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SUMMARY

In an effort to determine the feasibility of utilizing an on-off pitch and roll control for VTOL operation under visual conditions for hovering and low-speed flight and to evaluate the associated control requirements, a flight investigation was conducted with a variable-stability helicopter. For several maneuvering and precision tasks, pilot evaluations were obtained for selected combinations of control power, angular-velocity damping, static stability, out-of-trim conditions, and artificial random disturbances.

The results indicate that the use of an on-off control, with a properly sized dead band, reduced control power to approximately one-third of the level required for satisfactory maneuverability with the proportional control. The range of satisfactory combinations of angular-velocity damping and control power appears to be relatively small. Steady moments arising from the out-of-trim conditions and from static stability could be handled satisfactorily if such moments did not exceed 20 percent of the control power. Random disturbances which produced peak angular accelerations of less than 50 percent of the control power could be handled satisfactorily during the performance of precision tasks.

INTRODUCTION

The provision of control power for V/STOL aircraft during low-speed flight represents a substantial compromise between maneuverability and payload or range. Unlike the helicopter, in which pitching and rolling angular accelerations are produced economically simply by tilting the rotor tip-path plane, other V/STOL aircraft (particularly the jet lift types) generally must derive control from special moment-producing devices such as bleed-air jets which absorb large amounts of energy directly from the installed power. For example, the cost associated with providing control in large V/STOL aircraft is discussed in reference 1 which shows that doubling the reference control moment about all axes would increase the aircraft gross weight by about 10 percent for a 100,000-pound (0.44 meganewton) vehicle.

In view of the severe performance penalties associated with providing high levels of control power, it is evident that methods which offer potential for reducing control power requirements warrant consideration. One such method is the on-off, or bang-bang, type of control. Although on-off control methods have been applied in the area of space flight and for high-altitude flight by conventional research aircraft, the application of the on-off control to V/STOL aircraft has received little serious attention.

In an effort to determine the feasibility of utilizing on-off pitch and roll control for V/STOL operation and to evaluate the associated control-power requirements, a visual flight investigation was conducted with a variable-stability helicopter. Both maneuvering and precision tasks were performed. The initial flight was used to optimize the control dead band which was used throughout the remainder of the flight program. Next, the control-power requirements were established for zero angular-velocity damping and then, the effects of increased damping were determined for a selected control-power value. Finally, the effects of static stability, out-of-trim conditions, and random disturbances were investigated individually for selected combinations of control power and damping. Six pilots participated in the flight program – four NASA pilots (three from the Langley Research Center (LRC) and one from the Ames Research Center) and two U.S. Navy pilots. In general, the commentary and the ratings assigned by each of the pilots were in good agreement. Because of the limited participation by the non-Langley pilots, only the ratings provided by the three LRC pilots are presented herein; the commentary obtained from all the pilots is considered in the discussion.

SYMBOLS

Measurements for this investigation were taken in the U.S. Customary System of Units. Equivalent values are indicated herein in the International System (SI) in the interest of promoting the use of this system in future NASA reports.

M_{Y_Δ} pitching moment resulting from application of on-off control, lbf-ft (newton-meters)

M_{Y_q} pitching moment proportional to and opposing pitching angular velocity, $\frac{\text{lbf-ft}}{\text{rad/sec}} \left(\frac{\text{newton-meters}}{\text{rad/sec}}\right)$

M_{X_p} rolling moment proportional to and opposing rolling angular velocity $\frac{\text{lbf-ft}}{\text{rad/sec}} \left(\frac{\text{newton-meters}}{\text{rad/sec}}\right)$

M_{Y_u} pitching moment proportional to and opposing change in speed, $\frac{\text{lbf-ft}}{\text{ft/sec}}$ $\left(\frac{\text{newton-meters}}{\text{meters/sec}}\right)$

M_{Y_γ} pitching moment proportional to simulated disturbance, $\frac{\text{lbf-ft}}{\text{ft/sec}}$ $\left(\frac{\text{newton-meters}}{\text{meters/sec}}\right)$

M_{Y_c} pitching moment resulting from a trim change, lbf-ft (newton-meters)

I_X, I_Y moments of inertia about body X- and Y-axis, respectively, slug-ft^2 (kilogram-meters^2)

u forward component of velocity, ft/sec (meters/sec)

γ disturbance parameter, ft/sec (meters/sec)

$\dot{\theta}_M$ angular velocity of model aircraft, rad/sec

$\dot{\theta}_H$ angular velocity of test helicopter, rad/sec

δ control deflection, in. (cm)

V voltage output of on-off relay, volts

DEFINITIONS

Control power	angular acceleration resulting from control application, that is, M_{Y_Δ}/I_Y for pitch
Angular-velocity damping	angular acceleration proportional to and opposing the angular velocity, that is, M_{Y_q}/I_Y for pitch and M_{X_p}/I_X for roll
Static stability	angular acceleration proportional to and opposing translational velocity, that is, M_{Y_u}/I_Y for pitch
Disturbance susceptibility	angular acceleration proportional to disturbance, that is, M_{Y_γ}/I_Y for pitch
Out-of-trim	angular acceleration proportional to trim shift, that is, M_{Y_c}/I_Y for pitch

EQUIPMENT AND PROCEDURE

Test Vehicle and Simulation Method

The variable-stability helicopter used in this investigation is shown in figure 1. In this vehicle a computer-model simulation technique is employed which tends to eliminate

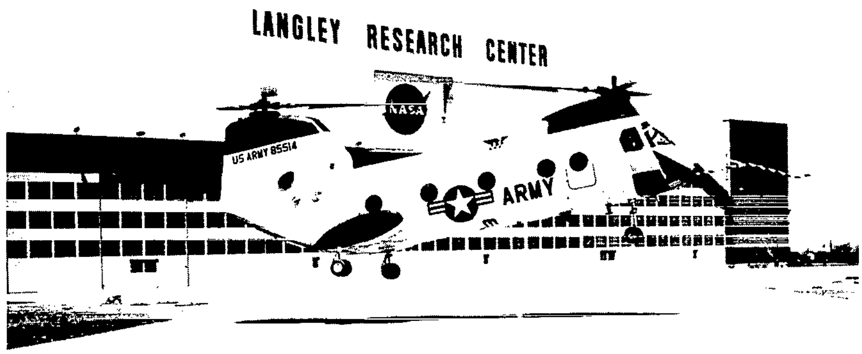

Figure 1.- Variable-stability helicopter. L-64-2891

the effects of external disturbances and unprogramed characteristics so that the individual effects of different parameters can be studied separately. A detailed description of the model simulation technique is given in references 2 and 3. Briefly, this technique is based on obtaining a continuous real-time solution of the equations of motion which describe the model aircraft (i.e., the aircraft being simulated) and using closed-loop servotechniques to force the test vehicle to follow the computer solution. In appendix A a signal flow diagram of this simulation is presented and the degree of accuracy which was achieved is indicated. A disturbance generator, which was used to provide an additional test parameter, is described in appendix B.

Most of the test combinations were evaluated for several visual tasks which were considered to represent basic and fundamental requirements for VTOL operation. The tasks may be separated into one of two categories – precision or maneuver. A description of each task is given in appendix C. The NASA pilot rating system used in the evaluation is presented in the following table:

Operating conditions	Adjective rating	Numerical rating	Description	Primary mission accomplished	Can be landed
Normal operation	Satisfactory	1	Excellent, includes optimum	Yes	Yes
		2	Good, pleasant to fly	Yes	Yes
		3	Satisfactory, but with some mildly unpleasant characteristics	Yes	Yes
Emergency operation	Unsatisfactory	4	Acceptable, but with unpleasant characteristics	Yes	Yes
		5	Unacceptable for normal operation	Doubtful	Yes
		6	Acceptable for emergency condition only[1]	Doubtful	Yes
No operation	Unacceptable	7	Unacceptable even for emergency condition[1]	No	Doubtful
		8	Unacceptable - dangerous	No	No
		9	Unacceptable - uncontrollable	No	No
	Catastrophic	10	Motions possibly violent enough to prevent pilot escape	No	No

[1]Failure of a stability augmenter.

Control Stick

Except for a portion of the first flight in which a center-mounted control (ref. 4) was used, the pitch and roll degrees of freedom were controlled using the side-mounted control shown in figure 2. The control had an available travel of about ±2 inches (5.08 cm) in pitch and roll as measured at the top of the stick. The length of the stick was about 7 inches (17.8 cm). The control had negligible friction and a spring gradient of about 1.3 pounds per inch (2.3 N/cm) in both pitch and roll. An arm rest which is visible in the figure was provided to minimize unintentional inputs. Conventional proportional controls were used for directional and height control.

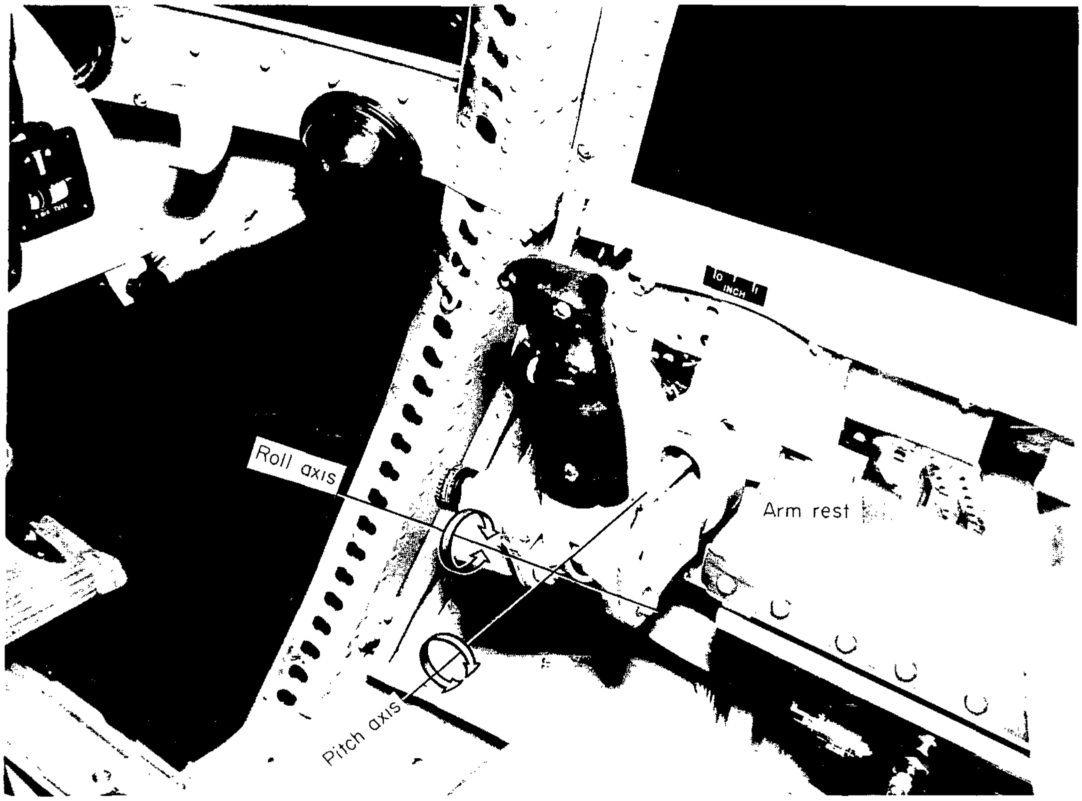

Figure 2.- Pitch and roll control stick. L-65-4265.1

RESULTS

Selection of Control Dead Band

On the initial flight, the control power and damping were held constant and variations were made in the magnitude of the control dead band by first using a center-mounted control and then a side-mounted control. Pilot commentary indicated that the use of either control stick would not appreciably affect the results but that the side control was preferable because of slightly improved precision attributed to the arm rest. Four different dead-band values ranging from ±1 inch (±2.54 cm) to ±1/8 inch (±0.3 cm) as measured at the top of the stick were investigated. The ±1/4-inch (±0.6-cm) dead band was rated satisfactory and was considered the best value because it felt comfortable and natural to the pilot; this ±1/4-inch (±0.6-cm) dead band was used throughout the remainder of the investigation. The ±1-inch (±2.54-cm) dead band was rated almost unflyable; the ±1/2-inch (±1.27-cm) dead band was flyable but unsatisfactory; the ±1/8-inch (±0.3-cm) dead band was unsatisfactory. With the ±1-inch (±2.54-cm) dead band and to a lesser degree with the ±1/2-inch (±1.27-cm) dead band, the pilot could not command the control precisely when it was needed because of the large control motion required. With the ±1/8-inch (±0.3-cm) dead band, aircraft motions and unintentional motion of the pilot's hand caused inadvertent triggering of the control. It is believed, however, that different stick-force characteristics might result in a somewhat different optimum dead band.

Effect of Control Power and Damping

Maneuver flight results.- Figure 3 provides an indication of the range of control-power and damping values investigated during the present on-off control studies as compared with the range of these parameters typically considered during studies of proportional control requirements (for example, ref. 4). It may be seen from figure 3 that the entire range of control-power values covered with the on-off control represents only a fraction of the range covered for conventional control systems.

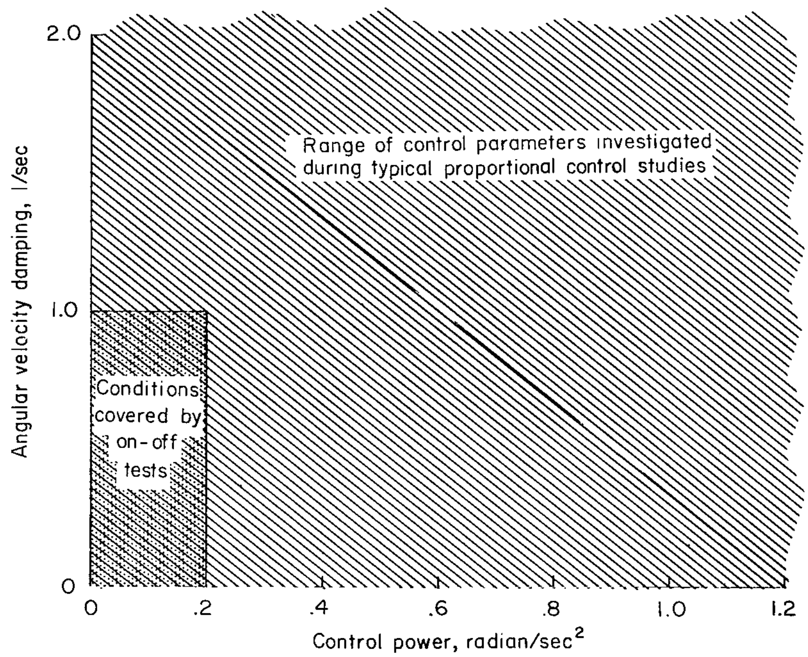

Figure 3.- Control power and damping range covered by the present on-off study compared with typical proportional control studies.

The control power and damping results which were obtained for the maneuver tasks are shown for the pitch and roll axes in figures 4 and 5, respectively. In order to conserve flight time and to provide control harmony, identical conditions were simulated in both axes simultaneously. The numbers shown inside the symbols were obtained by averaging the ratings assigned by the pilots during the task which yielded the poorest rating. The individual ratings assigned to each condition by the LRC pilots are shown outside the symbol for the most critical task. For discussion purposes each of the

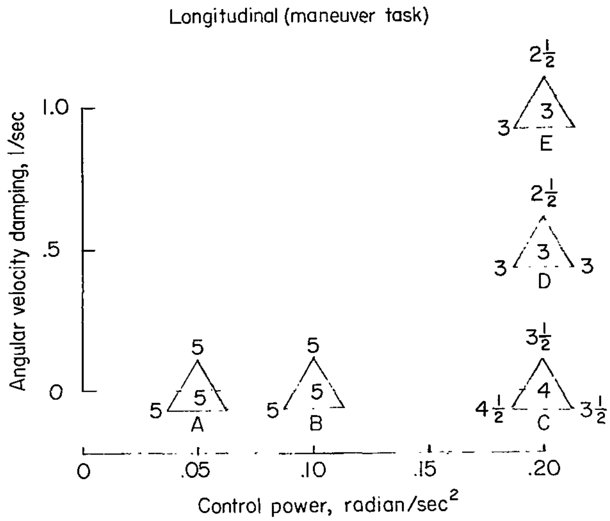

Figure 4.- On-off control power and damping results for longitudinal maneuver tasks in the absence of disturbances, trim changes, and static stability.

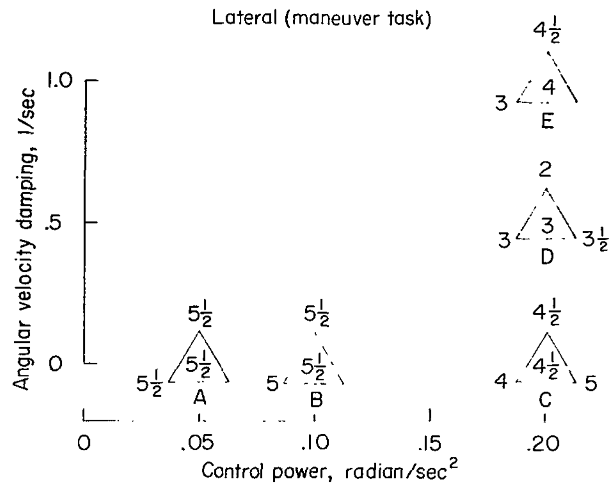

Figure 5.- On-off control power and damping results for lateral maneuver tasks in the absence of disturbances, trim changes, and static stability.

tested combinations of control power and damping is identified by a letter. In pitch, the poorest ratings were usually obtained during the quick start and stop maneuver; in roll, the turn reversals during forward flight were generally the most critical task.

The longitudinal-control results in figure 4 show that none of the conditions with zero damping were considered satisfactory. For condition A, and to a somewhat lesser extent for condition B, excessive time was required for initiating a maneuver. The control would be triggered for up to $2\frac{1}{2}$ seconds during the quick starts and would then have to be reversed for an equal amount of time to stop the angular velocity at the desired attitude. Such an operation, with full control applied in one direction and then in the other for long periods of time, was completely unsatisfactory because of the intense concentration required. The pilot was, therefore, restricted to the use of lower than desirable angular rates to prevent the development of dangerously large attitude changes. The low control power problem was even more severe when attempts were made to decelerate from about 45 knots to hover over a preselected spot on the ground. For this latter portion of the quick start and stop task, the stopping had to be carefully planned and the pilot had to be primarily concerned with the angular rates and attitude involved rather than being concerned only with arriving at the landing point. In other words the weak angular-acceleration capability provided by conditions A and B required the pilot to supply an amount of lead-time beyond his capability which resulted in potentially dangerous attitudes during gross maneuvering.

For condition C, which was also unsatisfactory, the control power was high enough to produce the desired angular rates but was not powerful enough to arrest these rates; therefore, uncomfortable attitudes tended to occur before the angular rate could be stopped. Consideration of the comments obtained for condition C indicated that addition of angular-velocity damping might be beneficial since the initial acceleration was considered good.

Increasing the damping from zero to the levels indicated by conditions D and E for the control power of 0.20 rad/sec^2 resulted in satisfactory ratings for all the maneuver tasks. For both these conditions, each pilot considered the initial response and the angular-velocity capability to be adequate for maneuvering. Two benefits seem directly attributable to the additional damping provided at conditions D and E. First, the damping prevented the occurrence of undesirably high angular rates. Second, as the desired attitude was approached, the damping, by opposing the angular velocity, effectively augmented the control power in reducing the the angular rate to zero.

With only slight variations, the preceding comments relative to the pitch axis may be applied directly to the results for corresponding test conditions in the roll axis which are presented in figure 5. In general the pilot's evaluations for the roll axis were 1/2 a rating unit poorer than for identical conditions in pitch with the result that only condition D was considered satisfactory for all the lateral maneuver tasks.

Precision flight results.- The results obtained for the precision flight tasks are presented in figures 6 and 7 for the pitch and roll axes, respectively. The trends indicated by the two figures are essentially identical and, therefore, will be discussed

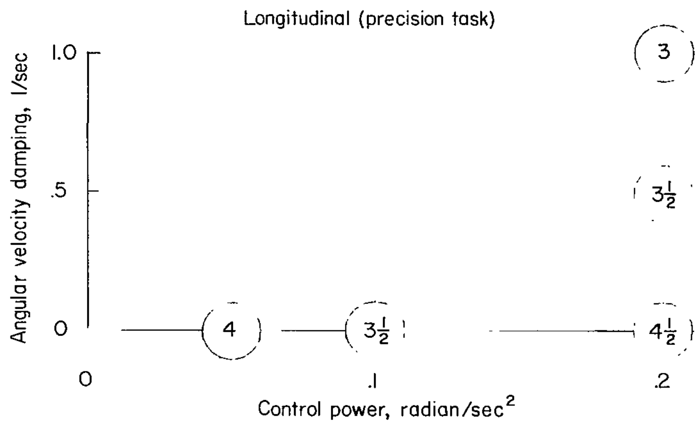

Figure 6.- On-off control results for longitudinal precision tasks in the absence of disturbances, trim changes, and static stability.

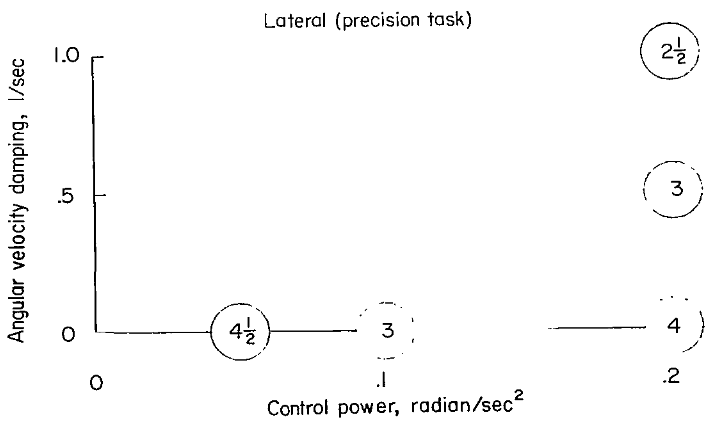

Figure 7.- On-off control results for lateral precision tasks in the absence of disturbances, trim changes, and static stability.

at the same time. With the exception of only one point, the pitch axis was rated 1/2 a unit poorer for precision flight than corresponding conditions in the roll axis. This rating difference was attributed to the position of the pilot relative to the axis of rotation which resulted in normal accelerations that somewhat aggravated precise control when pitch motions were commanded. The ratings presented in figures 6 and 7 were generally obtained from the landing task which was more critical than either hovering over a spot or the precision ground-track tasks.

For the conditions of zero damping, figures 6 and 7 indicate that the intermediate value of control power, 0.1 rad/sec^2, was satisfactory. This satisfactory rating at zero damping, however, is of academic interest only, since, as shown in the previous section, this condition is unsatisfactory for maneuvering. At the lowest control power, 0.05 rad/sec^2, the aircraft responded too slowly and at the highest value, 0.20 rad/sec^2, difficulty was experienced in making the input of sufficiently small duration for precise control. In other words, for the shortest control pulse that the pilot could command with high control power and zero damping, the aircraft attitude would overshoot and the ground position would be momentarily lost. By increasing the damping to 0.5/sec at the high value of control power, the ratings indicate that the precision improved to a satisfactory level. With the damping increased to 1.0/sec, further improvement in precise control was realized. For the conditions which were rated $3\frac{1}{2}$ or better, the pilots commented that the precision with which the aircraft could be landed was as good as that experienced during previous tests using a proportional control.

Effect of Out-of-Trim Condition

In an effort to determine the extent of potential control problems associated with an out-of-trim condition such as would be produced by a change in aircraft center-of-gravity position, trim changes corresponding to a maximum of 25 percent of the simulated control power were investigated. As opposed to a proportional control, a simple on-off control cannot produce a steady control moment to balance an out-of-trim moment, rather the effect of an out-of-trim moment must be balanced on a time-averaged basis by pulsing the control.

Out-of-trim variations were made in both the pitch and roll axes for two levels of damping and for a control power of 0.20 rad/sec^2. For the precision hovering task there was little or no deterioration in controllability (the deterioration which occurred in the maneuver task is discussed subsequently) even for the 25-percent out-of-trim condition, which corresponds to an angular acceleration of 0.05 rad/sec^2. In fact, for the hovering task the pilots were often unaware of the 25-percent trim change even though analysis of the flight records indicated that the pilot continuously pulsed the control with a time-on duration of about 0.5 second and with a frequency of once every 2 seconds.

Detrimental effects arising from the out-of-trim condition were quite apparent during performance of the maneuver tasks. The problems encountered during maneuvering appeared, however, to be related entirely to the effect of the trim change on the control symmetry rather than the constant moment which it produced when the control was in the off position.

For example, when the aircraft was rolled or pitched in the same direction as the trim change, the aircraft seemed very responsive and encouraged the pilot to use high angular rates. However, when the control was reversed to reduce the angular rate to zero, the amount of control power which was available in the direction opposing the trim change was not adequate to prevent overshooting the desired attitude.

Although the results indicate that an out-of-trim condition of up to 15 percent was no problem during maneuvering, the deterioration in controllability for the 25-percent case was about one pilot rating unit. These results seem to agree well with the results of reference 5 which show a rapid deterioration in controllability for constant disturbance torques greater than 20 percent of the moment provided for control. On the basis of these tests it appears that an automatic trimming device of some type would be required where out-of-trim moments greater than 15 percent to 20 percent of the control power could occur.

Effect of Static Stability

Tests were conducted to determine the effect of speed stability and dihedral effect (i.e., lateral speed stability) on handling qualities. The static-stability levels employed simultaneously in the pitch and roll axes were 0, 0.005, and 0.01 $\frac{\text{rad/sec}^2}{\text{ft/sec}}$ $\left(0, 0.016, \text{and } 0.033 \frac{\text{rad/sec}^2}{\text{m/sec}}\right)$ and the control power was held constant at 0.2 rad/sec^2. The tests were run at two levels of damping – 0 and 0.5/sec.

The level of speed stability present in an aircraft defines the amount of control moment required to balance the moment which results from a change in airspeed. The maximum speed change which can be trimmed with the primary control is given by the ratio of the control power to the static stability, so that the maximum trimmable speed change for the speed stability condition of 0.01 $\frac{\text{rad/sec}^2}{\text{ft/sec}}$ $\left(0.033 \frac{\text{rad/sec}^2}{\text{m/sec}}\right)$ was only about ±12 knots during these tests. In any practical application of the on-off control, however, some auxiliary trim device would, of course, be employed to compensate for the static moments resulting from speed changes. From an overall viewpoint, therefore, the present tests represented a hypothetical situation to the extent that there was no provision for establishing a new trim speed. Since the primary purpose of these tests was to determine the aircraft controllability near the trim speed, which in these tests corresponds to hovering, the lack of an auxiliary trim device should not detract from the generality of the results.

The results of these tests indicated that for precision flight and for maneuvering near the trim speed, the pilot experienced no increased control difficulty even at the highest level of static stability of the tests. As might be expected from the preceding section on the effects of out of trim, however, very severe control difficulties were encountered when the maneuvering involved translational speeds which approached the limiting trimmable speed. Under such conditions the control margin approaches zero so that even a small disturbance could be catastrophic. It should be noted that the same problem would exist with a proportional control and is mentioned here only to emphasize the need for adequate control margins at all times.

Based on an analogy with the out-of-trim tests, it appears that the on-off control in its simple form could be used satisfactorily only in those aircraft where the maximum speed change for the on-off control mode would produce pitching moments or rolling moments no greater than about 20 percent of the control power. Depending on the aircraft configuration, of course, alternate methods for handling moments arising from static stability would likely be available. For example, in a multiengine jet lift aircraft, it might be feasible to use differential thrust modulation to balance out the relatively slow variation in static moments, while using a faster-acting bleed air system in an on-off fashion to achieve the reduction of control power for maneuvering which this investigation has shown to be feasible when the on-off method is employed.

Effect of Disturbances

The effect of angular disturbances on controllability while using an on-off control was investigated for the lateral and longitudinal axes simultaneously during precision hovering and vertical landing. The effects of disturbances on maneuverability were not investigated during the present tests owing to the considerable flight time required to ensure that the peak disturbances would occur simultaneously with the most critical phases of the maneuver. It is believed, however, that under conditions of heavy disturbances the pilot would be most concerned with keeping the aircraft level (more nearly a precision task) and would do as little maneuvering as possible.

Since earlier flights in this investigation had shown that it was not possible to obtain satisfactory control for both the maneuver and the precision task with any of the tested combinations having control power values less than 0.20 rad/sec^2, only the higher control power was considered during the tests with disturbances. The effect of disturbances on performance of the precision tasks is illustrated for the pitch and roll axes in figures 8 and 9 which show a plot of the individual pilot ratings assigned by two NASA pilots against the angular acceleration produced by the peak disturbance. The nature of the disturbances is discussed in appendix B where it is shown that the peak disturbances occur several times per minute and have an amplitude of about $2\frac{1}{2}$ to 3 times the root-mean-square disturbance level. The variation of pilot rating with disturbance level was explored at the three levels of damping indicated in the figures.

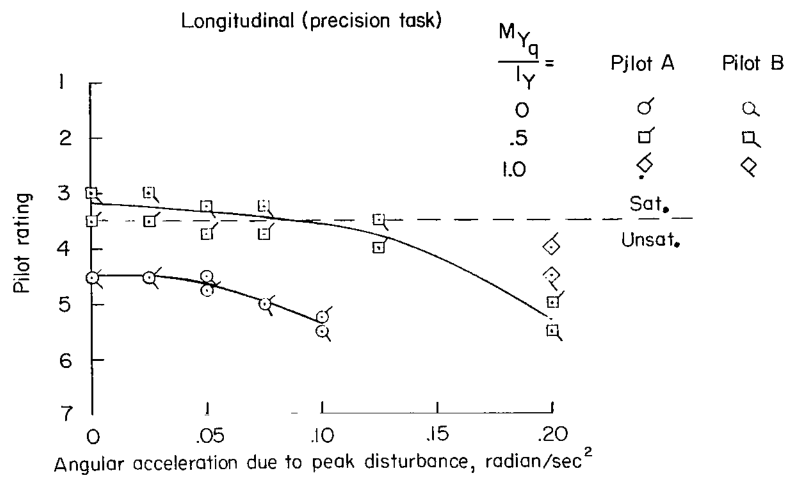

Figure 8.- Variation of pilot rating of longitudinal controllability during the precision tasks with control power equal to 0.20 radian/sec^2.

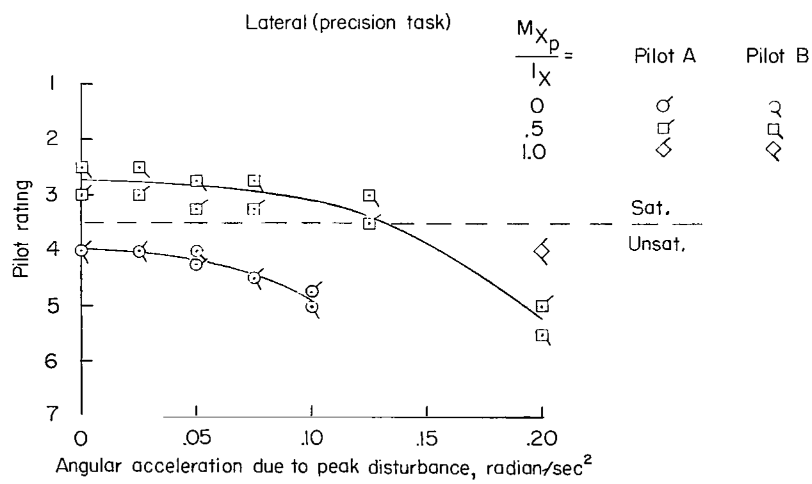

Figure 9.- Variation of pilot rating of lateral controllability during the precision tasks with control power equal to 0.20 radian/sec^2.

It is noted from figures 8 and 9 that both pilots indicate approximately the same rate of deterioration with increased disturbance level, and, as might be expected, the rate of deterioration occurs sooner and more rapidly for the conditions with the lower damping. For the condition with 0.5/sec damping, the deterioration was very gradual and, in fact, was still satisfactory up to peak disturbances of about 0.10 rad/sec^2 and 0.12 rad/sec^2 for the pitch and roll axes, respectively. The pilots agreed that the simulation of disturbances was very realistic and felt that the disturbance level wherein the peaks corresponded to about 0.10 rad/sec^2 was like a very gusty day. It is significant that the pilot was able to cope satisfactorily with disturbances which were on the order of 50 percent of the control power.

In general, it appears that the on-off control enabled the pilots to maintain precise control over the aircraft for all but the lowest conditions of damping or the most extreme levels of disturbance. The damping was beneficial in that it opposed the angular rates resulting from the disturbances and permitted the pilot more time to take corrective action. It should be noted, however, that the disturbance results might be very different if gross maneuvering had been attempted.

DISCUSSION

On the basis of pilot commentary and the data obtained, it appears that the range of satisfactory combinations of control power and damping encompasses a relatively small region. The narrowness of this satisfactory region results from the small difference in the control power value which was too sluggish for maneuvering and the slightly higher value which would tend to be too powerful for precision control. It is thought, however, that pilot location relative to the axis of rotation would be a significant parameter in establishing the upper limit on satisfactory control power. In figures 10 and 11 estimates of the minimum satisfactory regions (the areas enclosed by $3\frac{1}{2}$ pilot-rating boundaries) are presented for the pitch and roll axes, respectively. The regions, as shown, provide a satisfactory level for both maneuverability and precision flight.

The fact that the satisfactory region for the on-off control is both small and closed has also been confirmed by unpublished results which were obtained with the six-degree-of-freedom simulator at the NASA Ames Research Center. The small, closed nature of this region is likely to place a more exacting burden on the designer. For example, the reduction in aircraft inertia which accompanies expenditure of fuel and stores results in an increase in control power. While this would generally represent a shift in the favorable direction for proportional systems, such an increase in control power for the on-off control might shift the response into an unsatisfactory region. However, for the single test condition which was flown within the satisfactory region, the pilots commented that the precision was as good as for a proportional control and that the maneuverability was adequate.

Comparisons were made between the control power results of the present investigation and a similarly conducted study which employed a proportional control (ref. 4). Both investigations employed the same test vehicle and simulation technique and two of the pilots participated in both studies. Although the pilots utilized approximately the same angular rates in both investigations for conditions receiving similar ratings, the

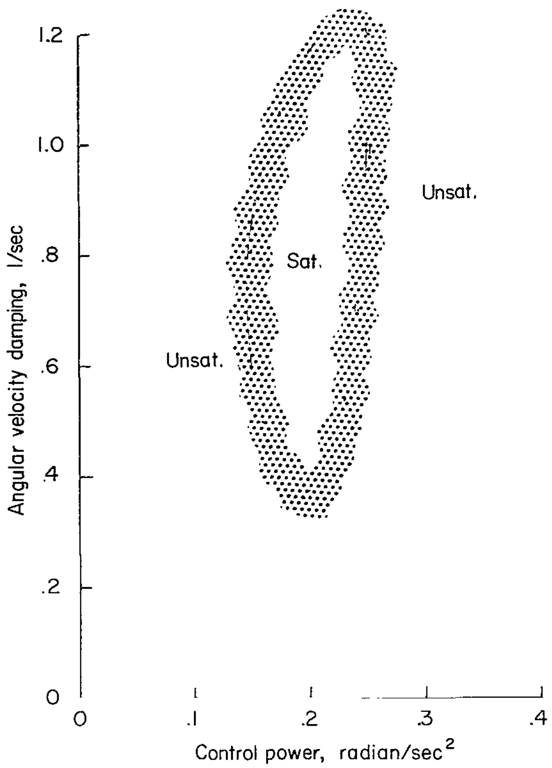

Figure 10.- Satisfactory longitudinal control power and damping region as estimated for on-off control for both maneuvering and precision flight.

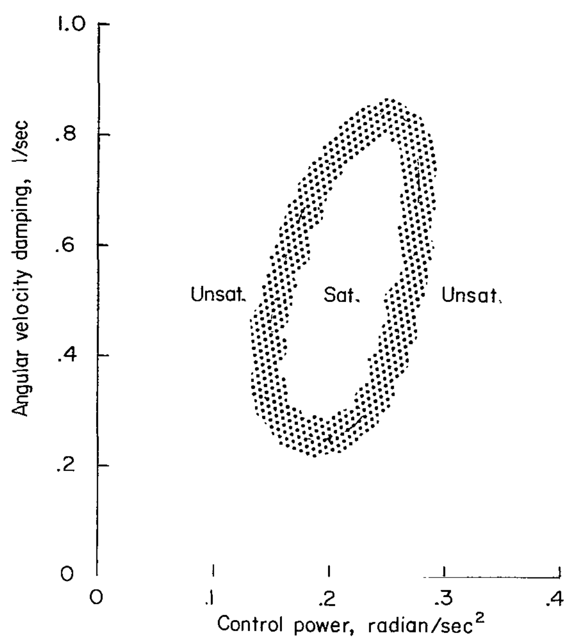

Figure 11.- Satisfactory lateral control power and damping region as estimated for on-off control for both maneuvering and precision flight.

control power requirements for the on-off control were reduced to about one-third the level required for satisfactory maneuverability with the proportional control.

The rapid adaptability by all the pilots to the on-off control for V/STOL operation was unexpected and emphasized the ease with which the system could be flown. The learning time for each of the five pilots was approximately 10 minutes. It seems significant that, for the conditions which were rated satisfactory, the pilot was not particularly aware that the control was of the on-off type. In contrast to what might be expected with an on-off control, observation of the stick motion and inspection of flight time histories indicated that the stick was moved continuously within the dead band even while no control moment was needed (rather than being held fixed until control was required). These small continuous control motions within the dead band appeared similar to the motions which are observed with proportional controls and seem to serve to keep the pilot in the "loop," so to speak, even when no control is imminently needed.

The control benefits realized from the on-off control appear to be related largely to the fact that it effectively provides a high sensitivity (angular acceleration per unit control travel) without causing the touchiness usually associated with high sensitivities. With the ±1/4-inch (0.64-cm) dead band and the control power of 0.2 rad/sec^2, the effective sensitivity is 0.8 rad/sec^2-in. (0.32 rad/sec^2-cm). During the proportional control study reported in reference 4, test combinations having sensitivities which approached this value were seriously downrated because of the control touchiness which allowed aircraft motions to feed back into the control system through the pilot's hand. In the case of the on-off control, the sensation of oversensitivity was eliminated by the dead band which provided a small region within which no acceleration was commanded. On the other hand, the dead band was small enough and the stick forces were light enough to permit the control to be triggered precisely when it was needed and with very little motion of the pilot's hand. In other words, use of the on-off control with a small dead band serves to prevent small inadvertent stick motions from disturbing the aircraft and still puts the full control capability at the pilot's fingertips where it can be used more efficiently.

CONCLUSIONS

A flight investigation of low-speed and hovering capability was conducted with an on-off control about the pitch and roll axes in combination with various levels of control power, angular-velocity damping, out-of-trim condition, static stability, and artificial disturbances. The test vehicle was a variable-stability helicopter. On the basis of these tests the following conclusions were drawn:

1. Use of an on-off relationship of control moment to stick motion with a properly sized dead band resulted in a control power reduction to approximately one-third of the level required for satisfactory maneuverability with a proportional control.

2. The pilots adapted to the on-off control very rapidly.

3. As deduced from pilot commentary, the region of satisfactory control on a plot of angular-velocity damping against control power appears to be relatively small, owing to the small difference in the value of control power which is too sluggish for maneuvering and the slightly higher value which tends to be too powerful for precise control, such as would be required for vertical landing and confined areas.

4. Angular accelerations resulting from an out-of-trim condition or from static stability should be handled by auxiliary means or by a more sophisticated on-off control when such moments exceed 15 percent to 20 percent of the control power.

5. For combinations of control power and damping rated satisfactory for maneuver, random disturbances which produce peak moments of less than 50 percent of the control

power can be handled satisfactorily during the performance of precision tasks. (The effects of such disturbances on maneuvering were not evaluated.)

6. The size of the control dead band has a significant effect on the pilot rating.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., March 25, 1966.

APPENDIX A

SIMULATION TECHNIQUE AND ACCURACY

In the signal-flow diagram in figure 12, the simulation technique and the essential features of the mechanization of the equations of motion for the on-off control study are

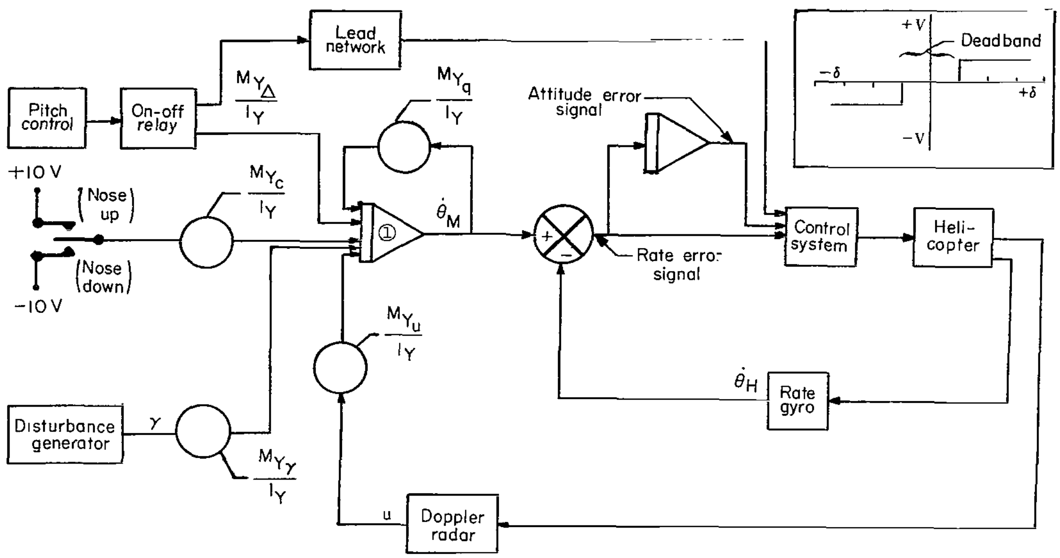

Figure 12.- Simplified signal-flow diagram for pitch axis.

illustrated for the longitudinal axis. A corresponding mechanization was used for the lateral axis. As indicated in figure 12, motion of the pilot's control stick produced a signal which was used to actuate an on-off relay whenever the stick motion exceeded a preselected travel, or dead band. As indicated in the insert at the top of figure 12, while the stick was maintained within the dead band, the relay output was zero; hence, no motion was commanded. When the stick was maintained outside the dead band, the relay was actuated and its output became plus or minus a constant value, depending on the direction of the stick displacement. The output of the relay was fed into integrator (1) which computed the model angular velocity on the basis of the control position and the other inputs to the integrator.

The simulation of aerodynamic angular-velocity damping was accomplished by feeding back a signal proportional to the model angular velocity into the summing junction of integrator (1).

Simulation of static stability (i.e., speed stability in the case of the longitudinal axis) was accomplished by feeding a signal proportional to the forward component of the aircraft velocity into integrator (1). In the absence of an accurate method for sensing airspeed at very low speeds including hovering, ground-referenced velocity signals, which were obtained from an onboard Doppler radar system, were used to provide the required velocity signals.

An electromechanical disturbance generator was used in the tests to provide a controlled disturbance level. A description of the disturbance generator and its characteristics is given in appendix B. The disturbance generator permitted the root-mean-square disturbance level to be varied from near zero (since the model simulation technique tended to eliminate external disturbances) to any desired magnitude.

Trim changes of the type which would be caused by a shift in aircraft center-of-gravity position were simulated by feeding a constant voltage into integrator (1).

The degree of simulation accuracy which was obtained during the on-off control investigation can be estimated on the basis of the time histories shown in figure 13. In

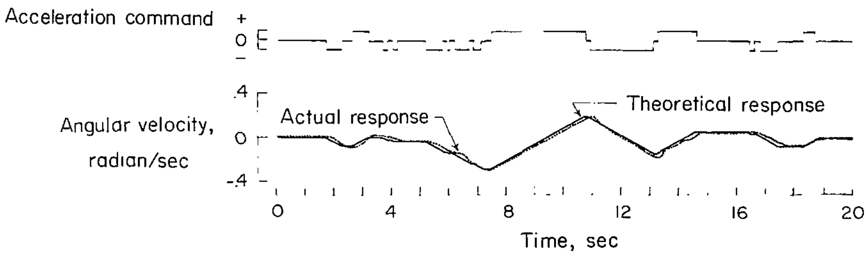

Figure 13.- Comparison of the actual and the theoretical responses for $M_{Y_\Delta}/I_Y = 0.15$ radian/sec^2.

this figure are shown the acceleration command, the theoretical angular-velocity response, and the actual angular-velocity response for the longitudinal axis. The time histories shown in the figure were obtained for a zero-order model, that is, a model aircraft having neutral stability so that the response to a control input was pure angular acceleration. The zero-order response was chosen for illustration here because it represents the most difficult case for the test helicopter to follow.

By taking the theoretical response in figure 13 as the reference, a time lag of about 0.2 second is observed in the actual response. The amplitude of the actual response, on the other hand, is considered to be reasonably accurate as may be observed by shifting the relative horizontal position of either of the time histories by about 0.2 second. Despite the fact that time delays on the order of 0.2 second are sometimes considered to be detrimental to handling qualities (and perhaps rightfully so), there was no evidence of such problems in these tests. Even though some of the pilots were forewarned of the existence of the time delay, none of them were able to detect it and it is not believed that this time delay had any adverse effect on the results.

APPENDIX A

There are several factors which might account for the acceptability of the time delay in the present case. First, the pilot had no cue as to precisely when the control dead band was exceeded since overtravel was available rather than having control stops at the edge of the dead band. Second, the dead band was quite small so that full control was commanded in a very small control travel and, hence, in a shorter time than the pilot normally takes to put in a large control deflection with a proportional control. Therefore, the total elapsed time from when the pilot senses the need for a control displacement to the time when the aircraft has a perceivable response is probably considerably shorter in the present simulation than in aircraft with conventional control systems. Also, in any aircraft there is some time delay in the response to control deflection. Third, a factor which might mask the effects of the time delay in the present case is that, on a long-term basis (greater than a few tenths of a second), the attitude error signal sees an error due to the lag and feeds in additional control so that the attitude change after a sufficient time will be nearly correct. It is seen from this last example that the time delay associated with this simulation lacks a clear analogy with classical time delays in that, on a long-term basis, the simulation technique used tends to overcontrol to compensate for the initial delay. All things considered, the results presented herein would perhaps correspond most closely to an aircraft with a control system having a combined first-order and transport time delay between 0.1 and 0.15 second.

APPENDIX B

DESCRIPTION OF DISTURBANCE GENERATOR

The disturbance generator used in the present investigation is an electromechanical device which employs the voltage outputs of three cam-driven potentiometers to provide random inputs to the model aircraft. Each of the three cams rotates at a slightly different frequency near 1 cycle per minute. The voltage output of each of the potentiometers is combined into a single output so as to obtain an output which repeats infrequently (owing to the slight variation in cam speeds). By superimposing the outputs of the cam-driven potentiometers on different fixed gains, it was possible to achieve somewhat independent sources of disturbances for the different aircraft axes.

A sample of the disturbance generator output is shown in figure 14. The maximum voltage peaks correspond to about $2\frac{1}{2}$ to 3 times the root-mean-square output of the generator which is in reasonable agreement with the familiar rule of thumb for natural turbulence. A power-spectral analysis of the simulated disturbances is shown in figure 15 for root-mean-square disturbance level of 0.04 rad/sec^2 for which case the peak disturbances were about 0.1 rad/sec^2. As can be observed from this plot, the intensity of the disturbances was concentrated at fairly low frequencies which would necessitate positive corrective action on the part of the pilot.

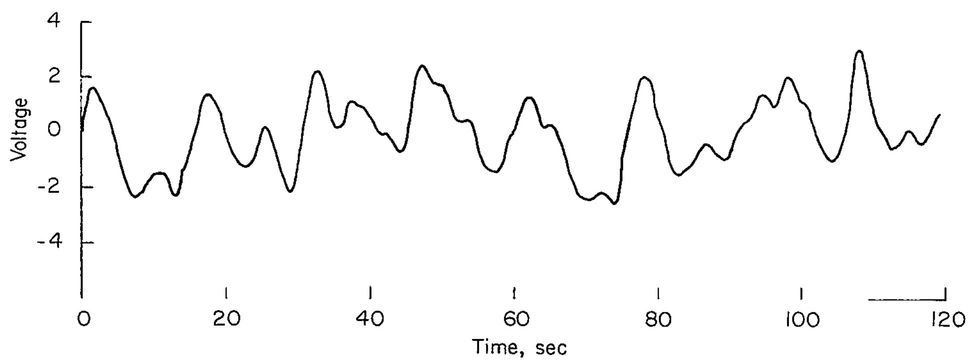

Figure 14.- Sample of disturbance-generator output. Root-mean-square level = ~1V.

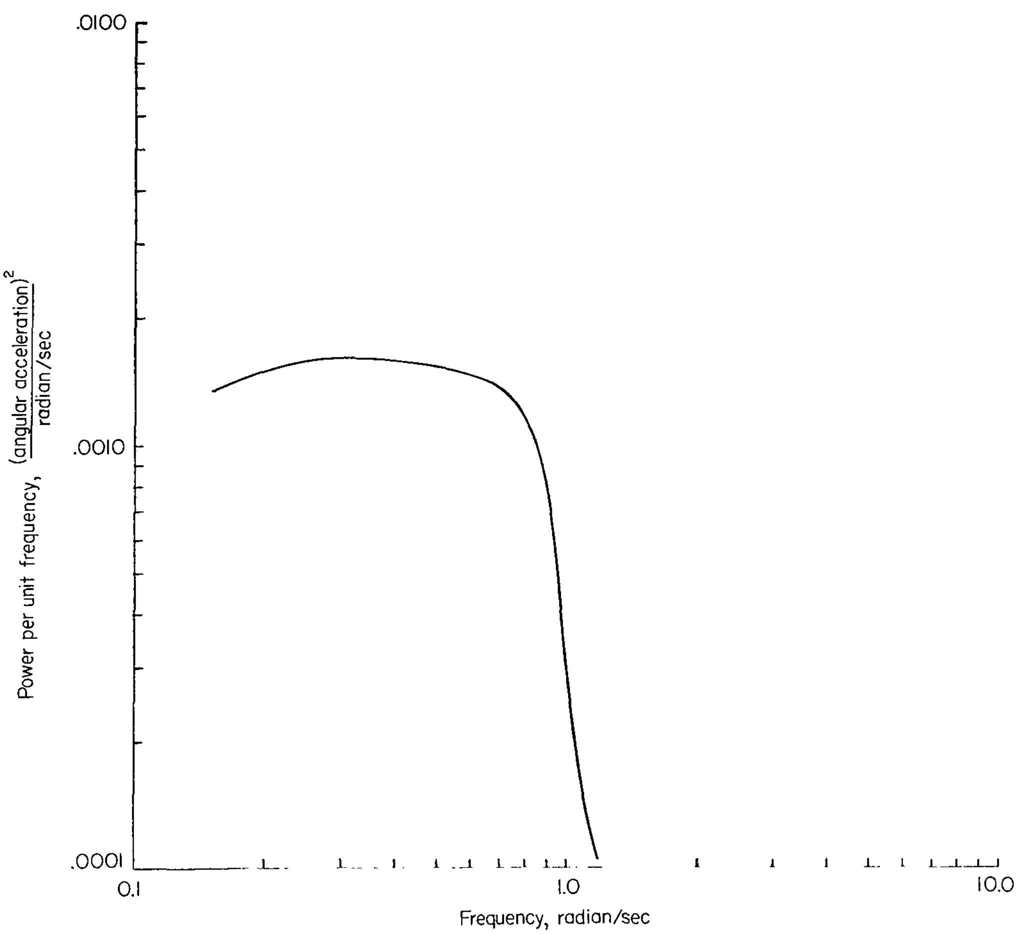

Figure 15.- Power spectral analysis of simulated disturbances scaled to provide a selected root-mean-square disturbance.

APPENDIX C

DESCRIPTION OF TASKS

Maneuvering Tasks

Longitudinal quick starts and stops.- During the longitudinal quick starts and stops, the helicopter was pitched down to accelerate to about 45 knots. After a short run at 45 knots, the aircraft was rapidly decelerated to a hover over a preselected spot.

Lateral quick starts and stops.- During the lateral quick starts and stops, the helicopter was rolled to the right or left to obtain a lateral acceleration until a sideward velocity of about 25 knots was attained. The aircraft was then rolled in the opposite direction and rapidly decelerated to a hover over a preselected point.

Turn reversals and landing approach.- The pilot made visual approaches to a specific landing spot at a speed of about 45 knots and an altitude of approximately 100 feet (30.5 m). During this approach the pilot executed rapid S-turn maneuvers, both to the left and to the right, before coming to a hover and descending to the intended landing point.

Precision Tasks

Precision hovering.- The precision hovering task involved attempts at proceeding a very short distance to a point over a spot on the ground, stopping, and accurately maintaining this position. In order to provide a close visual reference, the task was performed at heights no greater than 20 feet (6.1 m).

Vertical landing.- After gaining reasonable familiarity with the test conditions during the precision hovering task, vertical landings to touchdown were attempted.

Precision ground track.- The precision ground-track task involved attempts at flying in a straight line at an airspeed of 45 knots. This task was performed at a height of approximately 100 feet (30.5 m).

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