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CONTENTS

STRUCTURAL AND CONTROL DYNAMICS
Norman C. Taylor, Vertol Div., Boeing Co. - Chairman

Second Harmonic Control
by Rodney K. Wernicke, and Jan M. Drees, Bell Helicopter Co. 1

Application of an Absorber to Reduce Helicopter Vibration Levels
by Charles W. Ellis, and Robert Jones, Kaman Aircraft Corp. 8

Loads Prediction for Structural Rigid Rotor Systems
by J. E. Rhodes, and John A. Gaidelis, Lockheed-California Co. 17

A Method for Determining the Fully Coupled Aeroelastic Response of Helicopter Rotor Blades
by Edward R. Wood, and Kurt D. Hilzinger, Sikorsky Aircraft Div. 28

AERODYNAMICS
Ken Amer, Hughes Tool Co., Aircraft Div. - Chairman

A Linearized Theory for the Estimation of Helicopter Rotor Characteristics at Advance Ratios Above 1.0
by Peter J. Arcidiacono, David S. Jenney and Arthur F. Smith, Research Laboratories, United Aircraft Corp. 38

High Speed Helicopter Rotor Design
by Jan M. Drees, Bell Helicopter Co. 59

VTOL Automatic Flight Control
by L. Kaufman, and E. R. Schultz, Kaman Aircraft Corp. 68

Computed Induced Velocity, Induced Drag, and Angle of Attack Distributions for a Two-Bladed Rotor
by Raymond Piziali and Frank DuWaldt, Cornell Aeronautical Laboratory x

DEVELOPMENT AND TEST
Bartram Kelley, Bell Helicopter Co. - Chairman

An Integrated Hydraulic Control System Combining Boost, Control Fuel, Stability Augmentation, and Gust Alleviation
by R. P. Carroll, C. J. Wirth, and F. L. Smith, Kaman Aircraft Corp. 79

Flight Test Experiences in Obtaining the First Commercial Certification of a Helicopter for True Vertical Takeoff and Landing Capability
by John Diamond and H. H. Moser, Vertol Div., Boeing Co. x

Height Velocity Envelope Flight Test Project on the Bell 47G-3B Helicopter
by Gilbert Devore and William Hanley x

Design Philosophy of the OH-5A Helicopter
by Sidney S. Sherby, Hiller Aircraft Co. 85

Development and Test of the Bell OH-4A Light Observation Helicopter
by R. L. Lichten, Bell Helicopter Co. x

FLIGHT RESEARCH AIRCRAFT
Paul Carpenter, TRECOM, Ft. Eustis, Va. - Chairman

Research Flight Tests of the High Performance Iroquois
by Wesley L. Cresap and Robert R. Lynn, Bell Helicopter Co. 93

Design and Development of the Army XV-5A Lift Fan Powered V/STOL Aircraft
by H. B. Starkey, Ryan Aeronautical Co., and H. C. True, General Electric Co. 102

The Hummingbird Program
by E. B. Gibson, Lockheed-Georgia Co. 111

Results of the XH-51A Rigid Rotor Research Helicopter Program
by W. H. Statler, R. R. Heppe, and E. S. Cruz, Lockheed-California Co. 119

Design, Development and Flight Test of P-1127
by W. E. Gasich and G. Grogan, Norair Div., Northrop Corp. x

INSTRUMENTS AND ELECTRONICS
Ralph B. Lightfoot, Sikorsky Aircraft Div. - Chairman

Future Development of the ASW Helicopter
by Lt. Commander Robert E. Schock, USN, USNATC, Patuxent River, Md. 134

Designing for Compatibility
by Harold S. Oakes, Sikorsky Aircraft Div. 140

The ANIP Program
by H. W. Mitchell, Bell Helicopter Co. x

Inertial Navigation
by Robert J. Holm, Litton Systems, Inc. 143

The Evolution of Doppler Navigation for Steep Gradient Aircraft
by John J. Bowden, Ryan Aeronautical Co./Ryan Electronics 149

MILITARY AND COMMERCIAL OPERATIONS
Brig. General David B. Parker, USA, USATMC - Chairman

Helicopter Noise and Passive Defense
by C. R. Cox, Bell Helicopter Co. 156

Engineering Requirements for the Commercial Helicopter
by Robert L. Suggs, Petroleum Helicopters, Inc. 164

Partners in Progress - The Military and Commercial Helicopter
by M. Maxfield, Vertol Div., Boeing Co. 168

A Discussion of U. S. Army Aircraft Armament Program
by Major Donald J. Haid, Chief, U. S. Army Aviation Human Research Unit 177

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SECOND HARMONIC CONTROL

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SUMMARY

This paper presents a summary of an experimental investigation of the effects of a second harmonic feathering device on the dynamic and aerodynamic characteristics of a two-bladed semi-rigid rotor helicopter. The investigation was conducted by Bell Helicopter Company under the sponsorship of United States Army Transportation Research Command. A modified UH-1A (formerly HU-1A) was used as the test vehicle.

The amplitude of the second harmonic feathering produced and the phase relationship of the feathering with respect to the blade azimuth positions were adjustable in flight. This made it possible to probe for optimum settings for reducing vertical fuselage vibrations, reducing oscillatory rotor loads, and delaying retreating blade stall. During a portion of the tests, the air loads acting on the blade at six spanwise stations were measured simultaneously with the application of second harmonic control.

It is shown that even though the mechanism accomplished the anticipated changes in air load 2/rev thrust pulsations, the beneficial effects on vibration and load reduction were small. Second harmonic feathering was not effective when introduced for the delay of retreating blade stall. Subsequent analytical studies show this was probably caused by compressibility effects outside the stall region. Analyses of the test data indicate that the results apply specifically to the UH-1A fuselage and rotor system. The response of other helicopters to second harmonic feathering may be significantly different.

The mechanics of the experimental device used for producing second harmonic control, and the flight test program are described. The characteristics of the 2/rev thrust pulsations and accompanying 2/rev vibrations and loads inherent in the rotor and fuselage of the test helicopter are also defined.

INTRODUCTION

Two-per-rev feathering or second harmonic control (SHC) is not a new concept. In 1951, SHC was used (Reference 1) in testing rotors for fatigue life on a test stand. The use of SHC for the alleviation of retreating blade stall was proposed in 1952 (Reference 2). More recent studies at Bell (References 3 and 4) suggest the use of SHC to reduce main rotor blade 2/rev thrust pulsations that cause vibrations and loads in the rotor. The apparent mechanical complexity of a system for producing 2/rev blade feathering is probably the primary reason that an in-flight evaluation has not been conducted sooner.

In 1960, a simple passive system of producing SHC feathering was tested by Bell Helicopter Company. This system, shown schematically in Figure 1, featured a soft collective control linkage. The 2/rev feedback loads, normally present in the control system, introduced a certain amount of SHC depending on the stiffness of the mounts. However, it was suspected that the resulting 2/rev feathering was not at the proper phasing for optimum benefits.

An evaluation of SHC under controlled conditions became practical when a simple device for introducing second harmonic motions into the control system was conceived. This device had provisions for in-flight adjustments of both the amplitude and phasing of the 2/rev feathering. This made possible the investigation of both the delay of retreating blade stall and the reduction of rotor thrust pulsations.

For the delaying of retreating blade stall, the blade pitch is reduced when the blade is normal to the flight path and increased when the blade is aligned fore and aft. Thus, although the mean thrust of the rotor is maintained, the thrust on the fore and aft portions of the rotor disc is increased, and the thrust on the retreating and advancing side is reduced. Theoretical work (Reference 2) shows that this reduction of blade lift on the retreating blade can be accomplished with SHC.

To reduce 2/rev pulsations (encountered below retreating blade stall flight velocities) the mechanism must function in just the opposite manner. These pulsations peak when the blades are near the zero and 180-degree azimuth positions. This is shown in Figure 2 for the UH-1A helicopter at an airspeed of 113 knots. The data are taken from an earlier program for the measuring of dynamic air loads on the main rotor (Reference 3). The figure illustrates that for this test condition the 2/rev component constitutes the major portion of the total oscillatory forces on the blade. Notice also that the magnitude of the 2/rev air load is ± 920 pounds. Because of dynamic blade response, the oscillatory force transmitted to the mast is appreciably less than the exciting aerodynamic forces.

Figure 2 suggests that reducing the blade pitch at zero and 180 degrees azimuth angle would reduce the blade thrust pulsations. However, theoretical work (Reference 4) which includes both dynamic and aerodynamic considerations shows that optimum SHC phasing is 135 degrees earlier. The SHC phasing is defined as the blade azimuth position at which the minimum SHC feathering angle occurs. Figure 3 gives the results of these calculations, indicating that for the UH-1B, ± 2 degrees of SHC feathering phased at 45 degrees would eliminate the 2/rev rotor

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forces transmitted to the lift link. (The lift link is the connecting member between the bottom of the transmission and the fuselage.)

DESCRIPTION OF EQUIPMENT

Test Helicopter

The helicopter used during this program was the two-bladed, semi-rigid rotor U. S. Army UH-1A Iroquois. A photograph of the test vehicle with the second harmonic control mechanism installed is presented in Figure 4. Standard rotating controls were replaced with UH-1B components because of an anticipated increase in control loads at improper phasing of the SHC. The stabilizer bar was removed so that the output from the SHC mechanism could be introduced through the stabilizer bar mixing levers. Figure 5 presents a close-up photograph of the pylon assembly showing the rotor head and the controls of the SHC mechanism.

Second Harmonic Control Mechanism

The SHC mechanism was designed to add an oscillatory control motion occuring at twice per main rotor revolution which could be controlled in amplitude and phasing with respect to blade azimuth. For a two-bladed rotor, this is equivalent to a 2/rev collective control input. This was accomplished with a mechanism which converts the 2/rev rotary motion from the rotor mast into a 2/rev vertical slider motion. The operation of this SHC mechanism is illustrated in Figure 6. The two links between the SHC shaft and the slider "stretch" at twice per revolution as the slider rotates once per revolution with the rotor mast.

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. Both modes of control could be adjusted in flight.

The ability of the mechanism to produce an accurate 2/rev feathering motion has been investigated. Theoretically, it was found that the deviations from a pure harmonic motion are less than two per cent. Actually, larger deviations from the pure sine wave were measured. Tolerances in the bearings and control deflections permitted additional motions. Dynamic measurements indicated about 25 per cent higher feathering amplitudes than the static measurements with the rotor not rotating.

Instrumentation

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 the critical parts in the SHC system were instrumented; also, moments and 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 motions, blade feathering, and pylon positions. During part of the tests, an instrumented rotor blade was used that was fitted with NACA pressure transducers for measurements of differential pressures at six radial blade stations. From two to five transducers were used at each radial blade station. Only limited testing was possible with this blade because the installation of the transducers reduced its fatigue strength. A detailed description of this instrumentation is given in Reference 3.

DISCUSSION OF FLIGHT TEST RESULTS

Hovering

All hovering data were obtained in ground effect. The azimuth phase angle was found, as expected, to have no significance in zero relative wind. Rotor thrust determined from air load data taken during hovering is presented in Figure 7. It is shown that the oscillations are in phase with the feathering and the oscillatory air load is ± 550 pounds for ± .6 degrees SHC, or ± 920 pounds per degree of SHC feathering. Recall that the early work presented in Figure 2 showed that the 2/rev air load on each blade was ± 920 pounds or ± 1840 pounds for the rotor. Hence, it might be inferred that ± 2 degrees of SHC would be required to suppress 2/rev thrust pulsations at high speeds. Lift link loads are also shown in Figure 7. Vertical forces are normally transmitted to the fuselage through the lift link. By comparing the lift link loads with the rotor thrust pulsations, it may be seen that these two forces are out of phase and that the variation in lift link loads is .7 of the rotor thrust variation. This is caused by the 2/rev blade bending which is not in phase with the 2/rev air loads and the inertia effects of the pylon.

Effects of SHC on the fuselage vibrations and loads are shown versus second harmonic feathering amplitude in Figure 8. Note that in general, vibrations and loads increased linearly with SHC amplitude. Blade bending oscillatory moments, not shown, increased in like manner.

It was found that the power required increased with second harmonic feathering. This is believed to be caused by the change in lift distribution resulting in (a) two areas of high lift and increased induced velocities, and (b) two low lift areas of inefficient low blade loadings. The hovering power as a function of SHC amplitude is shown in Figure 9.

Forward Flight

General - The situation in forward flight is much more complex than in hovering. The phasing of the SHC becomes an important factor. Oscillatory loads were monitored very carefully since they were expected to increase for certain phase settings (see Figure 3). Because of high blade bending moments encountered during the tests, the maximum feathering amplitude was restricted to ± 1.1 degrees in forward flight while a further reduction to ± .6 degrees was deemed necessary when the shorter-life, air load blade was used. At an SHC phase setting of 5 degrees, the maximum amplitude

limit was lowered to ± .3 degrees because of the high control loads that developed at higher amplitudes.

Oscillatory Loads - The aerodynamic response to second harmonic feathering in forward flight is illustrated in Figure 10. Rotor thrust determined from air loads is shown for two SHC azimuth phase settings for 40 and 80 knots air speed. An azimuth phase setting of Ψ = 70 degrees is shown to increase rotor thrust pulsations at both 40 and 80 knots. It is shown that with SHC set at Ψ = 5 degrees, the total rotor thrust pulsation is nearly eliminated at 40 knots, but the thrust pulsations were only slightly reduced at 80 knots. Note, however, that in this instance (Ψ = 5 degrees) the amplitude of the 2/rev feathering is only ± .3 degrees because of the high control loads measured previously at the ± .6 degree setting. It is conceivable from Figure 10 that a larger SHC amplitude could have caused a noticeable reduction of the 2/rev thrust pulsations.

The effect of the second harmonic feathering on vibrations and rotor loads is summarized in Figure 11 for the 80 knot case. Each graph defines the envelope of operation with SHC. Each envelope was obtained from crossplots involving a large number of data points. Through this procedure, the effect of scatter of the data points is believed to be minimized. The SHC azimuth settings at the upper and lower limits are indicated in each graph of Figure 11. Notice that these settings vary from graph to graph. A compromise azimuth setting of 100 to 120 degrees and an SHC feathering amplitude of ± .3 degrees produced a reduction in c.g. vibrations and blade moments at the expense of increased pilot vibrations. The lift link loads are minimum for an azimuth setting of about 50 degrees (predicted was 45 degrees, see Figure 3).

It follows from these results that the expected simultaneous reduction in loads and vibrations are only realized near the center of the helicopter which responds mainly to vertical input forces from the rotor. The far forward located cabin of the UH-1, however, is also excited by inplane blade loads and control feedback loads. This could explain why no single optimum SHC azimuth angle was found.

The effects of SHC on vibrations and oscillatory loads as a function of air speed are presented in Figure 12 for ± .3 degrees second harmonic feathering amplitude. Here again, an envelope has been defined. Azimuth settings for the maximum reduction are indicated above each graph. It can be seen that to minimize any particular vibration or load it is necessary to adjust second harmonic control as air speed is increased.

Stall - At 100 knots and high gross weights, high blade moments precluded the measurement of air loads with the SHC phased for reducing retreating blade stall. However, from data taken during high gross weight flights with the standard blades, no beneficial effects were detectable. The amplitude of the SHC feathering had to be restricted for the 100-knot flights because of high control loads and random violent vibrations and excessive fore and aft pylon motions. These conditions prevented comprehensive evaluation of blade stall reduction. Because of this, analytical studies were conducted with the Bell Aerodynamic IBM Computer Program F35. This program is a further development of NASA procedures for calculating aerodynamic characteristics of lifting rotors (Reference 5). Results of these studies indicate that compressibility effects offset the benefits from the elimination of retreating blade stall. Figure 13 shows the calculated drag coefficient of the blade tip versus angle of attack with and without SHC. The effects of SHC on power required are also indicated. It follows from this figure that the drag of the retreating blade tip is 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 not realized.

OTHER CONSIDERATIONS

High Frequencies

On several occasions, high frequency vibrations were created by the second harmonic feathering mechanism. Figure 14 gives an example of a case where a 6/rev frequency was measured. The cause could not be definitely established, but it is believed that the looseness in the SHC mechanism caused by normal tolerance build up and the reduction in control system stiffness caused by the mechanism is responsible.

2/Rev Air Load Distribution

Figure 15 shows the air load pulsation and the lift link load for the 40-knot case with and without SHC. It is noted that the oscillations in the lift link load have increased in magnitude instead of decreased as might be expected. In the lower plot of Figure 15, the total rotor thrust has been divided into thrust acting on the outboard and inboard half of the blades. It can be seen that the reduction in total thrust pulsation was accomplished mainly by a phase shift in the 2/rev outboard air loads. Since there was no reduction in magnitude of local air loads, beneficial effects will be determined by the dynamics of the rotor. Blade flapping caused by the 2/rev pulsations of the outboard loads will result in forces transmitted to the fuselage which are not in phase with the outboard 2/rev air loads. These considerations suggest that separate SHC of the inboard portion of the blade may be more beneficial than the system tested.

Control Loads

The increase in oscillatory control loads may have caused an increase in the cabin vibrations which partly offsets the expected vibration reduction. The higher control loads could be due to both the inertia reactions to feathering the blades at 2/rev and the change in aerodynamic pitching moment characteristics resulting from the dynamic responses to the 2/rev feathering. It was concluded

from these test results that prediction methods of control loads and their effect on fuselage vibrations should be developed.

CONCLUSIONS

In general, the SHC was effective in creating changes in rotor thrust and accompanying changes in vibrations and loads. However, the effects are very complex in that the results on vibrations and oscillatory loads differ from one location in the helicopter to another. For optimum results, the phasing and amplitude of the SHC input have to be controlled continuously with forward speed.

At 80 knots, a simultaneous reduction in vertical vibration level at the cg and reduction in oscillatory blade bending moments occurred when the mechanism was phased at about 120 degrees. This is not beneficial to the pilot though, because of increased vibrations in the forward located cabin of the test helicopter. This indicates that a helicopter with its cabin located under the mast might realize benefits from a simultaneous reduction in vertical vibrations and oscillatory loads.

The aerodynamic response to SHC varied along the blade span. This suggests that SHC, applied to only a portion of the blade, might be superior to SHC of the whole blade. However, it was beyond the scope of the investigation to analyze such a system.

The investigation of SHC as a means for stall prevention was inconclusive. The theoretical results indicate that the test rotor does not respond beneficially to SHC when applied for the alleviation of retreating blade stall. The influence of SHC on other helicopters may be different since the response depends on detailed aerodynamic and dynamic characteristics of the rotor and fuselage.

It is believed that this program is an example of research in the true sense of the word. It is shown that the state of the art of present theoretical methods is still insufficient to accurately predict complex phenomena such as associated with the second harmonic feathering. This illustrates the need for experimental investigations of new devices and theories. New insight was obtained from this program which may lead to related analytical and test applications in the future.

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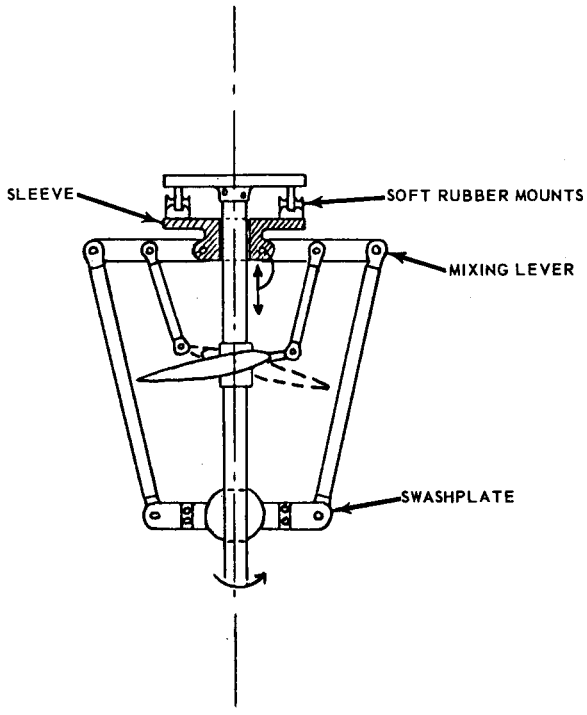


FIGURE 1. SOFT COLLECTIVE SYSTEM

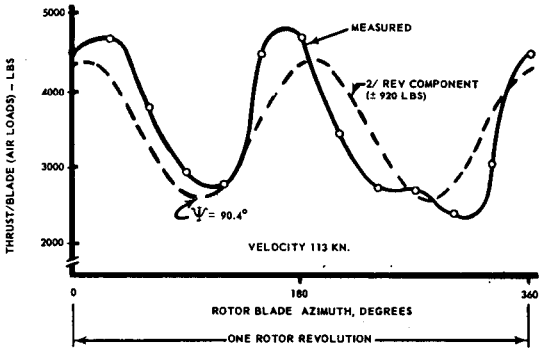


FIGURE 2. THRUST PULSATIONS UH-1A

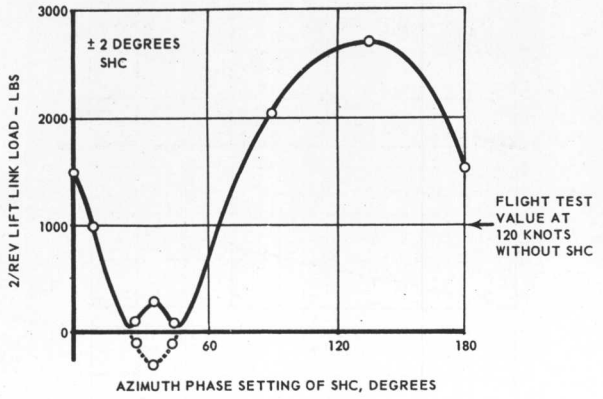


FIGURE 3. PREDICTED EFFECT OF SHC AT 120 KNOTS ON THE UH-1B LIFT LINK LOAD



FIGURE 4. THE UH-1A IROQUOIS TEST HELICOPTER WITH THE SECOND HARMONIC CONTROL MECHANISM INSTALLED

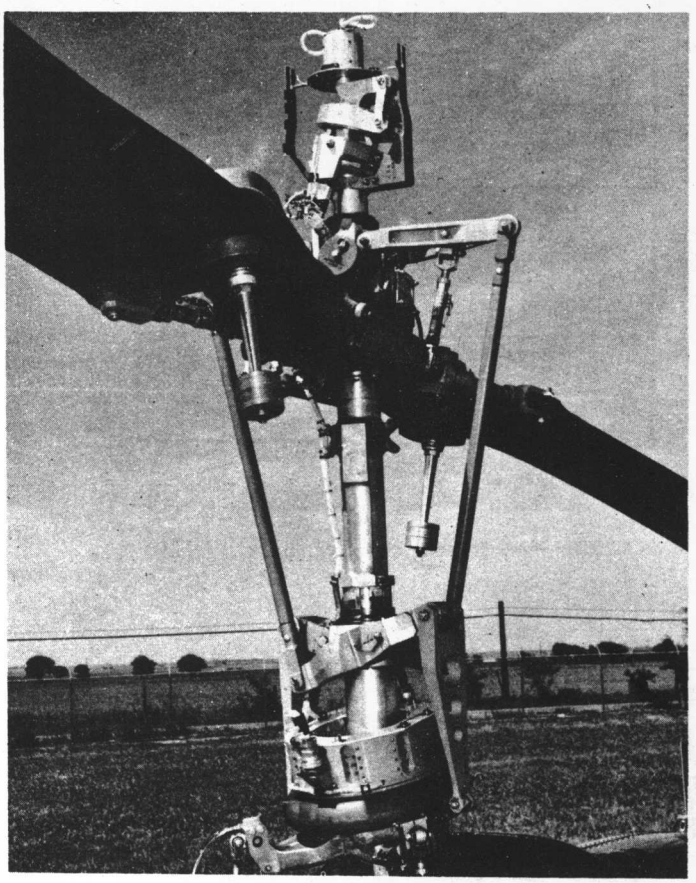


FIGURE 5. EXPERIMENTAL PYLON ASSEMBLY

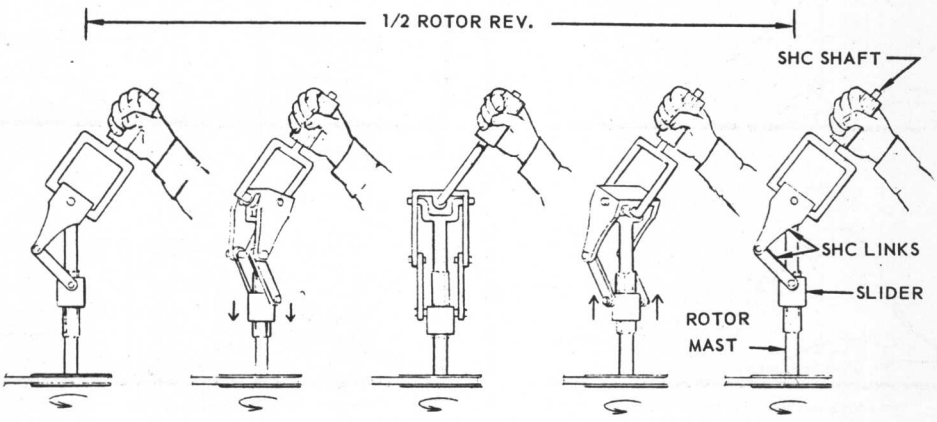


FIGURE 6. SCHEMATIC OF SECOND HARMONIC CONTROL MECHANISM

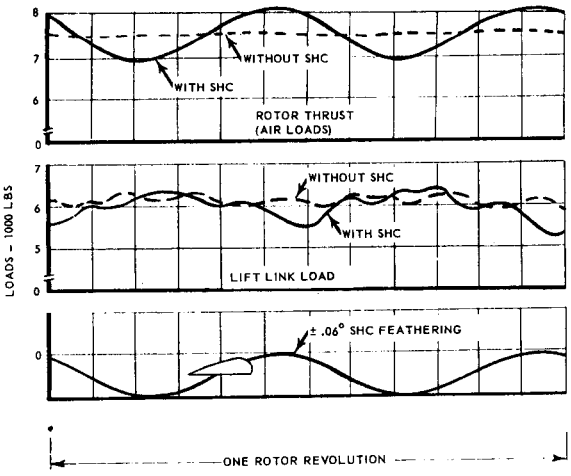


FIGURE 7. EFFECT OF SHC ON ROTOR THRUST AND LIFT LINK LOAD IN HOVERING

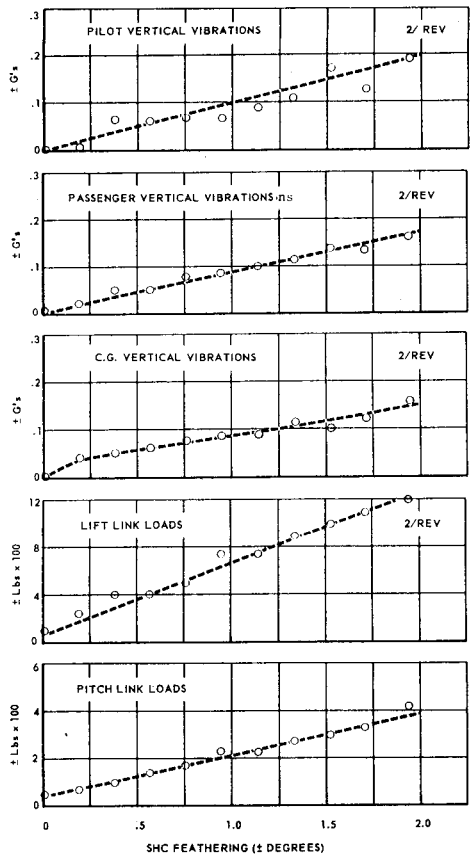


FIGURE 8. EFFECTS OF SHC ON OSCILLATORY LOADS AND VIBRATIONS IN HOVERING

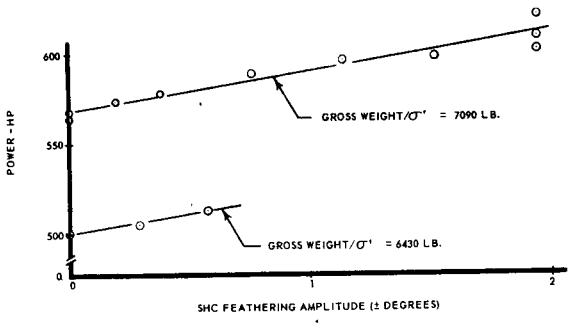


FIGURE 9. THE EFFECTS OF SECOND HARMONIC CONTROL ON POWER REQUIRED IN HOVERING (IGE)

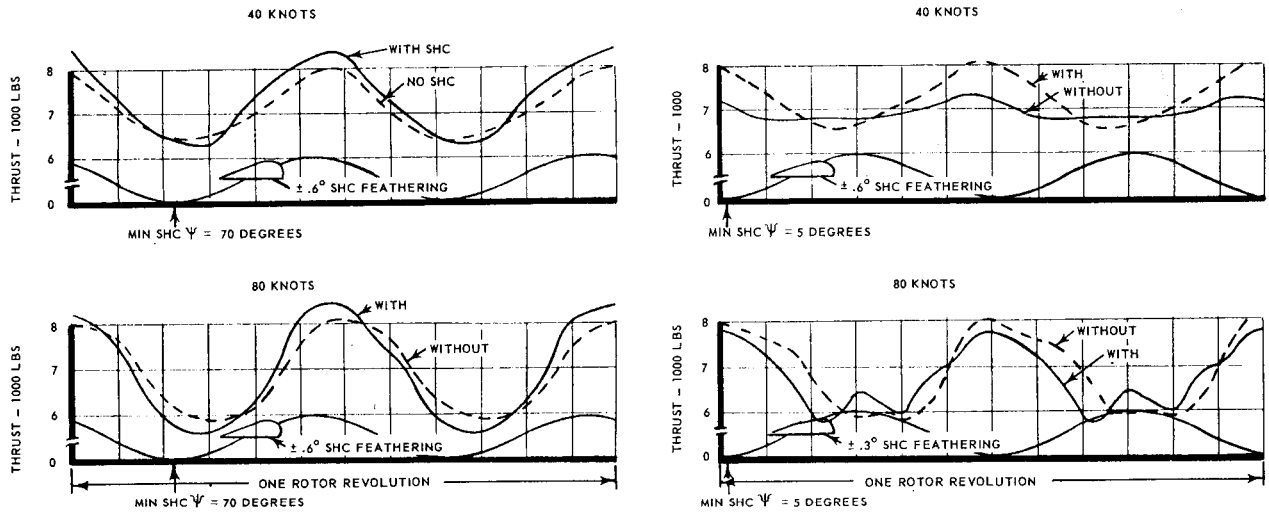


FIGURE 10. EFFECTS OF SECOND HARMONIC CONTROL ON ROTOR THRUST (AIR LOAD)

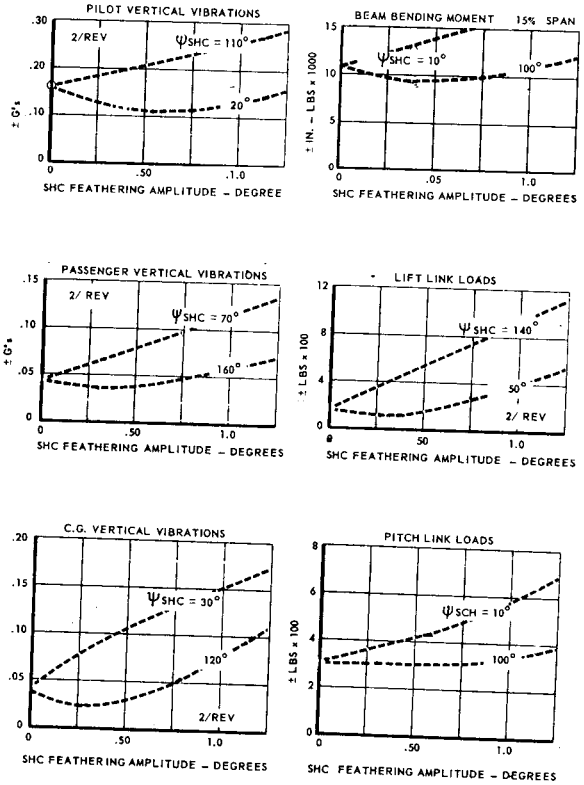


FIGURE 11. EFFECTS OF SHC AT 80 KNOTS

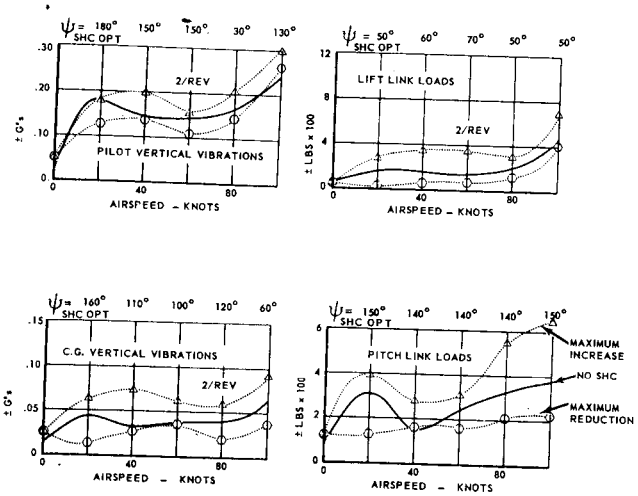


FIGURE 12. THE MAXIMUM REDUCTION AND MAXIMUM INCREASE OF VIBRATIONS AND LOADS DUE TO ± .3 DEGREES SHC AS A FUNCTION OF AIRSPEED

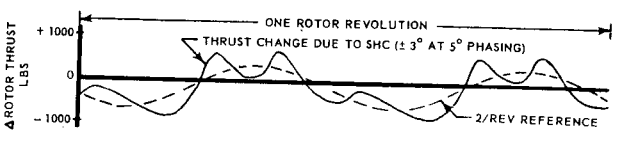


FIGURE 14. HIGHER HARMONIC THRUST DUE TO SHC MECHANISM

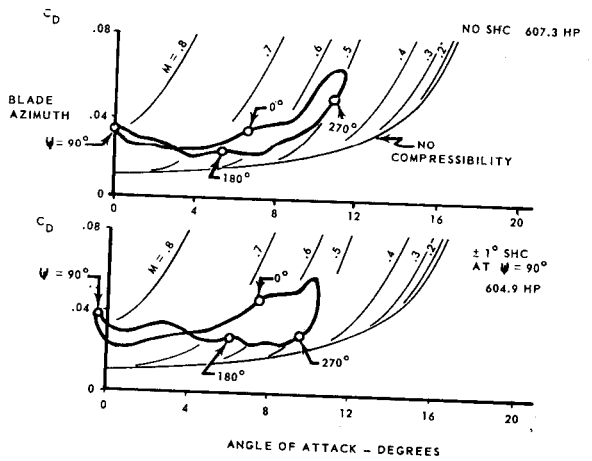


FIGURE 13. COMPUTED PROFILE DRAG COEFFICIENTS AT THE TIP OF THE BLADE V.S. ANGLE OF ATTACK. (UH-1A ROTOR AT 100 KNOTS)

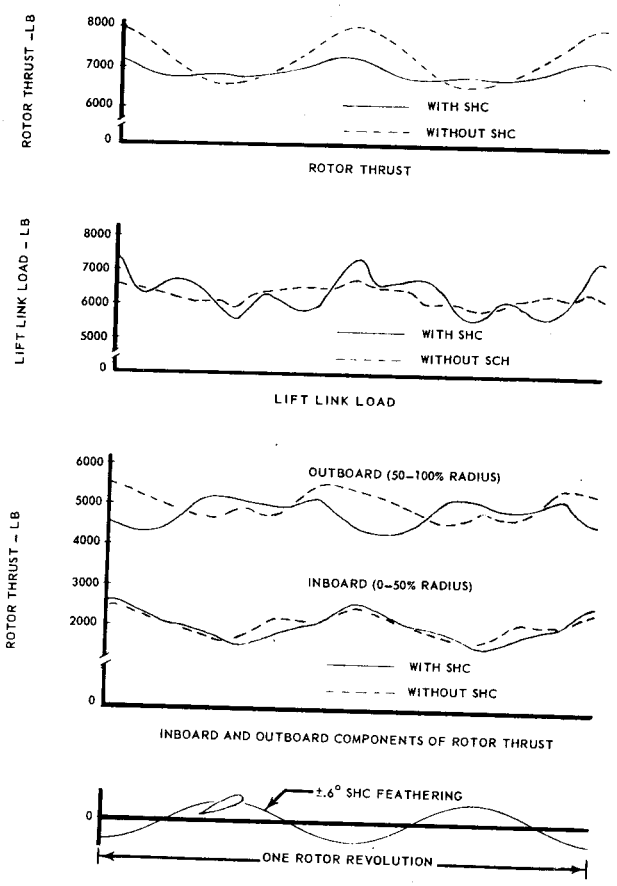


FIGURE 15. EFFECT OF SHC ON LIFT LINK LOADS AND SPANWISE AIR LOAD DISTRIBUTION AT 40 KNOTS

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APPLICATION OF AN ABSORBER TO REDUCE HELICOPTER VIBRATION LEVELS

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Abstract

The development of a vibration absorber for the reduction of helicopter vibration levels is described. Basic absorber theory is reviewed, and the limitations of such an approach are discussed. Design and development problems are reviewed; final results are also presented.

These results show that an absorber installation can effectively eliminate structural amplification of predominant flight vibration frequencies. This can be done with a very small penalty compared to other possible approaches, and without transferring the vibration problem to other areas in the aircraft. Design problems connected with the absorber installation are easily resolved, and service results are satisfactory.

Summary

While the basic low vibratory force levels of the UH-2A rotor system gave the UH-2A helicopter unusually low vibration levels throughout most of the operating speed range, the presence of large structural amplification at 4/rev (four times rotor speed) was found to create undesirable vibration levels at the front of the cockpit area around the pilot's feet and at the instrument panel locations while flying at very low speeds (15-30 kts). The basic fuselage configuration requirements made the correction of this structural amplification by structural means impractical, from both weight and functional considerations. A theoretical and experimental study of the use of vibration absorber techniques indicated that such methods could provide a significant reduction in structural amplification with a small weight penalty.

An absorber was designed and developed meeting the requirements. Flight testing of the device indicated vibration reductions in the forward end of the cockpit by a factor of more than 10 to 1. Reliability testing established the adequacy of the device from a serviceability standpoint. Service experience has confirmed test predictions in this regard.

While the basic helicopter characteristics produced satisfactory pilot seat vibration levels, use of the absorber allowed UH-2A vibration levels throughout the entire cockpit to meet the requirements of MIL-H-8501A, through the entire operating speed range of the aircraft.

Basic Aircraft Characteristics

The UH-2A (HU2K-1) helicopter has a 4-bladed main rotor, 3-bladed tail rotor, is powered by a General Electric T58-8b engine, and has an operating speed envelope extending to 150 knots. Designed as a utility helicopter, the aircraft fuselage was configured to provide the best design compromise between the speed requirements of the long range search and rescue mission and the rescue requirements of hoist pickups from the water. These two requirements together dictated the large rescue door immediately behind the pilot's door together with retractable landing gear. The latter primarily provides a clear, tangle-free area for hoist operations; as a secondary benefit, of course, is the drag reduction accomplished by retraction. The rescue door location was dictated by the requirement of placing the rescue hoist cable close alongside the pilot's head, to expedite aircraft positioning during hoist rescue operations.

As may be seen from Figure 1, this fuselage configuration results in a significant reduction in fuselage bending stiffness just aft of the pilot seat. Attempts were made to make up for the loss of beam stiffness across the wheel well area and through the rescue door area by stiffening the basic keel beam on the centerline of the

fuselage and by stiffening around the rescue door frame. However, the weight penalties involved in this approach made it necessary to limit the design criteria to the provision of adequate strength in this area.

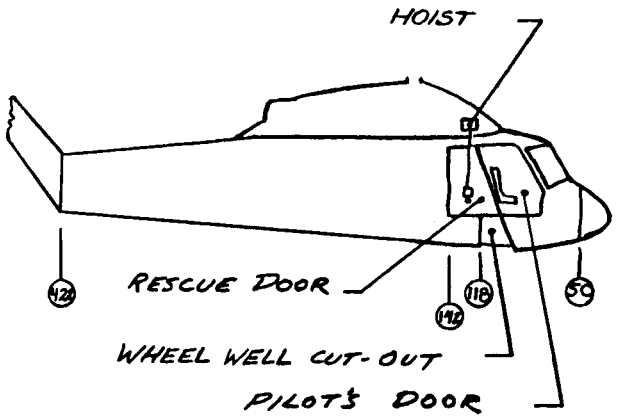


Fig. 1 Structural Configuration Sketch UH-2A Helicopter

Flight vibration data taken in this configuration is summarized in Figures 2 and 3. Figure 2a shows the variation of 4/rev pilot seat vibration (the only frequency showing any significant "g" levels) with airspeed. Figure 2b shows similar trends for the forward cockpit bulkhead, Station 50. Figure 3 shows the vibration levels of the same measurement stations at 20 knots as a function of rotor RPM. Figure 2 is data taken in level flight; Figure 3 data is shown for both level flight and a 200 foot per minute descent. This latter condition represented the area of maximum vibration level; large variations in levels were found in level flight at this speed, dependent on atmospheric turbulence, etc.; these variations could be minimized by recording data in the descent condition. It might be noted also that the UH-2A doppler radar system provided a valuable assist to the problem of getting repeatable test conditions in this area. Also shown on the plots are the spec. requirements of MIL-H-8501A.

Review of the vibration levels shown indicates that while pilot seat levels met requirements, significant amplification existed along the length of the cockpit, with levels at the forward end of the cockpit being about 4 to 5 times higher than pilot seat levels. Such levels contributed to pilot discomfort through both the feet and the visual stimulus of blurred instruments, particularly under red night lighting conditions. The RPM variation shown in Figure 3 indicates that while plane guard vibration levels are highest at 100% RPM (normal maximum operating RPM) the structural resonance was not sharply tuned and significant amplification existed at both ends of the operating range.

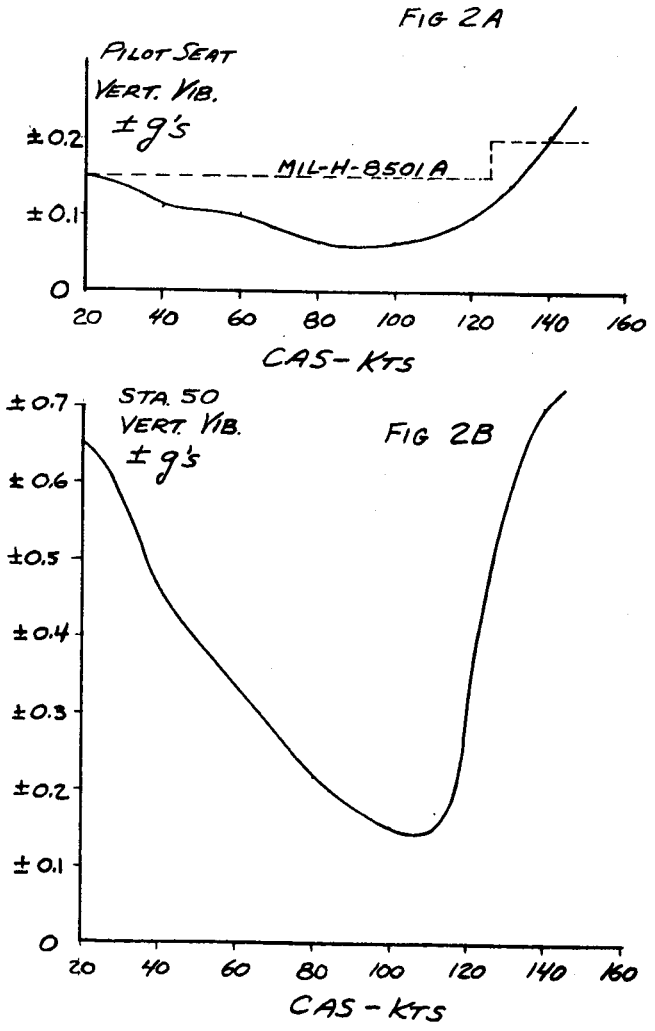


Fig. 2 UH-2A Vibration Level In Forward Flight Without Absorber 98% RPM

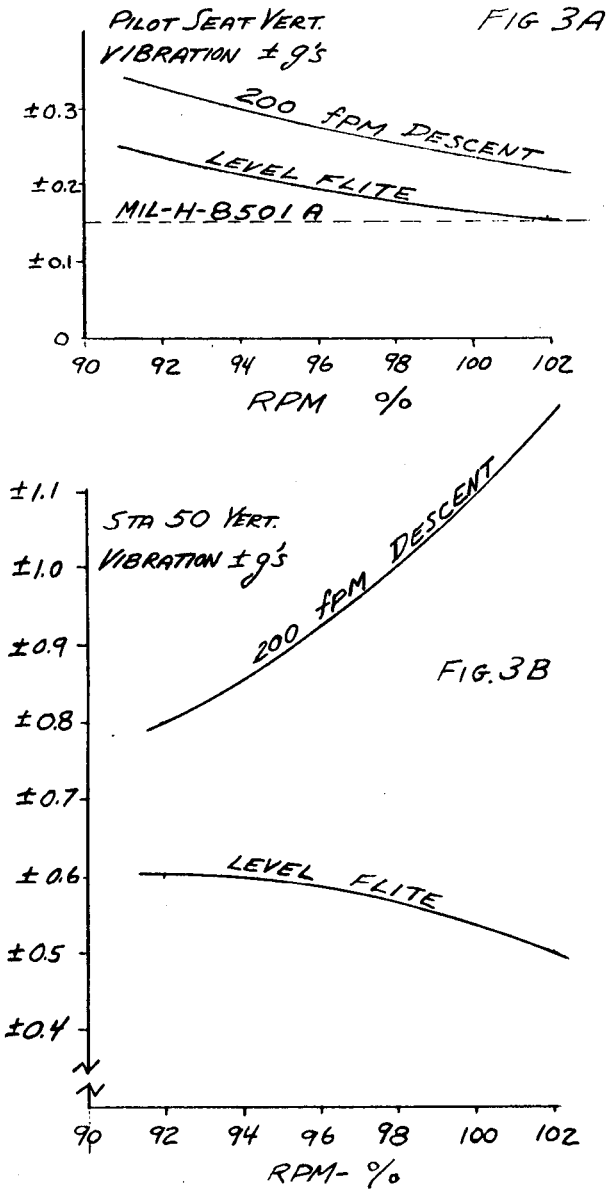


Fig. 3 UH-2A Vibration Level Vs RPM at 20 Knots Without Absorber

To provide additional guidance in improving this level of amplification, an in-flight fuselage mode shape was determined by judicious selection of accelerometer locations. Results of this survey are shown in Figure 4.

Review of the results shown in Figure 4 indicates significant bending in the forward fuselage structure between Stations 90 and 150. Reference to Figure 1 shows that this is the area of the wheel well cut out and the rescue door. Also shown is a relatively large bending deflection of the tail cone structure aft of Station 250.

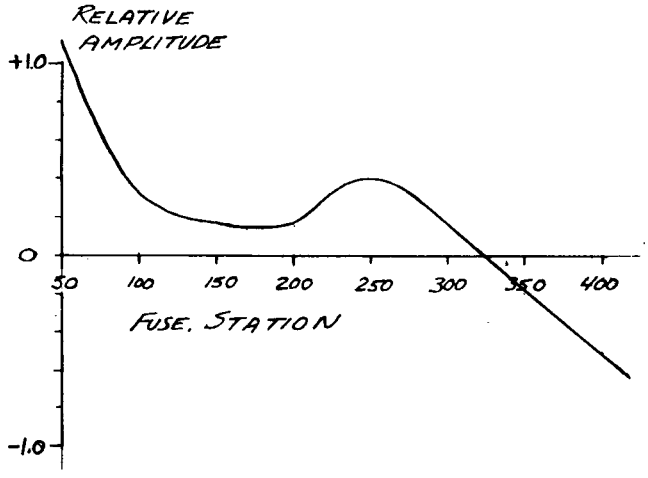


Fig. 4 UH-2A Fuselage Bending Mode Shape Without Absorber

Analysis and Shake Tests

Since flight test data indicated a bending mode of the fuselage to be the major source of the vibratory problem existing in the low plane guard speed regime, it was concluded that a static shake test program would be the quickest and easiest way to develop a satisfactory means of reducing the structural amplification. Therefore a vibration survey was conducted on a UH-2A helicopter in standard configuration except for the substitution of a dummy engine in place of the standard engine. The rotor was also removed and replaced by a rotary shaker assembly having the same weight as the rotor. This shaker had the capability of producing a 500 pound in-plane force. The helicopter was suspended at the hub by a soft bungee. This suspension system had a natural frequency of 60 CPM which was low enough to insure decoupling from any of the fuselage modes in the 1000 to 2000 CPM range that would be of interest.

Figure 5 shows the results of the vibration survey made on the unaltered UH-2A helicopter. It is seen from the response curve that a four per rev resonance of the fuselage occurs at 104% rotor RPM and that the mode shape of the fuselage taken at this frequency corresponds almost exactly to the mode

shape obtained in flight (Reference Figure 4). These results confirmed the suspicion that a near four per rev resonance of the fuselage existed at the 100% rotor RPM and the excessive vibration at Station 50 encountered at plane guard speeds causing large instrument panel motions and pilot feet discomfort was due to the amplification of the second mode of fuselage bending. It is interesting to note that pilot seat and cargo area vibrations were always low because of their proximity to the nodal points in this bending mode.

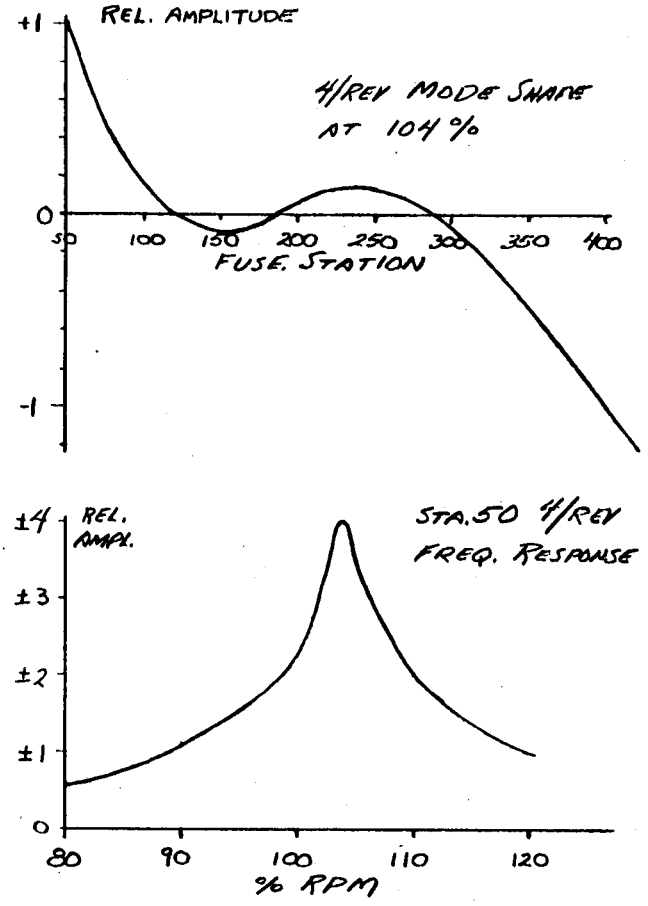


Fig. 5 Shake Test Results Without Vibration Absorber

Since the cause of the high vibratory level in the extreme forward section of the aircraft had been established, the major problem now existed in determining a design fix to eliminate this condition. One possible solution would be to alter the vibration characteristics of the fuselage by changing the natural frequency and since the natural frequency was already above the operating range, stiffening seemed to be in order. Preliminary shake test results with experimental stiffening showed that to change the structural characteristics of the fuselage would be expensive, a long lead time item, and a large increase in weight. Therefore, it was desirable to examine the use of a vibration absorber to reduce fuselage amplification. If the effective mass of the forward cockpit area were sufficiently small, a reasonably sized absorber could provide the desired vibration reduction at the least expense, time, and weight.

From previous vibration surveys, in which results were obtained with and without 220 pounds of ballast at Station 50, an estimated effective mass was calculated. Figure 6 shows the results in which ΔM (the known ballast mass at Station 50) is plotted versus $1/\omega_n^2$. By extending the curve to intercept the ΔM coordinate, the effective mass is known. The slope of this line is the effective spring rate. For the UH-2A helicopter in this mode of bending, the effective mass at Station 50 is 17.5 slugs or 564 pounds and the effective spring rate is 249,900 lbs/ft. Since the effective mass is small, a vibration absorber appeared to be a logical solution to the problem.

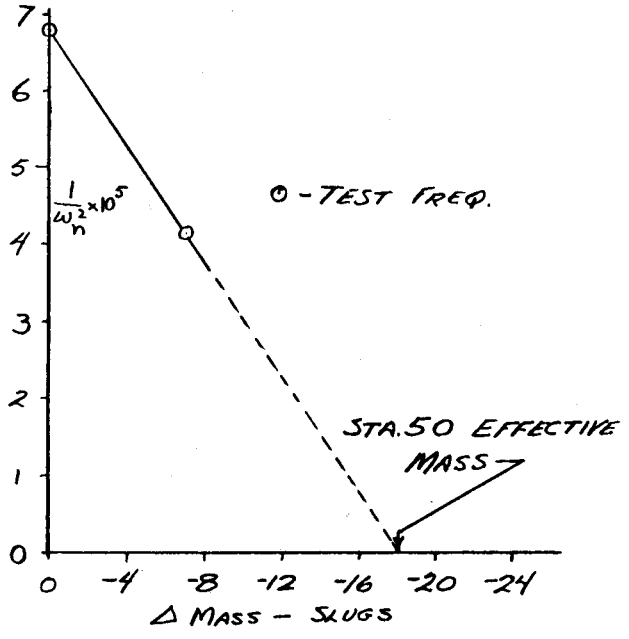


Fig. 6 Determination of Station 50 Fuselage Effective Mass at Resonance.

In order to determine the required size of the vibration absorber, some theoretical calculations were done. The theory of vibration absorbers is well known and can be found in any standard text on vibration, therefore the detailed theory will not be discussed in this paper. However, the use of a vibration absorber does introduce an additional degree of freedom in the system and therefore another resonant condition that must be considered in the design. The size or weight of the absorber will determine the practical range of operation; the lower the weight the smaller effective operating range of reduced vibration. The operating RPM of the UH-2A is between 92% and 100% and calculations were done for four per rev at these operating speeds. The vibration absorber was tuned to four per rev at 98%. This is tuned to an off resonance condition, relative to the resonance of the fuselage at 104%. However in the theory of undamped vibration absorbers (or any simple two degrees of freedom system), the forced mass amplitude goes to zero at the uncoupled natural frequency of the absorber mass, and therefore tuning the absorber to a point in the operating RPM range would reduce the vibration level to zero at that tuned frequency.

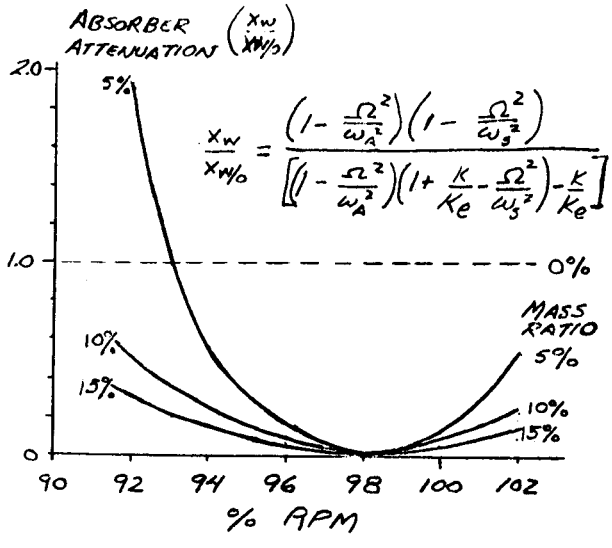


Fig. 7 Theoretical Absorber Performance Vs RPM and Ratio of Absorber Mass to Effective Mass

Figure 7 shows the results of the calculations. The equation used to obtain these results is also shown in this figure. It is seen that this equation is the response curve of the main mass of an undamped vibration absorber system normalized to the main mass without a vibration absorber. The results show that even with a vibration absorber having a weight of only 5% of the effective mass of the structure (28.2 pounds), reduced vibration occurs from 93% to 100% rotor RPM. With a vibration absorber of 56.4 pounds large reductions in vibration occur throughout the operating RPM range. Admittedly the results are unconservative in that calculations were done without damping; however, it was felt they were sufficient to size a vibration absorber for shake test purposes. From the calculations it was decided to first use a vibration absorber of approximately 50 pounds.

To simplify the design of an absorber for shake testing, beams simply supported at the right and left skin lines at Station 50 were used as springs, the frequency being changed by addition of weight or changing the dimension of the beam. Figure 8 shows the results of the shake test program.

Figure 8a shows the results of the vibration of the main body using a vibration absorber mass of 59.3 pounds. As expected two natural frequencies were introduced, one below the operating range and one above. However throughout the operating range the vibration is reduced and at the tuned frequency of the vibration absorber the resulting vibration has been reduced by approximately 10 to 1.

Figure 8b shows results obtained throughout the operating range for various absorbers normalized to results obtained without the absorber. Tuning in these tests was obtained by changing weight or beam dimensions, therefore tuning did not always occur at 98% rotor RPM. Tuning of the 54.3 pound weight occurred above the 100% rotor RPM and therefore shows amplification of the vibration condition from 92% to 94%. However the vibration absorbers of 33.5 and 54.3 pounds reduced the vibration level throughout the operating range. Even a vibration absorber of only 13.3 pounds showed excellent reduction in vibration level; however the range was not sufficient to warrant considering this low weight for a fixed frequency vibration absorber.

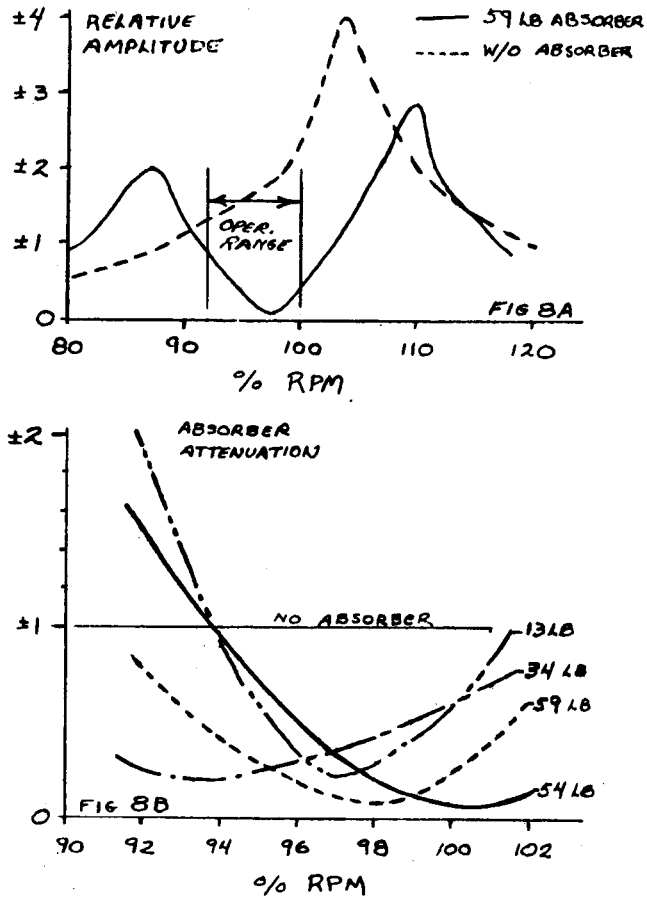


Fig. 8 Shake Test Performance of Several Vibration Absorbers

From the calculations and the shake test results, it was concluded that a vibration absorber would be an ideal solution to the vibration problem encountered on the UH-2A helicopter, providing at least a 10 to 1 reduction in Station 50 vibration levels.

For the vibration absorbers used in the shake test survey, no consideration was given to the loads or deflections that would be encountered. However for the final design of the vibration absorber the flight values of these parameters must be considered. Using the standard theoretical response curve of the absorber mass, but normalizing to main mass without the absorber, allows the amplification factor for the vibration absorber to be calculated. Therefore by knowing the g levels experienced on the aircraft for any flight condition without the vibration absorber, the g's and deflections expected on the absorber can be calculated directly. Using the above methods the vibration absorber was designed for 10 g's and ±1/2 inches of motion.

Absorber Design and Development

Having established the feasibility of using a tuned-mass vibration absorber as a means of controlling the structural amplification in the forward fuselage, the selection of an absorber design was undertaken. Since the aircraft battery was almost ideally located in the critical area, since the weight of the battery was large enough to provide a reasonably wide operating RPM range, and since both the battery design requirements (±9 g) and previous experience indicated the battery to be a suitable candidate for the vibrating mass, the design proceeded along these lines.

Review and test of mounting systems using either single cantilever or simply supported beams to provide the required spring rate indicated that considerable design trouble would be encountered from secondary problems with end attachment strength and rigidity. In addition, such mounting would compromise the accessibility and utility of the nose electrical-electronic area. Design attention was therefore switched to use of preloaded coil springs to provide the required mounting flexibility.

Development of this design posed several problems, the most pressing of which was that of separating the vertical, pitch, and roll frequencies of the absorber assembly. The finally evolved design, as sketched in Figure 9, provided a tray mount for the battery, with preloaded and opposed coil spring stacks at each of the four corners of the tray. The tray is U-shaped to place the c.g. of the battery close to the elastic axes of the spring system, minimizing the pitch and roll inertias and maximizing pitch and roll frequencies. The spring spacing is rectangular, rather than square, to separate pitch and roll frequencies. These design criteria were dictated by early test results on a flat, bottom-mounted tray, whose angular frequencies were very close to the desired vertical frequency. This resulted in large pitch and roll excursions, which used up the available spring travel but did not contribute to any vibration reduction.

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