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TECHNICAL EVALUATION

BOEING-VERTOL MODEL 347 ADVANCED TECHNOLOGY HELICOPTER

PHASE I

FINAL REPORT

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JANUARY 1972

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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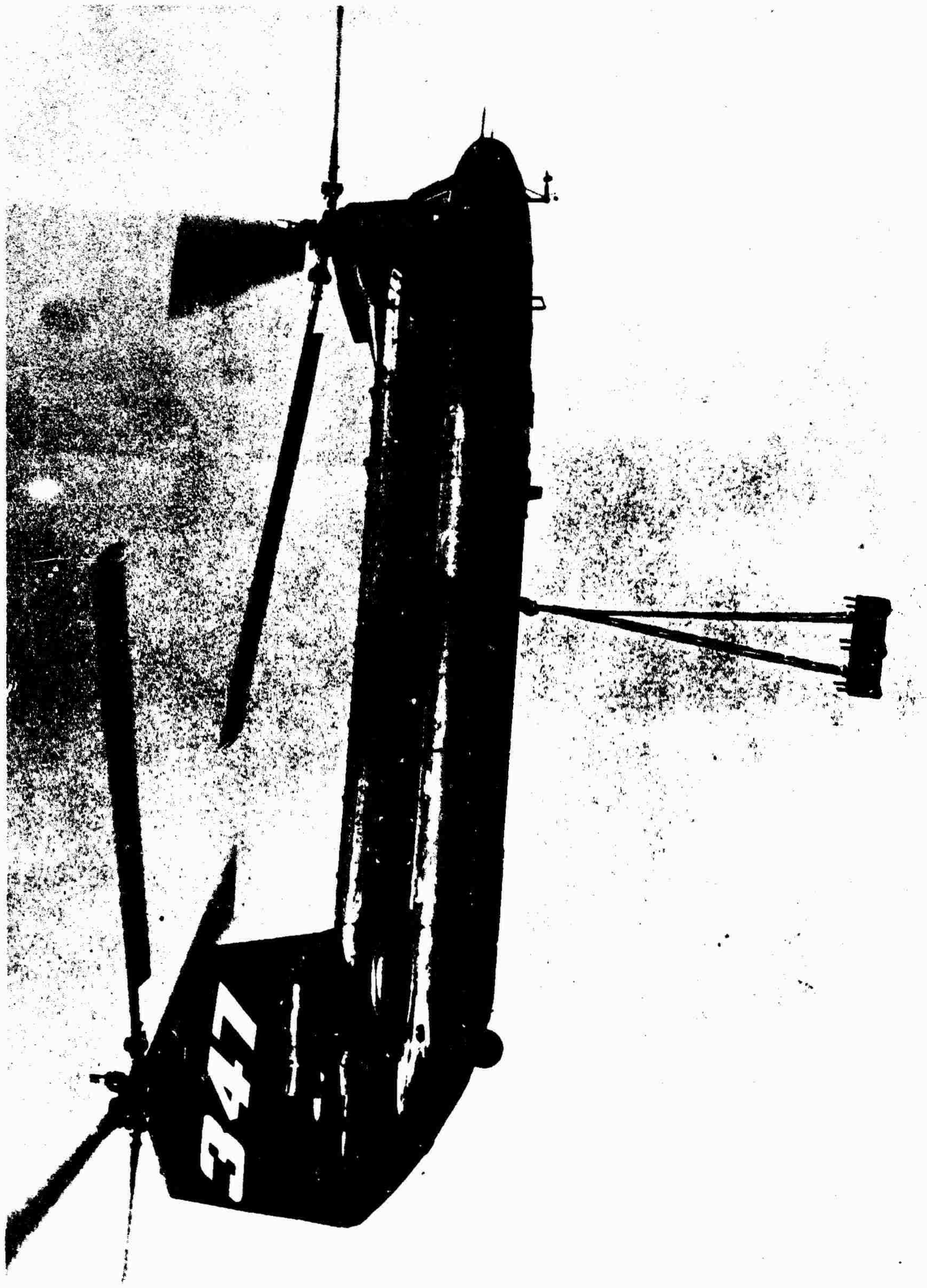
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ABSTRACT

The US Army Aviation Systems Test Activity (USAASTA) conducted the Phase I technical evaluation of the Boeing-Vertol Model 347 advanced technology research helicopter during the period 28 May to 19 June 1971. The Model 347, a derivative of the CH-47 transport helicopter, was tested at the contractor's facility near Philadelphia, Pennsylvania. Performance, handling qualities, vibration, and noise characteristics were evaluated to determine the improvements provided by incorporation of advanced technology systems in a large tandem-rotor transport helicopter. Compliance with the provisions of the contract statement of work and with military specifications, MIL-H-8501A and MIL-A-8806A, was determined. Several shortcomings identified during the testing were corrected by the contractor after the testing was completed. The effects of these corrections were evaluated during additional testing conducted on 11 and 12 August 1971. Level flight performance and out-of-ground-effect hover performance were significantly improved over that of the CH-47C helicopter. The excellent static longitudinal stability characteristics enhanced the mission capability of the aircraft. The steering and glide-path modes of the automatic flight path control system worked satisfactorily and reduced the pilot workload in instrument flight conditions. Cockpit noise and vibration characteristics were noticeably improved over those of the CH-47C. Correction of the faulty logic circuitry in the flight director steering command function was recommended to eliminate a hazardous flight condition. Correction of the following shortcomings was recommended to improve mission capabilities: downward slippage of thrust control rod, inadequate side-force characteristic in autorotation, excessively small turn needle and inclinometer, location of mode advisory lights, excessive long-term altitude error of altitude-hold system, oscillation of deviation indicator and flight director steering command bar, lack of heading synchronization indication on the pilot horizontal situation display, and vertical vibration level in aft cabin area. Five additional shortcomings of the cockpit displays and avionics systems were identified, and correction was recommended. The Model 347 helicopter failed to meet the requirements of six paragraphs of MIL-H-8501A and the cockpit vibration limits of the contract statement of work.

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INTRODUCTION

BACKGROUND

1. The Boeing-Vertol Model 347 helicopter is a derivative of the CH-47 helicopter currently used by the US Army. The Model 347 was developed to integrate and demonstrate advanced concepts in tandem-rotor helicopter technology. The intended purpose of these advanced concepts was to achieve improvements in handling qualities, vibration, noise, and performance. The research program consists of two phases. During Phase I, the basic airframe, rotor, and control system changes were incorporated, and testing was accomplished in the pure helicopter configuration. Phase II will consist of the addition of a high wing and modified rotor blades and subsequent testing to determine the effects of these changes.

2. The Model 347 Phase I program was conducted under contract with the US Army but was funded by The Boeing Company. Government participation in the program consisted of bailment to the contractor of a CH-47A helicopter, together with required modification kits, and conduct of flight testing at the contractor's facility. Authority for US Army Aviation Systems Test Activity (USAASTA) conduct of the Phase I flight test was provided by a test directive issued by the US Army Aviation Systems Command (AVSCOM) (ref 1, app I). The test plan for the conduct of the Phase I technical evaluation (ref 2) was prepared by USAASTA and approved by AVSCOM in May 1971 (ref 3).

TEST OBJECTIVE

3. The objective of the Phase I Army technical evaluation was to evaluate the improvements in handling qualities, vibration, noise, and performance provided by the incorporation of advanced technology systems in a tandem-rotor transport helicopter. Results of the evaluation (app II) were to be compared with test results obtained during previous USAASTA testing of the CH-47C helicopter.

DESCRIPTION

4. The Boeing-Vertol Model 347 helicopter flown during the Phase I Army technical evaluation was a modified CH-47A, serial number 65-7992, manufactured and modified by The Boeing Company, Vertol Division (Boeing-Vertol). Although the basic airframe was originally a CH-47A, Boeing production tab number B-164, the aircraft had been updated to the CH-47C configuration by incorporation of all significant engineering changes applicable to the current production CH-47C. The CH-47C helicopter is a twin-turbine engine, tandem-rotor aircraft designed to provide air transportation of internally loaded cargo and personnel and externally slung cargo. The Model 347 helicopter incorporates the following major changes to the CH-47C configuration:

a. Four-bladed rotors.

b. Fuselage lengthened by 110 inches.

c. Aft pylon height increased by 30 inches.

d. Retractable landing gear.

e. Left-hand cabin door incorporated.

f. Attachment structure for high wing provided.

g. Delta-three hinges on forward rotor.

h. Pitch and roll stability augmentation system (SAS) actuators relocated in the rotor pylons.

i. Redesigned control system which incorporated variable force feel and three-axis automatic stabilization systems.

j. Flight path control system which provided coupled steering and glide-path commands.

k. Acoustically treated cockpit.

l. Flight director and horizontal situation display on the pilot instrument panel.

m. Doppler navigator and map plotter.

n. Uprated T55-L-11 engines of 3,925 shaft horsepower (shp) each.

o. Additional vibration absorbers and structural detuning modifications.

5. Detailed descriptions of the test helicopter and installed systems are contained in references 4 through 10, appendix I. A general aircraft description, flight control system description, and photographs are contained in appendixes III, IV, and V, respectively. The design gross weight of the Model 347 is 45,000 pounds, and the alternate design gross weight is 54,500 pounds. Cockpit instrumentation was nonstandard, and aircraft loading was in accordance with the test plan (ref 2, app I) and the safety-of-flight release (ref 11).

SCOPE OF TEST

6. The Model 347 was evaluated as a research vehicle intended to demonstrate advanced concepts in tandem-rotor transport helicopter technology, as defined in

the statement of work contained in the contract agreement (ref 5, app I). The original evaluation of the helicopter required 25 hours of productive flight time accumulated in 31 test flights. As a result of the shortcomings identified during this evaluation, the contractor made several adjustments and modifications to the control system and requested that the effects of these corrections be evaluated (ref 12). Therefore, a reevaluation was performed, and the results were presented in a letter report (ref 13). These results are also incorporated in this report. This reevaluation required 4 hours of productive flight time accumulated in three test flights. Where appropriate, the nature and effects of the control system changes are described in the Results and Discussion section of this report. Data plots presented in appendix II which were accumulated during the reevaluation are identified by the legend "Reevaluation." All data not identified by this legend were accumulated during the original evaluation. Handling qualities, vibration, and noise were evaluated with respect to the requirements of the contract statement of work (ref 5), military specifications MIL-H-8501A (ref 14) and MIL-A-8806A (ref 15), and compared with previously determined characteristics of the CH-47C (refs 16, 17, and 18). Maneuvering characteristics were evaluated to provide a basis for comparison with future Phase II flight testing of the Model 347 with the wing installed. Limited out-of-ground-effect (OGE) hover and level flight performance data were obtained to permit evaluation of the effects of the four-bladed rotor system. An aircraft noise evaluation was performed by the US Army Aeromedical Research Laboratory (USAARL), and a brief summary of that evaluation is included in this report. The Model 347 was tested at the conditions shown in table 1.

7. Installation, calibration, and maintenance of the test instrumentation were performed by the contractor at the contractor's facility. Support and assistance in data reduction and analysis were provided by the contractor. The test aircraft was weighed by the contractor prior to the start of the test program. Empty weight of the helicopter with all test instrumentation installed was 30,215 pounds, and the center of gravity (cg) was at fuselage station (FS) 379.8 which is 6.2 inches forward of the midpoint between the rotor heads.

8. The flight restrictions and operating limitations applicable to this evaluation are detailed in the safety-of-flight release (ref 11, app I).

METHODS OF TEST

9. Standard test methods (refs 19 and 20, app I) were used to acquire and evaluate handling qualities and performance data. These test methods are briefly described in the Results and Discussion section of this report. A Handling Qualities Rating Scale (HQRS) was used to augment pilot comments relative to handling qualities (app VI). Vibration data were obtained concurrently with handling qualities testing. The vibration evaluation methods are described in the Results and Discussion section of this report. Noise-level data acquisition methods are described in reference 21, appendix I. Details of uncommon stability and control data reduction techniques utilized are described in appendix VII.

Table 1. Test Conditions.[1]

Test	Gross Weight (lb)	Longitudinal Center of Gravity[2] (in.)	Density Altitude (ft)	Temperature (°C)	Trim Calibrated Airspeed (kt)
Hover performance	34,790 and 50,180	22.6 fwd	330	18.1	Note[3]
Level flight performance	34,130 and 44,150	19.9 fwd and 8.4 aft	5,620 and 6,520	4.2 and 14.2	Note[4]
Controllability	42,900 to 45,670	7.6 aft to 10.8 aft	1,465 to 5,400	24.5 to 22.5	Zero to 131
Control trim positions	34,130 to 44,400	17.9 fwd to 8.8 aft	1,700 to 7,950	4.2 to 27.1	Note[5] Note[6] Note[7]
Static longitudinal stability	34,780 to 45,830	29.2 fwd to 10.9 aft	3,750 to 9,980	5.5 to 23.1	76 to 130
Dynamic longitudinal stability	44,150 to 45,290	24.0 fwd to 9.1 aft	1,820 to 10,540	10.0 to 27.7	Zero to 129
Static lateral-directional stability	33,010 to 45,140	18.0 fwd to 8.1 aft	4,590 to 7,330	12.7 to 20.0	73 to 130
Dynamic lateral-directional stability	44,150 to 45,290	24.0 fwd to 9.1 aft	1,820 to 10,540	10.0 to 27.7	Zero to 129
Maneuvering stability	43,000 and 45,670	8.0 aft	5,200	12.0 and 21.7	75 and 130
Single-engine failures	33,100 to 44,980	18.0 fwd to 9.0 aft	3,000 to 10,540	10.0 to 20.0	60 to 140
Noise characteristics[8]	45,000	9.0 aft	1,700 to 5,000	4.2 to 20.0	Zero to 145

[1]Doors, windows, and ramp closed. Rotor speed: 220 rpm. Extreme ranges of test parameters are shown. Only selected combinations of parameters within these ranges were tested.
[2]Center of gravity referenced to midpoint between rotors (FS 386).
[3]Out-of-ground-effect hover (150-foot aft wheel height). Rotor speed: 215 to 235 rpm.
[4]Referred knots true airspeed (KTAS): 52 to 172.
[5]Zero to 30 KTAS rearward, and 47 KTAS forward, OGE.
[6]Zero to 30 KTAS sideward, OGE.
[7]Thirty-nine to 154 KCAS in forward flight.
[8]Rotor speed: 220 and 230 rpm.

10. A detailed listing of the test instrumentation used in the Model 347 evaluation is contained in appendix VIII. Photographs 5 through 13, appendix V, show the cockpit and cabin instrumentation installed in the test aircraft.

CHRONOLOGY

11. The chronology of the Model 347 technical evaluation is as follows:

Test directive received	3	May	1971
Test aircraft received	28	May	1971
Test started	28	May	1971
Test completed	19	June	1971
Contractor debriefed	29	June	1971
Reevaluation started	10	August	1971
Reevaluation completed	11	August	1971
Reevaluation letter report submitted	1	October	1971

RESULTS AND DISCUSSION

GENERAL

12. Evaluations of performance, handling qualities, vibrations, and aircraft exterior and cockpit interior noise were conducted to determine improvements provided by incorporation of advanced technology systems in a tandem-rotor transport helicopter. Results of these evaluations were compared with the characteristics of the CH-47C helicopter, from which the Model 347 was derived. Level flight and OGE hover performance of the Model 347 were significantly improved over the CH-47C. At any given rotor shaft horsepower (rshp), the Model 347 could hover OGE at a heavier gross weight than the CH-47C and, at an equal gross weight, could cruise at a higher speed. The dualized differential airspeed-hold (DASH) system provided excellent longitudinal static stability characteristics which permitted sustained hands-off flight capability in any forward flight condition. The addition of a three-axis vernier trim (beep trim) capability, together with the roll and yaw attitude-hold functions of the SAS, enabled the pilot to quickly and easily obtain and maintain any desired trim attitude. The steering and instrument landing system (ILS) glide-path modes of the automatic flight path control system (FPCS) worked satisfactorily and reduced the pilot workload considerably during flight in instrument-flight-rule (IFR) conditions. The altitude-hold mode of the FPCS is, however, unsatisfactory for use in IFR conditions because of the excessive long-term altitude error permitted by the system. The cockpit four-per-revolution (4/rev) vibration levels were lower than the 3/rev levels of the CH-47C. The overall perceived vibration level in the cockpit in forward flight at 160 knots indicated airspeed (KIAS) was evaluated as being essentially the same as in the CH-47C at a 30-knot lower airspeed. The cockpit noise level was substantially reduced from that present in the CH-47C. This reduced noise level improved voice communications capabilities and reduced pilot fatigue. One deficiency was identified during the evaluation. Faulty logic circuitry in the flight director steering command function permitted false and misleading steering commands to be presented to the pilot. Correction of this deficiency is mandatory for acceptable aircraft operation under IFR conditions. Twelve shortcomings were identified during the evaluation. Excessive 4/rev vibrations in the rear portion of the cabin area at speeds above 130 knots true airspeed (KTAS) reduce the capability of the aircraft to perform passenger transport missions. Two handling qualities shortcomings, excessiv downward slippage of the thrust control rod at high collective settings anu inadequate side-force characteristic in autorotation, required moderate pilot compensation to achieve desired aircraft performance. The remaining shortcomings are all related to the pilot displays and avionics systems. Correction of all shortcomings is desirable for improved operation and mission capabilities.

PERFORMANCE

General

13. Limited level flight and OGE hover performance testing was accomplished to determine the effects of the configuration changes incorporated in the Model 347 helicopter, and to serve as a basis of comparison with future performance data to be acquired after installation of a wing and modified rotor blades on the test aircraft. At a constant rotor shaft horsepower, the Model 347 helicopter operating at a rotor speed of 220 rpm can hover OGE at a higher gross weight than can the CH-47C helicopter operating at 245 rpm. Using 6,000 rshp and at standard-day, sea-level conditions, the Model 347 can hover OGE at an 8-percent greater gross weight than the CH-47C. Level flight performance of the Model 347 is also improved. Using normal rated power (NRP) of the T55-L-11 engine at a 5,000-foot density altitude and 15°C, the Model 347 has a level flight airspeed 9 percent greater than the CH-47C helicopter.

Hover Performance

14. Out-of-ground-effect hover testing was accomplished at near sea-level conditions, at Millville, New Jersey, using a 150-foot tether line anchored to a concrete deadman, as shown in photograph A. A calibrated load cell was used to measure cable tension. Cockpit instrumentation permitted direct reading of cable tension and cable angle. The test was conducted by stabilizing the load cell reading, at predetermined engine torque values, up to the engine gas producer speed limit (N_1). This was done at a constant referred rotor speed (N/) of 219 rpm. In addition, data were also recorded at the high and low rotor speed limits (235 and 215 rpm) at the minimum and maximum aircraft gross weights of 34,790 and 50,180 pounds, respectively. The results of this test are presented in figure 1, appendix II. The results are also summarized and compared with the CH-47C helicopter test results (ref 16, app II) in figure A.

15. As shown in figure A, at any given constant rotor shaft horsepower, the Model 347 helicopter can hover OGE at a higher gross weight than is possible with the CH-47C helicopter. At sea-level, standard-day conditions, with 6,000 rshp, the Model 347 can hover OGE at 48,700 pounds compared to 45,100 pounds for the CH-47C. This is an increase of 3,600 pounds, or 8 percent, for the Model 347 helicopter.

Photograph A. Tethered Hover Test Rig.

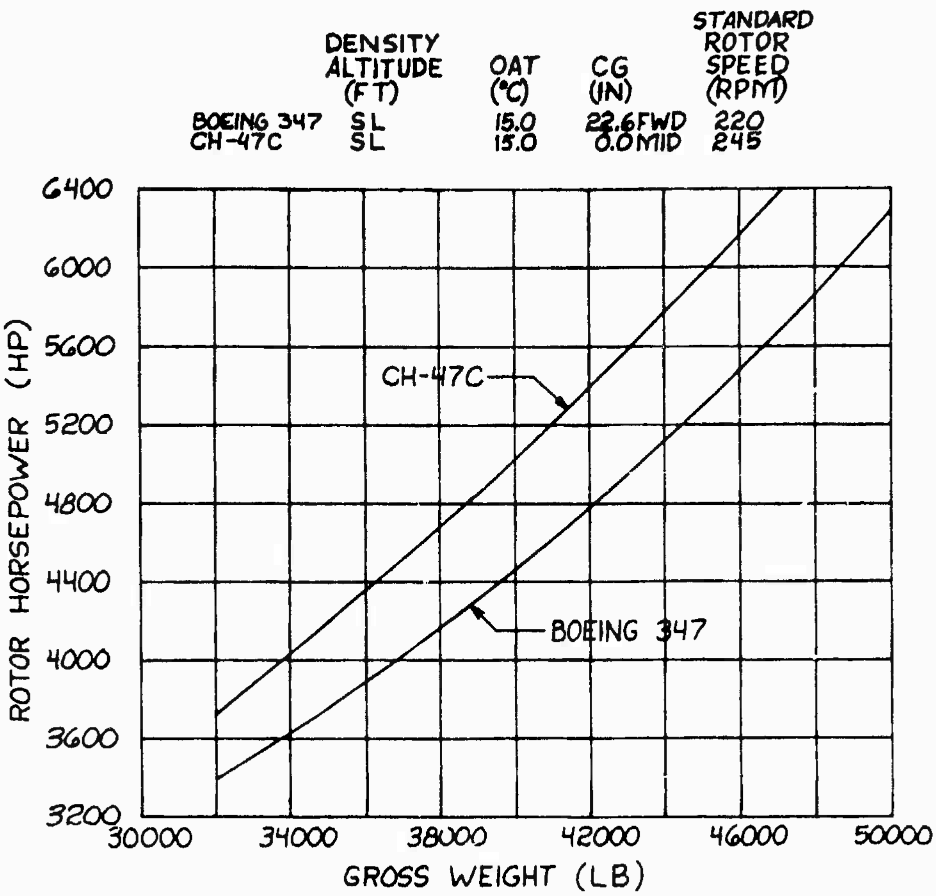


Figure A. Out-of-Ground-Effect Hover Performance Comparison.

Level Flight Performance

16. Level flight performance testing was conducted in two flights at average gross weights of 34,130 and 44,150 pounds. Data were obtained in stabilized level flight at approximately 10-knot speed increments from 40 KIAS to maximum level flight airspeed (V_H) while flying at the desired ratio of gross-weight/pressure-altitude (W/δ) and at a constant referred rotor speed of 220 rpm. The results of these tests are presented in figures 2 and 3, appendix II, in terms of generalized power required.

17. Compared with the CH-47C helicopter (ref 16, app I) at a rotor speed of 245 rpm, the Model 347 helicopter at 220 rpm required less power at any given weight and speed. Figure B presents a comparison of the level flight power required for the two aircraft at a 45,000-pound gross weight and a density altitude of 5,000 feet. At a constant 4,840 rshp, corresponding to NRP for the standard T55-L-11 engine at a 5,000-foot density altitude and 15°C, the level flight speed

of the Model 347 is 141 KTAS, an increase of 12 knots, or 9 percent, over the 129 KTAS of the CH-47C. This increase in NRP cruise speed would produce an approximate 9-percent increase in specific range for the Model 347 at an average mission gross weight of 45,000 pounds. Under the same level flight operating conditions at the speed for minimum power required (approximately 88 KTAS for both aircraft), the Model 347 required 3,550 rshp compared to 3,850 rshp required for the CH-47C. This is a decrease of 300 rshp, or 8 percent, for the Model 347 helicopter.

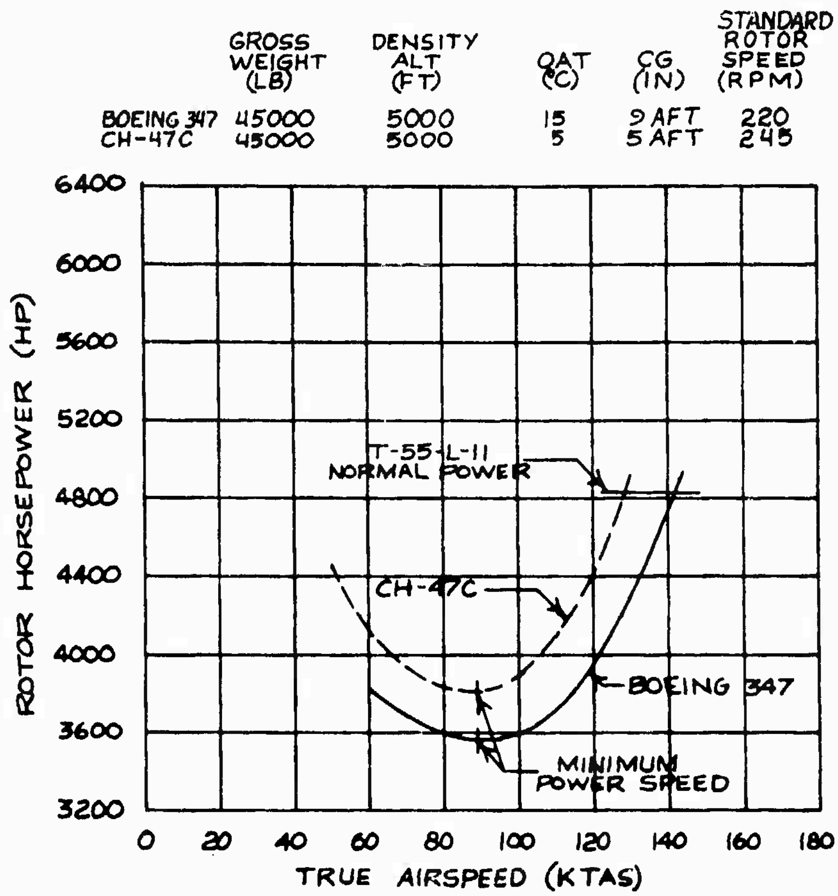


Figure B. Level Flight Performance Comparison.

STABILITY AND CONTROL

General

18. One of the objectives in the design of the Model 347 control system was to improve the handling qualities of the tandem-rotor transport helicopter. The

most significant improvement achieved, relative to the CH-47C, was the incorporation of a DASH system which provided excellent longitudinal static stability characteristics. The DASH system, which sensed both airspeed and pitch attitude, provided very powerful and consistent airspeed stabilization so that sustained hands-off flight could be accomplished, with very little airspeed variation, in any forward-flight condition. This excellent stability was achieved while retaining a degree of maneuverability and controllability entirely adequate for the transport helicopter mission. The addition of beep trim actuators to all three control axes aided the pilot in quickly and easily trimming the aircraft. The roll and yaw attitude-retention capability of the SAS contributed to the ease with which the pilot could maintain the trim conditions for extended periods. Two handling qualities shortcomings were identified during the evaluation. Downward slippage of the thrust control rod at high collective settings and inadequate side force in autorotation both required moderate pilot compensation to achieve desired aircraft performance. While the SAS-OFF and DASH-OFF longitudinal and directional static stability characteristics were improved slightly compared to the characteristics exhibited by the CH-47C, SAS-OFF flight is not recommended in IFR conditions because of the considerable pilot workload required. The handling qualities of the Model 347 helicopter are acceptable for Army use throughout the allowable flight envelope.

Control System Characteristics

19. The mechanical characteristics of the control system were evaluated on the ground with the rotors and engines stopped. Hydraulic and electrical power were provided by external sources. Control forces were measured by use of a hand-held force gage applied at the center of the cyclic control grip, thrust rod (collective control) grip, and directional pedals. Since the variable force-feel system produced increased cyclic control forces with increased airspeed, these forces were measured at zero airspeed and also with forward flight airspeed signals applied to the force-feel systems. In addition, a pitch rate signal was applied to the longitudinal system in order to measure the force contribution due to pitch rate. All switches and systems were set to duplicate normal inflight conditions. Control system characteristics in flight were essentially the same as those observed under the above described static test conditions.

20. The longitudinal control force characteristics are presented in figures 4, 5, and 6, appendix II, and are summarized in table 2. The average force gradient at zero pitch rate varied from 0.8 pound per inch (lb/in.) at zero airspeed to 1.6 lb/in. at 170 KIAS, an increase of 100 percent over the airspeed range tested. The gradient was further increased by 0.15 pound-per-inch/degree-per-second due to the pitch rate contribution of the variable force-feel system. The variation of force gradient with airspeed and pitch rate was smooth and free of any transients over the range tested. The longitudinal control characteristics met the requirements of MIL-H-8501A and are satisfactory for the transport mission.

Table 2. Longitudinal Control System Characteristics.[1]

Control Characteristics	Test Result
Free play	Less than 1/8 in.
Trim control displacement band	0.1 in.
Breakout, including friction	Hover ~ ±1.0 lb 170 knots ~ ±1.5 lb
Average friction band	0.8 lb
Average force gradient	Hover ~ 0.8 lb/in. 170 knots ~ 1.6 lb/in.
Force gradient due to pitch rate	$\frac{0.15 \text{ lb/in.}}{\text{deg/sec}}$

[1]Ground test data. Systems energized by external electrical and hydraulic power sources. Rotors stationary. Thrust control rod in detent.

21. The lateral control force characteristics determined during the original evaluation are presented in figures 7, 8, and 9, appendix II, and are summarized in table 3. The lateral force gradient increased from 0.8 lb/in. at zero airspeed to 1.1 lb/in. at 170 KIAS, an increase of approximately 40 percent. The lateral trim control displacement band, that range of control position in which no force was required to hold any control position displacement from trim, was 0.9 inch at zero airspeed and decreased to 0.2 inch at 170 KIAS. This large lateral trim control displacement band at hover and low speed increased the pilot workload required to accurately center the lateral control and to place the control in detent. With the lateral control out of detent, the roll-hold mode of the SAS was inoperative, and the pilot was required to continually apply corrective inputs to the lateral control in order to maintain a desired bank attitude (HQRS 4). The term "breakout force," as used hereafter, refers to the total breakout and friction force required to displace the cockpit control from the trim position. The lateral control originally lacked any measurable breakout force at speeds below 100 KIAS and failed to meet the 0.5-pound minimum breakout force requirement of paragraph 3.3.13 of MIL-H-8501A. The absence of breakout forces at hover and low speed made it moderately difficult to accurately center the lateral control during hover and slow-speed forward flight (HQRS 4).

22. Following the original evaluation, the contractor modified the lateral control system to correct the shortcomings identified by USAASTA. These corrections were

(1) increased breakout including friction, (2) widened roll-hold detent switch setting, and (3) reduced hysteresis of force-feel actuator (ref 12, app I). The effects of these changes were determined during a subsequent reevaluation of the Model 347 helicopter. The results of this reevaluation are presented in figures 10, 11, and 12, appendix II, and are summarized in table 3. The lateral trim control displacement band was reduced to 0.1 inch under all flight conditions, and the centering characteristics were greatly improved. The lateral force gradient varied from 0.9 lb/in. at zero airspeed to 1.3 lb/in. at 170 KIAS, an increase of approximately 40 percent over the airspeed range tested. The lateral control system characteristics, as determined during the reevaluation, met the requirements of MIL-H-8501A. Although the relatively high lateral force gradients were mildly unpleasant and not well harmonized with the longitudinal force gradients, only minimal pilot compensation was required to satisfactorily control the aircraft (HQRS 3). The modified lateral control characteristics are satisfactory for the transport mission.

Table 3. Lateral Control System Characteristics.[1]

Control Characteristic	Original Evaluation	Reevaluation[2]
Free play	Less than 1/8 in.	Less than 1/8 in.
Trim control displacement band	Hover ~ 0.9 in. 170 KIAS ~ 0.2 in.	Hover ~ 0.1 in. 170 KIAS ~ 0.1 in.
Breakout, including friction	Hover ~ none 170 KIAS ~ ±0.75 lb	Hover ~ ±1.0 lb 170 KIAS ~ 1.1 lb
Average friction band	Hover ~ 1.7 lb 170 KIAS ~ 1.0 lb	Hover ~ 1.5 lb 170 KIAS ~ 0.8 lb
Average force gradient	Hover ~ 0.8 lb/in. 170 KIAS ~ 1.1 lb/in.	Hover ~ 0.9 lb/in. 170 KIAS ~ 1.3 lb/in.

[1]Ground test data. Systems energized by external electrical and hydraulic power sources. Rotors stationary. Thrust control rod in detent.

[2]Lateral control system modified and adjusted prior to reevaluation. Following changes incorporated in lateral control system: increased breakout, widened roll hold detent switch, and reduced hysteresis of actuator.

23. During the original evaluation, the directional control trim mechanism slipped when the pedals were displaced more than 1 inch from trim. When this slippage occurred, the pedals would not return to the original trim position after all forces had been removed. If this slippage from trim occurred while maneuvering during night or IFR operations, recognition of the trim slippage and retrimming would require considerable pilot compensation (HQRS 5).

24. To correct the directional trim slippage, the contractor added a brake to the directional control trim motor to prevent slippage of the motor when the directional pedals were displaced from the trim position. The directional control characteristics were subsequently reevaluated during ground operations and in flight. The results of this reevaluation are presented in figure 13, appendix II, and are summarized in table 4. The directional pedal breakout forces of 10.0 pounds to the left and 14.0 pounds to the right exceeded by 3 pounds (42 percent) and 7 pounds (100 percent), respectively, the 7-pound limit permitted by paragraph 3.3.13 of MIL-H-8501A. The highest pedal forces encountered in flight occurred while making hovering turns. In this case, pedal displacements of up to 1.5 inches were used. This displacement required an estimated pedal force of approximately 20 pounds and exceeded by 5 pounds (33 percent) the 15-pound limit force permitted by paragraphs 3.3.11 and 3.3.12 of MIL-H-8501A. These directional control forces were not objectionable in flight and are satisfactory for the transport helicopter mission. No slippage from trim was observed during the reevaluation. The modified directional control system met the requirements of paragraphs 3.3.14, 3.5.10, and 3.5.11.1 of MIL-H-8501A and is satisfactory for the transport mission.

25. Thrust rod control force characteristics are shown in figure 14, appendix II. The limit forces for full-up and full-down thrust rod displacement, with the magnetic brake released, were 10 pounds and 8 pounds, respectively. Although these forces exceeded the 7-pound limit specified by paragraph 3.4.2 of MIL-H-8501A, they were satisfactory for the transport mission. The breakout forces within the center 50 percent of thrust rod travel were 2 to 4 pounds. With the magnetic brake engaged, the forces were 13 to 16 pounds in the center of the control travel. After applying an upward displacement beyond the midpoint of travel, the rod slipped downward approximately 0.2 inch after releasing the magnetic brake button and removing all forces from the handle. This slippage characteristic failed to meet the requirements of paragraph 3.4.2 of MIL-H-8501A. In flight, the slippage resulted in a loss of up to approximately 2 percent of engine torque and adversely affected the pilot's capability to make accurate torque changes, particularly under conditions where the pilot was operating at some engine or transmission limit (HQRS 4). Thrust control rod sensitivity was qualitatively evaluated as being reduced from that observed in the CH-47C. In addition, thrust control rod motion was pleasantly smooth, and the forces were slightly lower than the forces present in the CH-47C. The smoothness, reduced sensitivity, and reduced forces are improvements which reduce the pilot effort required in making collective changes.

26. After the original evaluation, the contractor modified the thrust control rod in an attempt to eliminate the downward slippage of the thrust rod and consequent

engine torque loss (ref 12, app I). During the reevaluation testing while hovering OGE at 45,000 pounds, the downward slippage of the thrust rod reoccurred and consistently produced a 2-percent loss in engine torque. This characteristic again violated the requirement of paragraph 3.4.2 of MIL-H-8501A, required moderate pilot compensation in making small torque adjustments, and was particularly annoying at high engine-torque settings (HQRS 4). Correction of this shortcoming is desirable for improved helicopter operation.

Table 4. Directional Control System Characteristics.[1]

Control Characteristic	Original Evaluation[2]	Reevaluation[3]
Free play	Less than 1/8 in.	Less than 1/8 in.
Trim control displacement band	0.1 in.	0.1 in.
Breakout, including friction	Left ~ 10.0 lb Right ~ 14.0 lb	Left ~ 10.0 lb Right ~ 14.0 lb
Average friction band	Could not be determined	Left ~ 3.0 lb Right ~ 5.0 lb
Average force gradient	Could not be determined	5.0 lb/in.

[1]Ground test data. Systems energized by external electrical and hydraulic power sources. Rotors stationary. Thrust control rod in detent.

[2]Characteristics not fully evaluated due to excessive slippage of force trim at any pedal displacement exceeding 1 inch from trim. Unbraked trim motor.

[3]Unbraked directional trim motor replaced with braked motor prior to reevaluation.

Longitudinal Cyclic Speed Trim

27. Operation of the longitudinal cyclic speed trim (LCST) was evaluated in stabilized level flight conditions. The position of the forward and aft head LCST actuators was recorded throughout the allowable airspeed envelope. The results of this test are presented in figure 15, appendix II. The design LCST schedule established by the contractor is also shown on this figure. Operation of the LCST was smooth and unobtrusive and did not cause any noticeable dynamic responses in the aircraft. Except for the difference in markings, the two LCST position gages

mounted on the instrument panel were similar to the gages used in the CH-47C and were easily observed from the crew stations. Selection of the desired mode of operation – automatic, manual, or taxi – was easily accomplished by use of a three-position toggle switch located on the center console.

Controllability

28. Controllability characteristics with all SAS and DASH systems operating were evaluated at a heavy weight, aft cg loading in hover and in forward flight. Single-axis control step inputs were applied to the longitudinal, lateral, and directional controls using mechanical fixtures to obtain the desired control input size. The step inputs were held steady while recording the subsequent aircraft angular rate (control response), and angular displacement (control power). The aircraft maximum angular acceleration (control sensitivity) was mathematically derived from the angular rate data. Three step inputs of increasing displacement in each direction were applied to each axis to establish controllability trends. The results of these tests are presented in figures 16 through 22, appendix II. The control power characteristics during OGE hover are summarized in table 5. Also shown in this table are the control power requirements of MIL-H-8501A.

Table 5. Out-of-Ground-Effect Hover Control Power.[1]

Axis	MIL-H-8501A Paragraph	Attitude[2] (deg)	
		MIL-H-8501A Minimum Requirement	Test Result
Longitudinal (deg in 1 sec)	3.2.13 3.6.1.1	VFR: 1.25 IFR: 2.04	Down: 2.1 Up: 2.5
Lateral (deg in 1/2 sec)	3.3.18 3.6.1.1	VFR: 0.75 IFR: 0.89	Left: 2.5 Right: 2.5
Directional (deg in 1 sec)	3.3.5 3.6.1.1	VFR: 3.06 IFR: 3.06	Left: 6.0 Right: 5.2

[1]Average gross weight: 45,500 pounds.
[2]Attitude change produced by 1-inch control input.

29. Longitudinal controllability characteristics are presented in figures 16 and 17, appendix II. Longitudinal control sensitivity (maximum angular acceleration) varied from a minimum of 5 deg/sec^2 per inch of control displacement in hover to a maximum of 13 deg/sec^2 per inch of control motion in forward flight at 131 KCAS. In hover, this was approximately one-half the sensitivity reported for

the CH-47C in APE II (ref 17, app I). Longitudinal control response (maximum angular rate) varied from 4 to 7 deg/sec per inch of control travel in hover and forward flight, respectively. The control response during hover was approximately two-thirds the angular rate reported for the CH-47C. Control power (angular displacement in 1 second) ranged from 2 to 4 degrees per inch of control travel at hover and forward flight, respectively. This was approximately one-half the control power reported for the CH-47C. The longitudinal controllability characteristics of the Model 347 permitted smooth, precise control of aircraft pitch attitude and airspeed, particularly during hover and approach to landing (HQRS 2).

30. Although longitudinal control sensitivity essentially doubled from hover to 131 KCAS, this variation with airspeed was compatible with the characteristics of the variable force-feel system and did not adversely affect controllability of the aircraft. The Model 347 helicopter met the requirements of paragraphs 3.2.1, 3.2.2, 3.2.5, 3.2.6, 3.2.9, 3.2.12, and 3.2.13 of MIL-H-8501A. The longitudinal controllability characteristics of the Model 347 helicopter are satisfactory for the transport mission.

31. Lateral controllability characteristics are presented in figures 18 and 19, appendix II. The average lateral sensitivity was 23 deg/sec^2 per inch of travel in hover and at the forward-flight airspeeds tested. The lateral control response varied from 9 to 16 deg/sec per inch of control travel, and the average displacement at 1/2 second (control power) was 2.5 degrees per inch of control travel. These lateral controllability characteristics were essentially similar to the characteristics of the CH-47C, as reported in APE III (ref 18, app I). Lateral controllability was qualitatively evaluated throughout the flight envelope, and no objectionable characteristics were observed. The lateral controllability characteristics met the requirements of paragraphs 3.3.4, 3.3.15, 3.3.16, and 3.3.18 of MIL-H-8501A. The lateral controllability characteristics of the Model 347 helicopter are satisfactory for the transport mission.

32. Results of the directional controllability tests are presented in figures 20 and 21, appendix II. Directional control sensitivity varied from an average of 16 deg/sec^2 per inch of control travel during hover to 11 deg/sec^2 per inch of travel during forward flight. Directional control sensitivity of the Model 347 in hover was about 50 percent greater than the control sensitivity of the CH-47C, as reported in APE II (ref 17, app I). Directional control response varied from an average of 14 deg/sec per inch of travel during hover to 9 deg/sec per inch of travel in forward flight. The directional control response of the Model 347 during hover was approximately 100 percent greater than in the CH-47C. Directional control power of the Model 347 varied from an average of 5.6 degrees in 1 second per inch of control travel during hover to an average of 3.3 degrees in forward flight. In hover, this control power was approximately twice that exhibited by the CH-47C.

33. During the original evaluation, directional pedal inputs were accompanied by excessive initial transient roll opposite to the direction of yaw. This phenomenon, which appeared to have been caused by the difference in height between the forward

and aft rotors and is usually identified as "roll due to directional control," is hereafter referred to as adverse roll. This characteristic is shown in the time history of aircraft response presented in figure 22, appendix II. In this example, a 1/3-inch left pedal displacement initially produced a 3-deg/sec right roll rate and resulted in a 2-degree right roll displacement 1 second after the control input. Roll acceleration, roll rate, and displacements were subsequently developed in the opposite, correct (left) direction. Although this adverse roll occurred with all directional control inputs under all flight conditions, it was particularly objectionable when stopping a pedal turn in hovering flight. In this case, it was difficult to stop a turn on a predetermined target heading, in that removing a pedal input to stop the turn introduced a large roll response. Typically, this unwanted roll response caused the pilot to miss the intended directional heading by 5 degrees and translate sideways for several feet (HQRS 4). Following this original evaluation, the contractor modified the lateral control to improve the operation of the roll-hold detent switch (para 22). Directional controllability characteristics were again examined during the reevaluation of the Model 347. During this reevaluation, directional control inputs of up to 1.5 inches in hover and up to 1.0 inch in forward flight produced approximately one-half of the transient adverse roll response observed during the original evaluation. This reduced value of adverse roll was not noticeable during normal operations, and hovering turns were easily accomplished (HQRS 2). The directional controllability characteristics of the Model 347 helicopter met the requirements of paragraphs 3.3.5, 3.3.6, 3.3.7, and 3.3.16 of MIL-H-8501A. The directional controllability characteristics of the Model 347 helicopter are satisfactory for the transport mission.

Trimmability

34. Within the normal operating envelope, all control forces could be trimmed to zero by use of the magnetic brake switch or the beep trim switches. Although the variable force-feel system permitted trimming of the longitudinal, lateral, and directional controls only within the center two-thirds of the full control travel, this trim range was adequate for all normal steady-state flight conditions and met the trim requirements of paragraphs 3.3.3 and 3.3.10 of MIL-H-8501A.

35. During the original evaluation, continuous operation of the lateral beep trim in forward flight produced long-term bank-attitude changes of approximately 3 degrees for each second of beep operation. Momentary operation of the lateral beep trim produced excessive initial roll acceleration and roll rate, and a transient bank-angle displacement which was subsequently washed out. A time history of a 1/4-second right lateral beep input is presented in figure 23, appendix II. In this case, the 1/4-second beep produced a transient right roll displacement of 2 degrees approximately 1 second after the start of the beep input. Five seconds after applying the lateral beep, the roll attitude returned to and stabilized at a 1/2-degree right displacement. The high initial roll acceleration and roll rate accompanying the lateral beep operation, together with the overshoot of the long-term attitude displacement, adversely affected the pilot's ability to make small

lateral trim changes. Typically, the pilot was required to apply numerous repeated momentary lateral beep inputs in order to obtain the desired attitude change (HQRS 5).

36. Prior to the reevaluation of the aircraft, the contractor modified the lateral beep circuitry to reduce the initial roll response and to reduce the long-term trim rate. Figure 24, appendix II, is an example of the aircraft response to a 1/4-second lateral beep obtained during the reevaluation tests. Although the roll acceleration and roll rate were oscillatory, as in the original evaluation, the oscillations were not objectionable. The roll attitude response was essentially deadbeat to the new long-term attitude with no perceptible overshoot. Operation of the lateral beep trim was very natural, and small trim changes could be easily accomplished with a single lateral beep input (HQRS 2). Continuous operation of the lateral beep trim produced a bank-attitude change of 1.6 degrees for each second of beep operation at all forward flight speeds. The lateral trimmability characteristics observed during the reevaluation test are satisfactory for the transport mission.

37. Continuous longitudinal beep trim operation resulted in 1/8 inch of stick movement per second, equivalent to a 6-knot change in airspeed at 130 KIAS. Operation of the longitudinal beep trim was very pleasant and aided the pilot in making minor airspeed changes in any flight condition (HQRS 2).

38. Continuous operation of the directional control beep trim moved the pedals at a rate of 1/4 inch for each second of operation, yawing the aircraft at a rate of 4 degrees for each second of operation at 130 KIAS. The directional beep trim was very useful in making small heading changes to obtain coordinated flight, especially under IFR conditions (HQRS 2). During the original evaluation, when retrimming with the magnetic brake, there was a noticeable delay of approximately 1/3 second between the time the magnetic brake button was depressed and the release of the force trim. After completion of the original evaluation, the contractor decreased the eddy current damping in the directional control. During the subsequent reevaluation of the helicopter, no delay in the release of the force trim was observed. The directional controls exhibited a noticeable recentering shock when releasing the magnetic brake while holding a pedal force. This characteristic required only minimal pilot compensation to overcome and did not adversely affect operation (HQRS 3). The directional trimmability characteristics of the Model 347 helicopter are satisfactory.

39. The change in longitudinal trim position when transitioning from climb at normal rated power to autorotation at 80 KIAS was 0.2 inch, aft. This trim change was 9 percent of the 2.25-inch change required in the CH-47C, as reported in reference 17, appendix I. The required lateral trim change under the same conditions was 0.5 inch, left. The longitudinal and lateral trim changes with power were satisfactory under all conditions and met the requirements of paragraphs 3.2.10.2 and 3.3.17 of MIL-H-8501A.

flight. From 25 to 30 KTAS in rearward flight, the gradient was neutral. The total longitudinal control travel over the 77-knot airspeed range was 0.9 inch, and the minimum remaining control margin was 44 percent at 47 KTAS in forward flight. The longitudinal control trim characteristics in rearward and slow-speed forward flight are satisfactory.

45. Increasing right lateral control displacement was required with increasing forward speed. The total control travel from 20 KTAS in rearward flight to 20 KTAS in forward flight was approximately 1 inch to the right. During the original evaluation, this large trim change was noticeable to the pilot, particularly while hovering in variable wind conditions, and required excessive lateral control corrective movements to prevent undesired sideward drift (HQRS 4). The effects of the lateral trim change were magnified by the excessive lateral control trim displacement band, poor lateral stick centering, and unsatisfactory lateral beep trim operation, all of which combined to emphasize the adverse effects of the lateral trim change with airspeed.

46. To correct this shortcoming, the contractor added a low-rate, automatic parallel trim compensation function to the already existing parallel actuator through which automatic turns were accomplished. Following this change, the lateral trim characteristics were again examined during the reevaluation of the helicopter. Although the lateral trim change characteristics were essentially the same as observed during the original evaluation, the automatic parallel trim compensation device greatly reduced the requirement for the pilot to make lateral trim corrections. As airspeed was increased or decreased from any trim condition, the trim compensation device automatically provided sufficient lateral control input to closely maintain the original trim bank attitude. While the lateral trim compensation device allowed some small bank angle error to persist as airspeed was increased or decreased from trim, the errors would be largely unnoticeable in normal operations. Starting from wings-level flight at zero airspeed, the helicopter could be accelerated to 50 KIAS, using the longitudinal beep trim only, without retrimming laterally. Over this 50-knot speed range, the bank attitude change was less than 1 degree, left wing down. If desired, this small bank-angle error could be easily corrected with the lateral beep trim or by releasing the force trim magnetic brake (HQRS 2). The lateral control trim characteristics in rearward and slow-speed forward flight are satisfactory for the transport mission.

47. Control positions and bank attitude in sideward flight are shown in figure 26, appendix II. The lateral control gradient in sideward flight was strongly positive (increasing lateral control displacement in the direction of flight) to 15 KTAS and slightly positive to neutral at higher speeds. This lateral characteristic, together with the essentially neutral longitudinal and directional trim gradients, provided good sideward flight characteristics. Steady sideward translation over relatively large distances could be accomplished with very minor and infrequent control inputs. Hovering over a spot in steady crosswinds with controls fixed was possible for periods of up to 10 seconds (HQRS 2). The minimum control margin remaining in sideward flight was 37 percent of lateral control at 30 KTAS in left sideward flight. The sideward flight characteristics of the Model 347 helicopter met the requirements of paragraph 3.3.2 of MIL-H-8501A and are satisfactory for the transport mission.

Takeoff and Landing Characteristics

40. Takeoff and landing characteristics were qualitatively evaluated throughout the allowable limits of loading conditions. Operations were limited to level paved surfaces and level grass areas. Surface winds observed during these tests ranged from calm to 20 knots, with maximum gusts to 30 knots. The most common takeoff and landing procedure, starting or ending at a 10-foot aft wheel hover height, was performed in a manner similar to that used in the CH-47C nelicopter. Except for a noticeable requirement to apply lateral control into crosswinds, the helicopter was relatively insensitive to wind direction or magnitude during liftoff from the ground. During liftoff in a direct left crosswind of 20 knots, the helicopter bank attitude changed from a wings-level attitude on the ground to an approximate 3-degree left-wing-down attitude at a stabilized 10-foot hover height. These characteristics did not degrade the pilot's ability to smoothly liftoff and maintain position over the ground.

41. Running takeoffs at ground speeds up to 35 knots were easily accomplished in any of the wind conditions which existed during the tests. With the aft landing gear swivel locks engaged, the aircraft could be accelerated on the ground to a 35-knot ground speed with less than a 5-degree heading change occurring during the acceleration. The helicopter met the requirements of paragraph 3.5.4.2 of MIL-H-8501A.

42. Running landings, with and without engine power, were performed at touchdown ground speeds of from 20 to 35 knots. The helicopter was landed on the aft gear and was allowed to roll initially with the forward gear clear of the runway surface. Pitch attitude and directional control immediately prior to and during touchdown were very natural and comfortable. The overall characteristics were quite similar to those of a tricycle-gear, fixed-wing airplane. Running landings with sideward drift were not evaluated. Within the scope of these tests, the requirements of paragraph 3.5.4.3 of MIL-H-8501A were met. The takeoff and landing characteristics of the Model 347 helicopter are satisfactory for the transport mission.

Low-Speed Flight Characteristics

43. Control trim characteristics in OGE hover were evaluated from 30 KTAS in rearward flight to 47 KTAS in forward flight and to 30 KTAS in sideward flight. The tests were conducted using a ground pace vehicle equipped with a calibrated wind indicator. Control trim positions were recorded in stabilized flight while tracking the pace vehicle at the desired airspeed. The results of these tests at aft cg and heavy weight loading are presented in figures 25 and 26, appendix II. Qualitative evaluations of other cg and weight conditions revealed similar trim characteristics.

44. As shown in figure 25, appendix II, the longitudinal control trim gradient was stable (increasing forward control position with increasing forward airspeed) and essentially linear from 25 KTAS in rearward flight to 47 KTAS in forward

Level Flight Trim Characteristics

48. Control trim characteristics were evaluated by trimming the helicopter in steady-heading, coordinated level flight at 10-knot speed increments from 39 KCAS to V_H. Data were recorded for each stabilized condition. Figures 27 and 28, appendix II, present the control position test data obtained during the original evaluation. Figure 29 presents the test data obtained during the reevaluation. The effects of the DASH system on the longitudinal control trim position were determined by measuring the change in the DASH actuator position at each stabilized airspeed and subtracting the DASH actuator equivalent control motion from the actual control position for that airspeed. The resulting DASH-OFF longitudinal trim characteristic data are presented in figure 30. Comparative data for the CH-47C are also shown on this figure.

49. The longitudinal control trim position gradient in level flight was stable and essentially linear throughout the tested airspeed ranges. As shown in figure C, large changes in weight and cg had only a minor effect on the longitudinal trim gradient. The total control travel, from 50 to 142 KCAS, was approximately 1.4 inches forward at the aft cg loading. Under similar conditions, the CH-47C trim control gradient was unstable, and the control travel over the same airspeed range was approximately 0.7 inch aft. The stable and consistent longitudinal control trim gradient of the Model 347 helicopter decreased pilot workload in changing airspeed and allowed rapid and accurate trimming at any desired airspeed (HQRS 2). This was particularly evident during flight under simulated IFR conditions. Extension or retraction of the landing gear at any trim speed required less than 0.1 inch of longitudinal control motion to maintain the trim airspeed.

50. The DASH-OFF longitudinal control trim position data shown in figure 30, appendix II, represent the conditions which would exist if the Model 347 augmentation system were turned OFF at 140 KCAS. If the DASH system were failed OFF at any other trim airspeed, the actual control position variation with airspeed would follow the gradient shown; however, the curve would originate from the DASH-ON trim position appropriate for that trim airspeed. The DASH-OFF gradient was stable below 50 KCAS, unstable between 50 and 120 KCAS, and slightly stable at faster speeds. Compared to the unaugmented CH-47C, the Model 347 longitudinal trim gradient was noticeably improved at high speed. Although flight under DASH-OFF conditions considerably increased pilot workload in trimming, such a dual failure would not preclude safe recovery of the aircraft. Failure of both DASH systems at high speed would create the most adverse condition, in that, by following the DASH-OFF trim gradient during subsequent deceleration prior to landing, the longitudinal control would be required to be trimmed at an unusually far forward position. When both DASH systems were turned OFF while trimmed at 140 KCAS and an aft cg, the stick position was 5.1 inches from full forward. During the subsequent landing approach, while trimmed at 50 KCAS, the control reached its most forward steady-state position, 3.5 inches from full forward. With this control position, the pilot was left with very little available arm reach with which to counteract nose-up gust disturbances or to apply other necessary forward control movement. It is recommended that the following "CAUTION" be placed in the operator's manual:

CAUTION

Following dual DASH system failure at high speed, unusually far forward longitudinal control positions will be required when trimming at lower airspeed. To preclude exceeding the available arm reach of the pilot, do not allow excessively high nose-up pitch rates or attitudes to occur when decelerating.

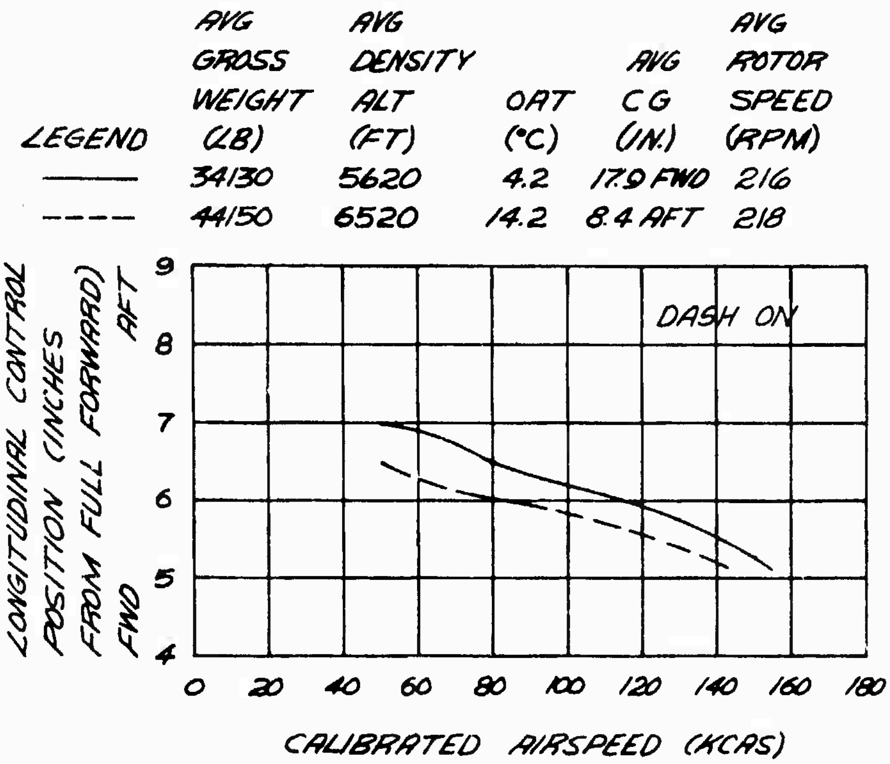


Figure C. Model 347 Static Trim Characteristics.

51. As shown in figures 27, 28, and 29, appendix II, the lateral control trim change from 100 to 150 KCAS was approximately 0.3 inch to the right. This trim change was about twice the lateral migration exhibited by the CH-47C over the same speed range. During the original evaluation, this lateral trim change with speed was objectionable, in that frequent lateral retrimming was required while making airspeed changes (HQRS 4). The effects of the lateral trim change with speed were magnified by the excessive lateral trim control displacement band, poor lateral stick centering, and unsatisfactory lateral beep trim characteristics, all of which combined to emphasize the adverse effects of lateral trim change with airspeed. As described in paragraph 46, the contractor modified the lateral control parallel actuator to provide automatic retrimming of the lateral control. Lateral trim change characteristics were again examined during the subsequent reevaluation

of the Model 347. While the lateral trim compensation device allowed some small bank-angle error to persist as airspeed was increased or decreased from trim, during normal operations the error would be unnoticeable for airspeed changes of up to 50 knots. Starting in trimmed, wings-level flight at 100 KIAS, the aircraft could be accelerated to 150 KIAS or slowed to 50 KIAS without requiring any pilot-supplied lateral retrimming. Starting at hover and accelerating to 150 KIAS without retrimming laterally, the change in bank attitude was 3 degrees, left wing down. The lateral trim corrections needed to maintain coordinated flight over this airspeed range were easily accomplished through the lateral beep trim or by releasing the force trim magnetic brake (HQRS 2). The lateral trim change characteristics of the Model 347 helicopter are satisfactory for the transport mission.

52. Directional trim change characteristics in forward flight were similar to the characteristics of the CH-47C helicopter at airspeeds between 70 and 140 KCAS. At speeds above and below this range, the Model 347 helicopter required slightly greater directional trim corrections with changes in airspeed. Trim corrections required to maintain coordinated flight were easily applied by the pilot through the directional beep trim parallel actuator (HQRS 2). The directional trim change characteristics of the Model 347 helicopter are satisfactory.

53. Pitch attitude variations with changes in trim airspeed were very small. From 50 to 100 KCAS, the change in pitch attitude was essentially zero. From 100 to 154 KCAS, the pitch attitude change was 4 degrees, nose down. Changes in gross weight and cg produced negligible variation in pitch attitude. The insensitivity of the pitch attitude to changes of weight and cg was helpful to the pilot in trimming the aircraft at speeds greater than 100 KCAS. Regardless of loading condition, it was relatively easy for the pilot to remember the unique pitch attitude, as presented on the artificial horizon, which is required for the desired trim airspeed (HQRS 2). The level flight trim attitude characteristics are satisfactory.

<u>Static Longitudinal Stability</u>

54. Static longitudinal stability characteristics were evaluated in level flight, NRP climb, and in autorotation. Tests were conducted at two weights and three cg loading conditions at density altitudes of approximately 4,000 and 10,000 feet. Longitudinal stability characteristics were evaluated by trimming the aircraft at the desired trim speed. While holding collective fixed, the helicopter was then displaced from the trim speed and again stabilized at incremental speeds greater and less than the trim speed. Data were recorded at each stabilized airspeed and are presented in figures 31 through 37, appendix II. Contrary to the characteristics of most aircraft, the simple variation of longitudinal control position with airspeed was not an indicator of static stability because of the contribution of the longitudinal control position transducer (control pick-off) to the DASH actuator. A more realistic indicator of longitudinal static stability was obtained by eliminating the effect of the control pick-off contribution at the off-trim airspeeds. This was done by mathematically subtracting the control pick-off contribution and plotting the results (figs. 31 through 37, app II). As a further confirmation of this method,

additional off-trim stabilized test conditions were obtained by holding the longitudinal control fixed at the trim position and increasing or decreasing airspeed by applying inputs directly to the longitudinal SAS actuators by use of the SAS pulser box. The SAS pulser box is a test device which can be used to apply pulse or step inputs to the rotor heads through the number-one SAS. Use of the SAS pulser box to produce control inputs to the rotors eliminates any influence which might be produced by the control pick-off and its associated circuitry. These inputs were held until the helicopter stabilized on a new airspeed and the results were recorded. The steep gradient (heavy solid line through each trim condition on figures 31 through 37) presents the resulting no pick-off equivalent longitudinal control variation with airspeed and is the best indicator of longitudinal static stability.

55. The longitudinal static stability of the Model 347 helicopter, as indicated by the variation of equivalent longitudinal control position with airspeed, was extremely stable and consistent under all conditions. The minimum gradient was approximately 0.077 inch of equivalent control travel per knot at a trim speed of 76 KCAS and at a heavy weight, aft cg loading (fig. 33, app II). The maximum gradient was approximately 0.111 inch per knot at a trim speed of 128 KCAS and at a light weight, forward cg loading (fig. 31). The variation of equivalent control position was essentially linear and constant about each trim speed and was unaffected by power changes in autorotation and NRP climb or by changes in flight altitude. The static longitudinal stability characteristics of the Model 347 were observed to be very powerful in correcting natural disturbances encountered in flight. The stability characteristics were unaffected by depressing the force trim magnetic brake or by holding the longitudinal control away from the trim position. The strongly stable and consistent static longitudinal stability characteristics of the Model 347 are highly desirable for all piloting tasks and enhance the pilot's ability to operate under IFR conditions (HQRS 1). The static longitudinal stability characteristics met the requirements of paragraphs 3.2.10, 3.2.10.1, and 3.6.3 of MIL-H-8501A and are satisfactory for the transport mission.

56. Static longitudinal stability characteristics with both DASH systems inoperative were determined mathematically by subtracting the DASH actuator motion from the control position data obtained during DASH-ON flight. The results of this computation at a heavy weight, aft cg loading are shown in figure 37, appendix II. A comparison with the CH-47C at similar test conditions is shown on the same figure. Qualitatively, static longitudinal stability with both DASH systems inoperative was neutral to unstable within the airspeed range of 50 to 120 KCAS and was stable at speeds above and below the range. Although flight with both DASH systems inoperative increased pilot workload significantly and required moderate pilot compensation to maintain airspeed and attitude, such a dual failure would not preclude safe continued operation of the helicopter under visual-flight-rule (VFR) conditions.

57. Static longitudinal stability characteristics with one DASH system inoperative were qualitatively evaluated throughout the flight envelope. The Model 347 helicopter was at least slightly stable under all flight conditions, and such a single failure would not preclude continued normal flight operations and mission completion under IFR conditions.

Dynamic Longitudinal Stability

58. Longitudinal dynamic stability characteristics were evaluated in OGE hover and in forward flight at airspeeds of 78 to 129 KCAS. Tests were conducted at two loading conditions at density altitudes of 1,800 feet, 5,000 feet, and 10,500 feet. To evaluate long-term response characteristics, the helicopter was trimmed in level flight at the desired airspeed, and the SAS pulser box was used to displace the aircraft from the trim speed. A 100-percent SAS step input was held until the helicopter stabilized at an off-trim airspeed and the step input was removed. The response of the helicopter in returning to the trim airspeed was then recorded. The results of these tests are presented in figures 38 through 42, apppendix II. Gust response characteristics were investigated by applying 1/2-second longitudinal pulses through the SAS pulser box. The SAS pulse inputs were 100 percent of the extensible link authority, equivalent to approximately 1 inch of the mechanical motion of the longitudinal control. Time histories of representative simulated gust responses are presented in figures 43 through 50. Longitudinal dynamic stability characteristics were also qualitatively evaluated with a 12,000-pound external load. The single point hook was located 55 inches forward of the midpoint between the rotors and, at the takeoff gross weight of 46,000 pounds, resulted in a maximum allowable forward cg condition: 24 inches forward. The concentrated load was attached to the single-point hook by a standard 16-foot military sling. Dynamic characteristics were evaluated by cycling the longitudinal control at the approximate natural frequency of the suspended load and observing the controls-fixed response of the aircraft.

59. As shown in figures 38 through 42, appendix II, the long-term response of the helicopter in returning to the trim airspeed was consistent under all test conditions. Airspeed was essentially deadbeat, to within 1 knot of the original trim speed. Return of the helicopter to trim airspeed and pitch attitude was smooth and positive (HQRS 2). The long-term dynamic longitudinal stability characteristics of the Model 347 helicopter are satisfactory for the transport mission.

60. Short-term longitudinal gust response characteristics of the helicopter were oscillatory and moderately damped, as is shown in figures 43 through 50, appendix II. At a 5,000-foot density altitude, the average damping ratio was 0.30. The period of the oscillation was approximately 1.0 second. At density altitudes of 5,000 feet or below, one overshoot of the trim pitch attitude was observed. Two overshoots were observed at density altitudes above 5,000 feet. The short-term response characteristics were similar to the characteristics of the CH-47C helicopter, as reported in reference 18, appendix I. With all stability augmentation systems operating, the Model 347 helicopter met the longitudinal requirements of paragraphs 3.2.11, 3.2.11.2, and 3.6.1.2 of MIL-H-8501A. During external load operations, ±20-degree longitudinal oscillations of the sling load were intentionally induced by pilot control inputs. In forward flight at 125 KIAS, the period of the free oscillation was approximately 3 seconds. With a 20-degree initial load oscillation, the maximum observed aircraft response was a ±1-degree pitch oscillation. The load oscillation was reduced to ±5 degrees in 3 cycles. At this time, no perceptible aircraft motion was caused by the oscillating load. The

helicopter short-term response characteristics permitted the pilot to readily distinguish between aircraft disturbances caused by motions of the load, and disturbances caused by gusts acting directly on the aircraft. These characteristics allowed the pilot to quickly recognize and respond to load oscillations (HQRS 2).

61. Dynamic longitudinal stability characteristics with one SAS and DASH system inoperative were evaluated qualitatively under the same conditions described in paragraph 58. Compared to the aircraft responses with all augmentation systems operating, the helicopter pitch damping was noticeably reduced. At density altitudes above 5,000 feet, a persistent, lightly damped, short-term pitch oscillation of ±1 degree occurred under all flight conditions tested. The period of this oscillation was approximately 3 seconds, and the oscillation could be visually observed for 2 cycles. This lightly damped oscillation did not significantly degrade the helicopter flying qualities and would not preclude safe mission completion under any circumstances. The dynamic longitudinal stability characteristics following single SAS and DASH system failure met the requirements of paragraph 3.6.1 of MIL-H-8501A and are satisfactory.

Static Lateral-Directional Stability

62. Static lateral-directional stability characteristics were evaluated in level flight, NRP climb, and autorotation. Tests were conducted at two loading conditions over the airspeed range of 73 to 129 KCAS. The tests were conducted by trimming the aircraft in coordinated (ball-centered) flight at the desired airspeed and recording the control positions and bank attitude. Holding collective fixed, the aircraft was then displaced to incremental sideslip angles on either side of the trim sideslip angle and stabilized in a steady-heading sideslip. The control positions and bank attitude were recorded at increasing sideslip angles up to the envelope limit. The results of these tests are presented in figures 51 through 59, appendix II. Comparisons of the static directional stability characteristics of the Model 347 and CH-47C helicopters are presented in figure 60.

63. Static directional stability, as indicated by the variation of directional control position with sideslip, was strongly positive up to sideslip angles of ±10 degrees from trim and was slightly less positive at greater sideslip angles. The pedal position gradient was not significantly affected by variations in density altitude or aircraft loading but was increasingly more positive with increasing airspeed. Qualitatively, directional stability was at least slightly positive at all airspeeds above 40 KIAS and neutral at all lower airspeeds. The SAS-ON static directional stability characteristics were qualitatively evaluated as being similar to the characteristics of the CH-47C helicopter.

64. Static directional stability characteristics with all stability augmentation systems inoperative were mathematically determined by subtracting the SAS extensible link contribution from the SAS-ON directional control position data and plotting the results as broken lines in figures 51 through 60, appendix II. With both stability augmentation systems inoperative, the variation of pedal position with sideslip was at least slightly stable for small sideslip angles (±3 degrees) about

trim at all airspeeds above 60 KIAS. At larger sideslip angles, the gradient was neutral in powered flight and slightly unstable in autorotation. At airspeeds of 60 KIAS or less, the gradient was neutral to slightly unstable at all sideslip angles. As shown in figure 60, at a heavy weight, aft cg loading condition the CH-47C pedal gradient (SAS OFF) is generally unstable. Under similar conditions, the Model 347 was slightly stable. Compared to the CH-47C helicopter, the slightly stable SAS-OFF pedal position gradient of the Model 347 noticeably improved the pilot's ability to continue flight under SAS-OFF conditions.

65. Dihedral effect, as indicated by the variation of lateral control position with sideslip, was positive and essentially linear at all test conditions. Dihedral effect was increasingly more positive as engine torque increased with increasing airspeed or increased gross weight. The minimum observed lateral control gradient occurred in autorotation at approximately 73 KCAS. At any given airspeed and loading condition, the lateral control gradient of the Model 347 was slightly less positive than the gradient of the CH-47C helicopter under similar conditions (ref 18, app I).

66. During the original evaluation, pedal-only turns resulted in consistent steady-state roll displacement into the turn. Left pedal displacement produced steady-state left bank, and right pedal input produced a right bank. This characteristic further demonstrated positive effective dihedral. Pedal-only turn characteristics were again investigated during the reevaluation after the lateral control system had been modified (para 22). During this reevaluation, pedal inputs of up to 1.0 inch in forward flight produced no steady-state roll displacement. With the lateral control free, pedal inputs produced essentially flat turns, indicating neutral effective dihedral. This change in the results of the pedal-only turn tests appeared to have been caused by the improved centering characteristics of the lateral control and the widened roll-hold detent switch setting incorporated in the control system prior to the reevaluation. Since no requirement exists for performing pedal-only turns with the lateral control free, as in this test, the observed neutral dihedral effect during pedal-only turns does not detract from the transport mission capability.

67. Side-force characteristics, as indicated by the variation of bank angle with steady-heading sideslip, were positive under all powered-flight conditions tested, and increased with increasing engine torque as airspeed or gross weight increased. As shown in figure 59, appendix II, the bank-angle gradient in autorotation was slightly positive at sideslip angles up to 5 degrees from trim and essentially neutral at higher angles of sideslip. This absence of significant side force in autorotation resulted in the helicopter being trimmed at large sideslip angles without the pilot being aware of this condition. This condition could cause degradation of autorotational descent performance, in that the sideslip trim error could result in erroneous airspeed indications and misleading side-drift indications. Trimming the aircraft within satisfactory sideslip angles required moderate pilot compensation (HQRS 4). Correction of the inadequate side-force characteristic in autorotation is desirable for improved helicopter operation.

68. Increasing sideslip angles in either direction from trim required aft displacement of the longitudinal control to maintain trim airspeed. Right sideslip

required slightly greater aft longitudinal control displacement than was required for equal left sideslip; however, this difference was not noticeable to the pilot. The pitch-with-sideslip characteristics were similar to the characteristics of the CH-47C, as reported in reference 18, appendix I.

69. The static lateral-directional stability characteristics of the Model 347 helicopter met the requirements of paragraphs 3.3.9 and 3.6.2 of MIL-H-8501A. Except for the inadequate side-force characteristic in autorotation (para 67), the static lateral-directional stability characteristics of the Model 347 helicopter are satisfactory.

Dynamic Lateral-Directional Stability

70. Dynamic lateral-directional stability characteristics were evaluated during OGE hover and in forward flight at density altitudes up to 10,500 feet. Quantitative data were obtained by introducing 1/2-second pulses into the SAS through a SAS pulser box. The pulses used were 100 percent of the lateral and directional SAS authority, equivalent to approximately 0.5 inch of lateral control displacement and 0.6 inch of directional control displacement. The results of these tests are presented in figures 61 through 73, appendix II.

71. As shown in figures 61 through 67, appendix II, the responses to lateral pulses were lightly damped, with an average period of 1.9 seconds. At a 5,000-foot density altitude, the average damping ratio was 0.15, and three overshoots of the trim bank attitude could be observed. The lateral pulses did not excite any noticeable directional response. Changes in altitude, gross weight, and speed had no significant effect on the aircraft response to lateral pulse inputs.

72. During the original evaluation, directional pulses, as shown in figures 68 through 73, appendix II, occasionally caused the lateral control to be forced out of the detent position. When this phenomenon occurred, lightly damped roll and yaw oscillations resulted, as shown in figure 68. During the reevaluation, after the lateral control had been modified, this phenomenon did not reoccur, and the roll and yaw oscillations were always moderately damped. At a 5,000-foot density altitude, the average damping ratio of directional oscillations was 0.25. The requirements of paragraph 3.6.1.2 of MIL-H-8501A were met.

73. Turns employing only lateral cyclic were evaluated at speeds above 40 KIAS with all augmentation systems operating, and also with one SAS inoperative. Turn characteristics were essentially the same under both conditions. A lateral cyclic control step input to produce a 30-degree roll displacement in 6 seconds resulted in a maximum adverse yaw of 2 degrees and caused the ball to slip one-quarter of a ball width from the center of the inclinometer. No reversal of the rolling velocity occurred under any test conditions. The initial adverse yaw subsided within 2 seconds of the lateral step input, and the turn was thereafter well coordinated. The cyclic-only turn characteristics met the requirements of paragraphs 3.3.9.1 and 3.3.9.2 of MIL-H-8501A. The forward-flight cyclic turn capabilities were effectively used during simulated IFR cruise and approach tasks, and no adverse

characteristics were observed (HQRS 2). The dynamic lateral-directional stability characteristics of the Model 347 helicopter are satisfactory for the transport mission.

Maneuvering Stability

74. Maneuvering stability characteristics were quantitatively evaluated at a heavy weight, aft cg loading condition at a density altitude of 5,200 feet. The variation of longitudinal control position with normal acceleration was determined by trimming the aircraft in coordinated level flight at the desired airspeed and then rolling the aircraft to incremental target bank attitudes to the left and right. The thrust rod was fixed at the original trim setting, and the indicated ship's system trim airspeed was maintained by allowing the aircraft to descend, if necessary. After stabilizing at the desired conditions, longitudinal control position and normal acceleration data were recorded. Data were also recorded while performing symmetrical steady pull-ups at the same trim speeds. The results of these tests are summarized in figure 74, appendix II. At trim airspeeds of 78 and 130 KCAS, the variation of longitudinal control position with normal acceleration was slightly stable and essentially linear in both symmetrical steady pull-ups and right turns. The gradients in left turns were essentially neutral. The data obtained during these particular tests may be unreliable due to the use of the ship's airspeed system to indicate airspeeds in turns. The boom airspeed system installed on the aircraft had been proven to be unreliable prior to this test, and the characteristics of the ship's airspeed system in turning flight were unknown. In order to clarify the validity of these results, it is recommended that a suitable airspeed boom be installed and properly tested on the Model 347 prior to initiation of Phase II testing. By comparing boom airspeed readings with the ship's system readings during Phase II testing, it may be possible to determine whether the difference in longitudinal control position data in left and right turns was the result of unsymmetrical maneuvering characteristics or a consequence of the airspeed system error in turns.

75. Qualitative evaluations of maneuvering characteristics at a 33,000-pound gross weight with a 16-inch forward cg indicated that the variation of longitudinal control position with normal acceleration was essentially neutral in both left and right turns at all airspeeds between 75 and 150 KIAS. From the wings-level trim condition, the aircraft could be rolled into a coordinated turn, up to the bank angle limit, with the longitudinal cyclic held fixed at the original trim position. In a right turn, under these conditions, the helicopter stabilized at indicated airspeeds which were 3 to 5 knots greater than the trim speed. In left turns, the aircraft stabilized within 1 or 2 knots of the original trim speed. The steady-state maneuvering characteristics were qualitatively evaluated as being similar to the characteristics of the CH-47C helicopter.

76. Response of the helicopter to aft longitudinal step inputs was evaluated under conditions similar to those established for the previously described turning maneuvers and symmetrical pull-up tests. For the step input tests, the aircraft was trimmed at the test airspeed, and a control fixture was adjusted to allow input of the desired control displacement. The aircraft angular velocity response and

normal acceleration response to aft step inputs were recorded. Results of this test are presented in figure 75, appendix II. At an initial trim speed of 75 KCAS, the angular velocity response was concave downward at approximately 0.5 second, and the normal acceleration was concave downward at 1.0 second. These response characteristics met the requirements of paragraph 3.2.11.1 of MIL-H-8501A.

77. Although longitudinal control force characteristics were not quantitatively determined during maneuvering tasks, this characteristic was qualitatively evaluated under all forward flight conditions. During operational tasks involving severe maneuvering, such as collision avoidance or terrain following, stick force per g was sensibly positive at all conditions evaluated. Satisfactory performance of any normal maneuvering task required minimum pilot effort (HQRS 3). The longitudinal maneuvering stability characteristics of the Model 347 helicopter are satisfactory for the transport mission.

Autorotational Characteristics

78. Autorotational flight characteristics were qualitatively evaluated at a gross weight of 45,000 pounds with a 9-inch aft cg. The aircraft was flown at airspeeds from 150 KIAS to an estimated 15 KIAS and within the density altitude range of 5,000 to 1,700 feet. Autorotational flight was entered by lowering the collective and simultaneously "beeping down" both engines to reduce engine torque to near zero. Landings were accomplished on a hard-surface runway at estimated touchdown speeds of 15 to 35 knots with wind speeds less than 3 knots.

79. Normal maneuvers were easily performed in autorotational flight. The inadequate side-force characteristic (para 67) degraded the pilot's capability to trim accurately in stabilized flight but did not adversely affect the maneuvering characteristics of the aircraft. Use of lateral cyclic alone to make turns produced satisfactory turn coordination. Landing deceleration and touchdown at ground speeds of 15 to 35 knots was comfortable and pleasant. The pilot's view of the touchdown area was unobstructed. With a landing field density altitude of 400 feet, the aircraft required approximately 400 feet to stop after touchdown at 35 knots. Although this stopping distance could be reduced with additional pilot proficiency, the Model 347 helicopter probably cannot meet the 200-foot maximum stopping distance requirement of paragraph 3.5.4.4 of MIL-H-8501A. The helicopter met the requirements of paragraphs 3.3.8, 3.5.4.3, and 3.5.7 of MIL-H-8501A. Except for the neutral side-force characteristic which adversely affected the pilot's ability to trim accurately, the autorotational flight and landing characteristics are satisfactory.

Simulated Single-Engine Failure

80. Failure of a single engine was simulated in level flight and in NRP climb at a gross weight of approximately 45,000 pounds with a 9-inch aft cg. Failure of the engine was simulated by moving the desired engine condition lever to ground idle. All pilot controls were held fixed following the simulated failure. A typical time history of a failure while climbing is shown in figure 76, appendix II. The

helicopter response to single-engine failures was extremely mild at all test conditions. The most noticeable motion was a ±5-degree roll oscillation. During the 15-second period following loss of power, the aircraft lost approximately 5 knots of airspeed. At the 140-KCAS trim speed, rotor speed loss was 7 rpm during the level-flight test and 15 rpm during the climb test. No pilot corrective action was required to counteract any airframe or rotor response characteristic (HQRS 2). Rising turbine inlet temperature values on the remaining engine required reduction of the thrust lever setting to preclude exceeding allowable temperature limits. Within the scope of this test, the helicopter met the requirements of paragraph 3.5.5.ı of MIL-H-8501A. The single-engine failure characteristics are satisfactory.

Stability Augmentation System Failure Characteristics

81. Single and dual SAS failures were evaluated throughout the allowable flight envelope. Two failure modes were evaluated: go-dead failures (SAS OFF), and hardover failures. For either mode of failure, the aircraft was evaluated both with one SAS still functioning and with one SAS in a preexisting OFF condition. Failures were introduced by turning OFF the desired SAS with the console switch or by introducing 100-percent hardover signals in the number-one SAS through the SAS pulser box.

82. Aircraft dynamic stability characteristics with one SAS OFF were not noticeably different from the characteristics with both stability augmentation systems ON. With both stability augmentation systems OFF, the dynamic stability characteristics were considerably degraded, but continued safe operation of the helicopter in VFR conditions was possible. The stability characteristics of the Model 347 with both stabiltiy augmentation systems inoperative were qualitatively evaluated as being improved over those of the CH-47C. As in the CH-47C, however, routine IFR operations in this condition are not recommended because of the high pilot workload required.

83. Single SAS hardover failures with the remaining SAS in normal operation produced noticeable aircraft response in the failed axis. Within the test airspeed range of 100 to 150 KIAS, no pilot action was required to correct the aircraft motion following this single failure. The effect of the failure was to rotate the aircraft through a small pitch, roll, or yaw displacement and stabilize at the new attitude. Recovery from this failure required only that the failed SAS be identified and turned OFF, following which the helicopter could continue normal flight with a single SAS operating. The most severe aircraft response to SAS failures was produced by applying a SAS hardover input with the other SAS already inoperative. In this dual-failure case, the roll axis hardover produced the least response, and the yaw axis hardover produced the most rapid response. The yaw hardover produced a yaw displacement of 10 degrees in approximately 1 second, and pilot corrective action was required after 1/2 second of additional delay in order to preclude exceeding sideslip limits. Roll and pitch hardovers required pilot corrective actions after approximately 4 and 3 seconds, respectively. Recovery from these

failures required that the SAS which introduced the hardover be identified and turned OFF, after which the aircraft could continue to be operated with both stability augmentation systems turned OFF.

84. The Model 347 helicopter met the SAS single-failure requirements of paragraphs 3.5.9(b) and 3.6.1 and the dual-failure requirements of paragraph 3.5.9(d) of MIL-H-8501A. The single-failure response characteristics are satisfactory for Army use under all conditions; however, the dual-failure response characteristics are sufficiently severe as to compromise continued safe operation under IFR conditions. Under IFR conditions, hardover failure in the yaw axis with the other SAS previously failed OFF would seriously affect the pilot's ability to recover the aircraft. It is therefore recommended that the following "WARNING" be placed in the Model 347 operator's manual:

WARNING

> Intentional operation into known instrument-flight-rule (IFR) conditions is not recommended unless both stability augmentation systems are operating properly prior to entry into such flight conditions. In the event that one SAS is inoperative, failure of the remaining SAS while flying in IFR conditions may result in loss of aircraft control.

Control System Hydraulic Power Failure Characteristics

85. Single failures of the control system dual hydraulic power systems were simulated by turning OFF the desired power source. As in the CH-47C, failure of a single system produced no adverse results. Turning OFF the number-one system caused the variable force-feel system control force gradient to be reduced to a constant lower level. This change in control force characteristics required only minimal pilot compensation to continue normal operations (HQRS 3). The power-operated control system characteristics met the requirements of paragraph 3.5.8 of MIL-H-8501A and are satisfactory for the transport mission.

MISCELLANEOUS

Cockpit Evaluation

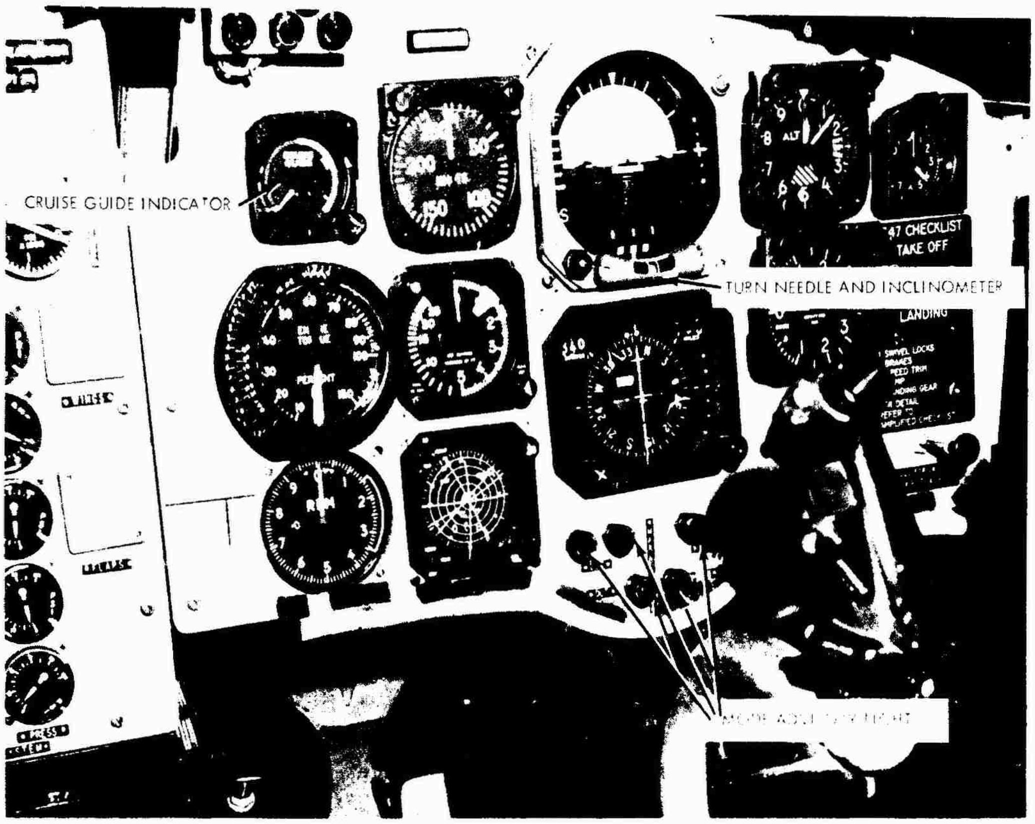
86. The pilot and copilot seats installed in the Model 347 helicopter provided four adjustments: fore and aft, up and down, seat back recline, and upper leg support. Photograph B is a side view of the pilot seat showing the locations of the adjustment release handles. The seat was easily adjusted to fit the pilot and provided excellent support. The upper leg support adjustment, which controlled the height of the forward portion of the seat bottom, was of particular benefit in reducing leg fatigue and allowing adequate blood circulation to the lower extremities.

Photograph B. Side View of Pilot Seat.

87. Engine condition levers, which are located on the center console of the CH-47C helicopter, were moved to the overhead console on the Model 347. Location of the condition levers in this overhead position did not adversely affect aircraft operation.

88. The pilot turn needle and inclinometer were integrated into the lower portion of the flight director case, as shown in photograph C. The small size of the turn needle and inclinometer prevented the pilot from quickly and accurately reading and interpreting these instruments. A larger-size turn needle and inclinometer are desirable for improved helicopter operation.

89. As shown in photograph C, mode advisory lights, indicating the status of the VHF navigation receivers and automatic flight path coupler, were located at the bottom edge of the pilot panel. In this position, the lights were out of range of the pilot's normal scanning pattern, and changes in the mode status could not easily be detected. Relocation of the mode advisory lights to a position within the pilot's normal scanning field of view is desirable for improved helicopter operation.



Photograph C. Pilot Instrument Panel.

90. As shown in photograph C, a cruise guide indicator (CGI) was mounted on the pilot panel to provide cockpit indications of the stress loads on the flight control system. The indicator face was divided into two color bands and a "barber pole" band. The two color bands, green and yellow, indicated allowable steady-state and transient loads, respectively, while the "barber pole" range indicated stress loads beyond allowable limits. A self-testing circuit was provided to check static calibration of the device. The CGI was useful to the pilot during maneuvering flight, especially when operating at heavy weights. The instrument was easily monitored in flight; and, under circumstances in which high load readings occurred, the pilot was able to quickly take corrective action to reduce the indicator reading.

Automatic Flight Path Control System

91. Qualitative evaluations of the automatic flight path control system (FPCS) were conducted under VFR and simulated IFR conditions. These tests were conducted within the airspeed range of 50 to 150 KIAS and throughout the allowable loading and altitude limits as specified in the safety-of-flight release (ref 11, app I). The FPCS provided the following aircraft control modes: heading mode for acquiring and maintaining any preselected heading; radio (VOR, ILS, FM) or doppler navigation to provide automatic heading control; and altitude mode

to provide automatic hold of barometric altitude or vertical tracking of ILS glide path. The FPCS operated through parallel actuators connected to the cockpit controls and, therefore, moved the cockpit controls when operating.

92. Operation of the altitude-hold function of the FPCS was evaluated by engaging the altitude hold in stabilized level flight and observing the indicated altitude variation after engagement. During the original evaluation, the altitude hold permitted long-term altitude errors of 200 feet to occur within 3 minutes of engagement. Such large errors rendered the altitude hold unusable for IFR operations. After modifications were made to the altitude-hold circuitry, the FPCS was reevaluated. As a result of the modifications, the long-term altitude error at constant airspeed was reduced to approximately 40 feet. When an airspeed change was made with the altitude hold engaged, the long-term altitude error increased considerably. In one case, the altitude hold was engaged at 100 KIAS, and while holding airspeed constant, the aircraft descended 40 feet in 2 minutes. After stabilizing at 40 feet below the original engagement altitude, the aircraft was slowly acclerated to 150 KIAS. While accelerating to the higher airspeed, the aircraft descended an additional 60 feet and again stabilized. The total long-term error in this case was 100 feet. Although the altitude-hold function is useful for VFR operations, the observed altitude errors are unsatisfactory for IFR operations. Correction of this shortcoming is desirable for improved helicopter operation.

93. Operation of the FPCS in the ILS glide-path tracking mode, heading mode, and radio navigation mode was satisfactory under all test conditions. Control inputs were smooth and provided comfortable rates of motion at all times. Interception of VOR radials at ranges of less than 3 miles from the station resulted in considerable overshoot of the radial; however, this characteristic required only minimal pilot compensation to adequately correct. Because of the physical location of the lateral control parallel actuator, the pilot was able to manually override the steering inputs of the FPCS merely by displacing the lateral control against the opposing force of the force-feel system. When the pilot no longer desired to override the FPCS, the lateral control could be released, and the FPCS smoothly resumed steering control. The ability to enter or leave the control loop at will, while leaving the FPCS coupled, enabled the pilot to easily perform any desired maneuvers (HQRS 2). Use of the FPCS to perform ILS glide-path tracking and to provide automatic steering commands to the flight control system reduced the pilot workload under all flight conditions and was particularly useful under IFR conditions. The ILS glide-path tracking mode and the steering modes of the FPCS are satisfactory for use during both VFR and IFR flight conditions.

<u>Avionics Systems</u>

94. Testing of the avionics systems consisted of qualitative evaluations of the installed communications and navigation systems as well as evaluations of the HZ-6B attitude indicator (flight director) and RD-100 radio deviation indicator (horizontal situation display). The evaluation included use of ground-based VOR and ILS facilities operated by the Federal Aviation Administration. The navigation systems installed in the Model 347 helicopter provided the following capabilities in both

coupled and uncoupled modes: doppler navigation, VOR navigation and approach, FM homing, and ILS approach. Two additional functions, programmed approach to hover and hover hold, were not evaluated due to prohibitions contained in the safety-of-flight release (ref 11, app I). During the course of the evaluation of the avionics systems, one deficiency and six shortcomings were observed on one or more occasions.

95. With a VOR station being received, the deviation indicator of the horizontal situation display and the steering command bar of the flight director oscillated at a frequency of 1 hertz. The oscillation of the deviation indicator was approximately one-half full scale deflection. When operating in the coupled mode, the FPCS attempted to follow the oscillating steering commands and produced an oscillatory rolling motion of the aircraft. This characteristic of the VOR receiver required moderate pilot compensation to track a desired VOR radial in the manual mode and precluded VOR tracking in the coupled mode. Correction of this shortcoming is desirable for improved aircraft operation.

96. When passing over a VOR transmitter, the deviation indicator of the horizontal situation display and the steering command bar of the flight director oscillated erratically over the full range of needle limits. When operating in the coupled mode, the FPCS attempted to follow the erratic steering commands and produced uncomfortable rolling motions. The pilot was therefore required to uncouple the FPCS prior to crossing a VOR and recouple after the station had been passed. Lack of capability to track across a VOR station in the coupled mode degraded the usefulness of the FPCS. Correction of this shortcoming is desirable for improved aircraft operation.

97. When executing ILS approaches, the flight director steering command bar repeatedly commanded flight through the localizer centerline. When the pilot followed the steering commands, the helicopter "S turned" all the way down the approach path. A similar result was produced by the FPCS when operating in the coupled mode. Correction of this shortcoming is desirable for improved aircraft operation.

98. Faulty operation of logic circuitry allowed false and misleading flight-path information to be presented to the pilot. After capture of a VOR radial near the transmitter station, the steering command bar was centered and the capture light was illuminated. Subsequent to capture, the helicopter was flown outbound from the station on a radial 30 degrees to the right of the radial selected on the course selector. At this time, the flight director steering bar was centered, incorrectly indicating that the aircraft was on the selected radial. This incorrect indication was confirmed by comparing the indication of the horizontal situation display deviation indicator with the steering command bar. This deficiency in flight director logic could be unsafe under IFR flight conditions, in that the pilot could be led to fly into hazardous off-course conditions. Correction of this deficiency is mandatory for safe aircraft operation under IFR conditions.

99. The horizontal situation display had no provision to warn the pilot that the heading indicator was not synchronized to the correct magnetic heading. Lack of proper synchronization could be determined only by looking at the heading reference system control panel located on the center console. For ease in checking the functioning of the heading system, an indication of heading synchronization should be provided within the pilot's normal scan area. Correction of this shortcoming is desirable for improved helicopter operation.

100. After the VOR/ILS capture light had once been illuminated, indicating capture of the selected VOR radial or ILS localizer, the light remained on regardless of the position of the aircraft with respect to the desired ground track. This characteristic could cause misinterpretation of the flight director capabilities, in that the flight director did not provide satisfactory steering commands when the aircraft was at a large angular displacement from the desired VOR radial or ILS localizer. In order to preclude possible misinterpretation, the VOR/ILS capture light logic should be modified so that the light will be extinguished at any time the aircraft is displaced beyond the recommended usable range limits of the flight director. Correction of this shortcoming is desirable for improved helicopter operation.

101. The horizontal situation display contained an annunciator window which displayed the signal source being used to drive the flight director. The numeral "1" was used to indicate a VHF navigation signal source, and the numeral "2" was used to indicate a doppler navigator signal source. No identification of signal source was provided when using an FM signal. The only means by which the pilot could determine that a valid FM signal was being received and processed was by observation of the motion of the deviation bar or steering command bar. Use of the existing annunciator device to indicate that a valid FM signal was being received would enable the pilot to easily recognize that a valid signal was being received and used by the system. Correction of this shortcoming is desirable for improved helicopter operation.

Airspeed Calibration

102. Because of the lack of an existing airspeed calibration, the ship's airspeed system was calibrated during the course of the evaluation. A measured ground course was used for this purpose. The results of this test are presented in figure 77, appendix II. The variation of indicated airspeed with calibrated airspeed was essentially linear over the tested airspeed range of 46 to 169 KIAS. The position error was zero at 110 KIAS, approximately 5 knots at 50 KIAS, and -4 knots at 169 KIAS. The position error characteristics of the ship's airspeed system are satisfactory.

Ground Operation Characteristics

103. Ground handling characteristics were evaluated on paved surfaces and on smooth sod surfaces in winds up to 20 knots. Taxiing was normally accomplished with the longitudinal cyclic speed trim (LCST) selector in the TAXI position. With

the selector switch in this position, the tip path planes of both rotors are tilted downward in front to increase the horizontal thrust vector and reduce the amount of collective blade angle required to taxi. This mode of operation of the LCST is not available in the CH-47C helicopter. Use of the TAXI position of the LCST significantly reduced the engine torque required during taxiing and aided in keeping the landing gear firmly on the ground during all maneuvers. Compared to the CH-47C helicopter, the ground handling characteristics of the Model 347 are significantly improved due to the incorporation of the TAXI position in the LCST selector. The Model 347 would probably meet the directional control requirements of paragraph 3.3.1 of MIL-H-8501A. No instance of droop stop pounding was observed during ground operations. The taxiing and pivoting requirements of paragraph 3.5.3 of MIL-H-8501A were met. Within the scope of this test, the ground handling characteristics of the Model 347 are satisfactory for the transport mission.

Vibration Characteristics

104. Vibration characteristics were evaluated with all installed vibration absorbers operating. Vibration sensors were installed at the following locations: pilot and copilot heel slides (station 50); pilot seat (station 95); front of cabin, immediately aft of companionway door (station 160); mid cabin (station 360); and rear of cabin, immediately forward of cargo ramp hinge (station 592). The locations of these sensors are described in further detail in appendix VIII. The measured vertical, lateral, and longitudinal vibration characteristics at frequencies corresponding to 4, 8, and 12 cycles per main rotor revolution are presented in figures 78 through 91, appendix II. These figures show the maximum and minimum amplitude which occurred over a 10-rotor-revolution (10/rev) data sample at each test condition. The 4/rev vertical vibration characteristics are summarized in table 6.

105. As shown in figures 78 through 81, appendix II, 4/rev vibrations in the cockpit area were less than 0.11g during all level flight tests. The 8/rev amplitude was considerably higher, with a maximum value of 0.50g vertical vibration recorded at the pilot heel slide in level flight at 153 KTAS. The 4/rev vibration levels in the cockpit were qualitatively evaluated as being significantly lower than the 3/rev vibration levels in the CH-47C helicopter at any similar test condition. In forward flight at 160 KIAS, the overall vibration level perceived by the pilot was evaluated as being equivalent to that perceived in the CH-47C at 130 KIAS. Although the 8/rev vibration amplitude (0.50g at 153 KTAS) exceeded the 0.15g and 0.20g limits of paragraph 3.7.1(b) of MIL-H-8501A, the 8/rev vibrations were not objectionable and did not cause any noticeable pilot discomfort. The contract work statement (ref 5, app I) specifies that the maximum cockpit vibration loads under steady-state flight conditions are not to exceed 0.05g at the 4/rev frequency and 0.10g at the 8/rev frequency. The maximum 4/rev vibration recorded in the cockpit (0.11g) exceeded the contract limit by 0.06g or 120 percent. The maximum observed 8/rev vibration (0.50g) exceeded the contract limit by 0.40g or 400 percent. A heavy, low-frequency vibration frequently observed in the CH-47C in sideward flight at heavy weights was entirely absent in the Model 347 helicopter. The cockpit vibration characteristics of the Model 347 are satisfactory.

Table 6. Level Flight 4/Rev Vertical Vibrations.[1]

Density Altitude: 5,100 feet
Gross Weight: 44,800 pounds

Rotor Speed: 219 rpm
Center of Gravity: FS 393.6 (7.6 inches aft)

Fuselage Station	True Airspeed					
	80 Knots		120 Knots		155 Knots	
	Maximum Value[2] (g)	Minimum Value (g)	Maximum Value[2] (g)	Minimum Value (g)	Maximum Value[2] (g)	Minimum Value (g)
50	0.11	0.09	0.08	0.05	0.09	0.03
95	0.05	0.04	0.03	0.02	0.07	0.03
160	0.24	0.21	0.18	0.15	0.08	0.02
360	0.14	0.12	0.09	0.07	0.19	0.17
592	0.31	0.26	0.27	0.26	0.50	0.39

[1]Accelerations measured over 10 rotor revolutions.
[2]Military specification: acceleration (g) between 30 knots rearward and V_{cruise} not to exceed 0.15g for frequencies up to 32 hertz (para 3.7.1(b) of MIL-H-8501A).

106. The 4/rev vibration levels were significantly higher in the cabin area than in the cockpit. The highest and most objectionable vibration levels occurred in the aft portion of the cabin, near station 592, just forward of the ramp hinge. As shown in figures 86 and 87, appendix II, the 4/rev vertical vibration maximum amplitude at station 592 exceeded 0.50g at 155 KTAS at a gross weight of 44,800 pounds. At a gross weight of 36,130 pounds, the maximum 4/rev vertical amplitude was 0.36g at 159 KTAS and a 4,130-foot density altitude. During flight at low altitude (density altitude of 1,280 feet) with the same loading conditions, the maximum vertical vibration level at 175 KTAS was 0.71g. In addition, high lateral vibration levels also occurred in this area of the cabin. At 155 KTAS, the lateral vibration level exceeded 0.15g under all loading conditions. The cabin floor vibration levels were qualitatively evaluated by standing and sitting at various cabin locations. The aft cabin area, that area between station 500 and station 592, was found to be unsatisfactory for passenger use at speeds above 130 KTAS. The vibration levels at speeds greater than 130 KTAS were physically uncomfortable and could not be tolerated for extended periods. Voices of persons standing in this area were distorted sufficiently by the vibrations to greatly reduce intelligibility of intercom transmissions. The excessive vibration levels in the aft area of the

cabin reduce the capability of the aircraft to perform passenger transport missions. Correction of the excessive aft cabin 4/rev vibration characteristics is desirable for improved aircraft capabilities.

107. Ground resonance and mechanical instability characteristics were evaluated throughout the allowable loading envelope. Only one instance of a mechanical instability occurred. This instance was under such unusual circumstances that reoccurrence during normal operations would be highly improbable. The phenomenon occurred during hover performance testing with the aircraft connected to a 150-foot steel cable which was anchored to a buried deadman. With an indicated cargo hook load of 16,000 pounds and an aircraft gross weight of 33,000 pounds (effective gross weight of 49,000 pounds), the helicopter was translated to the left of the deadman so that the cable was at an approximate 20-degree angle from the vertical. As the rotor speed was reduced from the normal 220 rpm to the minimum allowable speed of 215 rpm, both the thrust rod brake trigger and the cyclic force trim release button were depressed and held in. As the rotor speed reached 215 rpm, an observer noted that the external cable began to oscillate laterally. At approximately the same time, the aircraft began a roll oscillation at a frequency of approximately 2 hertz. The amplitude of the oscillation was observed to increase slightly over a period of approximately 5 seconds, at which time the pilot lowered the collective about 1/2 inch, and the oscillation almost immediately ceased. An attempt was made to duplicate the oscillation by repeating the test conditions under which it had first occurred. This attempt was unsuccessful, and no oscillation could be excited. Since the observed oscillation occurred under such unusual circumstances and was so easily stopped, this characteristic does not adversely affect the capabilities of the helicopter. The Model 347 helicopter met the intent of paragraph 3.7.3 of MIL-H-8501A.

Noise Characteristics

108. Interior and exterior noise characteristics were evaluated in forward flight and also in IGE and OGE hover. Tests were conducted at an average gross weight of 45,000 pounds with the cg at 7.6 inches aft of the midpoint between rotors. The door separating the cockpit from the cabin area was closed during all cockpit noise measurements. The cabin area (cargo compartment) was not acoustically treated and was not intended to represent the acoustical environment of an operational aircraft. The major purpose of the evaluation was to determine the cockpit noise and exterior noise characteristics. Quantitative evaluations of the noise characteristics were performed by the US Army Aeromedical Research Laboratory (USAARL), Fort Rucker, Alabama. The test methods and test results were published in a separate report published by USAARL (ref 21, app I). Selected results of these tests have been extracted from the USAARL report and are presented in tables 7 and 8.

Table 7. Cockpit Interior Noise Measurements.[1]

Average Noise Level at 100 KIAS in Level Flight (db)						
Data Source	Octave Band Center Frequency (Hz)					
	31.5	125	500	2,000	4,000	8,000
Boeing 347[2]	110	89	88	95	89	74
CH-47C[3]	117	100	98	108	100	89
Specification Limit[4]	111	111	109	100	94	94

[1]Measured between pilot and copilot at head level (data extracted from table XXV, ref 21, app I).
[2]Gross weight, 45,000 pounds; density altitude, 5,000 feet; rotor speed, 230 rpm.
[3]Rotor speed, 235 rpm; loading and flight conditions, unknown.
[4]Maximum acceptable noise level with protective helmets worn (MIL-A-8806A, para 3.1.3, table IIIA).

Table 8. Exterior Noise Measurements.[1]

Average Noise Level at a 10-Foot Hover Height (db)						
Data Source	Octave Band Center Frequency (Hz)					
	31.5	125	500	2,000	4,000	8,000
Boeing 347[2]	90	85	77	74	77	72
CH-47C[3]	91	85	84	74	72	72

[1]Measured 300 feet from front of helicopter in winds less than 5 knots (data extracted from table XXVII, ref 21, app I).
[2]Gross weight, 45,000 pounds; density altitude, 1,800 feet; rotor speed, 230 rpm.
[3]Rotor speed, 235 rpm; loading and flight conditions, unknown.

109. As shown in table 7, interior noise in the cockpit was significantly lower than measured in the CH-47C. In level flight at 100 KIAS, the Model 347 was from 7 to 15 decibels less noisy than the CH-47C over the frequency range of

31.5 to 8,000 hertz. Similar differences in noise measurements at other flight conditions were reported in reference 21, appendix I. Qualitatively, the cockpit noise environment was greatly improved over that found in the CH-47C. As one measure of comparison, the noise of the cockpit heater fan cannot be heard above the normal in-flight background cockpit noises in the CH-47C. By contrast, the noise of the apparently identical heater fan in the Model 347 helicopter could be clearly heard under all flight conditions. The relatively quiet cockpit noise environment improved the ability of the pilot to receive and transmit voice communications and appeared to have contributed to an overall decrease in pilot fatigue. A phenomenon known as "rotor bang," frequently heard in the cockpit of the CH-47C, was not observed in the cockpit of the Model 347 under any flight condition. Opening of the companionway door in flight resulted in a noticeable increase in high-frequency noise transmitted to the cockpit from the untreated aft cabin, but it did not materially degrade the pleasantly quiet cockpit area. The Model 347 cockpit met the sound-level requirements of MIL-A-8806A. The cockpit-noise characteristics are satisfactory for Army use. Measurement of the noise characteristics of the cargo area (cabin) should be accomplished after installation of acoustical treatment in that area.

110. Comparing the exterior sound pressure levels of the Model 347 and CH-47C, as shown in table 8, indicates that the Model 347 has equal or lower sound pressure levels at frequencies below 2,000 hertz and equal or greater sound pressure levels at 2,000 hertz and above. Limited qualitative evaluations of the exterior noise characteristics during engine start and ground operations indicated no significant difference between the CH-47C and Model 347. During observation of two landing approaches of the Model 347, no instance of "rotor bang" was heard.

Engine Characteristics

111. The inlet air source for the auxiliary power unit (APU) was located in the interior of the aircraft, adjacent to the APU. This air inlet location eliminated any restrictions associated with ingestion of main engine exhaust gases and permitted the APU to be operated even when both propulsion engines were operating. This capability permitted the pilot additional flexibility in the start-up and shutdown of the main engines.

112. Separation of the main engine beep trim functions, so that one switch controlled only one engine, aided the pilot in making rotor speed adjustments. The switches were more convenient and easier to use than the combined beep trim switch installed in the CH-47C.

113. The engine condition levers provided proportional control of engine and rotor speed at any control position between GROUND and FLIGHT. This capability reduced the tendency to overtorque the engines when moving the levers to FLIGHT and allowed the pilot to simultaneously move both levers if desired.

114. Because of the nonstandard nature of the engines installed on the test aircraft, a detailed evaluation of the engine characteristics was not performed. Within the scope of this evaluation, however, no objectionable engine characteristics were observed.

CONCLUSIONS

GENERAL

115. The following conclusions were reached upon completion of the technical evaluation of the Model 347 helicopter:

a. Out-of-ground-effect hover and level flight performance are significantly improved over the performance of the CH-47C (paras 15 and 17).

b. Thrust control rod smoothness, reduced sensitivity, and reduced control force characteristics are improvements over the CH-47C and reduce the pilot effort required in making collective changes (para 25).

c. Longitudinal, lateral, and directional beep trim capability aids the pilot in making minor trim corrections (paras 36, 37, and 38).

d. The stable and consistent longitudinal control trim position gradient decreases pilot workload in changing airspeed and contributes to the pilot's ability to quickly and accurately trim at any desired airspeed (para 49).

e. The insensitivity of the aircraft pitch attitude to changes in weight and cg is helpful to the pilot in trimming the aircraft, in that, regardless of loading conditions, it is relatively easy for the pilot to remember the unique pitch attitude, as indicated on the artificial horizon, which is required for the desired trim speed (para 53).

f. The strongly stable and consistent static longitudinal stability characteristics are highly desirable for all pilot tasks and enhance the pilot's ability to operate under IFR conditions (para 55).

g. Compared to the CH-47C, the SAS-OFF static directional stability characteristics are noticeably improved (para 64).

h. The FPCS automatic steering and glide-path tracking modes reduce pilot workload under all conditions and are particularly useful under IFR conditions (para 93).

i. Ground handling characteristics are significantly improved due to the incorporation of the TAXI position in the LCST selector (para 103).

j. Cockpit vibration levels are significantly lower than in the CH-47C (para 105).

k. Internal noise environment in the cockpit area is improved over that found in the CH-47C (para 109).

l. The separate main engine beep trim switches are more convenient and easier to use than the combined beep trim switch installed in the CH-47C (para 112).

m. One deficiency and 12 shortcomings were identified during the evaluation.

DEFICIENCY AND SHORTCOMINGS AFFECTING MISSION ACCOMPLISHMENT

116. Correction of the flight director logic circuitry deficiency, which permitted permitted false and misleading flight-path information to be presented to the pilot, is mandatory for safe aircraft operation under IFR conditions (para 98).

117. Correction of the following shortcomings is desirable for improved operation and mission capabilities.

a. Downward slippage of thrust control rod, which was annoying to the pilot at high engine torque settings (HQRS 4) (para 26).

b. Inadequate side-force characteristic in autorotation, which increased the pilot workload required in trimming the aircraft within satisfactory sideslip limits (HQRS 4) (para 67).

c. Inadequate size of turn needle and inclinometer at pilot station (para 88).

d. Poor location of mode advisory lights at the bottom edge of pilot panel (para 89).

e. Excessive long-term altitude error permitted by the altitude-hold system (para 92).

f. Excessive 1-hertz oscillation of deviation indicator and flight director steering command bar when tracking VOR radial (para 95).

g. Excessive random lateral oscillation of aircraft, in coupled VOR navigation mode, when passing over the VOR transmitter zone of confusion (para 96).

h. Excessive "S-turning" commands generated by the flight director steering command bar when operating on an ILS localizer signal (para 97).

i. Absence of heading synchronization indication on the pilot horizontal situation display (para 99).

j. Continued illumination of VOR/ILS capture light under conditions in which the aircraft is displaced beyond the usable limits of the flight director (para 100).

k. Excessive 4/rev vertical vibration level in the aft portion of the cabin area, between station 500 and station 592 (para 106).

SPECIFICATION CONFORMANCE

118. Within the scope of this test, the stability and control characteristics and vibration characteristics of the Model 347 helicopter failed to meet the following requirements of military specification MIL-H-8501A:

a. Paragraph 3.3.13 - The directional pedal breakout force of 10 pounds to the left and 14 pounds to the right exceeded the 7-pound limit by 3 pounds (42 percent) and 7 pounds (100 perc nt), respectively (para 24).

b. Paragraphs 3.3.11 and 3.3.12 - The limit directional control force of 20 pounds exceeded the 15-pound maximum allowable limit force by 5 pounds or 33 percent (para 24).

c. Paragraph 3.4.2 - The thrust rod limit forces of 10 pounds (UP) and 8 pounds (DOWN) exceeded the maximum allowable 7-pound limit by 3 pounds (43 percent) and 1 pound (14 percent), respectively (para 25).

d. Paragraph 3.4.2 - The thrust control rod slipped downward 0.2 inch after releasing the magnetic brake button and removing all forces from the handle (paras 25 and 26).

e. Paragraph 3.5.4.4 - Following autorotational touchdown at 35 knots, the helicopter would probably require more than the maximum allowable 200-foot ground roll distance to come to a stop (para 79).

f. Paragraph 3.7.1(b) - The maximum cockpit vertical vibration level (0.47g) at the 8/rev frequency exceeded the 0.15g and 0.20g allowable limits by 0.32g (212 percent) and 0.27g (135 percent), respectively (para 105).

119. Within the scope of this test, the cockpit vibration characteristics failed to meet the requirements of the contract work statement, in that the maximum 4/rev vertical vibration (0.11g) exceeded the contract limit (0.05g) by 0.06g (120 percent) and the 8/rev vertical vibration (0.47g) exceeded the contract limit (0.10g) by 0.37g (370 percent) (para 105).

RECOMMENDATIONS

120. The flight director logic circuitry deficiency, correction of which is mandatory, should be corrected prior to release of the helicopter for flight in IFR conditions (para 116).

121. The shortcomings, correction of which is desirable, should be corrected prior to initiation of Phase II testing (para 117).

122. A reliable and properly calibrated boom airspeed system should be installed on the aircraft prior to initiation of Phase II testing (para 74).

123. The following "CAUTION" should be placed in the operator's manual (para 50):

CAUTION

> Following dual DASH system failure at high speed, unusually far forward longitudinal control positions will be required when trimming at lower airspeed. To preclude exceeding the available arm reach of the pilot, do not allow excessively high nose-up pitch rates or attitudes to occur when decelerating.

124. The following "WARNING" should be placed in the operator's manual (para 84):

WARNING

> Intentional operation into known instrument-flight-rule (IFR) conditions is not recommended unless both stability augmentation systems are operating properly prior to entry into such flight conditions. In the event that one SAS is inoperative, failure of the remaining SAS while flying in IFR conditions may result in loss of aircraft control.

125. Measurement of cargo-area (cabin) noise characteristics should be accomplished after installation of sound attenuation treatment in that area (para 109).

APPENDIX I. REFERENCES

1. Letter, AVSCOM, 15 October 1970, subject: Test Directive, Evaluation of Boeing-Vertol Model 347.

2. Test Plan, USAASTA, Project No. 70-38, *Army Technical Evaluation, Boeing-Vertol Model 347 Helicopter*, May 1971.

3. Message, AVSCOM, AMSAV-R-F, 26 May 71, Unclas, subject: Test Plan of the Boeing-Vertol Model 347.

4. Report, The Boeing Company, Vertol Division, D8-2240-Z, *Model 347 Flight Development Proposal*, Revision A, April 1969.

5. Contract, Department of the Army, DAAJ01-70-A-0320(2M), "Flight Research Program," undated.

6. Report, The Boeing Company, Vertol Division, D347-10072-1, *Model 347 Flight Research Helicopter Performance, Stability, and Control Predictions*, May 1970.

7. Report, The Boeing Company, Vertol Division, D347-10065-1, *Flight Path Control System for Model 347 Flight Research Aircraft -- Design Report*, February 1970.

8. Report, The Boeing Company, Vertol Division, D347-10068-1, *Flight Demonstrator Flight Test Procedure*, March 1970.

9. Report, The Boeing Company, Vertol Division, D8-2238-1, *Model 347 Demonstrator -- Design Criteria and Requirements*, July 1969.

10. Report, The Boeing Company, Vertol Division, D347-10063-1, *Model 347 Demonstrator Variable Force Feel System*, February 1970.

11. Letter, AVSCOM, AMSAV-R-F, Subject: Safety of Flight Release for Boeing-Vertol Model 347, Project No. 70-38, 26 May 1971.

12. Interoffice Memorandum, The Boeing Company, Vertol Division, Reference Number 8-7586-1-85, "Corrective Action for Discrepancies in Flight Controls," 19 July 1971.

13. Letter, USAASTA, September 1971, subject: Reevaluation of Boeing-Vertol Model 347 Helicopter, Project No. 70-38.

14. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities, General Requirements For*, 7 September 1961, with Amendment 1, 3 April 1962.

15. Military Specification, MIL-A-8806A, *Acoustical Noise Level in Aircraft, General Specification For*, 11 July 1966, with Amendment 1, 12 September 1967.

16. Final Report, USAASTA, Project No. 66-29, *Airworthiness and Flight Characteristics Test, CH-47C Helicopter*, September 1971.

17. Final Report, USAAVNTA, Project No. 66-28, *Engineering Flight Test of the YCH-47C Medium Transport Helicopter, Army Preliminary Evaluation II*, February 1969.

18. Final Report, USAASTA, Project No. 66-28, *Engineering Flight Test of the YCH-47C Medium Transport Helicopter, Army Preliminary Evaluations III and IV*, October 1970.

19. Flight Test Manual, Naval Air Test Center, FTM No. 101, *Helicopter Stability and Control*, 10 June 1968.

20. Flight Test Manual, Naval Air Test Center, FTM No. 102, *Helicopter Performance Testing*, 28 June 1968.

21. Report, US Army Aeromedical Research Laboratory, USAARL-LR-72-2-2-2, *Noise Spectra of the Boeing-Vertol Model 347 Helicopter*, 1 September 1971.

APPENDIX II. TEST DATA

INDEX

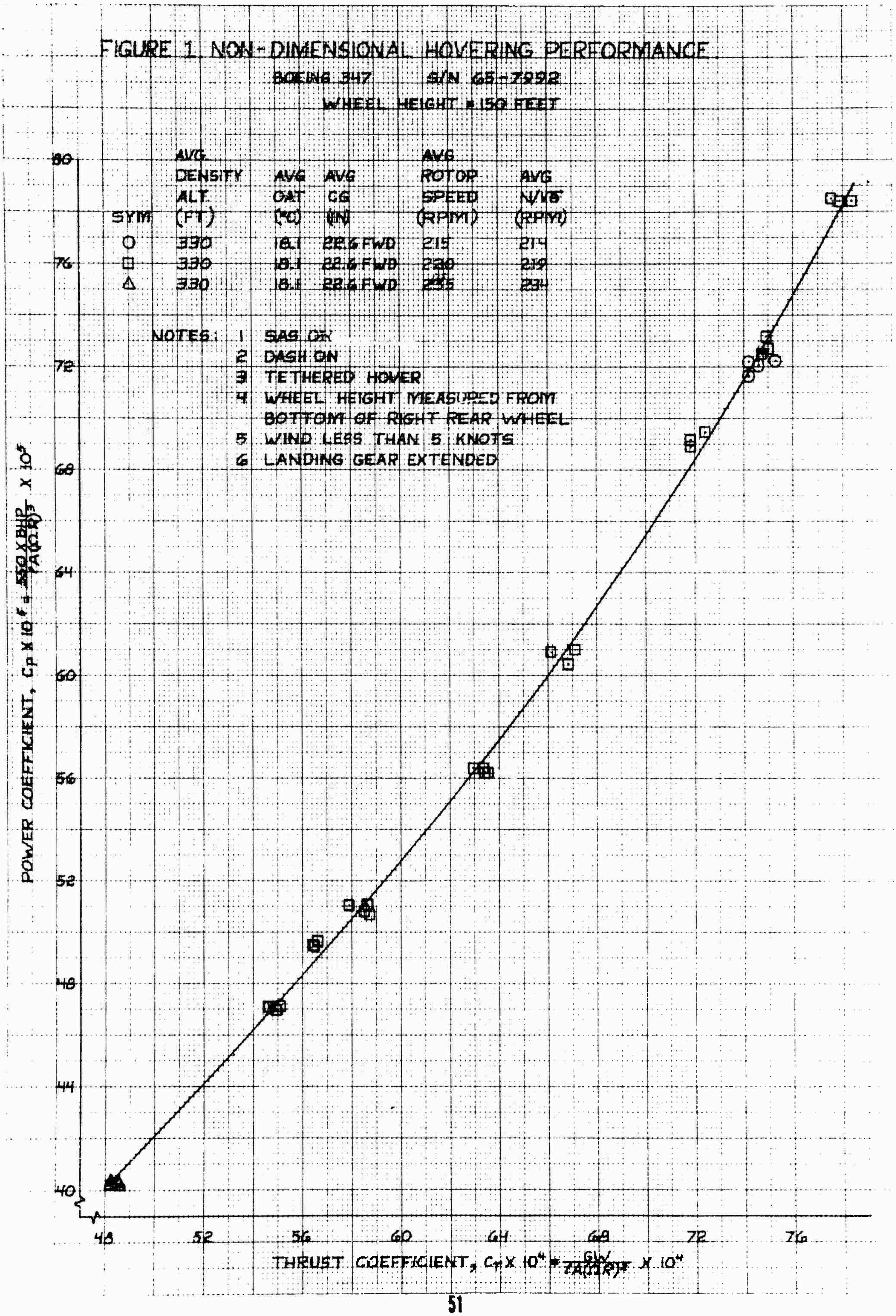


FIGURE 1 NON-DIMENSIONAL HOVERING PERFORMANCE
BOEING 347 S/N 65-7992
WHEEL HEIGHT = 150 FEET
SYM
AVG DENSITY ALT (FT)
AVG OAT (°C)
AVG CG (IN)
AVG ROTOR SPEED (RPM)
AVG N/√θ (RPM)
○ 330 18.1 22.6 FWD 215 214
⊡ 330 18.1 22.6 FWD 230 219
△ 330 18.1 22.6 FWD 235 234
NOTES: 1 SAS ON
2 DASH ON
3 TETHERED HOVER
4 WHEEL HEIGHT MEASURED FROM BOTTOM OF RIGHT REAR WHEEL
5 WIND LESS THAN 5 KNOTS
6 LANDING GEAR EXTENDED
POWER COEFFICIENT, $C_P \times 10^5 = \frac{550 \times BHP}{\rho A(\Omega R)^3} \times 10^5$
THRUST COEFFICIENT, $C_T \times 10^4 = \frac{GW}{\rho A(\Omega R)^2} \times 10^4$
80
76
72
68
64
60
56
52
48
44
40
48
52
56
60
64
68
72
76

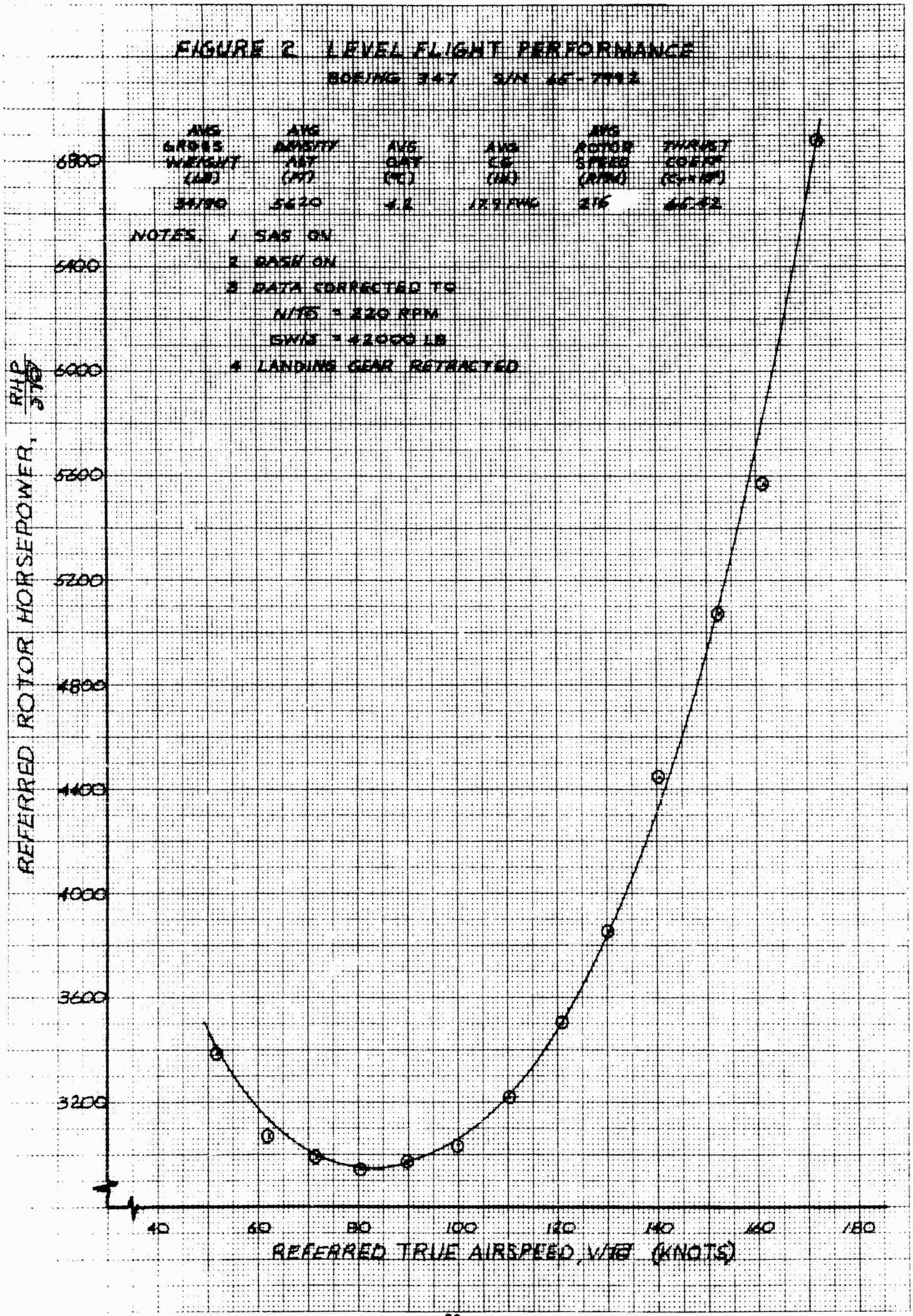


FIGURE 2 LEVEL FLIGHT PERFORMANCE
BOEING 347
AVG GROSS WEIGHT (LB)
AVG DENSITY ALT (FT)
AVG OAT (°C)
AVG CG (IN)
AVG ROTOR SPEED (RPM)
THRUST COEFF
5620
4.2
17.9 FWD
216
NOTES. 1 SAS ON
2 DASH ON
3 DATA CORRECTED TO
= 220 RPM
= 42000 LB
4 LANDING GEAR RETRACTED
REFERRED ROTOR HORSEPOWER
6800
6400
6000
5600
5200
4800
4400
4000
3600
3200
40
60
80
100
120
140
160
180
REFERRED TRUE AIRSPEED (KNOTS)

FIGURE 3 LEVEL FLIGHT PERFORMANCE

BOEING 347 S/N

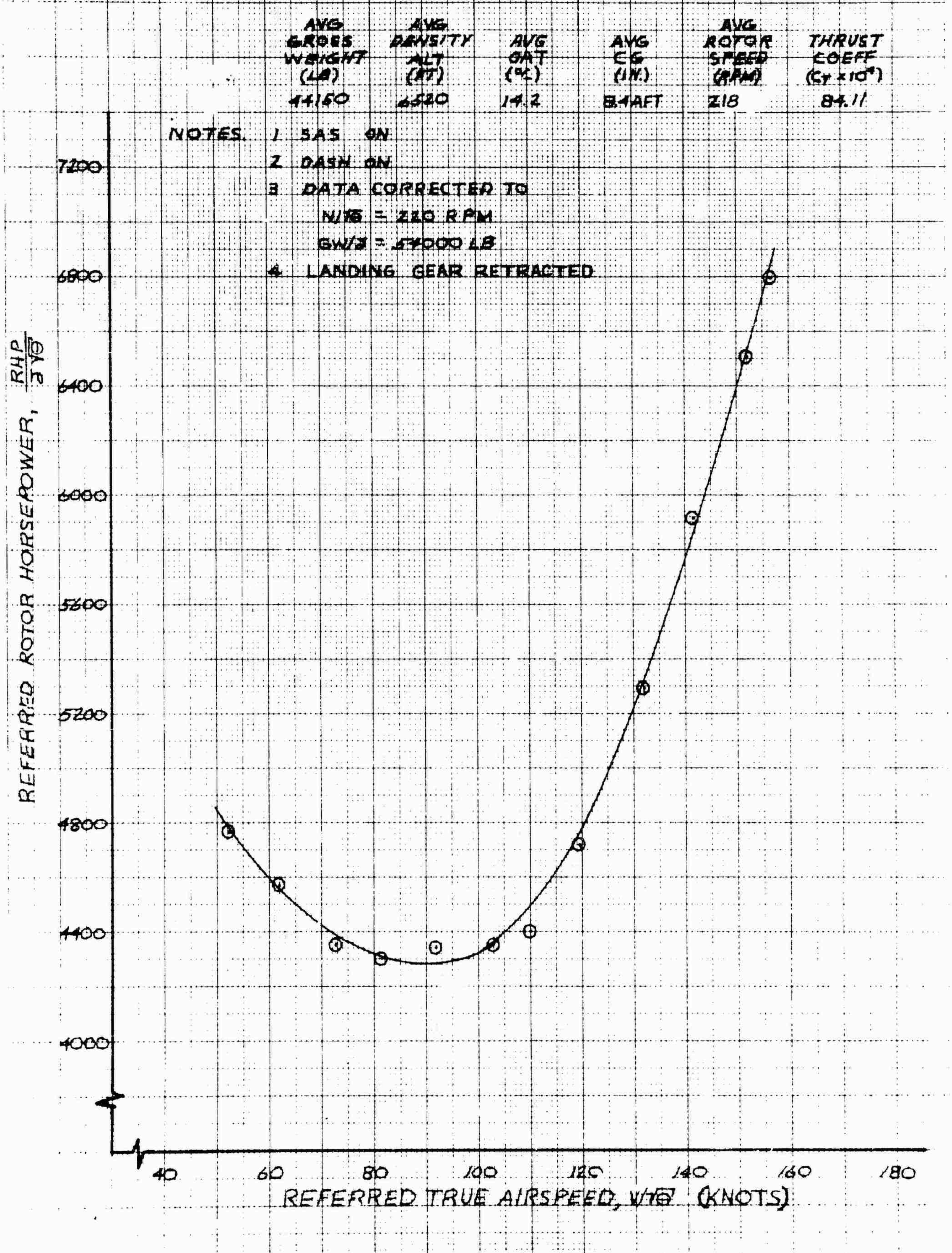


FIGURE 4. CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LONGITUDINAL CONTROL TRAVEL = 15.05 IN.
3 AIRSPEED SIGNAL WAS ZERO KIAS
4 SHADED SYMBOL DENOTE START POINT

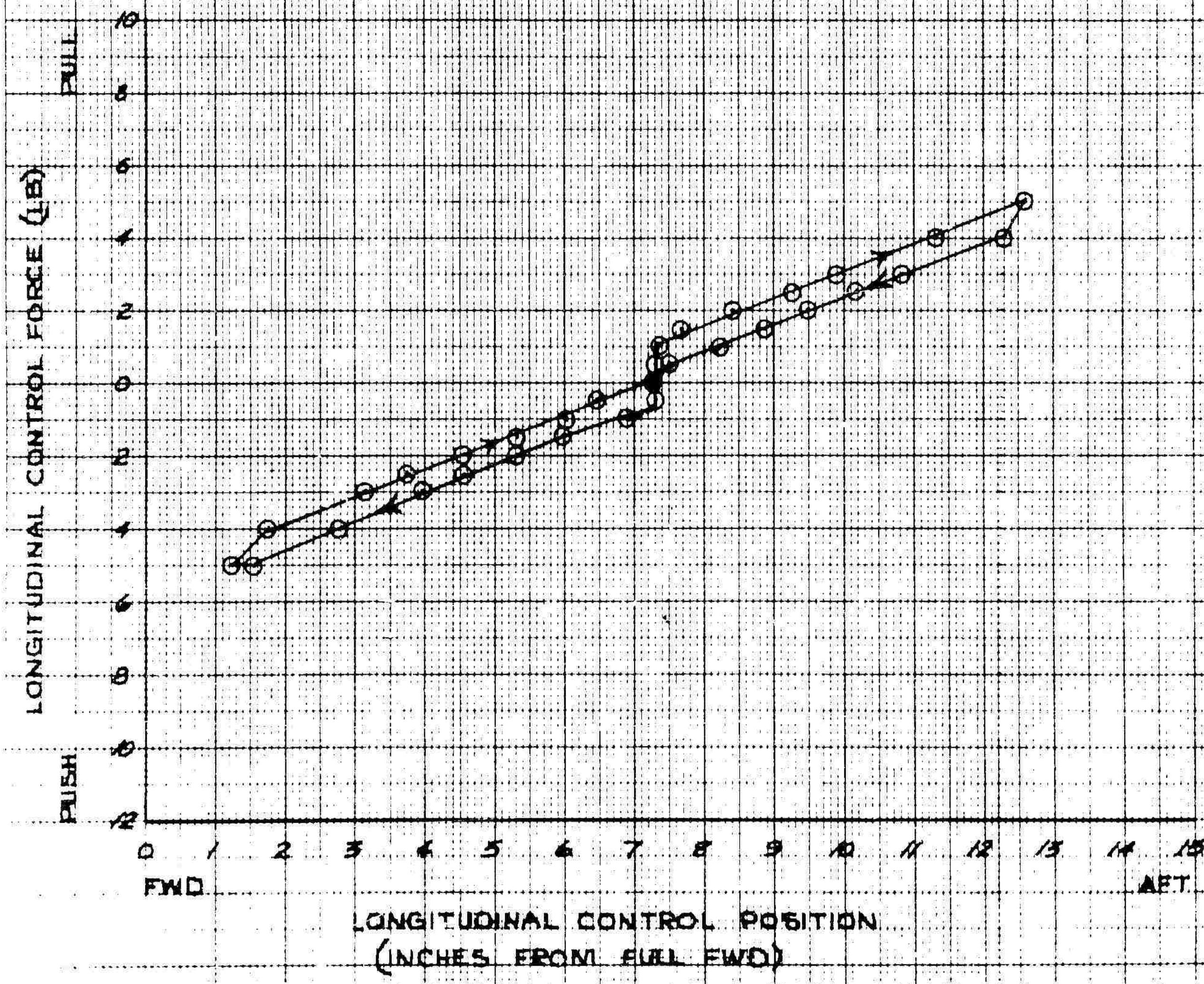


FIGURE 5 CONTROL SYSTEM CHARACTERISTICS

BOEING 347 S/N 65-7992

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LONGITUDINAL CONTROL TRAVEL = 15.05 IN.
3 AIRSPEED SIGNAL WAS 100 KIAS
4 SHADED SYMBOL DENOTE START POINT

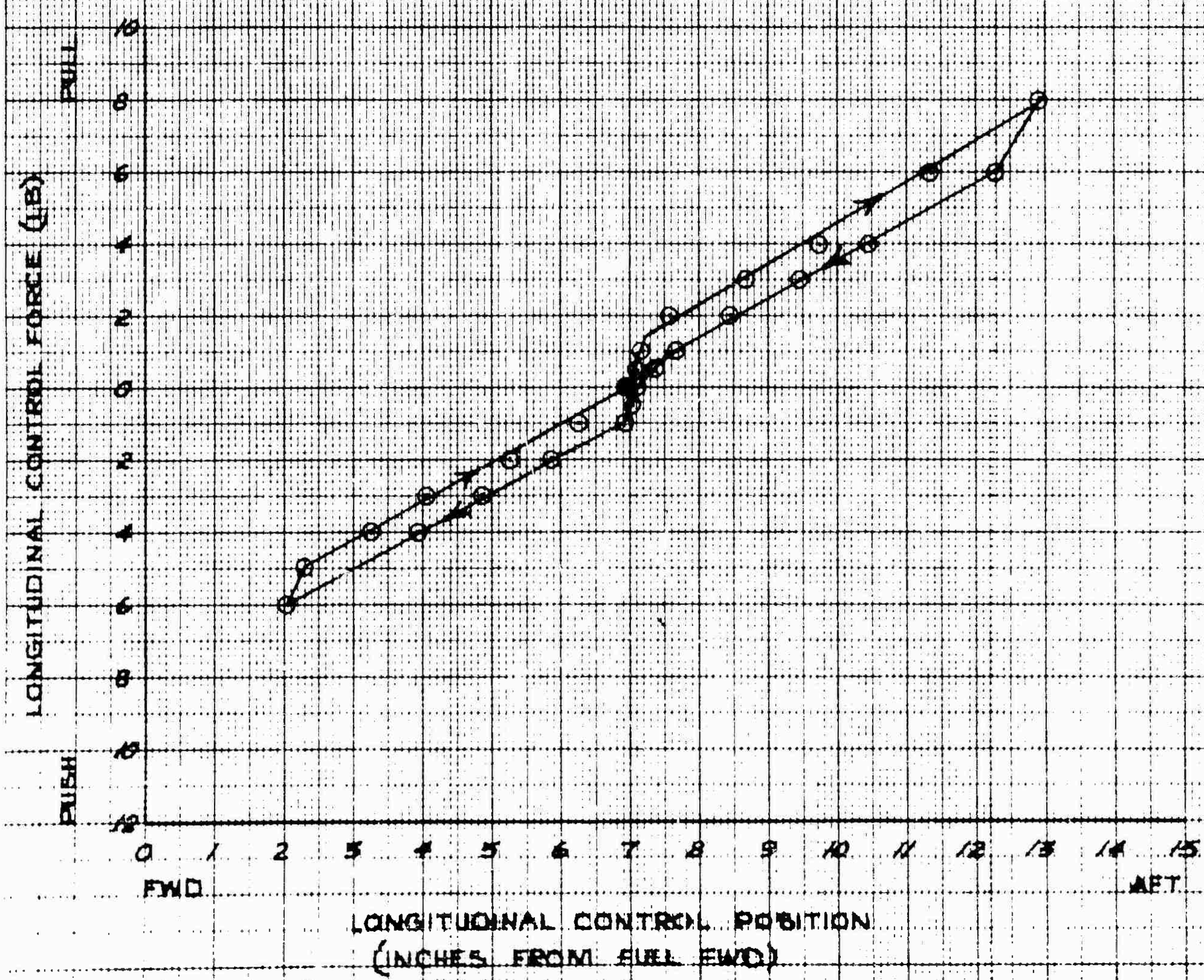


FIGURE 6 CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

NOTES: 1 TEST CONDUCTED ON GROUND EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LONGITUDINAL CONTROL TRAVEL = 15.05 IN.
3 AIRSPEED SIGNAL WAS 170 KIAS
4 SHADED SYMBOL DENOTE START POINT

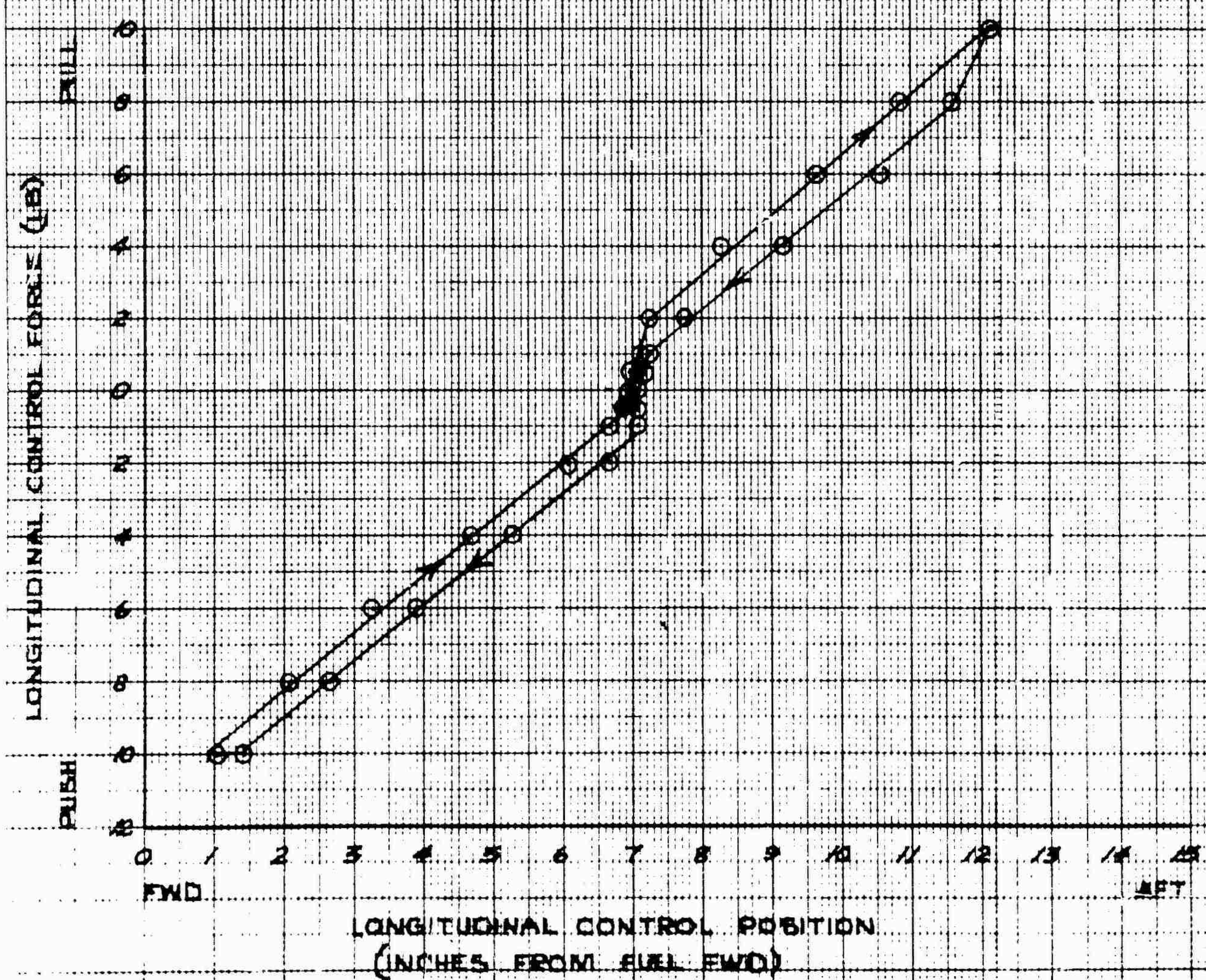


FIGURE 7 CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LATERAL CONTROL TRAVEL = 9.10 IN.
3 AIRSPEED SIGNAL WAS ZERO KIAS
4 SHADED SYMBOL DENOTE START POINT

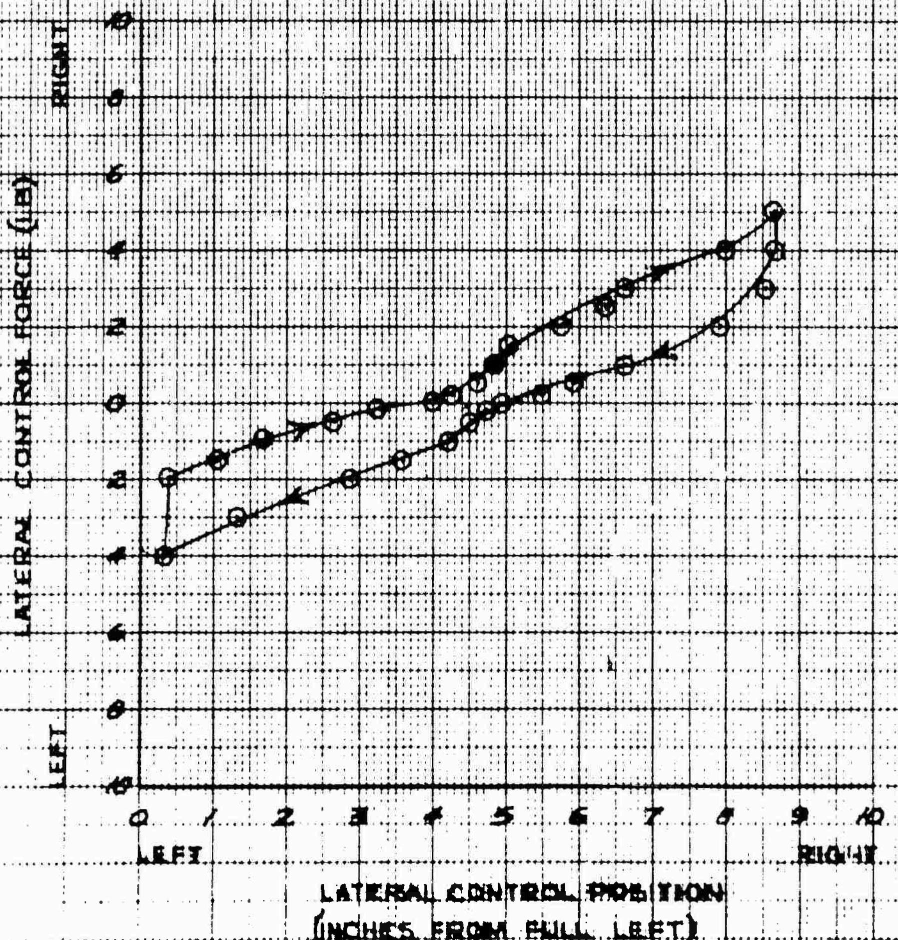


FIGURE 8 CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LATERAL CONTROL TRAVEL = 9.10 IN.
3 AIRSPEED SIGNAL WAS 100 KIAS
4 SHADED SYMBOL DENOTE START POINT

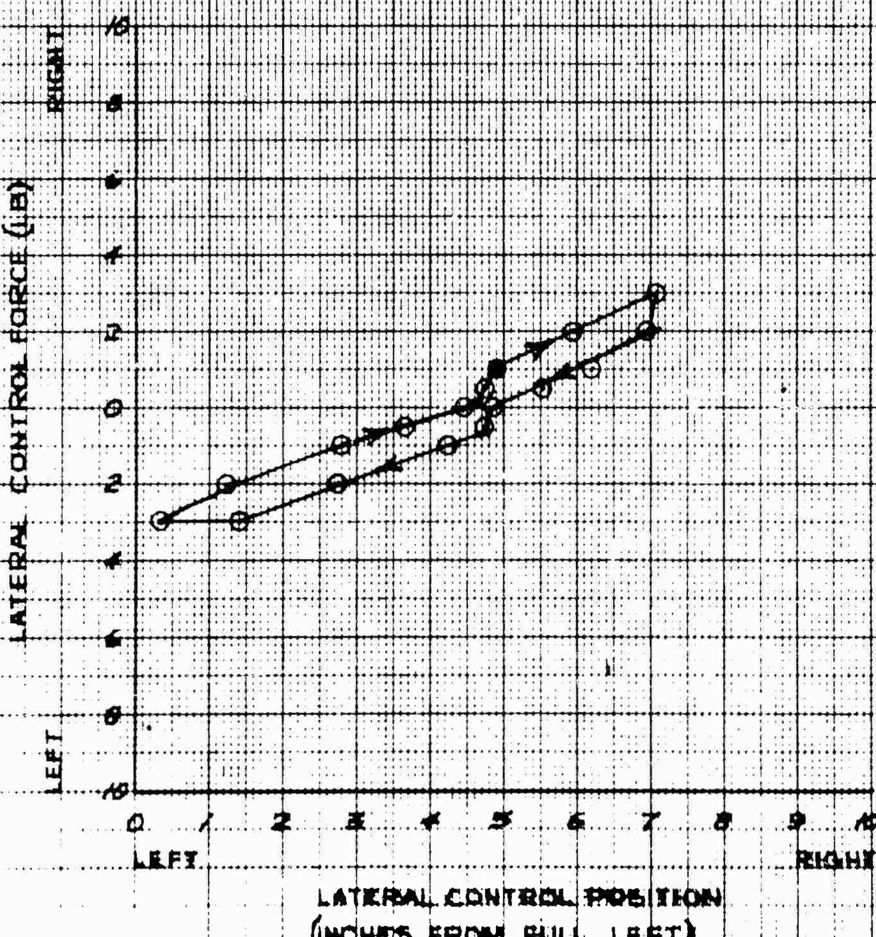


FIGURE 9 CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LATERAL CONTROL TRAVEL = 9.0 IN.
3 AIRSPEED SIGNAL WAS 170.0 KIAS
4 SHADED SYMBOL DENOTE START POINT

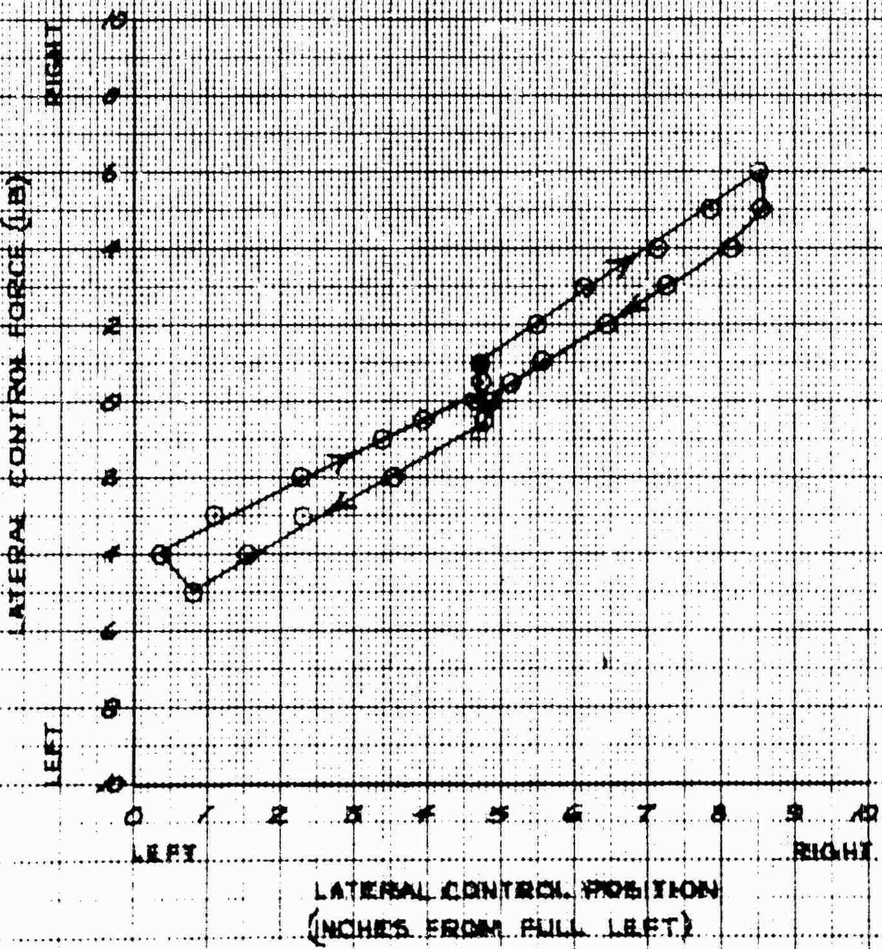


FIGURE 10 CONTROL SYSTEM CHARACTERISTICS.
BOEING 347 S/N 65-7992
REEVALUATION

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LATERAL CONTROL TRAVEL = 9.10 IN.
3 AIRSPEED SIGNAL WAS ZERO KIAS
4 SHADED SYMBOL DENOTE START POINT

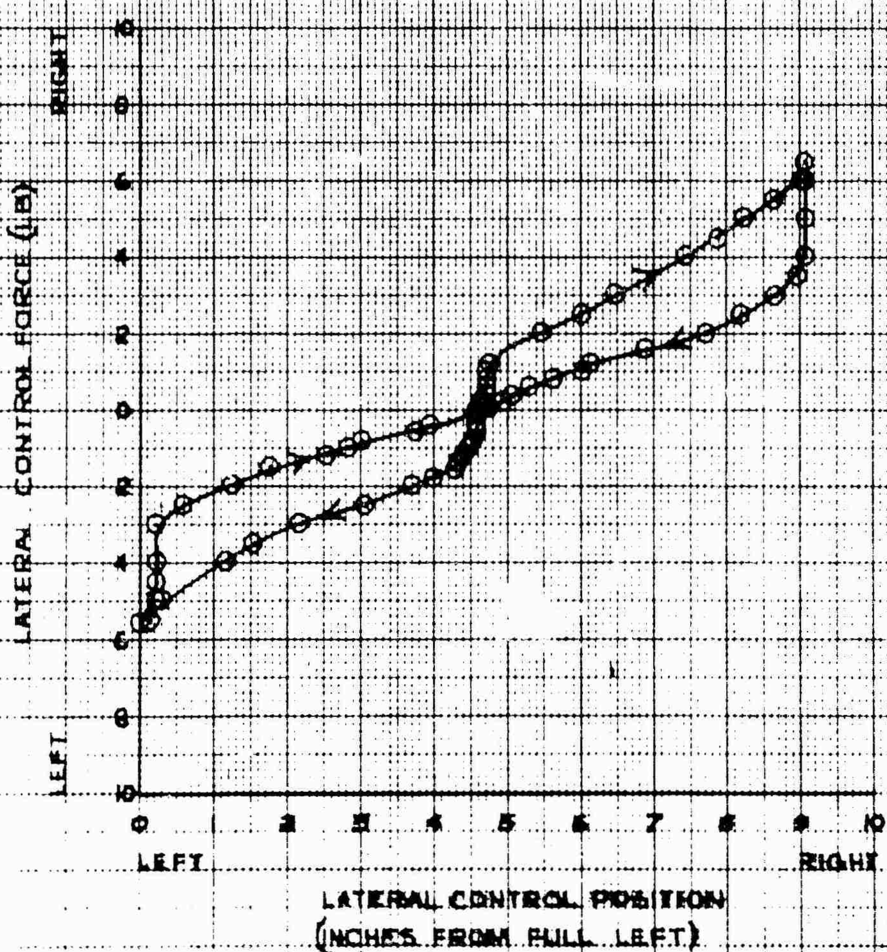


FIGURE 11 CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

REEVALUATION

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER.
2 TOTAL LATERAL CONTROL TRAVEL = 9.0 IN.
3 AIRSPEED SIGNAL WAS 100 KIAS
4 SHADED SYMBOL DENOTE START POINT

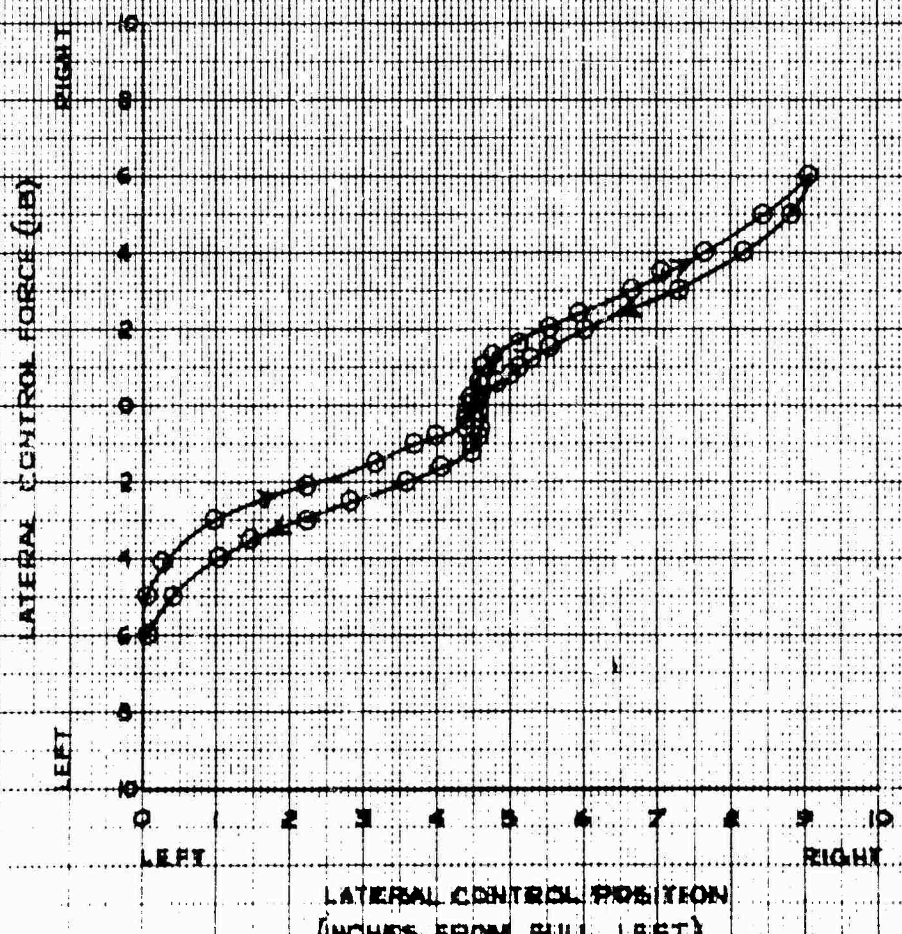
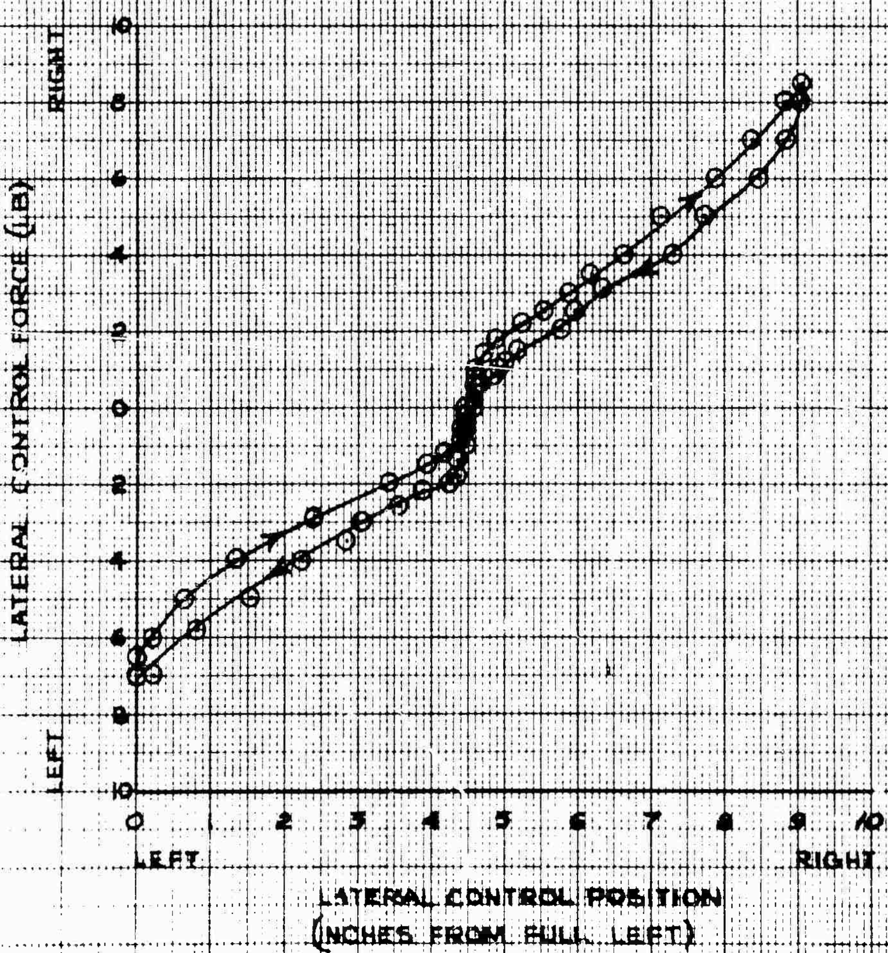


FIGURE 12 CONTROL SYSTEM CHARACTERISTICS.
BOEING 347 S/N 65-7992
REEVALUATION

NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL LATERAL CONTROL TRAVEL = 9.10 IN.
3 AIRSPEED SIGNAL WAS 170 KIAS
4 SHADED SYMBOL DENOTE START POINT



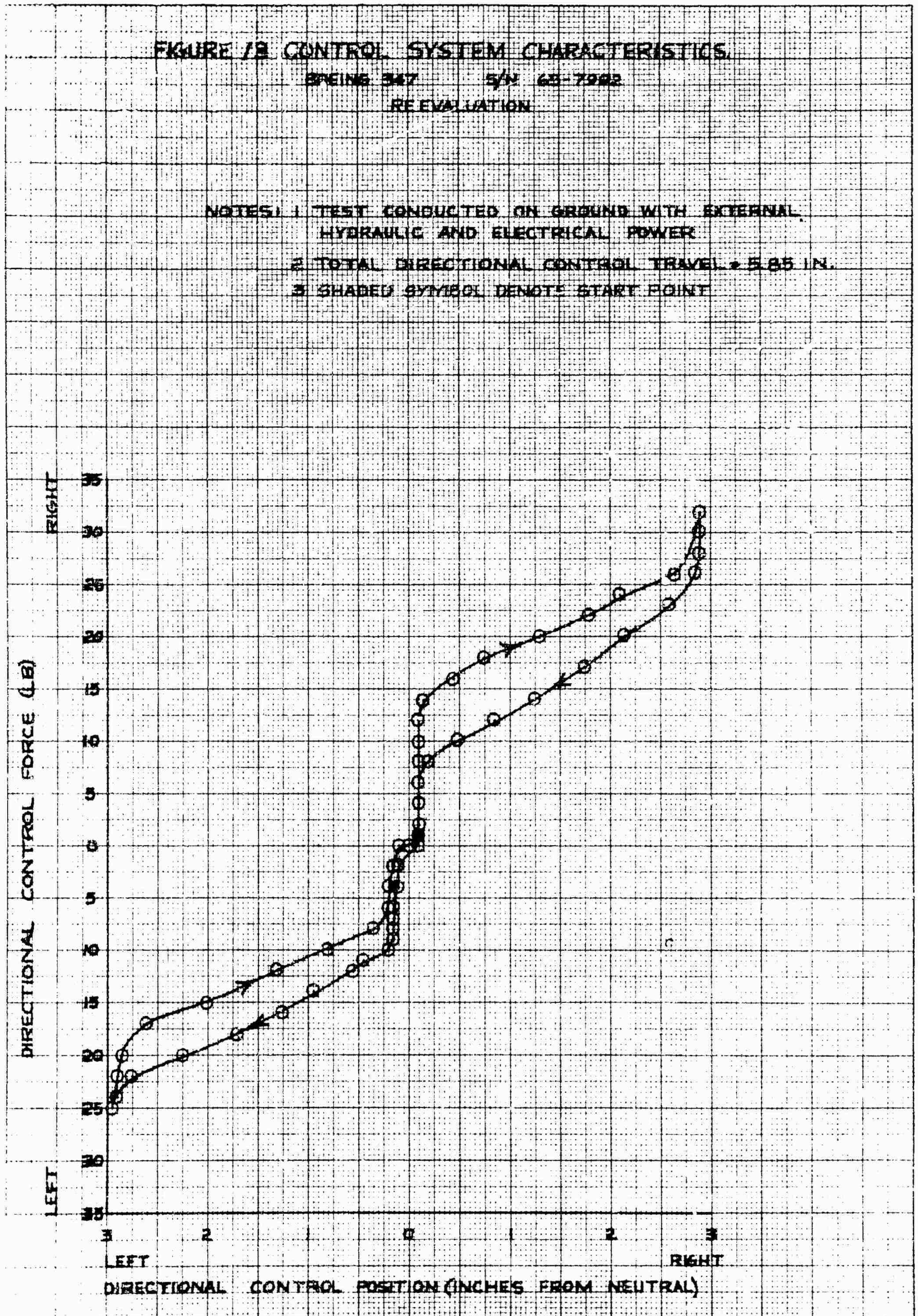


FIGURE 19 CONTROL SYSTEM CHARACTERISTICS
BOEING 347 S/N 65-7992
REEVALUATION
NOTES: 1 TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2 TOTAL DIRECTIONAL CONTROL TRAVEL = 5.85 IN.
3 SHADED SYMBOL DENOTE START POINT
DIRECTIONAL CONTROL FORCE (LB)
RIGHT
LEFT
35
30
25
20
15
10
5
0
5
10
15
20
25
30
35
3
2
1
0
1
2
3
LEFT
RIGHT
DIRECTIONAL CONTROL POSITION (INCHES FROM NEUTRAL)

FIGURE 18. CONTROL SYSTEM CHARACTERISTICS.

BOEING 347 S/N 65-7992

NOTES: 1. TEST CONDUCTED ON GROUND WITH EXTERNAL HYDRAULIC AND ELECTRICAL POWER
2. TOTAL THRUST ROD TRAVEL = 9.60 IN.

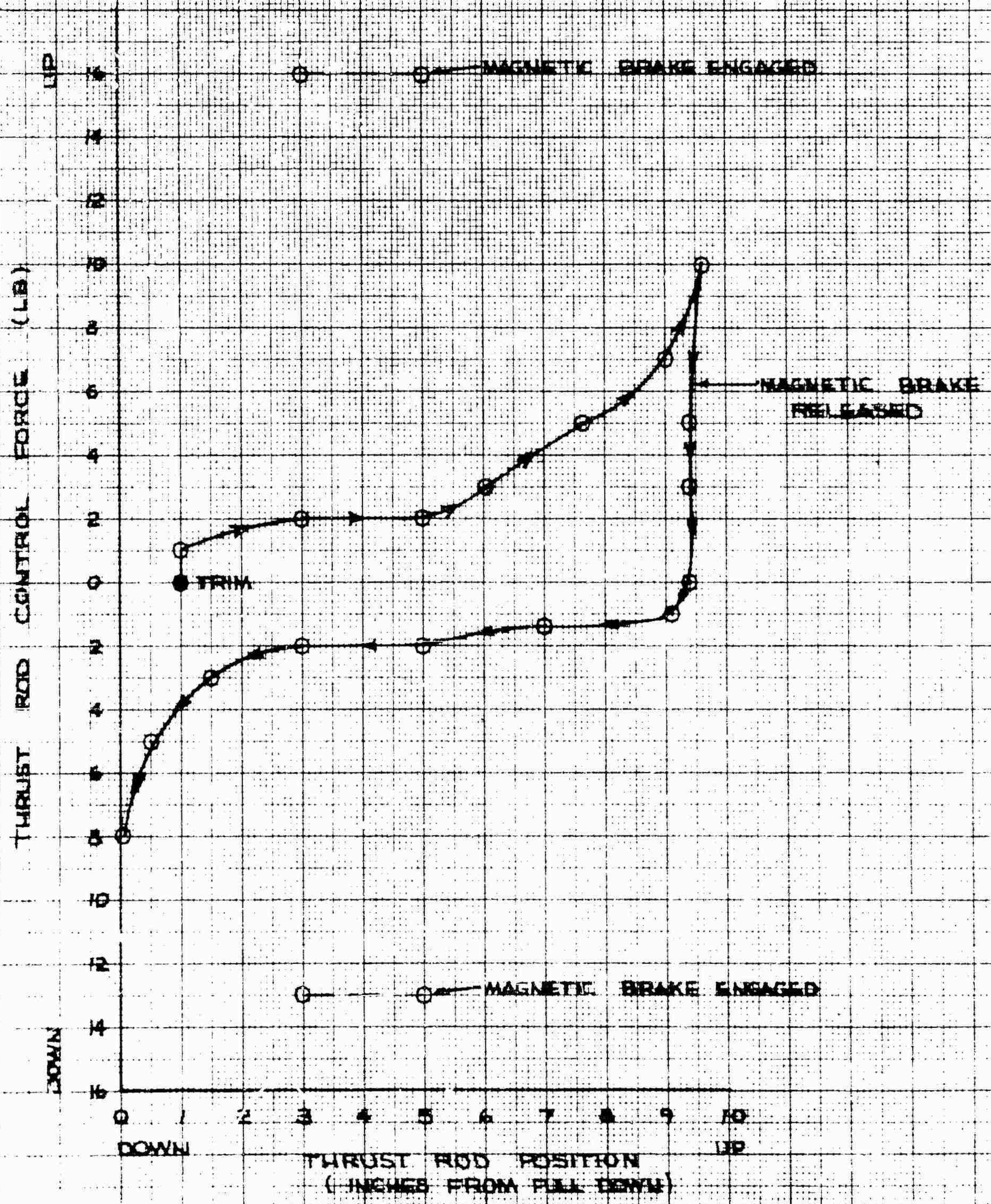


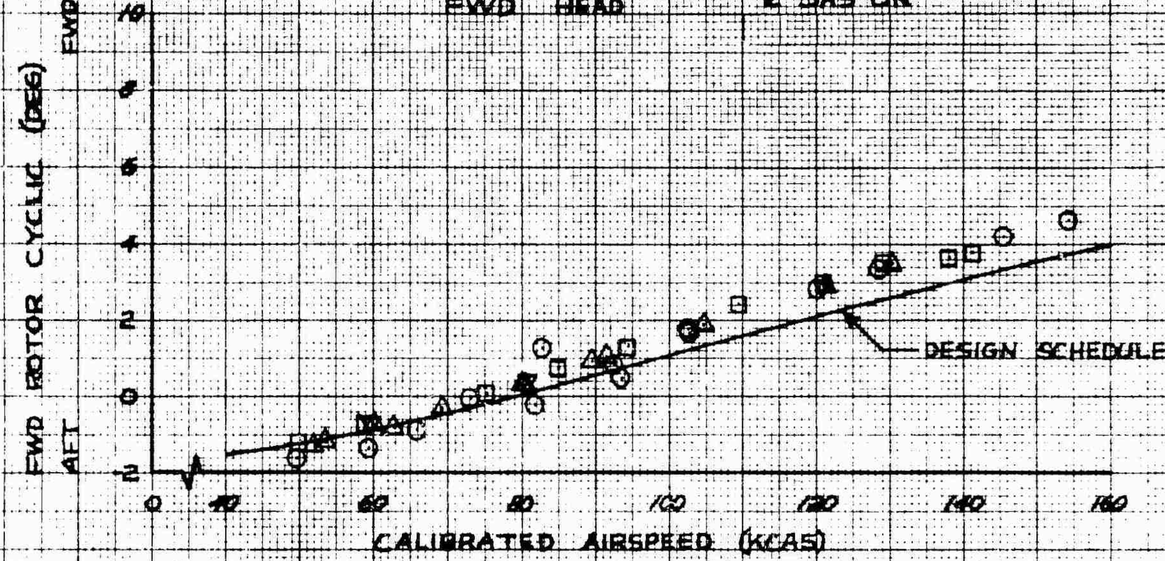
FIGURE 15 LONGITUDINAL CYCLIC SPEED TRIM SCHEDULE.

BOEING 347 S/N 65-7992

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF (C_T X 10^4)
⊙	34210	5620	4.2	17.0 FWD	217	65.42
⊞	44280	6520	14.2	8.4 AFT	219	84.11
△	36325	2000	17.0	2.0 FWD	220	60.01
▽	41290	330	18.1	22.4 FWD	220	64.81

NOTE: 1 DASH ON
2 SAS ON

FWD HEAD



AFT HEAD

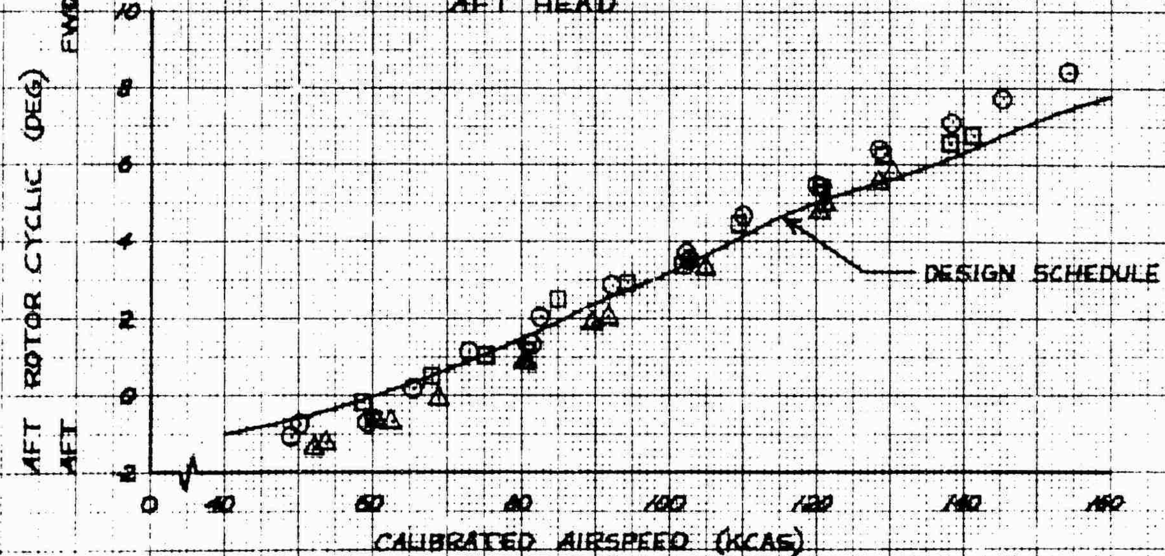


FIGURE 16. LONGITUDINAL CONTROLLABILITY
BOEING 347 S/N 65-7992
HOVER OGE

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF (C_Tx10^4)	TRIM A/S (KCAS)	AVG TIME TO MAX RATE (SEC)	AVG TIME TO MAX ACCEL (SEC)
45500	1520	29.3	8.1 AFT	220	74.54	0	1.20	0.45

NOTES: 1 SAS ON
2 DASH ON
3 LANDING GEAR EXTENDED

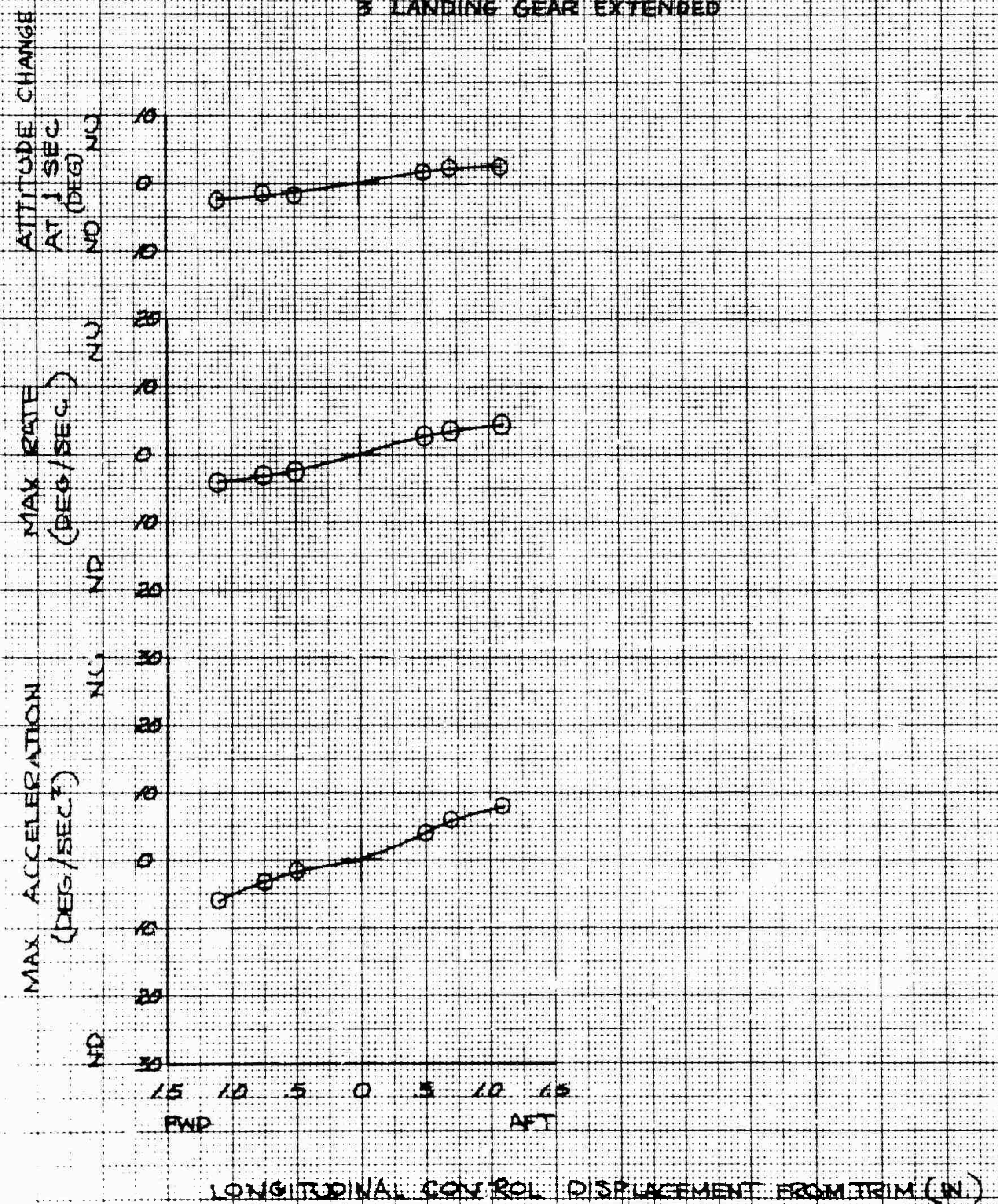


FIGURE 12. LONGITUDINAL CONTROLLABILITY
BOEING 347 S/N 65-7992
LEVEL FLIGHT

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF (C_Tx10⁴)	TRIM A/S (KCAS)	AVG TIME TO MAX RATE (SEC)	AVG TIME TO MAX ACCEL (SEC)
○	45670	5220	21.7	8.0 AFT	220	83.28	79	1.1	0.35
□	44970	5810	21.4	7.7 AFT	220	83.23	112	1.1	0.35
△	44150	5700	22.6	8.4 AFT	220	81.32	137	1.0	0.30

NOTES: 1. SAS ON
2. DASH ON
3. LANDING GEAR RETRACTED

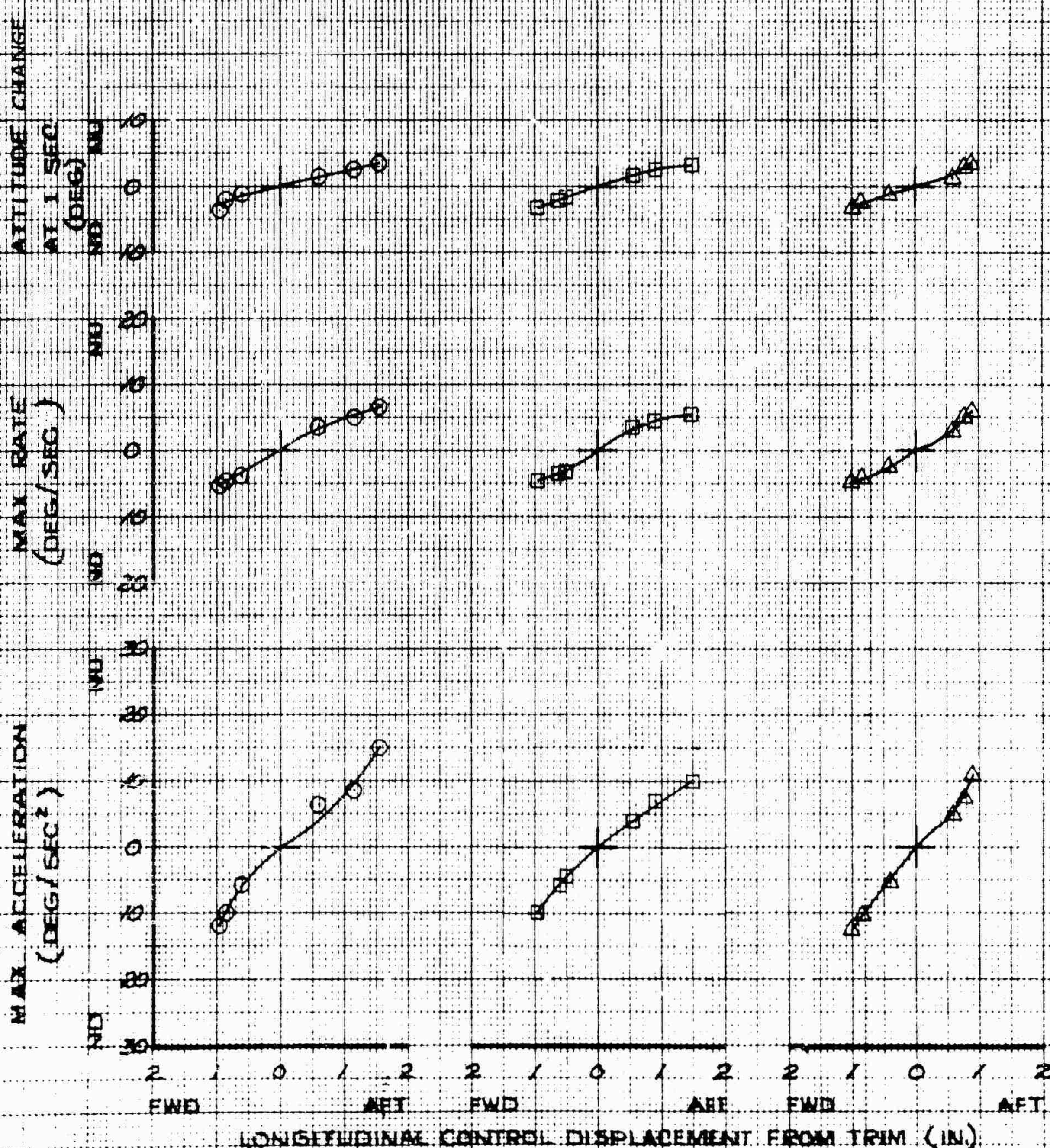


FIGURE 18 LATERAL CONTROLLABILITY

BOEING 347 S/N [illegible]

HOVER OGE

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST LEVER ([illegible])	TRIM A/S (KNOTS)	AVG TIME TO MAX RATE (SEC)	AVG TIME TO MAX ACCEL (SEC)
[illegible]	[illegible]	22.3	[illegible] AFT	220	74.54	0	0.40	0.15

NOTES — 1. SAS ON
2. DASH ON
3. LANDING GEAR EXTENDED

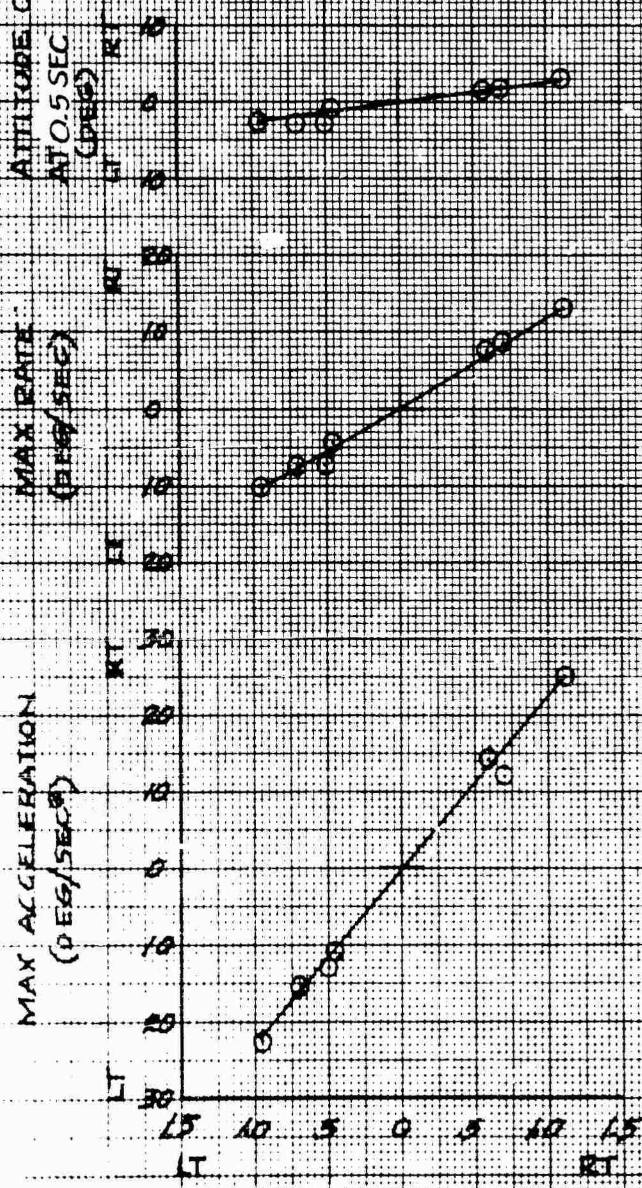


FIGURE 12. LATERAL CONTROLLABILITY

BOEING 347 S/N 65-7992

LEVEL FLIGHT

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	AVG TIME TO MAX RATE (SEC)	AVG TIME TO MAX ACCEL (SEC)
○	45530	5020	22.5	7.9 AFT	220	82.21	85	0.95	0.40
□	44760	5400	22.6	7.6 AFT	220	82.03	112	0.80	0.45
△	43900	5320	23.7	8.8 AFT	220	80.06	131	0.70	0.45

NOTES: 1 SAS ON
2 DASH ON
3 LANDING GEAR RETRACTED

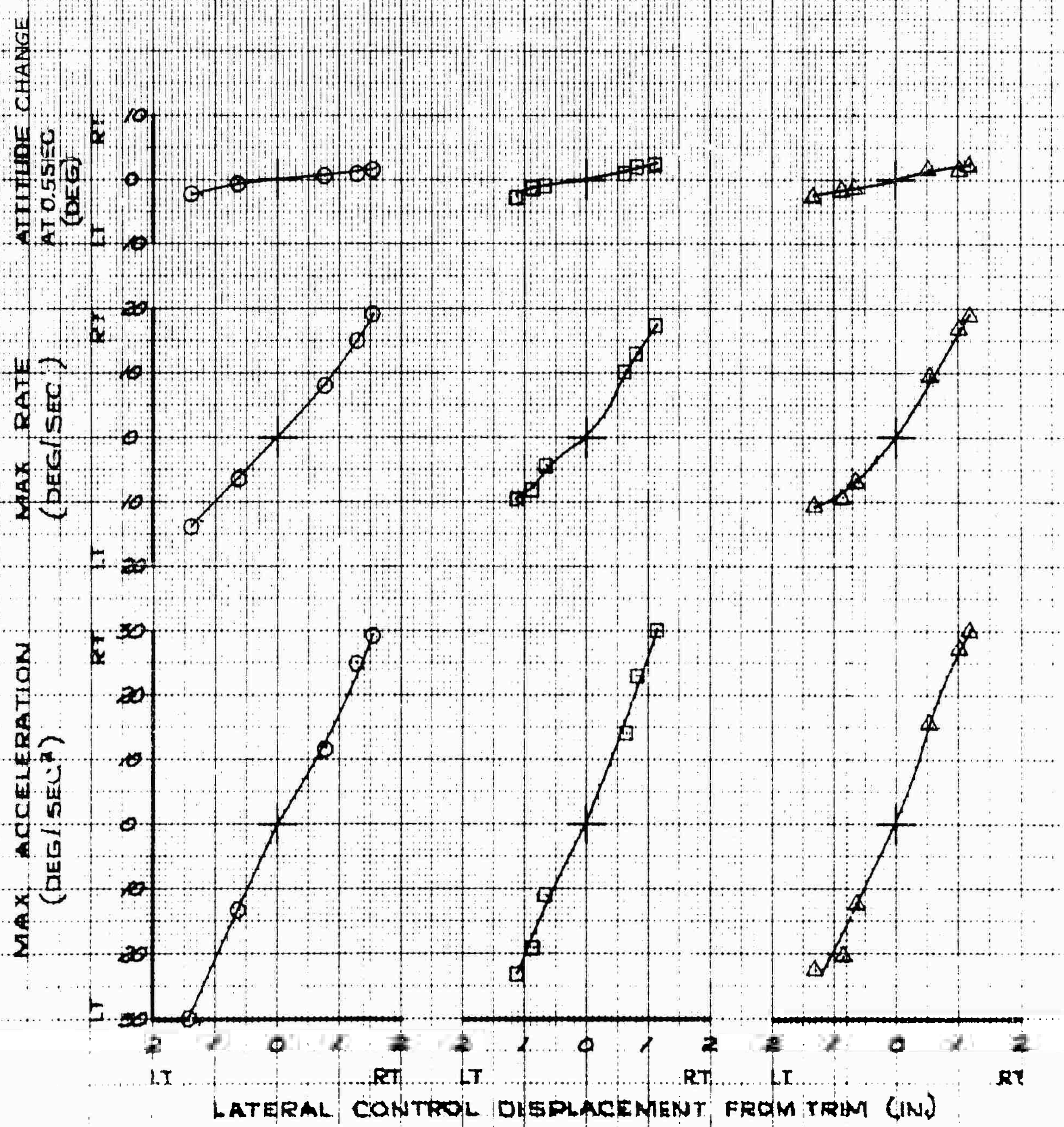


FIGURE 20 DIRECTIONAL CONTROLLABILITY

BOEING 347 S/N 65-7992

HOVER OGE

REEVALUATION

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	AVG TIME TO MAX RATE (SEC)	AVG TIME TO MAX ACCEL (SEC)
42900	1465	24.5	10.8 AFT	220	78.41	0	1.65	0.75

NOTES: 1 DASH ON
2 SAS ON
3 LANDING GEAR EXTENDED

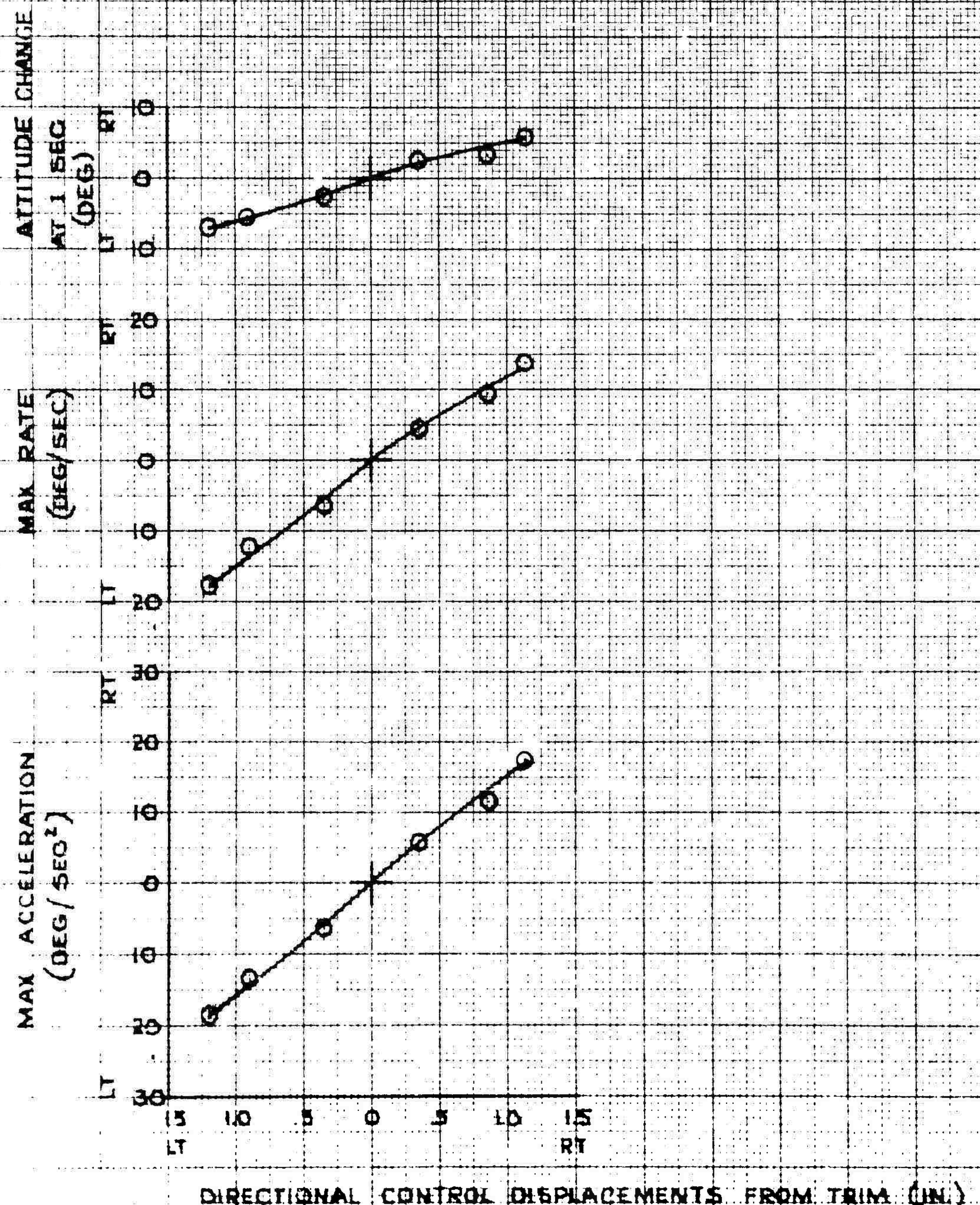


FIGURE 21 DIRECTIONAL CONTROLLABILITY

BOEING 347 S/N 65-7992

LEVEL FLIGHT

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	AVG TIME TO MAX RATE (SEC)	AVG TIME TO MAX ACCEL (SEC)
○	45270	5240	21.5	7.6 AFT	220	82.55	75	1.5	.55
□	44520	5390	22.7	7.7 AFT	220	81.60	113	1.5	.50
△	43540	5420	23.2	9.5 AFT	220	77.80	131	1.7	.60

NOTES 1 SAS ON
2 DASH ON
3 LANDING GEAR RETRACTED

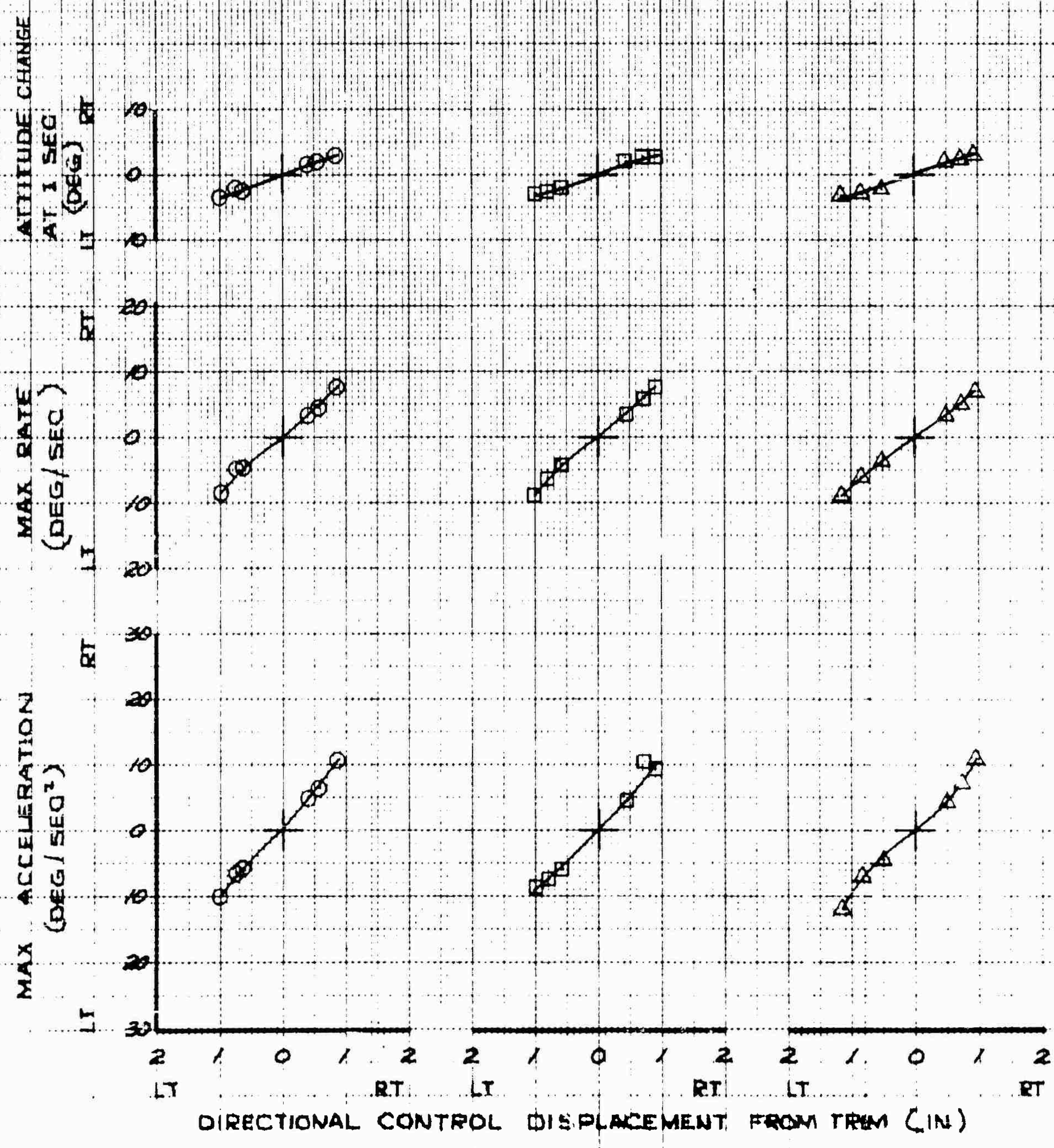


FIGURE 22 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL STEP

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF (C_T x 10^4)	TRIM A/S (KCAS)
44980	10540	10.0	9 AFT	221	96.25	78

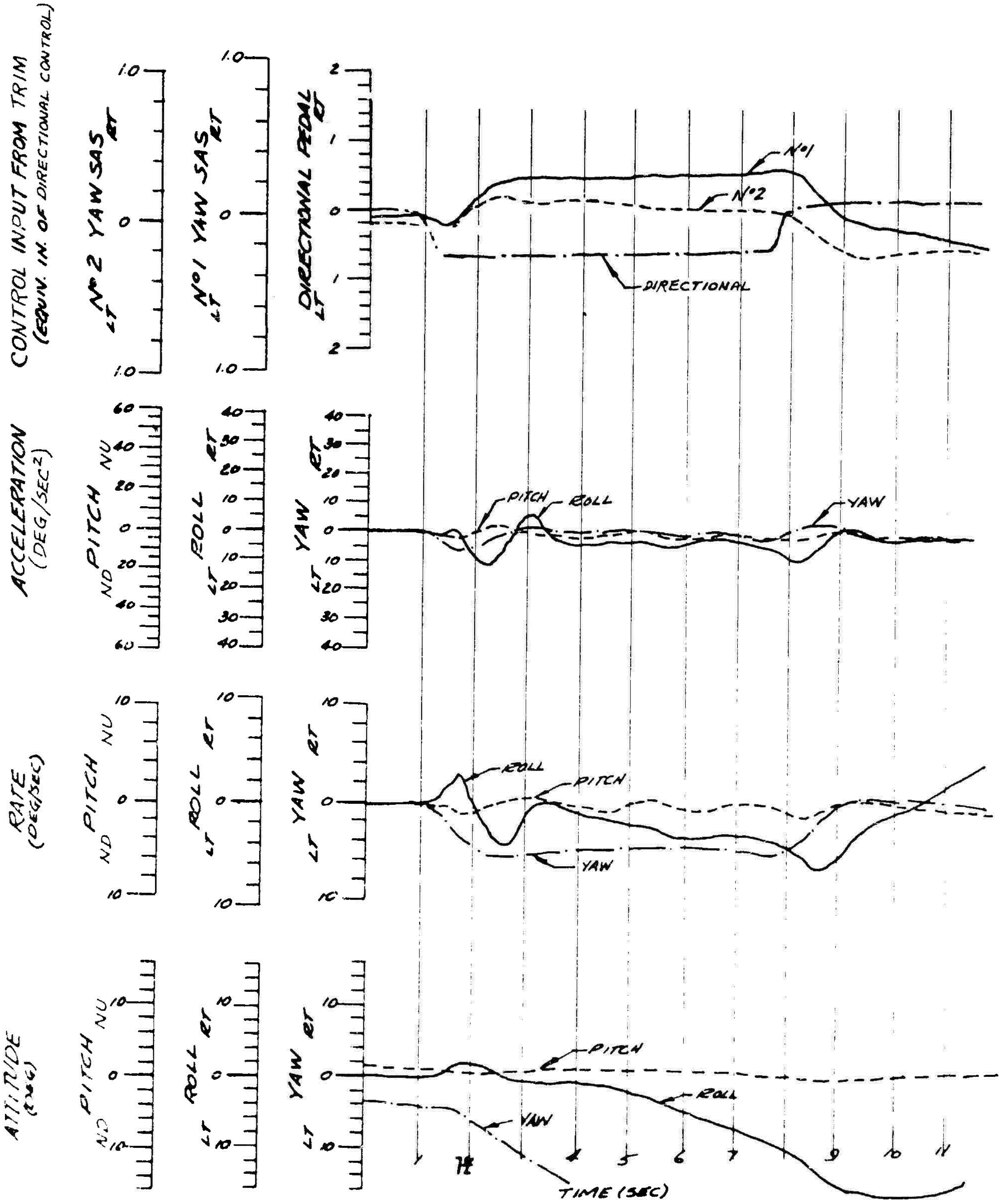


FIGURE 23 AIRCRAFT RESPONSE FOLLOWING RIGHT LATERAL TRIM INPUT
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
40180	2040	13.0	2 AFT	220	66.45	110

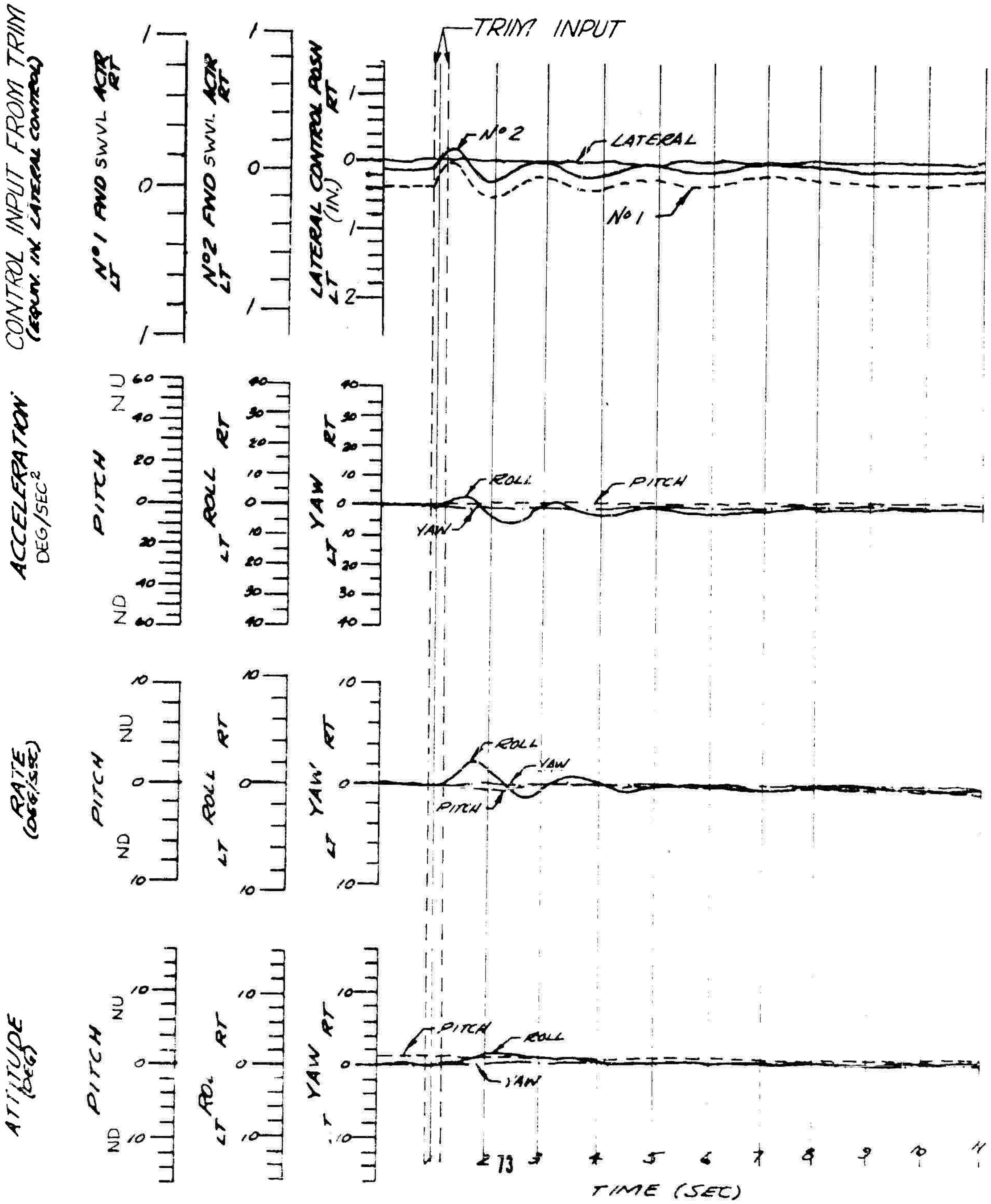


FIGURE 24 AIRCRAFT RESPONSE FOLLOWING RIGHT LATERAL TRIM INPUT

BOEING 347 S/N 65-7992

REEVALUATION

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
34400	8960	14.9	17.7 FWD	219	70.97	109

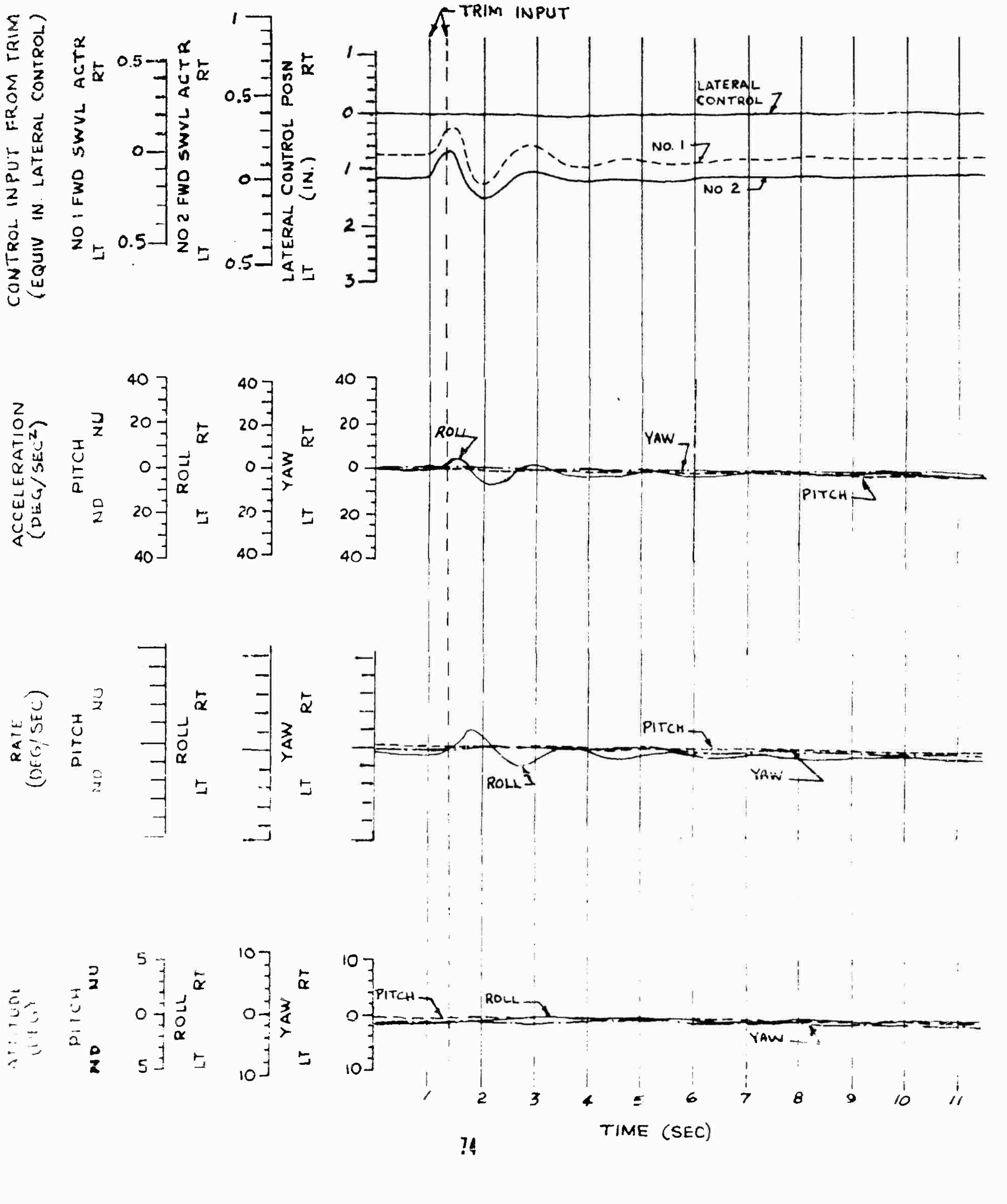


FIGURE 25. TRIM CONTROL POSITIONS IN FORWARD AND REARWARD FLIGHT

BOEING 347 S/N 65-7992

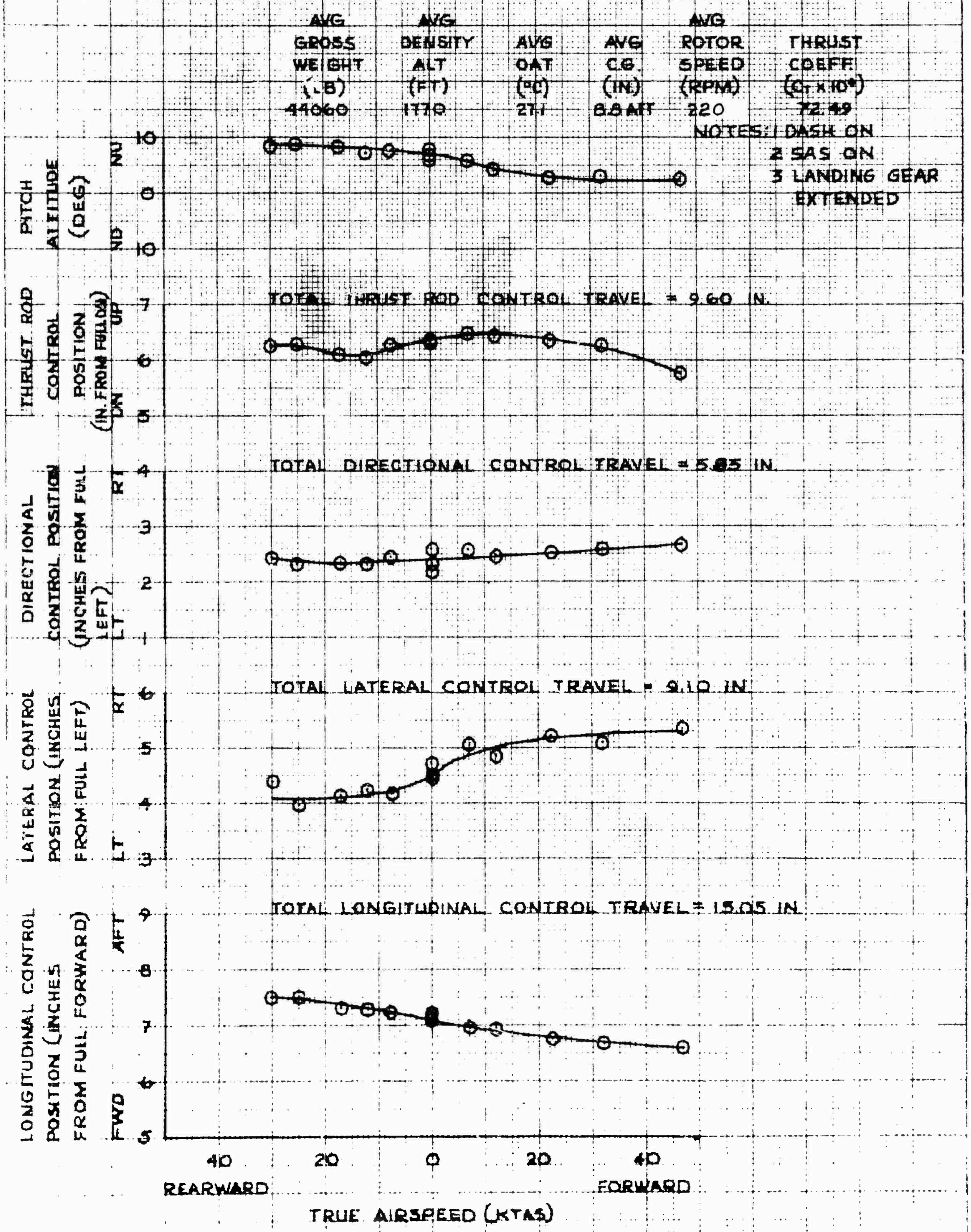


FIGURE 2.6 TRIM CONTROL POSITIONS IN SIDEWARD FLIGHT

BOEING 347 S/N 65-7992

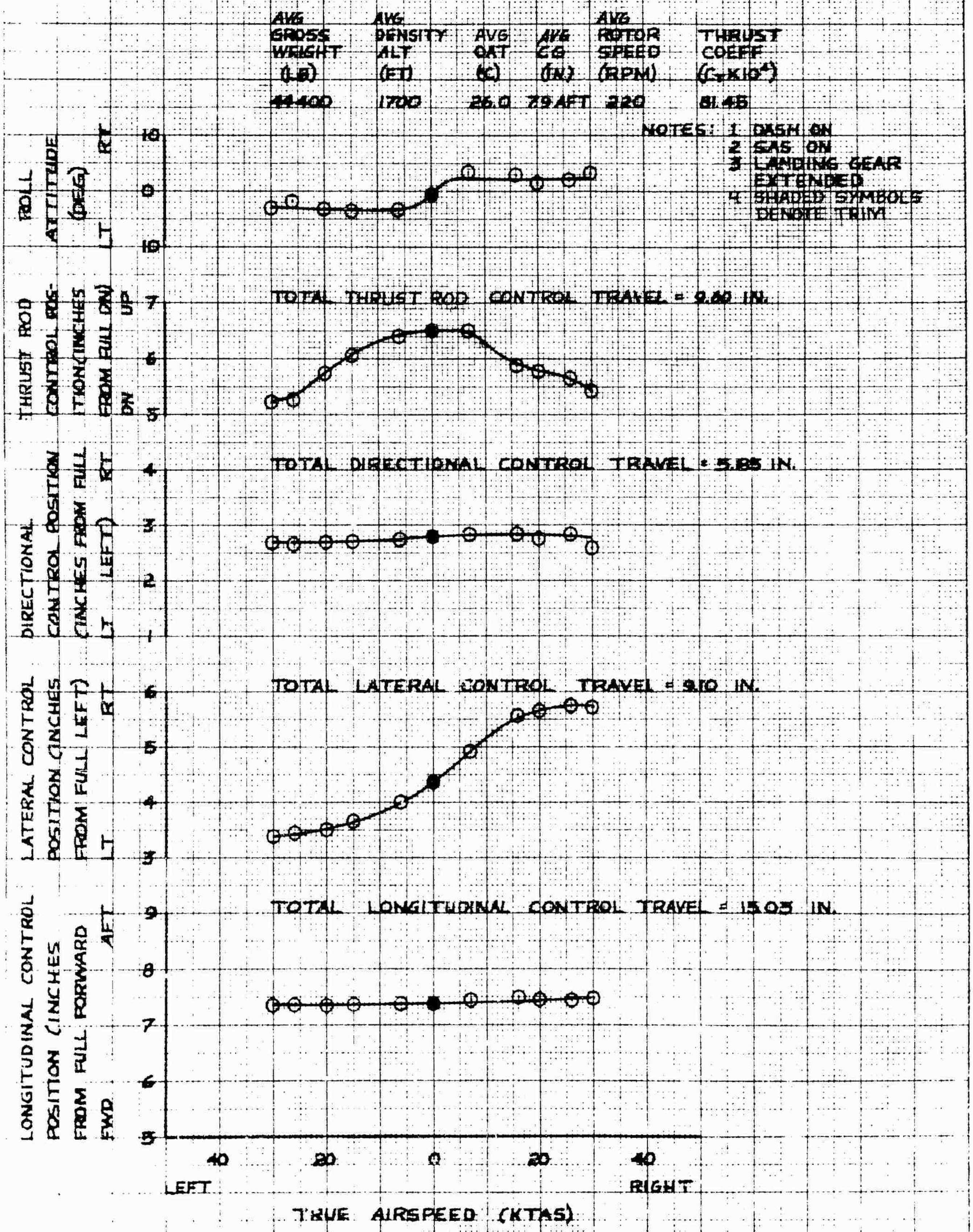


FIGURE 27 TRIM CONTROL POSITIONS IN LEVEL FLIGHT

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)
34130	5520	4.2	17.9 FWD	216	65.42

NOTES: 1 DASH ON
2 SAS ON
3 LANDING GEAR RETRACTED

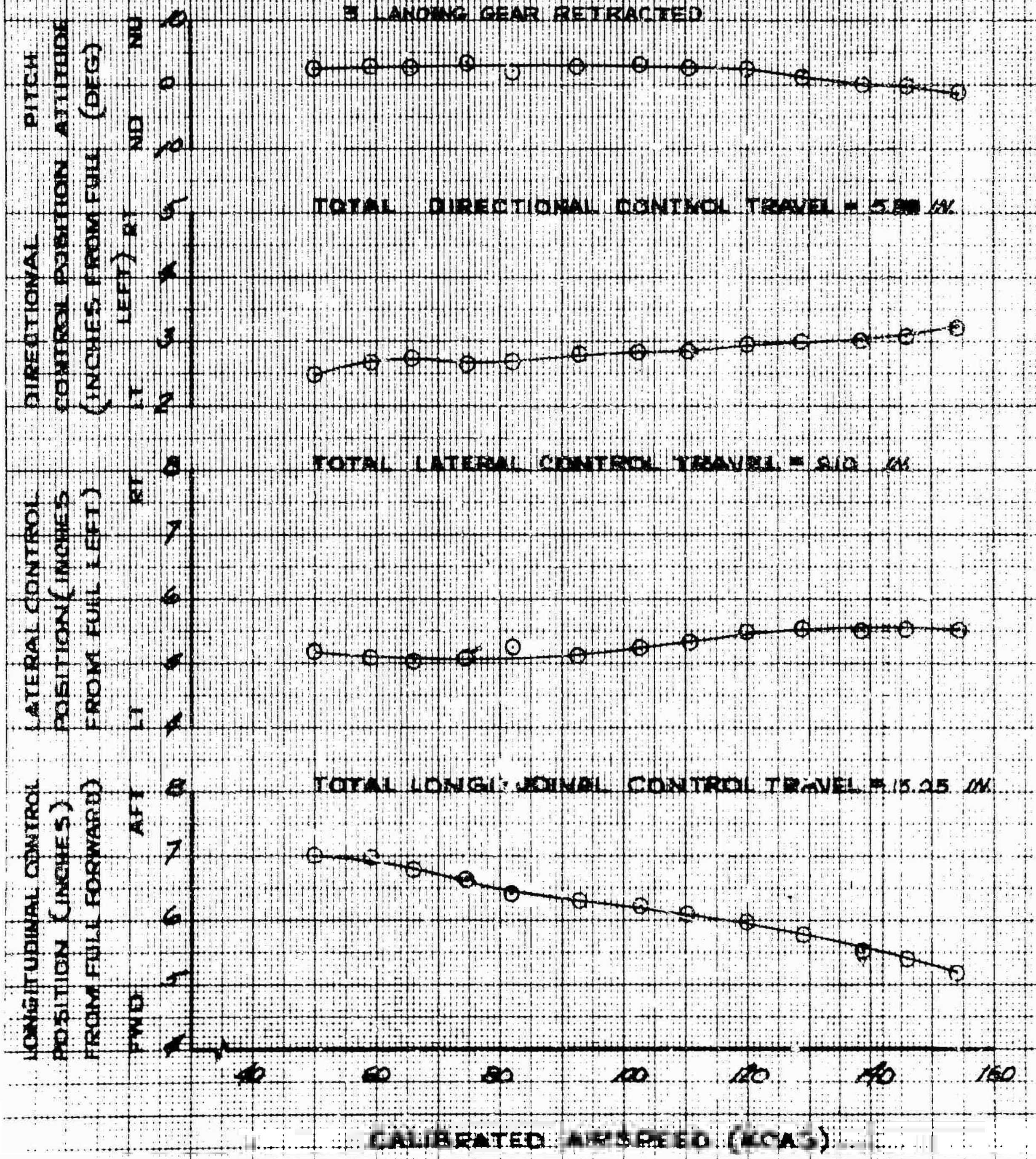


FIGURE 26 TRIM CONTROL POSITIONS IN LEVEL FLIGHT.
BOEING 347 S/N 65-7992

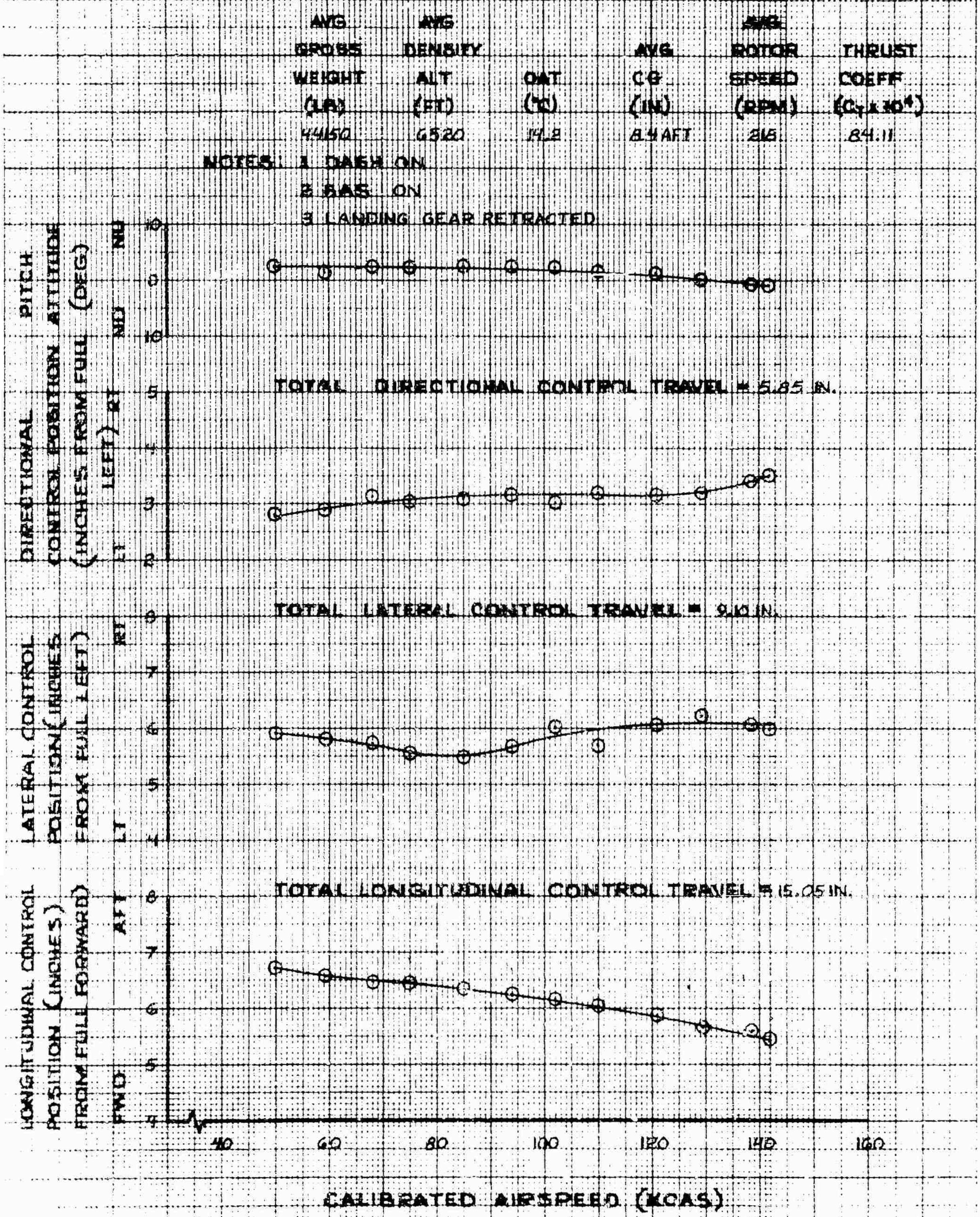


FIGURE 29 TRIM CONTROL POSITIONS IN LEVEL FLIGHT

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG C.G. (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)
34720	7950	16.0	17.3 FWD	219	6921

NOTES: 1 DASH ON
2 SAS ON
3 LANDING GEAR EXTENDED

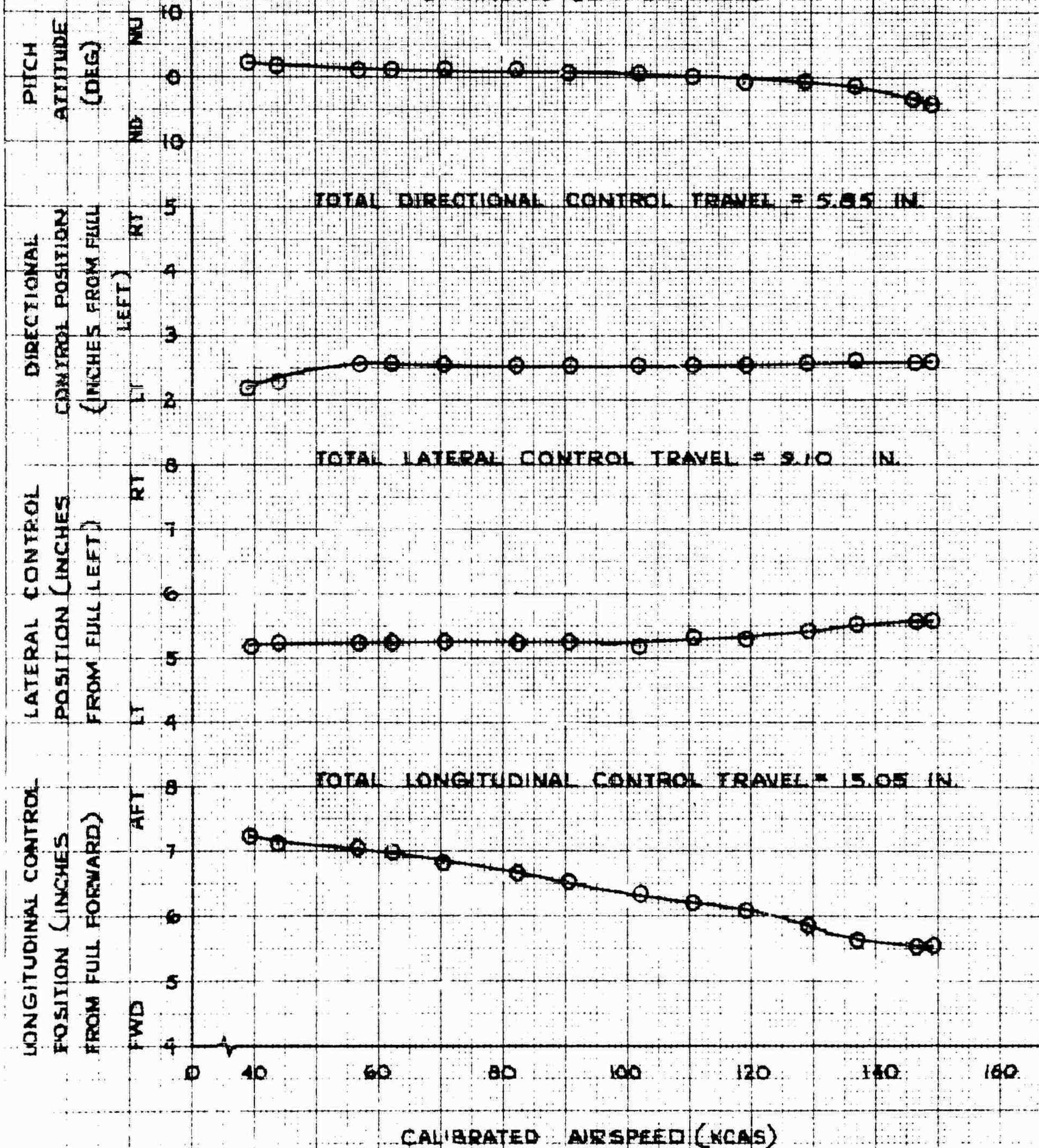


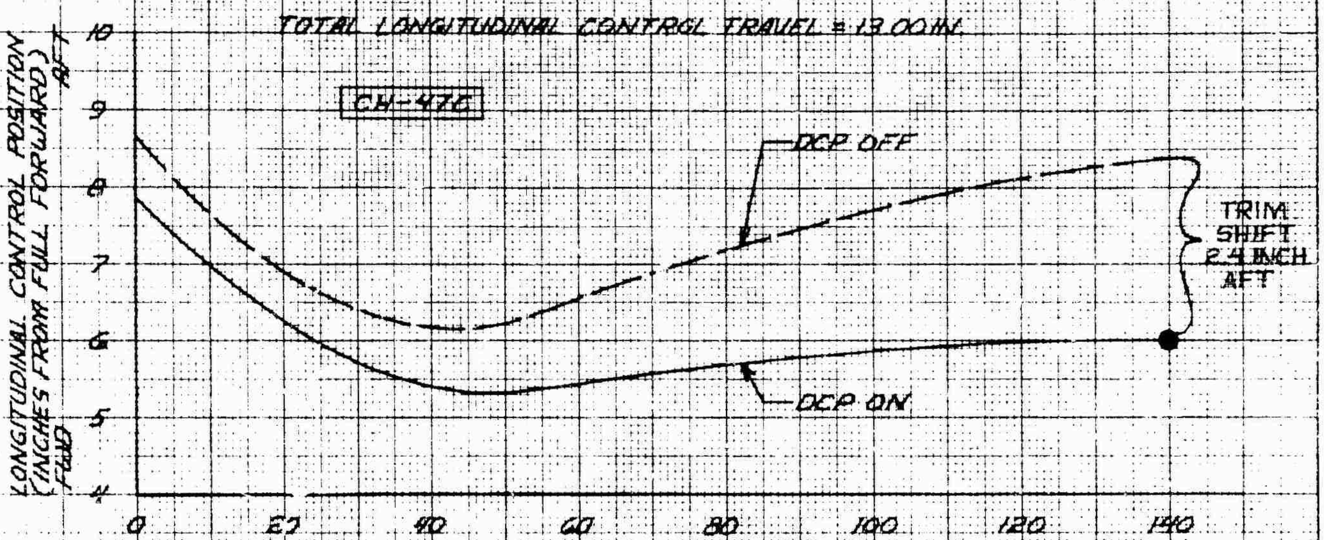
FIGURE 30 TRIM CONTROL POSITIONS COMPARISON IN LEVEL FLIGHT

CH-47C USA S/N 66-19121

BOEING 347 S/N 65-7992

	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
CH-47C	37850	2800	9.5	5.8 AFT	236	55.60	140
BOEING 347	44150	1520	14.2	8.3 AFT	219	84.11	140

NOTE: MODEL 347 CONTROL POSITIONS BELOW 50 KCAS ESTIMATED BASED ON OGE HOVER DATA WITH GEAR EXTENDED



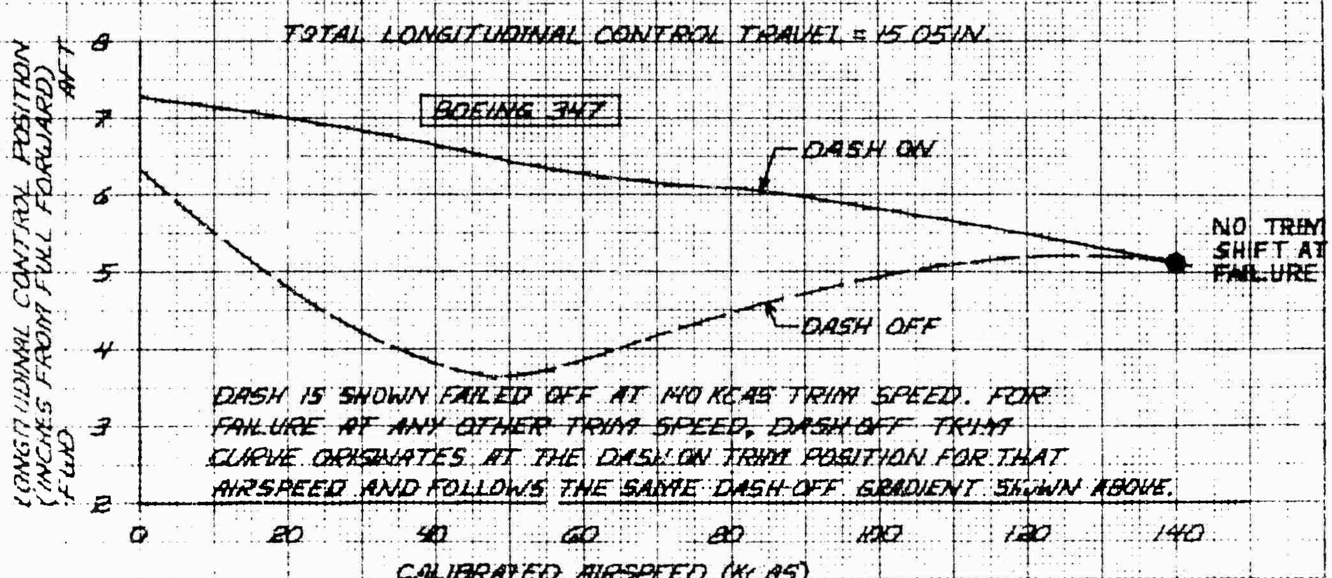


FIGURE 3 / STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY

BOEING 347 S/N 65-7992

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
○	35830	5140	12.7	18 FWD	220	65.02	78	LEVEL
□	35380	5320	13.5	18 FWD	220	64.52	110	LEVEL
△	34780	5640	14.2	19.8 FWD	220	64.06	128	LEVEL

NOTES:
1. DASH ON
2. SAS ON
3. THRUST ROD CONTROL POSITION FIXED AT EACH TRIM A/S
4. SHADED SYMBOLS DENOTE TRIM POINTS
5. LANDING GEAR RETRACTED

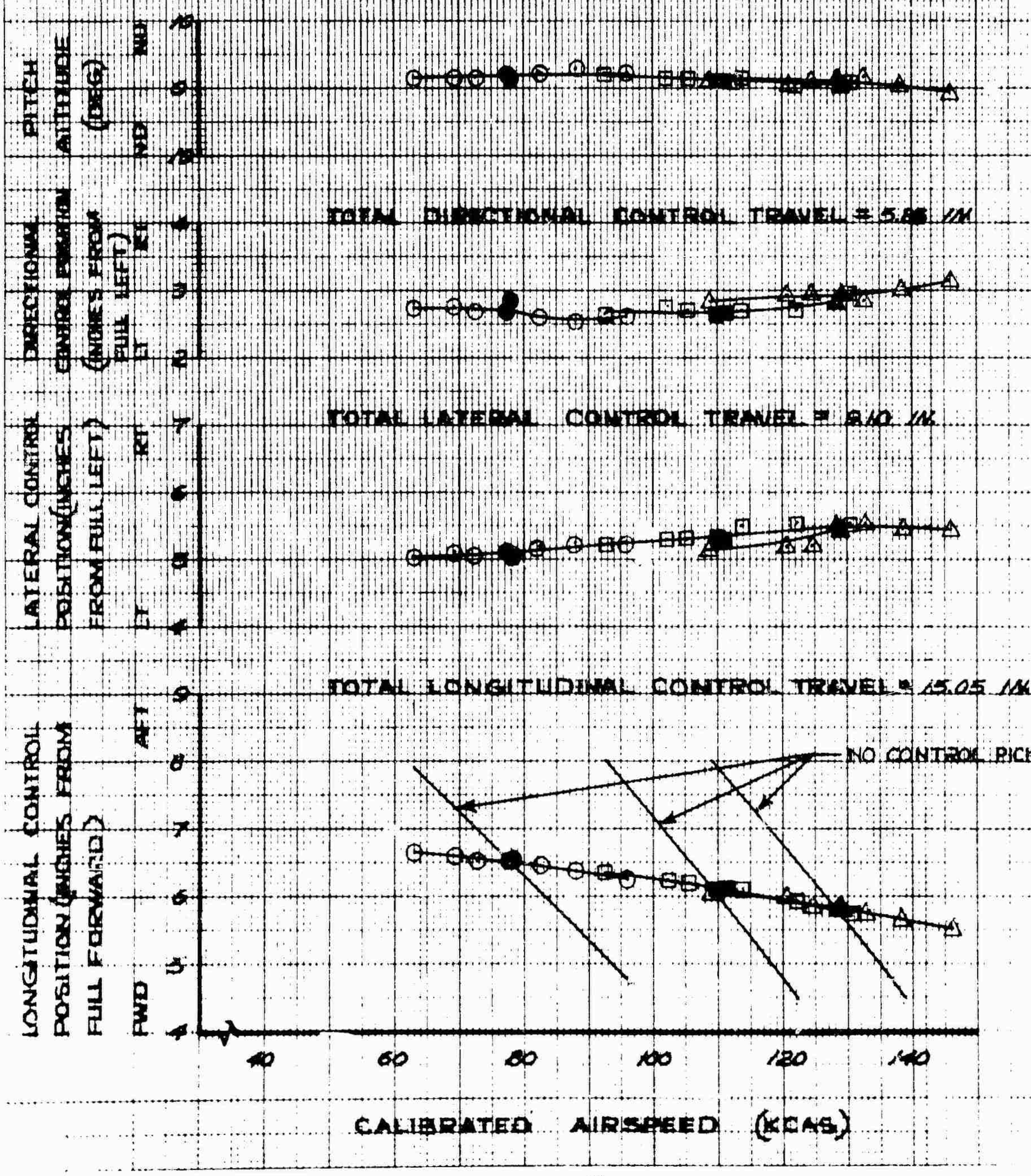


FIGURE 32. STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY.

BOEING 347 S/N 65-7992

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
○	45470	4850	5.5	28.2 FWD	220	81.70	81	LEVEL
□	44620	4850	8.0	29.2 FWD	220	80.19	113	LEVEL
△	43930	4450	7.1	28.6 FWD	220	78.19	130	LEVEL

NOTES:
1. DASH ON
2. SAS ON
3. THRUST ROD CONTROL POSITION FIXED AT EACH TRIM A/S
4. SHADED SYMBOLS DENOTE TRIM POINTS
5. LANDING GEAR RETRACTED

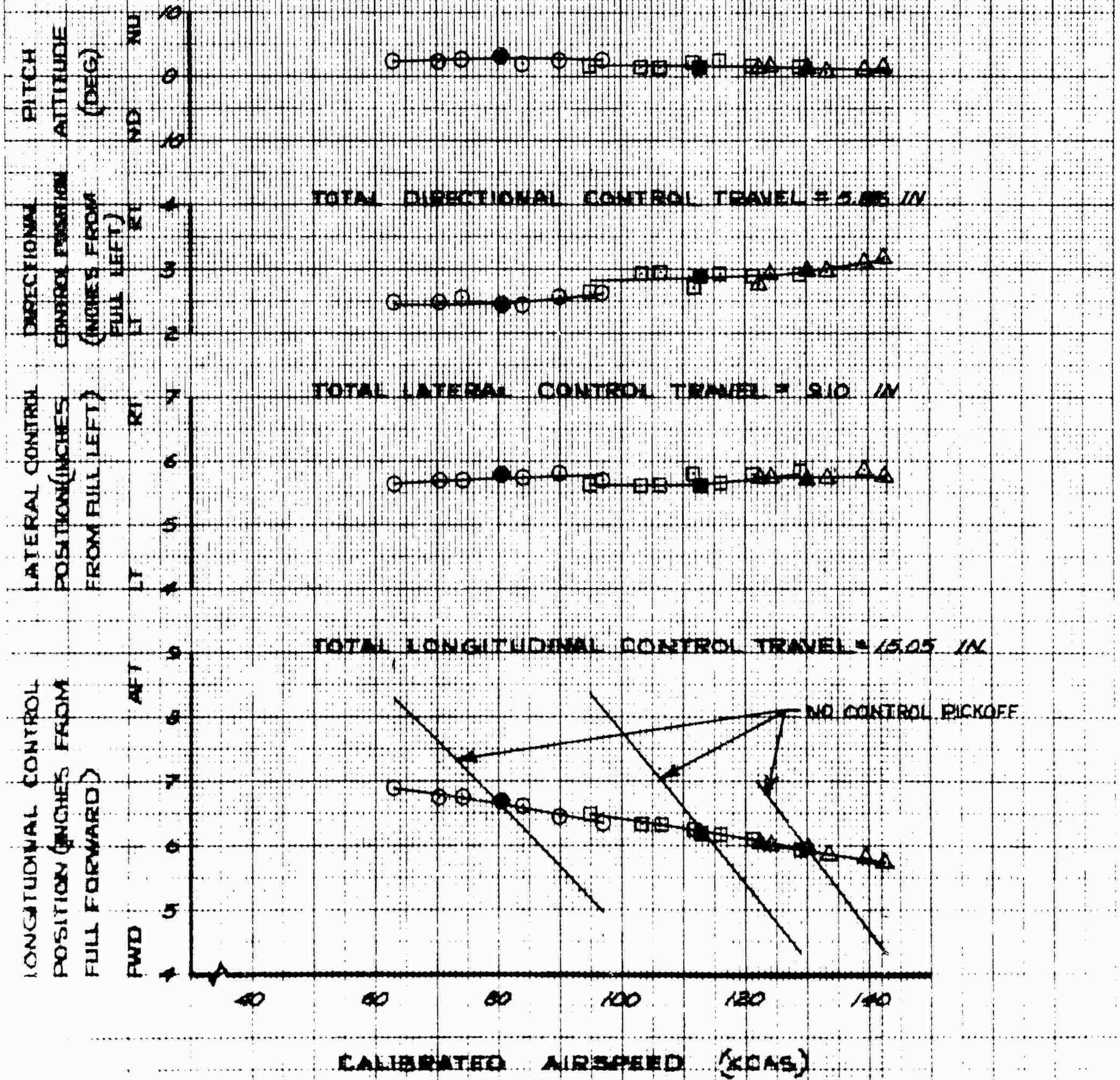


FIGURE 33 STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY

BOEING 347 S/N 65-7992

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
○	45640	3860	16.5	8.0 AFT	220	82.42	77	LEVEL
□	44930	3910	16.6	7.6 AFT	220	81.15	108	LEVEL
△	44500	3940	17.7	7.8 AFT	220	80.76	128	LEVEL

NOTES:
1 DASH ON
2 SAS ON
3 THRUST ROD CONTROL POSITION FIXED AT EACH TRIM A/S
4 SHADED SYMBOLS DENOTE TRIM POINTS
5 LANDING GEAR RETRACTED

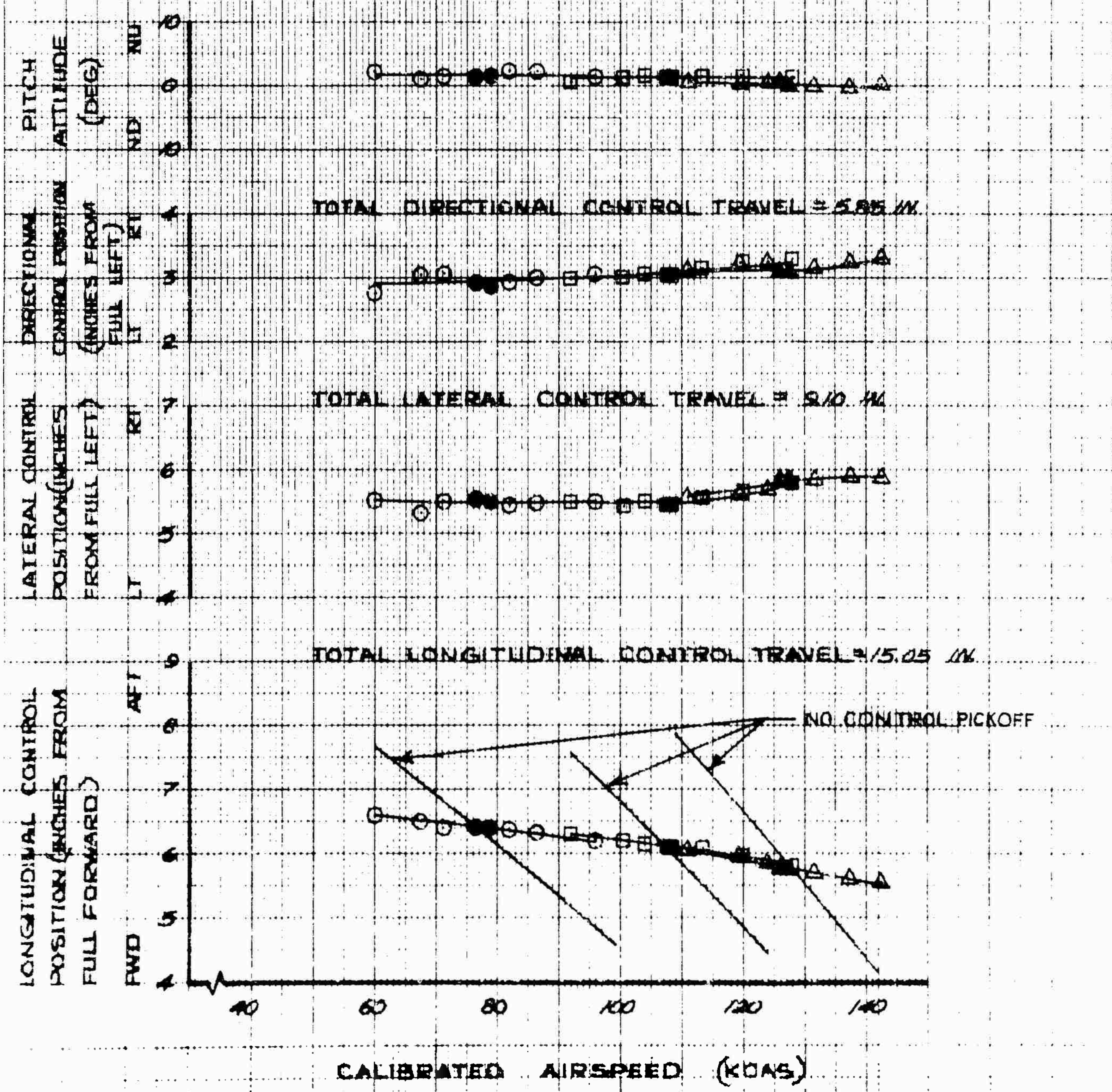


FIGURE 34 STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY

BOEING 347 S/N 65-7992

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
○	43420	[illegible]	5.7	9.7 AFT	220	91.73	76	LEVEL
□	42760	7980	6.8	10.9 AFT	220	89.93	108	LEVEL

NOTES:
1 DASH ON
2 SAS ON
3 THRUST ROD CONTROL POSITION FIXED AT EACH TRIM A/S
4 SHADED SYMBOLS DENOTE TRIM POINTS
5 LANDING GEAR RETRACTED

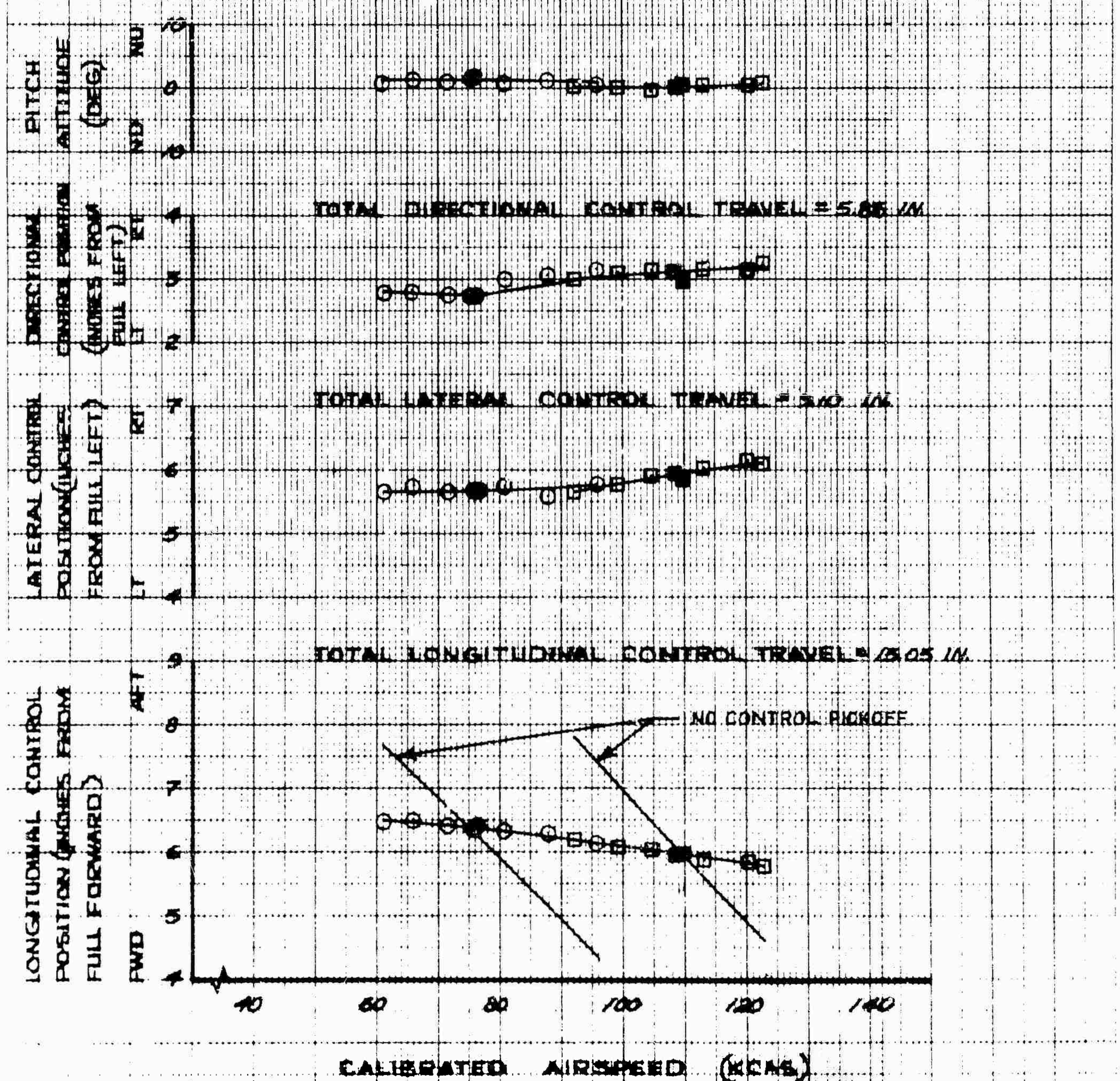


FIGURE 35 STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
45880	3750	22.4	8.1 AFT	220	72.65	82	CLIMB AT NRP

NOTES:
1 DASH ON
2 SAS ON
3 THRUST ROD CONTROL POSITION FIXED AT TRIM A/S
4 SHADED SYMBOLS DENOTE TRIM POINTS
5 LANDING GEAR RETRACTED

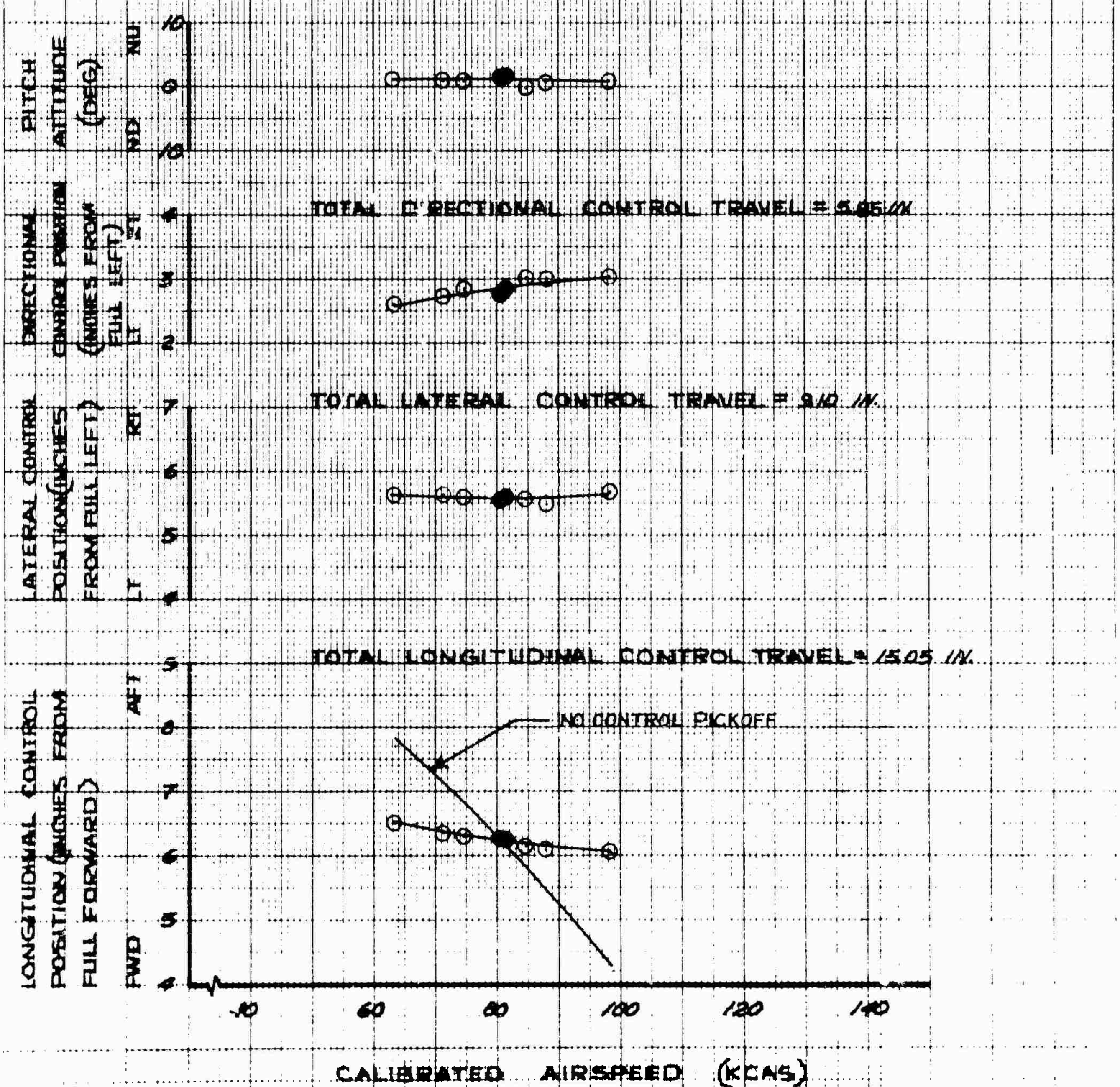


FIGURE 36 STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
45640	4650	23.1	8.0AFT	220	81.63	78	DESCENT AT AUTO

NOTES: 1 DASH ON
2 SAS ON
3 THRUST ROD CONTROL POSITION FIXED AT TRIM A/S
4 SHADED SYMBOLS DENOTE TRIM POINTS
5 LANDING GEAR RETRACTED

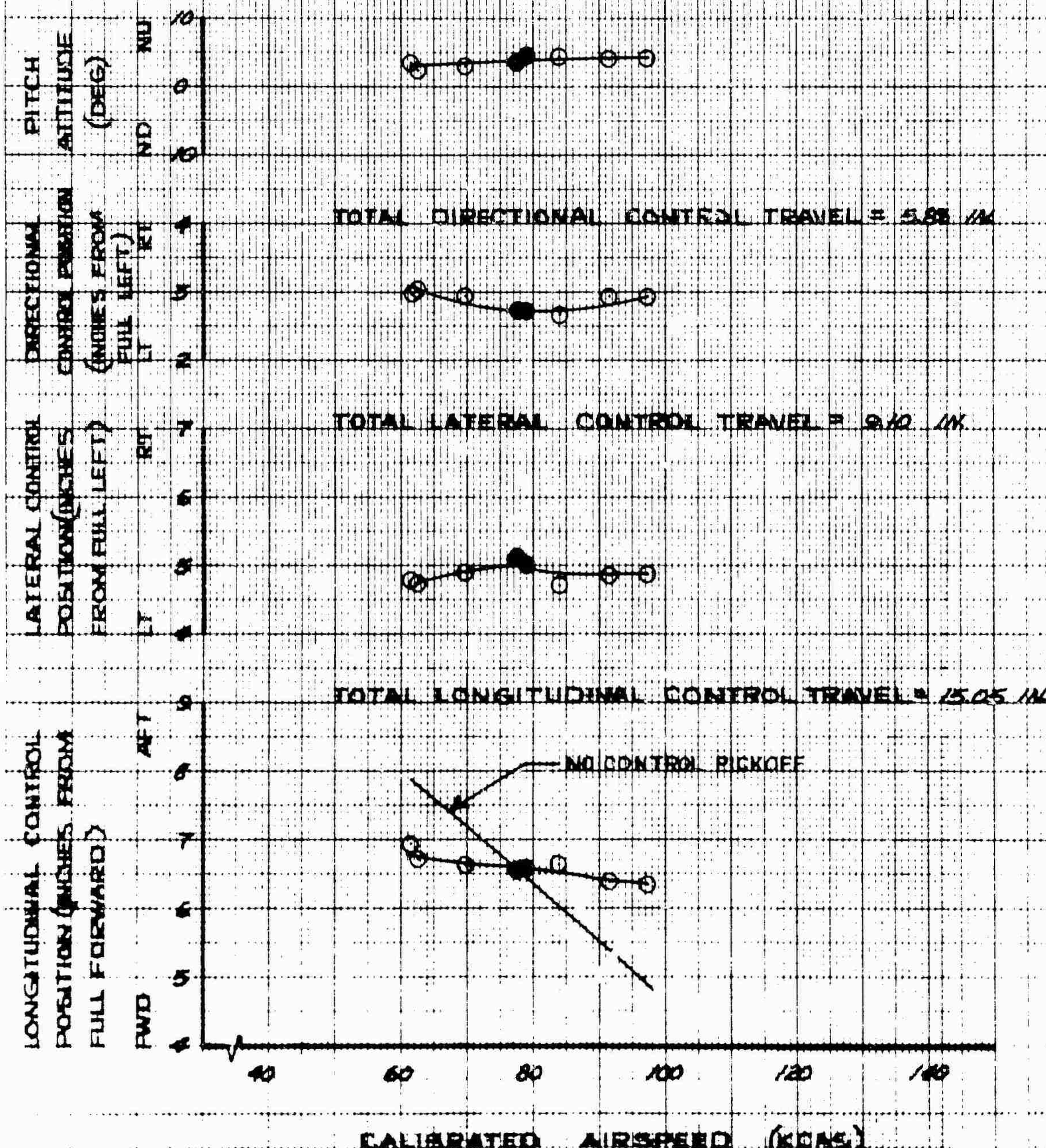


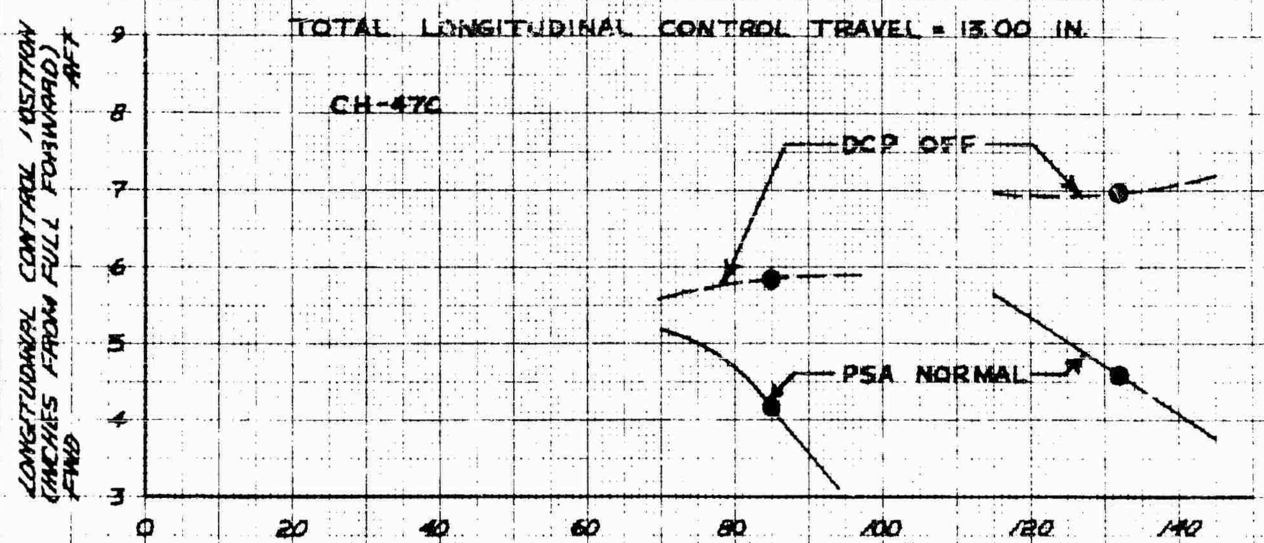
FIGURE 37. STATIC LONGITUDINAL COLLECTIVE-FIXED STABILITY COMPARISON.

CH-47C USA S/N 66-19121

BOEING 347 S/N 65-7992

	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
CH-47C	42630	3780	7.5	5.2 AFT	246	59.54	85	LEVEL
CH-47C	43165	5000	5.1	4.8 AFT	247	62.34	132	LEVEL
BOEING 347	45640	3860	16.5	8.0 AFT	220	82.42	77	LEVEL
BOEING 347	44500	3940	17.7	7.8 AFT	220	80.76	128	LEVEL

NOTE: 1. SHADED SYMBOLS DENOTE TRIM



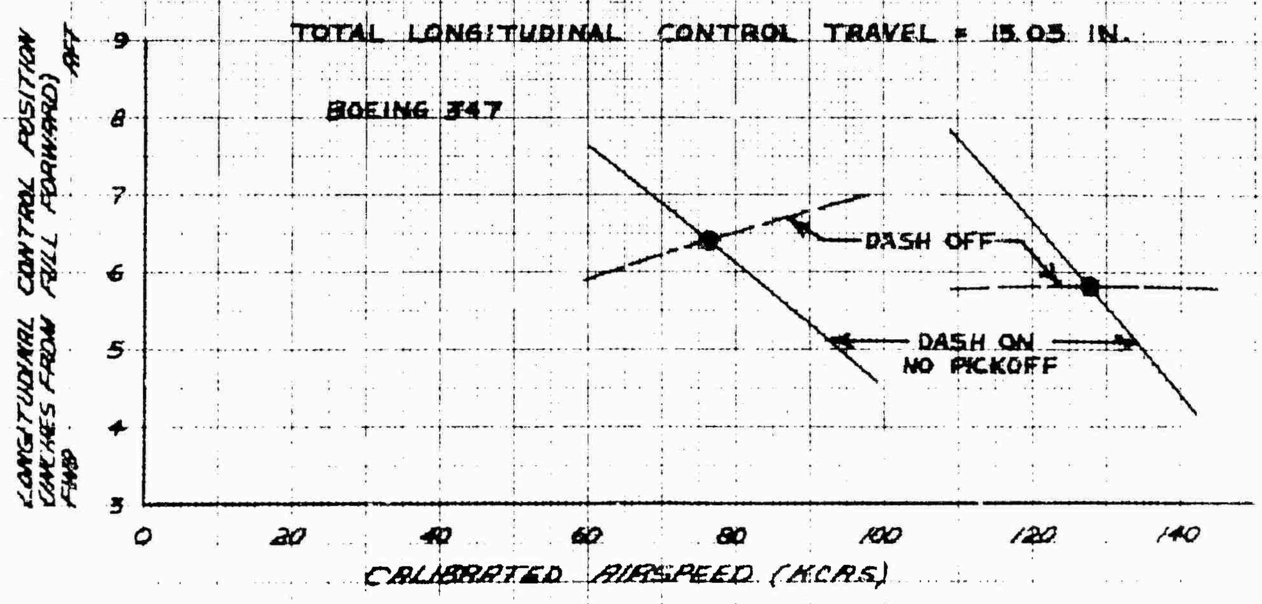


FIGURE 38. LONG TERM LONGITUDINAL RESPONSE FROM OFF TRIM CONDITION.

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF. ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	78

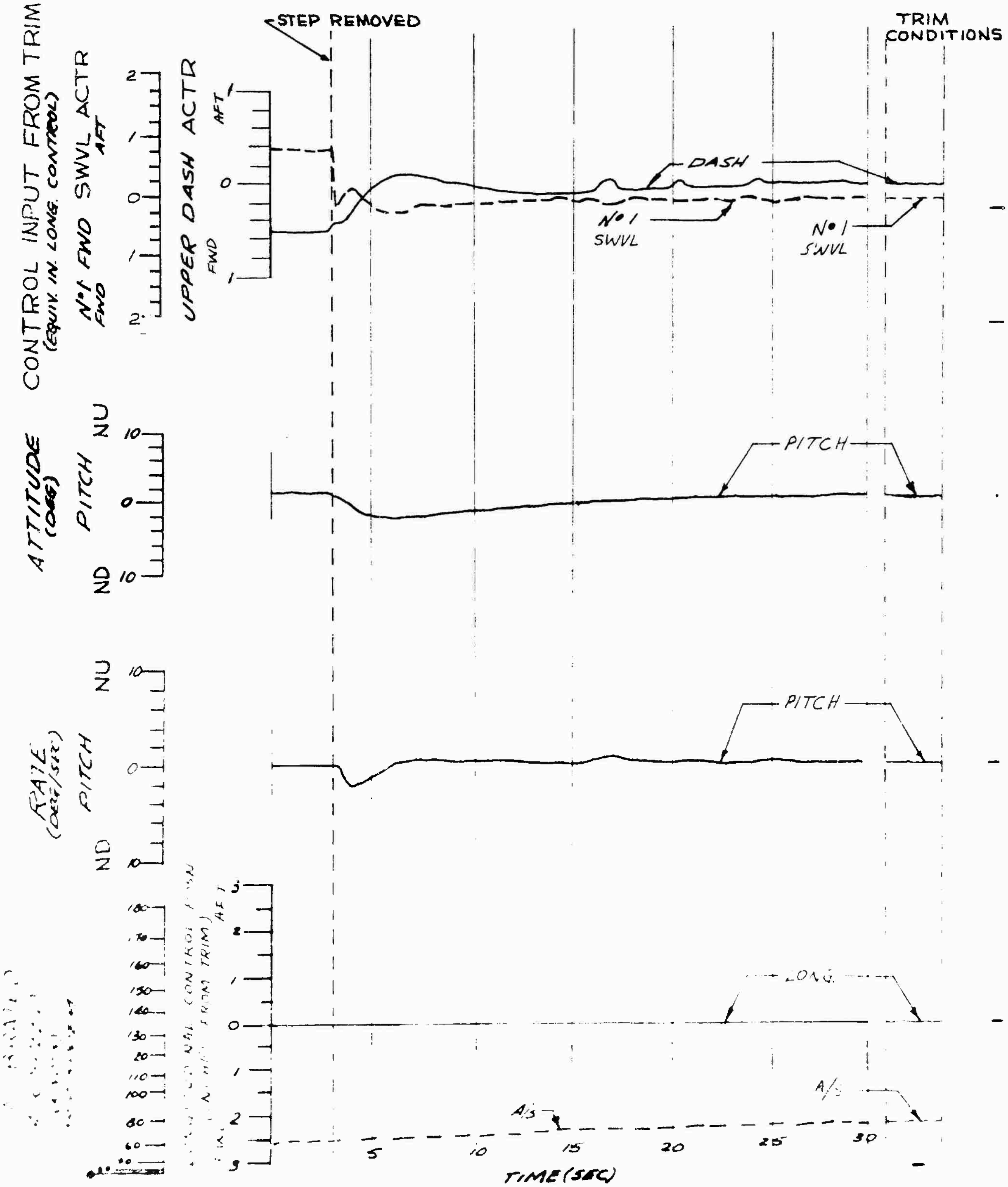


FIGURE 38. LONG TERM LONGITUDINAL RESPONSE FROM OFF TRIM CONDITION.

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	78

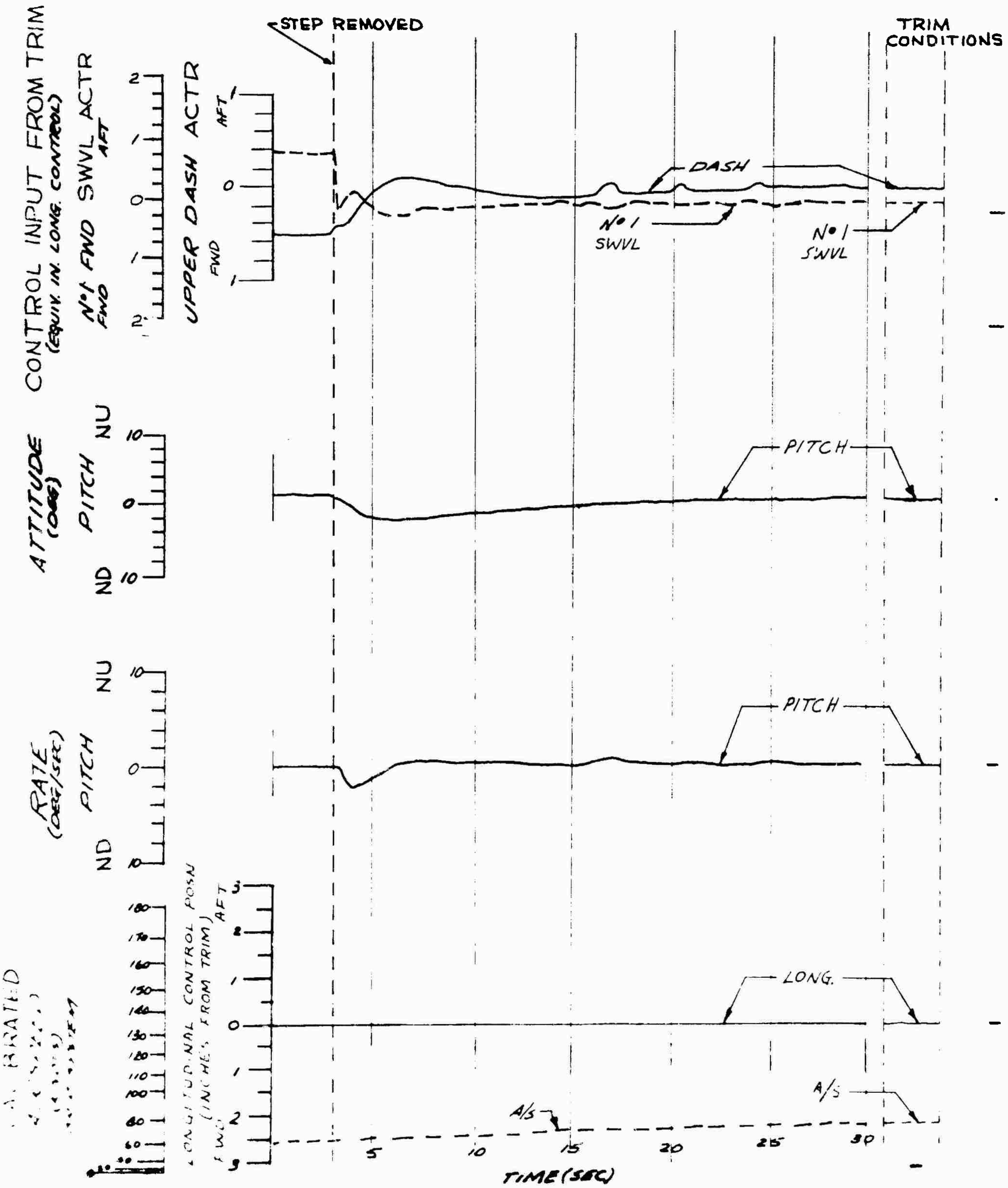


FIGURE 40. LONG TERM LONGITUDINAL RESPONSE FROM OFF TRIM CONDITION.

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	129

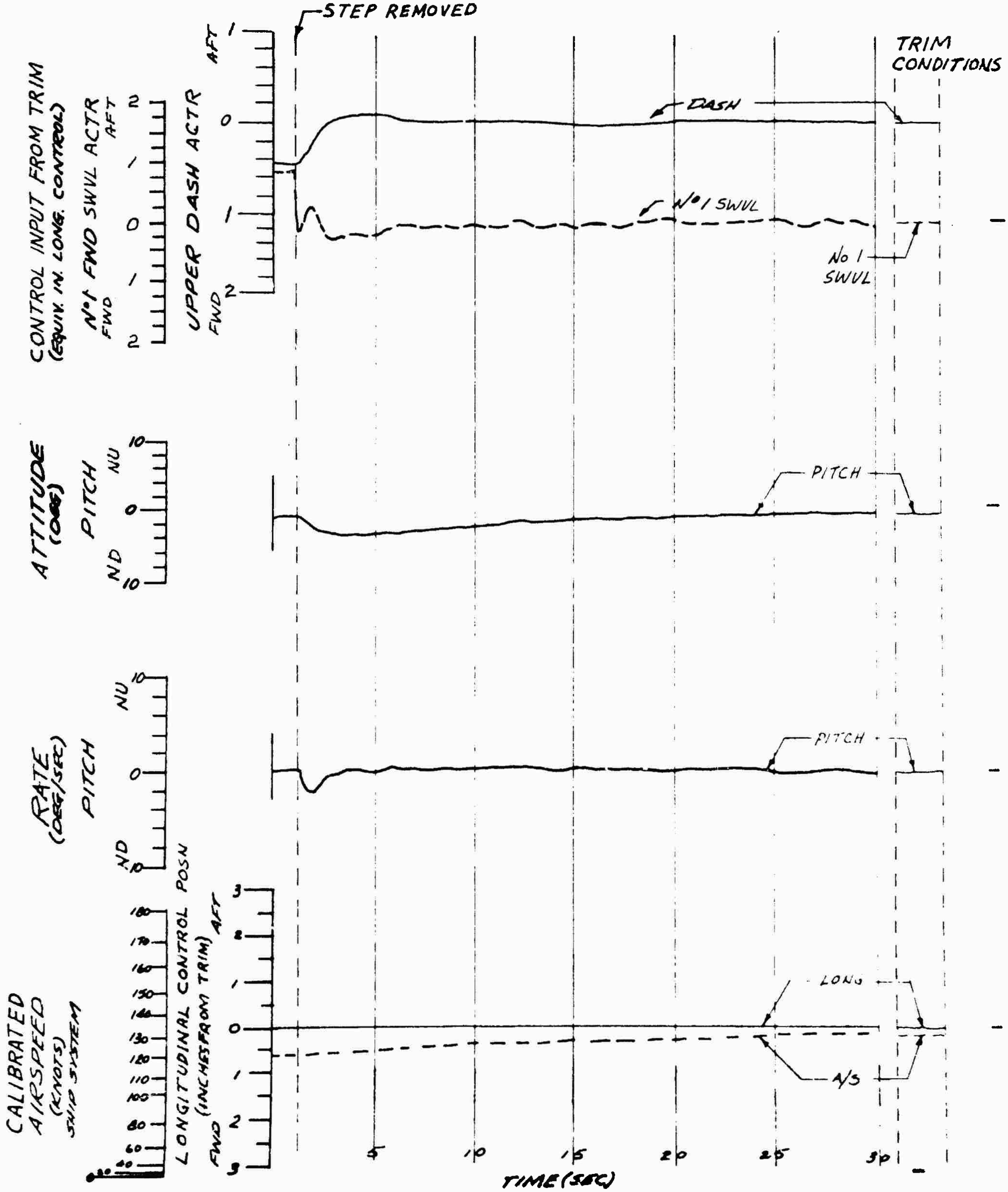


FIGURE 41 LONG TERM LONGITUDINAL RESPONSE FROM OFF TRIM CONDITION.

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9 AFT	221	96.25	78

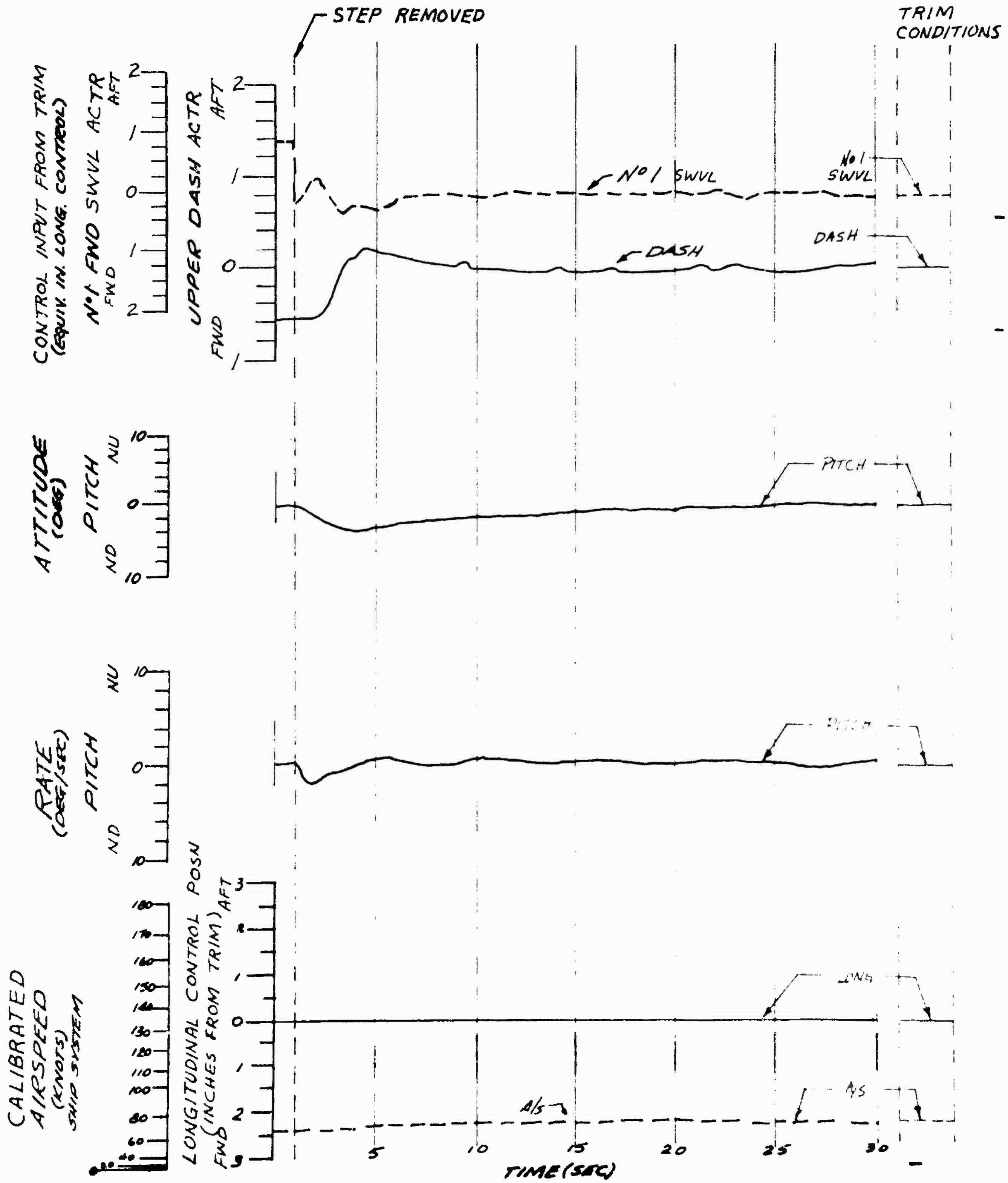


FIGURE 42. LONG TERM LONGITUDINAL RESPONSE FROM OFF TRIM CONDITION.

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9.1 AFT	220	96.25	110

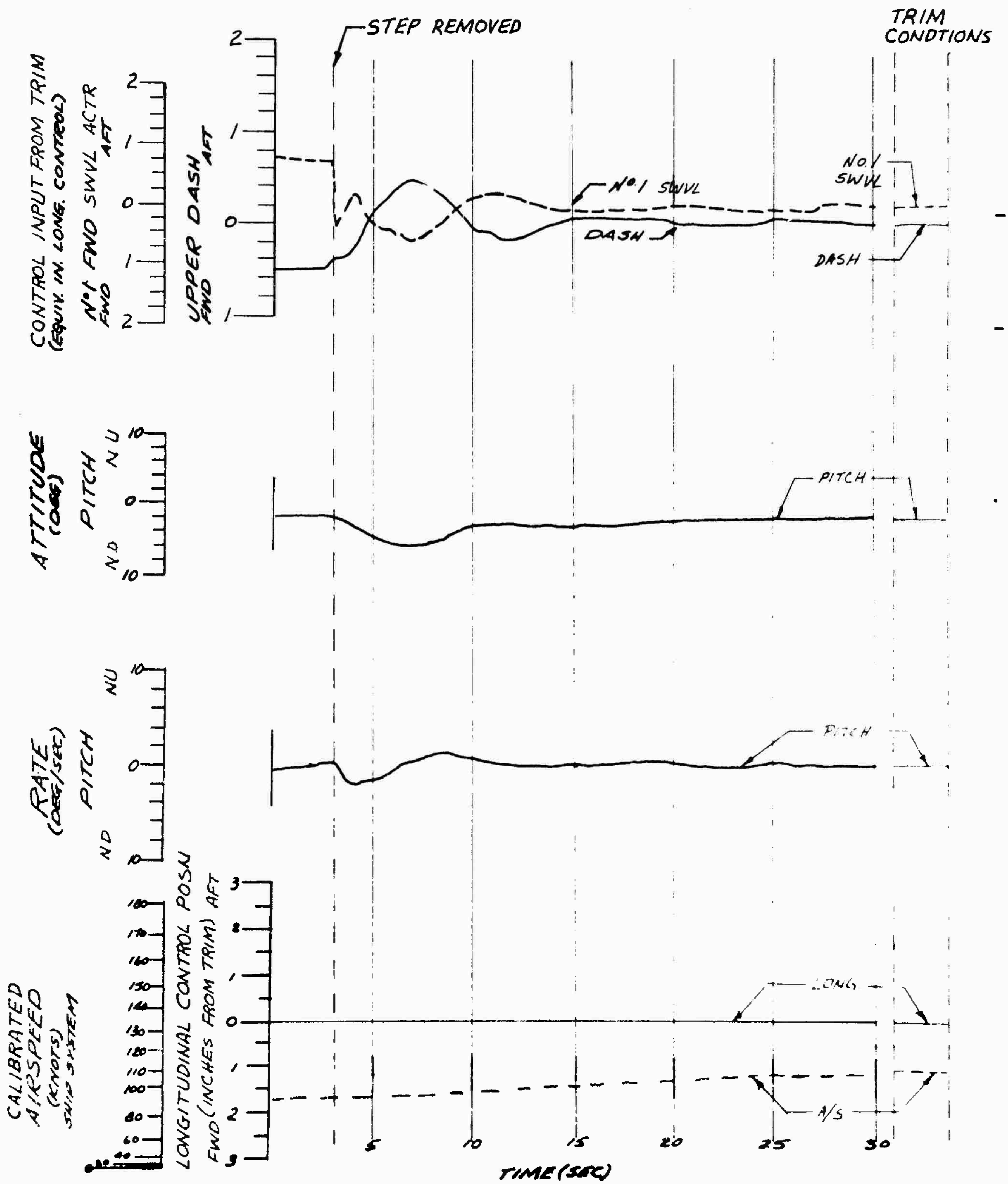


FIGURE 43 AIRCRAFT RESPONSE FOLLOWING FWD LONGITUDINAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45070	1820	27.7	7.7 AFT	220	74.16	35

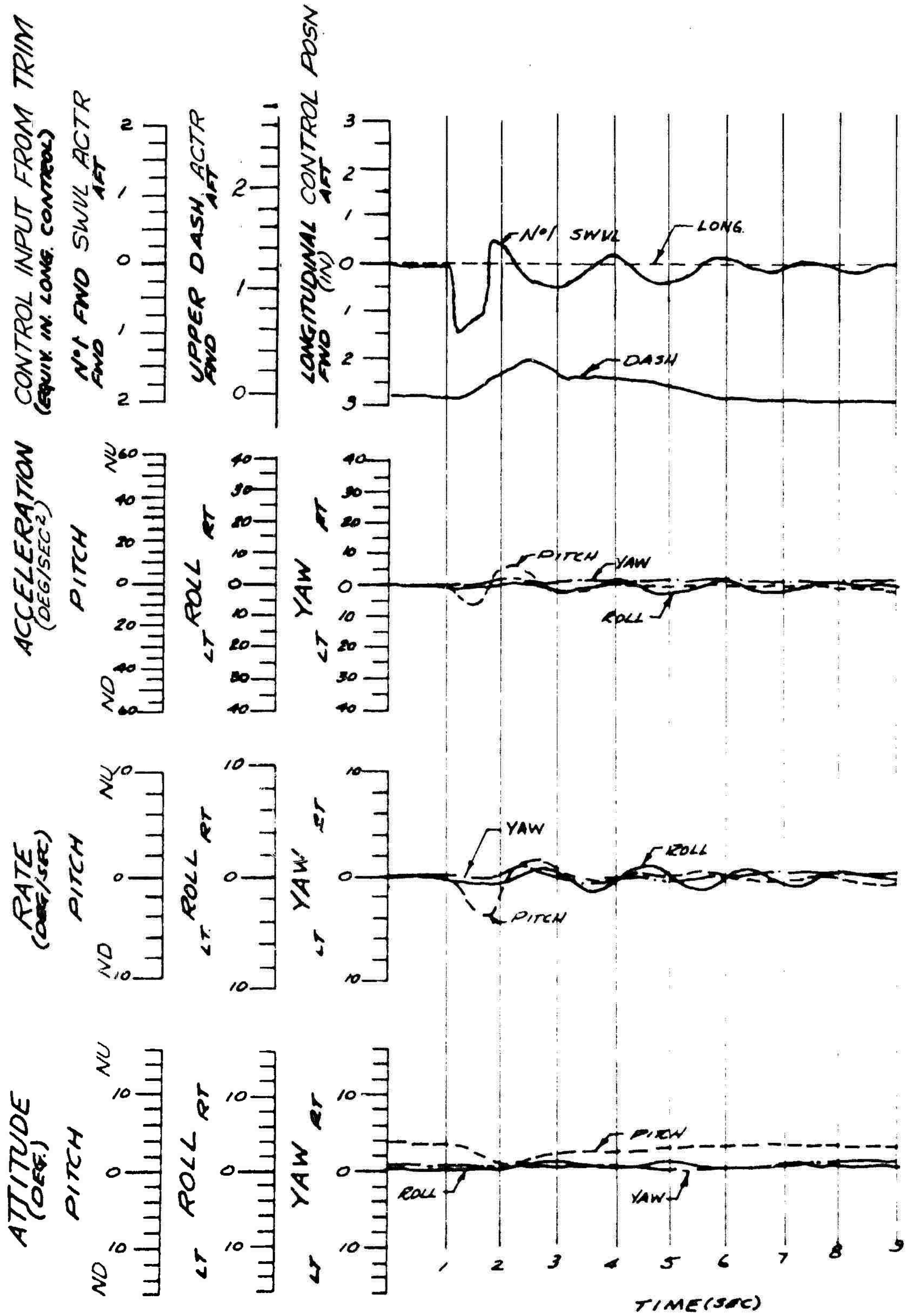


FIGURE 44 AIRCRAFT RESPONSE FOLLOWING AFT LONGITUDINAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	78

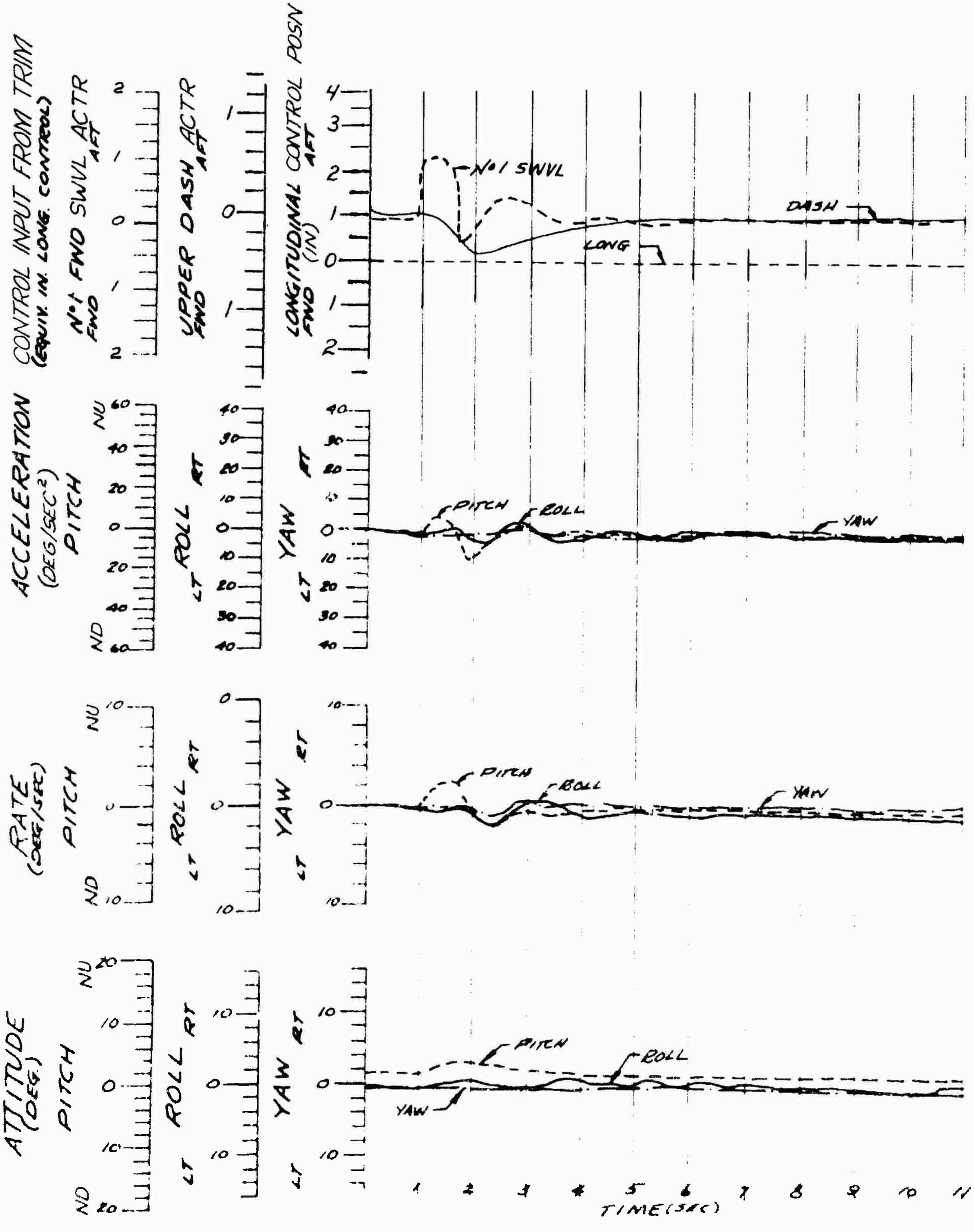


FIGURE 45 AIRCRAFT RESPONSE FOLLOWING FWD LONGITUDINAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	111

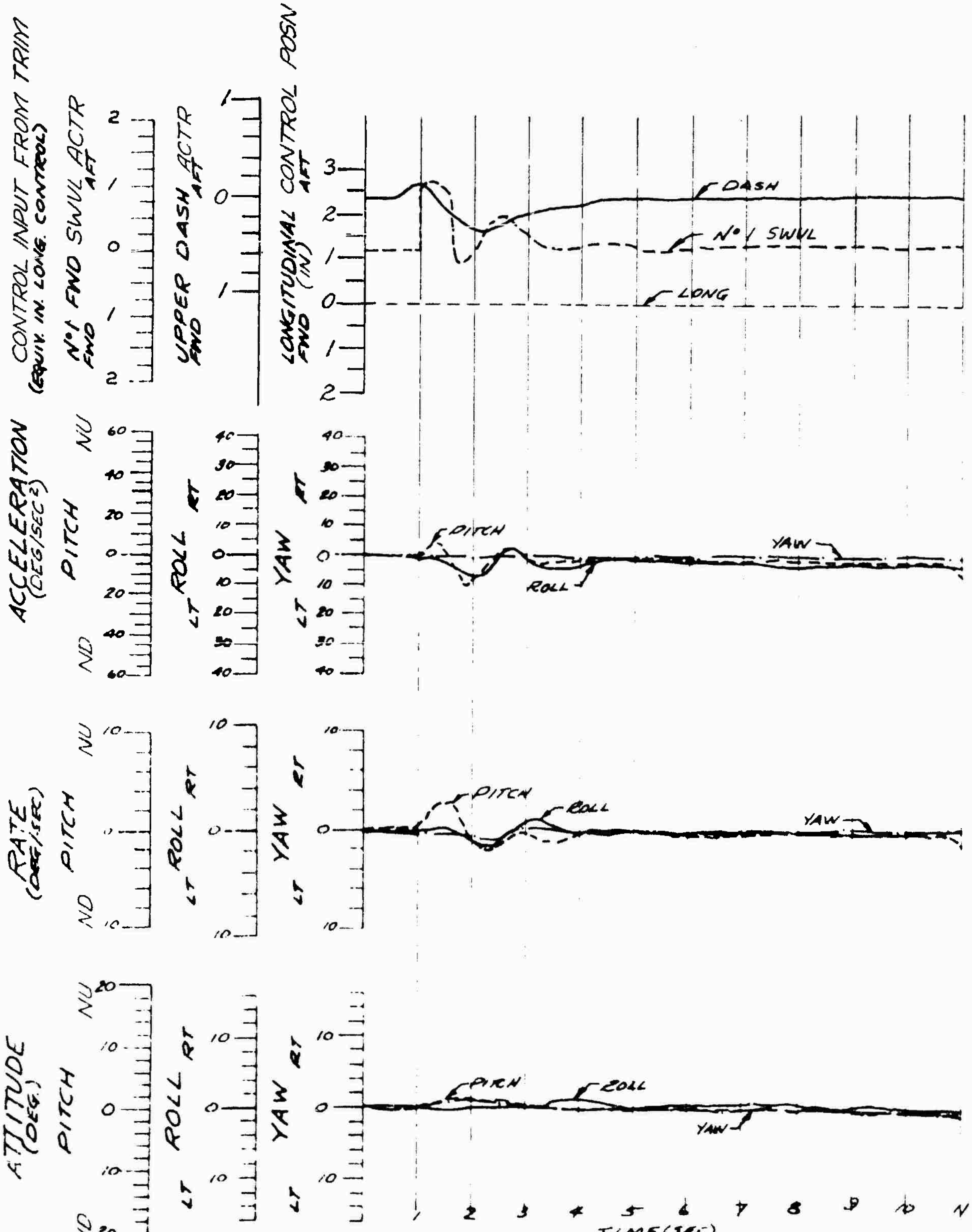


FIGURE 46 AIRCRAFT RESPONSE FOLLOWING AFT LONGITUDINAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	129

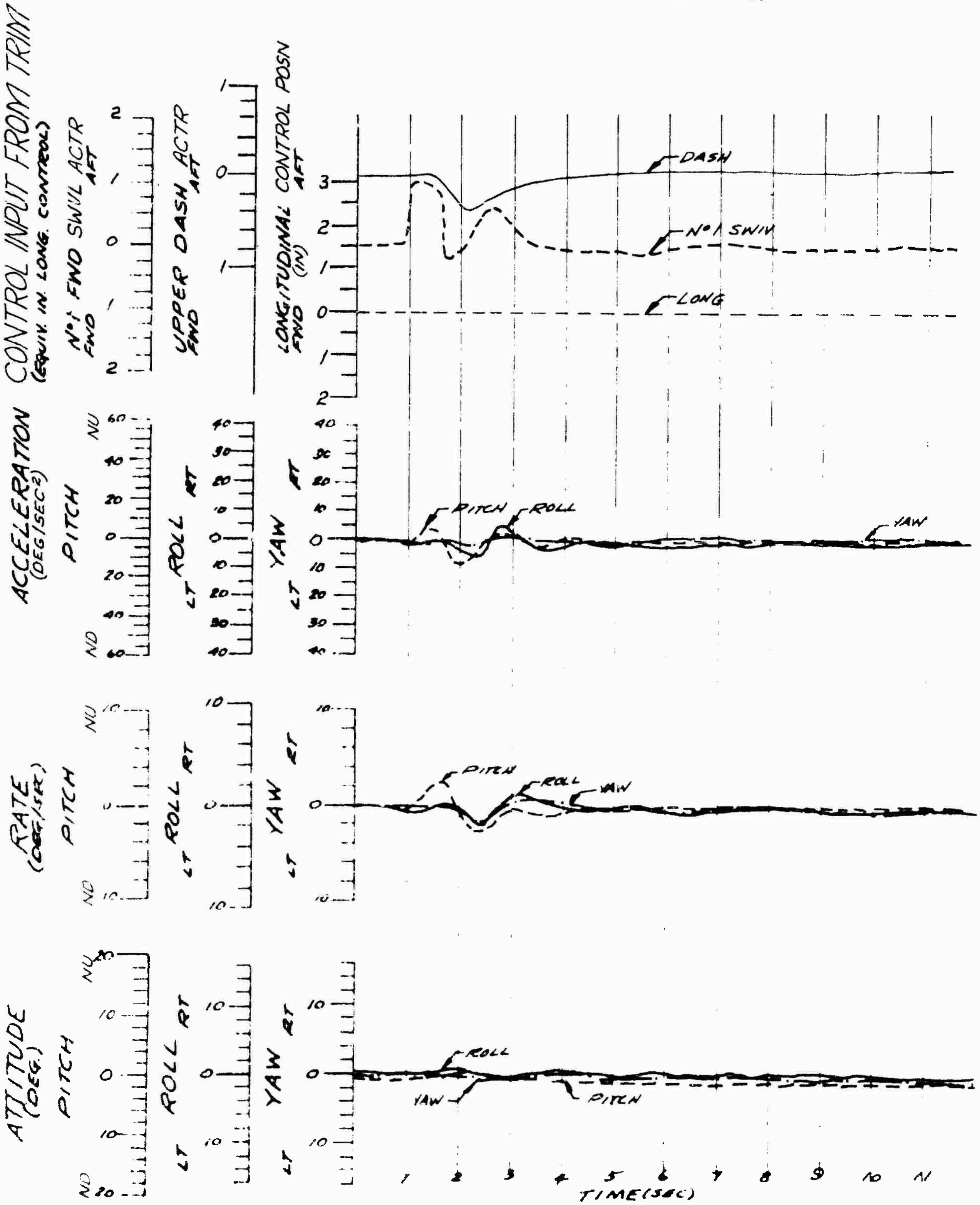


FIGURE 47 AIRCRAFT RESPONSE FOLLOWING FWD LONGITUDINAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45290	5040	16.0	7.8 AFT	220	81.79	78

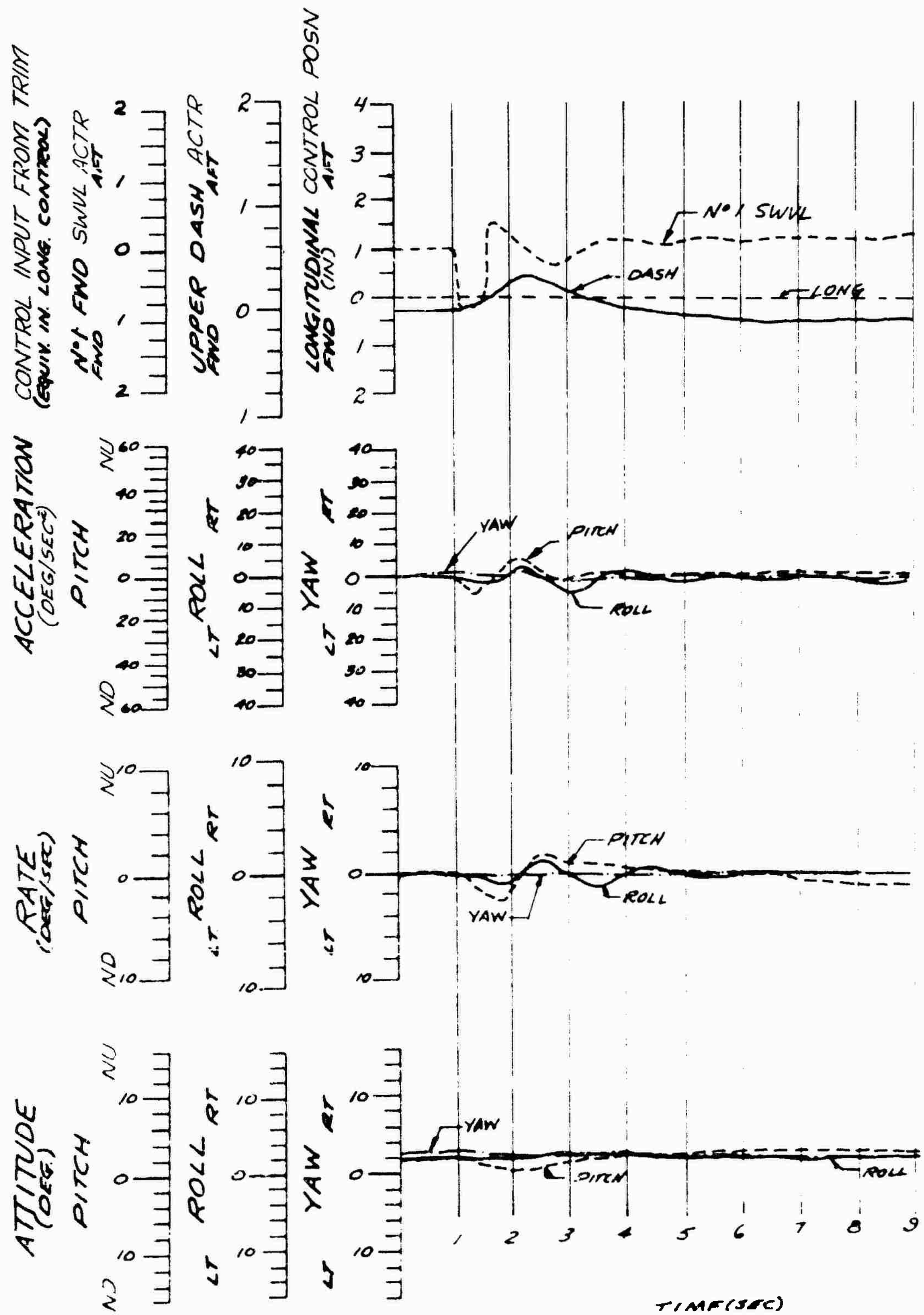


FIGURE 48 AIRCRAFT RESPONSE FOLLOWING AFT LONGITUDINAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45290	5040	16.0	7.8 AFT	220	81.79	129

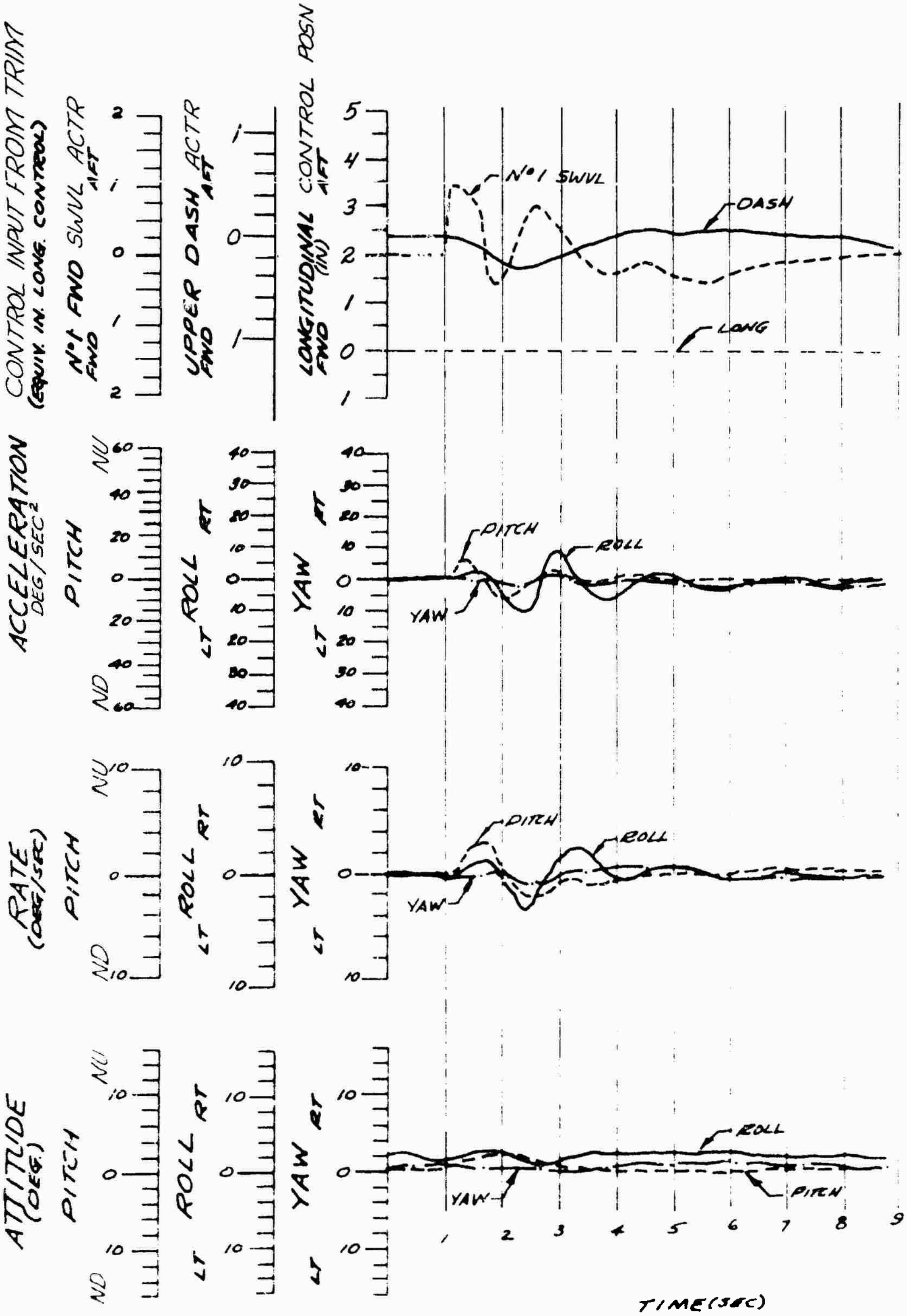


FIGURE 49 AIRCRAFT RESPONSE FOLLOWING AFT LONGITUDINAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9 AFT	22	96.25	78

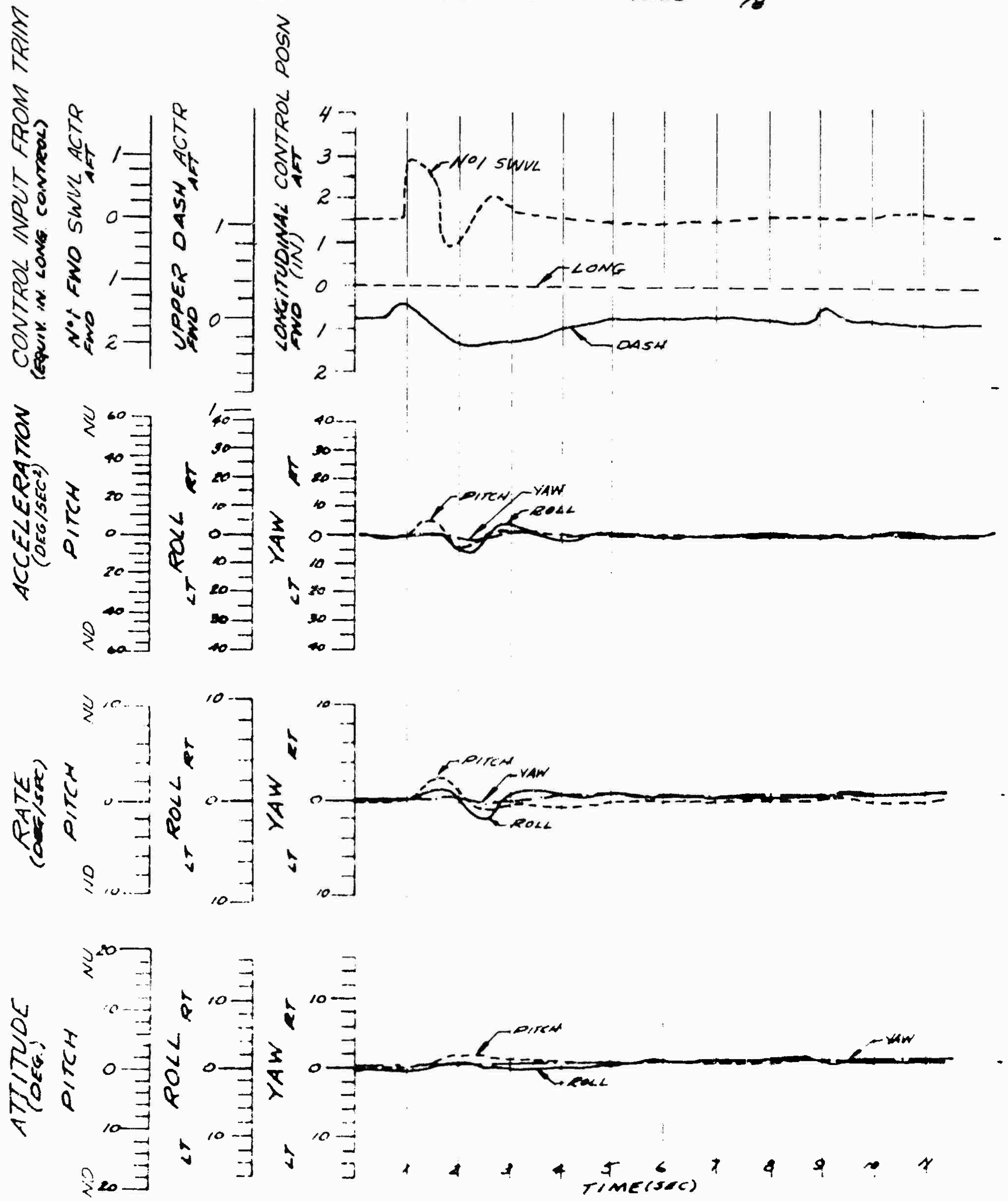


FIGURE 50 AIRCRAFT RESPONSE FOLLOWING AFT LONGITUDINAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9.1 AFT	220	96.25	111

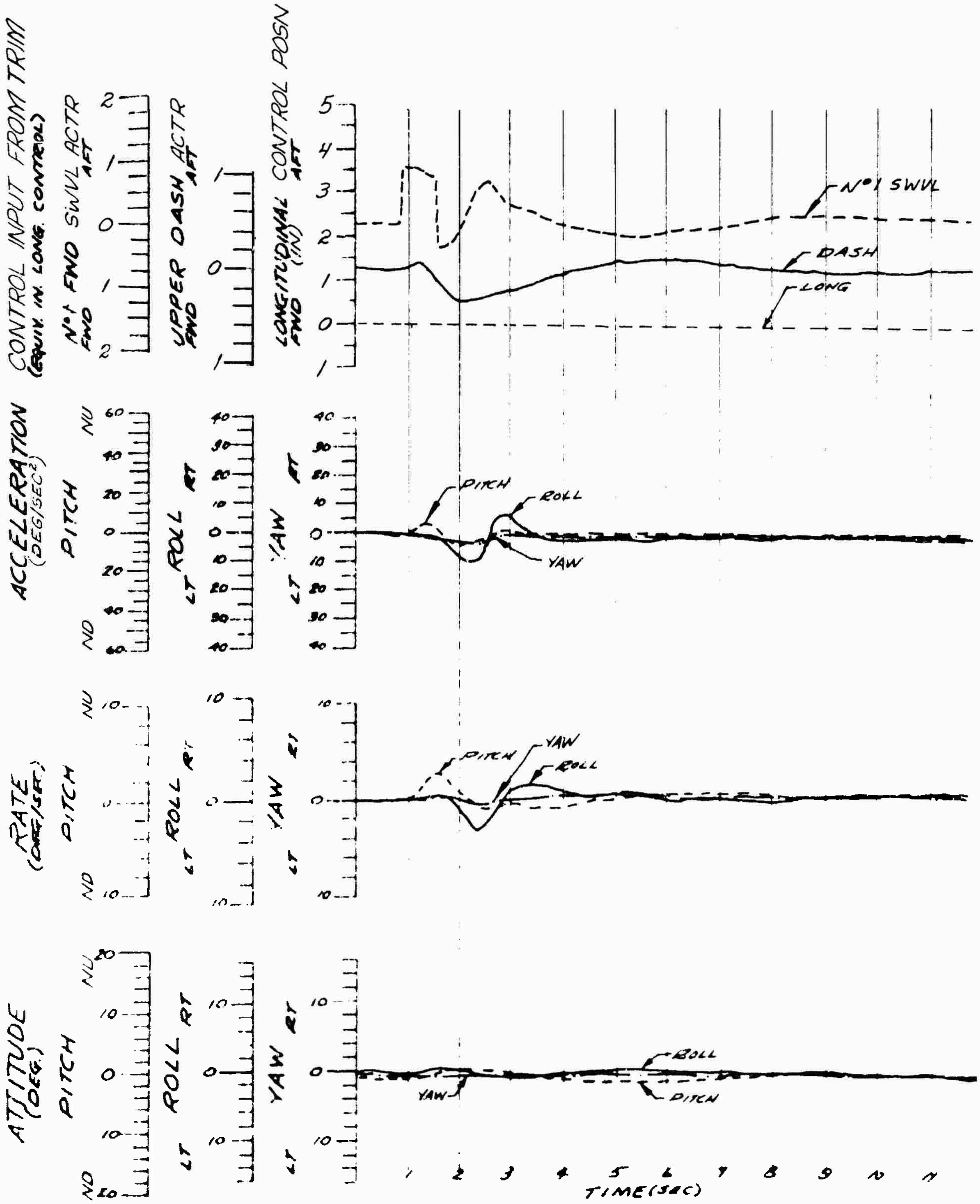


FIGURE 51. STATIC LATERAL DIRECTIONAL STABILITY
BOEING 347 S/N 65-7992

AVG. GROSS WEIGHT (LB)	AVG. DENSITY ALT (FT)	AVG OAT (°C)	AVG. CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
33010	5740	12.7	16.1 FWD	220	61.10	78	LEVEL

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

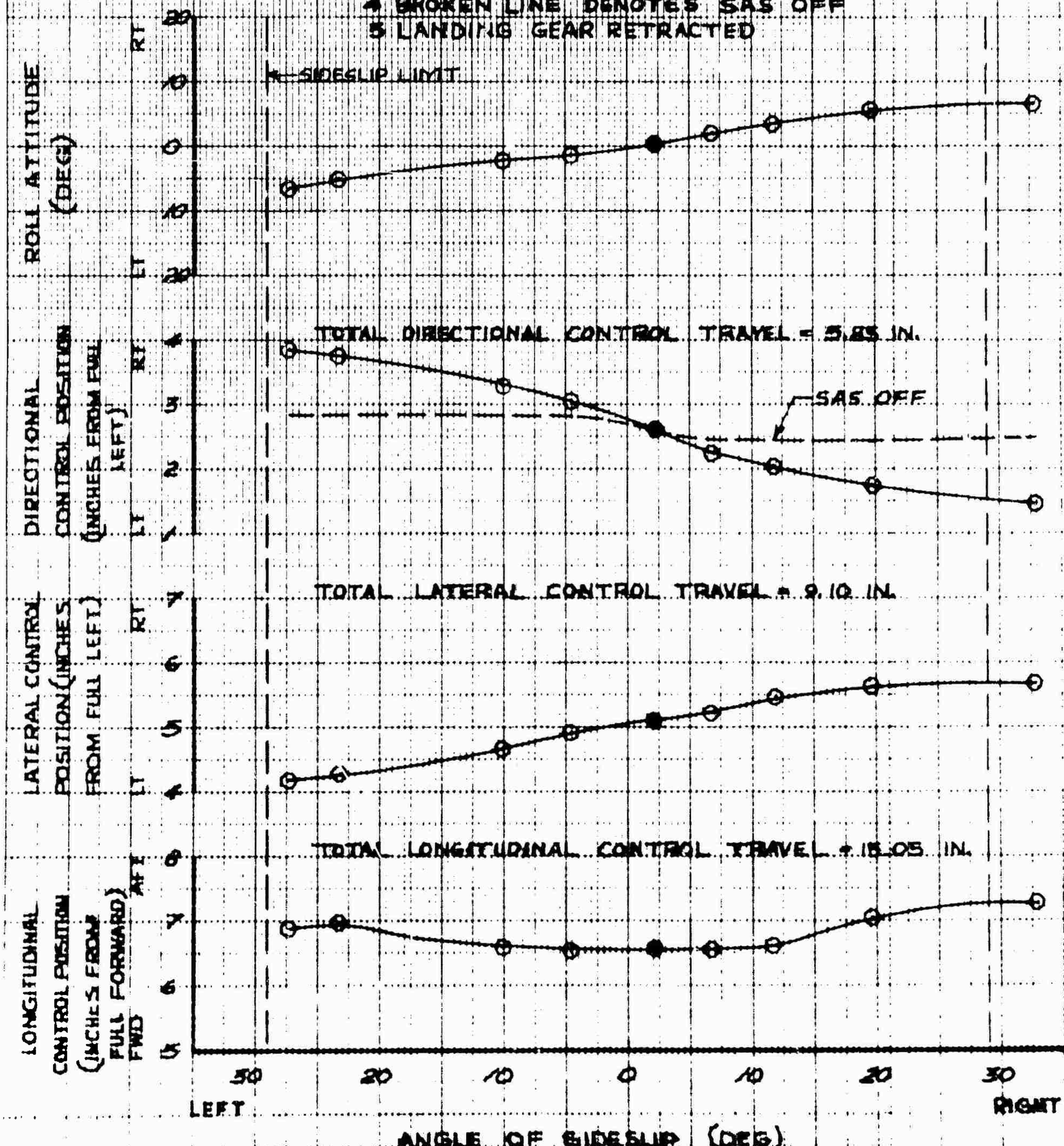


FIGURE 52 STATIC LATERAL DIRECTIONAL STABILITY

BOEING 347 S/N 65-7592

REEVALUATION

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG C G (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KEAS)	FLIGHT CONDITION
33715	7350	16.6	16.7 FWD	220.4	65.19	110	LEVEL

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 LANDING GEAR EXTENDED
5 BROKEN LINE DENOTES SAS OFF

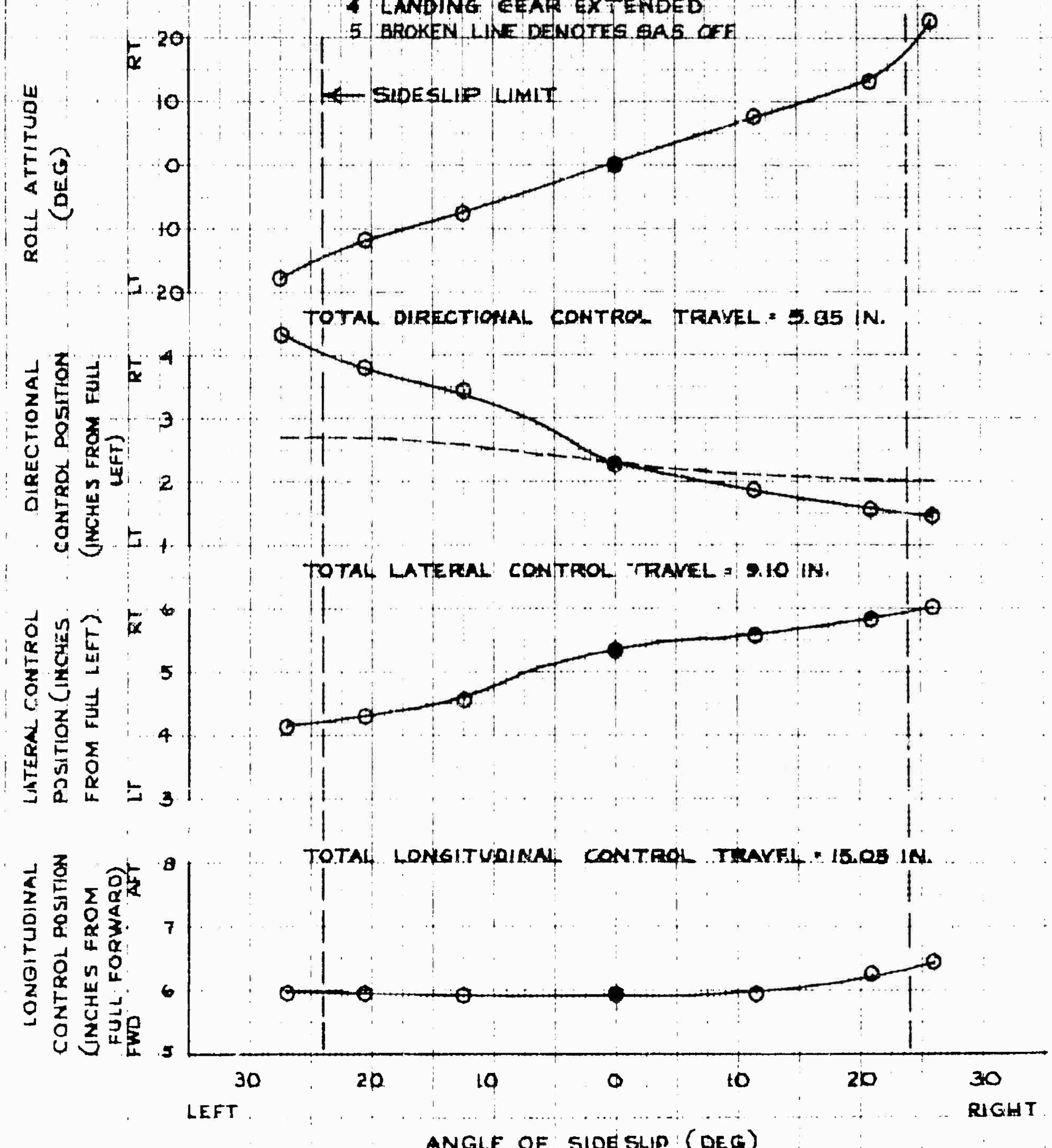


FIGURE 53 STATIC LATERAL DIRECTIONAL STABILITY

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
33530	5570	13.7	16.9 FWD	220	61.76	110	LEVEL

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

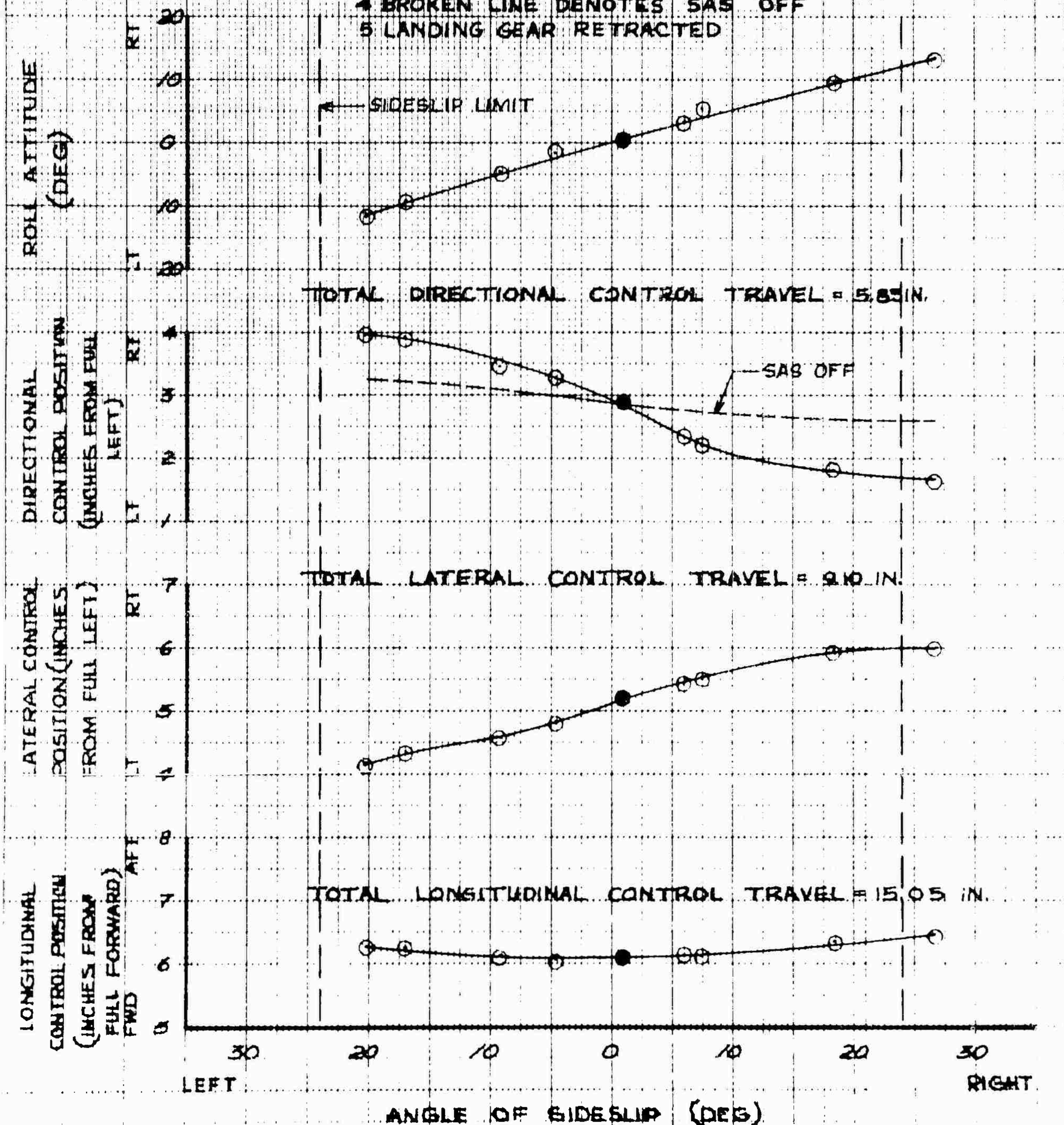


FIGURE 54 STATIC LATERAL DIRECTIONAL STABILITY.

BOEING 347 S/N 65-7992

AVG. GROSS WEIGHT (LB)	AVG. DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KTAS)	FLIGHT CONDITION
34220	5490	14.5	18.0 FWD	220	62.75	130	LEVEL

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

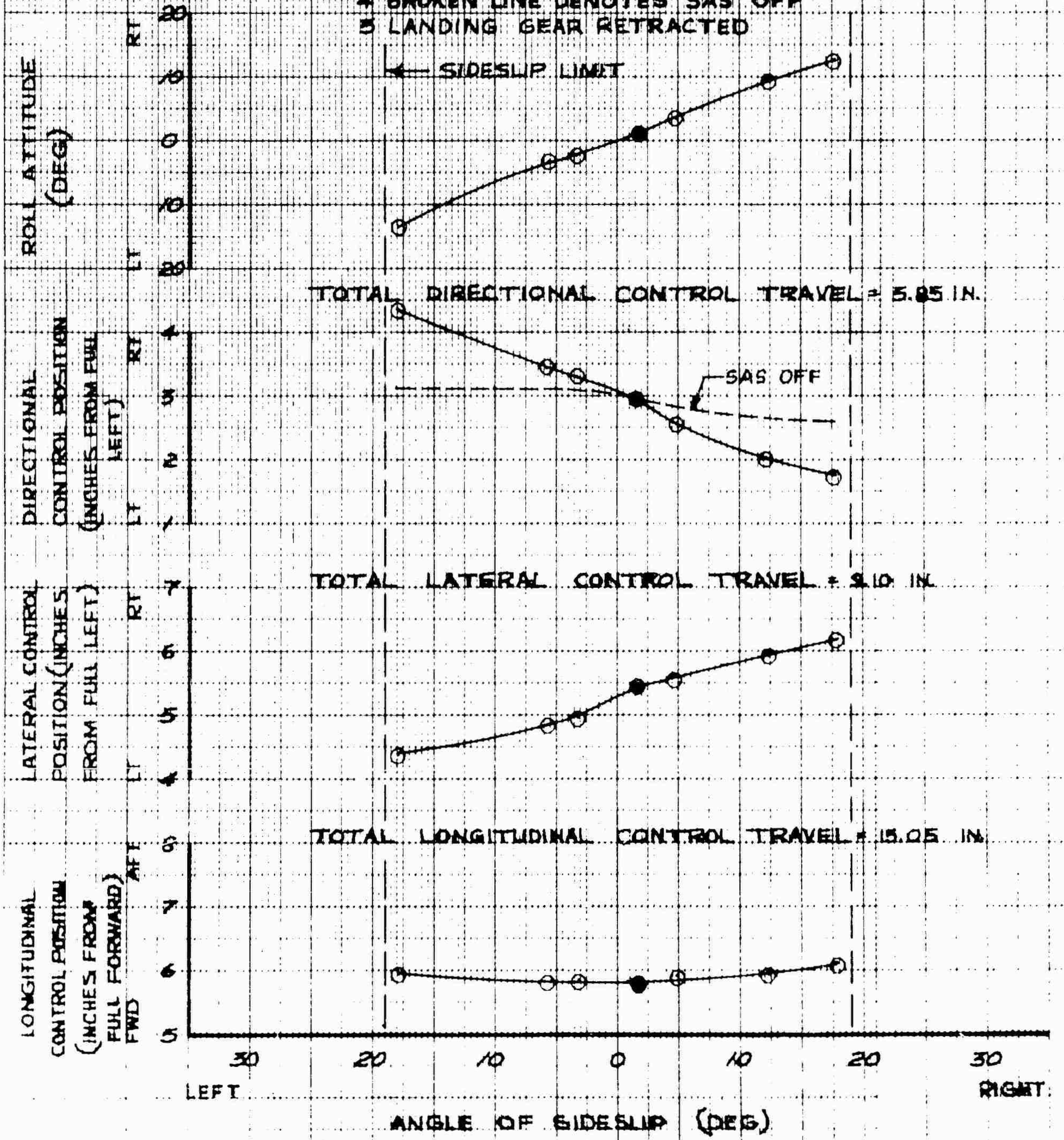


FIGURE 55 STATIC LATERAL DIRECTIONAL STABILITY
BOEING 347 S/N 65-7992

AVG. GROSS WEIGHT (LB)	AVG. DENSITY ALT (FT)	AVG OAT (°C)	AVG. CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
44330	4590	17.9	8.1 AFT	220	79.28	78	LEVEL

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

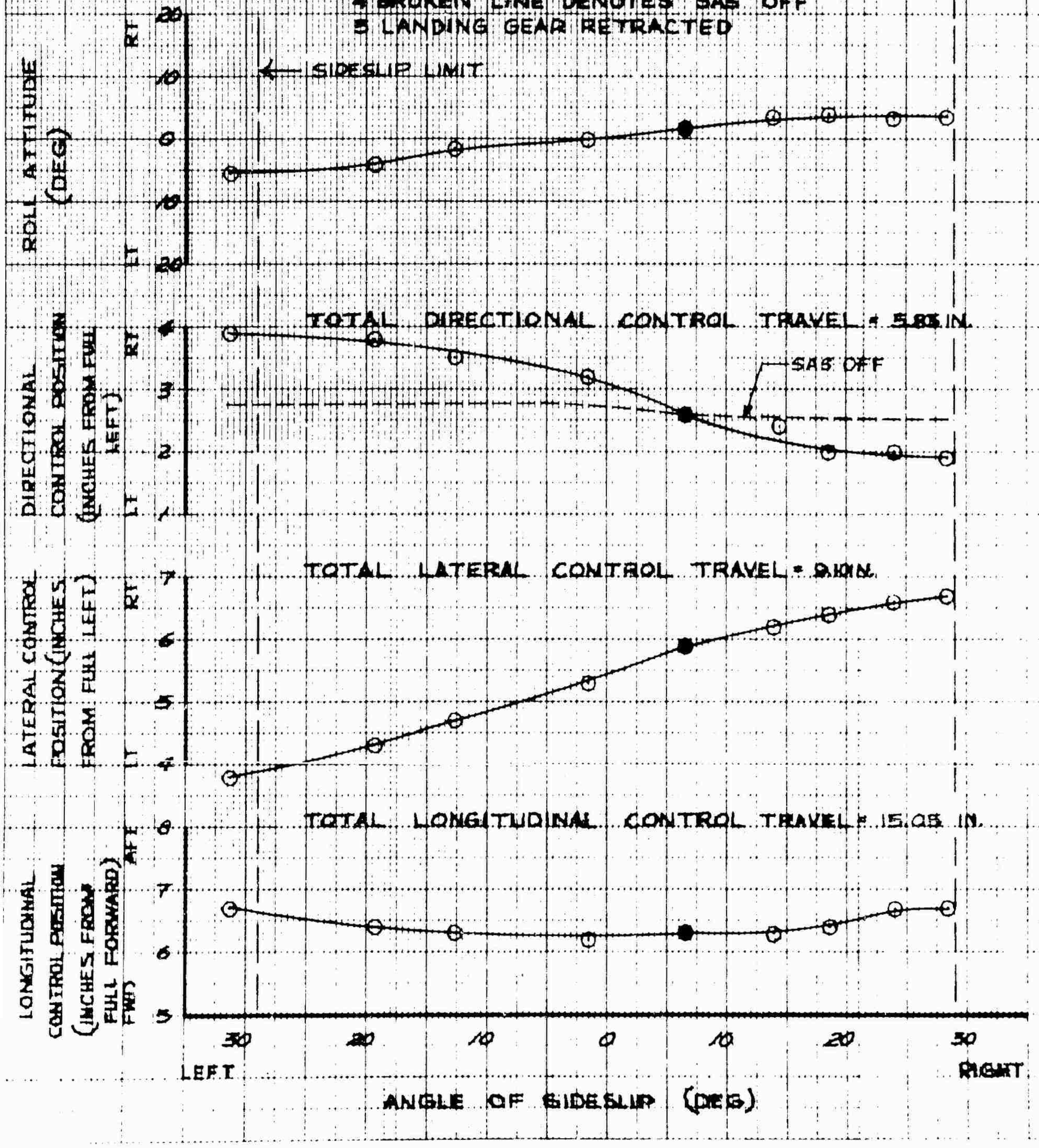


FIGURE 56 STATIC LATERAL DIRECTIONAL STABILITY

BOEING 347 S/N 65-7992

AVG. GROSS WEIGHT (LB)	AVG. DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
43300	4840	17.7	[illegible] AFT	220	7781	110	LEVEL

NOTES:
1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

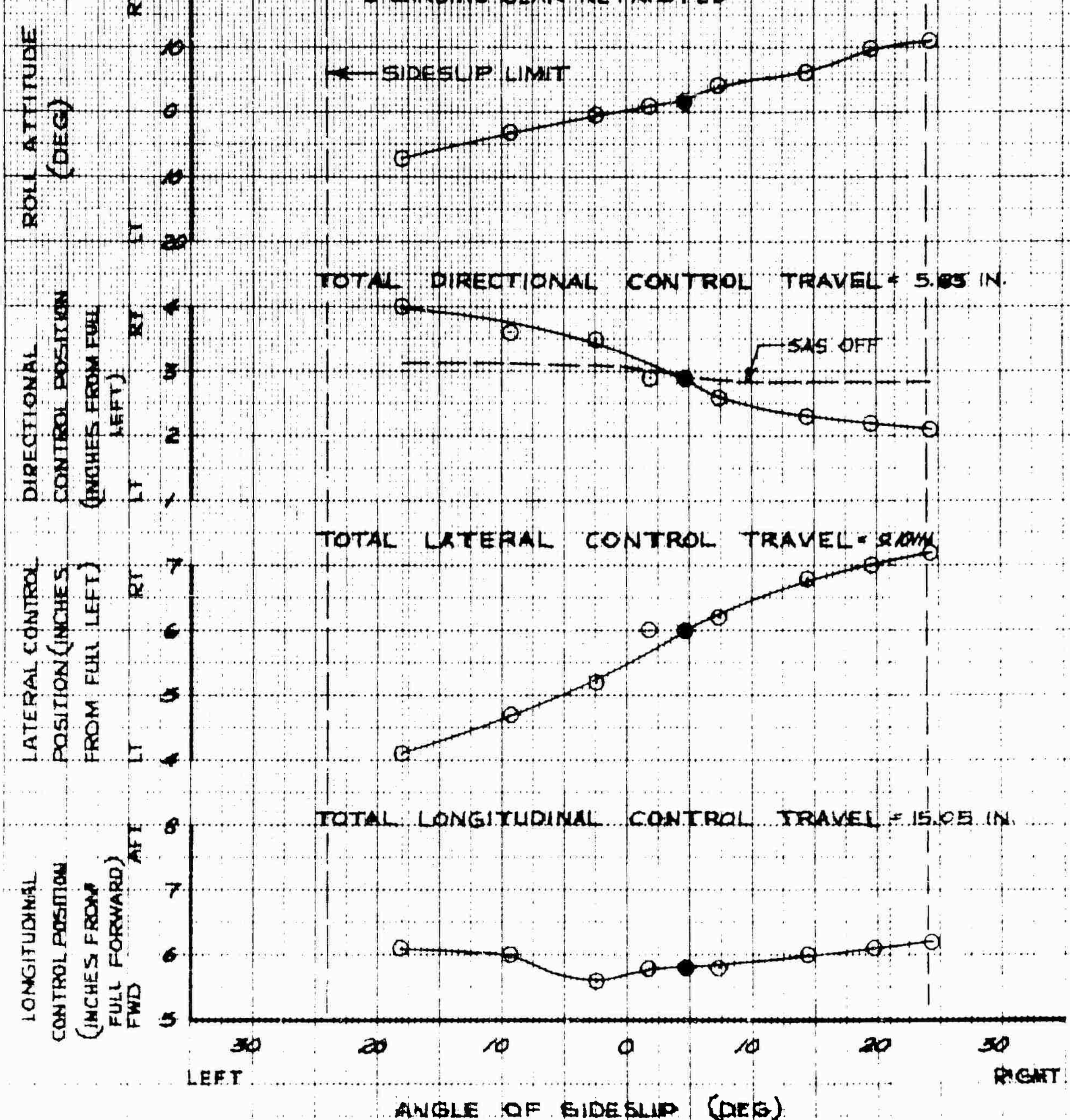


FIGURE 57 STATIC LATERAL DIRECTIONAL STABILITY.

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KIAS)	FLIGHT CONDITION
42630	4760	17.5	11.2 AFT	220	76.61	128	LEVEL

NOTES:
1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

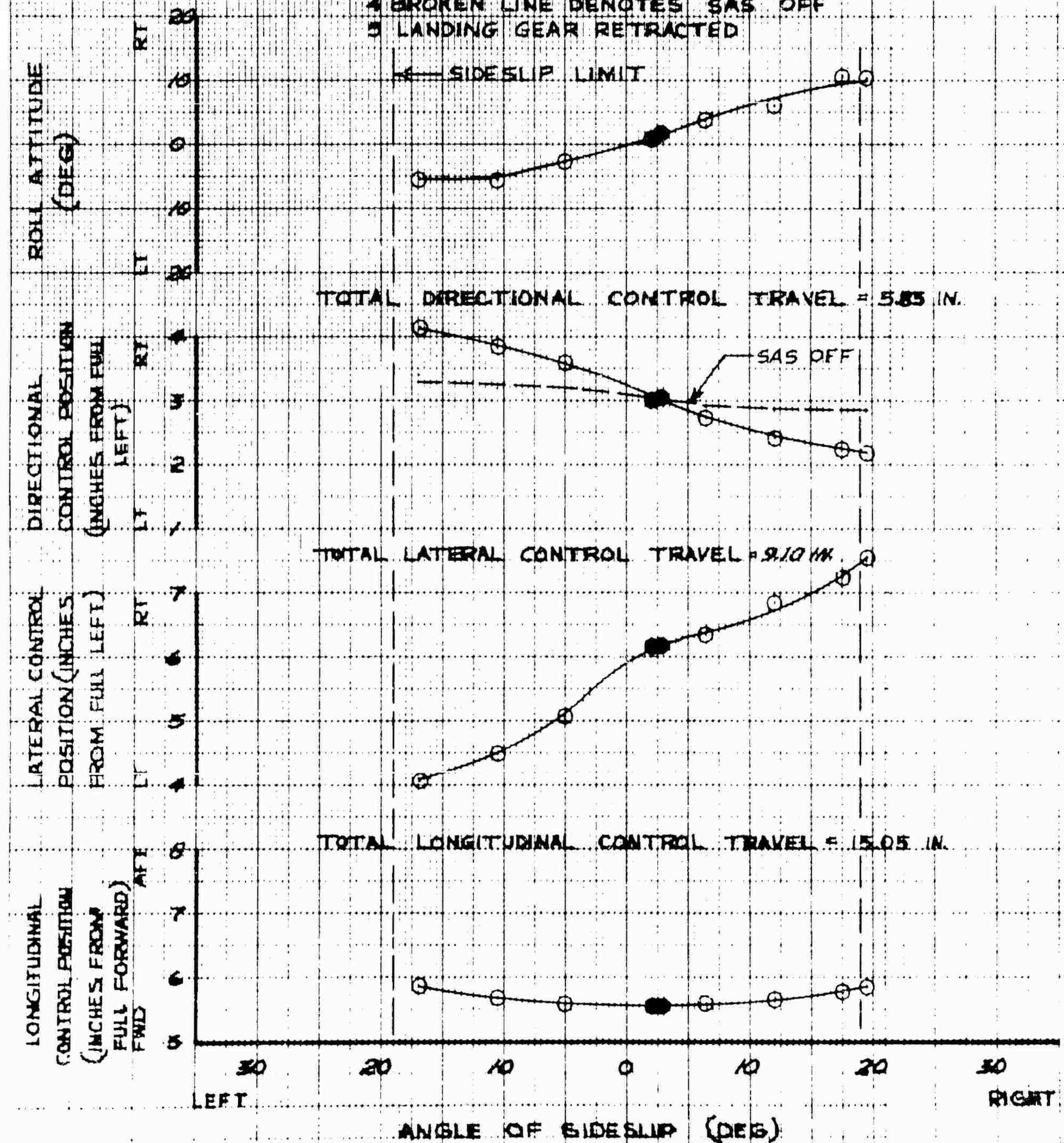


FIGURE 58 STATIC LATERAL DIRECTIONAL STABILITY

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
45140	6250	20.0	7.8 AFT	220	64.83	78	CLIMB AT NRP

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

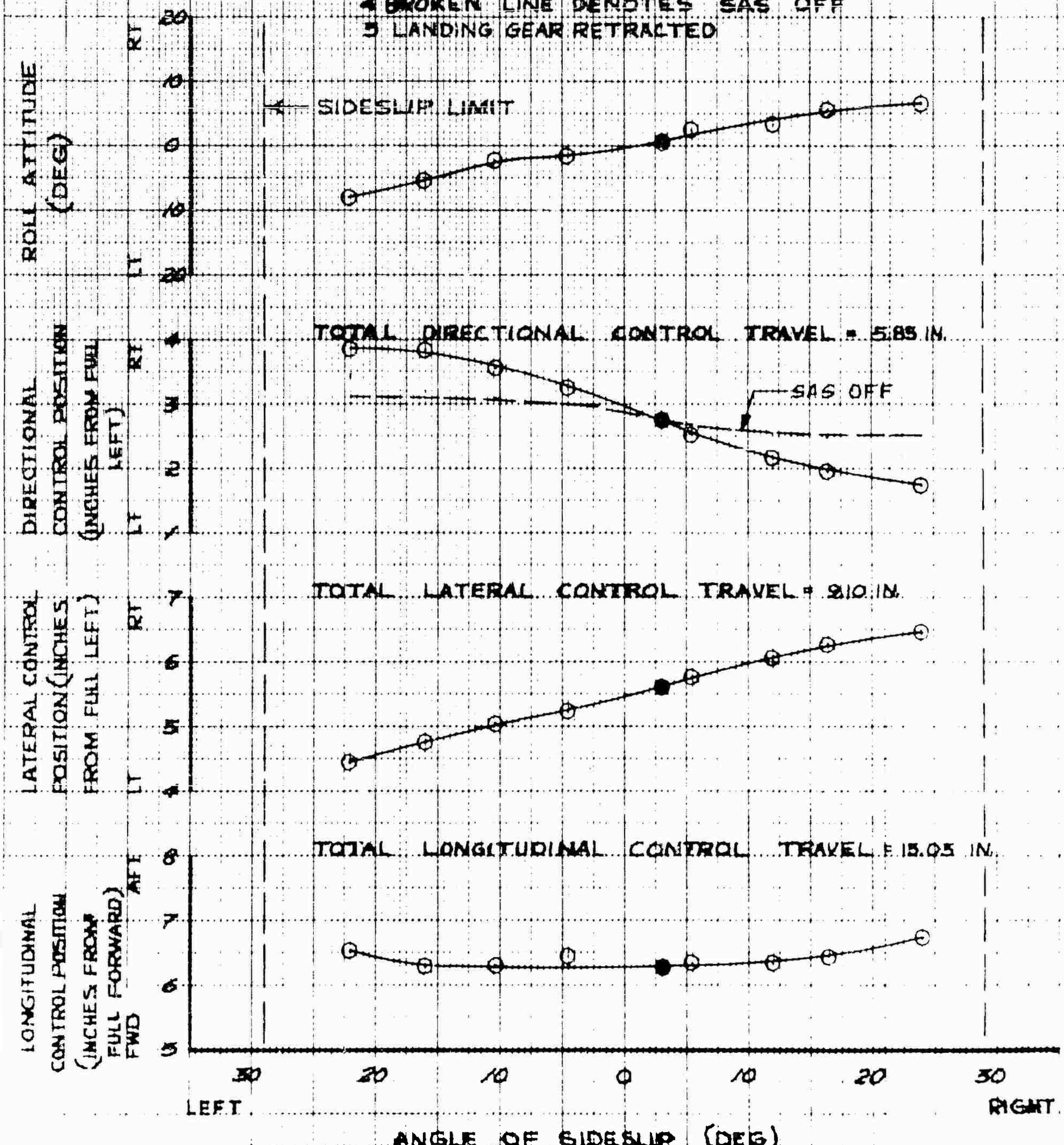


FIGURE 59 STATIC LATERAL DIRECTIONAL STABILITY

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)	FLIGHT CONDITION
45140	6250	20.0	7.8 AFT	220	81.14	73	DESCENT AT AUTO

NOTES: 1 DASH ON
2 SAS ON
3 SHADED SYMBOLS DENOTE TRIM POINTS
4 BROKEN LINE DENOTES SAS OFF
5 LANDING GEAR RETRACTED

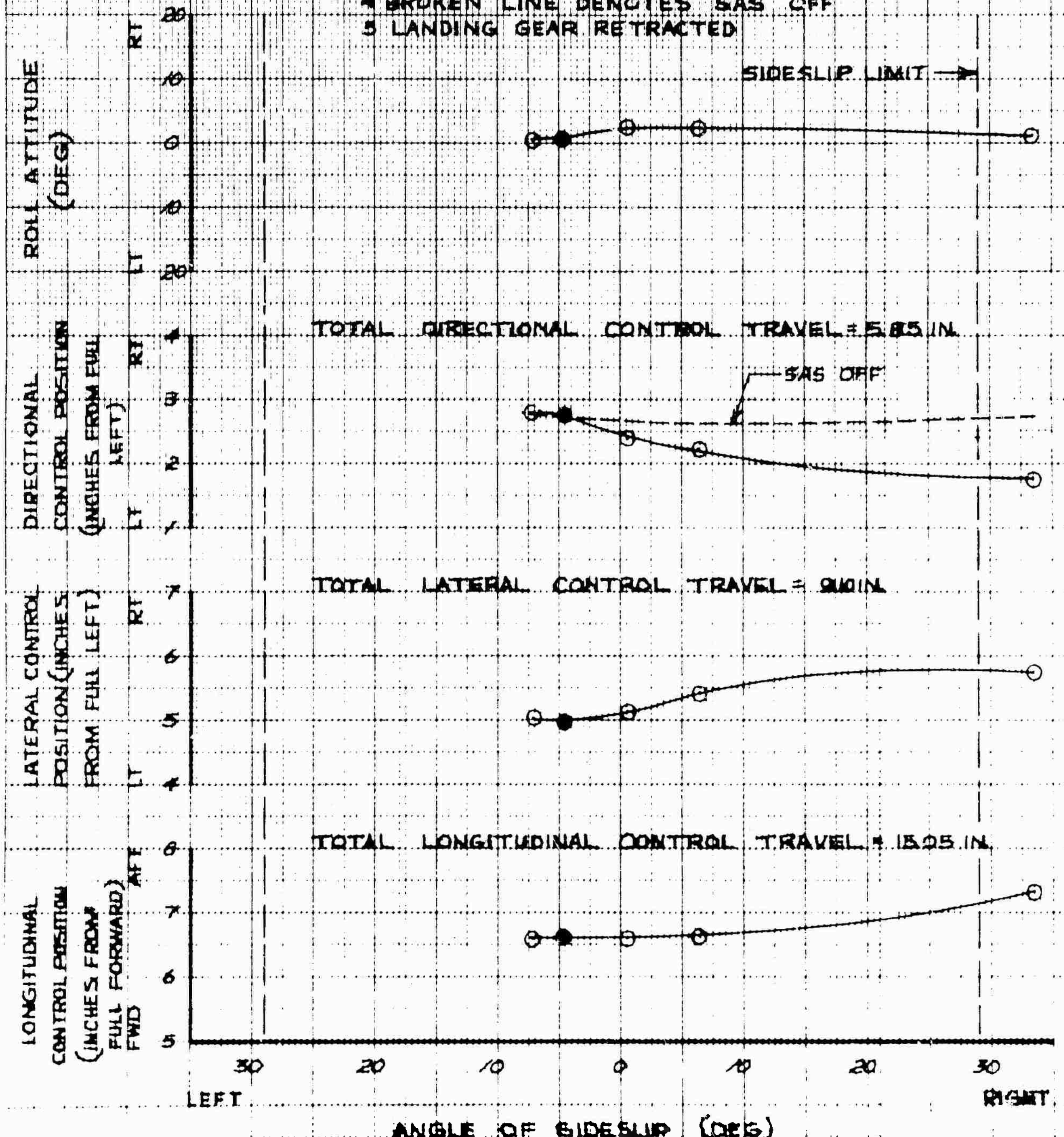
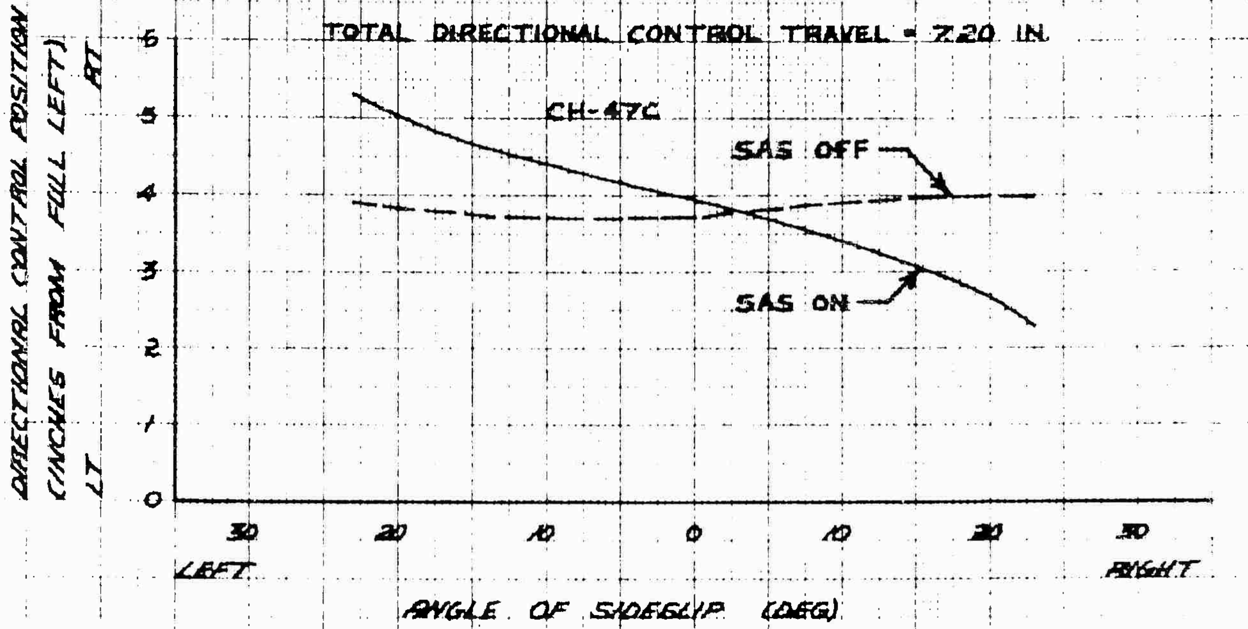


FIGURE 60. STATIC DIRECTIONAL STABILITY COMPARISON

CH-47C USA S/N 66-19121

BOEING 347 S/N 65-7992

	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF (C_T x 10^4)	TRIM A/S (KCAS)	FLIGHT CONDITION
CH-47C	42560	4490	6.1	6.2 AFT	246	60.58	115	LEVEL
BOEING 347	43300	4840	17.7	9.9 AFT	220	77.81	110	LEVEL



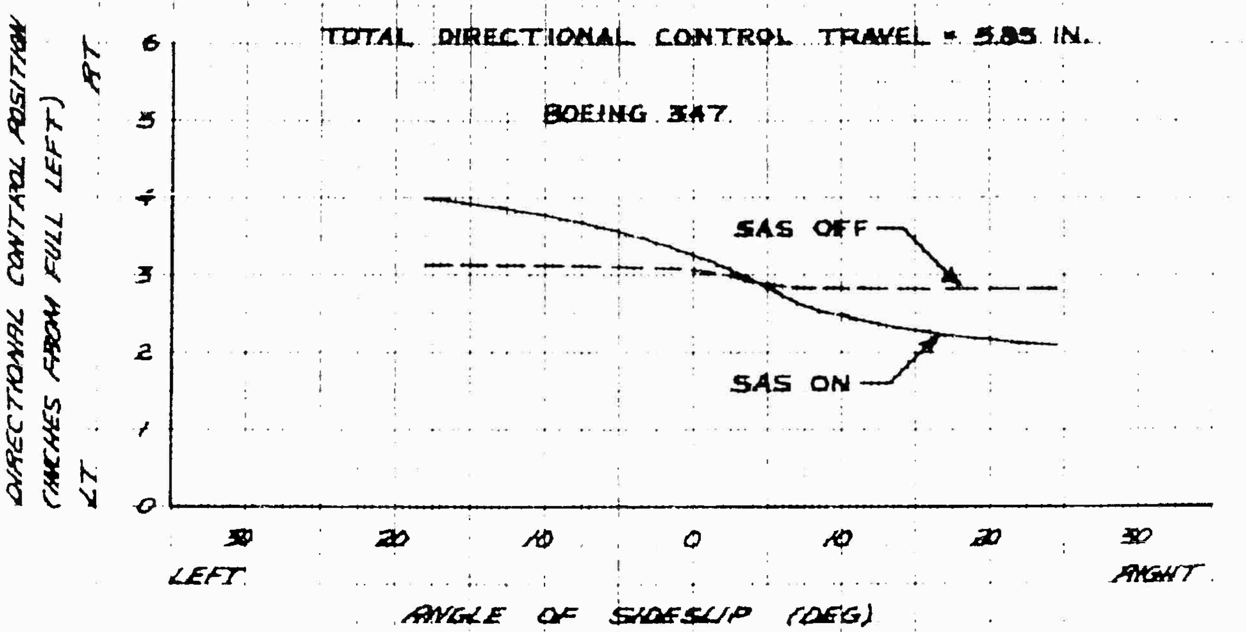


FIGURE 61 AIRCRAFT RESPONSE FOLLOWING RIGHT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45070	1820	27.7	7.7 AFT	220	74.16	30

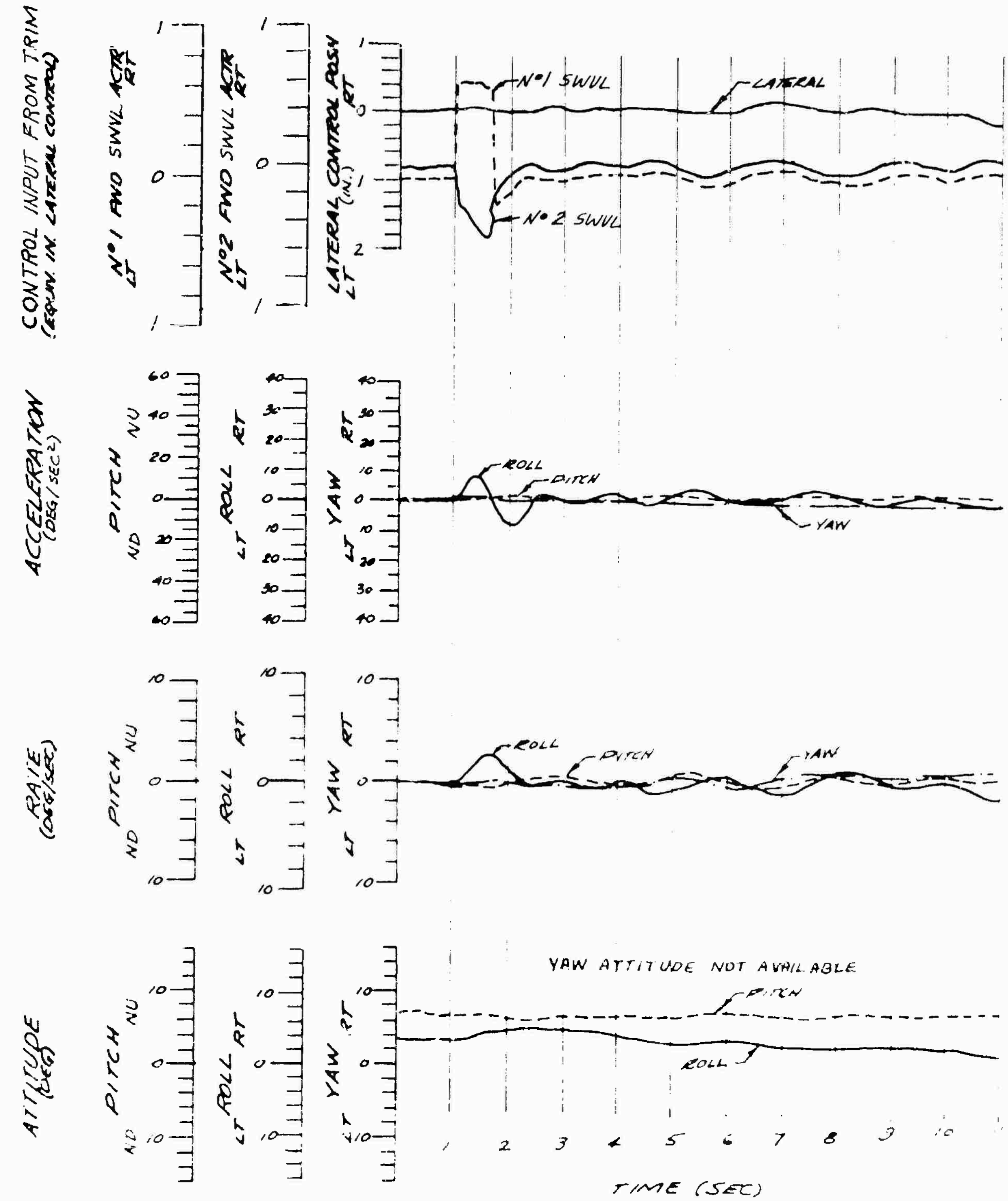


FIGURE 62 AIRCRAFT RESPONSE FOLLOWING LEFT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	78

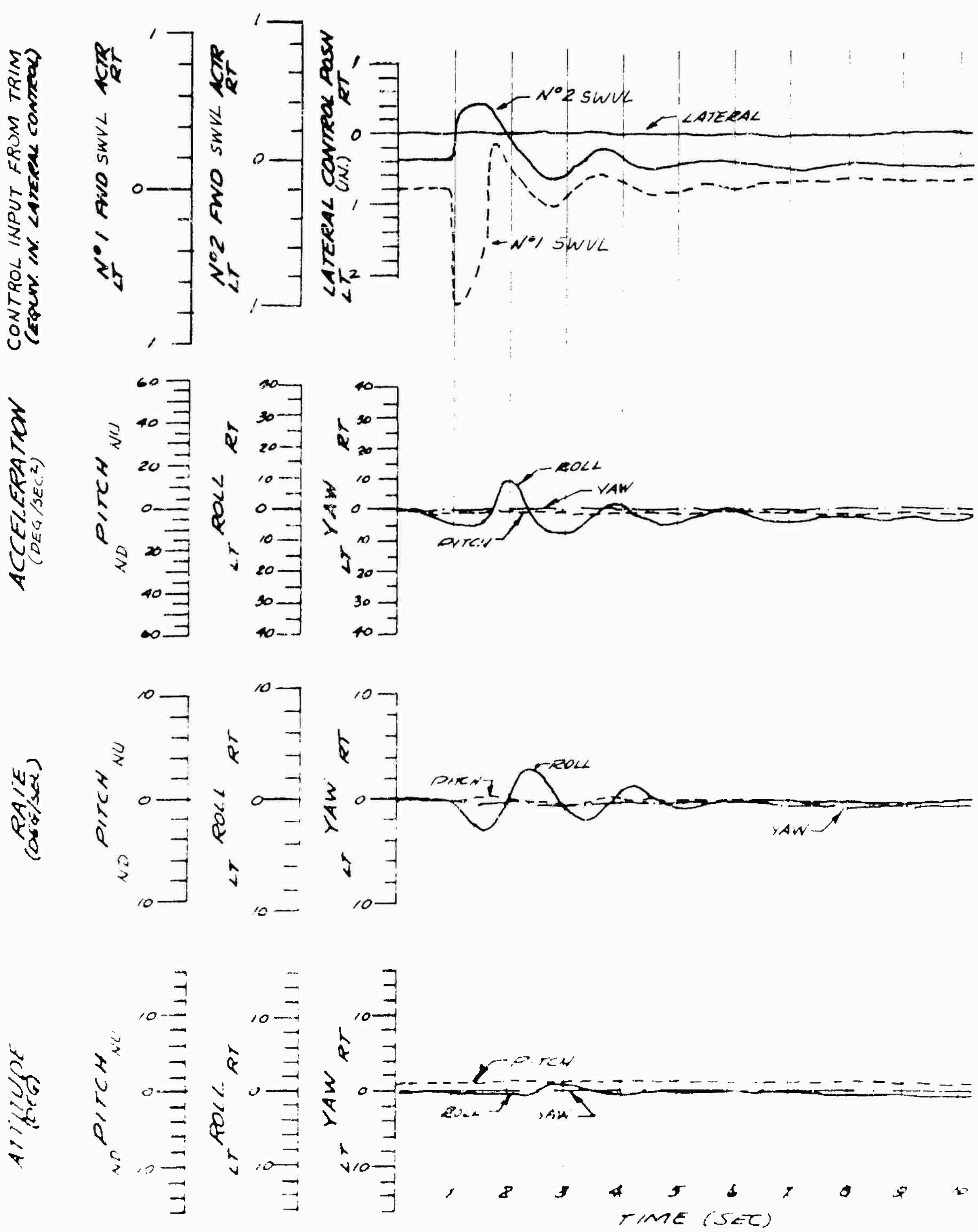


FIGURE 63 AIRCRAFT RESPONSE FOLLOWING LEFT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	111

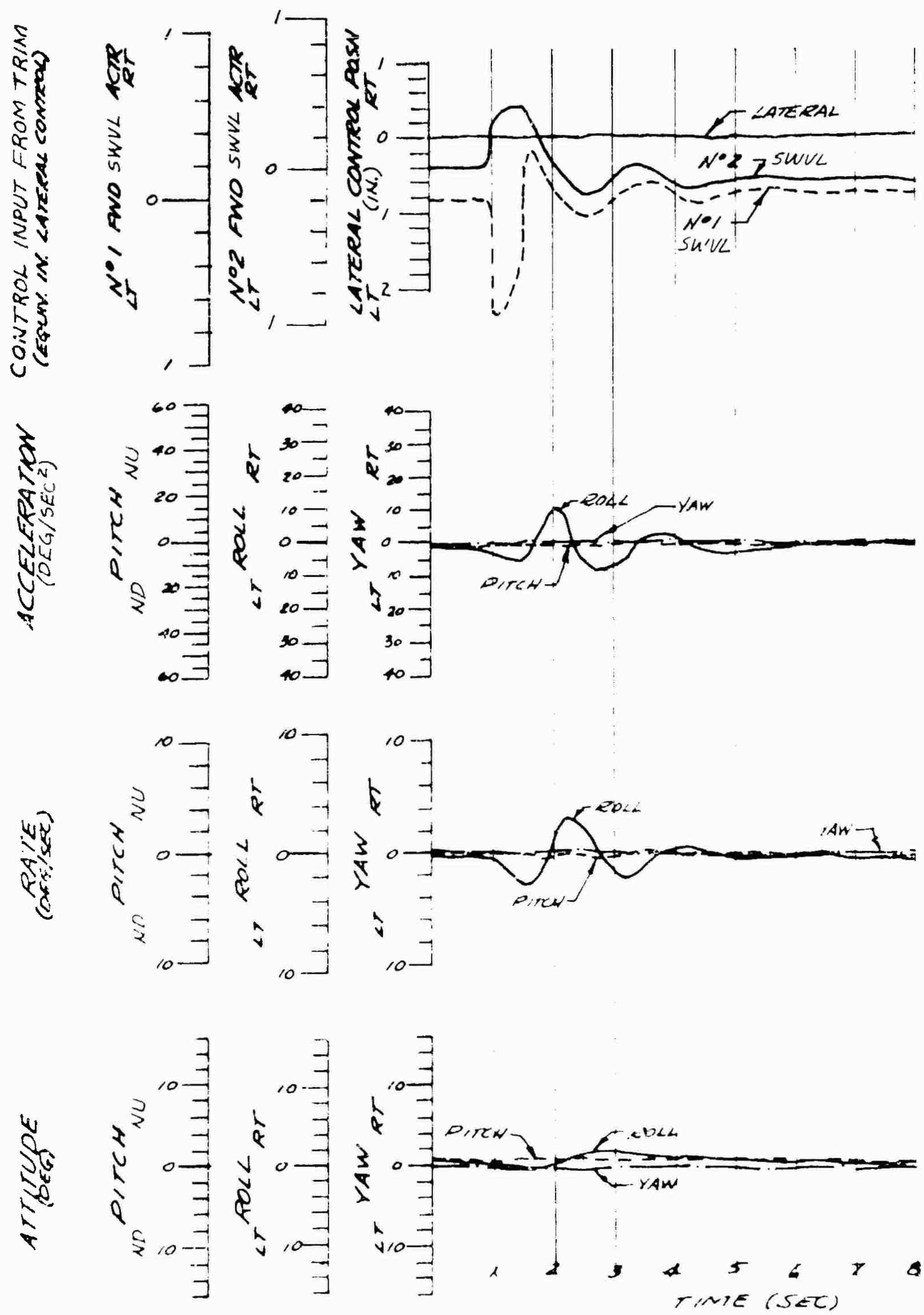


FIGURE 64 AIRCRAFT RESPONSE FOLLOWING LEFT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	7973	129

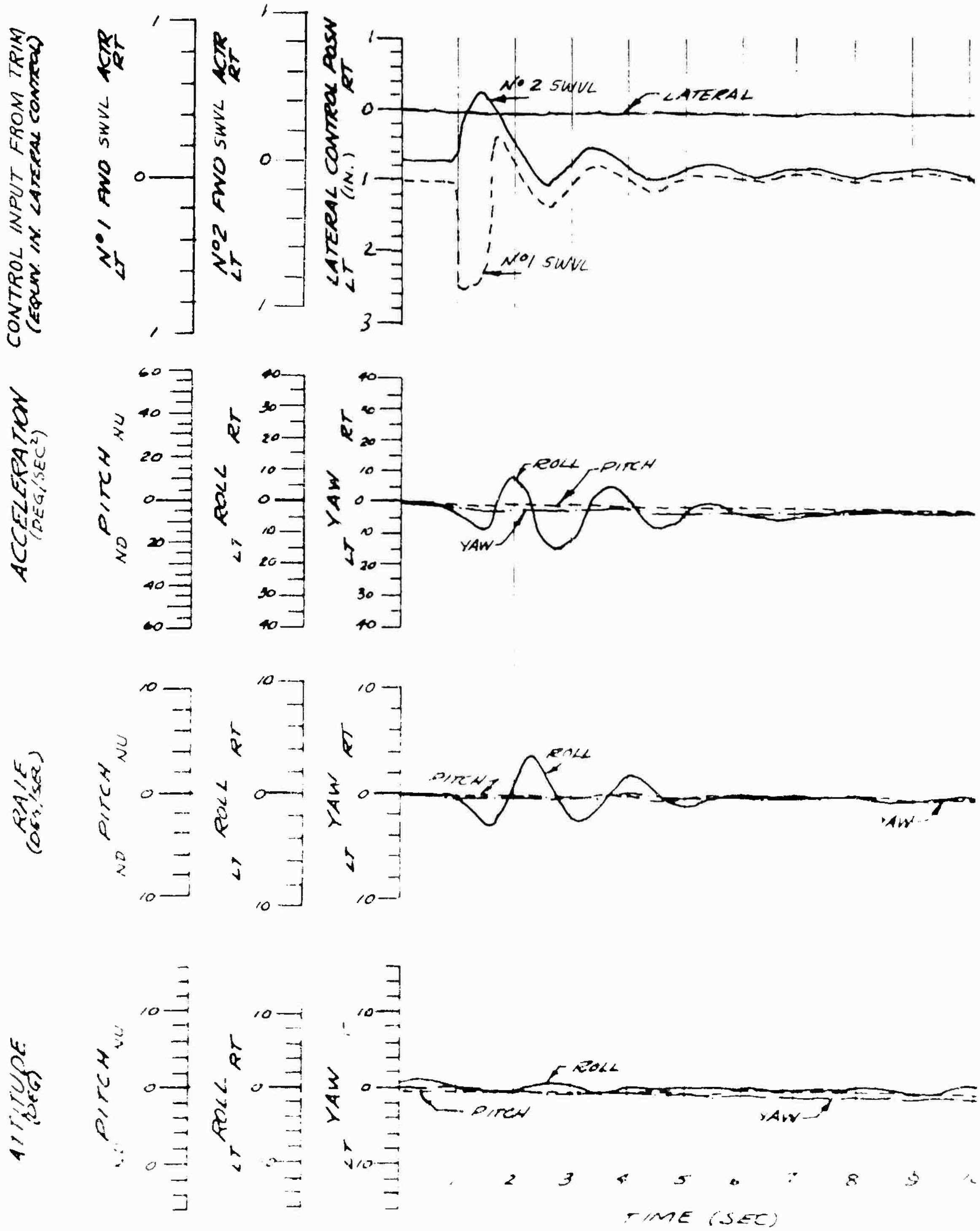


FIGURE 65 AIRCRAFT RESPONSE FOLLOWING RIGHT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45290	5040	16.0	7.8 AFT	220	81.79	78

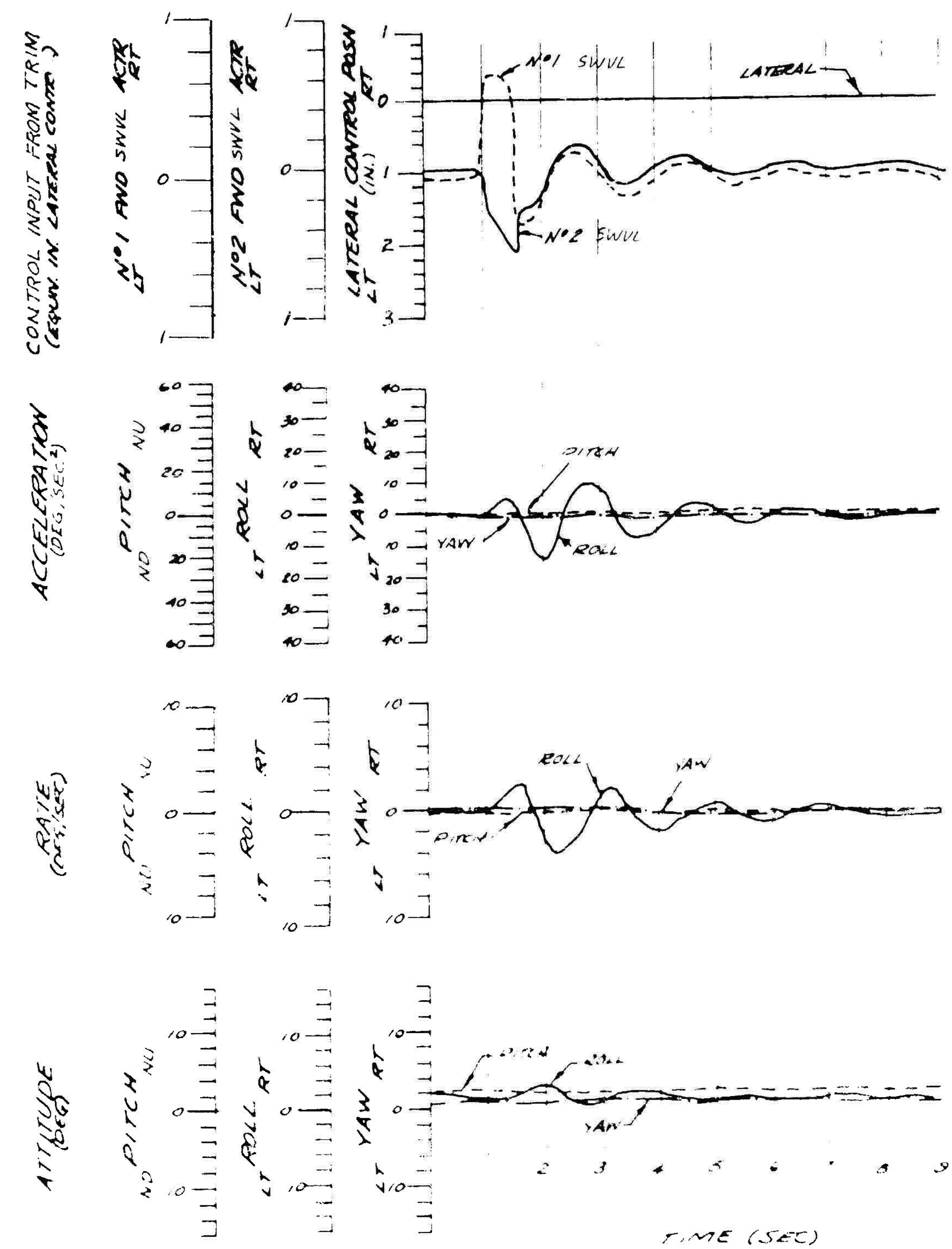


FIGURE 66 AIRCRAFT RESPONSE FOLLOWING LEFT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9 AFT	220.5	96.25	78

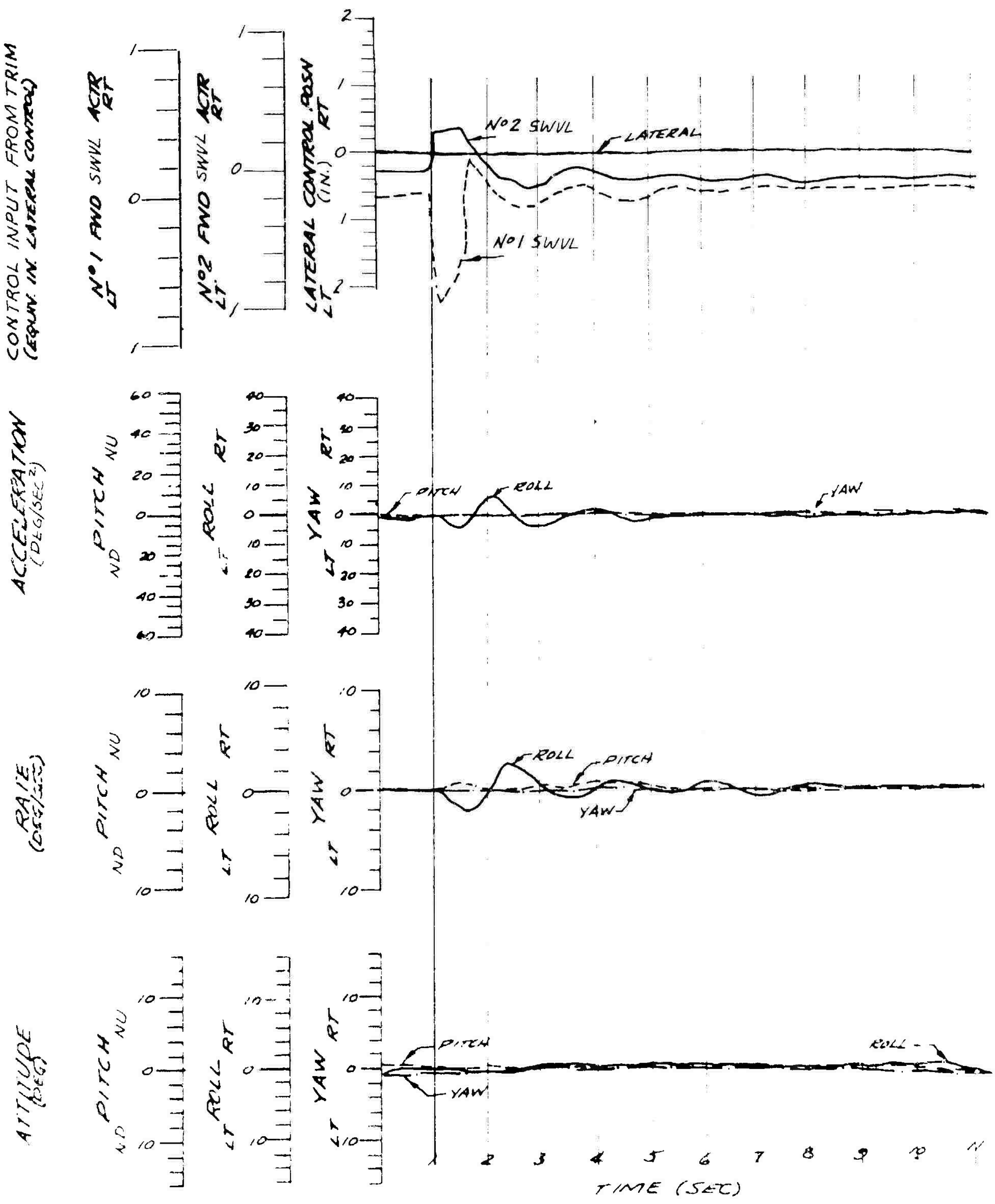


FIGURE 67 AIRCRAFT RESPONSE FOLLOWING LEFT LATERAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9.1 AFT	220	96.25	111

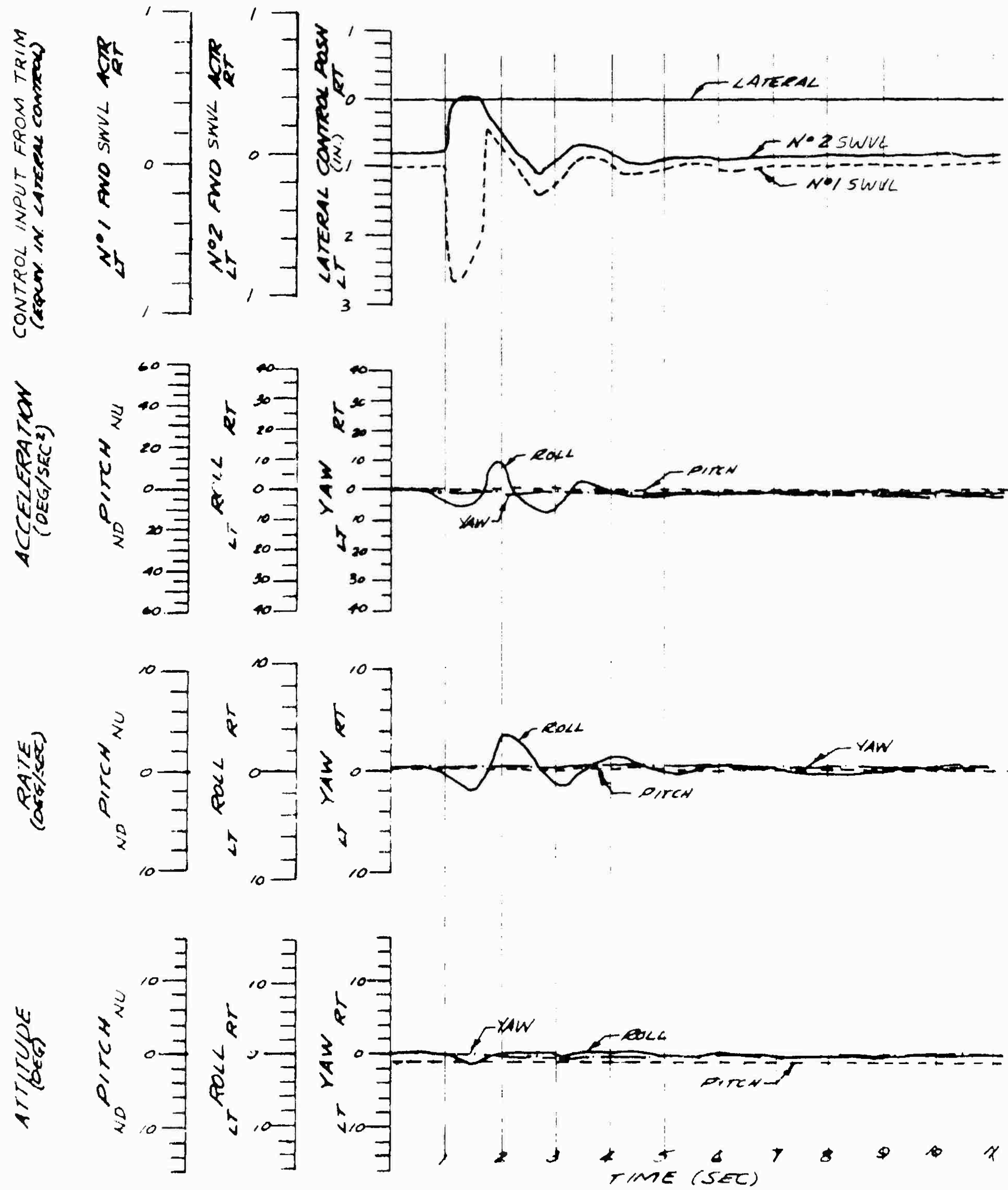


FIGURE 68 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45070	1820	27.7	7.7 AFT	220	74.16	35

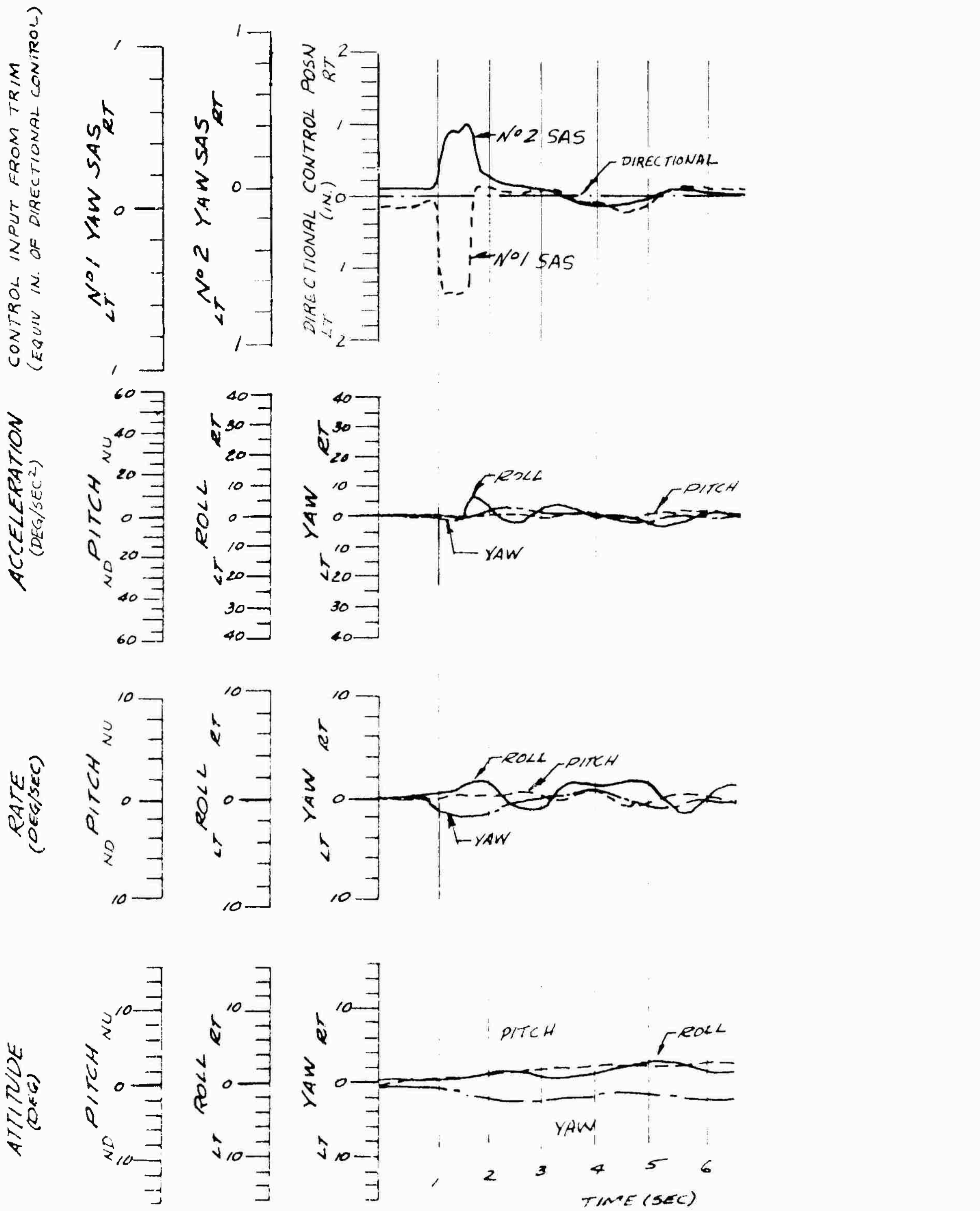


FIGURE 69 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	78

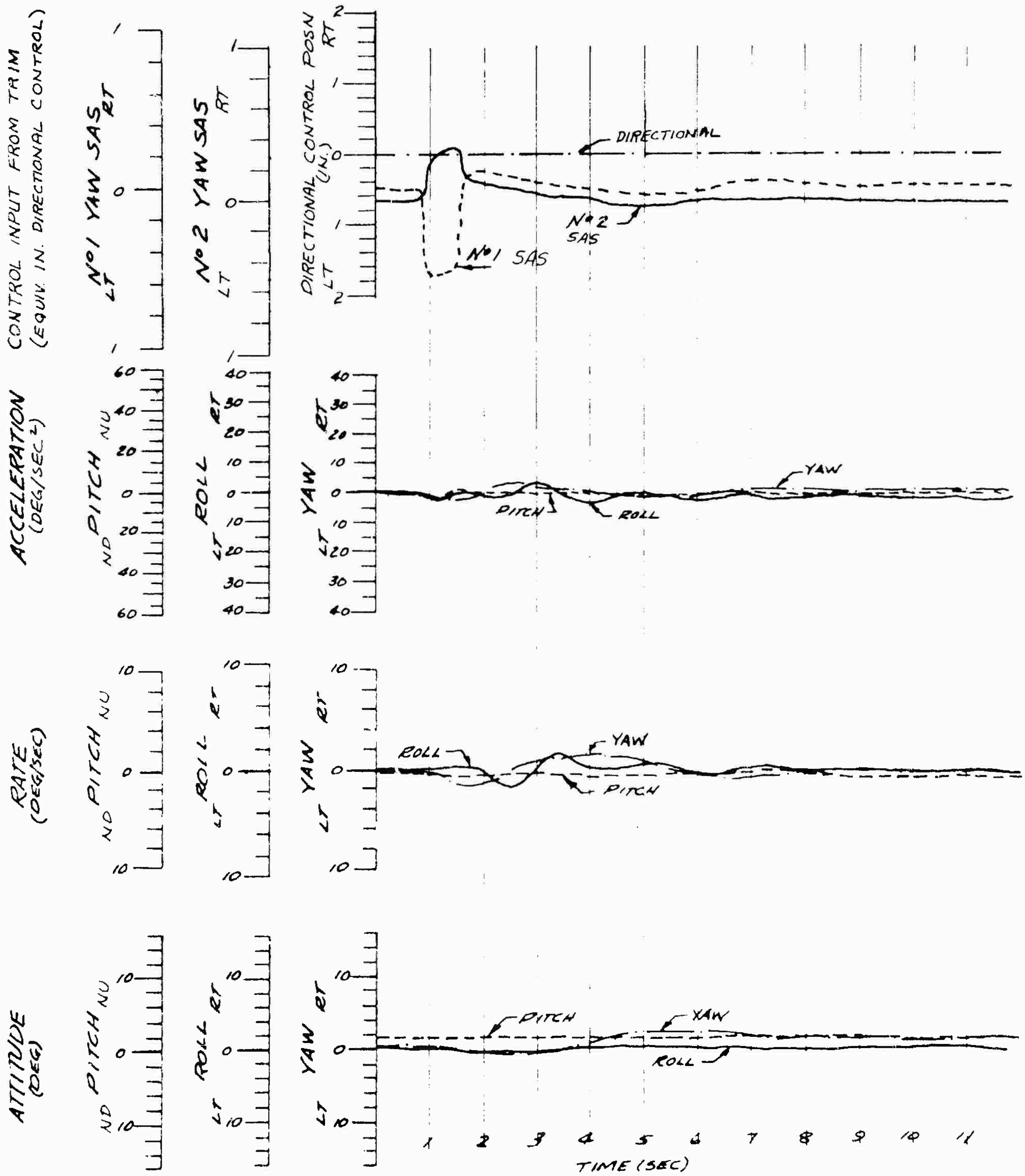


FIGURE 70 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	111

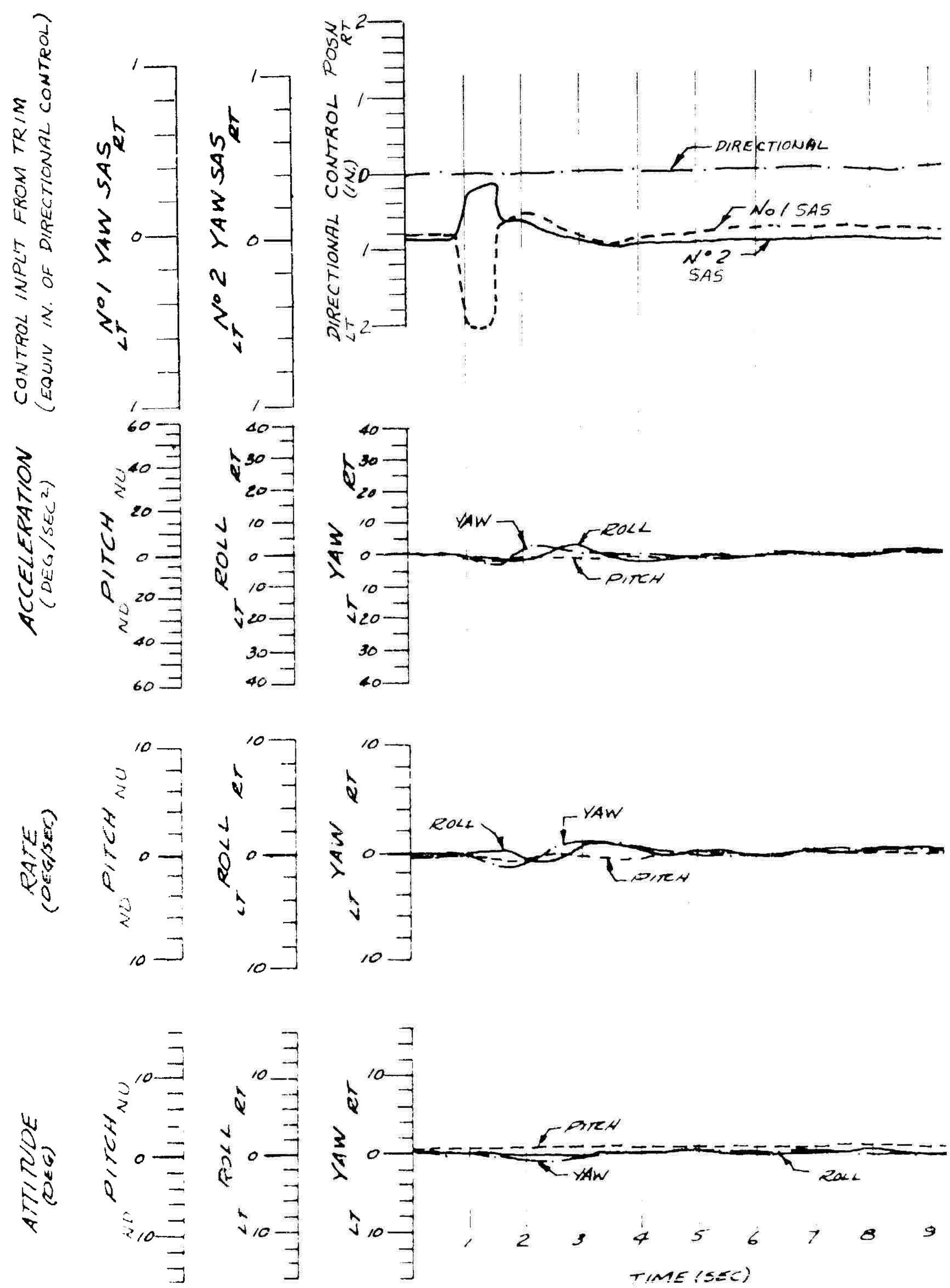


FIGURE 71 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44150	5000	20.0	24.0 FWD	220	79.73	129

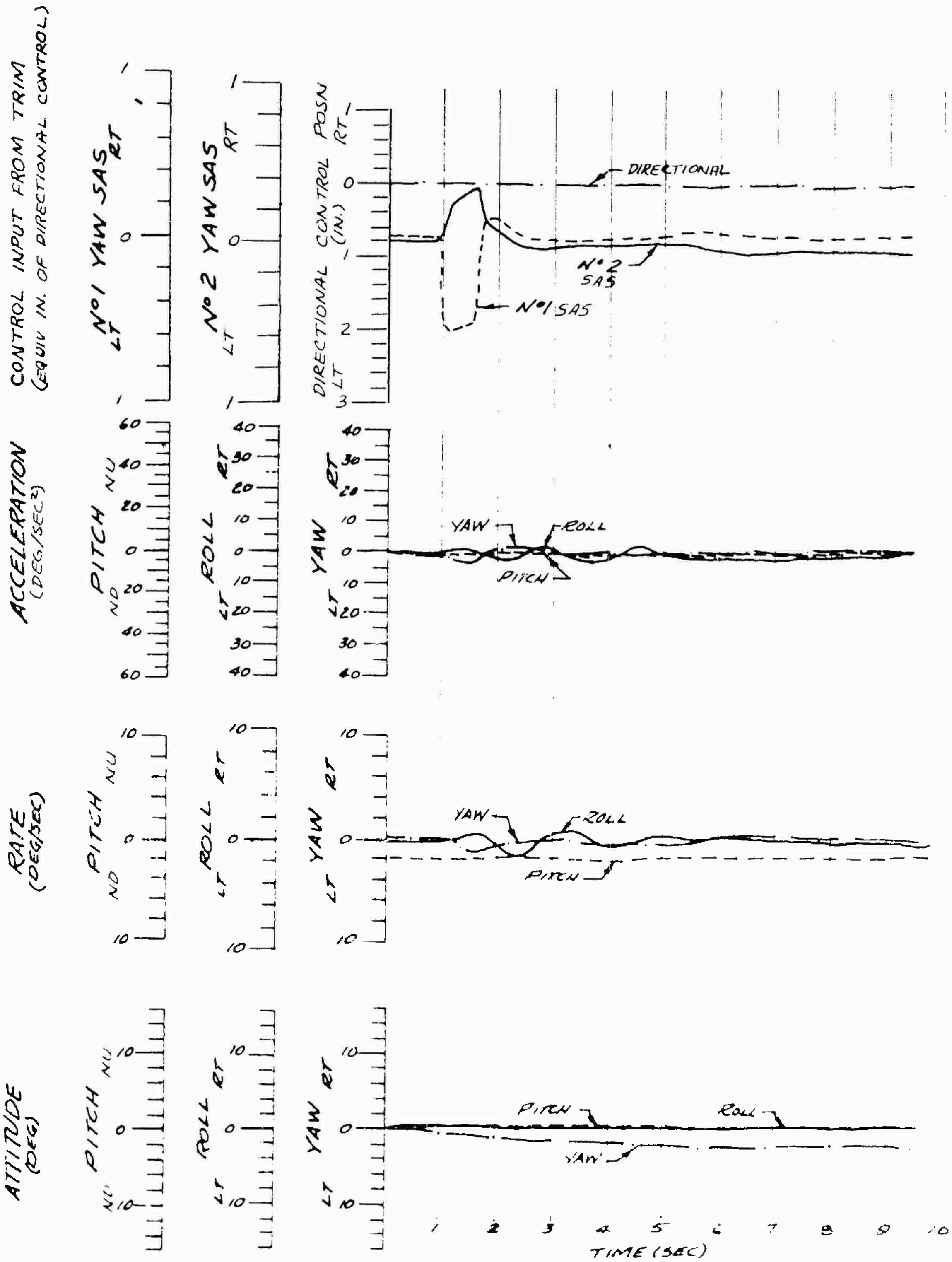


FIGURE 72 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL PULSE

BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9.1 AFT	220	96.25	78

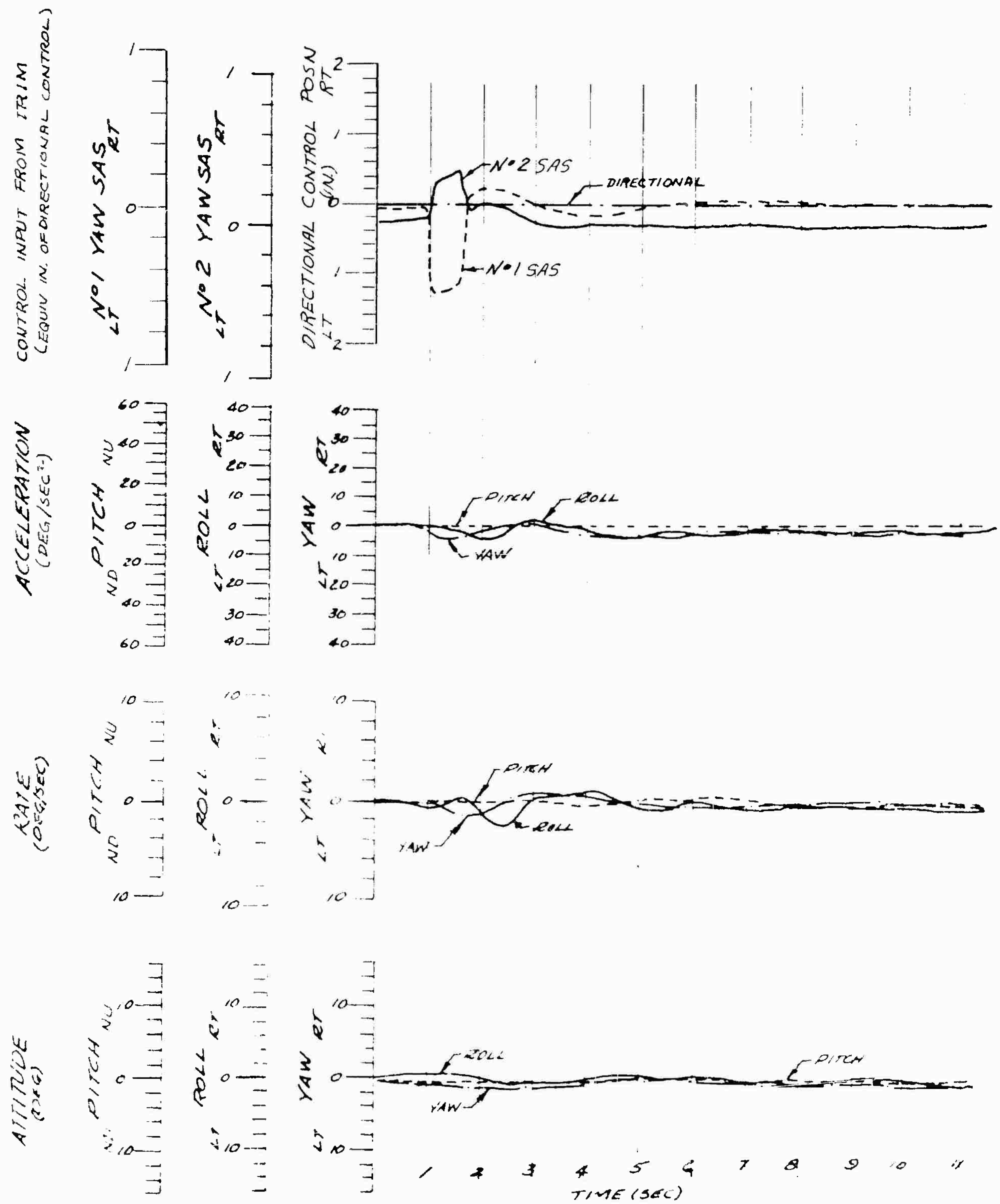


FIGURE 73 AIRCRAFT RESPONSE FOLLOWING LEFT DIRECTIONAL PULSE
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	C.G. (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9.1 AFT	220	96.25	111

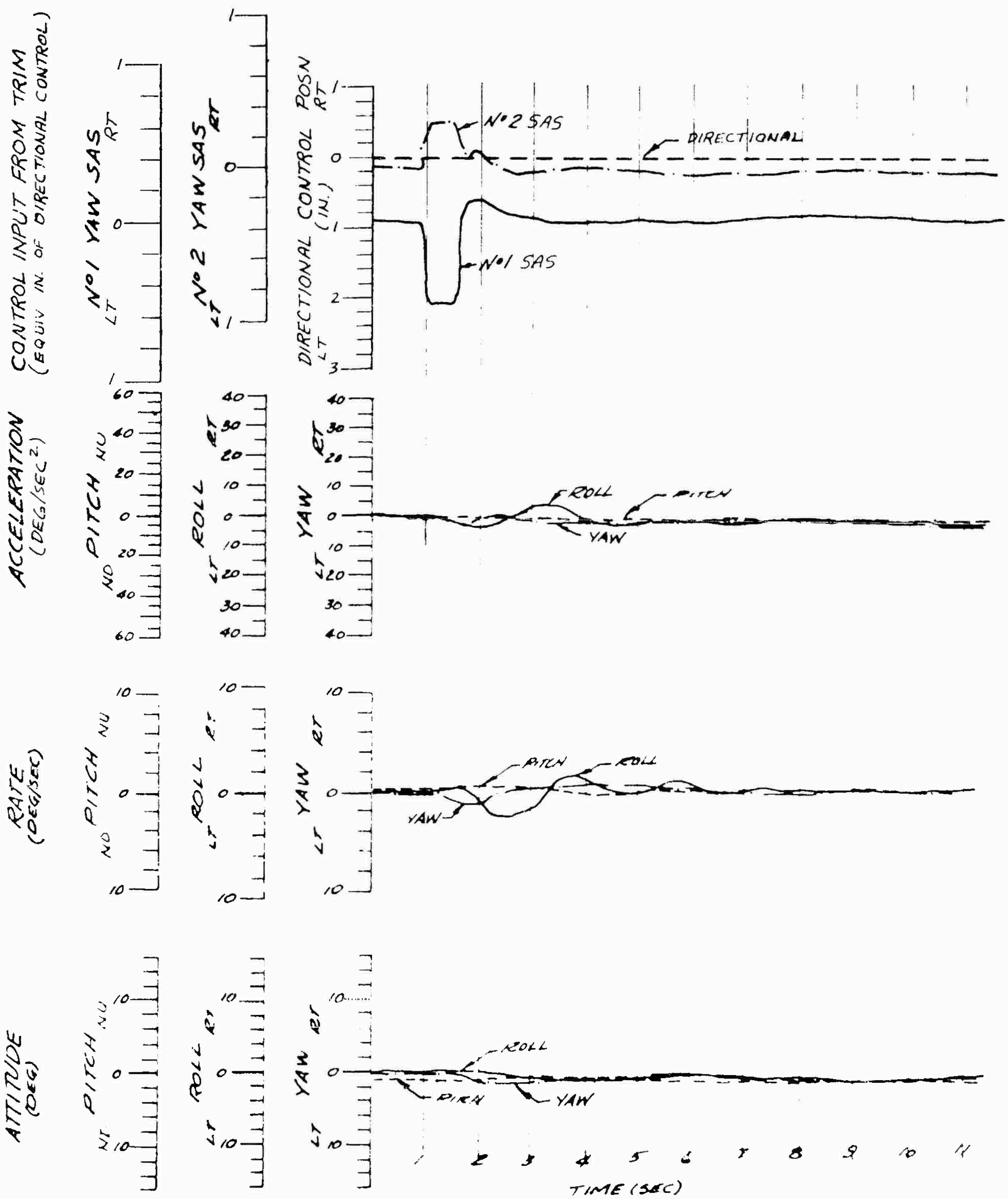


FIGURE 74 MANEUVERING STABILITY

BOEING 347 S/N 65-7992

SYM	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF (C_T X 10^4)	TRIM A/S (KCAS)
⊙	43000	5200	12.0	8.0 AFT	220	78.19	78
⊡	43000	5200	12.0	8.0 AFT	220	78.19	130

NOTES: 1 SAS ON
2 DASH ON
3 STANDARD SYMBOL DENOTES STEADY PULL UP
4 SHADED SYMBOL DENOTES RIGHT TURN
5 TAILS DENOTE LEFT TURN
6 LANDING GEAR RETRACTED

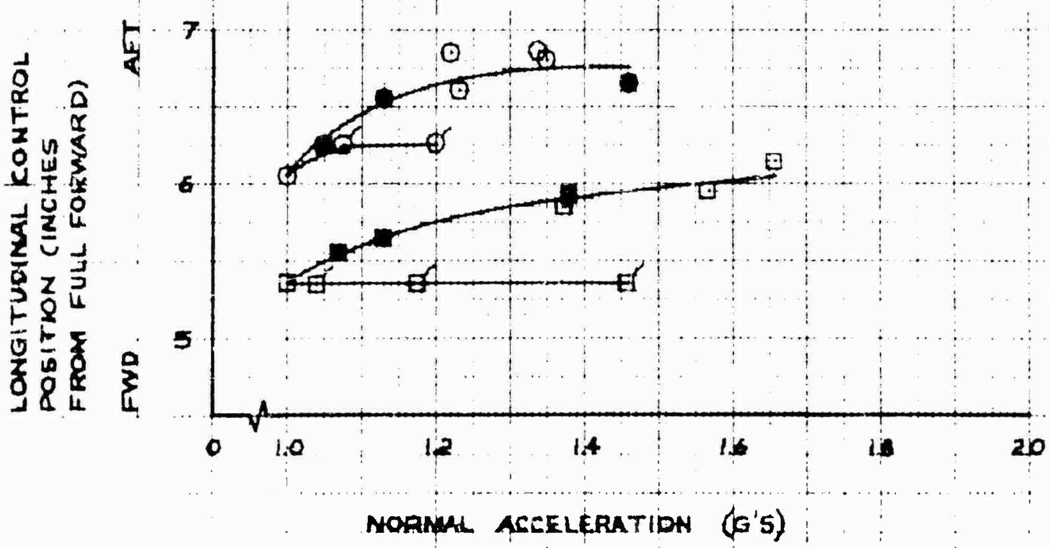
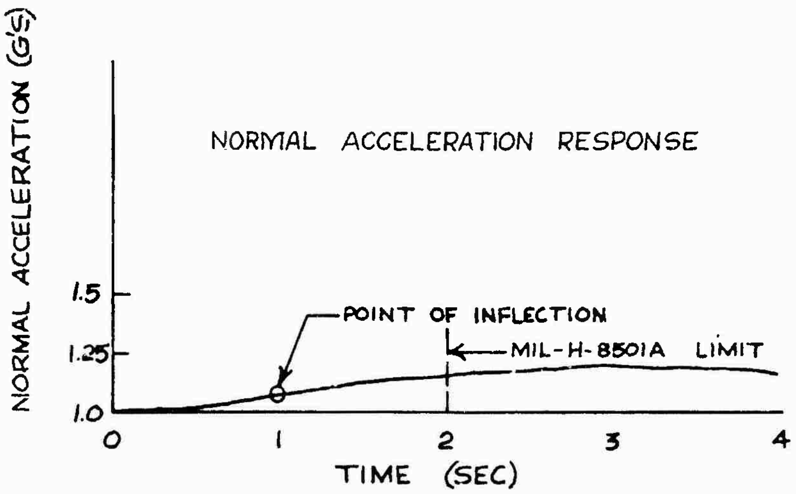


FIGURE 75. AIRCRAFT RESPONSE FOLLOWING AFT LONGITUDINAL CONTROL STEP INPUT.

BOEING 347 S/N 65-7992

AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
45670	5220	21.7	8.0 AFT	220	83.28	75

NOTE : LONGITUDINAL CONTROL STEP INPUT = 1.15 INCHES



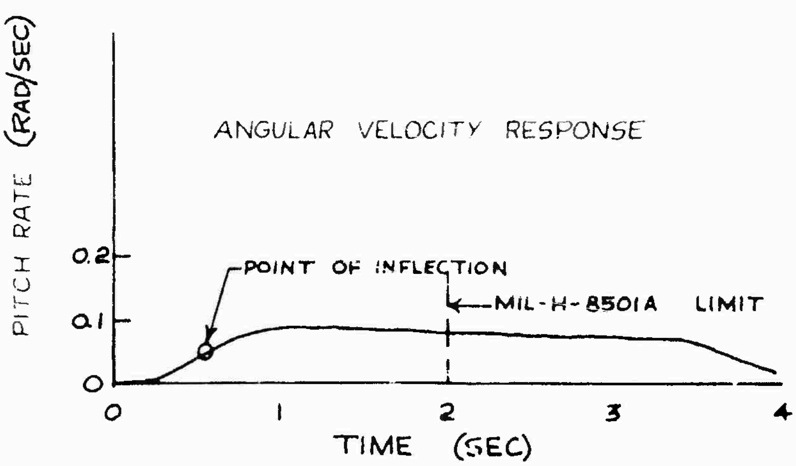
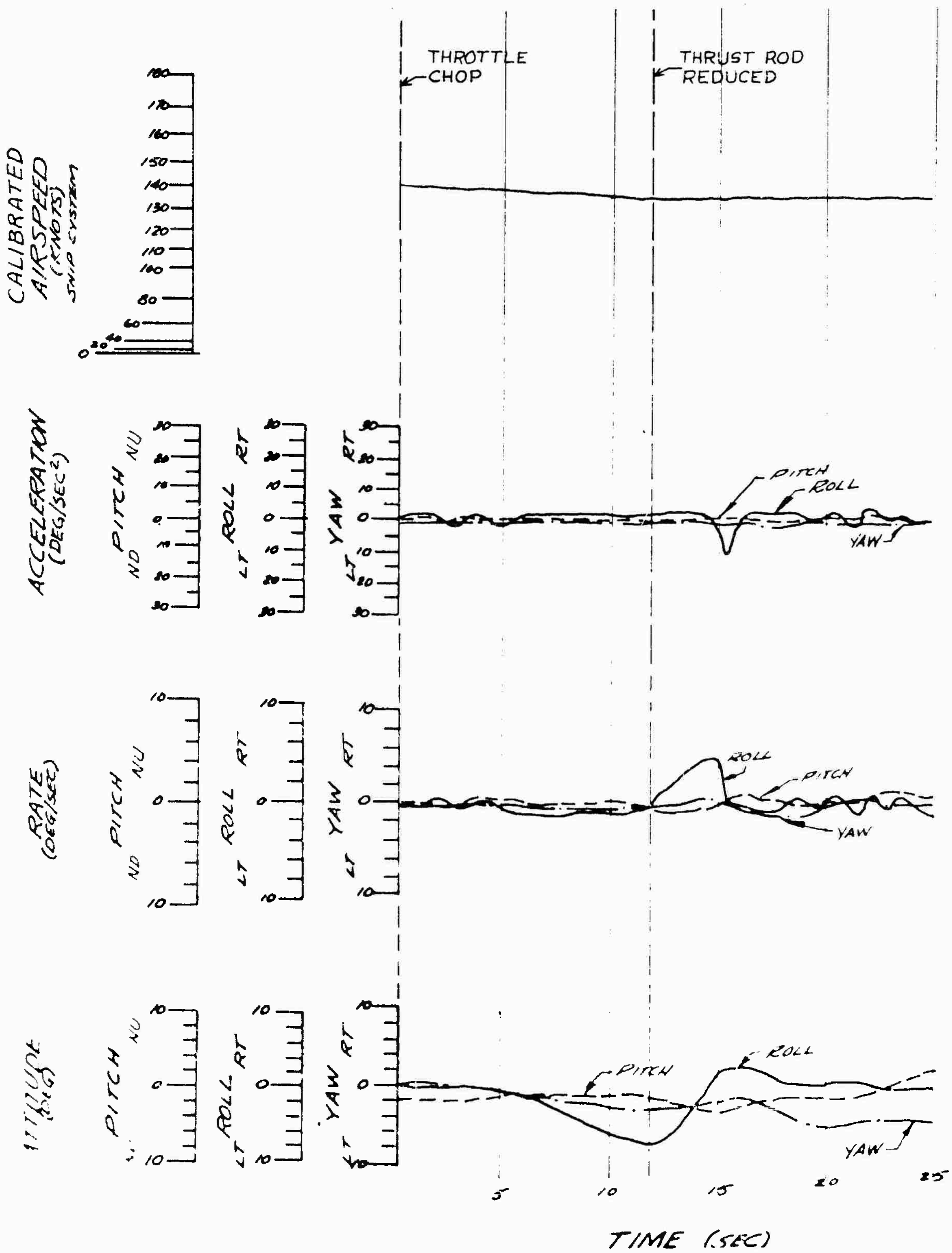


FIGURE 76 AIRCRAFT RESPONSE FOLLOWING NO. 1 THROTTLE CHOP
BOEING 347 S/N 65-7992

GROSS WEIGHT (LB)	DENSITY ALT (FT)	OAT (°C)	CG (IN)	ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	TRIM A/S (KCAS)
44980	10540	10.0	9 AFT	221	96.25	140



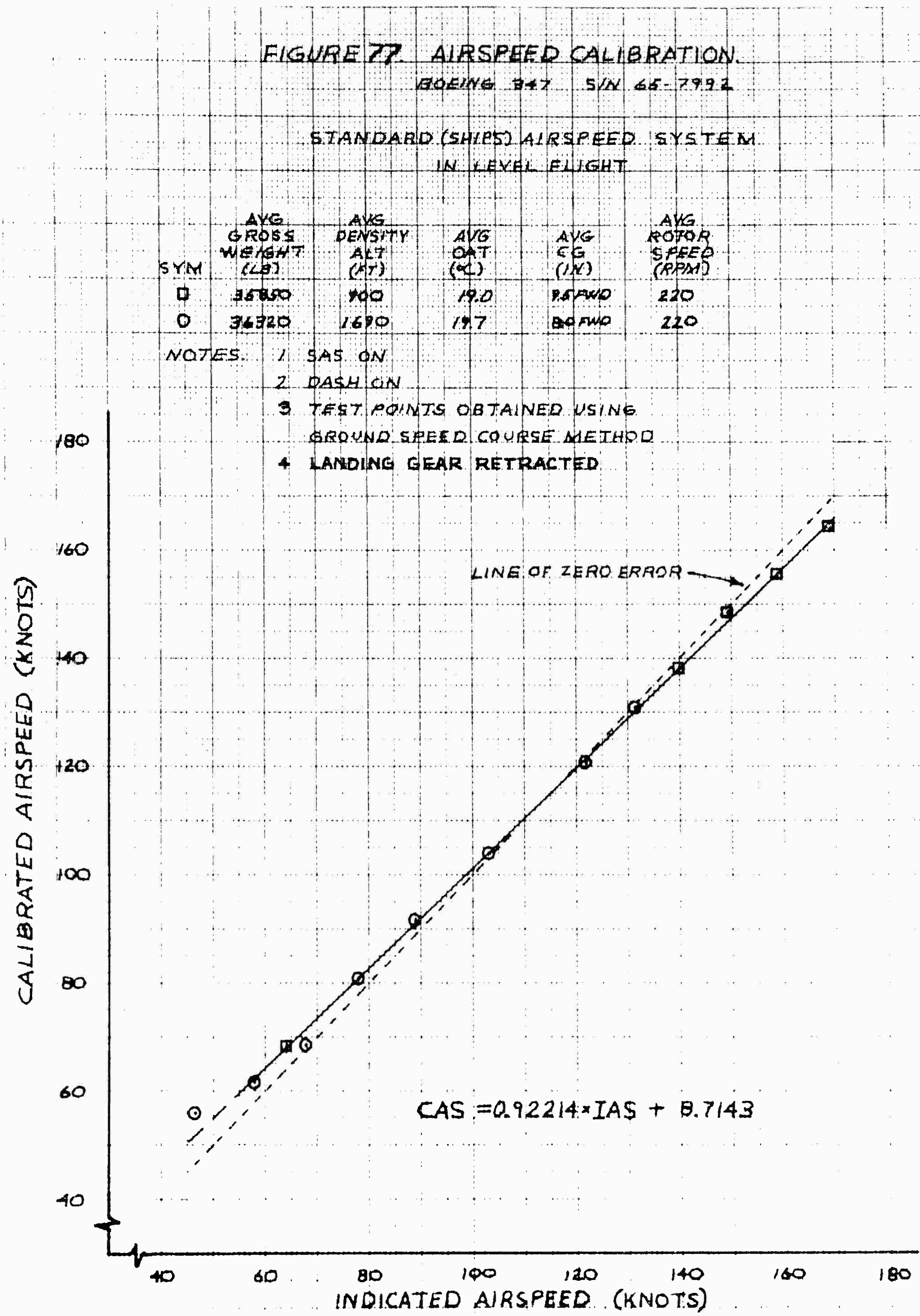


FIGURE 77 AIRSPEED CALIBRATION
BOEING 347 S/N 65-7992
STANDARD (SHIPS) AIRSPEED SYSTEM
IN LEVEL FLIGHT
SYM
AVG GROSS WEIGHT (LB)
AVG DENSITY ALT (FT)
AVG OAT (°C)
AVG CG (IN)
AVG ROTOR SPEED (RPM)
□ 35850 900 19.0 9.5 FWD 220
○ 36320 1690 19.7 8.0 FWD 220
NOTES. 1 SAS ON
2 DASH ON
3 TEST POINTS OBTAINED USING GROUND SPEED COURSE METHOD
4 LANDING GEAR RETRACTED
LINE OF ZERO ERROR
CAS = 0.92214 × IAS + 8.7143
CALIBRATED AIRSPEED (KNOTS)
40
60
80
100
120
140
160
180
INDICATED AIRSPEED (KNOTS)

FIGURE 78 STA 50 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	44800	5100	15.4	7.6AFT	219	81.96	LEVEL

NOTES: 1. SAS ON 3. ○ AT 9700 FT Hp 5. △ 1000 FPM DESCENT 7. ◇ 162 KTAS
2. DASH ON 4. □ 1500 FPM DESCENT 6. △ 500 FPM DESCENT

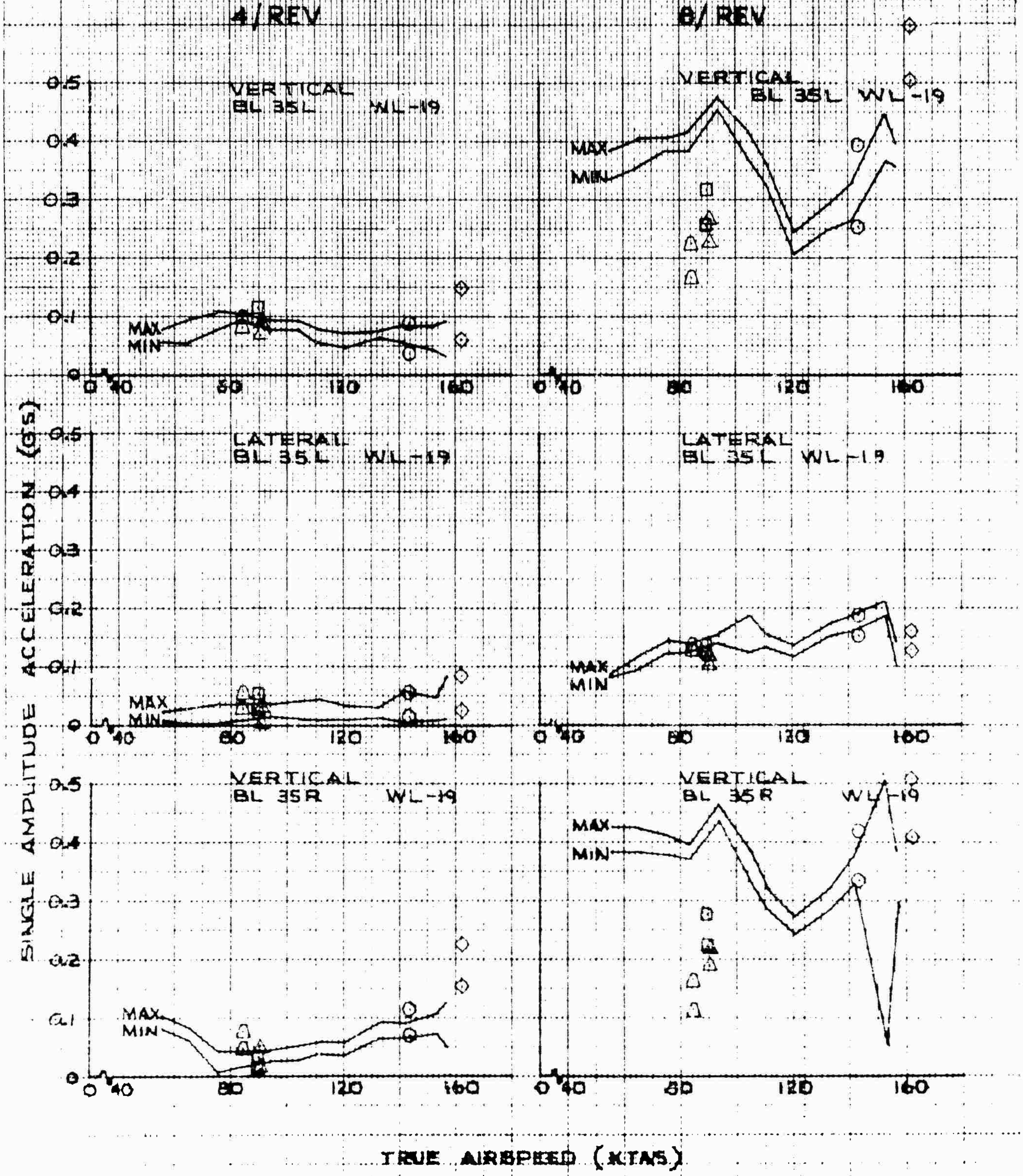


FIGURE 79 STA 50 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	36180	4130	17.6	17.2 FWD	219	65.02	LEVEL

NOTES: 1. SAS ON 3. ○ - AT 12800 FT H_D
2. DASH ON 4. □ - AUTOROTATION

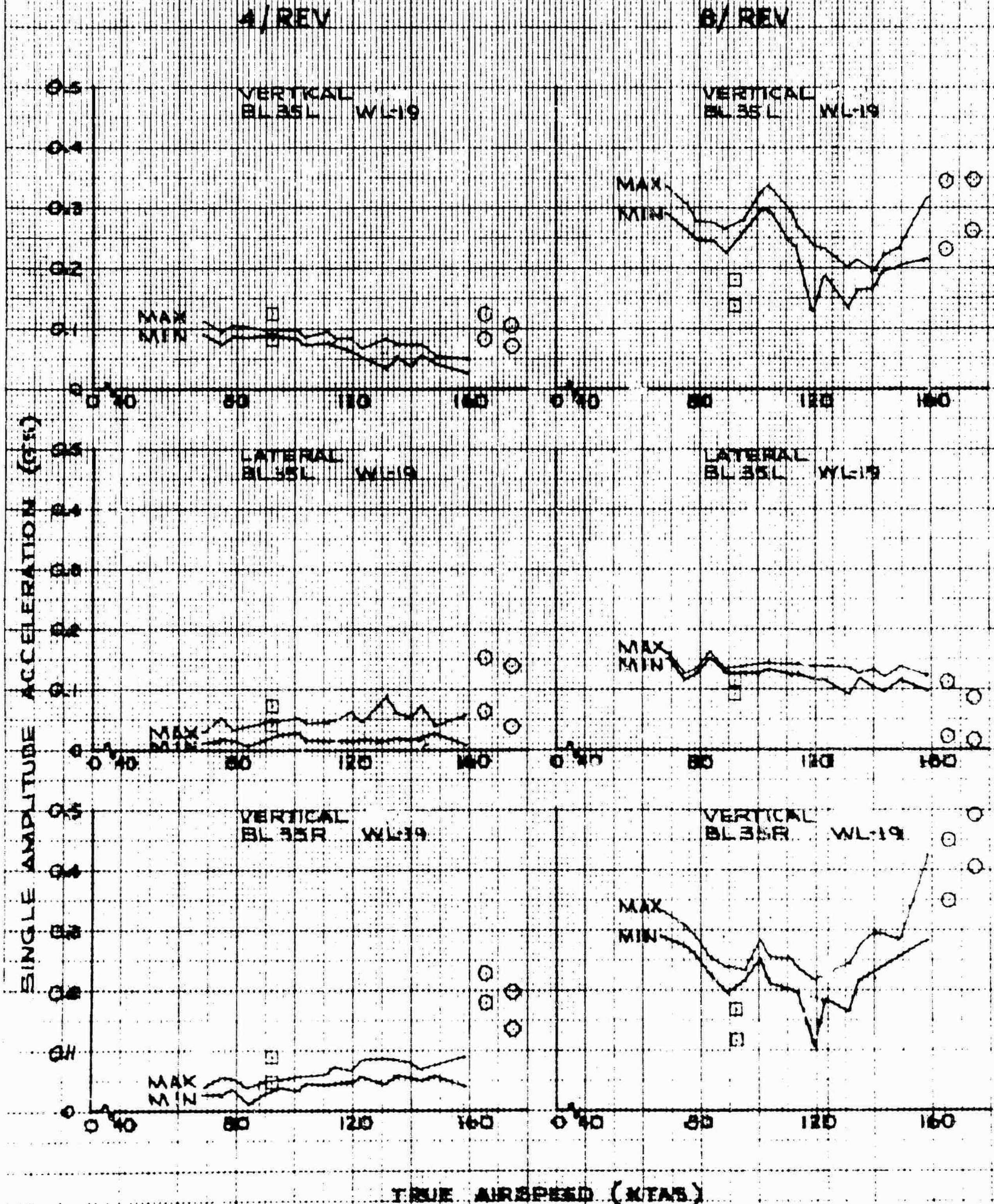


FIGURE 80 STA 95 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	44800	5100	15.4	7.6 AFT	219	81.96	LEVEL

NOTES: 1. SAS ON. 3. ○ AT 9700 FT H_D 5. △ 1000 FPM DESCENT
2. DASH ON. 4. □ 1500 FPM DESCENT 6. △ 500 FPM DESCENT
7. ◇ 162 KTAS

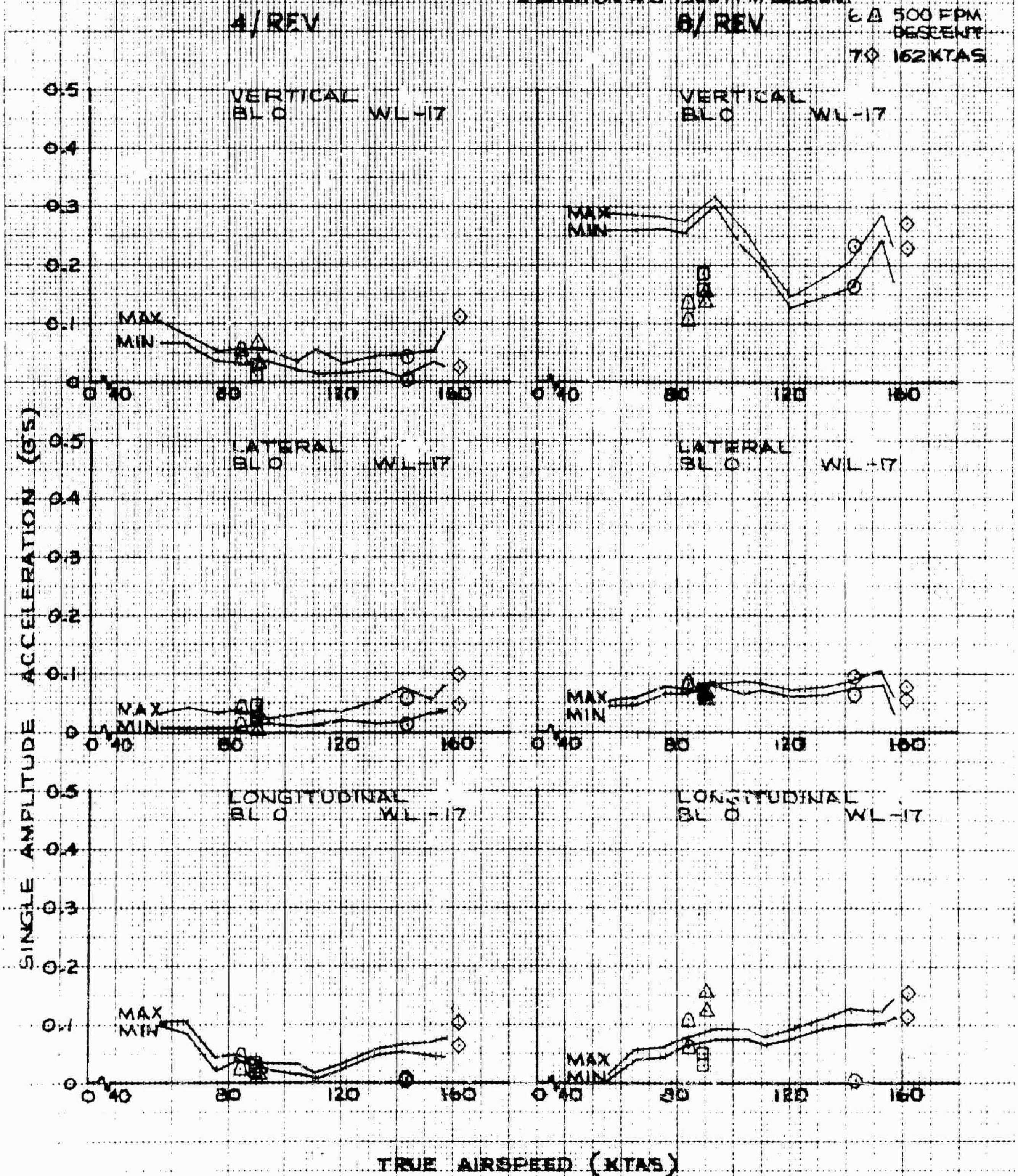


FIGURE 61. STA 95 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
————	36180	4130	17.6	17.2 FWD	219	65.02	LEVEL

NOTES: 1. SAS ON 3. ⊙ AT 1280 FT H_p
2. DASH ON 4. ⊡ AUTOROTATION

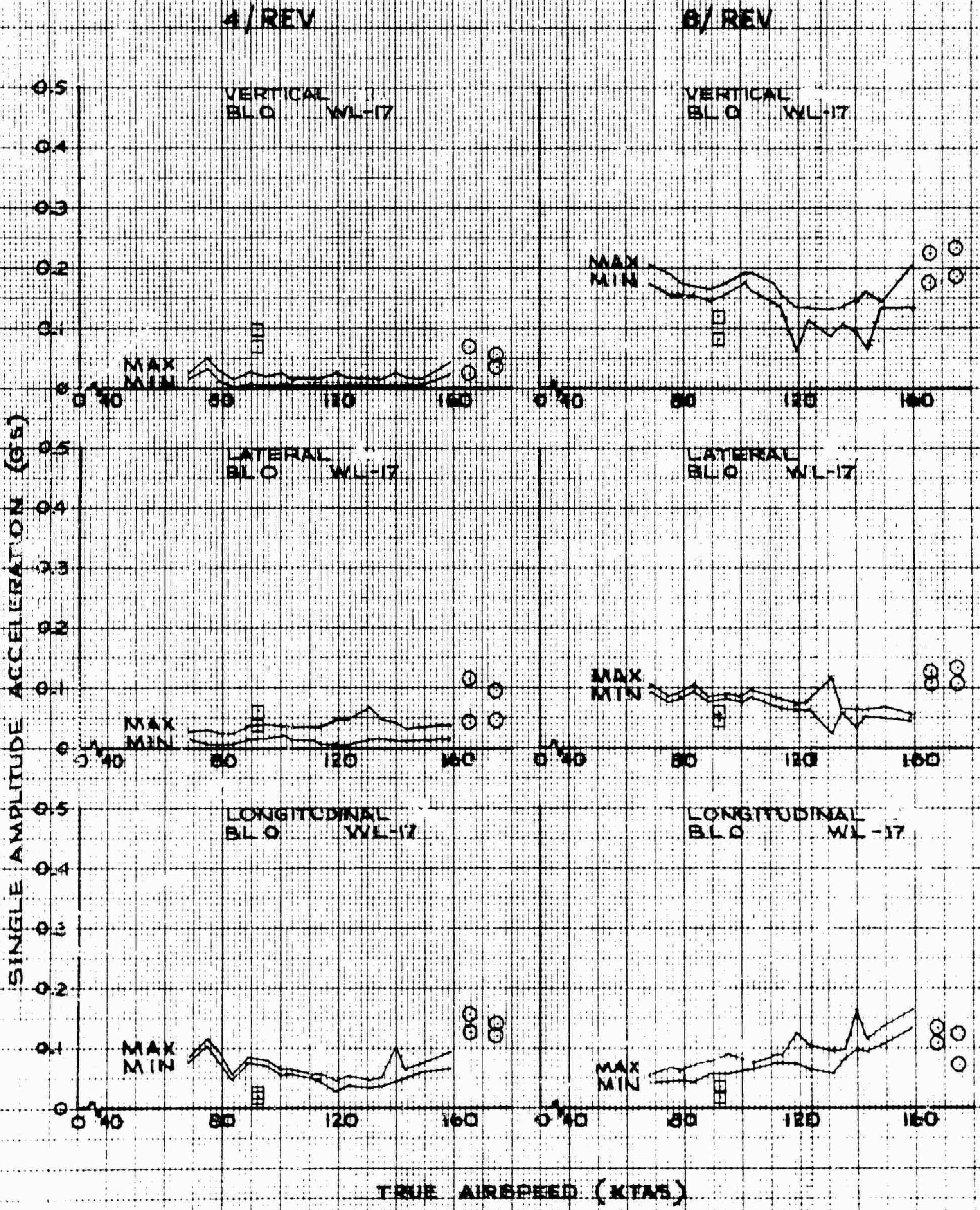


FIGURE 82 STA 160 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG C.G. (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	44890	5100	15.4	7.6 AFT	219	81.96	LEVEL

NOTES: 1. SAS ON 3.0 AT 9700 FT H_P 5. △ 1000 FPM DESCENT
2. DASH ON 4. □ 1500 FPM DESCENT
6. △ 500 FPM DESCENT
7. ◇ 162 KTAS

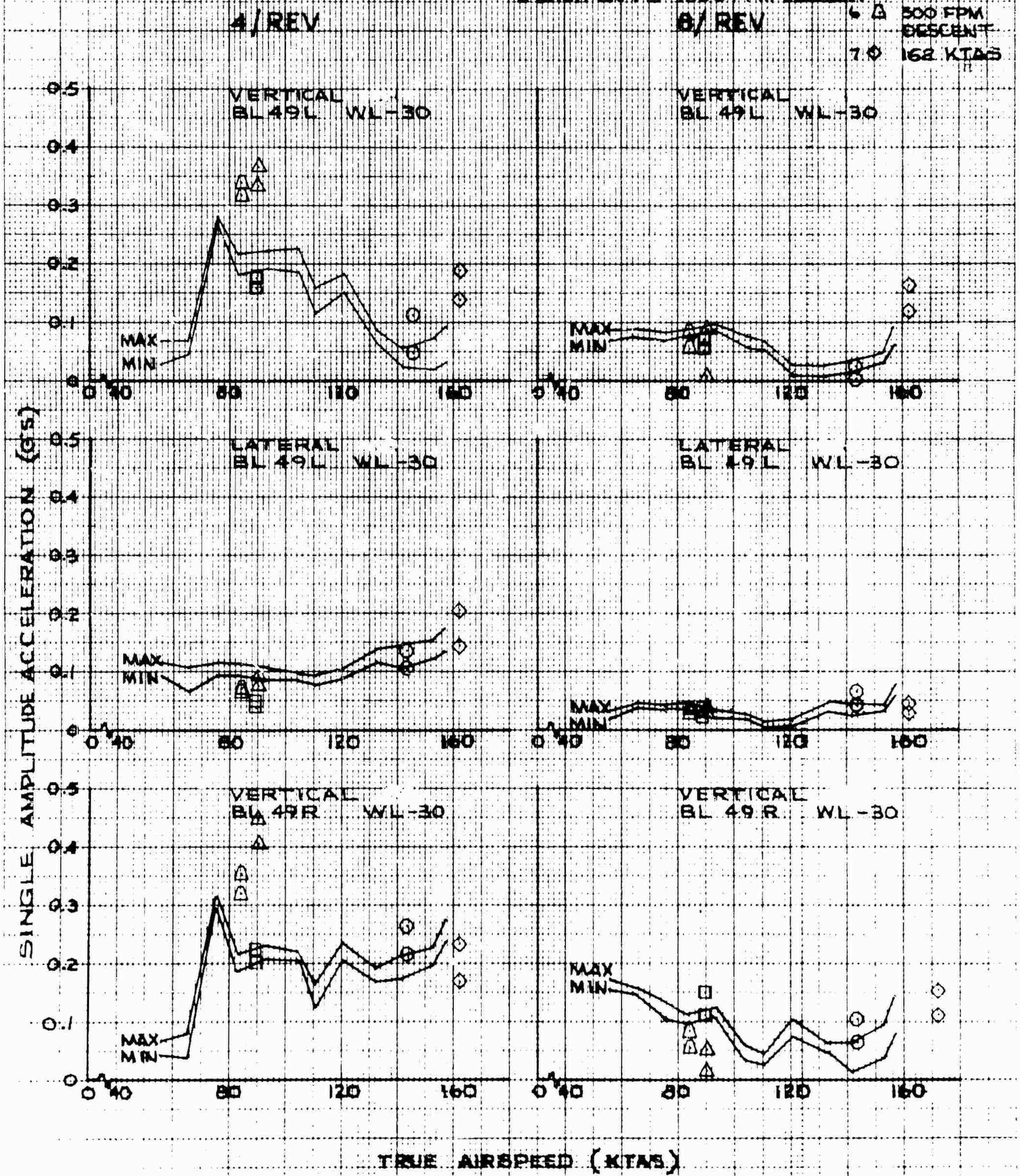


FIGURE 83 STA 160 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
——	36180	4130	17.6	17.2FWD	219	65.02	LEVEL

NOTES: 1. SAS ON 3. ○ AT 280 FT H_p
2. DASH ON 4. □ AUTOROTATION

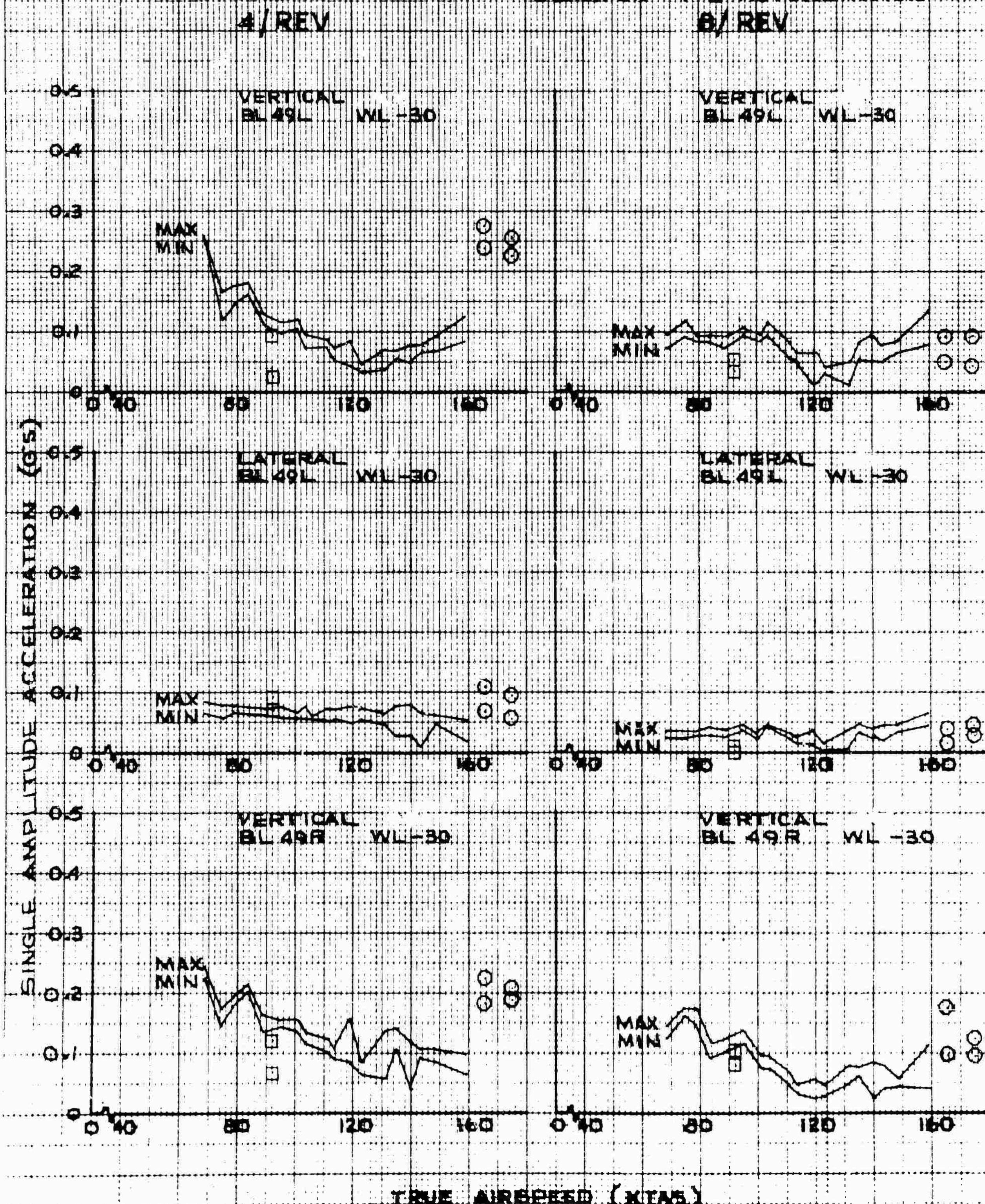


FIGURE 84 STA 360 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	44800	5100	15.4	7.6AFT	219	81.96	LEVEL

NOTES: 1 SAS ON 3 ○ AT 9700 FT H_p 5 △ 1000 FPM DESCENT
2 DASH ON 4 □ 1500 FPM DESCENT 6 △ 500 FPM DESCENT
7 ◇ 162 KTAS

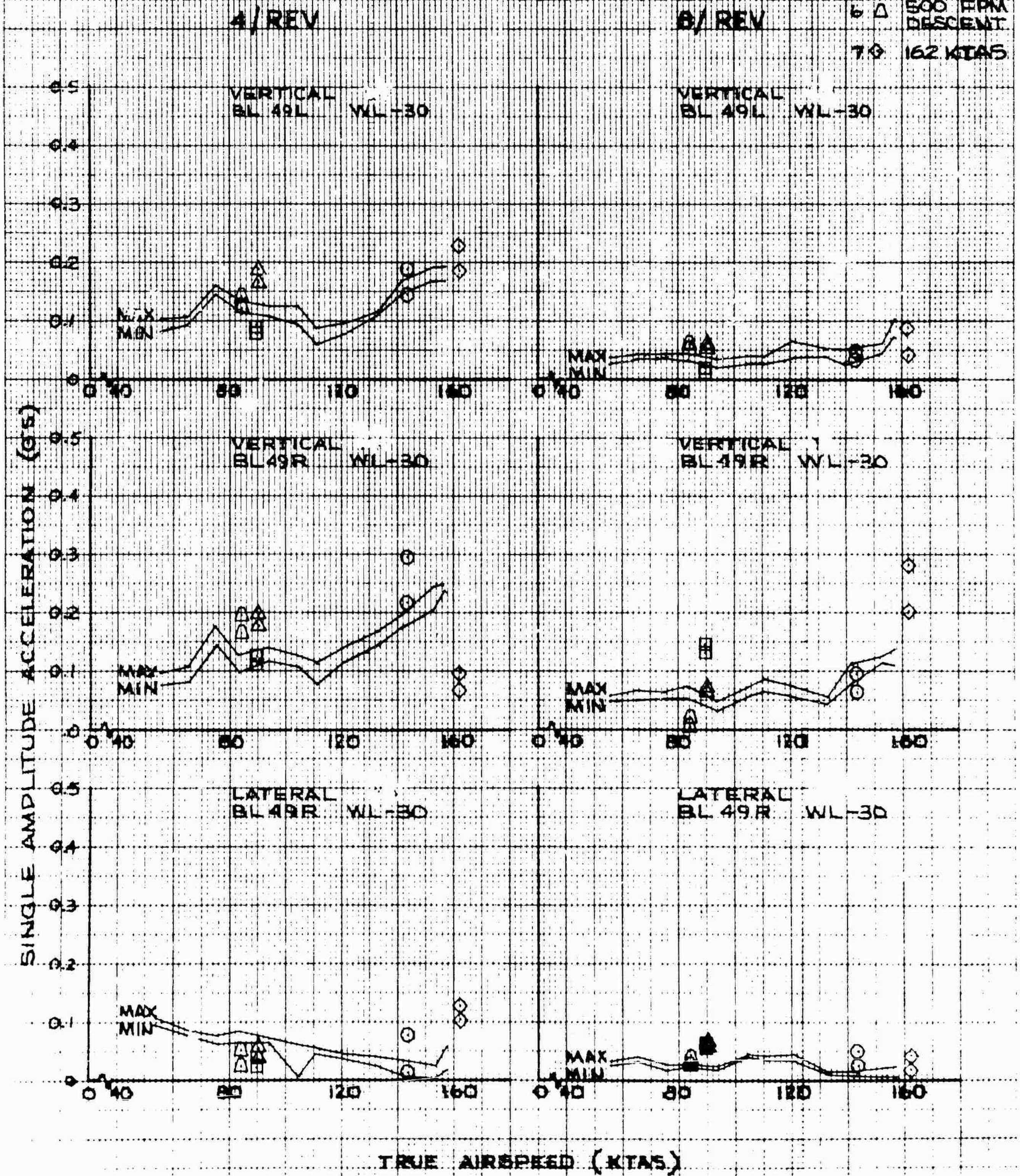


FIGURE 85 STA 360 VIBRATION CHARACTERISTICS

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG ROTOR SPEED (RPM)	THRUST COEFF (C_T x 10^4)	FLIGHT CONDITION
———	36180	4130	17.6	17.2 FWD	219	65.02	LEVEL

NOTES: 1. SAS ON 3. ○ AT 1280 FT H_D
2. DASH ON 4. □ AUTOROTATION

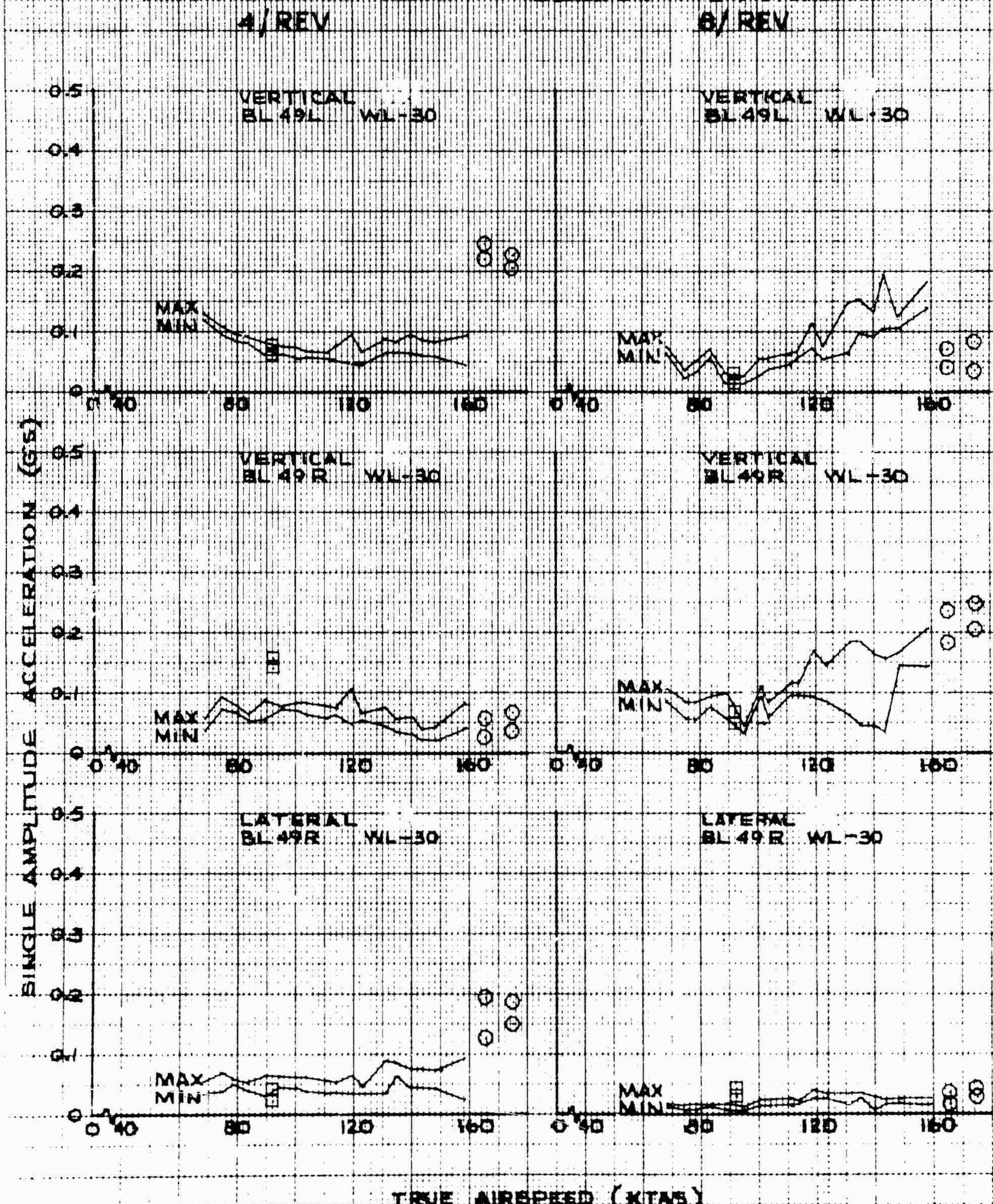


FIGURE 86 STA 592 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG. GROSS WEIGHT (LB)	AVG. DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN)	AVG. ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	44800	5100	15.4	7.6 AFT	219	81.96	LEVEL

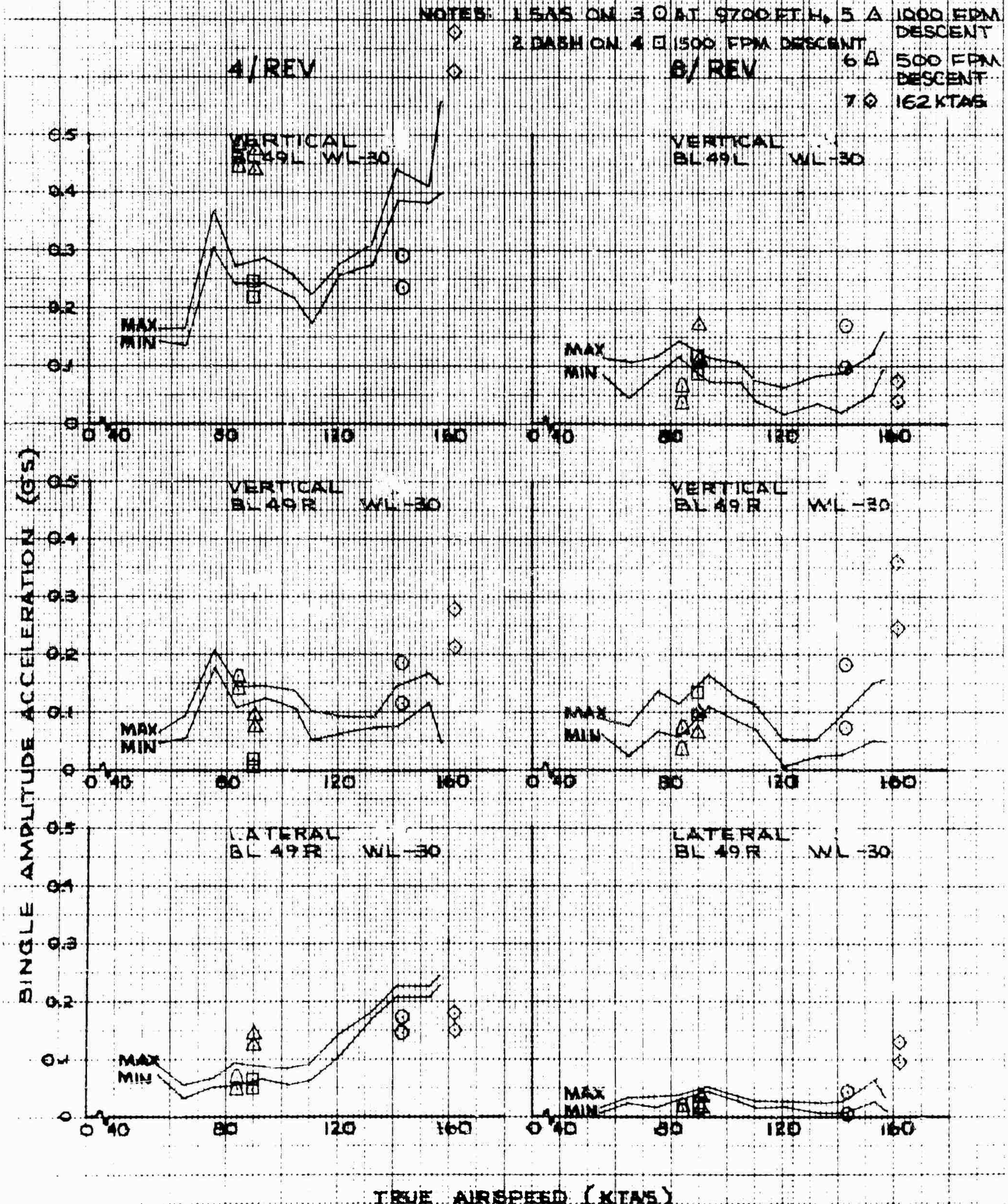


FIGURE 87 STA 592 VIBRATION CHARACTERISTICS.

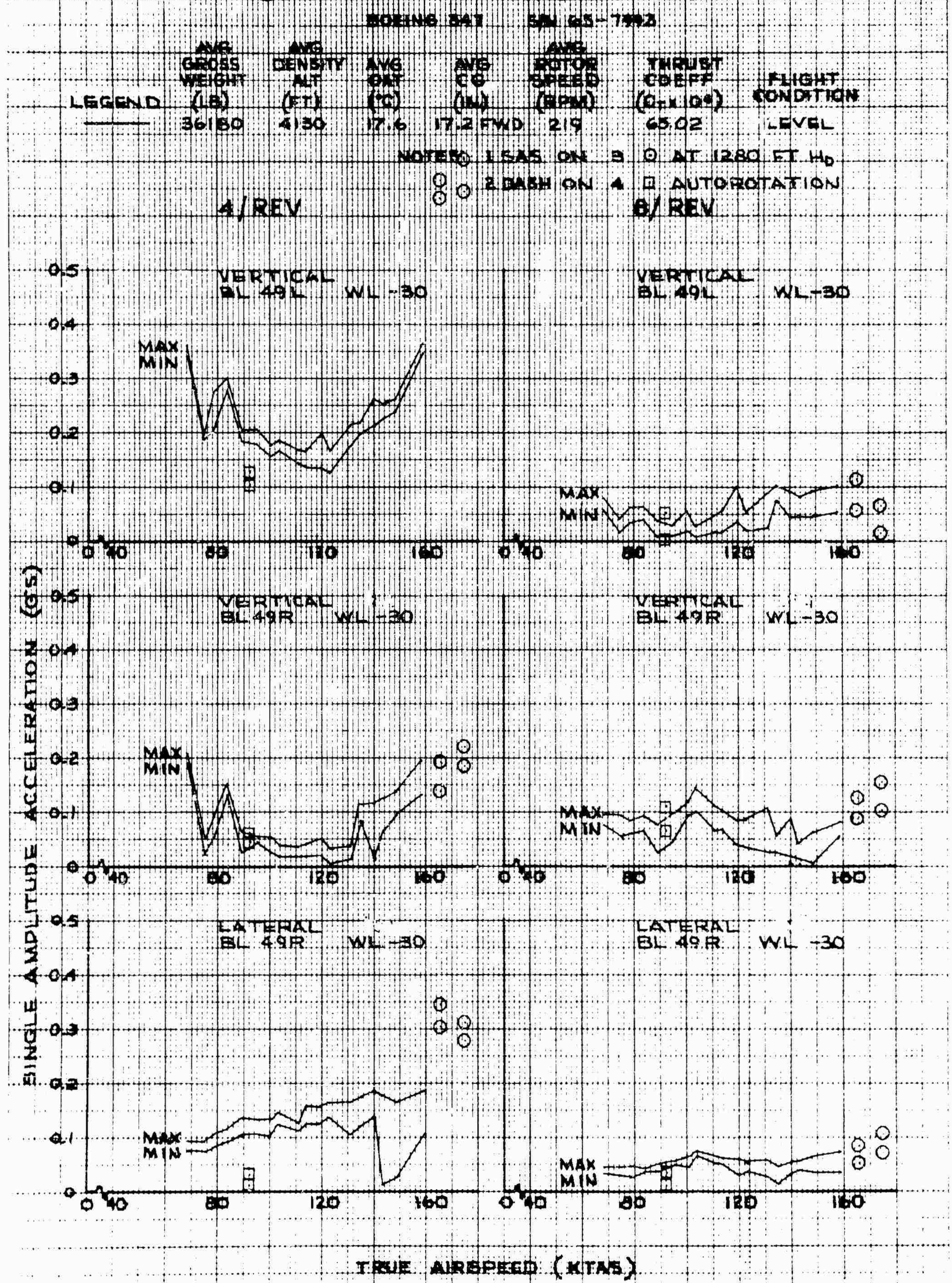


FIGURE 88. STA 95 VIBRATION CHARACTERISTICS.

BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF. ($C_T \times 10^4$)	FLIGHT CONDITION
———	44800	5100	15.4	7.6 AFT	219	84.1	LEVEL

NOTES 1. BAS ON
2. DASH ON

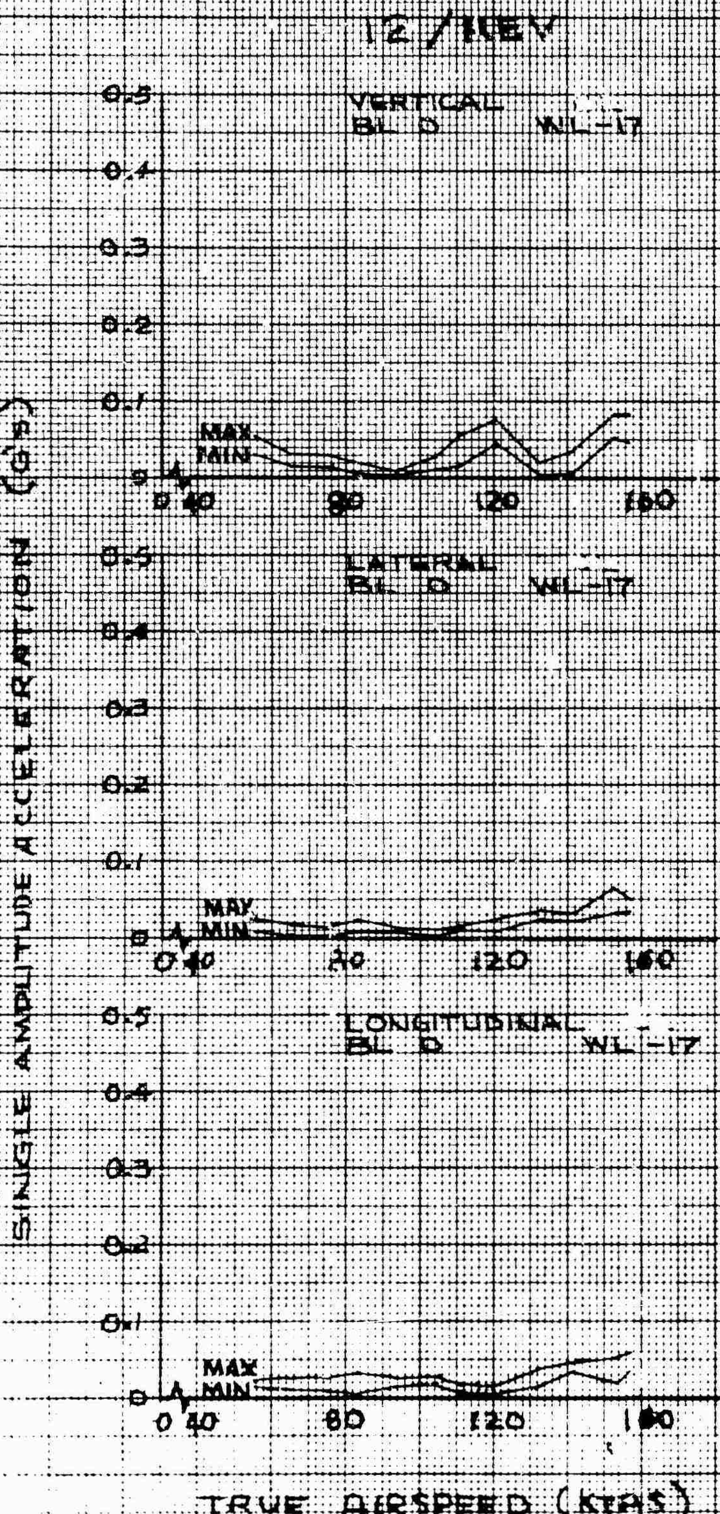


FIGURE 89. STA 95 VIBRATION CHARACTERISTICS.
BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	36180	4130	17.6	17.2 FWD	219	65.02	LEVEL

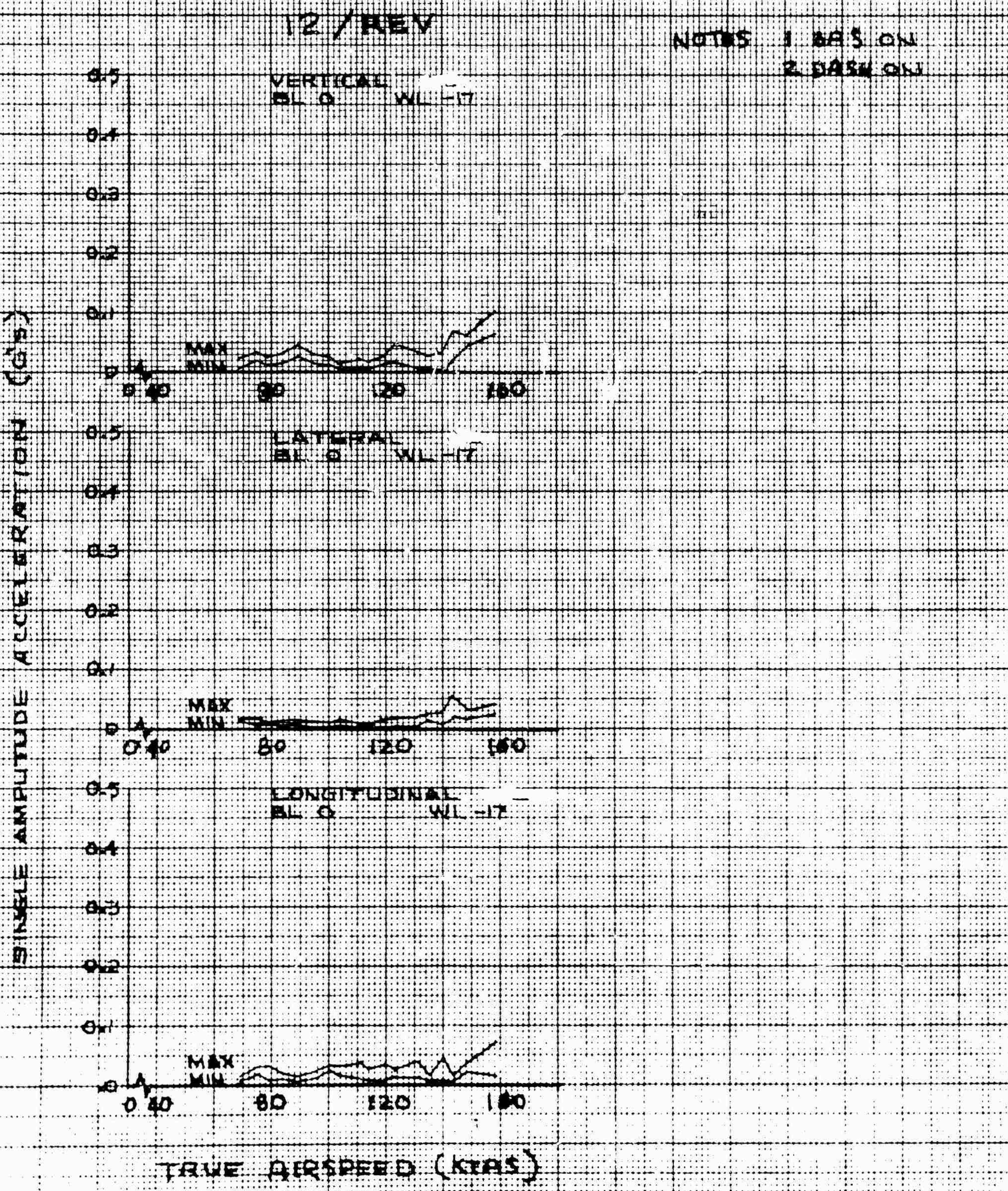


FIGURE 90. STA 592 VIBRATION CHARACTERISTICS.
BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG CG (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF ($C_T \times 10^4$)	FLIGHT CONDITION
———	41800	5100	15.4	7.6 AFT	219	84.1	LEVEL

12/REV

NOTES 1. BAS ON
2. DASH ON

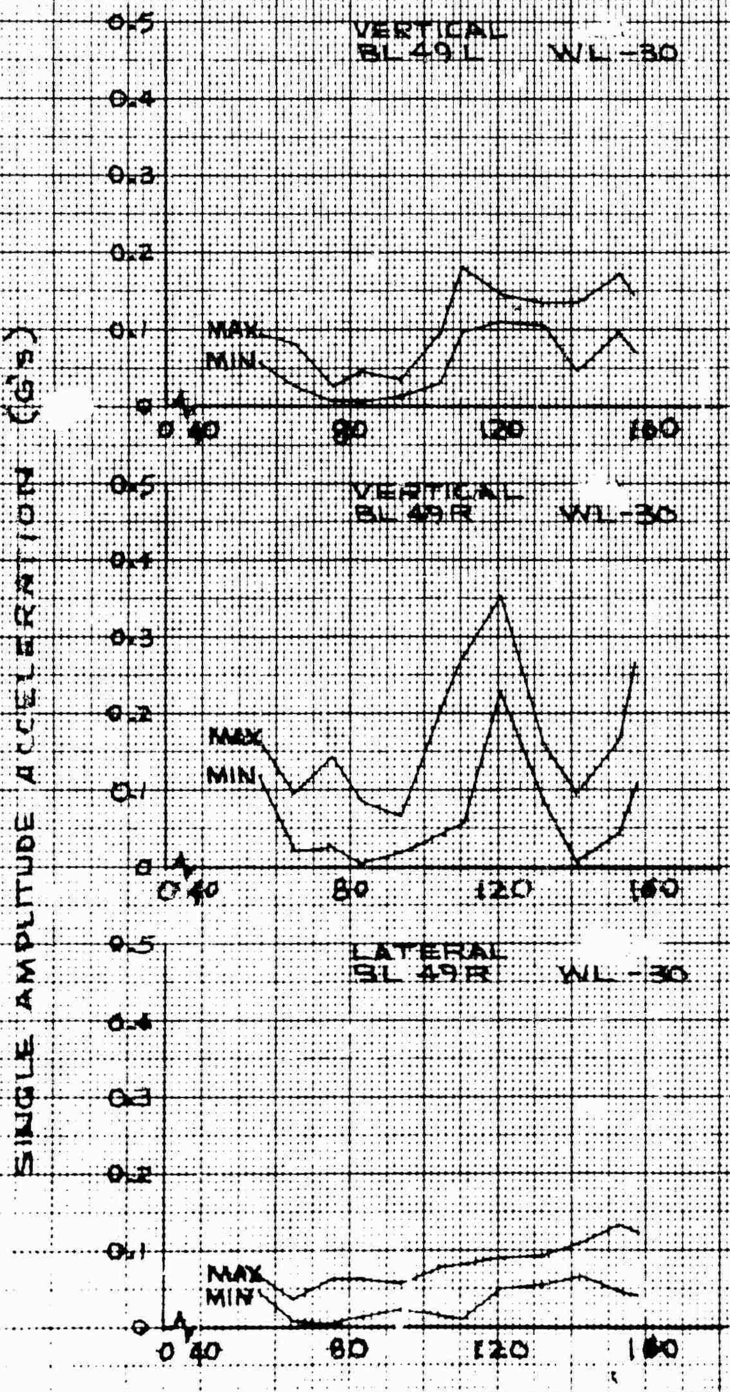
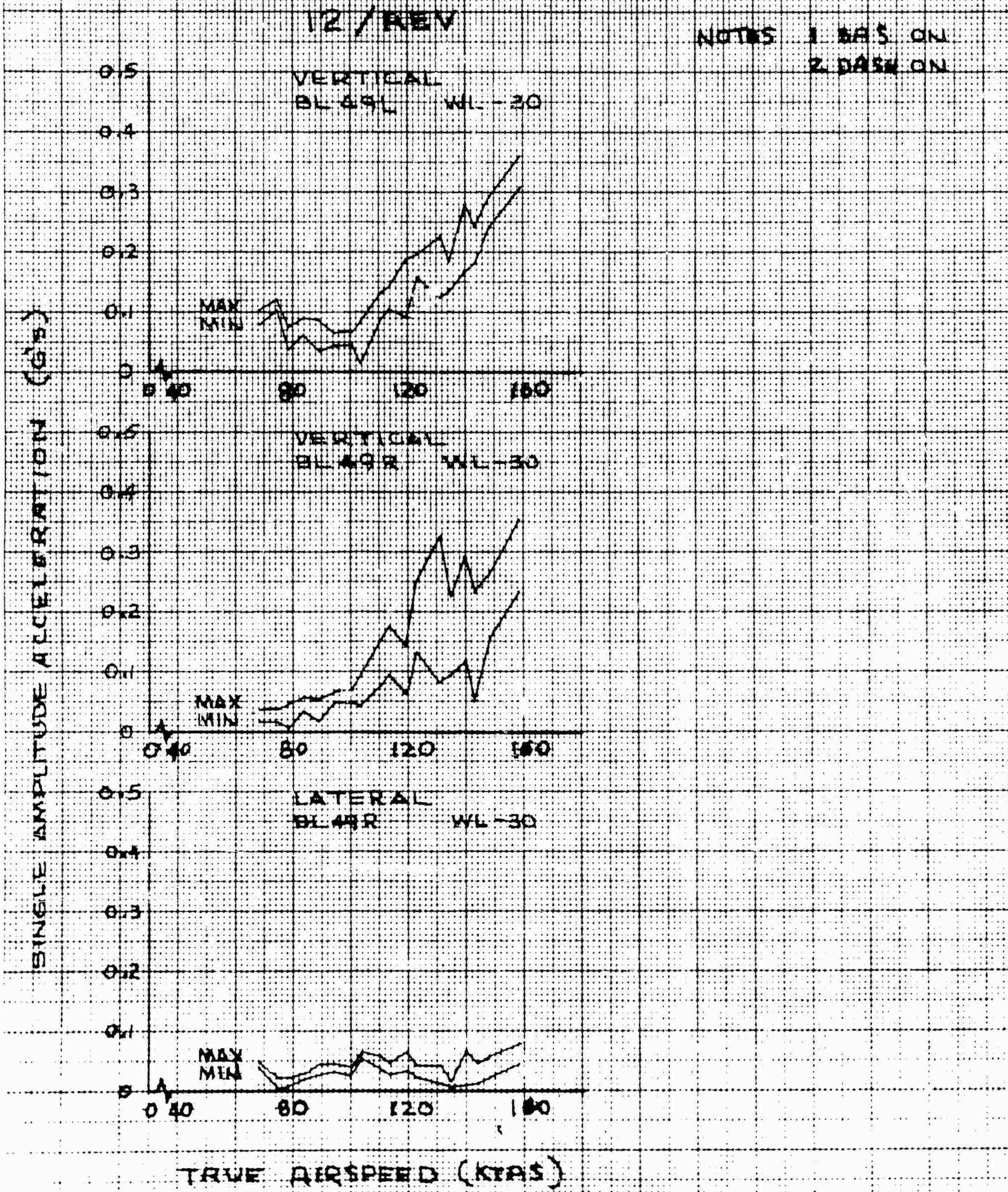


FIGURE 91 - STA 592 VIBRATION CHARACTERISTICS -
BOEING 347 S/N 65-7992

LEGEND	AVG GROSS WEIGHT (LB)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG C.G. (IN.)	AVG ROTOR SPEED (RPM)	THRUST COEFF. ($C_T \times 10^4$)	FLIGHT CONDITION
————	36180	4130	17.6	17.2 FWD	219	65.02	LEVEL



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APPENDIX III. DESCRIPTION OF TEST AIRCRAFT

GENERAL

1. The Boeing-Vertol Model 347 helicopter is derived from the CH-47 cargo helicopter family. Significant changes from the CH-47C are shown in the following photograph.

ROTOR SYSTEM

2. Four blades are installed on each rotor head. The blades are the same 30-foot-radius blades as those used on the CH-47C. The aft rotor hub, pitch housings, swashplate, and centrifugal droop stops are essentially the same as those on the CH-47C, except that they are rearranged as required for a four-bladed configuration. The forward head is an entirely new design incorporating centrifugal droop stops and Delta-three geometry. The Delta-three geometry on the forward head required the redesign of the hub, pitch housing, pitch shaft, swashplate, and pitch links.

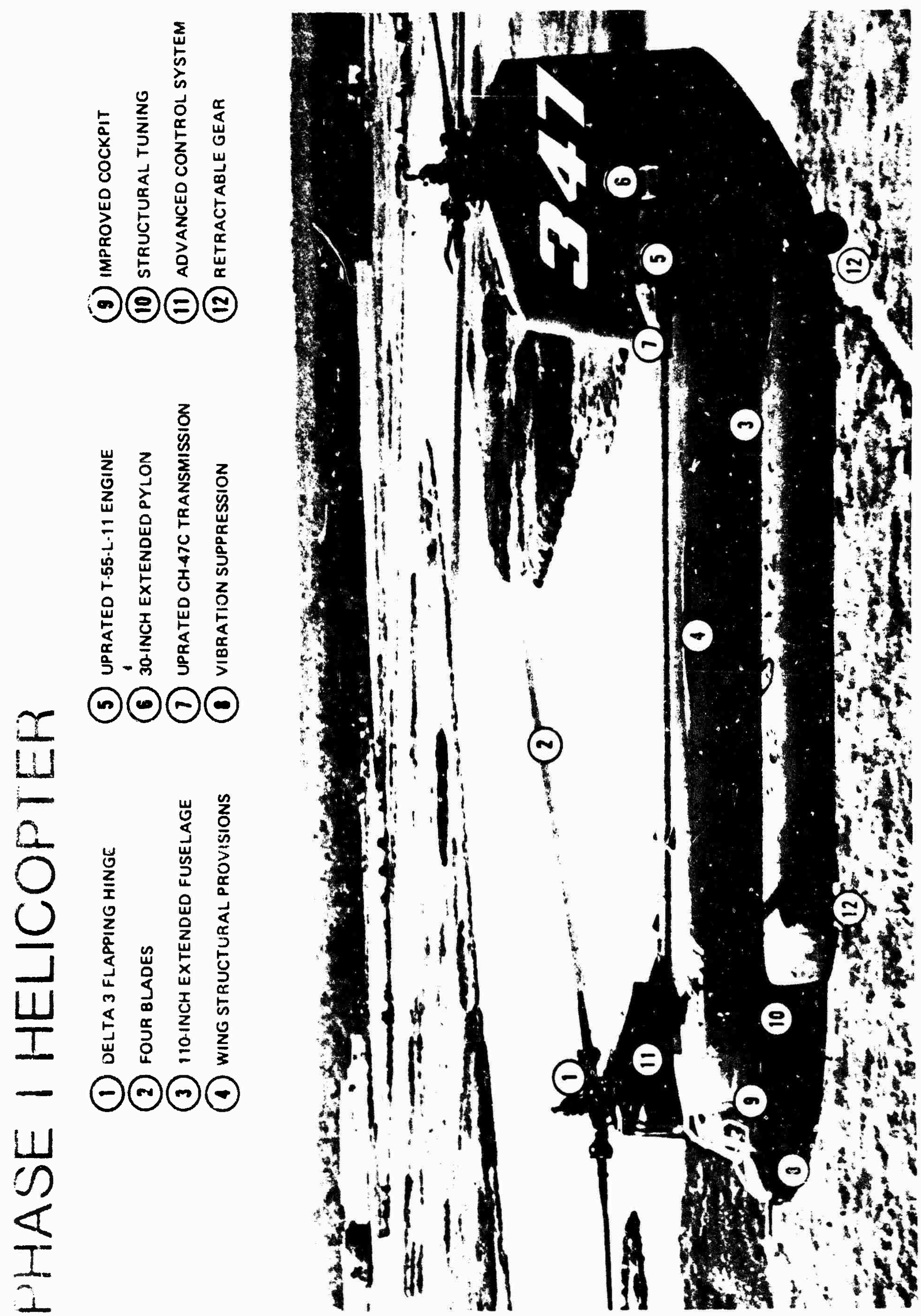
LANDING GEAR

3. The CH-47C forward and aft landing gear were converted to retractable gear by the addition of adapters, actuators, locks, and retracting mechanisms. The forward gear retracts fully into the revised pod section. The aft gear retracts partially into the revised pod, leaving approximately one-third of each wheel exposed. Each gear is provided with a positive uplock and downlock to retain the landing gear in the selected position. In addition to the normal mode of gear extension, the uplocks can be manually released to permit free fall partial extension of the landing gear. Full manual extension of the gear and engagement of the downlocks requires use of a special tool normally stored in the cabin area.

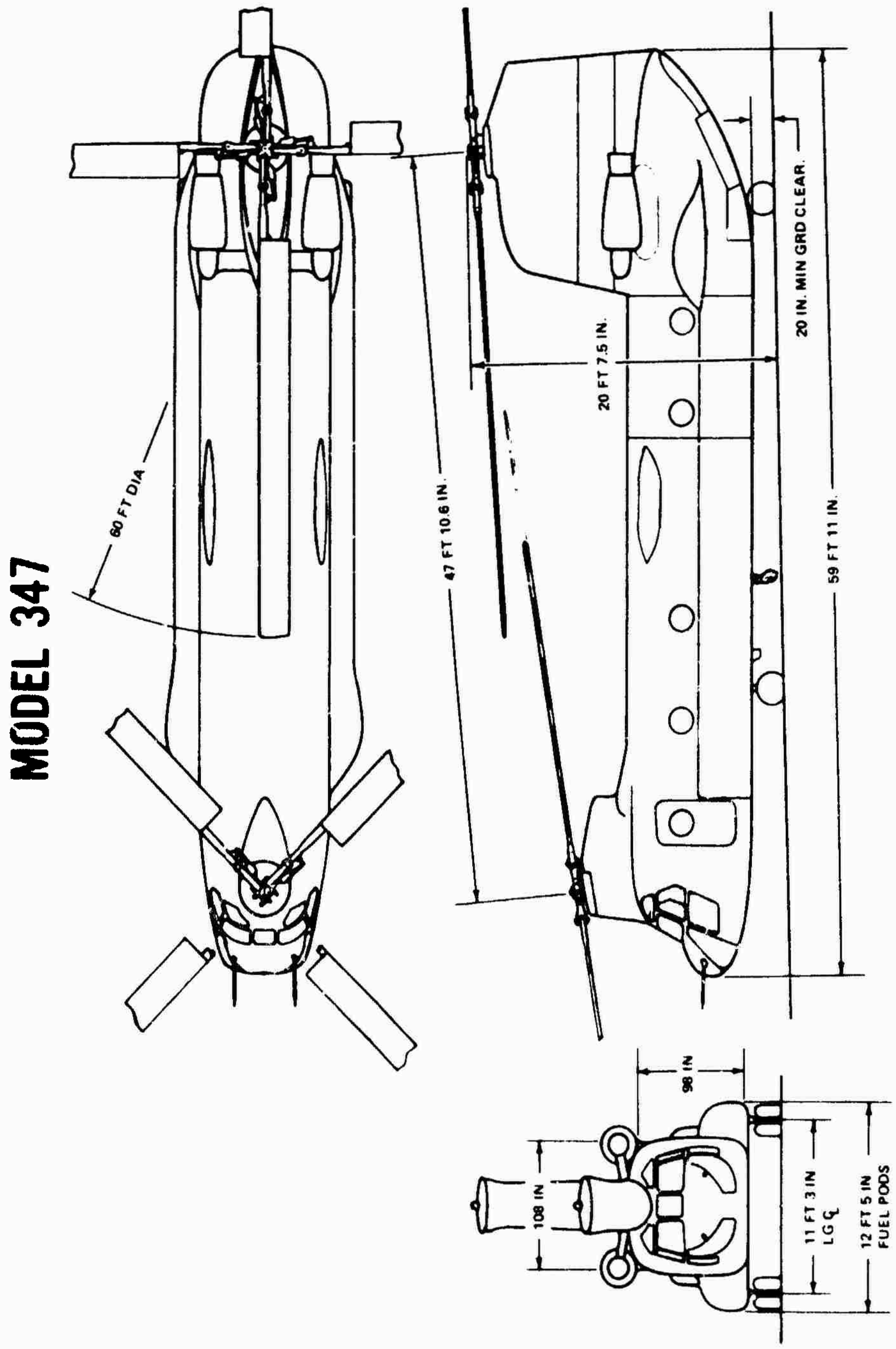
AIRFRAME

4. The structure is designed for a normal gross weight of 45,000 pounds at a limit load factor of 2.0g's. The alternate design gross weight with a 25,000-pound external load is 54,500 pounds at a limit load factor of 1.65g's.

5. The increased fuselage length was obtained by inserting a 110-inch adapter assembly at the manufacturing splice between the CH-47C constant cabin section and the aft fuselage assembly. The increased aft pylon height was obtained by adding a 30-inch adapter and new fairings to the CH-47C pylon.



Significant Changes in the Boeing-Vertol Model 347 Helicopter from the CH-47C Helicopter.



Three-View Drawing of the Model 347 Helicopter.

COCKPIT

6. Crew seats from the Boeing 737 transport are installed at the pilot and copilot stations. These seats provide adjustment of height, longitudinal position, seat back recline angle, and thigh support. The pilot instrument panel is equipped with a Sperry Model HZ-6F flight director (attitude indicator), a Sperry Model RD-100 horizontal situation display (radio direction indicator), and a Bendix Model ALA-51 radar altimeter. The center instrument panel was redesigned to eliminate the AC and DC loadmeters in order to permit installation of an expanded caution/warning annunciator panel. The center console accommodates displays and controls for the automatic flight path control system, Doppler navigation system, and the retractable landing gear. The overhead panel is increased in size to permit installation of the engine condition levers and electrical system monitoring instruments.

ACOUSTIC TREATMENT

7. The cockpit area is intensively treated to reduce the interior noise level. The treatment includes increased-thickness plexiglass windows and glass windshields, double-walled bulkhead at station 95, and a companionway door at station 120. In addition, all exposed sheet metal is treated with leaded vinyl material, the forward transmission drip pan is heavily insulated, and a cockpit floor covering is installed.

VIBRATION ATTENUATION

8. Five self-tuning vibration absorbers (STVA), similar to those used in the CH-47C, are installed in the aircraft. There is also one fixed-frequency absorber. Four of the STVA, three vertical and one lateral, are used to oppose the primary forcing frequency of 4 cycles per rotor revolution (cycles/rev). One STVA, at 2 cycles/rev, and one fixed-frequency absorber, at 8 cycles/rev, are also incorporated in the aircraft. The location and function of the individual absorbers are shown in table A.

Table A. Vibration Absorbers.

Frequency (Cycles)	Rotor Speed (rpm)	Direction	Location		
			Fuselage Station	Water Line	Buttline
4/rev	[1]216 and 239	Vertical	33	-7	Zero
4/rev	[1]216 and 239	Vertical	82	-30	25 right
4/rev	[1]216 and 239	Vertical	82	-30	17 left
4/rev	[1]216 and 239	Lateral	82	-30	40 right
2/rev	[1]216 and 239	Vertical	120	-2	30 right
8/rev	[2]220	Vertical	120	-18	30 right

[1]Indicates rotor speed range within which vibration absorbers are self-tuning.
[2]Fixed frequency absorber.

PHYSICAL CHARACTERISTICS

9. Additional descriptive data are contained in the following three-view drawing and in table B.

Table B. Basic Aircraft Information

<u>Airframe</u>

Overall length (rotors turning)	108 ft, 2.1 in.
Overall length of fuselage	59 ft, 11.0 in.
Distance between rotor masts	47 ft, 10.6 in.
Maximum fuselage width	12 ft, 5.0 in.
Height from extended aft landing gear to top of rear rotor hub	20 ft, 7.5 in.

<u>Rotors</u>

Rotor diameter	60 ft, 0.0 in.
Blade chord	25.25 in.
Blade twist (center line of rotor blade to tip)	9.233 deg
Blade section	Modified Ames droop snoot, t/c = 0.10
Rotor blade area	505 ft^2
Disc area (total)	5,654.9 ft^2
Disc loading at 46,000 pounds	8.1 lb/ft^2
Solidity ratio	0.0893
Normal operating speed (power on)	215 to 225 rpm
Tip speed at 220 rpm	691 ft/sec

<u>Engines</u>

Type	T55-L-11 (uprated)
Sea-level, standard-day rating at 16,000 rpm:	
Maximum power (10 minutes)	3,925 shp
Military power (30 minutes)	3,750 shp
Normal power (continuous)	3,000 shp

<u>Transmission</u>

Gear ratio	64:1
Torque limits:	
Dual engine	1,265 ft-lb (98 percent)
Single engine	1,300 ft-lb (100 percent)

APPENDIX IV. FLIGHT CONTROL SYSTEM

GENERAL

1. The basic control system of the Model 347 helicopter, including cockpit controls, lower dual-boost actuators, mechanical mixer, mechanical linkages, and upper dual-boost actuators, is essentially the same as in the CH-47C helicopter. Control runs are extended, as needed, to allow for the increased fuselage length and increased aft pylon height of the Model 347 helicopter. The irreversible electrohydraulic system utilizes conventional dual hydraulic lower boost actuators to transmit individual axis-oriented cockpit control motions to a mechanical mixing unit. The mixed outputs from the mechanical mixing unit provide inputs to the upper dual-boost actuators at the forward and aft rotor swashplates. In addition to pilot inputs through the cockpit controls, automatic inputs enter the flight control system by two means:

a. Differentially, through the SAS and DASH system actuators. These signals do not move the cockpit controls.

b. In parallel, through the FPCS actuator. These signals move the cockpit controls.

2. The Model 347 control system incorporates the following significant modifications and additions to the CH-47C control system:

a. Variable force feel in lateral and longitudinal axes.

b. Flight path control system providing automatic control of heading and altitude when engaged.

c. Vernier beep trim of bank attitude, heading, and airspeed.

d. The DASH system providing long-term retention of airspeed and pitch attitude.

e. Expanded SAS providing long-term bank attitude and heading hold.

f. Lateral and longitudinal SAS actuators relocated in the rotor pylons to improve fidelity and, if required, to permit use of the actuators in mode suppression. No mode suppression functions are presently performed by the system.

g. Cockpit control position transducers (control stick pick-offs) in the longitudinal, lateral, and directional control systems to improve maneuverability by overcoming the high stability provided by the SAS and DASH system.

h. Slow-rate automatic lateral trim actuator which moves the lateral control automatically to reduce the requirement for the pilot to retrim laterally as airspeed and thrust are varied.

FORCE-FEEL SYSTEMS

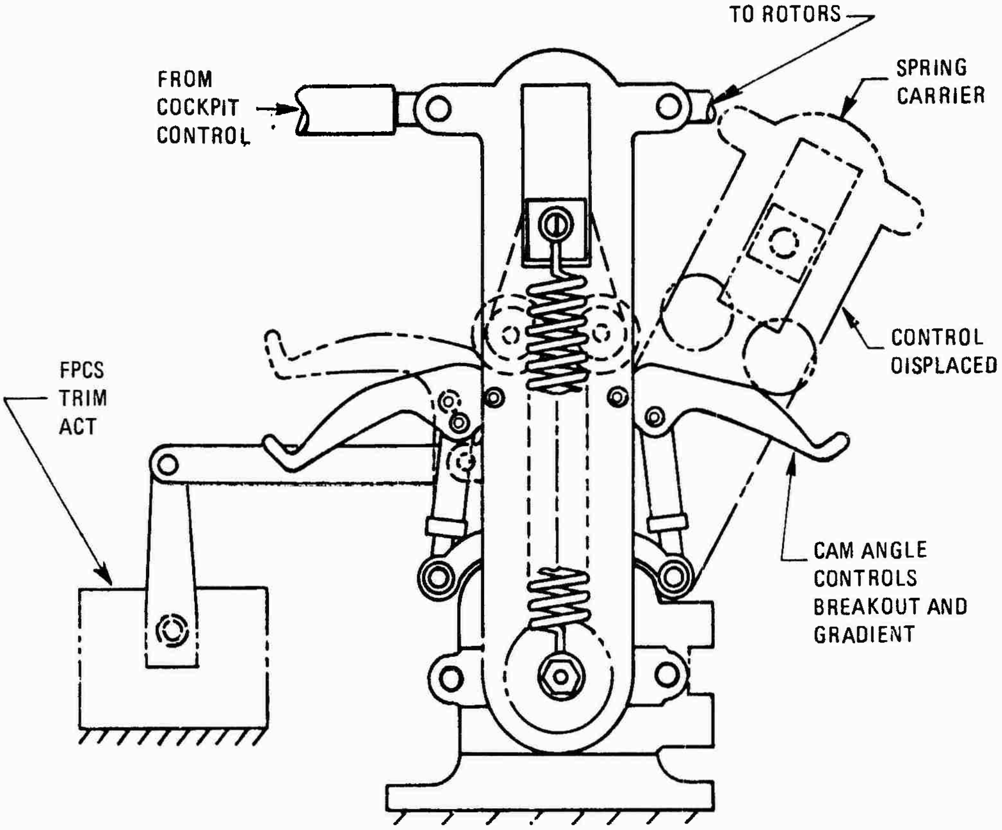
3. Variable force-feel systems provide longitudinal and lateral control pilot force cues which vary with airspeed. In addition, the longitudinal force-feel system also provides an increasing control force gradient with pitch rate. Directional control force feel is provided by a fixed spring capsule. The variable force-feel actuator used in the lateral and longitudinal systems is shown in figure 1. Also shown in figure 1 is an example plot of longitudinal control force characteristics under three different flight conditions. Lateral force characteristics are similar to those shown in figure 1 except that aircraft angular rates do not influence control forces.

4. Breakout and gradient scheduling is achieved by processing the output of an airspeed sensor. In the case of the longitudinal control, the output of a pitch rate gyro is added to the airspeed signal. The processed outputs serve as commands to the force-feel actuator servo loop, consisting of an amplifier, an electrohydraulic servo valve, and a feedback transducer. The servo positions the force-feel actuator cams in response to the airspeed and pitch rate inputs. Control forces are generated by displacing the center spring carrier against the force-feel cams. No forces are applied to the cockpit controls when the center spring carrier is centered in the detent position. The FPCS parallel actuator moves the trim reference point to retrim the control to zero force or to provide automatic lateral and longitudinal inputs to the control system. The lateral and longitudinal force-feel systems incorporate viscous dampers to apply forces proportional to control motion velocity. The directional and thrust control linkages include eddy current dampers for the same purpose.

LONGITUDINAL CONTROL AXIS

5. As shown in figure 2, pitch rate damping is provided by a vertical rate gyro, the output of which is lagged, electrically limited, and applied through an electrical mixer to the upper SAS actuators. Unlike the CH-47C, which has individual, axis-oriented SAS actuators for pitch and roll located prior to the mechanical mixer, the SAS actuators which provide pitch and roll stabilization in the Model 347 are located beyond the mechanical mixer, and the same actuators provide both pitch and roll inputs to the rotors. An electrical mixer is therefore required in the Model 347 to convert the individual axis-oriented SAS signals to upper SAS actuator mixed commands. The SAS electrical mixer performs the same function as is performed by the mechanical mixer in converting single-axis mechanical control inputs into properly mixed upper boost actuator commands.

VARIABLE FORCE FEEL



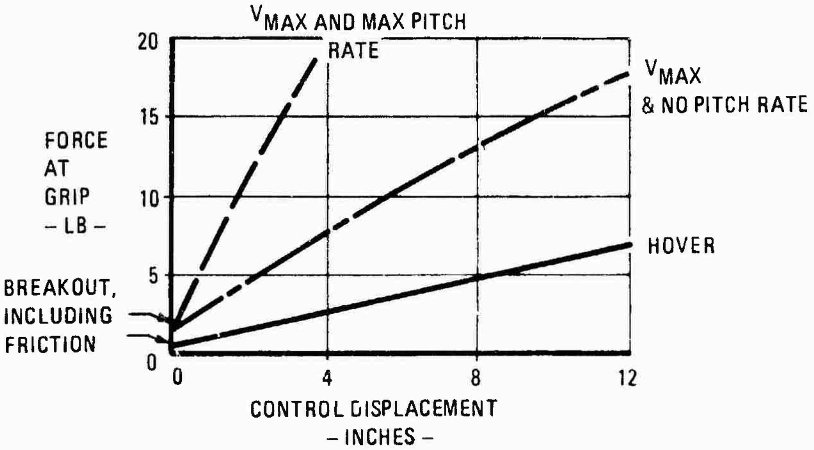


Figure 1. Variable Force-Feel System.

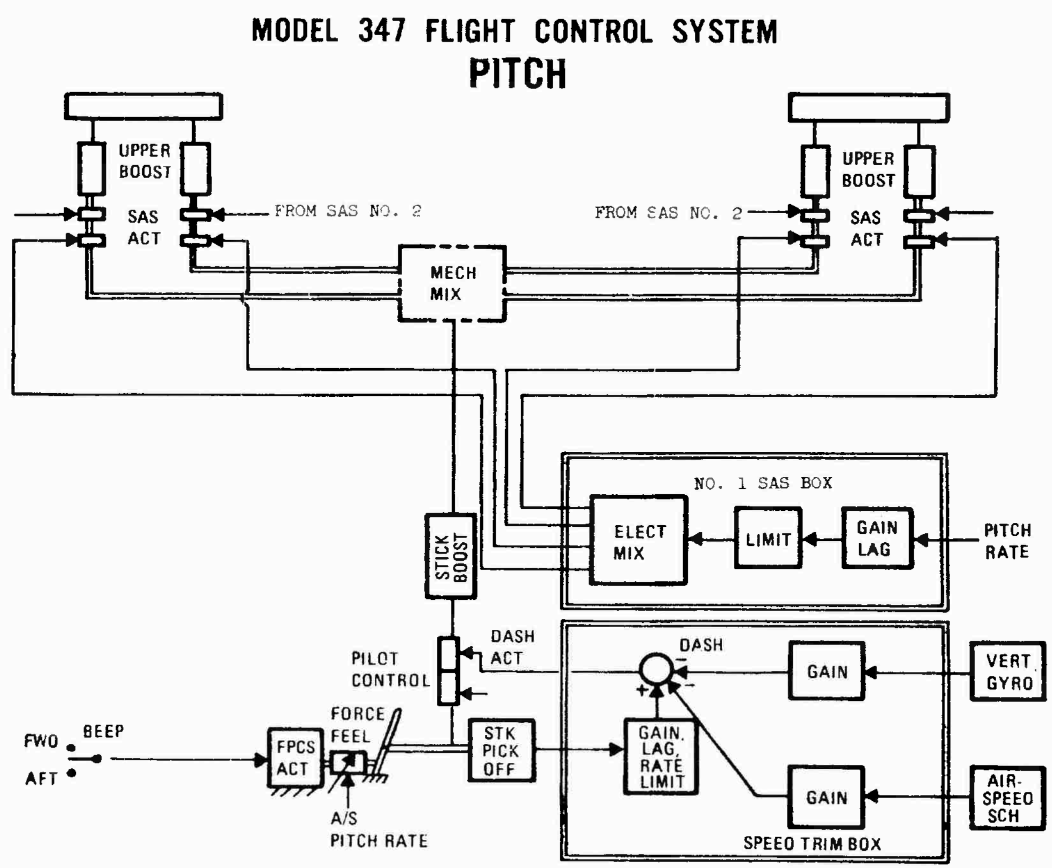


Figure 2. Model 347 Helicopter Flight Control System (Pitch).

6. By summation of aircraft pitch attitude, airspeed, and longitudinal control position, the DASH system actuator provides pitch attitude and airspeed hold as well as a stable gradient of longitudinal control position versus trim airspeed. Since neither pitch attitude nor airspeed are synchronized, their high gain settings would, if not otherwise modified, require very large longitudinal control displacements to oppose their effects. The incorporation of a longitudinal control position transducer (stick pick-off) cancels out most of the airspeed and attitude signal so that the trim control travel over the allowable operating speed range is reduced to approximately 2-1/2 inches.

7. The longitudinal control FPCS is used to provide vernier beep trim adjustment of airspeed and pitch attitude. Operation of the beep trim button causes the FPCS to be repositioned, moving the trim point of the variable force-feel capsule and, thereby, moving the pilot longitudinal control. Provision is made to use the FPCS actuator to introduce automatic commands to the longitudinal control during programmed approach to hover.

LATERAL CONTROL AXIS

8. A block diagram of the lateral control system is shown in figure 3. Rate damping is provided by shaped inputs from a roll rate gyro to the upper SAS actuators. As in the case of longitudinal SAS inputs, necessary processing of the lateral SAS signals is performed by the electrical mixing unit. A control position transducer opposes the rate gyro signals so that the high stability of the roll SAS does not degrade lateral maneuverability. Summation of the rate gyro signal and the control position signal produces a roll-rate response which is proportional to control displacement. Long-term hold of trim bank attitude is provided by bank-angle inputs from a vertical gyro. Bank-angle error signals from the vertical gyro are used to produce corrective control motions through the upper SAS actuators and through the slow rate trim actuator, which repositions the pilot lateral control. When the pilot displaces the lateral control out of the detent position during maneuvering, the trim bank-angle error is continuously synchronized to zero. When the pilot returns the control to the detent position and the aircraft roll rate is less than 1 degree per second, the synchronizer switch opens and the system holds the aircraft at the bank angle stored at the integrator output. Vernier beep trim of bank attitude is produced by operation of the lateral beep trim switch. Operation of the lateral beep trim switch adjusts the trim bank angle stored at integrator.

9. Coupled lateral steering commands are applied to the pilot control by the FPCS actuator. Input commands are derived from navigation error signals, heading error signals, and bank-angle error signals. During coupled flight, bank-angle and heading-hold error signals are synchronized to zero, and all long-term corrections are applied through the FPCS actuator. The variable force-feel system of the lateral control is programmed with airspeed only, in that lateral control position and, therefore, lateral control force is already proportional to the roll rate commanded.

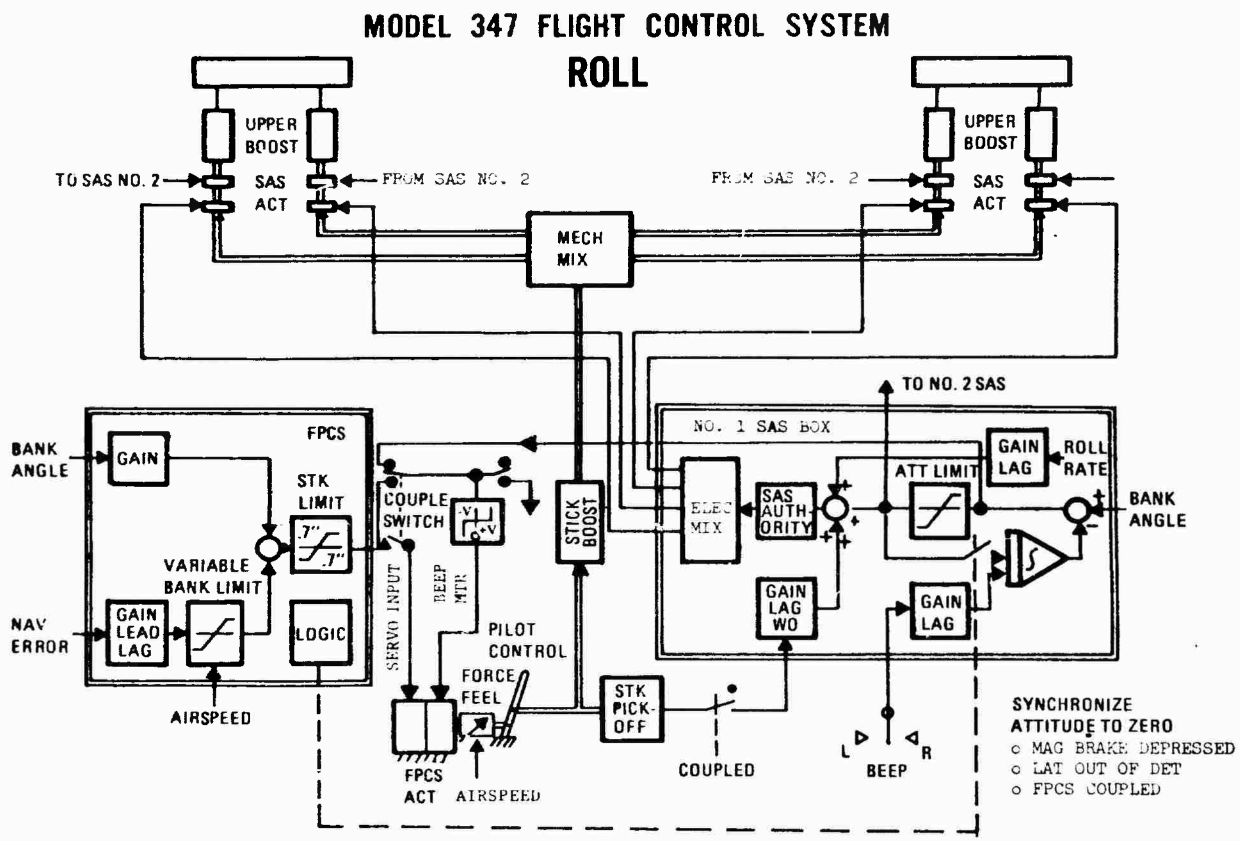


Figure 3. Model 347 Helicopter Flight Control System (Roll).

DIRECTIONAL CONTROL AXIS

10. As shown in figure 4, the directional control axis differs from the longitudinal and lateral axes, in that yaw SAS inputs are not processed by the electrical mixer. Since the yaw SAS actuator inputs occur before the mechanical mixer, yaw SAS inputs may be processed by the mechanical mixer. Rate damping and heading-hold functions are similar to those of the lateral axis. In addition, roll rate is coupled into the yaw SAS to provide automatic turn coordination.

11. Inputs from the directional control position transducer are used only at speeds below 40 KIAS. Yaw rate stabilization is also greatly reduced at speeds below 40 KIAS by a switching device. As on the CH-47C, static directional stability is artificially enhanced by sideslip sensing transducers. Vernier beep trim of aircraft heading is provided by a beep trim switch located on the thrust lever. Operation of the directional beep trim switch causes the yaw axis FPCS to reposition the directional pedal force-feel capsule, moving the trim position of the directional pedals.

THRUST AXIS

12. A block diagram of the thrust axis control system is shown in figure 5. Automatic inputs into the thrust control system are produced only by the thrust FPCS actuator. These FPCS inputs are introduced only when one of the coupled operating modes is selected by the pilot. The altitude-hold function is accomplished from a barometric altimeter reference which operates through a synchronizer. At any time that the altitude hold is turned off or the thrust brake trigger is depressed, the barometric reference is continually synchronized to the existing pressure altitude. When the altitude hold is engaged and the thrust lever trigger is released, the pressure altitude stored at the output of the integrator at the time of engagement is maintained by the FPCS. A vertical accelerometer provides a vertical damping input to the FPCS.

13. Engagement of the ILS coupled mode of operation causes the FPCS to move in response to ILS glide-slope signals. Interception of an ILS glide slope with both the barometric altitude hold and the ILS coupled mode engaged causes the altitude hold to be switched off automatically. Subsequent signals to the FPCS are provided only by the ILS receiver.

14. Two additional sources of automatic thrust control inputs are intended to be used in the Model 347. In the hover-hold mode, a radar altimeter reference is substituted for the barometric altitude hold. A programmed approach mode uses an onboard computer to derive a glide path which is used to automatically position the thrust FPCS to bring the helicopter to a hover at a reference point on the ground. At the time of the Model 347 Phase I evaluation, these two coupled modes of operation were not approved for use and were not evaluated.

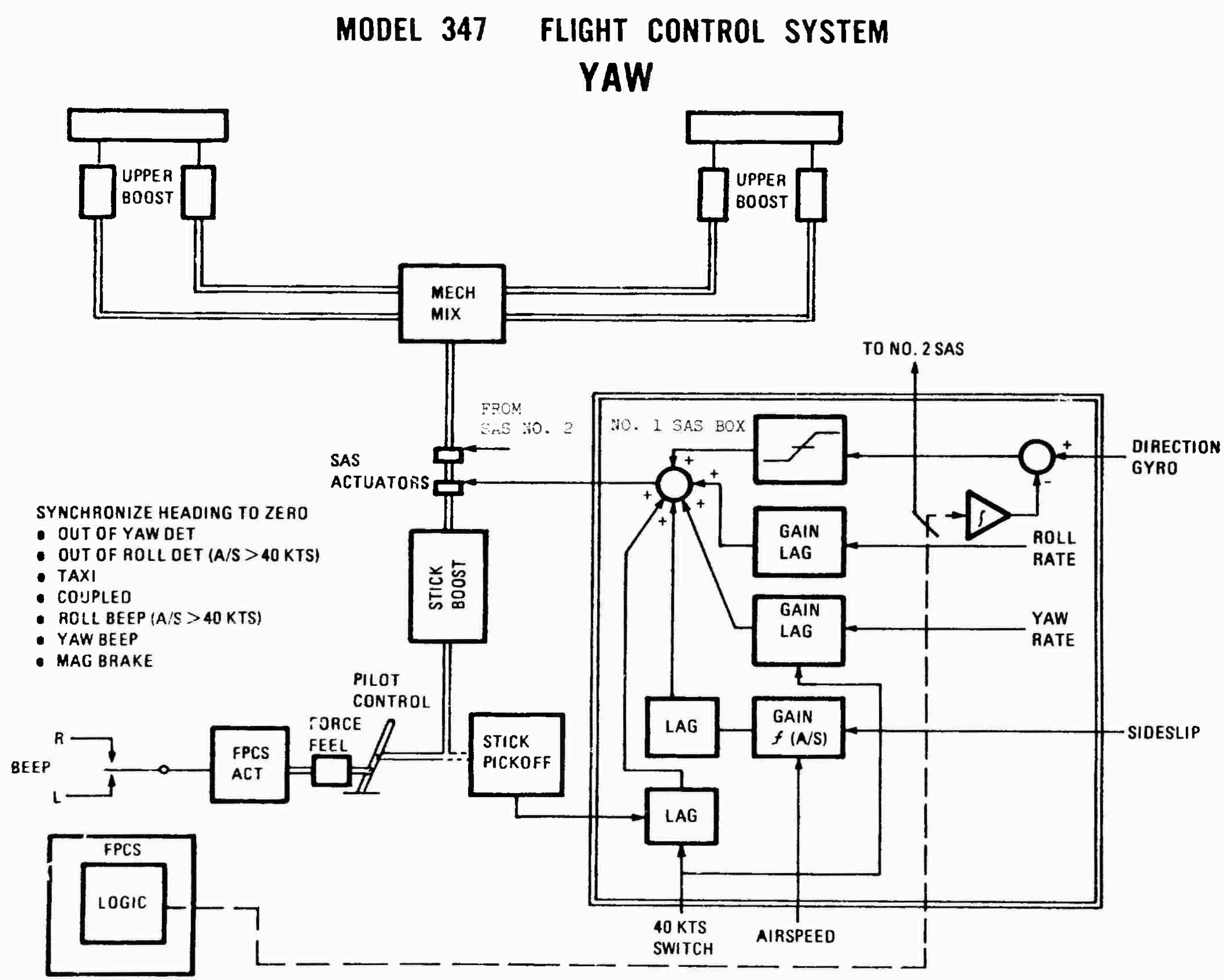


Figure 4. Model 347 Helicopter Flight Control System (Yaw).

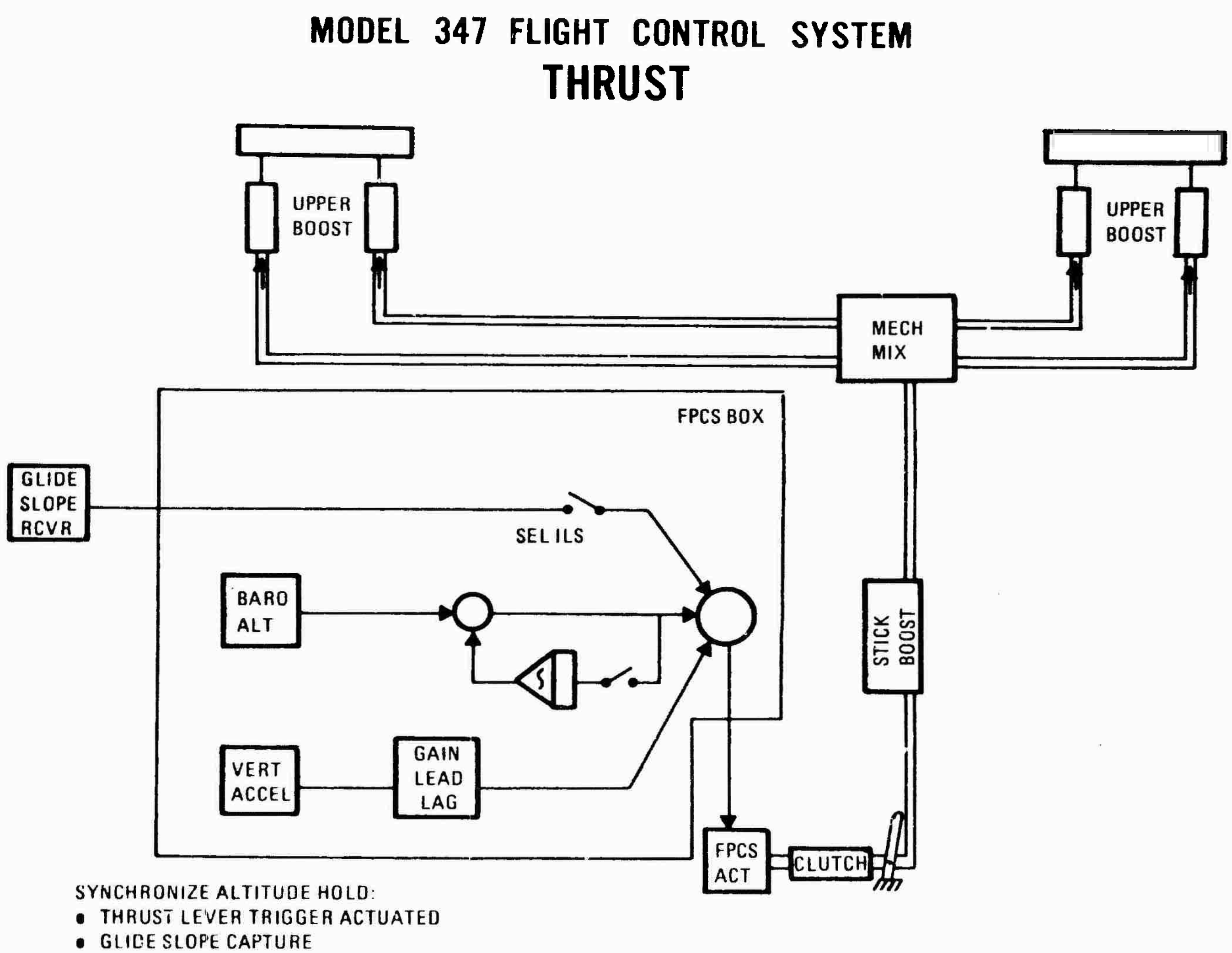


Figure 5. Model 347 Helicopter Flight Control System (Thrust).

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APPENDIX V. PHOTOGRAPHS

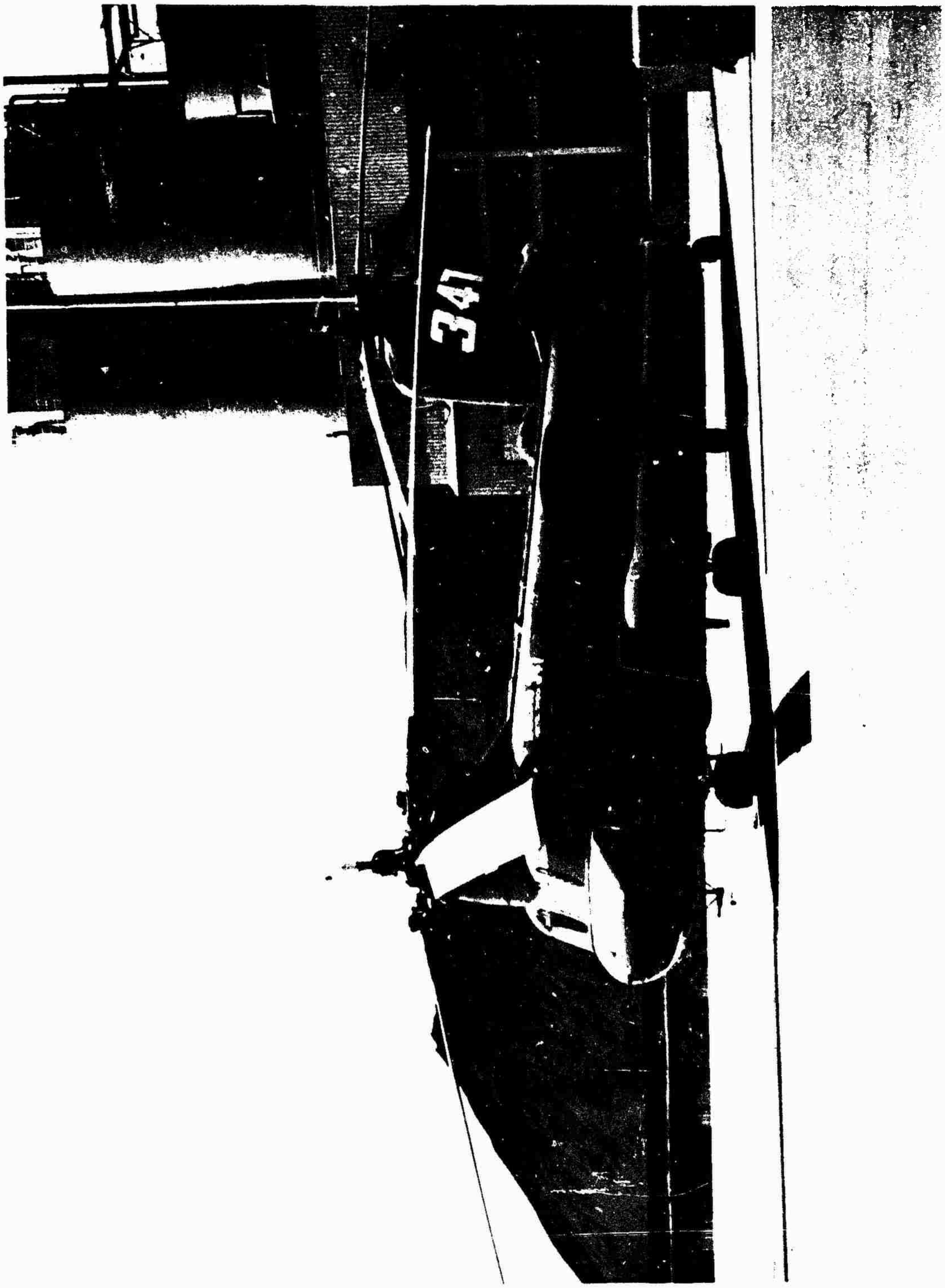


Photo 1. Left Front View.

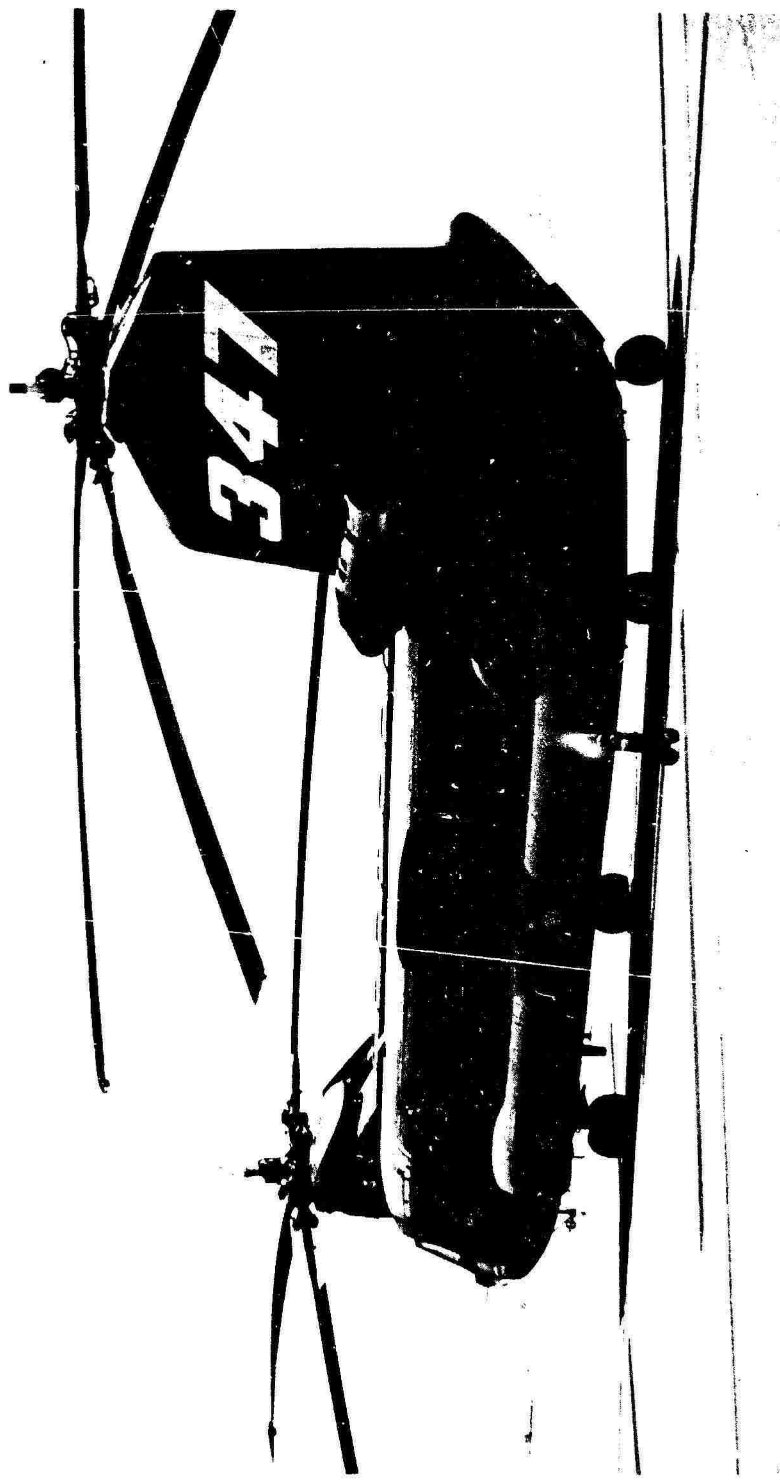


Photo 2. Left Rear View.

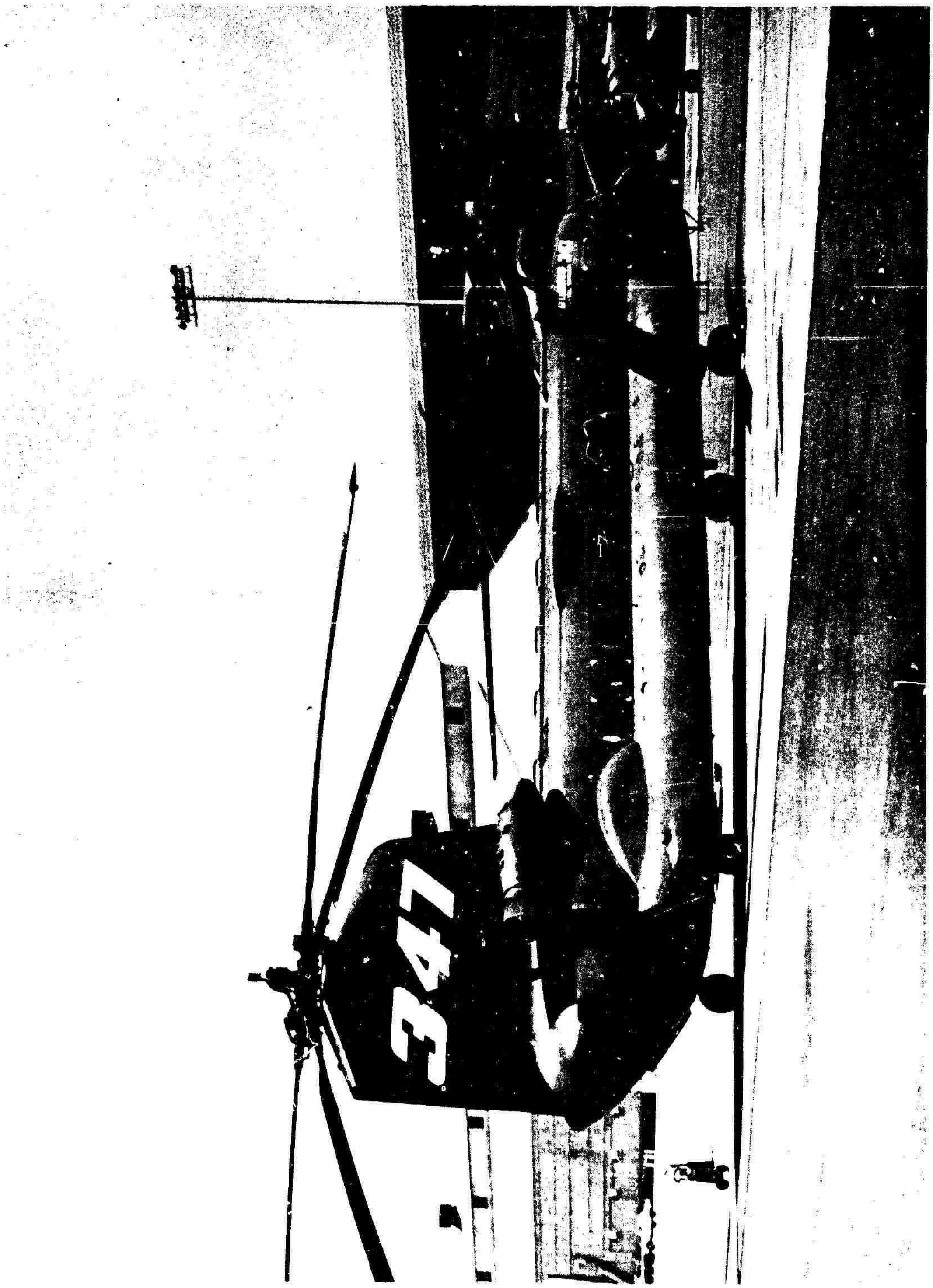


Photo 3. Right Rear View.

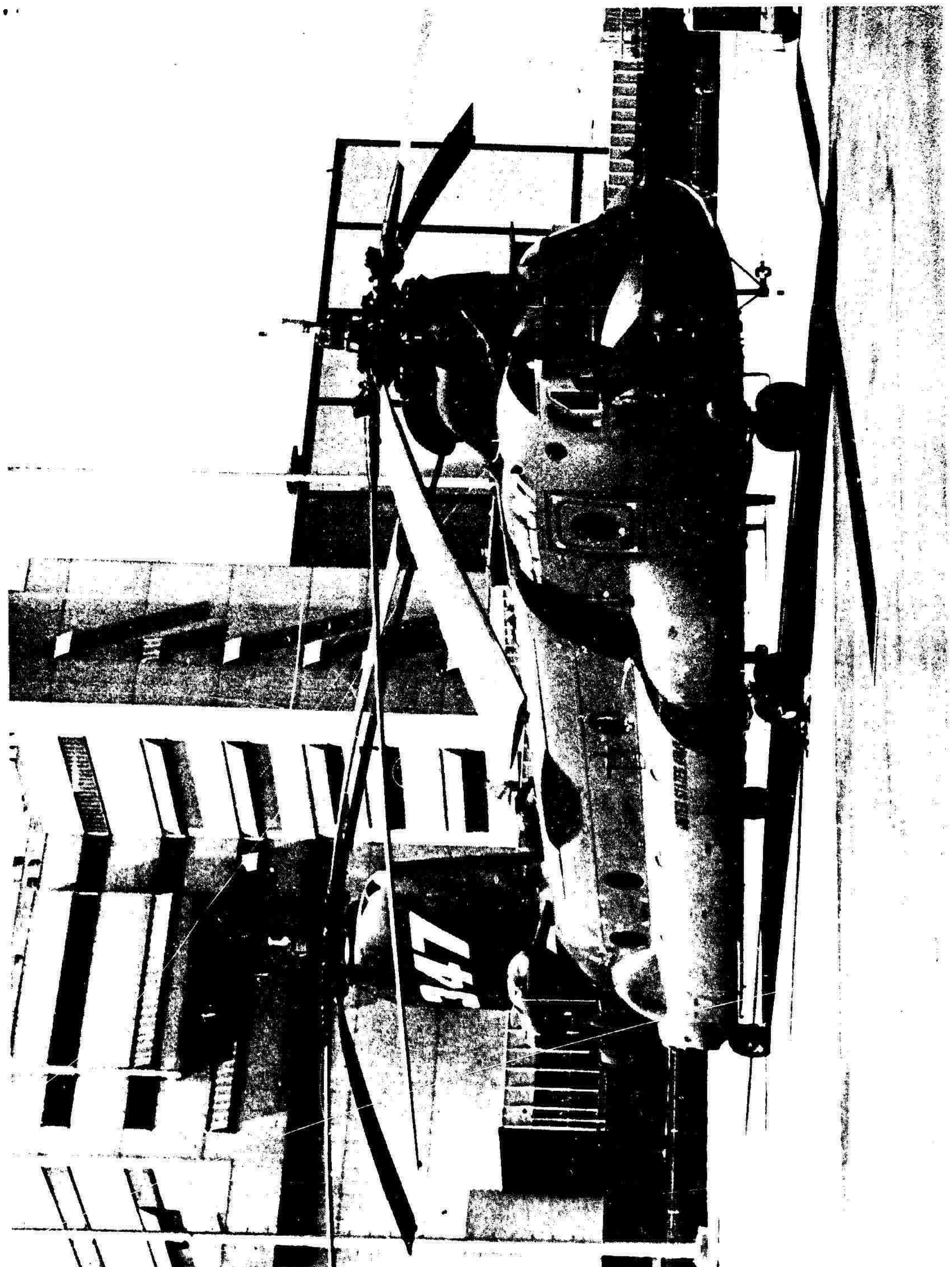


Photo 4. Right Front View.

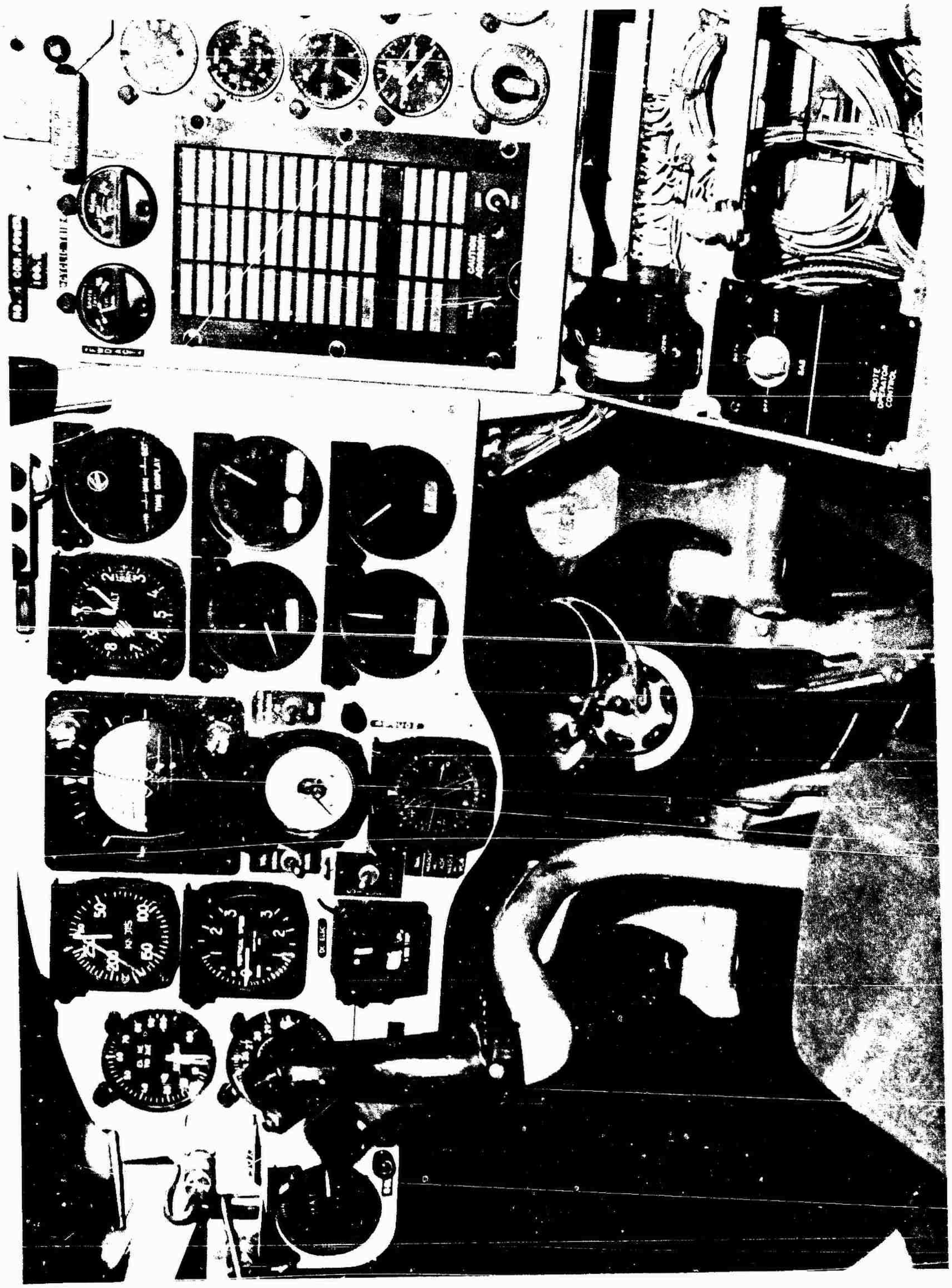


Photo 5. Copilot Instrument Panel.

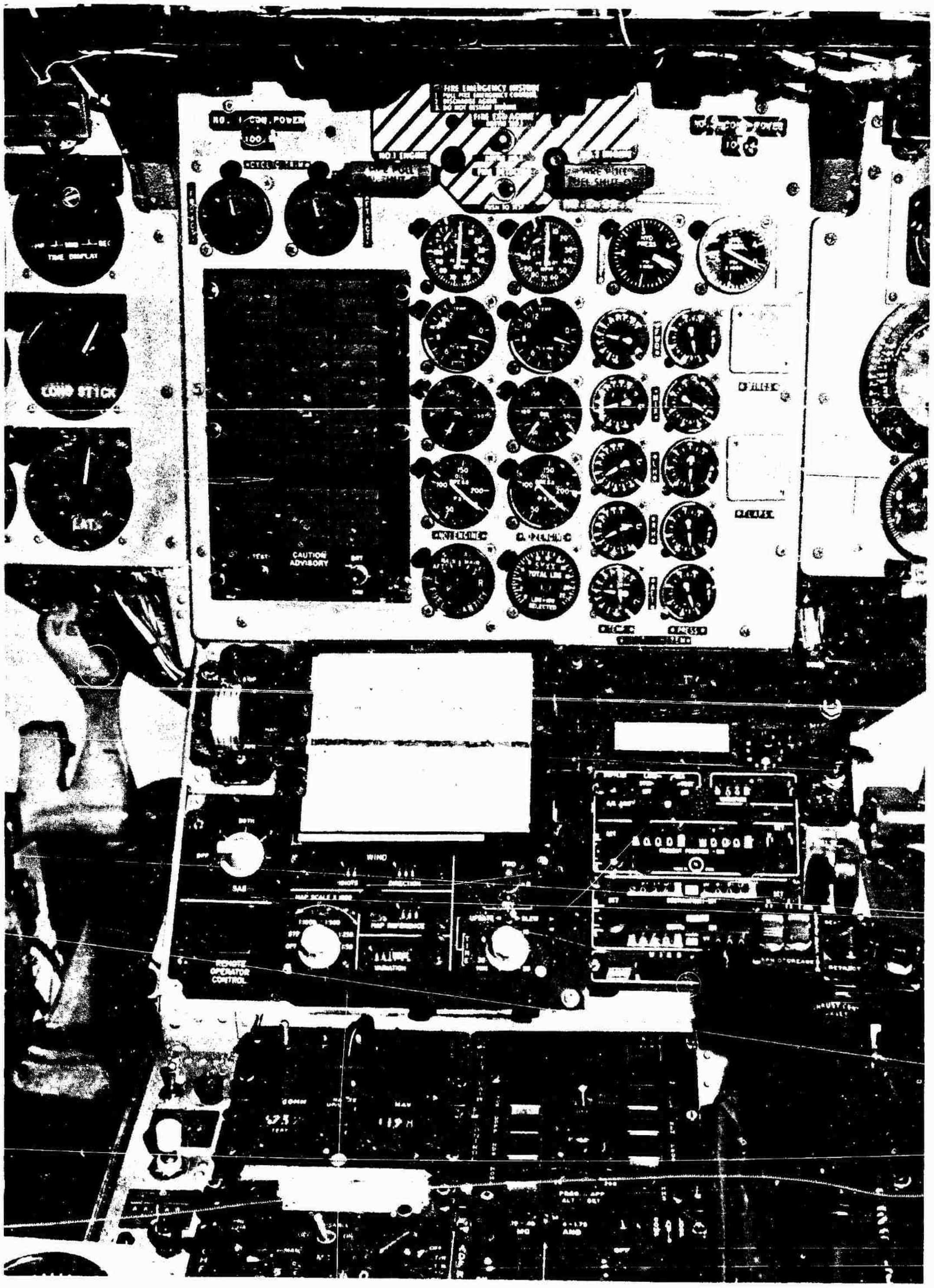


Photo 6. Center Instrument Panel.

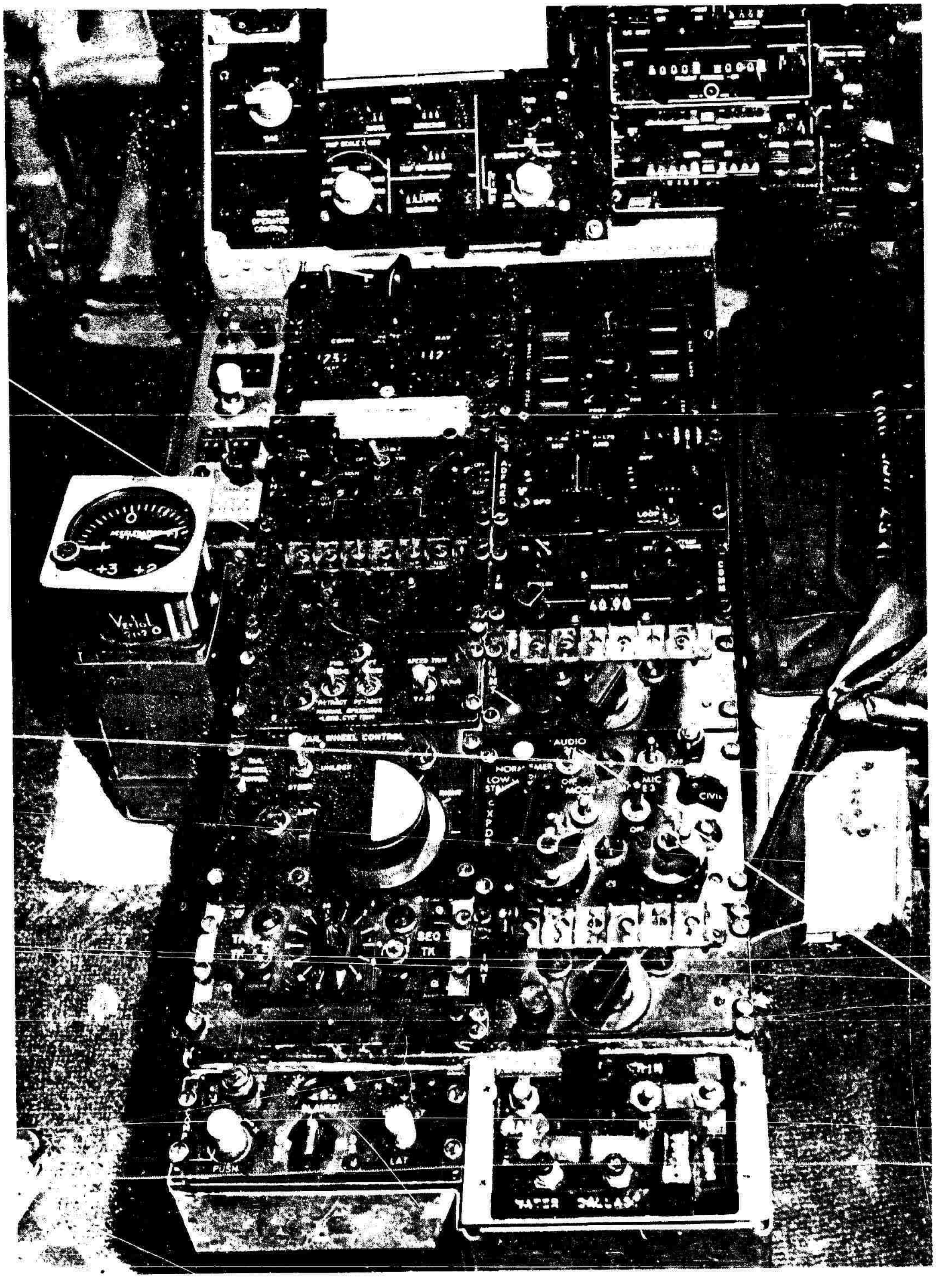


Photo 7. Center Console.

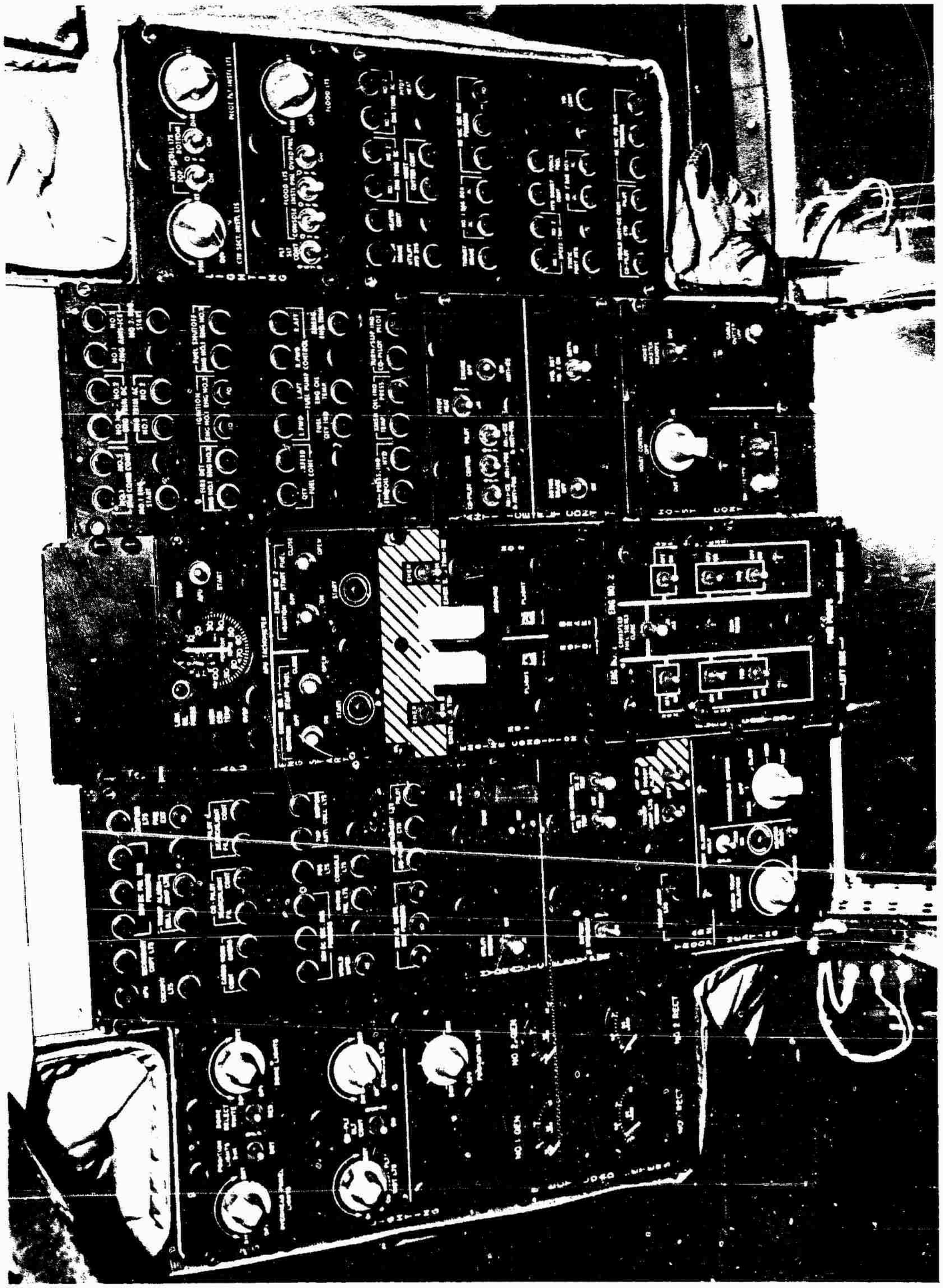


Photo 8. Overhead Console.

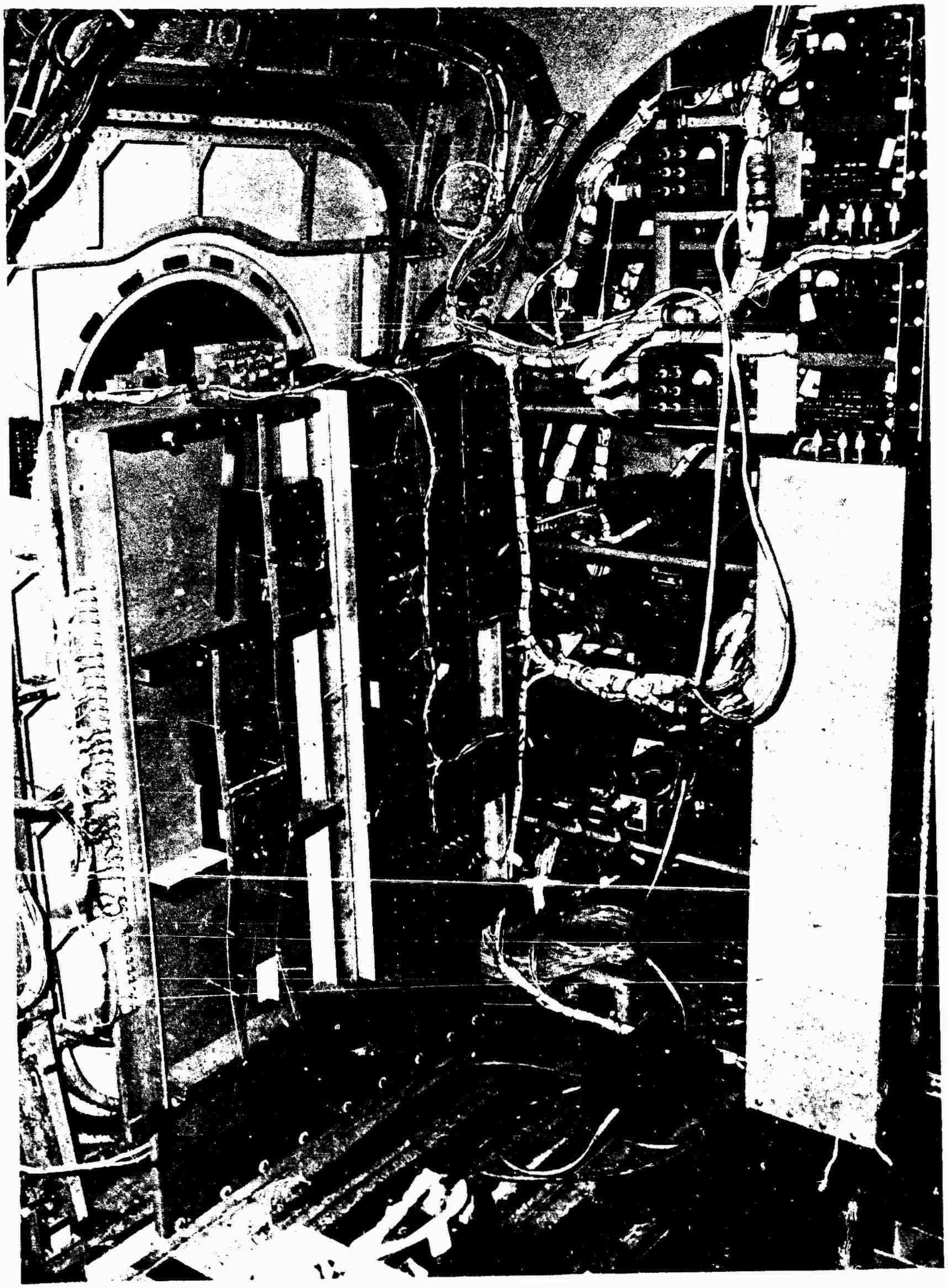


Photo 9. Left Front Corner of Cabin Showing Electronics Bay.

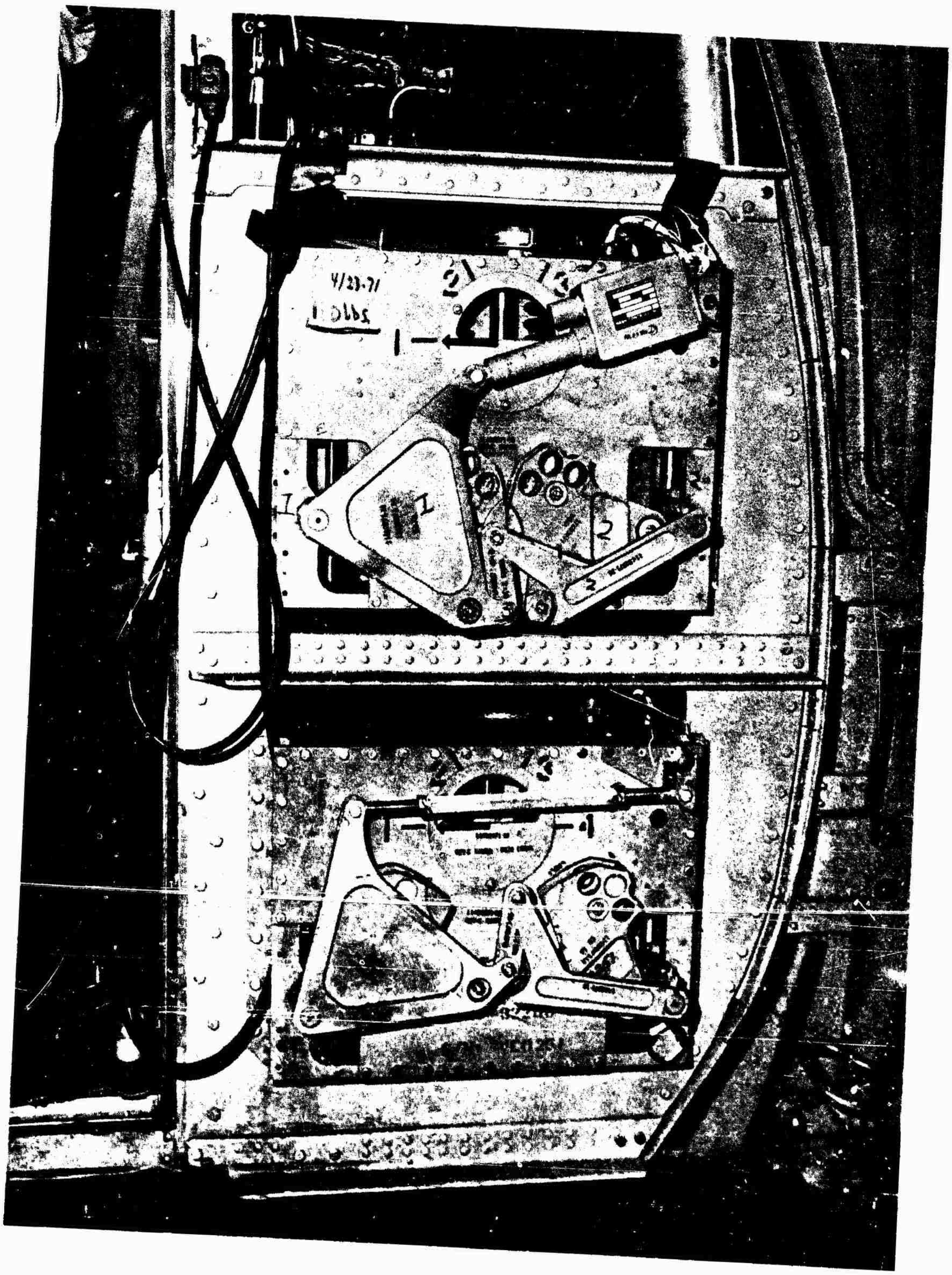


Photo 10. Right Front Corner of Cabin Showing Vibration Absorbers.

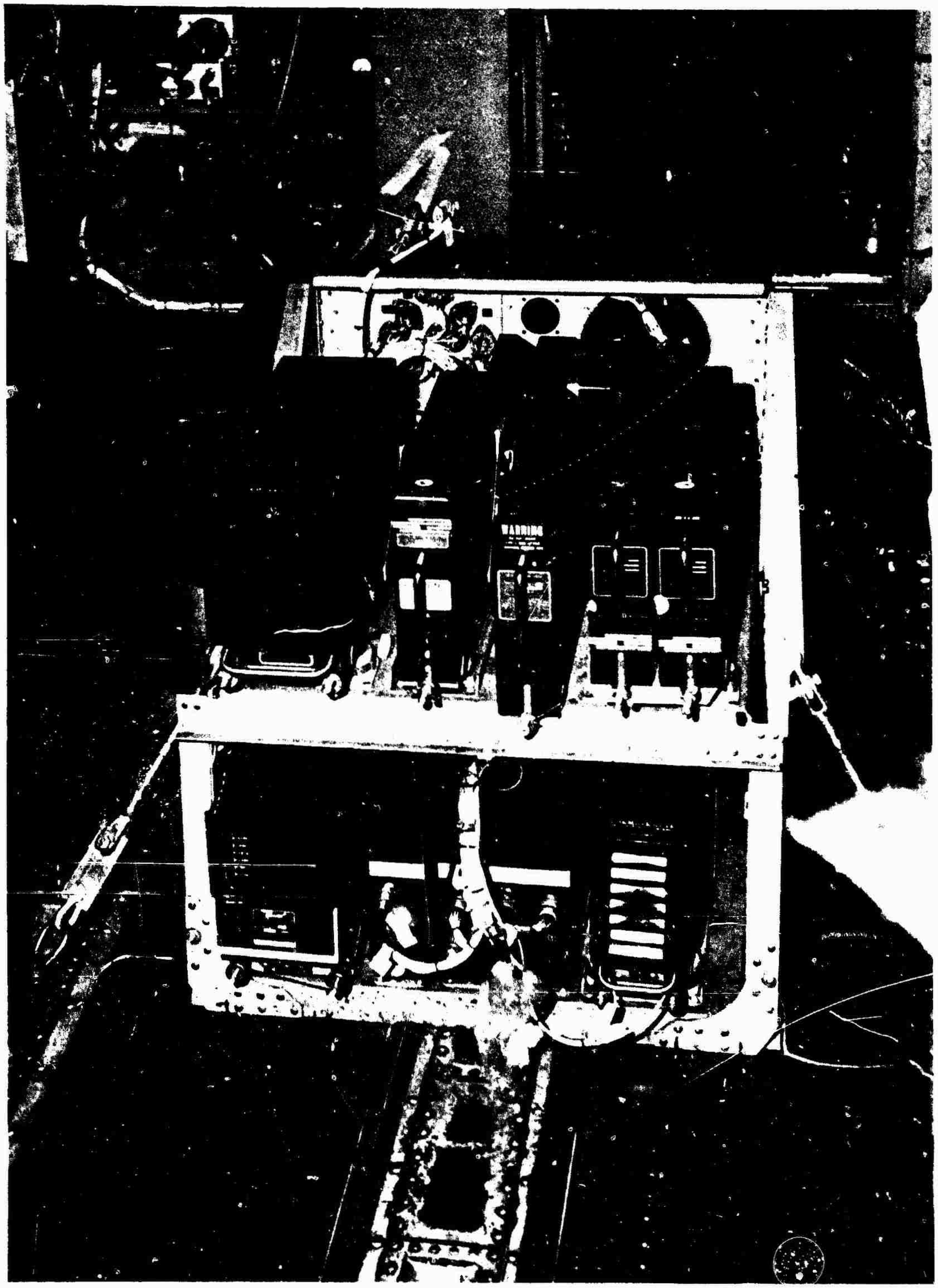


Photo 11. Test Instrumentation Mounted on Forward Cabin Floor.

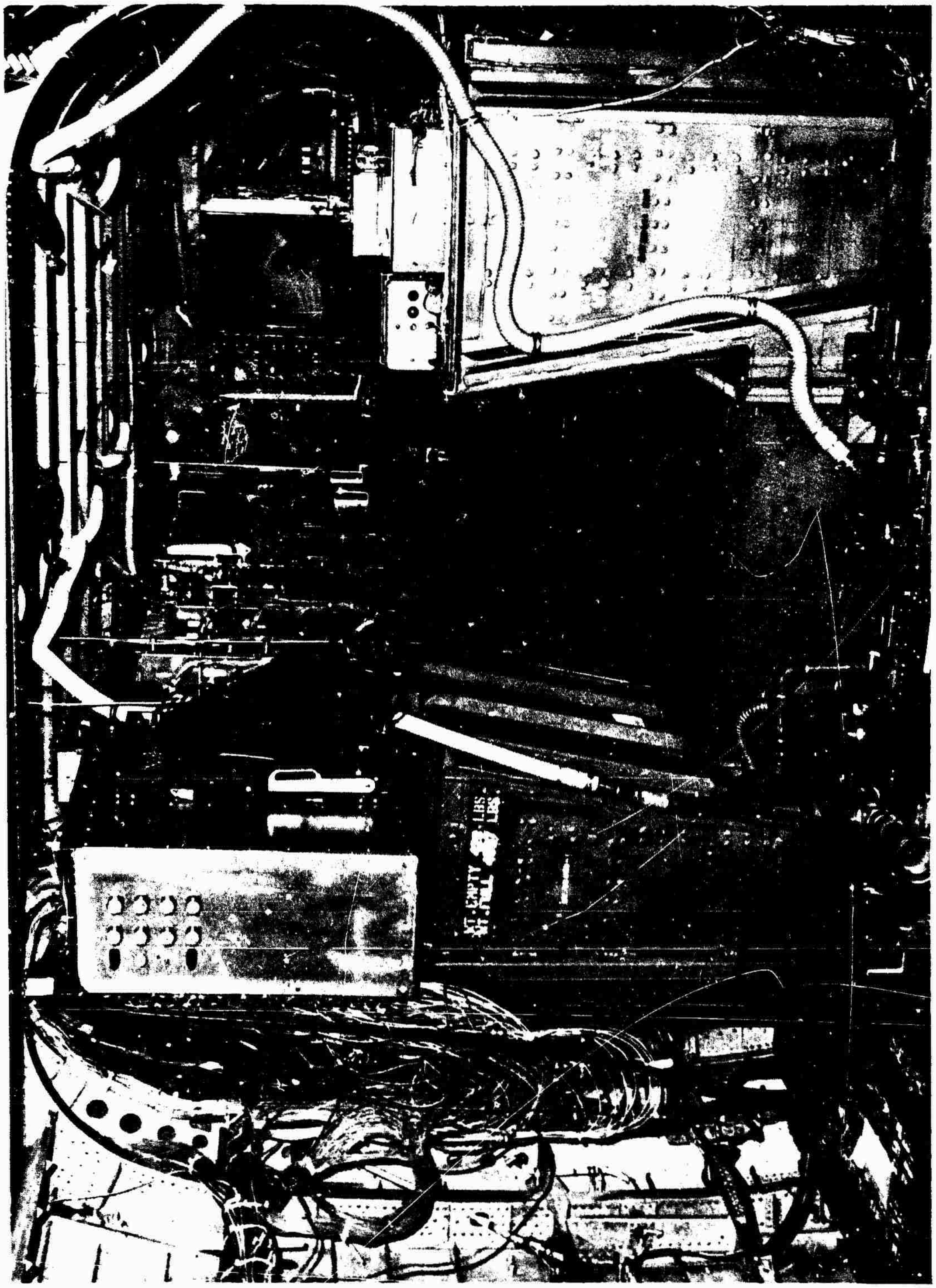


Photo 12. Water Ballast Tanks and Test Instrumentation Mounted at Mid-Cabin Area.

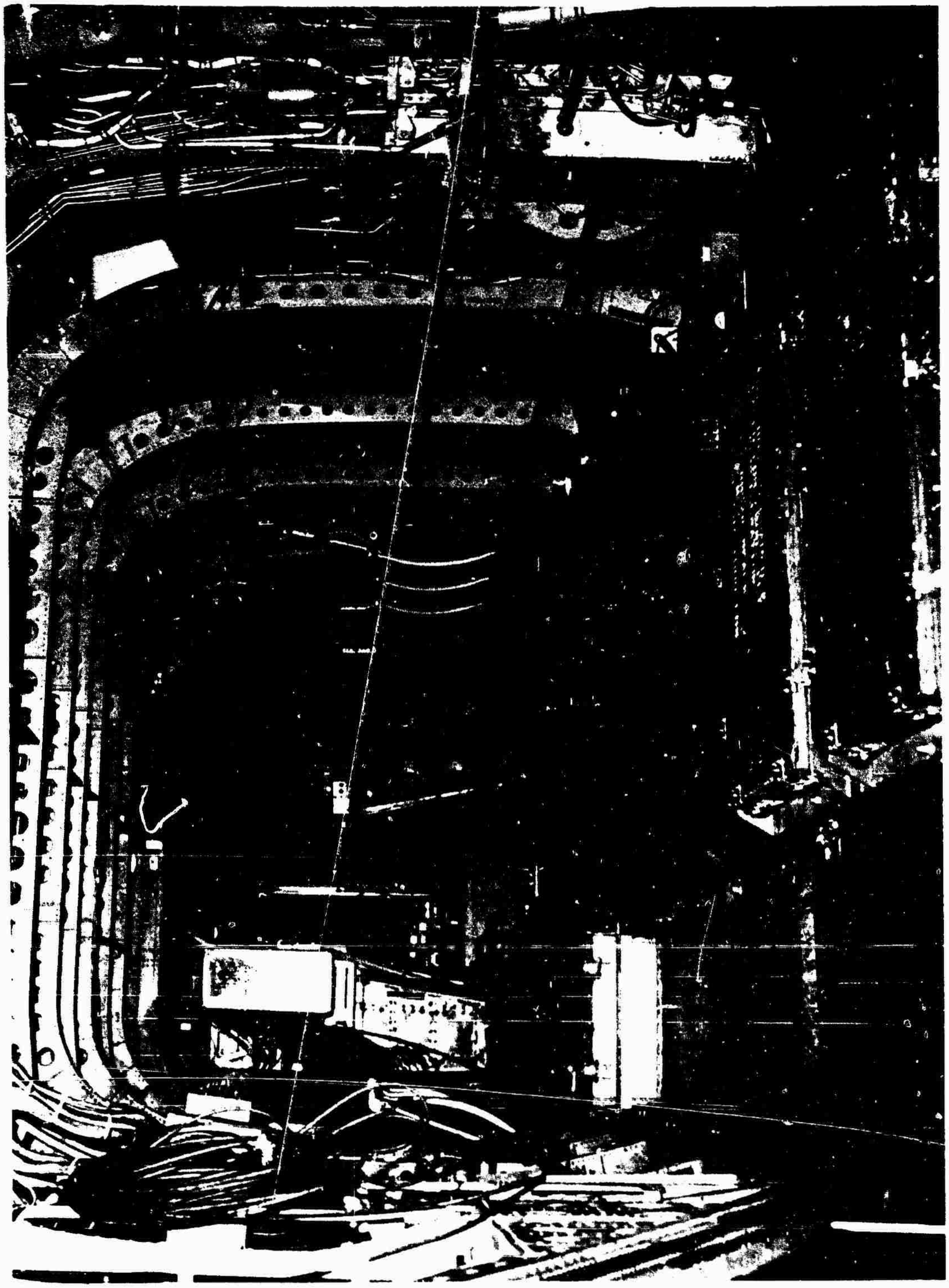
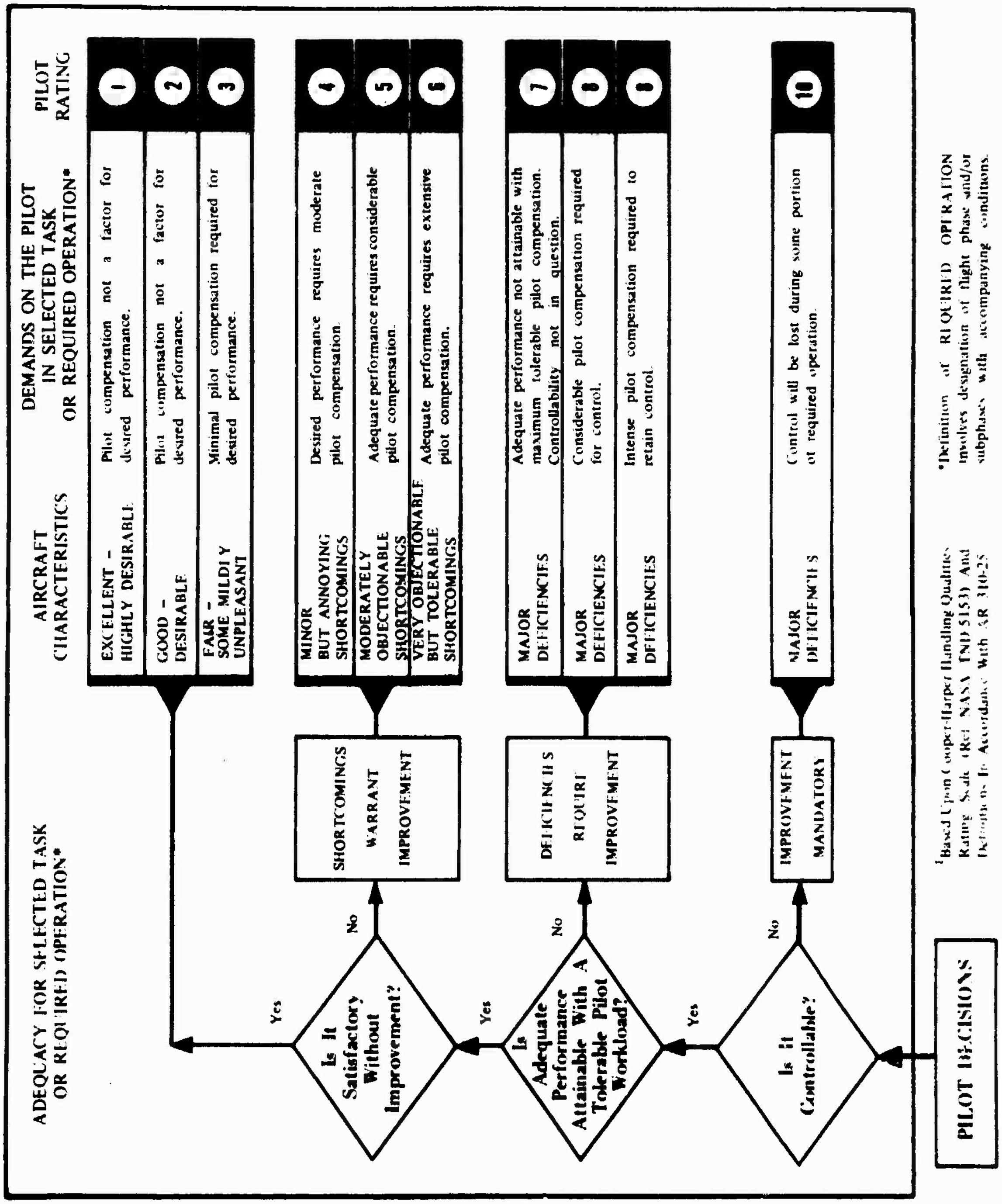


Photo 13. Lead Ballast Boxes Mounted at Aft-Cabin Area and Cargo Ramp.

APPENDIX VI. HANDLING QUALITIES RATING SCALE



[1] Based Upon Cooper-Harper Handling Qualities Rating Scale (Ref NASA TND 5153) And Definitions In Accordance With AR 310-25

*Definition of REQUIRED OPERATION involves designation of flight phase and/or subphases with accompanying conditions.

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APPENDIX VII. DATA REDUCTION AND ANALYSIS PROCEDURES

GENERAL

1. Nonstandard data reduction and analysis procedures were required in certain test areas, due to the unique characteristics of the Model 347 control system. The use of control position transducer (stick pick-off) inputs to modify the output of the augmentation systems, and the use of various augmentation devices to enhance static stability characteristics precluded the direct use of control position data to indicate static longitudinal stability, unaugmented static directional stability, and unaugmented longitudinal static trim characteristics.

STATIC LONGITUDINAL STABILITY CHARACTERISTICS

2. A dualized DASH actuator is located in the longitudinal control mechanical linkage. Airspeed and pitch attitude signal are fed into this series actuator to provide a high degree of stick-fixed speed and attitude stability. The airspeed and attitude gains are such that, if not otherwise modified, the DASH system would require excessively large longitudinal control motions to change airspeed and attitude. A control position transducer signal is, therefore, added to the airspeed and attitude signals to oppose the high static stability capability of the DASH system.

3. In order to present the static longitudinal stability in a manner which better indicates the true restoring moment existing at any off-trim airspeed, the stick pick-off contribution must be removed from the summation of control position factors. The total DASH system input to the control system may be written as:

DASH Input = Pitch Attitude Contribution + Airspeed Contribution + Pick-Off Contribution

Or:

$$\delta_{ACT} = G_\theta \theta + G_v (V - V_o) + G_s (\delta_{STICK} - \delta_o)$$

Where:

δ_{ACT} = DASH system actuator motion, expressed in terms of equivalent control (measured extension x 4.77)

G_θ = Pitch attitude gain (defined G = -0.32 in./deg)

θ = Pitch attitude, degrees (measured)

G_v = Airspeed gain (defined G_v = 0.11 in./kt)

V = Indicated airspeed, knots (measured)

V_o = Airspeed for zero speed command to actuator (defined V_o = 31.4 kt)

G_s = Actuator/stick pick-off gain (defined G_s = -6.6 in./in.)

δ_{STICK} = Longitudinal control position, inches (measured)

δ_o = Stick position for zero control position input to actuator, inches (defined δ_o = 0.2 in.)

Solving the DASH input for the pick-off contribution:

$$G_s (\delta_{STICK} - \delta_o) = \delta_{ACT} - G_\theta \theta - G_v (V - V_o)$$

Using the speed and attitude gains given above, the pick-off contribution is found and subtracted from the measured longitudinal control position to obtain the longitudinal control position without the pick-off, as follows:

$$\delta_{STICK} \text{ (without pick-off)} = \delta_{STICK} - G_s (\delta_{STICK} - \delta_o)$$

This procedure produces a value of equivalent control position at any airspeed. Equivalent control position data at various airspeeds are plotted to indicate the static longitudinal stability of the aircraft independent of variations of control pick-off position at off-trim airspeeds.

LONGITUDINAL STATIC TRIM CHARACTERISTICS

4. Actual longitudinal control position data accurately indicate the static trim characteristics of the Model 347 with the augmentation systems in operation. In order to indicate the static trim characteristics with augmentation systems inoperative, the variation of the DASH actuator motion with airspeed and attitude must be mathematically removed from the measured cockpit control position data. To accomplish this, the actual DASH actuator motion is measured and the equivalent control motion due to the DASH actuator is subtracted from the measured cockpit control position data in accordance with the following relationship:

Control Position (DASH OFF) = Control Position (DASH ON)
- DASH Actuator Equivalent Control Motion

Or, using the symbology defined in paragraph 3:

$$\delta_{STICK\ (DASH\ OFF)} = \delta_{STICK\ (DASH\ ON)} - \delta_{ACT}$$

Since the DASH system actuator remains fixed in the position in which the system is turned off, the origin of the DASH-OFF trim curve is dependent upon the airspeed and attitude existing at the instant it is turned off. If the longitudinal control and aircraft are not displaced from the conditions existing at the time the DASH system is turned off, the term "δ_{ACT}" is zero and $\delta_{STICK\ (DASH\ OFF)}$ equals $\delta_{STICK\ (DASH\ ON)}$.

STATIC DIRECTIONAL STABILITY CHARACTERISTICS

5. The static directional stability characteristic of the aircraft is indicated by the variation of directional control position with sideslip. The characteristic with both SAS operating is simply described by the measured control position data. In order to describe the SAS-OFF characteristics, it is necessary to mathematically remove the contribution provided by the yaw SAS actuators. The relation between yaw SAS actuator motion and directional control motion is known to be:

Equivalent Directional Control Motion = (1.75) (yaw SAS actuator motion)

The following relationship, therefore, describes the SAS-OFF directional control position and can be used to indicate SAS-OFF directional stability:

$$\text{Directional Control Position}_{\text{(SAS OFF)}} = \text{Directional Control Position}_{\text{(SAS ON)}} - (1.75)\ (\text{yaw SAS actuator motion})$$

Or:

$$\delta_{\text{Pedal (SAS OFF)}} = \delta_{\text{Pedal (SAS ON)}} - (1.75)\ (\delta_{\text{SAS}})$$

APPENDIX VIII. TEST INSTRUMENTATION

GENERAL

1. All test instrumentation was installed, calibrated, and maintained by the contractor at the test site. Except for the cg normal acceleration instrumentation and the airspeed instrumentation, all instrumentation was calibrated prior to the start of the test program. Airspeed and normal acceleration instrumentation were calibrated during the conduct of the test.

TEST PARAMETERS RECORDED

2. Quantitative data were obtained from both cockpit displays and from a magnetic tape recorder installed in the forward area of the cabin. The following test parameters were recorded:

Magnetic Tape

Airspeed (boom system)
Airspeed (ship's system)
Altitude (boom system)
Altitude (ship's system)
Outside air temperature
Time of day
Angle of sideslip
Rotor rpm
#1 engine fuel-flow rate
#2 engine fuel-flow rate
#1 engine fuel temperature
#2 engine fuel temperature
#1 engine N_1
#2 engine N_1
#1 engine fuel total
#2 engine fuel total
Forward rotor shaft torque
Aft rotor shaft torque
Event marker
Record counter
Longitudinal control position
Lateral control position
Directional control position
Thrust lever position
Pitch attitude
Roll attitude
Yaw attitude

Pitch angular rate
Roll angular rate
Yaw angular rate
Center-of-gravity normal acceleration
Differential airspeed-hold system actuator position (upper)
Differential airspeed-hold system actuator position (lower)
Longitudinal cyclic speed trim position (forward)
Longitudinal cyclic speed trim position (aft)
#1 yaw SAS extensible link position
#2 yaw SAS extensible link position
Swiveling actuator position (forward and aft head)
Pivoting actuator position (forward and aft head)
Vertical vibration:
- FS 50
- BL 35 left
- WL -19

Lateral vibration:
- FS 50
- BL 35 left
- WL -19

Vertical vibration:
- FS 50
- BL 35 right
- WL -19

Vertical vibration:
- FS 95
- BL zero
- WL -17

Lateral vibration:
- FS 95
- BL zero
- WL -17

Longitudinal vibration:
- FS 95
- BL zero
- WL -17

Vertical vibration:
- FS 160
- BL 49 left
- WL -30

Lateral vibration:
- FS 160
- BL 49 left
- WL -30

Vertical vibration:
- FS 360
- BL 49 left
- WL -30

Vertical vibration:
FS 360
BL 49 right
WL -30
Lateral vibration:
FS 360
BL 49 right
WL -30
Vertical vibration:
FS 592
BL 49 left
WL -30
Vertical vibration:
FS 592
BL 49 right
WL -30
Lateral vibration:
FS 592
BL 49 right
WL -30

Cockpit

Airspeed (boom system)
Airspeed (ship's system)
Altitude (boom system)
Altitude (ship's system)
Outside air temperature
Time of day
Angle of sideslip
Rotor rpm
#1 engine N_1
#2 engine N_1
#1 engine torque
#2 engine torque
Fuel quantity indicator
Event marker
Record counter
Longitudinal control position
Lateral control position
Directional control position
Thrust lever position
Center-of-gravity normal acceleration

3. Vibration sensors are mounted to the airframe as follows:

a. On canted deck, immediately forward of heel slide (canted deck is the extreme forward portion of floor where floor is connected to skin structure): FS 50.

b. On floor panel, immediately aft of pedestal: FS 95.

c. On floor panel, between floor outer tiedown and aircraft outer skin: FS 160, FS 360, and FS 592.

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PHASE I

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11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY US ARMY AVIATION SYSTEMS COMMAND ATTN: AMSAV-EF PO BOX 209, ST. LOUIS, MISSOURI 63166

13. ABSTRACT
The US Army Aviation Systems Test Activity (USAASTA) conducted the Phase I technical evaluation of the Boeing-Vertol Model 347 advanced technology research helicopter during the period 28 May to 19 June 1971. The Model 347, a derivative of the CH-47 transport helicopter, was tested at the contractor's facility near Philadelphia, Pennsylvania. Performance, handling qualities, vibration, and noise characteristics were evaluated to determine the improvements provided by incorporation of advanced technology systems in a large tandem-rotor transport helicopter. Compliance with the provisions of the contract statement of work and with military specifications, MIL-H-8501A and MIL-A-8806A, was determined. Several shortcomings identified during the testing were corrected by the contractor after the testing was completed. The effects of these corrections were evaluated during additional testing conducted on 11 and 12 August 1971. Level flight performance and out-of-ground-effect hover performance were significantly improved over that of the CH-47C helicopter. The excellent static longitudinal stability characteristics enhanced the mission capability of the aircraft. The steering and glide-path modes of the automatic flight path control system worked satisfactorily and reduced the pilot workload in instrument flight conditions. Cockpit noise and vibration characteristics were noticeably improved over those of the CH-47C. Correction of the faulty logic circuitry in the flight director steering command function was recommended to eliminate a hazardous flight condition. Correction of the following shortcomings was recommended to improve mission capabilities: downward slippage of thrust control rod, inadequate side-force characteristic in autorotation, excessively small turn needle and inclinometer, location of mode advisory lights, excessive long-term altitude error of altitude-hold system, oscillation of deviation indicator and flight director steering command bar, lack of heading synchronization indication on the pilot horizontal situation display, and vertical vibration level in aft cabin area. Five additional shortcomings of the cockpit displays and avionics systems were identified, and correction was recommended. The Model 347 helicopter failed to meet the requirements of six paragraphs of MIL-H-8501A and the cockpit vibration limits of the contract statement of work.

14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Phase I technical evaluation Boeing-Vertol Model 347 Performance, handling qualities, vibration, and noise characteristics were evaluated Determine improvements Compliance determined Several shortcomings corrected Excellent static longitudinal stability characteristics Five additional shortcomings identified Correction was recommended						