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

PRELIMINARY FLIGHT TEST DATA
UH-IB HIGH PERFORMANCE HELICOPTER

INTERIM REPORT NO. 5

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US ARMY MOBILITY COMMAND

FOREWORD

This report presents the results of flight investigations conducted in association with the primary flight test investigation of the high performance UH-1 helicopter. Results reported include main rotor blades with inboard trailing edge flaps, a two-bladed flex beam rotor, and tapered tip main rotor blades.

SECTION VI

MAIN ROTOR BLADES WITH INBOARD TRAILING EDGE FLAPS

A. Summary

Trailing edge flaps were installed on the inboard section of a UH-1B rotor and tested on the high-performance helicopter. The objective of the test was to determine a flap position which would offset the rotor blade nose-up pitching moments in the area of $\Psi = 270°$ and result in a reduction of the rotor control loads. The control load was found to be higher than those with the standard rotor for all flap positions investigated. The increase in control load resulted from the increase in blade pitching moments due to the increased blade chord length and moment arm due the flap installation. Although the flaps proved ineffective in reducing the control loads, the test results provided a clear understanding of the origin of the loads associated with the reverse flow region. The magnitudes of the control load and inboard flap section bending moment at $\Psi = 270°$ were found to be dependent on advance ratio and independent of flap position. Further, the magnitudes of the control load and inboard flap section bending moment at a given airspeed were approximately the same for all flap positions tested. The flap bending moments in the inboard section at $\Psi = 270°$ and the outboard section at $\Psi = 90°$ and $270°$ vary directly with the flap position.

The rotor control load follows the same trend as the inboard flap section bending moment, indicating that the primary cause of the high positive control load component at $\Psi = 270°$ is a result of a download (nose-up pitching moment) on the rotor blade in the reverse flow region. As a result of these tests, it is concluded that inboard blade cutouts and/or special blade cuffs can provide a significant reduction in the rotor control loads.

B. Introduction

Tests with high performance helicopters have indicated that high control loads may prove to be a major factor limiting high speed flight. As discussed in Section VIII, the control (pitch link) load includes two major components (see figure 8-6, page 19): a large negative component at $\Psi = 120°$ (identified as "A" in the figure) which is associated with advancing blade tip mach number effects, and a large positive component at $\Psi = 270°$ (identified as "B") which is associated with the reverse flow region on the retreating blade. The negative (advancing blade) component is non-existent below a tip Mach number of approximately .87; above .87 the load increases rapidly and accounts for almost 50% of the total oscillating load at approximately Mach .95. A significant reduction of this load component was achieved with the tapered tip blades

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discussed in Section VIII. The positive (retreating blade) component builds up steadily with increased airspeed. At speeds where the blade tip Mach number is less than .87, this component accounts for nearly all the oscillating control load. The tests described herein were directed toward the investigation on methods for reducing the main rotor control loads, and more specifically, the component associated with the reverse flow region of the retreating blade.

C. Description of Trailing Edge Flaps

Aluminum alloy, .063 inch thick, flaps were installed on the inboard section of a set of UH-1B main rotor blades. The flap was approximately 60 inches long and extended spanwise from rotor station 32.2 to rotor station 92.0. The flap was split into 10 equal spanwise sections by a .06 wide sawcut approximately 6 inches deep measured from the flap trailing edge to allow different flap angle settings to be made along the span, and to reduce the influence of adjacent loadings on the particular locations where load measurements were recorded. Figure 6-1 shows the flaps installed on the rotor blade. The chord of the blade was increased 7 inches by the addition of the flap. The focus for the flap angle settings was approximately 1 inch aft of the basic blade trailing edge. Strain gages to record flap bending moments were installed 5-1/2 inches forward of the flap trailing edge at rotor stations 41.2 and 83.0 (center of second and ninth flap section respectively). Tests with the trailing edge flaps were conducted during the last two weeks of February, 1964.

D. Test Results

Test flights were conducted with the flaps set at trail and at 13° up and 30° down from trail. A three inch section was removed from the trailing edge of the flaps prior to the test at the 30° down setting. As a result of reducing the chord length of the flap, the sensitivity of the bending moment traces was reduced. The 30° flap data presented herein are referred to in the equivalent bending moment for the 13° and trail flap lengths (i.e.: measured 30° flap data are multiplied by 4.85 which is the square of the ratio of the respective chord lengths assuming a uniform load). Figure 6-2 shows the pitch link load and the inboard and outboard flap bending moments as a function of blade azimuth for 120 knot level flight. The lowest pitch link load is with the flap at trail. It should be noted, however, that this load is approximately twice that of the standard rotor at the same airspeed. As shown in the figure, the flap position has a strong influence on the pitch link load component in the area of $\psi = 90°$ (advancing blade). In this area, the load is directly related to the flap bending moments.

The most important findings of these tests concern the load component in the area of $\psi = 270°$. In figure 6-2, it is seen that the inboard flaps section bending moments are all in the same direction, and approximately

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equal, regardless of flap position at all azimuth positions within the reverse flow region. The bending moments of the outboard flap section, which is outside of the reverse flow area, follow the same trends as for the steady moments in hover. Further, it can be seen that for azimuth positions where the inboard flap section is outside of the reverse flow region, that the curve formed by summing the inboard and outboard section moment will closely follow the control load trace. The flap section bending moments at $\psi = 270°$ are plotted against the rotor advance ratio ($V/\Omega R$) in figure 6-3. The inboard section is in the reverse flow region for all advance ratios shown. It is seen that the bending moments increase with increasing advance ratio, and that the moments are independent of flap position. The character of the curve is the same as for the pitch link load component with the standard rotor at $\psi = 270°$. The outboard flap section is outside of the reverse flow region for all advance ratios shown. In this case, the bending moments vary with flap position and are in the same sense as the flap position, i.e., up flap increases the blade nose up pitching moment, and down flap increases the blade nose down pitching. Based on the foregoing, it may be concluded that the pitch link load component which peaks at approximately $\psi = 270°$ results primarily from the blade nose up pitching moments within the reverse flow region. It is further concluded that this component can be significantly reduced by adding root cutouts to the rotor blade or by the installation of a fixed cuff, which is relatively insensitive to large angle of attack changes, or by adding air foil shaped feathering cuffs.

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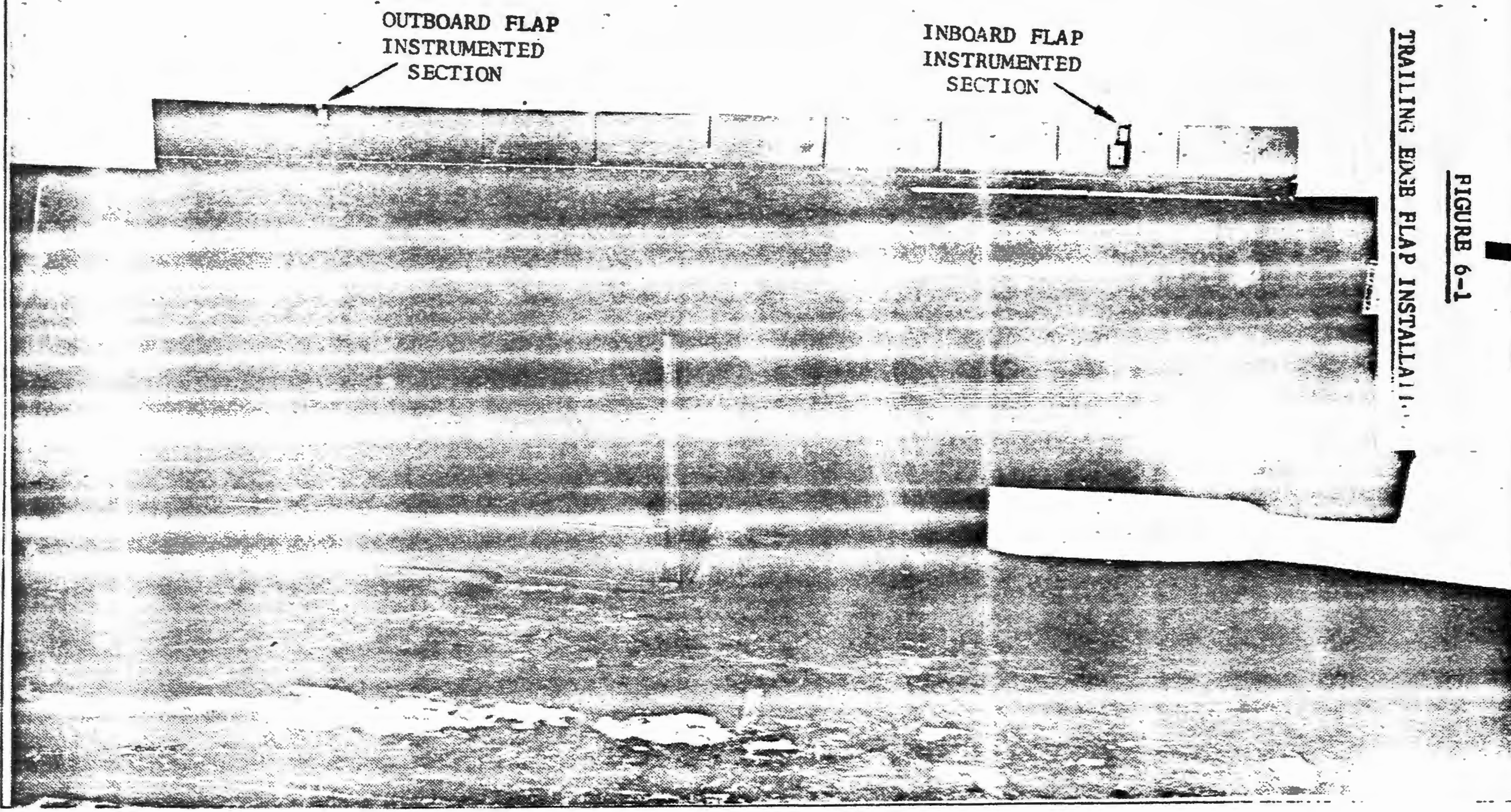

FIGURE 6-1

TRAILING EDGE FLAP INSTALLATI

FIGURE 6-2

FLAP & CONTROL LOADS VS. BLADE RADIAL POSITION AT 125 KNOTS

FLAP POSITION

—·—·— 13° Up

———— Trail

- - - - - 30° Down

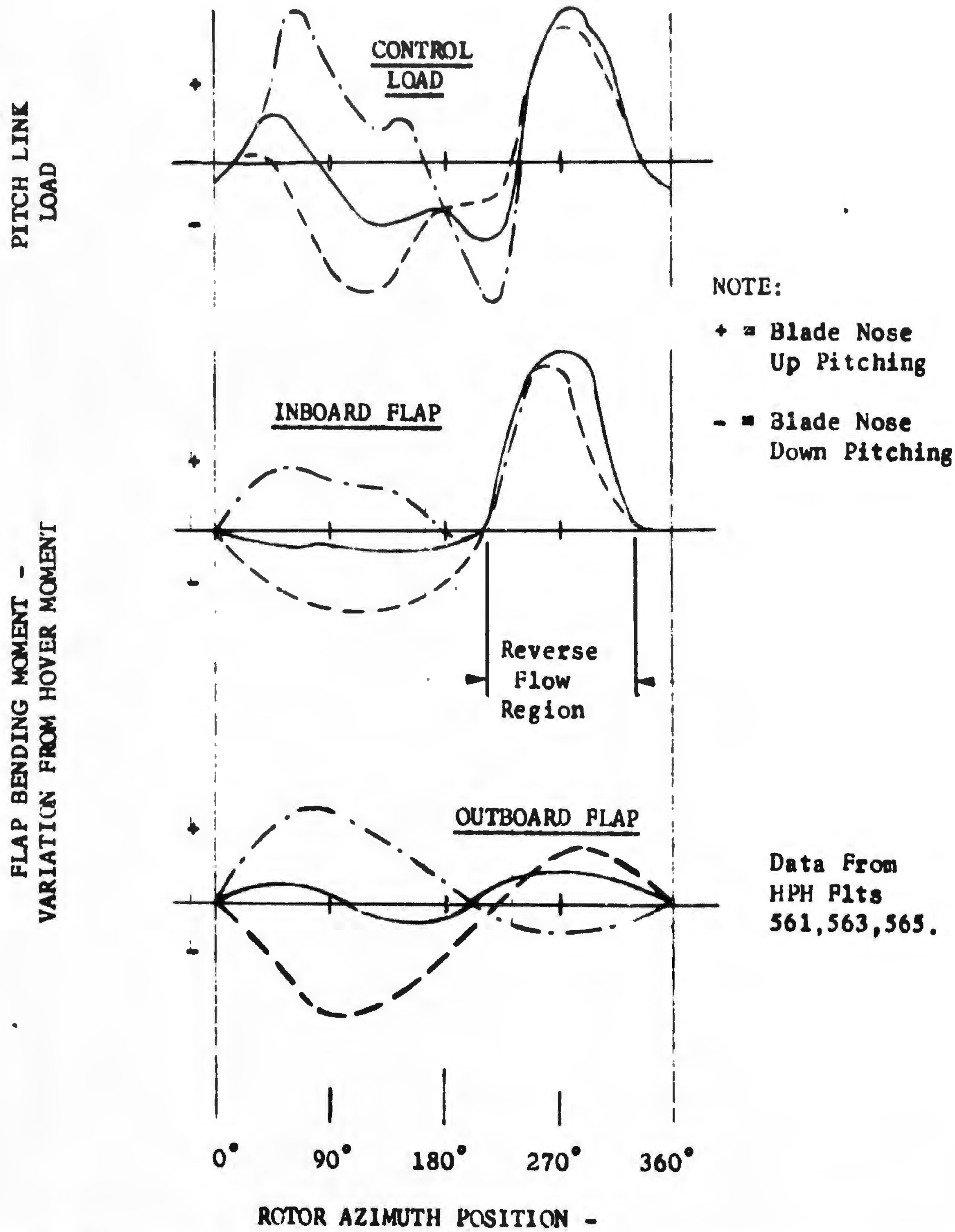

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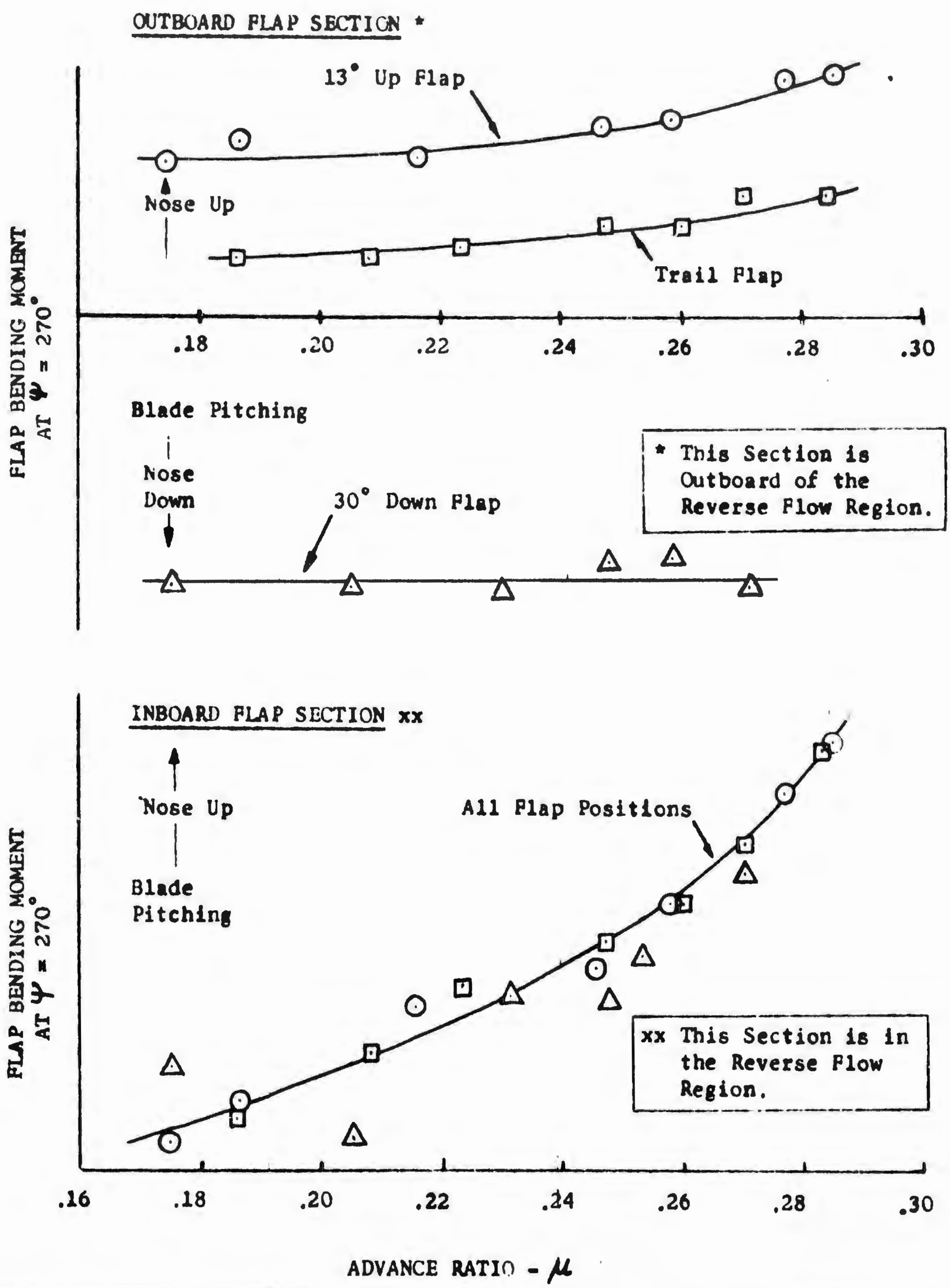
FIGURE 6-3

FLAP LOAD AT $\Psi = 270^\circ$ VS.

ADVANCE RATIO AND FLAP POSITION



SECTION VII

FLIGHT TEST OF A TWO-BLADED FLEX BEAM ROTOR

VII

FLIGHT TEST OF A TWO-BLADED FLEX BEAM ROTOR

A. Summary

A two-bladed, 44 foot diameter, flex beam rotor incorporating a soft beamwise bending and stiff chordwise bending section in the yoke was designed, fabricated and tested on a Model 204B helicopter. The rotor which utilizes production UH-1 grips, trunnion, and pillow blocks, was tested with both 21 inch and 27 inch chord blades. The purpose of the flight tests was to determine if this rotor design would result in a low vibration configuration similar to that experienced with the Phase II, Model 540 "doorhinge" rotor system.

As a result of the flight tests it was found that the pilot seat 2/rev vertical vibration level was comparable to that with the Phase II, 540 rotor at true airspeeds up to 130 knots. At speeds greater than 60 knots the copilot 2/rev vibration level was considerably worse than with the Model 540 rotor but better than the UH-1B rotor. The pilot and copilot seat 4/rev vertical vibration level was more predominant than the 2/rev level probably due to the fuselage characteristics of the test helicopter which was configured with a special long tailboom. The rotor and control loads with the 27 inch chord blades were comparable to those recorded from the Phase II (27 inch chord blades), Model 540 rotor.

B. Introduction

Test results with the Model 540 rotor, which combined a soft beamwise bending and stiff chordwise bending section in the hub, showed a significant reduction in the airframe vibration levels. The same design concept was applied to the flex beam hub and blade assembly discussed herein. The flex beam hub incorporated a simple inexpensive hub and production UH-1B grips, pillow blocks, and trunnions. The purpose of these tests was to determine if airframe vibration levels and rotor load levels comparable to the Model 540 rotor could be achieved with a more conventional rotor design using production UH-1B components.

C. Description of Rotor and System Hub Assembly

The flex beam hub assembly was designed to incorporate production UH-1B components wherever possible. Figure 7-1 shows the hub assembly. The major components making up the hub include a new yoke, the high performance three-bladed rotor spindles, and pitch horns, and production

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UH-1B grips, trunnion, pillow blocks and straps. With 21 inch chord blades the 25° δ_3 pitch horns were replaced by 0° δ_3 horns. The total weight of the hub assembly is 554.5 pounds.

Main Rotor Blades - The initial tests were conducted with the 21 inch chord blades from the high performance three-bladed rotor. These blades are similar to the UH-1B blade except for twist and length. The blades have 6° twist and are 1 foot shorter. The decreased blade length is required to maintain a 44 foot rotor diameter since the blade attachment is made 12 inches further outboard from standard for the flex beam hub. Additionally, 20 pound tip weights were installed in each blade. During the course of testing an additional 10 pounds (30 pounds total) of weight was added to the tip of each blade.

Later tests were conducted using the Model 540, 27 inch chord blades. These blades have a special NACA 0009-1/3 airfoil, 10° twist, and have 30 pounds of weight installed in each tip.

Stabilizer Bar Assembly - With the 21 inch chord blades installed, the stabilizer bar travel was restricted to ± 12° (± 15° standard) to prevent an interference between the stabilizer bar and pitch horns at the maximum blade angles.

The stabilizer bar assembly was removed when the 27 inch chord blades were installed to provide a higher minimum blade angle (6° versus 4° standard).

Test Helicopter - The tests were conducted with a company owned Model 204B (S/N 2010) helicopter, which was configured with a special honeycomb structure tail boom for a 48 foot diameter rotor. Instrumentation was installed to determine fuselage vibrations and rotor and control loads.

D. Test Results and Discussion

A limited flight test program was conducted to obtain data on pilot and copilot vibration, rotor and control loads, level flight performance and collective control response and stick forces.

Performance

Speed-power data were obtained with both the 21 inch and 27 inch rotor blade configuration. Figure 7-2 shows the power required for the 21 inch chord blade configuration at 324 rotor rpm. For comparison, data are also shown for the 27 inch chord blade configuration at the same rotor rpm and approximate gross weight.

Speed-power data were obtained from approximately 60 to 110 knots at rotor speeds of 300, 314 and 324 rpm with the 27 inch chord blades installed. These data are shown in Figure 7-3. The speed-power data at 324 rotor rpm are replotted and compared with UH-1B (Edward's Category II) and Model 540, Phase II data in Figure 7-4.

Rotor and Control Loads - Rotor and control load data were obtained for two gross weights at normal operating rpm (324 rpm) for both the 21 and 27 inch chord blade configurations. Additionally, the effect of rotor rpm on the 27 inch chord blade configuration rotor and control loads was evaluated in hover and at level flight speeds from approximately 84 to 99 knots.

Figure 7-5 shows the yoke beam and chord bending and pitch link oscillatory loads for two gross weights with the 21 and 27 inch chord rotor configurations. Data for the Phase II, Model 540 rotor are also shown for comparison. The effect of rotor rpm on the oscillatory loads and vibration level is shown in Figure 7-6.

Vibration - Pilot and copilot seat vertical vibration level data were recorded and harmonically analyzed for both the 21 and 27 inch chord blade configurations. For all rotor configurations, including the standard UH-1B, the 4/rev vibration level was predominant, indicating that the test helicopter (204B, S/N 2010) fuselage characteristics were changed due to the special long, honeycomb tailboom which was installed at the time of these tests. The 4/rev vibration levels for 21 inch and 27 inch chord flex beam rotor and standard UH-1B rotors are shown in Figure 7-7.

The 2/rev vibration levels are shown in Figure 7-8. From the figure it is seen that the pilot seat vibration level with the flex beam rotor is about 60 per cent of that with the standard UH-1B rotor at 130 knots true airspeed.

The 2/rev vibration level with the Phase II, Model 540 rotor is also shown for comparison. It is to be noted that the validity of direct comparison is questionable since the flex beam and UH-1B rotor data were recorded on a test vehicle configured with a special tailboom, while the 540 data were recorded on a production helicopter.

The vibration level while translating from in-ground effect hover through 40 knots, was very low and compared favorably to the Model 540 vibration levels through the same flight regime. During deceleration a significant increase in vibration level was noted from about 30 knots to hover (IGE). The vibration level was similar to a slightly less than normally encountered with the standard UH-1B.

Collective Control Response and Stick Forces - The collective control response during hovering flight and rotor speed decay checks was found to be equal to, and possibly better than, the standard UH-1B rotor. This is attributed to the rotor inertia and the fact that the control coupling resulted in an incremental blade angle increase with rotor coning. The collective stick forces with the 21 inch chord blade installed were entirely satisfactory with a minimum of collective counterweights required (1.4 pounds). With the 27 inch chord blades installed, satisfactory hydraulic boost-off operation was not possible even with the maximum allowable collective centerweight installed (14.1 pounds). To assure satisfactory stick force in the event of an inflight hydraulic system failure, an auxiliary hydraulic boost system was installed.

B. Conclusions

As a result of these tests it is concluded that the improvements in rotor and control loads, and airframe vibration levels demonstrated with Model 540 "doorhinge" rotor may also be attained with a flex beam rotor configuration employing a conventional spindle-bearing arrangement.

Specifically, it is concluded that:

- The 2/rev vertical vibration level at the pilot seat location showed a significant improvement over the UH-1B rotor and is comparable to the Model 540 "doorhinge" rotor.

- The 2/rev vertical vibration level of the copilot seat location was lower than that with the UH-1B rotor but higher than the Model 540.

- The 4/rev vertical vibration level at the pilot and copilot stations was the predominant excitation noted. This may have been due to the fuselage characteristics of the test vehicle with the special, long tailboom installed since the predominant excitation with a UH-1B rotor tested on the same helicopter was also 4/rev.

- During acceleration from hover through the transition airspeed range, the normal vibration buildup was not present with the flex beam rotor. However, during deceleration to hover (IGE) an increase of vibration level was noted.

- The rotor and control loads with the 27 inch chord flex beam rotor were comparable to those recorded during the Model 540, Phase II program.

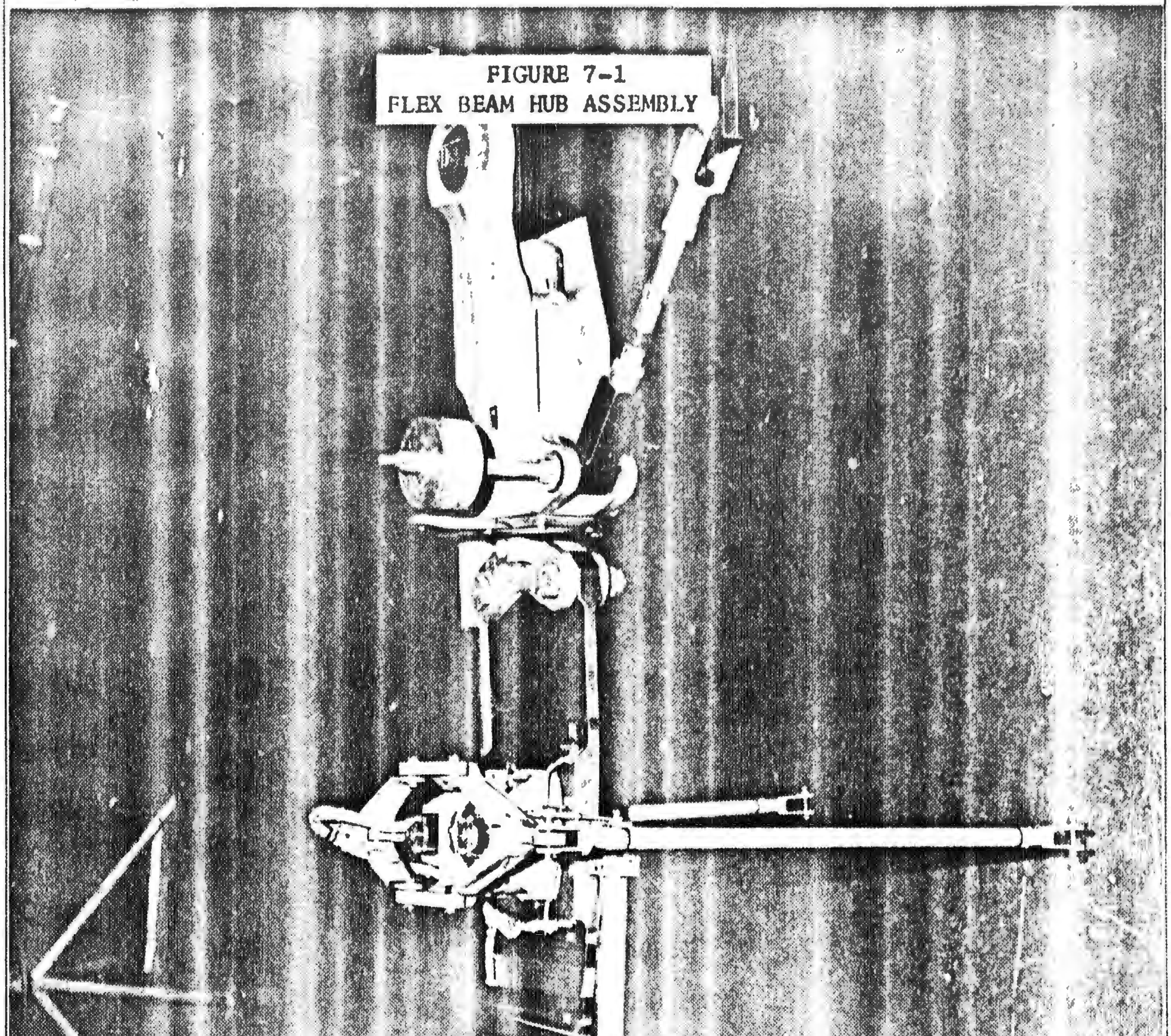

FIGURE 7-1
FLEX BEAM HUB ASSEMBLY

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FIGURE 7-2
COMPARISON OF POWER REQUIRED VS AIRSPEED
FOR 21 INCH CHORD AND 27 INCH CHORD
BLADES IN FLEX BEAM HUB

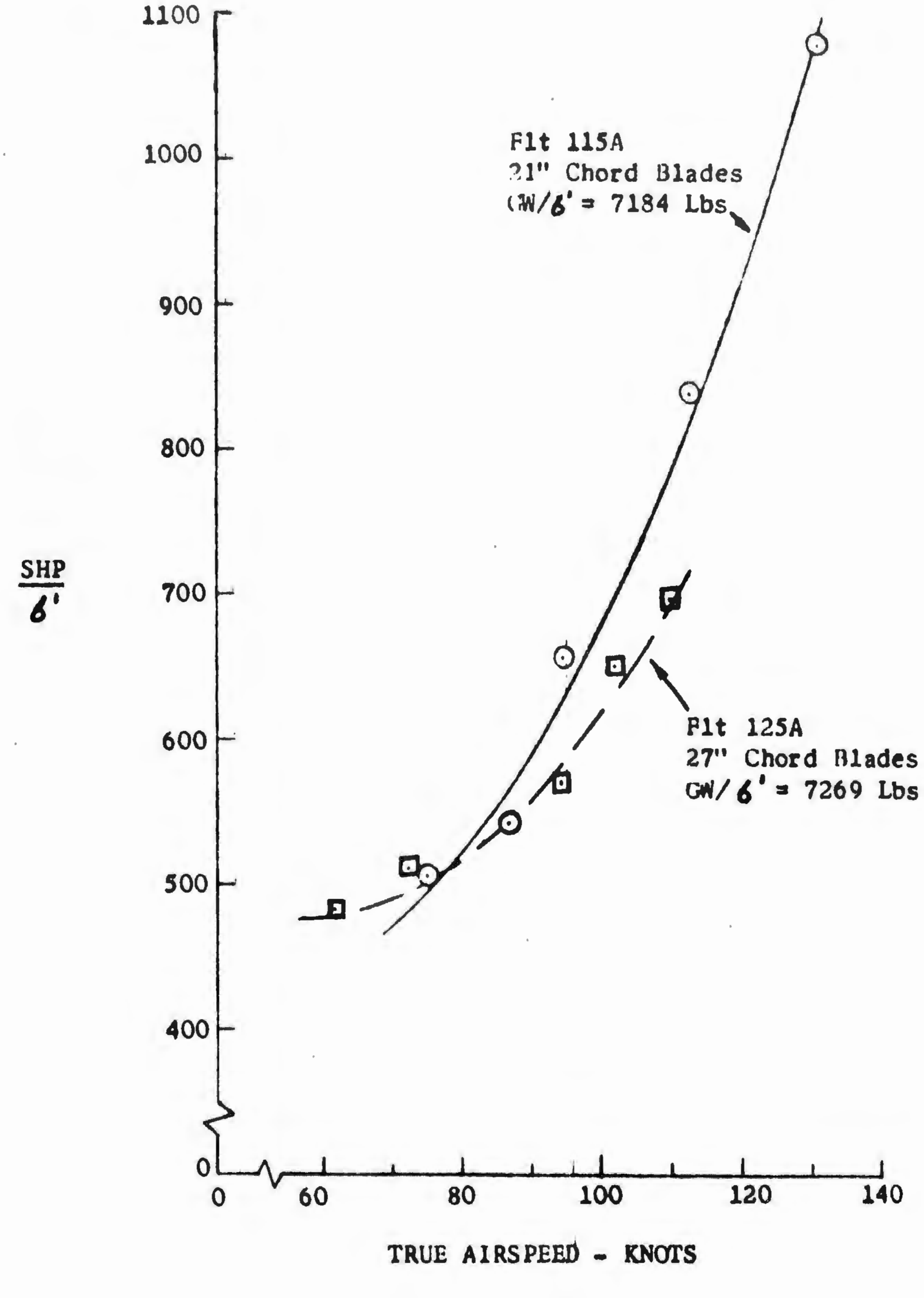

FIGURE 7-3
EFFECT OF RPM ON POWER REQUIRED
FOR FLEX BEAM HUB ROTOR WITH 27 INCH CHORD BLADES

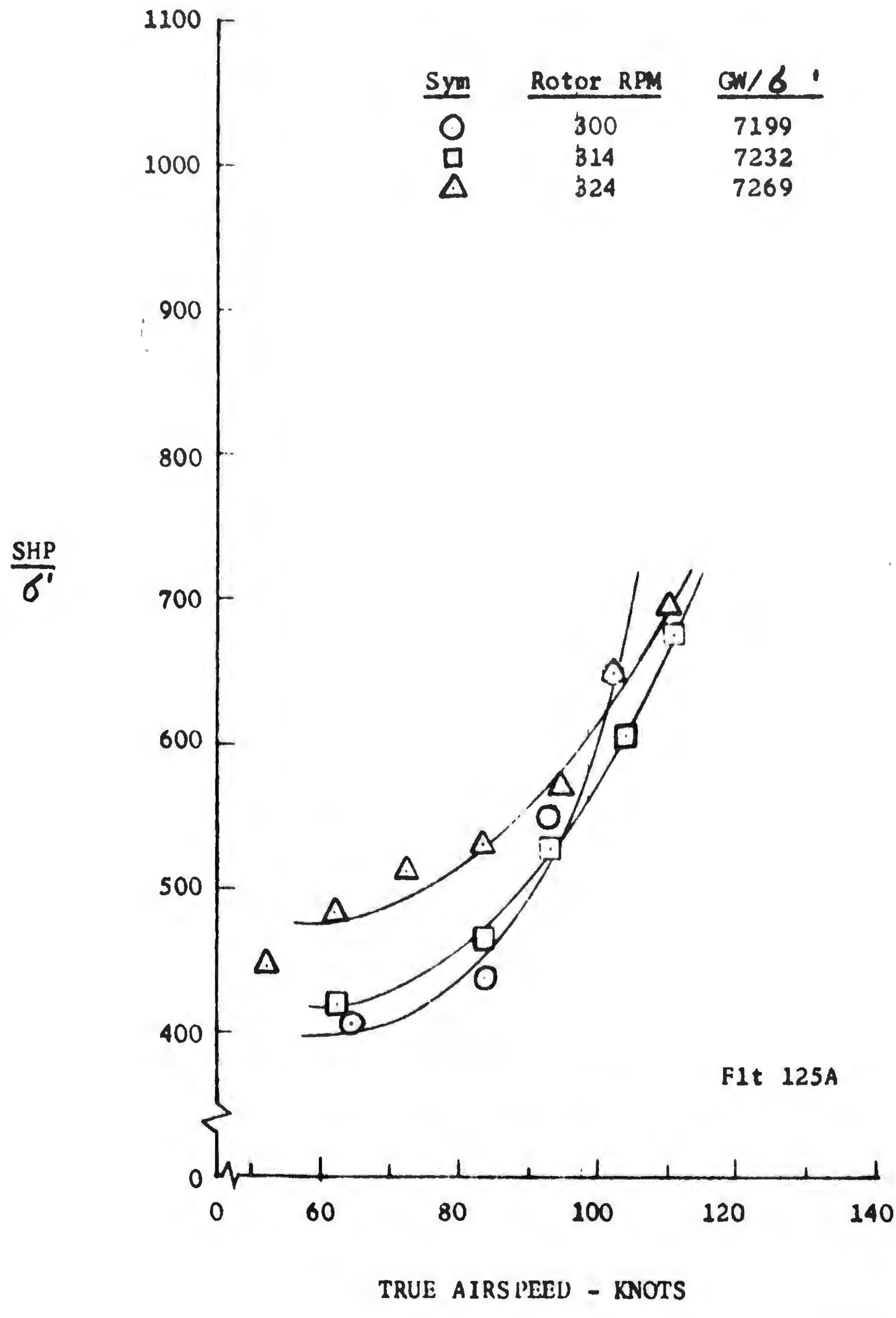

FIGURE 7-4
COMPARISON OF POWER REQUIRED VS AIRSPEED FOR THE
FLEX BEAM HUB ROTOR WITH 27" CHORD BLADES, THE UH-1B ROTOR
AND THE 540 ROTOR

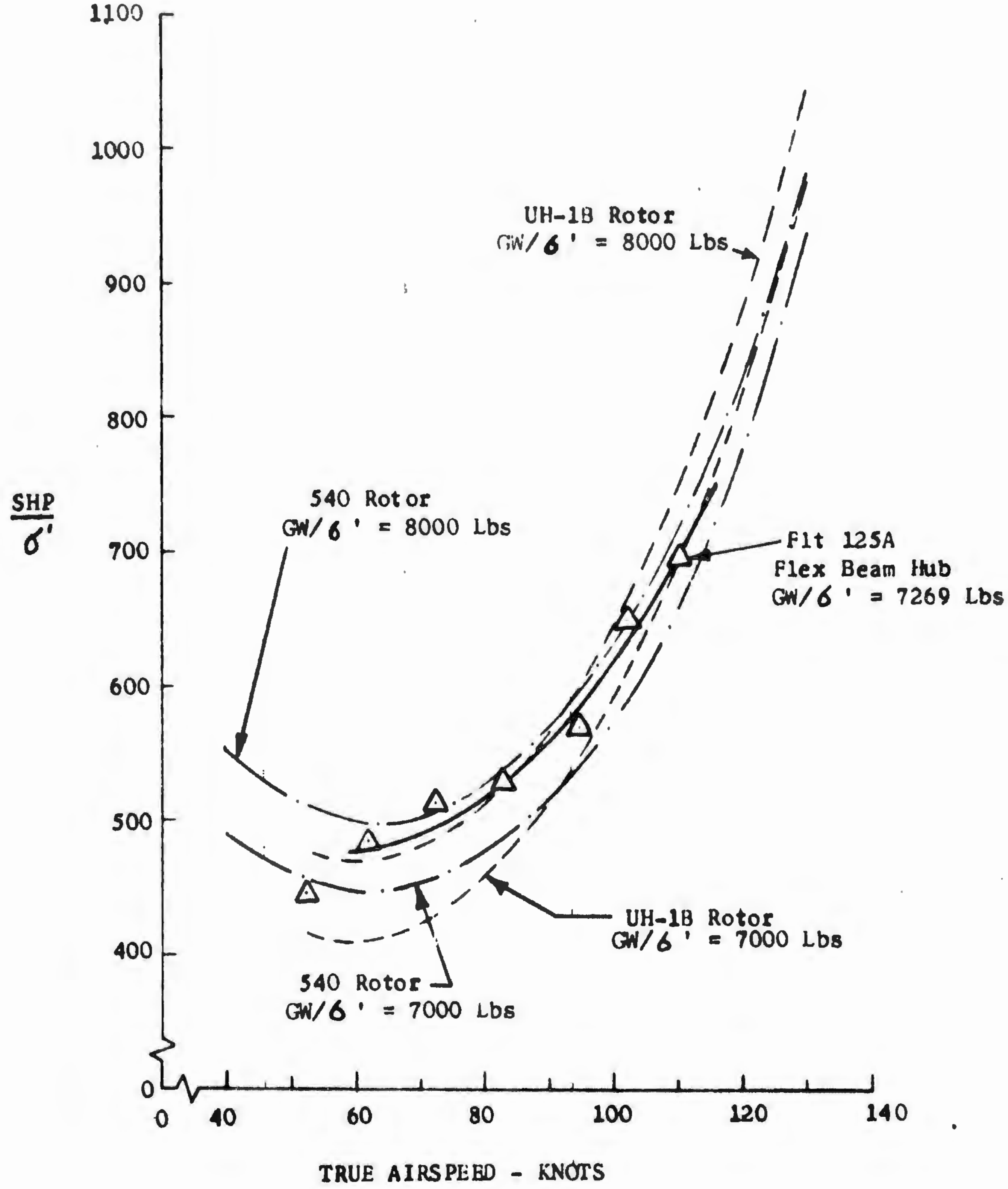

FIGURE 7-5
ROTOR AND CONTROL LOADS FOR THE FLEX BEAM
HUB WITH 21 INCH AND 27 INCH CHORD BLADES

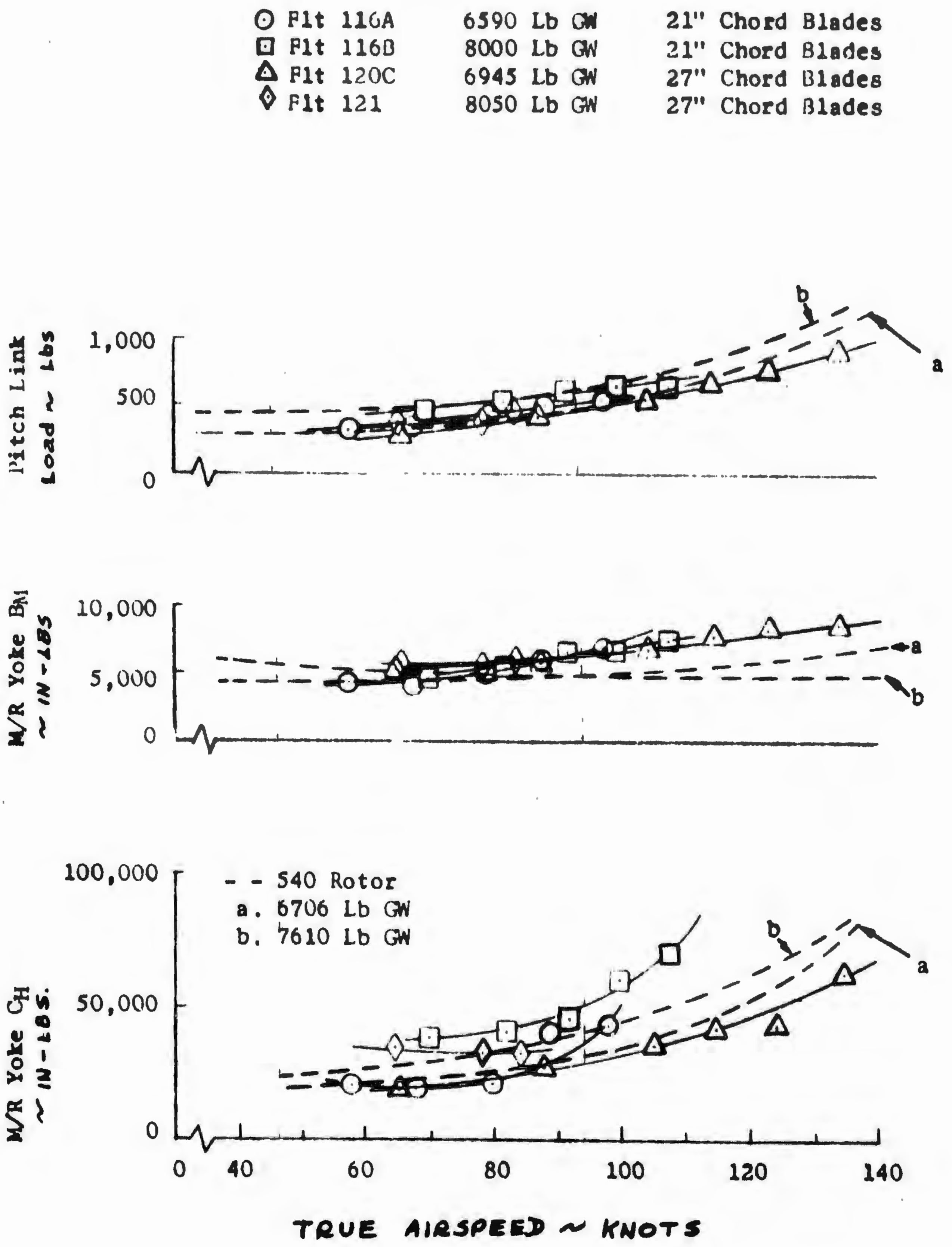

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FIGURE 7-6
EFFECT OF ROTOR SPEED ON THE ROTOR
AND CONTROL LOADS AND PILOT SEAT VIBRATION LEVEL

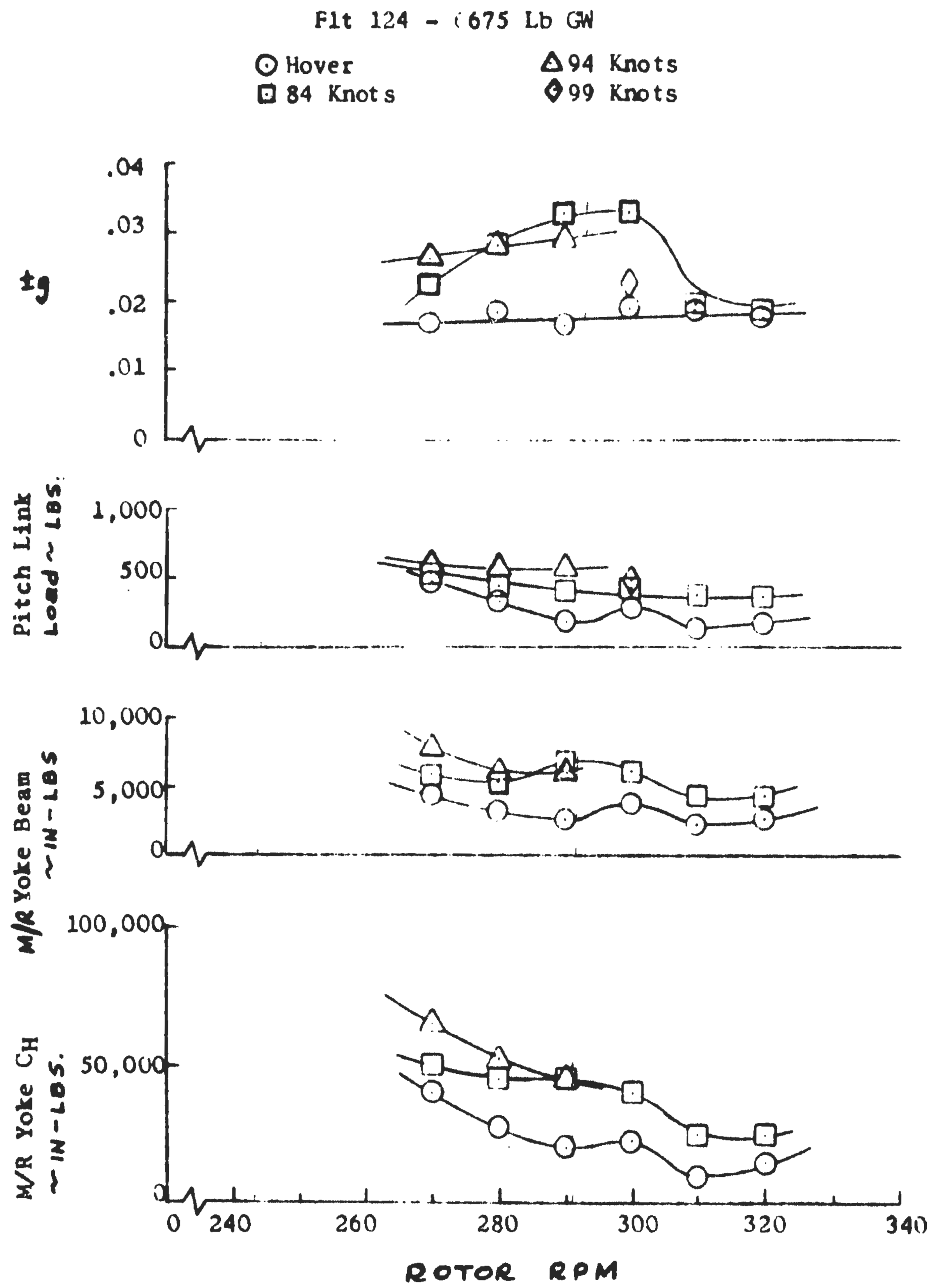


FIGURE 7-7
COMPARISON OF 4/REV VIBRATION LEVELS FOR FLEX BEAM ROTOR WITH 21 INCH AND 27 INCH CHORD BLADES AND THE UH-1B ROTOR

MODEL 204B EXPERIMENTAL HELICOPTER S/N 2010

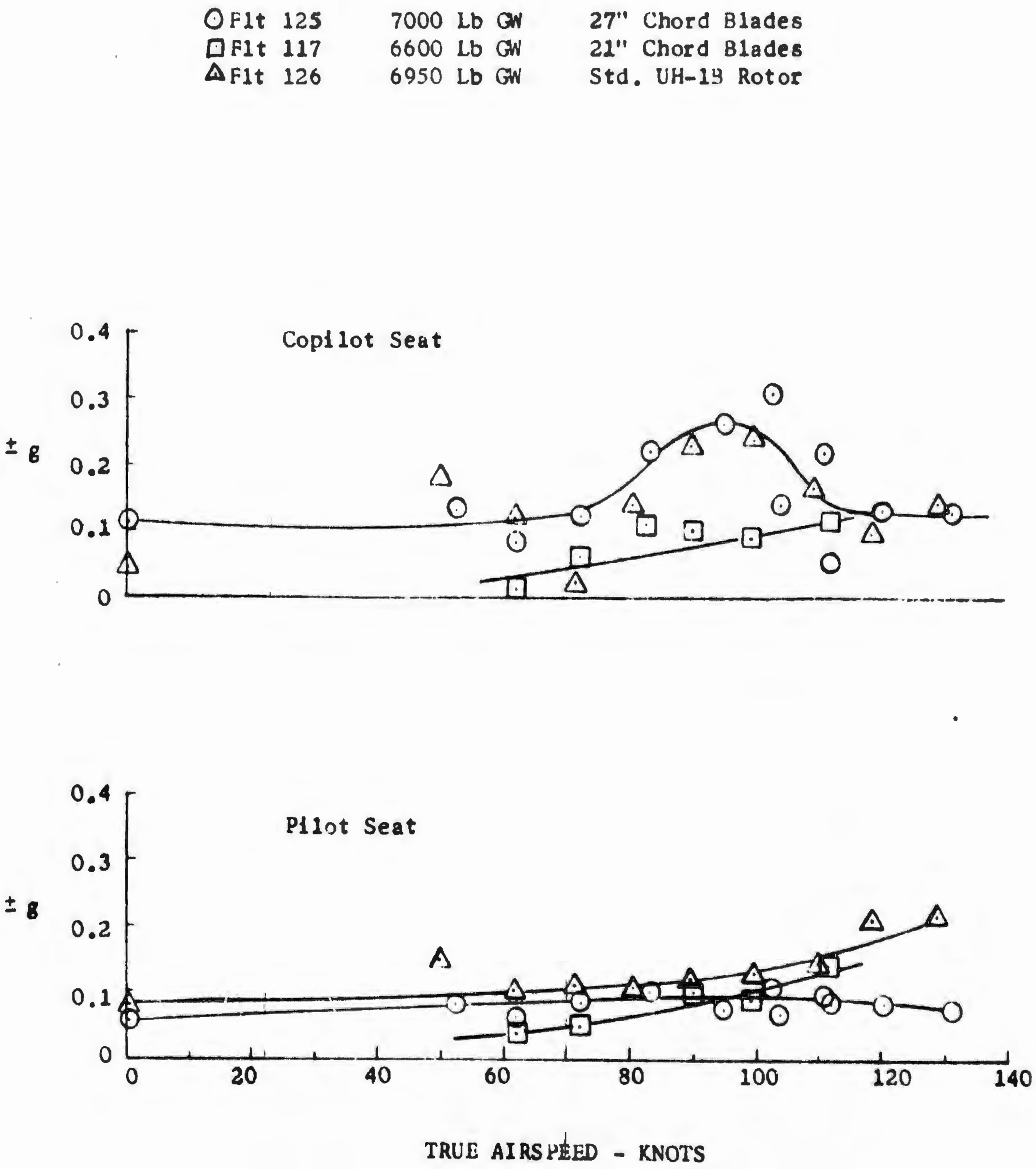
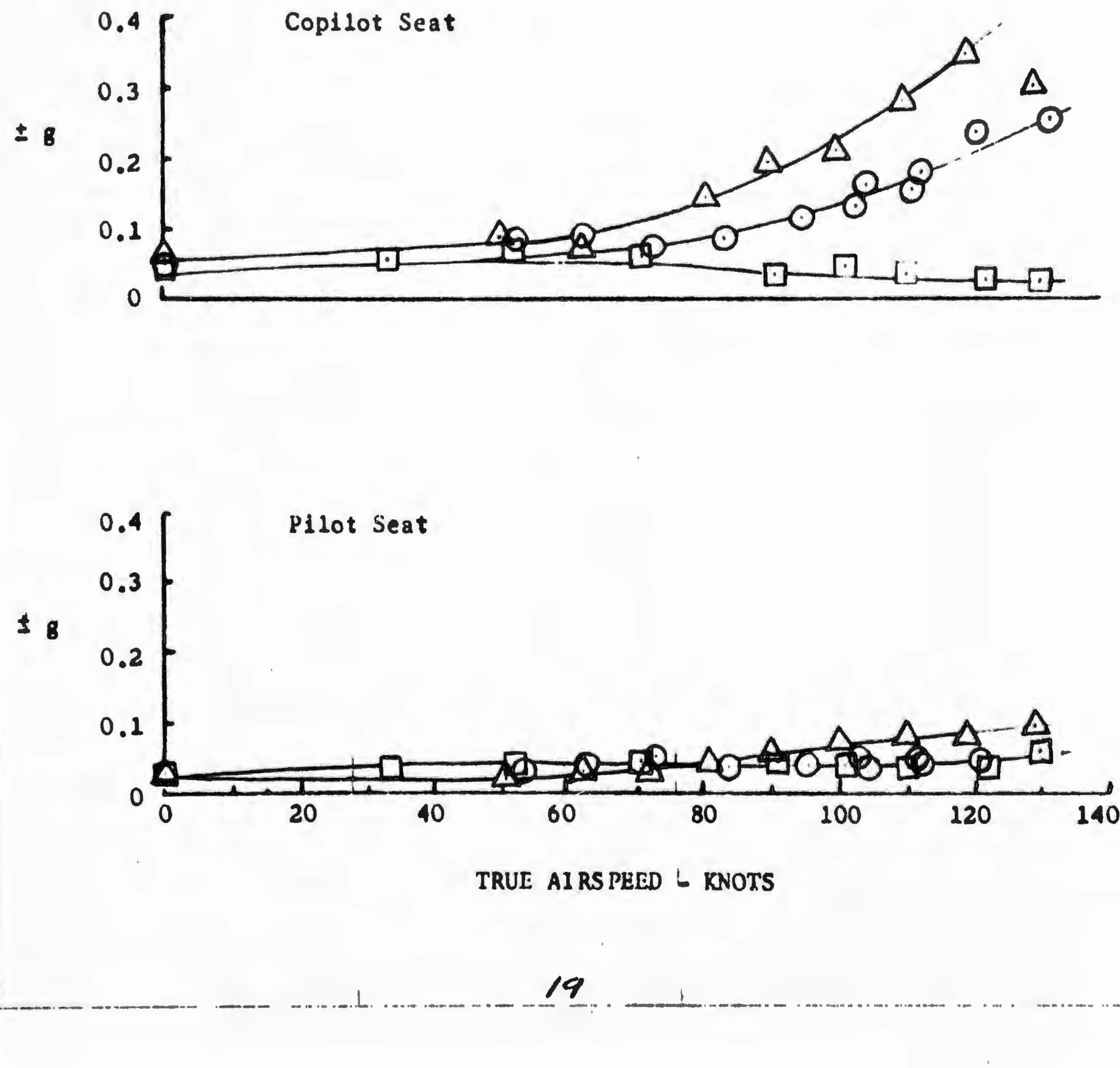


FIGURE 7-8
COMPARISON OF 2/REV VIBRATION LEVELS FOR FLEX BEAM
ROTOR WITH 27 INCH CHORD BLADES, THE UH-1B ROTOR AND THE MODEL 540 ROTOR

Flt. No.	Gross Wt.	Rotor	Helicopter
126	7000 Lb	UH-1B	204B S/N 2010
125	7000 Lb	Flex Beam	204B S/N 2010
5	7000 Lb	Model 540	UH-1B S/N 543



VIII

TAPERED TIP MAIN ROTOR BLADES

A. Summary

Main rotor blades with tapered tips (constant chord-tapered thickness) were designed, fabricated and flight tested. The initial flight tests were conducted using a company owned 204B helicopter. The results of these tests indicated that the performance was approximately the same as that of the UH-1B rotor within the airspeed range tested (130 knots). The oscillatory loads in the main rotor pitch links and yoke chord were lower than with the UH-1B rotor. The 4/rev, 5/rev and 6/rev vibration levels were higher than normal possibly due to the decrease in blade beam stiffness at the junction of the blade and tip sections. The joint stiffness was increased by bonding sheet metal across the joint and the blades were installed on the USA TRECOM high performance helicopter for evaluation at higher speeds. Flight tests with these blades on the high performance helicopter demonstrated a significant reduction in power required and control loads as compared with the UH-1B rotor. With the tapered tip blades the test vehicle was flown to a level flight true airspeed of 193 knots. The power required at 193 knots with the tapered tip blades was equivalent to the power required for the standard rotor at 182 knots. At 180 knots the taper tip blades required 200 HP less than the standard rotor, and also the oscillatory control load (pitch link) was approximately 30 per cent lower. For the same control load level the maximum speed of the test helicopter can be increased by 15 knots with the tapered tip blades; the tapered tip blades in reducing advancing blade Mach number effects provide a significant increase in speed with an attendant decrease in control loads.

B. Introduction

High tip speeds combined with high vehicle velocities cause power losses due to compressibility effects over large areas of the rotor disc. Significant power savings can be realized by reducing these losses. Main rotor blades with special wide chord tips were fabricated and installed on the high performance helicopter to determine if that type of tip section would prove beneficial in reducing compressibility losses. During tests with these blades weaving was encountered at 120 knots, and since the power required trend for these blades indicated that power would be higher than the standard blades throughout the flight regime, the tests were terminated. It was concluded from that program that further testing should be accomplished using tip configurations with a less abrupt transition in airfoil section. Accordingly, main rotor blades with tapered (constant chord) tips were designed, fabricated and flight tested. The flight test results with these blades are presented herein.

C. Description of Rotor Blades

Standard UH-1B rotor blades were modified by cutting the blade off at Station 211.2 and installing a 53-inch span tapered section. The tapered section had a constant taper from a NACA 0012 airfoil section at the junction to a 6 per cent thick airfoil section at the tip. The chord length of the tapered section was 21 inches constant, the same as the standard blade. Figure 8-1 shows the tapered section.

Solid metal machined spars were used for the leading and trailing edges of the tapered section. The inboard section of the leading edge spar fitted into the boxbeam of the blade and was fastened by screws that went through the boxbeam and threaded into the spar extension. The trailing edge spar extended into the blade shell and was fastened with through bolts.

D. Test Results

The initial evaluation of the tapered tip blades was conducted on a Bell-owned model 204B helicopter (S/N 1501). The test helicopter was in standard 44 foot diameter rotor configuration except for the main rotor hub which was for a 48 foot diameter rotor, and therefore, had a precone angle of 2-3/4°.

After completion of evaluation on the Model 204B helicopter the rotor was installed and tested on the USA TRECOM high performance helicopter. Instrumentation was installed on both test helicopters to measure shaft horsepower, control loads, vibrations and main rotor yoke and blade bending moments.

Evaluation flights on the 204B helicopter were conducted in February 1964. The rotor was installed and tested on the high performance helicopter from 28 April through 11 May 1964.

Ground Run - Ground run tests with the helicopter tied down were performed to track and balance the rotor and to observe its behavior. During the ground runs the cyclic and collective controls were operated to their maximum displacement within the limits of safe tiedown operation. The collective and cyclic control stick forces were satisfactory with and without hydraulic boost.

It was noted during the ground tests that the blade tip path plane as seen from the pilot's seat was very sharp and normal at low power, but that as the collective was increased to give engine powers greater than approximately 400 horsepower, the tip plane appeared to the pilot to lose definition (sharpness). There were no apparent changes in vibration, noise or control stick forces associated with the loss of the visual tip path. Tip weight and trim tab changes had no effect on the visual definition of the tip path.

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High speed motion pictures were taken and a stroboscopic light was used during a night ground run. No abnormal behavior was noted and it was seen that the rotor stayed in very good track at all azimuth positions and power settings. From these observations it was concluded that the rotor behavior was satisfactory and normal. The phenomenon causing the loss of tip path plane definition is still unexplained.

Flight Evaluation - Flight evaluations were conducted to obtain level flight performance, rotor and control load and vibration data. Additionally, hover (OGE) performance data were recorded with the rotor installed on the 204B helicopter.

The out-of-ground hover performance data are shown in Figure 8-2. These data are compared with the UH-1B Category II data and show that the hovering performance with the tapered tip and standard rotors is approximately the same.

Level flight performance data with the special rotor installed on the 204B helicopter are shown and compared with UH-1B Category II test data in Figure 8-3. A maximum advancing tip Mach number of .885 was attained during tests with this helicopter; however, the UH-1B data are for tip Mach numbers of less than .85. Since the UH-1B data are below the threshold Mach number effects become significant, the benefits of the tapered tip blades are not apparent.

With the tapered tip blades installed on the high performance compound helicopter, level flight performance data were obtained at true airspeeds up to 193 knots. These data and UH-1B rotor data are shown in Figure 8-4. At higher Mach numbers, above M = .87, the tapered tip blades showed an increasingly significant reduction in power required. At 193 knots the power required with the tapered tip blades was equivalent to the power required from the standard rotor at 182 knots. At 180 knots the tapered tip blades required 200 horsepower less than the standard rotor.

It should be noted that the data presented for the standard rotor are for 9725 pound GW/δ' as compared to the 10,058 pound GW/δ' for the tapered tip blades. Also the rotor speed is lower (314 versus 324) with the standard rotor. If the weight and rotor speed for the standard rotor are increased to the same values as the tapered tip blades, the power required would be increased and as a result the power reduction as shown in the figure may be considered conservative.

Oscillatory loads were measured in the main rotor pitch link, yoke, and blades. The oscillatory pitch link loads data are shown in Figure 8-5. When compared to the UH-1B rotor the tapered tip blade shows a significant decrease in the pitch link load for both the standard and high performance helicopters, particularly at the high speeds. At approximately 185 knots the load is reduced by about 40 per cent with the tapered tip blades. The load reduction is even more significant when the origin of these loads is considered.

Figure 8-6 shows the pitch link loads as a function of rotor azimuth for the high performance helicopter with standard blades at 186 knots. It is seen that the pitch link load is comprised of two major components, a large negative load at approximately $\psi = 120°$ ("A") and a large positive load at approximately $\psi = 270°$ ("B"). The large positive load component is associated with the reverse flow region of the retreating blade and builds up slowly with increasing air speed (see Section VI for additional discussion). The large negative load may be attributed to advancing blade Mach number effects. Figure 8-6b shows this load component for the standard and tapered tip blades plotted as a function of the advancing blade Mach number.

Below a Mach number of .87 this load component is non-existant, after Mach .87 it increases very rapidly such that it accounts for almost half of the total oscillatory load with the standard blades at a Mach number of approximately .95. With the tapered tip blades the rise is less rapid, and at a Mach number of approximately .95 this load component is approximately 70 per cent lower than with standard blades.

The main rotor yoke beam and chord oscillatory loads are shown in Figure 8-7. The loads for the standard rotor blades are shown for comparison. The beam loads are about the same for the standard and tapered tip blades. The chord loads with the tapered tip blades on the 204B are lower than with the standard rotor. With the tapered tip blades on the high performance helicopter the chord loads are about the same as with the standard rotor.

Vibrations - The pilot and copilot vibration levels were measured for all flights. The overall and 2/rev vibration levels were comparable to those with the standard rotor installed. At high gross weights, above 8000 pounds, the higher frequency (4, 5 and 6/rev) vibration levels were greater than those measured with the standard rotor. The high frequency vibration is believed to have been caused by the relatively low beam stiffness of the blades at the tapered tip attachment.

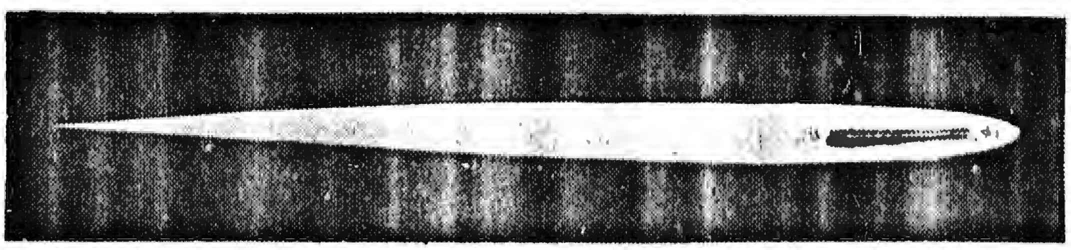
Conclusions

As a result of these tests, it is concluded that the performance of the tapered tip rotor blades is comparable to that of the standard blades in the low speed range, and that the tapered tip blades show significant reductions in the power requirements and control loads at the higher speeds. The speed, range, and production of the high performance helicopter can be significantly improved by installation of tapered tip main rotor blades.

FIGURE 8-1

TAPER TIP BLADE INSTALLATION AND
COMPARISON WITH STANDARD BLADE

Taper Tip
Blade Installation



Tapered
Tip

Standard
Tip

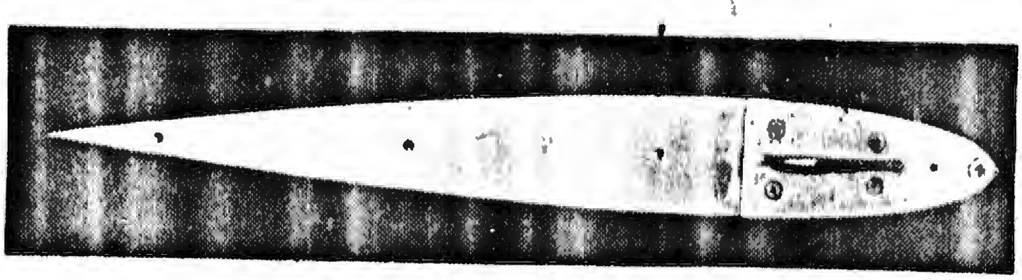


FIGURE 8-2

HOVERING PERFORMANCE TAPERED TIP MAIN ROTOR BLADES

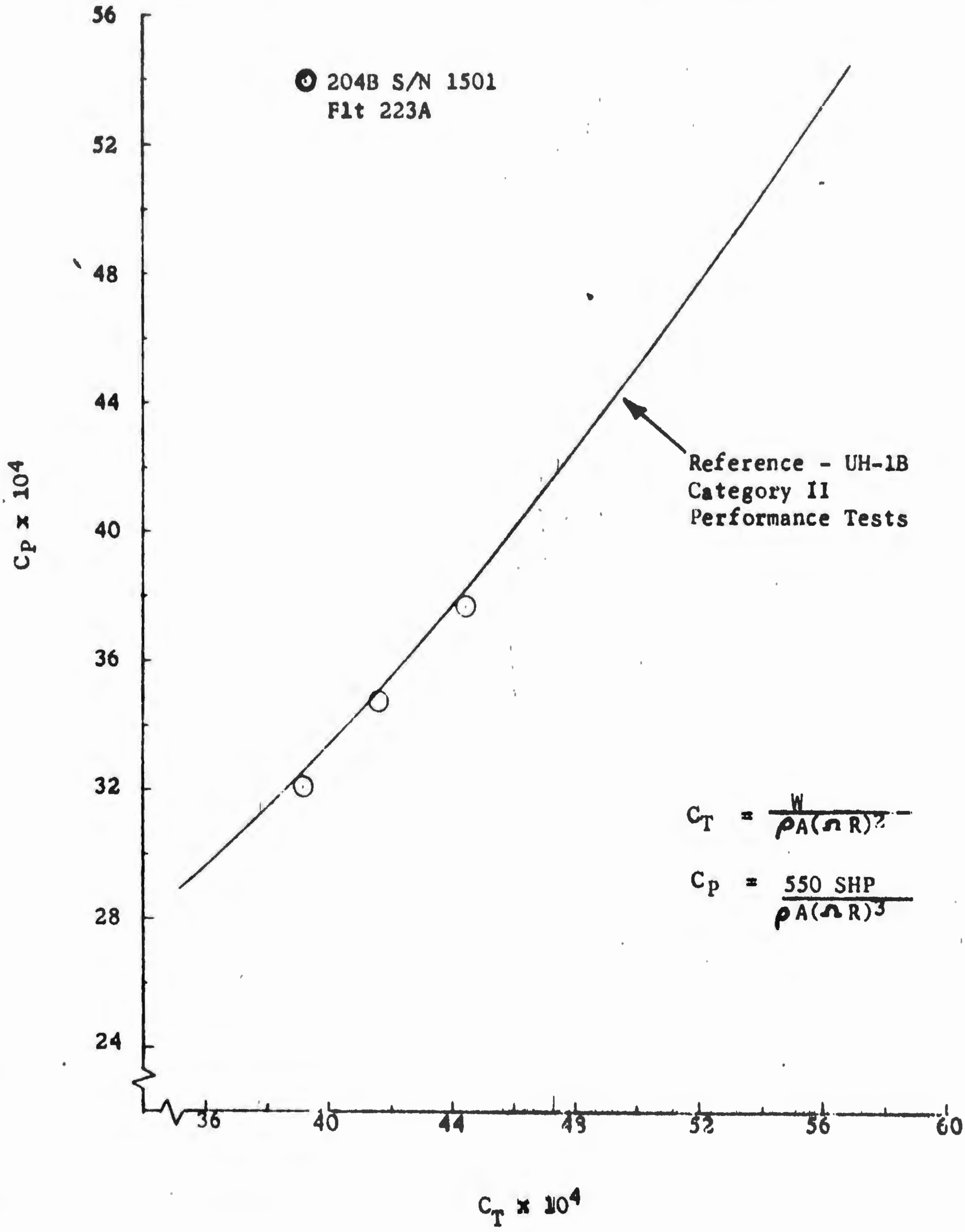


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FIGURE 8-3

POWER REQUIRED VS AIRSPEED FOR TAPERED TIP MAIN ROTOR BLADES INSTALLED ON 204B S/N 1501

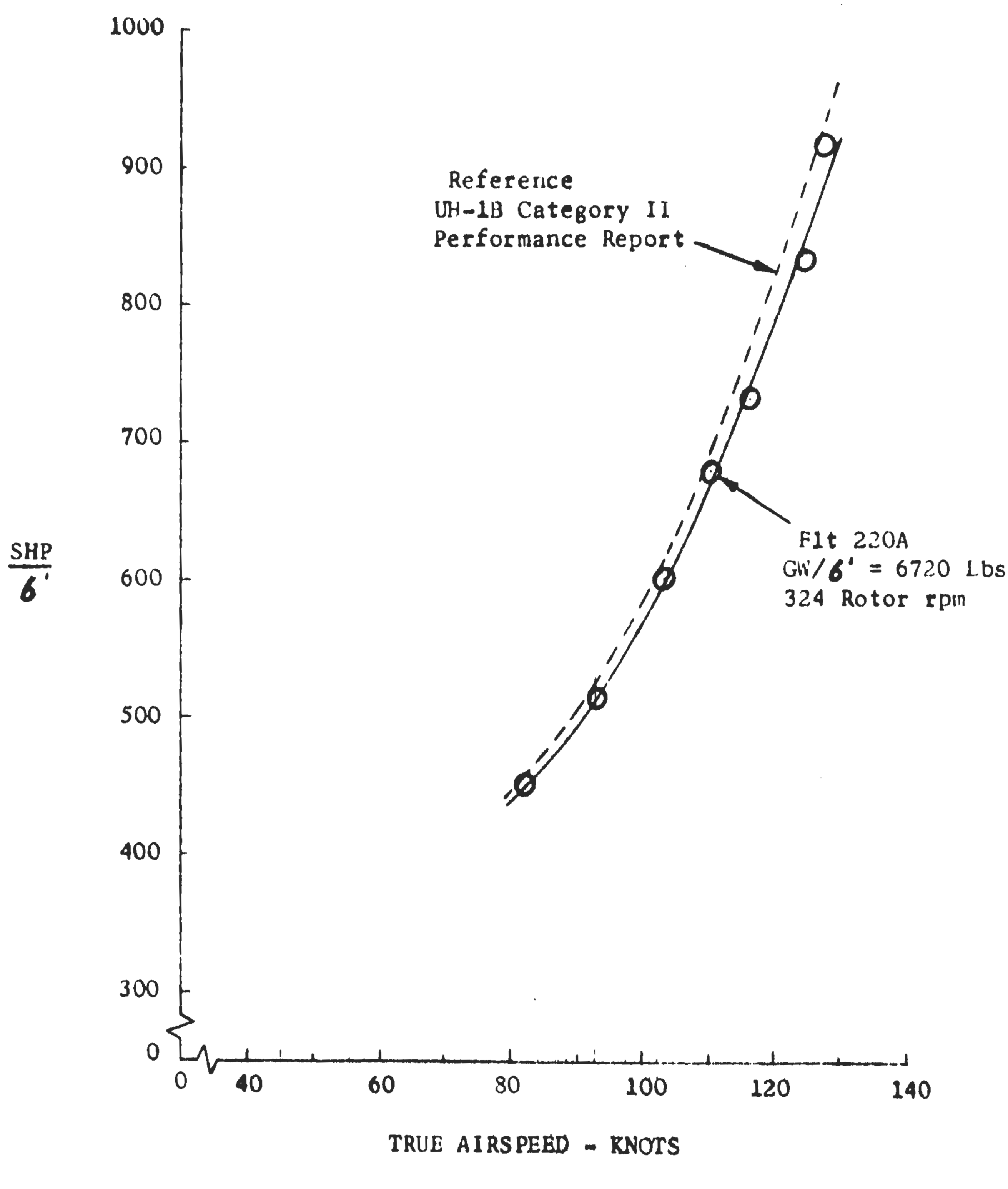


FIGURE 8-4

COMPARISON OF POWER REQUIRED VS AIRSPEED FOR THE TAPERED TIP AND STANDARD MAIN ROTOR BLADES INSTALLED ON THE HIGH-PERFORMANCE HELICOPTER

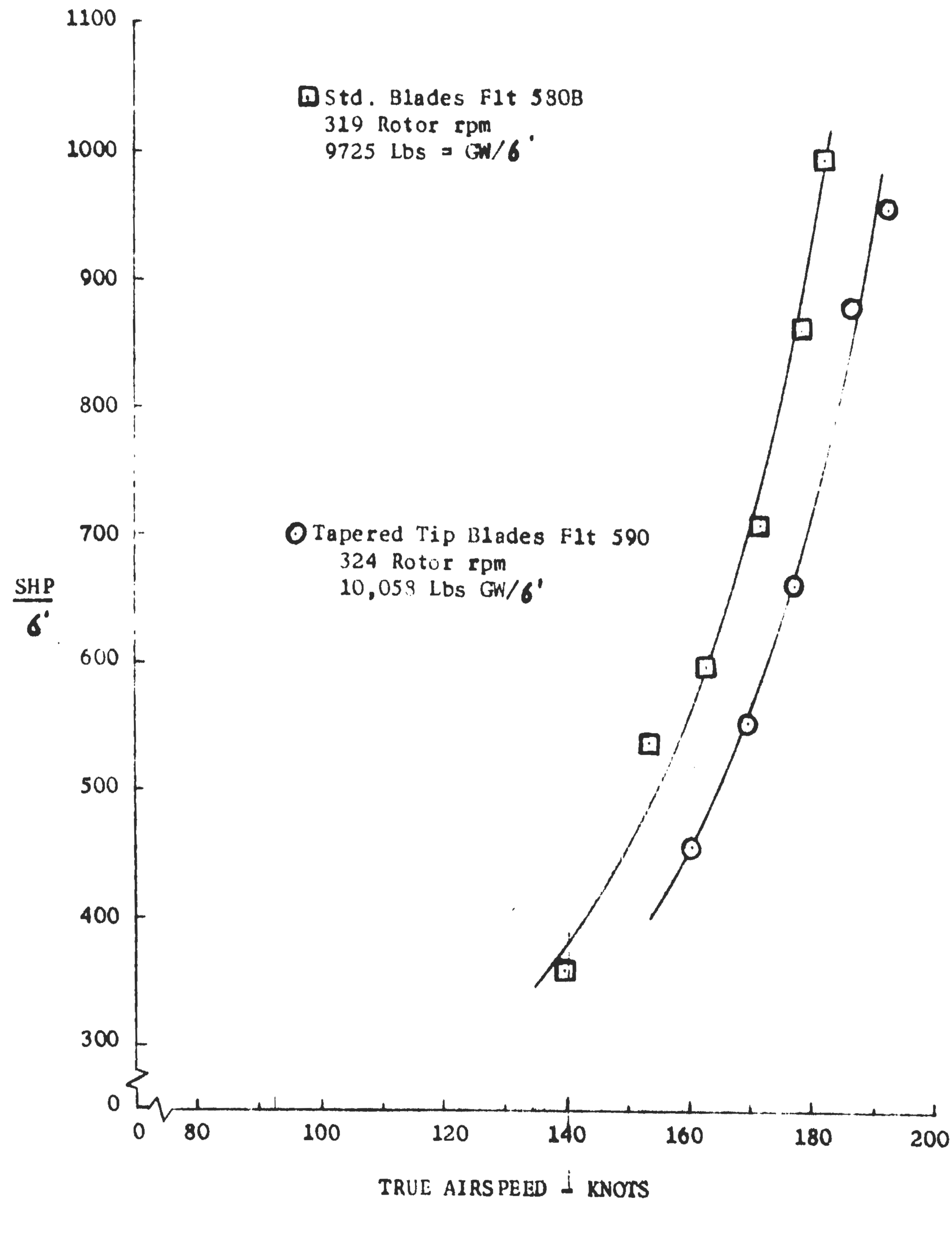
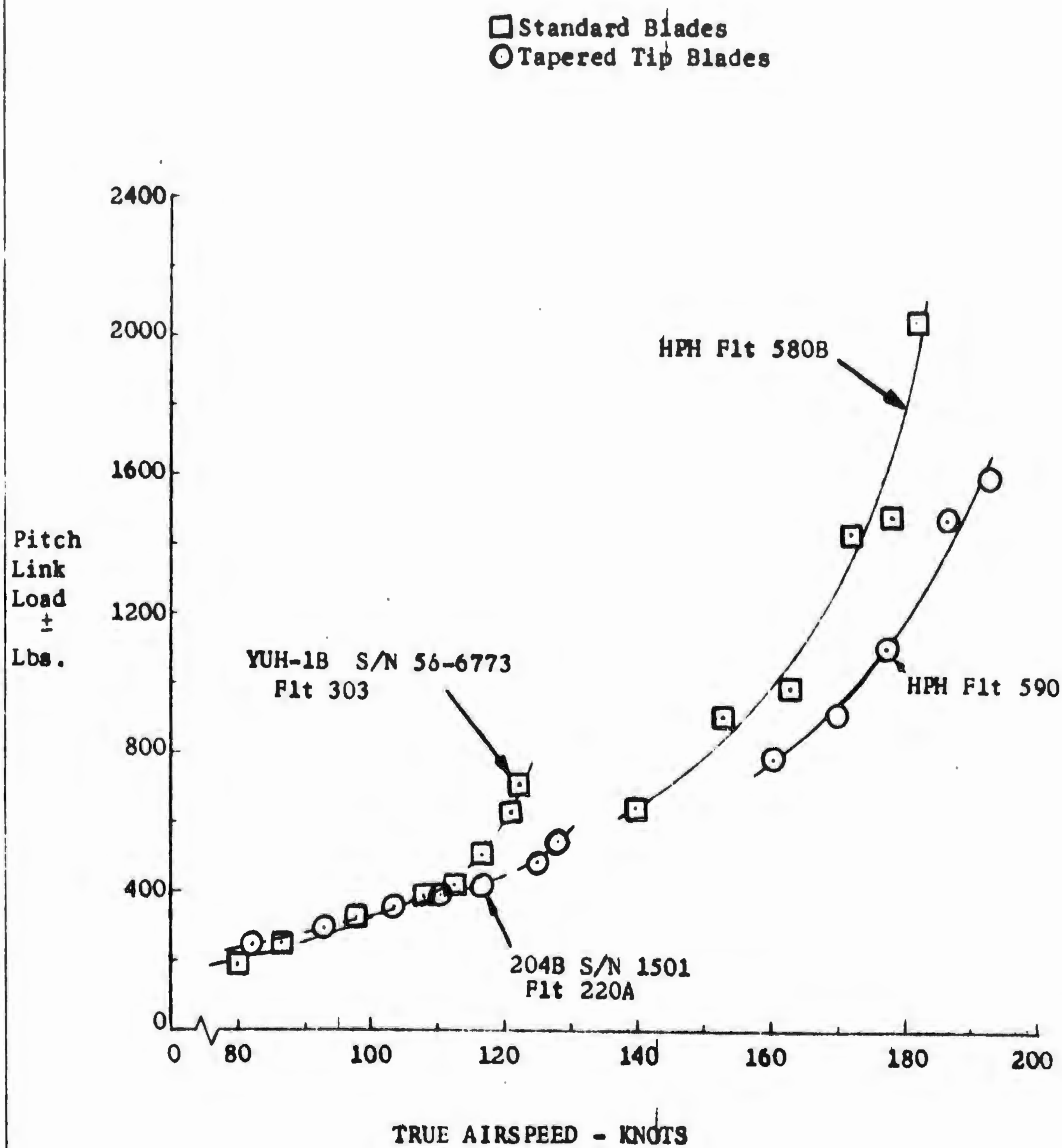


FIGURE 8-5

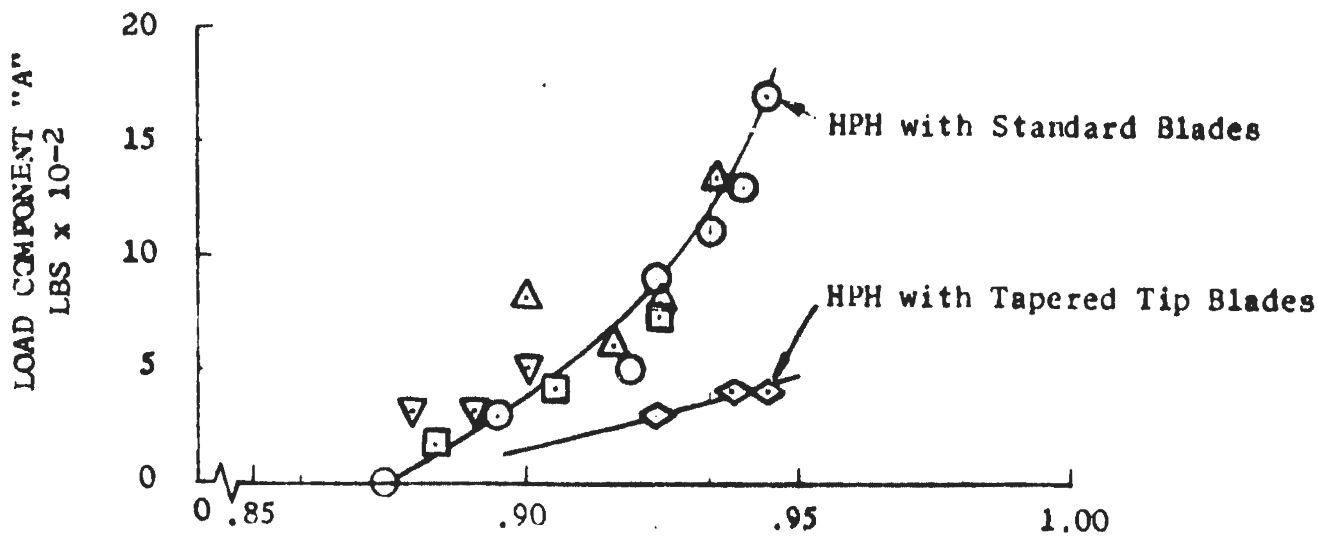
COMPARISON OF PITCH LINK OSCILLATORY LOAD VS AIRSPEED FOR STANDARD AND TAPERED TIP MAIN ROTOR BLADES



7672 55425

FIGURE 8-6

COMPARISON OF ADVANCING BLADE LOAD COMPONENT VS TIP MACH NUMBER FOR STANDARD AND TAPERED TIP BLADES



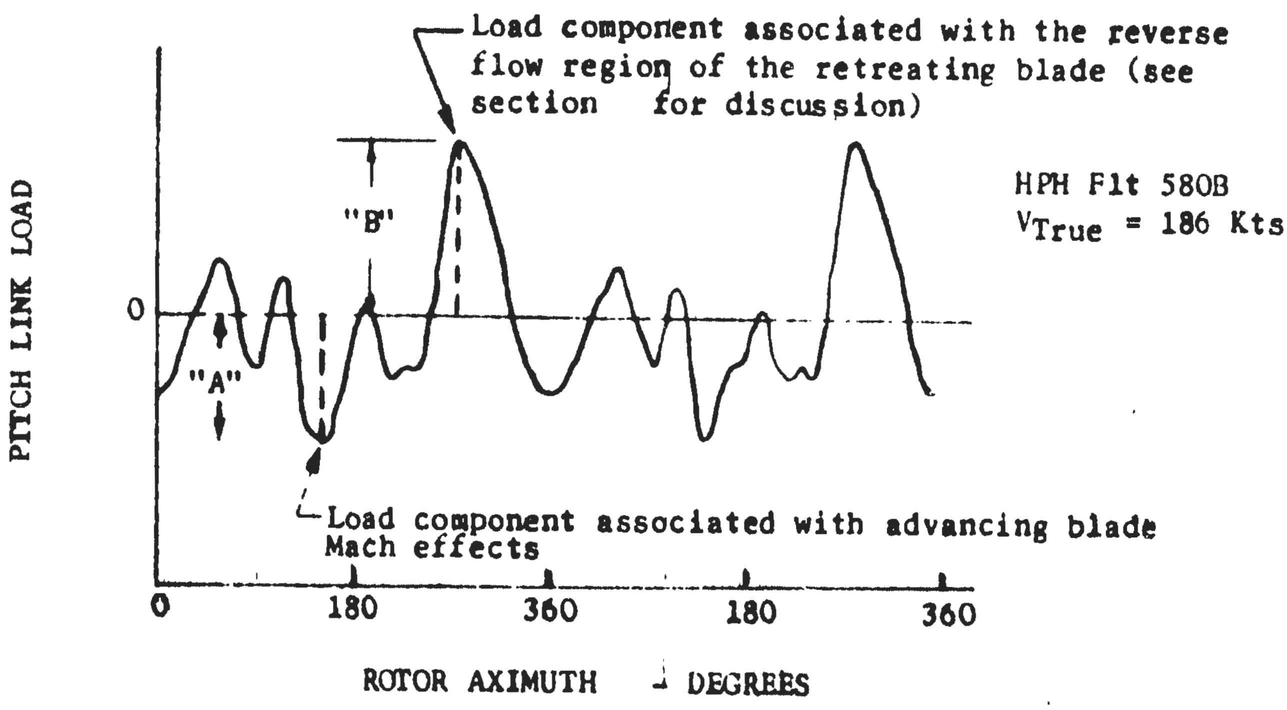


FIGURE 8-7

COMPARISON OF MAIN ROTOR YOKE OSCILLATORY BENDING MOMENTS FOR STANDARD AND TAPERED TIP MAIN ROTOR BLADES