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**AN EXPERIMENTAL INVESTIGATION
OF A SECOND HARMONIC FEATHERING DEVICE
ON THE UH-1A HELICOPTER**

Task 1D121401A14604
(Formerly Task 9R38-01-019-04)

Contract DA-44-177-TC-806

June 1963

prepared by:

BELL HELICOPTER COMPANY
Fort Worth, Texas



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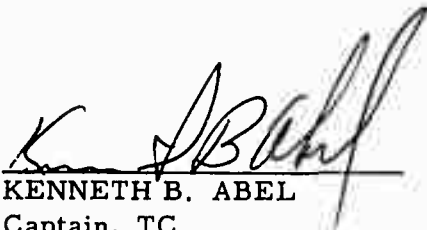
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U. S. ARMY TRANSPORTATION RESEARCH COMMAND

Fort Eustis, Virginia

This report has been reviewed by the U. S. Army Transportation Research Command and is considered to be technically sound. The report is published for the exchange of information and the stimulation of ideas.

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Task 1D121401A14604
(Formerly Task 9R38-01-019-04)

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June, 1963

AN EXPERIMENTAL INVESTIGATION
OF A SECOND HARMONIC FEATHERING DEVICE
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Bell Helicopter Report 537-099-001

Prepared by



for

U. S. ARMY TRANSPORTATION RESEARCH COMMAND

Fort Eustis, Virginia

FOREWORD

This report outlines the results of an experimental research program to investigate a second harmonic feathering mechanism as a means to reduce the fuselage vibrations and/or oscillatory rotor loads. The program was conducted by Bell Helicopter Company under U. S. Army Transportation Research Command Contract DA 44-177-TC-806 (Reference 1), and was carried out under the technical cognizance of Mr. John E. Yeates, USATRECOM, Fort Eustis, Virginia.

The flight tests were conducted with a UH-1A helicopter modified to accommodate the experimental Second Harmonic Control Mechanism and instrumentation. Following a suggestion from USATRECOM personnel, USATRECOM-furnished UH-1A rotor blades, instrumented for air load measurements, were used.

Personnel associated with this program were Mr. R. K. Wernicke, Research Program Project Engineer, and Messrs. E. C. Darlington, J. A. DeTore, J. M. Drees, K. W. Froberg, C. L. Livingston, and R. R. Lynn. Mr. Drees is the inventor of the mechanism used in this program to produce second harmonic main rotor blade feathering.

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SYMBOLS

G	Acceleration due to gravity; used as a unit for vibration measurement
V_T	True airspeed
T	Thrust (pounds) - aerodynamic force normal to the rotor blade mean chord line
Δ	Change resulting from actuation of the second harmonic control mechanism; all other conditions held constant
σ'	Density ratio, referred to sea level standard day
ψ	SHC phase angle (degrees). Azimuth angle of instrumented blade at which second harmonic feathering reaches a minimum, measured counterclockwise from the aft position when looking down on the rotor disc.
Ψ	Azimuth angle (degrees) of instrumented blade, measured counterclockwise from the aft position

I. SUMMARY

This report presents results of an experimental investigation of the effects of a second harmonic feathering device on a two-bladed, semi-rigid rotor helicopter. Flight tests were conducted with an Army UH-1A helicopter modified to incorporate a Bell-designed experimental Second Harmonic Control (SHC) Mechanism.

Prior theoretical studies indicated that second harmonic feathering would be beneficial in delaying retreating blade stall and in reducing oscillatory loads and vertical vibrations in the cockpit. Air load data of this program are presented which show that pulsations in rotor thrust can be reduced with second harmonic feathering. Reductions in cockpit vertical vibrations and structural loads are shown for certain settings of the SHC; however, simultaneous reduction of vibrations measured at different locations in the fuselage and of loads in the rotor blades and the control system was not realized.

A complete experimental evaluation of second harmonic feathering for the mitigation of retreating blade stall was hampered by unexplained, abnormally high vibrations and control loads encountered at high speeds. Analyses of such a system show that second harmonic feathering can reduce stall and the accompanying high drag on the retreating blade; however, increased drag from compressibility effects occurs at other points around the rotor azimuth, thus offsetting the effects from reducing retreating blade stall.

The evaluation of the data of this program suggests that high control loads reduce the benefits of SHC. A need for further development in computing control loads is indicated.

The data in this report pertain only to the specific parameters of the UH-1A test helicopter. The results indicate that the response of other helicopters to second harmonic feathering may be influenced considerably by rotor parameters such as blade loading and tip speed, cabin arrangement, and fuselage dynamic characteristics.

II. CONCLUSIONS

Following are five principal conclusions that can be drawn from the discussions of the test results of the SHC system:

A. GENERAL

The over-all effects of SHC are very complex in that the results on vibrations and oscillatory loads differ from one location to another. For optimum results, the phasing and amplitude of the SHC input have to be controlled continuously with forward speed.

B. AERODYNAMIC EFFECTS

The change in aerodynamic rotor thrust due to SHC is of the predicted order of magnitude and in phase with the second harmonic blade feathering. However, the change in rotor thrust with rotor azimuth, due to second harmonic feathering, varies along the blade span.

C. DYNAMIC EFFECTS

The SHC, basically, performs as expected in creating changes in fuselage vibrations and in the dynamic rotor loads measured in the blades and in the lift link (supporting link from base of transmission to fuselage). However, it was also found to create a number of unpredicted and important side effects. The principal side effects were found to be in the large oscillatory control and blade loads which occurred at SHC settings where air load thrust pulsations were found to be a minimum.

D. EFFECTS ON BLADE STALL AND COMPRESSIBILITY

The evaluation of SHC as a means for stall reduction was inconclusive. Analytical work indicates that the test rotor did not benefit from SHC because of compressibility effects outside the stalled region. It is possible that the situation will be different for the UH-1B and other helicopters since the phenomenon depends on detailed aerodynamic and dynamic characteristics.

E. BENEFICIAL EFFECTS

At 80 knots, the vertical vibration level at the c.g. was reduced 50 per cent when the mechanism was phased at 120 degrees; oscillatory control loads and oscillatory blade bending loads were also reduced. But the vertical vibration level of the cabin was increased at this phasing, thus adversely affecting the pilot's comfort.

III. RECOMMENDATIONS

Based on the foregoing conclusions, the following subjects are recommended for further study:

A. SHC OF A PORTION OF THE BLADE

A study of SHC applied to only a portion of the blade is recommended (reference Conclusion B).

B. METHODS FOR PREDICTING CONTROL LOADS

Further development of methods to theoretically predict dynamic control loads and their effect on fuselage vibrations are recommended (reference Conclusion C).

C. REDUCTION OF RETREATING BLADE STALL

It is recommended that second harmonic feathering be considered a potential method for reducing retreating blade stall of rotors with low tip speeds (reference Conclusion D).

D. SHC FOR OTHER HELICOPTERS

It is recommended that the application of SHC to a helicopter with its cabin located under the mast be considered as a means for simultaneously reducing vertical vibrations and oscillatory loads (reference Conclusion E).

IV. INTRODUCTION

Second harmonic blade feathering or second harmonic control (SHC) has received considerable attention in the past. Early investigations are reported in Reference 2, where SHC was used as a means for testing rotors for fatigue life. The application of SHC for use in helicopters was proposed by Stewart (References 3 and 4) for use in preventing stall on the retreating blade at high forward speeds. More recent theoretical studies (References 5, 6, 7, and 8) reveal that for a two-bladed rotor operating below the stall limit, twice per revolution (2/rev) rotor thrust pulsations can be minimized with second harmonic feathering. Thus, it could be expected that oscillatory stresses and fuselage vibration would be minimized. The several theoretical treatments of second harmonic feathering in previous investigations are discussed in more detail in the following section entitled, "Theoretical Background."

In spite of the potential benefits from SHC, there have been no previously reported in-flight evaluations of such a mechanism. This is probably because of the mechanical complexity of the system which was believed to require adjustable phasing and amplitude control to obtain optimum results under all flight conditions.

The development of a relatively simple SHC mechanism by this contractor made possible the experimental evaluation presented herein. This device permitted in-flight amplitude control from zero to ± 2 degrees second harmonic blade feathering. In-flight control of phasing was also possible. The effective azimuth phase angle range encompassed a full 360 degrees.

The use of USATRECOM-furnished UH-1A rotor blades, instrumented for air load measurements, offered a unique opportunity for studying the aerodynamic effects of second harmonic feathering. The use of the UH-1A instrumented 15-inch chord blades instead of the UH-1B 21-inch chord blades, however, is believed to have resulted in a less conclusive demonstration of the benefits of the principle of second harmonic feathering. This is due to the lower stall and compressibility limits of the UH-1A blades as compared to the higher solidity UH-1B rotor blades which are operated at the same r.p.m. as the UH-1A blades.

The program illustrates the compelling necessity to investigate experimentally devices, theories, etc., which appear to be promising analytically. It is hoped that this investigation will provide insight into the complex phenomena associated with second harmonic feathering so that future work may lead to the realization of the potential benefits of such a system.

V. THEORETICAL BACKGROUND

The first thorough theoretical study of the effect of SHC on helicopter rotors was made by Stewart (References 3 and 4). From these studies it was concluded that SHC could be effective in delaying retreating blade stall. The effect of rotor inertia on second harmonic blade flapping will reduce the influence of the SHC. However, for high inertia rotors such as the UH-1 series this reduction is not significant.

The Bell theoretical studies (References 6, 7, and 8) were focused on the possibility of reducing the 2/rev vertical rotor forces. The computing program used was a variant of the digital analysis described in Reference 9 in which second harmonic feathering was introduced in the blade pitch equation. Second harmonic flapping was taken into account by following the iterative process between the aerodynamic and dynamic parts of the analysis.

The results of the studies by Bell Helicopter Company pertain to the effect of SHC on the UH-1B helicopter. Figure 1 shows the predicted oscillatory 2/rev total shear loads as measured in the lift link. The lift link is the connective member between the main rotor transmission and the fuselage. Figure 2 shows the predicted reduction of the 2/rev hub spindle moments in the beamwise direction as a function of speed and SHC amplitude and Figure 3 gives the 2/rev cabin vibration. It can be seen from these figures that at 120 knots an SHC amplitude of about ± 2 degrees, phased at 45 degrees azimuth setting, would produce optimum results. (The azimuth setting is defined as the blade azimuth where the blade angle is minimum.) From Figure 1 it is also obvious that an incorrect setting of the SHC phasing would cause a considerable increase in oscillatory loads over the base line condition.

It was concluded from these studies that the maximum capability of the SHC mechanism for the UH-1 helicopter should be ± 2 degrees of second harmonic feathering.

At a later stage an aerodynamic computer program (Bell IBM Program F35) was also used to calculate the effects of compressibility in connection with SHC. That program is a further development of NASA procedures for calculating aerodynamic characteristics of lifting rotors (Reference 10). The results of these calculations are discussed in this report in connection with Figure 26.

VI. DESCRIPTION OF EQUIPMENT

A. TEST HELICOPTER

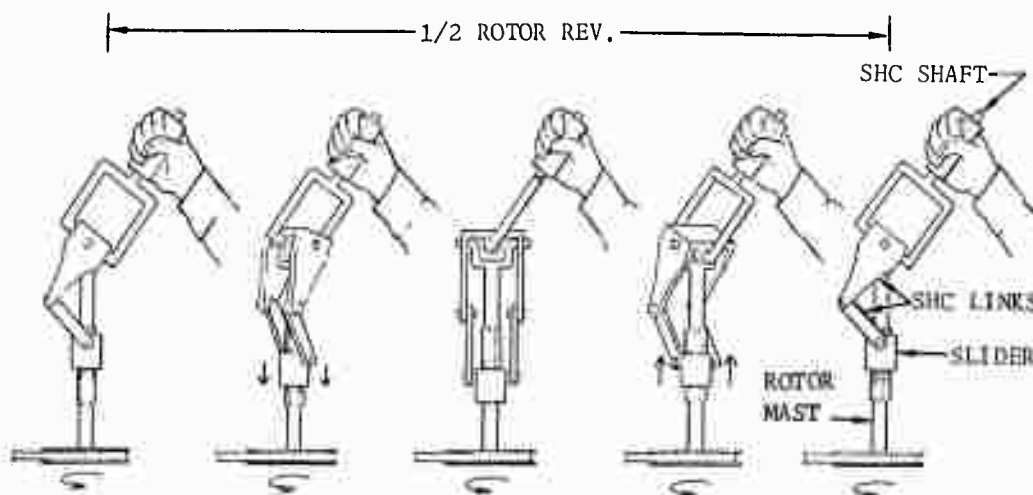
The helicopter used during this program was the two-bladed, semirigid rotor United States Army UH-1A Iroquois, Serial Number 59-1616. A photograph of the test vehicle with the second harmonic control mechanism installed is presented in Figure 4. A three-view drawing of the basic helicopter with dimensional data is given in Figure 5. The standard UH-1A rotating controls were replaced with UH-1B components because of an anticipated increase in controls loads. The stabilizer bar was removed from the test helicopter so that the output from the SHC mechanism could be introduced through the stabilizer bar mixing levers. Previous tests (Reference 11) indicate that elimination of the stabilizer bar should not affect rotor behavior in steady state flight.

Figure 6 presents a close-up photograph of the pylon assembly showing the rotor head, the controls of the SHC mechanism, a small slip ring on top of the rotor mast, and a slip ring on the swash plate. Figure 7 presents a sketch of the pylon assembly wherein parts are coded for identification.

B. SECOND HARMONIC CONTROL MECHANISM

1. Description of the Principle

The SHC mechanism was designed to mix into the normal control system of the helicopter, a twice per main rotor revolution oscillation, which could be controlled in amplitude and phasing with respect to blade azimuth. This was accomplished with a mechanism which converts the rotary motion from the rotor mast into a 2/rev vertical slider motion. The operation of the SHC mechanism can be explained with the sketch below.



The SHC shaft is held in space by a swash plate (not shown) which is controlled with a nonrotating standpipe through the hollow rotor mast. The two links between the SHC shaft and the slider "stretch" at twice per revolution as the slider rotates once per revolution around the rotor mast axis.

Amplitude control of the SHC was obtained by adjusting the degree of tilt of the SHC shaft. Phase control was obtained by selecting the azimuth direction in which the shaft tilts.

The position of the SHC mechanism on top of the mast was selected for the test installation to avoid mechanical interference problems; however, it is feasible and advantageous from an aerodynamic standpoint, to locate the mechanism beneath the rotor mast.

2. Description of the Actual Mechanism

Figures 8a and 8b present two views of the SHC mechanism installed on the helicopter and adjusted so that no second harmonic feathering is produced. Figures 8c and 8d present two views of the installed SHC mechanism with the actuator arm in the position at which maximum second harmonic amplitude is produced. The SHC azimuth phase angle, defined as the blade azimuth position at which minimum SHC feathering occurs, is 90 degrees in these views. As in Figures 8a and 8b, the rotor blades are oriented perpendicular to the nonrotating actuator arm. In this orientation, the blade pitch increase is a maximum for both blades. In Figures 8e and 8f, the actuator arm has been rotated to be in line with the blades. (The actuator arm is still depressed for maximum feathering.) This returns the blades to minimum blade pitch. Figures 8e and 8f present views of the SHC mechanism phased at 180 and zero degrees, respectively.

3. Deviations from a Perfect Second Harmonic Motion

Calculations of the kinematics of the system (Reference 8) show that the theoretical error of the SHC motion compared with a perfect 2/rev sine wave is of the order of a few per cent. Measurements taken during the subject program revealed that the mechanism deviated from the theoretical motion, due to some looseness in the SHC mechanism caused by machine tolerance. This is illustrated by Figure 9 which shows a small amount of slider motion detectable when the SHC mechanism was placed in the "off" position. The second harmonic feathering due to the looseness was found to be small compared to the slider position trace at maximum second harmonic feathering. This is illustrated on Figure 9.

Figure 10 shows that the SHC feathering amplitude measured in flight differs from measurements taken with the rotor stopped. The reasons for this difference are believed to be, (a) dynamic deflections of the control system due to oscillatory loads, (b) tolerances in bearings, (c) possible cocking of the slider. In all data presentations throughout this report, the static measurements are used as a basis for the blade feathering angle.

It is felt that this is more realistic, due to the probability that at least part of the difference between the static and dynamic measurements is caused by a small cyclic (rather than collective) input resulting from the looseness in the SHC mechanism.

4. Controls for Amplitude and Phasing

The controls for the second harmonic feathering mechanism were mounted in the cabin. These controls were connected with flexible shafting leading to levers underneath the hollow rotor mast which controlled the orientation and tilt of the SHC swash plate on top of the mast. Figure 11 shows the azimuth and amplitude control wheels, operated at the Flight Engineer's Station in the cabin. Azimuth control through a 45-degree phase angle range was possible in flight. Ground adjustment was used to reposition the mechanism for various ranges of phasing.

C. INSTRUMENTATION

1. General

Instrumentation was provided to evaluate the effects of the SHC and to assure that the tests were within load level limits defined by prior stress analysis. All of the critical parts in the SHC system were instrumented; also, loads were measured in the blades and control system. Accelerometers were installed in the cabin and near the center of gravity. Position indicators recorded controls, blade feathering, pylon positions, etc. A detailed list of the instrumentation used during the flight test program is given in Table 1.

2. Air Load Measuring Blades

During Phase II the differential pressures on a blade were measured at six radial blade stations. A detailed description of the air load measurement is given in Reference 11. The locations of the pressure transducers used during the subject program are given in Table 1.

3. Slip Rings

Two slip rings were used to transmit the data from the rotating system to recording equipment in the cabin (Figure 7). The 15-channel slip ring on top of the mast is a Bell-designed unit normally used for tail rotor tests. The lower swash-plate-mounted, 11-channel slip ring was designed and constructed for this program. Both slip rings were manufactured by the Instrument Engineering Company, Austin, Texas.

4. Recording Equipment

All data were recorded on Consolidated Electrodynamic Corporation oscillographs, Model 5-114-P3-18. A photo panel was used to record data for performance measurements. Table 1 lists the instruments installed in the photo panel.

VII. DATA REDUCTION PROCEDURES

The data obtained from the oscillograph records were read in three ways:

- (1) Peak to peak readings were taken for blade chord and beam loads and loads in the control system to insure that acceptable stress levels were not exceeded.
- (2) Vibrations in the cabin and at the c.g. and the lift link loads were analyzed to obtain the 2/rev component to show the fuselage response and the dynamic vertical rotor forces as a result of the 2/rev feathering.
- (3) Pressure transducer trace readings were made for 12 and 24 increments per rotor revolution for numerical integration along the blade span.

Based on the accuracy of similar studies the data presented in this report are believed to be accurate within 10 per cent.

The method for numerical integrations, developed for the Air Load Measurement Program (Reference 11) on the same helicopter and related instrumentation, involved the use of empirically determined weighting factors to calculate thrust per inch of blade at each of the instrumented stations along the span. This numerical integration method used from 5 to 10 pressure transducers distributed chordwise at each span station. In this program, however, fewer pressure transducers were monitored at each span station. Therefore, it was necessary to revise the method for calculating the thrust per inch. Assuming that pressure variations at a single pickup point might be directly representative of the thrust per inch variation at its span station, conversion factors were calculated based on data from the earlier investigation to convert differential pressure at a specific point to thrust per inch.

The validity of this approach is illustrated by three examples shown in Figure 12. The thrust per inch calculated with the numerical integration method is compared with thrust per inch determined by using a conversion factor and differential pressure at a single point. The figures show that a single pickup point can be used to determine blade loading at 40 and 55 per cent radius, but air loads for stations outboard of the 55 per cent radius cannot be represented by a single pickup. It was found that if the blade loadings determined separately with each of several pickups at one blade station were averaged, the average correlated satisfactorily with blade loadings determined with the numerical integration method. An example of this is shown by Figure 13. The conversion factors and location of each transducer used are shown in Table 2.

The thrust per blade was found by a numerical integration of the thrust per inch as determined at the 40, 75, 85, 90, and 95 per cent radial stations. This method is identical to that developed in the previous investigation (Reference 11). From the thrust per blade, the thrust per rotor can be calculated as well as the change of thrust due to SHC feathering.

VIII. FLIGHT TEST PROGRAM

The flight test program included:

- (1) Preliminary Ground Runs - Tiedown tests were made to check out rotor balance and track, and operation of the SHC mechanism.
- (2) Phase I - Phase I flights were conducted with standard strain gaged UH-1A blades. Loads were measured throughout the flight regime.
- (3) Phase II - During Phase II, flight tests were conducted with the air load instrumented blades. Data were taken for selected positions of the SHC at speeds from hovering to 100 knots.

For all flight regimes except hovering, a maximum second harmonic feathering of ± 1.1 degrees was established based on control loads obtained during preliminary tiedown tests. In some instances, as indicated in Section IX, further reduction of the amplitude was dictated by the test results. In hovering, a maximum feathering of ± 1.9 degrees was tested and was limited only by the mechanical stops of the SHC device. The entire range of SHC phasing was most thoroughly investigated at 80 knots indicated airspeed.

The test helicopter gross weight, corrected to standard sea level conditions (GW/ σ '), varied between 6317 and 7206 pounds. The speed range investigated was from hovering to 100 knots indicated airspeed (approximately 107 knots true airspeed). A list of Phase I and Phase II flight conditions is given in Table 3.

Flights made during Phase I were of an exploratory nature, the purpose of which was to determine gross effects from second harmonic feathering. To establish the gross effects of the device, flights were conducted throughout the day. Phase II flights were made in calm air in the early morning. Data from both Phase I and Phase II are presented in the appendix. Although there is some scatter in Phase I data, the results are considered adequate to mix with Phase II to provide a complete picture.

IX. DISCUSSION AND EVALUATION OF FLIGHT TEST RESULTS

A. GENERAL

The flight test results presented in this section are from both the Phase I and Phase II flights. All significant Phase I and Phase II data are given in the appendix. Summary plots of the data in the appendix, as well as typical examples, are presented herein to illustrate the principal results. An attempt is made to analyze the data and to explain the results. As will be seen, the effects of second harmonic feathering are very complex. Interpretation of the results is, therefore, not a simple matter. The analysis is further complicated by considerable scatter noted in the flight test data. In addition, high control loads restricted the measurement of air loads for certain flight conditions.

B. TEST OBJECTIVE

The purposes of the tests were to determine experimentally the characteristics of an SHC mechanism for the prevention of retreating blade stall and reduction of rotor thrust pulsations. A brief explanation of the function of the mechanism in each of these applications is given below.

1. Delay or Reduction of Retreating Blade Stall

It was expected that stall occurring on the retreating blade at high speed could be eliminated by producing a 2/rev feathering that reduces blade pitch when the blade is normal to the flight path and increases pitch when the blade is aligned fore and aft. In doing this, the mean thrust of the rotor is maintained, but the thrust on the fore and aft portions of the rotor disc is higher, thus reducing thrust requirements on the retreating blade.

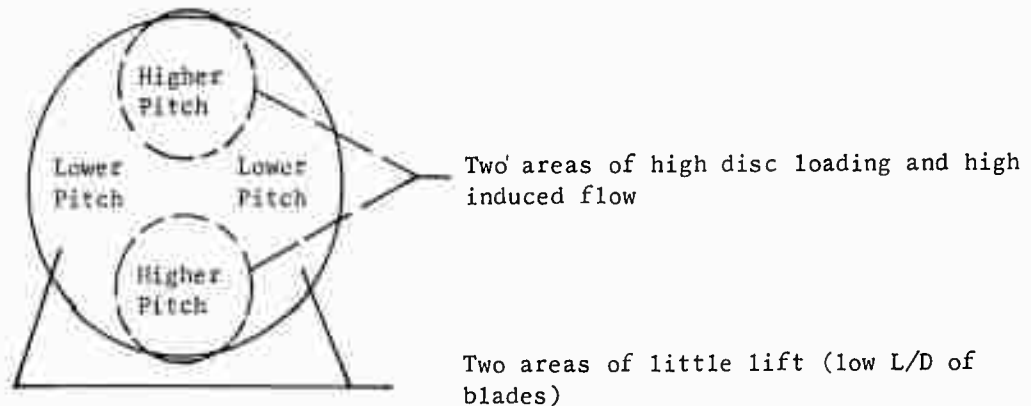
2. Reduction of Thrust Pulsations

Two-bladed rotors produce 2/rev thrust pulsations that increase with forward speed. These thrust pulsations produce undesirable oscillatory loads in the rotor and control systems and 2/rev vertical vibrations in the cockpit. These thrust pulsations are shown, in Reference 11, to peak when the blades are near the zero and 180-degree azimuth positions. It has been theorized that 2/rev feathering, which decreases pitch when the blades are fore and aft and increases pitch when the blades are laterally disposed, will smooth out the rotor thrust. It is realized that this phasing will produce higher angles of attack on the retreating blade, causing stall to occur at lower airspeeds.

C. HOVERING

All the hovering data were obtained in ground effect for maximum safety. These data provided a check on control loads and on the general effectiveness of the system. In this flight condition, only the amplitude of the SHC is of interest. The azimuth phase angle was found, as expected, to have no significance in a zero relative wind. Air load data taken during hovering are presented in Figure 14, showing the total thrust of the rotor and the SHC motion of the blades as a function of rotor azimuth. It is shown that there is considerable thrust pulsation created by second harmonic feathering. The air loads are in phase with the feathering, and the vertical force amplitude per degree of SHC is about ± 1000 pounds (for both blades).

It was found that power required in hovering increased with second harmonic feathering. Power as a function of SHC amplitude is shown in Figure 15. Calculations of power to drive the mechanism indicate that only a small portion of the power increase shown in Figure 15 can be attributed to power requirements of the mechanism. The power increase is believed to be caused by the change in lift distribution resulting in two areas of high downwash velocities, and thus an apparent increase in disc loading, and two inefficient areas of low lift (see sketch).



In Figure 16, the 2/rev oscillatory lift link loads are given as a function of SHC amplitude. That figure shows that the increase in oscillatory lift loads is directly proportional to the SHC input. By comparing the lift link loads with the rotor thrust pulsations (Figure 17) it may be seen that these two forces are out of phase and that the variation in lift link load is .7 of the rotor thrust variation. The difference between the steady rotor thrust and the lift link load (corrected for rotor and pylon weight) is caused by loads carried through the pylon mounts and control tubes. The difference in amplitude of the oscillatory force in the lift link and the oscillatory rotor thrust is caused by the dynamics of the rotor blades. A phase shift between the oscillatory rotor thrust and oscillatory lift link load occurs because the first symmetrical beamwise blade bending mode is below the 2/rev rotor frequency.

Vertical thrust pulsations induced in hovering created vibrations in the fuselage. Figure 18 shows the measured vibration levels at several locations in the helicopter, as a function of the SHC amplitude. Figure 18 shows that oscillatory accelerations of $\pm .15$ to $\pm .2$ g's are within the capability of this SHC system. It is noted that the fuselage is not near a resonant condition near the 2/rev frequency.

Vertical acceleration levels of about .1 g at the pilot's seat have been recorded in shake tests (with the rotor replaced by an equivalent weight) when 2/rev oscillatory forces of 1000 pounds are applied at the top of the mast. This correlates reasonably well with data given in Figures 14 and 18, where it is shown that $\pm .57$ degrees SHC gives ± 550 pounds thrust pulsation, which, in turn, causes a $\pm .06$ g 2/rev vibration in the helicopter. It was concluded that the SHC follows basically the expected trends.

The 2/rev pulsation in rotor thrust (reference Figure 14) indicates that mechanically the SHC device performed satisfactorily. Recall, however, that it was pointed out in the description of the mechanism that the slider motion deviated from the theoretical motion (reference Figure 9).

Investigations of the loads experienced during initial hovering flights resulted in a decision to limit the amplitude of the SHC to a maximum of ± 1.1 degrees on initial forward flights. Since subsequent tests showed that maximum benefits from second harmonic feathering occurred below this amplitude, this value was not exceeded on later flights.

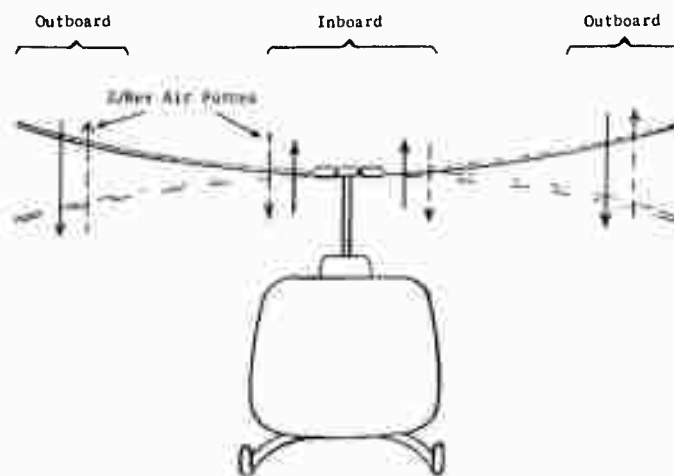
D. FORWARD FLIGHT

1. 40 Knots

At 40 knots airspeed, the effects of second harmonic feathering on the air loads are presented in Figure 19. For this case, however, the phasing of the SHC is significant. Figure 19 shows the air loads versus blade azimuth position with and without SHC. It may be seen that with this particular phasing (5 degrees) and amplitude ($\pm .57$ degrees) the second harmonic feathering reduces the 2/rev rotor thrust pulsations. As in hovering, it was determined that a 1000-pound oscillatory difference in thrust is developed per degree of SHC. Other data given in the appendix show that at other phase angles the SHC increased the 2/rev rotor thrust pulsations.

The effect of SHC on the lift link loads is given in Figure 20. It is shown that the 2/rev lift link loads increase rather than decrease when the SHC is applied. Recall that in hovering (refer to Figure 17) the lift link loads are out of phase with the induced rotor thrust pulsations.

A closer study of the air loads at 40 knots reveals that the SHC affects mainly the outboard portions of the blade. In Figure 21, total rotor thrust has been divided into thrust acting on the outboard half of the blades and the inboard half. The total rotor force, as shown in Figure 19, is the sum of the inboard and outboard air loads for both blades. Note that the SHC affects the inboard air loads only very slightly but is more effective in the outboard portion of the blade. Notice that the effect on the outboard section is basically a shift in the phase of the 2/rev pulsations. Numerically, the outboard pulsations cancel the inboard pulsations. However, the inboard 2/rev air loads enter the fuselage directly through shear forces at the hub. The outboard loads, on the contrary, cause an out-of-phase 2/rev rotor flapping, resulting in the force transmitted to the fuselage being out of phase with the outboard load. The lift link loads, therefore, are found by the difference between



the inboard lift and outboard lift times an unknown dynamic amplification factor. These considerations suggest that separate SHC of the inboard or outboard portion of the blade may offer interesting possibilities.

2. 80 Knots

For the 80-knot case, the most complete data are available. The characteristics of thrust pulsations, phasing, and magnitude at 80 knots were similar to that at 40 knots, although no condition was measured whereby the thrust pulsations were significantly reduced. Air load data in the appendix indicate that a reduction in thrust pulsations occurs at the same control settings (± 5.57 degrees and 5 degrees phasing) that reduced pulsations at 40 knots. But because of high control loads encountered during Phase I, SHC feathering amplitude was restricted to ± 2.28 degrees when measuring air loads with SHC phasing at 5 degrees.

Figure 22 shows an example of the effect of SHC phasing and amplitude on vibrations of the helicopter. Similar data presentations are given in the appendix for vibrations at other fuselage locations and for blade and control loads. As a next step, all such data as given in Figure 22 are synthesized by cross-plotting the maximum and minimum responses for each SHC amplitude. The resulting plots are given in Figure 23 and are discussed in the following paragraphs.

In Figure 23 it is shown that the SHC is capable of reducing vibration levels and loads to a certain extent, as well as increasing the vibrations and loads. Figure 23 also shows that the phase angle for optimum conditions differs from one item to another, i.e., control loads, vibrations, etc., being considered. For instance, minimum pilot vibrations are obtained at 20 degrees phasing while blade loads, some control loads, and the c.g. accelerations are increased. This points to a basic difficulty. It is impossible to optimize everything at the same time.

Theoretical predictions in Reference 8 showed that 2/rev lift link loads and 2/rev hub bending moments would be minimized at an azimuth phasing of 45 degrees (see Figure 1). Experimental results presented in Figure 23 indicate that the maximum reduction in 2/rev lift link loads occurs when the mechanism is phased at 50 degrees.

It appears to be impossible to find, even at the optimum phase angle, an amplitude for the SHC where the 2/rev cabin vibrations go to zero. Originally, it was expected that the SHC vibrations could be superimposed upon the base line vibrations of the helicopter so that there would be always one condition where 2/rev cabin vibrations would be eliminated (see Figure 3). A number of possible explanations as to why this was not the case have been investigated. These are discussed in the following paragraphs:

- (1) The higher in-plane (chord) loads, as shown in Figure 23, give rise to additional vibrations which may occur at the same time that minimum vertical thrust vibrations are expected due to the SHC. However, by measuring the oscillatory pylon mount motions which isolate in-plane rotor vibrations, this effect has been excluded. In Figure 24, it is shown that the mount deflections with and without SHC are identical.
- (2) Higher control loads were measured when pilot vibrations were reduced because of the SHC (see Figure 23). Since the control loads enter the fuselage through the control system, it is possible that they cause an increase in vibration level which partly offsets the expected vibration reduction.
- (3) During the testing, it was noted that on occasion higher frequencies than 2/rev were generated by the SHC mechanism. A 6/rev oscillation can be seen in the lift link load presented previously in Figure 20. Figure 25 shows an example where the change in thrust due to SHC as measured by the air load blades at 80 knots is shown as a function of the rotor azimuth. A 6/rev vibration is indicated. The cause for this high frequency is not known, but it is conceivable that slight motions of the mechanism such as described in Section V contribute to deviations from the expected results. Present theoretical methods are inadequate for analyzing 6/rev vibrations and loads.

On the other hand, the results in Figure 23 also show that the vibration level at the c.g. of the helicopter can be reduced with a simultaneous reduction in blade and control loads. The SHC phase angle for this case is about 120 degrees. This finding indicates that an SHC mechanism may be advantageous for a fuselage where all the cargo is situated close to the c.g. of the aircraft.

3. 100 Knots

The amount of SHC introduced at 100 knots was restricted because of high control loads, excessive pylon motions, and unexplained high amplitude vibrations encountered occasionally when SHC was applied. At one time (phase angle of 91.5 degrees at an amplitude of ± 3.38 degrees) excessive vibrations, including larger than normal fore and aft pylon motions, were experienced. At another time, a violent vibration was experienced at 100 knots (129 degrees phase angle and a ± 7.76 degrees amplitude) in which case it was necessary to reduce airspeed to 80 knots and the SHC amplitude to zero in order to eliminate the vibrations. Unfortunately, no records were taken during this period. These conditions were not repeated throughout the remainder of the tests. The reason for these vibrations and the associated pylon motions is not understood, but it is possible that the reduction in stiffness of the control system due to the addition of more parts to accommodate the SHC function was influential.

As discussed earlier, a beneficial effect was expected in reducing blade stall. From the data taken during high gross weight flights, no such effects were detectable. Analytical studies, conducted by Bell, indicate that compressibility effects could offset the gains due to the elimination of retreating blade stall. Figure 26 shows the calculated rotor drag at several azimuth stations with and without SHC. Computed drag coefficients at the rotor blade tip are presented in Figure 27 showing that the tip operates continuously in compressibility. The drag coefficient is shown to decrease on the retreating side, but increases at the fore and aft blade positions when the SHC is applied. It follows from these analyses that retreating blade stall can indeed be reduced; but in the fore and aft sectors of the rotor disc, the drag is increased due to compressibility. For the subject test rotor, the beneficial effects of SHC on vibrations and power are therefore small.

E. OPTIMUM CONDITIONS AS A FUNCTION OF SPEED

Figure 28 summarizes the effects of the SHC as a function of forward speed. Indicated are the base line zero SHC data and the maximum and minimum conditions that can be obtained through an SHC amplitude of ± 2.28 degrees. To make this plot, the phase angles for optimum results were used so that the phase angle as a function of speed changes constantly. The amplitude of ± 2.28 degrees is, at very low speeds, apparently too large,

giving rise to resulting vibrations higher than the base line vibrations. The benefits of SHC at high speeds seem to diminish to almost zero, probably because of the aforementioned side effects such as high control loads and compressibility.

The data of Figure 28 illustrate the difficulty of practically applying the SHC system. Both the amplitude and the phase angle have to be changed with forward speed; while, as shown before, it is impossible to satisfy all optimum conditions for vibrations and loads simultaneously.

F. CONCLUDING DISCUSSIONS

A limited comparison of test results with the data predicted for the SHC can be made. In Reference 8, the effect on vibrations and rotor loads was predicted for an SHC system on the UH-1B at 120 knots. Since the actual program was flown with the UH-1A with 15-inch chord blades instead of the UH-1B with 21-inch chord blades, the calculated results do not apply to the flight tests. There is, for instance, a difference in the base line vibration level between the UH-1A and UH-1B. In Figure 29, plots of the 2/rev lift link load versus SHC amplitude are given for several forward speeds as measured during this flight test program. The calculated value is taken from Reference 8. It is shown that in this case, a reasonable correlation exists between the calculated data and the measured data even though the calculations refer to the UH-1B and do not include considerations of changes in oscillatory control loads.

It is believed that refinements of the present theoretical methods are needed. In particular, methods for prediction of control loads should be developed and evaluated by subsequent flight testing. The effects of control loads on fuselage vibrations seem to be another important subject for further study.

It is likely that the SHC principle will have different effects on other rotors and other helicopters. It may be useful to investigate this in more detail. Also, second harmonic feathering of only portions of the blades could show promise.

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TABLE 1

LIST OF INSTRUMENTATION

Items Instrumented		Used During	
Measurement	Part or Position	Phase I	Phase II
Loads	SHC Slider Drive Links	X	
	SHC Swash Plate Control Link	X	
	SHC Phase Control Lever	X	X
	Pitch Links	2	1
	Drag Brace	X	
	Left & Right Cyclic Boost Tubes	X	X
	Collective Boost Tube	X	X
	Lift Link	X	X
Blade Bending Moments	15% Beam	X	X
	28% Beam	X	
	45% Beam	X	
	65% Beam	X	
	80% Beam		X
	15% Chord	X	X
	28% Chord	X	
	65% Chord	X	
Vibrations	Pilot Position	X	X
	Passenger Compartment	X	X
	Center of Gravity	X	X
Position Indicators	Pylon Mounts	4	4
	Instrumented Blade-Azimuth	X	X
	Blade Pitch	X	X
	Cyclic Position	X	X
	SHC Slider	X	X
	*SHC Actuator - Amplitude & Azimuth	X	X
Performance (Photo Panel)	Airspeed	X	X
	Engine Output Shaft Speed	X	X
	Engine Gas Producer Turbine Speed	X	X
	Altitude	X	X
	Outside Ambient Temperature	X	X
	Engine Torquemeter Pressure	X	X
Main Rotor Air Loads	<u>Blade Span</u>		
	40%	4, 17%	X
	55%	2, 17, 34%	X
	75%	2, 9, 35%	X
	85%	2, 7, 34%	X
	90%	2, 9, 17, 23, 34%	X
	95%	2, 9, 17, 23, 63%	X

Notes: *These data recorded manually

TABLE 2
CONVERSION FACTORS USED TO DETERMINE
BLADE LOADING (THRUST/INCH)

SPAN STATION	CHORD STATION	CONVERSION FACTOR
40%	4%	3.8 lb-in/psi
* 55%	17%	8.39 lb-in/psi
75%	2% 9%	3.46 lb-in/psi 5.78 lb-in/psi
85%	2% 17%	3.8 lb-in/psi 7.5 lb-in/psi
90%	2% 9% 17%	3.4 lb-in/psi 5.9 lb-in/psi 5.9 lb-in/psi
95%	2% 23% 63%	3.2 lb-in/psi 9.5 lb-in/psi 39.0 lb-in/psi

*Note: After reducing data from this investigation, pressure transducers at 55% were determined to be unreliable.

TABLE 3

LIST OF PHASE I AND II FLIGHT CONDITIONS

Level Flight

Center of Gravity: Between
Stations 128.1 & 129.7

Flight No.	T.O.GW (lb.)	SHC Settings		Indicated Airspeed (Knots)	Pressure Alt feet		Temperature °C		Category
		Lever Index	Amplitude (In.)		Take-off	Flight	Take-off	Flight	
21A	6390	3*	0-1.0**	0	560	≈ 560	27	27	Phase I (Standard Blades)
22A	6390	3	0,.2,.4,.6	60	500	1000	34	34	
22B	6390	3	0,.2,.4,.6	80	550	1000	38	36	
23A	5780	3	0,.2,.3,.4	0,40,60,80,100	520	1000	34	34	
23B	6370	2	0-1.0	0	500	≈ 500	43	43	
23C	6390	2	0,.2,.4	80	540	1000	44	35	
23D	6320	2	0,.2,.6	80,100	560	1300	39	39	
24A	6320	4	0,.2,.4,.6	80	540	1000	33	30	
24B	6280	1	0,.2,.4,.6	80	540	1000	37	33	Phase II (Air Load Measuring Blades)
26B	6031	1	0,.15,.3	0,20,40,60,80,100	500	1000	36	30	
26C	6031	3	0,.15,.3	40,60,80	550	900	35	34	
27A	6031	3	0,.15,.3	0,20,40,60,80,100	520	1200	26	25	
27B	6031	1	0,.15,.3	0,20,40,60,80,100	500	1350-1500	33	28-27	

*Three or more azimuth positions were tested for each lever index. Nominal phase ranges for each lever index are:

1 45°-90° 3 135°-180°
2 90°-135° 4 180°-225°

**See Figure 10 for blade feathering corresponding to amplitude in inches.

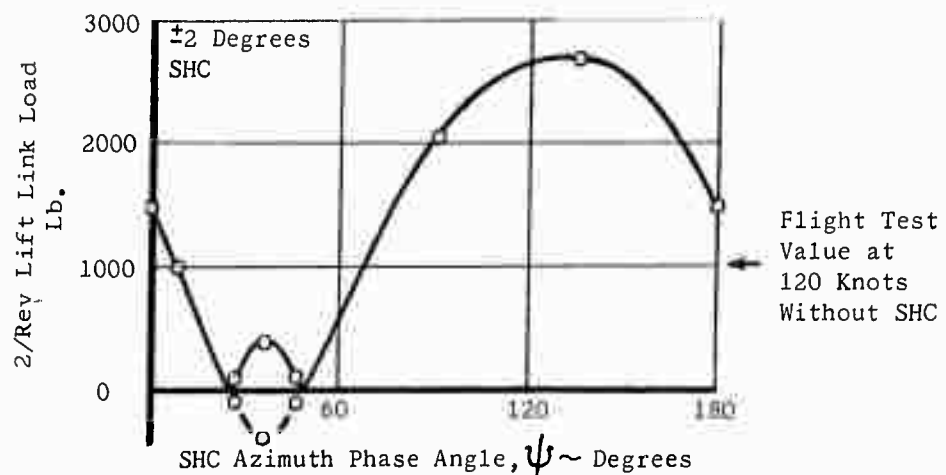


Figure 1. Predicted Effect of SHC at 120 Knots on the UH-1B Lift Link Loads.

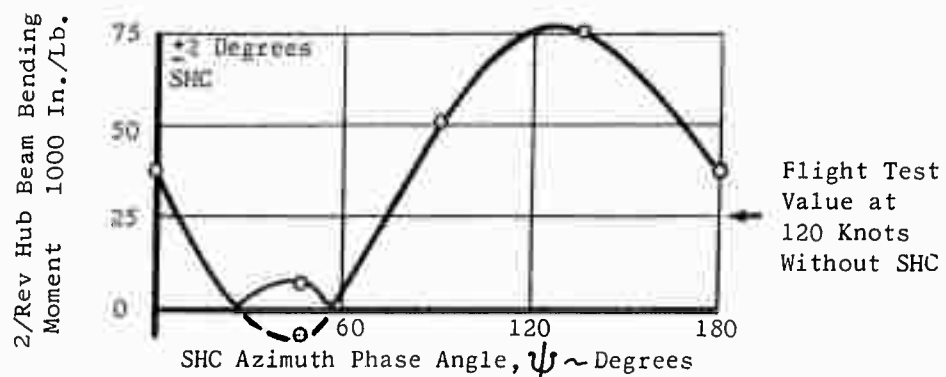


Figure 2. Predicted Effect of SHC at 120 Knots on the UH-1B Hub Spindle Moments.

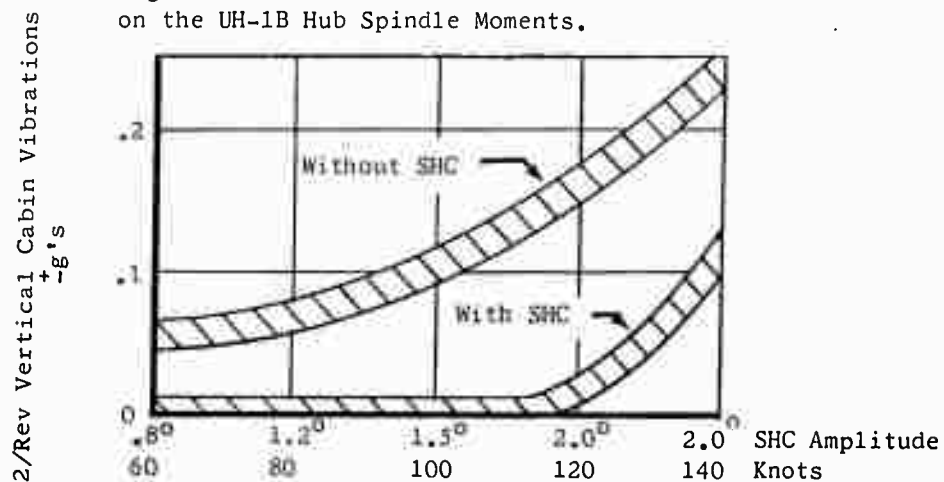


Figure 3. Predicted Effect of SHC on the UH-1B Cabin Vibrations.



Figure 4. The UH-1A Iroquois Test Helicopter with the Second Harmonic Control Mechanism Installed.

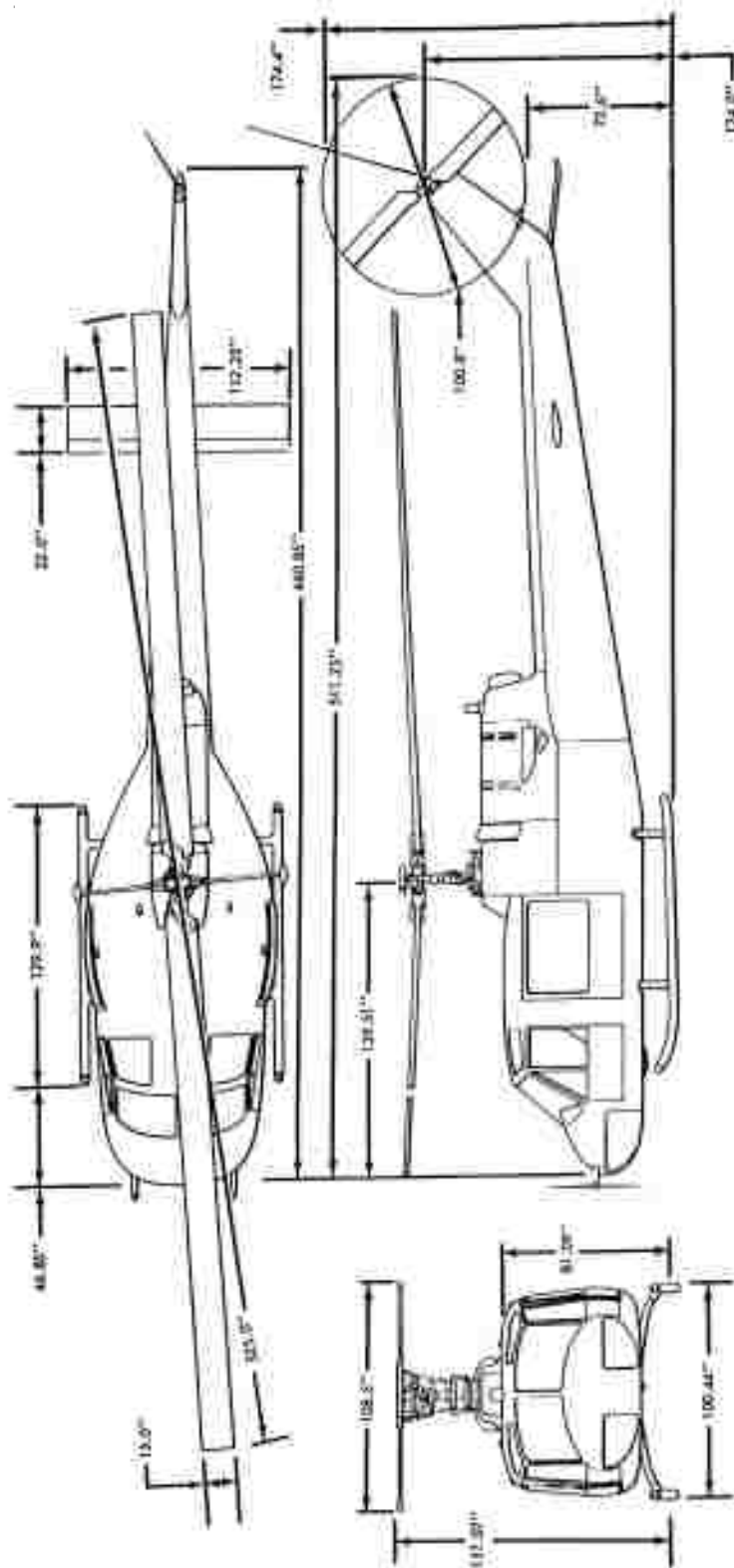


Figure 5. UH-1A Helicopter - Three-View and Dimensional Data.

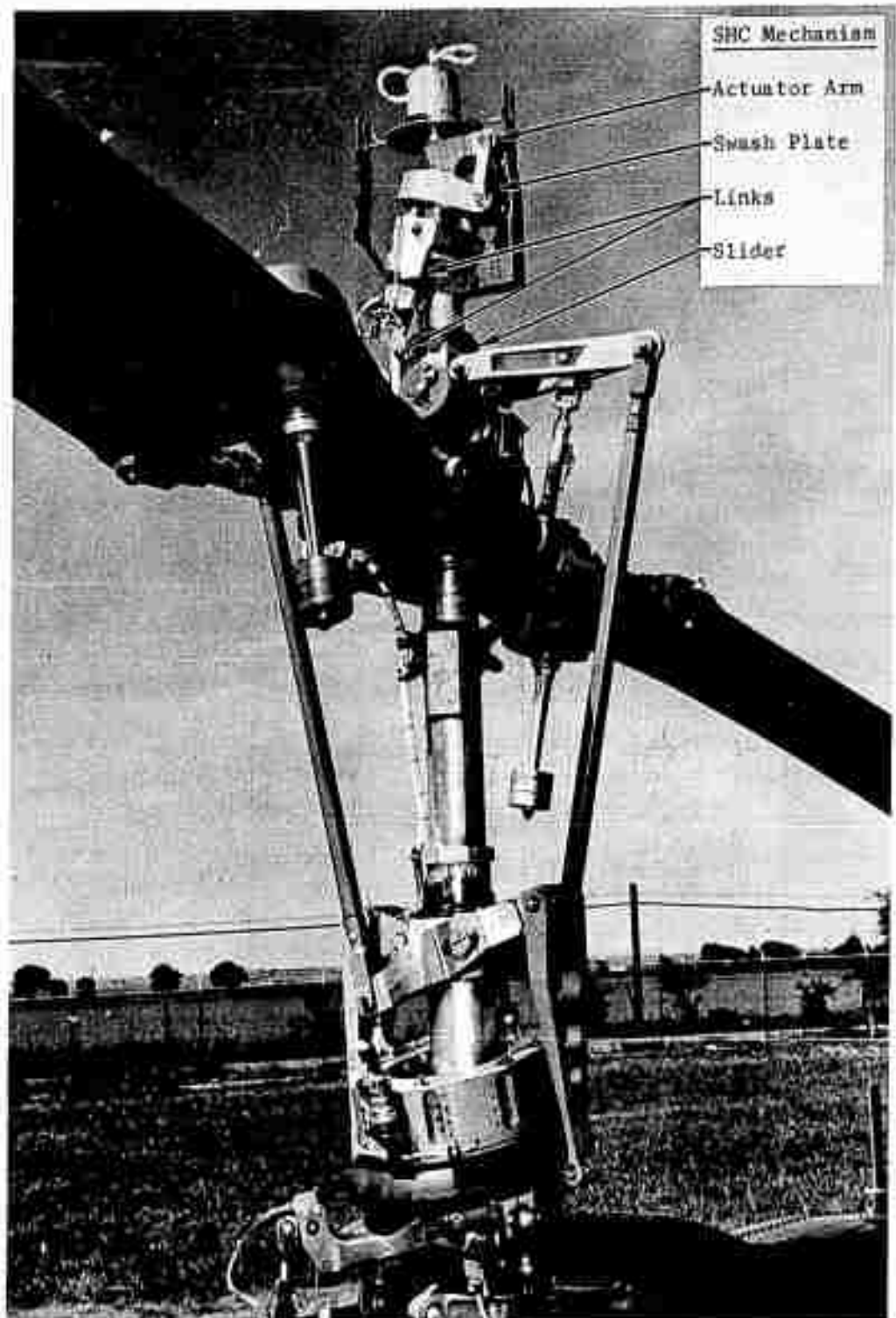
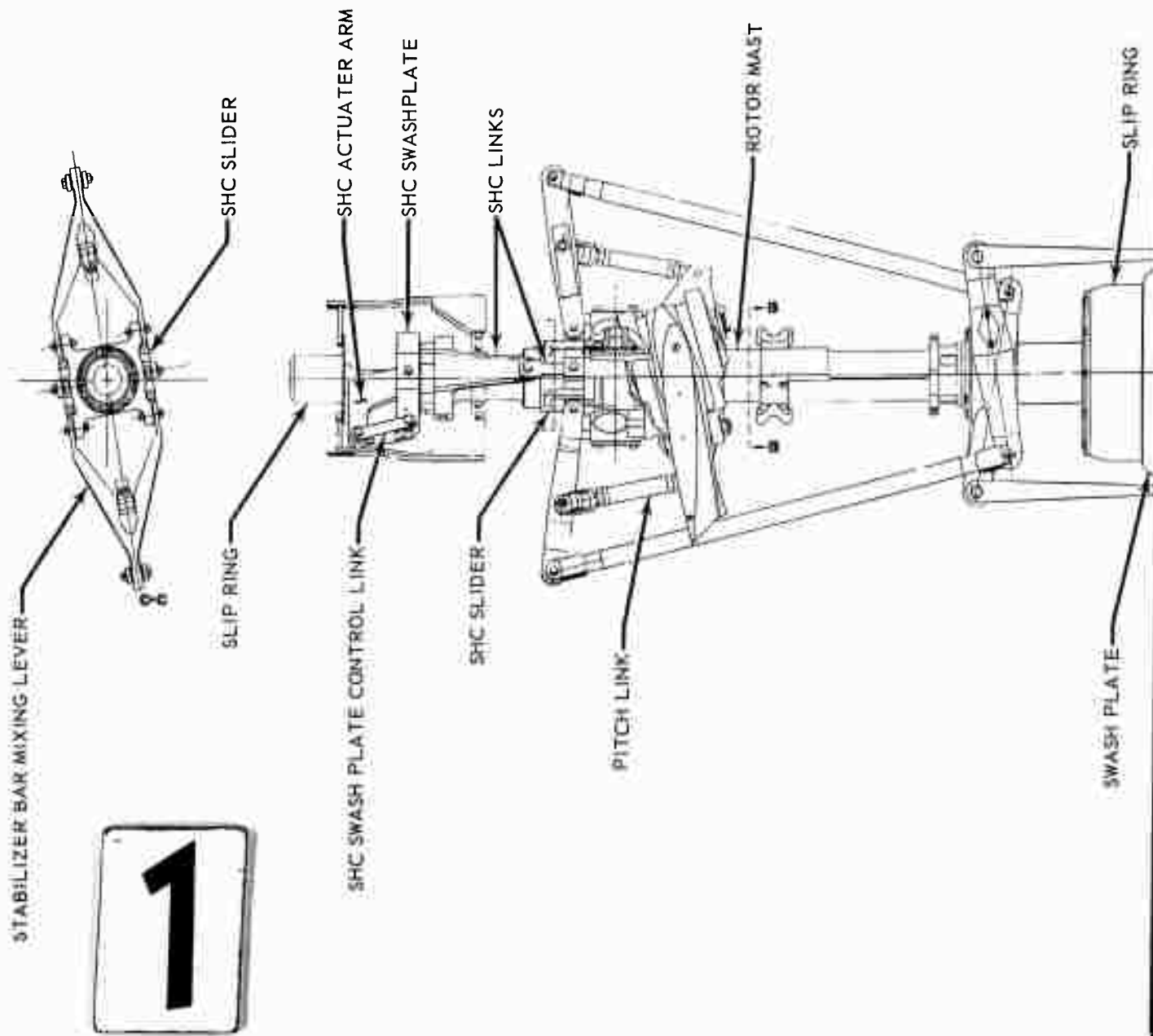


Figure 6. Photograph of Experimental Pylon Assembly



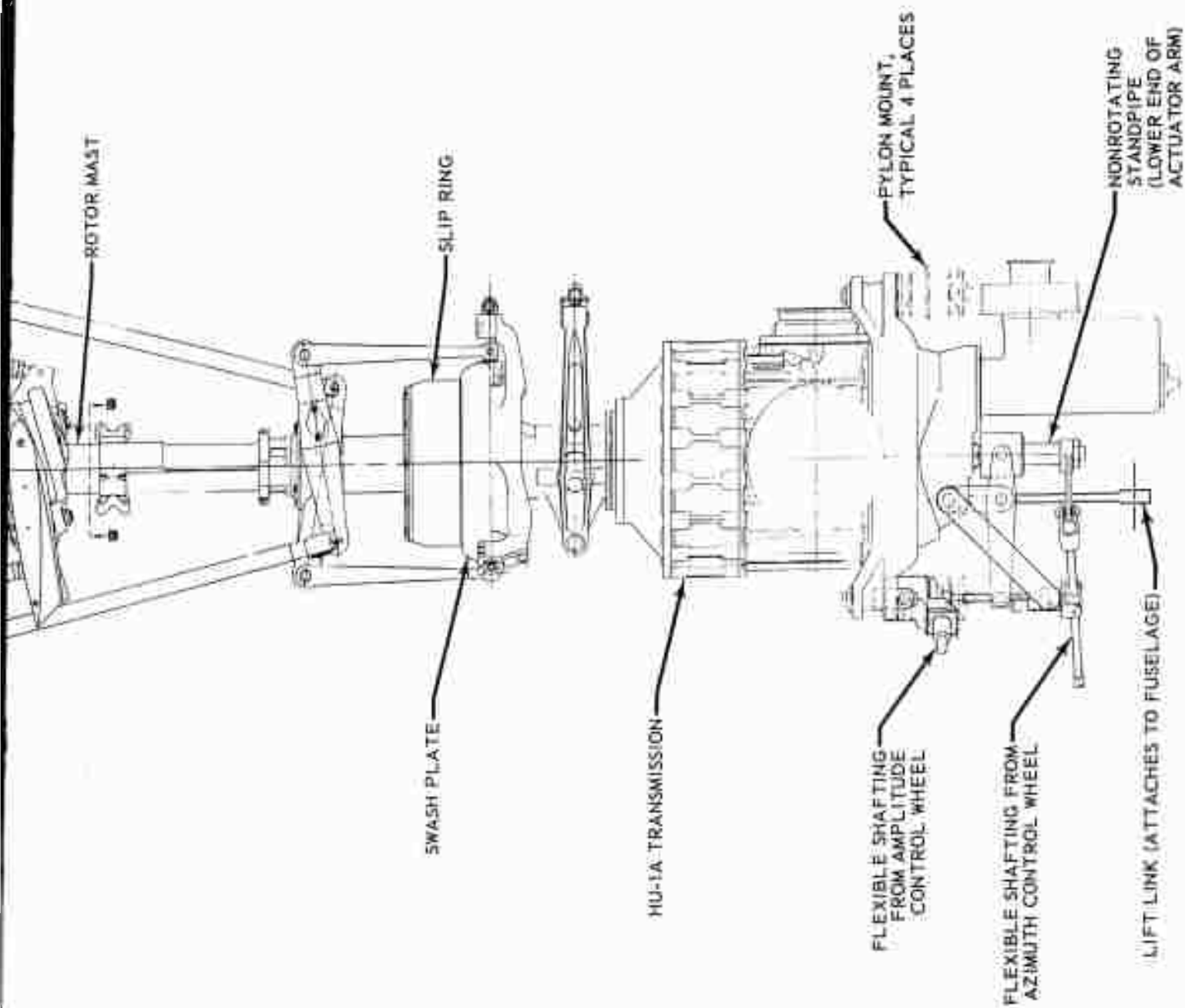


FIGURE 7. DRAWING OF EXPERIMENTAL PYLON ASSEMBLY



Figure 8a. Looking Fwd
No SHC



Figure 8b. Right Side
No SHC



Figure 8c. Looking Fwd
Maximum SHC $\psi = 90^\circ$

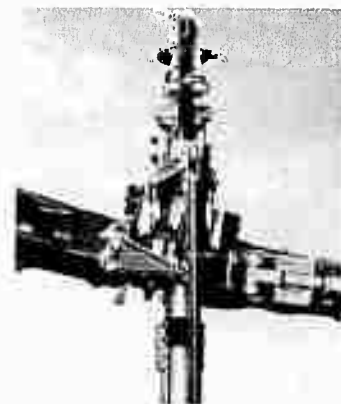


Figure 8d. Right Side
Maximum SHC $\psi = 90^\circ$

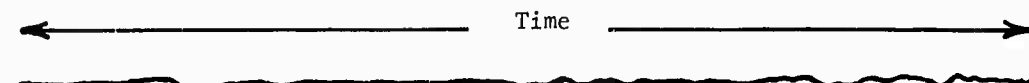


Figure 8e. Looking Fwd
Maximum SHC $\psi = 180^\circ$



Figure 8f. Right Side
Maximum SHC $\psi = 0^\circ$

Figure 8. Views of the Second Harmonic Control (SHC) Mechanism
at Several Control Settings. Blades are Aligned Fore and Aft.



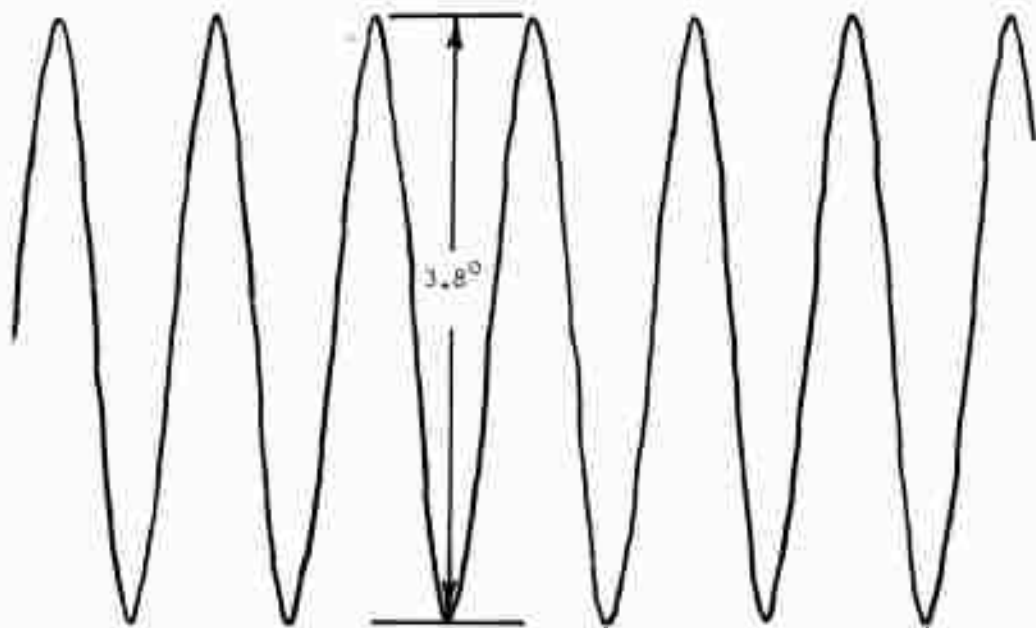
Condition: First Flight; Hover; Zero SHC Amplitude



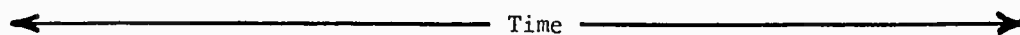
Condition: Last Flight; Hover; Zero SHC Amplitude



Condition: Last Flight; 100 Knots; Zero SHC Amplitude



Condition: First Flight; Hover; $\pm 1.9^\circ$ SHC Amplitude



(All traces have same vertical scale)

Figure 9. Oscillograph Traces of the Second Harmonic Control Slider Position.

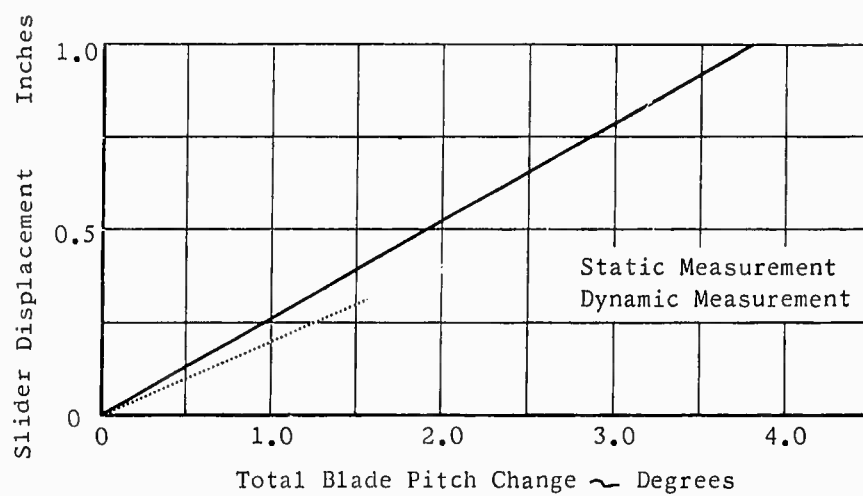


Figure 10. Pitch Change Versus Slider Displacement.

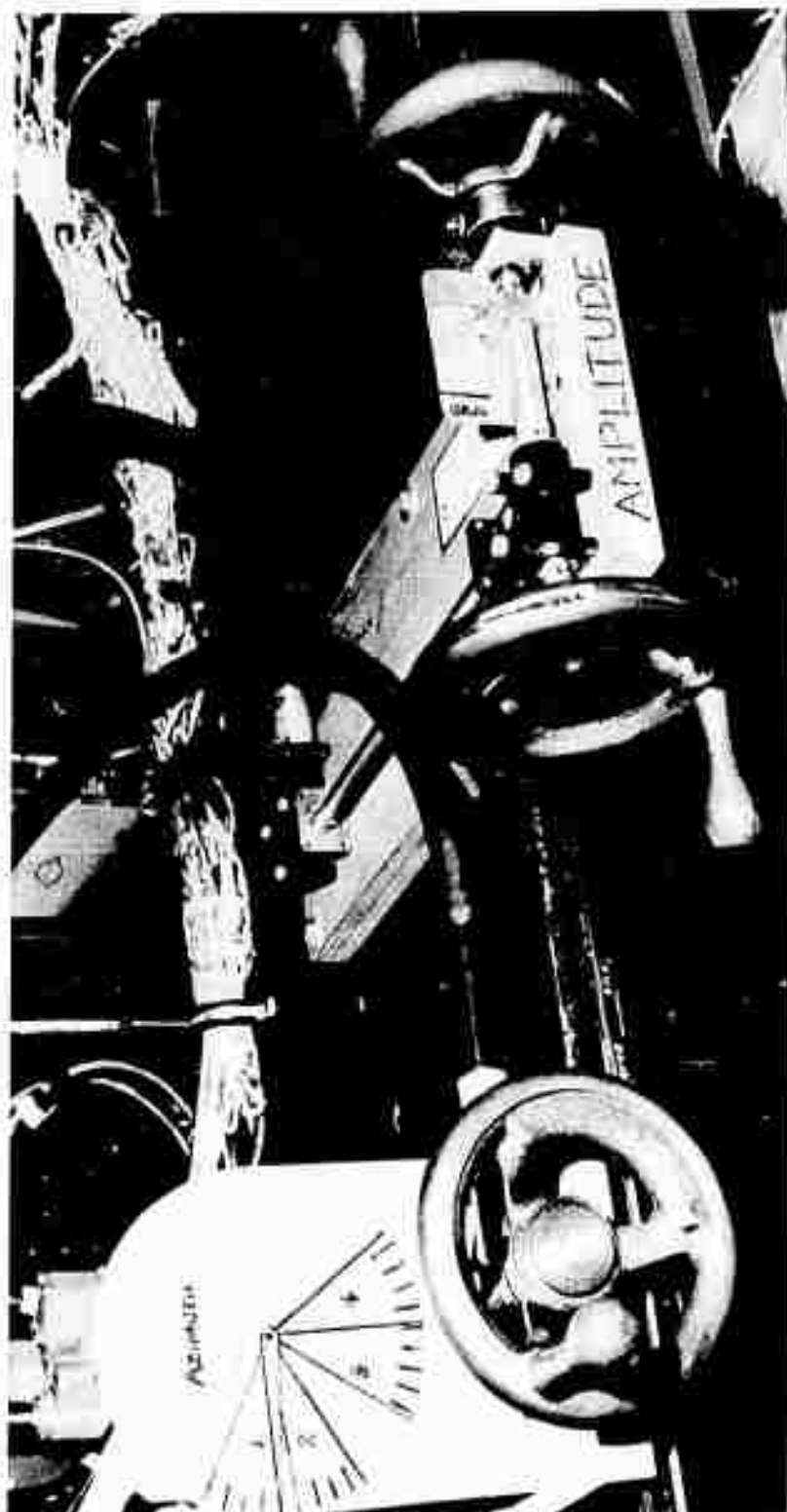


Figure 11. Hand Wheels for In-flight Adjustment of the Second Harmonic Control Mechanism

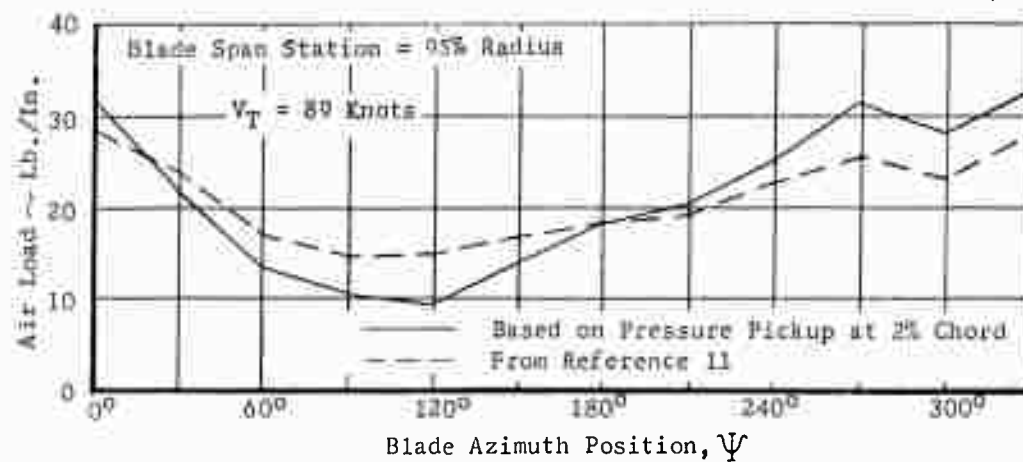
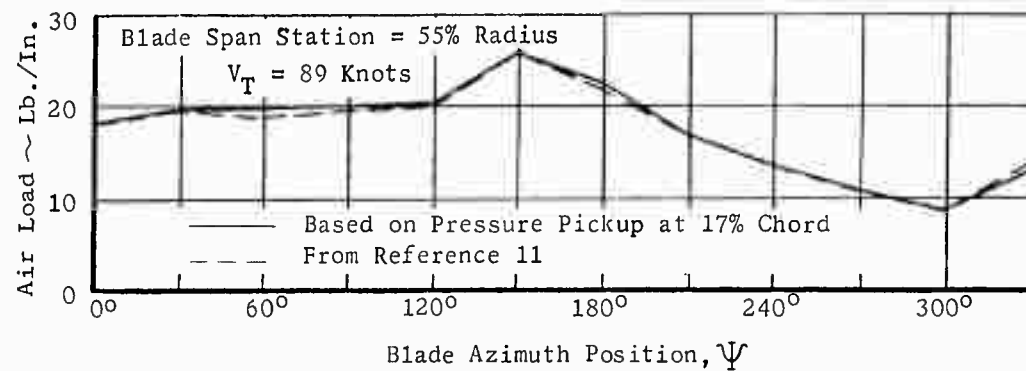
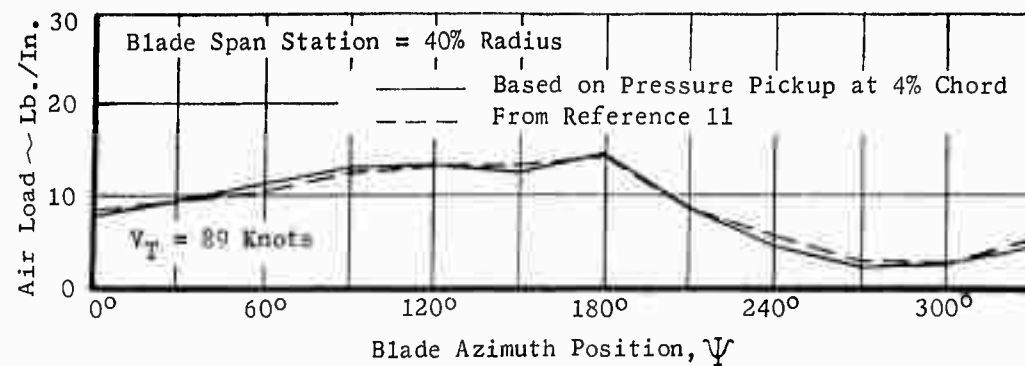


Figure 12. A Comparison of the Air Load Reduction Method Utilizing One Pressure Pickup with the Method Used in Reference 11.

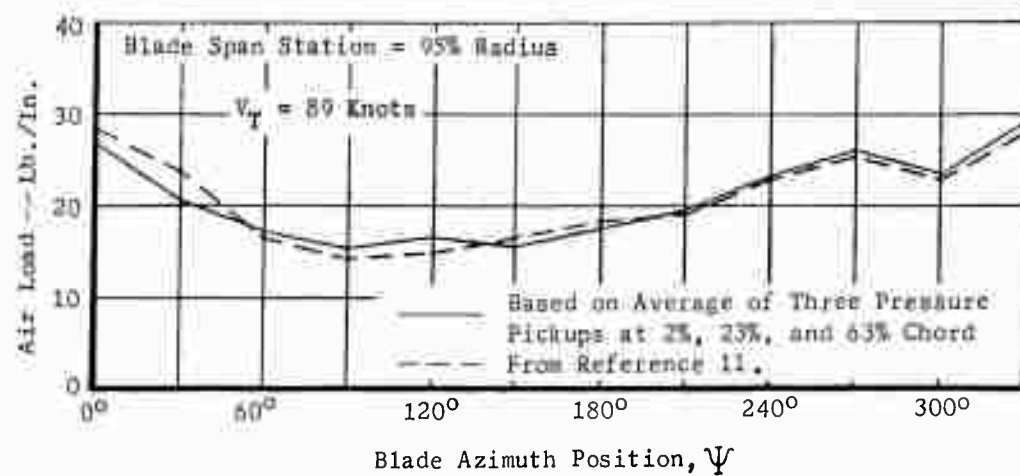


Figure 13. A Comparison of the Air Load Reduction Method Utilizing Three Pressure Pickups with the Method Used in Reference 11.

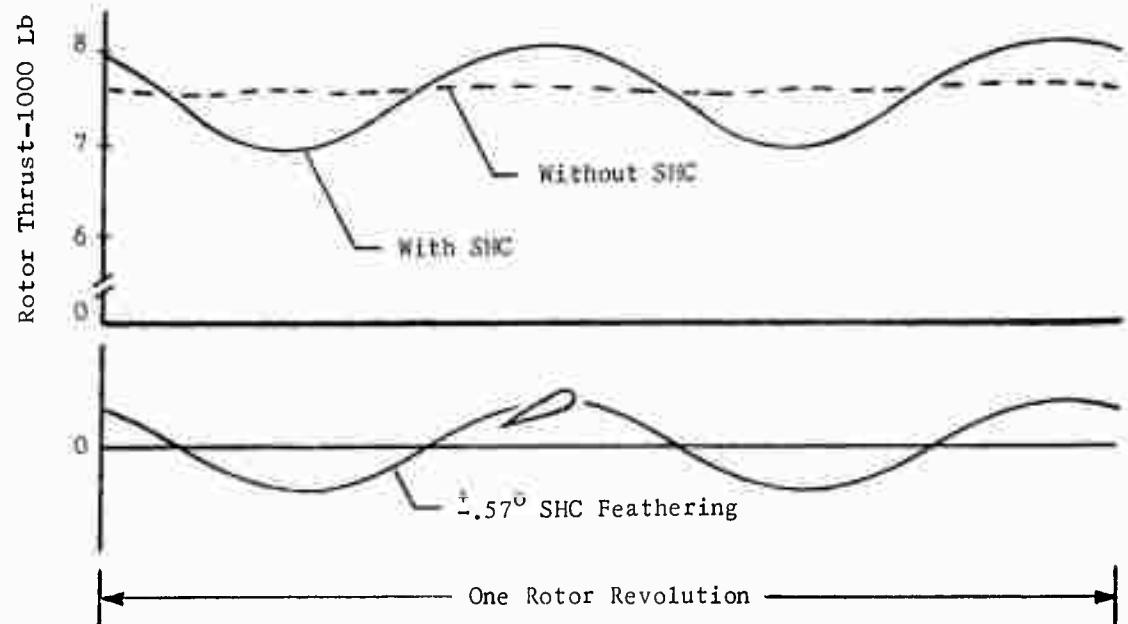


Figure 14. A Comparison of Rotor Thrust With and Without Second Harmonic Control in Hovering.

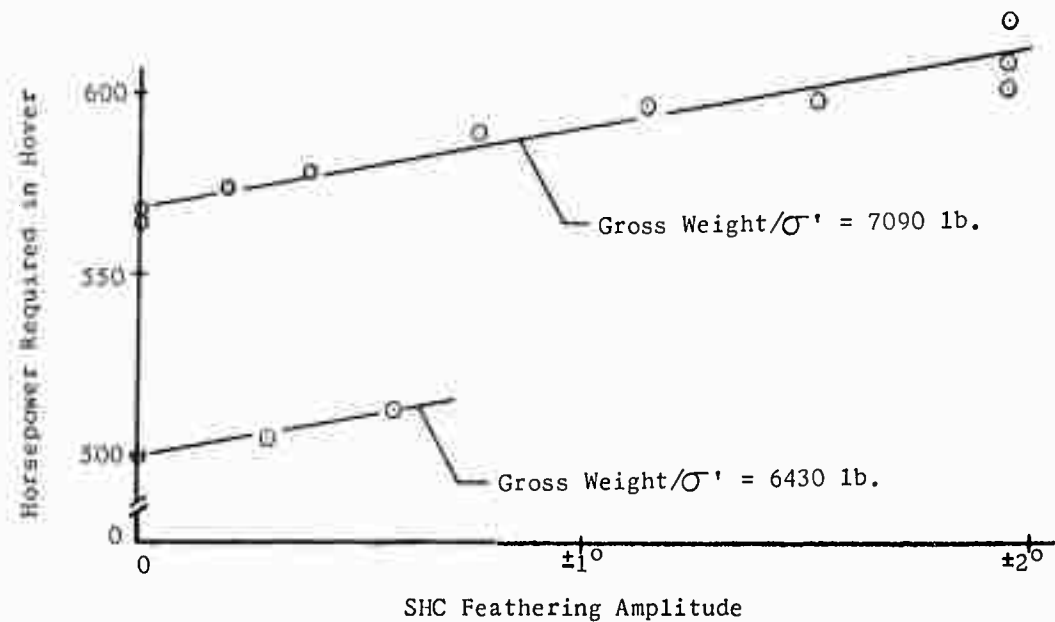


Figure 15. The Effects of Second Harmonic Control on Power Required in Hovering.

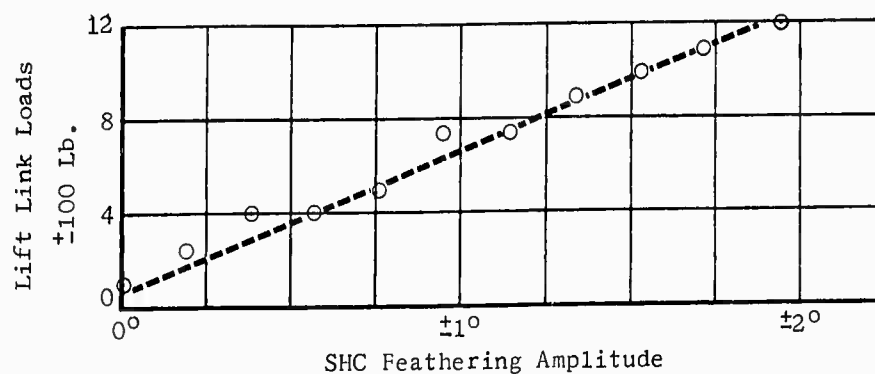


Figure 16. The Effects of Second Harmonic Control on 2/Rev Lift Link Loads in Hovering.

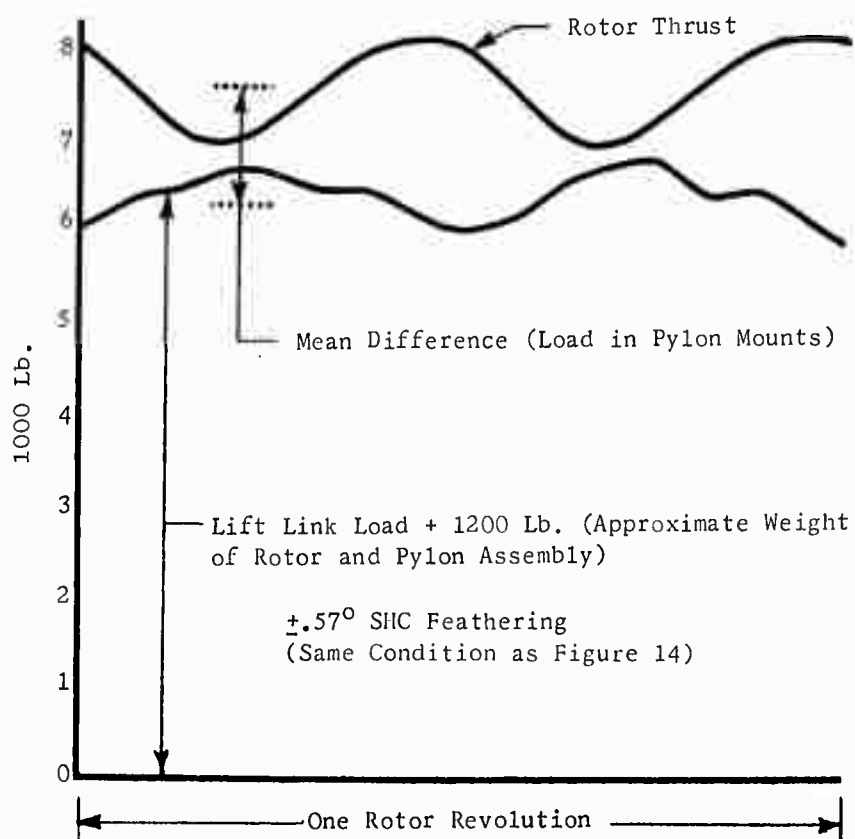


Figure 17. Rotor Thrust Compared with Lift Link Load as a Function of Blade Azimuth Position in Hovering.

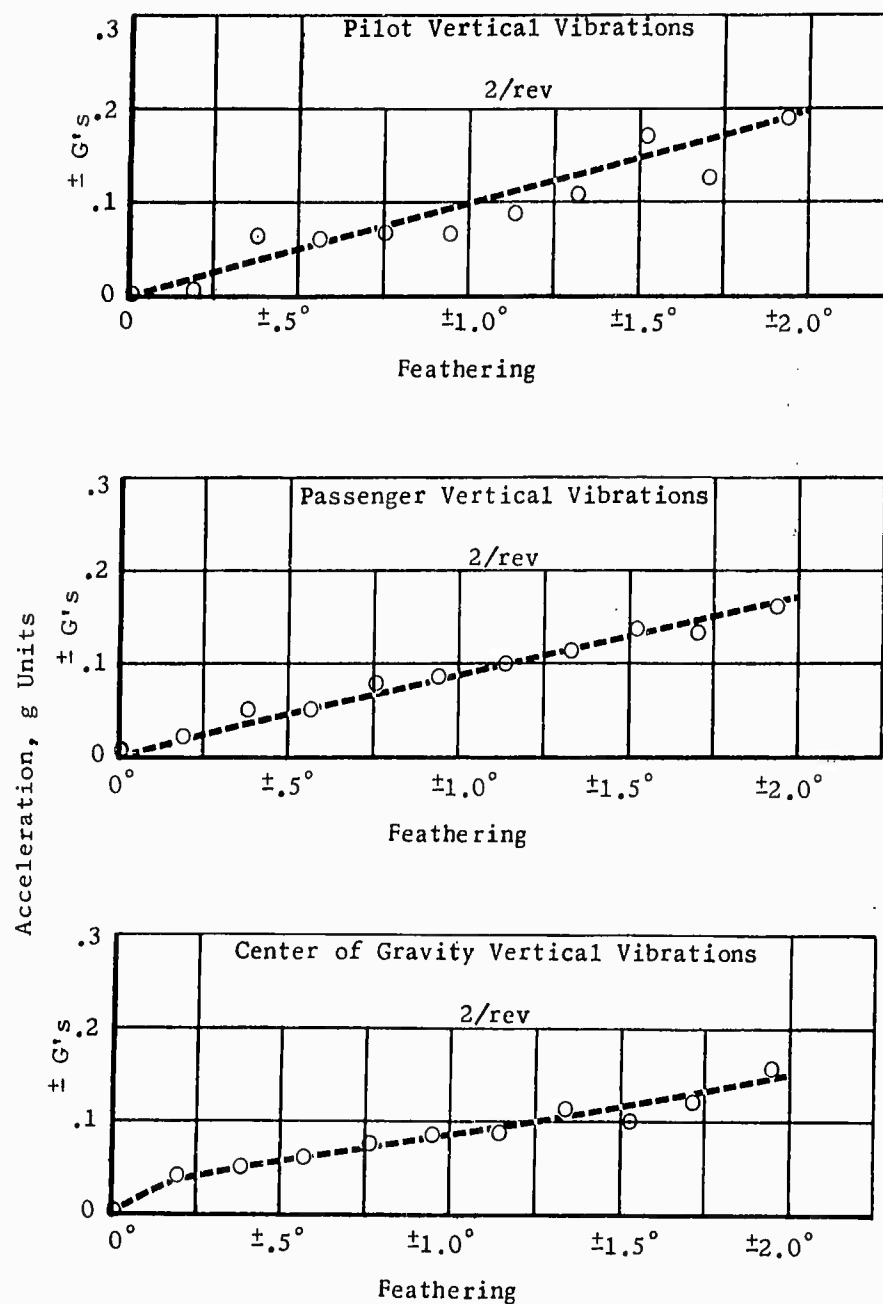


Figure 18. The Effects of Second Harmonic Control on Fuselage Vibrations in Hovering.



Figure 19. A Comparison of Rotor Thrust With and Without Second Harmonic Control at 40 Knots.

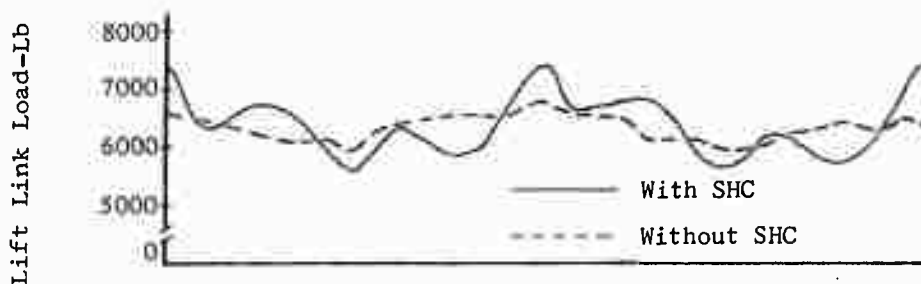


Figure 20. The Effects of Second Harmonic Control on 2/rev Lift Link Loads at 40 Knots.

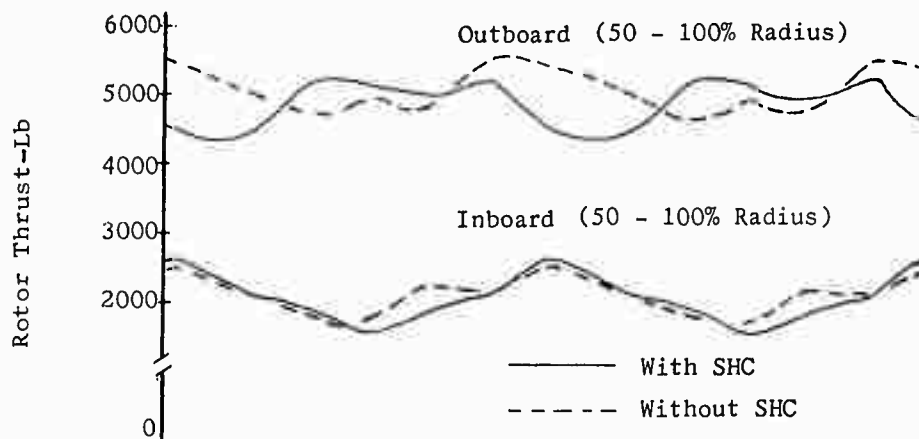
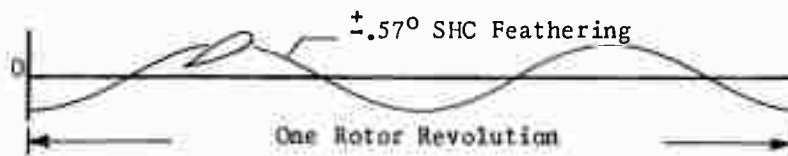


Figure 21. A Comparison of Inboard and Outboard Components of Rotor Thrust at 40 Knots.



Second Harmonic Feathering Reference for Figures 19, 20, and 21. SHC Phasing 5° .

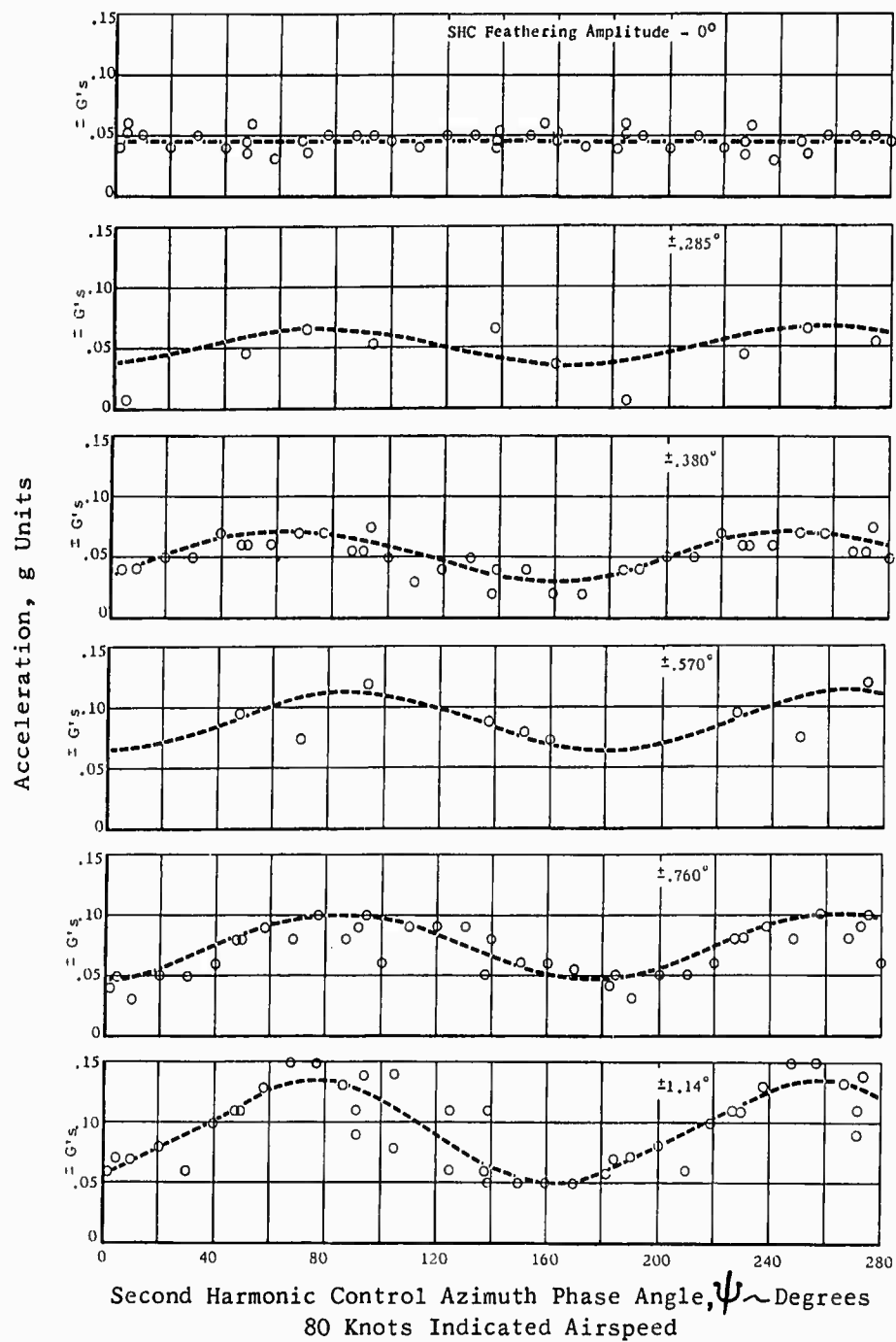
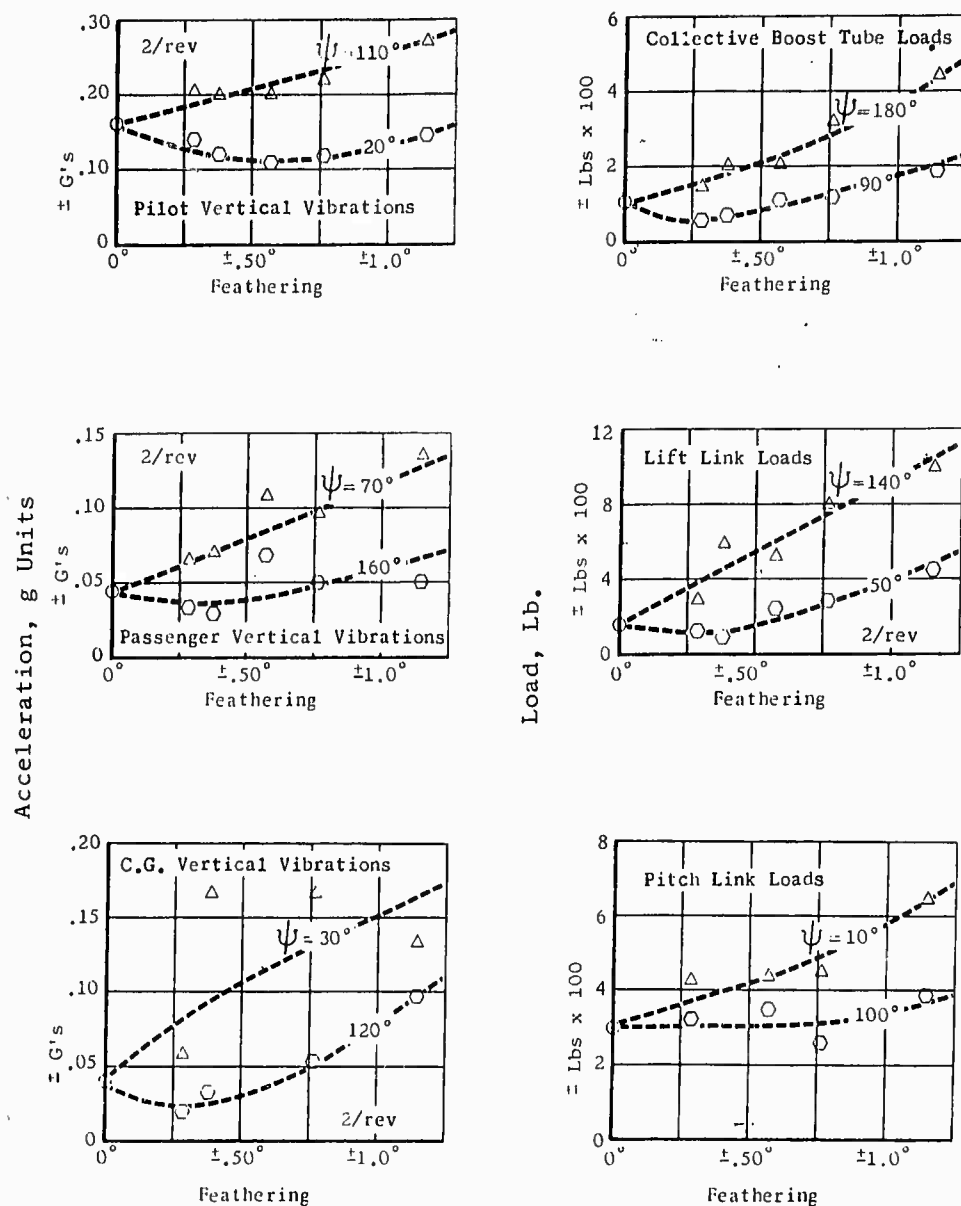
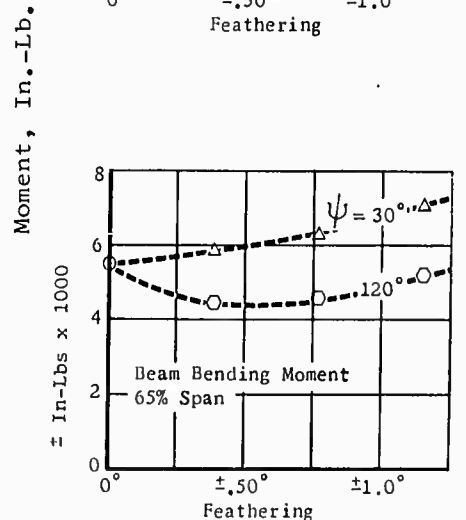
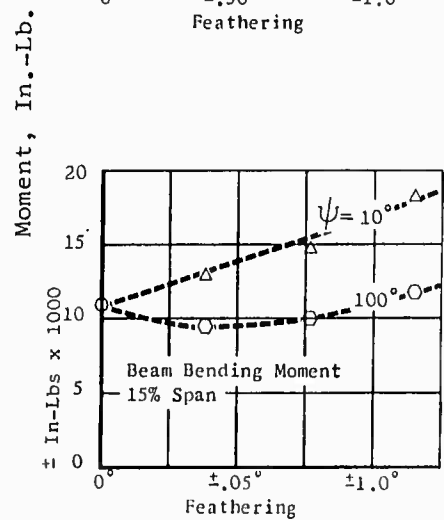
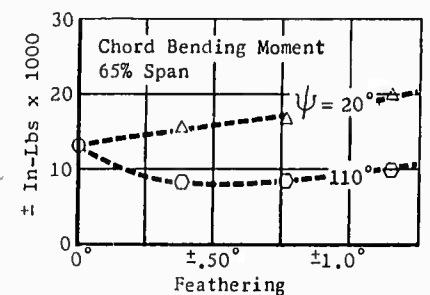
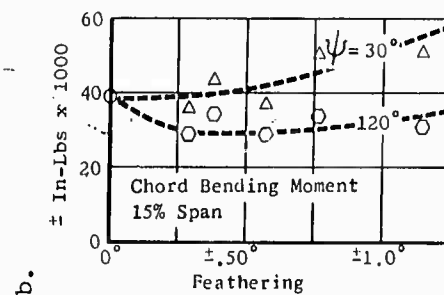
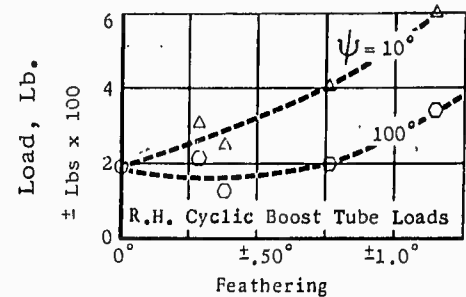
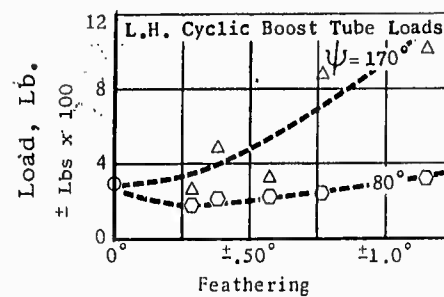


Figure 22. The Effect of Second Harmonic Control Phasing and Amplitude on 2/rev Vertical Vibrations at the Passenger Seat. (A Typical Example of Flight Test Results Presented in the Appendix.)



Cross Plots of Data Presented in the Appendix

Figure 23. Maximum and Minimum Vibrations and Oscillatory Loads Versus Second Harmonic Control Feathering Amplitude at 80 Knots.



Cross Plots of Data Presented in the Appendix

Figure 23 (Continued). Maximum and Minimum Vibrations and Oscillatory Loads Versus Second Harmonic Control Feathering Amplitude at 80 Knots.

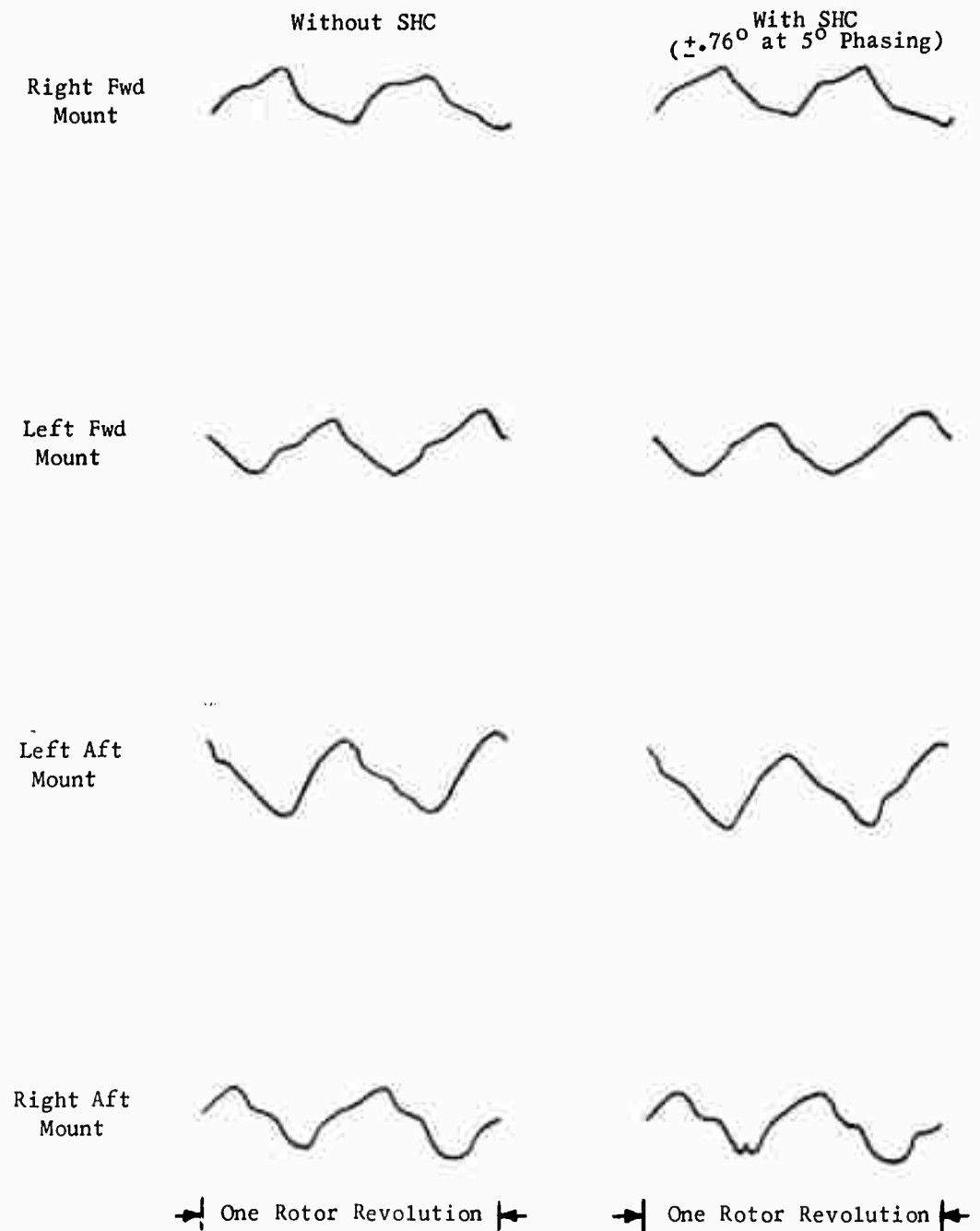


Figure 24. A Comparison of Pylon Motion With and Without Second Harmonic Control at 80 Knots.

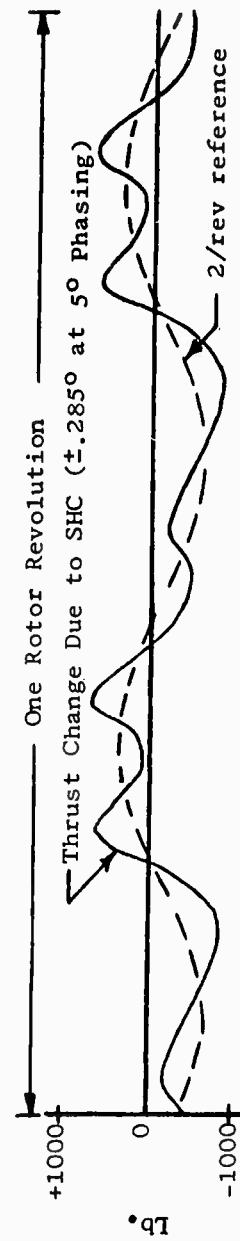


Figure 25. The Change in Rotor Thrust Due to Second Harmonic Control at 80 Knots; the Presence of Higher Harmonics Illustrated

----- Lines of Constant Angle of Attack

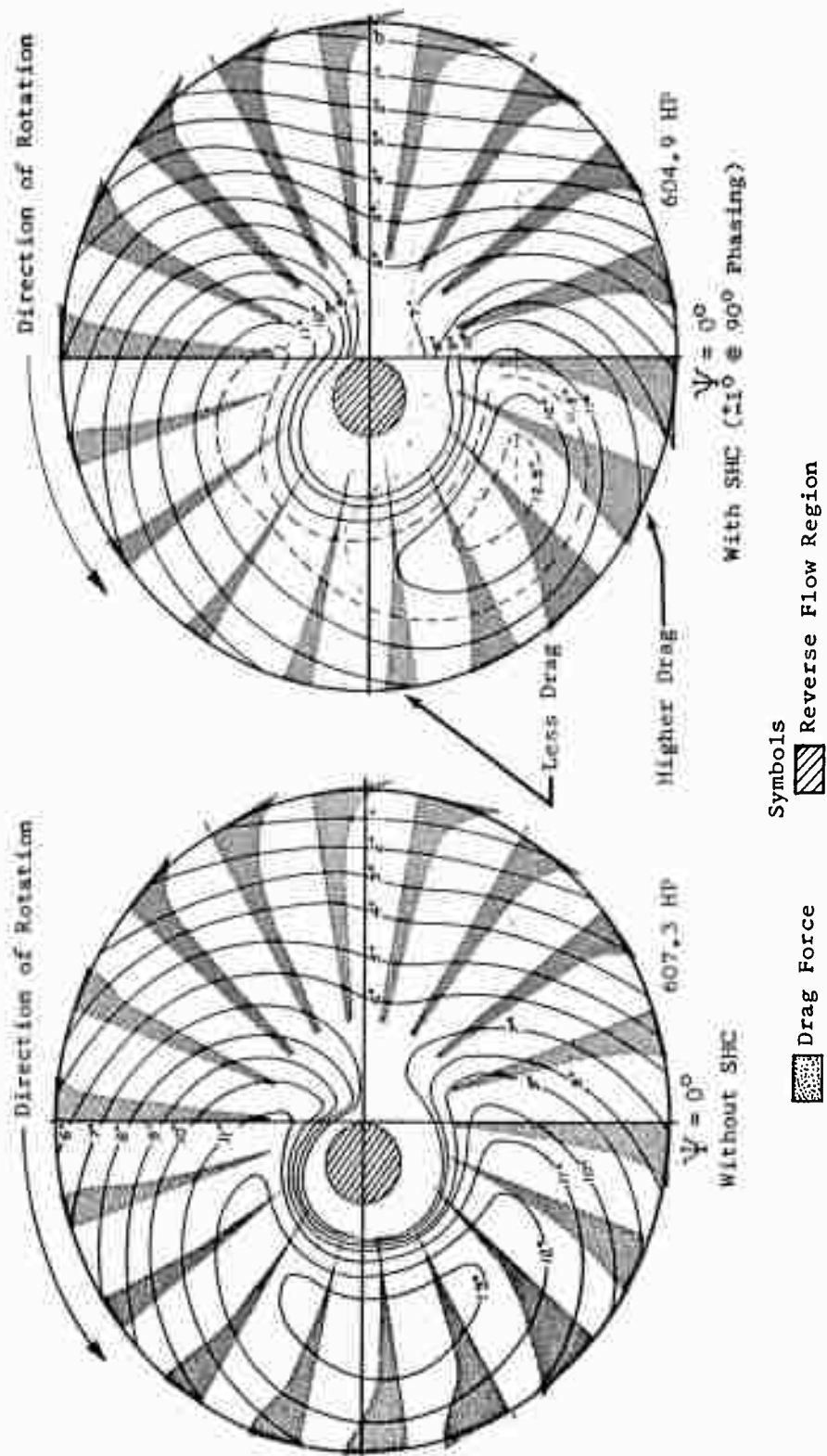


Figure 26. Computed Rotor Drag With and Without SHC at 100 Knots, Gross Weight/ $\sigma = 7150$ lb.

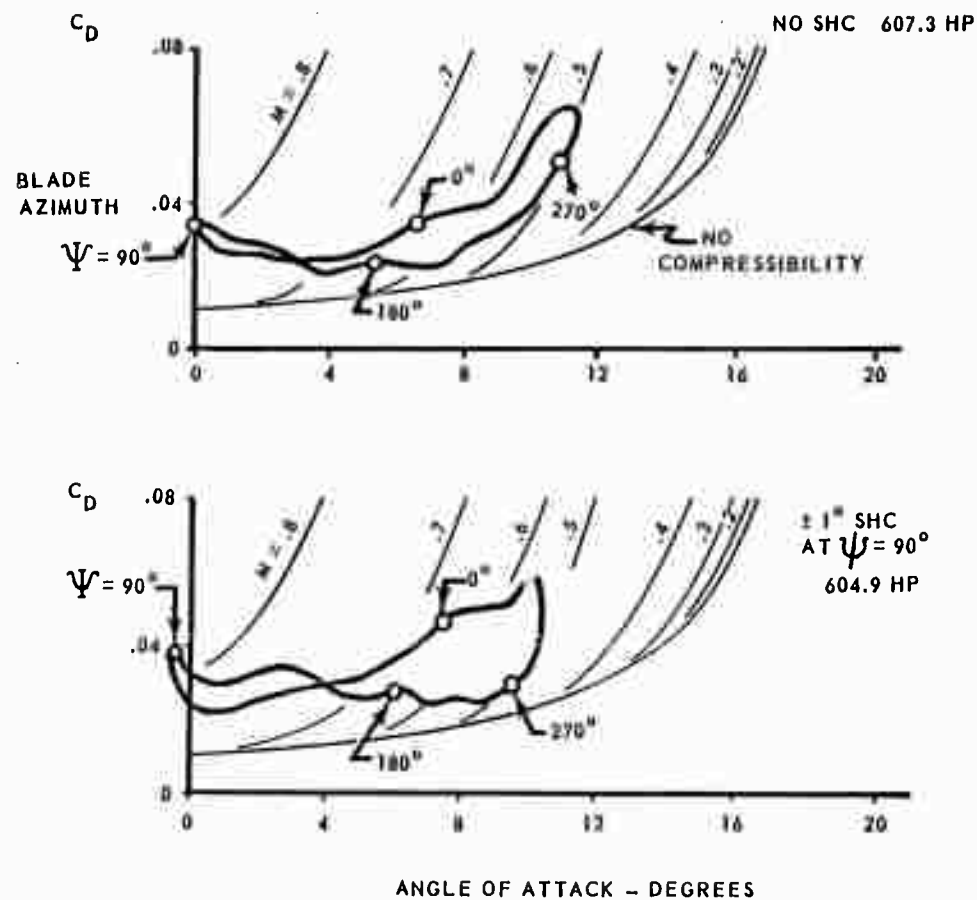
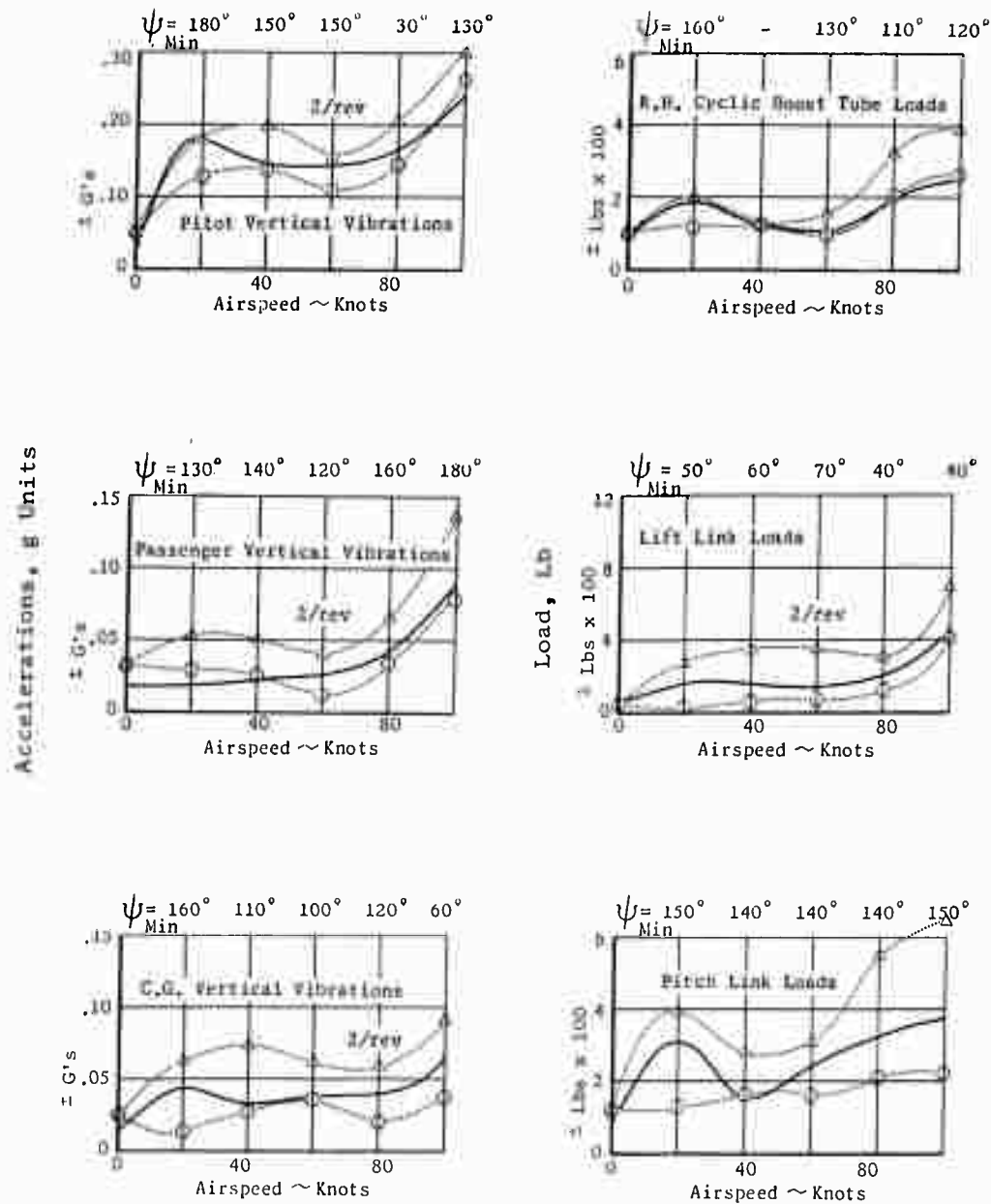


Figure 27. Computed Profile Drag Coefficients at the Tip of the Blade Versus Angle of Attack (UH-1A Rotor at 100 Knots)



Cross Plots of Data Presented in the Appendix

SHC Feathering Amplitude

- $\pm 0^\circ$
-○ $\pm 2.85^\circ$ @ $\psi_{\min}$ (azimuth phase angle for minimum vibrations and loads)
-△ $\pm 2.85^\circ$ @ $\psi_{\max}$ ($\psi_{\max} \approx \psi_{\min} \pm 90^\circ$)

Figure 28. The Effects of Second Harmonic Control on Vibrations and Oscillatory Loads as a Function of Airspeed.

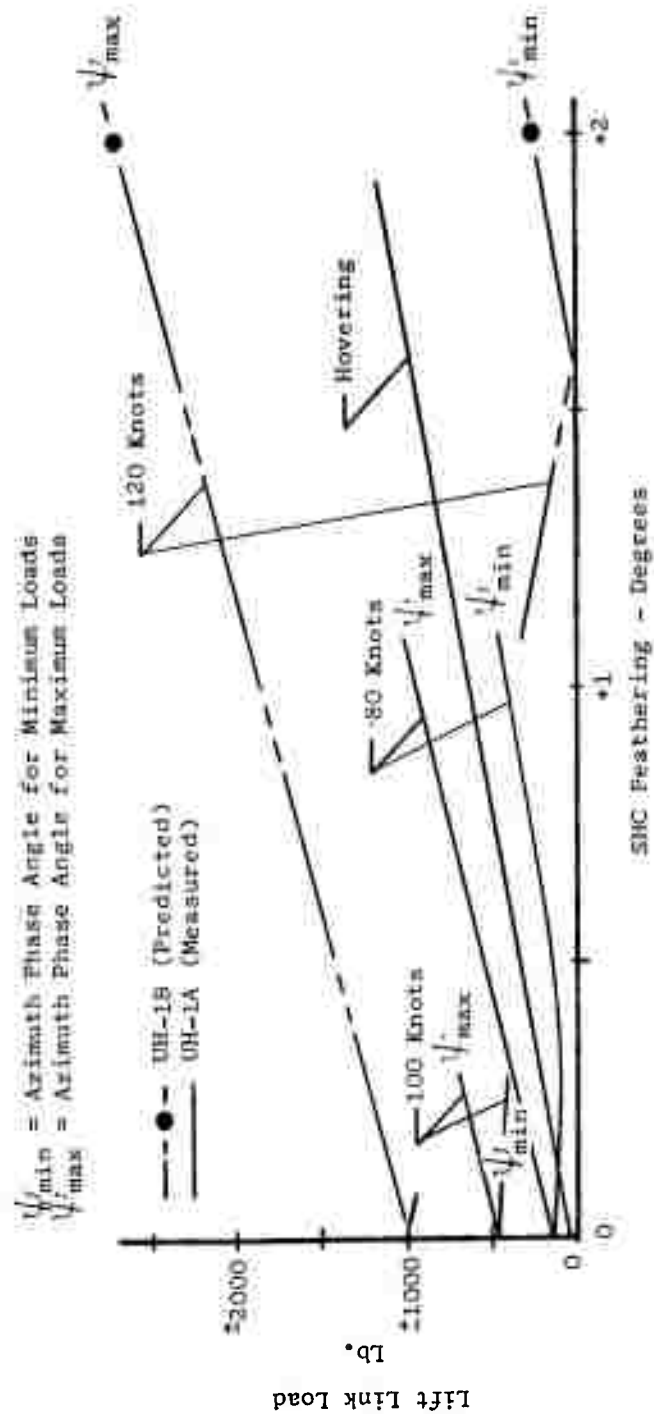


Figure 29. A Comparison of Predicted and Measured 2/Rev Lift Link Loads.

APPENDIX - Test Data

In this appendix the available measured data are presented. In some cases, when indicated, the data were analyzed to show only the 2/rev component. In other cases the peak-to-peak values are given. An attempt has been made to fair curves through the data points even when considerable scatter is apparent or when only a few data points are available. In such cases trends found for slightly different conditions (such as SHC amplitude) were often used to guide a curve through the few available data points. A faired curve could not justifiably be drawn through the scattered data in several presentations. It also was found that by plotting the data points from zero to 270 degrees SHC phase angle helped in fairing the curves, although zero to 180 degrees is sufficient to describe one full cycle of possible SHC phase angles.

The faired curves were used in preparing the summary data plots, in which for selected SHC phase settings the vibrations and loads are given as a function of SHC feathering amplitude. Similarly, summary plots have been constructed to determine the effect of speed.

The air load data are presented as a function of the rotor azimuth as explained in Section VII.

The following page lists all the figures given in this appendix. Some of these figures were used and repeated in the discussion of the results (Section IX).

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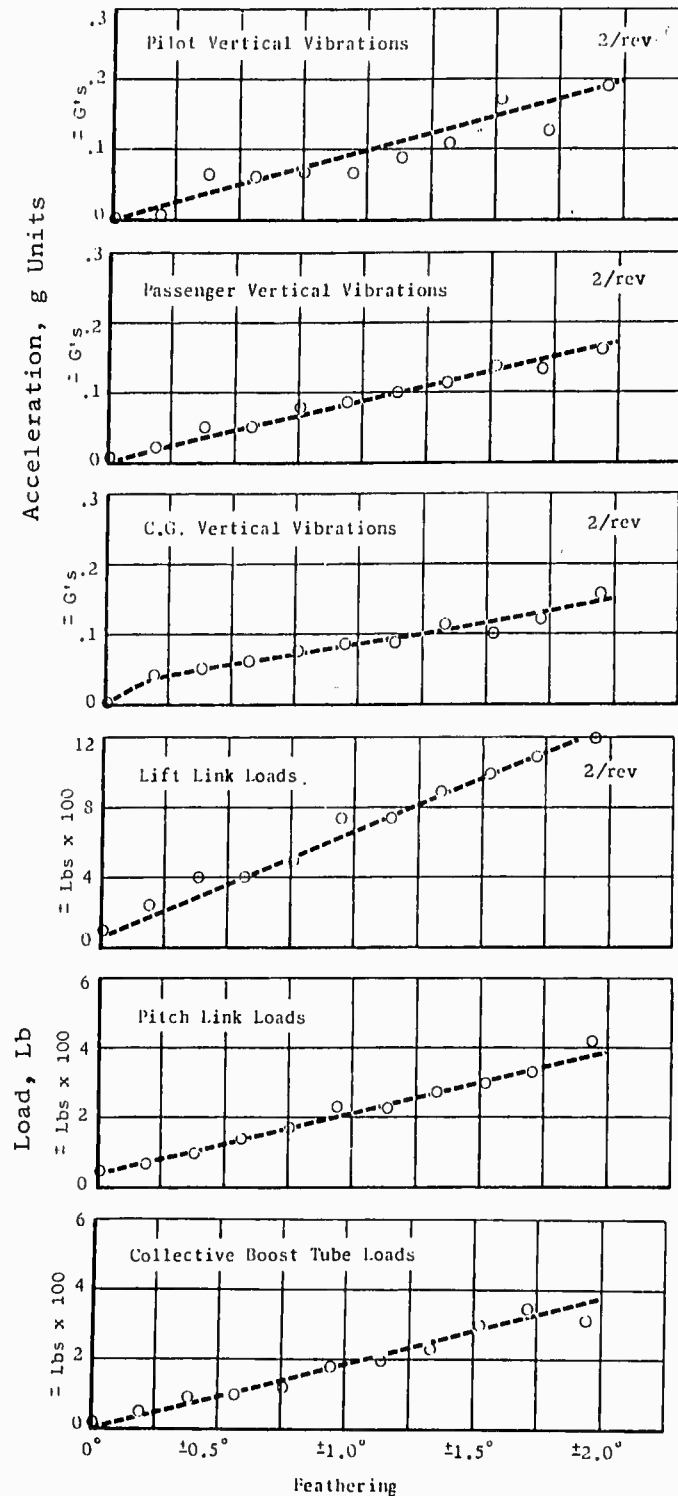


Figure 30, Vibrations and Oscillatory Loads Versus SHC Feathering, Hovering.

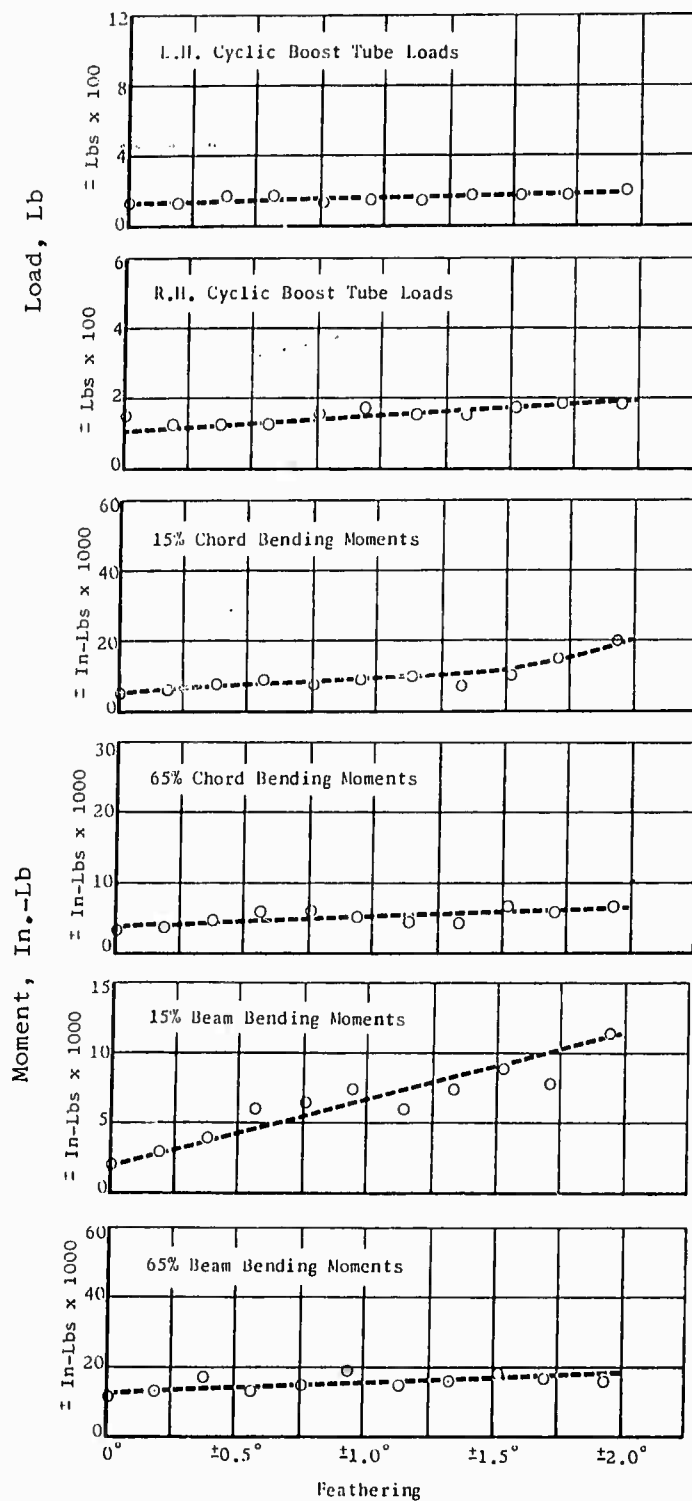


Figure 30 (cont'd). Vibrations and Oscillatory Loads Versus SHC Feathering, Hovering

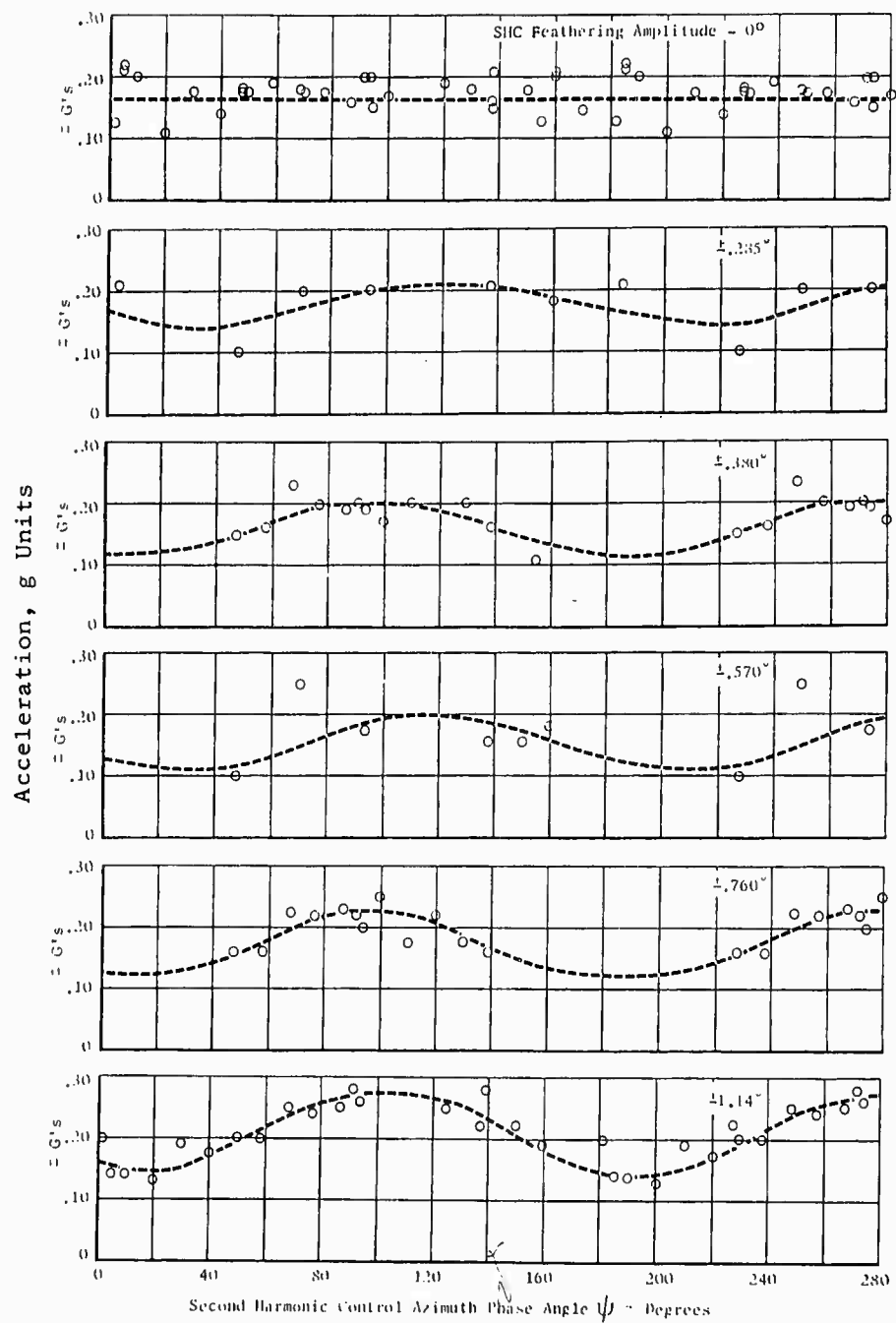


Figure 31. Pilot 2/Rev Vertical Vibrations
Versus SHC Phasing, 80 Knots Indicated Airspeed.

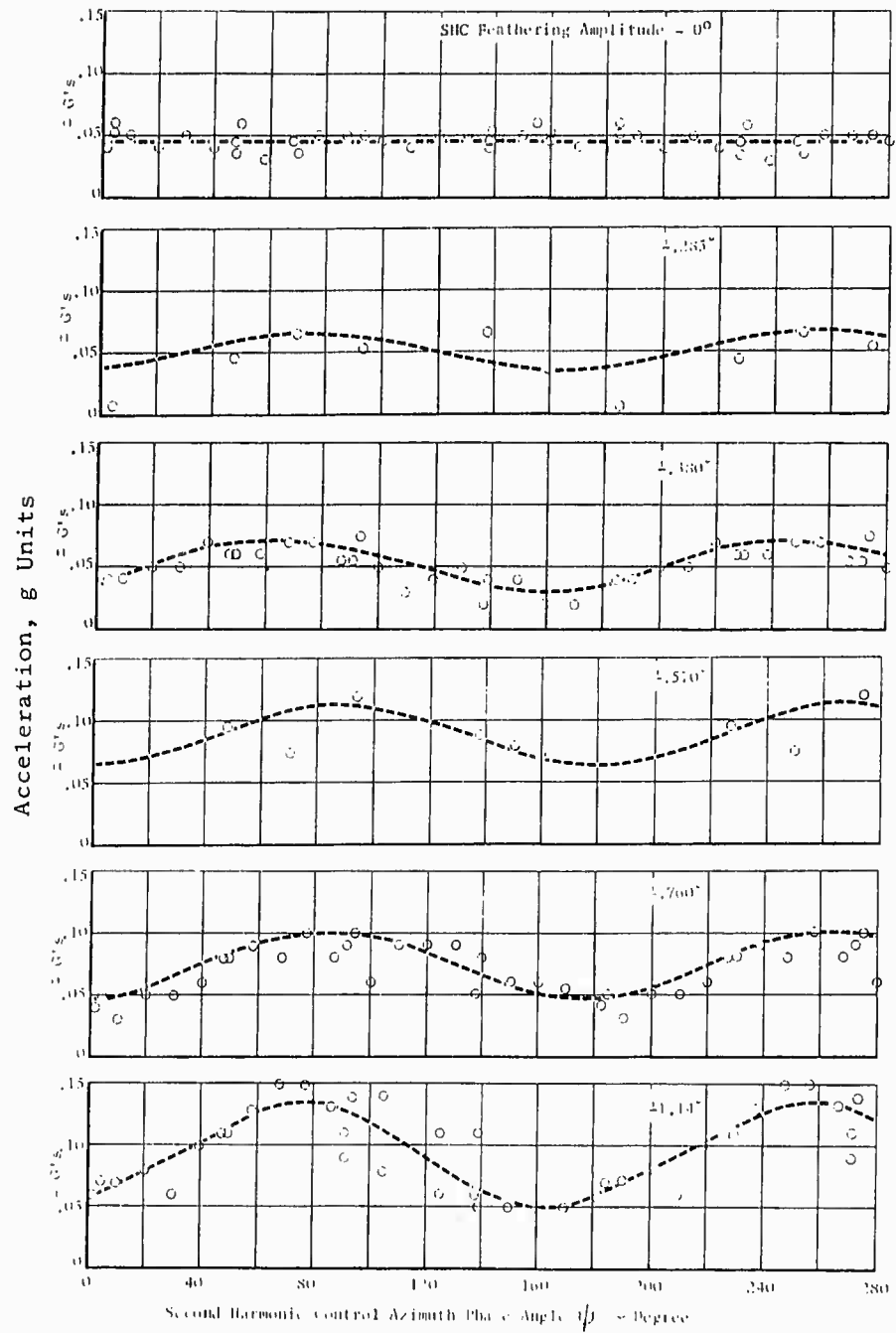


Figure 32. Passenger 2/Rev Vertical Vibrations Versus SHC Phasing, 80 Knots Indicated Airspeed.

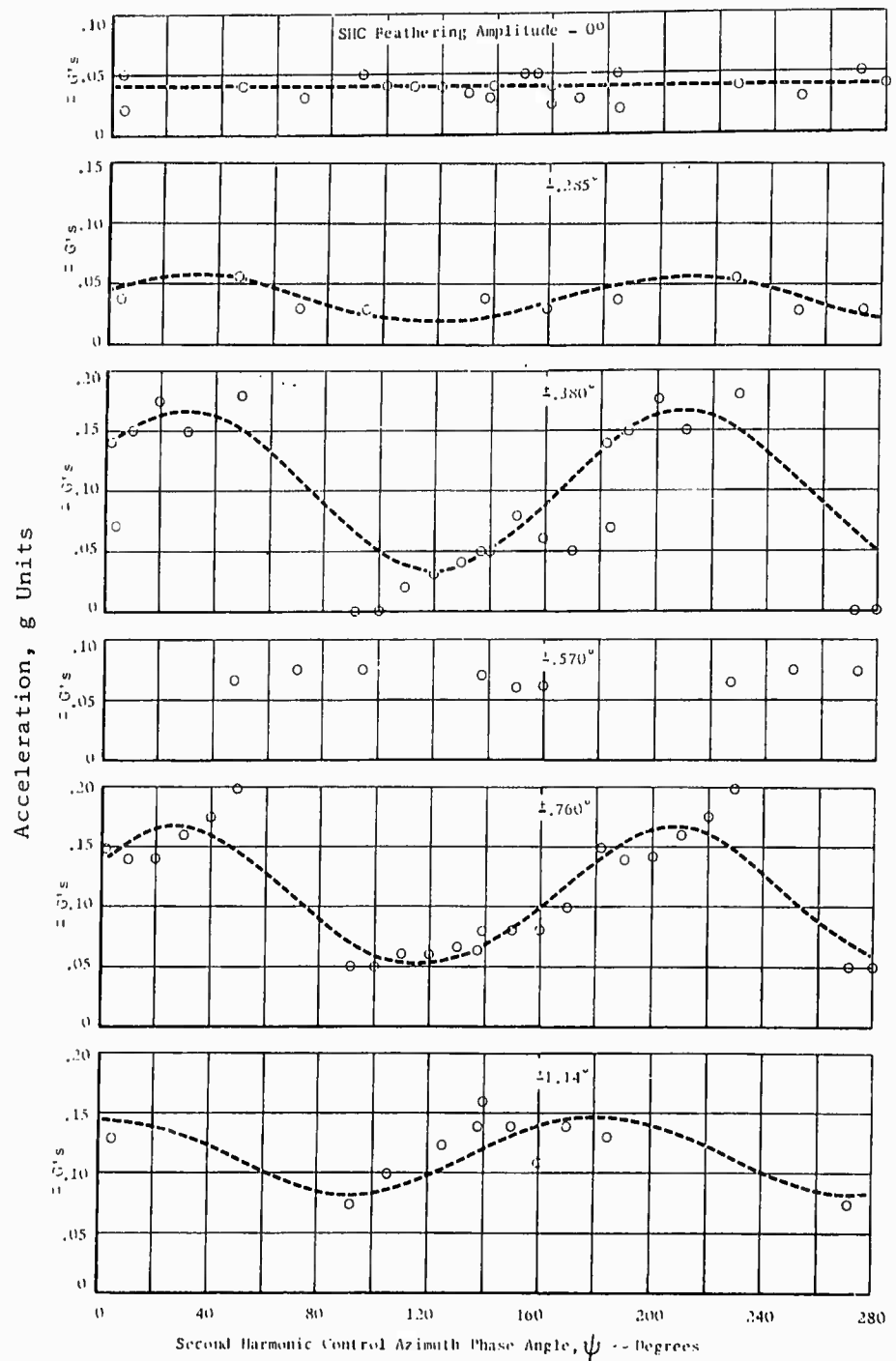


Figure 33. CG 2/Rev Vertical Vibrations
Versus SHC Phasing, 80 Knots Indicated
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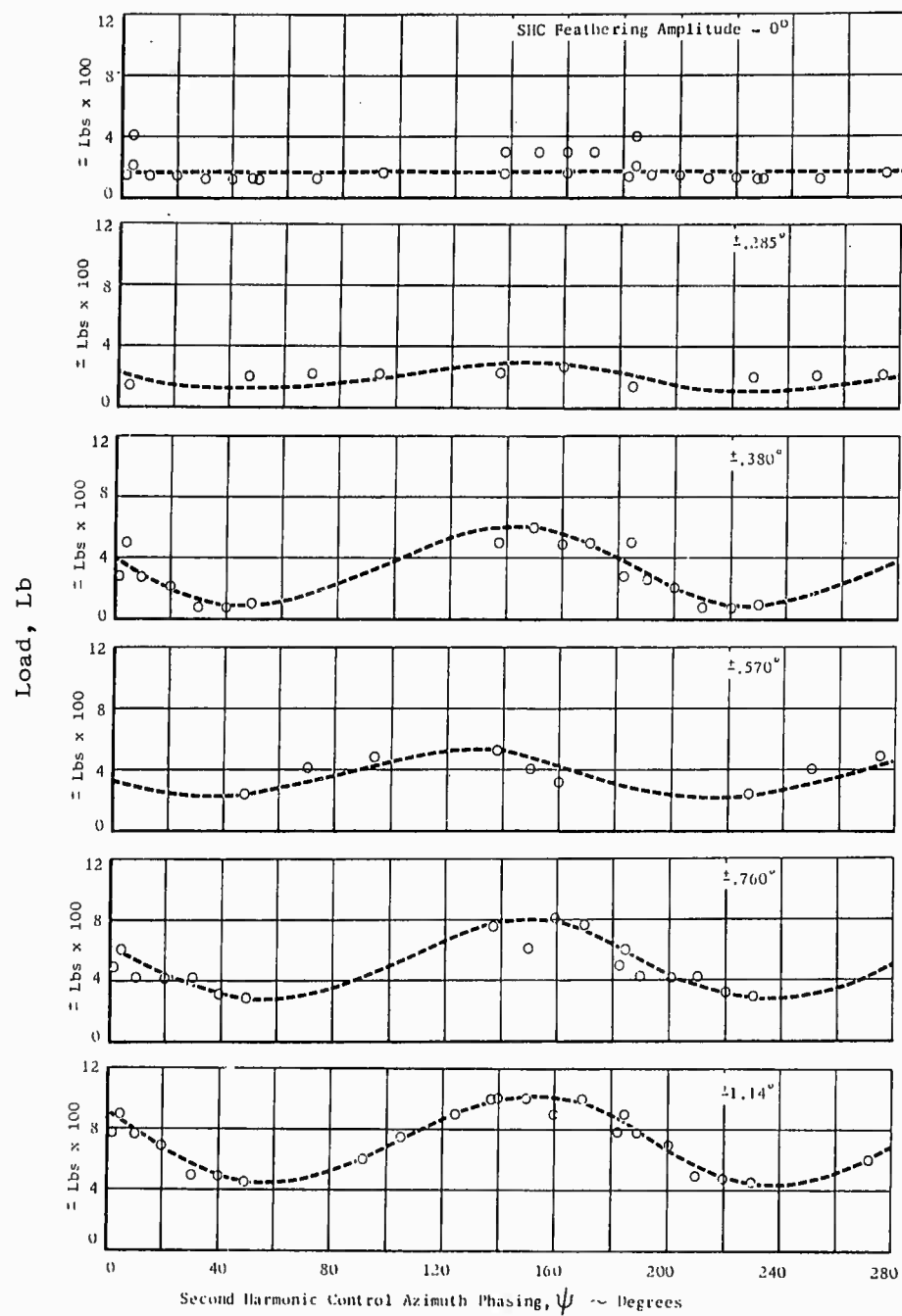


Figure 34. Lift Link 2/Rev Oscillatory Loads Versus SHC Phasing, 80 Knots Indicated Airspeed.

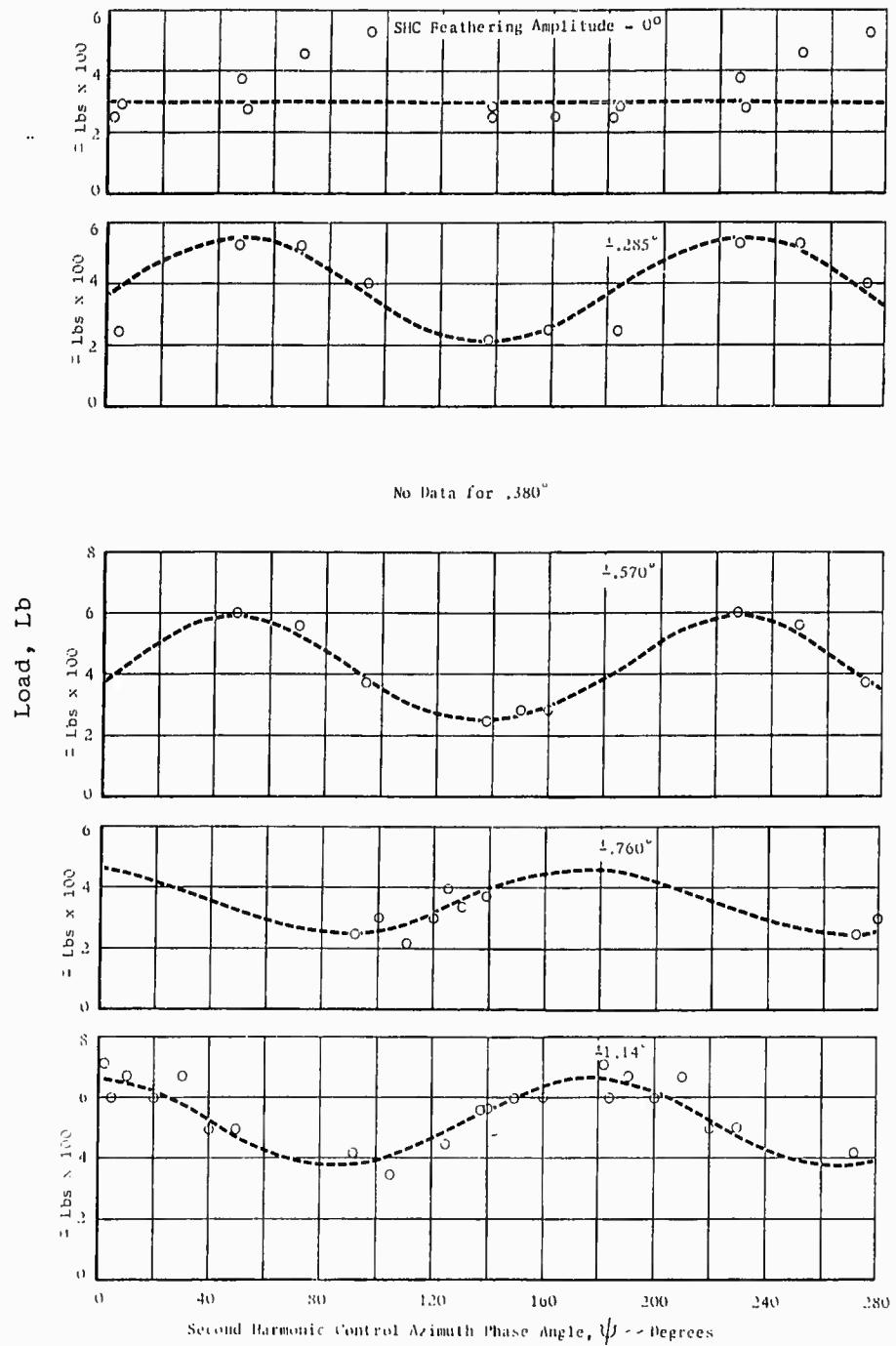


Figure 35. Pitch Link Oscillatory Loads
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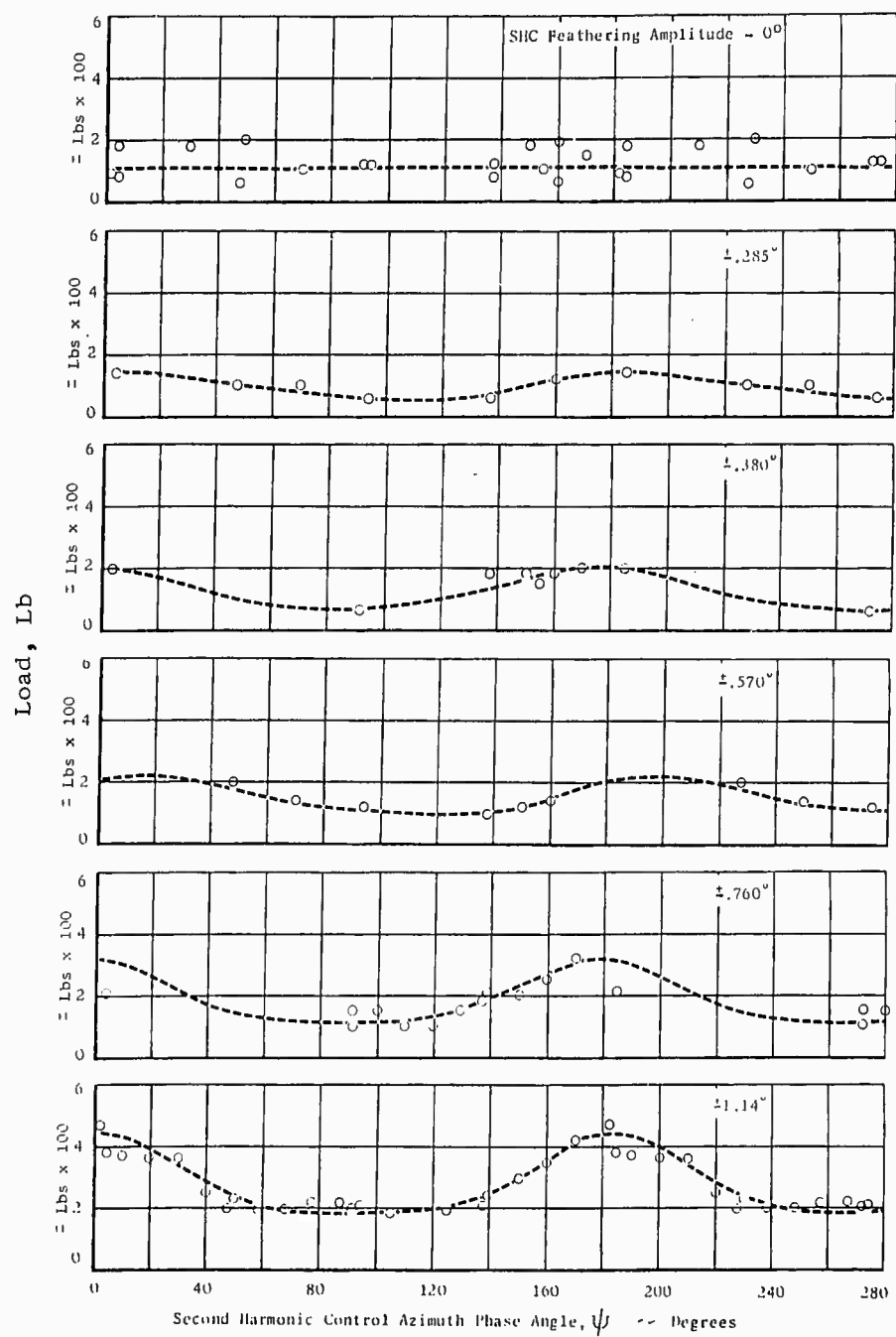


Figure 36. Collective Boost Tube Oscillatory Loads Versus SHC Phasing, 80 Knots Indicated Airspeed.

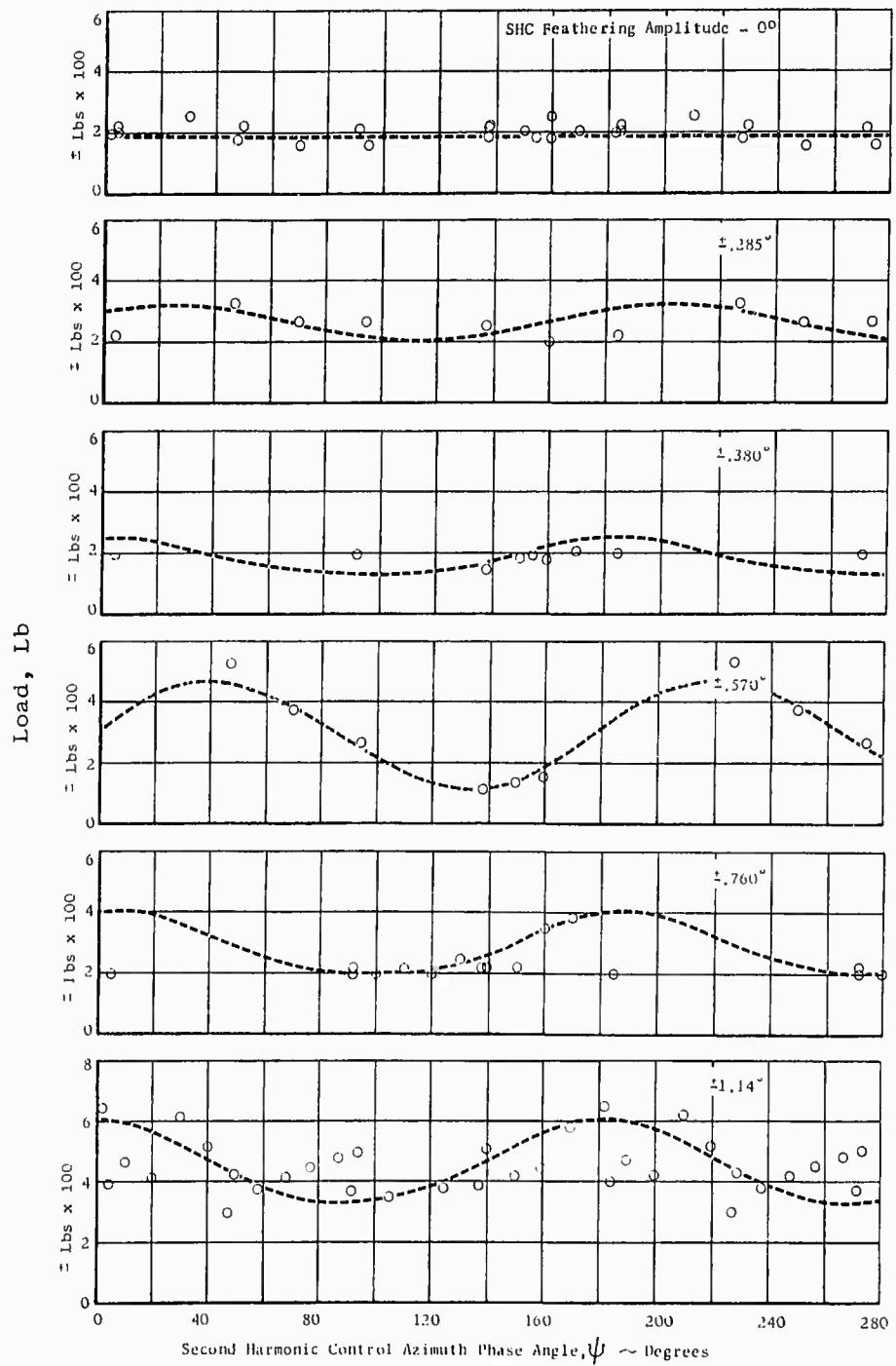


Figure 37. Right Hand Cyclic Boost Tube Oscillatory Loads Versus SHC Phasing, 80 Knots Indicated Airspeed.

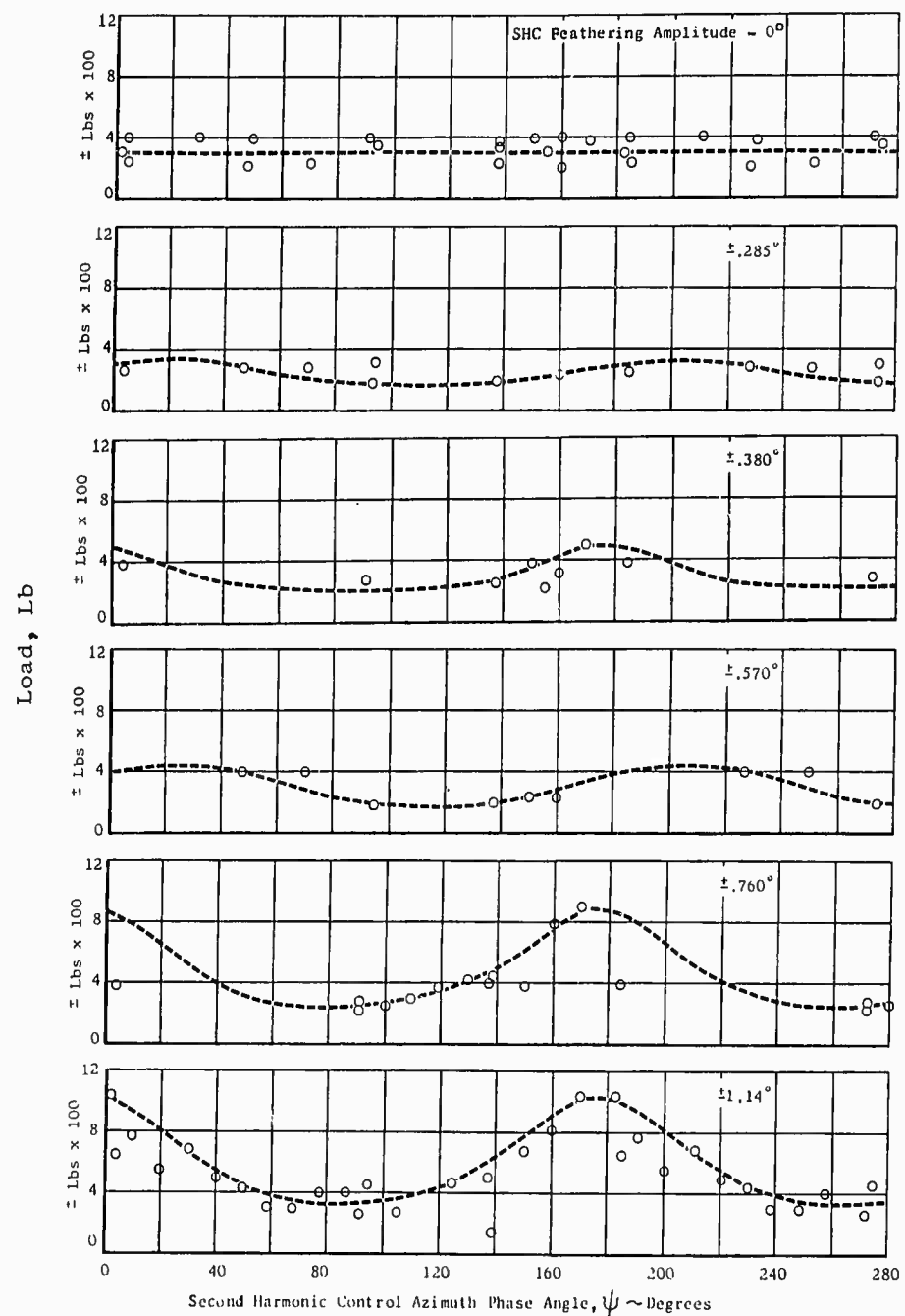


Figure 38. Left Hand Cyclic Boost Tube
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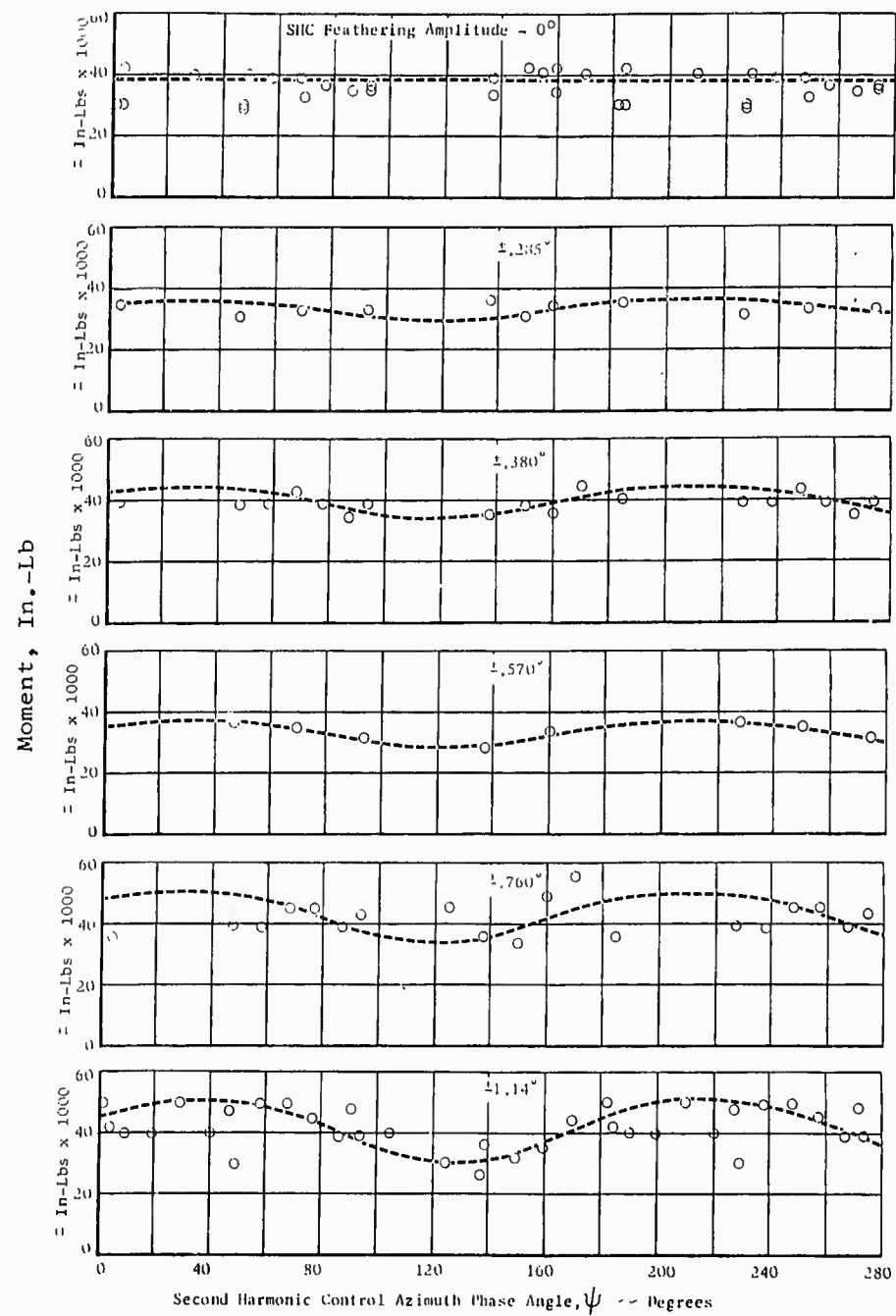
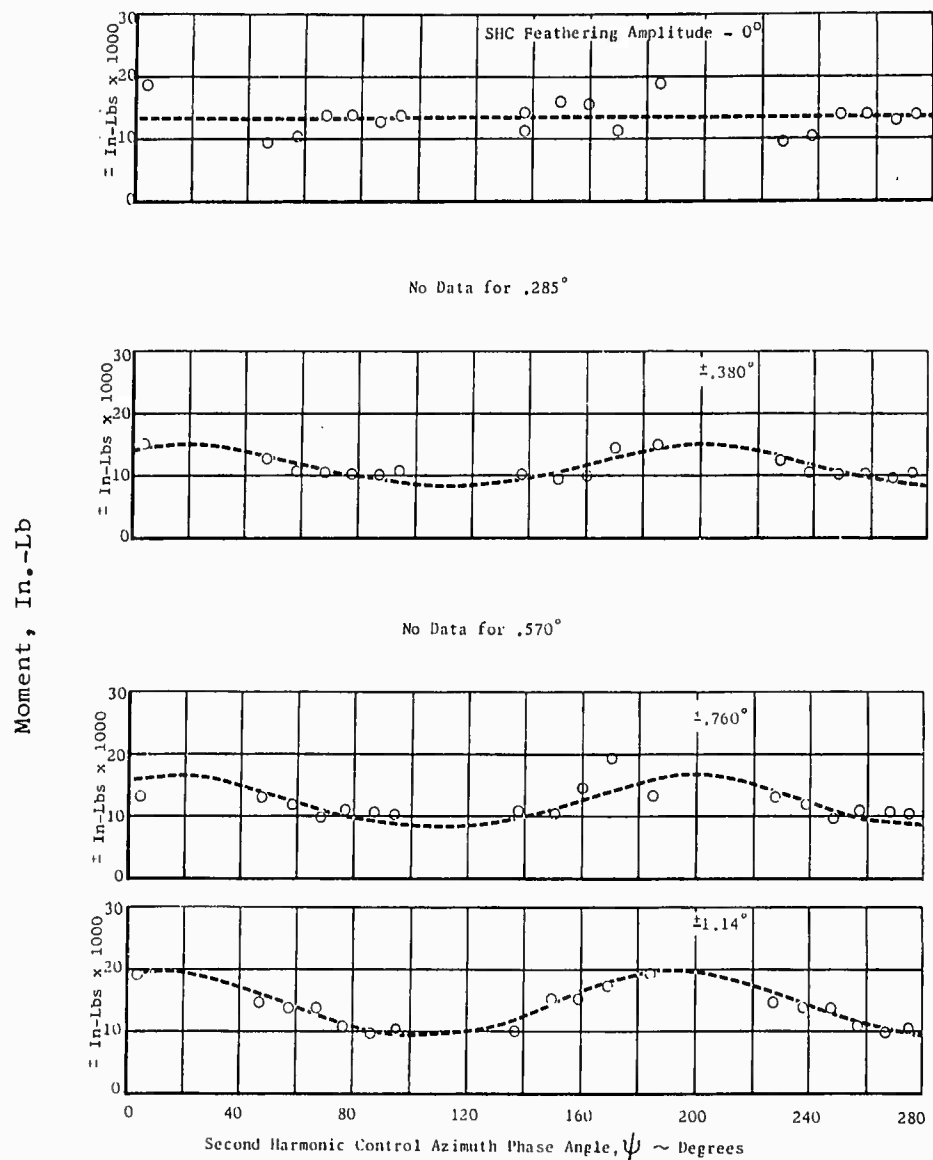


Figure 39. Chord Bending Oscillatory Moments at 15% Radius Versus SHC Phasing, 80 Knots Indicated Airspeed.



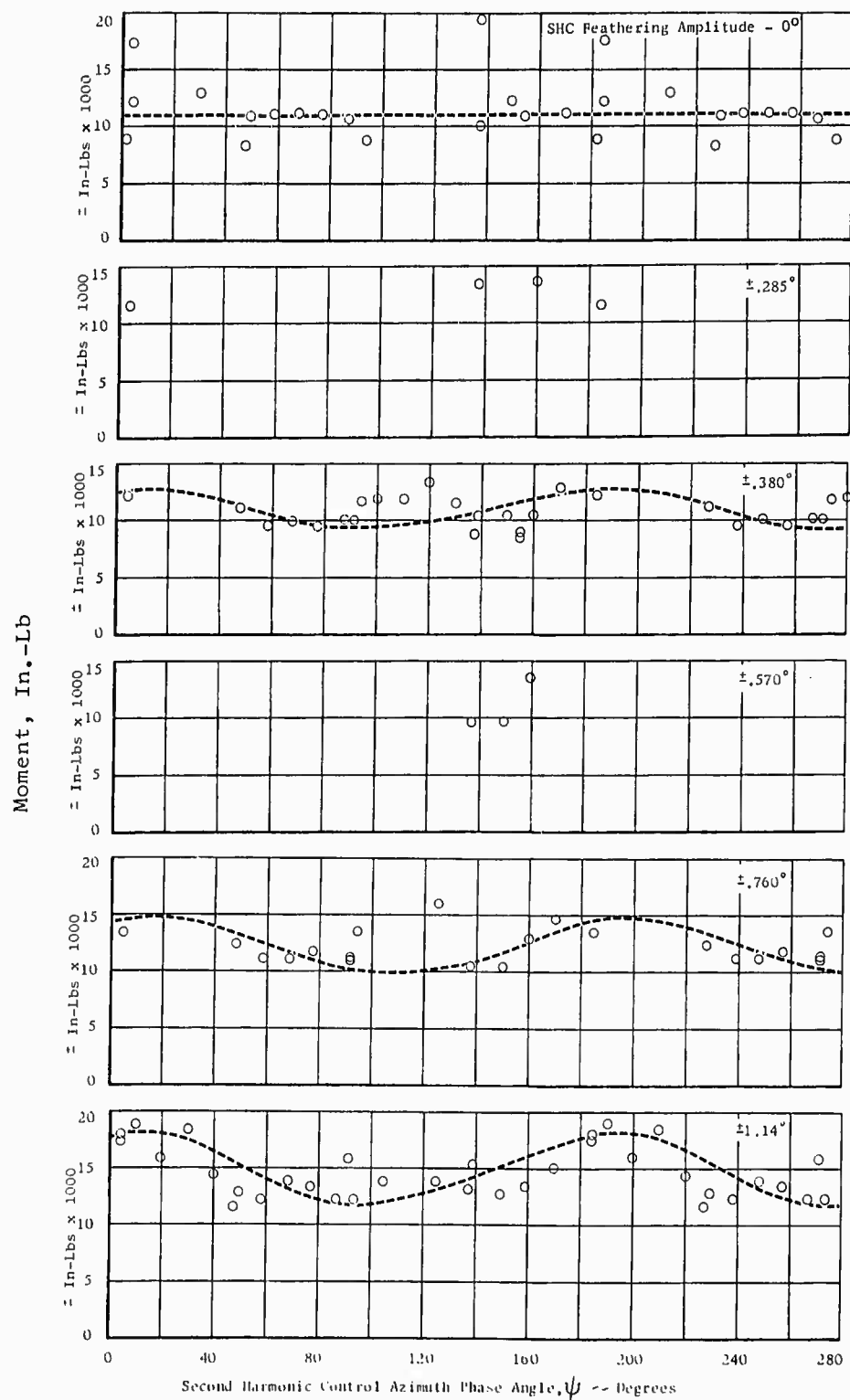


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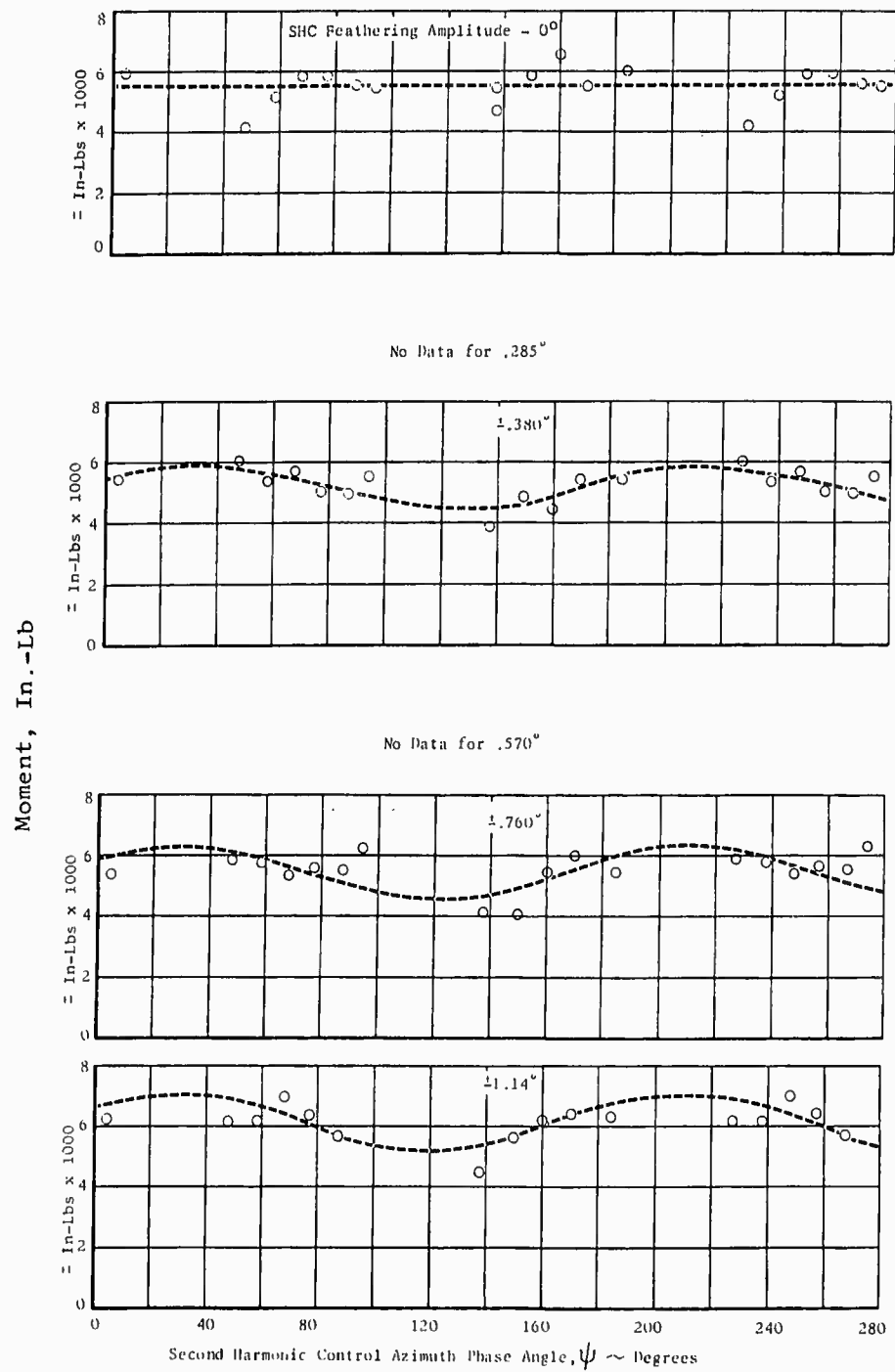


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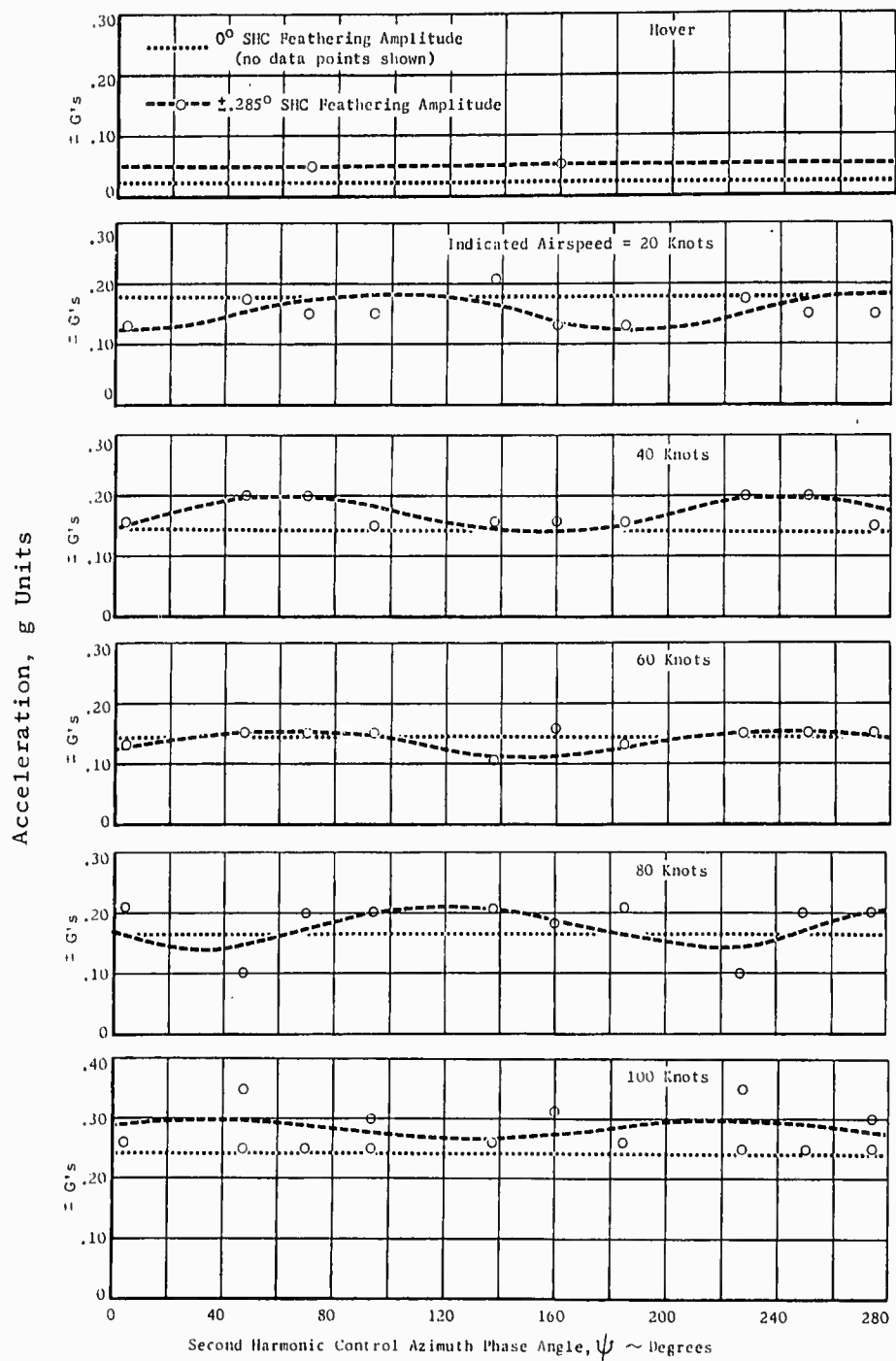


Figure 43. Pilot 2/Rev Vertical Vibrations
Versus SHC Phasing.

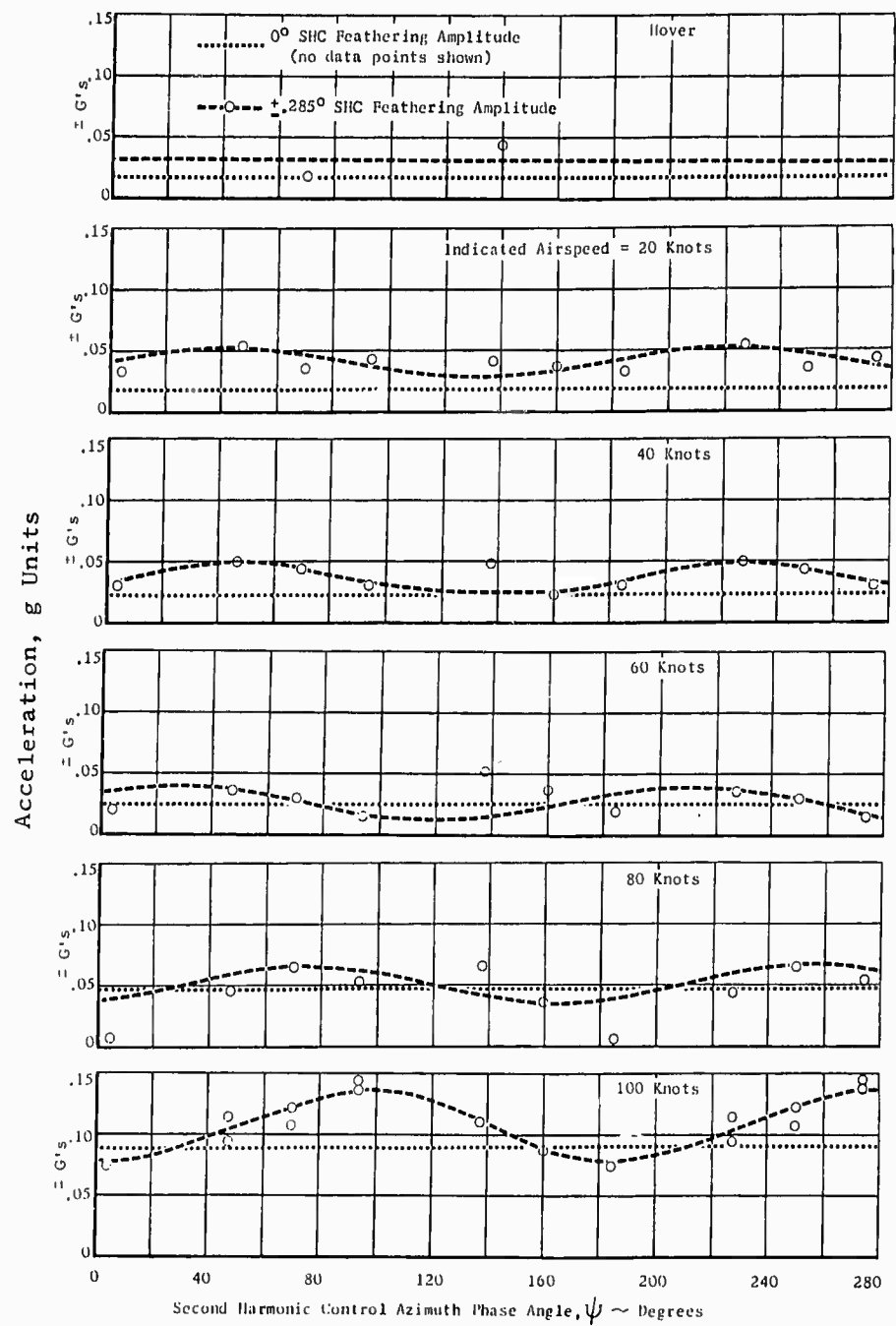


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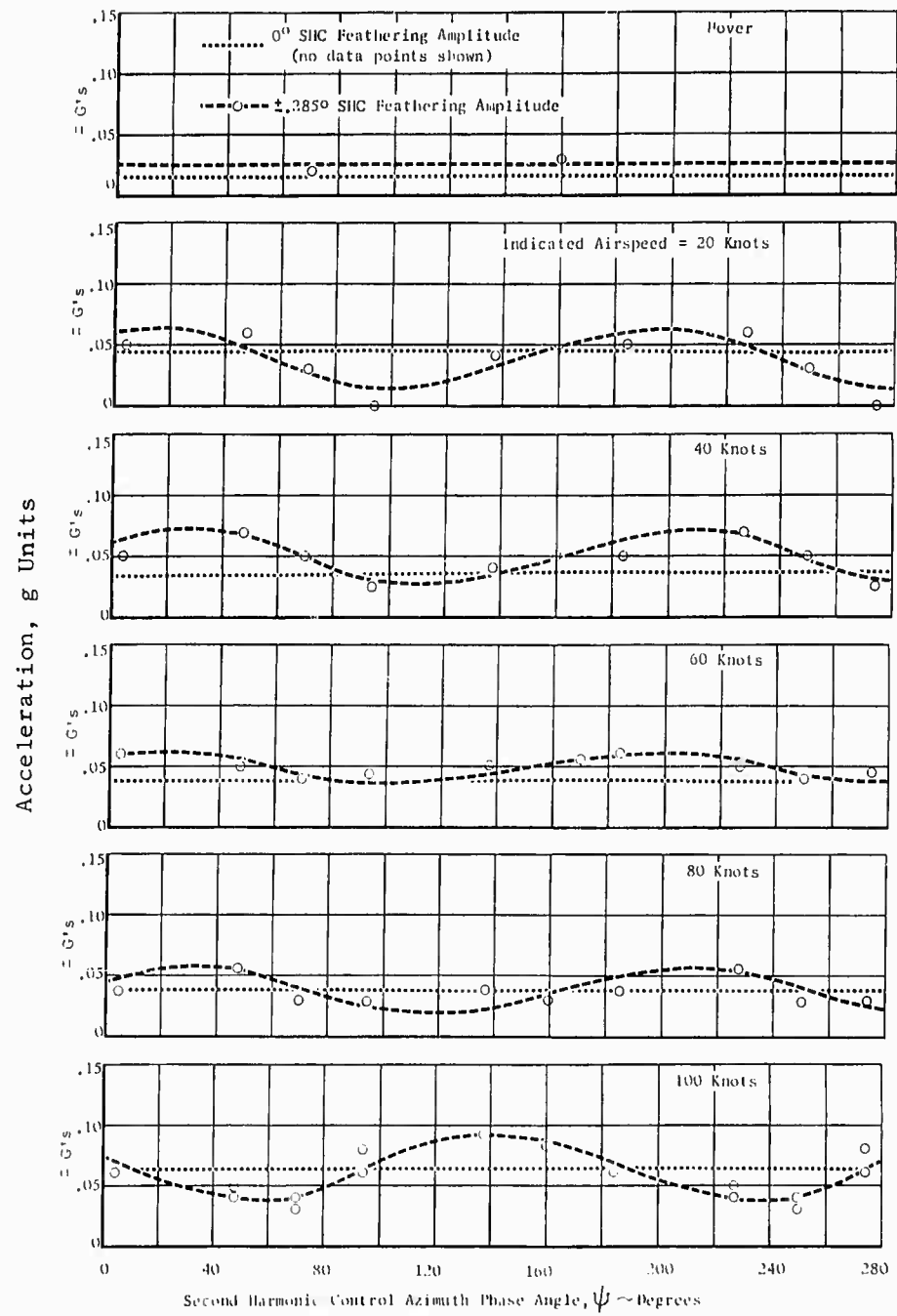


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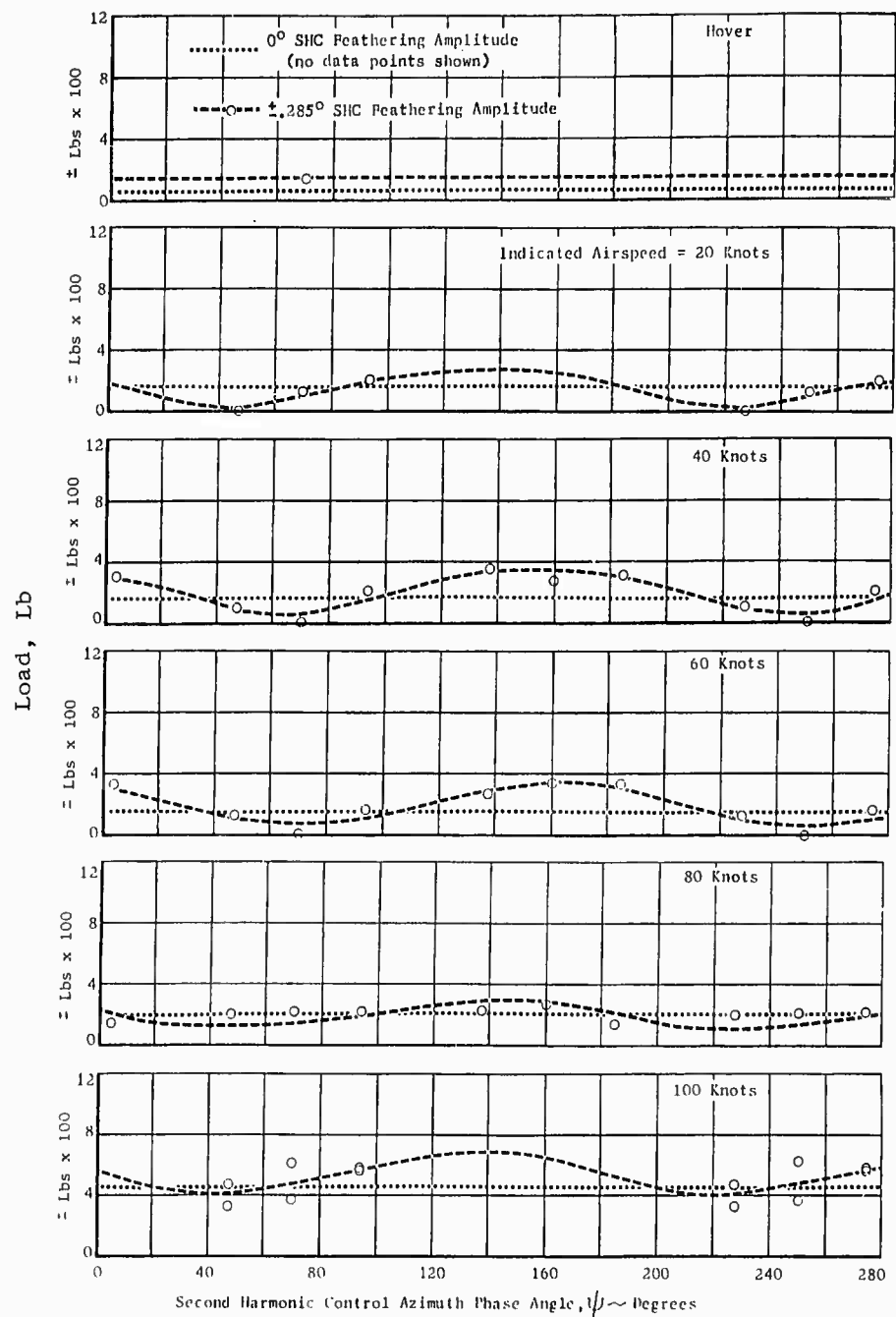


Figure 46. Lift Link 2/Rev Oscillatory Loads Versus SHC Phasing.

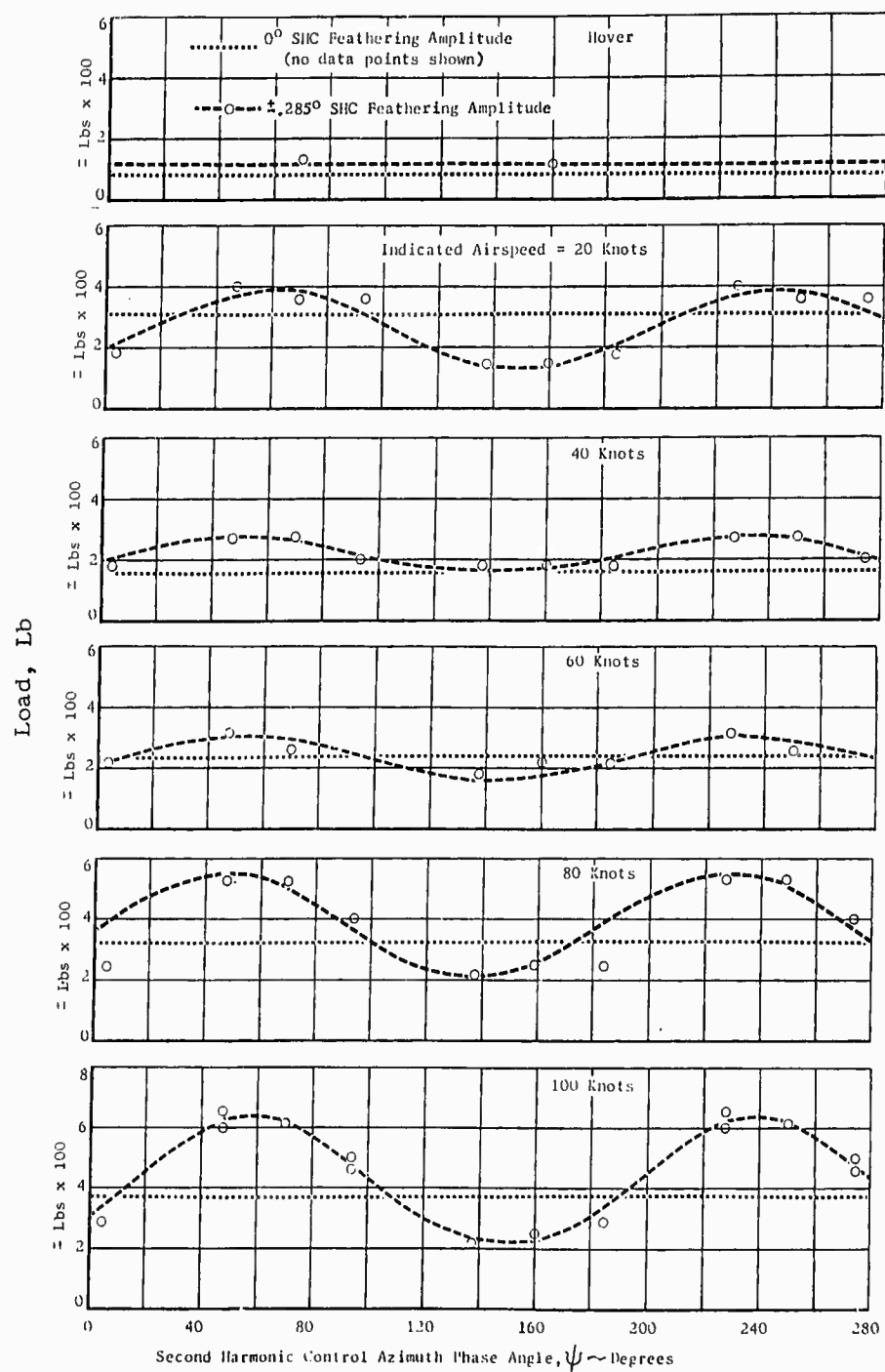


Figure 47. Pitch Link Oscillatory Loads Versus SHC Phasing.

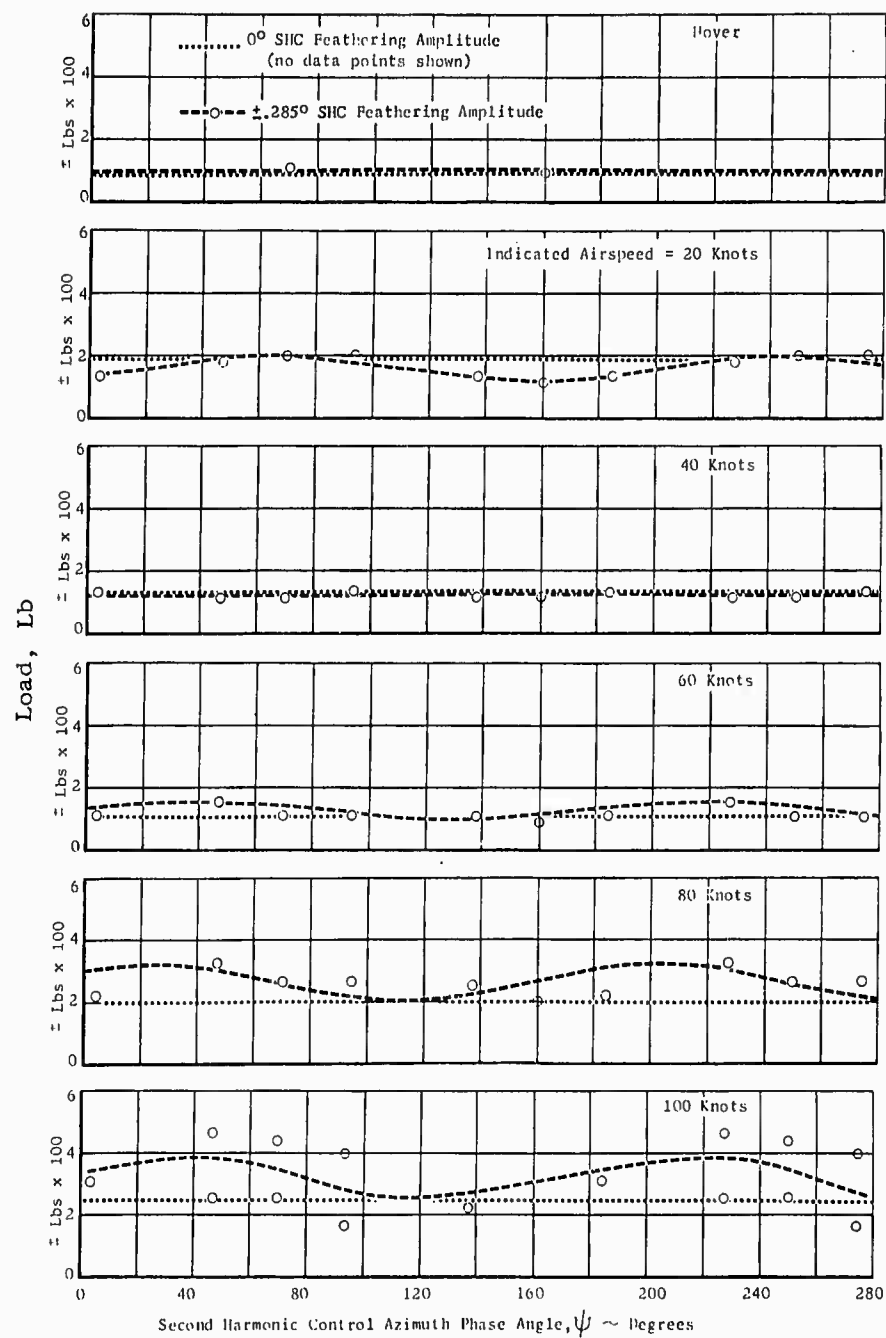
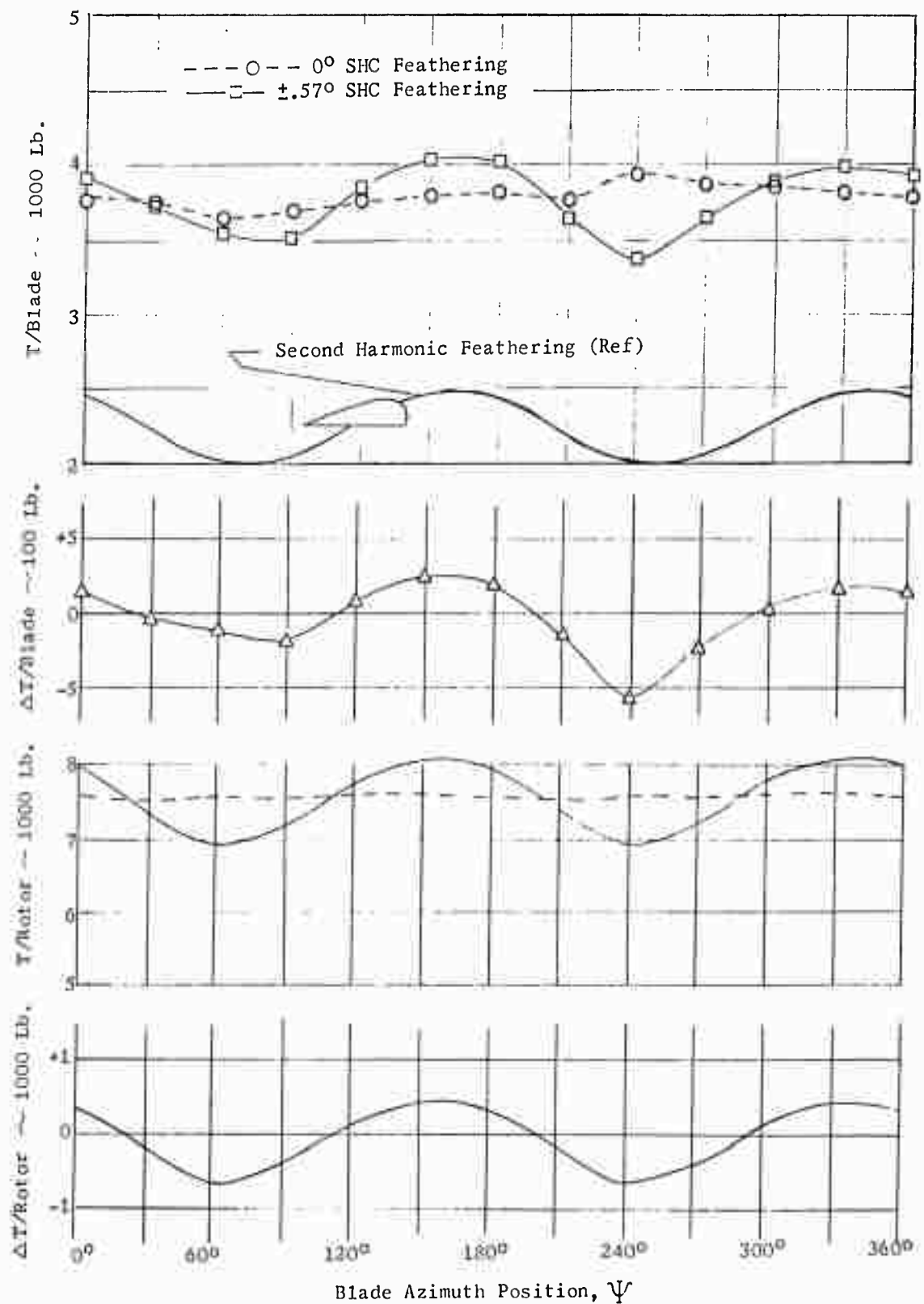


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Blade Azimuth Position, Ψ
 Figure 49. A Comparison of Thrust at 0 Knots
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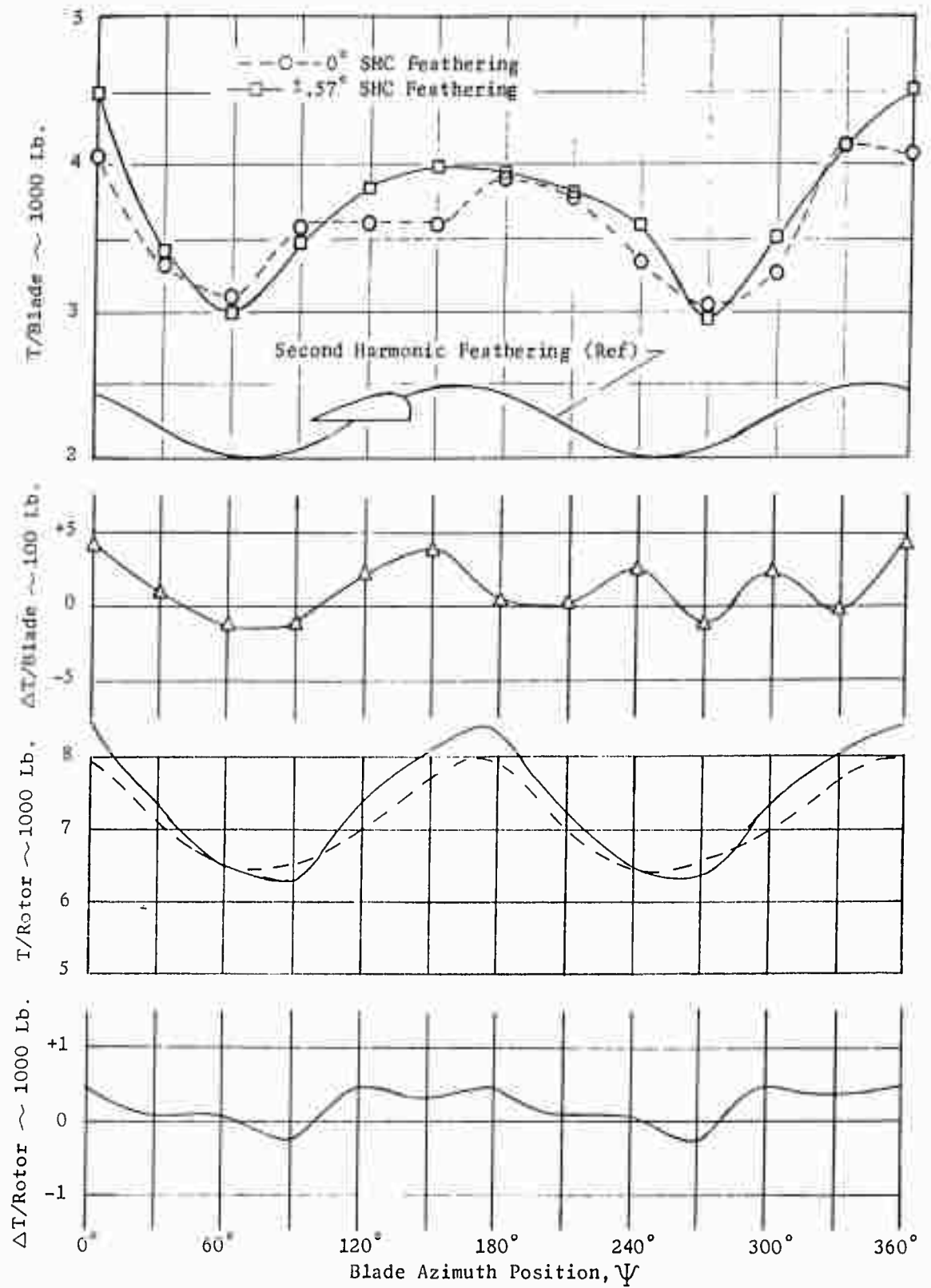
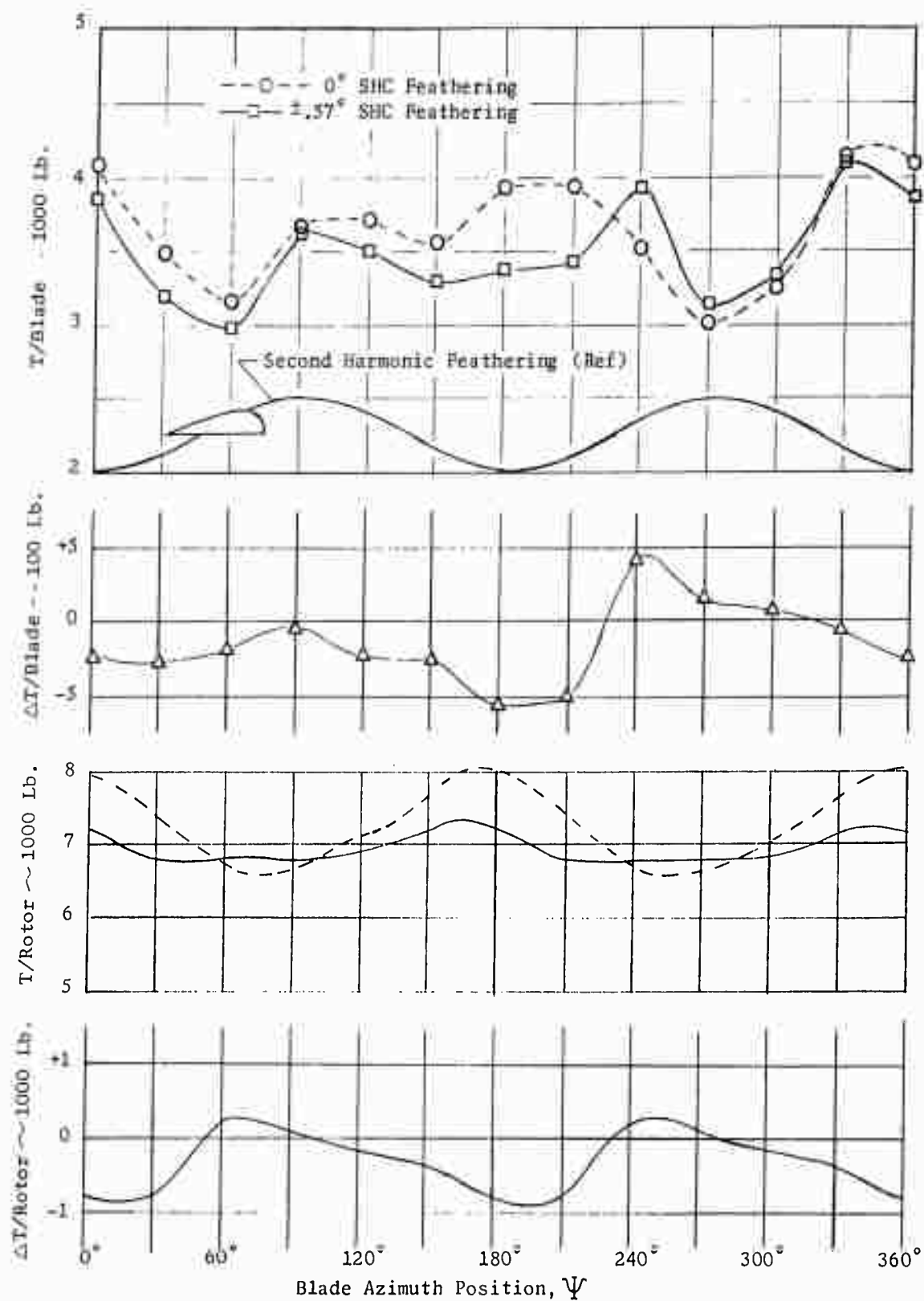


Figure 50. A Comparison of Thrust at 40 Knots for SHC Phasing at 70°



Blade Azimuth Position, Ψ
 Figure 51. A Comparison of Thrust
 at 40 Knots for SHC Phasing at 5%

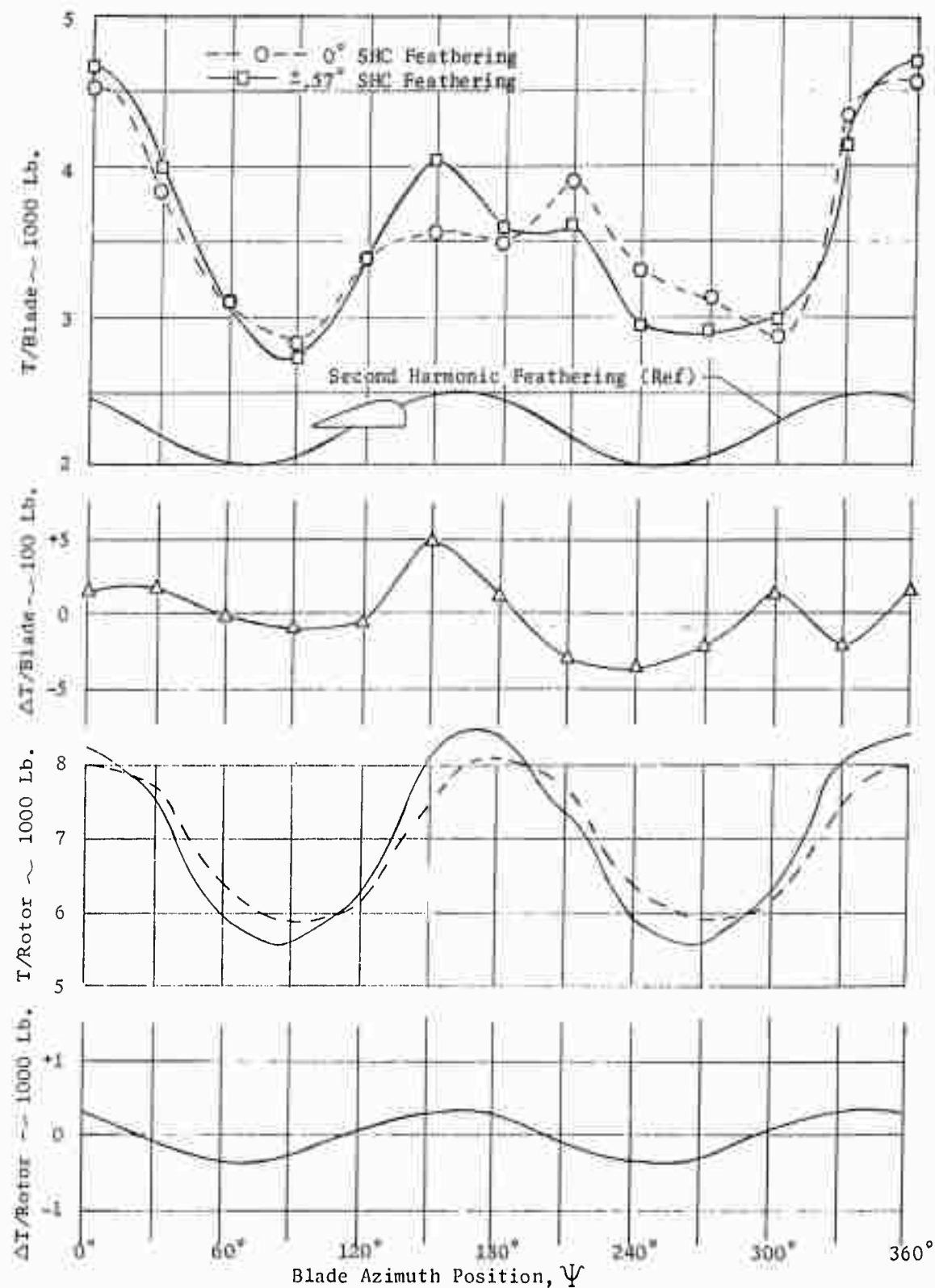
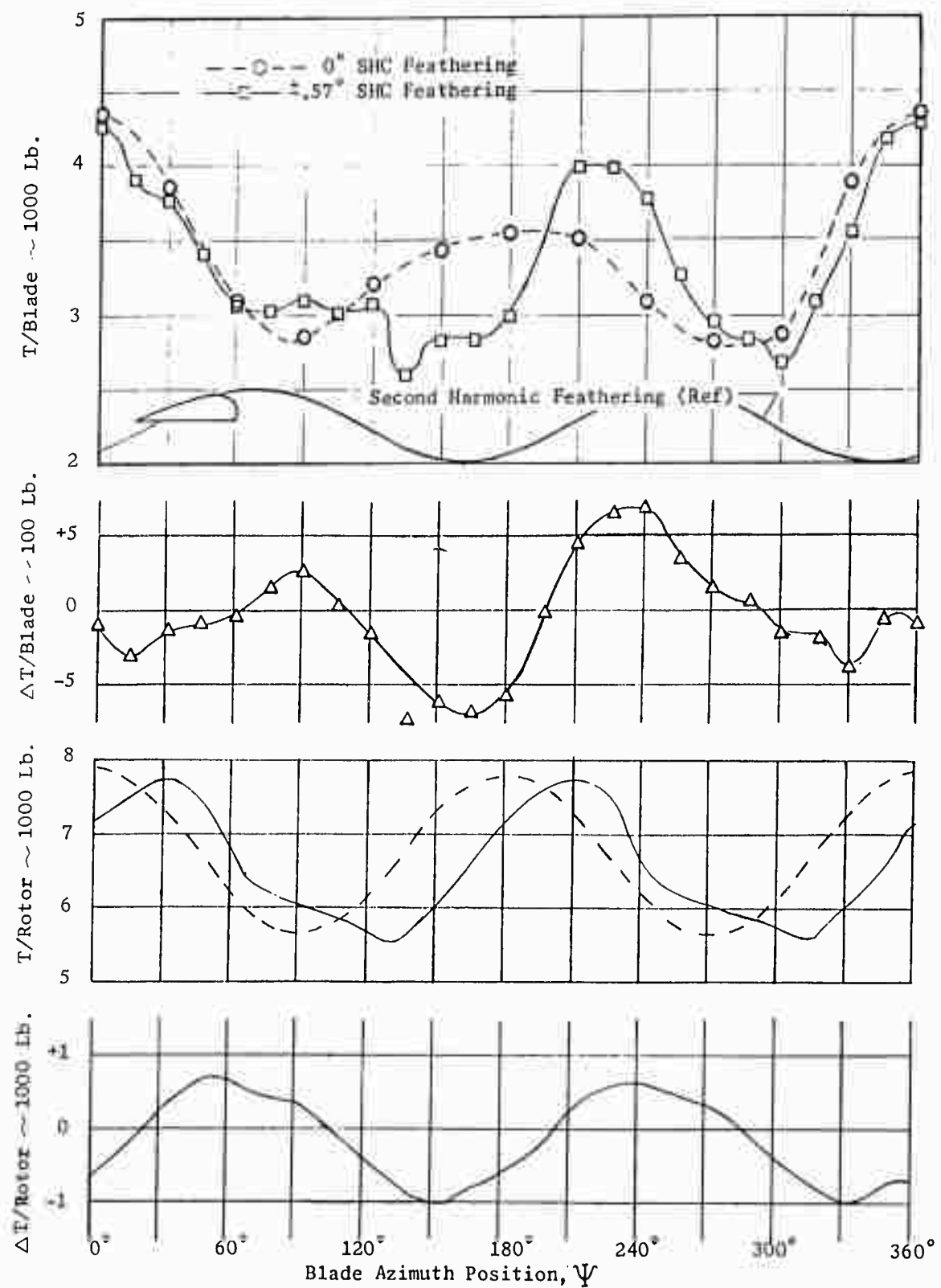


Figure 52. A Comparison of Thrust at 80 Knots for SHC Phasing at 70° .



Blade Azimuth Position, Ψ
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 at 80 Knots for SHC Phasing at 160°.

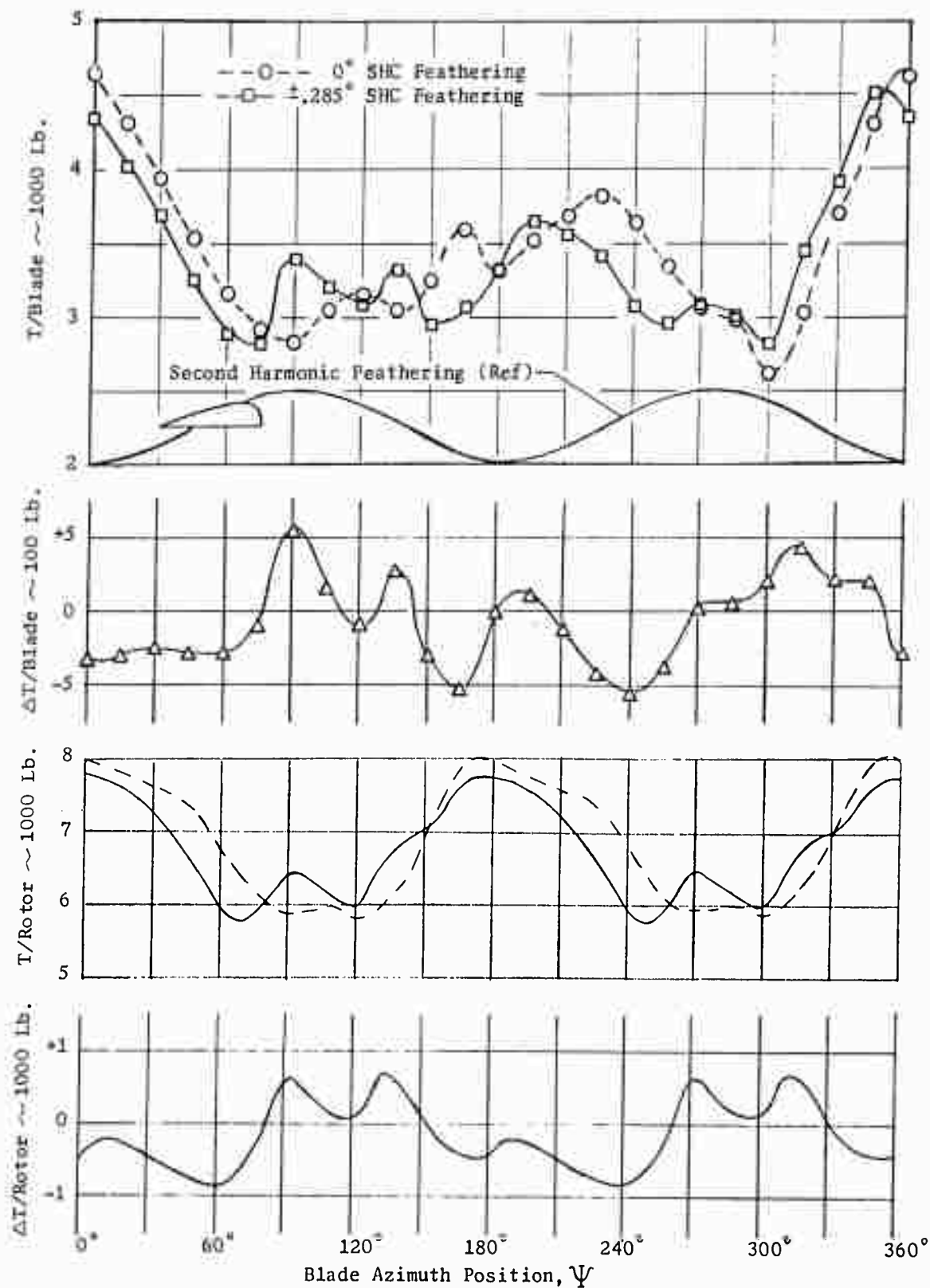


Figure 54. A Comparison of Thrust
at 80 Knots for SHC Phasing at 5° .

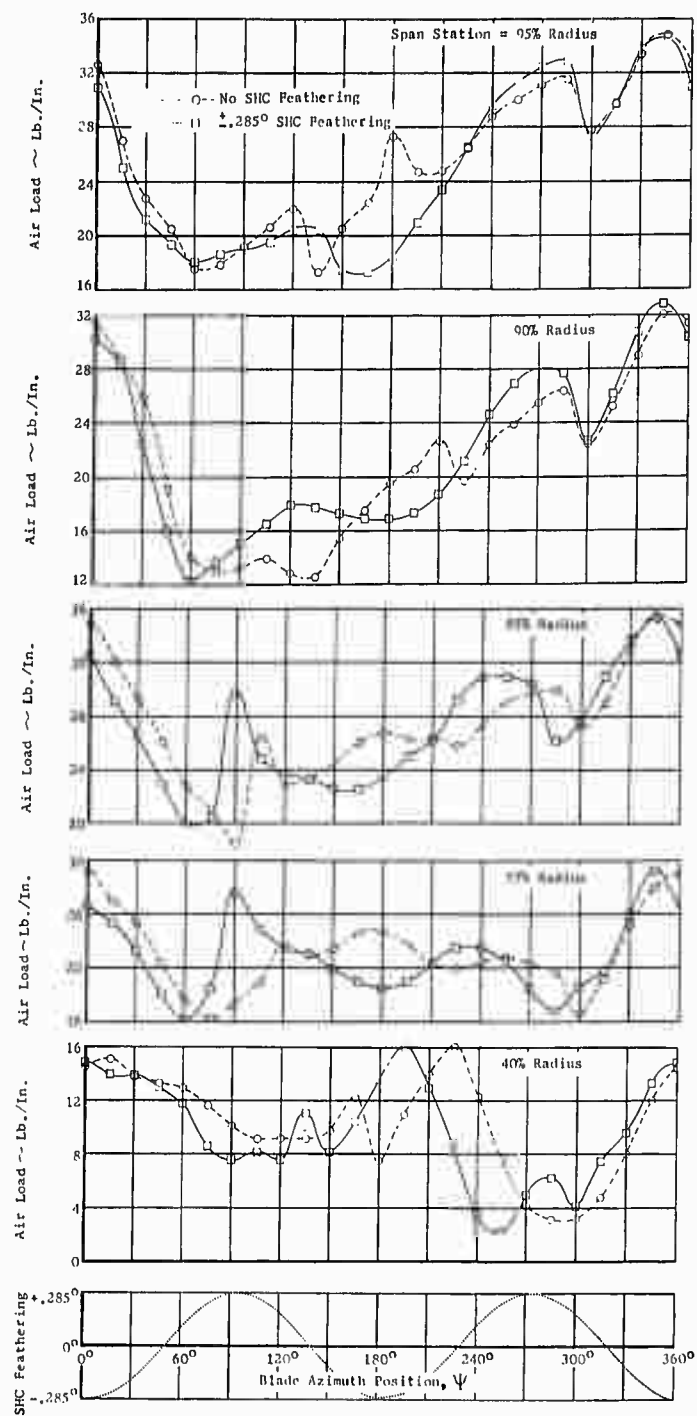


Figure 55. Air Load Versus Blade Azimuth at 80 Knots for SHC Phasing at 5°.

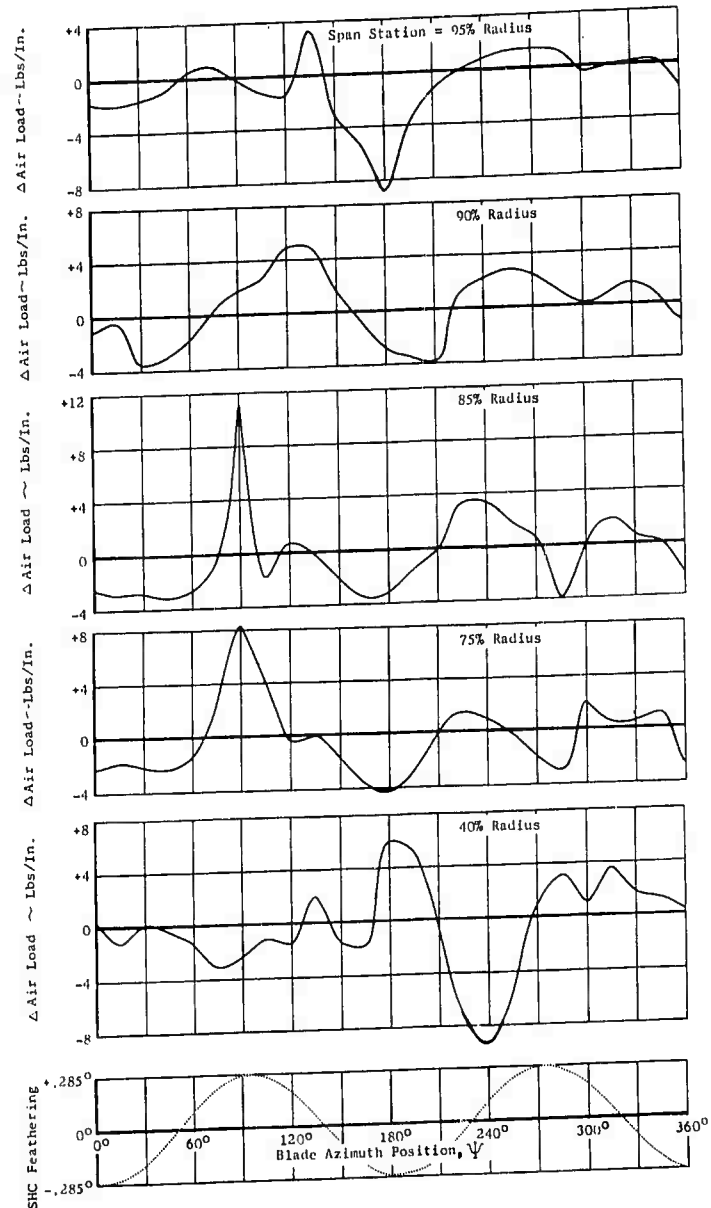


Figure 56. Change in Air Load Versus Blade Azimuth at 80 Knots for SHC Phasing at 5°.

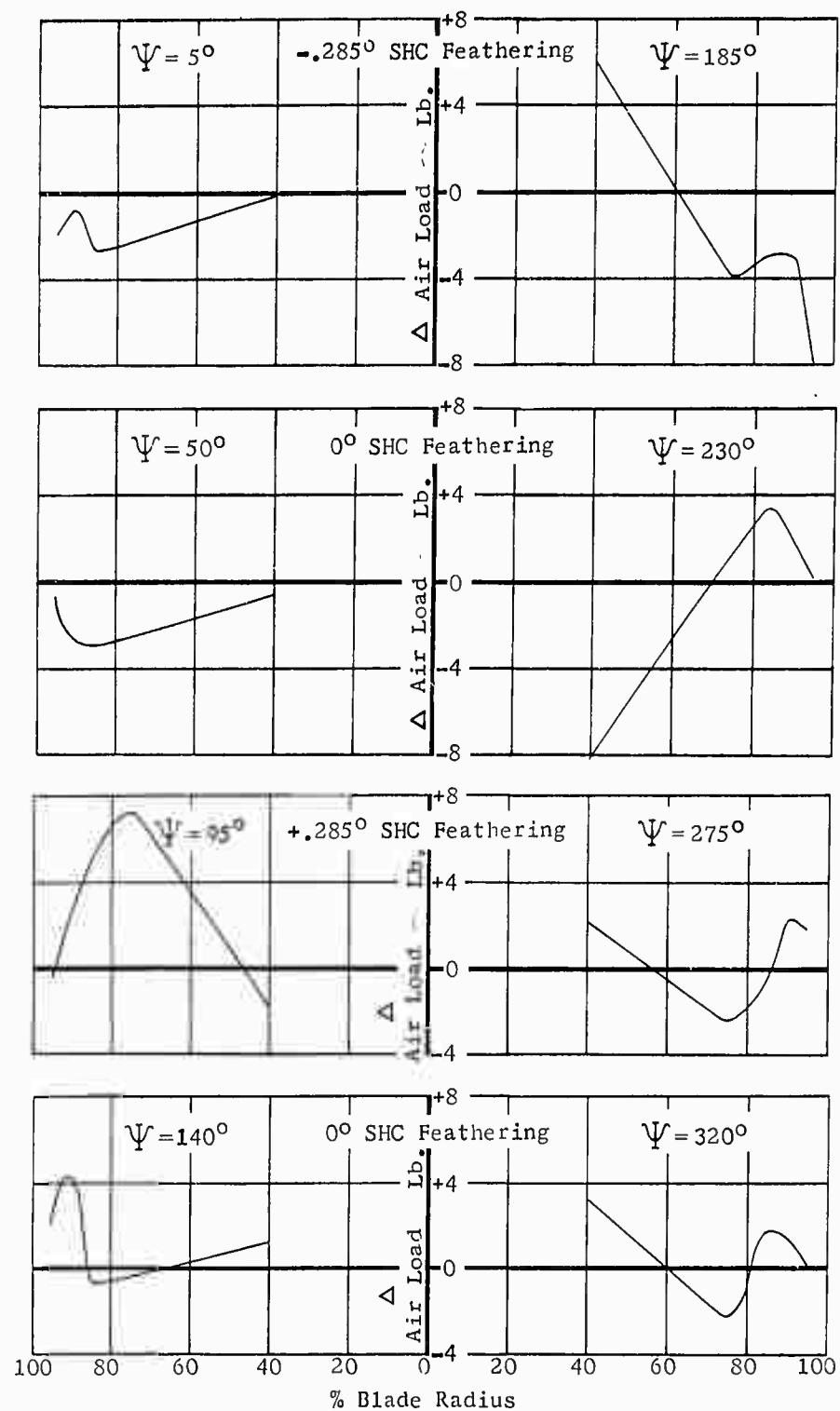


Figure 57. Change in Air Load Versus Blade Span at 80 Knots for SHC Phasing at 5° .

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The results of an experimental investigation of a second harmonic feathering device on the UH-1A helicopter are presented. The mechanism was expected to reduce vertical fuselage vibrations resulting from

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The test program included measurement of fuselage vibrations, rotor and control structural loads, and air loads acting on the blade as a function of second harmonic feathering amplitude and phasing. It is shown that the mechanism accomplished the anticipated changes in air load thrust pulsations. Benefits from blade stall delay or significant suppression of cockpit vertical vibrations and oscillatory rotor loads were not realized; however, for certain SHC settings vertical vibrations at the helicopter center of gravity and oscillatory blade and control loads were reduced simultaneously.

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3. Dynamic Rotor Air Loads Data
4. Contract DA 44-177-TC-806

Bell Helicopter Company, Fort Worth, Tex., AN EXPERIMENTAL INVESTIGATION OF A SECOND HARMONIC FEATHERING DEVICE ON THE UH-1A HELICOPTER - J. M. Drees and R. K. Wernicke - Report No. 537-099-001, June, 1963 81 pp, 57 illus, 3 tables (Contract DA 44-177-TC-806) Task 98 38-01-019-04, TCREC Technical Report 62-109

The results of an experimental investigation of a second harmonic feathering device on the UH-1A helicopter are presented. The mechanism was expected to reduce vertical fuselage vibrations resulting from

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2/rev rotor thrust pulsations and also to delay retreating blade stall. Both the amplitude of the second harmonic feathering produced and the phase relationship of this feathering to the blade azimuth position were controllable in flight.

The test program included measurement of fuselage vibrations, rotor and control structural loads, and air loads acting on the blade as a function of second harmonic feathering amplitude and phasing. It is shown that the mechanism accomplished the anticipated changes in air load thrust pulsations. Benefits from blade stall delay or significant suppression of cockpit vertical vibrations and oscillatory rotor loads were not realized; however, for certain SHC settings vertical vibrations at the helicopter center of gravity and oscillatory blade and control loads were reduced simultaneously.

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