

Flight Number	Period	Flown By	Blade Configuration Inertia		Flight Time Hrs
			I_b Slug-ft^2	γ	
166 177	14 Feb 77 6 Apr 77	BHT	672	2.61	9.3
178 180	20 Apr 77 26 Apr 77	Army	672	2.61	5.8
181 184	26 May 77 2 Jun 77	BHT	550	3.19	3.3
185 193	20 Jun 77 3 Aug 77	BHT	323	5.43	3.5

2. AUTOROTATIONAL LANDINGS

One of the objectives of the High Energy Rotor System was the elimination of the height-velocity restrictions shared by all conventional helicopters. This section of the report describes the techniques needed to perform safe autorotation landings in the low altitude and airspeed flight regions and the basic theory behind increasing the rotor inertia to eliminate these restrictions. Test data are presented for autorotational landings performed at three levels of main rotor Lock number and a summary of results in terms of the effect of Lock number on pilot technique.

2.1 TYPICAL H-V RESTRICTIONS

Most helicopters have a region of operation in which, if an engine failure occurs, a safe autorotational landing cannot be executed. This area of limited autorotational landing capability exists for both single and multiple engine helicopters and is described by the altitude above the ground and airspeed. It is commonly illustrated with a height-velocity diagram (or deadman's curve). As seen in Figure 4, the diagram for a standard OH-58A, for maximum performance as recorded in Reference 2, two restricted regions are indicated and are typical for conventional helicopters.

The restricted areas of the height-velocity diagram are normally determined by flight test. Using a build-up technique, the pilot performs a series of simulated engine failures starting at entry conditions expected to be well outside the restricted area. Subsequent entries are made at more critical conditions until the pilot feels a safe landing could not be performed from conditions more critical than the last. These tests are performed to establish limits at enough altitude and airspeed combinations to allow construction of the height-velocity diagram.

The establishment of the restricted area limits is a strong function of pilot opinion. The test pilot must judge what average pilot reaction time and skill level are required to perform a successful landing and adjust his established restriction limits accordingly. Since the judgement is subject to variation with individual test pilots, the data presented here are based on the maximum performance of the helicopter, allowing comparison with the data in Reference 2.

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[2] J. C. Watts, et al., HEIGHT-VELOCITY TEST OF THE OH-58A HELICOPTER, USAASTA Technical Report 69-16, U.S. Army Aviation Systems Test Activity, Edwards Air Force Base, California, June 1971, AD884973.

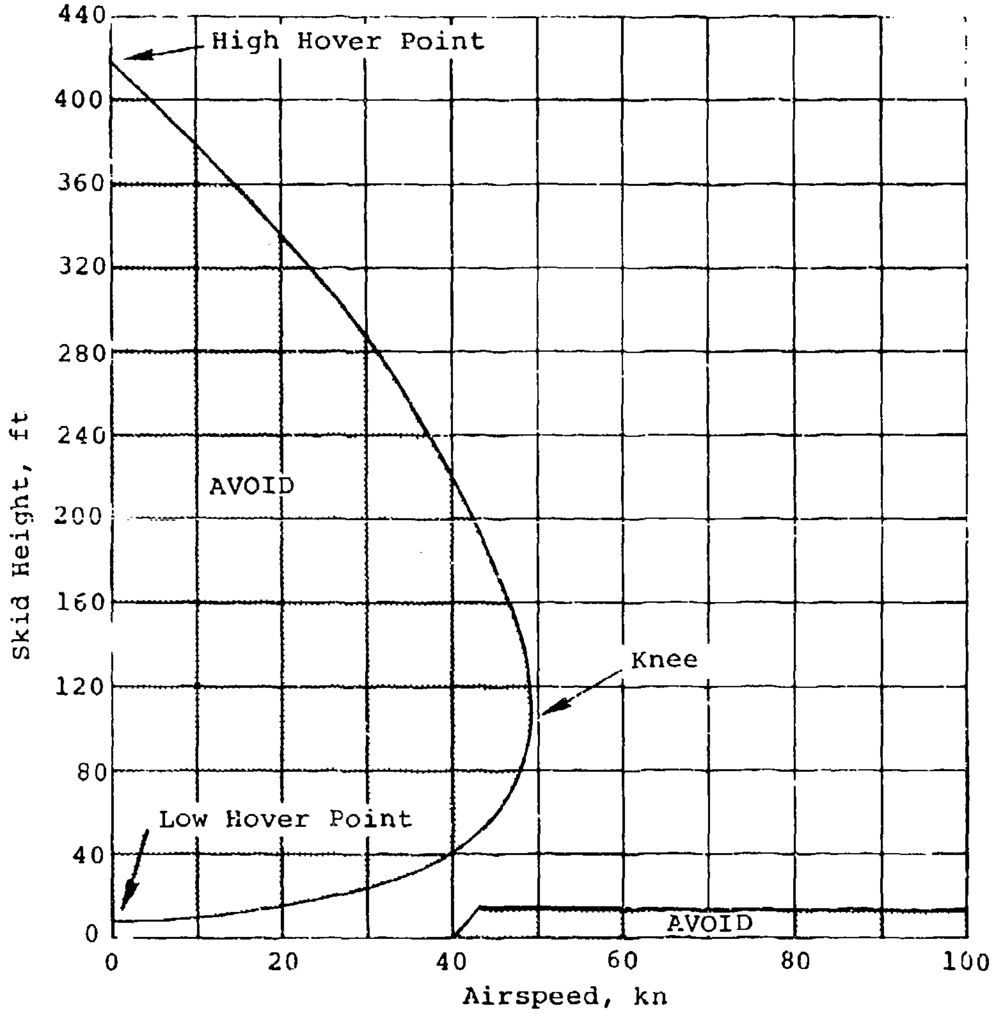

Figure 4. OH-58A maximum performance height-velocity diagram.

Another important factor affecting the height-velocity restriction is the effect of wind. The published restricted areas are to be shown for zero wind conditions. The presence of a headwind when performing an autorotational landing will significantly reduce the difficulty of the task. Thus, the tests performed in this program were done in winds of less than 3 knots.

One more factor that can have an effect on the restricted area is the technique used to perform the landings. For the tests reported here, the pilot's primary objective was to attain a zero rate-of-sink at touchdown and to accept a safe horizontal velocity. The technique was to level the aircraft 1 or 2 feet off the ground with no rate-of-descent and then gradually sink to the ground as rpm continued to decrease. During this phase of the maneuver, small amounts of aft cyclic were applied to reduce the horizontal velocity. This technique yielded consistent zero rate-of-descent touchdowns while the horizontal velocity varied somewhat.

Since height-velocity restrictions are established by a test pilot opinion and adjusted for factors reflecting the abilit of typical pilots, some variation in the restricted area may be expected if different pilots test the same helicopter. However, these restrictions generally are good indicators of operating conditions in which a safe autorotational landing would be difficult or impossible to achieve.

Referring to Figure 4, the low-speed area is usually described by 3 points: the high hover point, the low hover point, and the knee or highest speed point. The high-speed area indicates an area of operation in which there is insufficient clearance between the tail of the helicopter and the ground to allow sufficient flare attitude to decelerate and control rotor RPM prior to ground contact. The low-speed area was of primary interest in this work.

For the standard OH-58 helicopter, the required autorotational landing technique depends on the flight condition before engine failure. At low altitude and airspeed, below the knee of the curve, the pilot technique required for a safe landing is to use increased collective to reduce the sink rate as the helicopter approaches 10 to 15 feet above the ground. At about 10 feet above the ground, the ship is leveled and collective is increased as the helicopter settles. At higher airspeeds (still below the airspeed at the knee) the collective may be reduced somewhat to regain or maintain rotor RPM while the helicopter is decelerated using a cyclic flare.

In higher altitude entries (above the altitude at the knee of the curve), the collective is reduced and the cyclic moved forward to pitch the nose down in order to gain forward airspeed. The objective is to increase airspeed to that required to build rotor speed during a decelerating flare, usually close to the speed defined at the knee of the curve. As the speed increases, altitude rapidly decreases until the pilot initiates a flare. The purpose of this flare is to first reduce the rate-of-sink to a minimum at a low altitude; second, to reduce the forward airspeed before touchdown; and third, maintain or increase rotor RPM. As the helicopter slows down, forward cyclic is applied to lower the fuselage to the proper landing attitude and, simultaneously, the collective is raised to cushion the landing. The techniques described above are taught to Army aviators during training and are also described in the OH-58A Flight Manual (Reference 3).

2.2 ENERGY CONSIDERATIONS

In order to better understand the pilot techniques required during an autorotational landing, consideration of the helicopter energy states is necessary. After an engine failure in flight, the helicopter has three basic sources of energy. These are (1) the potential energy of altitude (E_h), (2) the kinetic energy of flight path velocity (E_V), and (3) the rotational energy of the main rotor (E_R). Power for flight may be extracted from these sources by decreasing the energy level over some time period, thus converting this energy into a useful form. Listed below in Table 2 are the equations describing these energy sources and potential levels of power extraction.

TABLE 2. ENERGY SOURCES

Source	Energy	Power
Altitude	$E_h = Wh$	$P_h = W\frac{dh}{dt}$
Velocity	$E_V = 1/2\frac{W}{g}V^2$	$P_V = \frac{W}{g}V\frac{dV}{dt}$
Rotor	$E_R = 1/2 I_R\Omega^2$	$P_R = I_R\Omega\frac{d\Omega}{dt}$

[3] OPERATOR'S MANUAL, ARMY MODEL OH-58A HELICOPTER, Technical Manual 55-1520-228-10, Headquarters, Department of the Army, Washington, D. C., October 1970.

After an engine failure, the energy state of the helicopter may be written as

$$E_{TOTAL} = Wh + 1/2\,\frac{W}{g}V^2 + 1/2\,I_R\Omega^2 \qquad (1)$$

The power that can be extracted from these energy levels is limited by the difference between the initial energy level and the minimum usable energy level. The minimum potential and kinetic energy levels are zero when the helicopter is on the ground and has no translational velocity. The minimum rotor energy is not as easily defined. As the rotor speed decreases, the rotor thrust coefficient $t_c = \dfrac{\text{Thrust}}{1/2\rho bcR(\Omega R)^2}$ increases if a constant thrust is required, as during the final stage of landing. At some rotor speed, the maximum usable value of t_c ($t_{c_{max}}$) will be reached and further decreases in rotor speed will result in reduced thrust. This minimum rotor speed (Ω_{min}) will define the minimum usable rotor energy. Thus, the maximum energy that can be extracted from the helicopter's initial energy state at engine failure is governed by

$$E_{max} = Wh_i + \frac{1}{2}\,\frac{W}{g}\,V_i^2 + \frac{1}{2}\,I_R\,(\Omega_i^2 - \Omega_{min}^2) \qquad (2)$$

As seen in this equation, the importance of the rotor energy component of the total helicopter energy level increases as the altitude and airspeed at the engine failure decrease. The pilot technique used during an autorotational landing from any entry condition governs the management of these energy resources during the time available between engine failure and ground contact. Of primary importance to the pilot is to arrive at a point just above the ground with a minimum rate of sink and horizontal speed with a maximum amount of rotor energy. The final descent will be controlled by extracting this rotor energy with a collective pull to cushion the touchdown. This final extraction of rotor energy just before touchdown is common to all autorotation landings.

Several indices have been developed from these energy expressions to evaluate the autorotational performance of a helicopter. One such index, as discussed in Reference 4, is based

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[4] T. L. Wood, HIGH ENERGY ROTOR SYSTEM, presented at the 32nd Annual Forum of the American Helicopter Society, Washington, D.C., May 1976.

on performance during the final landing phase of the autorotational maneuver and is expressed as the time the helicopter can remain airborne after a hovering throttle chop. This time becomes a convenient theoretical and practical measure by which the autorotational landing performance may be estimated.

The time delay used for the autorotational index can be approximated by considering the rotational energy available when the rotor speed drops from the hover rpm to the rotor stall rpm, and the energy spent by the rotor during the hold-off time. This index is written as

$$t/K = \frac{I_R \Omega^2 \left(1 - \frac{t_c}{.8\, t_{c_{max}}}\right)}{1100\ SHP_{hover_{OGE}}} \qquad (3)$$

The factor K is introduced since this equation does not take into account: (1) the potential energy that becomes available during the descent, (2) the fact that the maneuver takes place in ground effect, and (3) the residual rate of descent that may exist at touchdown.

This program provided a unique opportunity to evaluate the worth of such an index since the effect of rotor inertia could be evaluated independently of all other variables by testing at a constant entry rpm and gross weight/density ratio (thus holding the power term and t_c constant).

The theoretical values of the autorotational index for the rotor inertias tested are calculated as

I_b	t/K
323 slug-ft^2	1.37 sec
550 slug-ft^2	2.33 sec
670 slug ft^2	2.85 sec

where $SHP_{hover_{OGE}}$ = 283 HP, $.8\, t_{c_{max}}$ = .25, t_c = .13, and Ω = 37.07 rad/sec

Thus for the range of rotor inertias used, the time that a pilot can remain airborne after a hovering throttle chop may be expected to double for the highest inertia rotor. As will be presented in subsequent sections of this report, experimental results agree with this finding.

2.3 STANDARD INERTIA TEST RESULTS

In the following subsections, data gathered during the lowest rotor inertia (323 slug-ft^2) tests will be used to discuss pilot techniques for the standard OH-58A. While not precisely comparable, since the main rotor blades and hub are modified, the techniques used in these data are similar to those required for the Model OH-58A.

2.3.1 Autorotational Index Throttle Chop

As discussed previously, pilots can obtain a good indication of overall autorotational landing capabilities by performing a throttle chop from a low altitude (about 4 feet skid height) hover and by increasing collective to remain airborne for as long as possible. For this condition, the kinetic energy is zero and the potential energy is almost zero. Thus, all the power required to cushion the landing must come from a reduction in the rotor energy. The length of time the pilot can remain airborne after the throttle chop provides an indication of the rotor energy he may extract during the final phase of any autorotational landing. This low altitude (4 feet skid height) hovering throttle chop will be referred to as an autorotational index measurement. Figure 5 presents the time history of this maneuver for the low inertia (323 slug-ft^2) rotor.

Immediately after the throttle chop, the main rotor rpm began decreasing as the power supplied by the engine diminished. Right pedal was required to decrease tail rotor thrust in response to the decrease in yawing moment due to drive torque to the main rotor. As the rpm and altitude decreased, collective pitch was supplied to maintain a relatively constant rotor thrust. This was accomplished, as shown by the g-level trace in Figure 5, since the g-level remained at essentially 1.0 throughout the maneuver. The helicopter settled gently to the ground with a main rotor speed of 250 rpm, 4.4 seconds after initiation of the throttle chop. The power that was required to generate the rotor thrust was obtained by the extraction of rotor energy through the reduction of rotor speed.

2.3.2 Low Hover Point Throttle Chop

As altitude increases, eventually the power available from rotor energy will be insufficient to allow a gentle descent. This altitude defines the low hover point as indicated in Figure 4. For the low inertia (323 slug-ft^2) rotor, the low hover point was determined to be 27 feet. The data obtained during this limit-case hovering throttle chop are presented in Figure 6.

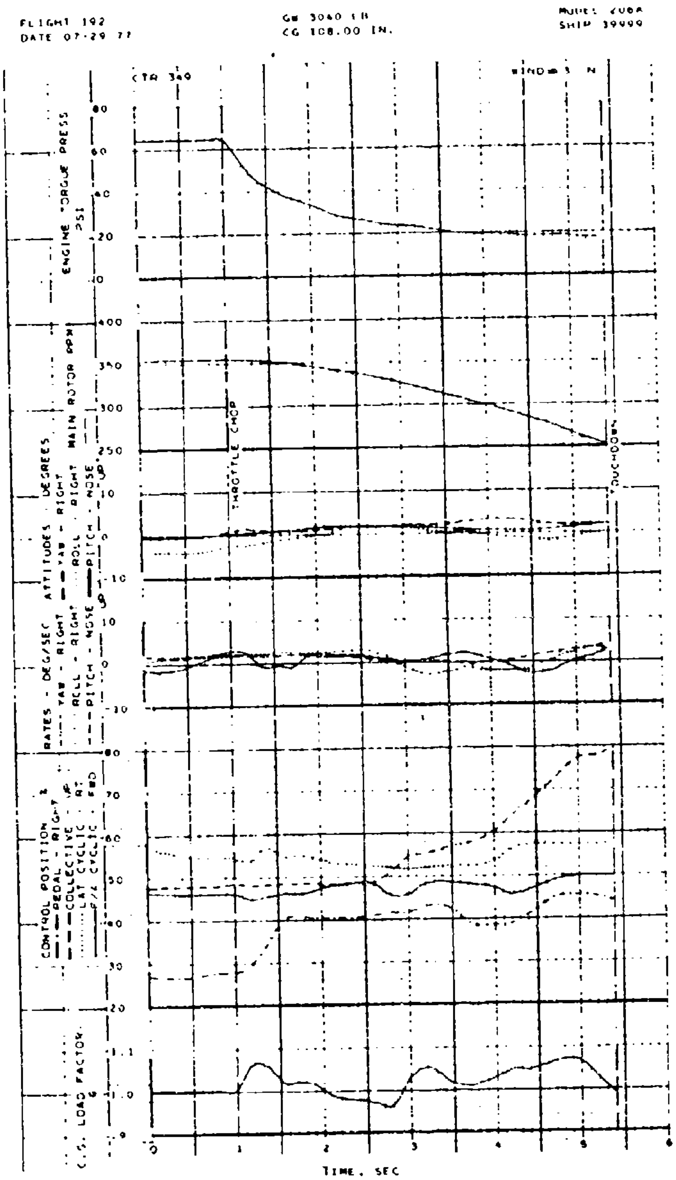
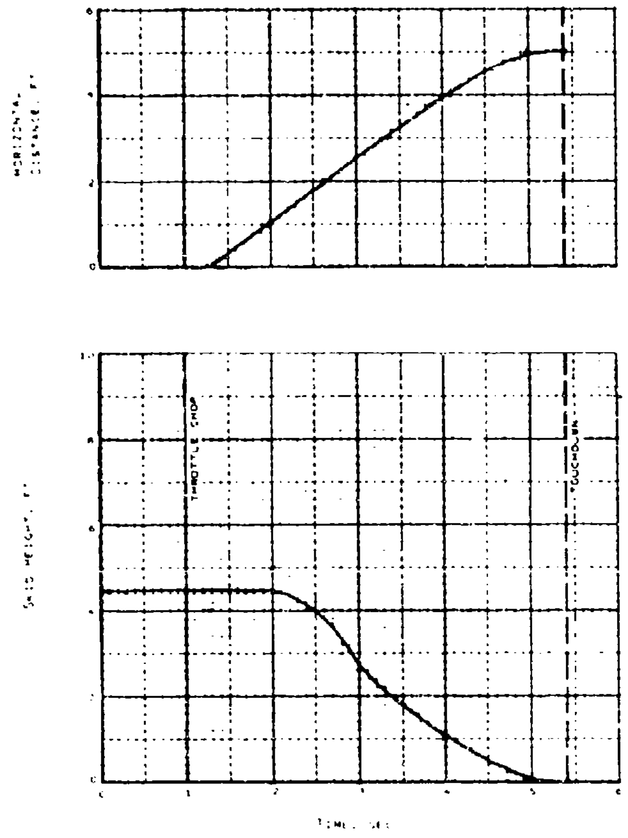


Figure 5. Autorotational index throttle chop - standard inertia rotor.



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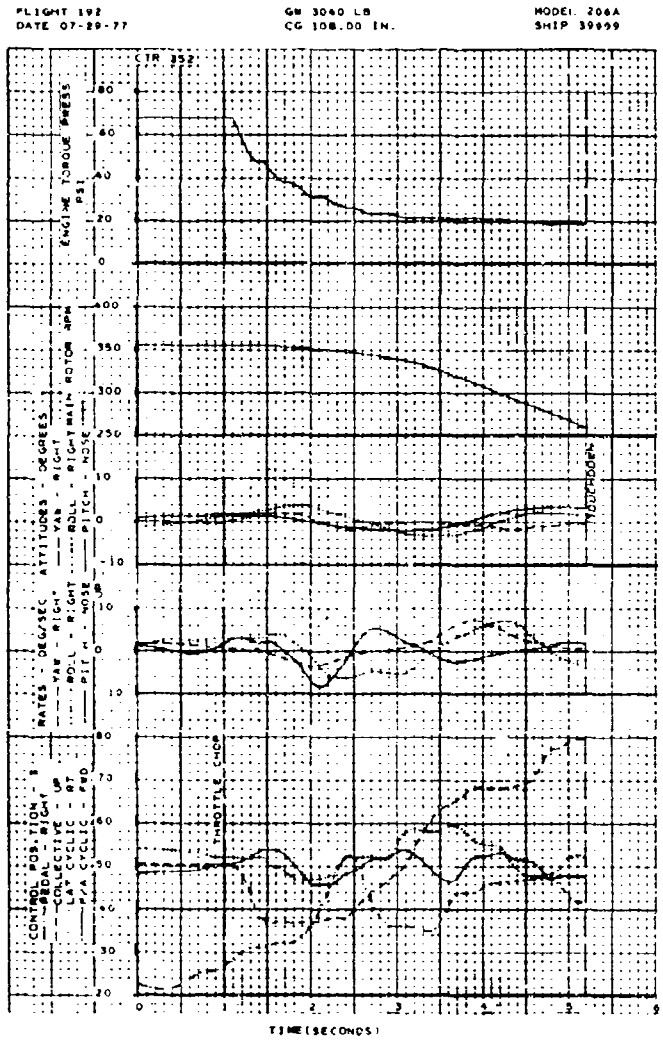


Figure 6. Low hover point throttle chop from 27 feet - standard inertia rotor.

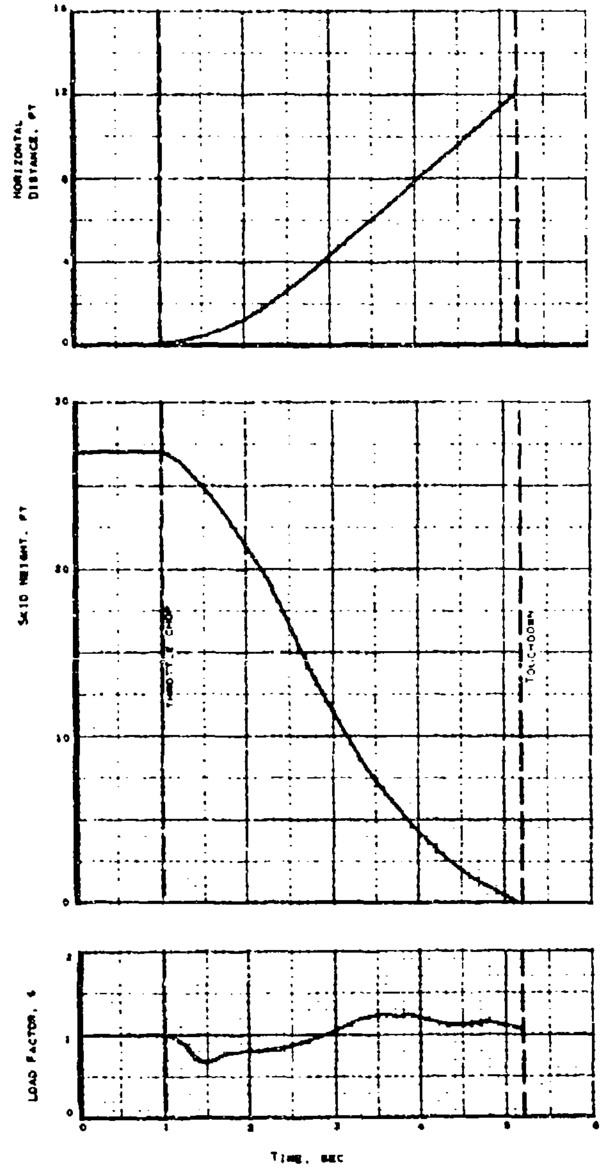


Figure 6. Concluded.

Immediately after the throttle chop, the pilot reduced collective by 13 percent to control the rate of main rotor rpm reduction. At about 15 feet and a rate of sink of 12 feet per second, a steady collective pull was started to cushion the landing. Touchdown occurred with a rate of sink of 2 feet per second and a main rotor speed of 260 rpm. In the test pilot's opinion, this was the highest altitude from which a successful landing could be performed and thus defined the maximum performance low hover point.

2.3.3 <u>High Hover Point Throttle Chop</u>

As hovering altitude increases above the low hover point, the potential energy contribution to total helicopter energy becomes more important. Eventually, an altitude will be reached where this potential energy may be managed to allow a safe autorotational landing. This altitude is defined as the high hover point as shown in Figure 4. For the standard inertia configuration, this altitude was determined to be 350 feet and a time history of a throttle chop at this condition is presented in Figure 7. In this case, since altitude was available, the collective was lowered and the nose pitched over immediately after the simulated engine failure. The normal acceleration reached a minimum of 0.6g during the collective drop and a nose-down pitch attitude of -25 degrees was attained. The initial portion of the maneuver was out of range of the grid camera which covered only up to about 100 feet above the landing site. However, when the helicopter reached grid camera range, its rate of descent was 46 feet per second, the horizontal velocity was 70 feet per second, and the rotor speed was back to 354 rpm. This is indicative of the trade of potential energy for both kinetic and rotor energy. A 12-degree nose-up flare was initiated at about 100 feet altitude where the peak g-level of 1.4g was reached. The flare was maintained to about 20 feet above the ground, by which time the rate of descent was reduced to 15 feet per second and the horizontal velocity to about 45 feet per second. As the nose of the ship was lowered toward a level attitude, a collective pull was used to bleed rotor rpm to provide power to maintain rotor thrust sufficient to reduce the rate of descent to zero by touchdown. However, since the nose was lowered, the horizontal deceleration was not maintained and the horizontal velocity at touchdown was 30 feet per second.

The rotor speed was reduced to 330 rpm at touchdown indicating that, based on a rpm of 250 for the hovering throttle chop of Figure 5, not all the available rotor energy was used in this landing. This is primarily because of the high horizontal speed at touchdown, which reduced the power required to remain

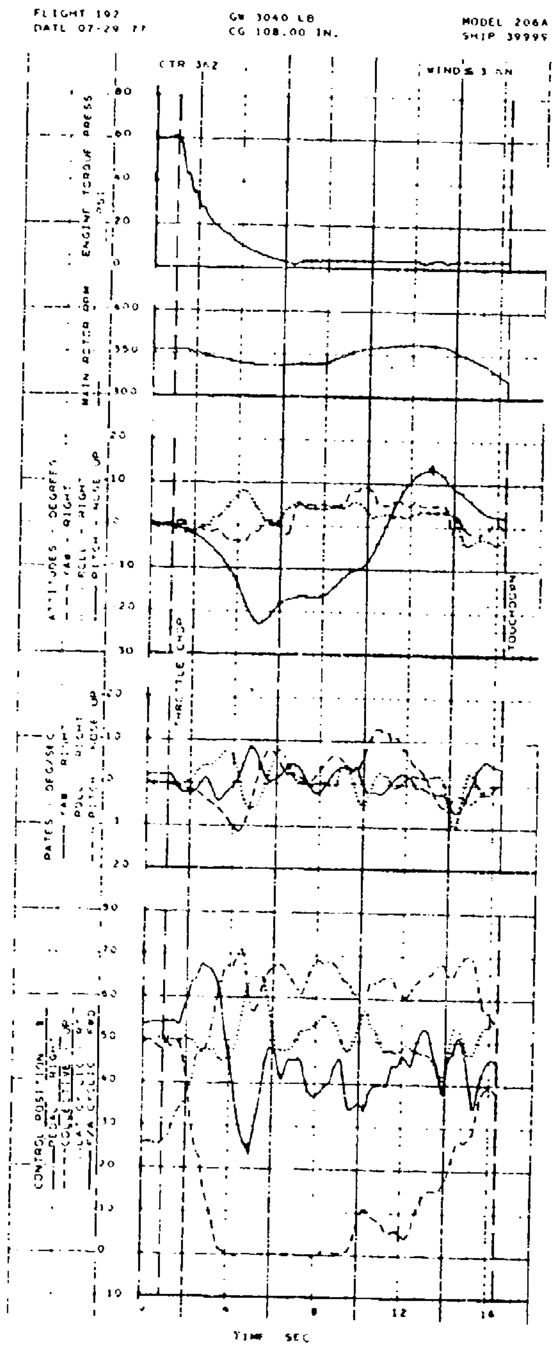


Figure 7. High hover point throttle chop from 350 feet - standard inertia rotor.

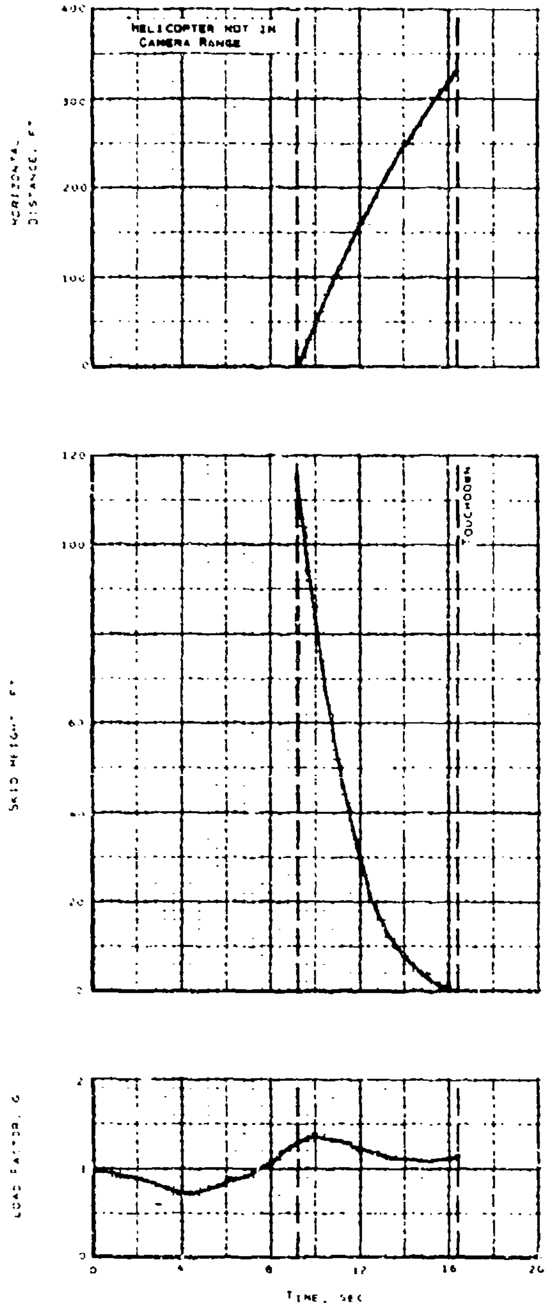


Figure 7. Concluded.

airborne. Comments by pilots performing these autorotations indicate that the landing task becomes much more difficult as the touchdown horizontal speed approaches zero.

2.3.4 Low Altitude, Low Airspeed Throttle Chop

Another example of a standard autorotational landing technique is shown in Figure 8. In this case, the simulated engine failure was initiated near the knee of the height-velocity diagram, at 100 feet altitude and a forward airspeed of 45 knots.

The pilot actions were similar to those used in the high hover case. Initial control motions were to drop collective to maintain rotor speed and to nose over to maintain airspeed. Since the entry condition was with forward airspeed, sufficient kinetic energy was available for the landing flare and the no of the helicopter was lowered to only 8 degrees nose down as opposed to the 23 degrees used for the hover entry case.

A nose-up flare was used to slow the rate of descent and produce a normal acceleration of 1.5g. With a peak nose-up attitude of 19 degrees, this indicates a deceleration of both rate of descent and horizontal velocity. This flare exchanges helicopter kinetic energy for rotor thrust and additional rotor energy.

At ab t 10 feet skid height, the nose of the helicopter was lowe to a landing attitude and the collective was raised to cu on the touchdown. The rotor speed was reduced to 280 rpm at uchdown, indicative of extracting most of the available rotor energy. The touchdown was made at a horizontal velocity of 20 feet per second.

Reviewing the results of these standard rotor inertia maneuvers, it is apparent that the pilot will use the kinetic energy of the helicopter to reduce the rate of descent if possible. Only after reaching an altitude of about 10 feet will the extraction of rotor energy be started. The energy in the rotor must be used to support the helicopter during the leveling of the aircraft after the flare, until touchdown. This maneuver must be coordinated precisely to avoid tail skid contact and subsequent nose-down pitching during final descent.

In assessing the difficulty of successfully accomplishing an autorotational landing, it was determined that increasing the available rotor energy would increase the safety margin in these maneuvers.

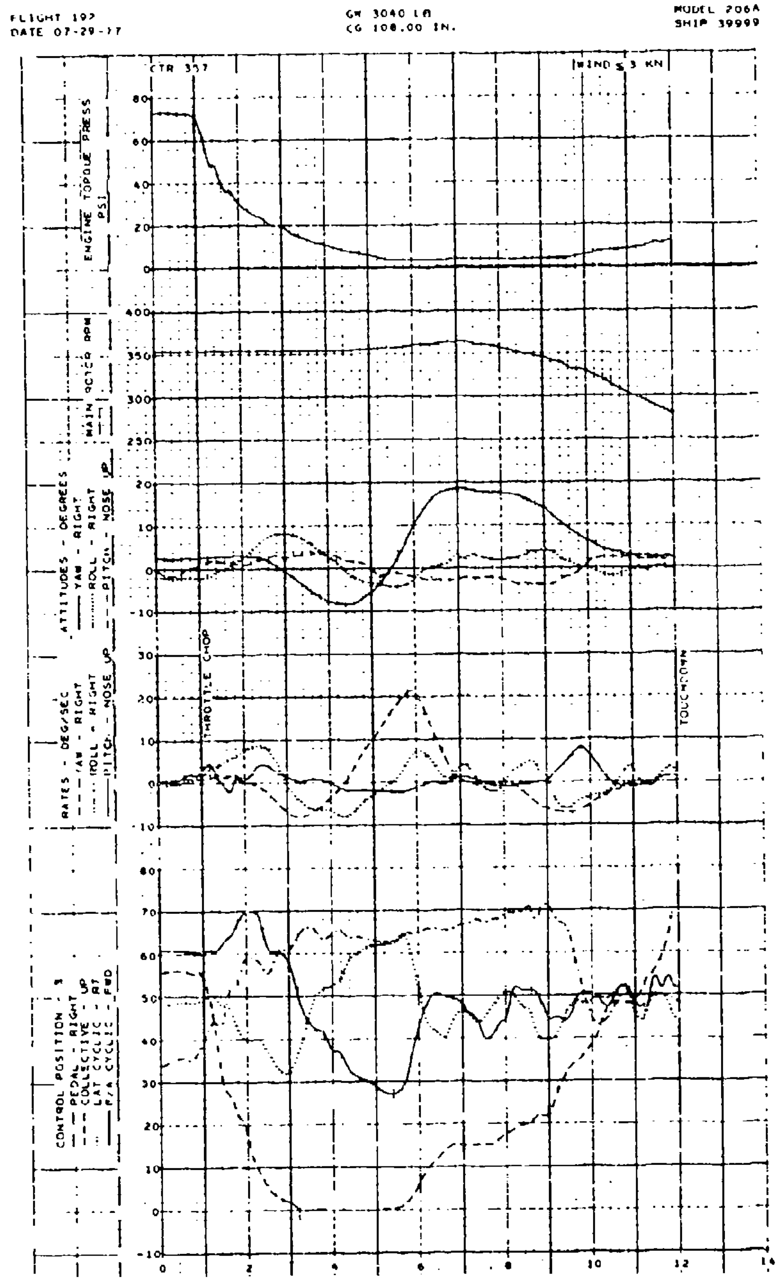


Figure 8. Throttle chop from knee at 100 feet, 45 knots - standard inertia rotor.

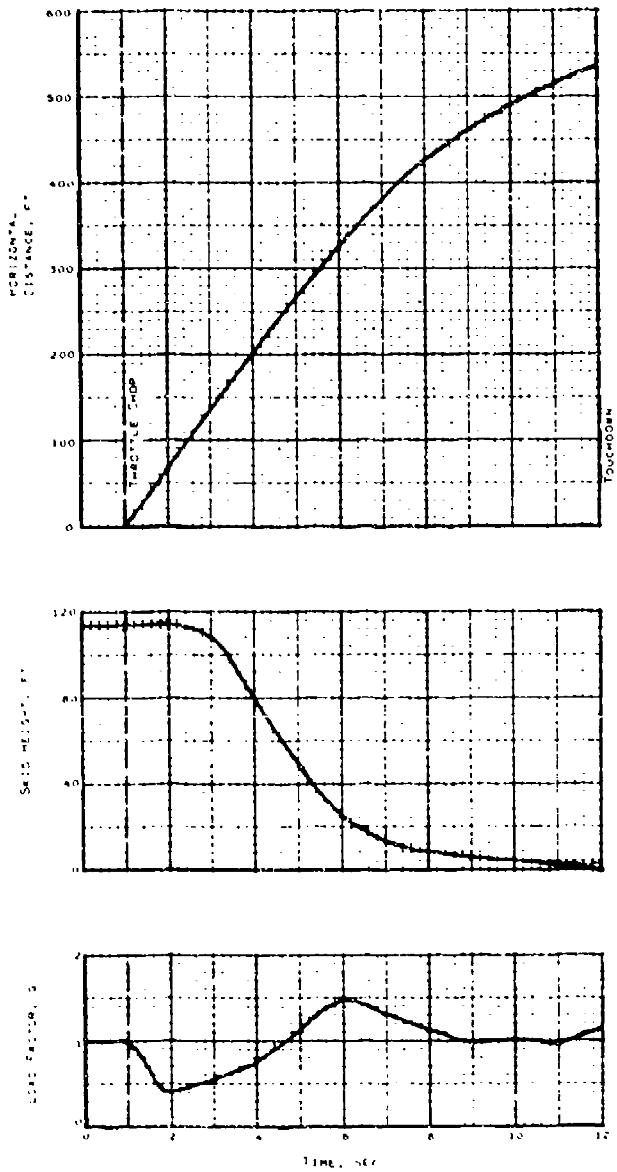


Figure 8. Concluded.

2.4 INCREASED ROTOR INERTIA TESTING

To investigate the effect of rotor inertia on autorotational landing performance, the rotor inertia of an OH-58A prototype helicopter was varied as described in Section 1. During the contracted flight test program more than 100 autorotational entries and landings were performed at three gross weights and three levels of rotor inertia. The entry conditions for these tests are indicated in Figures 9 and 10. Based on the results of these tests, the low-speed height-velocity restrictions for the three inertia levels are presented in Figure 11 along with the diagram for the standard OH-58A.

The highest inertia rotor allowed complete elimination of any height-velocity restrictions. Autorotational landings were accomplished safely from all entry conditions attempted, including high power climbs at low rotor speed.

For the mid-inertia rotor, a small restricted region remained between 75 and 125 feet altitude with airspeed less than 5 knots. Safe landings were made from hovering throttle chops at 85 and 100 feet, but the pilot felt the workload was too high for elimination of the restriction.

The lowest inertia rotor showed some differences from the standard OH-58A. This is to be expected since the chord of the rotor was increased over that of the OH-58A and, as discussed in Reference 4, the reduced blade loading of the test rotor will make a significant difference in the touchdown phase of the landing. The higher low hover point is a direct result of the lower blade loading.

2.4.1 Autorotational Landings with High Inertia Rotor

During this contracted program, the highest inertia rotor (I_b = 672 slug-ft^2) was used for both the BHT and the Army pilot evaluation. The following sections present data gathered during the contracted effort. It should be noted that all data recorded during Flight 171 (the BHT flights) were obtained under zero wind speed, the most demanding conditions for H-V testing.

2.4.1.1 Autorotational Index Throttle Chops

As discussed in Section 2.3.1, a throttle chop for a 4-foot hover provides an index of overall autorotational landing characteristics. With the high inertia rotor (I_b = 672 slug-ft^2) the pilot was able to hold the helicopter off the ground for 8.5 seconds after the throttle chop. This compares

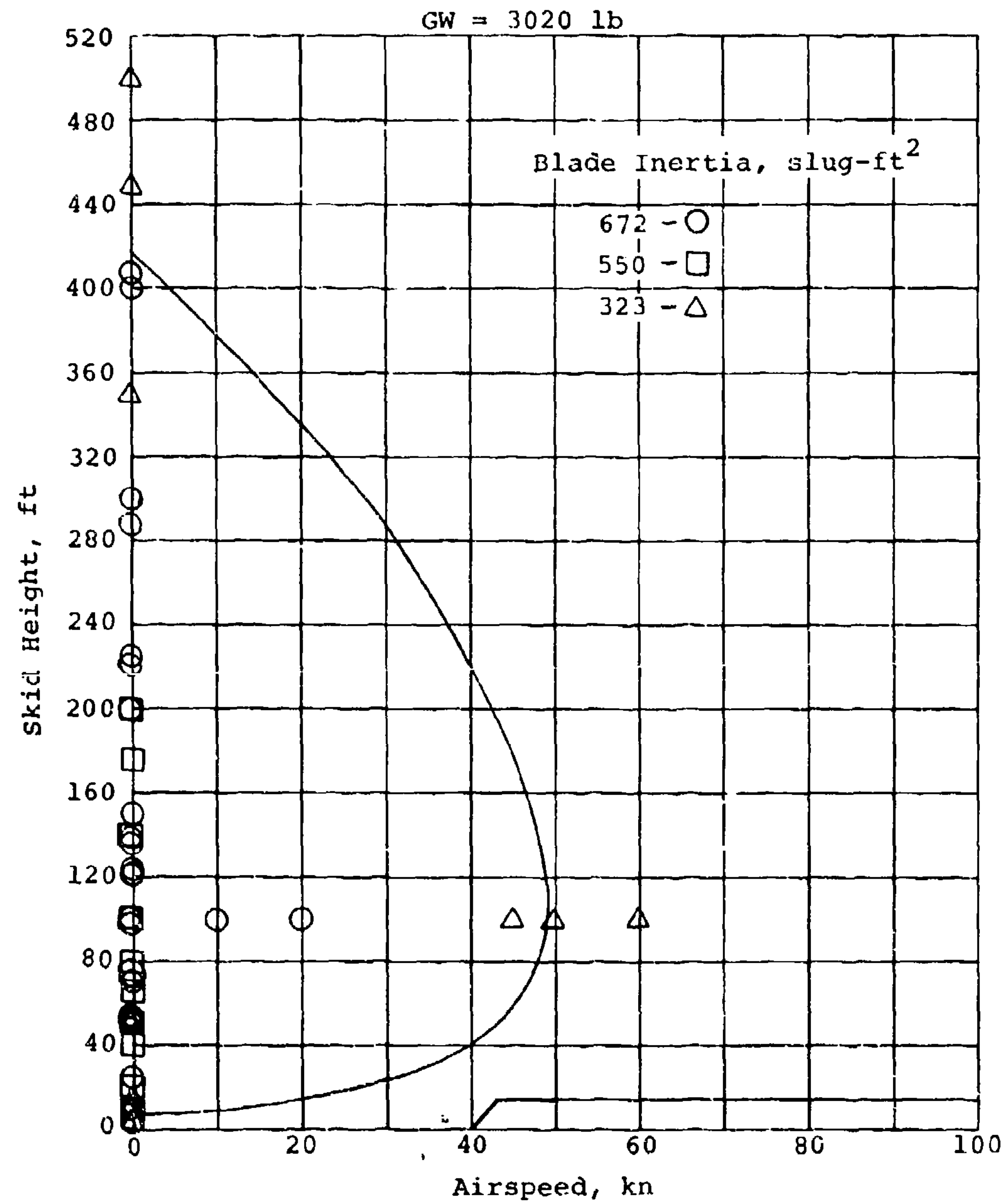


Figure 9. Summary of entry conditions, heavy gross weight.

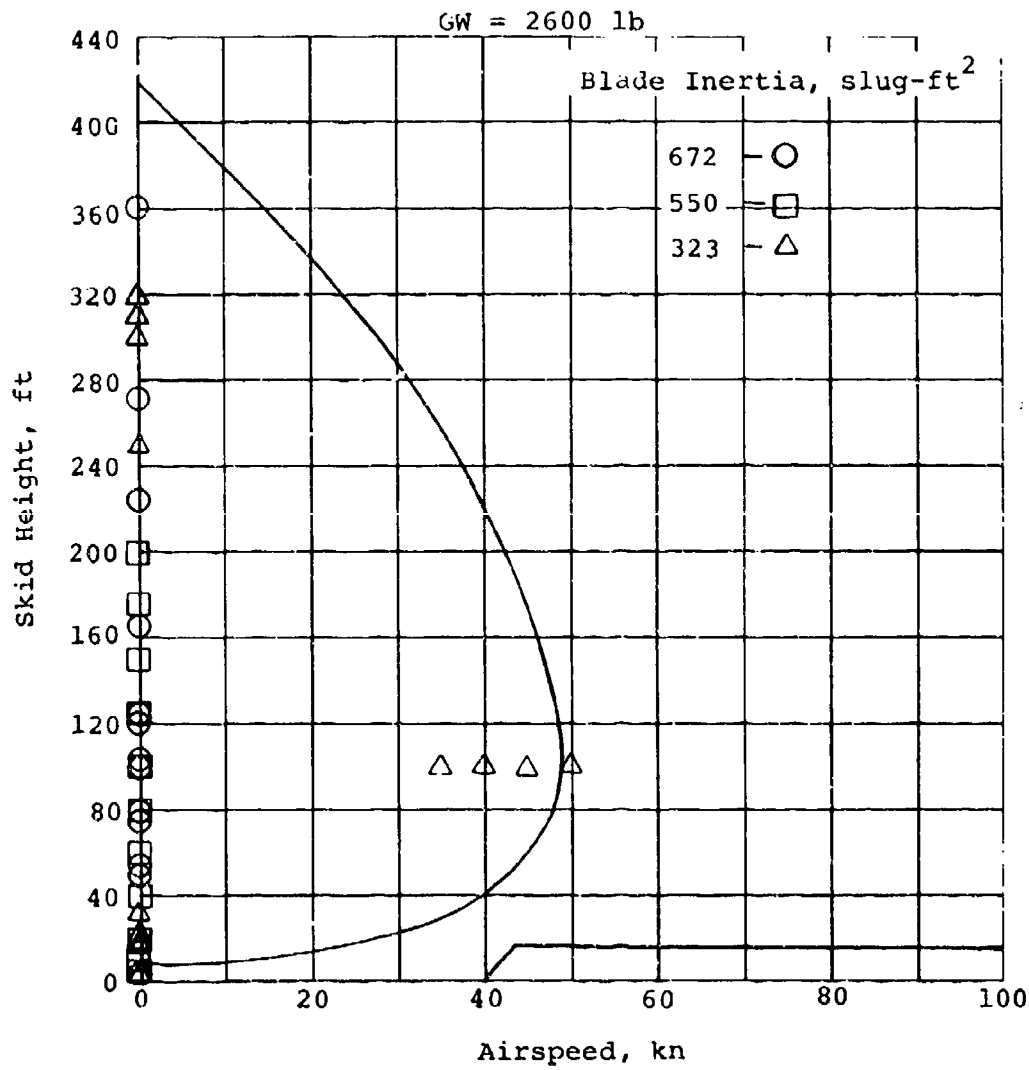


Figure 10. Summary of entry conditions, light gross weight.

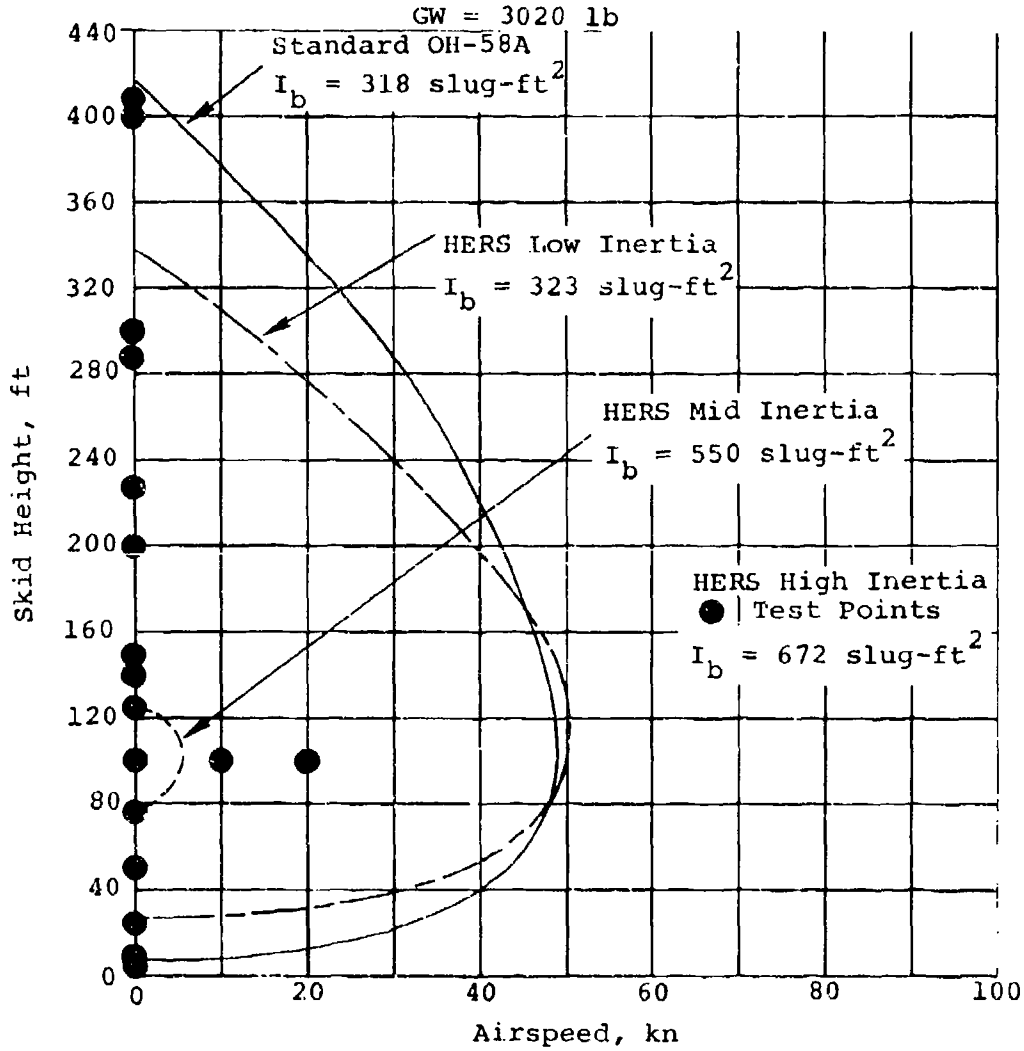


Figure 11. Comparison of HERS test results with standard OH-58A.

with 4.4 seconds for the standard inertia rotor. Figure 12 is an example of this maneuver.

2.4.1.2 Low-Altitude Hovering Throttle Chops

As the hovering altitude increases, a difference in pilot technique is noticeable. Up to a hovering altitude of about 50 feet, it was possible to drop the collective somewhat after the throttle chop in order to reduce the amount of rpm decay. This increased the rate-of-sink initially, but with the high rotor inertia, a collective pull near the ground provided a sufficient deceleration to allow a safe landing. Figure 13 presents a time history of a 50-foot hovering throttle chop in which the initial collective movement is a 15-percent drop. No comparable data exist for the standard inertia rotor since this flight condition is deep within the restricted area of the height-velocity diagram.

At hovering altitudes above 50 feet, the pilot technique transitioned into the technique described in the next section.

2.4.1.3 High-Altitude Hovering Throttle Chops

For the high-altitude hovering conditions, the pilot technique required for successful autorotational landings is altered significantly from that used with low inertia. A comparison of Figure 14, a simulated engine failure for a hover at 300 feet with the high inertia rotor, with a similar maneuver using the standard inertia rotor, as shown in Figure 7, illustrates the difference in techniques.

With the high inertia rotor (Figure 14), the collective is lowered and the nose of the helicopter is pitched down after the throttle chop. While similar to the low-inertia technique, this pitchover is primarily to avoid a vertical descent into the helicopter's own rotor wake. Consequently, the severity of the pushover was less than that required with the low inertia rotor. The high inertia case entered the grid camera range with a horizontal velocity of 40 ft/sec and a vertical velocity of 36 ft/sec compared to the 70 and 46 ft/sec respectively for the standard inertia case of Figure 7. The lower horizontal speed with the HERS is an indication that the pilot feels the rotor energy is sufficient without having to build kinetic energy during the descent.

At approximately 140 feet, the pilot initiates a steady collective pull to reduce the rate of descent. A cyclic flare is also started shortly after the collective pull in order to reduce the horizontal velocity. This contrasts with the standard

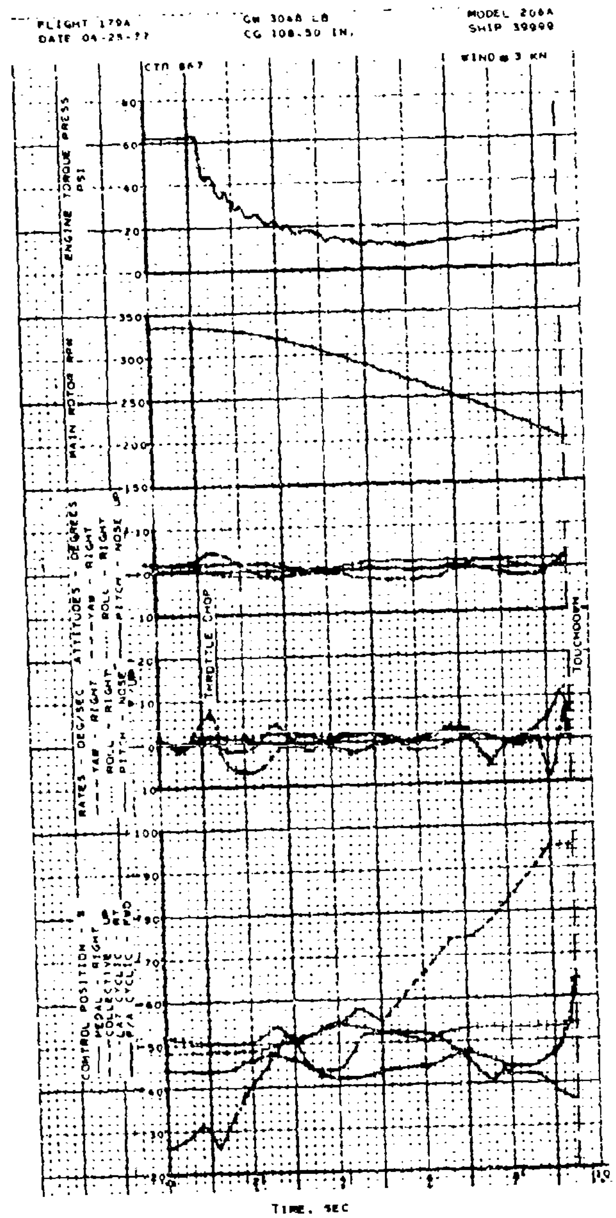


Figure 12. Autorotational index throttle chop - high inertia rotor.

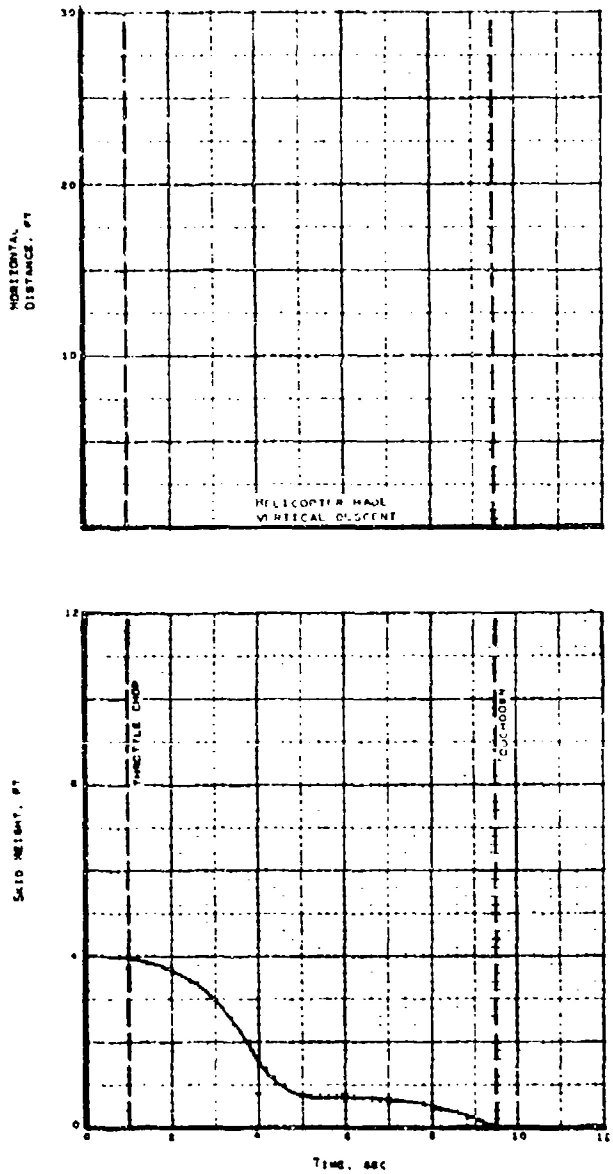


Figure 12. Concluded.

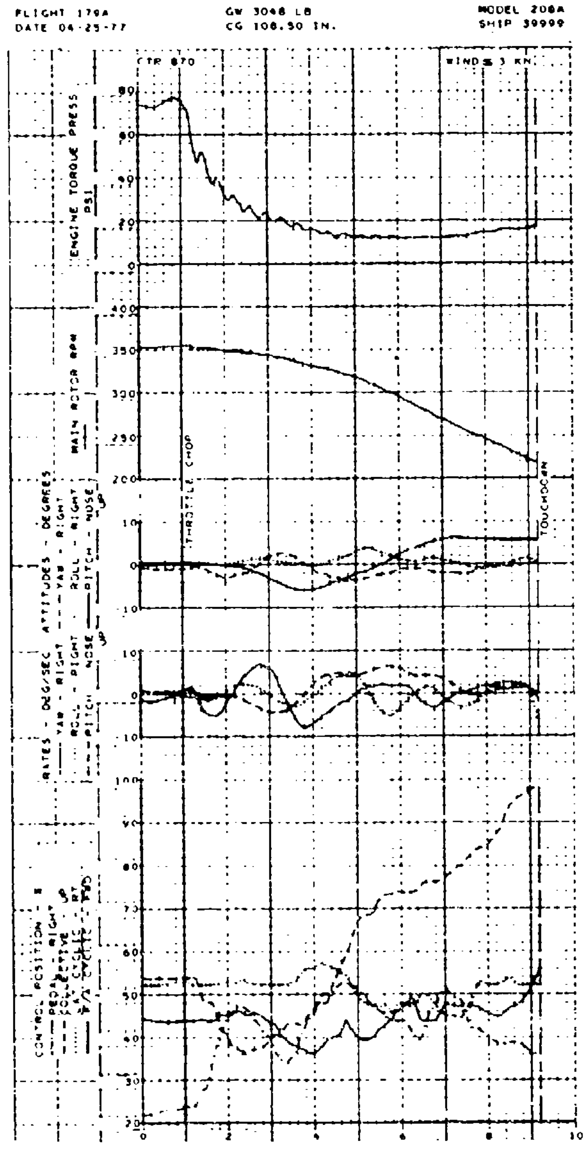
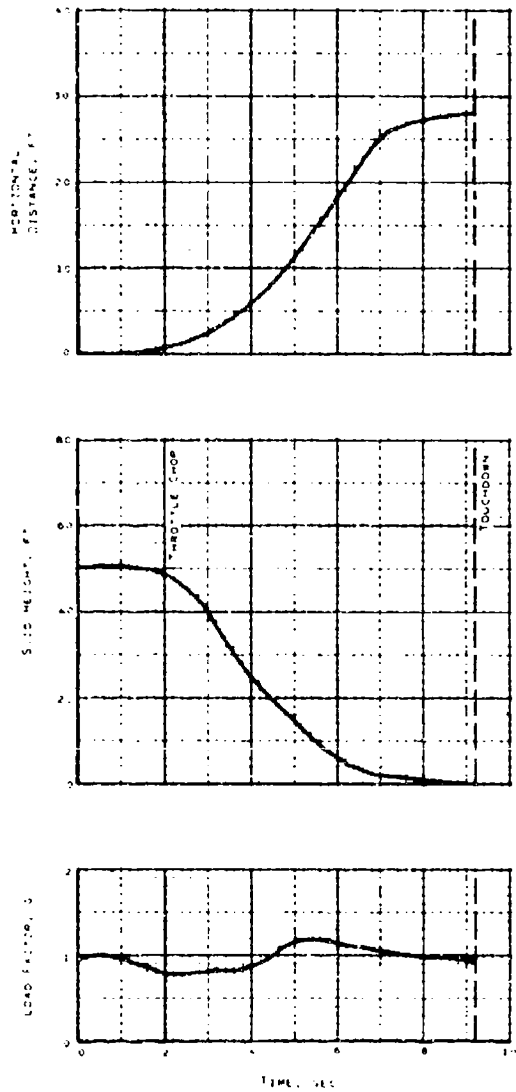


Figure 13. Hovering throttle chop from 50 feet - high inertia rotor.



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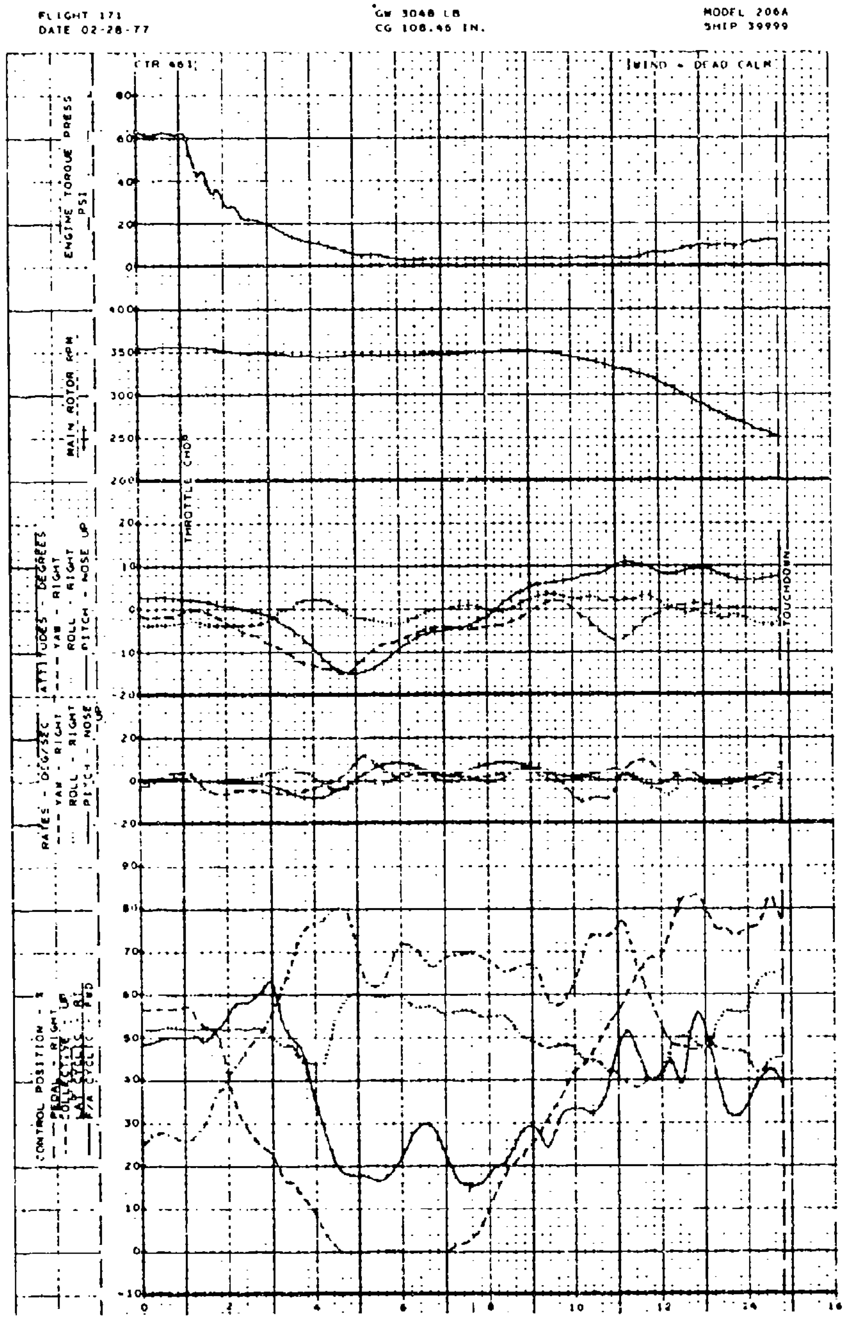

Figure 14. Hovering throttle chop from 300 feet - high inertia rotor.

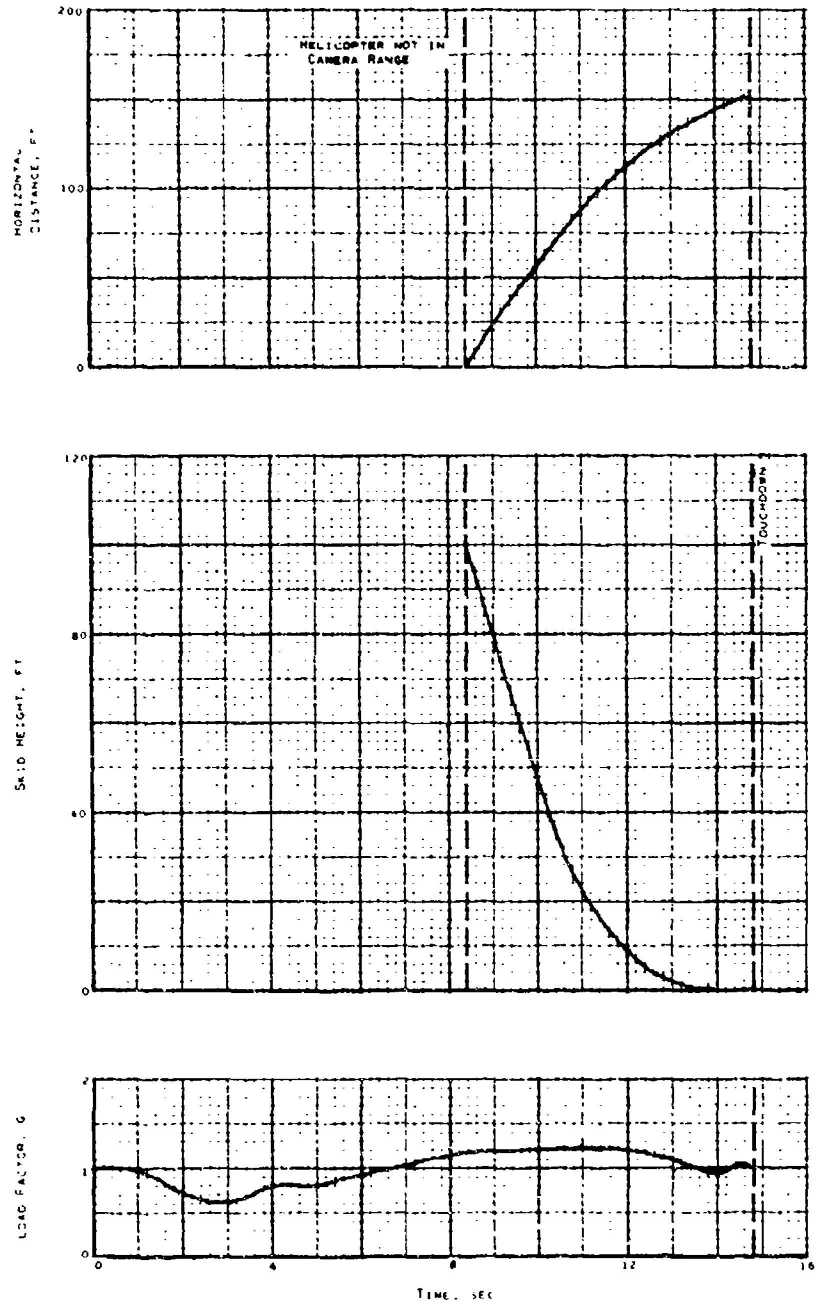

Figure 14. Concluded.

inertia case where the final collective pull was not started until the altitude was just under 30 feet. The standard inertia technique is to reduce the rate of descent primarily with a cyclic flare and to save the use of collective until as late as possible in landing.

Touchdown occurred after the rate of descent was reduced to zero about 1 foot above the runway. The horizontal touchdown velocity was 14 ft/sec and the main rotor speed reduced to 250 rpm.

The high-altitude hovering throttle chop illustrates the major differences in landing techniques between that used with the HERS and the standard inertia rotor. With HERS, the pilot requires less kinetic energy and thus less horizontal speed. The collective is used to reduce rate of descent and the cyclic flare is primarily to reduce touchdown horizontal velocity. This technique allows a lower pilot workload when performing autorotational landings.

2.4.1.4 High-Speed, Low-Altitude Throttle Chops

An autorotational entry and landing from the high-speed and low-altitude condition is also easier to perform with the HERS although demonstration in this region of the deadman curve was not performed. To investigate this condition, a series of throttle chops were performed from level flight entry conditions at speeds up to 90 knots at 40 to 60 feet altitude. An example of these maneuvers is shown in Figure 15.

From the initial altitude of 66 feet, the helicopter was able to climb to a safe altitude while decelerating and was set up for an autorotational landing before descending to the initial altitude. This indicates that recovery from low-altitude, high-speed engine failures would be enhanced with the HERS.

2.4.2 Autorotational Landings with Mid-Inertia Rotor

After completion of the highest inertia rotor testing, one tip weight was removed from each rotor blade to obtain a mid-inertia (I_b = 550 slug-ft^2) configuration. Test techniques were similar to those employed during the high inertia tests and pilot technique was modified very little.

An example of hovering throttle chop at 4 feet skid height is presented in Figure 16. No horizontal or vertical displacement time history is available for this record due to an instrumentation problem. It is apparent that the pilot applied a gradual increase in collective pitch and remained airborne for 5.8 seconds after the throttle chop.

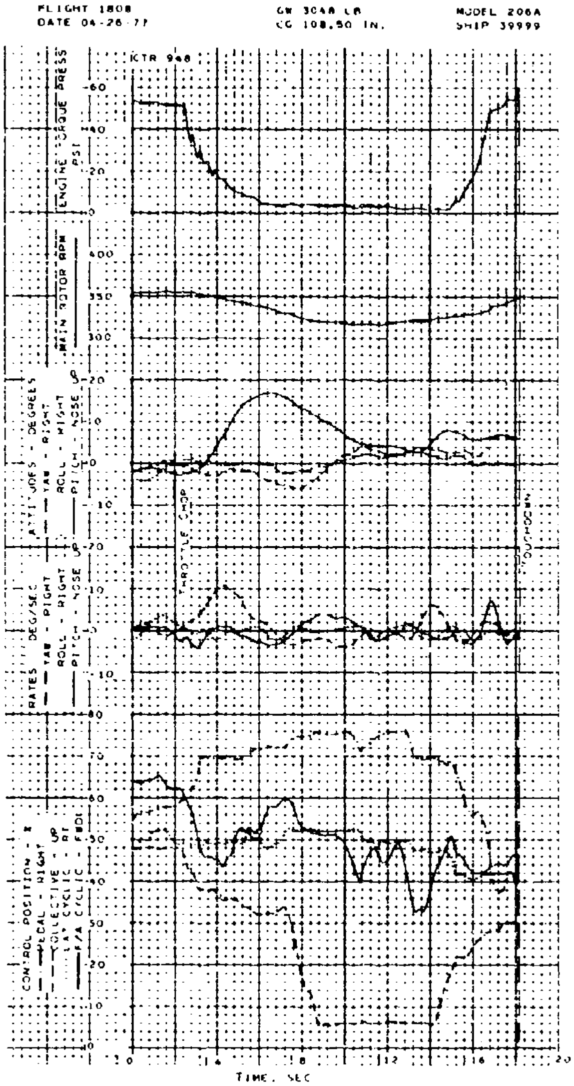

Figure 15. High-speed, low-altitude throttle chop - high inertia rotor.

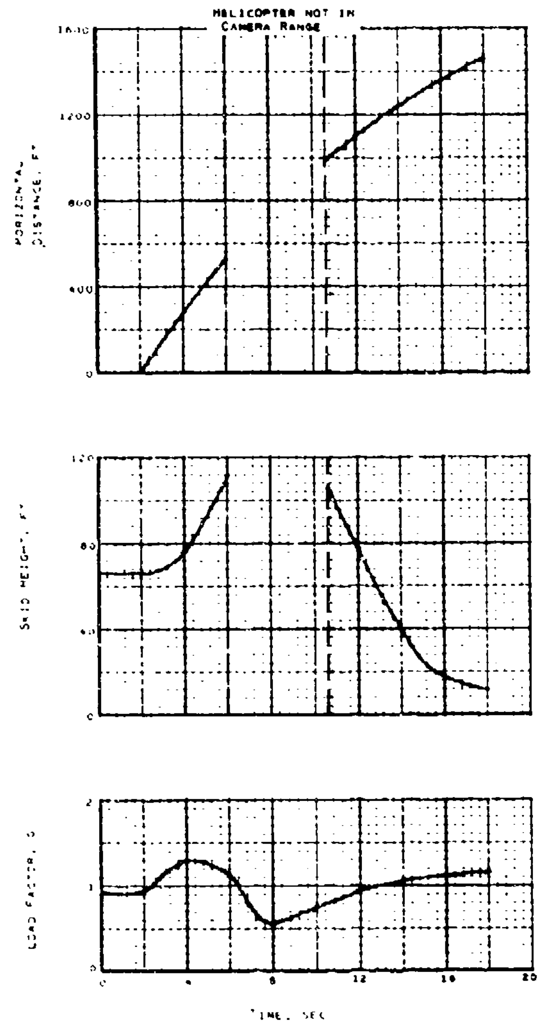

Figure 15. Concluded.

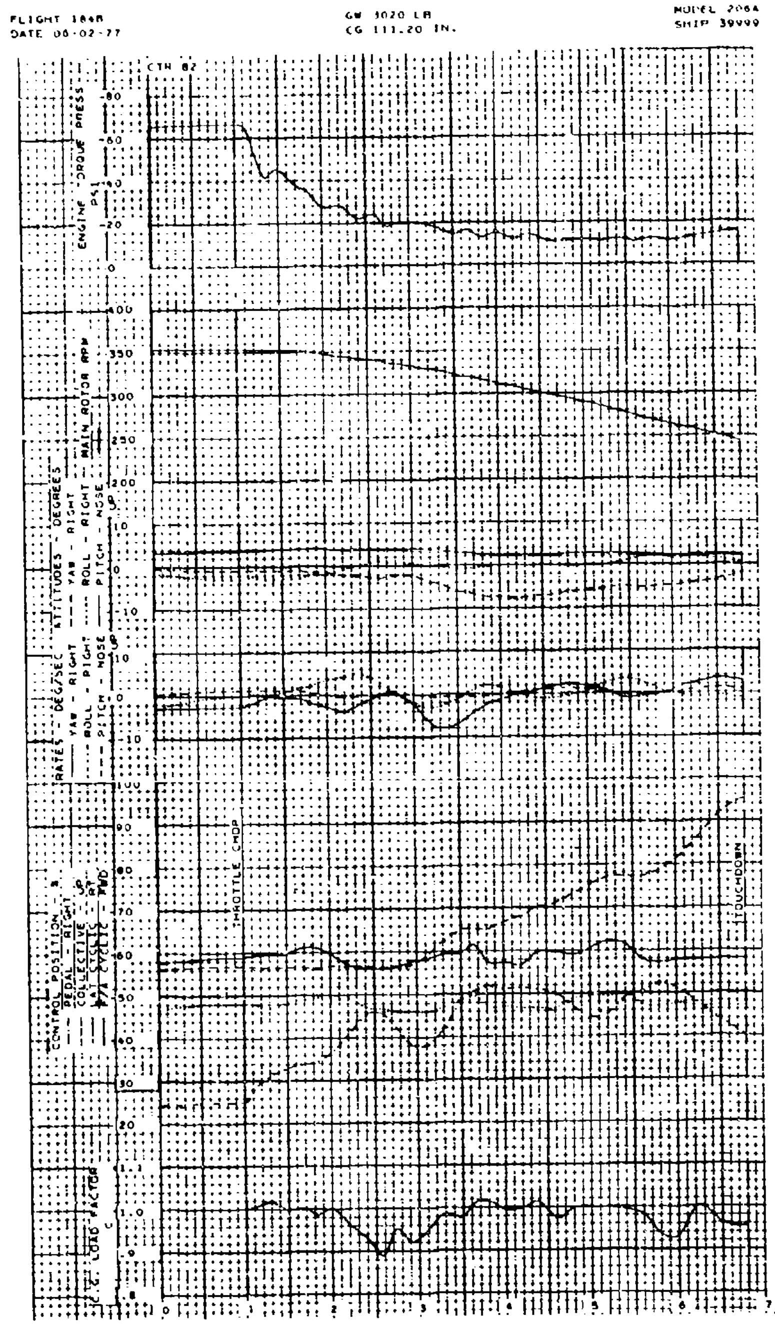

Figure 16. Autorotational index throttle chop - mid-inertia rotor.

Another sample of the mid-inertia rotor is presented in Figure 17. This record demonstrates a successful autorotational landing from a hovering throttle chop at an altitude of 100 feet. Even though a landing was accomplished, the test pilot judged that an engine failure from this condition would be more difficult than the average pilot would likely tolerate, and consequently a restriction on the height-velocity diagram was required.

2.5 HERS EFFECTS ON PILOT WORKLOAD FACTORS

During the course of testing these rotor configurations, many factors were identified as important with regard to a successful autorotational landing. The following subsections will discuss these factors both as they are affected by the HERS and the autorotation landing maneuver in general.

2.5.1 Rotor RPM Bleed Rates

Starting with the initial throttle chop, the rate at which the main rotor speed bleeds down to 330 rpm (low rpm warning), with collective held constant, is decreased significantly with increasing rotor inertia as shown in Figure 18. The time before reaching 330 rpm varies inversely with the power required, and the higher the blade inertia the longer it takes to bleed the rotor speed down. This allows the pilot more time for recognition of the engine failure as the helicopter will still yaw nose left as the main rotor torque goes to zero. Thus, the HERS will provide additional time to identify the problem before rotor speed becomes critically low.

2.5.2 Effect of Collective Time Delays

The effect of delaying the collective drop after a simulated engine failure was investigated at 50 and 200 feet hover entries. Figure 19 presents a time history of a hovering throttle chop at 50 feet with a collective delay of 0.7 second while Figure 20 illustrates a similar maneuver with a 1.7 second delay. With the longer delay, the rpm decays further but the pilot simply drops the collective further to regain it. Differences in pilot technique for the pitchover between these two maneuvers result in a very similar landing condition. Pilot opinion rated the longer delay landing only slightly more difficult. Table 3 summarizes the data obtained during this evaluation.

The majority of these tests were performed with delays in control inputs consistent with the external cues presented to the pilot. Consequently, test results are felt to be realistic in terms of operational use of the HERS.

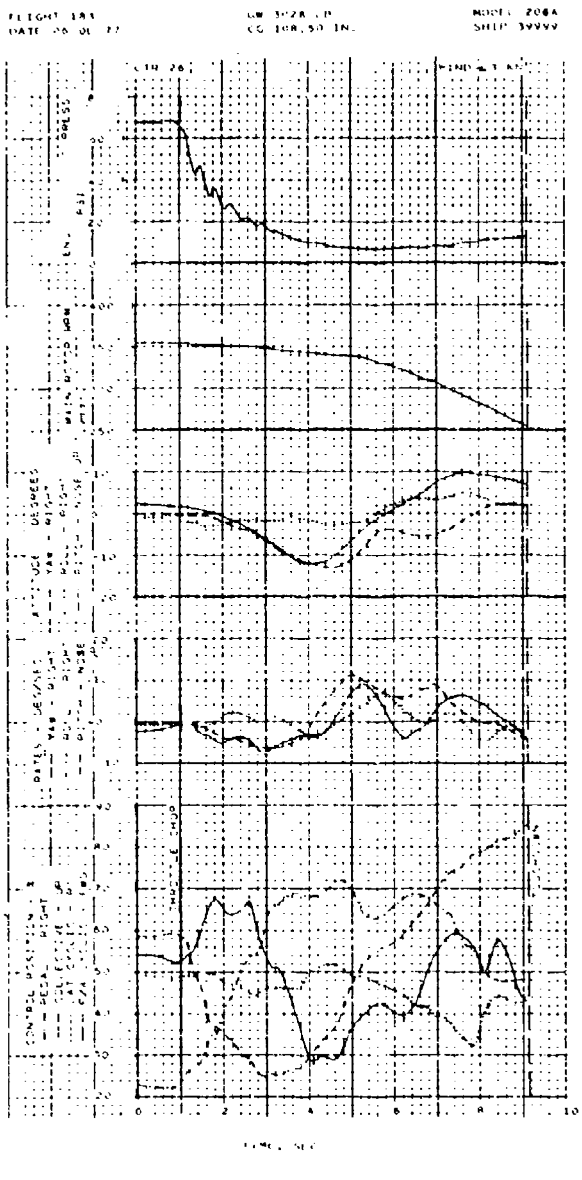

Figure 17. Hovering throttle chop from 100 feet - mid-inertia rotor.

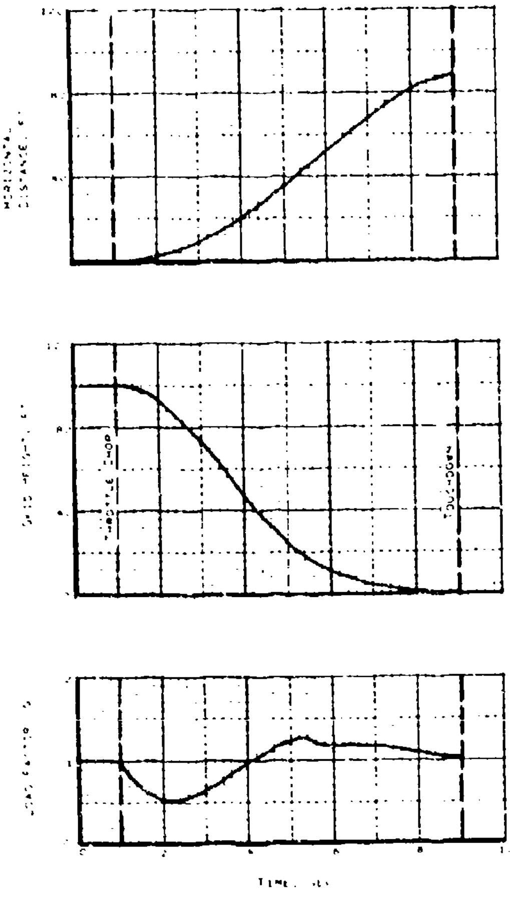

Figure 17. Concluded.

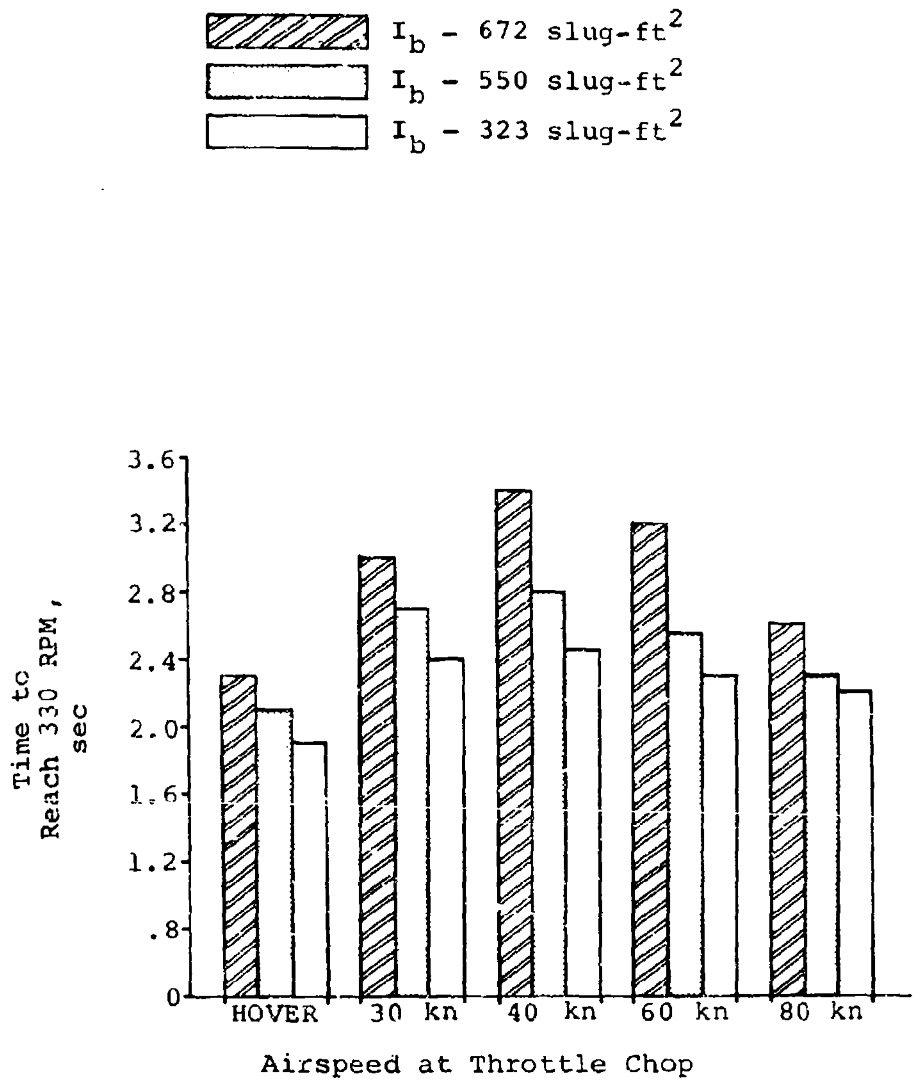

Figure 18. Time required to bleed rotor speed to low RPM warning.

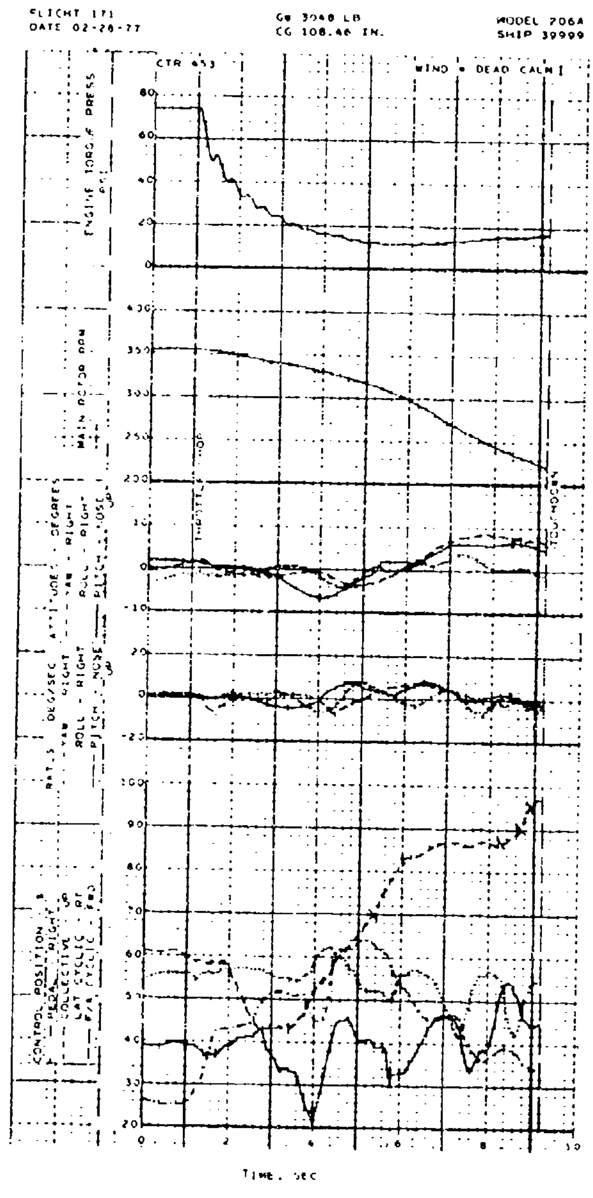

Figure 19. Hovering throttle chop from 50 feet, 0.7-second collective delay - high inertia rotor.

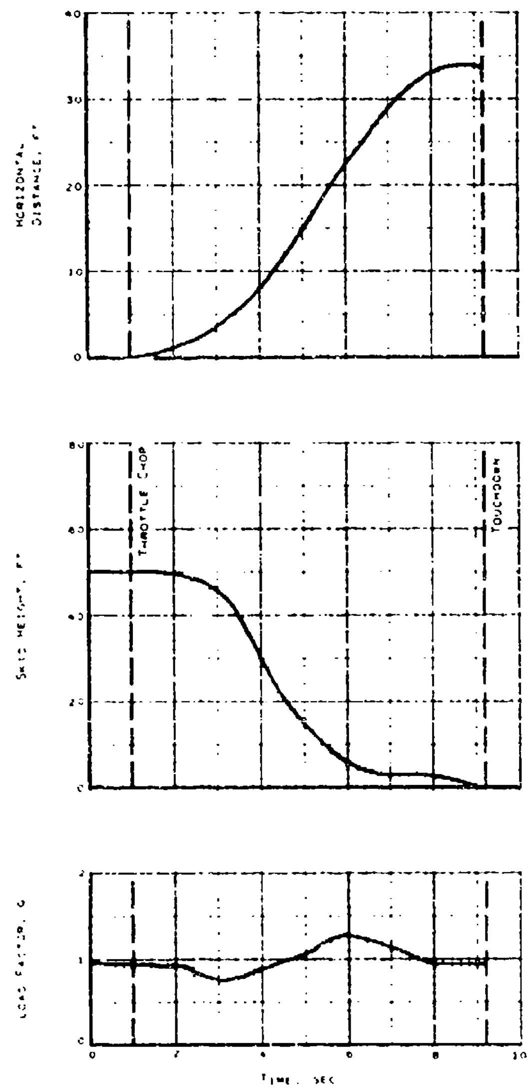

Figure 19. Concluded.

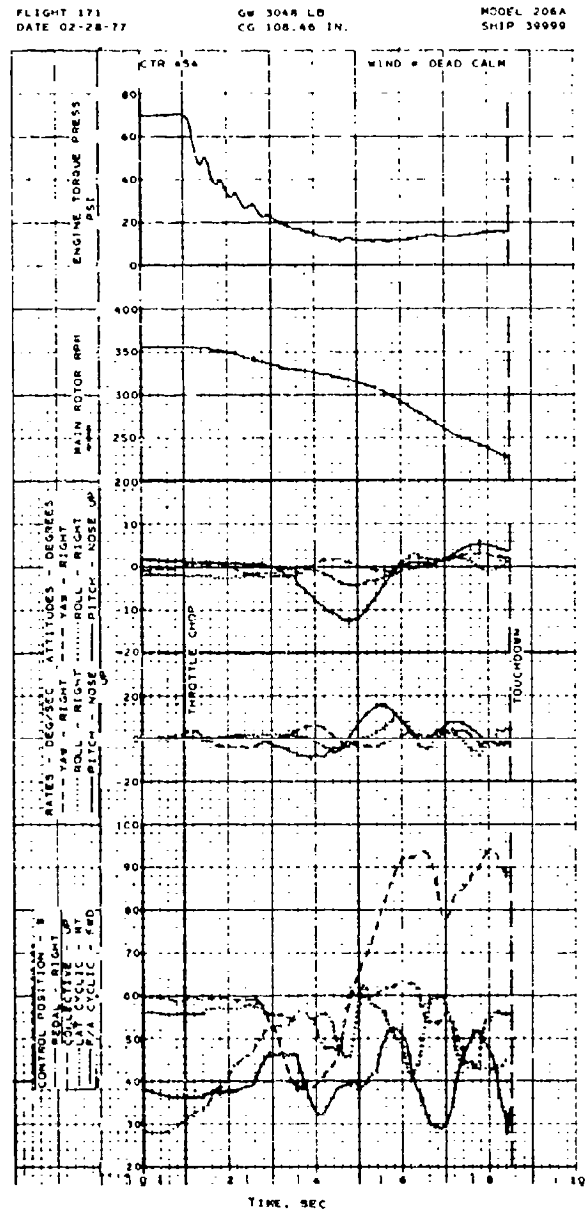

Figure 20. Hovering throttle chop from 50 feet, 1.7-second collective delay - high inertia rotor.

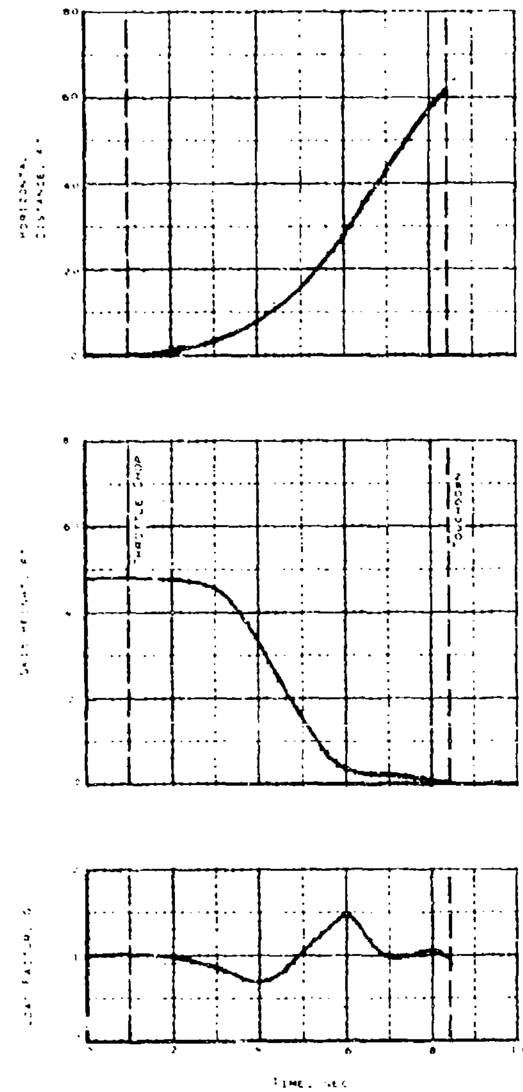

Figure 20. Concluded.

TABLE 3. EFFECT OF TIME DELAY ON HOVERING SIMULATED ENGINE FAILURES

	50 Feet Hover		200 Feet Hover	
Collective Delay (sec)	0.7	1.7	0.25	0.8
RPM at Collective Drop (rpm)	350	340	355	347
Time of Touchdown (sec)	8.1	7.6	12.3	11.9
Minimum Pitch Attitude (deg)	-7.0	-12.5	-12	-16
Maximum Rate of Descent (fps)	15	19	34	39
Maximum Horizontal Speed (fps)	8	16	35	40
Touchdown Horizontal Speed (fps)	0	11	10	32
RPM at Touchdown	221	227	247	277
Touchdown Vertical Speed (fps)	≅0	≅0	≅0	≅0

2.5.3 Sink Rates in Autorotation

In order to compare steady autorotation performance with the standard OH-58A, a series of steady autorotational descents were made at varying airspeeds and main rotor speeds. The results, plotted in Figure 21 and compared with data from Reference 5, indicate no significant difference in autorotational rate of descent with airspeed at constant rotor speed. However, the variation of rate of descent with rotor speed is higher than the standard OH-58 due to the increased profile power required of the modified rotor blades.

2.5.4 Control of Rotor Speed

During autorotational descents, pilot control of rotor speed with collective stick inputs was found to present no problems. No tendencies toward potential pilot-induced oscillations were found, indicating that the aerodynamic controls affecting rotor speed were more important than rotor inertia effects for this configuration.

2.5.5 Avoiding Vertical Descents

The importance of maintaining some forward airspeed during the autorotational descent is well known, as indicated in Reference 6. If a vertical descent is attempted, a region of roughness is encountered as the rotor descends into its own wake when the rate of descent is near the rotor-induced velocity magnitude. In this region, the helicopter is difficult to control.

Normally, the pilot maintains some forward airspeed to avoid this region. However, during the test program with the highest inertia rotor, an inadvertant vertical descent was caused by an unexpected gust. The test was being performed in calm wind conditions with occasional small gusts from variable directions. A gust caused an initial left yaw which was countered with a large right pedal input as shown in Figure 22 and a near vertical descent ensued. As the helicopter settled into its own wake, the rate of descent increased rapidly and full up collective was applied. The rate of descent was arrested prior to ground contact, but enough thrust was generated by the full up collective for the helicopter to lift-off again. A softer touchdown was made following the lift-off.

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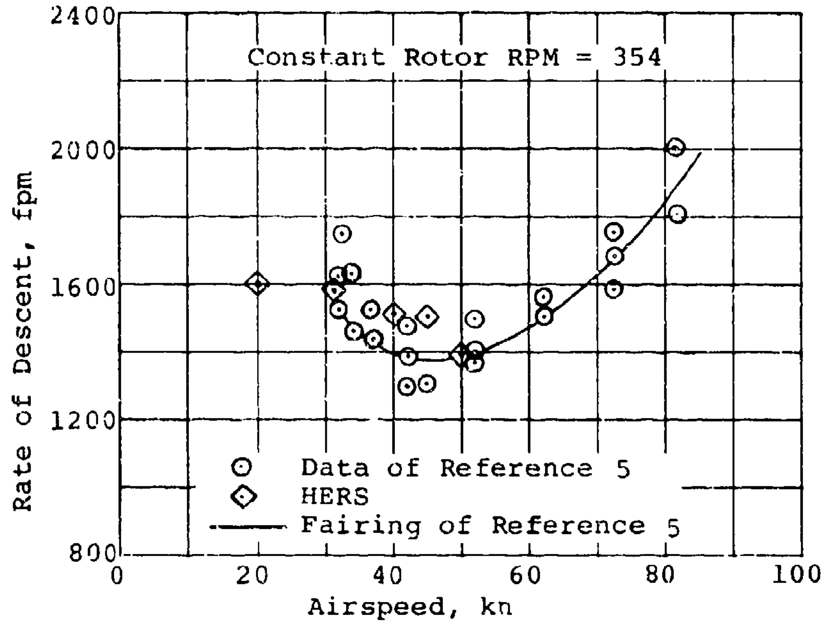
[5] G. M. Yamakawa, J. C. Watts, AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST - PRODUCTION OH-58A HELICOPTER UNARMED AND ARMED WITH XM27E1 WEAPON SYSTEM, PERFORMANCE, USAASTA Technical Report 68-30, U.S. Army Aviation Systems Test Activity, Edwards AFB, California, September 1970, AD875793.

[6] J. M. Drees, A THEORY OF AIRFLOW THROUGH ROTORS AND ITS APPLICATION TO SOME HELICOPTER PROBLEMS, The Journal of the Helicopter Association of Great Britain, Vol. 3, No. 1, 1949.



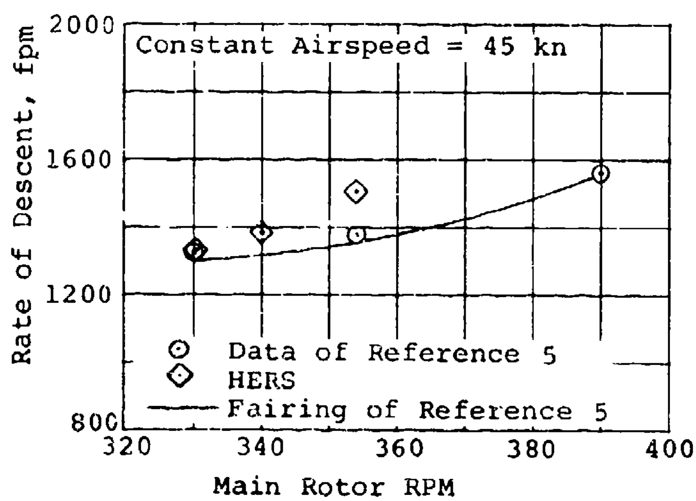


Figure 21. Comparison of steady autorotational sink rates with standard OH-58A.

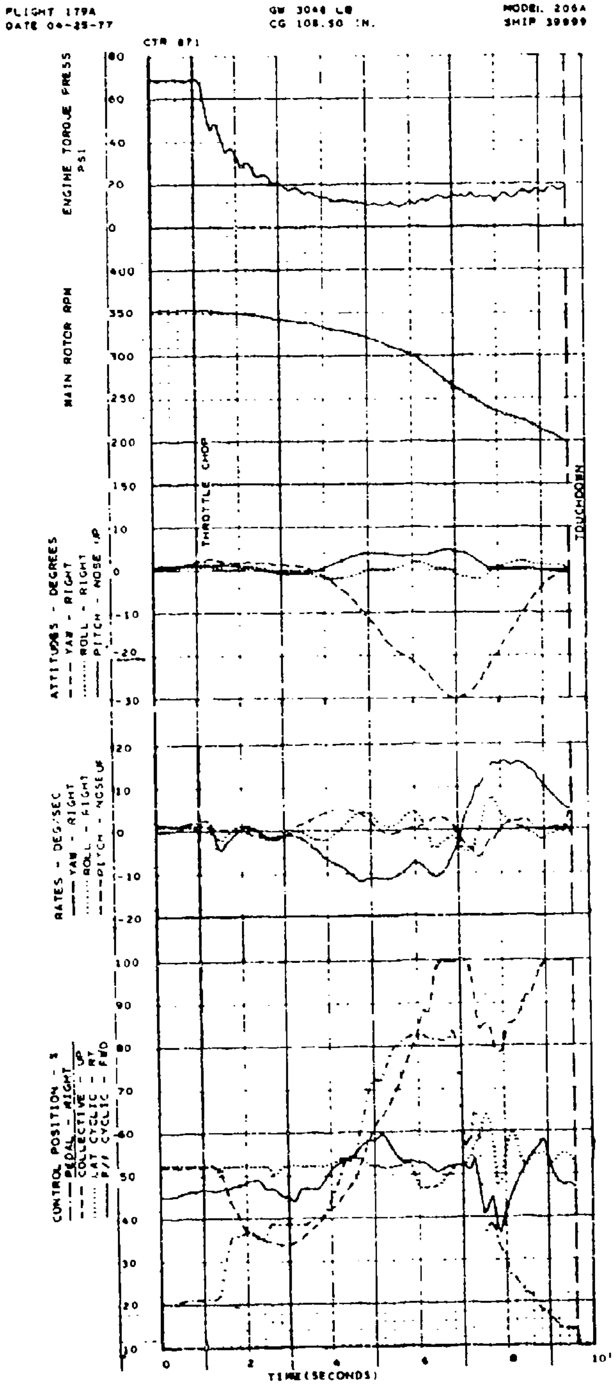


Figure 22. Hovering throttle chop from 75 feet with gust disturbance - high inertia rotor.

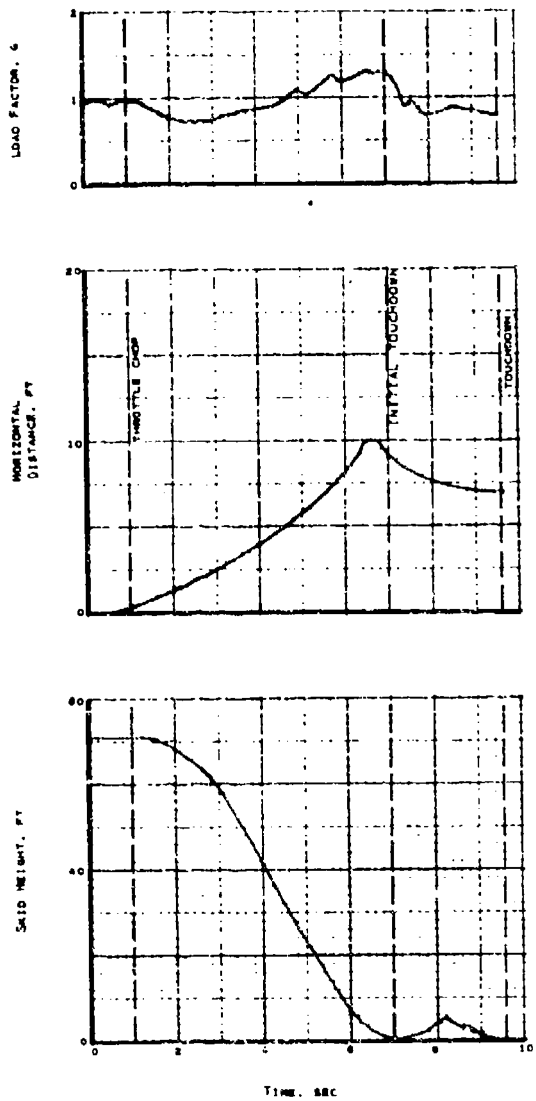


Figure 22. Concluded.

2.5.6 Autorotational Landings

The most critical phase of the H-V test is the final approach to a landing. Several factors affect the pilot workload required for a safe execution of the landing. The following subsections outline factors which have been found to significantly influence the autorotational landing and that are affected by the HERS.

2.5.6.1 Landing Techniques

The use of the High Energy Rotor System allows a modification of pilot technique that increases the safety margin for landing. Several autorotational landings were performed to investigate the modified technique and to compare it to the technique employed with the standard OH-58A.

Figures 23 and 24 show the pilot control motions and flight profile for the standard and HERS techniques. Superimposed on the flight profile are reference lines indicating the helicopter pitch altitude at one-second intervals. Close spacing of the reference lines indicates a slow maneuver while larger spacing indicates a faster maneuver. Both records were for an autorotational entry condition of 40 to 50 knots level flight at more than 100 feet altitude. For the standard technique (Figure 23) a cyclic flare is initiated just below 100 feet and held to about 20 feet above the ground. This resulted in a nose up attitude of 18 degrees. The collective was increased slightly during the flare to prevent rotor overspeed. Only after reaching 15 feet does the final collective pull begin.

In contrast, with the HERS (Figure 24) the collective is initially pulled at an altitude of about 60 feet and is raised almost continuously until touchdown. A short cyclic flare is applied at about 20 feet altitude to reduce the horizontal velocity. The absence of a cyclic flare allows the pilot to hold the maximum pitch attitude to just over 10 degrees nose up.

The HERS technique provides an additional safety margin during autorotational landing from almost any entry condition. Figure 25 compares the pitch attitude and rotor speed for the two techniques previously shown in Figures 23 and 24. The decrease in maximum pitch attitude due to HERS from 18 to 10.5 degrees affords the pilot a much better view of the landing site during the last portion of the landing. In addition, the high pitch attitude must be reduced to near zero before ground contact to prevent a tail first touchdown and subsequent pitch-down to a hard landing. The elimination of the high

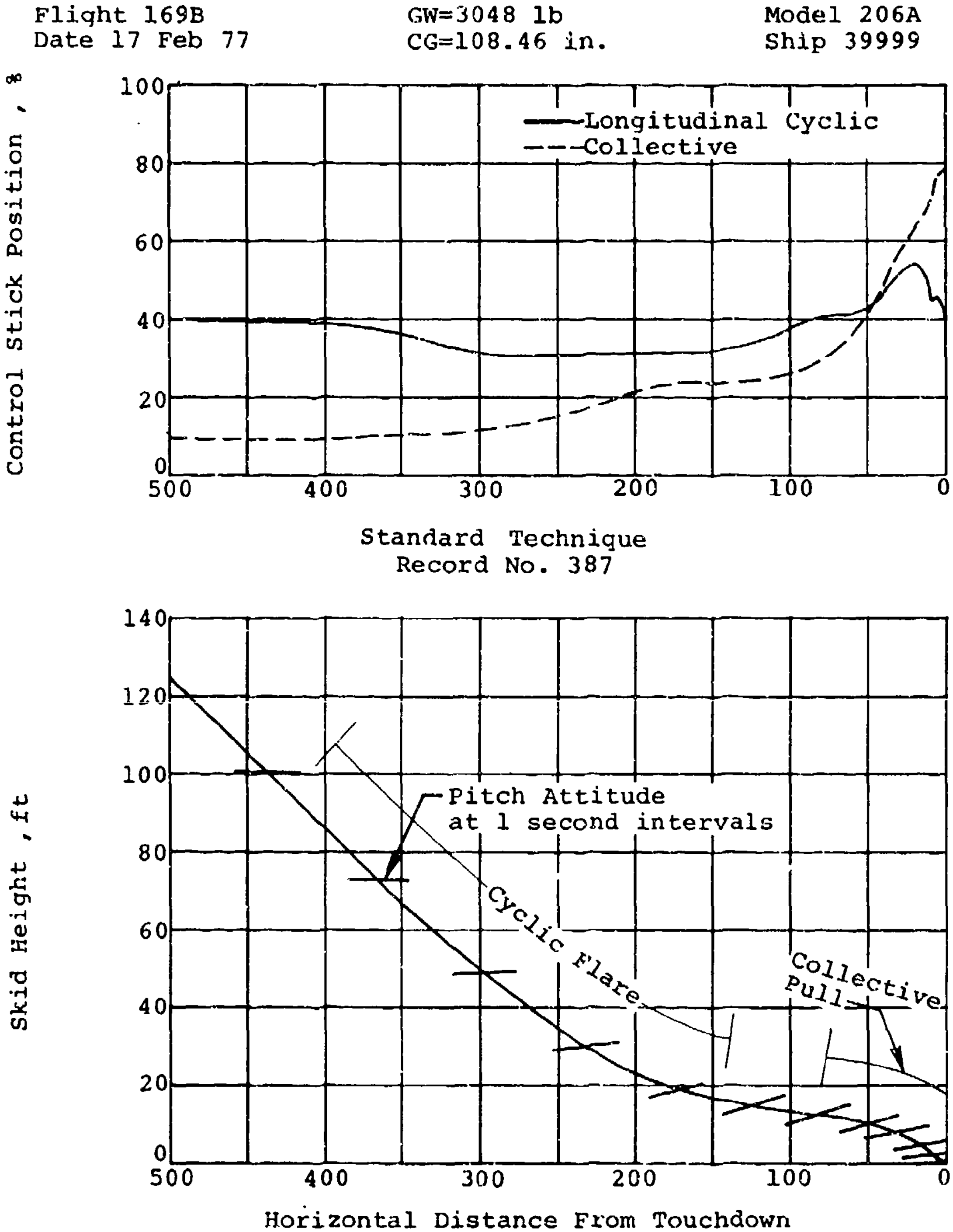


Figure 23. Autorotation landing - standard technique.

Flight 169B
Date 17 Feb 77

GW=3048 lb
CG=108.46 in.

Model 206A
Ship 39999

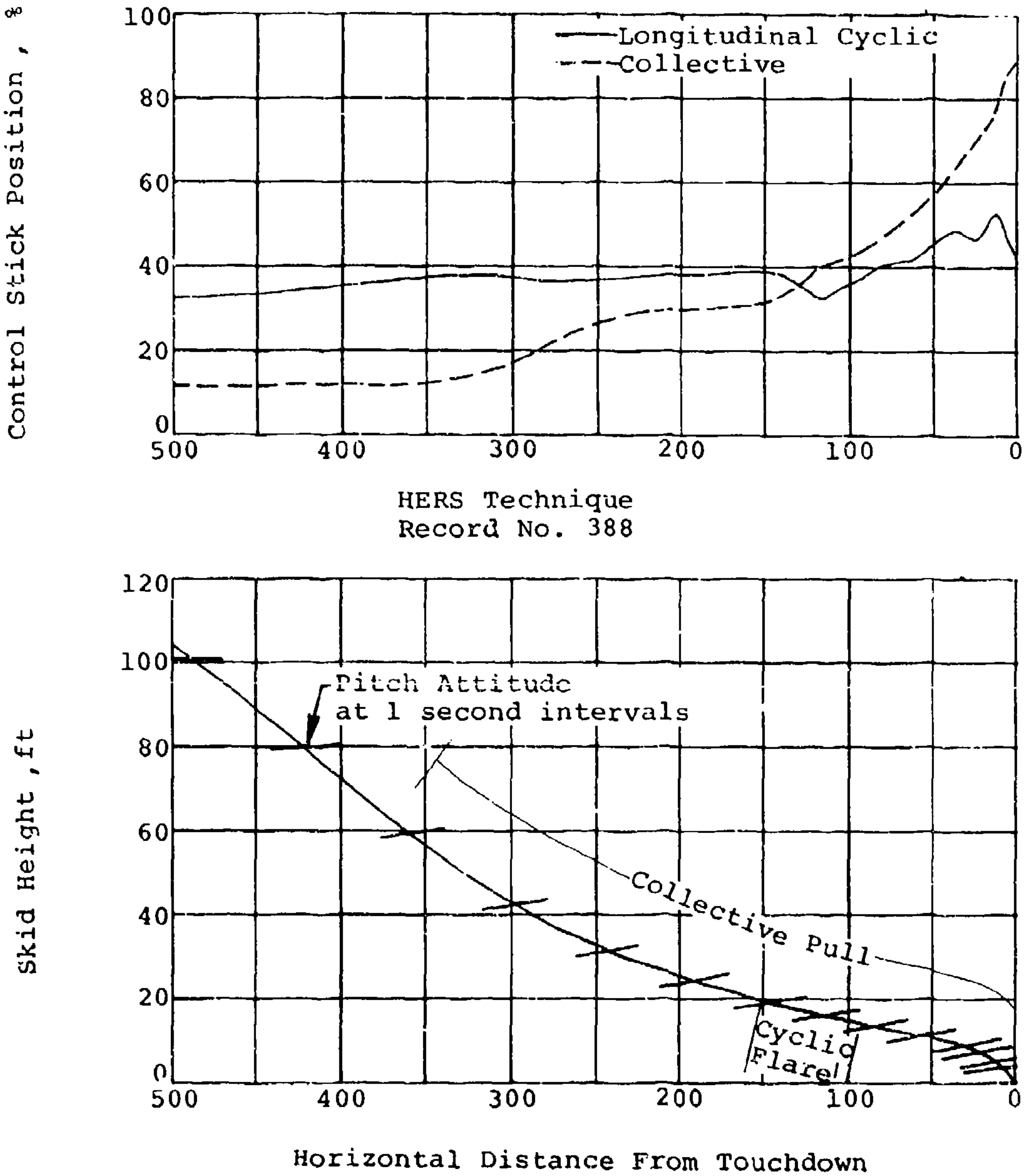


Figure 24. Autorotation landing - HERS technique.

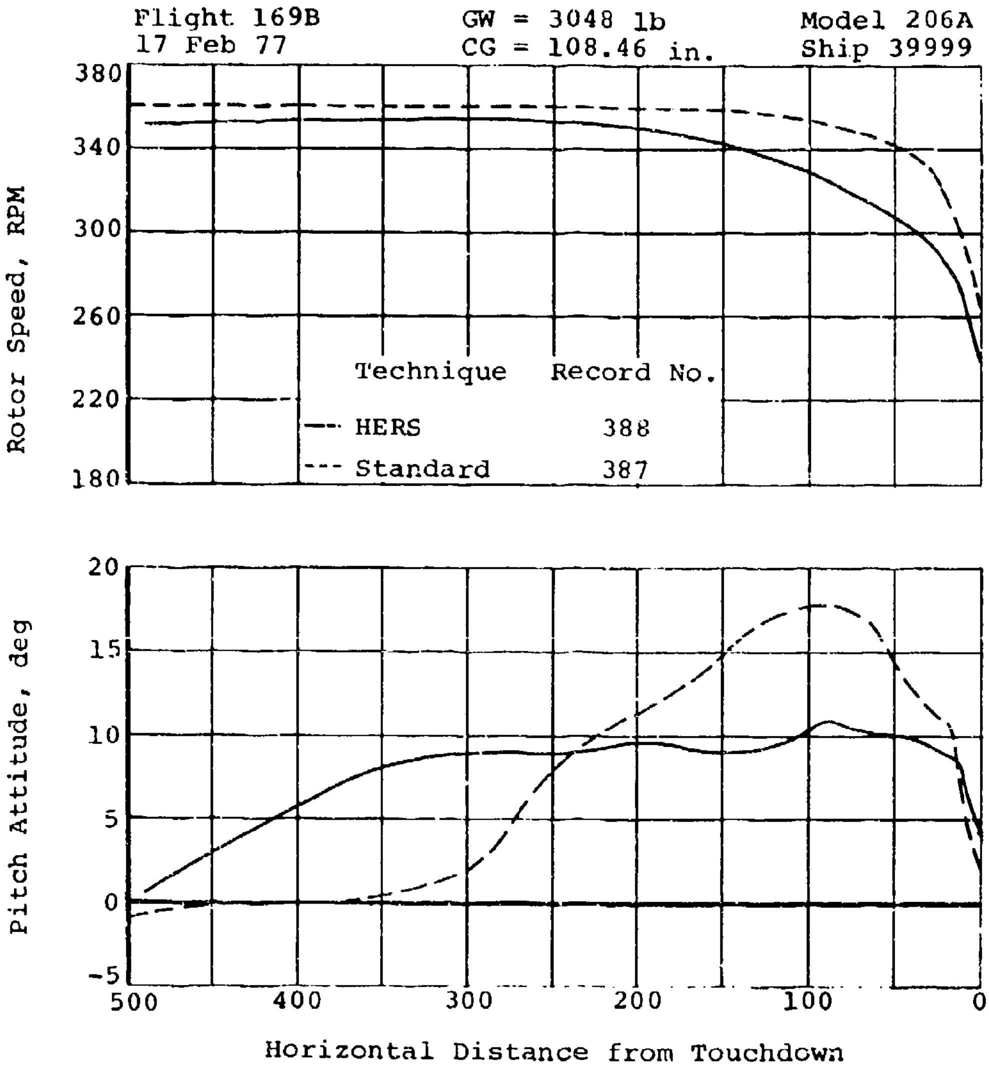


Figure 25. Comparison of autorotational flare tec ıniques.

nose-up attitude during the landing reduces pilot workload significantly and increases the likelihood of a safe landing with the HERS.

2.5.6.2 Touchdown Conditions

Other very important factors bearing on pilot workload are the horizontal and vertical velocities at touchdown. As the pilot approaches the landing, the most important factor is to ensure the vertical velocity at touchdown is as slow as possible to avoid a hard landing. Once he is sure that the vertical velocity is low enough, the horizontal velocity is considered. On a hard, paved runway the upper limit of acceptable horizontal velocity at touchdown is governed by his ability to control yaw with the tail rotor thrust during deceleration on the ground. This maximum acceptable speed is difficult to quantify, but is generally no more than about 20 knots. In general, the faster the acceptable touchdown horizontal velocity, the easier it is to perform the landing. If the helicopter is equipped with steerable wheeled landing gear, autorotational landing on smooth surfaces will be easier than with helicopters with skid gear.

When the landing surface is not hard and flat, the importance of a slow horizontal velocity at touchdown increases. On a rough landing surface, a fast touchdown speed may lead to a sudden deceleration and tipping of the helicopter or loss of control due to obstacle contact. In this situation, a touchdown speed at no more than a walking pace is desired.

The results of these contracted flight tests indicate a slow touchdown horizontal velocity is much easier to attain with high rotor inertia than with low inertia. The additional rotor energy allows the pilot more time to decelerate and thus reduce horizontal speed before touchdown.

Due to the ideal landing surface (the airport runway) used during this test, the pilot did not attempt a zero airspeed landing every time. Thus, the effect of increased inertia on reducing the pilot workload in attaining a slow horizontal touchdown speed is the pilot's qualitative opinion.

2.5.6.3 Wind Effects

One of the most important factors affecting the H-V test was that of wind. During the BHT contracted flights with the highest inertia rotor, a series of hovering throttle chops from 4 to 400 feet were performed in winds that were progressively slower than that of the previous flight. Initial test data were acquired with a headwind of 8 knots or less. The next flight gathered H-V data in headwinds of 5 knots or less. Finally, the tests were repeated with dead calm winds.

With such a progression of wind speeds, the pilot was able to accurately judge the workload change with wind speed. The dead calm wind condition was significantly more difficult than the case where winds were less than 5 knots. This wind effect reflected the pilot's ability to land at a low horizontal touchdown speed and was one of the most noticeable factors affecting the pilot workload during landing. However, even under conditions of dead calm winds, successful autorotational landings were performed with the HERS to demonstrate the elimination of the low-speed height-velocity restriction.

2.5.6.4 Collective Control Input Rates

One of the favorable pilot comments that describes an autorotational landing with the HERS is that "things happen so slowly." The use of rotor energy for additional power in landing requires the pilot to bleed rotor speed using a collective control input. The rate at which this collective input is applied varies as the rotor inertia is changed as indicated in Figure 26.

The rate of pilot collective stick control input during the last 5 feet of the autorotational descent was measured for all test conditions. The range of rates applied also varied with conditions of wind, horizontal speed, and pilot techniques used earlier in the maneuver. However, the trend of slower collective rates with increased inertia is apparent.

The BHT pilot stated that the rate used in the final collective pull was related directly to the vertical response of the helicopter to the control input. With the high inertia rotor, a relatively slow collective rate was required to maintain a slow descent in the last 5 feet of the landing. As rotor inertia decreased, a higher rate was required. Pilot sensing of vertical response to collective inputs coupled with the amount of collective he felt was left before rotor lift effectiveness was lost and the time required to touchdown were the factors governing collective input rates.

2.6 SUMMARY OF AUTOROTATIONAL LANDING TESTS

The benefits of increased rotor inertia in autorotational landing performance has been demonstrated in the initial and final phases of the simulated engine failure maneuver. The low-speed restricted area of the height-velocity diagram has been eliminated for the high inertia rotor and was reduced to a very small area with the mid-inertia configuration. Although little work was done addressing the high-speed region, tests indicated that HERS will improve the safety margins in this area also.

No adverse effects were noted in performing the autorotational tests. Control of rotor speed was easy and the techniques used for maximum performance were not difficult to apply.

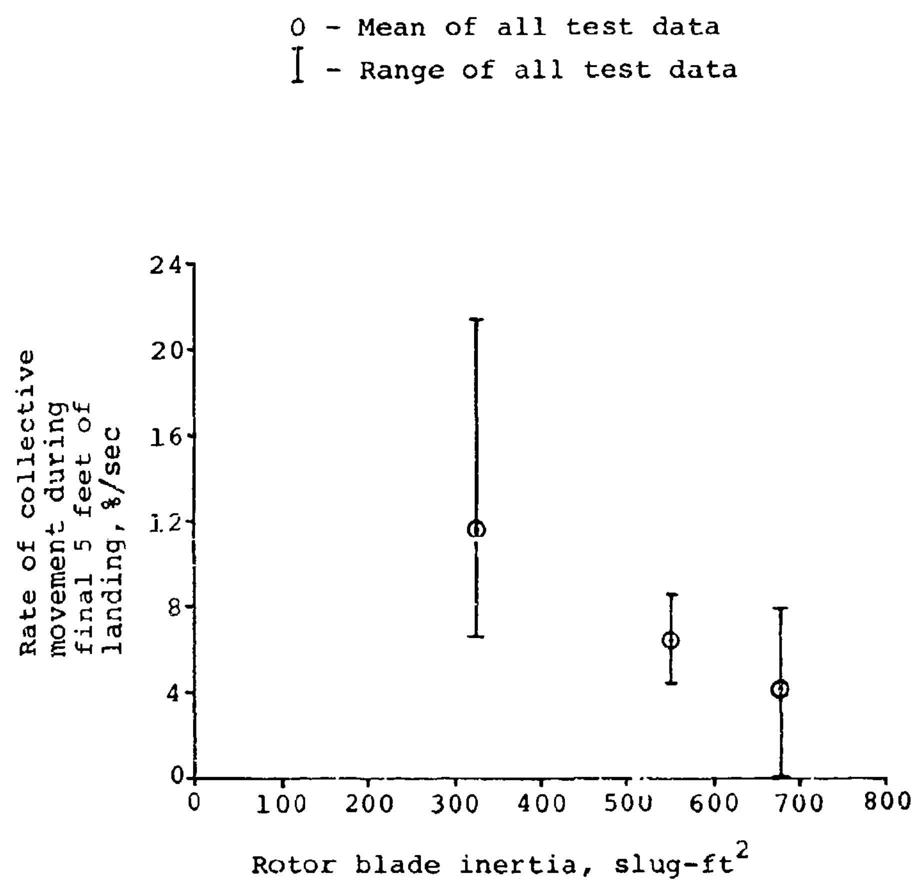


Figure 26. Effect of rotor blade inertia on collective rates in final autorotation landing phase.

3. PERFORMANCE

The use of the High Energy Rotor System significantly enhances the transient performance capabilities of the helicopter. This increased performance would be used to offset the payload loss due to increased rotor weight. The transient performance benefits are obtained by using some of the stored rotor energy power which is extracted from the rotor by bleeding rotor rpm. With the high inertia (I_b = 672 slug-ft^2) blades, a reduction in rotor speed from 354 to 330 rpm represents approximately 220 horsepower-seconds. The mid-inertia rotor provides 180 horsepower-seconds for the same rotor speed change.

Maneuver testing was performed to evaluate the use of transient rotor power and to verify the operational suitability and safety of the bleed rpm techniques.

3.1 PERFORMANCE ENHANCEMENT WITH HERS

The use of a bleed rpm technique to provide transient power for maneuvers is not uncommon. However, for standard inertia rotors, the rpm reduction required to develop excess power leaves little margin for error or for recovery from engine failures. As shown in Figure 27, higher rotor inertia allows more power to be extracted, even at low rotor rpm. The low inertia case indicates that high bleed rates and large changes in rotor speed are necessary for moderate power contributions.

Power extracted from the rotor while bleeding rpm is also an efficient means of producing power. If the additional power supplied by bleed rpm techniques were supplied instead by the engine, the transmission would require increased strength and efficiency losses would reduce the power to the rotor. The additional transmission power would in turn require higher antitorque thrust to be produced by the tail rotor, which would impact on pilot workload and provide another source of power loss. Since, for the HERS technique, the power is extracted in the rotor and used in the rotor, these transmission losses are not present. Since the additional power does not go through the transmission, a potential weight savings in the drive train may help offset the increased rotor weight. This weight savings will be realized only if the helicopter is designed to an additional installed engine power equal to the transient power that could be extracted for the rotor.

3.2 PERFORMANCE TEST RESULTS WITH HERS

Specific tests that were performed to demonstrate the bleed rpm technique are presented in this section. The maneuvers

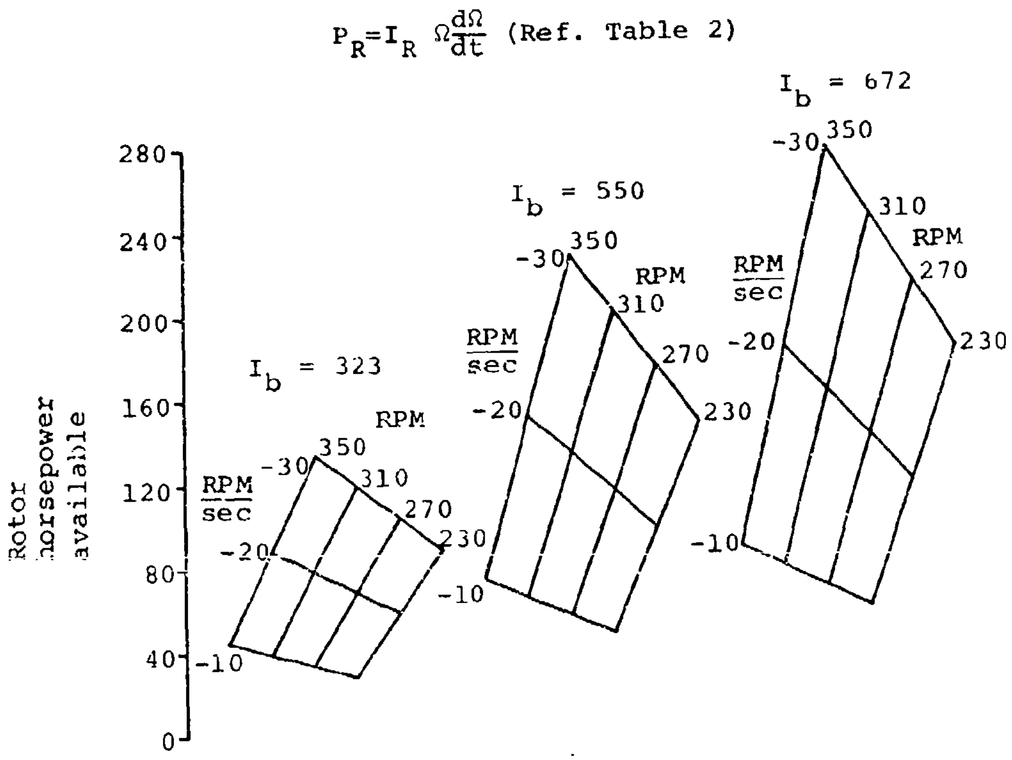


Figure 27. Rotor power variation with inertia.

of interest are those normally required to perform the NOE mission such as takeoffs, unmasking maneuvers such as bob-ups, pop-ups, and translational accelerations. Each maneuver was performed using the bleed rpm technique and then repeated using constant rotor rpm. This provides a comparison of the HERS technique performance to that of a standard inertia rotor. While not directly comparable with the standard OH-58A since rotor airfoil and solidity are modified, the results are indicative of the increased performance potential of the HERS.

3.2.1 Improved Takeoff Performance with HERS

To demonstrate the improvement in takeoff performance with the HERS, the engine governor was modified as described in Section 1.1.4, to restrict the power available to just that required for a 4-foot hover.

A time history of a typical takeoff using the HERS bleed rpm technique is shown in Figure 28. The pilot opinion of employing this technique was favorable. As can be seen in this figure, no unusual control motions were required in performing the takeoff and the rpm recovery after takeoff was considered easy and natural.

A comparison of the takeoff performance using the bleed rpm technique with that of a constant rpm takeoff is presented in Figure 29. As shown here, the pilot demanded this power at a rate of 30 horsepower for 2.5 seconds and lower power for about 6 seconds. The significant improvement of takeoff performance with a small additional amount of power is evident. The Army evaluation pilot expressed surprise in the change in flight path possible using this technique.

The general pilot opinion of employing this technique was favorable. However, the standard use of this technique during takeoffs raises the question of an aborted takeoff in the event of an engine failure.

The most critical condition would be that of a power failure during a high vertical rate of climb and low rotor rpm. Throttle chops were made in this condition to demonstrate that successful autorotational landings could be made. These throttle chops were demonstrated at a gross weight of 3050 pounds with rates of climb between 540 to 900 feet per minute and altitudes of 31 to 150 feet at the time of the throttle chop. Figure 30 presents a time history of an aborted pop-up maneuver using bleed rpm where the throttle chop occurred at a main rotor rpm of 330 and a rate of climb of 900 feet per minute. A successful landing was made 10 seconds after the throttle chop. The peak altitude attained was 78 feet and the touchdown rate of sink was reduced to almost zero. A flight profile of this maneuver is presented in Figure 31.

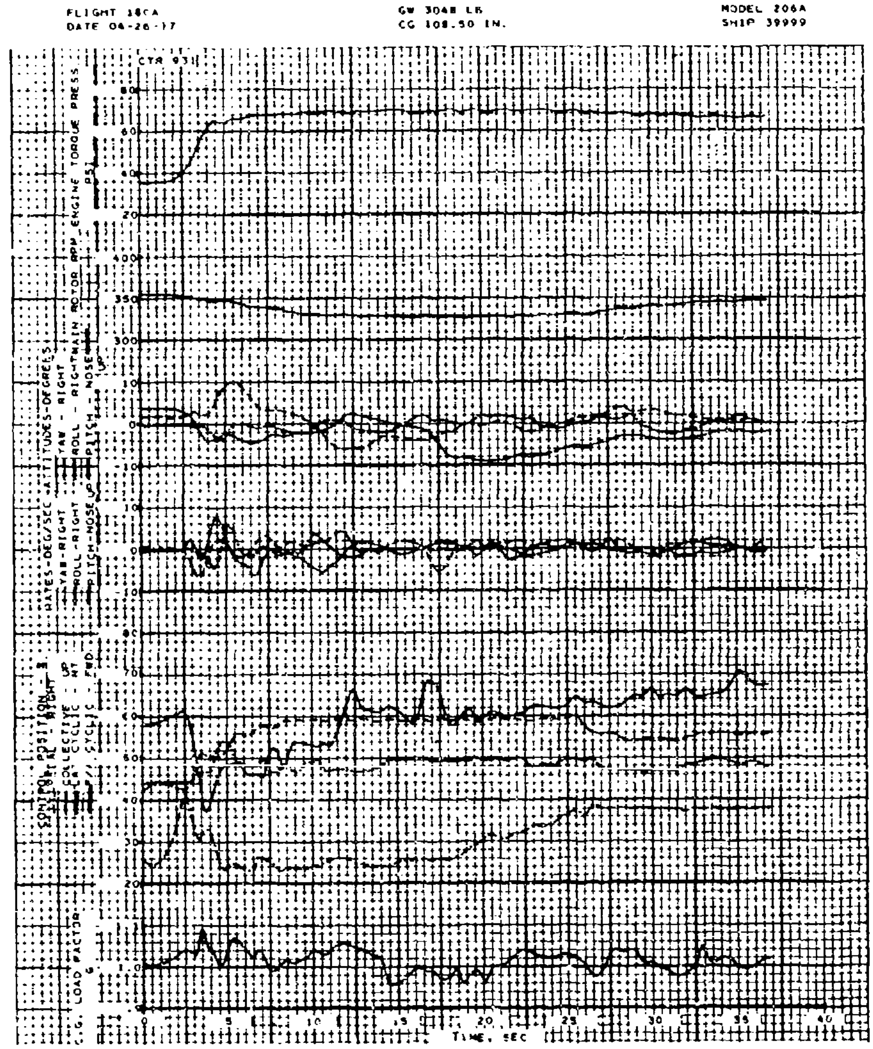
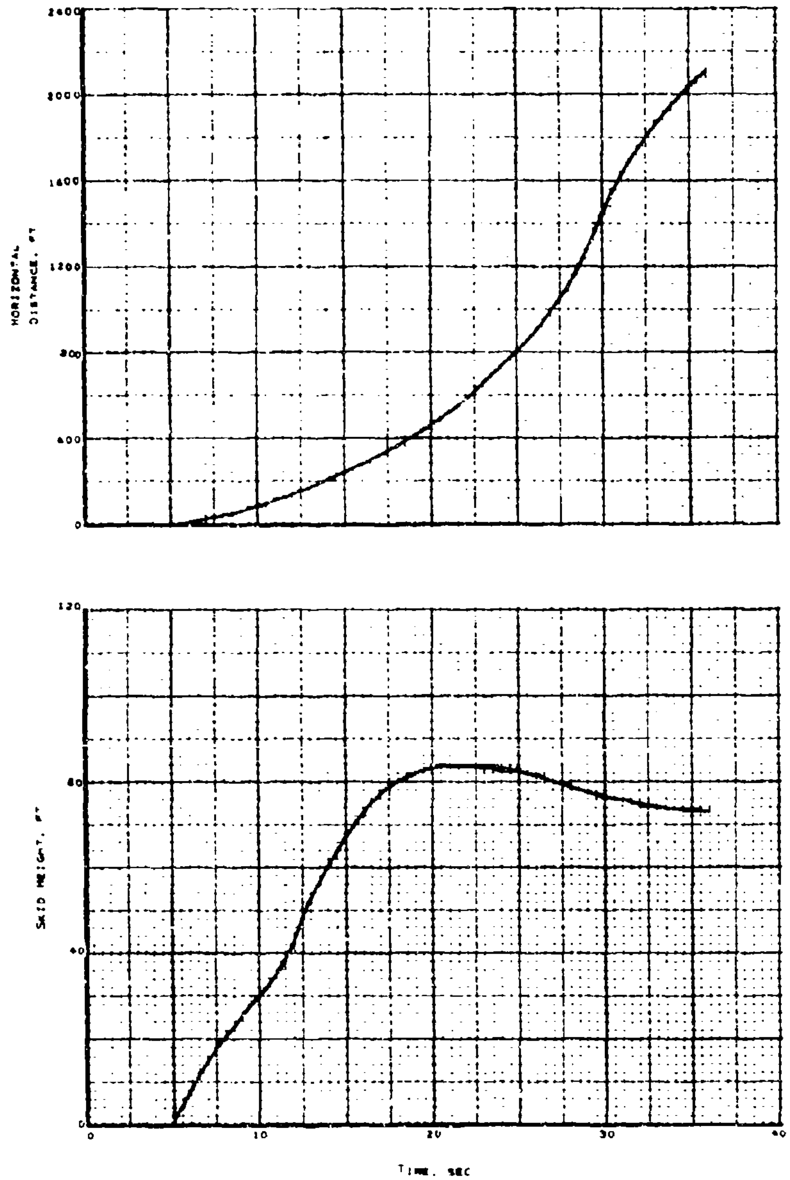


Figure 28. Takeoff using bleed RPM technique - high inertia rotor.



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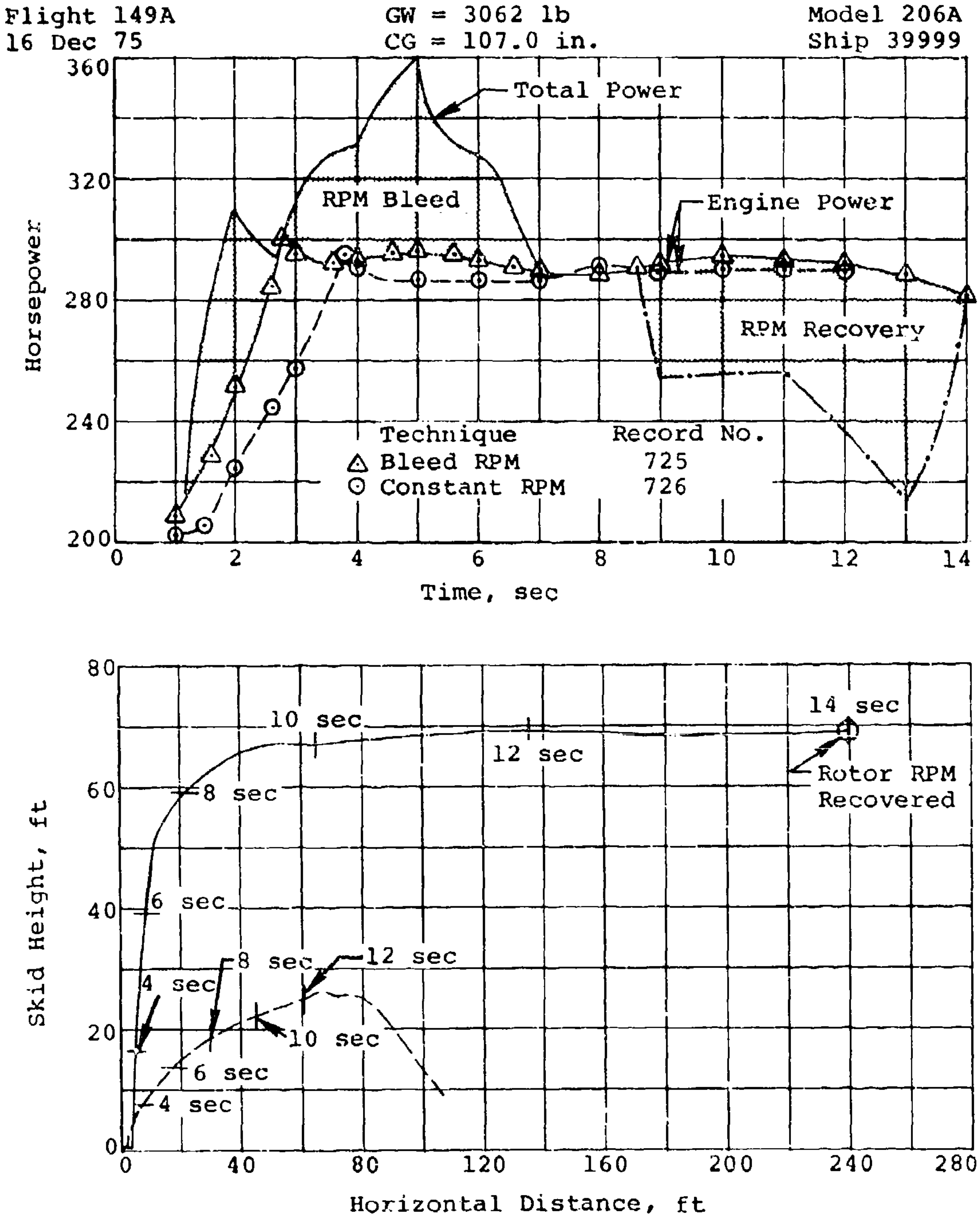


Figure 29. Comparison of bleed RPM takeoff performance with constant RPM technique.

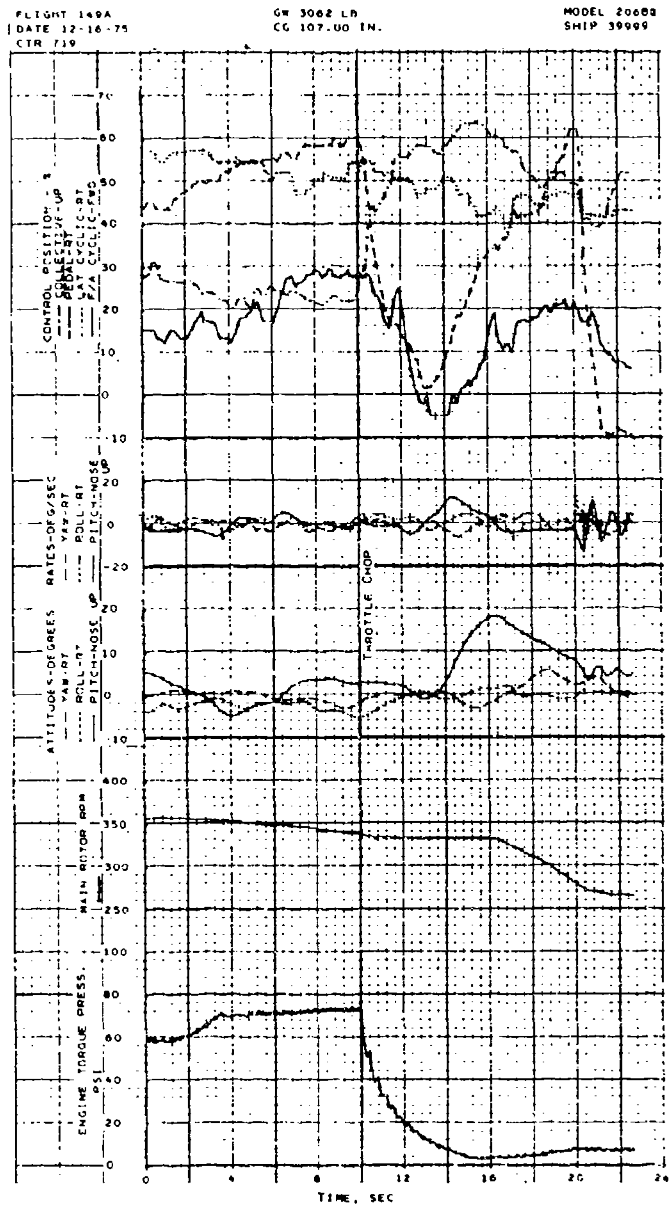


Figure 30. Throttle chop at low rotor RPM and 900 fpm rate-of-climb - high inertia rotor.

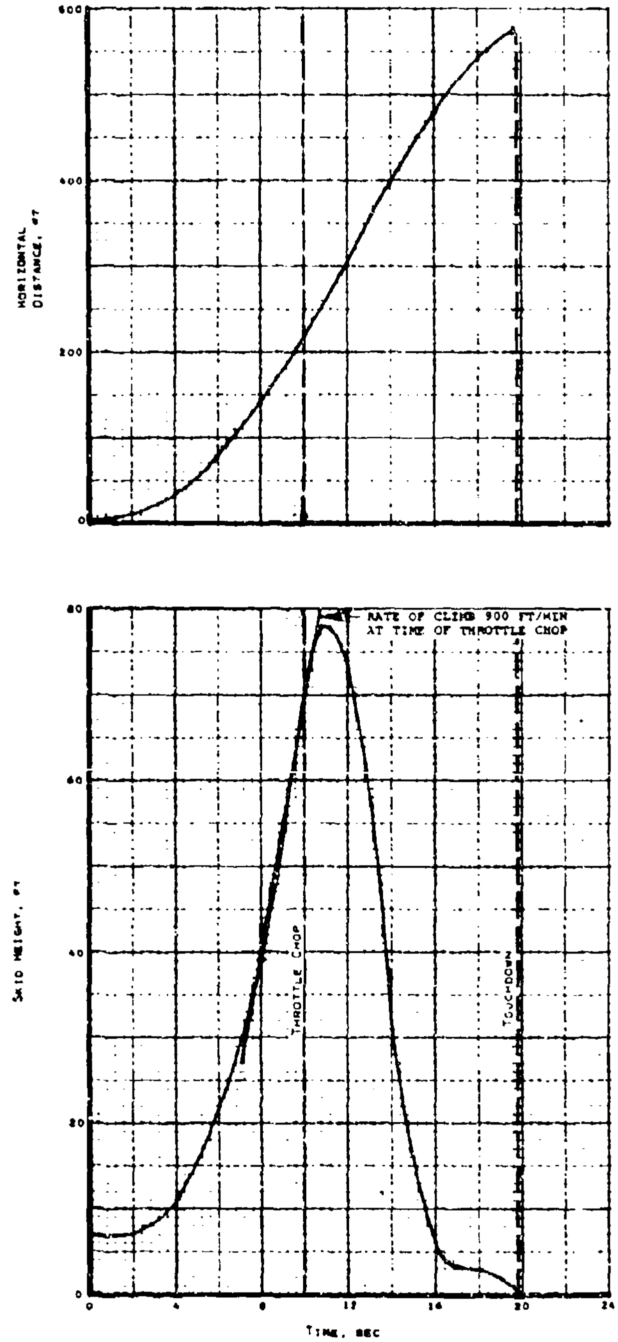


Figure 30. Concluded.

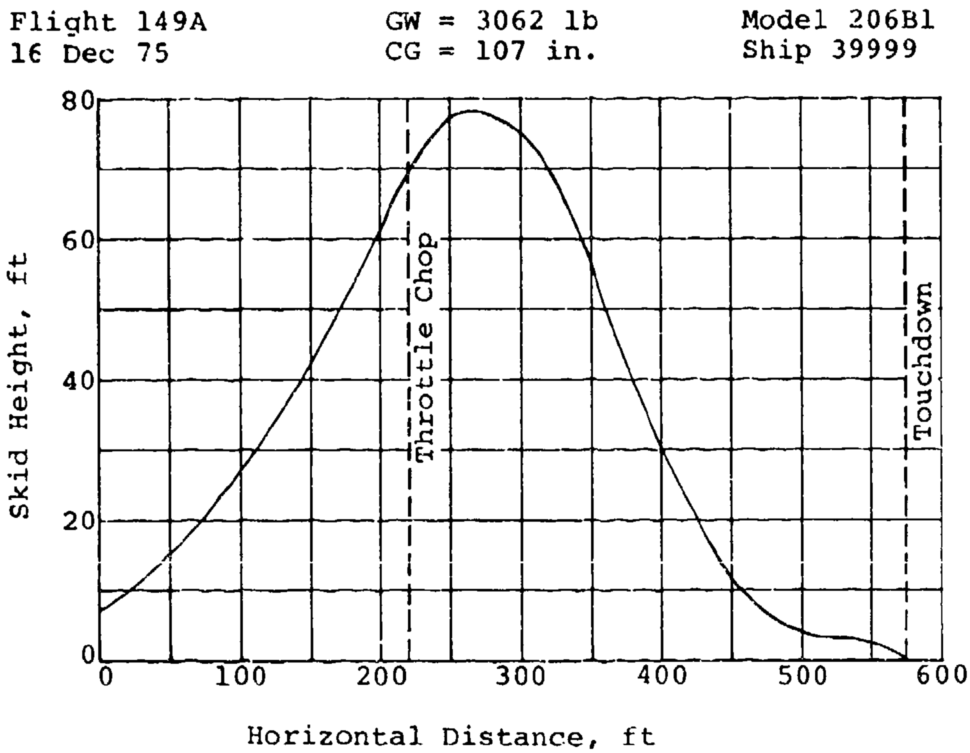


Figure 31. Flight profile of throttle chop at low rotor RPM and 900 fpm rate-of-climb - high inertia rotor.

The pilot's workload during an aborted takeoff was increased over that required for a throttle chop in level flight at the same airspeed and altitude, but was considered acceptable for an average pilot's capabilities.

In general, the workload is considerably reduced since the pilot does not have to fear exceeding engine and/or transmission redlines. Only main rotor rpm has to be watched.

3.2.2 NOE Maneuverability

The primary objective of these tests was to determine the effect of using transient rotor power to provide short-term performance improvements during NOE maneuvers. This application of HERS bleed rpm technique is advantageous since it is the nature of NOE missions to require intermittent high power levels for short periods of time. The maneuvers selected for this evaluation were pop-ups and bob-ups, forward acceleration, and lateral accelerations.

3.2.2.1 Unmasking Maneuvers

The basic purpose of the pop-up or bob-up maneuver is to reach a specified altitude as quickly as possible to allow a survey of surrounding terrain. The maneuvers are similar except that the pop-up is started from a hover and the bob-up from a forward flight condition.

For these tests, the engine governor was set to restrict engine power so that the helicopter could not hover out of ground effect (OGE). This was done to simulate performance at high gross weight and density altitude conditions. Figure 32 presents a time history of a pop-up using the bleed rpm technique with the HERS. The control motions required for this maneuver were smooth and easy to perform.

A comparison of the bleed rpm performance with that of the same helicopter using constant rpm is presented in Figure 33. Using engine power alone (constant rpm), the helicopter was able to reach a maximum altitude of 18 feet with an average rate of climb of 103 ft/min. When the rotor rpm was bled down at a rate of 1.6 rpm/sec to 327 rpm, the helicopter attained a peak altitude of 56 feet, 3 times the altitude attained using constant rpm. The average rate-of-climb using bleed rpm was 202 ft/min, double that of the constant rpm case.

The helicopter could not sustain hovering flight at the peak altitude in either case as the engine power was insufficient for an OGE hover. However, the pop-up was possible and the peak altitude was sustained for a short period of time consistent with an unmasking maneuver necessary in the NOE environment.

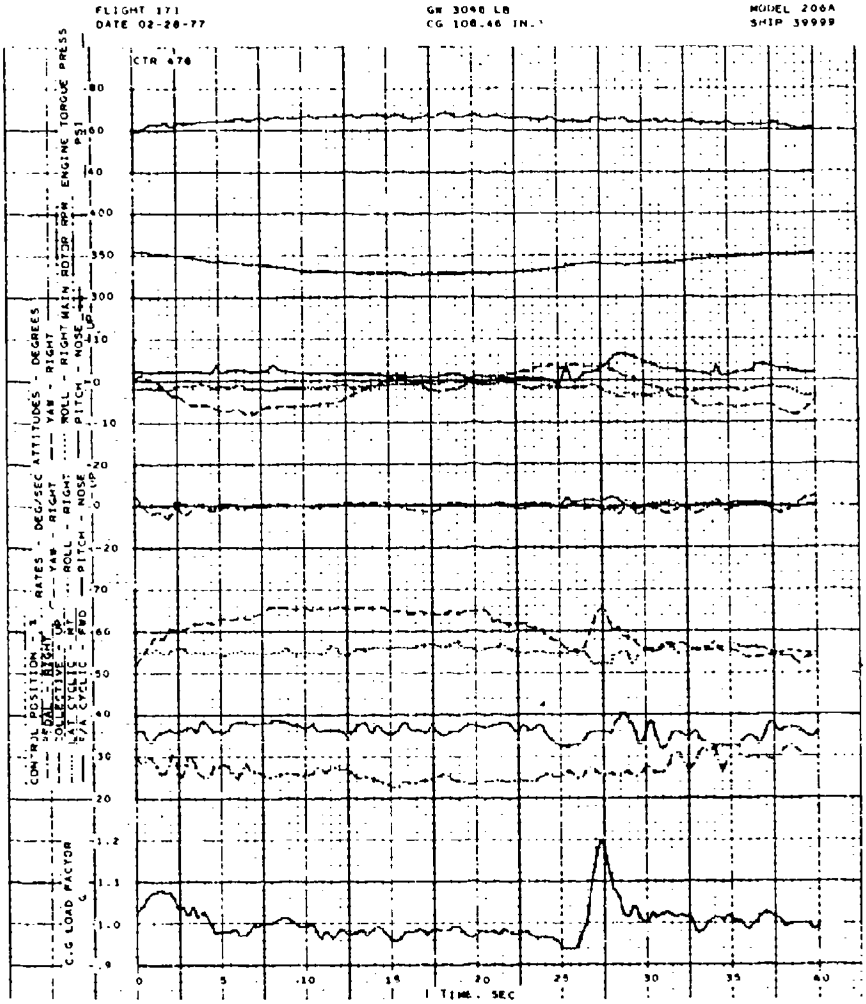


Figure 32. Unmasking maneuver using bleed RPM technique - high inertia rotor.

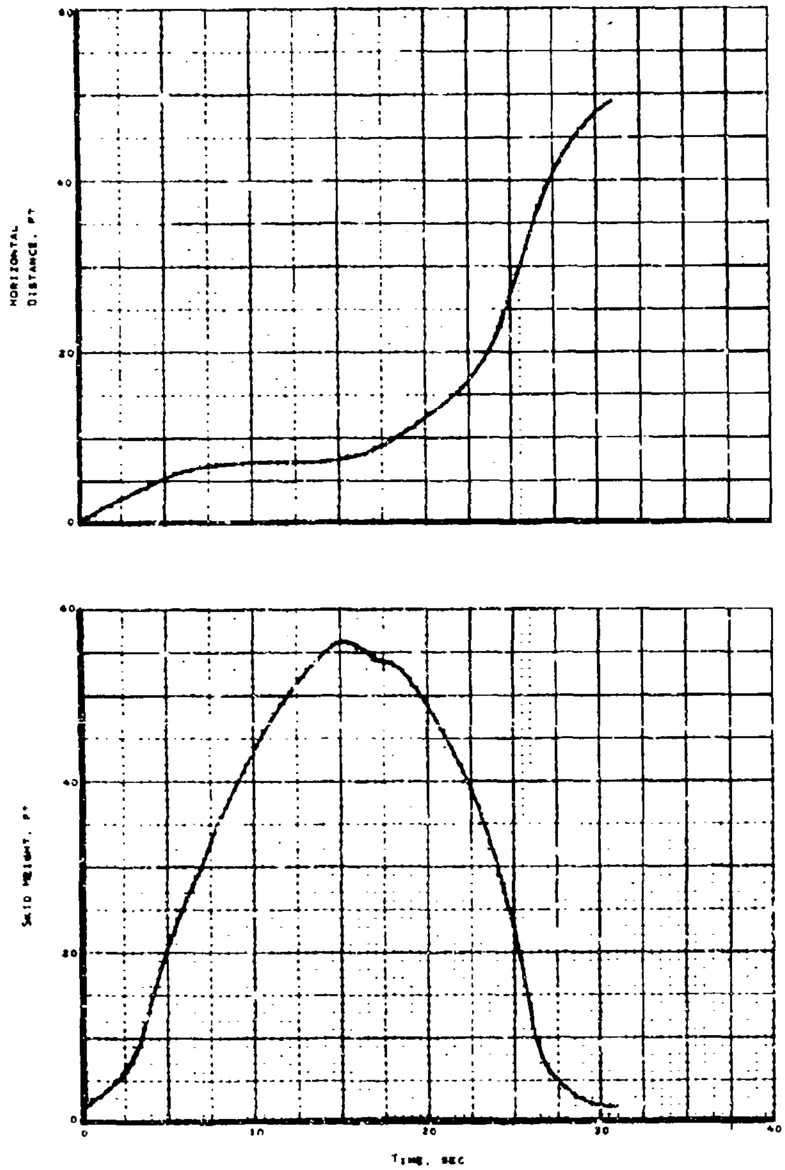


Figure 32. Concluded.

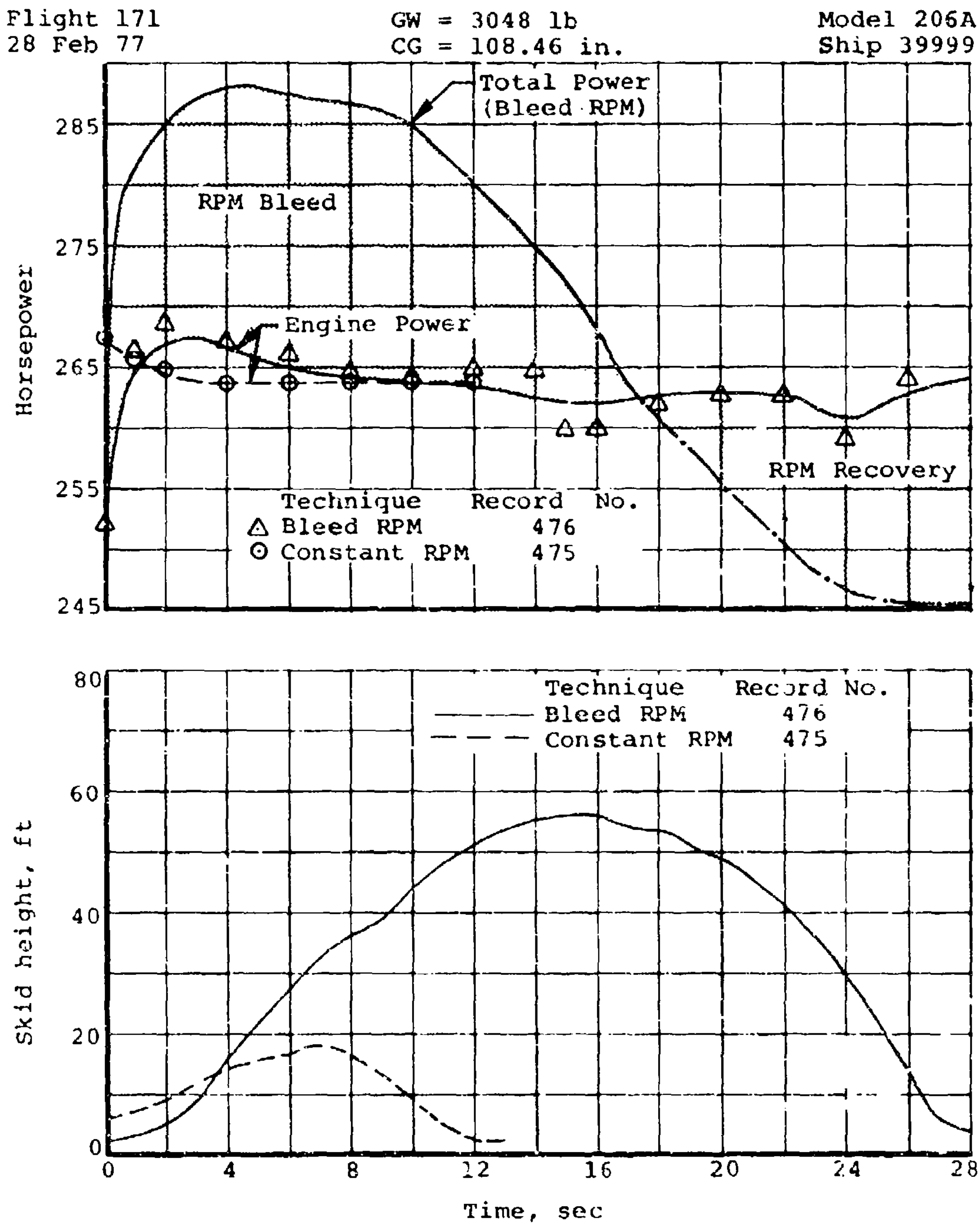


Figure 33. Comparison of bleed RPM and constant RPM unmasking maneuvers.

3.2.2.2 Longitudinal Accelerations

Longitudinal acceleration maneuvers were also evaluated during this program. The objective of these maneuvers was to start from hover or some low airspeed, and while maintaining constant altitude, accelerate the helicopter to a higher airspeed. Longitudinal accelerations were flown from hover up to speeds that ranged from 35 to 75 knots, and from level flight at 40 knots up to 80 knots forward speed. These maneuvers simulate a dash across an open field.

Figure 34 compares a bleed and constant rpm acceleration from hover. The longitudinal acceleration using bleed rpm provided between 20 and 30 extra horsepower for approximately 9 seconds over that supplied by the engine alone. By bleeding the rotor speed to 332 rpm at an average rate of 2.5 rpm/sec, the pilot was able to generate an initial longitudinal acceleration of 6 ft/sec^2, which was approximately 4 times greater than the acceleration produced by engine power alone. This increased acceleration was maintained for the entire acceleration portion of the maneuver, allowing the helicopter to attain 27 knots in 4.5 seconds with bleed rpm compared to 12 seconds required at constant rotor rpm.

A time history of a typical bleed rpm longitudinal acceleration is shown in Figure 35. No directional control problems were encountered during the maneuver, and pilots indicated that the bleed rpm technique for longitudinal accelerations was natural and did not increase workload to any significant extent.

Several accelerations were also performed starting from approximately 40 knots up to about 80 knots to provide an indication of the helicopter dash capability. Some difficulty was encountered in comparing bleed and no-bleed maneuvers in that entry conditions were not always matched. An example of this maneuver is presented in Figure 36. Note that the oscillograph was shut off before the end of the bleed rpm record resulting in no-power data for the last 2 seconds of the maneuver.

The difficulty in comparing maneuvers of this nature is apparent when comparing the engine power time history for these maneuvers. In order to maintain rotor speed, the pilot was unable to apply collective as rapidly as could be done with the bleed technique. Toward the end of the maneuver, the engine power output was the same for both techniques.

The effect of bleed rpm is apparent when comparing the time required to accelerate to 80 knots. The bleed rpm maneuver required 15.5 seconds to accelerate from 39 knots to 80 knots, for an average acceleration of 1.89 kn/sec.

Flight 171
28 Feb 77

GW = 3048 lb
CG = 108.46 in.

Model 206A
Ship 39999

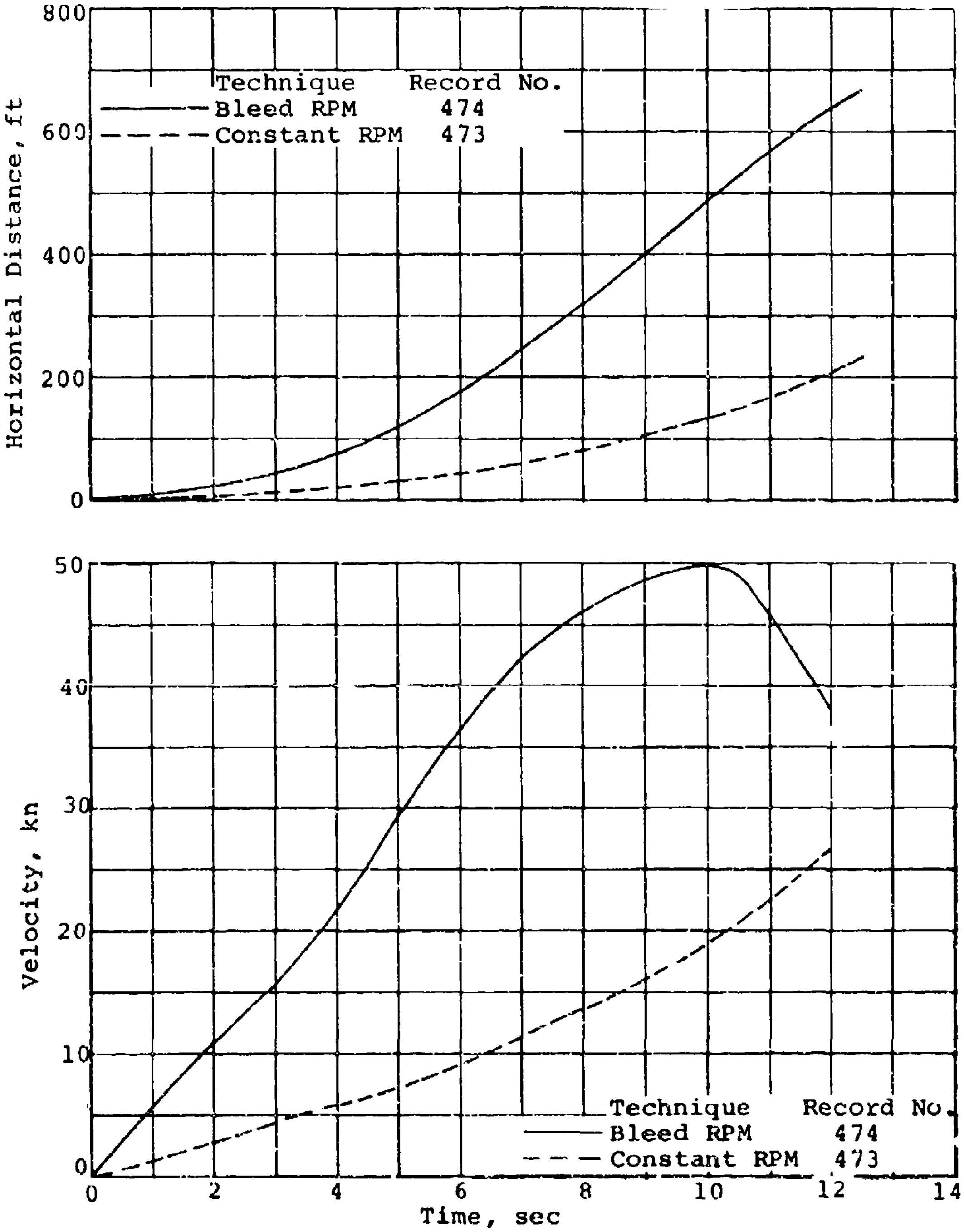


Figure 34. Comparison of bleed RPM and constant RPM longitudinal acceleration

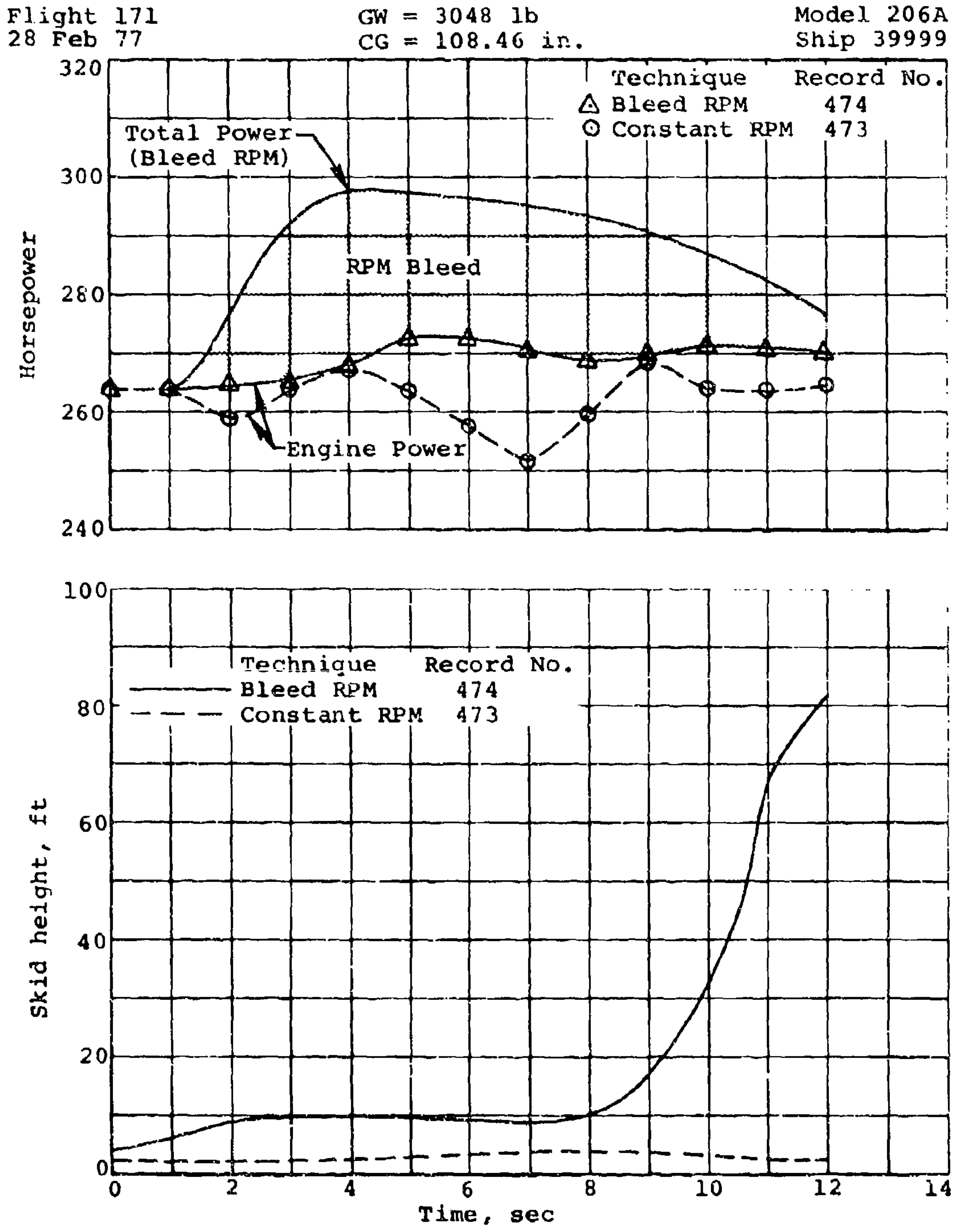


Figure 34. Concluded.

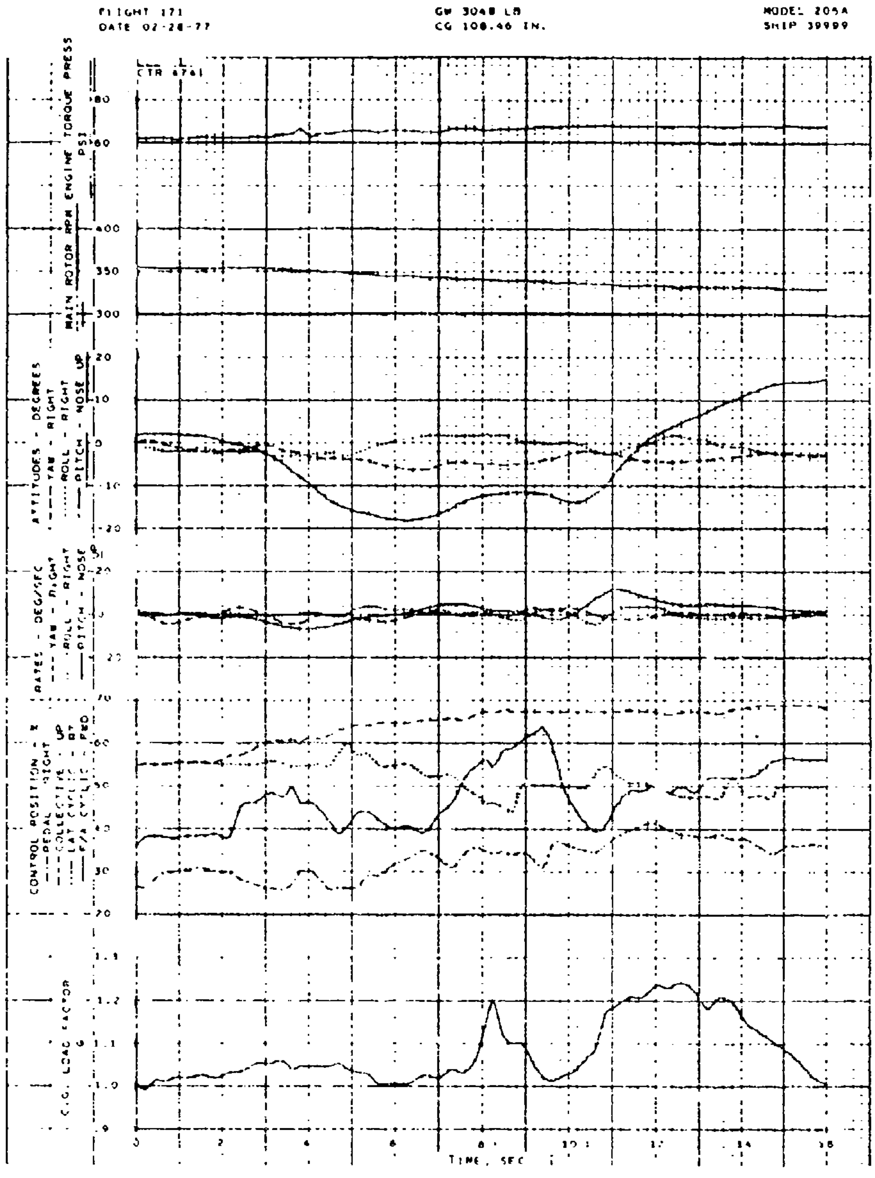
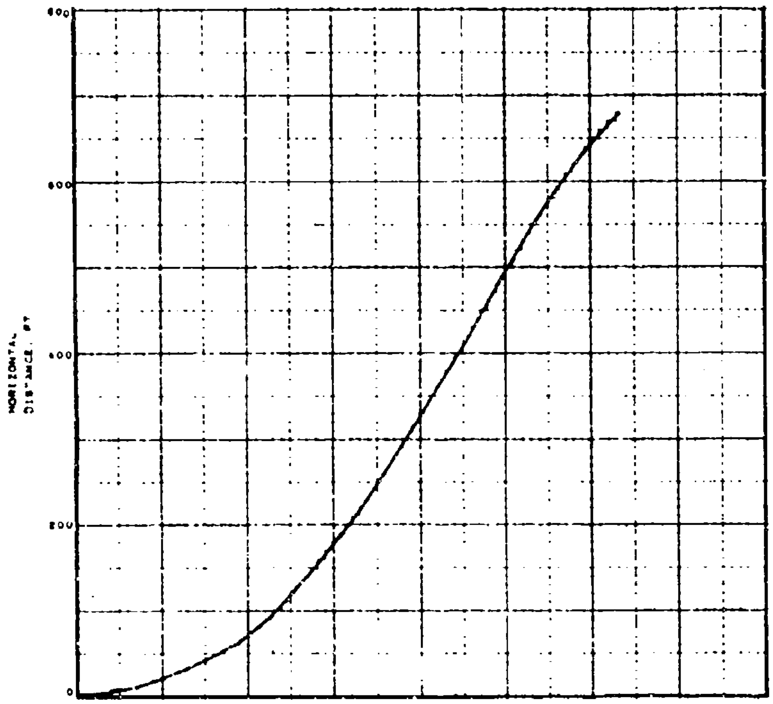
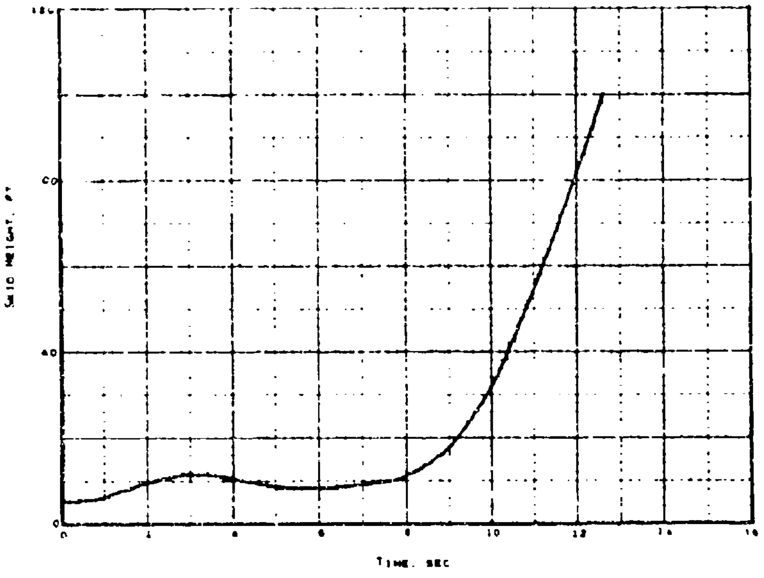


Figure 35. Longitudinal acceleration using bleed RPM technique - high inertia rotor.





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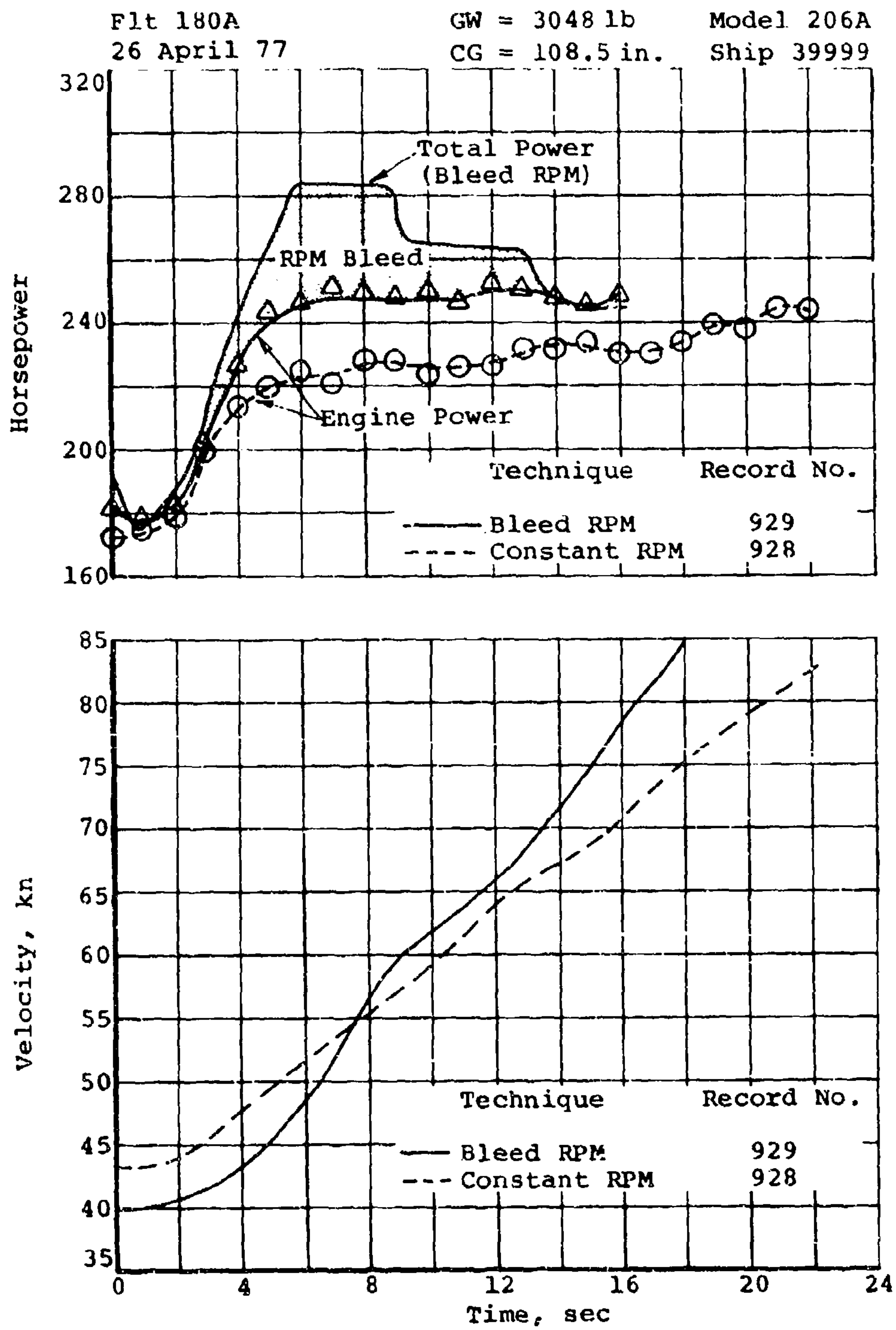

Figure 36. Comparison of bleed RPM and constant RPM dash capability - high inertia rotor.

The constant rpm technique resulted in a relatively constant acceleration while the acceleration during the bleed rpm case varied with bleed rate. This is reflected in the variation of total horsepower during the maneuver.

3.2.2.3 Lateral Accelerations

A limited investigation of the use of bleed rpm during lateral acceleration maneuver was also conducted. A comparison of a bleed and constant rpm left lateral acceleration is shown in Figure 37. Both bleed and constant rpm maneuvers in these tests were not flown as maximum performance maneuvers, but rather to explore the effect of increasing the aircraft bank angle by the use of excess horsepower obtained from bleed rpm. The average bleed rate of 2.5 rpm/second allowed an increased bank angle 9 degrees greater than that obtained in the constant rpm acceleration. The increased bank angle is translated as shown in Figure 37 into increased acceleration, allowing the pilot to reach 25 knots in one-half the time that was required for the constant rpm case. In all of these maneuvers, bleed rpm was recovered by reducing bank angle and executing a pedal turn into the wind. Figure 38 shows a typical bleed rpm lateral acceleration maneuver.

3.3 SUMMARY OF PERFORMANCE TESTS

The employment of bleed rpm techniques with the high inertia rotor was demonstrated to improve the transient performance and maneuverability of the test helicopter.

The extraction of relatively low horsepower levels over several seconds provided an additional power source allowing increased takeoff performance at equal gross weight and engine power available conditions. The pilot techniques required for this increased performance were natural and easy to perform. Rotor speed recovery after a bleed rpm takeoff was easily accomplished and simulated engine failures at critical times during the takeoff were successfully demonstrated.

The use of bleed rpm techniques also provided increased maneuverability in transient maneuvers typical of NOE operations. The extraction of rotor power provided an additional power margin to be used for increased acceleration in both the horizontal and vertical directions. Since the additional power is supplied by the rotor, with a properly designed engine governor the pilot would not have to monitor the engine and transmission limits during the maneuver. Rotor speed is the only parameter to monitor, thus a reduction in pilot workload can be realized with the HERS.

Flight 171 GW = 3048 lb Model 206A
28 Feb 77 CG = 108.46 in. Ship 39999

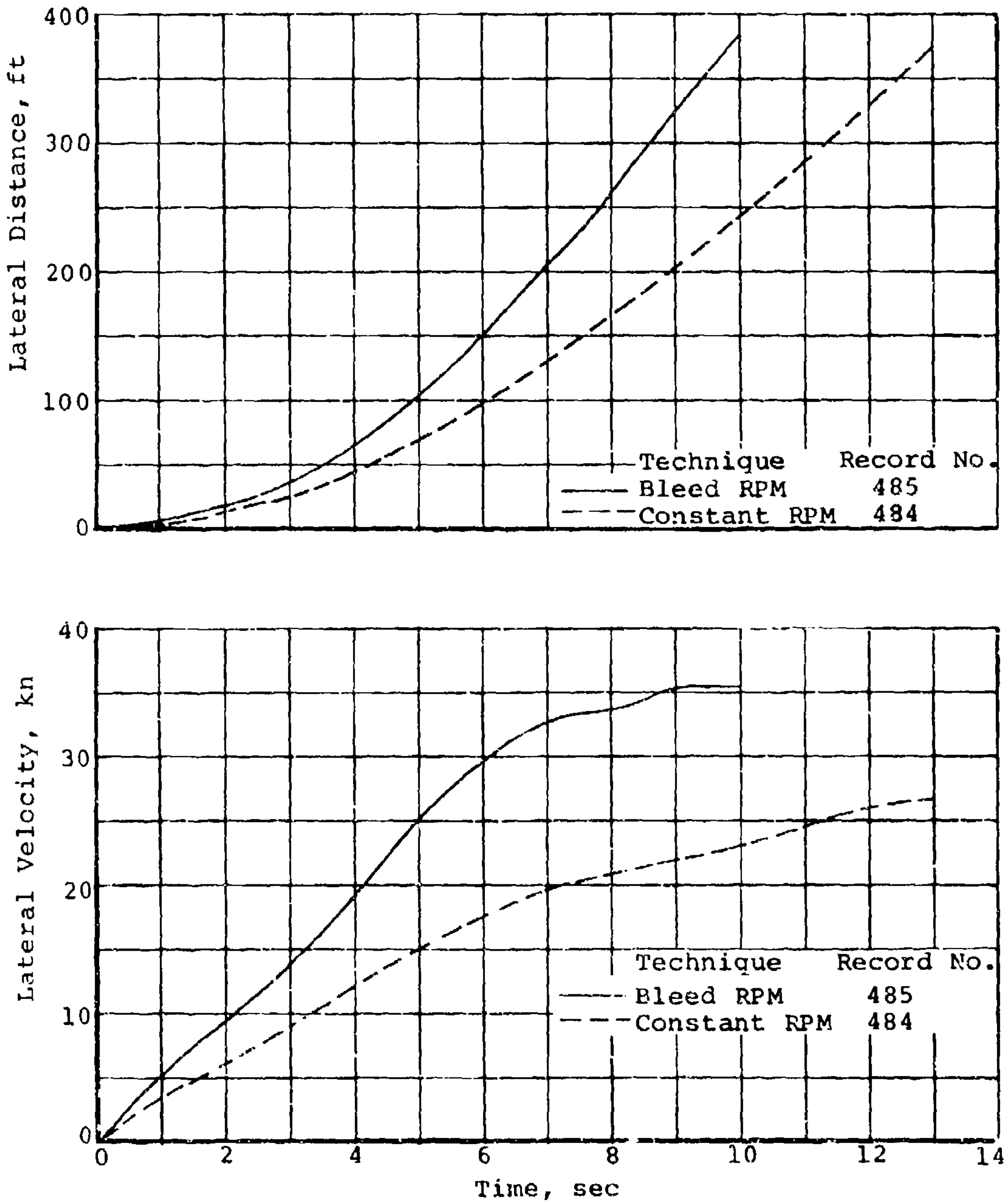

Figure 37. Comparison of bleed RPM and constant RPM lateral acceleration - high inertia rotor.

Flight 171
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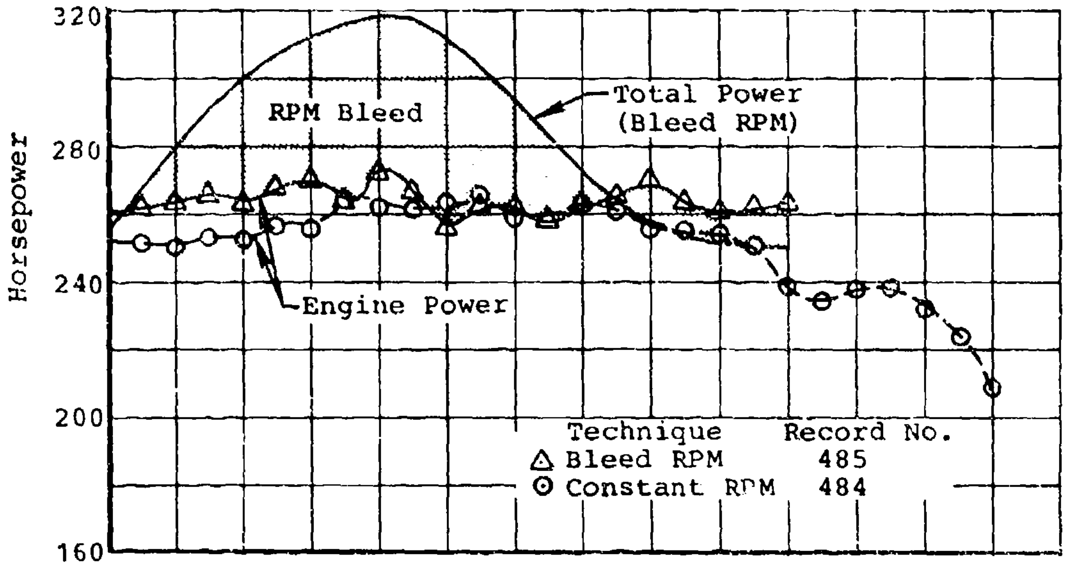
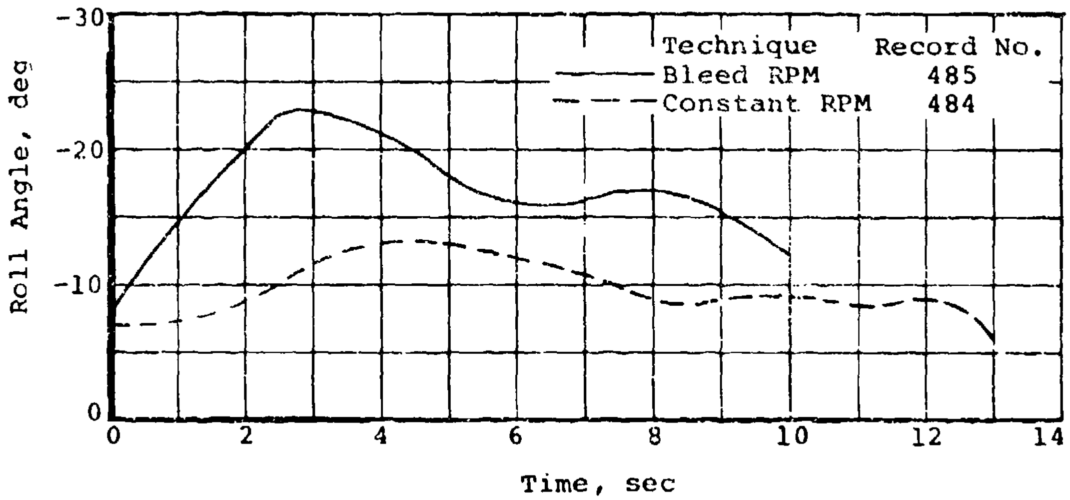
GW = 3048 lb
CG = 108.46 in.

Model 206A
Ship 39999





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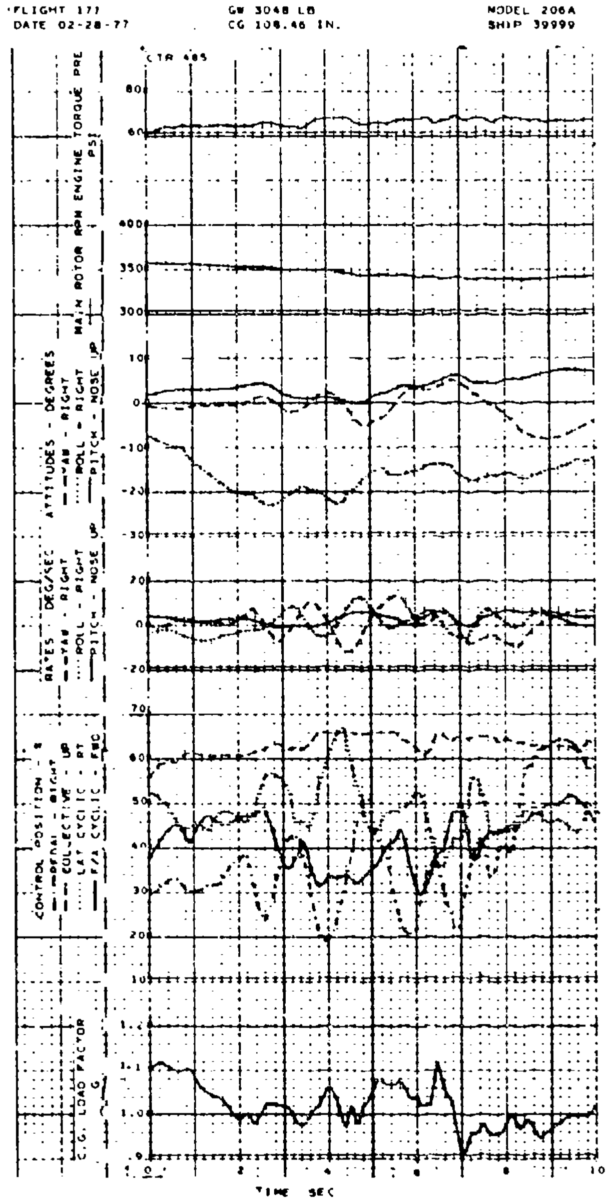

Figure 38. Lateral acceleration using bleed RPM technique - high inertia rotor.

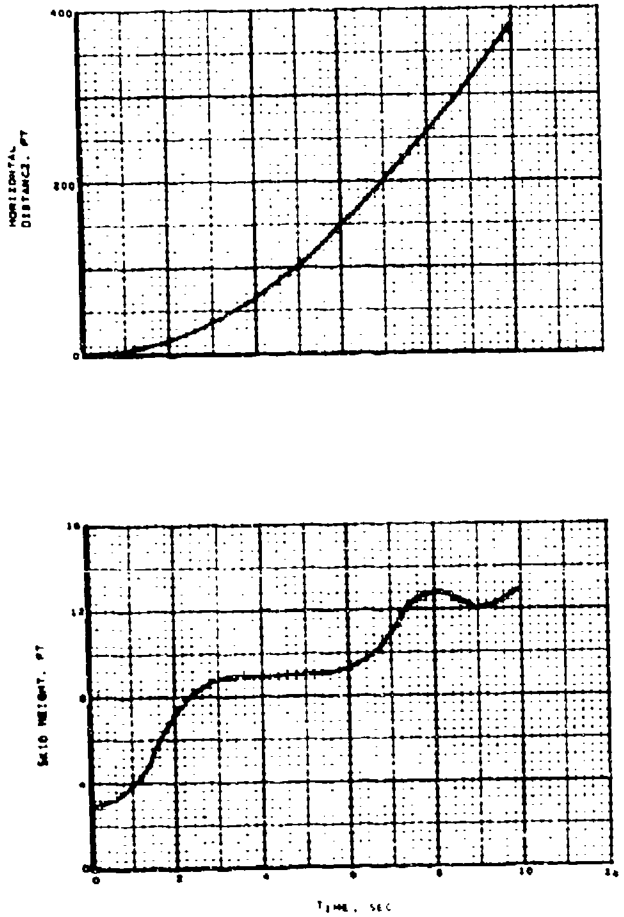

Figure 38. Concluded.

4. HANDLING QUALITIES

The effect of rotor inertia (or Lock number) on the handling qualities of the helicopter was evaluated during this test program. The testing compared the response of the helicopter to control steps and pulse inputs for all three Lock numbers along with handling qualities in pullups, pushovers, and turns.

4.1 EFFECT OF ROTOR INERTIA ON HANDLING QUALITIES

The major effect of rotor inertia on handling qualities may be theoretically examined through the flapping equation for centrally hinged blades as derived, for example, in Reference 7.

This equation may be written as

$$\ddot{\beta} + \frac{\gamma\Omega}{8}\dot{\beta} + \Omega^2\beta = f(t) \tag{4}$$

where

$$\gamma = \frac{\rho a c R^4}{I_b} \quad \text{(Lock number)}$$

and

$$f(t) = \text{forcing function}$$

If the forcing function is of the form

$$f(t) = 2q\Omega\sin\Omega t$$

as would result from a pure angular rotation of the hub at a rate q, the rotor flapping may be expressed as

$$\Delta\beta = -\frac{16}{\gamma\Omega}q \tag{5}$$

---

[7]Bramwell, A.R.S., HELICOPTER DYNAMICS, John Wiley & Sons, New York, 1976.

$\Delta\beta$ can be thought of as the lag in rotor flapping due to the angular rate q imposed on the rotor. This lag in flapping is representative of a rotor damping moment which resists the angular rate q. This phenomenon is discussed in Reference 7.

As Equation 5 indicates, the lag in flapping (and thus rotor damping) is inversely proportional to Lock number, or directly proportional to rotor inertia. This change in damping with Lock number was measured during the flight test program by examining the helicopter response to step control inputs. Figure 39 indicates the variation of helicopter roll damping with rotor inertia for the OH-58A test vehicle as measured during the contracted flights. The additional rotor damping improves the basic stability of the helicopter but, as was discovered in initial tests of the HER system, makes the helicopter response to control inputs more sluggish. In order to improve this response, control system modifications were required.

4.2 EFFECT OF CONTROL SYSTEM MODIFICATIONS

The initial modification to the OH-58 control system was to incorporate a quickener to increase control sensitivity in both pitch and roll. Components from a BHT Model 206 (Jet-Ranger) Stability and Control Augmentation System (SCAS) were utilized to provide a feedforward loop as indicated in Figure 40. Pilot control motions were sensed and a quickening signal was supplied to an electrohydraulic actuator in the control system. The quickener initially magnified the pilot control command and then washed out the quickening signal as shown in Figure 41. A typical time history (Figure 42) of a lateral step input with and without the control quickener illustrates the effect of the quickener.

While the control quickener provides increased control sensitivity, the higher damping due to blade inertia results in a lower fuselage angular rate per inch of pilot control input than with the standard inertia rotor. In order to provide the most favorable control characteristics, a pilot stick-to-swashplate gearing change was also made as presented in Table 3. This gearing change reduced the amount of pilot cyclic stick travel to produce the full swashplate travel without changing the total swashplate range.

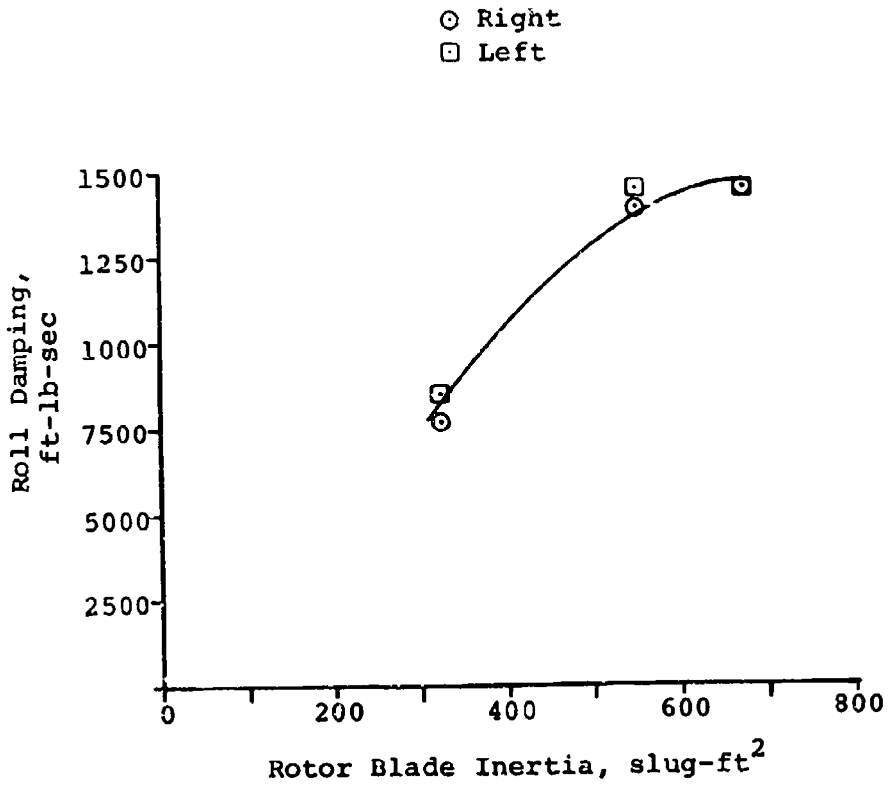

Figure 39. Effect of rotor inertia on roll damping in hover.

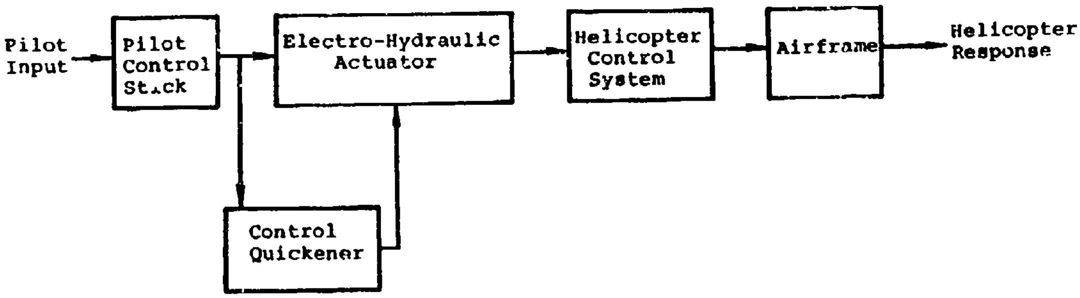

Figure 40. Control quickener block diagram.

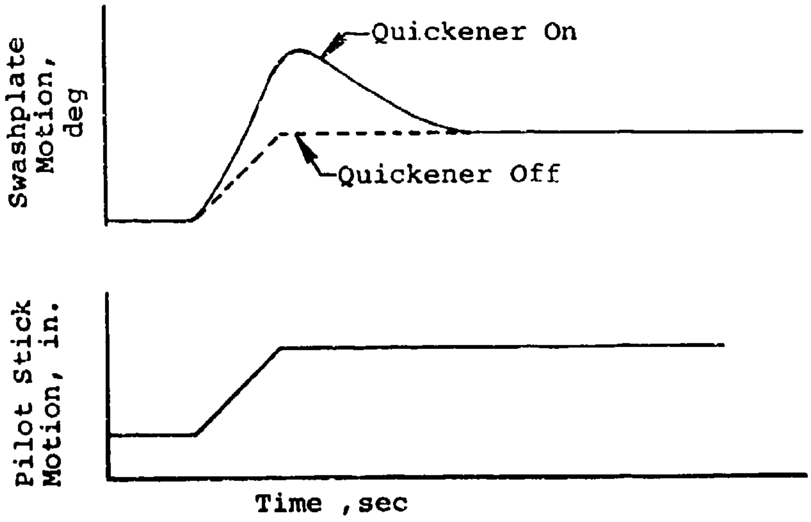

Figure 41. Rotor control moment as affected by control quickener.

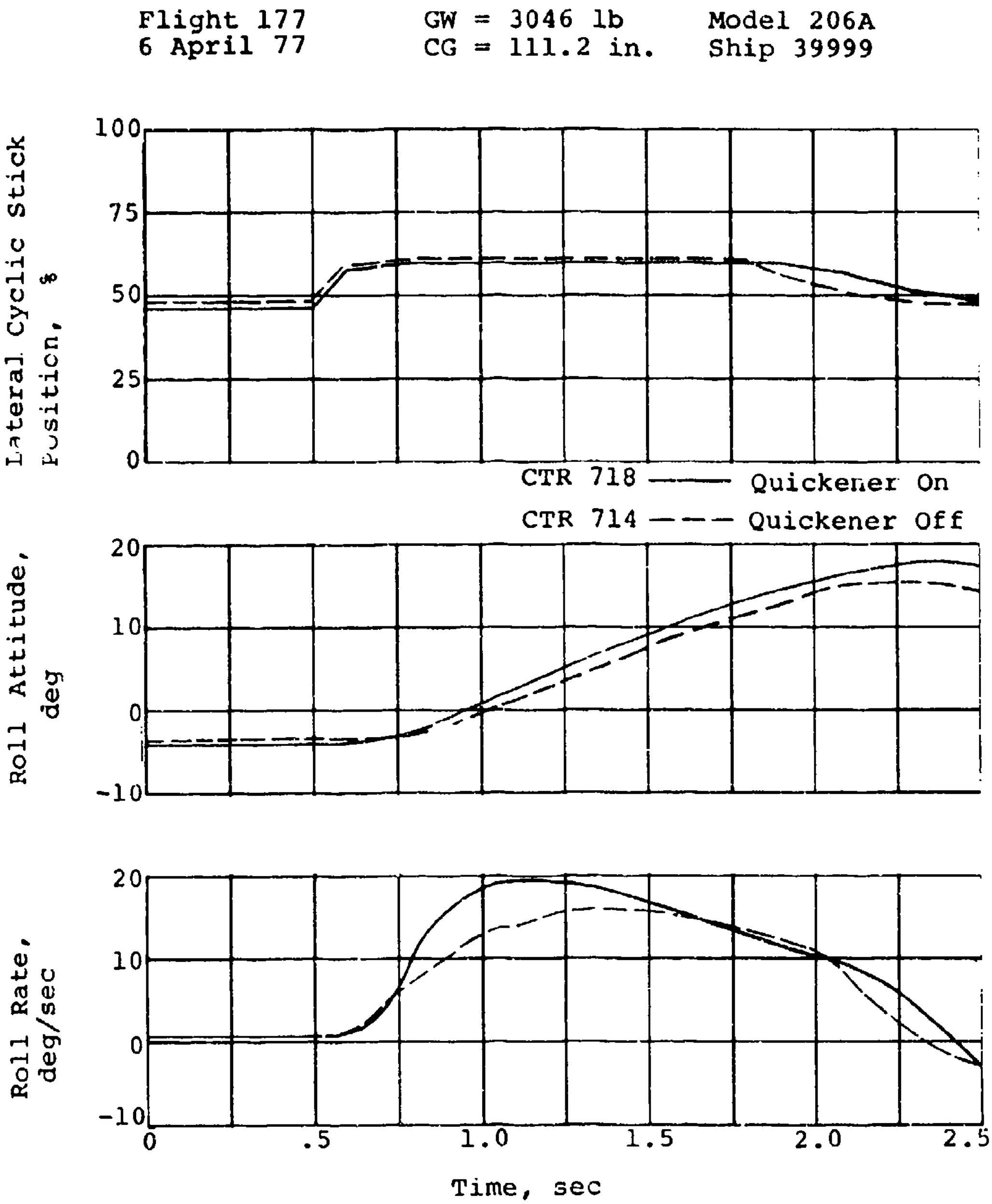

Figure 42. Effect of control quickener on roll response.

TABLE 4. CONTROL GEARING FOR HERS TESTS

Rotor Inertia (slug-ft^2)	Longitudinal Cyclic Control Gearing $\left(\frac{\text{deg swashplate}}{\text{inch stick}}\right)$	Lateral Cyclic Control Gearing $\left(\frac{\text{deg swashplate}}{\text{inch stick}}\right)$
672, 550	2.37 (9% increase)	1.65 (15% increase)
323	2.18 (standard)	1.44 (standard)

The control system modifications were tailored mainly toward the roll axis, which is the most sensitive in the basic ship. The effect of these modifications can be seen in Figures 43 and 44 which show the hovering control response data as measured during the flight test program.

The increased damping due to increased rotor inertia is easily seen in the roll response data. Both high and mid-inertia rotors showed higher damping and slightly lower control sensitivity even with the modified control gearing than the standard inertia case. When the control quickener was used, the high and mid-inertia data both indicate improved sensitivity and damping.

The pitch axis data indicate more scatter and less conclusive trends than indicated in roll data. Since the pitch axis characteristics were considered better than roll in the basic ship, less tailoring of quickener and gearing characteristics was performed.

4.3 HANDLING QUALITIES TEST RESULTS

Once the helicopter control system was modified for the HERS, control response characteristics were measured for the three inertia levels in hover and at 60 knots and 95 knots level flight. Figures 45 and 46 present the measured results for the hover and 60-knot cases. At 95 knots, only the standard and high inertia cases were tested with results similar to that obtained at 60 knots.

4.3.1 Longitudinal Characteristics

With the control quickener off, control response decreases with increased rotor inertia. This would be expected since the additional rotor damping would result in a lower fuselage

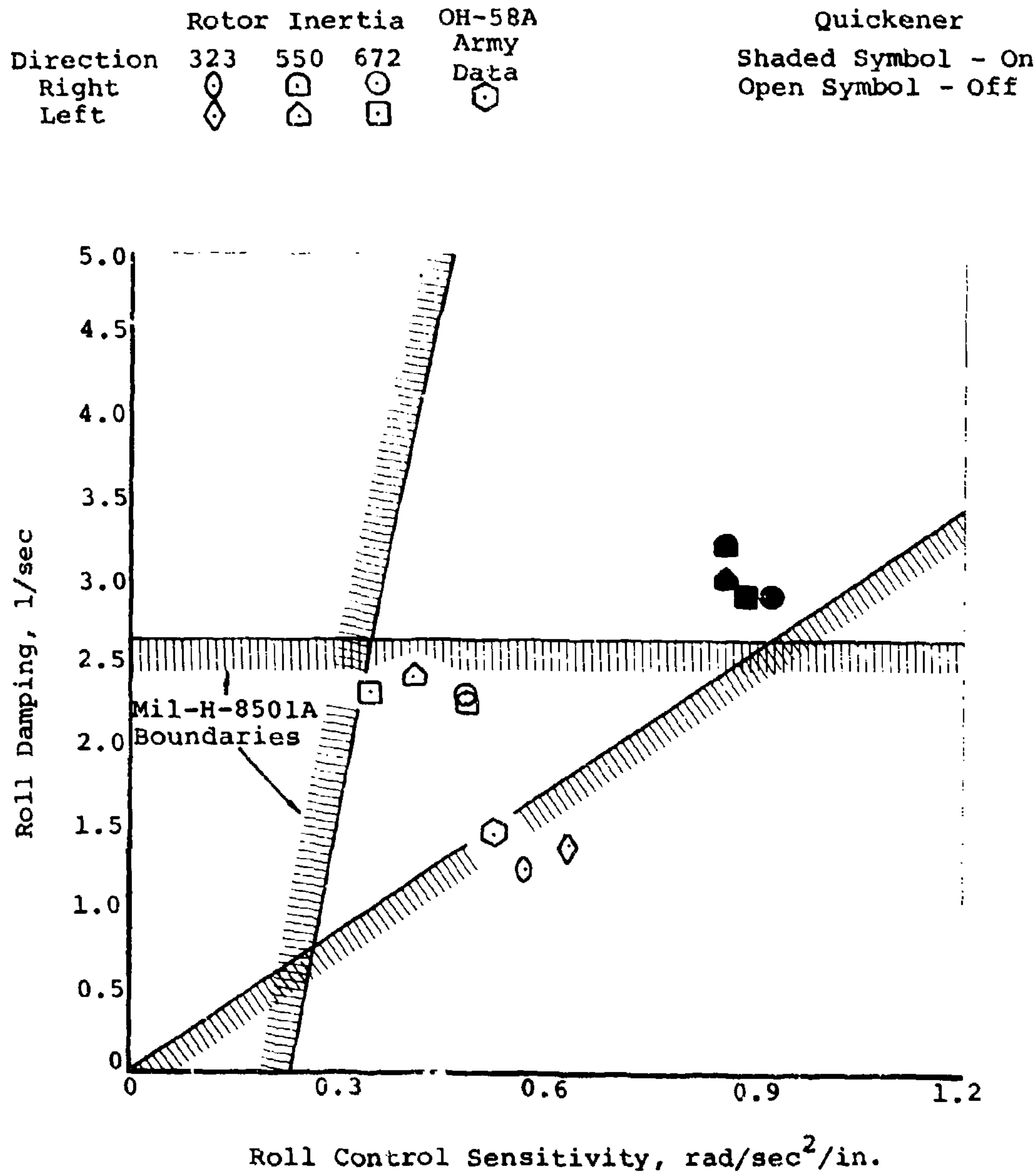

Figure 43. Hovering roll control characteristics.

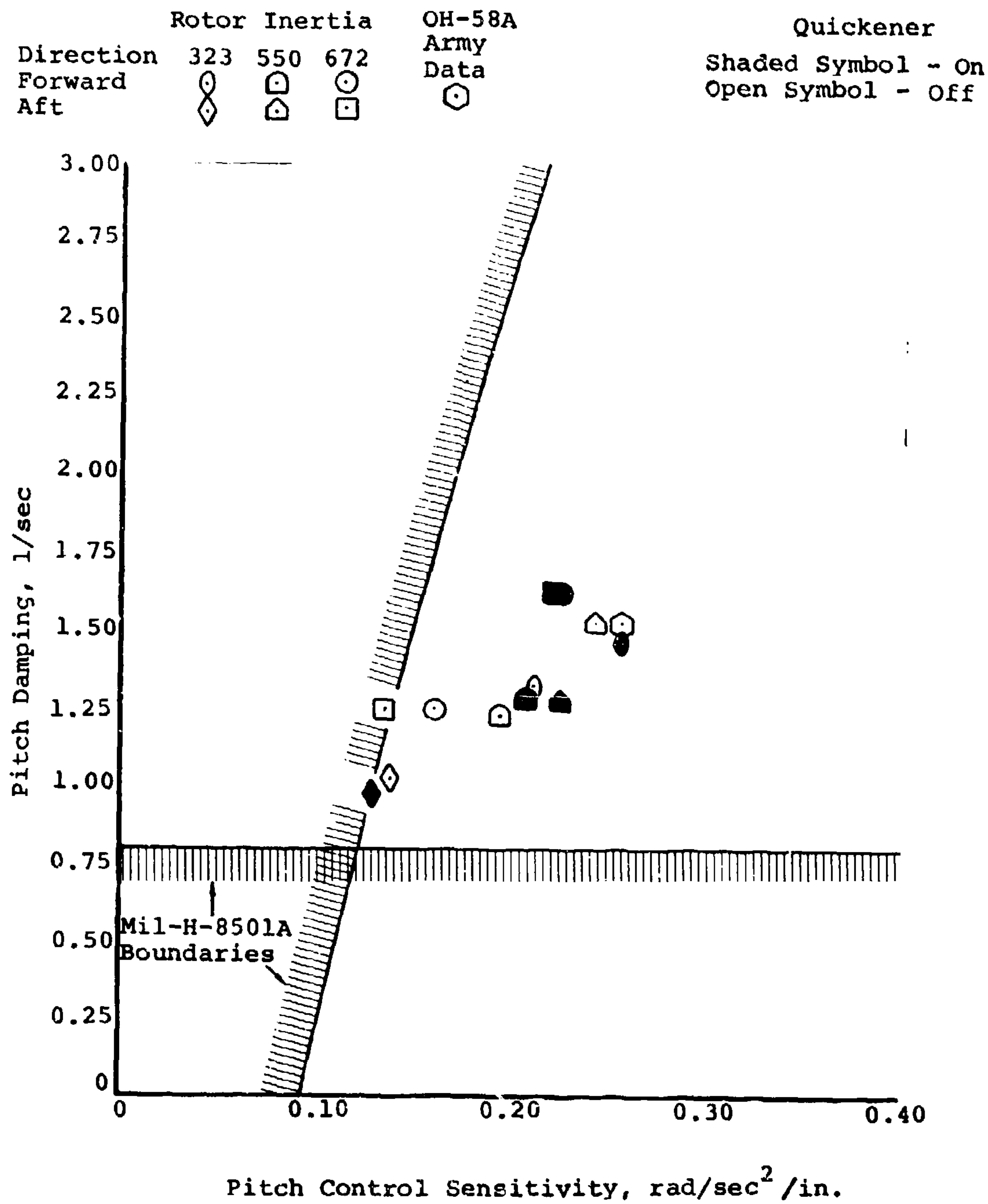

Figure 44. Hovering pitch control characteristics.

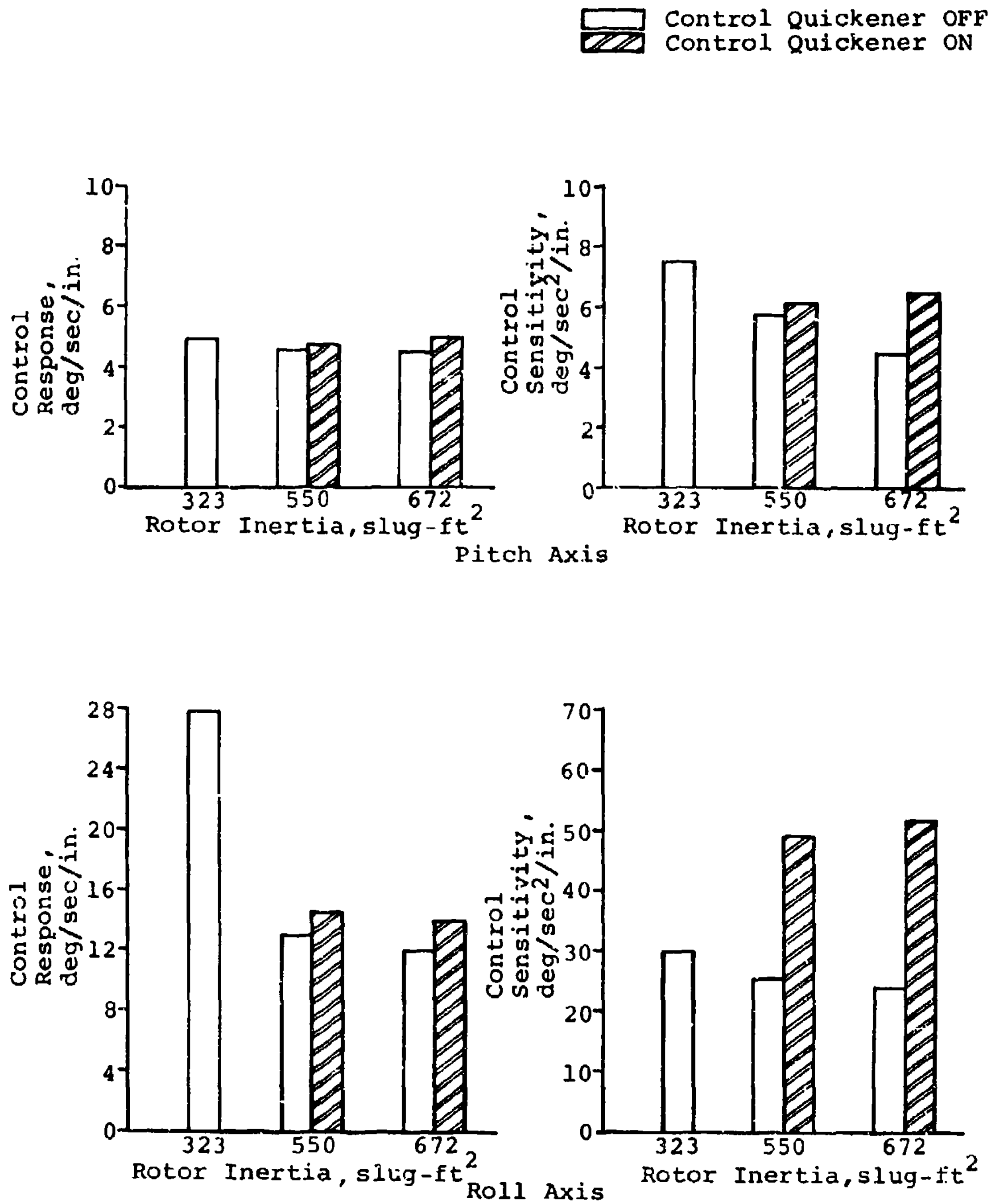

Figure 45. Hover control response and sensitivity variation with rotor inertia and control quickening.

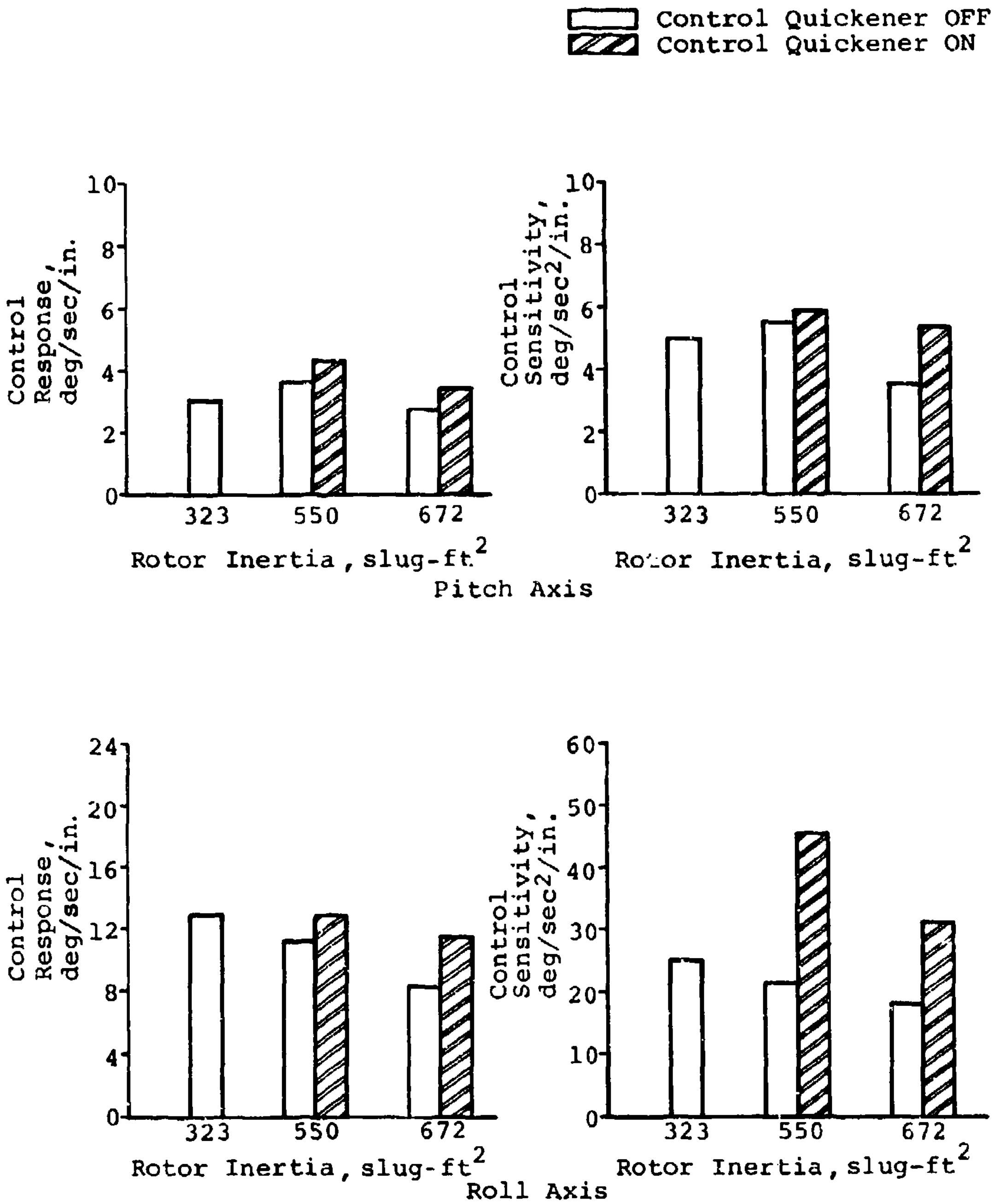

Figure 46. 60-knot level flight control response and sensitivity variation with rotor inertia and control quickening.

rate required to produce a damping moment equal to the control input moment. This general result doesn't hold for the 60-knot pitch response in which the higher control sensitivity produced by the modified control gearing gives a higher input moment and thus higher fuselage response.

When the control quickener is used, both sensitivity and response are almost equal to that of the standard OH-58A. The slightly lower response is noticeable to the pilot and indicates that more tailoring is needed in the pitch axis control quickener.

4.3.2 Lateral Characteristics

With the control quickener inoperative, roll response and sensitivity are noticeably reduced. However, the standard OH-58A roll response exceeds the MIL-H-8501A maximum rate of 20 degrees per second per inch which indicates a reduction in response is desirable. In fact, the control gearing changes and control quickener characteristics were tailored to provide good roll response.

4.3.3 Dynamic Stability

The additional rotor damping caused by increased rotor inertia was discussed in Section 4.1. This rotor damping affects the controls-fixed dynamic stability of the helicopter as well as its response to control inputs. In general, increased rotor damping will increase both the period and the damping ratio of the unaugmented single rotor helicopter in both hover and in forward flight. In order to evaluate the effect of rotor inertia, both longitudinal and lateral cyclic control pulse inputs were used to excite the short period response of the helicopter in hover and forward flight. When possible, long period response was also evaluated.

For the roll axis in hover, lateral cyclic pulses resulted in an oscillatory roll response for all three rotor inertias tested, as illustrated in Figure 47. With low rotor inertia, the roll rate tended to diverge unless corrective cyclic inputs were made. These corrective inputs made it impossible to evaluate the period or damping quantitatively for this inertia.

For mid and high inertia, the roll response is much more acceptable. A well-damped oscillation with a period of 3.8 and 4.5 seconds for mid and high inertia respectively resulted from the control pulse. The damping ratio for both cases is approximately 0.5.

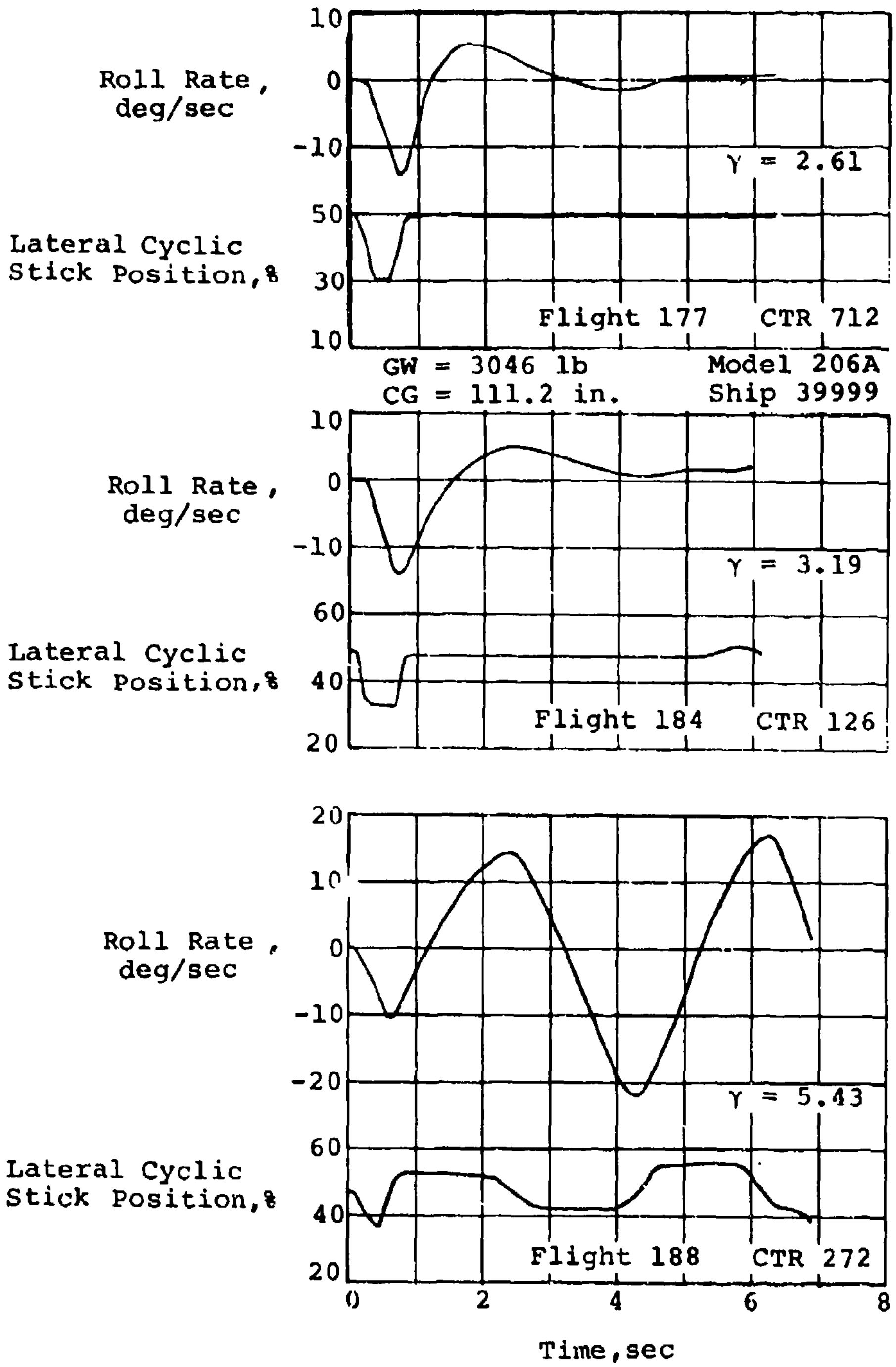

Figure 47. Roll response to lateral pulse inputs for three-rotor inertias.

In the longitudinal axis at hover, a damped short period oscillation was noted for all three rotor inertias. With low inertia, longitudinal control pulses produced an oscillatory response with a period of about 4 seconds and light damping (damping ratio of less than 0.1). The mid-inertia rotor resulted in a 4-second period and a damping ratio of about 0.4. The high inertia rotor gave a 4.2-second period and a 0.5 damping ratio.

At forward speeds of 60 and 95 knots, the increased inertia gave improved dynamic stability over the low inertia rotor, which was apparent to the pilot but difficult to quantify. At 60 knots, the short period longitudinal response was deadbeat for both forward and aft control pulses for all three rotor inertias. A long period oscillation of approximately 24 seconds with light damping was determined for the mid-inertia case but the data for the other inertias are inadequate for period and damping determination. Lateral response was deadbeat for all rotor inertia levels tested.

In general, the effect of increased rotor inertia was very apparent to the pilot. Lower workload in hover was particularly noted but was also apparent in forward flight. Quantitative measurement of dynamic stability characteristics was unsuccessful for several configurations due to the inability to return the cyclic stick to the exact trim position after the pulse input. This caused airspeed and attitude changes which altered the result of the measurement, particularly for long period response. The quantitative results obtained do agree with the pilot opinions and substantiate the improved dynamic stability of HERS.

4.4 GENERAL HANDLING QUALITIES RESULTS

Since the principal objective of this program was to evaluate the effect of rotor inertia on the height-velocity restriction and on NOE maneuverability, the control characteristics were tailored to provide acceptable rather than optimal handling qualities. In particular, the use of the control quickener to increase control sensitivity and response was found to produce a mild but noticeable discontinuity in aircraft response to cyclic control inputs. This resulted from washing out the quickener input (refer to Figure 41) too rapidly. Further modification of the quickener circuitry could have reduced this mild discontinuity but, since the control characteristics were considered acceptable, this was not accomplished.

After completion of the contracted flight test program, BHT conducted a short flight test investigation using a hub spring in conjunction with the HERS. Previous hub spring tests, reported in Reference 8, were conducted with the same hub used in the HERS program. Two spring rates were available, 132 and 210 ft-lb/deg, and results are presented for only the higher spring rate. Only tests of controllability and dynamic stability were performed since the effect of hub restraint was not expected to significantly alter the previous results for height-velocity and NOE maneuverability.

The increased control sensitivity due to the hub spring resulted in hover controllability characteristics approximately equal to those obtained with the high inertia rotor and control quickener on. Figures 48 and 49 illustrate the effect on control response and sensitivity produced by the hub spring or quickener, as compared to the standard OH-58A.

One additional pilot comment noted that the mild discontinuity in aircraft response to cyclic control inputs was not present with the hub spring. This further enhanced the pilot opinion of this hardware combination.

[8]Sonneborn, W., and Yen, J., HUB MOMENT SPRINGS ON TWO-BLADED ROTORS, Proceedings, Specialists Meeting on Rotorcraft Dynamics, American Helicopter Society and NASA Ames Research Center, Moffett Field, California, February 1974.

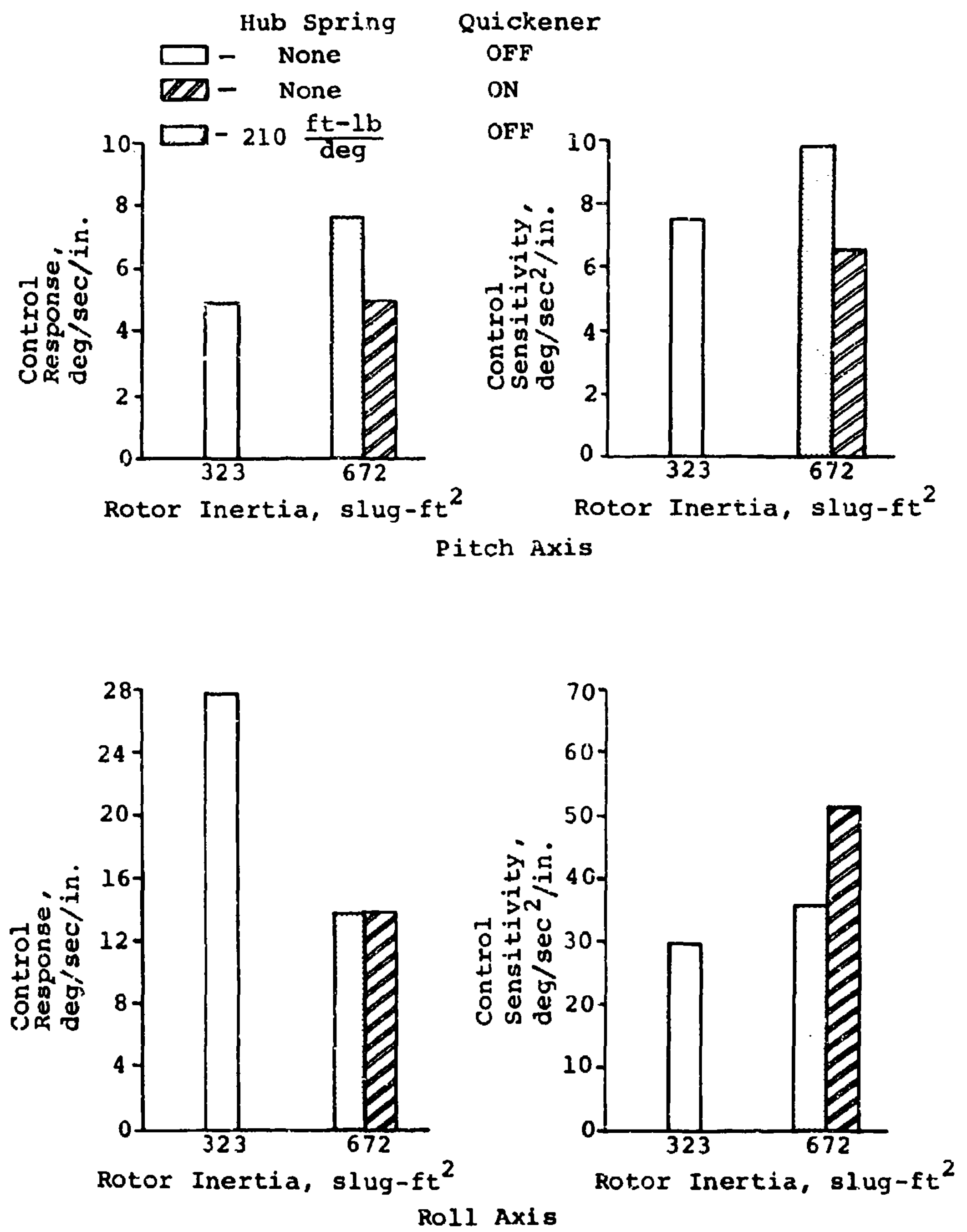

Figure 48. Hover control response and sensitivity variation with rotor inertia and hub spring.

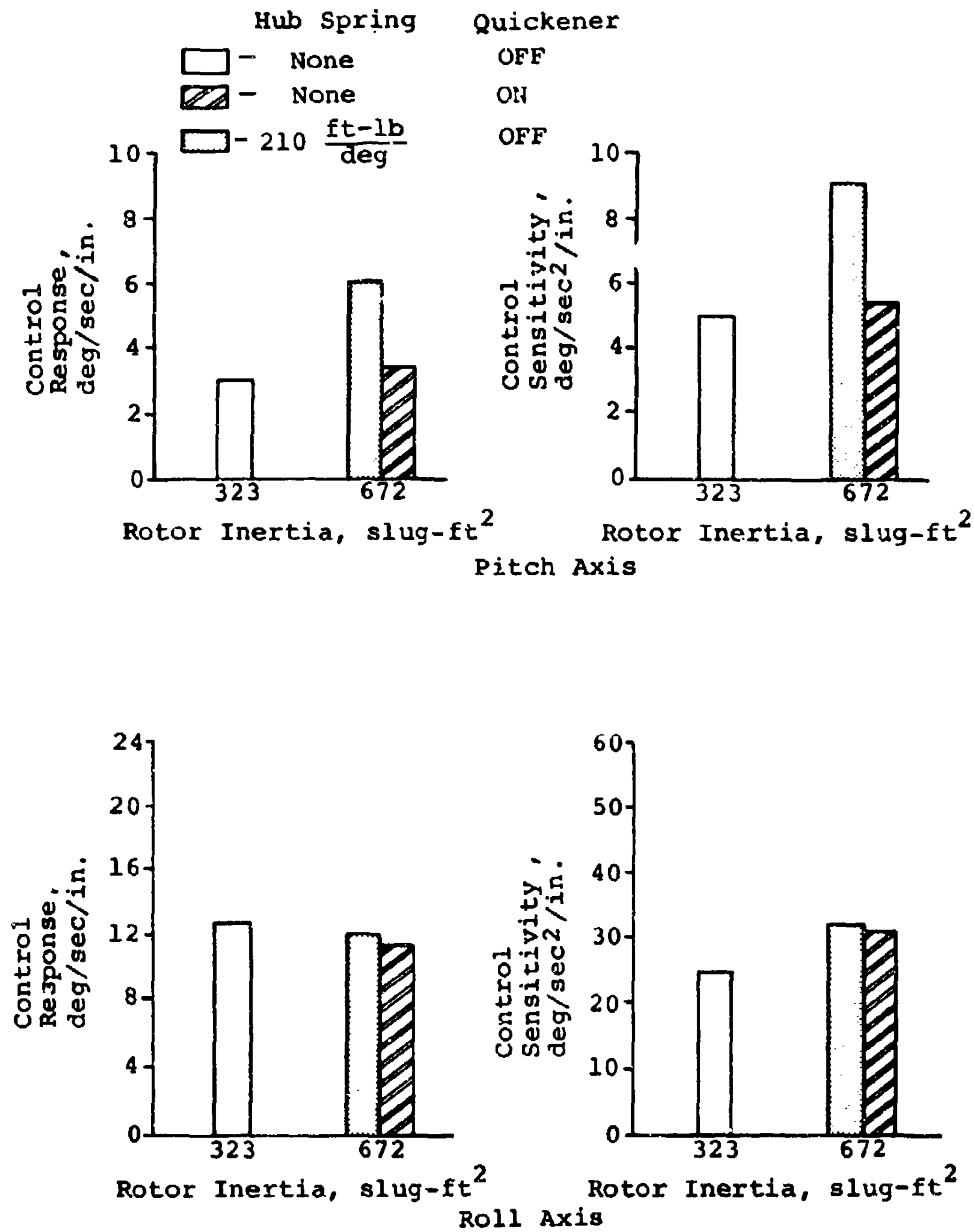

Figure 49. 60-knot level flight control response and sensitivity variation with rotor inertia and hub spring.

5. CONCLUSIONS AND RECOMMENDATIONS

5.1 CONCLUSIONS

The High Energy Rotor System has demonstrated the ability to reduce or eliminate the height-velocity restrictions on autorotational landings. Through the use of a simplified pilot technique, normal autorotational landing procedures were much easier and safer to employ. Low-speed maneuverability was greatly enhanced through significant transient power extraction from stored rotor energy. Longitudinal acceleration from hover was quadrupled and lateral acceleration was doubled with the HERS technique. Vertical climb rate was doubled with the HERS. This transient power is available to offset the weight of the HERS. The effect of the HERS on helicopter handling qualities was evaluated and determined to be improved over the basic helicopter.

The employment of a High Energy Rotor System on operational helicopters will enhance the aircraft's safety and ability to operate in the NOE environment. The stored energy of the rotor can be used easily and efficiently to improve takeoffs, bob-ups, and accelerations. It can accomplish this without increasing the design torque of the main rotor transmission. Recovery from the bleed rpm NOE maneuvers, following engine power failure, can readily be accomplished. The characteristics of this rotor are such that the rpm bleed is slow and easy to control with collective motion. The autorotation landing technique is simplified, and the height-velocity restrictions can be significantly reduced or eliminated.

The combination of the HERS with the hub spring provided a good match for low-speed handling qualities. The increased damping and control sensitivity reduced pilot workload in all flight regimes tested.

5.2 RECOMMENDATIONS

To fully assess the weight increase required for a High Energy Rotor System, a complete rotor design is needed to establish a weight for a production configuration. This program modified existing hardware to achieve high rotor inertia requiring both structural weight and aerodynamic penalties that are not necessary if new designs are made.

The combination of high rotor inertia (low Lock number) and some degree of hub restraint resulted in enhanced handling qualities. The use of the HERS on a hingeless rotor appears to be a promising combination that needs investigation.

REFERENCES

1. Wood, T. L., Bull, H., SAFETY OF FLIGHT DATA FOR THE HIGH ENERGY ROTOR SAFETY OF FLIGHT VERIFICATION REVIEW (SOFVR) EVALUATION, Bell Helicopter Textron Report Number 699-099-051, Fort Worth, Texas, March 1977.

2. Watts, J. C., Condon, G. W., and Pincavage, J. V., HEIGHT-VELOCITY TEST OF THE OH-58A HELICOPTER, USAASTA Technical Report Number 69-16, U. S. Army Aviation Systems Test Activity, Edwards Air Force Base, California, June 1971, AD884973.

3. OPERATOR'S MANUAL, ARMY MODEL OH-58A HELICOPTER, Technical Manual 55-1520-228-10, Headquarters, Department of the Army, Washington, D. C., October 1970.

4. Wood, T. L., HIGH ENERGY ROTOR SYSTEM, presented at the 32nd Annual Forum of the American Helicopter Society, Washington, D. C., May 1976.

5. Yamakawa, G. M., and Watts, J. C., AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST - PRODUCTION OH-58A HELICOPTER UNARMED AND ARMED WITH XM27E1 WEAPON SYSTEM - PERFORMANCE, USAASTA Technical Report 68-30, U. S. Army Aviation Systems Test Activity, Edwards Air Force Base, California, September 1970, AD875793.

6. Drees, J. M., A THEORY OF AIRFLOW THROUGH ROTORS AND ITS APPLICATION TO SOME HELICOPTER PROBLEMS, The Journal of the Helicopter Association of Great Britain, Vol. 3, No. 1, 1949.

7. Bramwell, A. R. S., HELICOPTER DYNAMICS, John Wiley & Sons, New York, 1976.

8. Sonneborn, W., and Yen, J., HUB MOMENT SPRINGS ON TWO-BLADED ROTORS, Proceedings, Specialists Meeting on Rotorcraft Dyanmics, American Helicopter Society and NASA/Ames Research Center, Moffett Field, California, February 1974.

APPENDIX A

INSTRUCTIONS INSTRUMENTATION LIST

Channel No.	Item
1	Main Rotor RPM
2	Pedal Position
3	Collective Position
4	Radar Altimeter
5	Left Cyclic Boost Tube Force
6	Center of Gravity Vertical Accelerometer
7	Lateral Cyclic Stick Position
8	Longitudinal Cyclic Stick Position
9	Yaw Rate
10	Roll Rate
11	Pitch Rate
12	Engine Torque Pressure
13	Yaw Attitude
14	Roll Attitude
15	Pitch Attitude
16	Collective Boost Tube Force
17	Throttle/Touchdown Mark
18	Main Rotor Azimuth

APPENDIX B

FLIGHT LOG

Date	Blade Inertia	Flight Number	Time~ Hr	GW,lb/cg,in.	Density Ratio	Objective of Flight
14 Feb 77	672	166	0.6	-	-	Shakedown
15 Feb 77	672	167A	0.8	2820/107.5	1.022	Height Velocity
15 Feb 77	672	167B	0.3	2618/110.3	1.027	Height Velocity
16 Feb 77	672	168	0.2	-	-	Instrumentation Shakedown
17 Feb 77	672	169A	0.4	2618/110.3	1.022	Height Velocity
17 Feb 77	672	169B	0.9	3048/108.5	0.995	Height Velocity
17 Feb 77	672	169C	0.2	-	-	Instrumentation Shakedown
24 Feb 77	672	170	0.5	3048/108.5	0.973	Full Power Maneuver Test
28 Feb 77	672	171A	1.3	3048/108.5	0.977	Height-Velocity Variable RPM Maneuver Test
28 Feb 77	672	171B	0.2	3048/108.5	0.977	Demo. to Army
04 Mar 77	672	172	0.6	3087/108.7	0.999	Dynamic Stab. & Controllability
08 Mar 77	672	173	1.2	-	-	Adjust Control Quickener
22 Mar 77	672	174	0.7	3046/111.2	-	Dynamic Stab. & Controllability

APPENDIX B (Continued)

Date	Blade Inertia	Flight Number	Time~ Hr	GW,lb/cg,in.	Density Ratio	Objective of Flight
23 Mar 77	672	175	0.4	-	-	Instrumentation Shake-down
29 Mar 77	672	176	0.2	-	-	Instrumentation Shake-down
06 Apr 77	672	177	0.8	3046/111.2	0.956	Dynamic Stab. & Controllability
20 Apr 77	672	178A	0.8	3048/108.5	0.948	Familiarization Army Pilot Evaluation
20 Apr 77	672	178B	1.1	3048/108.5	0.948	Dynamic Stab. & Controllability
25 Apr 77	672	179A	1.1	3048/108.5	0.962	Height Velocity
25 Apr 77	672	179B	1.0	3048/108.5	0.962	Fwd A/S T.O., Power-On Maneuvers
26 Apr 77	672	180A	1.0	3048/108.5	0.987	Reduced Power Maneuvers
26 Apr 77	672	180B	0.7	3048/108.5	0.976	Reduced Power Maneuvers
26 May 77	550	181	0.4	2800/108.0	-	Load Level - Reduced Inertia
31 May 77	550	182	0.1	-	-	Rotor Work
01 Jun 77	550	183	1.3	3028/108.5	0.952	Height-Velocity Testing

APPENDIX B (Concluded)

Date	Blade Inertia	Flight Number	Time~ Hr	GW,lb/cg,in.	Density Ratio	Objective of Flight
02 Jun 77	550	184A	0.5	2600/109.0	0.948	Height-Velocity Testing
02 Jun 77	550	184B	1.0	3028/111.2	0.935	Handling Qualities Reduced Power Testing
20 Jun 77	323	GR24	0.3	-	-	Rotor Shake-down
22 Jun 77	323	185	0.4	-	-	Rotor Shake-down
30 Jun 77	323	186	0.4	3000/108.4	0.930	Load Level
01 Jul 77	323	187	0.2	-	-	Instrumenta-tion
18 Jul 77	323	188	0.4	3050/111.2	0.922	Dynamic Stab. & Controlla-bility
19 Ju 77	323	189	0.3	3050/111.2	0.921	Dynamic Stab. & Controlla-bility
26 Jul 77	323	190	0.1	-	-	Shakedown
27 Jul 77	323	191	0.2	-	-	Shakedown
29 Jul 77	323	192	1.1	3040/108.0	0.940	Height Velocity
03 Aug 77	323	193	0.5	2650/109.0	0.932	Height Velocity

LIST OF SYMBOLS

a	Lift-curve slope of rotor airfoil, c_ℓ/rad
b	Number of blades
c	Chord of blades, ft
E_H	Potential energy of altitude, ft-lb
E_R	Rotational energy of the main rotor, ft-lb
E_{TOTAL}	Total energy of the helicopter, ft-lb
E_V	Kinetic energy of velocity, ft-lb
g	Acceleration of gravity, ft/sec^2
h	Height above a reference level, ft
I_b	Rotational inertia of one blade and hub (1/2 I_R), slug-ft^2
I_R	Rotational inertia of total rotor with hub, slug-ft^2
N_I	Engine power turbine rotational speed
q	Fuselage rate of rotation, rad/sec
P_H	Power extracted from potential energy, ft-lb/sec
P_R	Power extracted from rotational energy, ft-lb/sec
P_V	Power extracted from kinetic energy, ft-lb/sec
R	Rotor radius, ft
t/k	Autorotational index, sec
t_c	Rotor thrust coefficient
$t_{c_{max}}$	Maximum usable thrust coefficient

LIST OF SYMBOLS (Concluded)

V	Helicopter flight path velocity, ft/sec
W	Helicopter gross weight, lb
β	Rotor blade flapping angle, rad
γ	Lock number $= \dfrac{\rho acR^4}{I_b}$
ρ	Air density, slug/ft^3
Ω	Main rotor rotational speed, rad/sec