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FLIGHT TESTING OF AN ASW HELICOPTER

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ABSTRACT

A program of flight trials carried out with a Sea King Mk.50 anti-submarine warfare (ASW) helicopter is described. An outline is given of the aircraft instrumentation, the data acquisition system and the data analysis procedures. Results are presented for conditions of trimmed level flight together with typical examples of dynamic response tests, transition manoeuvres and Doppler and cable hover tests. The flight trials program was designed to provide results both for use in the validation of a mathematical model of the Sea King helicopter and for more general use in helicopter flight behaviour studies.

1. INTRODUCTION

A mathematical model of the Sea King Mk.50 helicopter has been developed by Aeronautical Research Laboratories (ARL) and is reported in [1] and [2]. The model is an attempt to represent both the performance and dynamic flight behaviour of this helicopter over a wide range of conditions. These conditions include the use of a dunking sonar in the anti-submarine warfare (ASW) role. It is in this operational role that the Sea King Mk.50 is used by the Royal Australian Navy (RAN) and the model is considered to be a useful tool in supporting operation of both the aircraft and its training simulator.

To establish confidence in the mathematical model, a set of flight test data are required for comparison with model behaviour predictions. The problems of obtaining adequate data to satisfy the needs of simulation have been noted by Hatfield [3]. Only a limited amount of data for the Sea King Mk.50 were available from the manufacturers [4,5], and as the results of Perkins [4] pertain to the earlier Mk.1 version, it was decided to conduct flight trials on the RAN Mk.50 version.

The purpose of this document is to describe the trials program undertaken and to present a representative selection of results. These include trim flight data, dynamic responses to pilot inputs, and an ASW transition manoeuvre. Full descriptions of the instrumentation, data acquisition system, and data reduction procedures are given elsewhere [6-16] and are only briefly outlined herein. Whilst the acquisition of data for mathematical model validation was of prime importance in devising the test schedule, the opportunity was taken to include tests useful in the more general field of helicopter flight behaviour studies.

2. FLIGHT TEST PROGRAM

2.1 General

In order to achieve the program objectives, a wide range of tests was scheduled which involved performance, stability and control, and ASW manoeuvres. The latter included automatic transitions in both altitude and velocity, together with dynamic responses to disturbances while the helicopter was in the ASW Doppler hover and cable hover modes. No previous data relating to ASW dynamic response tests have been found in the literature.

In drawing up the test schedule, procedures outlined by the US Naval Air Test Centre [17,18] were followed.

2.2 Scope of Flight Tests

The schedule was detailed into a number of sorties (flights), the majority being specified to take place in low wind conditions, although certain transition and dunking manoeuvres were specified at higher wind conditions for performance evaluation purposes. An adequate range of all-up weight (AUW), centre of gravity (c.g.) position and wind combinations was achieved. Generally the trials may be classified as relating either to performance or stability and control aspects. Some extra flight time was involved with the calibration of certain data channels and this activity is discussed in Section 4.

2.2.1 Performance Tests

In the present context, performance tests connote the achievement of steady conditions with all time derivatives equal to zero. Table 1 lists the performance tests in the flight schedule together with the extent of the data bank.

2.2.2 Stability and Control

In an analogous fashion, Table 2 lists the stability and control tests performed. In tests on the dynamic response to flying control inputs, a variety of inputs including steps, pulses, and doublets was specified with the automatic flight control system (AFCS) both ON and OFF. Other dynamic responses studied were the effects of beeper trim switching, recovery from disturbances while in Doppler and cable hover, and the simulated failure of one engine. Included also were manoeuvring stability tests and the more slowly varying aircraft manoeuvres such as transitions, sonar transducer raising and lowering, and the landing phase.

3. INSTRUMENTATION

3.1 Flight Instruments

While it may have been desirable to have an instrumentation package completely independent of the aircraft instrumentation, the wide range of variables measured during the trials program rendered this impracticable. In particular, signals relating to certain variables could be derived only from the special role equipment carried in the aircraft, e.g. Doppler groundspeed components and ASW cable information. Further reliance on the aircraft instrumentation was imposed by funding restrictions which limited both the number and quality of transducers contributed by ARL.

Figure 1 is a sketch of the aircraft flying controls. Although it would be desirable to measure the pitch angles of the main and tail rotor blades directly, and hence avoid backlash and bending effects in the control linkages and swash plate mechanism, this was not practicable. Instead, the angles had to be deduced from a knowledge of the relationships (i.e. gearings, mixing unit cross-couplings, and servo dynamic response characteristics) between the components of the flying controls (see Section 4.1) as determined by calibrations. In this case, linear displacement transducers were used to measure the auxiliary servo jack displacements of each control channel. The yaw and collective flying control positions were similarly measured, while, for the pitch and roll stick positions, the inbuilt linear variometers (LINVARS) were used. The AFCS contribution to each channel may be calculated from a knowledge of the control and jack positions. Meters indicating each control position were also installed in the cockpit to assist the pilot during dynamic response testing.

By measuring aircraft angular rates as well as attitudes, some redundancy was deliberately achieved. Pitch and roll rate signals were obtained from rate gyros supplied and fitted by ARL, whilst the yaw rate signal was derived from the aircraft AFCS. The aircraft system also supplied pitch and roll attitude signals from the vertical gyro and the yaw attitude (heading) from the gyro-compass. The retention of this degree of redundancy has been justified by kinematic consistency checking [16], which revealed not only errors in calibration expressions but indicated inadequacies in some instruments under certain conditions.

ARL also supplied and fitted triaxially mounted linear accelerometers near the aircraft c.g. position. Other linear motions measured were the longitudinal and lateral groundspeed components, obtained from the aircraft Doppler radar.

Motion of the aircraft with respect to the airmass was measured by an instrumented boom projecting from the nose of the aircraft (Fig. 2). Both sideslip and angle of attack vanes were fitted in addition to the pitot-static probe. While these instruments were not directly affected by rotor downwash, for airspeeds above 30 knots it was necessary to determine the probe position error by calibrations against a trailing pitot-static probe [8].

Altitudes up to about 450 ft were measured by the aircraft radio altimeter, both raw and smoothed signals being recorded. The boom static pressure reading, corrected for position error, was used to determine the higher altitudes.

Other quantities which were important in determining the flight dynamic behaviour of the helicopter were rotor speed and the torque of each of the two engines. In addition, those quantities which illustrated the performance of the aircraft in the ASW sonar dunking mode of operation were measured: i.e. longitudinal and lateral cable angles, cable length, and ball depth. To complete the information on air data, outside air temperature was also recorded.

Table 3 is a list summarizing the quantities measured together with their sampling rates and transducer types. It should be noted that channel 18 can be manually switched in flight to record either of two quantities. In addition, channels 31 and 32 can be connected to either of two quantities depending on the type of test being performed. Sampling rates were chosen to cope with the maximum expected rates of change of the quantities measured. Additional quantities derived from measured values are listed in Table 4 and will be referred to when discussing data reduction procedures (Section 4.3).

3.2 Data Acquisition System

This system was developed by ARL and is described fully in [9]. The system allows, without sub-multiplexing, 20 channels of data to be recorded in 12-bit form at a sampling frequency of 60 Hz for each channel. By sub-multiplexing six of these channels (three at 2:1 and three at 4:1), up to 32 data sources are recorded. The sources with limited high frequency content are assigned to submultiplexed channels where the lower sampling rates are quite adequate. Recording was on magnetic tape, with 15 minutes of recording per tape and an overall voltage measurement accuracy better than 0.2%.

A Nagra IV-SJ tape recorder was chosen because of its proven capability in the severe vibration environment which exists in a helicopter. This recorder has two data tracks plus one voice commentary track. To efficiently utilize its capacity, data were recorded on each of the two data tracks using a self-clocking code. The recording density, however, was set at a conservative value in order to limit tape "drop-outs".

In the recording process, the analogue signals are sample by a multiplexer, digitised, converted to serial form, encoded, and written onto the tape as serial digital words, each of 12 bits. The operation of the system is synchronized by a control and timing unit, Because the analogue input signals are provided by a wide variety of transducer types, a variety of signal conditioning circuits, which incorporate six-pole Butterworth active filters, are employed. A quick-look facility is also provided in the airborne system which enables the operator to monitor, in analogue form on an oscilloscope, any selected input channel during recording or on playback.

Post-flight examination of the flight record at the test venue was possible using ground-based quick-look facilities [11]. This enables hard copy analogue output records, for any of the 32 recorded channels, to be produced simultaneously on a (six-channel) paper trace recorder. This quick-look facility is sufficiently accurate for the output traces to be used for preliminary analysis. It also provides a decoded digital output with appropriate timing pulses, for use with a transcriber [12]. The latter device converts the data into a form suitable for recording on seven-track computer-compatible tape for subsequent processing.

3.3 General Comments on Performance of Instrumentation

Experience during these trials suggests that much extra time and effort is required for data processing and assessment when the calibrations of certain transducers vary from flight to flight. Some pressure transducers were deficient in this respect and a discrepancy encountered during the boom probe calibration [8] could only be explained by a change in calibration, possibly caused by the high vibration environment. Linear accelerometer outputs were also affected by this enviornment and required both analogue and digital filtering to remove unwanted rotor and blade frequency components. In future work, more care should be taken to site these transducers in a region of reduced vibration level. Adequate results were obtained from the rate gyros but could be improved further by better siting together with improved calibration and checking facilities.

Linear displacement transducers used to measure control and auxiliary servo displacements were generally satisfactory. However, in future work, the direct measurement of blade root angles would be desirable and, failing this, the primary jack displacements. Other improvements, such as a reliable low airspeed measuring system to obtain more consistent trim flight data, and a better jig for "cleaner" pilot inputs would also be advantageous.

4. DATA REDUCTION

4.1 Calibration Ground Tests

Ground tests were needed for the calibration of many of the channels shown in Table 3 and also for the measurement of the characteristics of the flying controls of the aircraft. Each data channel requires calibration to enable digitally recorded data to be expressed in engineering units. Channel calibrations derived from ground tests are fully described in [13] and [19]. In all cases the data acquisition system was used to record calibration data while the transducer signal was varied incrementally over its operating range. At the same time, the digital voltmeter of the quick-look facility on the data acquisition system was read, allowing preliminary calibrations to be derived prior to magnetic tape processing.

The measurement of the aircraft flying controls characteristics provided essential data for the control system part of the mathematical model. Two main types of tests were involved. The first was to measure the dynamic response characteristics of the primary and auxiliary servos in the flying controls, and the second was to measure the gearings in the flying controls. To measure dynamic response characteristics, transducers were mounted on the output shafts of the appropriate servos, and photographic records were made of oscilloscope traces of the transducer outputs. While some tests were performed using manually stepped stick and pedal inputs, better results were obtained using the "hard-over" test facility of the aircraft flight control system to produce step inputs.

To obtain the gearing relationships, each channel of the flying controls was investigated separately. Sticks and pedals were moved incrementally over their complete ranges and measurements of angles and positions made. For each angle or position, the corresponding positions of servo input and output shaft and blade angles were measured and recorded. Cross-coupling between the collective controls and the other flying controls necessitated these measurements being done at both extremes of the collective stick travel. The calculation of blade angles from auxiliary servo positions measured during the flight testing used a simplified model of the flying controls [19].

4.2 Calibration Flight Tests

Flight tests were performed for calibrating Doppler velocity, radio altitude, yaw attitude and rate, sonar transducer depth, cable length and boom dynamic pressure measurements. All quantities, with the exception of the boom dynamic pressure, are derived from the aircraft instrumentation system. In some cases, the calibration involves acceptance of the instrument reading as being correct. However, in calibrating the Doppler velocity channels it was possible to determine the true velocity by the use of the pacer vehicle technique or, in the case of high speed runs, measuring elapsed time of flight between runway markers at known spacing.

The calibration of the nose boom pitot-static probe against a reference towed probe (Fig. 3) is described in [8]. A range of flight conditions covering climb, descent, and level flight was used from which the boom probe airspeed calibration and position error correction were derived.

4.3 Data Processing

The data reduction procedures described fully in [14] are now briefly outlined. In order to process the data on a mainframe computer, two transciptions are performed. Using the transcriber [12], twin-track NAGRA data are transferred to computer compatible seven-track magnetic tape. A special program then converts each flight record on the tape to a disk file, at the same time providing automatic file labelling and error checking. Disk files of raw data can then be processed by other special programs which provide a variety of facilities thus:

(a) Correction of "drop-outs", i.e. misplaced data bits.

(b) Corrections for instrument time delays and delays arising from phase shifts introduced by use of analogue and digital filters.

(c) Conversion of data to engineering units assuming linear calibration.

(d) Digital filtering to remove noise [15]. Smoothing by least squares also available.

(e) Derivation of additional quanitites (see Table 4):

- (i) from boom-mounted pitot-static probe;
- (ii) blade angles;
- (iii) Euler angles;
- (iv) kinematic consistency quantities.

(f) Output of processed data in tabular form and in a variety of graphical formats such as strip plots, multi-curve overlay plots, and cross plots.

Kinematic consistency checking methods, as described fully in [16], have been applied to a reasonable cross-section of all data, including data presented in this report. This has led to the resolution of uncertainties in the calibration constants of some channels by comparing measured quantities and equivalent kinematic quantities derived by integration or differentiation of other measurements. These methods have also enabled the identification of suspect data which may arise from overloading and instrument malfunction under certain flight conditions. Derived quantities can then be substituted for such invalid data.

5. RESULTS

5.1 Level Flight Trim Curves

The dependence of a selection of relevant trim flight variables on airspeed is shown in Figure 4 for a nominal range of - 16 kn to 110 kn. The variables of main interest are:

(a) attitude angles in pitch and roll;

(b) engine torque;

(c) flying controls positions; and

(d) corresponding blade angles.

Because of the difficulty in achieving steady state conditions, particularly at low speeds, the plotted points represent the mean of the maximum and minimum excursions of a variable throughout a nominally "steady" sample of order of 10 seconds duration. An indication of the average excursion throughout the speed range is given for each variable.

Although relatively calm conditions prevailed during the low speed trim and Doppler tests, the presence of a light wind blowing directly across the runway meant that the aircraft could not be headed along its flight path without sideslip occurring. Data were therefore not obtained for pure symmetrical flight. In particular, some of the scatter apparent in Figure 4b for roll attitude may be attributed to the need to offset the sideslip by a change in roll attitude. Generally the scatter is of the same order as that of data reported in [20] and [21] for the Wessex and Puma respectively. The nose-up "hump" in the pitch attitude curve at low speeds, caused by downwash impingement on the tailplane, is well in evidence and occurs in aircraft of similar configuration [20,21,22]. The dashed line represents data from [5] for a Sea King with the same c.g. position. While both curves are of similar form the present results are always higher than those of [5]. The reasons for this discrepancy are not known at present. The engine torque variation has also been included in Figure 4 and exhibits the classical minimum at mid-range airspeeds.

Figure 4d shows the variation with speed of the longitudinal cyclic stick position. The main point to be noted is that the undesirable non-linear characteristic with AFCS "OFF" is improved when the AFCS is "ON". Also a "hump" is apparent at lower forward speeds and is a result of the pitch moment caused by rotor downwash impingement on the tailplane previously described. The corresponding variation in cyclic blade angle B_{1S} with airspeed is illustrated in Figure 4e. Data are presented for AFCS "ON" and "OFF" and ideally should be coincident. This is generally borne out, except for B_{1S} at low speeds where the maximum gradient occurs. The differences are therefore attributed to difficulties in controlling the aircraft in this flight regime where considerable vibration occurs.

5.2 Dynamic Response Tests

Because of the large number of dynamic response tests carried out on the aircraft, only representative samples are included here. Both Figure 5 and 6 show the time histories of relevant variables when a transient input is applied to one of the flying controls.

Flight records have been processed for each manoeuvre to allow approximately one second of near-steady flight conditions before pilot inputs are introduced. Digital filtering and smoothing of control and push-pull movement has been avoided so as not to degrade the record of the input transient. Control inputs are, in fact, not always of the desired form, indicating the pilot's difficulty in prodcuing these even with the assistance of a jig in which to move the stick. Generally, the attempt to produce an uncoupled input by moving one control only is well achieved.

Results of the two tests will now be briefly commented on.

(a) Lateral cyclic pulse input, 80 kn AFCS "ON" (Figure 5) - The difference between the input pulse to the cyclic stick and the ensuing motion of the push-pull rod (i.e. the auxiliary servo output position) represents the AFCS contribution to the roll auxiliary servo. A small AFCS contribution to the yaw channel can also be seen, as might be expected of an attitude stabilizing system. The net result is that there is little change in the aircraft yaw attitude, whilst the roll attitude is quickly restored to its value prior to the pulse input. The rotor r.p.m. also increases during the initial rolling motion, suggesting that the rotor is being unloaded under these conditions. Subsequent rotor r.p.m. variation is related to the characteristics of the engine fuel control system.

(b) Collective stick, doublet input, 80 kn AFCS "OFF" (Figure 6) - The primary response of the helicopter is reflected by the immediate change in normal acceleration. Because of collective/yaw cross feed, the increased tail rotor thrust offsets the fuselage reaction to the main rotor torque increase so that the initial change in yaw rate is small. Subsequent damped oscillations in torque, rotor r.p.m., and yaw rate result from strong interactions. For example, yaw rate changes produced by torque variations are sensed as rotor speed changes, which in turn demand torque variation to maintain the governed rotor r.p.m.. Other tests, not included in the examples reported here, show that with AFCS "ON", the duration of these transients is greatly reduced by the presence of yaw stabilization.

5.3 Transition Manoeuvres

The time histories generated during a "transition down" manoeuvre [1] are illustrated in Figure 7. Nominal entry gate conditions for this manoeuvre are 90 kn and 200 ft (61 m), both speed and altitude being reduced until hover is achieved at the set height of 40 ft (12.2 m). This being an automatic manoeuvre, all flying control movements arise from AFCS action via the auxiliary servo units. Pitch and Roll cyclic stick movements occur through operation of the automatic beeper trim system, while collective stick and pedal movements occur through action of the open-loop spring system [23].

Other points to note are:

(a) Pitch attitude varies in the range 2 deg to 8 deg and is a consequence of the transition deceleration requirement, together with the inherent variations with airspeed (see Fig. 4).

(b) Periodic variations of approximately four seconds duration in the torque, rotor r.p.m. and roll attitude channels, particularly during the deceleration/descent phase. Examination suggests that θ_{C75}, the main rotor collective blade angle at 75% radius is being varied in response to the altitude control signals of the AFCS. These in turn produce torque changes and consequent coupling with rotor r.p.m. via the engine speed control/fuel management system.

5.4 Doppler and Cable Hover Tests

Results of a Doppler hover disturbance test are shown in Figure 8. Initially the aircraft is in a stable Doppler hover condition (0 to 10 s). It is then moved via pilot stick movements until a steady lateral velocity of about 12 kn is reached (10 to 21 s). At 21 s the AFCS is again switched to Doppler hover control and a smooth recovery to a stable hover is acheived by about 30 s.

Recovery from an aft disturbance when in cable hover mode is shown in Figure 9. It should be noted that for these results the longitudinal and lateral cable angles with respect to the aircraft are plotted. Initially the aircraft is in a steady cable hover (0 to 1 s) and is then moved backwards via pilot stick movements (1 to 8 s) and allowed to make its own recovery under cable hover control starting at about 8 s. It should also be noted that the cable is on the forward edge of the funnel when the cable hover mode is reselected. The recovery itself is fairly slow, taking until about 33 s, and exhibits overshoot in the longitudinal cable angle time history. Unfortunately a system malfunction in a sub-multiplexed channel which affected the Doppler and radio altitude channels, limits the usefulness of the data.

6. CONCLUDING REMARKS

A flight test program involving measurements of both the performance and dynamic response of a Sea King helicopter has been completed. The data bank, which is more comprehensive than hitherto available, will serve as a reference for validating a mathematical model of the Sea King helicopter, whether by graphical comparison or systems identification techniques.

Some level flight low speed trim data exhibit more scatter than at higher speeds, thus reflecting the difficulties associated both with airspeed measurement and holding the aircraft in a good steady state flight condition. Generally the trim flight curves are of similar form to those presented elsewhere for helicopters of similar configuration.

The dynamic response of the aircraft to rapid control movements has been recorded for various combinations of input waveform, airspeed, and autostabilizer mode. Some typical results are presented. Additional samples are shown of recordings made during operation in the ASW mode. These include transition manoeuvres and the effect of AFCS corrective actions when in the Doppler and cable hover modes.

The application of kinematic consistency checking to the data has resulted in the identification of inconsistent data, arising from the calibration variability of certain transducers or occasional instrument malfunctioning under stringent flight conditions. Possible improvements to the instrumentation package for future work have been outlined.

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TABLE 1

Summary of Performance Flight Tests

Test	Objective
Hover	1. To determine the relationship between power (torque) and altitude when the aircraft is hovering at various altitudes both in and out of ground effect (IGE and OGE). 11 samples. 2. To determine the relationships between power, blade angle, and rotor speed (N_r) when hovering both in and out of ground effect. 13 samples.
Level flight	To determine power, blade angle control settings, and pitch and roll attitude angles at hover and over a range of steady forward, backward, and sideways airspeeds covering the operating flight envelope. 32 samples.
Vertical flight	To determine power and blade angles over a range of steady vertical climb and decent rates. The autorotation case is included. 5 samples.
Forward flight climb	To determine power, blade angles and aircraft incidence angle at varying airspeed and climb rates. 7 samples.
Forward flight descent	To determine power, blade angles, and aircraft incidence during steady power-on descent at varying airspeeds. 6 samples.
Autorotation	To determine blade angles and aircraft incidence under power-off descent conditions at varying airspeed. 4 samples.
Steady heading sideslip	To establish power and blade angles for a range of sideslip angles at different airspeeds. 14 samples.
Spot turns	To determine power and tail rotor blade angle for a range of spot turn rates in both directions. 4 samples.
Banked turns	To determine power, blade angles, and control settings for a series of no-sideslip turns over a range of roll angles and airspeeds. 14 samples.
Sonar dunking	To determine cable angles and aircraft attitude angles in pitch and roll for a range of windspeeds and transducer depths when the aircraft is performing the sonar dunking manoeuvre. 3 samples.

TABLE 2

Summary of Stability and Control Flight Tests

Test	Objective
Dynamic response to flying control inputs	To determine aircraft dynamic flight behaviour at various airspeeds resulting from specified flying control inputs, with AFCS both ON and OFF. Step, pulse and doublet inputs were applied to the fore-aft cyclic, lateral cyclic and collective sticks, and the pedals. 139 samples.
Dynamic response to beeper trim switch inputs	To determine aircraft dynamic flight behaviour at various airspeeds resulting from specified beeper trim switch inputs. 12 samples.
Manoeuvring stability	To determine the aircraft dynamic flight behaviour and the fore-aft cyclic control position required to develop steady pitch rates or accelerations in wings-level pull-ups at various airspeeds. 8 samples.
Dynamic response in Doppler hover	To determine aircraft dynamic flight behaviour resulting from specified perturbations in Doppler hover. 8 samples.
Dynamic response in cable hover	To determine aircraft and cable dynamic behaviour resulting from specified perturbations in cable hover. 8 samples.
Sonar transducer (ball) raising and lowering	To determine aircraft and cable dynamic behaviour resulting from raising and lowering the ball. 7 samples.
Automatic transitions (demand up)	To determine aircraft dynamic flight behaviour during transition (DOWN and UP) manoeuvres. 8 samples.
Engine cuts	To determine aircraft dynamic flight behaviour following an engine cut at various airspeeds. 8 samples.
Take-off and landing	To determine pilot control inputs and aircraft dynamic response behaviour during typical take-off and landing manoeuvres. 5 samples.

TABLE 3

Quantities Measured by Data Acquisition System

Channel number	Quantity measured	Transducer type	Sampling frequency (Hz)	Analogue filter cut-off frequency (Hz)
1	Cyclic stick position - pitch	Linvar (a/c) (linear variometer)	60	12
2	Cyclic stick position - roll	Linvar (a/c)	60	12
3	Collective stick position	Position transducer	60	12
4	Angle of attack (vane)	Potentiometer	60	12
5	Fore-aft (pitch) push-pull rod position	LVDT (linear variable differential transformer	60	3†
6	Lateral (roll) push-pull rod position	LVDT	60	12
7	Collective push-pull rod position	LVDT	60	12
8	Pitch rate	Rate gyro	60	12
9	Sideslip angle (vane)	Potentiometer	60	12
10	Roll attitude	Attitude gyro (a/c)	60	12
11	Roll rate	Rate gyro	60	12
12	Longitudinal acceleration	Accelerometer	60	12
13	Lateral acceleration	Accelerometer	60	12
14	Normal acceleration	Accelerometer	60	12
15	Pitch attitude (Euler angle)	Attitude gyro (a/c)	30	6
16	Pedal position	LVDT	30	6
17	Yaw rate	Rate gyro (a/c)	30	6
18/1*	Yaw attitude (heading)	Gyro compass (a/c)	30	-
18/2	Torque - Engine 2	Synchro (a/c)		
19	Lateral cable angle	Linvar (a/c)	30	6
20	Longitudinal cable angle	Linvar (a/c)	30	6
21	Doppler longitudinal velocity	Doppler radar (a/c)	15	6
22	Doppler lateral velocity	Doppler radar (a/c)	15	6
23	Boom probe dynamic pressure	Pressure transducer	15	12
24	Radio altitude (raw)	Radio altimeter (a/c)	15	3
25	Radio altitude (smooth)	Radio altimeter (a/c)	15	3
26	Boom probe absolute pressure	Pressure transducer	15	3
27	Yaw push-pull rod position	LVDT	15	12†
28	Ambient temperature	Resistance thermometer	15	-
29	Torque - Engine 1	Synchro (a/c)	15	-
30	Rotor r.p.m.	Tachometer (a/c)	15	-
31/1‡	Sonar transducer depth	Potentiometer (a/c)	15	3
31/2	Towed probe dynamic pressure	Pressure transducer		
32/1‡	Cable length	Potentiometer (a/c)	15	3
32/2	Boom - towed differential pressure	Pressure transducer		
33	Clock time - in octal (five least-significant digits)		60	-

a/c - Denotes aircraft transducer.

● Switch selectable in flight.

† Filters inadvertently interchanged.

‡ Pre-flight changeover required.

Push-pull rod positions represent auxiliary servo displacements

TABLE 4

Additional Quantities Calculated

Channel number	Quantity calculated
34	Airspeed from boom probe at standard sea level conditions, corrected for position error
35	Altimeter setting, sea level value, QNH
36	Sea level temperature
37	Altitude from boom probe static pressure
38	True airspeed at aircraft altitude
39	Wind velocity
40	Direction from which wind is coming
41	Longitudinal cyclic blade pitch angle
42	Lateral cyclic blade pitch angle
43	Collective blade pitch angle
44	Tail rotor collective blade pitch angle
45	Collective blade pitch angle at 75% rotor radius position
46	Yaw Euler angle
47	Roll Euler angle
48	Yaw Euler angle derivative - by differentiation
49	Pitch Euler angle derivative - by differentiation
50	Roll Euler angle derivative - by differentiation
51	Roll rate - by differentiation
52	Pitch rate - by differentiation
53	Yaw rate - by differentiation
54	Longitudinal inertial acceleration
55	Lateral inertial acceleration
56	Normal inertial acceleration
57	Longitudinal velocity derivative
58	Lateral velocity derivative
59	Normal velocity derivative
60	Longitudinal velocity (in vehicle carried vertical axes) - by integration
61	Lateral velocity (in vehicle-carried vertical axes) - by integration
62	Vertical velocity - by integration
63	Horizontal velocity magnitude
64	Height - by integration
65	Yaw Euler angle derivative
66	Pitch Euler angle derivative
67	Roll Euler angle derivative
68	Yaw Euler angle - by integration
69	Pitch Euler angle - by integration
70	Roll Euler angle - by integration

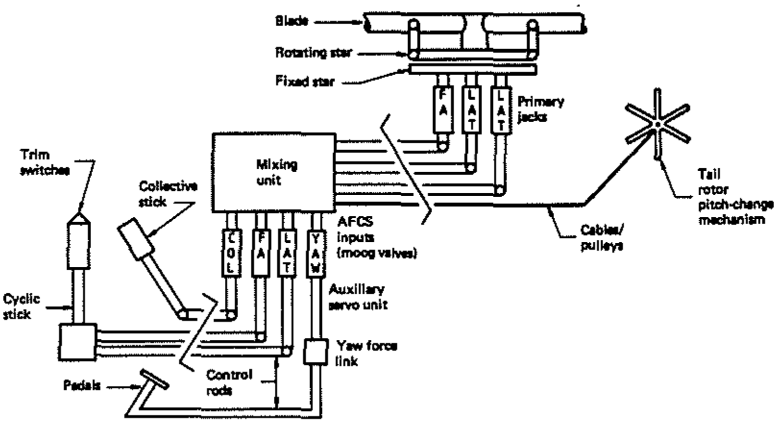

FIG. 1 AIRCRAFT FLYING CONTROLS

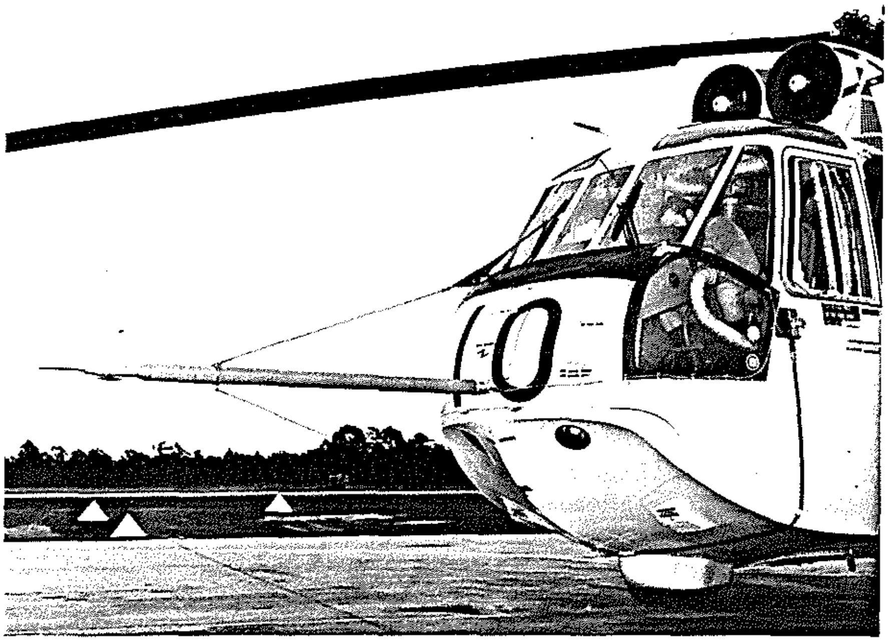

FIG. 2 INSTRUMENTATION BOOM

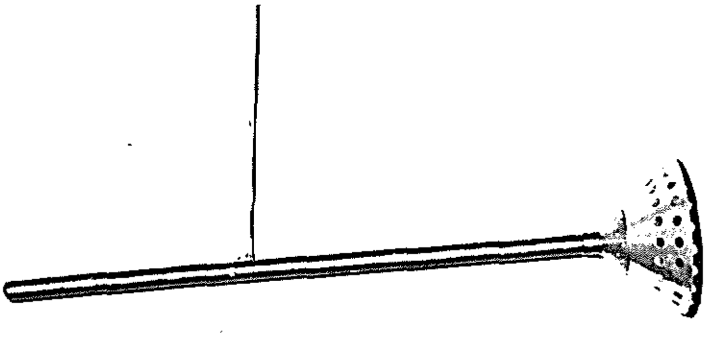

FIG. 3 PITOT – STATIC TRAILING PROBE

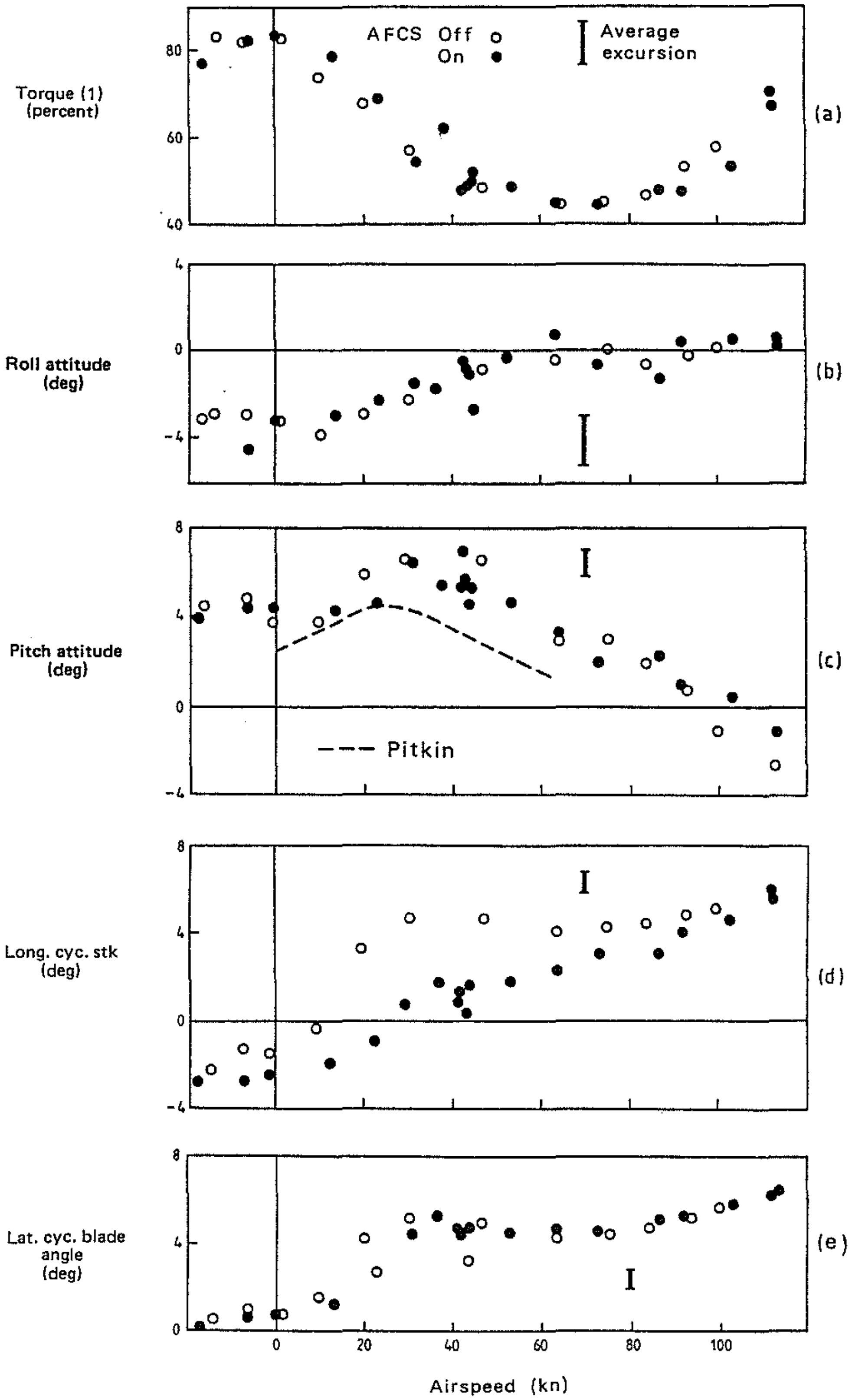

FIG. 4 DEPENDENCE OF SOME TRIM FLIGHT VARIABLES ON AIRSPEED

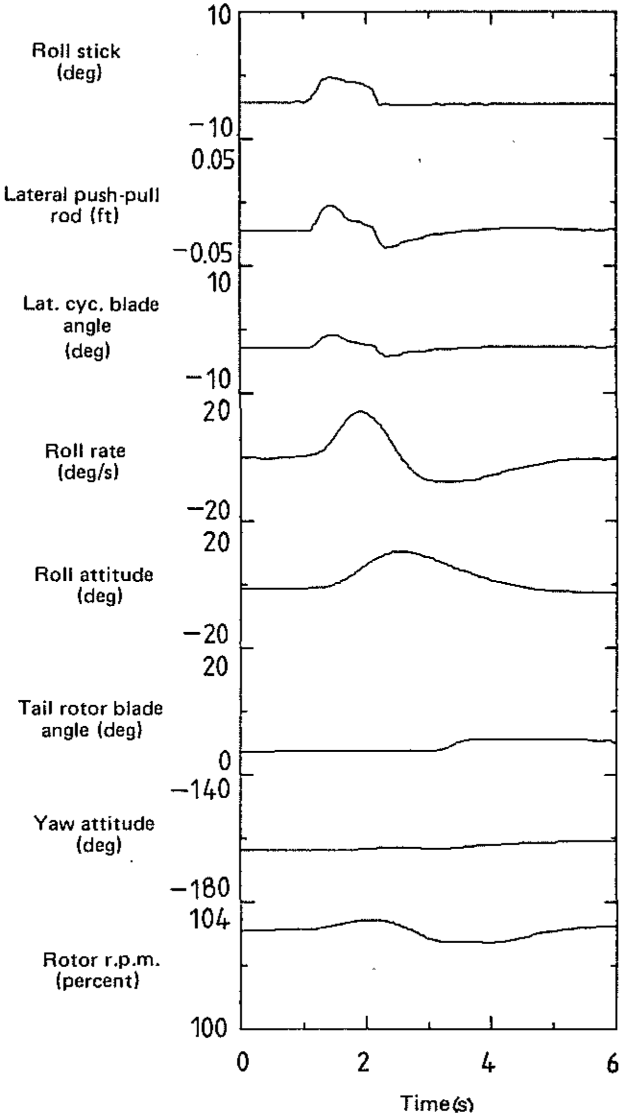

FIG. 5 LATERAL CYCLIC PULSE INPUT – 80 KN-AFCS ON

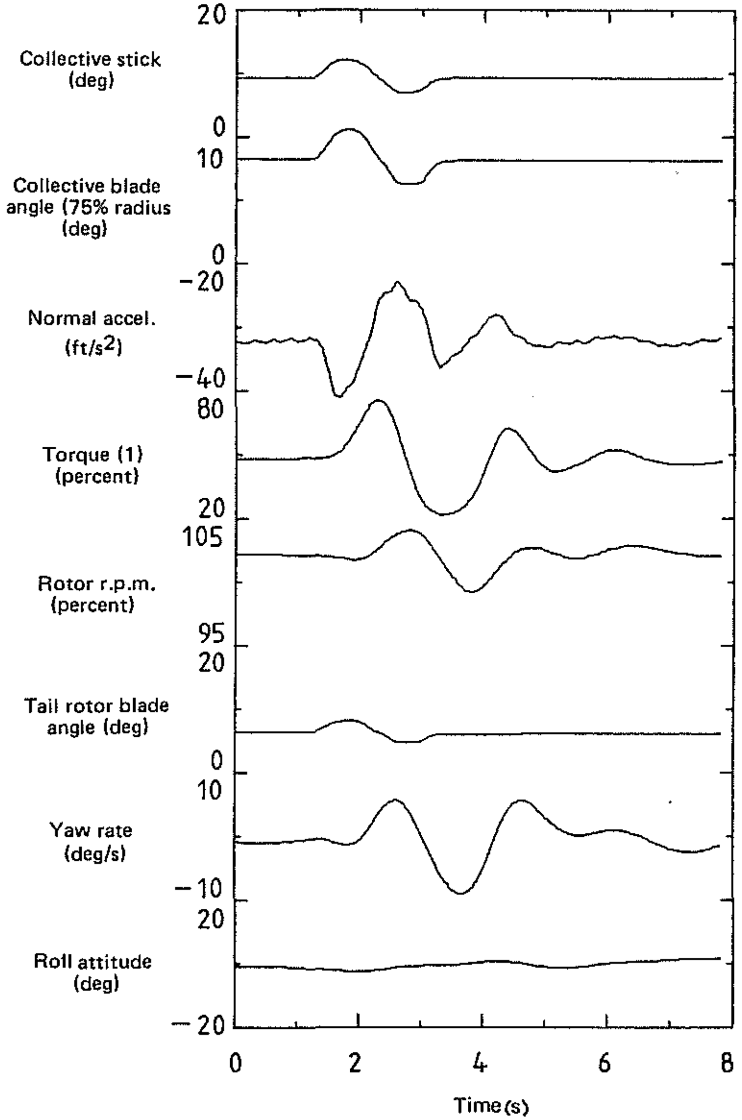

FIG. 6 COLLECTIVE DOUBLET INPUT – 80 KN-AFCS OFF

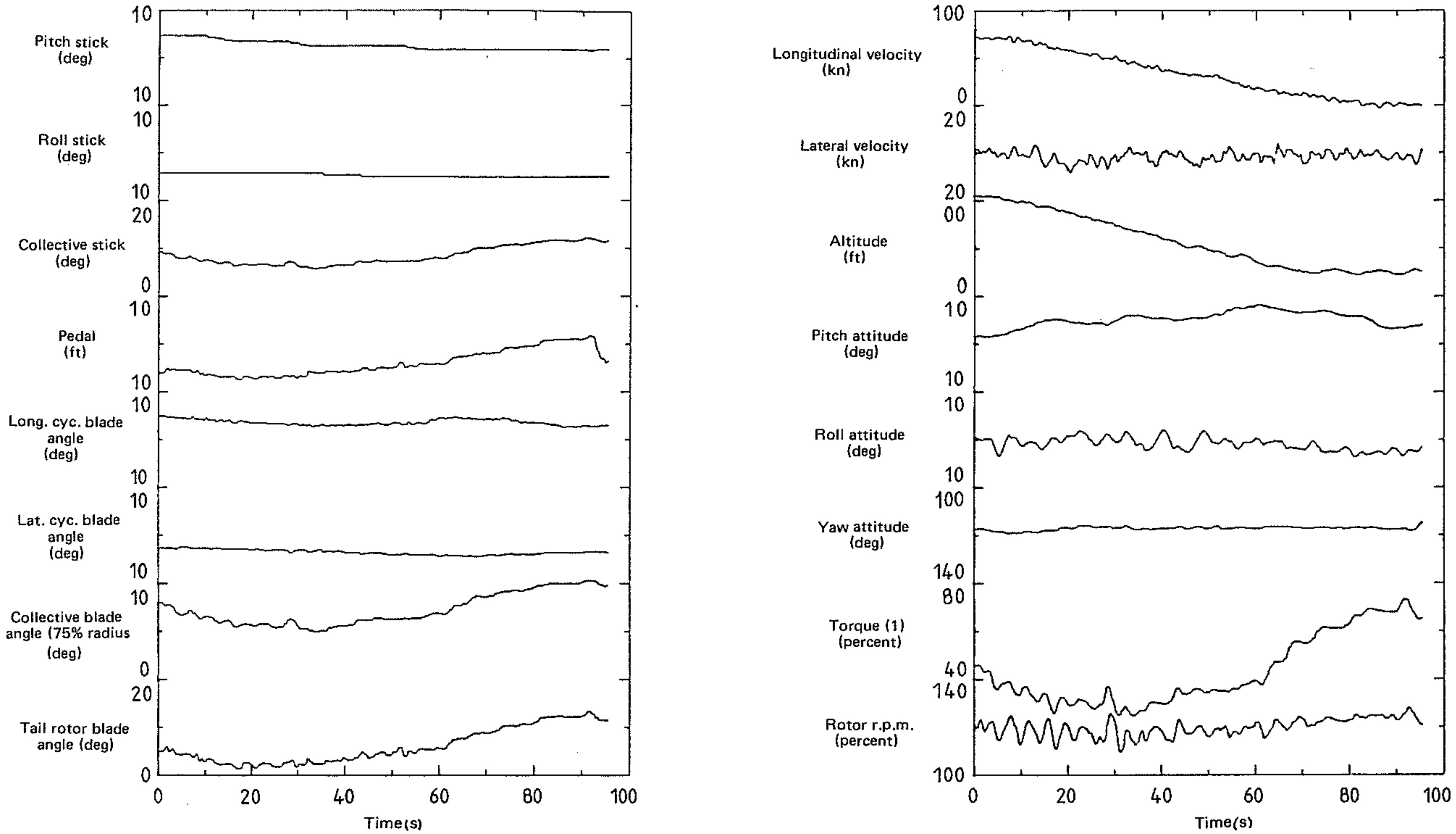

FIG. 7 TRANSITION DOWN TO HOVER AT 40 FT FROM 90 KN AT 200 FT

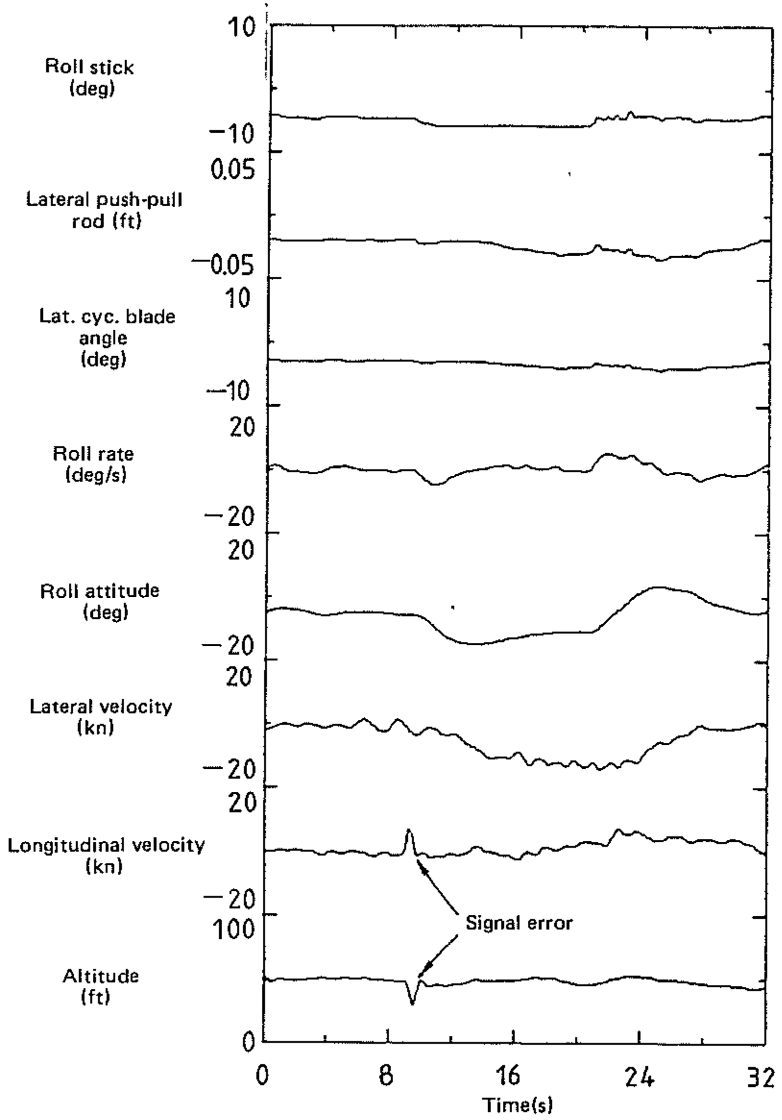

FIG. 8 RECOVERY FROM PORT DIRECTION DISTURBANCE IN DOPPLER HOVER

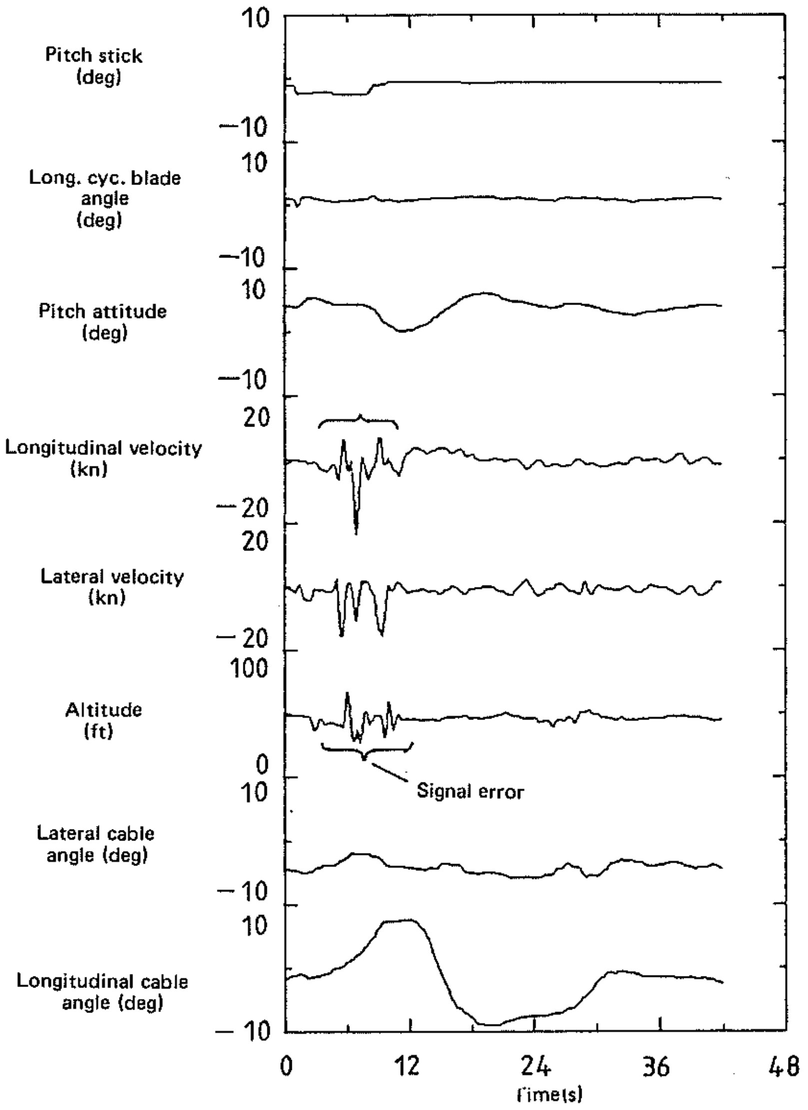

FIG. 9 RECOVERY FROM AFT DIRECTION DISTURBANCE IN CABLE HOVER