ICAS-88-6.1.1 EXPERIMENTAL INVESTIGATION OF STRONG IN-FLIGHT OSCILLATION ON HELICOPTERS AND ITS PREVENTION

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Abstract

A strong random in-flight oscillation failure on helicopters directly threatens flight safety. Therefore, research and analysis of such a failure will play an important role in its prevention and flight safety.

This paper is a summary of research on this kind of the failures. Its main phenomena and features are summarized, followed by the discription of the experimental investigation of the failure including the research approaches, the flight dynamic model of the helicopter, experimental investigation on the control system-autopilot loop oscillation and the pilot-induced helicopter oscillation, and the research results. Especially, by coupling a true helicopter to a ground flight simulator and an analog computer, a series of ground helicopter simulation tests have been conducted, the reappearance of the failure phenomena achieved and some causes of the failure found. Accordingly, some failure pervention principles and measures for design and operation are proposed. Finally, some useful conclusions are presented.

Introduction

Due to expansion of the flight envelope, complexity of the flight control system and wide use of the autopilot, a strong random in-flight oscillation failure has happened frequently on modern aircraft and helicopters, which affects normal operation and threatens flight safety. Related to such a failure are complicated causes and a variety of factors. In the fixed wing aircraft field much research on the failure has been done [Refs. 1, 2, 3]; however, in the helicopter field, little. With the development of domestic helicopters, especially the introduction of exotic helicopters, this failure has occurred again and again. Therefore, research and analysis of such an oscillation failure will be of great significance in its prevention, flight safety and development of new helicopters.

Features of the In-flight Oscillation Failure

According to the statistical and comprehensive analysis of large quantities of original failure records, the failure has the following features.

(1) Obvious change of the helicopter attitude (mainly a longitudinal oscillation). A "horse-riding" longitudinal oscillation is usually felt in a failure case, occasionally even followed by a lateral one. In the slight case the pilot can feel the toss or shake of the helicopter and in the serious case he may be vibrated out of the seat and cannot see the instrument panel clearly.

(2) Obvious abnormal control feeling, such as light control stick force, empty control displacement, supersensitive control sensing and shaking of the control stick.

(3) Suddenness and randomness. The failure may happen in various flight conditions, such as level flight, climbing, gliding and manoeuvering.

(4) Being closely related to the control system and pilot's control action. For example, it occurs in case of large longitudinal servo valve friction, small control system friction, serious hydraulic fluid contamination, small and unmatched $L\dot{\theta} / L\theta$ ratio (the ratio of longitudinal autopilot angular speed to angular transfer coefficient), rude and frenquent control action for flight state change, and so on. The control system vibration and control stick shake are often found during the ground post-failure inspection.

Experimental Research

1. Research approaches

According to the features of the failure, research work was carried out by coupling a true helicopter to a ground flight simulator and an analog computer and conducting a series of ground simulation tests to achieve the reappearance of the failure phenomena and study the effects of various factors on the failure (see Fig. 1).

The steps were as follows,

(1) Deriving the flight dynamic model of the helicopter, i.e., its motion equation; conducting the control and stability calculation on an analog computer in order to find out the response characteristics of the helicopter; and, in the meantime, correcting the stability and control

derivatives and builting up flight test curve fitting to obtain satisfactory results.

(2) Experimenting on the control system-autopilot loop oscillation on a true helicopter by using different nonlinear factors of the control system (such as gap and friction) and autopilot parameters.

(3) Experimenting on the pilot-flight control system-helicopter loop oscillation to integrate the effect of pilot's control action as shown in Fig.1.

(4) Analyzing the failure mechanism.

(5) Proposing some prevention principles and measures and testing them on helicopters grounded for in-flight oscillation failure.

2. The Flight Dynamic Model of the Helicopter

In any ground simulation test, the flight dynamic response characteristics of the helicopter must be mathematically modeled. The choice of the model directly affects the accuracy of the test and the creditability of the failure reappearance. This is not a complicated problem for domestic helicopters. But for imported ones, difficulties arise due to lack of some original design data, such as helicopter moment of inertia, fuselage aerodynamic characteristics and helicopter's center of gravity (C.G) position. Herein, for the needs of failure study and with the consideration of scheduling, economy and reliability, a modified engineering calculation method was presented. The stability and control calculation method for typical articulated helicopters was used with some of the original data being approximately processed or practically measured. Then by means of flight test curve fitting through the correction of the stability and control derivatives on the analog computer, a good agreement was achieved between the computational and the flight test results. Details of the method were presented in Ref.[4]. A comparison of the results showed that the main stability and control derivatives to be corrected were static speed stability Mu, drag damping Xu, pitch damping $M\dot{\theta}$ and helicopter pitch moment of inertia $M\ddot{\theta}$, among which $M\dot{\theta}$ and $M\ddot{\theta}$ only affected the initial state of the control responses, and, Mu and Xu mainly affected its steady value, with Mu's effect being the greatest. Major differences between the calculated and flight test results lay in the steady values of control responses. Research results showed that by the correction of Mu/2 to Mu/3 all the computational results of various flight condition were in good agreement with the flight test data with an error of 10% to 20%. The typical comparison curves were shown in Fig.2.

3. Experimental Investigation of the Control system－Autopilot Loop Oscillation [Ref.5]

Analysis of the main features of the failure showed that the oscillation is always closely related to the helicopter control system and the autopilot. Hence, by coupling a true helicopter control system to an analog computer the test of the loop oscillation was conducted with emphasis on the loop oscillation condition, the effects of various parameters on it, its mechanism, boundary conditions and effect of the pilot on it, and the reappearance of the failure phenomena was achieved. The test schematic is shown in Fig.3. The test showed that, under normal condition, the movement of the control stick produced deflection signals on the swashplate both directly through controls/hydraulic booster and indirectly through compensation sensor/autopilot. The latter was opposite to the control angle produced by the horizon- and rate gyro-sensed signal and offset it to ensure the normal control state. Here the compensation sensor coordinated the control of the pilot and the autopilot, and it was this manner of coordination that provided the possibility of the control system loop oscillation. When the control force of servo valve including static and dynamic friction, cohesion of fluid molecules, dimension error and hydraulic fluid contamination, became greater at the control input end than the accumulative friction of the control rod and the partical feel spring force, the booster housing motion no longer caused the relative movement of the servo valve but forced the control rod to move together with the booster housing so that a reversed electric signal was generated on the compensation sensor, which was then amplified by the autopilot control components to command the booster motion . Because the direction of this commanded motion was always opposite to the original booster movement, such a cyclic movement formed the control system－autopilot loop oscillation.

In conclusion, the necessary and sufficient conditions for this loop oscillation were as follows:

(1) The autopilot, which has the control system with a compensation sensor , is on.

(2) The valve control force Fv is greater than the control rod system friction Fc (including feel spring force).

(3) The longitudinal amplification $K\theta$ and the angular transfer coefficient $L\theta$ are large enough when the autopilot is at work.

In the test, a specific fixture was mounted on the longitudinal booster servo valve to provide an additional valve friction. Then, with different combination of autopilot parameters, a perturbation was applied to the control system so that the loop oscillation was observed. Under the condition of the loop oscillation, control stick and swashplate vibration curves were both recorded without and with pilot's normal operation.

The test results showed that,

(1) The major affecting factors are the booster servo valve friction Fv, the longitudinal autopilot amplification $K\theta$ and longitudinal angular transfer coefficient $L\theta$. The relationship among these three parameters and the typical oscillation boundary curves, which were obtained experimentally, are shown in Fig.4. The so-called "critical oscillation" is a state to which these three parameters are adjusted so that the loop oscillation can be started. If the combination of the parameters lies above the boundary curve, the loop oscillation occurs. Otherewise it does not.

(2) The minor affecting factors are the hydraulic pressure, control system friction and gap. The loop oscillation tends to happen at high pressure, small friction and large gap.

(3) For the control system with an autopilot and compensation sensor, the effect of the servo valve friction on the oscillation becomes more sensitive. The loop oscillation may happan when $K\theta$ and $L\theta$ are large enough and $Fv/Fc \geqslant 1.0$. Whereas for the conventional hydraulically powered control system, the oscillation can happen only when $Fv/Fc \geqslant 3 \sim 4$.

(4) In general, when $Fv < 6$kg, the loop oscillation can cause the vibration of the control stick and the swashplate at a frequency of $6 \sim 7$ Hz (see Fig.5a), but can not cause the helicopter attitude oscillation. If the pilot catches hold of the control stick, the oscillation can be immediately stopped.

(5) When $Fv = 6 \sim 10$kg, which is a large servo valve friction, apart from vibration of the control stick and the swashplate, the loop oscillation is able to cause an additional control stick and swashplate vibration with a frequency of $0.5 \sim 1.0$Hz and a fairly big amplitude (see Figs.5b,5c), as a result of the feel spring and nonlinearities of the control system. In this case, pilot's catching hold of the control stick can not stop the oscillation, but cause a large swashplate deflection change of about 3.6° (see Fig.5c) as well as the helicopter pitching oscillation. According to further computational results the pitching attitude change can be over 10°.

(6) when $Fv > 10$kg, no loop oscillation is observed and there is a phenomenon of stick self-shifting as a result of servo valve sluggishness.

Repeated tests showed that the above test results were reproduceable. The general trend is the same though the specific values may vary a little for different helicopters.

4. Experimental Investigation of The Pilot-Control System-Helicopter Loop Oscillation.

The phenomena and features of the failure showed that consideration of the effects of the pilot's action must be included in the failure investigation. Therefore experimental investigation of the pilot-control system-helicopter loop oscillation was conducted (see Fig.1) by using a true helicopter coupled to a ground flight simulator and an analog computer; the pilot and helicopter were mathematically modeled by an analog computer with input and output interfaces; the control system, autopilot control components, servo booster, actuator and swashplate were on a true helicopter, and the autopilot sensors (horizon, pitch, roll and yaw gyro) were mounted on the ground flight simulator. The response signal output of the helicopter movement from the analog computer controled the flight simulator and pilot model simultaneously. The movement of the flight simulator was sensed by the autopilot sensors to drive the swashplate through the control components and the servo booster. The deflection of the sawshplate was then turned into electric signal by the mechano-electronic transfer unit to control the helicopter response characteristies. At the same time, electric control signal output of the pilot model controled the movement of the stick by the servo simulator and deflected the swashplate again through the real control system. Thus, the large loop oscillation test was realized.

The test results showed that,

(1) an unmatched relationship between the autopilot longitudinal angular speed and the angular transfer coefficient, i.e., a small $L\dot{\theta}/L\theta$, is the main cause of the induced large loop oscillation. Under a certain flight state, a small $L\dot{\theta}/L\theta$ ($L\dot{\theta}/L\theta < 1.0$) leads to a small damping and a supersensitive control response, which may cause a strong induced oscillation due to the pilot's repeated inconscient corrective action for certain internal or external disturbances. The test results in Fig.6 showed the effects of the longitudinal autopilot ratio $L\dot{\theta}/L\theta$ on this induced in-flight oscillation. For this reason, it is favorable to switch off the autopilot in order to eliminate this kind of induced oscillation.

(2) Nonlinearities of the helicopter control system, such as friction, gap and pilot's response retardation time, which can result in a large phase lagging between the stick force or movement and the swashplate deflection or helicopter response, has a great effect on this

induced oscillation. In analytical computation these nonlinearities can be converted into pilot's response retardation time. From simulation test results shown in Fig.7, it is observed that a greater pilot's response retardation time τ may cause a strong induced pitching oscillation of the helicopter as long as the other conditions are the same.

(3) Flight parameters of the helicopter, such as speed, weight and center of gravity location, have their effects on this kind of induced oscillation as well. The greater the flight speed, the smaller its damping; the lighter the helicopter, i.e., the smaller its moment of inertia, the higher response the helicopter has; the more aft the center of gravity location, the greater the unstability of the helicopter. Correspondingly, this kind of induced oscillation can happen more easily.

Prevention Measures

The experimental research shows that prevention of strong in-flight oscillation on helicopters is a problem involving all phases of helicopter's life cycle: design, manufacturing, operation and maintenance. Detailed prevention principles and measures are proposed as follows:

(1) As for helicopter design, allowable tolerance ranges of all channel parameters of the autopilot, especially the matching between the angular speed and the angular transfer coefficient, must be strictly specified. In principle, the ratio $L\dot{\theta} / L\theta$ should not be less than 1.0 while $L\theta$, not very large. In addition, a high precision hydraulic filter should be utilized in order to reduce the hydraulic fluid contamination under class 9 of the NAS-1638 Specification.

(2) As for manufacturing, operation and maintenance, all the nonlinearities, such as friction, gap and dead zone of the control system, electromagnetic stop and the hydraulic booster, tightness and hard-going degrees of the bearing in key positions, and autopilot parameters should be strictly controled and checked regularly. Especially, the matching between the servo valve and control system friction (inclucding feel spring) should be strictly controled to ensure that the Fv/Fc is less than 1.0. In addition, the hydraulic fluid contamination must be strictly restricted in hydraulic fluid transportation and preservation, on the booster calibration device, in the hydraulic power unit, and so on.

(3) The pilot should control accurately and mildly to avoid rude and frequent control actions.

(4) Once the in-flight oscillation happens, the pilot should switch off the autopilot immediately, take hold of the stick, and reduce the flight speed and the collective pitch (as long as the flight height allows). Artificial corrective control actions with the stick are not allowed. Operational practice has proved the effectiveness of these principles and measures.

Conclusions

1. The research and analysis of strong in-flight oscillation on helicopters will play an important role in flight safety and helicopter design. And ground helicopter simulation test is an effective research method.

2. In case of lack of original data (for imported helicopters), a reliable and accurate mathematical helicopter model can be obtained to study the failure by means of the approximation technique and a method of flight test curve fitting through correction of stability and control derivatives.

3. The servo valve friction, autopilot longitudinal amplification and angular transfer coefficient are the main factors causing the control system-autopilot loop oscillation, which, in general, can lead to the vibration of the control system, but not to helicopter pitching oscillation. If there exists a large friction of about 6~10kg in the servo valve, the oscillation may be very strong in case of improper control action.

4. An unmatched relationship between the longitudinal angular speed and the angular transfer coefficient, i.e., a small ratio $L\dot{\theta} / L\theta$, is an important cause of induced in-flight oscillation. In addition, the nonlinearities of the control system and the pilot's lagging correction have also a great effect on it.

5. For the control system with an autopilot and compensation sensors, the effect of the servo valve friction becomes more sensitive. Hence a high precision hydraulic filter should be used to restrict the fluid contamination.

6. The prevention of strong in-flight oscillation on helicopters involves all phases of helicopter's life cycle: design, manufacturing, operation and maintenance. Most importantly, the autopilot longitudinal parameters ($L\dot{\theta} / L\theta \nless 1.0$) and the control system frictions ratio ($Fv/Fc \ngtr 1.0$) should be strictly restricted.

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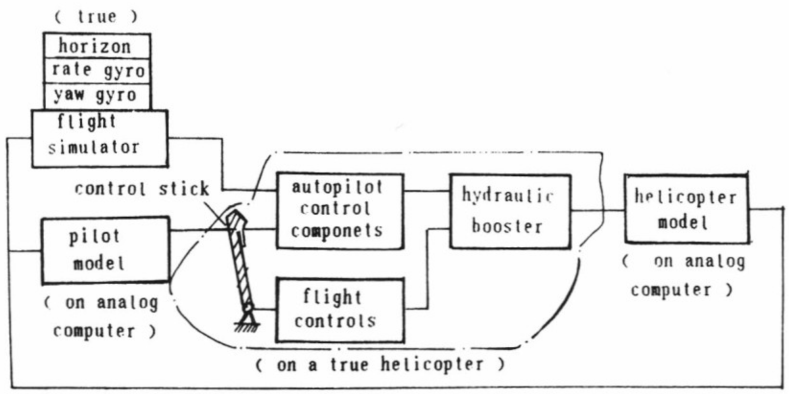

FIGURE 1 THE HELICOPTER GROUND SIMULATION TEST DIAGRAM

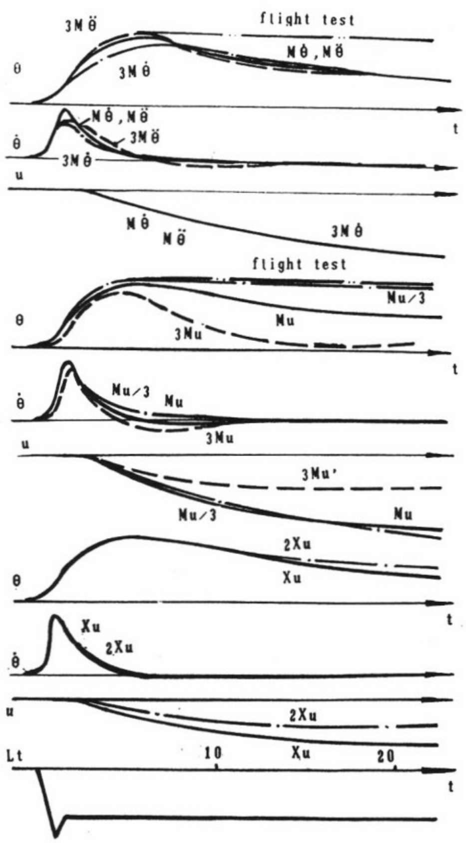

FIGURE 2 EFFECT OF MAIN DERIVATIVES ON THE HELICOPTER DYNAMIC RESPONSE AND THE COMPARISON OF COMPUTATIONAL RESULTS WITH THOSE OF FLIGHT TEST

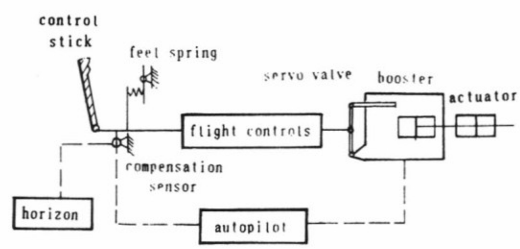

FIGURE 3 CONTROL SYSTEM--AUTOPILOT LOOP OSCILLATION TSET DIAGRAM

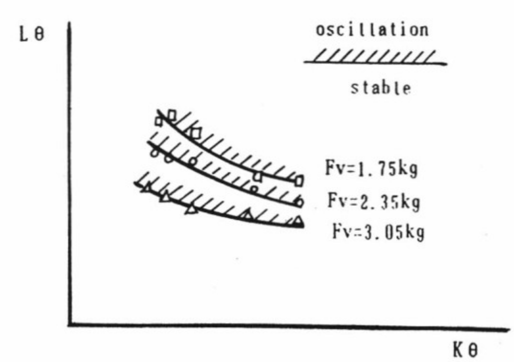

FIGURE 4 BOUNDARY OF CONTROL SYSTEM--AUTOPILOT LOOP OSCILLATION (TEST RESULTS)

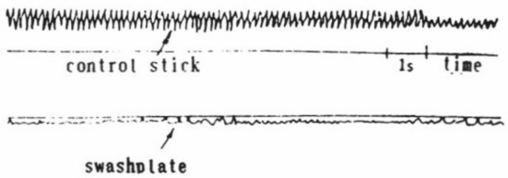

figure 5.a Fv=4.5kg free stick

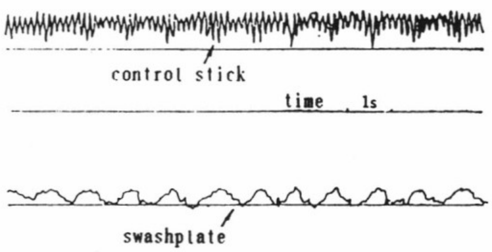

figure 5.b Fv=6.2kg free stick

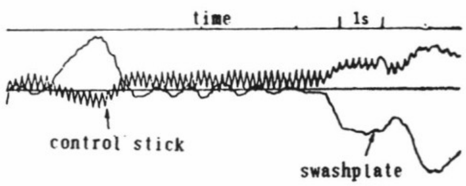

figure 5.c Fv=9.0kg hold stick

FIGURE 5 EFFECT OF SERVO VALVE FRICTION ON CONTROL SYSTEM--AUTOPILOT LOOP OSCILLATION (TEST RESULTS)

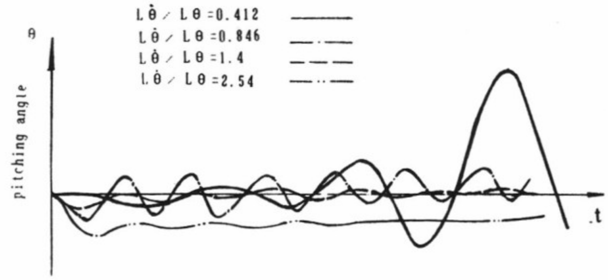

FIGURE 6 EFFECT OF AUTOPILOT RATIO $L\dot{\theta} / L\theta$ ON IN-FLIGHT INDUCED OSCILLATION (GROUND SIMULATION TEST RESULTS)

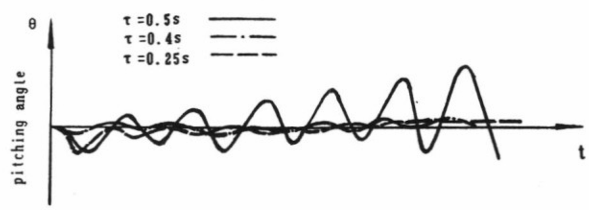

FIGURE 7 EFFECT OF NONLINEARITY OF THE CONTROL SYSTEM AND PILOT'S RESPONSE RETARDATION TIME τ ON IN-FLIGHT INDUCED OSCILLATION (GROUND SIMULATION TEST RESULTS)