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AIRWORTHINESS AND FLIGHT CHARACTERISTICS TEST (A&FC) OF THE BHTI 214ST HELICOPTER

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FINAL REPORT

UNITED STATES ARMY AVIATION ENGINEERING FLIGHT ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

The Airworthiness and Flight Characteristics Test of the Bell 214ST helicopter (S/N N3186W) was conducted by the US Army Aviation Engineering Flight Activity (USAAEFA). The test was conducted at Edwards Air Force Base (elevation 2302 feet), Bakersfield (elevation 488 feet), Bishop (elevation 4120 feet) and Coyote Flats (elevation 9980 feet), California. A total of 62.6 hours (27.5 productive hours) were flown between 1 July and 27 August 1985. Hover, level flight and autorotational descent performance tests were conducted. Handling

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SECURITY CLASSIFICATION OF THIS PAGE (When Data Entered) qualities tests included static and dynamic stability, maneuvering stability, power management, systems failures, mission maneuvers and flight in simulated Instrument Meteorological Conditions (IMC). The airspeed for best glide distance determined from the autorotational descent performance test results of this evaluation was significantly less than the best glide distance airspeed presented in the flight manual. The out-of-ground effect hover test results when compared to the flight manual indicated less hover capability at standard day conditions and altitudes below 8060 feet but better hover capability at higher altitudes. The maximum difference between the flight manual and the test results of this evaluation was 250 pounds, below 8060 feet. Two major conclusions reached were: the 214ST is an excellent IMC aircraft with its IMC capabilities exceeding those of any US Army helicopters and its cockpit design significantly reduces pilot workload in most modes of flight. The most significant enhancing characteristics were: the low level of cockpit vibrations; the installation of an air data computer; the incorporation of a rotor brake and the low noise levels in the cockpit. The one unsatisfactory characteristic identified was the position of the collective during the first engine start which could result in a hazardous operating condition. Sixteen undesirable characteristics were identified of which the most significant were: the limited field of view during decelerations; increased pilot workload at various wind conditions during low speed flight; inaccuracies of the radio altimeter due to masking by relatively small external loads. Although this aircraft was not designed to meet military specifications, compliance with MIL-H-8501A was noted where data were available. Some flight control parameters did not comply with MIL-H-8501A, however, none of the noncompliances were considered to be significant. Several recommendations are made for the incorporation (in military aircraft) of various subsystems which were installed in the 214ST helicopter.

CLARIFICATION

The BHTI 214ST helicopter is a commercial aircraft and was not designed for US military use or compliance with US military specifications. However, in some cases, comparison of test results was made with the appropriate portions of military specification MIL-H-8501A to aid Army readers in their interpretation of the results.

During this evaluation, certain mission maneuvers were performed to assess the flight characteristics of a modern two-bladed (semi-rigid) helicopter while performing these maneuvers. The results of this evaluation will be used to expand the US Army's helicopter technical data base. The adequacy of the BHTI 214ST helicopter to perform as a military aircraft should not be inferred from the results of this evaluation.

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INTRODUCTION

BACKGROUND

1. Bell Helicopter Textron Inc. (BHTI), Fort Worth, Texas has produced a medium weight civilian transport helicopter, the 214ST, which is a twin engine growth version of the 214A helicopter. BHTI offered the use of a 214ST to the United States Army Aviation Systems Command (AVSCOM) for the conduct of an Airworthiness and Flight Characteristics (A&FC) evaluation. On 18 June 1985, the United States Army Engineering Flight Activity (USAAEFA) was tasked by AVSCOM (ref 1, app A) to plan, conduct, and report on the findings of a limited A&FC evaluation of the 214ST.

TEST OBJECTIVE

2. The objective of this evaluation was to provide test results which will be incorporated in the technical data base established by AVSCOM in support of technical advancements.

DESCRIPTION

3. The test 214ST helicopter, N3186W (photos 1 through 8, app B), is a twin engine, single main rotor configured helicopter with nonretractable wheel-type landing gear. A movable fly-by-wire horizontal elevator is located on the lower portion of the tail rotor pylon. The main and tail rotors are two-bladed semi-rigid systems. The delta-hinged, tractor-type tail rotor is attached to the right side of the tail pylon. Maximum gross weight of the helicopter is 17,500 pounds. The 214ST is powered by two General Electric (GE) CT7-2A turboshaft engines having an uninstalled power available (5 minute limit) of 1625 shaft horsepower (shp) at a power turbine speed of 21,000 revolutions per minute (rpm) each at sea level, standard day static conditions. Installed dual engine power is transmission limited to 2350 shp (5 minute limit). The aircraft also has an automatic flight control system (AFCS), a flight director system, and a two-axis (pitch and roll) attitude/altitude retension system (AARS). A more detailed description of the 214ST is included in appendix B and additional descriptions can be found in the FAA Approved Rotorcraft Flight Manual (ref 2, app A), 214ST Rotorcraft Operators Handbook (ref 3), Training Manual (ref 4) and Product Data Manuals (ref 5).

TEST SCOPE

4. The major portion of flight testing was conducted at Edwards Air Force Base, California (2302 feet). Portions of the level

flight performance testing and level flight airspeed calibration were conducted at Bakersfield, California (488 feet). Low airspeed flying qualities and tethered hover performance tests were conducted at Bishop (4120 feet) and Coyote Flats (9980 feet), California. A total of 34 flights were conducted between 1 July and 27 August 1985, totalling 62.6 flight hours (includes 20 ferry flight hours). Total productive flight test time was 27.5 hours. BHTI personnel installed, calibrated (verified by USAAEFA personnel) and maintained all the test instrumentation and performed all required maintenance on the helicopter. Flight restrictions and operating limitations observed during the A&FC evaluation are contained in the FAA Approved Rotorcraft Flight Manual (referred to as "Flight Manual"). Testing was conducted in accordance with the test plan (refs 6 and 7) at the conditions shown in tables 1 and 2.

TEST METHODOLOGY

5. A detailed listing of test instrumentation is contained in appendix C. Established flight test techniques and data reduction procedures were used (refs 8 and 9, app A) and are described in appendix D. A Handling Qualities Rating Scale (HQRS) (fig. 1, app D) and a Vibrations Rating Scale (VRS) (fig. 2) were used to augment pilot comments relative to aircraft handling qualities and vibrations. The flight test data were obtained from test instrumentation displayed on the instrument panel and recorded on magnetic tape installed in the aircraft. Real time telemetry monitoring of selected data parameters was used during certain tests.

Table 1. Performance Test Conditions[1]

Test	Gross Weight (lb)	Longitudinal Center of Gravity (FS)	Density Altitude (ft)	Trim Airspeed (KCAS)[2]
Hover	11,580 to 17,740[3]	242.0	5640 to 11,480	Zero[4]
Level Flight	13,040 to 15,710	232.4	3780 to 13,200	38 to 155
Autorotational Descent[5]	16,000	232.2	7960	57 to 99

NOTES:

[1]Tests were conducted at an approximate mid lateral center of gravity location with the stability and control augmentation system ON in the clean configuration (fuselage mounted passenger steps not installed) and at 100% (287 rpm) main rotor speed unless otherwise noted.
[2]KCAS: Knots calibrated airspeed
[3]Aircraft gross weight plus tether cable tension.
[4]Winds: 3 knots or less.
[5]Main rotor speed was varied from 276 rpm to 302 rpm at 79 KCAS.

Table 2. Handling Qualities Test Conditions

Test	Gross Weight (lb)	Longitudinal Center of Gravity (FS)	Density Altitude (ft)	Trim Airspeed (KCAS)[2]	Flight Condition
Control Positions in Trimmed Forward Flight	13,340 to 15,710	232 to 243.5	3780 to 13,200	35 to 149	Level
	16,220	232.4	7450	54 to 108	Climb with maximum continuous power
	16,100	232.1	7700	57 to 99	Descent at 1000 fpm
Static Longitudinal Stability	14,880	243.8	7230	82 and 112	Level
	16,730	237.2	6770	82 and 97	
	14,040	243.8	6120	107	Climb with maximum continuous power
	16,580	237.2	7510	80	
	14,020	243.8	6220	110	Descent at 1000 fpm
	16,580	237.2	6800	83	
Static Lateral-Directional Stability	14,490	243.8	6250	80 and 111	Level
	16,840	237.2	6980	82 and 109	
	16,580	237.0	6780	102	Climb with maximum continuous power
	16,580	237.0	6940	102	Descent at 1000 fpm
Maneuvering Stability	14,410	243.8	6240	80 and 111	Left and right turns
	17,060	237.2	6390	80	
Dynamic Stability	16,700	237.5	7400	75 and 102	
Controllability	13,970	244.2	4500	Zero	Hover
	17,040	237.4	7190	80 and 107	Level
Mission Maneuvers	14,000	243.7	3000	N/A	Pinnacle operation, confined area operation, terrain flight, accelerations and decelerations
Low Speed Flight Characteristics	13,860	232.3	5040 and 11,120	Zero to 40 KTAS[3]	Wheel height: 15 ft
Simulated Engine Failures	15,500	232.3	6500	75 to 111	Level
Simulated Stability and Control Augmentation System Failures	15,600	232.3	6600	75 to 111	Level and maneuvering flight

NOTES:

[1]Test conditions at an approximate mid lateral center of gravity location, with the stability and control

RESULTS AND DISCUSSION

GENERAL

6. The Airworthiness and Flight Characteristics test of the BHTI 214ST helicopter was conducted by the USAAEFA. The test was conducted at Edwards Air Force Base, (elevation 2302 feet), Bakersfield (elevation 488 feet), Bishop (elevation 4120 feet) and Coyote Flats (elevation 9980 feet), California. Hover, level flight and autorotational descent performance tests were conducted. Where possible, the test results were compared to the Federal Aviation Administration (FAA) Approved Rotorcraft Flight Manual. The airspeed for best glide distance determined from the autorotational descent performance test results of this evaluation was significantly less than the best glide distance airspeed presented in the flight manual. The out-of-ground effect hover test results when compared to the flight manual indicated less hover capability at standard day conditions and altitudes below 8060 feet but better hover capability at higher altitudes. The maximum difference between the flight manual and the test results of this evaluation was 250 pounds, below 8060 feet. Handling qualities tests included static and dynamic stability, maneuvering stability, power management, systems failures, mission maneuvers and flight in simulated Instrument Meteorological Conditions (IMC). Two major conclusions reached were: the 214ST is an excellent IMC aircraft and its cockpit design signficantly reduces pilot workload in most modes of flight. Several enhancing characteristics were noted with the most significant being: the low level of cockpit vibrations; the installation of an air data computer; the incorporation of a rotor brake and the low noise levels in the cockpit. The one unsatisfactory characteristic identified was the position of the collective during the first engine start which could result in a hazardous operating condition. Sixteen undesirable characteristics were identified of which the most significant were: the limited field of view during decelerations; increased pilot workload at various wind conditions during low speed flight; inaccuracies of the radio altimeter due to masking by relatively small external loads. Although this aircraft was not designed to meet military specification, compliance with MIL-H-8501A (ref 10, app A) was noted where data were applicable. Some flight control characteristics did not comply with MIL-H-8501A, however, none of the noncompliances were considered to be significant. Several recommendations are made for the incorporation (in military aircraft) of various subsystems which were installed in the 214ST helicopter.

PERFORMANCE

Hover Performance

7. Hover tests were conducted utilizing the tethered hover method at the conditions of table 1. The 5-foot main wheel height in-ground effect (IGE) and 100-foot main wheel height out-of-ground effect (OGE) tests were conducted at test sites with field elevations of 4120 and 9980 feet. Data were also obtained using a free flight hover technique. A production loadmeter (strain gauged cargo hook) was calibrated and used to measure the tension in the tether cable during the hover tests. Variations in thrust coefficient (C_T) were attained by varying tension in the cable. Power available was provided by BHTI. The power available in hover includes a 3 percent reduction in maximum engine power that is used by the FAA in approval of rotorcraft manuals. Takeoff power is limited to 2350 shaft horsepower by the main transmission up to 8760 feet, standard day conditions.

8. A summary of the OGE hover test results is shown in figure 1, appendix E. Nondimensional test results are presented in figures 2 and 3. The standard day, sea level hovering capability of the 214ST helicopter was determined to be 17,360 pounds (lb) using takeoff power. This is a difference of 140 lb from that presented in the 214ST Flight Manual (17,500 lb). Further comparison between the USAAEFA test results and the Flight Manual shows varying differences in hovering capability from sea level to 14,000 feet. Above 8060 feet the USAAEFA test results indicate an increase in hovering capability over that presented in the Flight Manual by as much as 450 lb. Below this altitude, a maximum of 250 lb of reduced hovering capability was determined.

9. The graphical presentation of hover capability in the Flight Manual (pages 4A-2 through 4A-20) could not be read precisely. The inaccuracy of this presentation can result in predictions that are optimistic with regard to the actual hovering capability of the helicopter by as much as 200 lb. The Flight Manual should be changed to allow more precise prediction of hovering capability.

10. The ratio of gross weight to density ratio (W/σ) was limited to 21,800 lb on the 214ST by BHTI. The IGE hover capability at standard day conditions exceeds the W/σ limit in that it can hover up to 7320 feet at 17,500 lb gross weight and at reduced gross weight up to its maximum approved density altitude of 14,000 feet with less than takeoff power. These results confirmed that the aircraft could hover at the limit shown in the Flight Manual.

Level Flight Performance

11. Level flight performance tests were conducted to determine power required and fuel flow for a range of airspeeds, altitudes and gross weights within the operational envelope of a 214ST. Tests were conducted to evaluate one aircraft configuration, described in appendix B, at a forward longitudinal center of gravity (cg) location and constant main rotor speed (100%) operation. Four flights were flown in coordinated (ball-centered) flight. A constant ratio of W/σ was maintained during each level flight performance test. No corrections were made for the drag of external instrumentation, fuselage mounted passenger steps, or associated fairings that were not installed on the test aircraft. These steps are normally included on the 214ST.

12. Nondimensional test results are presented in figures 4 and 5, appendix E. These test results were used to determine the power required for figures 6 through 9, the dimensional test results. Two transport helicopter missions were selected for the 214ST to summarize the level flight performance. The first condition selected was 6000 feet, standard day conditions, 100% main rotor speed and a gross weight of 17,500 lb (maximum takeoff gross weight). The other was a 12,800 lb gross weight (representative ferry flight) at the same flight conditions. Long range cruise speeds were determined to be 126 (limited to 122 knots by never-exceed velocity (V_{NE})) and 145 knots true airspeed, respectively. Power required and specific range data for these conditions are presented in figure 10, appendix E. At the conditions selected for the ferry flight consideration and allowing for a 20 minute fuel reserve; 250 pounds of fuel for starting, run-up, takeoff and climb; no wind, the range of the 214ST was 437 nautical miles. This range was determined using the standard ships fuel capacity of 435 gallons usable fuel and exceeds the ferry range of any current US Army helicopter similarly configured (without auxiliary fuel tanks).

Autorotational Descent Performance

13. Autorotational descent performance tests were conducted near 16,000 lb gross weight and at the conditions listed in table 1. The tests were conducted by retarding the power control throttles to the idle position and then stabilizing on airspeed and rotor speed. At the normal operating rotor speed of 100 percent (287 rpm), airspeed was varied to determine the airspeed for minimum rate of descent ($V_{min\ R/D}$). At the approximate airspeed for minimum rate of descent, rotor speed was varied to determine

the effect on descent performance. Test results are presented in figures 11 and 12, appendix E.

14. The minimum rate of descent at 16,000 pounds gross weight was 1900 ft/min at an airspeed of 79 knots calibrated airspeed (KCAS). A calibrated airspeed of 79 knots is equivalent to 75 knots indicated airspeed (KIAS). The airspeed for maximum glide distance was 91 KCAS (87 KIAS) and the rate of descent was 2060 ft/min. This airspeed differs with the 109 KCAS (105 KIAS) by 18 knots recommended in the emergency section of the Flight Manual (page 3-16) for dual engine failure. The 214ST is also limited to 104 KCAS (100 KIAS) in autorotation under the airspeed limitations section page 1-3 of the same manual. BHTI should resolve the conflict of maximum glide distance (autorotation) airspeed in the Flight Manual with USAAEFA's test results.

HANDLING QUALITIES

General

15. Handling qualities of the BHTI 214ST helicopter were evaluated at the various longitudinal cg and gross weight configurations listed in table 2. All tests were flown at a mid lateral cg location. The 214ST was found to be an excellent IMC aircraft with many aircraft subsystems reducing pilot workload significantly. One unsatisfactory characteristic, the position of the collective during the first engine start which could result in a hazardous operating condition, was found. Sixteen undesirable characteristics were found. The most significant undesirable characteristics were the limited field of view during decelerations and the increased pilot workload in low speed flight at various wind speeds.

Flight Control System Mechanical Characteristics

16. The flight control system mechanical characteristics were evaluated on the ground with external hydraulic and electrical power applied to the aircraft, engines and rotors static, force trim ON, and the stability and control augmentation system (SCAS) engaged. The results of the flight control system tests, obtained by on-board instrumentation and a calibrated hand-held force gauge, are presented in figures 13 through 17, appendix E. The mechanical characteristics are summarized in tables 3 through 6. These tables include compliance of the mechanical characteristics with Mil-H-8501A. Control forces were measured at the center of the pilot's cyclic and collective grips and pedals. The flight control system characteristics were also qualitatively evaluated

Table 3. Longitudinal Control System Characteristics

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Control Centering	Absolute	Para 3.2.3 Positive, self-centering	Yes
Stick Jump[2] (in.)	0.1 to 0.2 Fwd and Aft	Para 3.2.3 Stick jump undesirable	No
Force Gradient for First Inch of Travel from Trim (lb/in.)	Slope greater than or equal	Para 3.2.4 Slope for first inch of travel from trim must be greater than or equal to the slope for the remaining stick travel	Yes
	3.0 (aft) 2.5 (fwd)	Para 3.2.4 Longitudinal force gradient for the first inch of travel from trim must be greater than 0.5 o and no more than 2.0	No
Breakout Force Including Friction (lb)	2.0 (aft) 2.0 (fwd)	Para 3.2.7 .5 minimum, 1.5 maximum	No
Force of One Inch of Travel from Trim (lb)	3.0 (aft) 2.5 (fwd)	Para 3.2.4 Force produced by one inch of travel from trim shall not be less than breakout force plus friction	Yes
Force vs. Displacement Gradient (lb/in.)	2.5 (aft) 1.4 (fwd) Positive	Para 3.2.4 Slope of curve shall be positive at all times and display no undesirable discontinuities	Yes
	No undesirable Discontinuities		Yes
Limit Control Force[3] (lb)	16 (aft) 11.5 (fwd)	Para 3.2.6 8.0	No No
Control Forces Trimmable to Zero	Yes	Para 3.2.3 Required	Yes
Control System Freeplay[4] (in.)	0.1	Para 3.5.10 No greater than 0.2	Yes
Total Control Travel (in.)	11.1	-------	----
Control Dynamics[5]	2 overshoots (aft) 3 overshoots (fwd)	-------	----
Trim Authority (in.)	9.7	-------	----

NOTES:

[1]Military Specification, MIL-H-8501A, 7 September 1961.
[2]Defined as the amount of control movement after trim activation following a 20 percent cyclic displacement.
[3]Control trimmed to 50 percent position.
[4]Defined as the amount of control travel without corresponding blade travel.
[5]Defined as the number of oscillations of the control about trim following a 10 percent

Table 4. Lateral Control System Characteristics

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Control Centering	Absolute	Para 3.3.10 Positive, self-centering	Yes
Stick Jump[2] (in.)	0.1 to 0.2 (fwd and aft)	Para 3.3.10 Stick jump undesirable	No
Force Gradient for First Inch of Travel from Trim (lb/in.)	Slope greater than or equal	Para 3.3.11 Slope for first inch of travel from trim must be greater than or equal to the slope for the remaining stick travel	Yes
	1.1 (lt) 1.2 (rt)	Para 3.3.11 Force gradient for the first inch of travel from trim must be greater than 0.5 and no more than 2.0	Yes
Breakout Force Including Friction (lb)	2.5 (lt) 2.5 (rt)	Para 3.3.12 .5 minimum, 1.5 maximum	No
Force of One Inch of Travel from Trim (lb)	3.6 (lt) 3.8 (rt)	Para 3.3.11 Force produced by one inch of travel from trim shall not be less than breakout force plus friction	Yes
Force vs. Displacement Gradient (lb/in.)	1.1 (lt & rt) Positive	Para 3.3.11 Slope of curve shall be positive at all times and display no undesirable discontinuities	Yes
	No undesirable Discontinuities		Yes
Limit Control Forces[3] (lb)	9.0 (lt) 8.5 (rt)	Table II 7.0	No No
Control Forces Trimmable to Zero	Yes	Para 3.3.10 Required	Yes
Control System Freeplay[4] (in.)	0.1	Para 3.5.10 No greater than 0.2	Yes
Total Control Travel (in.)	10.7	----	----
Control Dynamics[5]	4 overshoot (lt) 4 overshoot (rt)	----	----
Trim Rate (in./sec)	0.5 (lt) 0.4 (rt)	----	----
Trim Authority (in.)	10.0	----	----

NOTES:

[1]Military Specification, MIL-H-8501A, 7 September 1961.
[2]Defined as the amount of control movement after trim activation following a 20 percent cyclic displacement.
[3]Control trimmed to 50 percent position.
[4]Defined as the amount of control travel without corresponding blade travel.
[5]Defined as the number of oscillations of the control about trim following a 10 percent displacement.

Table 5. Directional Control System Characteristics

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Control Centering	Positive	Para 3.3.10 Positive self-centering	Yes
Stick Jump[2] (in.)	1.0	Para 3.3.10 Stick Jump Undesirable	No
Breakout Force Plus Friction (lb)	3.0 (lt) 3.5 (rt)	Para 3.3.13 3.0 minimum 7.0 maximum	Yes
Limit Control Forces[3] (lb)	27.0 (lt) 28.0 (rt)	Para 3.3.11 15.0 maximum	No No
Control Forces Trimmable to Zero	Yes	Para 3.3.10 Required	Yes
Control System Freeplay[4] (in.)	0.2	Para 3.5.10 No greater than 0.2	Yes
Total Control Travel (in.)	5.8	----	----
Control Dynamics[5]	3 overshoots	----	----

NOTES:

[1]Military Specification, MIL-H-8501A, 7 September 1961.
[2]Defined as the amount of control movement after trim following a 20 percent pedal displacement.
[3]Control trimmed to 50 percent position.
[4]Defined as the amount of control travel without corresponding blade travel.
[5]Defined as the number of oscillations of the control about trim following a 10 percent displacement.

Table 6. Collective Control System Characteristics

Test Parameter	Test Results	Specification Requirement[1]	Specification Compliance
Sustained Force for First inch of Travel from Trim (lb/in.)	6.0 (up) 11.0 (down)	------	----
Breakout Force Plus Friction (lb)	2.4 (up) 2.9 (down)	Para 3.4.2 1.0 minimum, 3.0 maximum	Yes Yes
Limit Control Force[2] (lb)	11.0	Para 3.4.2 7.0 maximum	No
Control System Freeplay[3] (in.)	0.1	Para 3.5.10 No greater than 0.2	Yes
Total Control Travel (in.)	10.9	----	----

NOTES:

[1]Military Specification, MIL-H-8501A, 7 September 1961.
[2]Maximum control force over entire throw of the collective.
[3]Defined as the amount of control travel without corresponding blade travel.

in flight with SCAS engaged, SCAS and AARS engaged and in several degraded modes. The results of the flight evaluation qualitatively agreed with the ground results.

17. A summary of cyclic control travel limits is presented in figure 13, appendix E. Cyclic control travel is limited by mechanical stops located at the base of both cyclic control sticks. The cyclic control envelope was not a function of either collective or directional control position. The cyclic control forces could be trimmed to zero within 91% of the longitudinal and 85% of the lateral control travel using the electrical trim system.

18. The longitudinal cyclic control was trimmed to the 50 percent position to obtain the longitudinal control system characteristics. The longitudinal control did exhibit some "stick jump", but it was not sufficient to degrade the flying qualities. The longitudinal cyclic control system did not meet the requirements of paragraphs 3.2.3, 3.2.4, 3.2.6 and 3.2.7 of MIL-H-8501A (table 3). However, none of these noncompliances adversely affected the flying qualities of the 214ST. Therefore, the longitudinal cyclic control system characteristics are satisfactory.

19. The lateral cyclic control was trimmed to the 50 percent position for most of the lateral control system characteristics evaluation. The lateral flight control exhibited some "stick jump", but it was not significant. The lateral cyclic control system did not meet the requirements of paragraphs 3.3.10, 3.3.11 and table II of MIL-H-8501A (table 4). However, these noncompliances did not adversely affect the flying qualities of the 214ST. Therefore, the lateral cyclic control system characteristics are satisfactory.

20. The directional control pedals were trimmed to the 50 percent position using on-board instrumentation. The directional control pedals exhibited "stick jump", however, neither taxi nor flight handling qualities were degraded. The requirements of paragraphs 3.3.10 and 3.3.11 of MIL-H-8501A (table 5) were not met. However, these noncompliances did not adversely affect the flying qualities of the 214ST. The directional control system characteristics are satisfactory.

21. The collective control system was evaluated with the collective friction OFF. The sustained pull and limit control force upward were essentially constant at 6.0 lb once the collective was moved more than 0.4 in. The sustained push control force downward varied between 8.5 and 11.0 lb. The down limit force was 11.0 lb and occurred at 6.5 in. from full down (approximately

mid-collective position). In flight with collective friction OFF, collective control forces were sufficient to hold the collective in any position indefinitely. However, the high forces required to initiate and sustain movement make small torque changes (1-3%) difficult, which resulted in torque overshoots and multiple collective adjustments. The requirements of paragraph 3.4.2 of MIL-H-8501A were not met in that the maximum effort for collective control movement (limit force) was exceeded by 4.0 lb (table 6). The high force required to initiate and sustain movement of the collective control (friction OFF) is an undesirable characteristic.

Control Positions In Trimmed Forward Flight

22. Trimmed flight control positions were evaluated in conjunction with the level flight and autorotative descent performance tests and during maximum continuous power climbs and descents (1000 fpm). Representative data are presented in figures 18 through 20, appendix E. Control position and attitude trends were essentially unchanged by cg location, gross weight or flight condition. The trim longitudinal control position for all flight conditions, greater than 45 KCAS, was conventional in that increased forward cyclic control position was required to trim at increased airspeeds. The cyclic control position variation with airspeed was essentially linear at airspeeds greater than 60 KCAS for all configurations and flight conditions. Maximum lateral control trim changes of approximately 1.7 in. in level flight from 30 KCAS to 140 KCAS were noticeable but not objectionable. Directional control trim changes were small (less than 1.5 in.) for all forward level flight conditions. Pitch attitude remained essentially constant, (near zero degrees) once the trimmed flight condition was attained. The constant pitch attitude exhibited by the 214ST is a desirable characteristic in that passenger comfort will be increased. Additionally, the pilot will not be "searching" for the correct attitude under IMC conditions. The trimmed flight control position characteristics are satisfactory.

Static Longitudinal Stability

23. The static longitudinal stability characteristics were evaluated at the conditions listed in table 2. The helicopter was stabilized in ball-centered flight at the desired trim airspeed and flight condition. The collective was held fixed while airspeed was varied approximately ± 20 KCAS about trim in 5 knot increments. Test results are shown in figures 21 through 24, appendix E. The static longitudinal stability, as indicated by the variation of longitudinal cyclic control position with

airspeed, was positive (forward longitudinal cyclic control position with increasing airspeed). Control force cues of longitudinal cyclic control displacement about trim were weak but sufficient for airspeed control within ±2 knots. The static longitudinal stability characteristics were essentially the same during climbs, descents, and autorotation in similar configurations. Desired cruise airspeed was maintained within ± 2 knots in light turbulence conditions with small (±0.1 in.), infrequent (every 5 to 7 seconds) longitudinal cyclic inputs (HQRS 3). The static longitudinal stability characteristics are satisfactory and met the requirements of MIL-H-8501A.

Static Lateral-Directional Stability

24. The static lateral-directional stability characteristics were evaluated at the conditions listed in table 2. The helicopter was stabilized in ball-centered flight at the desired trim airspeed and flight condition. The collective control stick was held fixed and sideslip angle was varied in approximately 5 degree increments (left and right) while maintaining constant airspeed. Test results are shown in figures 25 through 27, appendix E. Static directional stability, as indicated by the variation of directional control position with sideslip angle, was positive (left pedal for right sideslip angles) and essentially linear. Dihedral effect, as indicated by the variation of lateral cyclic control position with sideslip angle, was positive (right cyclic control for right sideslip angles) and essentially linear. The gradient of lateral cyclic control position with sideslip angle was approximately the same for all configurations tested. Sideforce cues in uncoordinated flight were sensed by the pilot as sideslip angles increased beyond approximately 3 degrees left or right at both trim airspeeds. The bank angle gradient with sideslip in level flight increased slightly with increased airspeed.

25. Longitudinal cyclic control trim changes during steady heading sideslips in level and descending flight were not objectionable and were characterized by a slight requirement for aft longitudinal cyclic control as sideslip was increased left or right. Left sideslip angles required slightly more aft longitudinal cyclic control than did right sideslip angles. In climbing flight, the longitudinal cyclic control position trend changed in that right sideslip required a slight amount of forward longitudinal cyclic control. The static lateral-directional stability characteristics were essentially the same during climb, descent, and autorotation except as stated above. The pilot workload required to maintain coordinated flight was low (HQRS 2) when SCAS was engaged. During cruise flight, the aircraft trim condition was maintained within

±2 degrees of heading and bank angle, and ±1/2 ball width with little pilot compensation (HQRS 2). The static lateral-directional stability characteristics in all flight conditions are satisfactory and met the requirements of MIL-H-8501A.

Maneuvering Stability

26. Maneuvering stability was evaluated for two aircraft configurations, lightweight/aft cg and heavyweight/aft cg at the conditions presented in table 2. The maneuvering stability tests were accomplished by initially stabilizing the helicopter in ball-centered level flight at the trim airspeed and then incrementally increasing the normal acceleration (g) by increasing the bank angle in left and right turns. Constant collective control position was maintained in the maneuvers and the pilot attempted to maintain a constant airspeed. Test results are presented in figures 28, 29 and 30, appendix E. At the density altitude evaluated, the normal acceleration was limited to 1.4 g in the lightweight configuration and 1.3 g in the heavyweight configuration by BHTI. For the present military applications these normal acceleration limits are low, and consideration should be given to expanding the envelope if the aircraft is considered for military application.

27. The stick-fixed maneuvering stability, as indicated by the variation of longitudinal cyclic control position with g, at the lower airspeed (80 KCAS) for both lightweight and heavyweight configurations was positive (increasing aft cyclic control with increasing g). There were no significant differences in the handling qualities characteristics between right and left turns. The variation in longitudinal control positions with g was linear and the lateral cyclic control position remained essentially constant at all bank angles. The gradients of longitudinal control position versus normal acceleration was 4.0 in./g and 1.5 in./g in the heavyweight and lightweight, respectively. The longitudinal control force cues were adequate at bank angles greater than 15 degrees. The maneuvering stability characteristics at 80 KCAS are satisfactory.

28. Maneuvering stability at 111 KCAS was only evaluated in the lightweight configuration due to the airspeed limit at maximum gross weight. The stick-fixed maneuvering stability was perceived as neutral (no change in longitudinal control position with increasing g) by the pilot (gradient approximately 0.6 in./g). The direction of turn did not significantly change the characteristics of the longitudinal cyclic control. However, the lateral cyclic control in left turns indicates a control reversal at 1.1 g. The lateral cyclic control reversal was very apparent in

flight and increased pilot workload. The pilot workload while maintaining constant bank angle (bank angles greater than 20 degrees) and constant airspeed was significantly greater at 111 KCAS than at 80 KCAS. The maneuvering stability characteristics at load factors greater than 1.3 were such that airspeed could only maintained within ±5 knots and bank angle within ±4 degrees. The poor longitudinal position and force cues for maneuvering stability at 111 KCAS is an undesirable characteristic.

Dynamic Stability (Gust Response)

29. The dynamic stability characteristics were evaluated at the conditions listed in table 2. The helicopter response was investigated in all control axes in both forward flight and at a hover using one-inch 0.5-second control pulses. Steady heading sideslip releases in level, climbing and descending flight were also evaluated. Representative time histories are shown in figures 31 and 32, appendix E. Pedal releases at a 10 degree sideslip angle (left and right) resulted in a rapid return to within 1/2 ball width of trim and then slowly (after 10 to 15 seconds) returned to trim (1 degree right sideslip angle) with one overshoot. Additionally, the gust response was evaluated qualitatively in calm to light turbulence conditions as defined in the FAA Flight Information Handbook (ref 11, app A). With SCAS ON and AARS OFF, the aircraft was flown "hands off" for extended time periods (greater than 1 minute) in light turbulence. Only small transient airspeed and altitude fluctuations were noted. The dynamic stability characteristics remained essentially the same for all flight conditions (level, climbing and descending flight). The dynamic stability characteristics are satisfactory and met the requirements of MIL-H-8501A.

Controllability

30. Controllability tests were conducted in forward flight at 80 and 107 KCAS and during hover to evaluate the control power, response, and sensitivity characteristics. Controllability was measured in terms of aircraft attitude displacement (control power), maximum angular velocities (control response), and maximum angular accelerations (control sensitivity) about an aircraft axis following a control step input of a measured size. Following the input, all controls were held fixed until a maximum rate was established or until recovery was necessary. The magnitude of inputs was varied by using an adjustable rigid control fixture on the cyclic control and by the co-pilot physically blocking the range of movement of the directional pedals. Real time telemetry monitoring was utilized to confirm the desired input size

and shape. Controllability tests were conducted at the conditions listed in table 2.

31. Longitudinal controllability characteristics are presented in figures 33 through 37, appendix E. Neither longitudinal control power (pitch attitude change after one second following a one inch input) nor longitudinal control response (maximum pitch rate per inch of control input) varied with the direction of input and only slightly with airspeed. The slight roll coupling noted during longitudinal controllability testing did not significantly increase pilot workload during mission maneuvers. The longitudinal controllability characteristics are satisfactory and met the requirements of MIL-H-8501A.

32. Lateral controllability characteristics are presented in figures 38 through 42, appendix E. The lateral control power, response, and sensitivity did not change with the direction of input. However, both control response and sensitivity varied slightly with airspeed. Very slight pitch coupling was noted but did not increase pilot workload during the mission maneuver evaluation. The lateral controllability characteristics are satisfactory and met the requirements of MIL-H-8501A.

33. Directional controllability characteristics are presented in figures 43 through 47, appendix E. The directional control characteristics at 80 and 107 KCAS were essentially the same. The hover control power and response were approximately 50 percent greater than measured in forward flight. The high control power and response may have contributed to the "lack of hover stability" discussed in paragraphs 60 and 62. These characteristics did not change with direction for inputs less than 0.75 in. However, for right inputs greater than 0.75 in. the maximum yaw rates were not reached prior to initiating recovery, therefore, neither control response or sensitivity could be quantitatively evaluated. The directional controllability characteristics are satisfactory and met the requirements of MIL-H-8501A.

Ground Handling Characteristics

34. The ground handling characteristics were evaluated concurrently with other tests. The following areas were evaluated; aircraft repositioning by ground personnel, preflight procedures, engine starting, runup procedures/systems checks, ground taxiing, engine/rotor shutdown, aircraft servicing, and rotor blade tie-downs.

35. The wheeled landing gear of the 214ST made aircraft repositioning extremely easy utilizing a standard aircraft tow tractor

and towbar. The towbar was easily attached to the aircraft nosegear by one man in less than 30 seconds. The average time required to reposition the 214ST from inside the hangar to the starting pad (approximately 50 yards) was less than 2 minutes. The aircraft repositioning characteristics are satisfactory.

36. The preflight procedures, as prescribed in the Flight Manual were relatively easy to follow but could not be completed without ground support equipment. The average preflight time for one person was approximately 15 minutes, which included a walk-around inspection of the outside of the helicopter and climbing on top of the aircraft to inspect the main rotor system and main transmission area. The most difficult area to preflight was the tail rotor assembly. The preflight checklist requires that the pilot insure that the 90 degree gearbox filler cap is secured. This check required the assistance of a maintenance work stand or step ladder because the tail rotor is located 11 feet above the ground and no integral access means is provided. The requirement to utilize ground support equipment to properly preflight the tail rotor assembly is undesirable.

37. During night preflights, the pilot's inspection was enhanced by the installation of illuminated sight gages. All engines, transmissions and hydraulic reservoirs were externally illuminated by lights powered by the aircraft's batteries. The illumination of aircraft's fluid reservoir sight gauges should be incorporated in all military aircraft.

38. The 214ST was equipped with two features which allowed the pilots (from the pilot's station) to insure all aircraft doors (including aft baggage compartment doors) were closed and locked. All doors had microswitches which illuminated a cockpit advisory light whenever any door was opened. Additionally, all doors (except the baggage compartment doors) were equipped with a mechanical device which extended a red indicator pin when the doors were unlocked. As a result of these two devices, it would have been very difficult for the pilot to be unaware of any door status. A door status indicator should be incorporated in all military aircraft.

39. With the rotor static, the weight of the main rotor blades force the collective control stick to increase to approximately the 50% position. During the starting sequence of the first engine, the pilot must lower the collective to the full down position as soon as the rotors are turning and hydraulic pressure is adequate to power the collective servo. Failure to lower the collective to the full down position will result in the aircraft becoming light on the landing gear and possibly becoming airborne

as the pilot increases the throttle of the operating engine to achieve 95% main rotor speed. During the first engine start the pilot is required to monitor several engine parameters, time, starter configuration and control positions. Several times during the evaluation the pilot inadvertently failed to lower the collective. On one occasion the aircraft became airborne at less than 90% rotor speed with ground personnel near the aircraft. Pilot action to resolve this hazardous condition was to lower the collective control abruptly. This resulted in sudden contact with the ground, subsequent flexing of the airframe which broke both overhead windows. The position of the collective during the first engine start which could result in a hazardous operating condition is unsatisfactory.

40. The procedures for systems checks, as prescribed in the Flight Manual were long and laborious. Total time required to accomplish the step-by-step procedures using the call and response method of cockpit communication was in excess of 10 minutes. The procedure requiring the longest period of time (approximately 1.5 minutes each) was the fly-by-wire elevator system checks which were required prior to all flights. Additionally, there was no "Thru Flight Checklist" which would have allowed the pilots to check systems during the first runup and then only the critical systems for following flights. Consideration should be given to the incorporation of a "Thru Flight Checklist". The excessive time required to perform the aircraft runup procedure is an undesirable characteristic.

41. The ground taxiing characteristics of the 214ST were evaluated throughout the A&FC. The aircraft was very maneuverable on the ground. However, the high sensitivity of the directional pedals during ground taxi operations initially (during pilot training) resulted in overcontrol. This may be due to the light breakout plus friction force presented in table 5. Some difficulty was experienced judging the turning radius of the aircraft when trying to park the aircraft precisely. With some practice, satisfactory performance was achieved. Intermittent vibrations were encountered during taxi and are discussed in paragraph 68. Except for the intermittent vibrations, the ground taxi characteristics are satisfactory.

42. The aircraft shutdown procedures were simple, and relatively easy to accomplish. The most time consuming procedure was the engine cool-down period of 2 minutes. Maintaining 75% main rotor speed during the cool-down period was a nuisance and difficult to accomplish in that throttle position and main rotor speed appear to have poor correlation (except when full OFF or full ON). This required several (3-4) small throttle adjustments to

set and maintain the proper speed during the cool-down period. The poor throttle and main rotor speed correlation during manual throttle operation is discussed in greater detail in paragraph 54. The aircraft shutdown procedures are satisfactory.

43. The rotor stopping procedure was simple and effective. The main rotor hydraulic brake (similar to the optional UH-1 installation) was applied as main rotor speed decreased below 40%. The rotor was slowed and stopped at varying rates, depending on the pressure applied to the rotor brake handle, with stop times of 10 to 15 seconds. The incorporation of a rotor brake is an enhancing characteristic.

44. The location of the rotor brake handle in the cockpit made operation awkward but acceptable. The rotor brake handle may be placed in the full ON position by pushing the handle past an over-center cam to the full forward (locked) position. If the handle is bumped aft (unrestrained), it swings aft at a very high rate. Due to the close proximity of the pilot's head to the rotor brake handle, there is a possibility of injury. The operational design of the rotor brake handle is an undesirable characteristic.

45. The rotor blade tie-down devices consist of a sock-type device which is designed to be placed over one blade and a wide cloth strap-type device which is looped over the opposite blade to pull the blade downward. Both tie-down devices are awkward to use and difficult to install. A maintenance work stand was required for installation due to the height of the rotor blades. The poor design of the rotor blade tie-down devices is an undesirable characteristic.

Low Speed Flight Characteristics

46. The low speed flight characteristics of the 214ST were evaluated at the conditions in table 2. Testing was performed at speeds up to 35 KTAS in sideward, 30 KTAS in rearward, and 40 KTAS in forward flight utilizing a ground pace vehicle as a speed reference. It should be noted that the right sidewards flight limit was inadvertantly exceeded by 5 knots during the low altitude portion of the evaluation, however, no anomalies were seen. The helicopter was flown IGE at a wheel height of approximately 15 feet. The low speed flight test data are presented in figures 48 through 54, appendix E.

47. The flight control trends in forward and rearward flight (fig. 48, app E) were essentially unchanged by density altitude. There were no unusual trends or significant discontinuities in any flight controls. Between 10 and 20 KTAS rearward, the pitch

attitude changed from 0 to 4 degrees nose down and required approximately 1.2 in. more aft longitudinal cyclic. Additionally, the control excursions increased significantly in all axes. The large pitch attitude change and additional control excursions increased the pilot workload from a nominal HQRS 3 to HQRS 5 during precise hover tasks. The increased pilot workload during hover tasks in tailwinds greater than 10 KTAS is an undesirable characteristic.

48. The flight control trends during left and right sideward flight were essentially unchanged by density altitude (fig 49, app E). The requirement for right directional flight control increased significantly between 20 and 30 KTAS in left sideward flight. However, a minimum of 17 percent directional control remained at the worst case evaluated. Pitch attitude varied approximately 2 deg. Roll attitude varied linearly from a 2 deg right wing low (40 KTAS right) to 3 degs left wing low (35 KTAS left). During left sideward flight while attempting to maintain a steady heading at 20 KTAS (density altitude 5040 ft) and between 10 and 20 KTAS (density altitude 11,080 ft) the pilot workload increased significantly from a nominal HQRS 3 to HQRS 5 due to aircraft pitch, roll and yaw oscillations. Figures 50 and 51 depict the oscillations experienced at both altitudes. The characteristics of the oscillations were similar at both density altitudes except that the magnitude of the rate and flight control excursions were larger at the higher density altitude. The aircraft oscillations and the resultant flight control excursions will increase pilot workload during any low speed maneuver which would require precise position control. The high pilot workload during low speed flight in left crosswinds (10 to 20 KTAS) at high density altitudes is an undesirable characteristic.

49. A critical azimuth evaluation was performed at a constant 25 KTAS at both test altitudes. There were no unusual flight control trends or discontinuities, figure 52, appendix E. The minimum directional control margin occurred at 60 deg azimuth and minimum longitudinal control margin at 190 deg azimuth. At azimuths between 40 and 70 deg (right quartering headwind), aircraft oscillations similar to those described in the previous paragraph occurred. As with the previously discussed oscillations, relatively large flight control excursions (±0.4 in.) were required to counter these oscillations (figs. 53 and 54). For the same reasons stated in the previous two paragraphs, the pilot's workload will significantly increase during tasks requiring precise hovering in right quartering head winds. The high pilot workload during low speed flight in right quartering head winds is an undesirable characteristic.

50. During right sideward low speed flight it was noted that main rotor flapping increased to 90% (100% = 10 deg hub flapping) at airspeeds above 30 KTAS. Additionally, the critcal azimuth evaluation revealed that maximum main rotor flapping occurred between 90 and 180 deg wind azimuth (fig. 52, app E). At 100% main rotor flapping the hub will strike a "frangible stop" which allows an additional 10% (1 deg) flapping prior to mast contact. The pilot cannot always control direction of approach or hover direction, therefore, the aircraft must be capable of operating in all wind azimuths. Consideration should be given to evaluating main rotor flapping during all low speed maneuvers at wind azimuths between 90 and 180 degrees with wind speeds greater than 25 KTAS.

Power Management

Engine Torque Matching:

51. Engine torque matching characteristics were qualitatively assessed throughout the A&FC tests. The maximum torque difference noted throughout static and dynamic testing was approximately 3%, which only occurred at low power settings. At stabilized power settings greater than 70%, the maximum torque difference was one percent. The torque matching of the General Electric CT7-2A engines is satisfactory.

Rotor Speed Droop Characteristics:

52. Rotor speed droop characteristics were evaluated in all modes of flight and at several gross weight configurations. Some rotor overspeeding and underspeeding (3 to 5%) was noted (fig. 55, app E), but was not considered to be significant for the transport helicopter mission. The rotor speed droop characteristics are satisfactory.

Torque/Measured Gas Temperature/Gas Producer Indicators:

53. The torque, Measured Gas Temperature (MGT) and Gas Producer (Ng) indicators are equipped with inner and outer scales which are used to monitor the engines when operating in single or dual engine configuration. The torque indicator's outer scale is used for dual engine operation and the inner scale for single engine operation. The MGT and Ng indicators are configured so that the outer scale is used for single engine operation and the inner scale for dual engine operation. The reversing of configuration on primary cockpit engine indicators will result in confusion during high pilot workload tasks. The reversal of torque/MGT/Ng indicator marking logic is undesirable.

Electrical Control Unit Lockout Operation:

54. Main rotor speed (N_R) control during Electrical Control Unit (ECU) lockout operations was evaluated during vertical takeoff to a hover, normal takeoff, traffic patterns, normal approach to a hover, and vertical landing. To enter ECU lockout, the pilot depressed the Engine Stop Release switch and rotated the throttle full open, which activated the ECU lockout mode and placed the engine trim control in manual operation. The pilot then immediately reduced the throttle to control N_R within limits. The poor correlation between throttle position and N_R setting (discussed in para 42) and the high sensitivity of the throttles made N_R control (within normal operating limits of 99-100%) very difficult. The pilot's attention was so distracted by N_R control requirements that flying the aircraft with precision was very difficult. Additionally, in the ECU lockout mode the normal MGT and overspeed limiting devices are disabled and the pilot is responsible for MGT and overspeed control. Frequent (every 2-3 seconds), small (0.1 in.) throttle adjustments were required to maintain N_R within the normal operating range (HQRS 5). Collective control adjustments in flight increased the throttle adjustment requirements significantly. The high throttle sensitivity and the poor throttle correlation with main rotor speed during manual throttle operations are undesirable.

Instrument Flight Capability

55. Instrument flight characteristics were qualitatively evaluated during various phases of the A&FC evaluation. Enroute tasks and approaches were conducted in several configurations:

a. AARS engaged and coupled with the flight director, SCAS engaged and force trim ON.

b. ARRS engaged but not coupled to the flight director, SCAS engaged and force trim ON.

c. AARS disengaged, SCAS engaged and force trim ON.

d. AARS engaged but not coupled with the flight director, SCAS disengaged, force trim ON.

e. AARS and SCAS disengaged, force trim ON.

The pilot workload in all configurations was significantly lower in the 214ST than in other US Army transport helicopters (in similar configurations). In configurations a and b, all instrument tasks were assigned HQRS's of 1 or 2. Other than occasional pilot inputs in the directional and collective flight controls, the pilots were only required to monitor the flight controls.

56. The 214ST is designed such that the SCAS is used for short term rate damping and the AARS is used for long term attitude control. Both systems are independent of each other (paras 30 through 38, app B). The 214ST is unique in that the AARS may be engaged without the SCAS. With only the AARS engaged, the aircraft does exhibit degradation in short term attitude hold, however, it can be flown "hands off" for extended time periods. The aircraft's degraded short term response can best be described as a "dutch roll" motion. In calm air, attitudes were held within one degree and in light turbulence within three degrees.

57. The most degraded configuration flown during the IMC evaluation was with AARS and SCAS both disengaged and force trim ON. The highest pilot workload (HQRS 4 to 5) was noted during an Instrument Landing System (ILS) approach as the aircraft approached minimums. The flight director significantly reduced the pilot workload in that all that was required was to "line up" three reference indicators to be on course and on glideslope. Only one approach was flown in this configuration and the pilot workload would be reduced significantly had there been time to fly additional practice approaches. The low pilot workload during IMC flight (both "full-up" and degraded modes) is an enhancing characteristic.

58. During the A&FC evaluation it became apparent that the 214ST was well equipped for extensive operation in the IMC environment. The 214ST had Area Navigation (RNAV), very high frequency omni range (VOR) and automatic direction finder (ADF) receivers. The RNAV could be preprogramed with 10 way points, the VOR and ADF with two frequencies each allowed the pilots to "set up" navigation points prior to take off. The flight director had several modes which are not normally seen in US Army helicopters, but significantly reduce pilot workload when utilized. The RNAV and the capability of coupling it to the flight director is an additional feature not seen in US Army helicopters. With this capability, the 214ST was capable of flying direct to destination rather than "zig-zagging" VOR to VOR. The integral fuel capacity (approximately four hours, without auxiliary tanks, in a ferry configuration) enabled the 214ST to fly longer straight line legs than any current US Army helicopter (without auxiliary fuel tanks). These features coupled with those discussed in paragraphs 55, 56, and 57 gave the 214ST helicopter IMC capabilities that exceed those of any current US Army helicopter.

Mission Maneuver Characteristics

59. Several utility/cargo mission maneuvers were performed during the A&FC evaluation. These included confined area operations, pinnacle operations, external loads, simulated IMC flight, approaches, external loads and accelerations/decelerations at both light and heavy gross weights. The maneuvers are described in the Utility Helicopter, UH-60 Aircrew Training Manual (ref 12, app A).

60. Takeoff and landing characteristics were evaluated throughout the A&FC tests. Vertical takeoffs to a hover and forward flight, vertical landings from a hover, and rolling takeoffs and run-on landings were performed. The aircraft was basically stable during all takeoffs and landings. In a hover, no unusual flight characteristics were noted and relative position drift was less than two feet. Pilot workload was evaluated as moderate (HQRS 4) due to the requirement for frequent (every 3-4 seconds), small ($\pm$0.1 in.) pedal inputs to maintain aircraft heading within $\pm$2 deg. Run-on landings were performed with the longitudinal cyclic positioned as necessary to establish the appropriate deceleration and landing attitude. The run-on landing characteristics were undesirable in that on every attempt the aircraft bounced back into the air (1 to 2 feet) immediately following the initial touchdown (occurred at heavy and light gross weights, fwd and aft cg's). The poor touchdown characteristics during run-on landings are undesirable.

61. During approaches to unimproved areas (confined and pinnacle operations) and decelerations the forward field of view was very limited. The field of view was restricted due to a high pitch attitude, design of the cockpit glare shield, and the limited size of the "chin bubble". The pilots were forced to sideslip the aircraft to maintain an adequate view of the approach area. This compensation is a relatively minor problem when operating at low C_T's. However, when at high C_T's or during sling load operations, due to power margins and higher pilot workload, the pilot is unable to compensate by sideslipping. The limited forward field of view is not unlike those seen on other state of the art military helicopters. The limited forward field of view during decelerations is an undesirable characteristic.

62. Two external load evaluations were accomplished. The evaluation included a high density (approximately 3'x 3'x 3', lead alloy), 3100 lb load and a low density (7'x 6'3"x 6'8" hollow container), 2715 lb load. The high density load increased the aircraft gross weight to approximately 17,000 lb requiring a torque of 75% at approximately 1500 ft density altitude (load

10 feet above ground). Aircraft gross weight with the low density load was 16,250 lb requiring a torque of 98% at approximately 4500 ft density altitude. The pilot workload during the "hookup" phase was fairly high due to less than optimal hover stability (para 60). Vibration levels increased slightly (VRS 4) as the load was picked up (para 70). The radio altimeter was rendered useless by the low density load's horizontal surface (7'x 6'3") due to the masking of the radio altimeter signals. The absence of accurate radio altimeter indications during external load operations significantly reduces cues for hover heights. This problem did not occur with the high density load. During both evaluations the loads were flown to 80 KIAS and 30 degrees of bank angle with rapid rollouts. In both configurations the aircraft was very stable with very little feedback to the airframe. No feedback was noted in the flight controls even though the low density load was spinning and the load was flown in light turbulence. As airspeed was increased with the low density load, more forward longitudinal cyclic control was required than with a "clean" aircraft. The increased forward longitudinal cyclic control was very apparent and was an excellent cue for airspeed management. The inaccuracy of the radio altimeter indication due to masking by a relatively small external load is an undesirable characteristic.

63. The 214ST is equipped with a direct reading cockpit loadmeter. This device accurately displayed the actual load weight placed on the aircraft cargo hook. It allowed the pilot to make real time decisions concerning external load operations. All military cargo helicopters should incorporate a direct reading cockpit loadmeter.

AIRCRAFT SYSTEM FAILURES

Simulated Engine Failure

64. Simulated single engine failures were evaluated at 15,600 lb average gross weight at airspeeds between 75 and 110 KIAS (ship) in level flight. A representative time history is presented in figure 56, appendix E. The engine failures were simulated by rotating one throttle from the "full open" to the "idle" position and delaying for a minimum of 2 seconds prior to moving any flight control. There were no differences (handling qualities or failure cues) noted between a "failed" left engine or a "failed" right engine. The simulated engine failures were detected by "splits" in cockpit engine parameters and a noticeable 2 to 3 deg left yaw. Other than the yaw excursion no unusual attitude changes or control forces were observed during the simulated failures and

the subsequent transition to single engine flight. The simulated single engine failure characteristics are satisfactory.

Fly-By-Wire Elevator Failure

65. Several fly-by-wire (FBW) elevator (single system) failures were experienced during the A&FC evaluation. Most of these failures occurred during hover, sideslips in excess of 10 deg or at flight speed near V_{NE}. All failures were immediately detected due to the illumination of the AFCS caution light and the elevator annunciator lights. No excursions in attitude or flight control forces were noted. To re-engage the failed FBW elevator system, the effected elevator power switch was cycled OFF and back ON. All re-engagements were successful with no attitude or flight control transients. Though there were no dual system FBW elevator failures evaluated, it should be noted, that the V_{NE} (due to the FBW elevator failure) is not less than 75 KIAS for any combination of gross weight and density altitude. This V_{NE} is significantly higher than the V_{NE} for the UH-60A Black Hawk helicopter with similar failures. The FBW elevator failure (single system) characteristics are satisfactory.

Stability and Control Augmentation System (SCAS) Failures

66. SCAS failures (disengagements) with AARS OFF were evaluated at a hover and at 75 and 111 KCAS. The disengagements were performed in level flight, pitching flight from level flight, rolling flight from level flight, and a combination of pitching and rolling from level flight. All failures were simultaneous three axes failures which were induced by turning SCAS power off. The initial indications of the disengagement were the immediate illumination of the AFCS caution and SCAS annunciator lights. In level flight there were no immediate aircraft rate excursions, flight control feedback or force disharmony. In the other three modes of flight, the roll rate increased significantly approximately one-half second after the SCAS power was removed. Recovery was necessitated due to high roll acceleration. The high roll accelerations were similar to those seen in other military aircraft. A representative time history is presented in figure 57, appendix E. Three axis SCAS disengagement characteristics are satisfactory.

67. Continued flight with three axes of SCAS disengaged was evaluated during pilot training, mission maneuvers and IMC evaluations. The flight regimes ranged from hover to high speed forward flight, takeoffs and landings, wings level climbs and descents, and banked climbs and descents. The pilot workload was rated approximately 2 HQRS higher when maintaining aircraft attitudes, airspeeds,

and altitudes within the same tolerances for flight with SCAS engaged. Both pilots tended to induce aircraft oscillations in the lateral axis. At high C_T's or high power settings, the 214ST appeared to be slightly less stable than in other regimes of flight. During the mission maneuvers evaluation, it was determined that a three axis SCAS failure would not abort any day VMC missions except those requiring very precise aircraft control (landing in tight confined areas or pinnacles). The characteristics of continued flight with a three axes SCAS failure are satisfactory.

VIBRATION CHARACTERISTICS

68. Cockpit vibration characteristics were evauated qualitatively in all modes of ground operation and flight. During ground taxi, vibrations randomly increased from a VRS of 2 to VRS 4. Several attempts were made to determine the aircraft configuration (aircraft weight or center of gravity), taxi speed or particular flight control position which induced the increased vibration. The increased vibration was experienced in all configurations and appeared to be self-excited. There were no divergent tendencies.

69. The 214ST helicopter was equipped with a limited inflight main rotor tracking system (para 6, app B). The rotor tracker had a total travel of 0.2 degrees of blade pitch change (four flats on the pitch change links). Through the use of the inflight rotor tracker, the pilots were able to significantly reduce small 1/rev vertical vibration increases which occurred due to varying temperature/gross weight/airspeed. Both pilot and passenger comfort were increased due to the reduced vibration levels. The limited inflight main rotor tracking system is an enhancing characteristic.

70. The vibration levels in most modes of flight were given a VRS of 1 or 2. The vibrations were significantly lower than either pilot had experienced in other two bladed helicopters and lower than most 4 bladed helicopters. Two areas in which the vibrations increased were during hover operations and flights near V_{NE} at high C_T's. During both operations the VRS increased from level 2 to 4. Vibrations reverted to a VRS 2 once the aircraft transitioned into forward flight (from a hover) or airspeed was decreased from V_{NE}. The low level (VRS 1 to 2) of cockpit vibration is an enhancing characteristic.

HUMAN FACTORS

Cockpit and Cabin Evaluation

General:

71. The cockpit and cabin were qualitatively evaluated throughout the test program, including a night evaluation. The presence of test instrumentation and equipment was considered during the assessment. Cockpit arrangement, pilot ingress/egress, comfort, and readability of gauges and notations are satisfactory, except as discussed in the following paragraphs.

Day Lighting:

72. The light intensity of several cockpit displays, when exposed to direct sunlight, was insufficient. The following displays were affected:

All radio digital displays
Caution/warning/advisory panel
Both annunciator panels
Both V_{NE} airspeed limit panels
King KDI 572 Indicator (DME display)

In all cases the pilots, by shifting body position or utilizing their hands, were able to shade the affected display. This is undesirable since the possibility of pilots not observing an illuminated caution or warning segment is increased significantly. The insufficient day lighting intensity for various cockpit panels and radio displays is undesirable.

Radio Altimeter:

73. The 214ST was equipped with a Sperry AA335 radio altimeter (radar altimeter) which was mounted on the lower right side of the pilot's instrument panel. The Sperry AA335 radio altimeter was equipped with analog display, but was not equipped with a digital display. Because of the location of the radio altimeter (in relation to the pilot's eyes) it was very difficult for the pilot to read when it indicated less than 100 feet. The radio altimeter is most valuable to the pilot when operating in close proximity to the ground (external loads, approaches, etc) and of very little value when operating above 100 feet. The installation of a radio altimeter in the 214ST is desirable, however, its use is significantly reduced due to its location. Recommend that the Sperry AA335 radio altimeter be repositioned so that altitudes less than 100 feet may be more easily discerned by the pilot.

Annunciator Panels:

74. The 214ST is equipped with 13 different caution/warning/advisory panels, not counting those mounted on the copilot instrument panel. They include:

- Chip Indicator Panel
- Engine Fire Annunciators
- Baggage Fire Annunciators
- Caution/Warning/Advisory Panel
- V_{NE} Warning Light
- Flight Control System Panel
- Engine Starter-Engaged Annunciators
- Pilot's "Master" Annunciator Panel
- Flight Director Panel
- Three Miscellaneous Advisory Lights (right side of pilot's clock)
- AFCS Caution Light
- KA 35A Marker Beacon Indicators
- Hydraulic Actuator Bypass Panel

As a result of the large number of different panels, the instrument panel is unnecessarily crowded which forces the pilot to "search" for various panels. Several panels could be combined which would result in a more systematic cockpit presentation. Consideration should be given to combining various annunicator panels to reduce cockpit clutter.

Air Data Computer:

75. The 214ST is equipped with an air data computer (ADC) which gives the pilots a continuous updated readout of V_{NE}, density altitude and present gross weight (para 13, appendix B). Since the 214ST exhibits very low vibration cues at airspeeds near V_{NE}, the continously updated V_{NE} is essential. However, the ADC was used for other purposes during the conduct of the A&FC evaluation. Several airborne mission decisions were made based on the current density altitude and gross weight. These decisions were very timely and simple when compared to making the same decisions using manuals in military aircraft. The air data computer as installed in the 214ST is an enhancing characteristics. Consideration should be given to similar installations in current military helicopters.

Cockpit Seating:

76. During the course of the A&FC evaluation the pilots flew two ferry flights between Arlington, Texas and Edwards AFB,

California, each lasting approximately 8 flight hours. Some flight legs were in excess of three hours. On both flights, the cockpit seats were very comfortable. The pilots did not experience any "hot spots" nor did the seats appear to cause body stiffness as has been experienced in most military aircraft. It was noted that the seat restraint system lacked an anti-submarine belt (crotch belt). Consideration should be given to installing an anti-submarine belt on both pilot seats.

Cockpit Ventilation:

77. The cockpit ventilation system was evaluated during all phases of the A&FC evaluation. The system is equipped with two vent blowers (rather than one found in military aircraft) which signficantly increases air flow within the cockpit. In most military helicopters the vent blower is of little use, however, in the 214ST the vent blowers significantly increased the comfort level in the cockpit during hot weather operations. Consideration should be given to installing a cockpit ventilation system as effective as that installed in the 214ST in all military aircraft.

Night Evaluation

78. A night flight (in conjuction with a ferry flight) was conducted in order to evaluate the cockpit lighting. Most of the cockpit lighting was in accordance with BHTI's instrument lighting specification (ref 13, app A). All cockpit lighting (except radios) was white rather than the red or blue lighting found in most military aircraft. The white lighting allowed the pilots to clearly read all gauges at low light intensities. All white lighting was balanced so that all gauges and background lighting were perceived by the pilots to be of the same intensity. The level of pilot adjustable light intensity was excellent, from full bright down to no perceivable display. Though not quantitatively measured, the intensity level for white lighting was much less than needed when operating with red. Neither pilot felt night vision was degraded through the use of white lighting. The white lighting did not result in glare or reflection on any window, although the amber lighting of the pedestal radios caused reflections on the pilot's windshield (para 79). The white cockpit night lighting is satisfactory.

79. The radios (King Gold Crown series) installed on the 214ST were equipped with an automatic dimming feature for the amber digital displays but were not pilot adjustable. The displays were too bright and not compatible with the white lighting used for all other cockpit night lighting. As mentioned in the previous paragraph, the displays were reflected on the top center of the

pilot's windshields. The lack of cockpit night lighting harmony is undesirable and degraded what otherwise was an excellent night cockpit.

80. Two other pilot aids found on the 214ST not found on US Army aircraft were lit circuit breaker panels and approach plate/map holders. The overhead circuit breaker panel was equipped with background lighting thus allowing the pilots to "read" the circuit breaker panel without fumbling with a flashlight or the utility light. Identification of a particular circuit breaker was very easy and will significantly reduce identification time during a night emergency. The installation of a background lit circuit breaker panel is enhancing and should be incorporated in future military aircraft.

81. In addition to the normal utility lights mounted overhead, the 214ST was equipped with a pilot and copilot lit approach plate/map holder. These approach plate holders (mounted on either side of the instrument panel) are lit with pilot adjustable lights which are mounted on the front door post. These lights allow either pilot to read a map or approach plate without causing glare for the other pilot. The installation of lit approach plate/map holders is enhancing and should be incorporated in future military aircraft.

Noise Evaluation

82. The noise level in and around the 214ST was qualitatively evaluated during all phases of the A&FC tests. The noise level at either pilot station was significantly lower than in US Army helicopters. One pilot wore a HGU-26P helmet and one pilot a HGU-55/P. Both pilots flew without earplugs and all radios and the intercom system were monitored at approximately three-quarters of full volume. All intercom and radio transmissions, both outgoing and incoming, were easily understood. The low noise level in the cockpit is an enhancing characteristic.

AVAILABILITY AND MAINTAINABILITY

83. During the course of the A&FC evaluation, the availability of the 214ST was qualitatively evaluated as excellent. The aircraft performed remarkably well with minimal daily maintenance effort. During the 2 month long evaluation only two test days were lost due to maintenance. The high availability of the 214ST is enhancing.

84. Most cowlings on the 214ST were secured with Dzus fasteners or aircraft bolts. As a result, the opening or removal of most

cowlings was time consuming and required various types of screwdrivers. Consideration should be given to incorporating "quick opening" latches on the 214ST's cowlings.

AIRSPEED CALIBRATION

85. Airspeed calibration tests were conducted to determine the position error of the 214ST airspeed system. The aircraft's pitot-static system was calibrated using a T-28 fixed-wing pace aircraft and a calibrated trailing bomb (finned pitot-static system) in level flight. The trailing bomb was also used to calibrate the aircraft pitot-static system in climbs with maximum continuous power and autorotational descents. Results of these tests are presented in figures 58 and 59, appendix E. These test results differ from the data in the 214ST Flight Manual. However, the data presented in the manual represents an average for various aircraft configurations and only two longitudinal cg's and one aircraft external configuration were tested during this evaluation.

CONCLUSIONS

GENERAL

86. The following conclusions were reached upon completion of the BHIT 214ST helicopter A&FC evaluation:

a. Nine enhancing, one unsatisfactory and 16 undesirable characteristics were identified.

b. The 214ST hover performance determined in this evaluation differs from that presented in the FAA Approved Rotorcraft Flight Manual for the 214ST (BHT-214ST-FM).

c. The 214ST helicopter's IMC capabilities exceed those of any current US Army helicopter.

d. The overall design of the 214ST's cockpit significantly reduced pilot workload.

ENHANCING CHARACTERISTICS

87. The following enhancing characteristics (as defined in app D) were identified:

a. The low pilot workload during IMC flight (para 58).

b. The incorporation of a rotor brake (para 43).

c. The installation of a limited inflight main rotor tracking system (para 69).

d. The low level of cockpit vibration (para 70).

e. The installation of an air data computer (para 75).

f. The installation of background lit circuit breaker panels (para 80).

g. The installation of lit approach/map holder (para 81).

h. The low noise levels in the cockpit (para 82).

i. The high availability of the 214ST helicopter (para 83).

UNSATISFACTORY CHARACTERISTIC

88. The following unsatisfactory characteristic (as defined in appendix D) was identified: The position of the collective during the first engine start which could result in a hazardous operating condition (para 39).

UNDESIRABLE CHARACTERISTICS

89. The following undesirable characteristics (as defined in apppendix D) were identified and are listed in the order of importance:

a. The limited forward field of view during decelerations (para 61).

b. The increased pilot workload during rearward flight (greater than 10 KTAS) (para 47).

c. The high pilot workload during low speed flight in left crosswinds (10 to 20 KTAS) at high density altitudes (para 48).

d. The high pilot workload during low speed flight in right quartering head winds (para 49).

e. The inaccuracy of the radio altimeter indication due to masking by a relatively small external load (para 62).

f. The poor longitudinal position and force cues for maneuvering stability at 111 KCAS (para 28).

g. The insufficient day lighting intensity for various cockpit panels and radio displays (para 72).

h. The excessive time required to perform the aircraft run-up procedure (para 40).

i. The poor touchdown characteristics during run-on landings (para 60).

j. The high throttle sensitivity and the poor throttle correlation with main rotor speed during manual throttle operation (para 54).

k. The poor design of the main rotor tie-down devices (para 45).

l. The operational design of the rotor brake handle (para 44).

m. The requirement to utilize ground support equipment to properly preflight the tail rotor assembly (para 36).

n. The high force required to initiate and sustain movement of the collective control (para 21).

o. The reversal of torque/MGT/Ng indicator marking logic (para 53).

p. The lack of cockpit night lighting harmony (para 79).

RECOMMENDATIONS

90. The following recommendations are made:

a. The enhancing characteristics noted in paragraph 87 should be incorporated in future military aircraft designs.

b. Incorporation of a direct reading cockpit loadmeter in all military cargo helicopters (para 63).

c. Incorporation of cockpit ventilation systems as effective as that installed in the 214ST in all military aircraft (para 77).

d. Incorporation of illuminated aircraft fluid reservoir sight gauges in all military aircraft (para 37).

e. Incorporation of door status indicators in all military aircraft (para 38).

f. BHTI resolve the conflict of maximum glide distance (autorotation) airspeed in the FAA Approved Rotorcraft Flight Manual with USAAEFA's test results (para 14).

91. The following recommendations are made if the 214ST helicopter is considered for military application:

a. The unsatisfactory and undesirable characteristics reported in paragraphs 88 and 89, respectively, should be corrected.

b. Evaluate main rotor flapping during all low speed maneuvers at wind azimuths between 90 and 180 degrees with wind speeds greater than 25 KTAS (para 50).

c. Install anti-submarine belts (crotch belts) on both pilot seats (para 76).

d. The normal acceleration limits should be expanded for military application (para 26).

e. Incorporation of "quick opening" latches on the 214ST's cowling (para 84).

f. Combine various cockpit annunciator panels to reduce cockpit clutter (para 74).

g. Reposition the Sperry AA335 radio altimeter so that altitudes less than 100 feet may be more easily discerned by the pilot (para 73).

h. Incorporate a "thru flight checklist" (para 40).

i. The hover performance section of the FAA Approved Rotorcraft Flight Manual should be changed to allow more precise prediction of the 214ST hovering capability.

APPENDIX A. REFERENCES

1. Letter, AVSCOM, AMSAV-ED, 11 June 1985, subject: Airworthiness and Flight Characteristics (A&FC) Test of the BHTI 214ST Helicopter. (Test Request).

2. Flight Manual, BHT-214ST-FM, *FAA Approved Rotorcraft Flight Manual, 214ST Helicopter,* Bell Helicopter Textron, Incorporated, Revision 2, dated 12 February 1982.

3. Supplemental Flight Manual, BHT-214ST-OH, *214ST Rotorcraft Operators Handbook,* Bell Helicopter Textron, Incorporated, 20 October 1982, with revision 4.

4. Training Manual, *Bell 214ST Pilot Training Manual,* Flight Safety International, with Revision 3, April 1985.

5. Product Data Manual, *Bell 214ST Product Data, Volumes 1 and 2,* Bell Helicopter Textron, Incorporated, March 1985.

6. Test Plan, USAAEFA Project No. 85-10, *Airworthiness and Flight Characteristics Test of the BHTI 214ST Helicopter,* 8 July 1985.

7. Letter, AVSCOM, AMSAV-ED, 1 August 1985, subject: Test Plan, Airworthiness and Flight Characteristics (A&FC) Test of the BHTI 214ST Helicopter, USAAEFA Project No. 85-10.

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10. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For,* 3 April 1962.

11. DOD Flight Information Publication (Enroute) Flight Information Handbook, 14 February 1985.

12. Field Circular, FC 1-212, Aircrew Training Manual, *Utility Helicopter, UH-60,* 31 August 1984.

13. Specification, BHT Specification 299-947-131, *General Specification for Intergal, 5 Volt, Instrument White Lighting Systems,* Revision C, 22 March 79.

14. Maintenance Manual, BHT-214ST-MM, *Bell 214ST Maintenance Manual, Volumes 1 and 2,* Bell Helicopter Textron, Incorporated, 4 January 1982.

APPENDIX B. DESCRIPTION

TABLE OF CONTENTS

GENERAL

1. The 214ST is a twin-turbine engine, two-bladed helicopter (photos 1 through 8) manufactured by Bell Helicopter Textron Incorporated. The 214ST is designed for internal and external cargo during visual and instrument, day and night operation. The normal crew consists of two pilots and allows seating for up to 18 passengers. The fuselage consists of a forward section (includes the cabin area, two baggage compartments (limited to 1100 pounds), the engine and transmission compartments) and the tailboom which supports the tail rotor drive system and the Fly By Wire (FBW) elevator system. The engines drive the semi-rigid main rotor system and the two bladed delta-hinged tail rotor. The drive train system on the 214ST includes a combining gearbox, main drive shaft, main transmission, main rotor mast and a tail rotor drive system. The tail rotor drive system includes six tail rotor drive shaft sections, an intermediate gearbox, and a tail rotor gearbox. The helicopter is equipped with two flight control hydraulic systems with only one required for safe flight. The primary electrical systems are normally powered by two 30-volt, 400-ampere generators which are driven by the combining gearbox. The fuel (jet fuel) is stored in seven rupture-resistant fuel cells located under the passenger compartment floor and below the combining gearbox. The helicopter can be configured with either skid type or nonretractable wheel type landing gear. The helicopter is equipped with standard flight controls and a Stability and Control Augmentation System (SCAS). Additionally, it is equipped with an Attitude/Altitude Retention System (AARS). The general helicopter arrangement and dimensions are shown in figures 1 and 2.

AIRFRAME

Fuselage

2. The airframe, as shown in figure 3, consists of two major assemblies: the forward fuselage and the tailboom. The forward fuselage is a combined semi-monocoque and reinforced-shell structure with transverse bulkheads and metal and fiberglass covering. Two longitudinal main beams provide the primary structural support. The forward fuselage includes the cabin area, two baggage compartments, the engine and transmission compartments, and the fuel tank enclosures. The work and engine decks, some main beam panels, the cabin floor panels, and the cabin roof panels are of sheet metal and honeycomb construction. The cabin floor structure (limited to 100 pounds per square inch) is equipped with several tie-down rings for securing cargo and allows up to 18 passenger seats.

Photo 1. Front View, 214ST Helicopter

Photo 2. Right Rear View, 214ST Helicopter

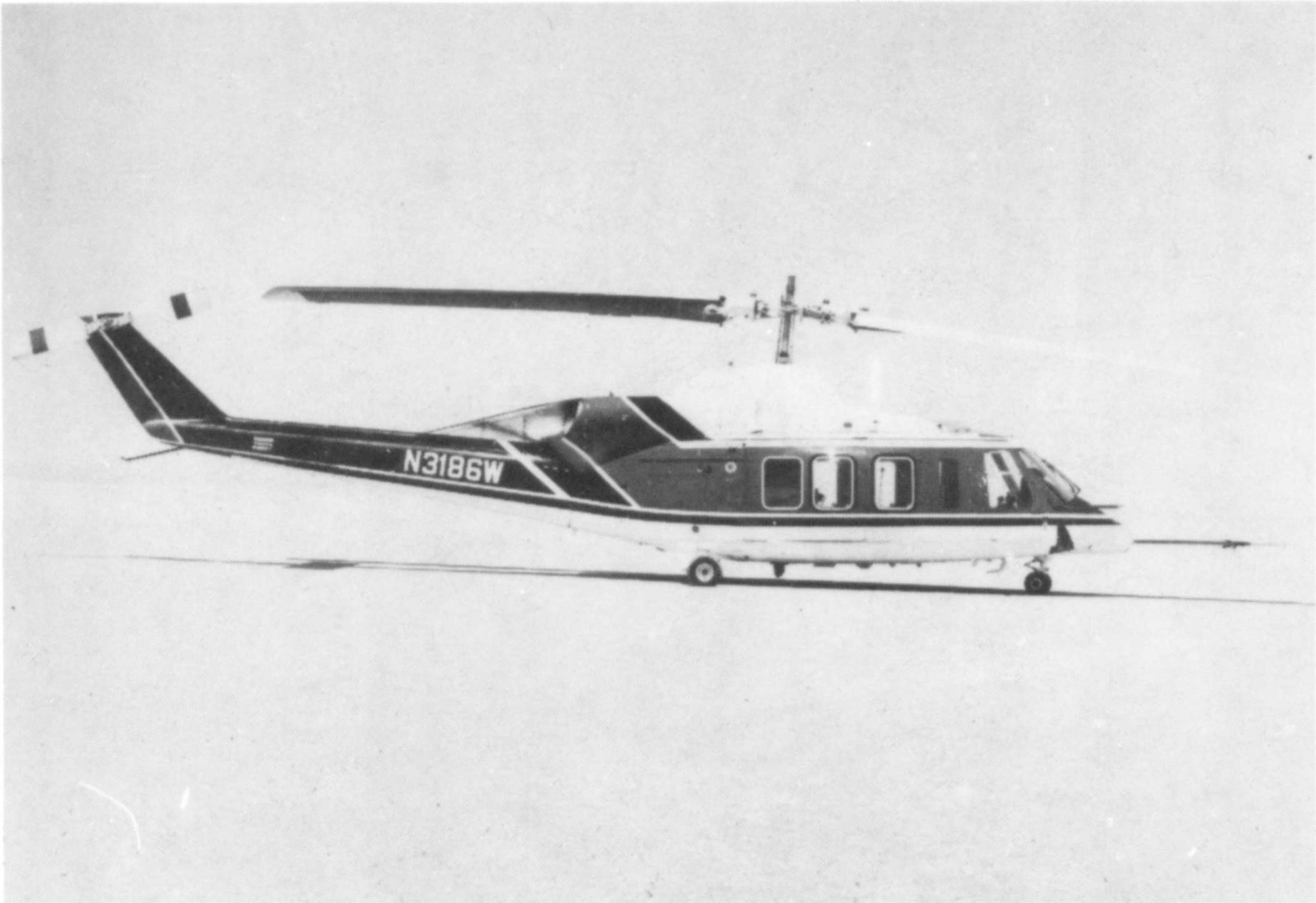


Photo 3. Right View, 214ST Helicopter

Photo 4. Right Front View, 214ST Helicopter

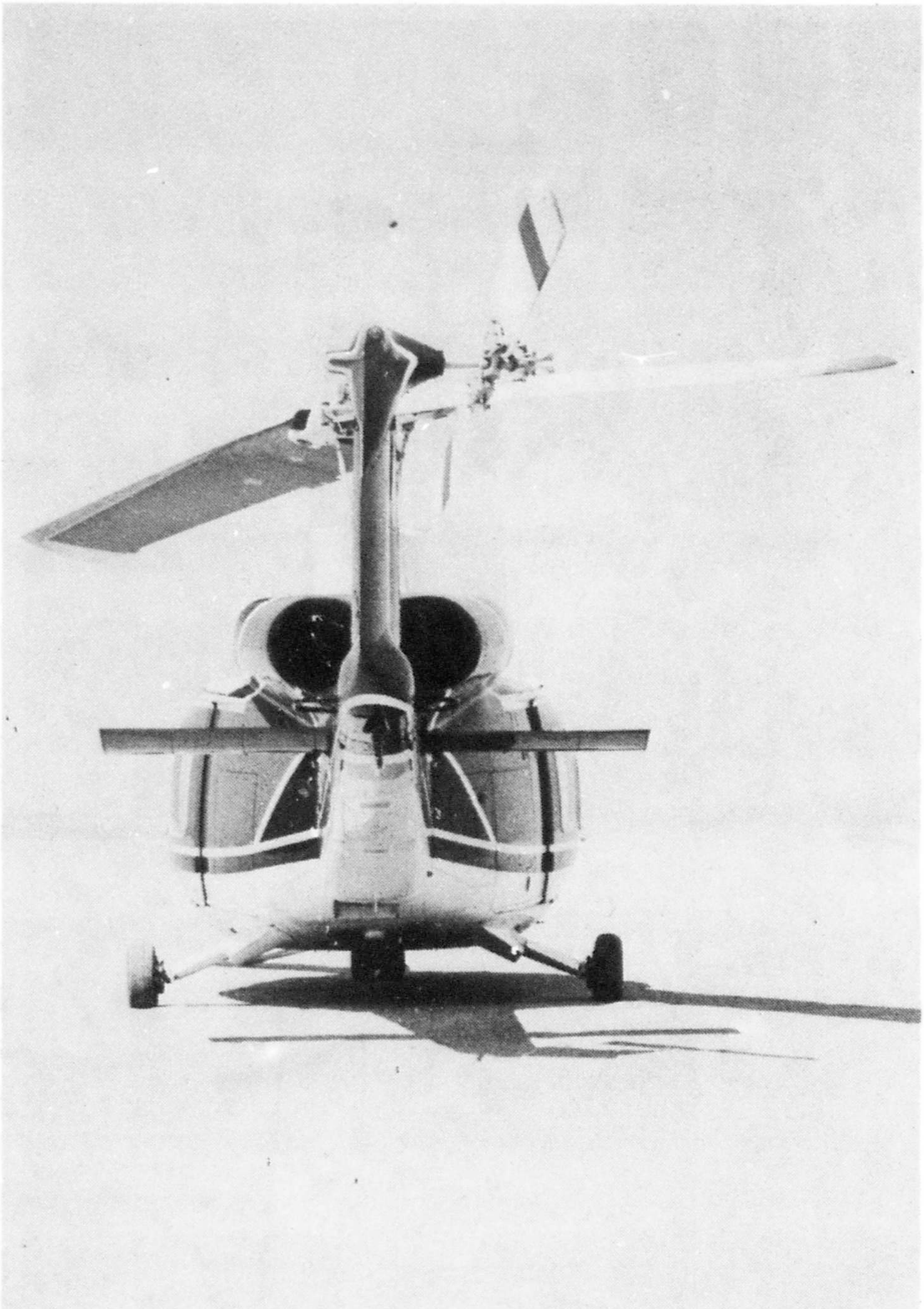


Photo 5. Rear View, 214ST Helicopter

Photo 6. Left Front View, 214ST Helicopter

Photo 7. Left View, 214ST Helicopter

Photo 8. Left Rear View, 214ST Helicopter

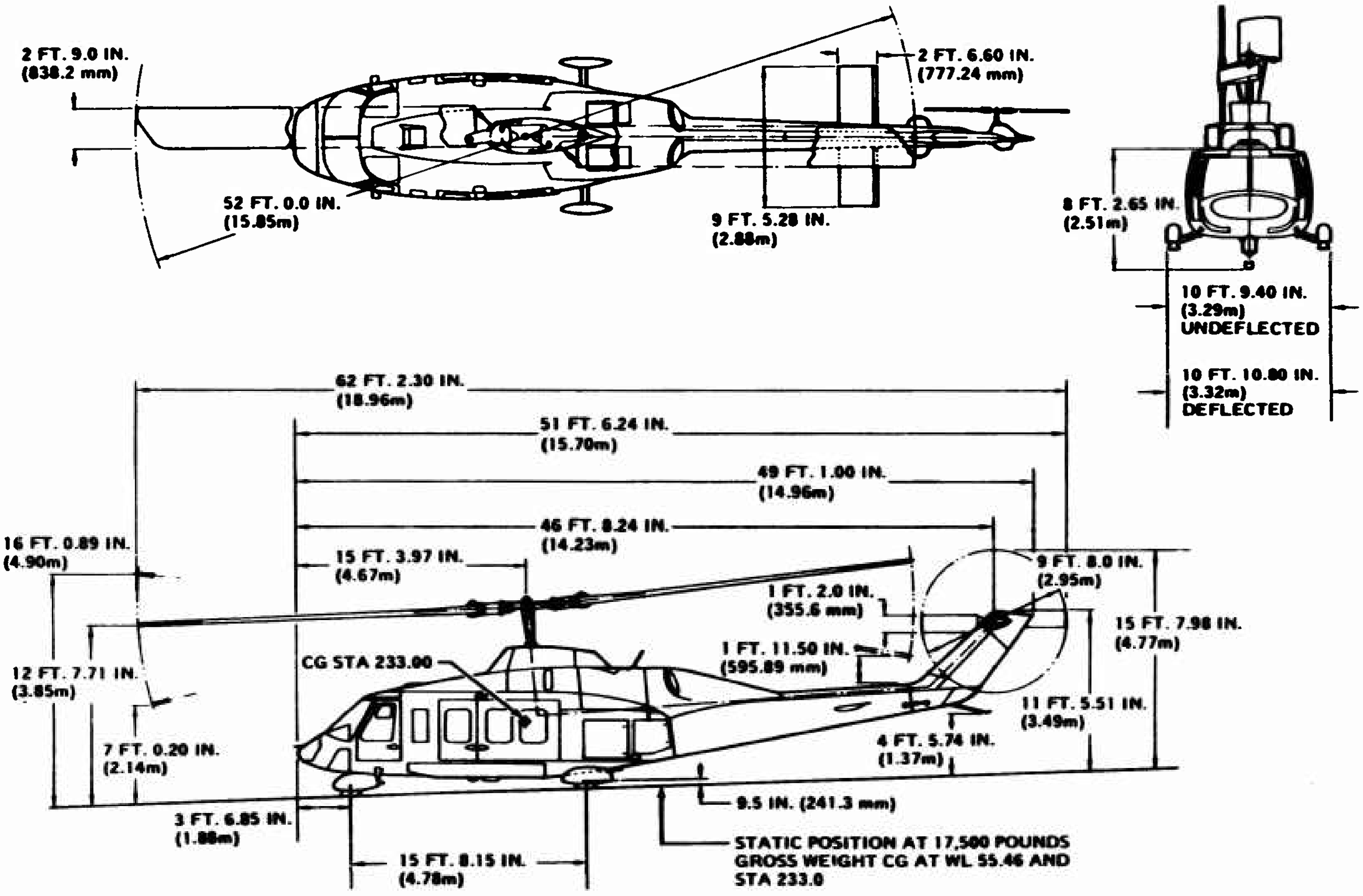


Figure 1. Principal Dimensions - Wheeled Landing Gear

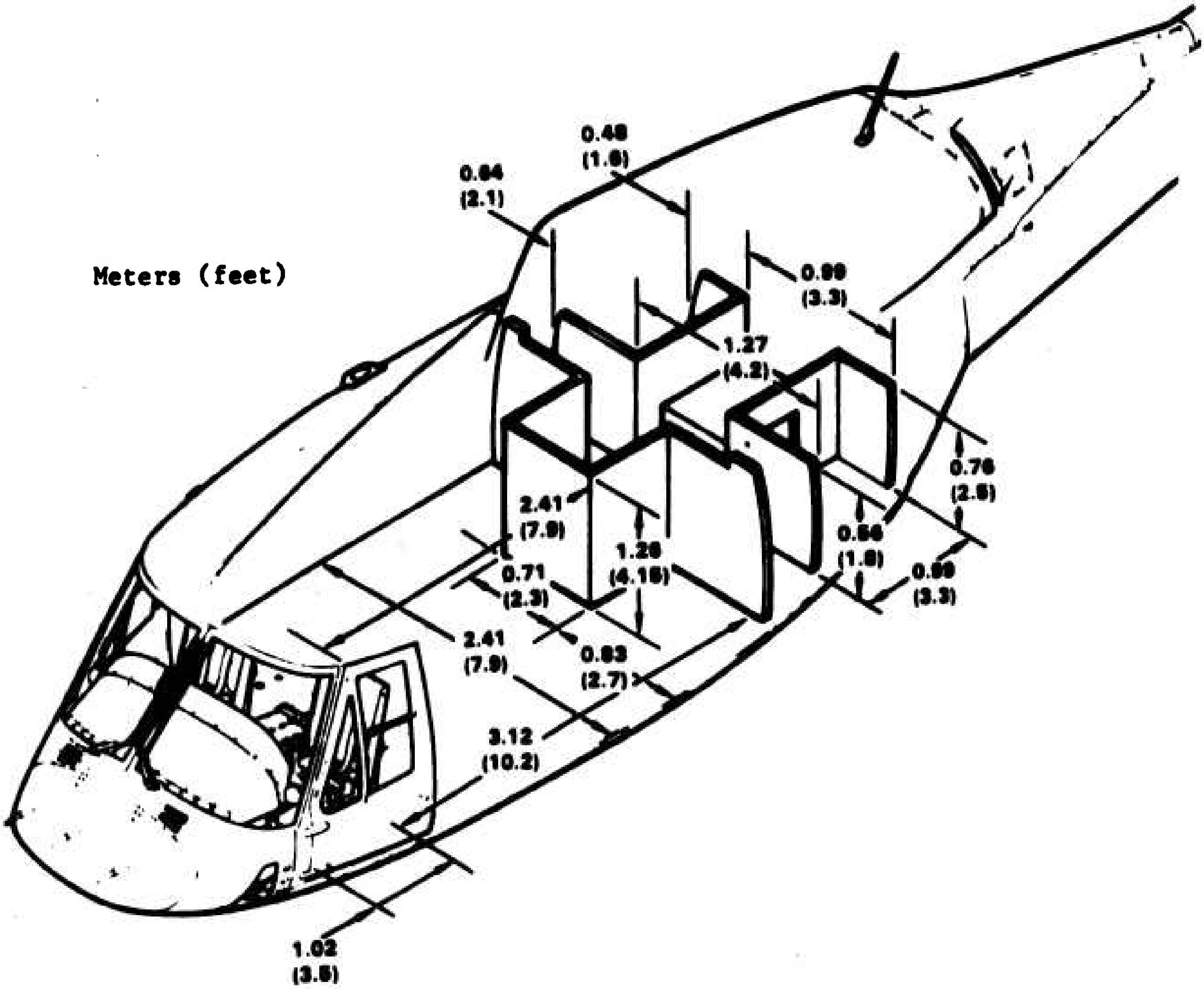
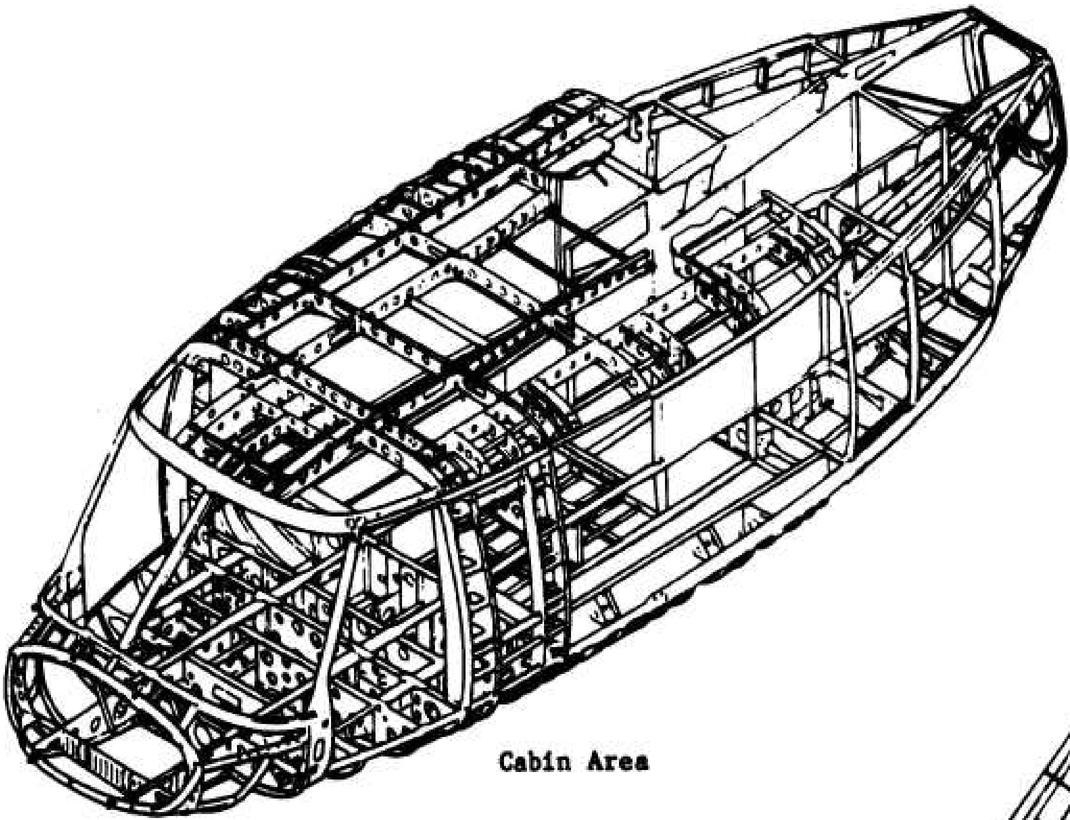
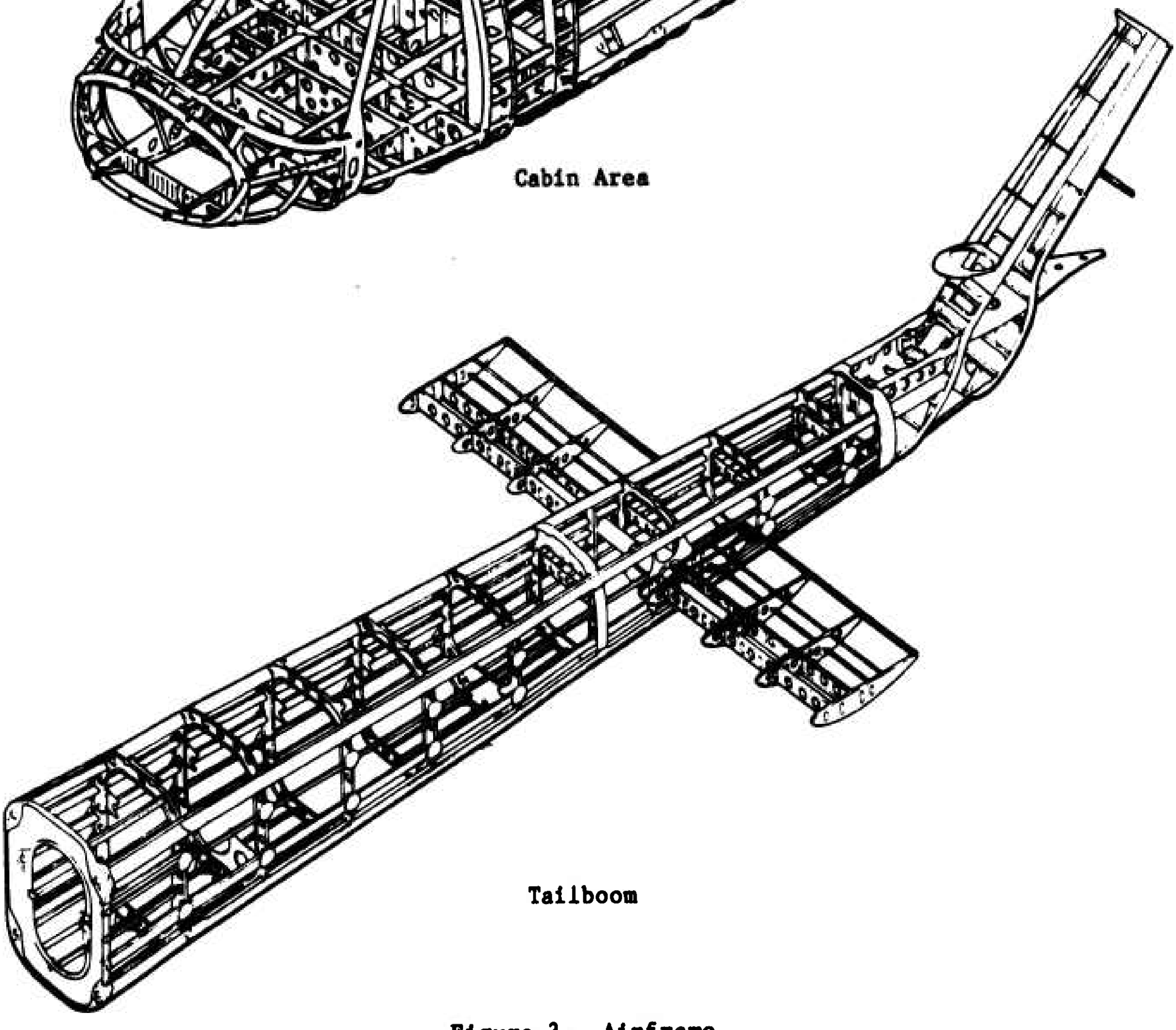


Figure 2. Principal Interior Dimensions



Cabin Area



Tailboom

Figure 3. Airframe

3. The tailboom is a semimonocoque structure with internal support members, skin, and driveshaft covers made of aluminum alloy. The tailboom structure has transverse bulkheads, longitudinal stringers and longerons. The tailboom supports the tail rotor system, vertical fin, elevator, and tailskid. The vertical fin is cambered to reduce tail rotor thrust requirements in forward flight.

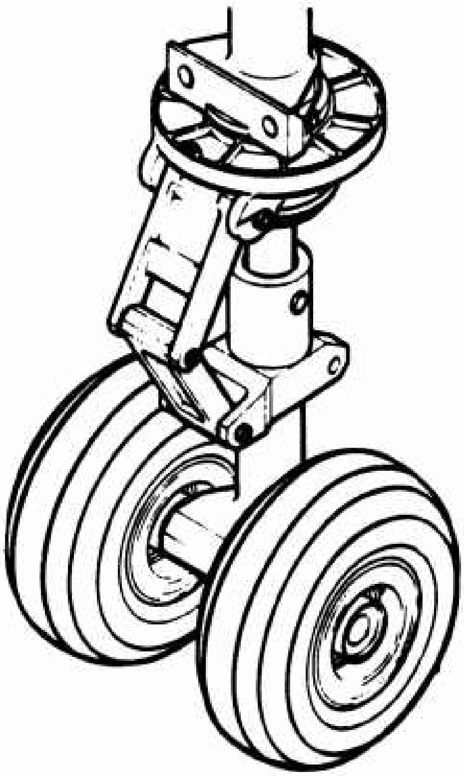
Landing Gear

4. The wheeled landing gear (fig. 4) is a fixed, tricycle landing gear installation. The nose gear consists of a pair of wheels mounted with an air/oil shock strut and incorporates a shimmy damper to reduce nose gear vibration. With the aircraft on the ground, the nose gear shock strut is compressed and allows the nose gear to swivel 360 degrees. Inflight, the nose gear shock strut extends and allows a centering cam to align the nose gear longitudinally.

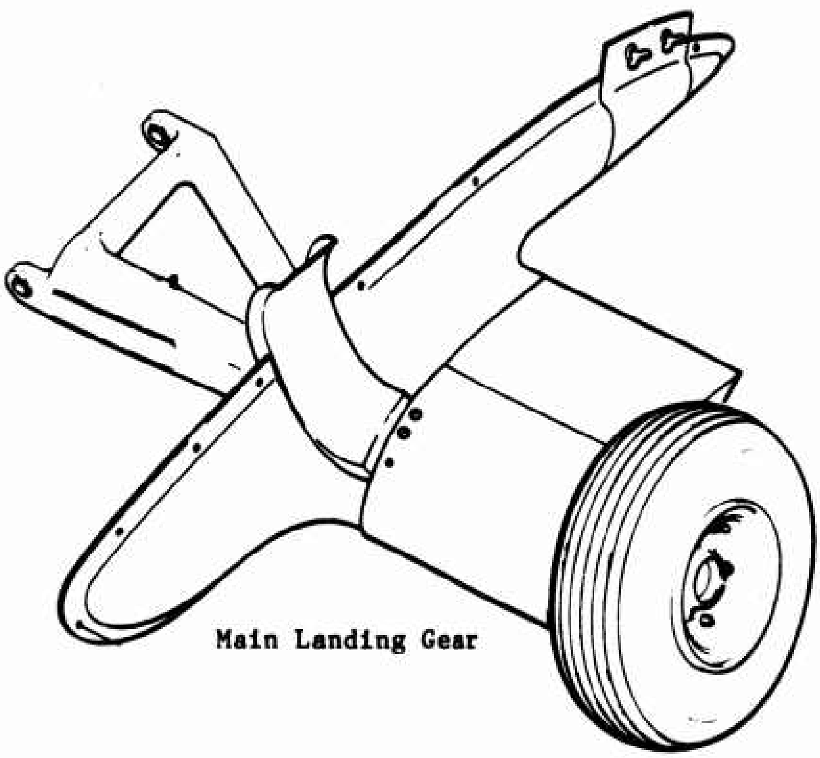
5. Each main gear consists of a cantilevered beam and an air/oleo assembly. The beam assembly attaches to a fuselage fitting on the inboard end and pivots to allow vertical displacement. The wheel assembly incorporates a tubeless tire and differential hydraulically operated disc breaks mounted to an integral axle. The upper end of the oleo assembly mounts to the fuselage fitting and the lower end supports the beam assembly. Each beam is fitted with an aerodynamic fairing.

Main Rotor

6. The 214ST helicopter is equipped with a two-bladed semi-rigid main rotor system (fig. 5). The major components are: a titanium yoke attached to the main rotor mast through a pair of elastomeric bearings (similar to AH-1 design); aluminum-alloy blade grips (similar to UH-1H design); and two main rotor blades. The hub assembly is unusual in that the pilot has a limited inflight tracking capability. The limited blade tracking can be accomplished through a small DC motor mounted on the pitch horn on the main rotor blades. Through a worm gear arrangement, the motor causes the pitch horn to raise or lower. This changes the pitch of the rotor blade without moving the pitch link tubes up to 0.2 degrees. The main rotor blades are constructed primarily of fiberglass. A fiberglass main spar and a fiberglass trailing-edge strip form the framework for each blade. A Nomex honeycomb core fits between the main spar and the trailing edge and supports a skin composed of layers of nonwoven fiberglass. The outboard half of the spar assembly incorporates a lead noseblock to increase rotational inertia and maintain chordwise balance. The leading edge of each blade is protected by a stainless steel and



Nose Landing Gear Wheel and Tire Assembly



Main Landing Gear

Figure 4. Landing Gear

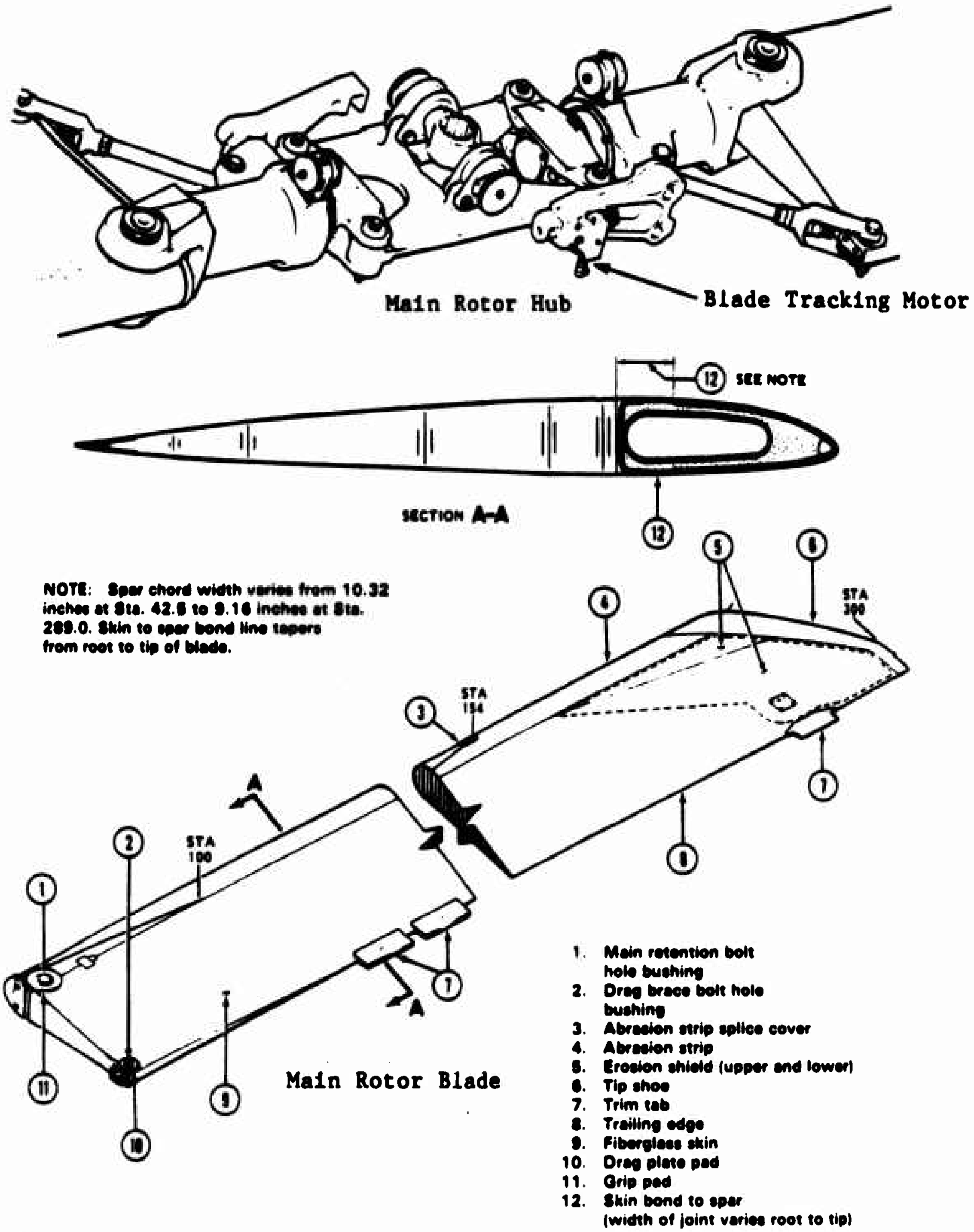


Figure 5. Main Rotor Hub and Blade Assembly

titanium abrasive strip. The blade tip area is covered with a replaceable stainless tip shoe. The paint system incorporates a semiconductive graphite layer to aid in the dissipation of static electricity. Each blade is equipped with three trim tabs which are used for blade tracking.

Tail Rotor

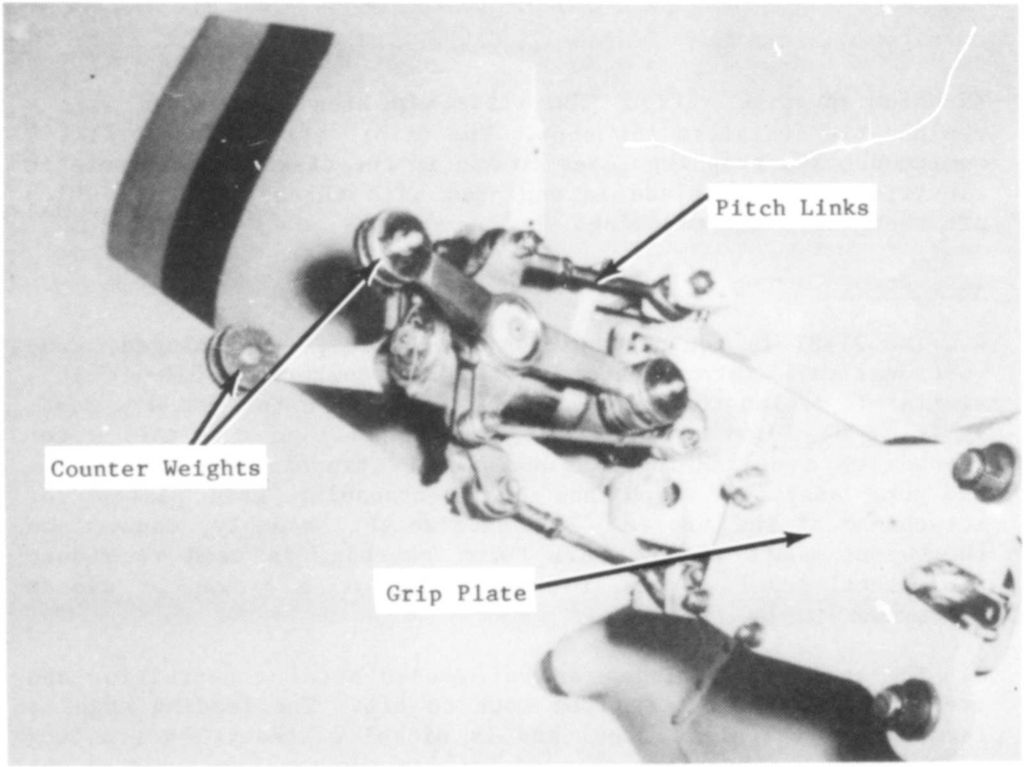
7. The 214ST is equipped with a two-bladed, delta-hinged, conventional tail rotor system (Bell part number 214-010-800-105) similar in design to the AH-1S tail rotor. The tail rotor assembly (fig. 6) is mounted on the output shaft of the tail rotor gearbox by a delta-hinged trunnion. The trunnion is mounted on the yoke assembly which has two rectangular grip plates for attachment of the blades. A counterweight assembly, mounted on the output shaft of the tail rotor gearbox, is used to reduce directional pedal forces in the event of a hydraulic system failure.

8. The tail rotor blades are of bonded metal construction and are tapered in thickness from root to tip. The leading edge is formed from stainless steel and is nickel plated. The trailing edge strip is fiberglass, bonded between the trailing edge of the skin. The core is an aluminum honeycomb material, and the upper and lower skin is made from stainless steel. A fiberglass crack-arrester strap is installed within the blade directly behind the leading edge. Stainless-steel doublers are bonded to the blade roots to add structural integrity at the blade attaching points.

Fire Detection and Protection System

9. The fire detection system consists of a separate fire-warning system for each engine and a smoke detector in each of the two baggage compartments. Each system provides visual cockpit warning. The engine fire detection system is similar to those found in other Bell helicopters. An engine fire is detected by a single heat sensing element (Inconel tubing) in each engine compartment. A cockpit warning light is illuminated whenever the Inconel tubing temperature exceeds 500 degrees Fahrenheit. The warning system is designed to ensure that short or open circuits will not illuminate the cockpit warning lights.

10. The baggage compartment smoke detectors are light-sensitive systems. When smoke in the compartment reduces light transmission to 30 to 35 percent below that of clear air, the smoke detector sends a signal which turns on the appropriate BAGGAGE FIRE warning light in the cockpit. The baggage compartment smoke detector system has a "press to test" feature which checks bulb and system electrical continuity.



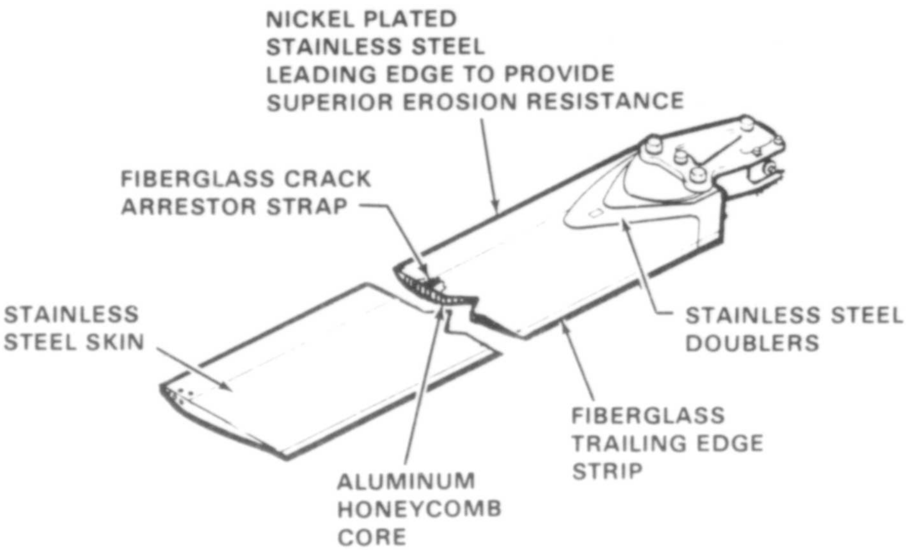


Figure 6. Tail Rotor System

11. The helicopter is equipped with an engine fire-extinguishing system. The system is similar to engine fire-extinguishing systems found on several Army aircraft. Two spherical fire-extinguishing bottles, which are pressurized to approximately 600 pounds per square inch, are filled with Halon 1301 (Freon) and nitrogen. Each bottle has two discharge outlets equipped with electrically actuated cartridges which allow both bottles to be discharged into either engine compartment. The fire-extinguishing system is controlled by a fire control panel (two arming switches and one agent release switch). The arming switches have three functions: arming the extinguishing system, closing the fuel shutoff valve to the appropriate engine, and closing both engine bleed air valves.

Pitot-Static System

12. The pitot-static system (fig. 7) consists of three heated pitot tubes, six heated static ports, and associated components. The three pitot tubes (fig. 8) are located on the nose of the aircraft and the static ports are located just forward of the cockpit doors (three static ports on each side). The right pitot tube (#3) and the upper left and lower right static ports are used for the pilot's pitot-static instruments and the air data computer. It is also used in a voting capacity in the event of a mismatch between #1 and #2 systems. The center pitot tube (#2) and center static ports are used to supply data to the FBW elevator system. The left pitot tube (#1) and the lower left and upper right static ports supply input to the copilot's pitot-static instruments and the FBW elevator system. The copilot's pitot and static lines are equipped with pilot controlled bypass valves. By placing the valves in the "bypass" position the airspeed transducer (copilot's only) is removed from the pitot-static system.

Air Data Computer

13. The air data computer (ADC) (fig. 9) provides present gross weight, density altitude, and velocity never exceed (V_{NE}) information to the pilots. Additionally, the ADC will provide the results of "power assurance" checks during the aircraft runup. The ADC receives input from: # 3 airspeed transducer, fuel quantity indicator, outside air temperature sensor, pilot's power turbine tachometer, pilot's gas producer speed, measured gas temperature gauge, and the dual torque gauge. After the aircraft's gross weight is set into the ADC during runup, the ADC continually updates for fuel burn, temperature and pressure altitude changes and displays the current gross weight, density altitude, and V_{NE}. The pilot is also provided a red warning

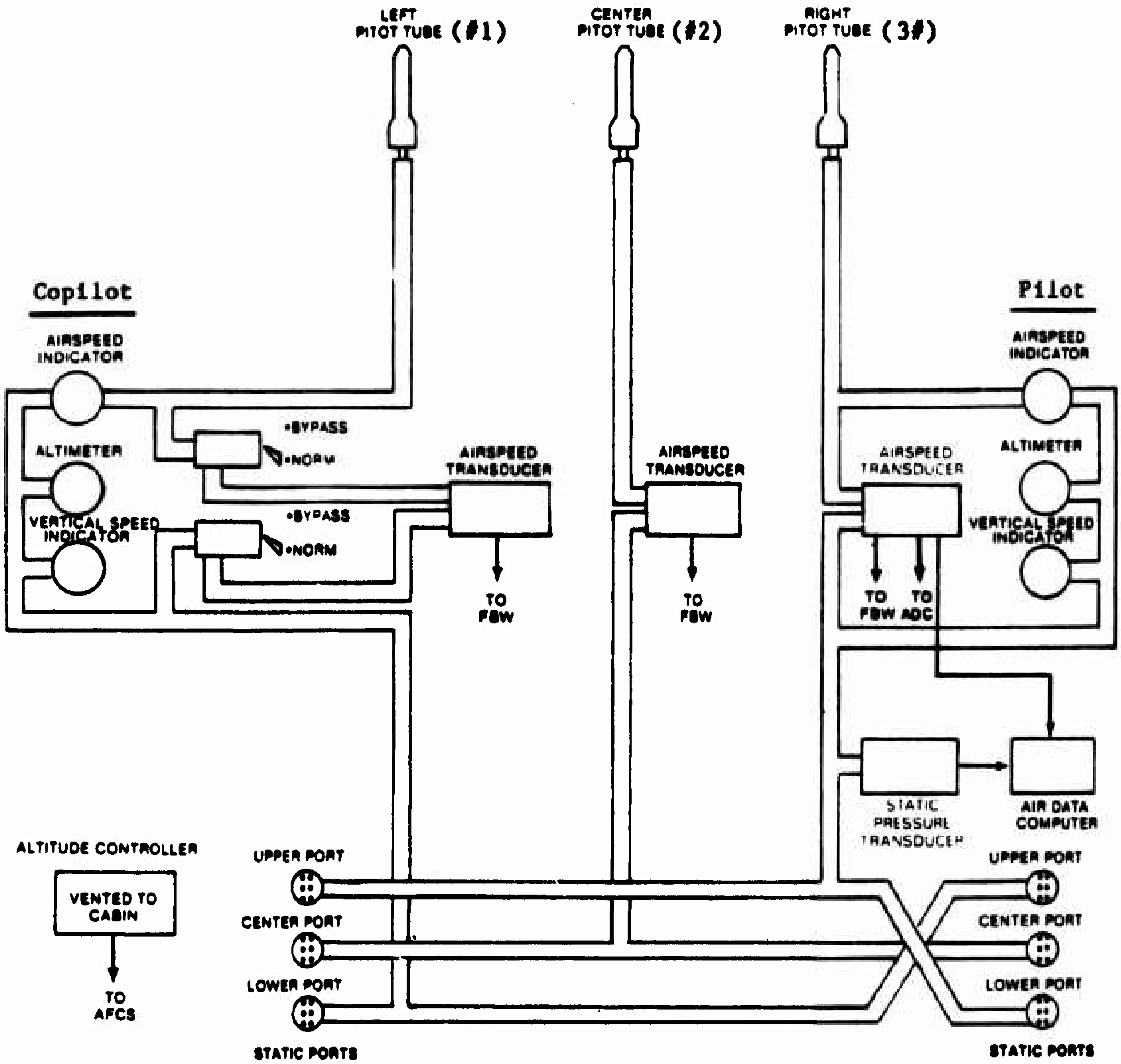
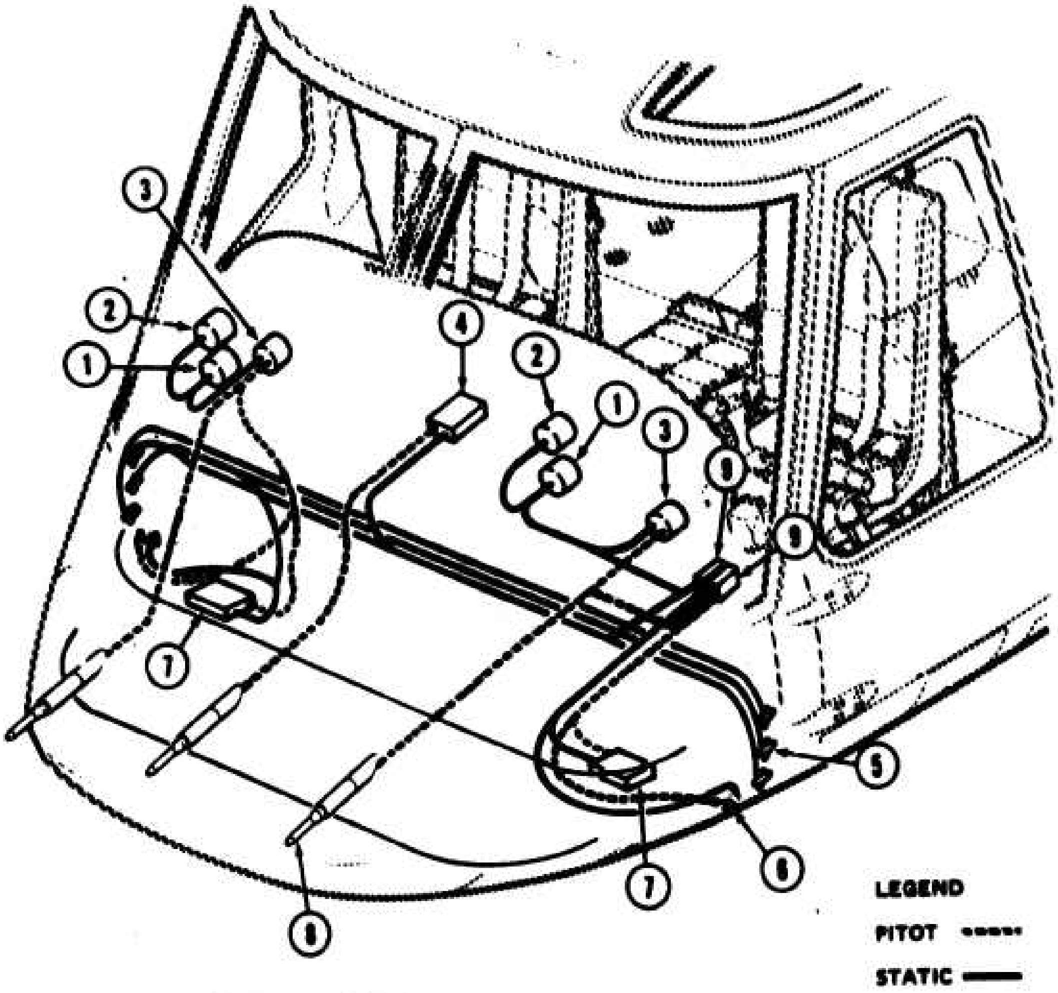
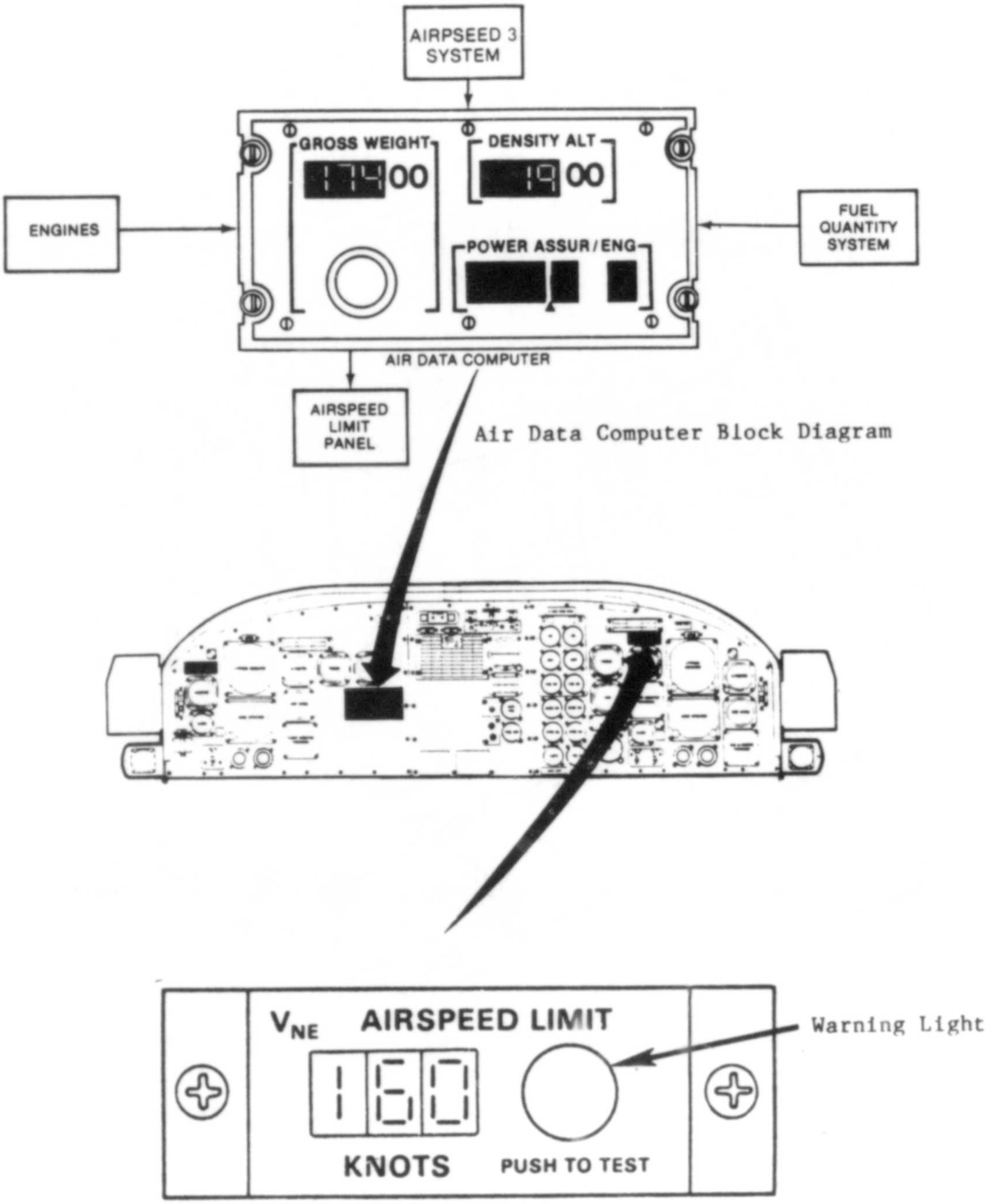


Figure 7. Pitot-Static Schematic



1. Vertical speed indicator
2. Altimeter
3. Airspeed indicator
4. Air data computer
5. Static vents (3 each side)
6. Drain caps (2 each side)
7. Airspeed transducer
8. Pitot tube (3 each)
9. Valve

Figure 8. Pitot-Static System



Airspeed Limit Panel

Figure 9. Air Data Computer Components

light on the V_{NE} indicator which flashes if the aircraft is flown at airspeeds greater than V_{NE} + 2 knots.

Fly-By-Wire Elevator System

14. The 214ST utilizes an elevator, similar to those installed on other Bell helicopters, for longitudinal stability and to reduce inflight pitch attitude changes. However, the elevator is unusual in that it is controlled by a Fly-By-Wire (FBW) system rather than a series of push-pull tubes. Additionally, the right elevator is set approximately 3 degrees more trailing edge down (TEDN) than the left. The elevators are moved by dual, electrically powered actuators which are connected in parallel. During normal operation (above 50 KIAS), the actuators respond in unison to equally share the aerodynamic loads. The FBW elevator system (fig. 10) includes two elevator control amplifiers, three airspeed sensors, four control motion transducers (CMT), an elevator position indication system, a control panel and failure lights.

15. Flight control input signals from collective and longitudinal CMT's, together with airspeed data, are electronically mixed by the elevator amplifiers and produce a signal to drive the elevator actuator. Each amplifier has dual signal channels and differences between them are detected as failures. A detected failure will automatically disengage that amplifier and remove power from the corresponding actuator. The remaining system will continue to operate in a normal manner (fail-operate) with no system degradation. A subsequent failure in the remaining amplifier or third airspeed system will lock the elevator in place (fail-passive). In the event of a FBW elevator system failure, airspeed is limited by the type failure (single or dual), aircraft gross weight and density altitude as presented in the operator's manual. The V_{NE} is not less than 75 KCAS for any elevator failure.

16. The FBW elevator system incorporates all three airspeed systems, #1, #2, and #3 (fig. 7). The signals from the "primary" systems transducers (#1 and #2) are compared with each other for disagreement thru monitor circuits located in the elevator control amplifiers. The #3 airspeed system is used only in a "voting" capacity to determine which airspeed system (#1 or #2) is correct and the "incorrect" elevator system is shutdown.

17. The cockpit controls and indicators consist of an elevator control panel, an elevator position indication system and various failure lights integrated in the FLT CONTROLS panel. The pilots can turn power ON or OFF and test or monitor either elevator system, however, the pilot cannot manually slew the elevators.

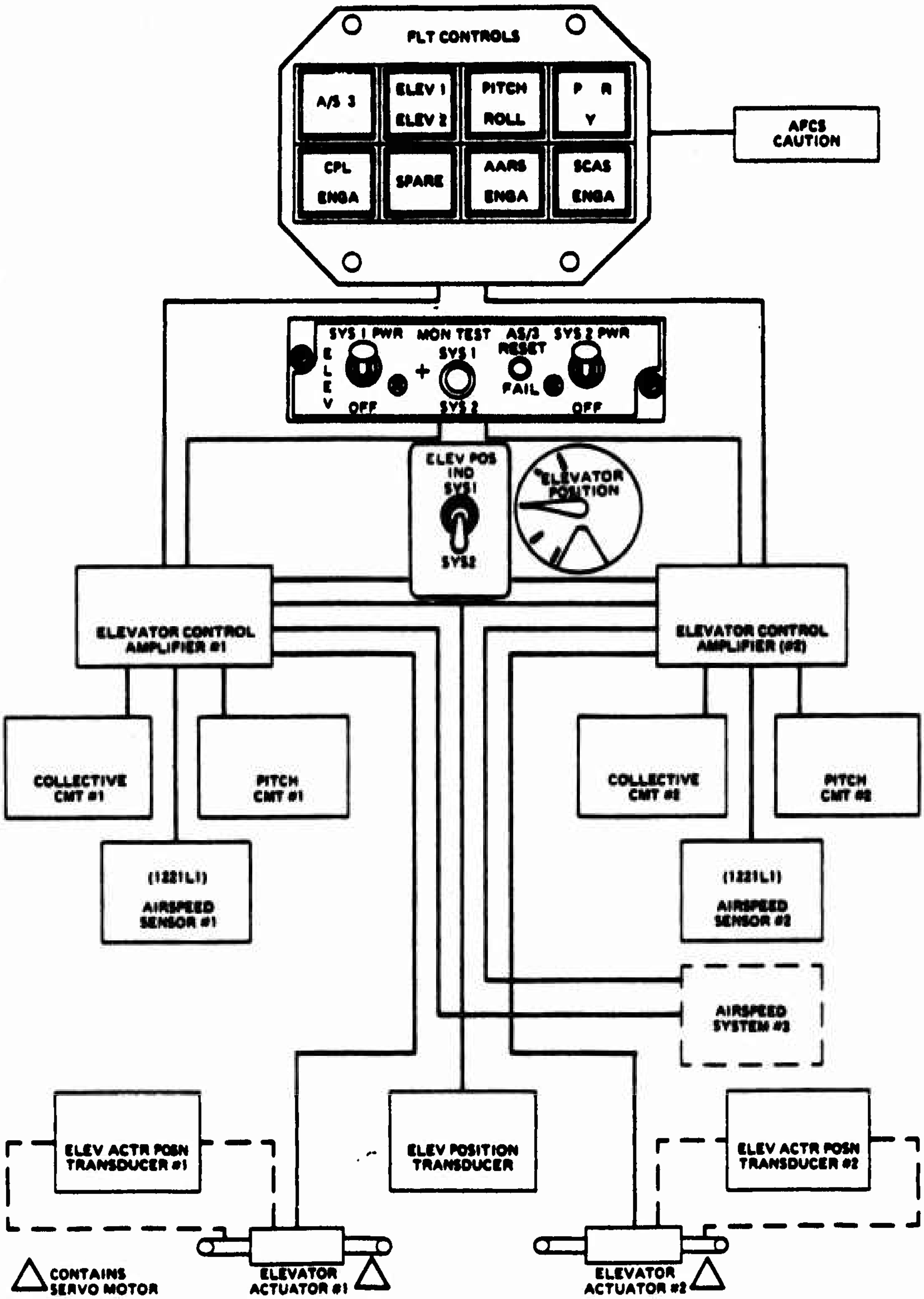


Figure 10. FBW Elevator System Block Diagram

Transmission System

18. The transmission systems of the 214ST (fig. 11) consist of a combining gearbox, a main transmission, an intermediate (42 degree) gearbox, and a tail rotor (90 degree) gearbox. Under dual engine conditions, the drive system is rated at 2350 shp (5-min), with a maximum continuous rated power of 1950 shp. During single-engine operations, the drive system is rated at 1725 shp (2.5-min), 1625 shp (30 min) and 1530 shp (maximum continuous).

19. Power from the engines is transmitted directly into the combining gearbox. The combining gearbox is mounted forward of the engines and aft of the main transmission. The combining gearbox provides speed reduction and combines the power of the two engines. Overrunning clutches are incorporated which will disengage should an engine fail to allow autorotation or single-engine operation. Two flight control hydraulic pumps and both main generators mount to the combining gearbox.

20. The main transmission is mounted on top of the cabin fuselage just forward of the combining gearbox. A special NODAL-BEAM transmission mount (fig. 12) is incorporated which was specifically designed to reduce two-per-rev vertical vibrations. The main transmission further reduces the speed of the drive system, provides an angle change for the main rotor, and supports the main rotor. A manual (hand-operated) hydraulic rotor brake system is incorporated for slowing and securing the main rotor during engine shutdown (below 40 percent rotor speed).

21. The tail rotor drive system incorporates six shaft sections (five are interchangeable), an intermediate (42 degree) gearbox and a tail (90 degree) gearbox. Both gearboxes further reduce the speed of the drive system and change the angle of drive to the tail rotor.

22. A chip detector system consisting of nine chip detectors is incorporated in the drive system to warn the pilot of impending transmission failures. A fuzz-burn feature allows the pilot three attempts to clear a chip indication (burn-off the metal particles when they are insignificant) and continue the flight. A chip detector memory unit (latch-type) is mounted in the avionics bay. Any chip indication will cause the appropriate latch indicator to latch and the indicator will not reset until maintenance personnel have checked and reset the indicator manually.

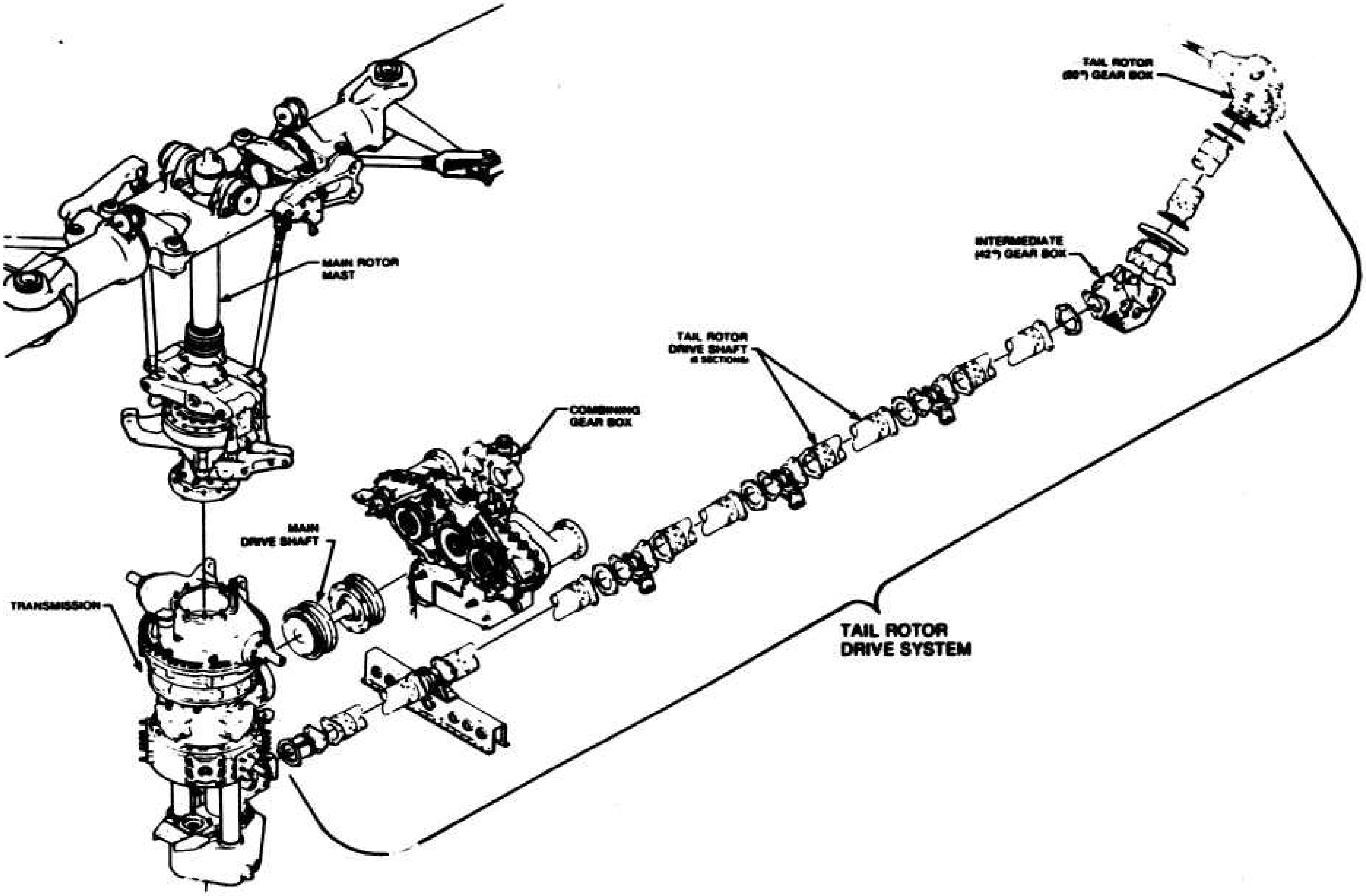


Figure 11. Powertrain

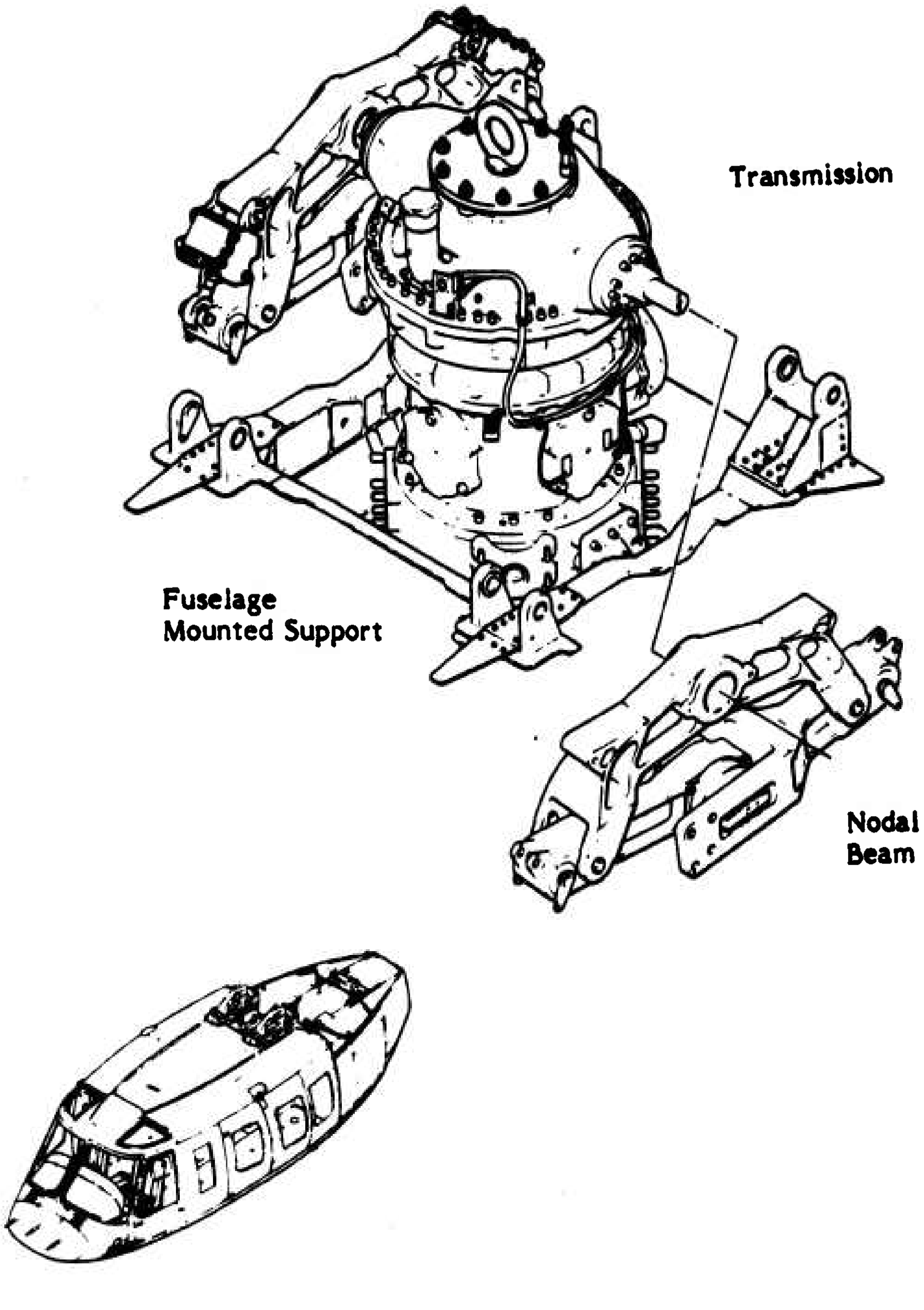


Figure 12. Nodamatic Transmission Mount

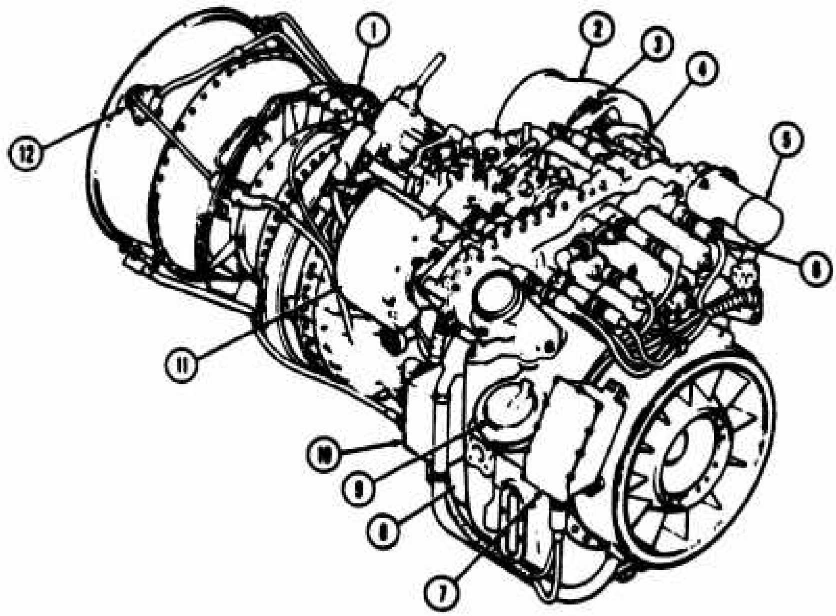
POWERPLANTS

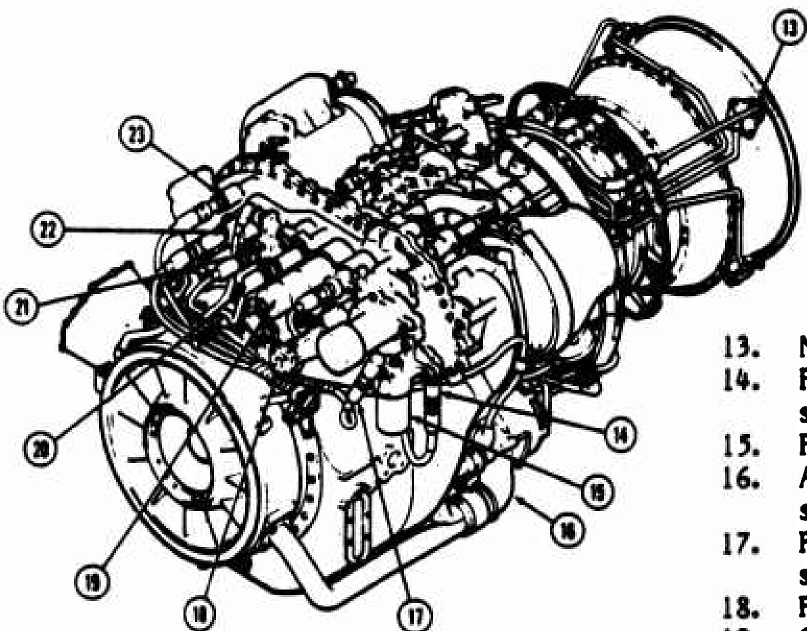
General

23. The primary power plants for the 214ST helicopter are two General Electric CT7-2A front drive turboshaft engines, rated at 1,625 shaft horsepower (shp) at 21,000 RPM (takeoff-5 minutes, sea level, standard day, uninstalled). The CT7-2A engine is a civilian version of the military T700-GE-700 engine with the following modifications (required for FAA Certification); steel filter bowls for oil and fuel, quartz transparent oil level sight glasses, shrouded and drained fuel lines (to prevent coking), cockpit indicators for impending fuel and oil filter bypass and inoperative scavenge blower, and a dual engine total torque limiting device (part of the fuel control). The engines are mounted in nacelles on either side of the main transmission. Each engine has four modules: cold section, hot section, power turbine section, and accessory section (fig. 13). Design features of each engine include an axial-centrifugal flow compressor, a through-flow combustor, a two stage air-cooled high pressure gas generator turbine, a two stage uncooled power turbine, and self-contained lubrication and electrical systems. The compressor has a bleed air capability which provides heated air for engine inlet anti-icing, cockpit/cabin heating, and crossbleed engine starting. In order to reduce sand and dust erosion and foreign object damage, an integral particle separator (IPS) operates when the engine is running. The CT7-2A engine also incorporates a history recorder (similar to the T700-GE-700) which records total engine events. Pertinent engine data are shown below.

Model..............................	CT7-2A
Type...............................	Turboshaft
Rated Power........................	1625 shp uninstalled at sea level, standard day, static conditions at 21,000 RPM, takeoff power (5-min)
Compressor.........................	Five axial stages, 1 centrifugal stage
Variable geometry..................	Inlet guide vanes, and stages 1 and 2 stator vanes
Combustion chamber.................	Single annular combustor with axial flow
Gas generator stages...............	2
Power turbine stages...............	2
Direction of engine rotation (aft looking fwd)................	Clockwise
Weight (dry).......................	432 lb max

1. Thermocouple harness
2. Particle separator scavange blower
3. HMU
4. Sequence valve
5. Oil cooler
6. Chip detector
7. History recorder
8. ECU
9. Oil filler cap
10. Ignition exciter
11. Starter
12. Torque/overspeed sensor





13. Np speed sensor
14. Fuel pressure switch
15. Fuel switch
16. Anti-icing and start bleed valve
17. Fuel bypass switch
18. Fuel boost pump
19. Oil filter
20. Oil filter bypass sensor
21. Alternator starter
22. Oil pressure sensor
23. Oil temperature sensor

Figure 13. General Electric CT7-2A Turboshaft Engine

Length..............................	46 in.
Maximum diameter....................	25 in.
Fuel................................	MIL-T-5624 (grade JP-4 or JP-5), ATSM D-1655, (Jet A, A-1 or B)
Lubricating oil.....................	MIL-L-7808H or MIL-L-23699C or approved Type II oils

Engine Power Control System

24. The engine control system consists of a load demand system, power control system, and a speed control system. The load demand system supplies a collective signal to the load-demand spindle on the engine hydromechanical unit (HMU) which controls fuel flow. The power control system for each engine controls the HMU through the engine throttles in the cockpit. The engine speed control system operates through the engine trim wheels located on the center pedestal and the electrical control unit (ECU). The ECU receives inputs from the engine alternator, thermocouple harness, power turbine (Np) sensor, torque and overspeed sensor, torque signal from the opposite engine for load matching, and feedback signals from the HMU for system stabilization, and a demand speed from the engine trim wheels. The temperature limiting system operates when the turbine temperature reaches 924 ± 5°C. At this point, the temperature limit channel overrides the speed governing of the load sharing channel and fuel flow is reduced to hold the MGT to 924 ± 5°C. The overspeed system provides Np overspeed protection. Fuel flow is shut off by the sequence valve if Np reaches 117 percent. If the ECU fails, it can be overridden by advancing the throttles beyond the normal 120-degree flight position to the 130-degree stop and manually controlling engine trim with the throttles. The ECU can be reset and the ECU LOCKOUT caution light extinguished by moving the throttle to the idle stop position.

FLIGHT CONTROLS

General

25. The 214ST is equipped with dual flight controls. Each control system, except the FBW elevator, uses conventional push-pull control linkages which operate dual, redundant, hydraulic actuators. The FBW elevator is operated by redundant electromechanical actuators. The cockpit controls for the basic flight control system consist of cyclic and collective hand controls, and antitorque control pedals (fig. 14).

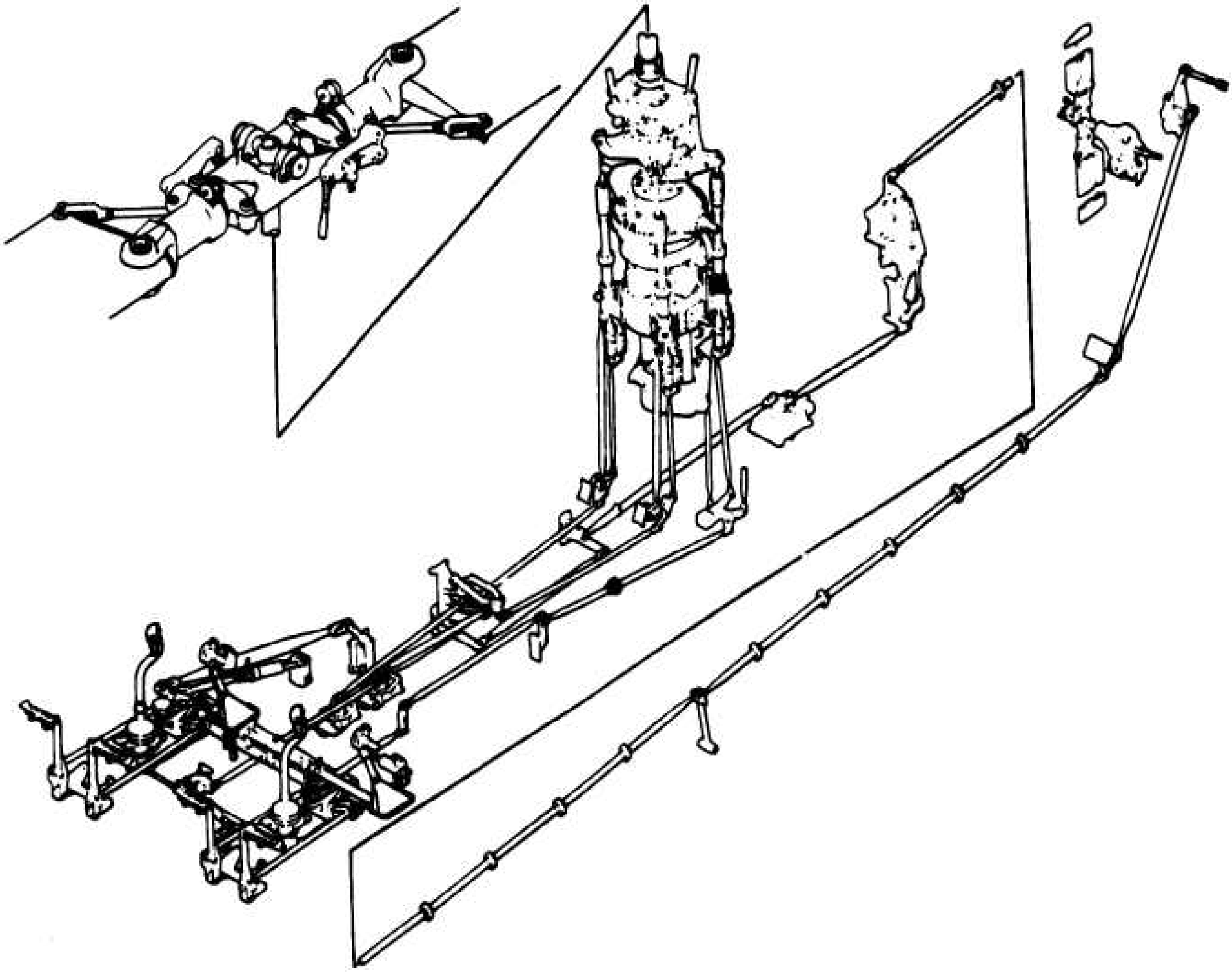


Figure 14. Flight Control Systems

Force Trim System (AARS OFF)

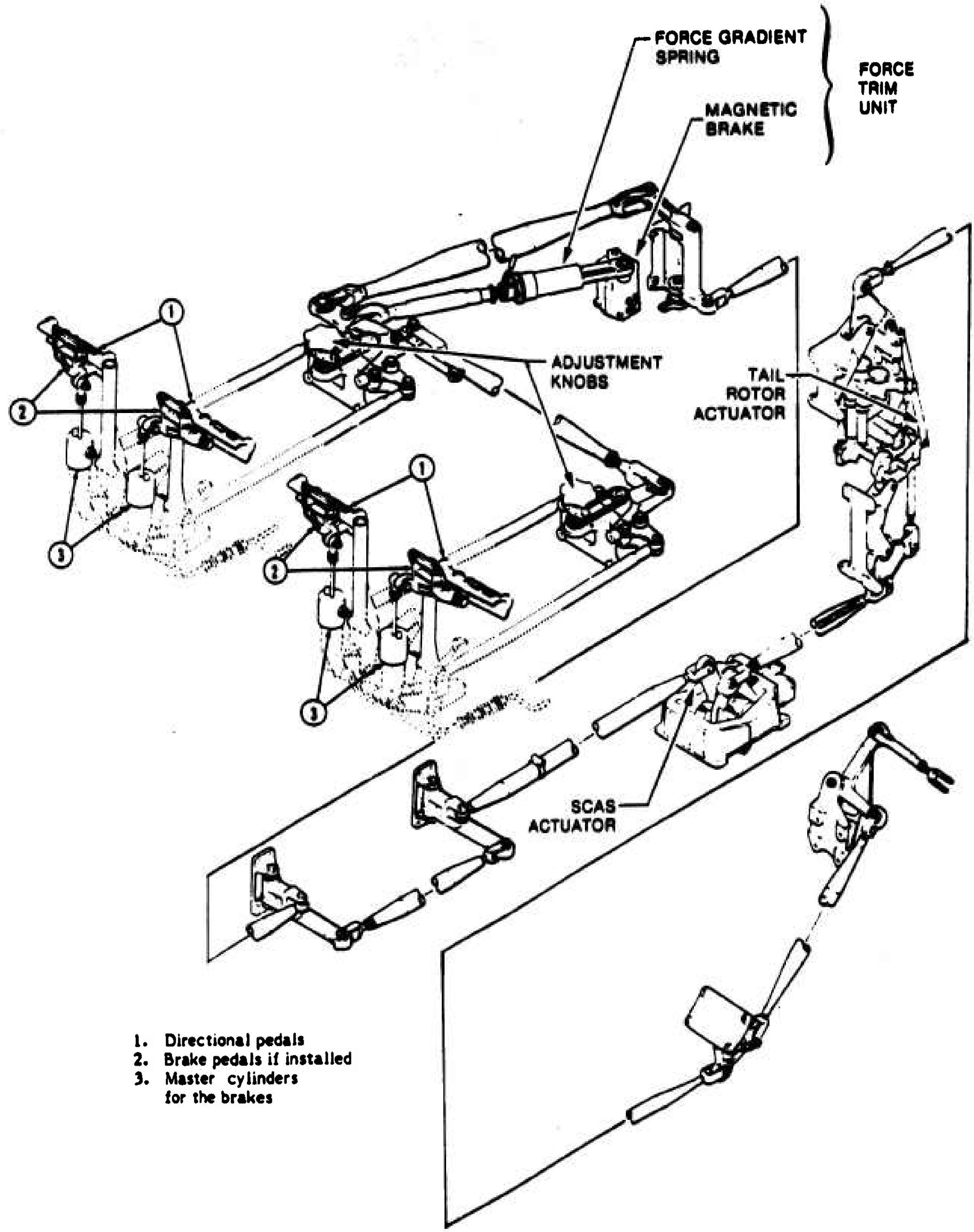
26. The force trim provides the pilots with an artifical "feel" system to generate force cues in the longitudinal, lateral, and directional flight controls. The directional force trim system is similar to that found in the UH-1H and consists of a force gradient spring and a magnetic brake (fig. 15). The directional force trim system is somewhat unusual in that the directional force trim can be released through the use of the yaw trim release switch mounted on the collective control panel (fig. 16). The yaw trim release switch allows the directional trim to be referenced without interrupting the longitudinal or lateral force trim system. The longitudinal and lateral force trim systems consist of force gradient springs and parallel trim actuators (PTA) mounted parallel with the flight controls. The PTA's function as magnetic brakes when the AARS is OFF. All axes of force trim may be referenced simultaneously thru the use of the force trim button mounted on both cyclic control heads. Additionally, the entire force trim systems may be turned OFF or ON with the force trim switch located on the pedestal mounted control panel.

Cyclic Controls

27. Cyclic controls are mounted through the cockpit floor and through separate pitch and roll subsystems transmit control movement to a mixing-lever assembly. From this assembly, separate but identical linkages transmit combined movement to the hydraulic servo actuators. The cyclic control linkage is equipped with force-trim units which act as spring-loaded centering mechanisms for the cyclic controls while providing the pilot with artificial feel. An adjustable friction control (similar to the UH-1H) is installed at the base of the cyclic control, which allows the pilot to set the desired control friction or lock the cyclic control. In addition, SCAS actuators are installed in series in each subsystem to dampen the short term response of the helicopter.

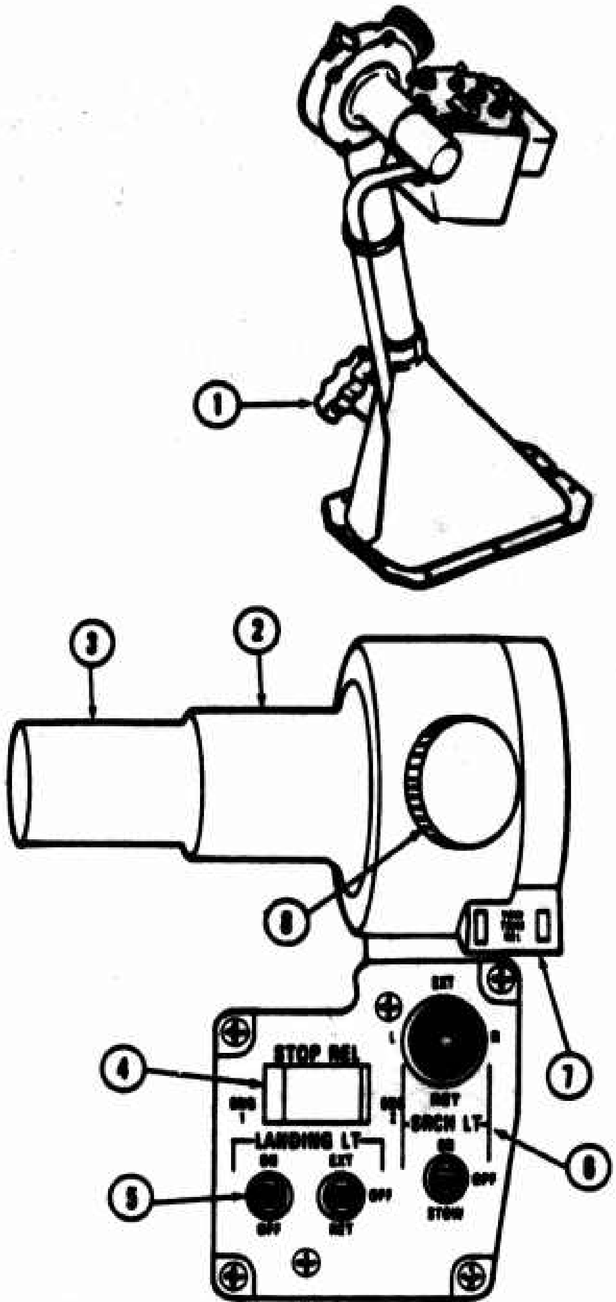
Collective Pitch Control

28. The collective pitch controls are mounted through the cockpit floor and transmit pilot input through a series of push-pull tubes and bell cranks to a dual hydraulic servo-actuator. As with the cyclic actuators, the collective actuator is mounted in the main transmission area below the stationary swashplate. The 214ST collective control design is somewhat unusual in that the twist grip throttles are mounted horizontally (fig. 16). The left throttle controls the Power Available Spindle (PAS) on the #1 engine and the right throttle controls the PAS on the #2 engine. In addition to the normal landing and search light controls, a



1. Directional pedals
2. Brake pedals if installed
3. Master cylinders for the brakes

Figure 15. Directional Control System



Pilot's Collective Lever

1. Collective friction knob
2. No. 2 Engine throttle
3. No. 1 Engine throttle
4. Throttle stop release switch
5. Landing light switches
6. Searchlight switches
7. Yaw trim release switch
8. Throttle friction knob

Figure 16. Pilot's Collective Lever

yaw trim release switch has been mounted on each collective control head. The pilot and copilot's collective are identical in appearance, motion, and feel, except for the addition of a control panel and a throttle friction knob on the pilot's collective. The pilot's collective has a friction control similar to that of the cyclic control stick and can be adjusted.

Tail Rotor Pitch Control

29. The tail rotor pitch control pedals (fig. 15) are conventional in appearance and operation. The pedals are connected to a dual hydraulic servoactuator through a series of push-pull tubes and bell cranks. The pedal position can be adjusted approximately 3 inches with a pedal adjuster located on the cockpit floor just aft of the pedals. The pilot's and copilot's pedals are equipped with toe brakes.

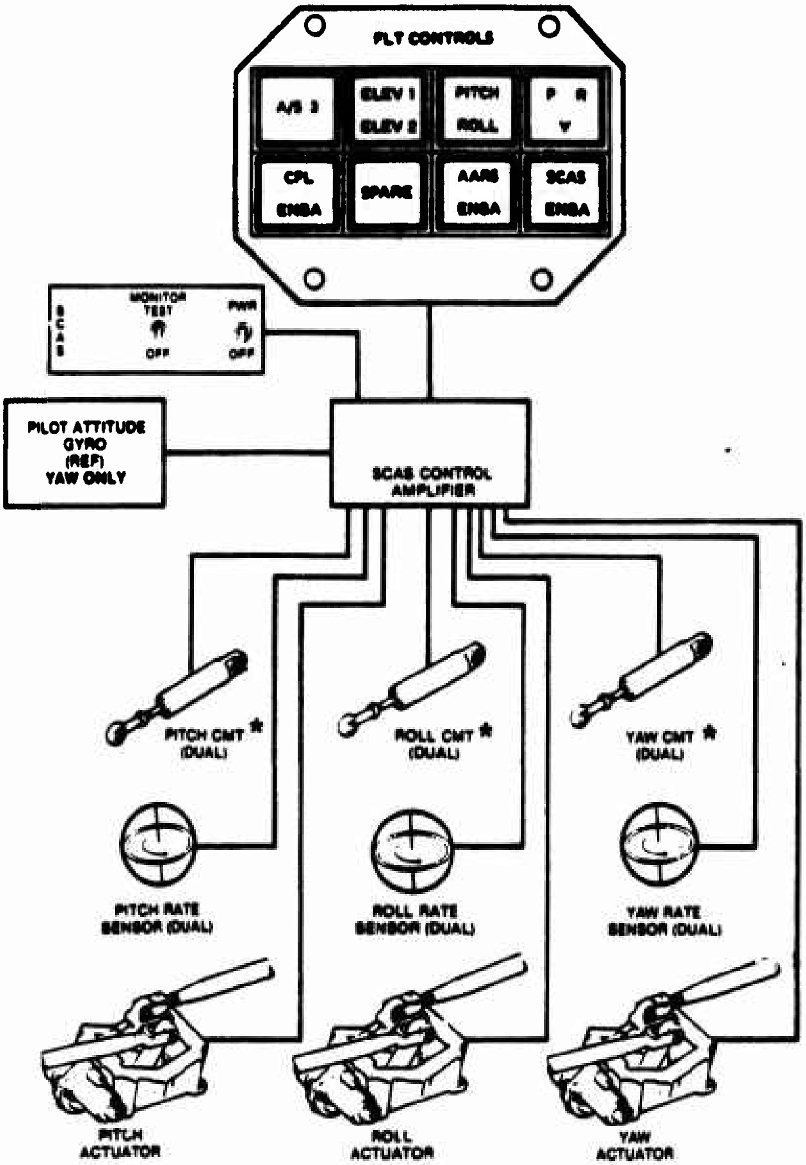
Stability and Control Augmentation System

30. The SCAS is a three-axis (pitch, roll, and yaw) limited authority stabilization system (fig. 17) which use electromechanical actuators. Each axis of the SCAS utilizes signals from rate gyros and control motion transducers to provide drive signals to the actuators. The authority of the SCAS is $\pm 11\%$ in pitch and roll and $\pm 12.5\%$ in yaw. The pilot's roll attitude gyro is utilized by the yaw axis to improve turn coordination (limited to 50% yaw actuator authority). Each axis of the SCAS is independent except for electrical power.

31. Each axis consists of dual electronic channels driving a single actuator. The dual channels provide inherent hardover protection in that both channels must agree on a command before the actuator can respond. When a disagreement between the two channels is detected by a monitor circuit the affected axis is disengaged (fig. 18). When the failed axis is disengaged, the actuator will center (at slower than normal rate) to prevent control offset or reduced control authority. Each axis of the SCAS has the following independent failure detectors:

a) Tach-Command (Compares actuator output to command signal)
b) DC Tracking (Monitors current in the motor control phase winding)
c) Lock Current (Monitors current in actuator lock solenoid)

After a SCAS axis is disengaged by the monitor circuits, the AFCS annunciator panel will display the failed axis and the AFCS caution light will illuminate.



*CMT Control Motion Transducer

Figure 17. Stability and Control Augmentation System

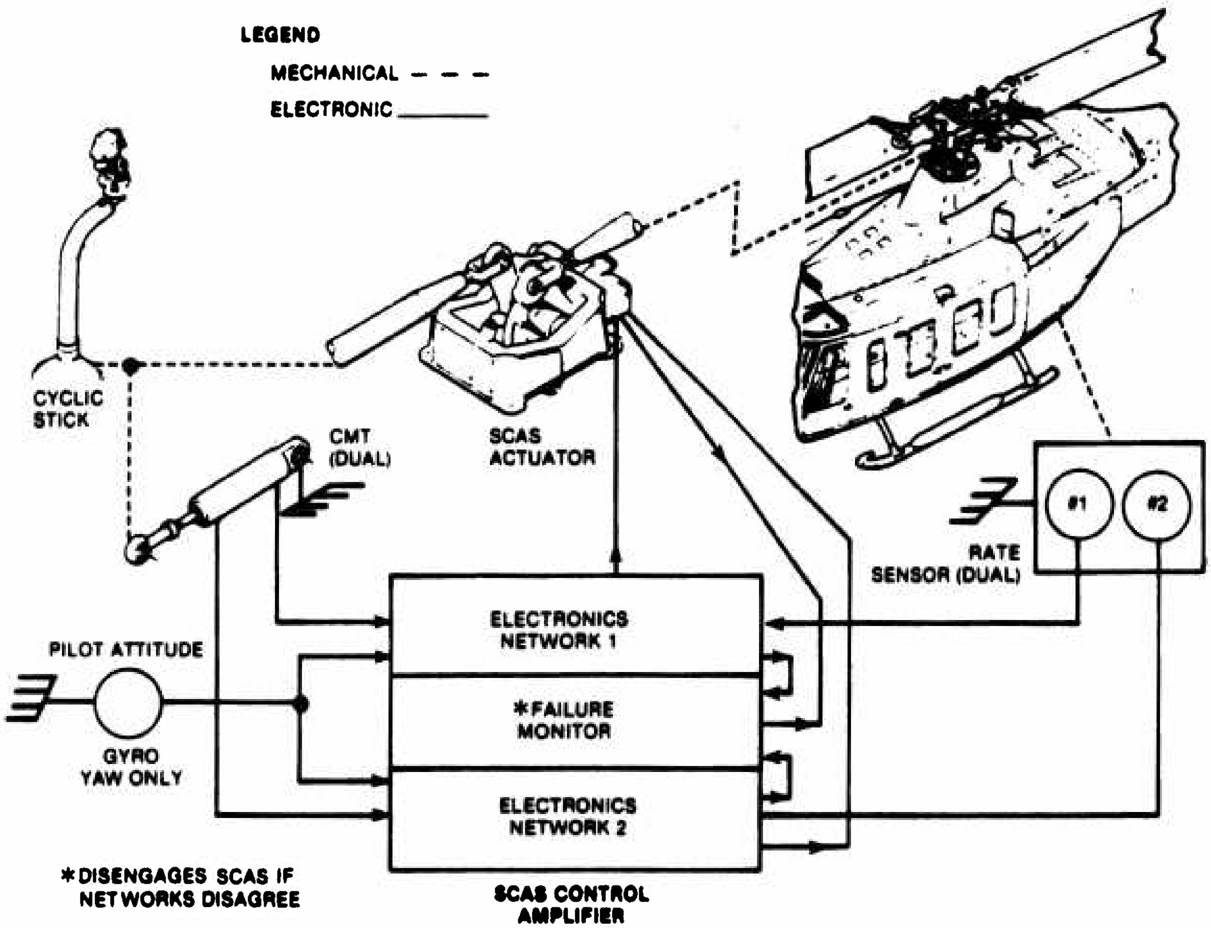


Figure 18. Typical SCAS Channel

32. The SCAS contains a pilot-initiated monitor test function to verify proper operation of the failure monitor circuits. Both positive and negative test signals are used to ensure that the monitors will function for either polarity or error signal. The MONITOR TEST switch for the above mentioned test and the main power switch are located on the SCAS control panel.

Attitude/Altitude Retention System

33. The Attitude/Altitude Retention System (AARS) shown in figure 19 is a two-axis attitude retention system (pitch and roll). The basic AARS has three modes, attitude, heading, and altitude, which can be engaged in any combination except heading and altitude only. If the aircraft is equipped with a Nav Coupler Mode (test aircraft was so configured), the ARRS has several additional coupled navigation modes and a vertical speed mode, in addition to the three basic modes. Both axes of AARS has 100% lateral and longitudinal flight control authority. Each axis of AARS has dual channels of input to drive a single parallel trim actuator (PTA). The dual channels provide hardover protection in that both channels must agree on a command prior to the PTA responding. If a disagreement is detected by either channel, the affected axis will be disengaged automatically. The AARS utilizes signals from the pilot and copilot attitude gyros and altitude controller; airspeed and collective signals from the fly-by-wire elevator; and heading signals from the horizontal situation indicator. These signals are processed in the AARS control amplifier and the processed signals are then provided to the PTA's (same actuators used in the pitch and roll force trim systems) which in turn input, longitudinal and lateral flight control inputs to maintain aircraft attitude.

34. The AARS is independent of the Stability and Control Augmentation System (SCAS), however, it is normally operated in conjunction with SCAS. With either pitch or roll SCAS disengaged, a logic signal is automatically sent to AARS to modify the short-term system gain, maintaining limited aircraft stability in the absence of SCAS inputs.

35. The AARS control panel (fig. 19) is located on the upper right-hand portion of the center pedestal. The control panel provides a means of controlling input power to the AARS amplifier and the flight controls annunciator panel. Two control switches are located on the AARS control panel; The PWR switch, which controls input power to the AARS amplifier, and the MONITOR TEST switch which routes a DC voltage to the AARS amplifier failure monitor when placed in the TEST position.

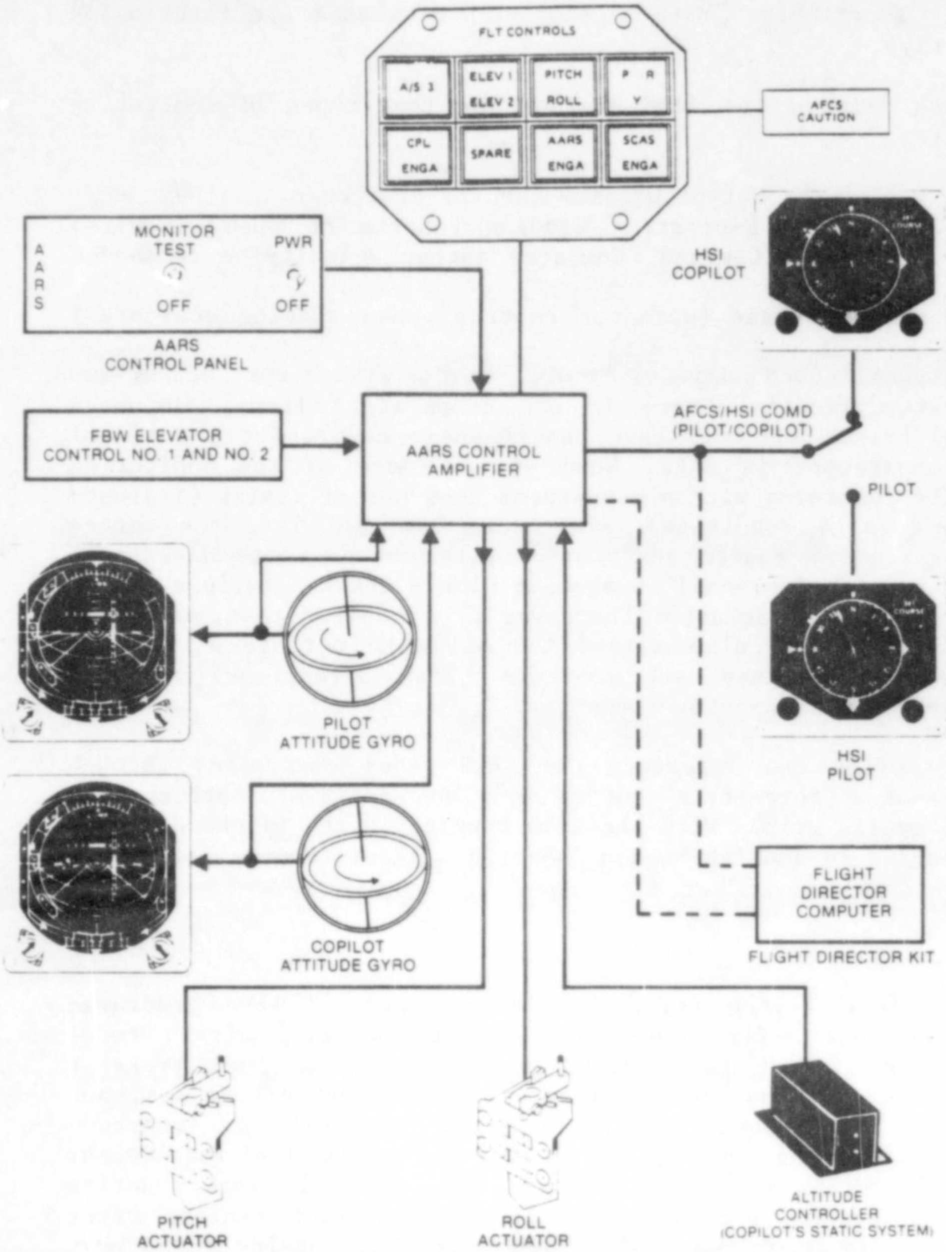


Figure 19. AARS System

36. The engagement/disengagement of the AARS is controlled by two "press to engage/disengage" buttons on the Automatic Flight Control System (AFCS) annunciator panel. The basic attitude hold is controlled by the "AARS ENGA" button and the flight director is coupled through the "CPL ENGA" button. The PITCH and ROLL lights above the "AARS ENGA" button indicate which axis of the AARS is functioning (both lights OUT, both axes are functioning correctly).

37. Each axis of the AARS monitor has four types of monitoring circuits:

a) Tachometer Output Winding (shorts or open circuits)
b) Tachometer Excitation Winding (shorts or open circuits)
c) Tachometer Command (Compares output velocity to commanded velocity)
d) Control Phase (actuator control phase winding monitored)

The output of each monitor circuit combine with either one or two comparator circuits. There is one comparator circuit for each channel (pitch and roll have two channels each), for a total of four comparators in AARS. When one or more of the monitoring circuits combining with a comparator goes out of limits (indicating an invalid condition), the output is routed to the engage logic circuitry causing that axis (pitch or roll) to disengage. Due to the "dual channel" feature a single channel failure cannot result is a AARS actuator "hardover". A monitor test switch on the control panel also allows the pilot to perform a monitor test which exercises all monitors (both directions) and the corresponding comparator circuitry.

38. The pilot can reference the ARRS (when uncoupled) through the use of a "force trim" button or a "Beeper Trim", both mounted on the cyclic grip. With the ARRS coupled to the flight director referencing is done through a "heading bug" or "course selector".

FUEL SYSTEM

39. The fuel system (fig. 20) is composed of seven rupture-resistant fuel cells (neoprene-impregnated fabric) with a total capacity of 440 U S. gallons, fuel lines, two fuel transfer pumps, two prime solenoid valves, two fuel shut-off valves, and a gravity interconnect valve. The seven fuel cells are interconnected in such a way as to provide automatic fuel management (fig. 21) while minimizing changes in center of gravity during operation and to establish two independent fuel systems after the fuel supply is partially depleted. The transfer pumps have

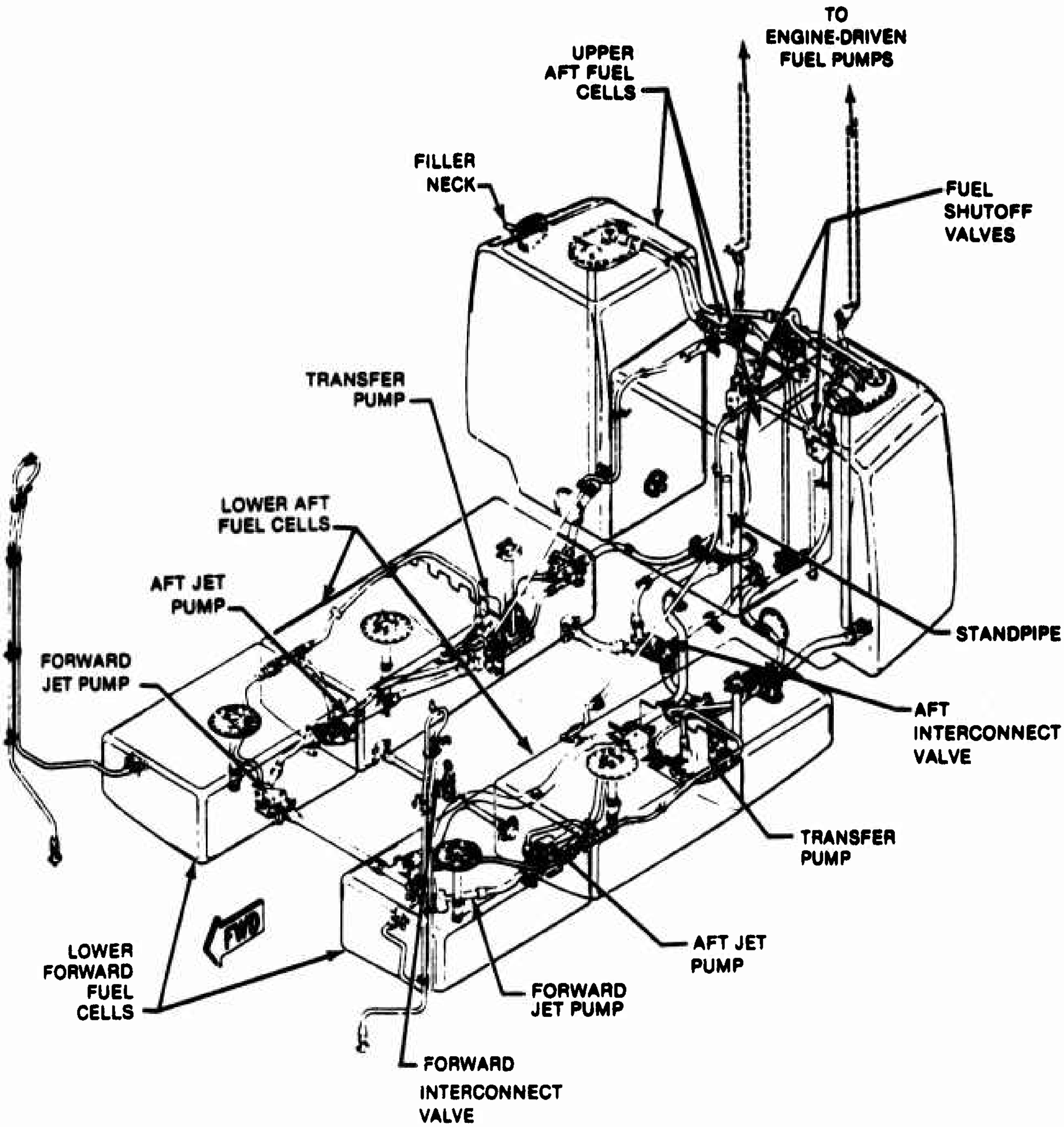


Figure 20. Airframe Fuel System

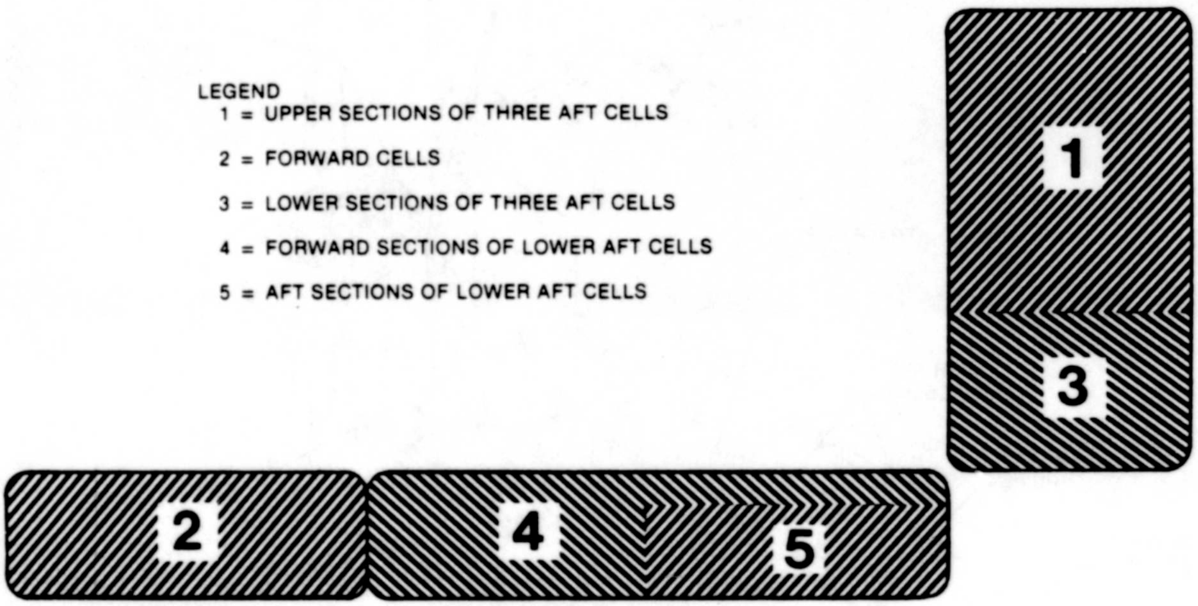


Figure 21. Fuel Burn Sequence

two functions: (1) to provide positive pressure through the prime solenoid valve to the engine-driven boost pump during engine starting and (2) to provide fuel flow for operation of the jet pumps to transfer fuel from the forward cells to the rear compartment of the lower aft cells. Reverse fuel flow is prevented by check valves and one-way flapper valves. Two cockpit-controlled FUEL PRIME switches open the respective engine solenoid to allow pressurized fuel from the transfer pumps to be routed into the engine inlet lines for priming of the engine-driven fuel boost pumps during starting and for high altitude operations (above 15,000 feet mean sea level). Two fuel switches are incorporated to control the operation of each engine fuel shut-off valve. Additionally, arming the fire extinguisher system for the engine will close the fuel shut-off valve, regardless of the cockpit switch position. A gravity interconnect switch is incorporated to allow the fuel to flow by gravity between the fuel cells of the two systems.

40. Fuel quantity data is measured by eight DC activated capacitance probes in the fuel cells and are transmitted electrically to a two-pointer indicator which indicates the quantity of fuel remaining in pounds in the left and right fuel systems. A digital display window, on the lower part of the instrument, shows total fuel remaining in pounds. A yellow FUEL VALVE caution light (one for each engine) illuminates anytime the respective fuel valve is in transit or not in the selected position. A yellow FUEL TRANS caution light (one for each engine) illuminates any time fuel transfer is not taking place. The light is deactivated when no fuel is available for transfer. A yellow FUEL LOW caution light (one for each engine) illuminates when approximately 170 pounds of fuel is remaining in the respective lower aft cell. A single yellow caution light illuminates when the cell interconnect valves are either in transit or not in the selected position. A yellow FUEL PRESS caution light illuminates when fuel pressure downstream of the engine-driven boost pumps falls below normal limits. A yellow FUEL FILTER caution light (one for each engine) illuminates when the associated engine fuel filter is becoming clogged. Two range extension kits are available for the aircraft: (1) two 87.5 U.S. gallon tanks and (2) two 28.0 U.S. gallon tanks. The range extension kits may be used separately or together.

ELECTRICAL SYSTEM

41. The 214ST uses DC as primary electrical power (fig. 22) which is supplied by two independent primary DC power systems (No. 1 and No. 2 main generators). Secondary DC power is provided by

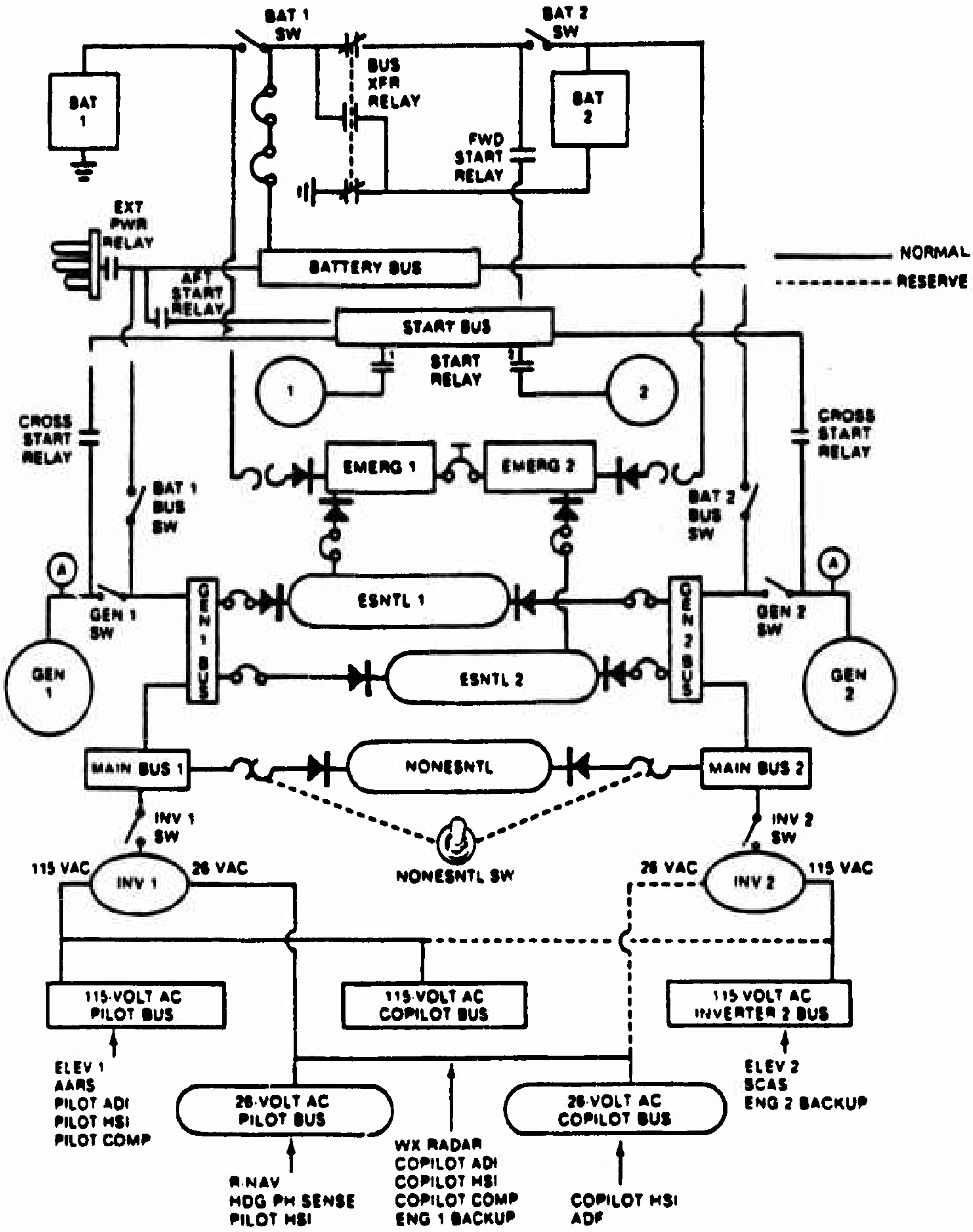


Figure 22. Electrical Power Distribution

two 24-volt nickel-cadmium batteries. The 214ST also has provision for an external DC power supply for system operation on the ground and to serve as a secondary source of power for engine start (when battery charge is low). Primary AC power is obtained from two solid state inverters.

42. Primary DC power (28-volt) is supplied by two 30-volt, 400-ampere, self-cooled generators which are driven by the combining gearbox. The generators are designed to equally share the electrical load during normal operation, however, either generator is capable of carrying the entire load in the event of a single generator failure. Each generator's output is monitored by a generator control unit which provides generator output voltage regulation and protection from overvoltage, undervoltage, ground faults, and system isolation through control of the respective battery bus relay.

43. Secondary DC power is provided by two 24-volt, 34.5-ampere, 20-cell batteries which are monitored, controlled, and charged by charging monitors. The batteries provide engine starting power (primary source) and are used as an emergency power source in the event of a dual generator failure. Following a dual generator failure, the batteries are capable of carrying normal aircraft electrical loads (excluding the nonessential bus) for approximately 30 minutes (batteries fully charged at time of failure). Separate red BATT HOT warning lights illuminate when either battery temperature is 68°C (155°F) or above. Separate red BATT SYS warning lights illuminate when: (1) battery temperature has exceeded 145°F, (2) there is a voltage imbalance between battery cells, or (3) a charger/monitor failure has occurred.

44. An external DC power connector is located on the left side of the helicopter and accepts a ground source of 28 volts DC with a capability of 600 to 1300 amperes. A yellow EXT PWR caution light illuminates when the external power receptacle door is opened (it does not indicate when external power is applied).

45. Primary AC power is supplied by two identical 800-ampere, 115/26-volt, 400-Hz, single-phase, solid-state inverters which are powered by 28 volts DC. During normal operations, inverter No.1 provides power to the pilot and copilot 115 and 26-volt buses. Inverter No. 2 provides 115 volts to the No. 2 inverter bus and back-up power to the other 115 and 26-volt buses. The illumination of the INV 1 or INV 2 yellow caution light indicates that the respective inverter has failed (115 volt output). Illumination of the 26 VAC yellow caution light indicates that the 26-volt buses are not powered. Loss of 26-volt power will

disable both Horizontal Situation Indicators (HSI), the Automatic Direction Finder (ADF), the Area Navigation (RNAV) and the SCAS heading reference.

FLIGHT DIRECTOR SYSTEM

46. The flight director system (fig. 23) incorporated in the 214ST provides the pilot with pitch and roll commands using the steering bars in the Attitude Director Indicator (ADI) and supplies the same commands to the AARS for coupled automatic flight control (when desired). The system includes two mode selectors (one at each pilot station) and a flight director computer. The flight director system has three basic operational modes: pitch, navigation, and standby.

a. Pitch Modes. The pitch modes can be selected by depressing any of three switchlights in the top row of the mode selector (fig. 24). These modes are ALT (altitude hold), IAS (indicator airspeed hold), and VS (vertical speed hold). Any one of these modes may be selected simultaneously with the navigation modes.

b. Navigation Modes. The navigation modes can be selected by depressing any of the six switchlights in the middle two rows of the mode selector. These modes are: HDG (heading select - for manual heading control utilizing the heading select bug on the Horizontal Situation Indicator (HSI); NAV (navigation - for automatic maintenance of the enroute course selected in the HSI course window); ILS (instrument landing system - automatic approach course and glideslope information); BC (backcourse - similar to ILS mode except that the deviation and course signals are reversed); VOR APR (VOR approach - for automatic approach course information only) and GA (go-around used during missed approach to command wings level and climb rate of approximately 750 feet per minute at 65 KIAS).

c. SBY Mode. Standby mode can be selected by depressing the switchlight in the bottom row of the mode selector. Standby mode disables all flight director modes and uncouples the AARS (when coupled). The three modes of AARS (attitude hold, heading hold and altitude hold) are coupled to the flight director system when the appropriate flight director modes are selected ON when the AARS is also selected ON.

Attitude Director Indicator

47. The ADI is a remote indicator for the attitude gyro which serves as the pilot's primary attitude reference. Lateral and

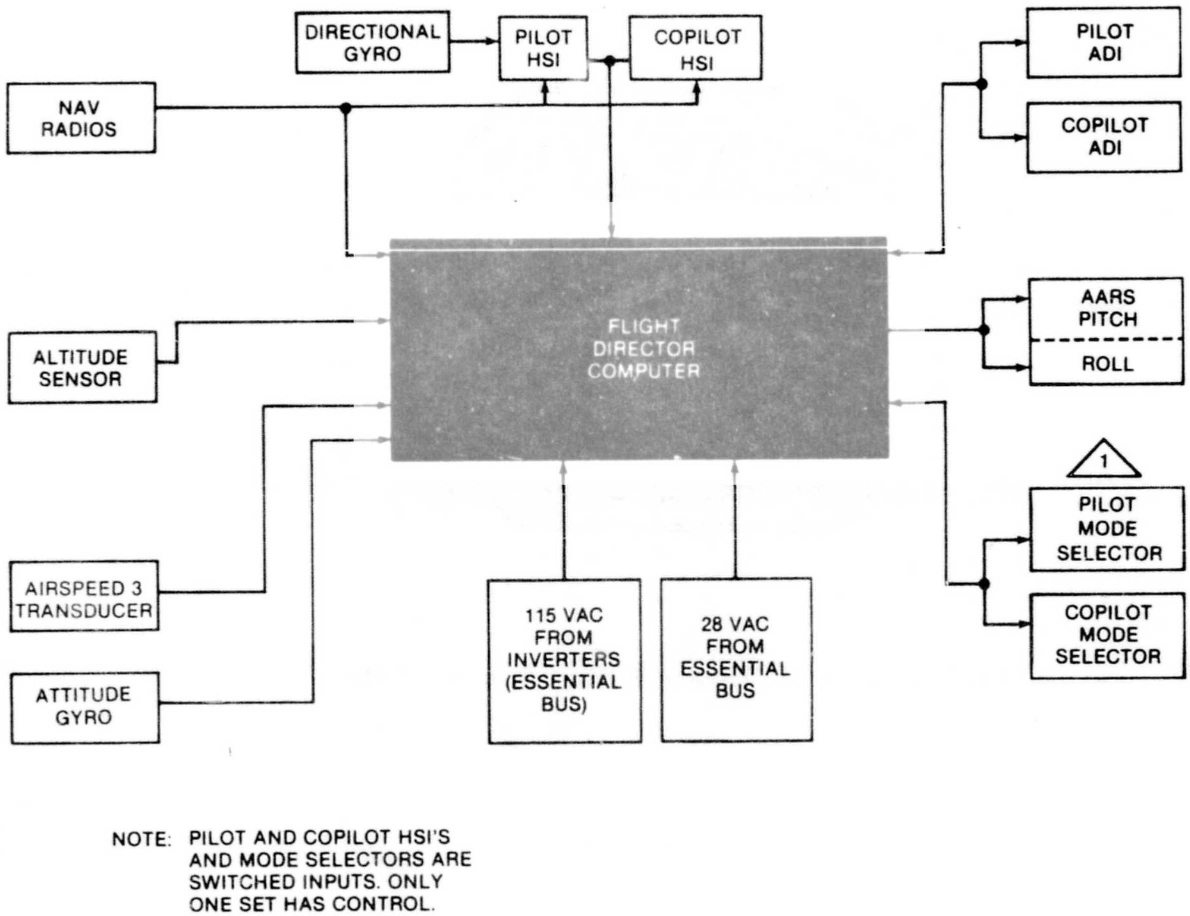


Figure 23. Flight Detector System Block Diagram

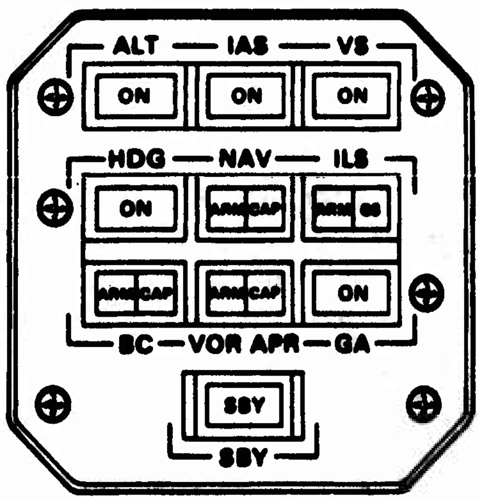


Figure 24. Flight Director Mode Selector Panel

vertical commands are displayed by the steering bars. The ADI has a fixed roll scale and a movable attitude sphere. Steering bars are biased out of view when the flight director system is OFF. The rate-of-turn pointer indicates in what direction and at what rate the helicopter is turning. The inclinometer provides the pilot with a conventional display of slip or skid. The glidescope pointer indicates the helicopter's relative position in relation to a glideslope beam when ILS mode is selected.

Horizontal Situation Indicator

48. The HSI serves as the pilot's primary source of navigation information. The rotating compass card provides the helicopter's magnetic heading which is displayed under the upper lubber line. The white heading bug is used to select the desired heading and to supply heading data to the flight director system. The No. 1 bearing pointer indicates the bearing to the selected VOR or waypoint (RNAV). The No. 2 bearing pointer indicates the bearing to the selected ADF station (no VOR capability). The COURSE SET indicator displays the selected course and the course knob positions the course pointer appropriately. The course deviation indicator shows the position of the selected course relative to the helicopter actual position. The TO - FROM indicators tell the pilot if the selected course will carry him to or from the VOR/waypoint station. The glideslope pointer shows the position of the glideslope beam relative to the helicopter when an ILS radio frequency is selected.

Area Navigation System

49. Area navigation (RNAV) provides a method of point-to-point navigation along an air route. The course is defined by waypoints (electronically defined positions using a single bearing and distance from a VOR/DME station) that behave as fixed VOR/DME stations. RNAV may be used to fly directly to destinations or conventionally to fly along existing airways. Course deviation data is continually displayed on the pilot's and copilot's HSI's. RNAV mode is selected and programmed via the KNS 81 radio mounted in the center console.

BASIC AIRCRAFT INFORMATION

50. Principal dimensions and general data of the 214ST helicopter are as follows:

Airframe

Length:
- Maximum (rotor blades turning)..................62 ft 2.3 in.
- Fuselage (nose to vertical tail)................50 ft 1.4 in.
- Main rotor to tail rotor clearance..............6.03 in.

Width:
- Main rotor blades turning........................52 ft 0 in.
- Main landing gear (outside of tire to outside of tire-full fuel, no load)........10 ft 9.4 in.

Height:
- Maximum (tail rotor blades turning).............15 ft 10.6 in.
- Main rotor clearance (ground to tip, rotor static against stops)....................8 ft 3.7 in.
- Tail rotor clearance (ground to tip, rotor turning)................................6 ft 2.4 in.

Horizontal stabilizer:
- Span...113.3 in.
- Chord (at root/tip)................................30.6 in.
- Area (total, including tailboom)..................21.4 sq ft
- Airfoil..Inverted Clark Y-type
- Range of travel (reference to fuselage water line)..................................21 deg

Vertical tail fin:
- Effective span.....................................72 in.
- Chord (average)....................................43.4 in.
- Area...21.7 sq ft
- Airfoil..BHT Cambered

Gross Weight:
- Maximum gross weight...............................17,500 pounds
- Minimum weight.....................................9,200 pounds
- Fuel capacity (no auxiliary tanks)................440 U.S. Gallons

Main Rotor
- Number of blades...................................2
- Diameter...52 ft
- Disk area..2123.7 sq ft
- Disk loading (17,500 pounds gross weight)..........8.24 psf
- Blade chord..33 in.
- Blade area (both blades)...........................143.0 sq ft
- Solidity (blade area/disc area)....................0.067
- Airfoil section....................................Wortmann 098 modified

Blade twist.......................................-19.2 min/ft
Hub precone angle.................................2.5 deg
Flapping angle....................................±10 deg
Tipspeed at a hover (287 rotor rpm, 100%)..........781 fps

Tail Rotor

Number of blades..................................2
Diameter..9 ft 8 in.
Disc area...73.4 sq ft
Blade chord.......................................14 in.
Blade area (both blades)..........................11.3 sq ft
Solidity..0.154
Airfoil section...................................FX69-HL-083
Blade twist.......................................0
Tipspeed at a hover (1455 rotor rpm, 100%)........736 fps

Main Rotor RPM

	Power ON	Power OFF
Minimum	99% / 281 rpm	90% / 272 rpm
Normal	100% / 287 rpm	100% / 287 rpm
Maximum	100% / 287 rpm	105% / 301 rpm
Design-maximum	100% / 287 rpm	110% / 317 rpm
Design-minimum	96% / 276 rpm	90% / 258 rpm

Tail Rotor RPM

	Power ON	Power OFF
Minimum	99% / 1424 rpm	90% / 1379 rpm
Normal	100% / 1455 rpm	100% / 1455 rpm
Maximum	100% / 1455 rpm	105% / 1526 rpm
Design-maximum	100% / 1455 rpm	110% / 1607 rpm
Design-minimum	96% / 1399 rpm	90% / 1309 rpm

Drive System (see fig. 25)

Engine to combining gearbox ratio.....................1:1
  Direction of shaft rotation — CW looking forward
  Engine speed at 100% — 21,000 rpm
Engine to main transmission ratio.....................3.0325:1
  Direction of shaft rotation — CW looking forward
  Shaft speed at 100% — 6,925 rpm
Engine to main rotor ratio............................73.1523:1
  Direction of main rotor rotation — CCW looking down
  Main rotor speed at 100% — 287 rpm
Engine to 42-degree gearbox ratio.....................4.3999:1
  Direction of shaft rotation — CCW looking forward
  Shaft speed at 100% — 4773 rpm

Engine to 90-degree gearbox ratio........................4.4956:1

Direction of shaft rotation — CCW looking forward

Shaft speed at 100% — 4671 rpm

Engine to tail rotor ratio..............................14.4331:1

Main rotor to tail rotor ratio..........................1:5.0604

Direction of rotation — Lower blade advancing

Tail rotor speed at 100% — 1455 rpm

Rotational Speed Signals at 100%

	RPM	Frequency, Hz
Main rotor, N_R	287.0	3447.8
Power turbine, N_p	21,000	1399.9
Gas producer, N_g	44,700	2135.7

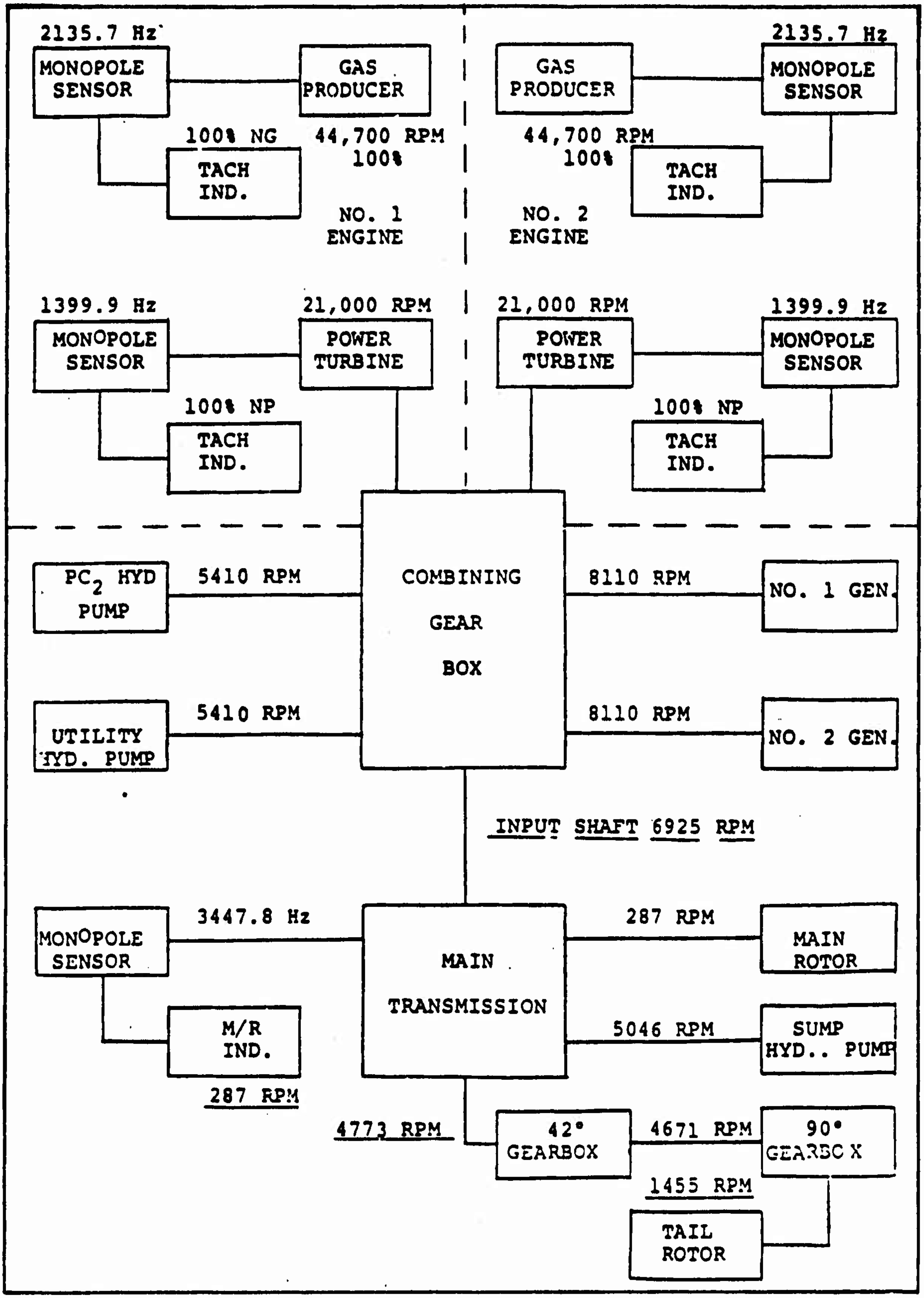


Figure 25. Powertrain Speed Schematic

APPENDIX C. INSTRUMENTATION

GENERAL

1. The test instrumentation was installed, calibrated and maintained by Bell Helicopter Textron, Inc. personnel. The airborne data acquisition system utilized pulse code modulation (PCM) encoding with provisions for telemetry transmission of various parameters. A swiveling pitot-static tube (fuselage station -56), an angle of attack vane (fuselage station -35.5), a sideslip vane (fuselage station -39.5), and an outside air temperature probe (fuselage station -19) were installed on a test boom (butt line 0 and waterline 19.6) mounted on the nose of the aircraft. Equipment required only for specific tests was installed when needed and is discussed in the section on special equipment. Pilot's qualitative comments were recorded on a voice activated tape recorder installed in the cockpit. Data was obtained from calibrated instrumentation and displayed or recorded as indicated below.

Pilot Panel

Airspeed(boom)
Airspeed(ship)
Pressure altitude(boom)
Pressure altitude(ship)
Angle of sideslip
Rate of climb (IVSI)*
Main rotor blade flapping

Copilot Panel

Normal acceleration
Main rotor speed
Event switch
Run Number
Time code display
Fuel used
Instrumentation controls

* Not calibrated

2. PCM data parameters recorded on board the aircraft include the following:

Airspeed (boom)
Airspeed (ship)
Pressure altitude (boom)
Radar altimeter (ship)
Outside air temperature(sensitive)

Angle of sideslip
Angle of attack
Main rotor speed
Fuel used (2)
Fuel flow (2)
Engine torque (2)
Gas turbine speed (2)
Power turbine speed (2)
Measured gas temperature (2)
Longitudinal control position (copilot)
Lateral control position (copilot)
Collective control position (copilot)
Directional control position (pilot)
Elevator position (ship)
Pitch attitude
Roll attitude
Yaw attitude
Pitch rate
Roll rate
Yaw rate
CG normal acceleration
Main rotor flapping angle
Event
Record number
Time of day

AIRSPEED CALIBRATION

3. The standard ship's and test boom airspeed system were calibrated during level flight, climbs, and descents. A calibrated pace vehicle (T-28) and trailing bomb was used to determine position error in level flight. In climbs and decents, only calibrated trailing bombs were utilized. The position error of the boom airspeed system is presented in figures 1 and 2.

SPECIAL EQUIPMENT

Weather Station

4. A portable weather station (provided by USAAEFA), consisting of an anemometer, sensitive temperature gage, and barometer, was used to record wind speed, wind direction, ambient temperature, and pressure altitude at selected heights up to 50 feet above ground level.

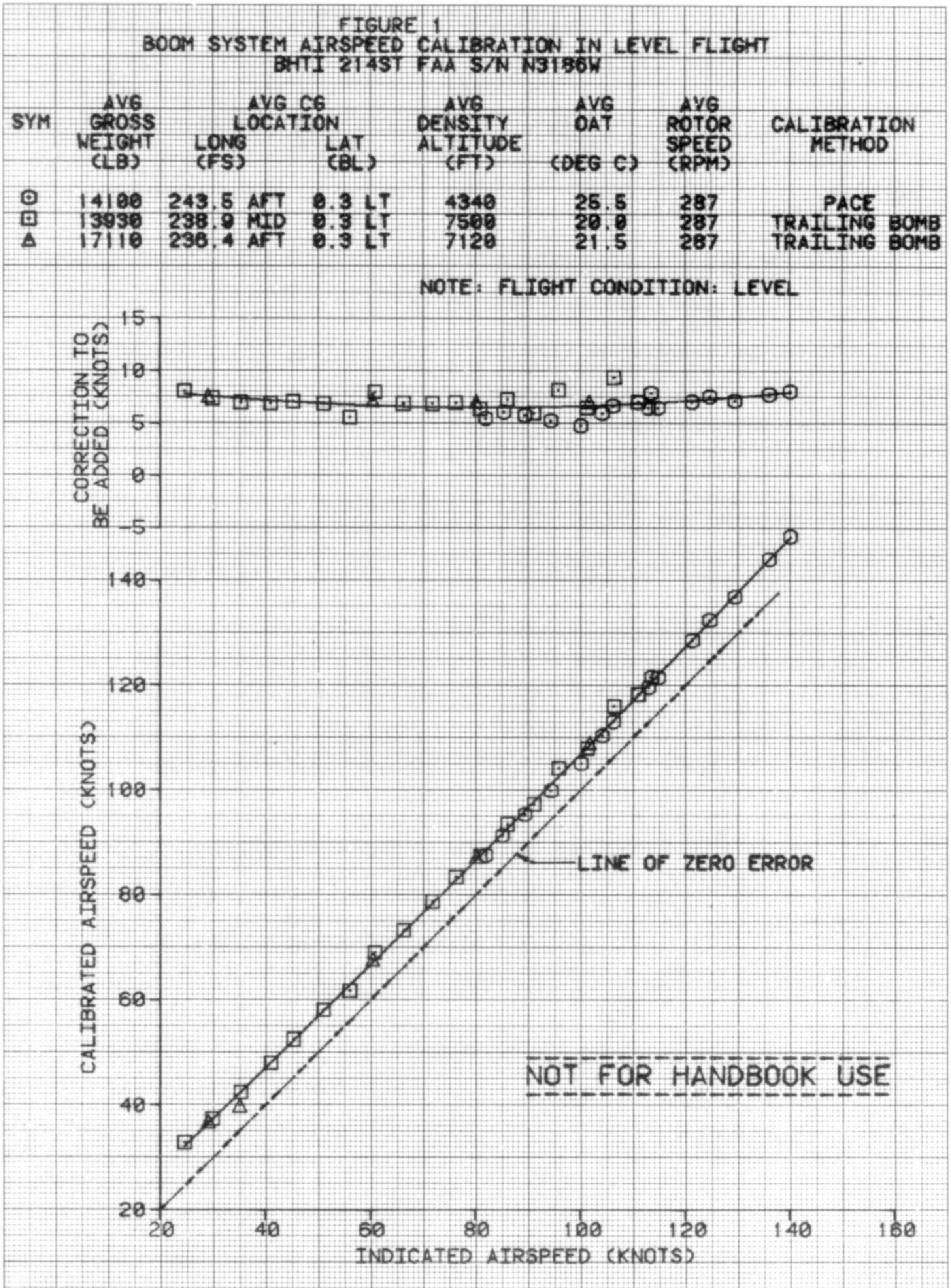
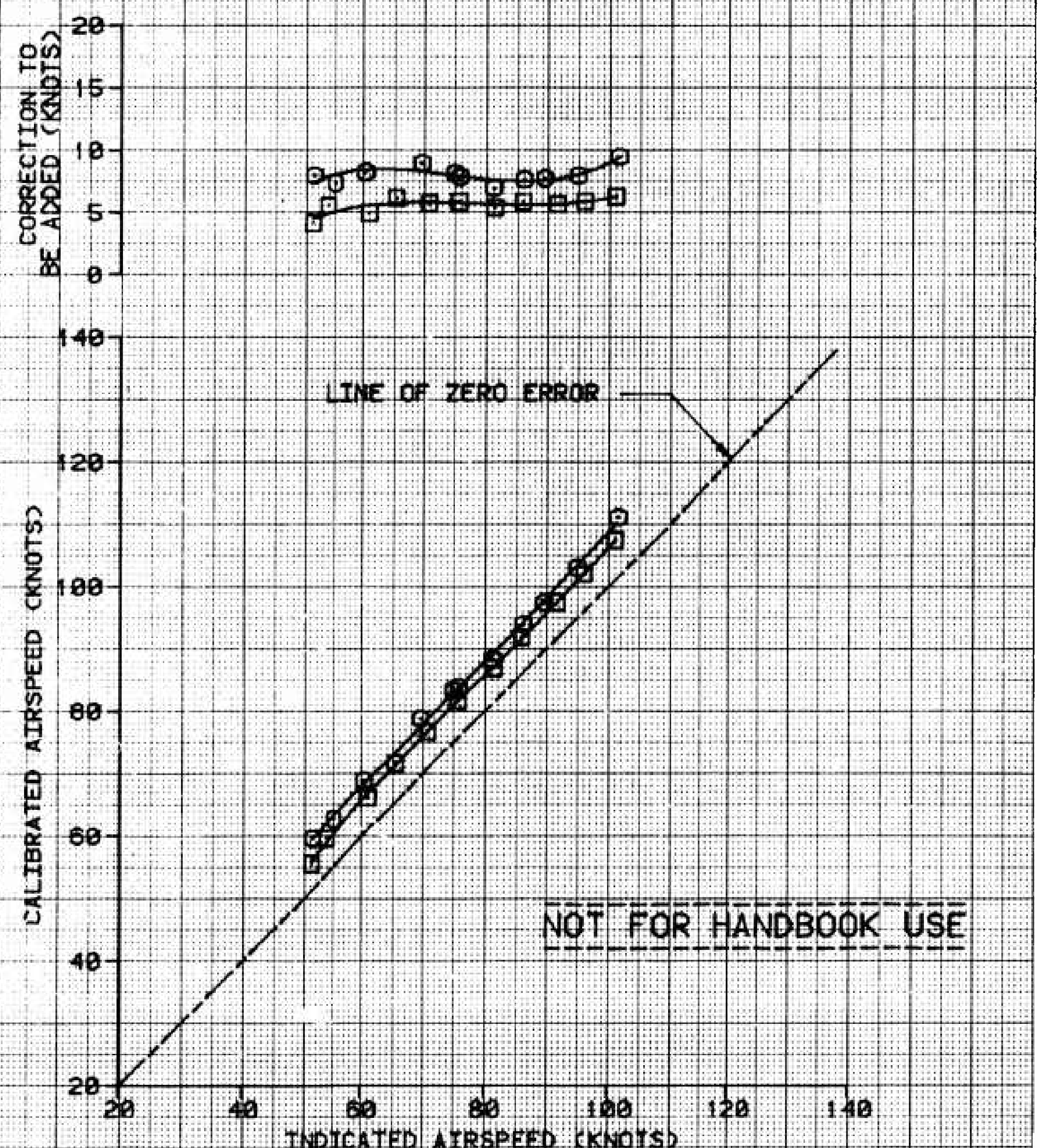


FIGURE 1
BOOM SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT
BHTI 214ST FAA S/N N3186W
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
CALIBRATION METHOD
14100 243.5 AFT 0.3 LT 4340 25.5 287 PACE
13930 238.9 MID 0.3 LT 7500 20.0 287 TRAILING BOMB
17110 236.4 AFT 0.3 LT 7120 21.5 287 TRAILING BOMB
NOTE: FLIGHT CONDITION: LEVEL
CORRECTION TO BE ADDED (KNOTS)
CALIBRATED AIRSPEED (KNOTS)
LINE OF ZERO ERROR
NOT FOR HANDBOOK USE
INDICATED AIRSPEED (KNOTS)

FIGURE 2
BOOM SYSTEM AIRSPEED CALIBRATION IN CLIMBS AND DESCENTS
BHTI 214ST FAA S/N N3188W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	17380	236.7 AFT	0.3 LT	8380	20.0	287	CLIMB
□	17300	236.5 AFT	0.3 LT	8300	20.0	287	AUTOROTATION

NOTES: 1. TRAILING BOMB METHOD
2. CLIMBS CONDUCTED USING MAX CONTINUOUS POWER



Camera

5. Video, 35MM, and 120MM cameras (provided by USAAEFA) were used for documentation purposes, however, no data parameters were recorded with this equipment.

Load Cell

6. The ship's load cell (calibrated with 5 known weights) incorporated with the ship's cargo hook was used during tethered hover tests. A ship's indicator provided the copilot with cable tension and outside observers were used to maintain vertical alignment of the cable.

Ground Pace Vechicles

7. Pace vehicles (provided by USAAEFA), with "fifth wheels" (calibrated ground speed devices) installed, were used to establish precise airspeed during low airspeed handling qualities test.

Control Fixtures

8. Control fixtures (provided by USAAEFA) for longitudinal and lateral cyclic controls with associated hardpoints installed in the copilot station were used during controllability and dynamic stability testing.

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. Performance data were obtained using basic methods described in Army Materiel Command Pamphlet (AMCP) 706-204 (ref 9, app A). Performance testing was conducted in coordinated (ball-centered) flight. Handling qualities were evaluated using standard test methods described in Naval Air Test Center Flight Test Manual FTM No. 101, (ref 10).

AIRCRAFT RIGGING

2. A flight control rigging check (in accordance with maintenance manuals, ref 14, app A) was performed on the main and tail rotors to insure representative handling qualities and compliance with established limits. The Fly-By-Wire (FBW) elevator system was only checked to ensure that maximum deflections were within limits. The flight control blade angles and FBW positions are listed in table 1. BHTI has no "engineering" checks for the rigging of the FBW elevator systems. Several attempts were made to document the scheduling of the FBW elevator. However, all were unsucessful due to the various combinations of parameters which affect FBW elevator programming. Documentation of stabilator scheduling was not available from BHTI.

AIRCRAFT WEIGHT AND BALANCE

3. The aircraft was weighed in the instrumented configuration with normally serviced oil reservoirs and fuel drained (except trapped fuel) prior to the start of the program. The initial weight of the aircraft was 10,342 pounds with a longitudinal center of gravity (cg) located at FS 247.6 and a lateral cg located at Butt Line 0.43 left. The aircraft's fuel gauge was calibrated using graduated fuel cans. Additionally, a correction for the calibrated cockpit fuel totalizer was determined. The fuel weight for each test flight was determined prior to engine start and after engine shutdown by using the fuel totalizer to determine volume and a hydrometer to determine the specific weight of the fuel.

PERFORMANCE

General

4. Helicopter performance was generalized through the use of non-dimensional coefficients using the 1962 US Standard Atmosphere as follows:

Table 1. Flight Control Rigging

Main Rotor	Angle (deg)	Ideal Rigging/Tolerance[1] (deg)
Longitudinal Cyclic		
Full forward	14° 22'	14° to 14° 30'
Full aft	12° 16'	-----
Total Travel	26° 38'	27° ±1°
Lateral Cyclic		
Full right	10° 47'	-----
Full left	10° 55'	10° 30' to 11°
Total Travel	21° 42'	20° ±1°
Collective		
Low blade angle	7° 56'	7° ±15' zero[2]
High blade angle	25° 26'	-----
Total Travel	17° 30'	18° ±1
Tail Rotor		
Pedal full right	-8° 46'	-----
Pedal full left	26° 44'	26° 30' +15'
Total Travel	35° 30'	36° ±1°
Fly-by-wire elevator		
Leading edge down (electrical limit)	-11° 48'	-11° 30' ±30'
Stow position	-2° 11'	-2° ±15'
Leading edge up (electrical limit)	9° 36'	10° +30'
Total Travel	21° 24'	21° 30' ±1°

NOTES:

[1] Values listed in this column were obtained from BHTI and represent the prototype 214ST helicopters.

[2] This value is used as the starting point for proper collective rigging. The collective low blade angle is further adjusted by an in flight autorotation check. (BHT-214ST-MM, Maintenance Manual, 62-00-04, page 13/14)

a. Coefficient of Power (C_P):

$$C_P = \frac{SHP\ (550)}{\rho A(\Omega R)^3} \tag{1}$$

b. Coefficient of Thrust (C_T):

$$C_T = \frac{GW}{\rho A(\Omega R)^2} \tag{2}$$

c. Advance Ratio (μ):

$$\mu = \frac{V_T\ (1.6878)}{\Omega R} \tag{3}$$

Where:

SHP = Engine output shaft horsepower (both)

ρ = Ambient air density ($lb\text{-}sec^2/ft^4$) = $\rho_o \sigma$

ρ_o = 0.0023769 ($lb\text{-}sec^2/ft^4$)

$\sigma = \delta/\theta$

$\delta = [1. - (6.8755856\ E\text{-}06)xH_p]^{5.255863}$

$\theta = \dfrac{OAT + 273.15}{288.15}$

H_p = Pressure altitude (feet)

A = Main rotor disc area = 2123.7 ft^2

Ω = Main rotor angular velocity (radians/sec)

R = Main rotor radius = 26.0 ft

GW = Gross weight (lb)

V_T = True airspeed (kt) = $\dfrac{V_E}{1.6878\sqrt{\rho/\rho_o}}$

1.6878 = Conversion factor (ft/sec-kt)

V_E = Equivalent airspeed (ft/sec) =

$$\left\{ \frac{7(70.7262\ P_a)}{\rho_o} \left[\left(\frac{Q_c}{P_a} + 1 \right)^{2/7} - 1 \right] \right\}^{1/2}$$

70.7262 = Conversion factor (lb/ft^2-in.-Hg)

Q_c = Dynamic pressure (in.-Hg)

P_a = Ambient air pressure (in.-Hg)

At the normal operating rotor speed of 287.0 (100%), the following constants may be used to calculate C_P and C_T:

$\Omega R = 781.419$

$A(\Omega R)^2 = 1{,}296{,}764{,}944.$

$A(\Omega R)^3 = 1.0133170 \times 10^{12}$

5. The engine output shaft torque was determined by use of the engine torque sensor. The power turbine shaft twists as a function of applied torque. A concentric reference shaft is secured by a pin at the front end of the power turbine drive shaft and is free to rotate relative to the power turbine drive shaft at the rear end. The relative rotation is due to transmitted torque, and the resulting phase angle between the reference teeth on the two shafts is picked up by the torque sensor. The torque sensors for engines installed in the aircraft during this evaluation were not calibrated. Engine torque sensor data obtained from normal engine test cell qualification for each engine by GE were used to correct the torque sensor data in this report. These data and the corrections used are presented in figures 1 and 2. The output from the engine sensor was recorded on the onboard data recording system. The output SHP was determined from the engine's output shaft torque and rotational speed by the following equation.

$$SHP = \frac{Q(N_P)}{5252.113} \tag{4}$$

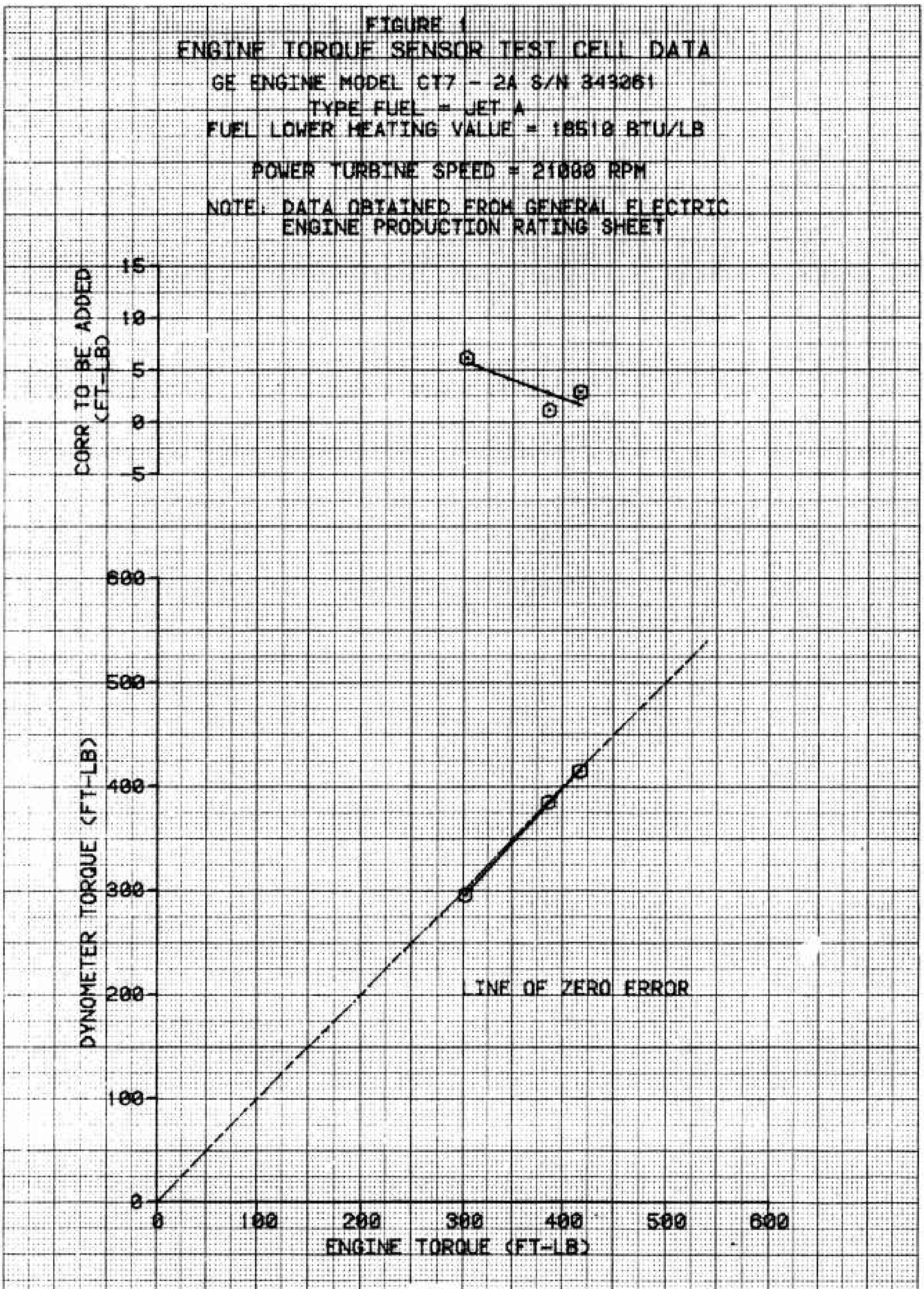


FIGURE 1
ENGINE TORQUE SENSOR TEST CELL DATA
GE ENGINE MODEL CT7 - 2A S/N 343261
TYPE FUEL = JET A
FUEL LOWER HEATING VALUE = 18510 BTU/LB
POWER TURBINE SPEED = 21000 RPM
NOTE: DATA OBTAINED FROM GENERAL ELECTRIC
ENGINE PRODUCTION RATING SHEET
CORR TO BE ADDED (FT-LB)
15
10
5
0
-5
DYNOMETER TORQUE (FT-LB)
600
500
400
300
200
100
0
LINE OF ZERO ERROR
0
100
200
300
400
500
600
ENGINE TORQUE (FT-LB)

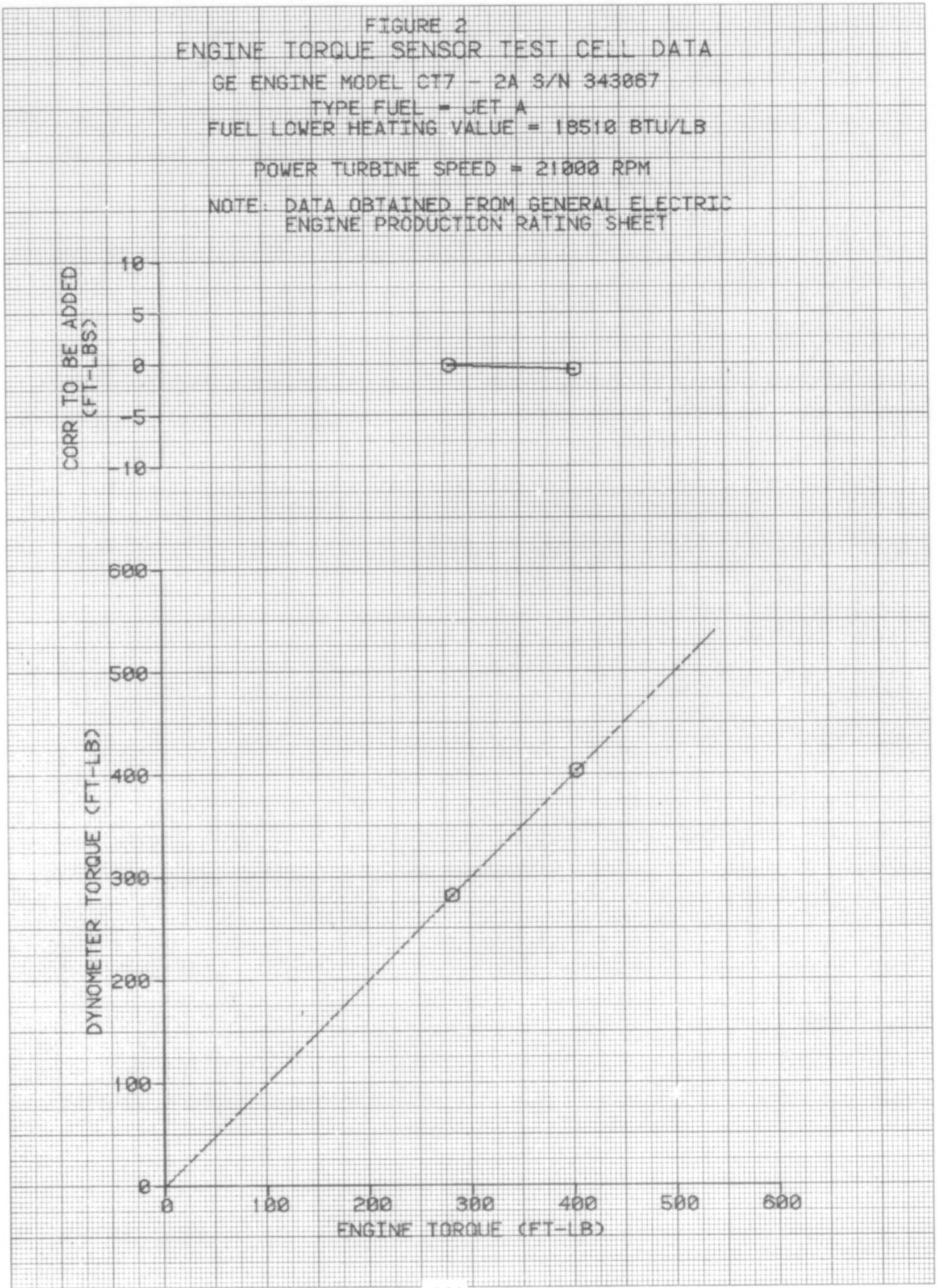


FIGURE 2
ENGINE TORQUE SENSOR TEST CELL DATA
GE ENGINE MODEL CT7 - 2A S/N 343067
TYPE FUEL = JET A
FUEL LOWER HEATING VALUE = 18510 BTU/LB
POWER TURBINE SPEED = 21000 RPM
NOTE: DATA OBTAINED FROM GENERAL ELECTRIC ENGINE PRODUCTION RATING SHEET
CORR TO BE ADDED (FT-LBS)
10
5
0
-5
-10
DYNOMETER TORQUE (FT-LB)
600
500
400
300
200
100
0
0
100
200
300
400
500
600
ENGINE TORQUE (FT-LB)

Where:

Q = Engine output shaft torque (ft-lb)

N_p = Engine output shaft rotational speed (rpm)

5252.113 = Conversion factor (ft-lb-rev/min-SHP)

No corrections for instrumentaton electrical power consumption or reductions in power required for the effect of external instrumentation drag were made.

Hover Performance

6. Hover performance was obtained both in and out-of-ground effect by the tethered hover technique. All hover tests were conducted in winds less than 3 knots. Atmospheric pressure, temperature, and wind velocity were recorded from a ground weather station. The hover tests consisted of tethering the helicopter to the ground with a cable and load cell, and stabilizing at predesignated power settings. The power settings were varied to obtain increments of cable tension from minimum to maximum.

Level Flight Performance

7. Level flight performance data were obtained in coordinated (ball-centered) flight using the ship's turn and slip indicator as the flight reference. Each test was conducted maintaining a constant ratio of gross weight to density ratio (W/σ) by increasing altitude as gross weight decreased due to fuel consumption. Main rotor speed was maintained at 100% due to the limited operational main rotor speed range.

Autorotational Descent Performance

8. Autorotational descent performance data were acquired at incremental airspeeds with constant rotor speed and incremental rotor speeds with constant airspeed. The tapeline rates of descent were calculated by the expression:

$$R/D\ \text{tapeline} = \frac{dH_p}{dt} \times \frac{T_t}{T_s} \qquad (9)$$

Where:

R/D tapeline = Tapeline rate of descent (ft/sec)

$\frac{dH_p}{dt}$ = Change in pressure altitude per given time (ft/sec)

T_t = Test ambient air temperature (°K)

T_s = Standard ambient air temperature (°K)

HANDLING QUALITIES

Control Positions in Trimmed Forward Flight

9. Control positions and aircraft attitudes as functions of airspeed were determined during level flight performance tests.

Static Longitudinal Stability

10. The static longitudinal stability tests were accomplished by establishing the trim condition in ball-centered flight and then varying control positions to obtain airspeed changes about the trim airspeed with collective control held fixed at the trim value. The airspeed range of interest was approximately ± 20 knots from trim. Altitude was allowed to vary as required during the test.

Static Lateral-Directional Stability

11. These tests were conducted by establishing the trim condition and then varying sideslip angle incrementally up to the pre-established limits. During each test, collective control position, airspeed, and aircraft ground track were held constant and altitude allowed to vary as required.

Maneuvering Stability

12. This test was accomplished by establishing the trim condition and then incrementally increasing load factor by increasing roll attitude (in both directions) while holding airspeed and collective control position constant and allowing altitude to vary as necessary.

Dynamic Stability

13. Dynamic longitudinal and lateral-directional stability were qualitatively evaluated to determine both the short- and long-period characteristics. The short-period response was evaluated by use of longitudinal, lateral, and directional pulse inputs and by releases from a steady-heading sideslip. The long-period dynamic response was evaluated longitudinally by slowly returning the flight controls to trim position following a decrease of 10 knots indicated airspeed (KIAS) from the trim airspeed.

DEFINITIONS

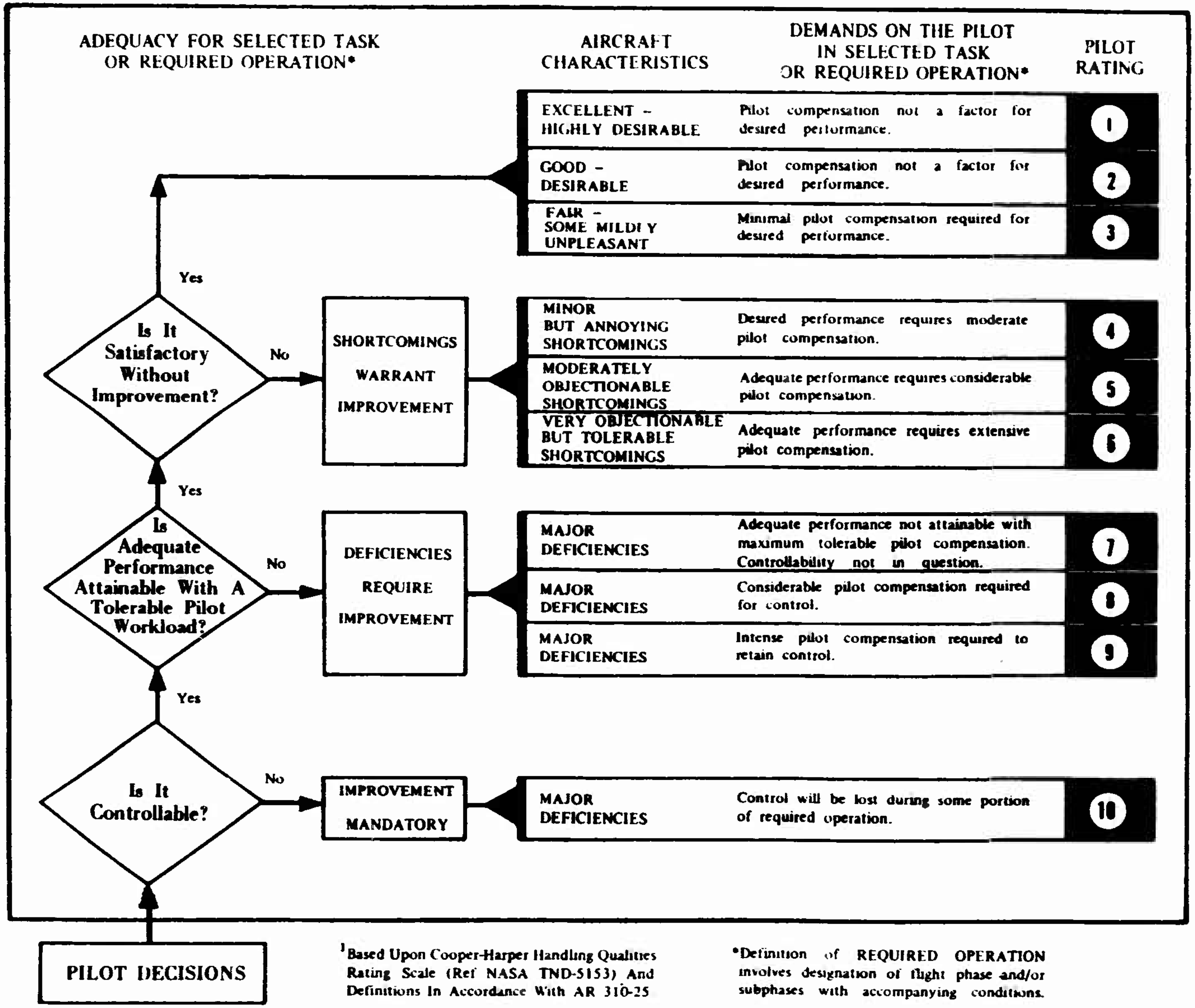
14. An unsatisfactory characteristic is defined as a feature of the equipment that could compromise safety, result in damage to the equipment or injury to personnel if operation is continued, or jeopardizes the satisfactory completion of an assigned mission.

15. An undesirable characteristic is defined as a feature of the equipment that reduces efficiency, results in increased workload; or provide an unnecessary annoyance. It generally should not cause an immediate breakdown; jeopardize safety, or materially reduce the usability of the material or end product.

16. An enhancing characteristic is defined as a characteristic of the test article which will enhance the accomplishment of the mission or is a marked advancement in the state of the art.

QUALITATIVE RATING SCALES

17. A Handling Qualities Rating Scale (HQRS) was used to augment pilot comments and is presented in figure 1. The Vibration Rating Scale (VRS) was used to augment pilot comments on vibration and is presented in figure 2.



[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.

Figure 1. Handling Qualities Rating Scale

DEGREE OF VIBRATION	DESCRIPTION[1]	PILOT RATING
No vibration		0
Slight	Not apparent to experienced aircrew fully occupied by their tasks, but noticeable if their attention is directed to it or if not otherwise occupied.	1 2 3
Moderate	Experienced aircrew are aware of the vibration but it does not affect their work, at least over a short period.	4 5 6
Severe	Vibration is immediately apparent to experienced aircrew even when fully occupied. Performance of primary task is affected or tasks can only be done with difficulty.	7 8 9
Intolerable	Sole preoccupation of aircrew is to reduce vibration level.	10

[1] Based upon the Subjective Vibration Assessment Scale developed by the Aeroplane and Armament Experimental Establishment, Boscombe Down, England.

Figure 2. Vibration Rating Scale

APPENDIX E. TEST DATA

INDEX

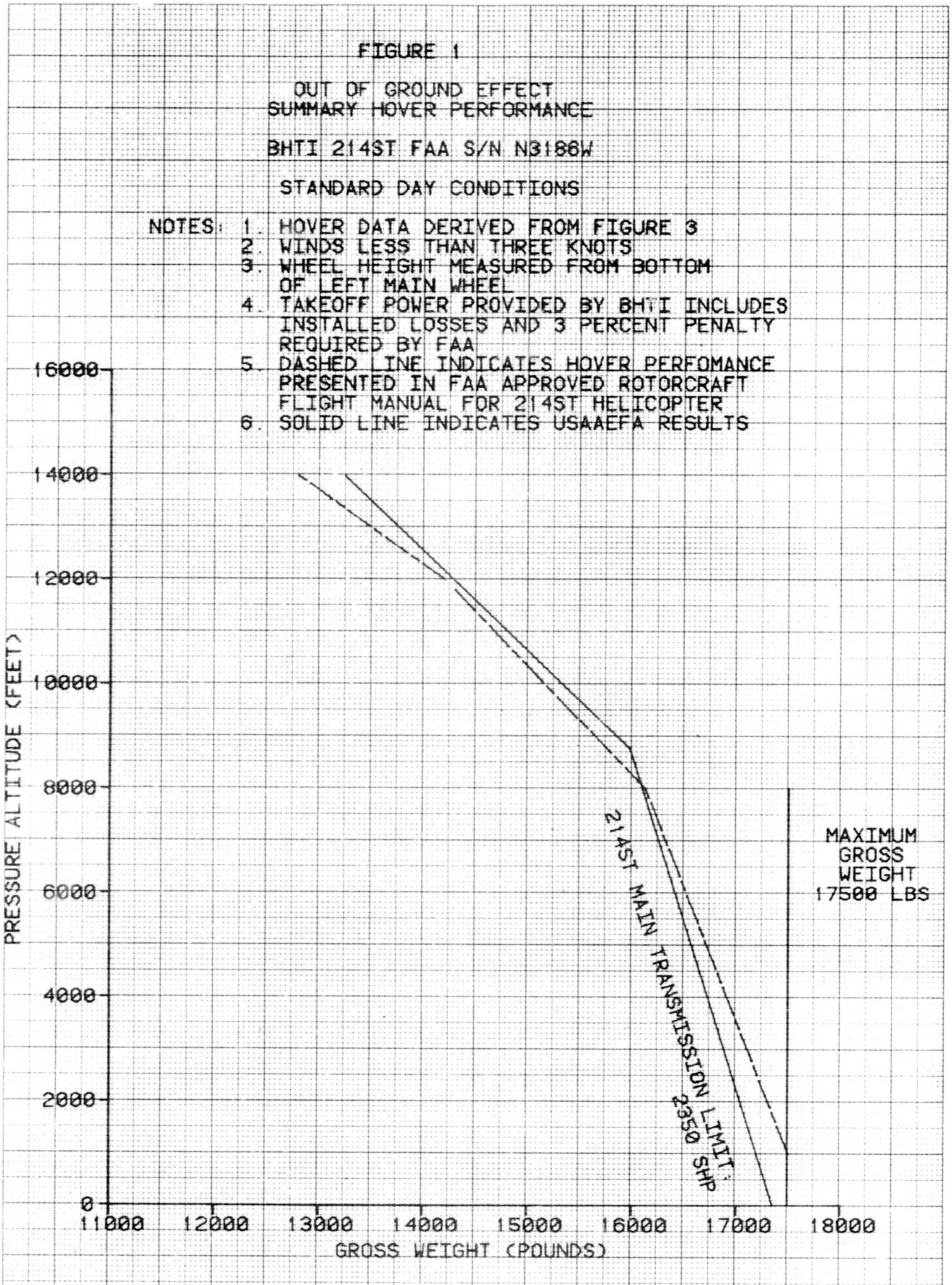


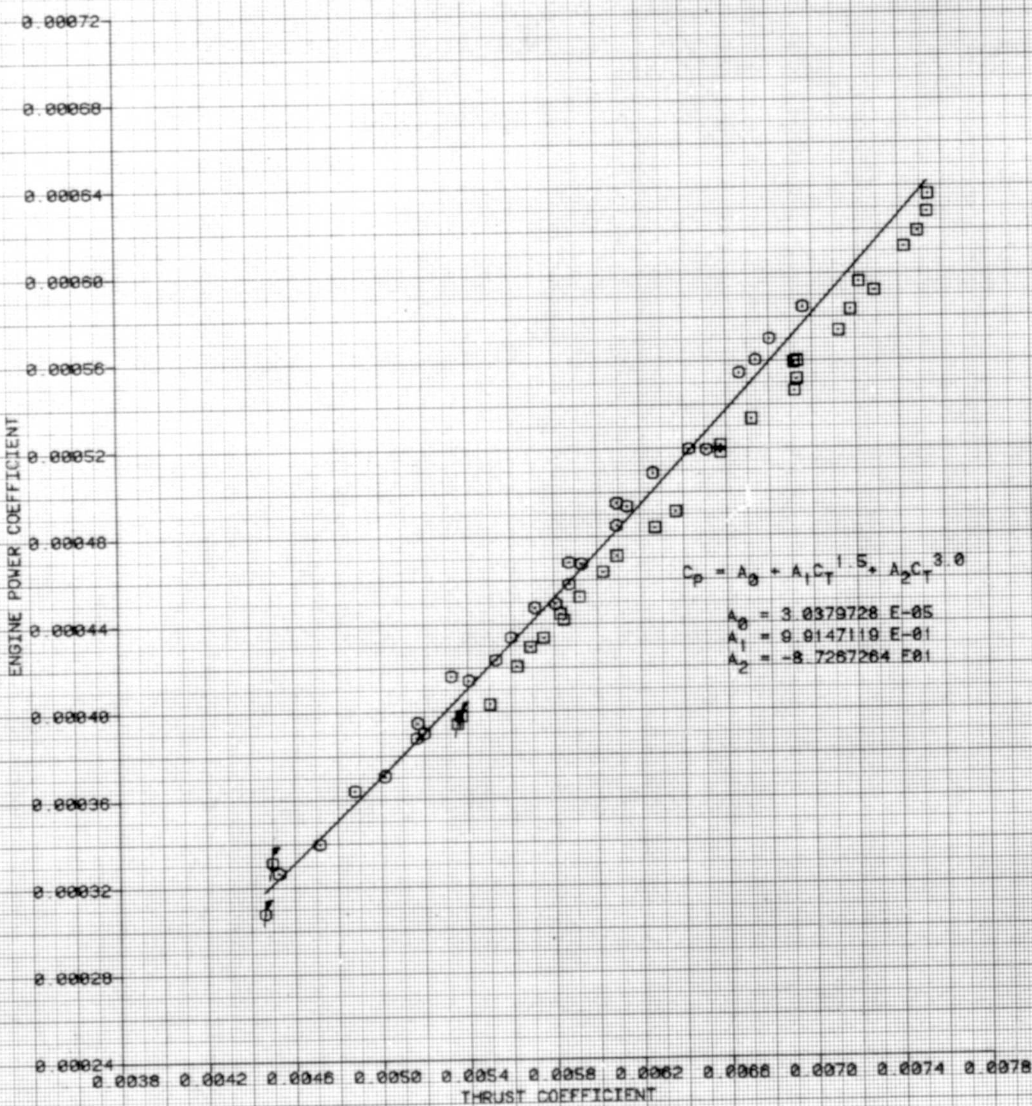
FIGURE 1
OUT OF GROUND EFFECT
SUMMARY HOVER PERFORMANCE
BHTI 214ST FAA S/N N3186W
STANDARD DAY CONDITIONS
NOTES: 1. HOVER DATA DERIVED FROM FIGURE 3
2. WINDS LESS THAN THREE KNOTS
3. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
4. TAKEOFF POWER PROVIDED BY BHTI INCLUDES INSTALLED LOSSES AND 3 PERCENT PENALTY REQUIRED BY FAA
5. DASHED LINE INDICATES HOVER PERFOMANCE PRESENTED IN FAA APPROVED ROTORCRAFT FLIGHT MANUAL FOR 214ST HELICOPTER
6. SOLID LINE INDICATES USAAEFA RESULTS
PRESSURE ALTITUDE (FEET)
16000
14000
12000
10000
8000
6000
4000
2000
0
11000
12000
13000
14000
15000
16000
17000
18000
GROSS WEIGHT (POUNDS)
214ST MAIN TRANSMISSION LIMIT: 2350 SHP
MAXIMUM GROSS WEIGHT 17500 LBS

FIGURE 2
NONDIMENSIONAL HOVER PERFORMANCE
BHTI 214ST FAA S/N N3186W

WHEEL HEIGHT = 5 FT

SYMBOL	DENSITY ALTITUDE (FEET)	OUTSIDE AIR TEMPERATURE (DEG C)
⊙	6120	24.0
⊡	11480	12.5

NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. MAIN ROTOR SPEED: 287 RPM
3. DATA WAS OBTAINED USING TETHERED HOVER TECHNIQUE UNLESS OTHERWISE NOTED BY FLAGS WHICH DENOTE FREE FLIGHT HOVER
4. WINDS LESS THAN THREE KNOTS



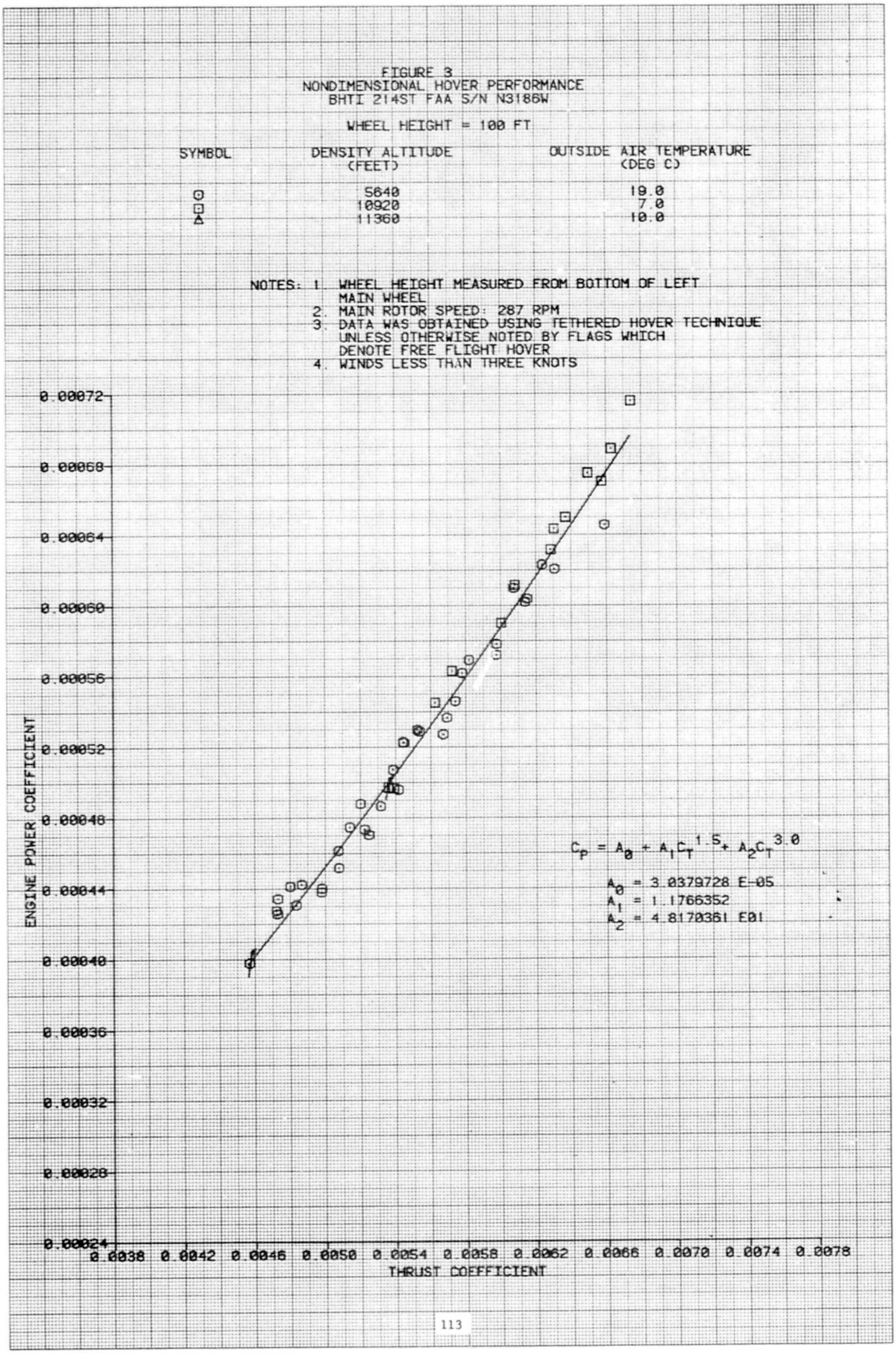


FIGURE 3
NONDIMENSIONAL HOVER PERFORMANCE
BHTI 214ST FAA S/N N3186W
WHEEL HEIGHT = 100 FT
SYMBOL
DENSITY ALTITUDE (FEET)
OUTSIDE AIR TEMPERATURE (DEG C)
5640
10920
11360
19.0
7.0
10.0
NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL
2. MAIN ROTOR SPEED: 287 RPM
3. DATA WAS OBTAINED USING TETHERED HOVER TECHNIQUE UNLESS OTHERWISE NOTED BY FLAGS WHICH DENOTE FREE FLIGHT HOVER
4. WINDS LESS THAN THREE KNOTS
ENGINE POWER COEFFICIENT
0.00072
0.00068
0.00064
0.00060
0.00056
0.00052
0.00048
0.00044
0.00040
0.00036
0.00032
0.00028
0.00024
$C_P = A_0 + A_1 C_T^{1.5} + A_2 C_T^{3.0}$
A_0 = 3.0379728 E-05
A_1 = 1.1766352
A_2 = 4.8170361 E01
0.0038
0.0042
0.0046
0.0050
0.0054
0.0058
0.0062
0.0066
0.0070
0.0074
0.0078
THRUST COEFFICIENT

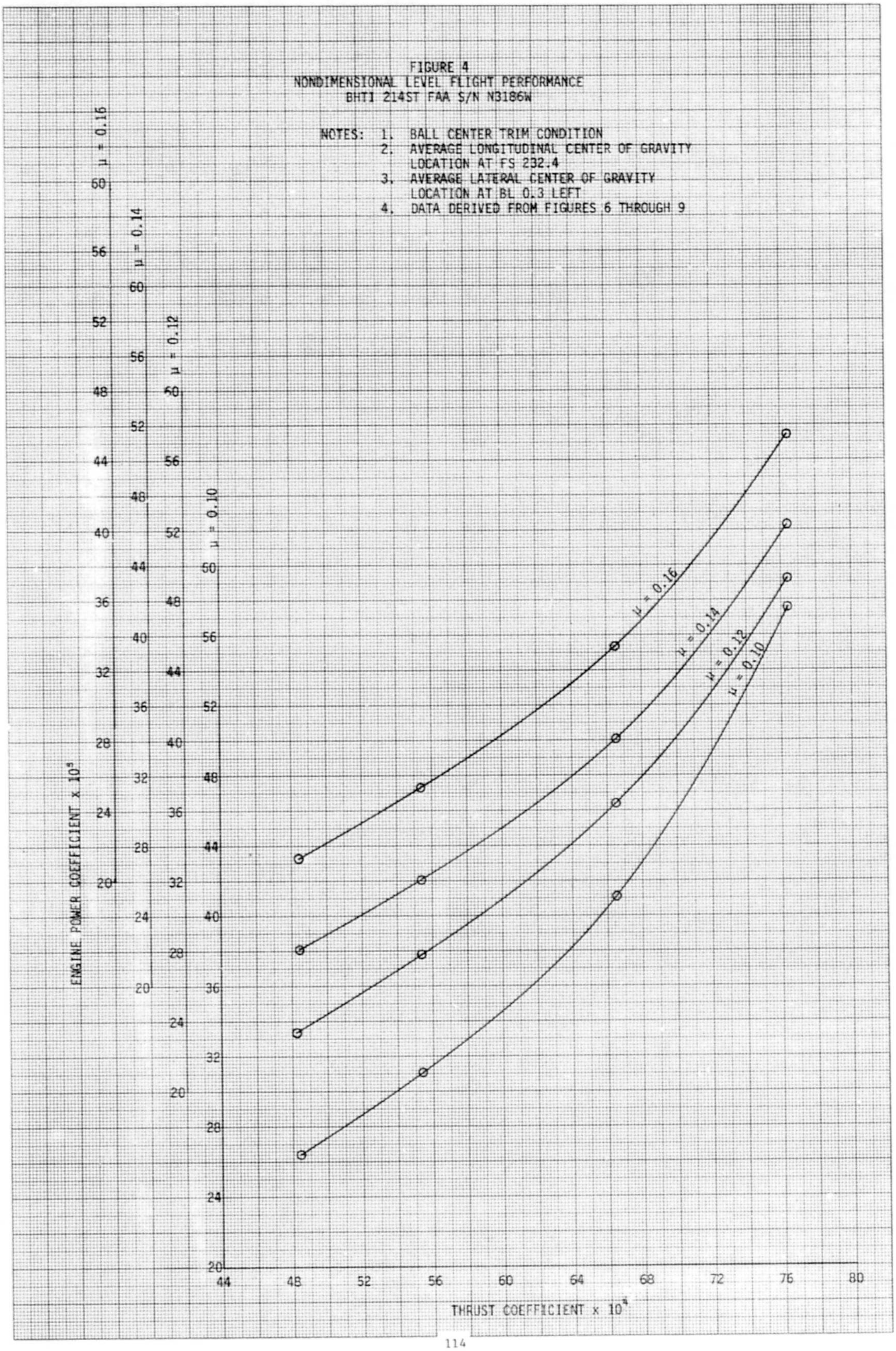
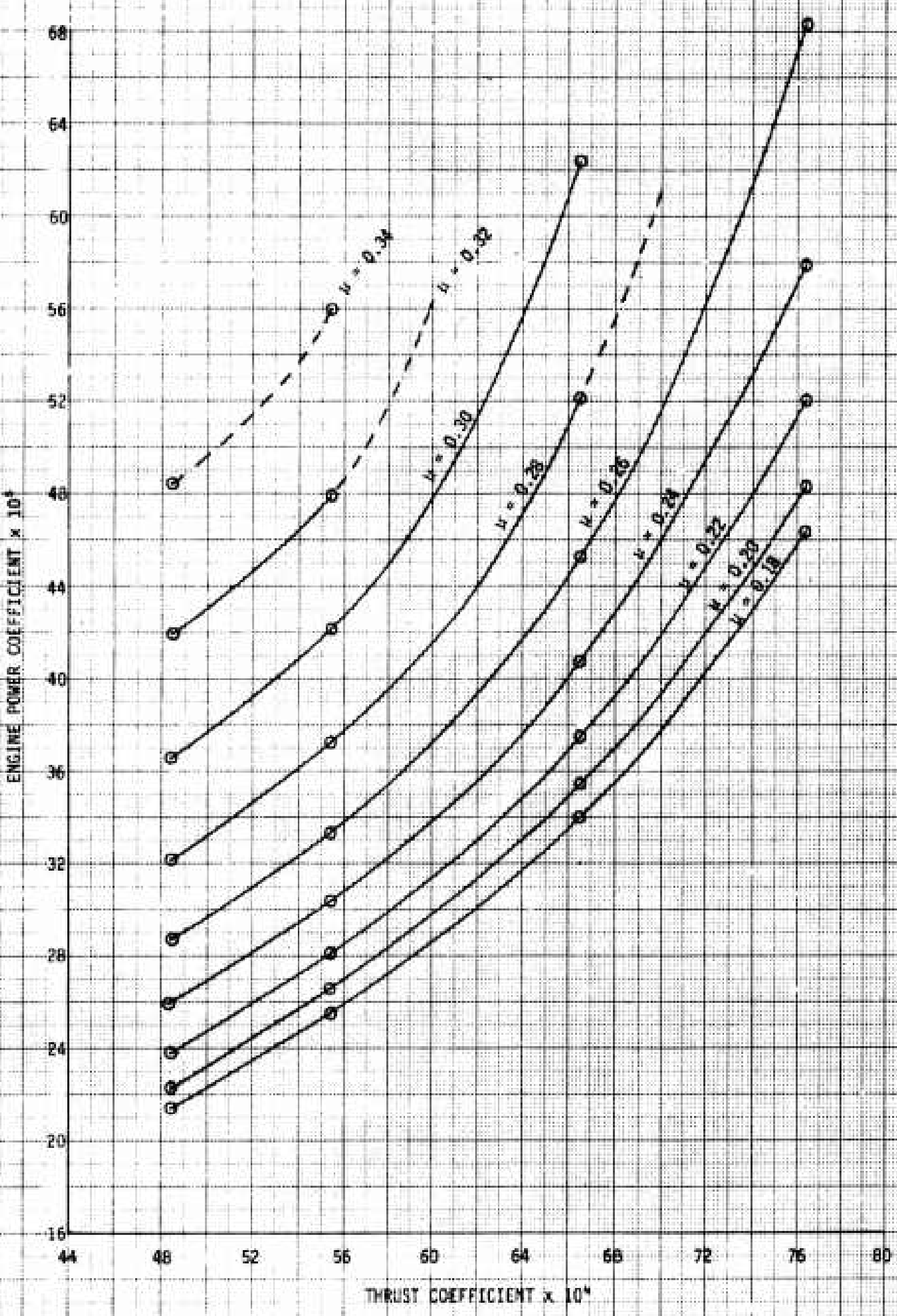


FIGURE 4
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
BHTI 214ST FAA S/N N3186W
NOTES: 1. BALL CENTER TRIM CONDITION
2. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 232.4
3. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.3 LEFT
4. DATA DERIVED FROM FIGURES 6 THROUGH 9
μ = 0.16
μ = 0.14
μ = 0.12
μ = 0.10
ENGINE POWER COEFFICIENT x 10⁶
THRUST COEFFICIENT x 10⁴

FIGURE 5
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
BHTI 214ST FAA S/N N3186W

NOTES: 1. BALL CENTER TRIM CONDITION
2. AVERAGE LONGITUDINAL CENTER OF GRAVITY LOCATION AT FS 232.4
3. AVERAGE LATERAL CENTER OF GRAVITY LOCATION AT BL 0.3 LEFT
4. DATA DERIVED FROM FIGURES 6 THROUGH 9



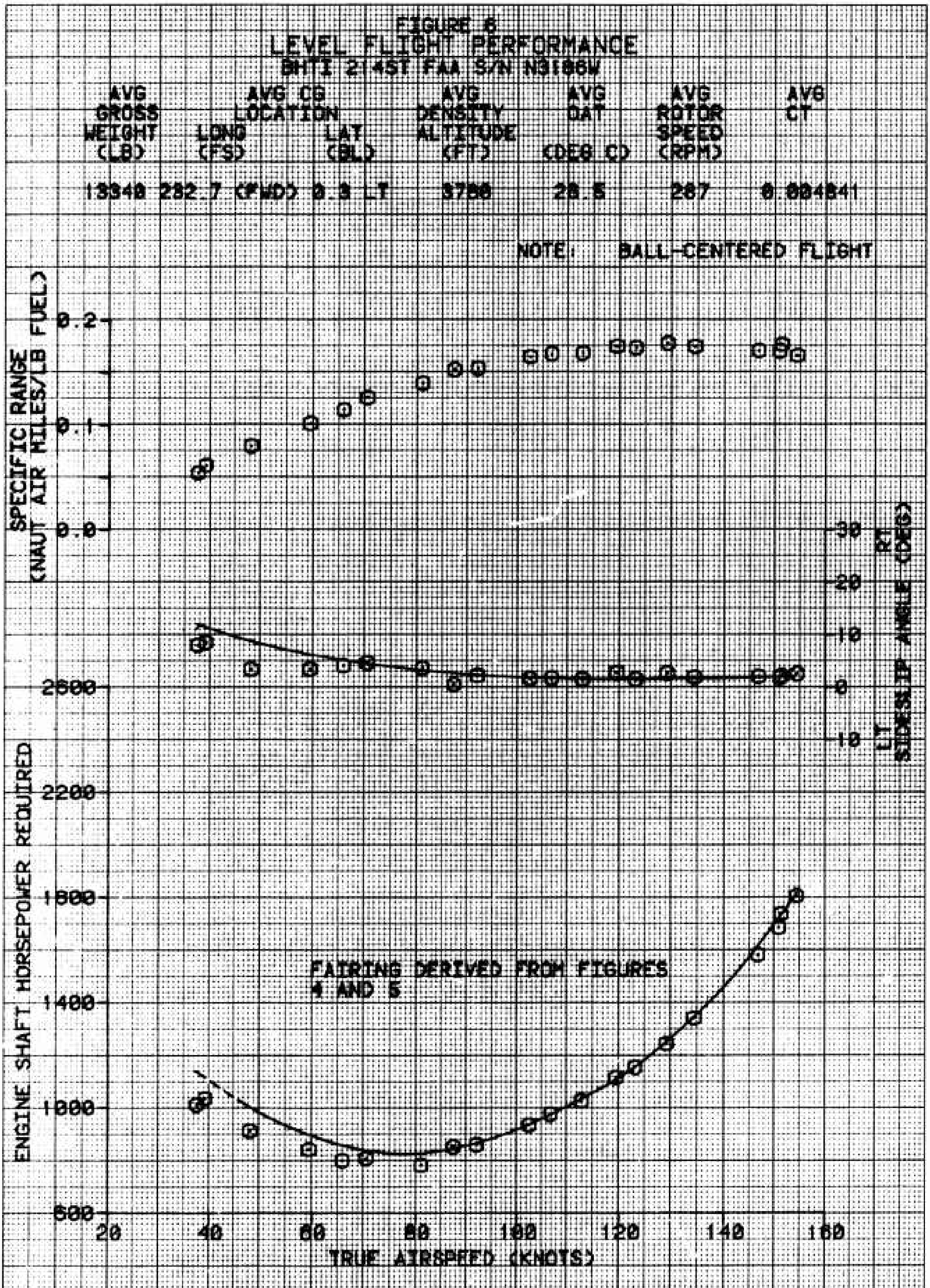


FIGURE 6
LEVEL FLIGHT PERFORMANCE
BHTI 214ST FAA S/N N3186W
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG (FS) LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
AVG CT
13340 282.7 (FWD) 0.3 LT 3700 28.5 287 0.004841
NOTE: BALL-CENTERED FLIGHT
SPECIFIC RANGE (NAUT AIR MILES/LB FUEL)
0.2
0.1
0.0
RT
30
20
10
0
10
LT
SIDESLIP ANGLE (DEG)
ENGINE SHAFT HORSEPOWER REQUIRED
2600
2200
1800
1400
1000
600
FAIRING DERIVED FROM FIGURES 4 AND 5
20
40
60
80
100
120
140
160
TRUE AIRSPEED (KNOTS)

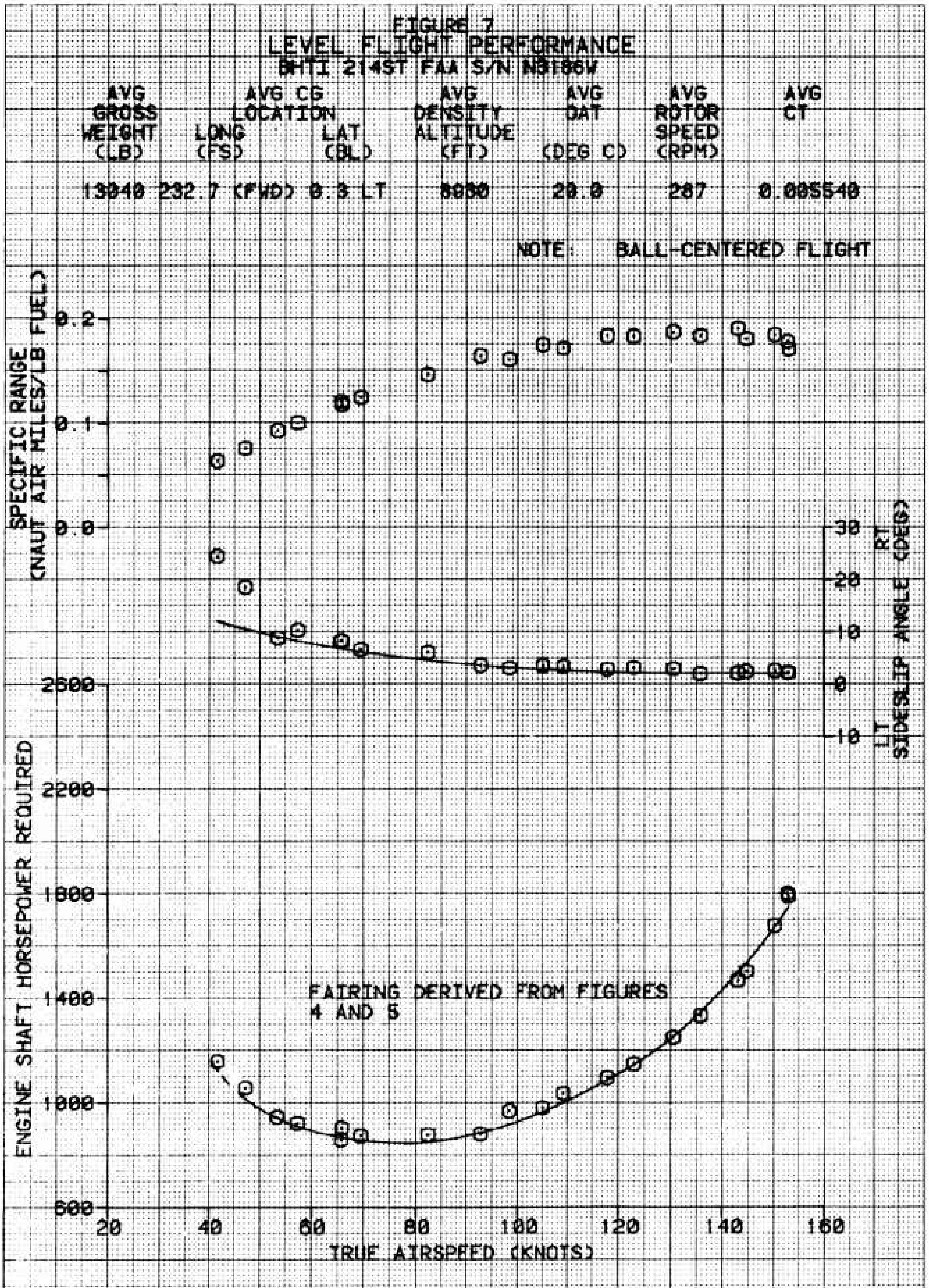


FIGURE 7
LEVEL FLIGHT PERFORMANCE
BHTI 214ST FAA S/N N3186W
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
AVG CT
13040 232.7 (FWD) 0.3 LT 8030 20.0 287 0.005540
NOTE: BALL-CENTERED FLIGHT
SPECIFIC RANGE (NAUT AIR MILES/LB FUEL)
0.2
0.1
0.0
RT
SIDESLIP ANGLE (DEG)
LT
30
20
10
0
10
ENGINE SHAFT HORSEPOWER REQUIRED
2600
2200
1800
1400
1000
600
FAIRING DERIVED FROM FIGURES 4 AND 5
20
40
60
80
100
120
140
160
TRUE AIRSPEED (KNOTS)

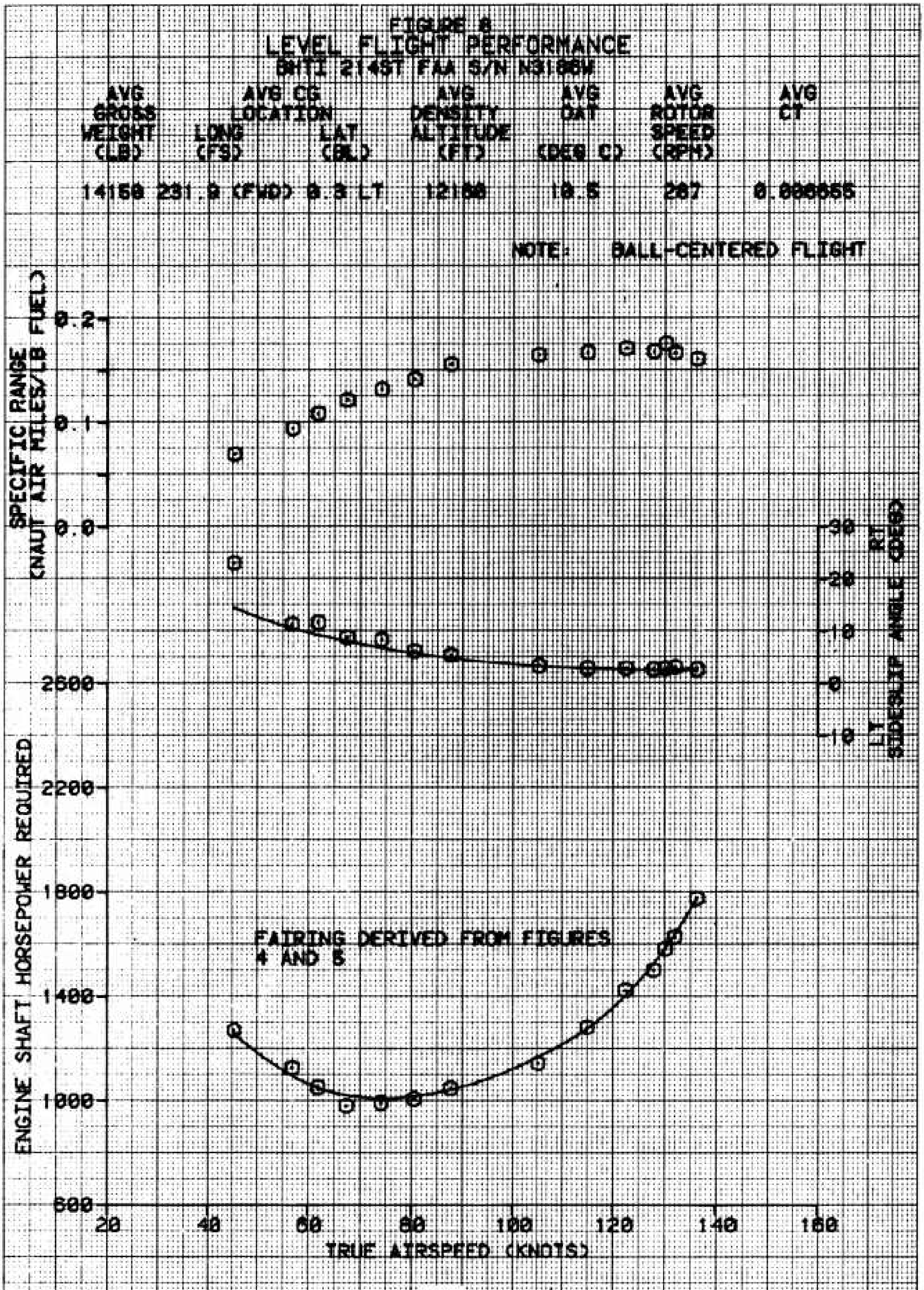


FIGURE 8
LEVEL FLIGHT PERFORMANCE
BHTI 214ST FAA S/N N3186W
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
AVG CT
14150 231.9 (FWD) 0.3 LT 12100 10.5 287 0.000655
NOTE: BALL-CENTERED FLIGHT
SPECIFIC RANGE (NAUT AIR MILES/LB FUEL)
0.2
0.1
0.0
RT
30
20
10
0
10
LT
SIDESLIP ANGLE (DEG)
ENGINE SHAFT HORSEPOWER REQUIRED
2600
2200
1800
1400
1000
600
FAIRING DERIVED FROM FIGURES 4 AND 5
20
40
60
80
100
120
140
160
TRUE AIRSPEED (KNOTS)

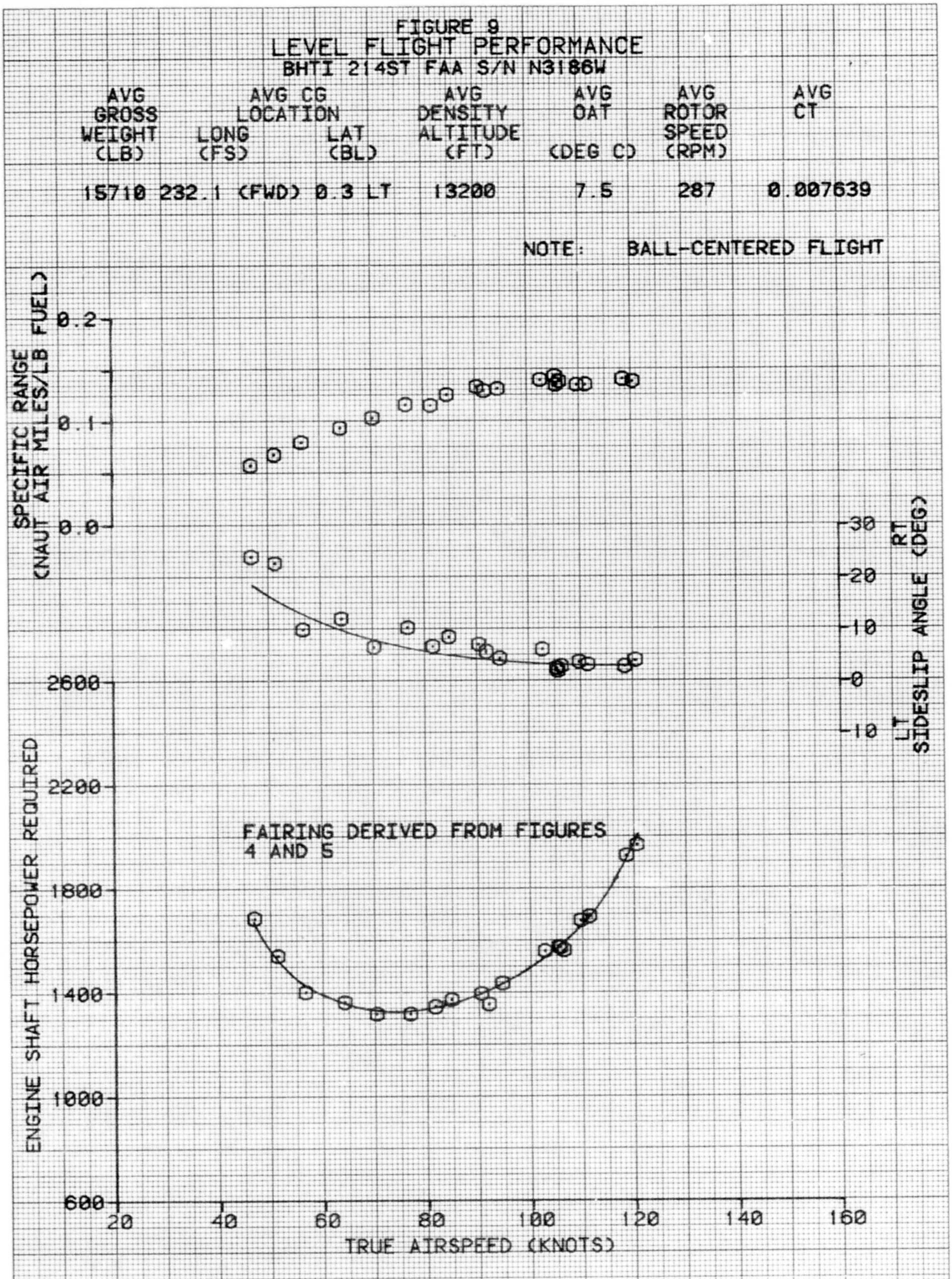


FIGURE 9
LEVEL FLIGHT PERFORMANCE
BHTI 214ST FAA S/N N3186W
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
AVG CT
15710
232.1 (FWD)
0.3 LT
13200
7.5
287
0.007639
NOTE: BALL-CENTERED FLIGHT
SPECIFIC RANGE (NAUT AIR MILES/LB FUEL)
0.2
0.1
0.0
RT
LT
SIDESLIP ANGLE (DEG)
30
20
10
0
10
ENGINE SHAFT HORSEPOWER REQUIRED
2600
2200
1800
1400
1000
600
FAIRING DERIVED FROM FIGURES 4 AND 5
20
40
60
80
100
120
140
160
TRUE AIRSPEED (KNOTS)

FIGURE 10
LEVEL FLIGHT PERFORMANCE SUMMARY
BHTI FAA S/N N3186W

STANDARD DAY CONDITIONS
DENSITY ALTITUDE = 6000 FT MAIN ROTOR SPEED = 287 RPM

NOTE: BALL-CENTERED FLIGHT

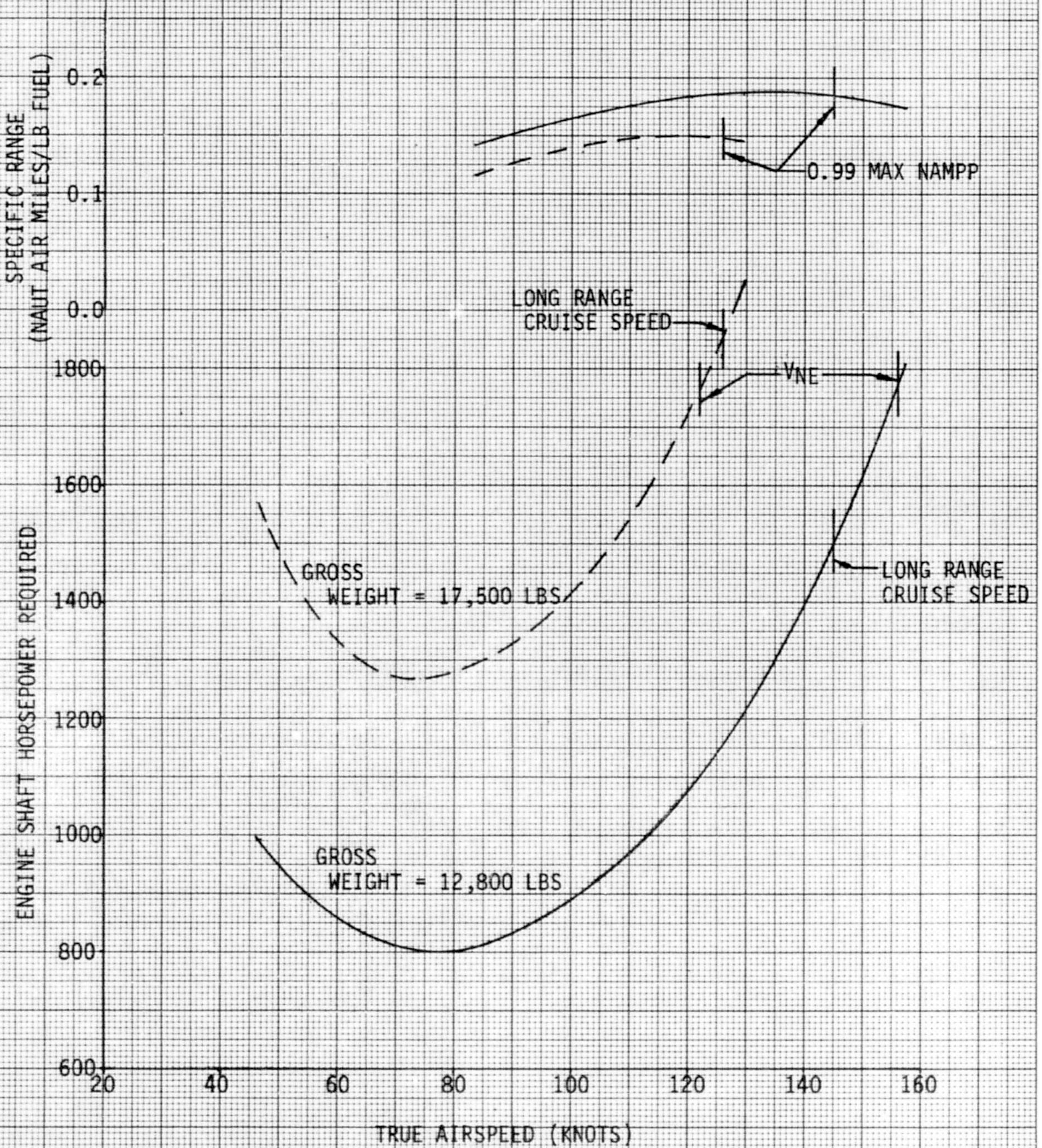


FIGURE 11
AUTOROTATIONAL DESCENT PERFORMANCE
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)
16100	232.2(FWD)	0.3 LT	7700	18.0	287

NOTE: BALL-CENTERED FLIGHT

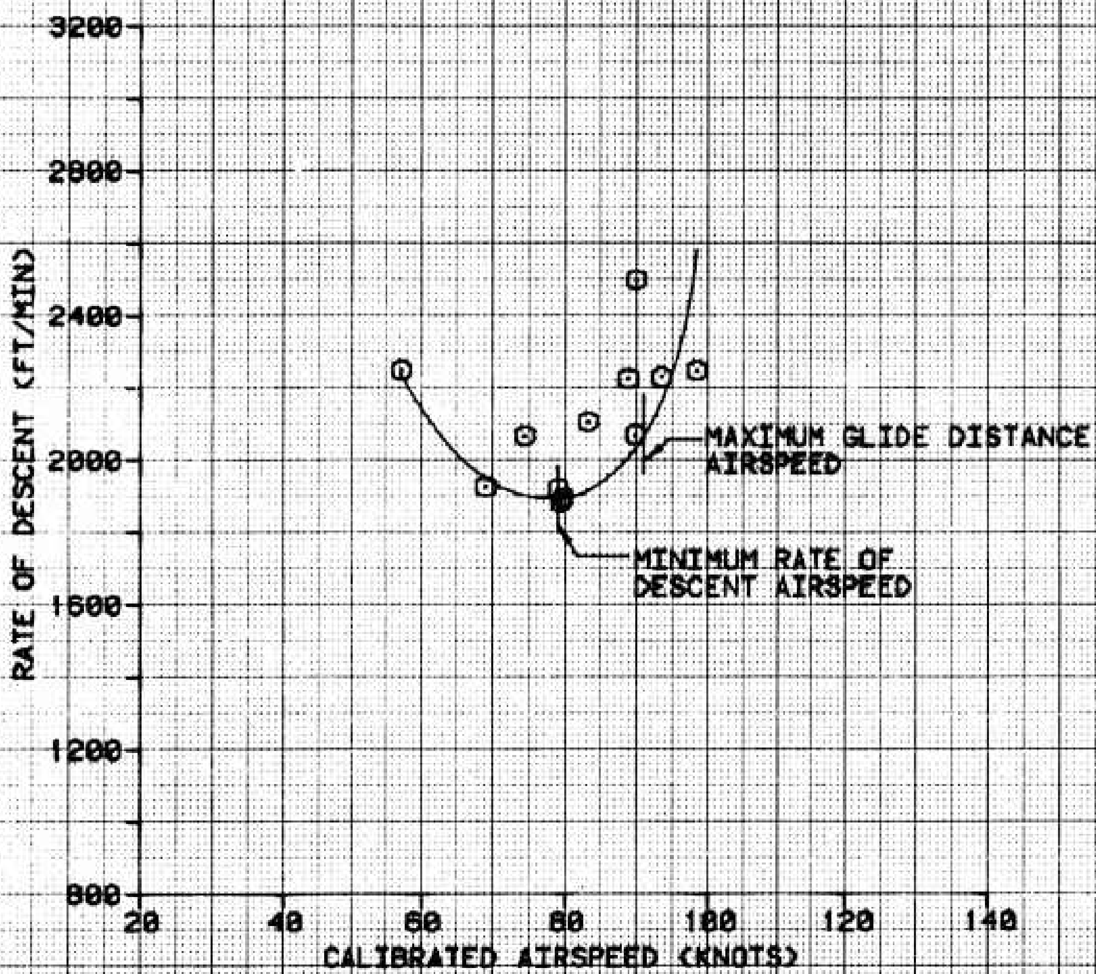


FIGURE 12
AUTOROTATIONAL DESCENT PERFORMANCE
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG CALIBRATED AIRSPEED (KT)
15900	232.3(FWD)	0.3 LT	8220	17.0	79

NOTE: BALL-CENTERED FLIGHT

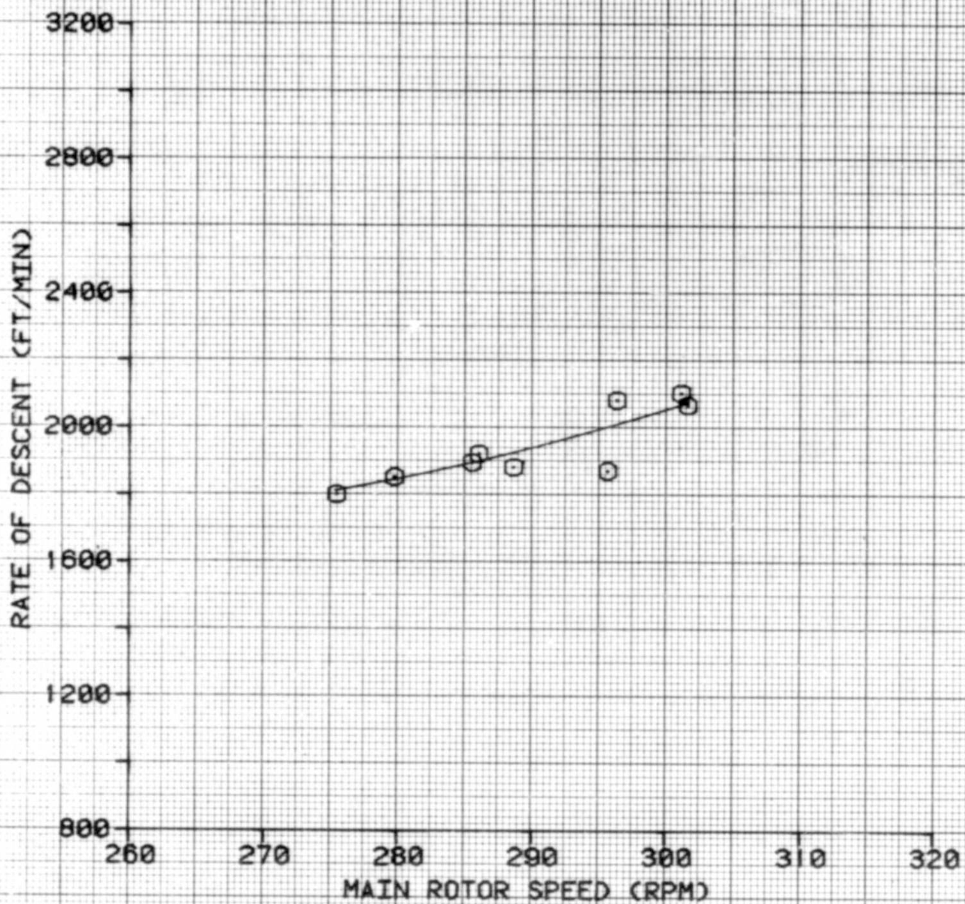


FIGURE 13
LIMITS OF CYCLIC CONTROL TRAVEL
BHTI 214ST S/N N3186W

NOTES: 1. ROTOR STATIC
2. COLLECTIVE AND DIRECTIONAL CONTROLS DO NOT AFFECT CYCLIC CONTROL POSITIONS

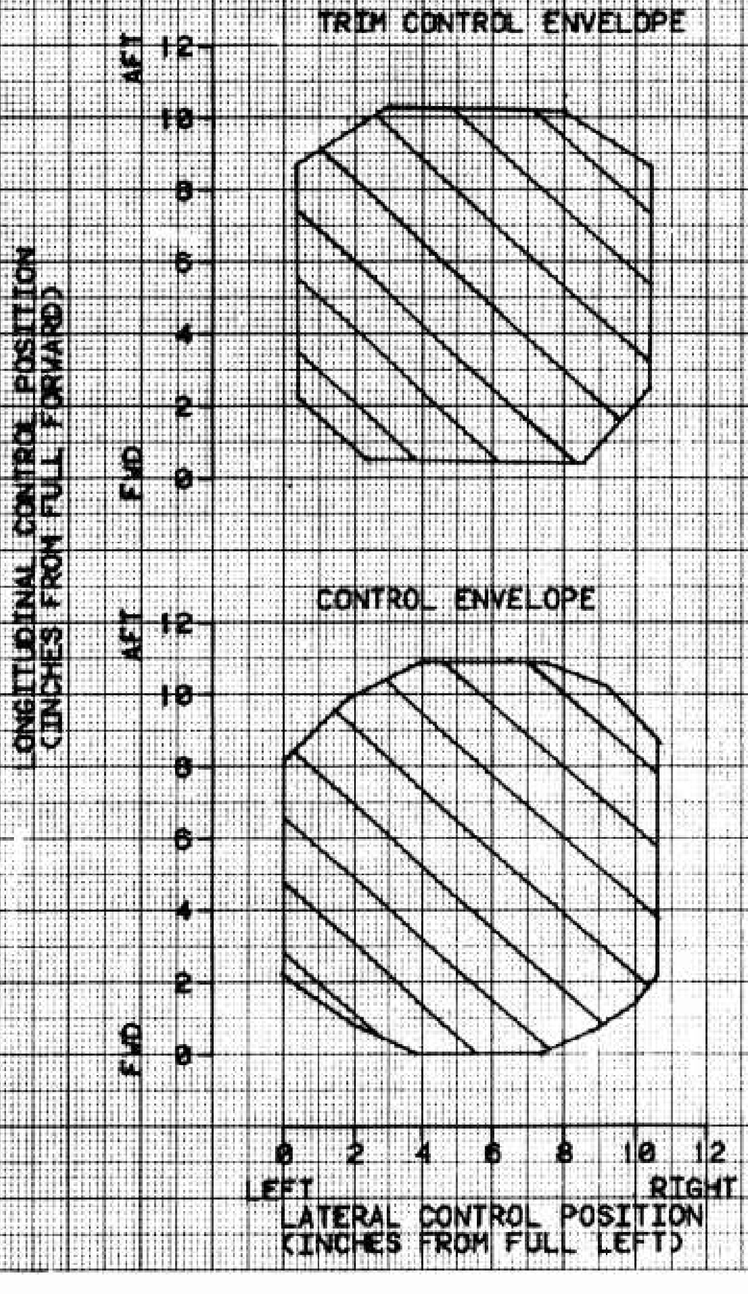


FIGURE 14
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
BHTI 214ST FAA S/N N8186W

NOTES:
1. ROTOR STATIC
2. FORCE MEASURED AT CENTER OF CONTROL GRIP
3. SHADED SYMBOL DENOTES TRIM POSITION
4. TOTAL LONGITUDINAL CONTROL TRAVEL EQUALS 11.1 INCHES

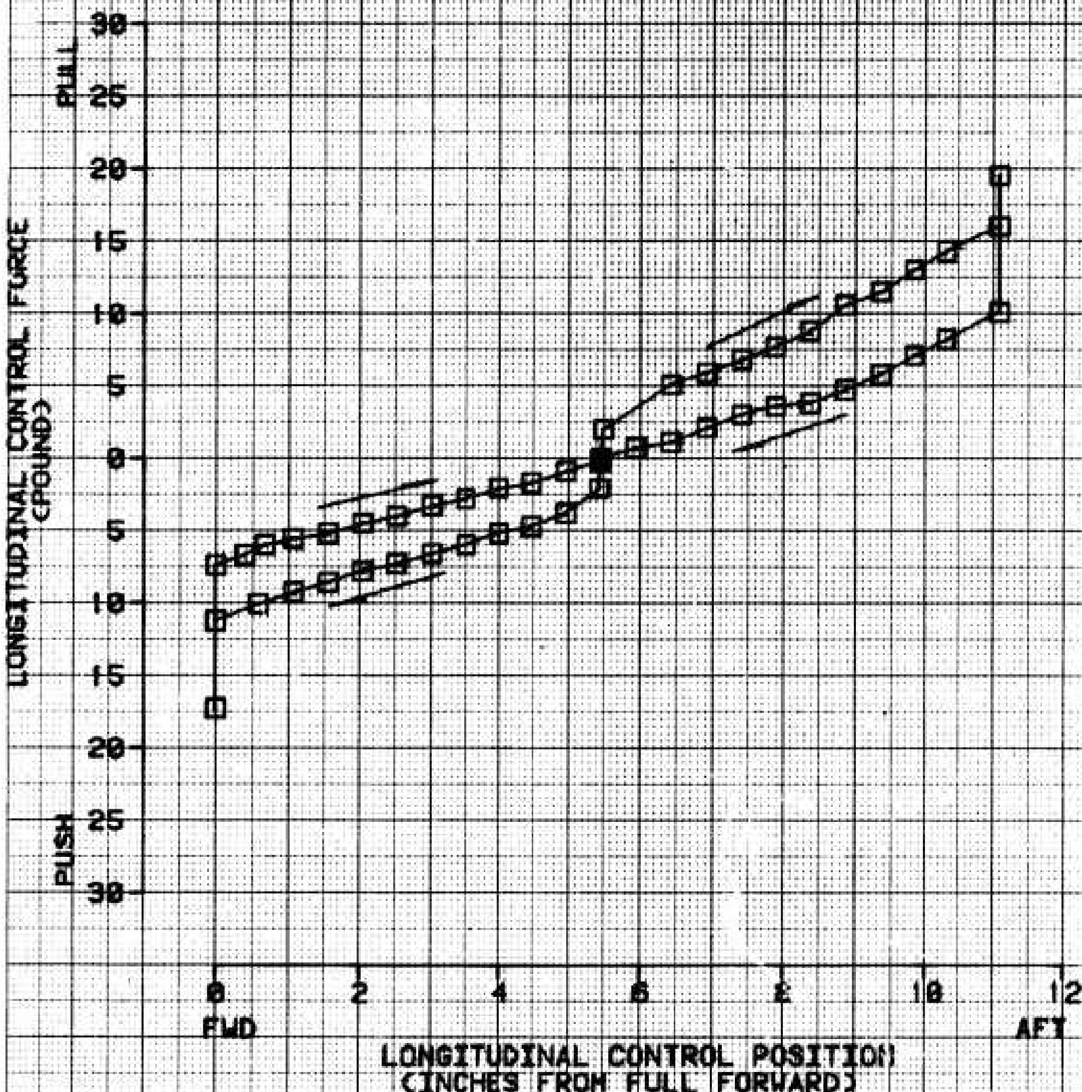


FIGURE 15
LATERAL CONTROL SYSTEM CHARACTERISTICS
BHTI 214ST FAA S/N N3186W

NOTES: 1. ROTOR STATIC
2. FORCE MEASURED AT CENTER OF CONTROL GRIP
3. SHADED SYMBOL DENOTES TRIM POSITION
4. TOTAL LATERAL CONTROL TRAVEL EQUALS 10.7 INCHES

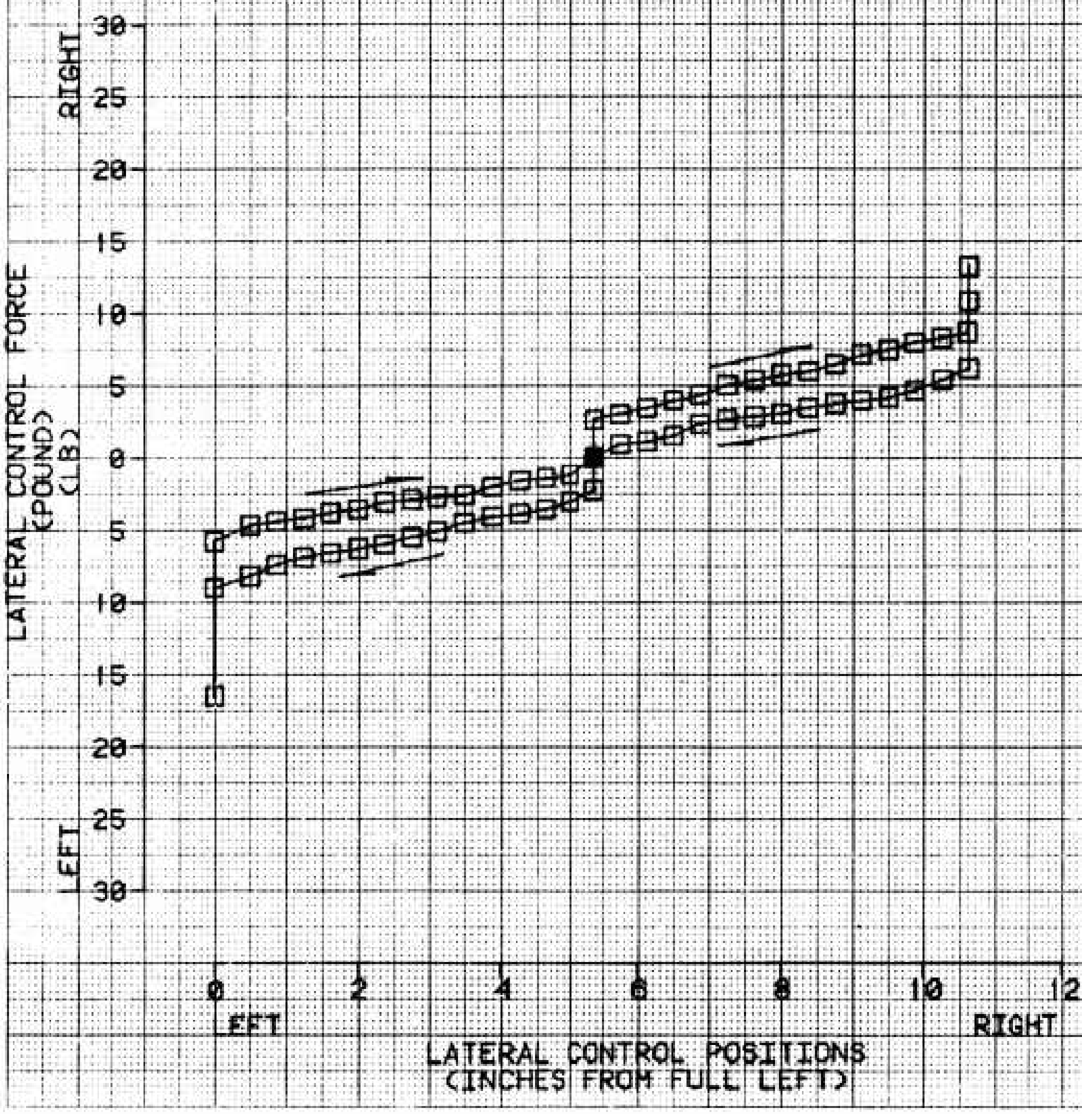


FIGURE 16
DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS
BHTI 214ST FAA S/N N3186W

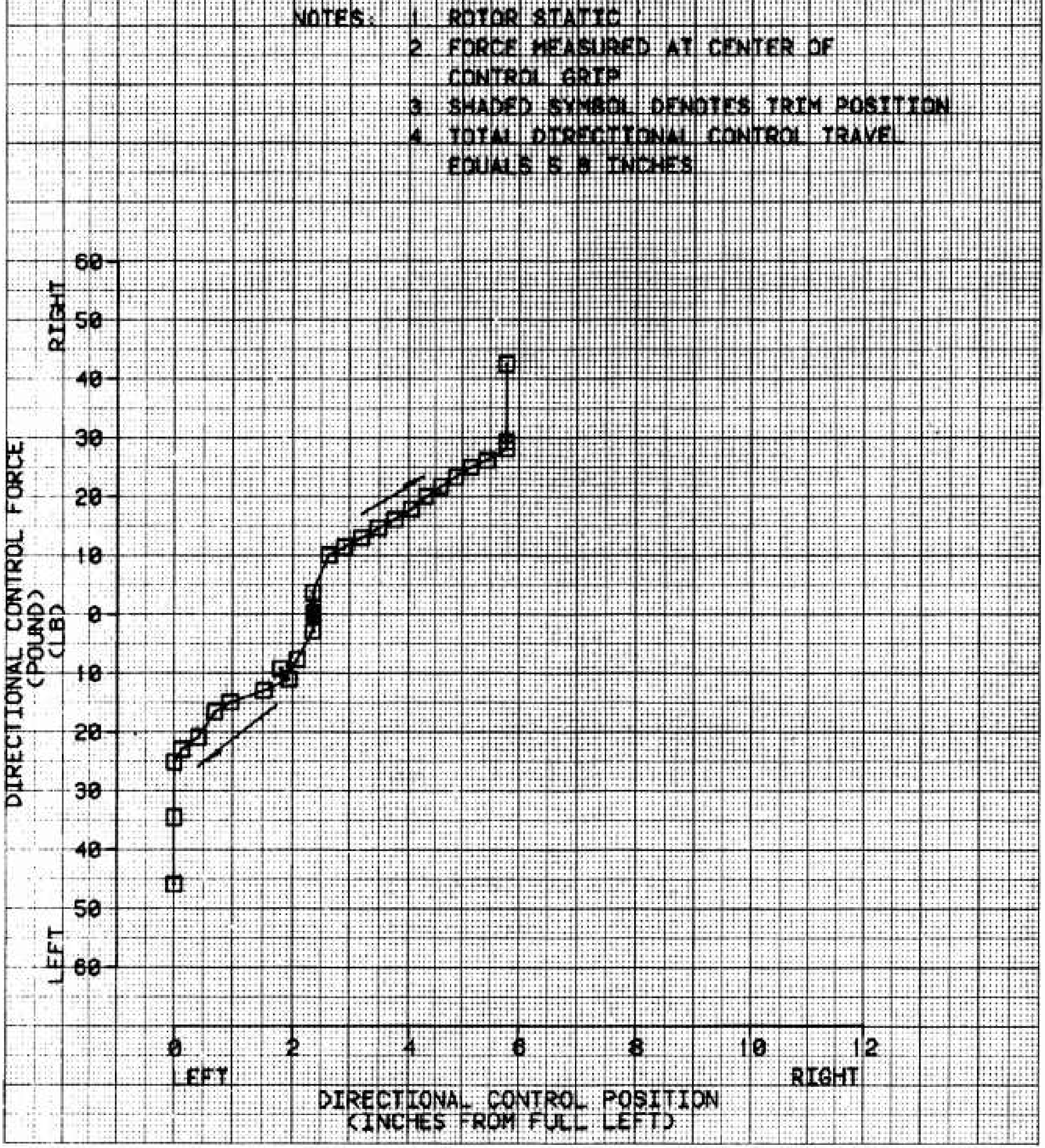


FIGURE 17
COLLECTIVE CONTROL SYSTEM CHARACTERISTICS
BHTI 214ST FAA S/N N3186W

NOTES:
1. ROTOR STATIC
2. FORCE MEASURED AT CENTER OF CONTROL PEDAL
3. SHADED SYMBOL DENOTES BEGINNING POINT
4. TOTAL COLLECTIVE CONTROL TRAVEL EQUALS 10.9 INCHES
5. FRICTION OFF

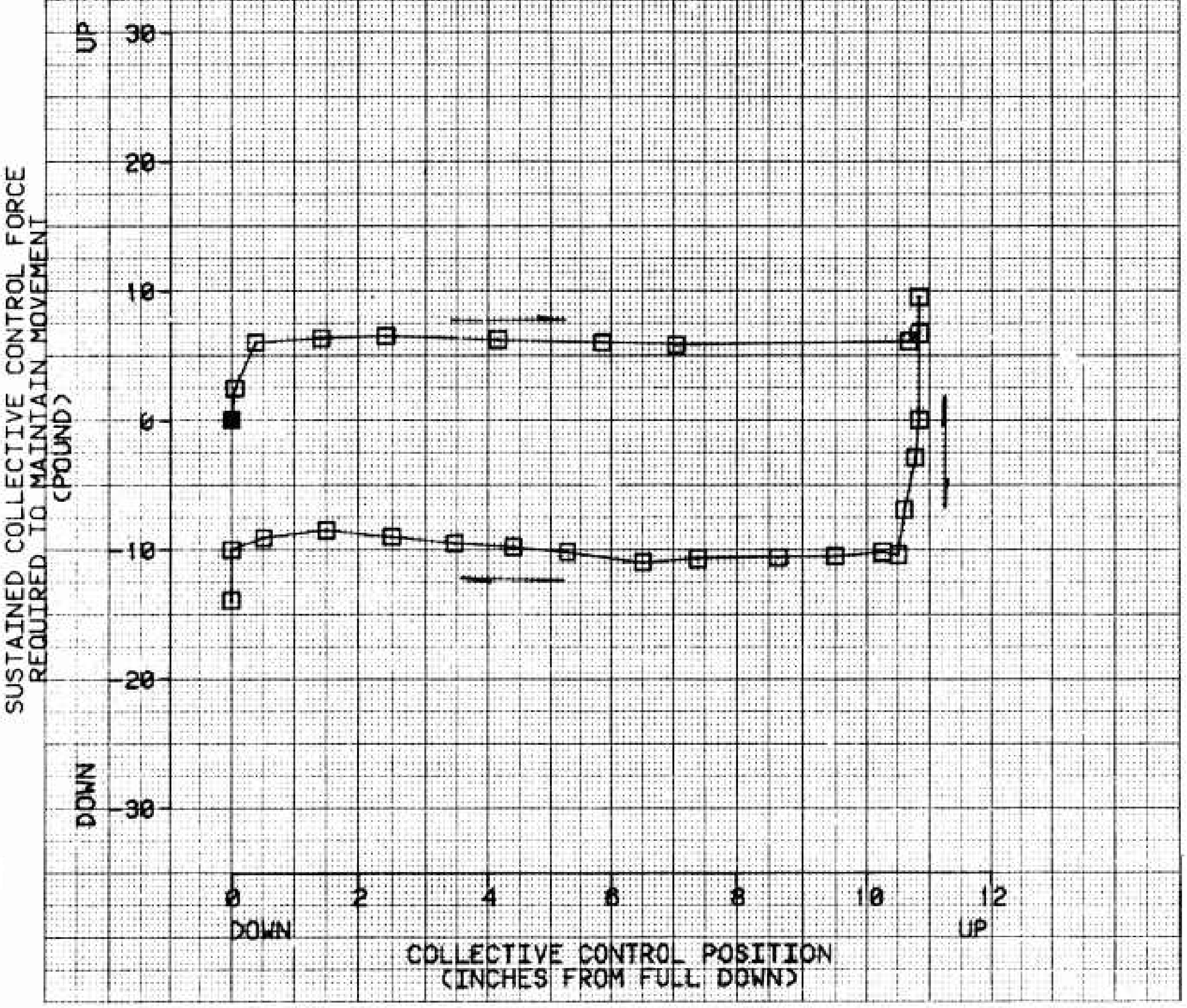


FIGURE 18
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊙	13340	232.7 (FWD)	0.3 LT	3780	26.5	287	LEVEL
⊡	14100	243.5 (AFT)	0.3 LT	4420	25.5	287	LEVEL

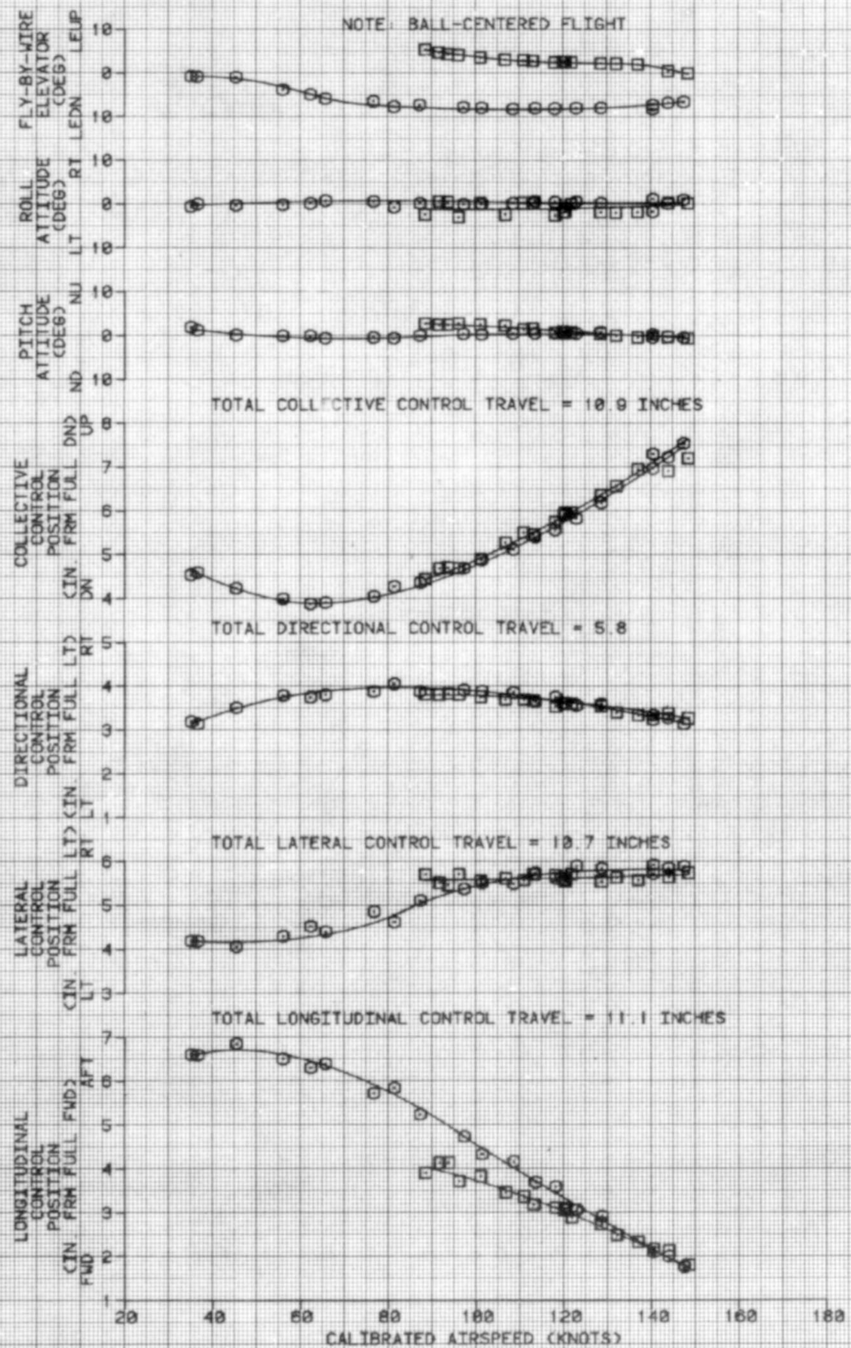


FIGURE 19
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊙	13840	232.7 (FWD)	0.3 LT	3780	26.5	287	LEVEL
⊡	15710	232.1 (FWD)	0.3 LT	13200	7.5	287	LEVEL

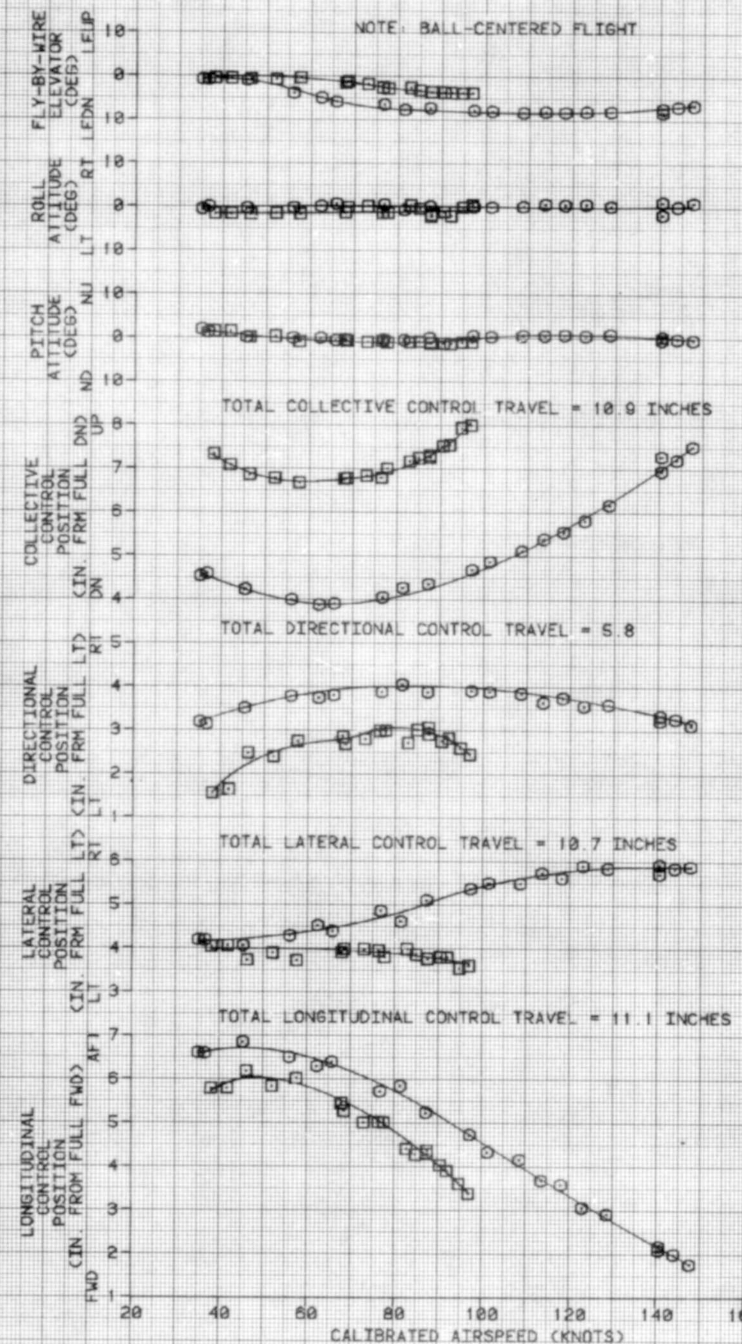


FIGURE 20
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
BHTI 214ST FAA S/N N3180W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	16220	232.4(FWD)	0.3 LT	7450	18.5	287	CLIMB
⊡	16180	231.1(FWD)	0.3 LT	7790	18.0	287	DESCENT AT 1000 FPM

NOTES: 1. BALL-CENTERED FLIGHT
2. CLIMBS CONDUCTED USING MAX CONTINUOUS POWER

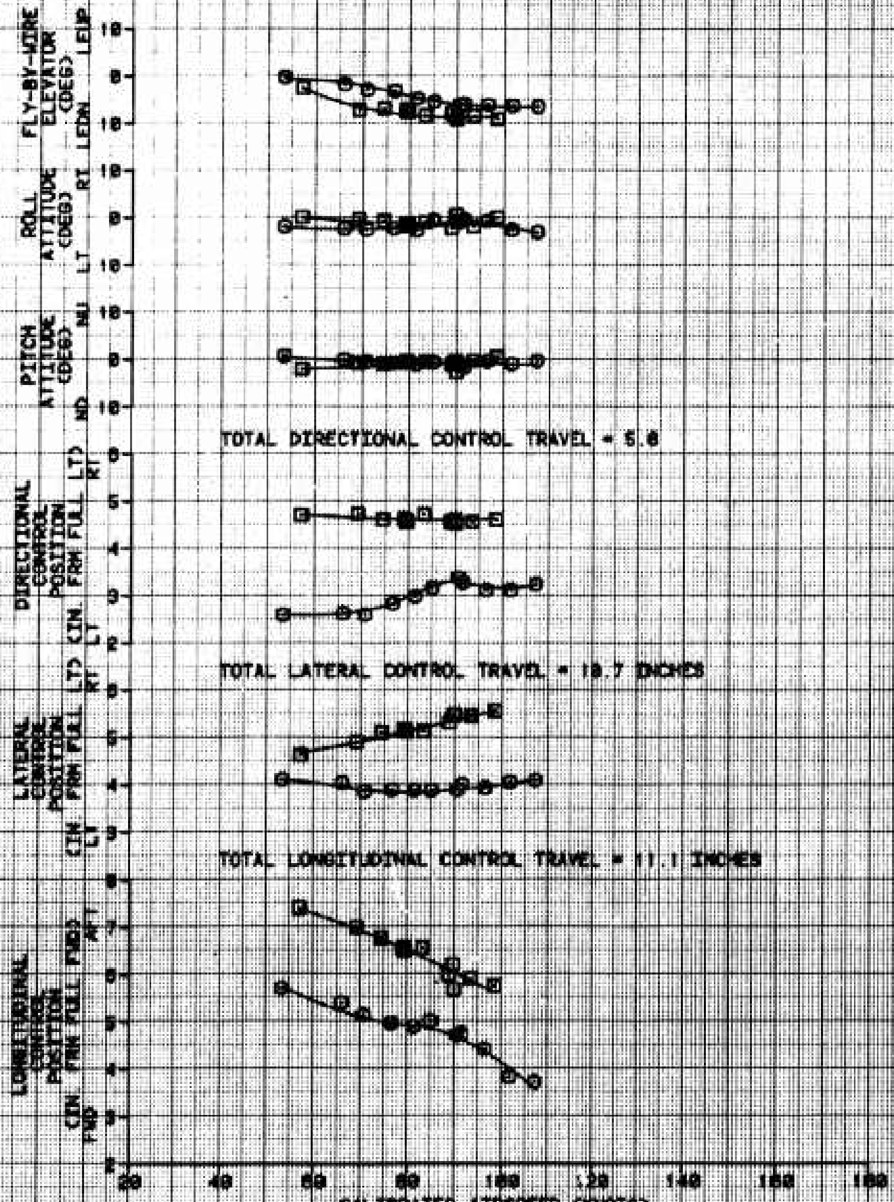


FIGURE 21
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
BHTI 214ST FAA S/N N3180W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	14900	243.8(AFT)	0.3 LT	7520	24.5	287	LEVEL
⊙	14800	243.7(AFT)	0.3 LT	8840	28.0	287	LEVEL

NOTE: SHADED SYMBOLS DENOTE TRIM

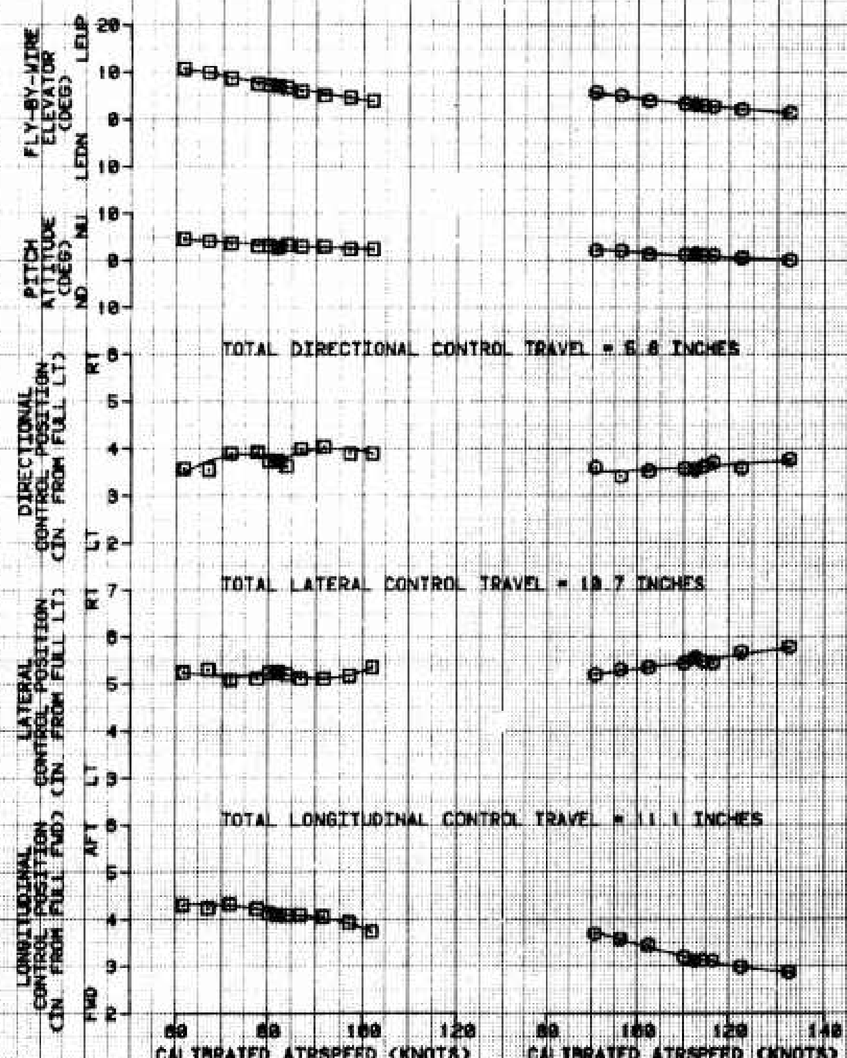


FIGURE 22
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
□	16860	237.2(AFT)	0.3 LT	6740	23.0	288	LEVEL
○	16600	237.2(AFT)	0.3 LT	6800	23.0	288	LEVEL

NOTE: SHADED SYMBOLS DENOTE TRIM

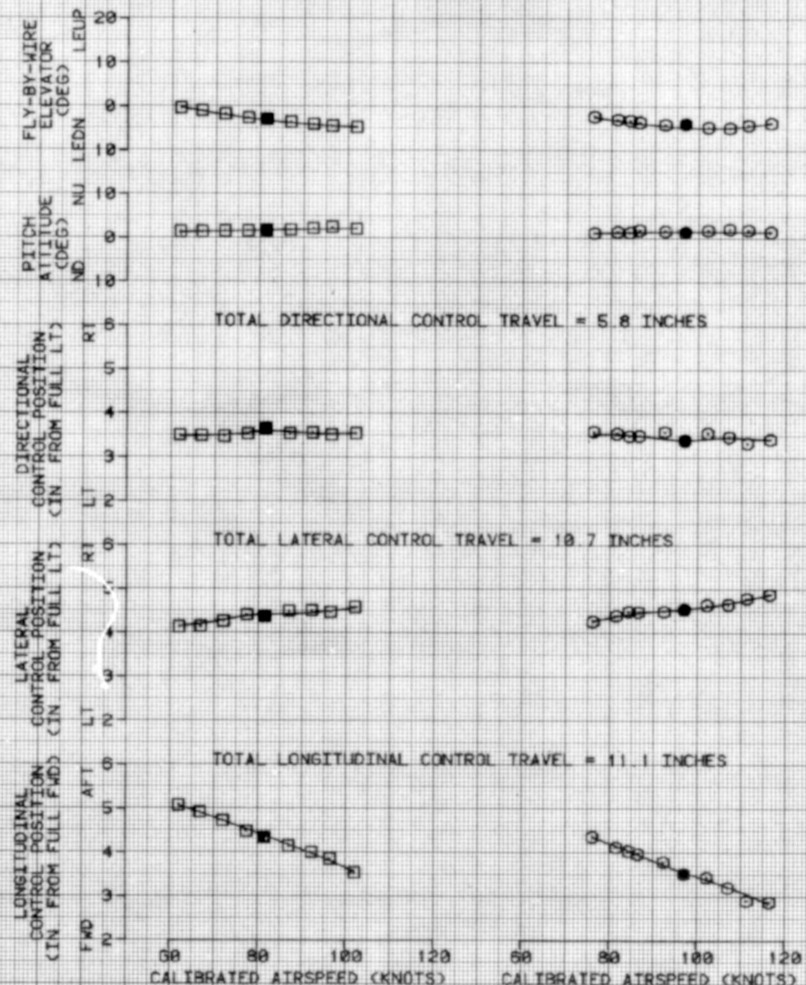


FIGURE 28
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
BHTI 214ST FAA S/N N3180W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	14840	243.8 (AFT)	0.3 LT	6130	25.8	287	CLIMB AT MAX CONTINUOUS POWER
⊙	14620	243.5 (AFT)	0.3 LT	6230	25.6	287	DESCENT AT 1000 FPM

NOTE: SHADED SYMBOLS DENOTE TRIM

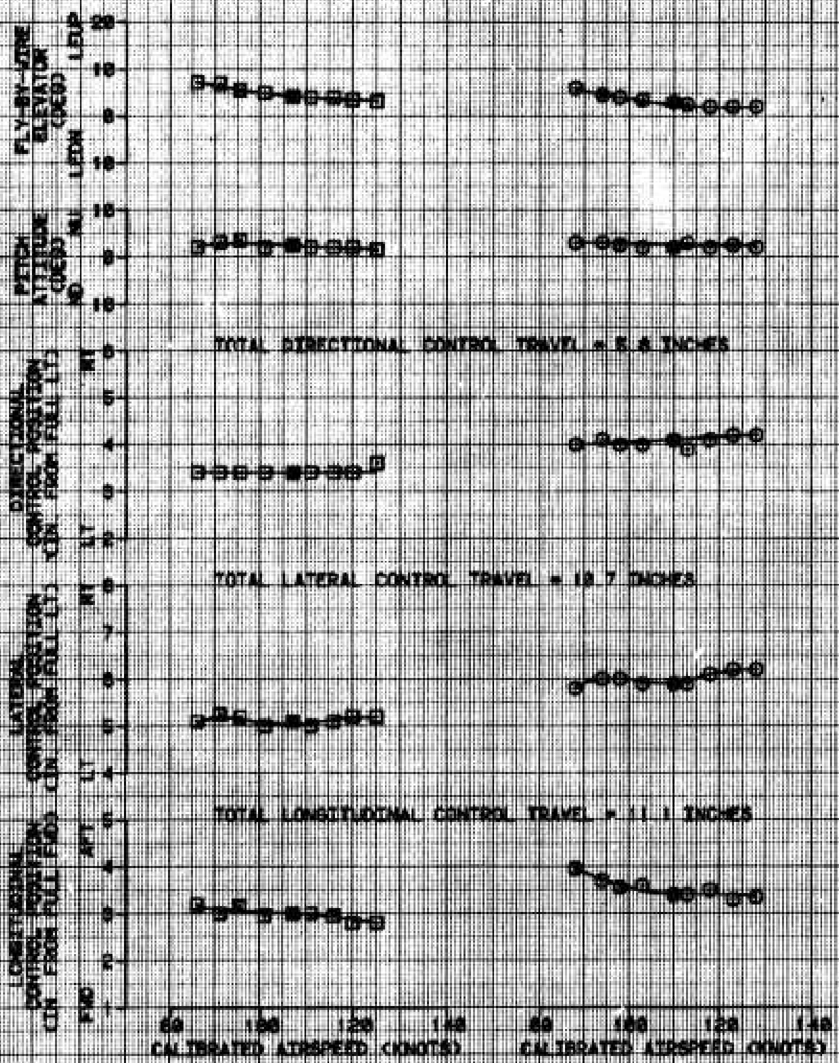


FIGURE 24
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
BHTI 214ST FAA S/N N3185W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
⊡	16500	237.2 (AFT)	0.3 LT	7510	20.0	287	CLIMB AT MAX CONTINUOUS POWER
⊙	16500	237.2(AFT)	0.3 LT	6000	21.5	287	DESCENT AT 1000 FPM

NOTE: SHADED SYMBOLS DENOTE TRIM

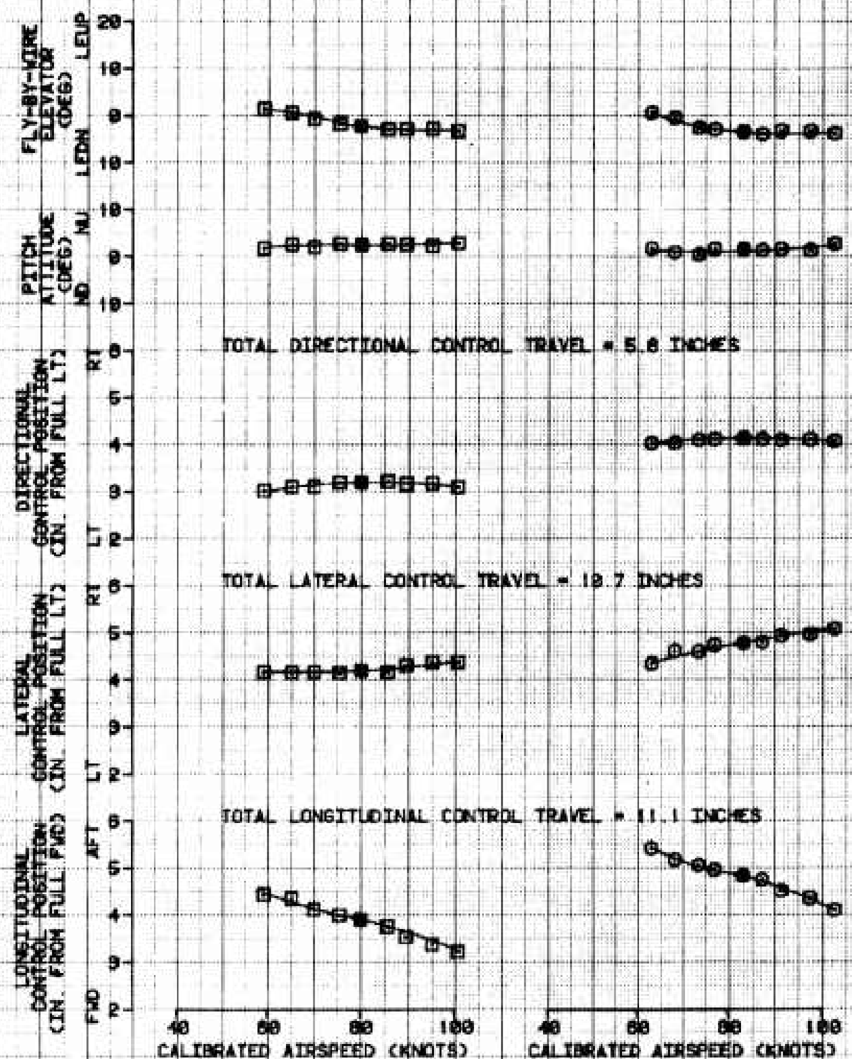


FIGURE 26
STATIC LATERAL-DIRECTIONAL STABILITY
BHTI 214ST FAA S/N N3180W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)
⊙	14600	243.8(AFT)	0.3 LT	6200	26.0	287	80
⊡	14380	243.8(AFT)	0.3 LT	6300	25.0	287	111

NOTES: 1. TRIM FLIGHT CONDITION: LEVEL
2. SHADED SYMBOL DENOTES TRIM
3. AARS OFF

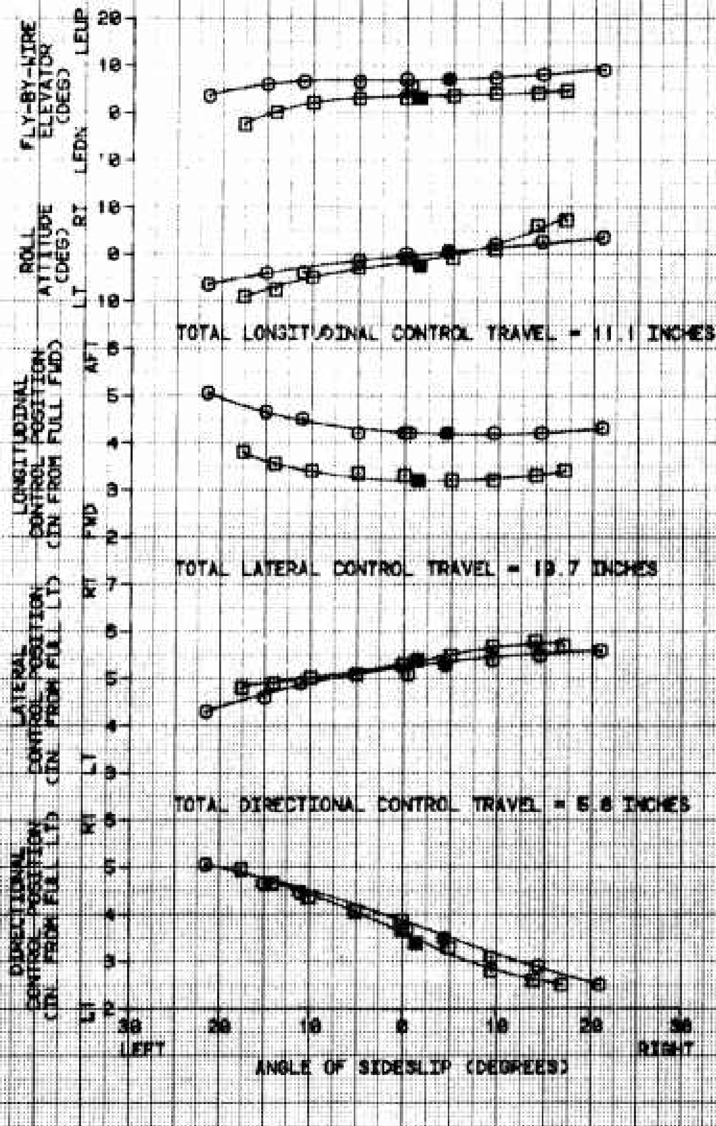


FIGURE 28
STATIC LATERAL-DIRECTIONAL STABILITY
BHTI 214ST FAA S/N N3106W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)
⊙	16840	237.2(AFT)	0.9 LT	7890	22.0	288	82
⊡	16840	237.2(AFT)	0.9 LT	8020	22.0	288	100

NOTES: 1. TRIM FLIGHT CONDITION: LEVEL
2. SHADED SYMBOL DENOTES TRIM
3. AARS OFF

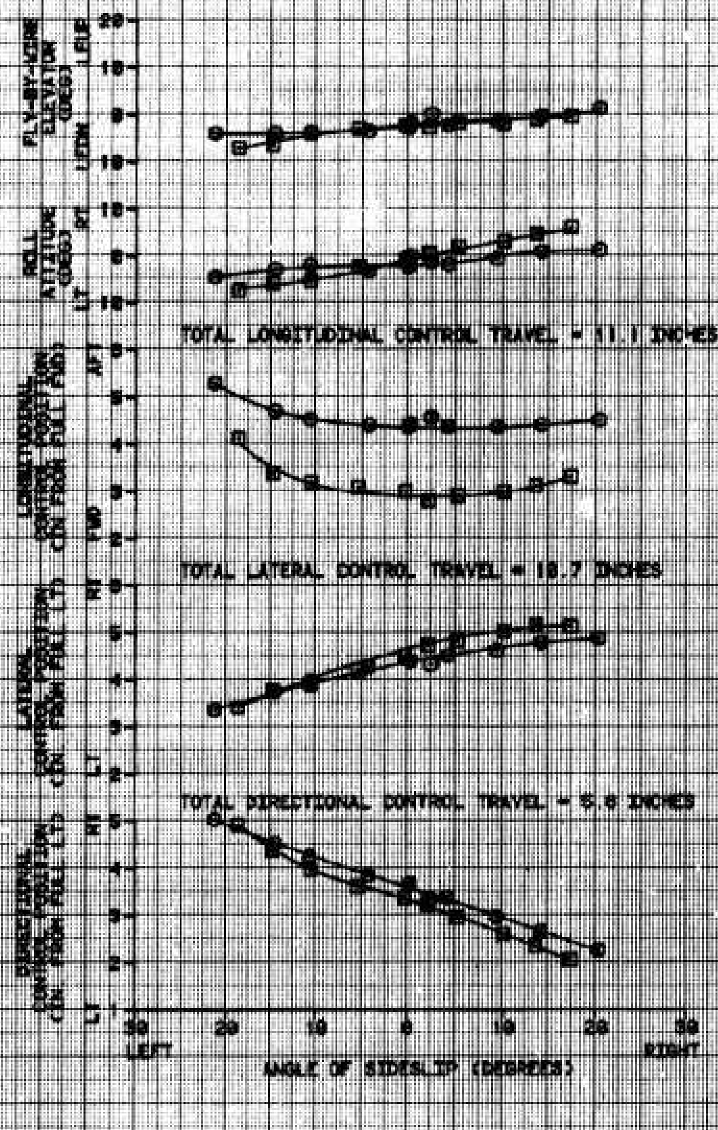


FIGURE 27
STATIC LATERAL-DIRECTIONAL STABILITY
BHTI 214ST FAA S/N N3180W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
○	16000	287.9 (AFT)	0.5 LT	5780	21.0	287	CLIMB AT MAX CONTINUOUS POWER
□	16000	287.9 (AFT)	0.5 LT	6040	20.5	287	DESCENT AT 1000 FPM

NOTES: 1. AVERAGE CALIBRATED AIRSPEED: 102 KTS
2. SHADED SYMBOL DENOTES TRIM
3. AARS OFF

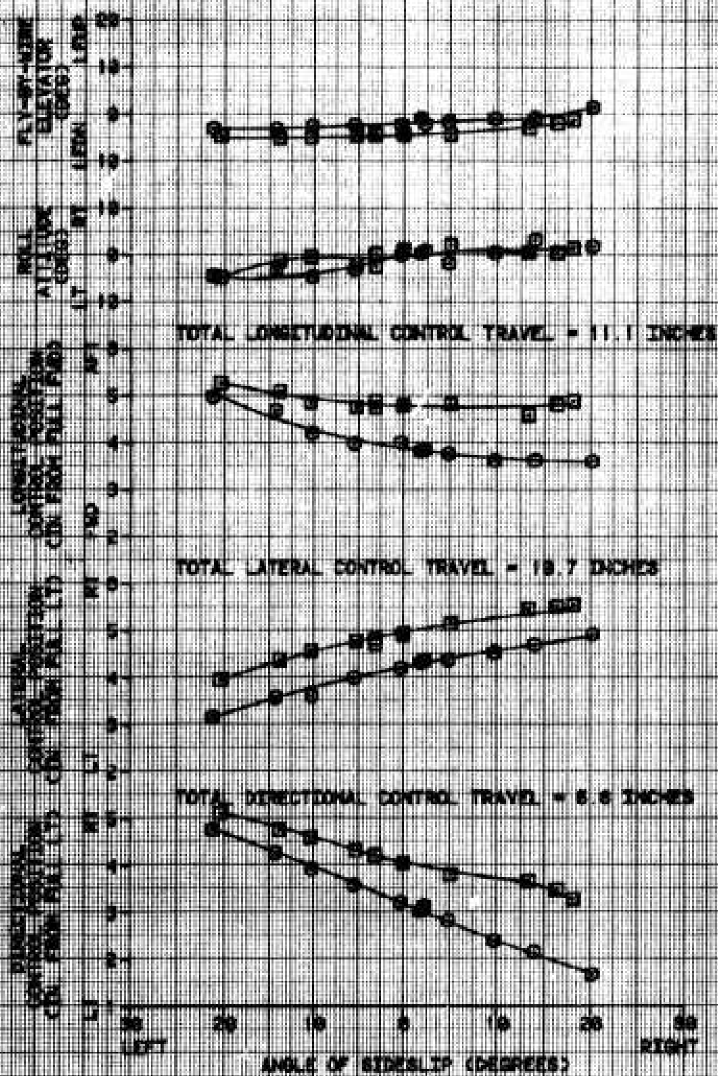


FIGURE 28
MANEUVERING STABILITY
BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
○	14340	243.8(AFT)	0.3 LT	5120	22.0	287	LT TURNS
□	14220	243.8(AFT)	0.3 LT	5320	22.0	287	RT TURNS

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. TRIM CALIBRATED AIRSPEED: 80 KTS

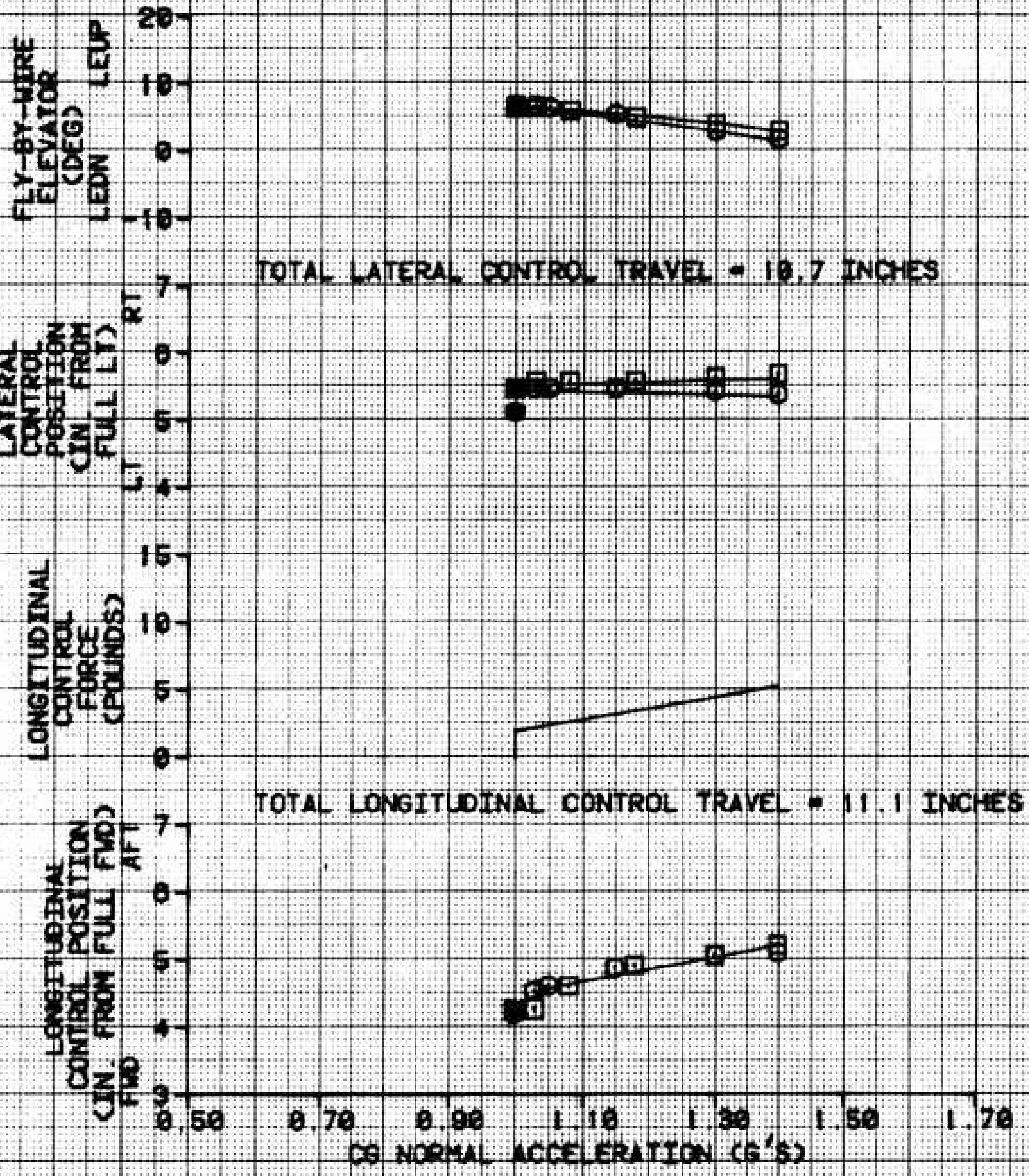


FIGURE 20
MANEUVERING STABILITY
BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
○	14600	243.8(AFT)	0.3 LT	6320	22.0	287	LT TURNS
□	14400	243.8(AFT)	0.3 LT	6200	22.0	287	RT TURNS

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. TRIM CALIBRATED AIRSPEED: 111 KTS

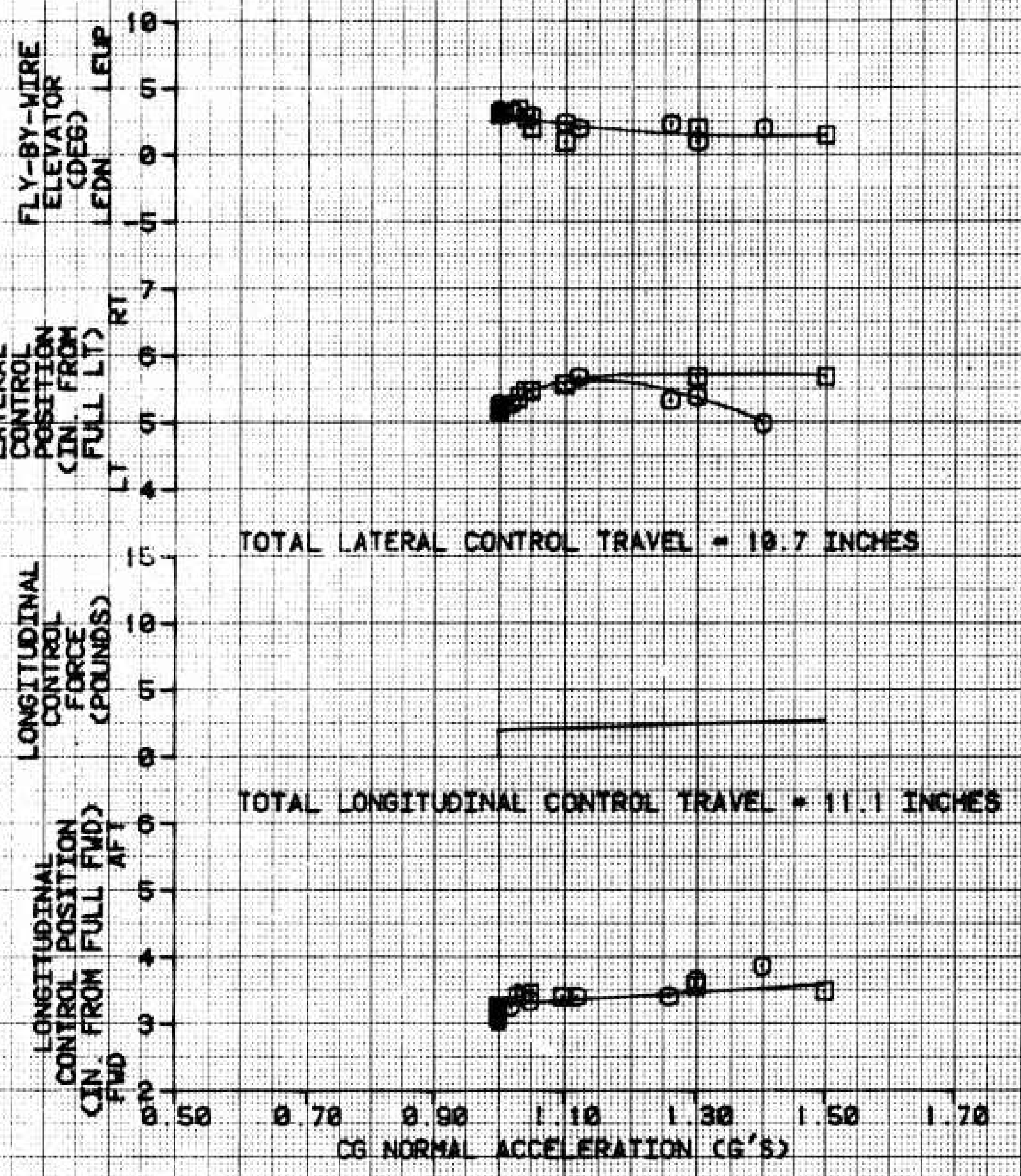


FIGURE 30
MANEUVERING STABILITY
BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION
○	17100	237.2(AFT)	0.3 LT	6900	22.0	287	LT TURNS
□	16960	237.2(AFT)	0.3 LT	6400	22.0	287	RT TURNS

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. TRIM CALIBRATED AIRSPEED: 90 KTS

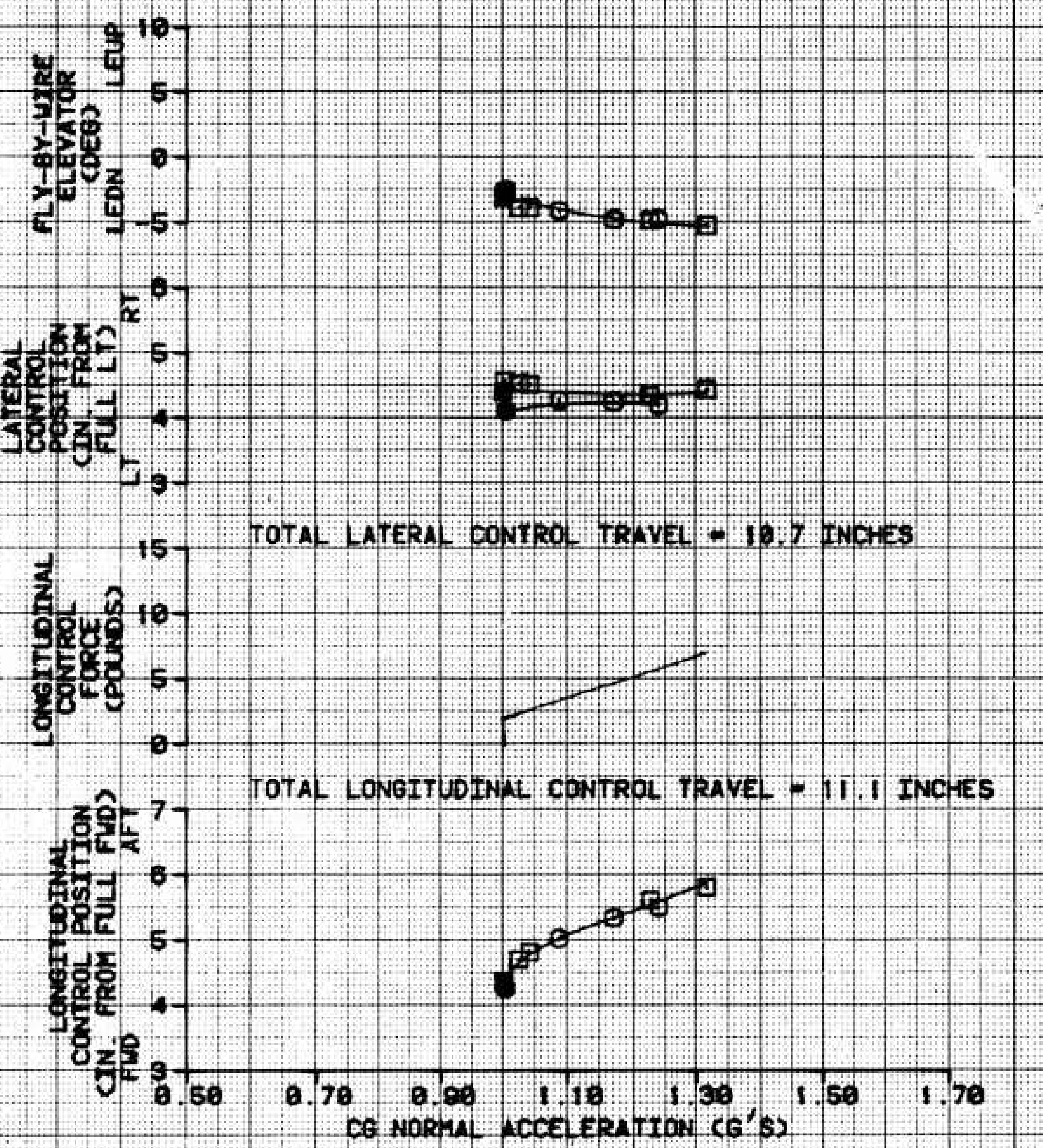


FIGURE 31
AFT LONGITUDINAL PULSE
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
16730	237.5(AFT)	0.3 LT	7520	20.0	287	102	LEVEL

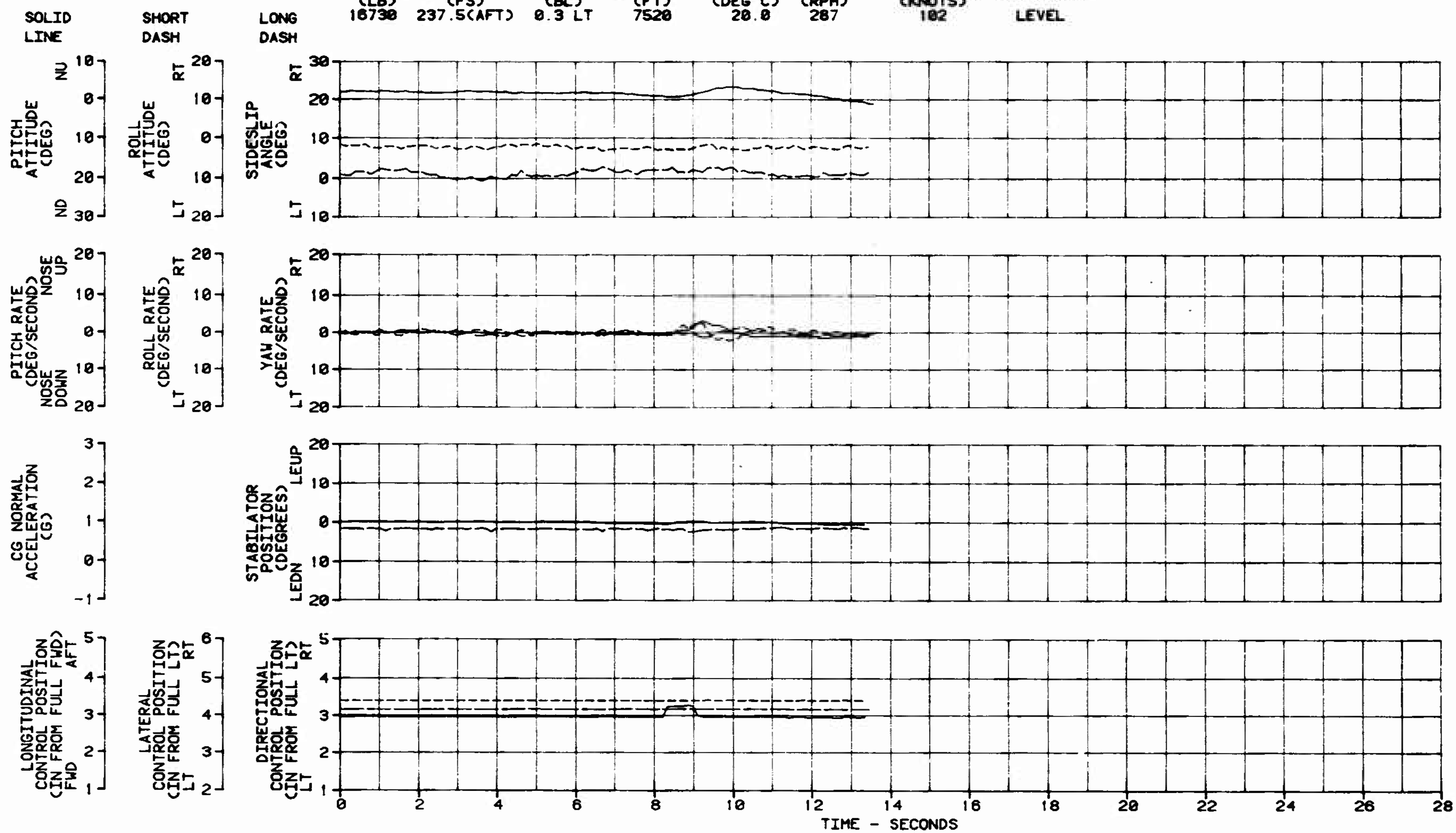


FIGURE 32
RELEASE FROM STEADY HEADING SIDESLIP
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
16730	237.5(AFT)	0.3 LT	7520	20.0	287	102	LEVEL

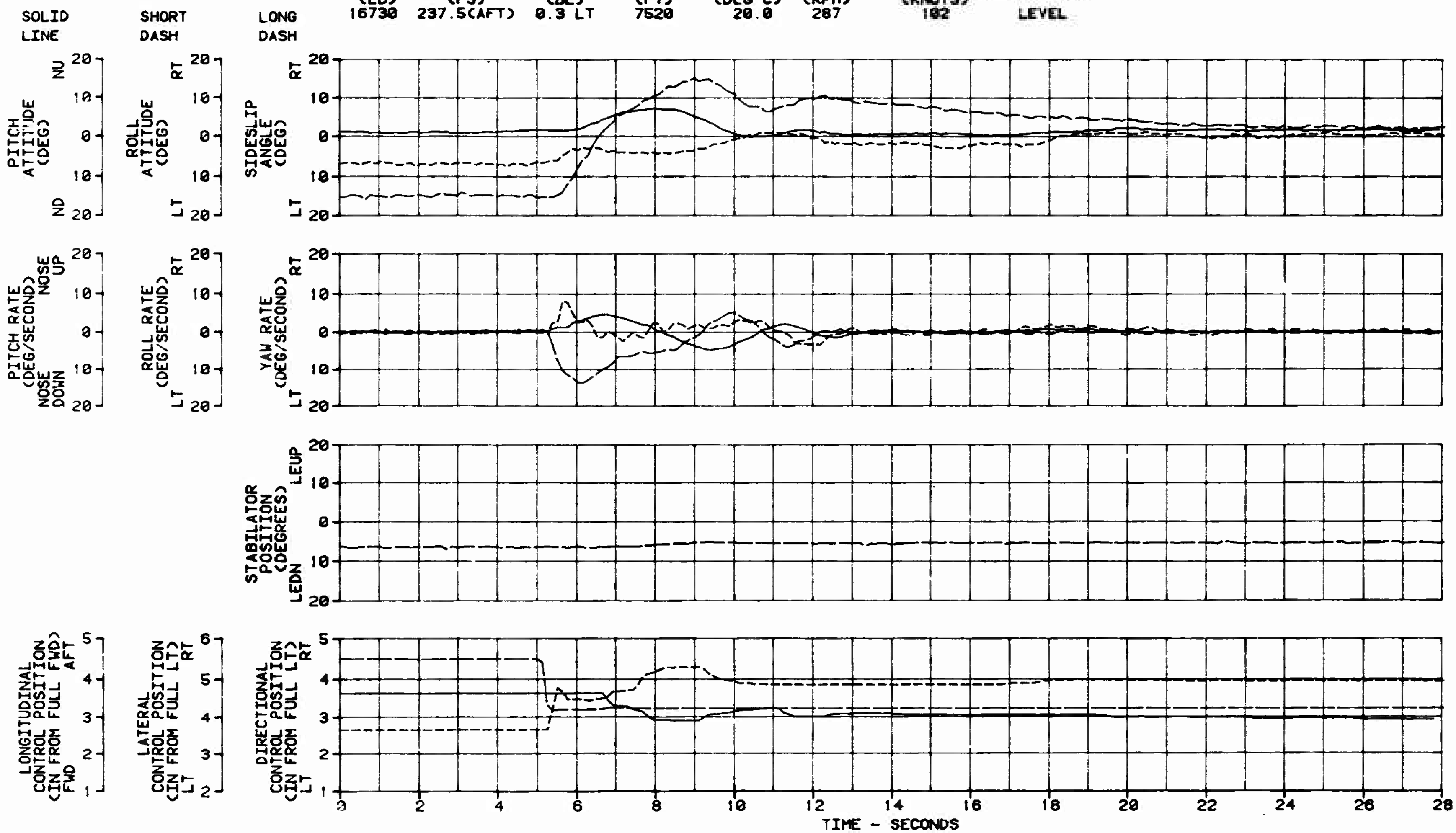
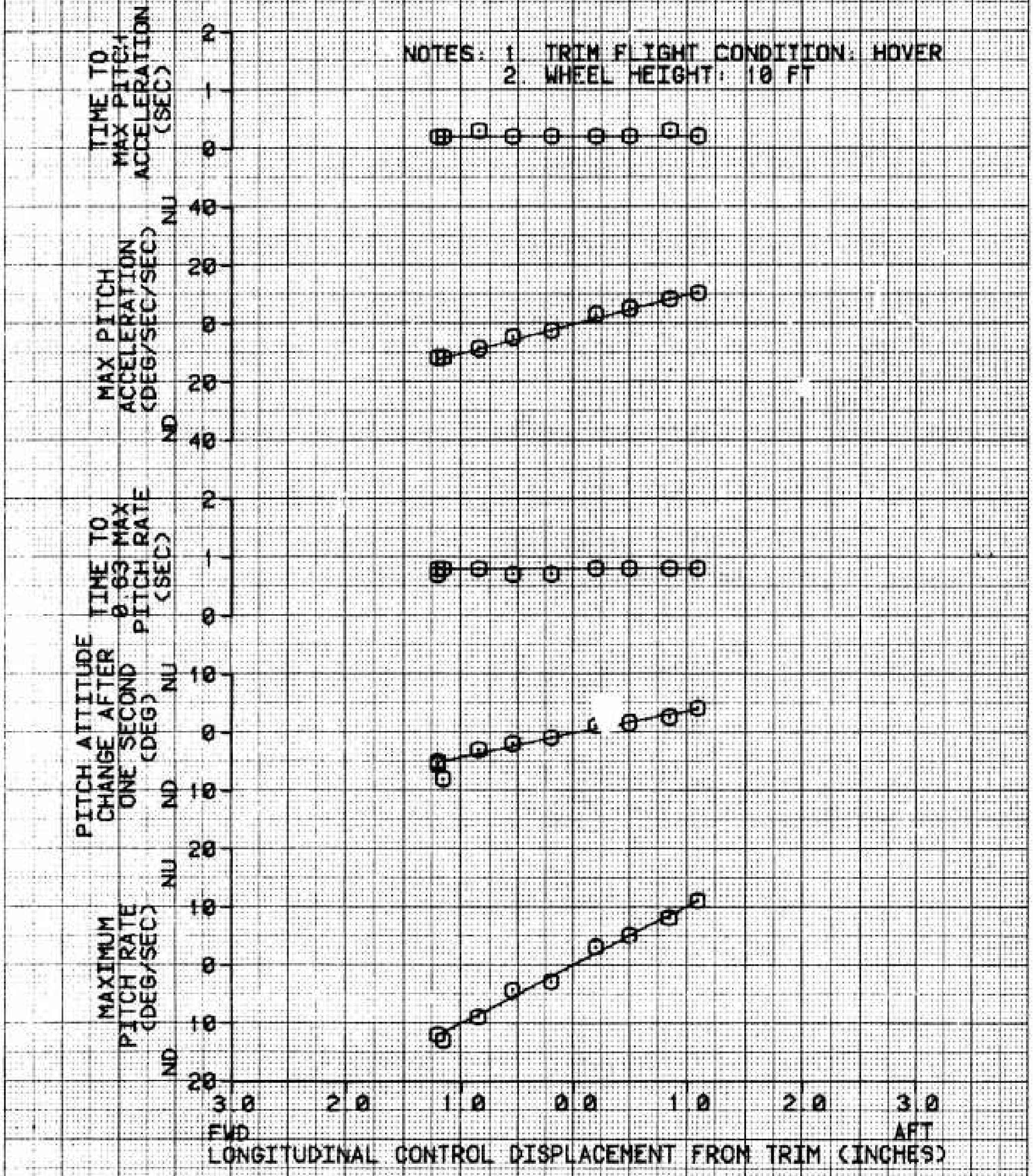


FIGURE 33
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)
14170	244.2(AFT)	0.3 LT	4500	12.0	287	ZERO



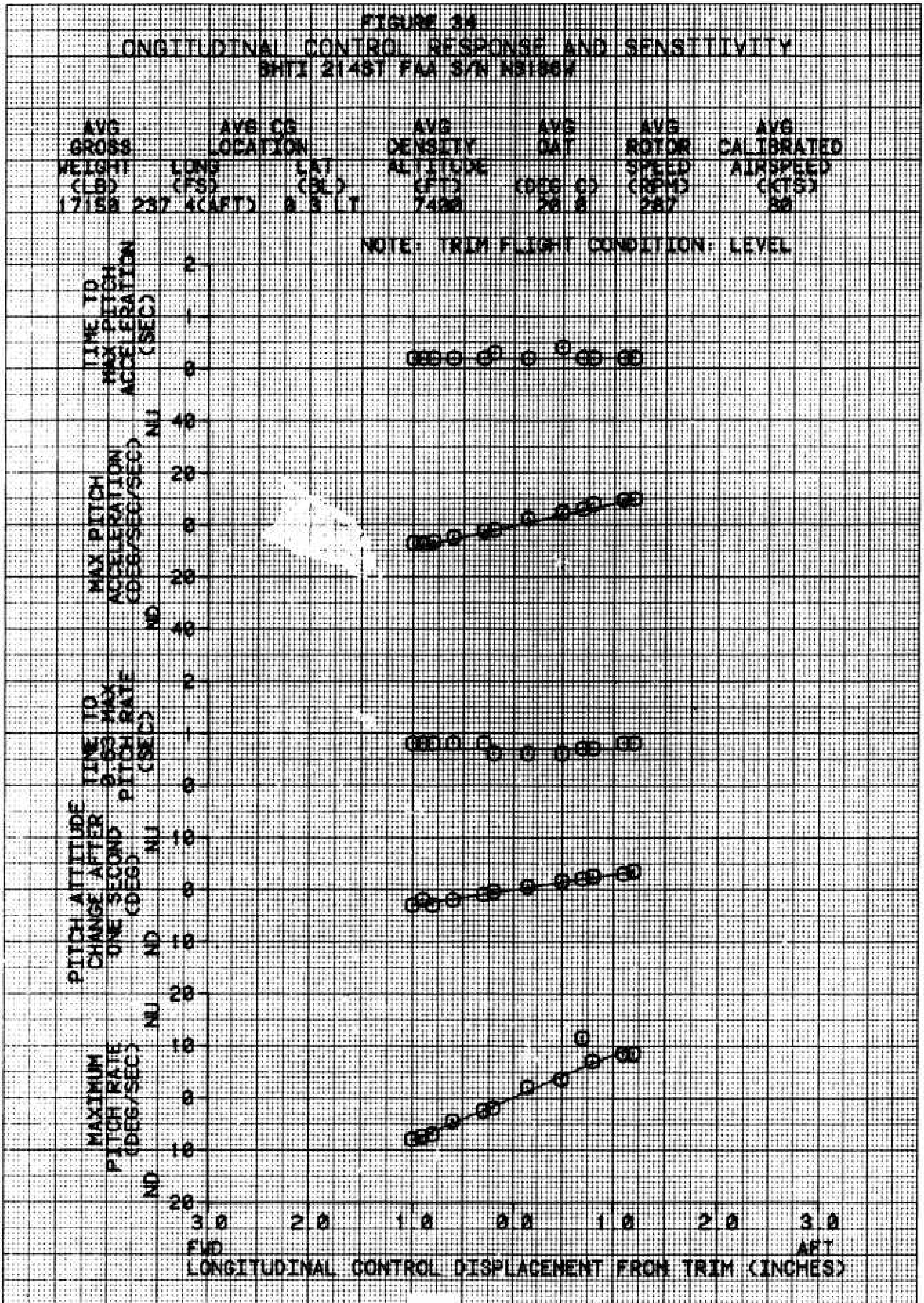


FIGURE 34
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3186W
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
AVG CALIBRATED AIRSPEED (KTS)
17150
237.4(AFT)
0.3 LT
7400
20.0
287
80
NOTE: TRIM FLIGHT CONDITION: LEVEL
TIME TO MAX PITCH ACCELERATION (SEC)
MAX PITCH ACCELERATION (DEG/SEC/SEC)
TIME TO 0.63 MAX PITCH RATE (SEC)
PITCH ATTITUDE CHANGE AFTER ONE SECOND (DEG)
MAXIMUM PITCH RATE (DEG/SEC)
NU
ND
FWD
AFT
LONGITUDINAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

FIGURE 35
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N NS196W

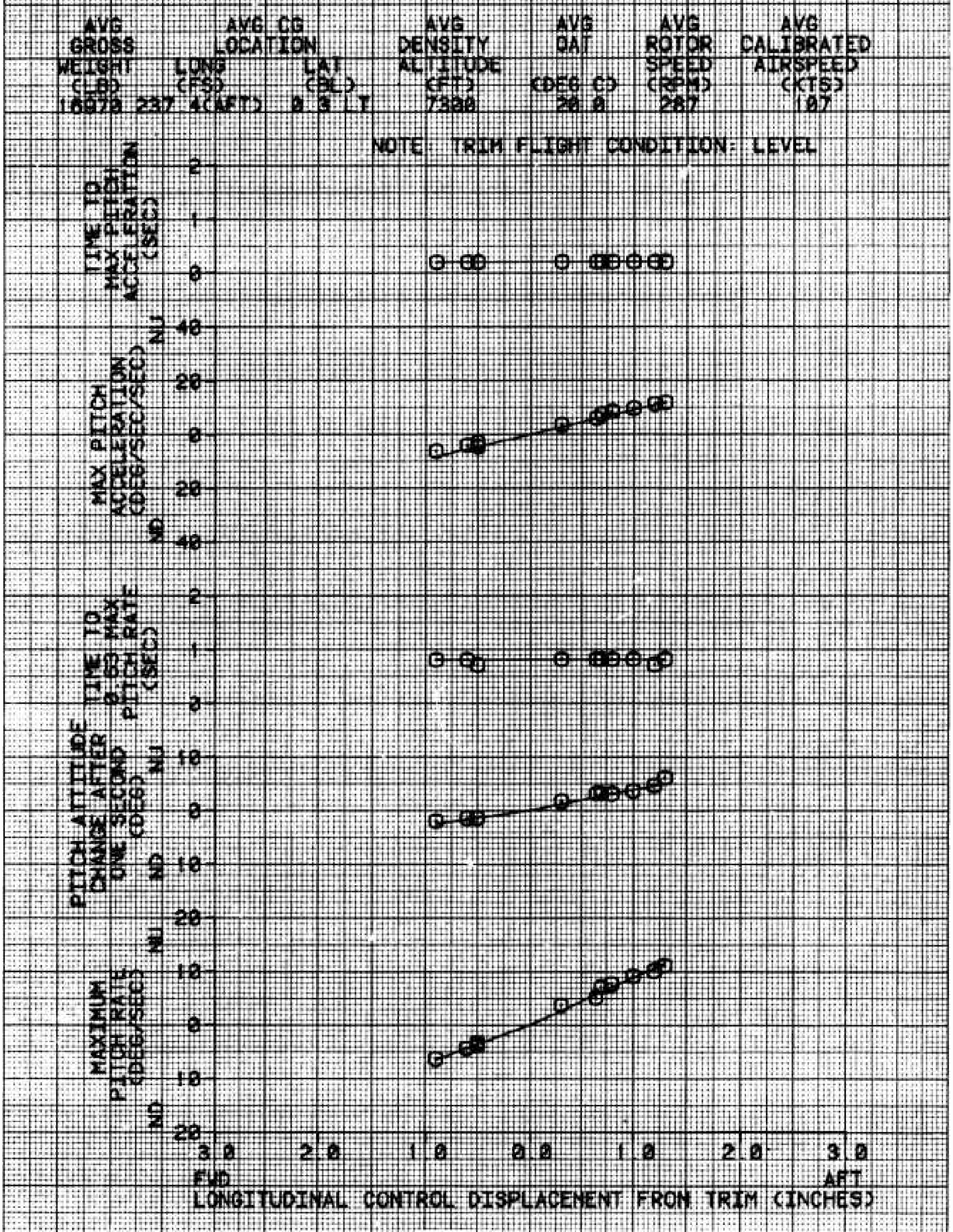


FIGURE 36
FORWARD LONGITUDINAL STEP
BHTI 214ST FAA S/N N3180W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
17180	237.8(AFT)	0.3 LT	7220	20.0	287	80	LEVEL

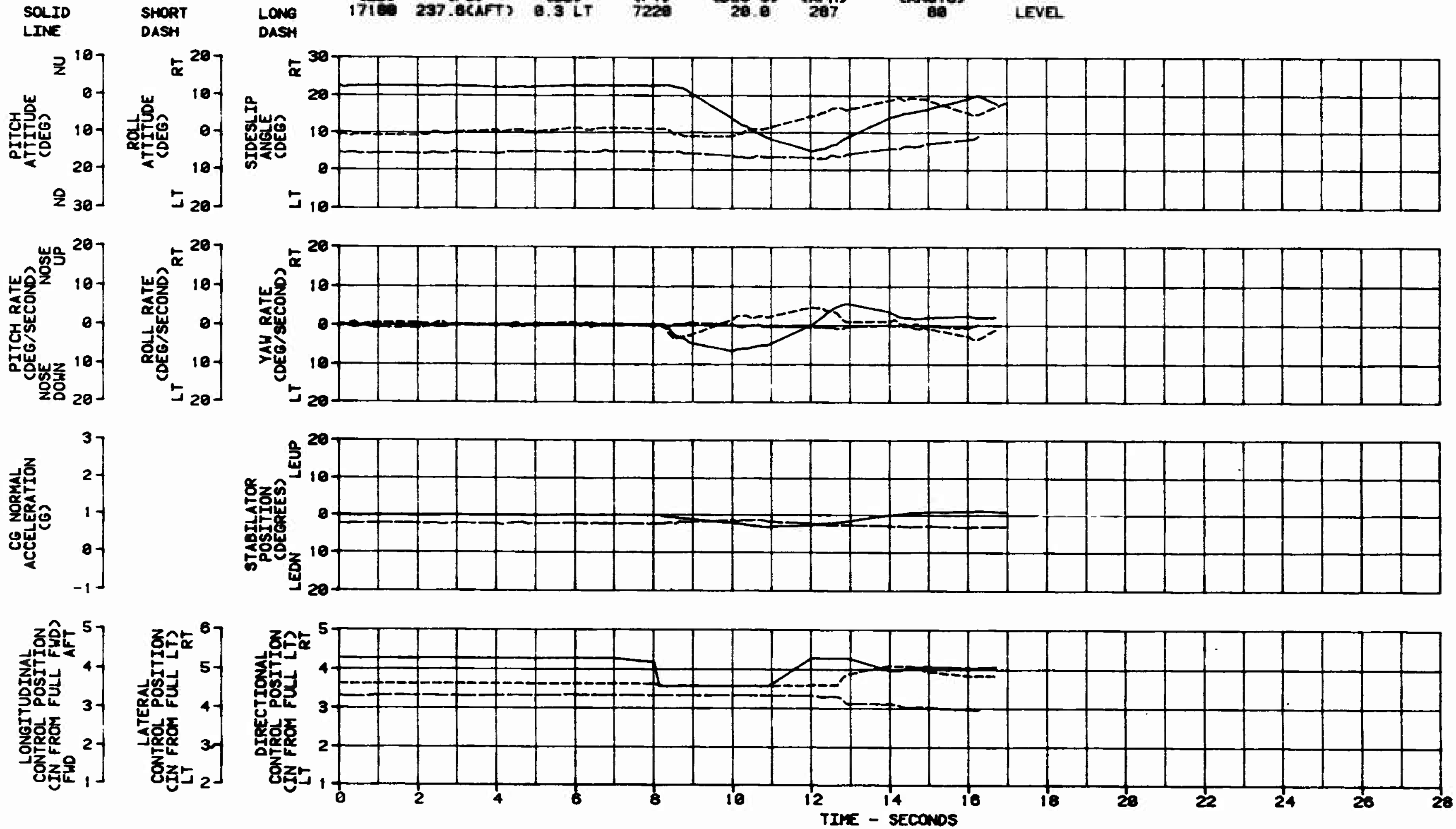


FIGURE 37
AFT LONGITUDINAL STEP
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
17300	237.6(AFT)	0.3 LT	7320	20.0	287	80	LEVEL

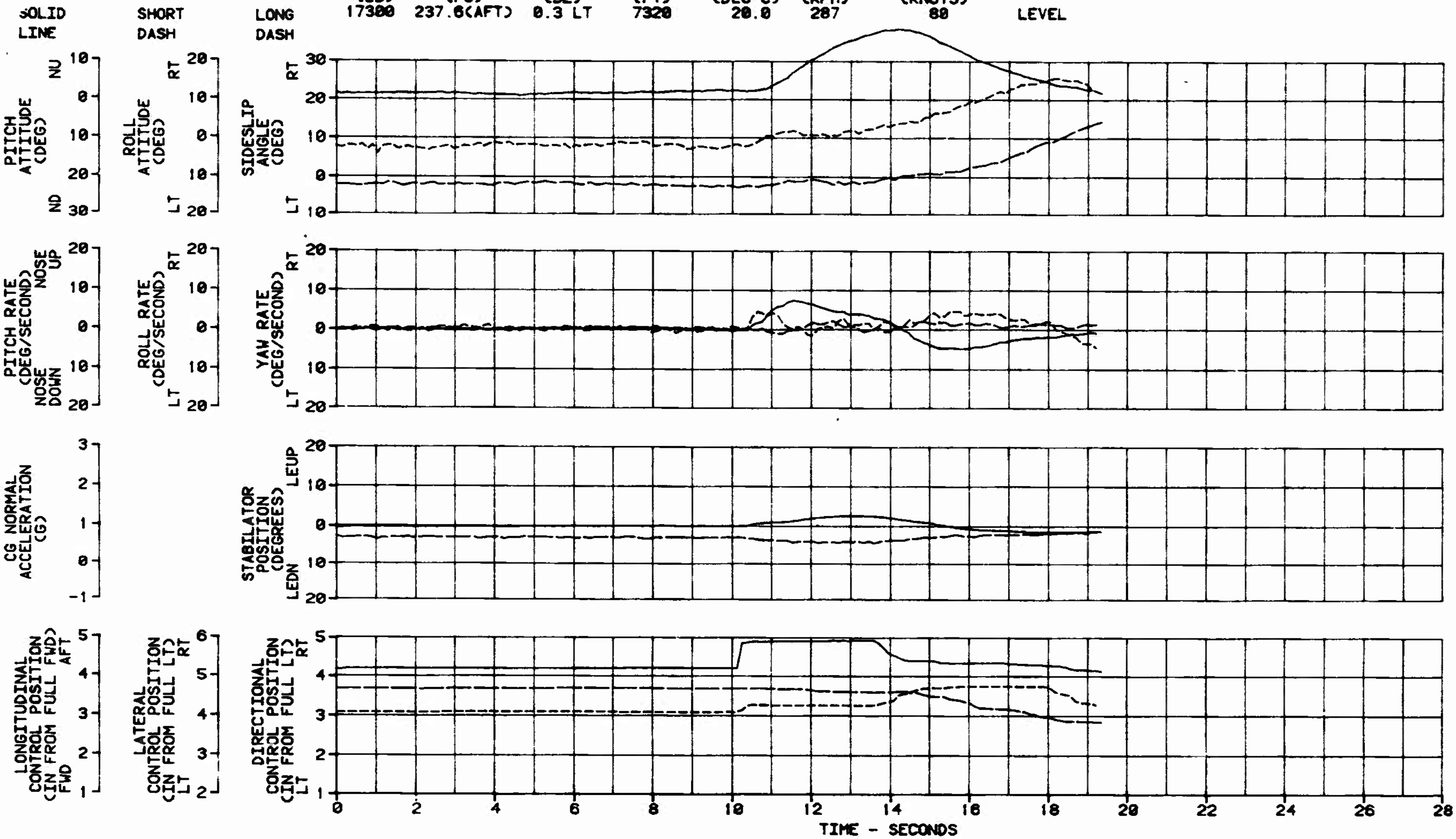


FIGURE 38

LATERAL CONTROL RESPONSE AND SENSITIVITY

BHTI 214ST FAA S/N N3186W

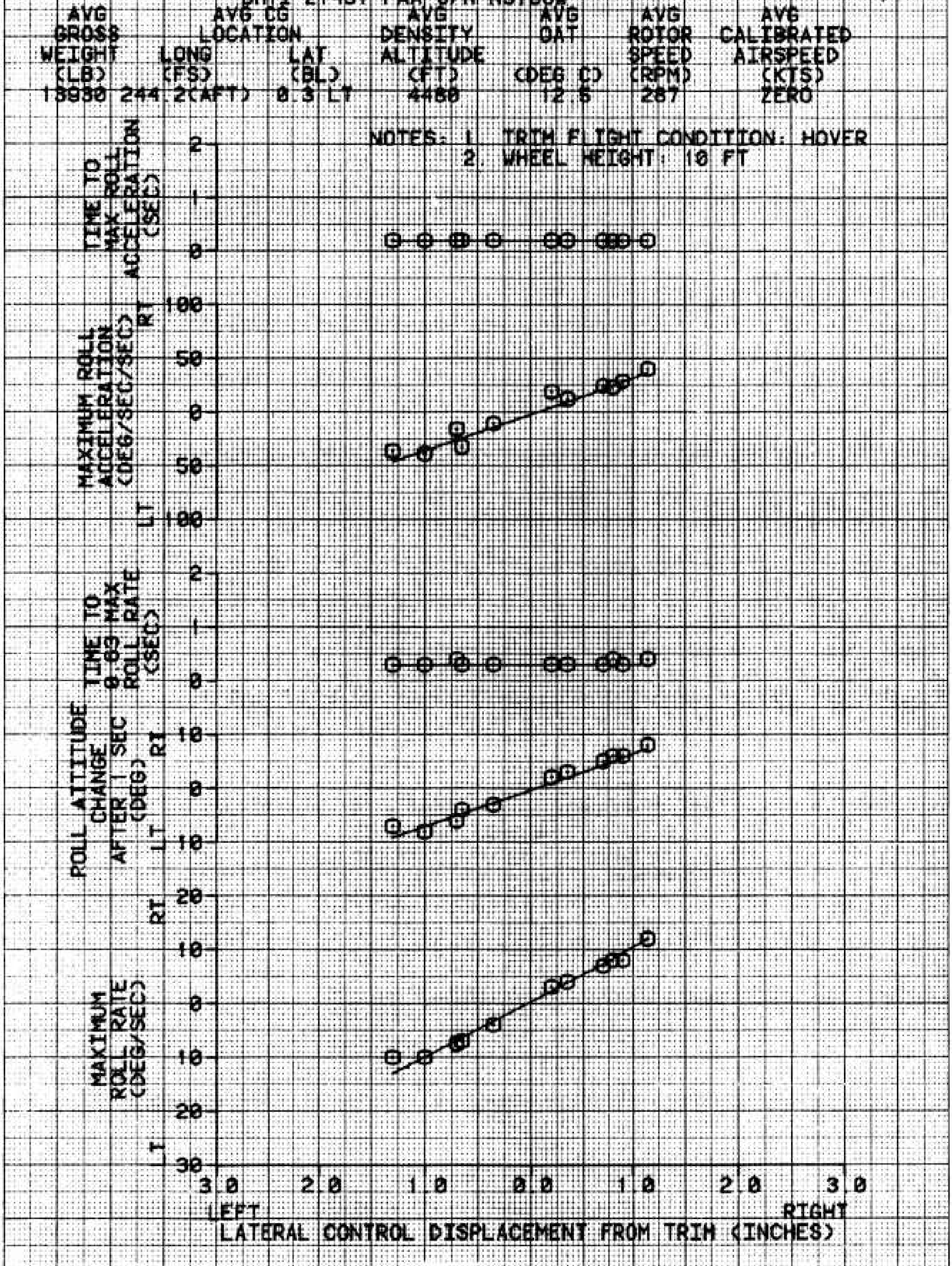
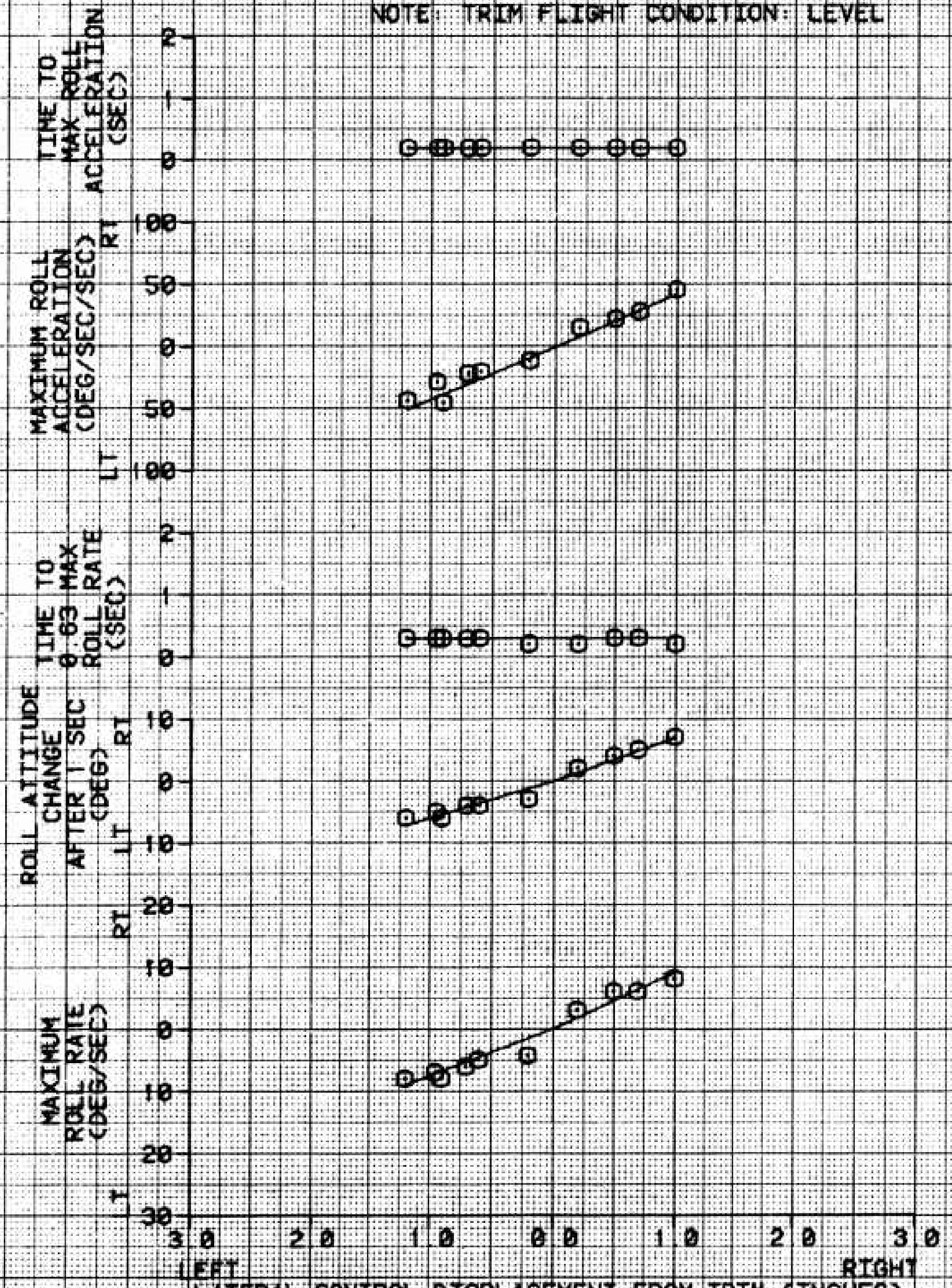


FIGURE 39
LATERAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)
17050	237.4(AFT)	0.3 LT	7420	20.5	287	80

NOTE: TRIM FLIGHT CONDITION: LEVEL



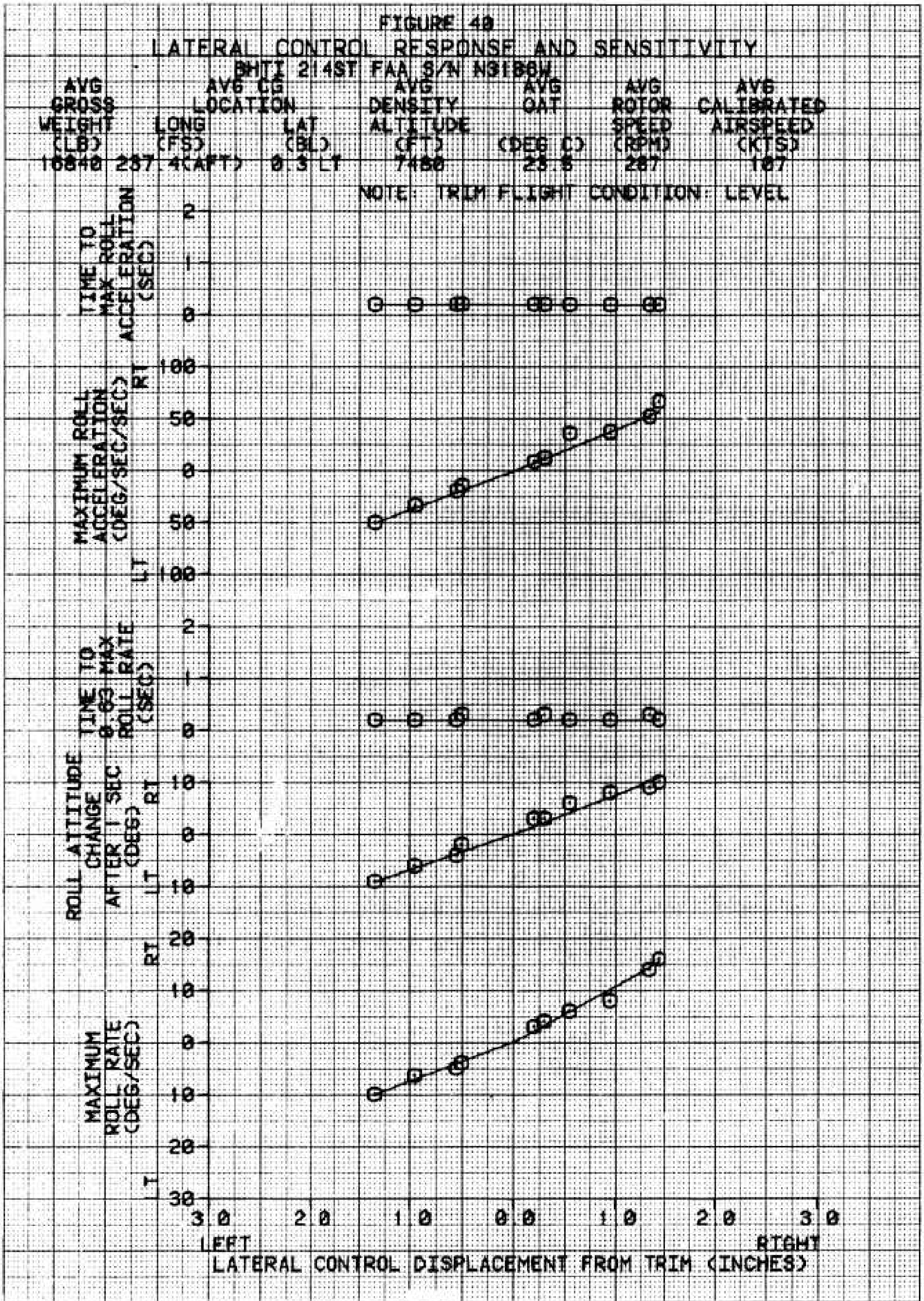


FIGURE 40
LATERAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3186W
AVG GROSS WEIGHT (LB) 16840
AVG CG LOCATION LONG (FS) 237.4(AFT) LAT (BL) 0.3 LT
AVG DENSITY ALTITUDE (FT) 7480
AVG OAT (DEG C) 23.5
AVG ROTOR SPEED (RPM) 287
AVG CALIBRATED AIRSPEED (KTS) 107
NOTE: TRIM FLIGHT CONDITION: LEVEL
TIME TO MAX ROLL ACCELERATION (SEC)
MAXIMUM ROLL ACCELERATION (DEG/SEC/SEC)
RT
LT
TIME TO 0.63 MAX ROLL RATE (SEC)
ROLL ATTITUDE CHANGE AFTER 1 SEC (DEG)
MAXIMUM ROLL RATE (DEG/SEC)
LEFT
RIGHT
LATERAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

FIGURE 41
LEFT LATERAL STEP
BHTI 214ST FAA S/N N3180W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
17100	237.5(AFT)	0.3 LT	7320	20.0	287	80	LEVEL

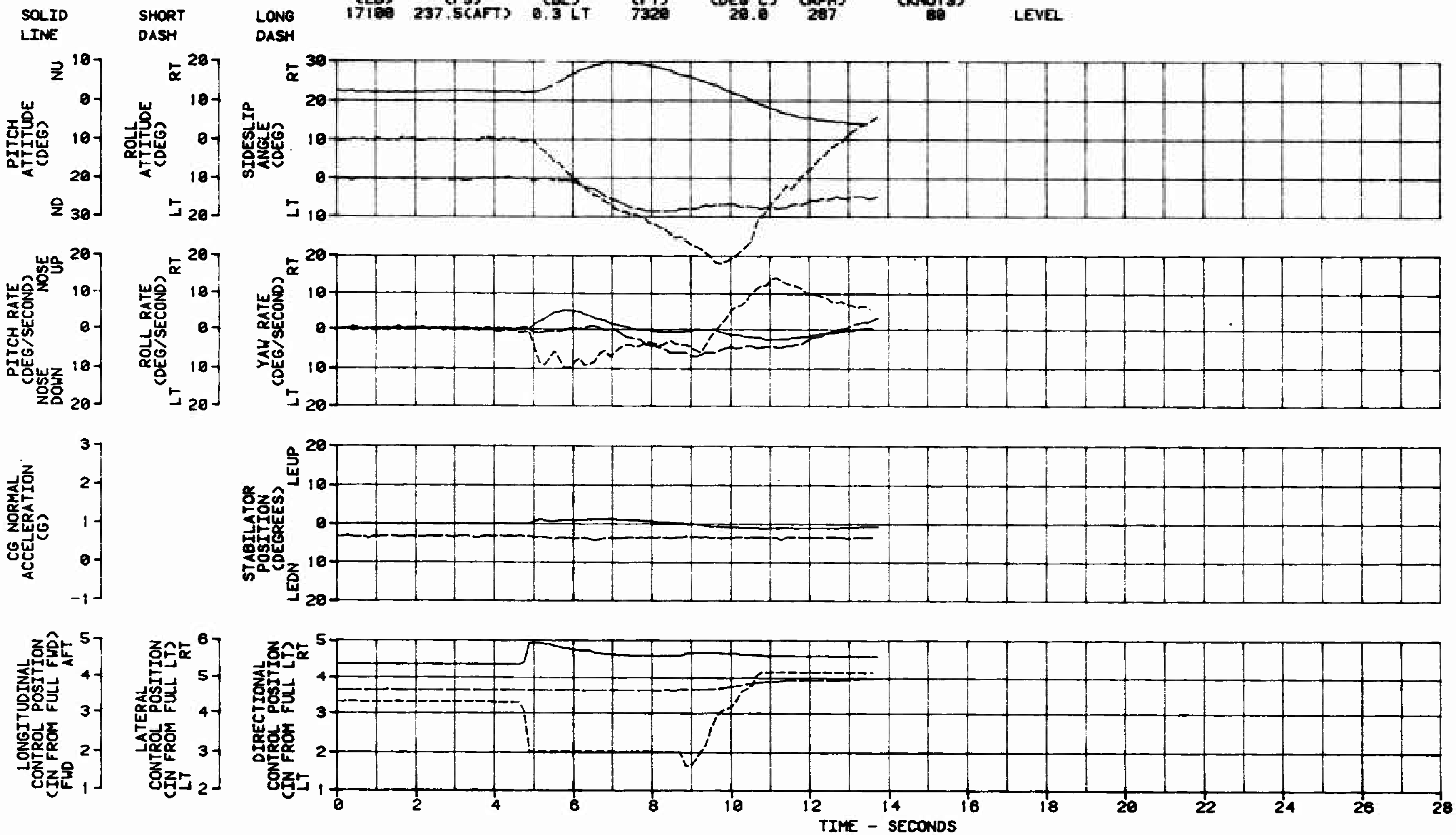


FIGURE 42
RIGHT LATERAL STEP
BHTI 214ST FAA S/N N3180W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
17140	237.5(AFT)	0.3 LT	7200	20.0	287	80	LEVEL

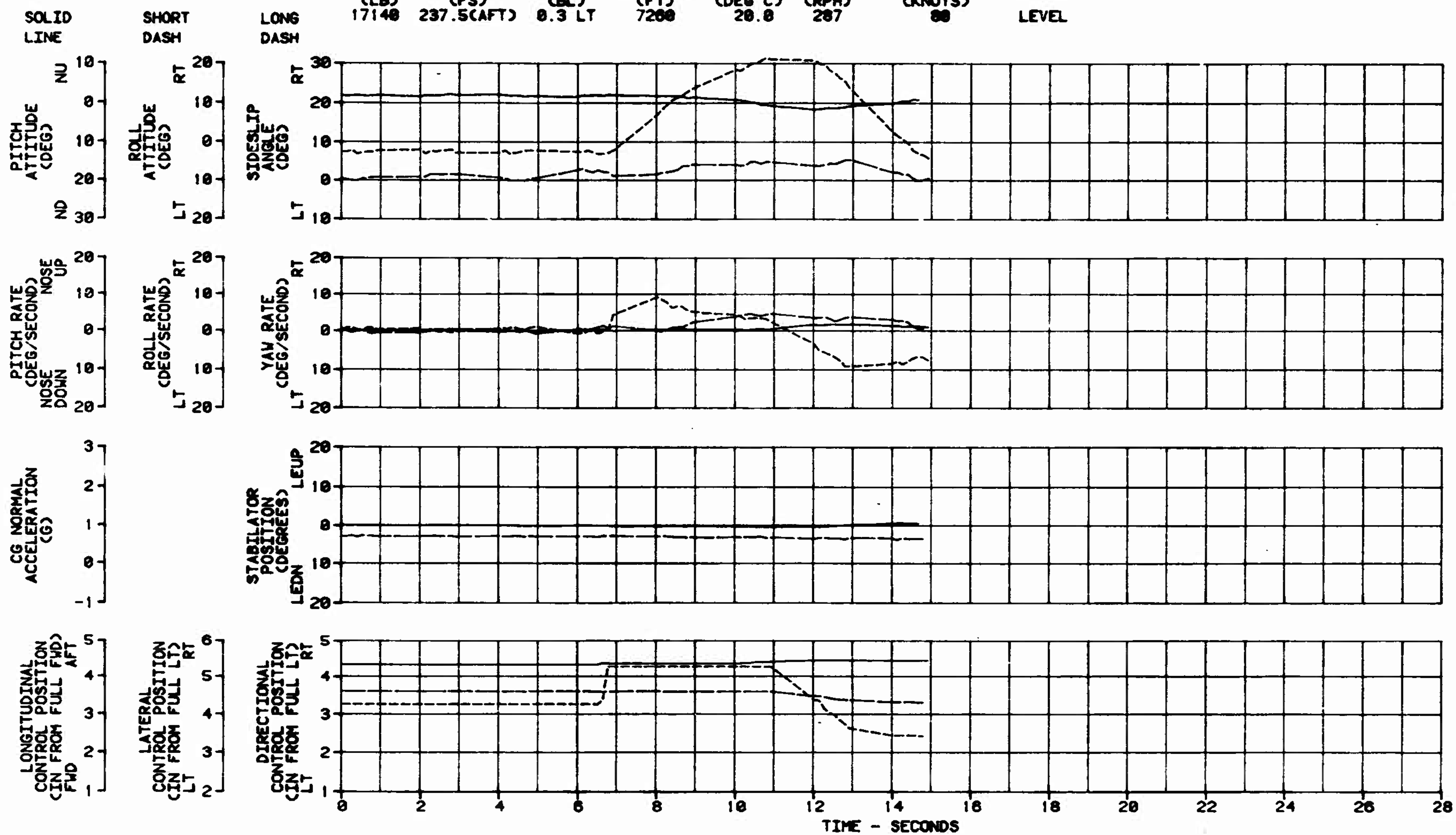


FIGURE 43
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)
13800	244.2(AFT)	0.3 LT	4520	12.0	287	ZERO

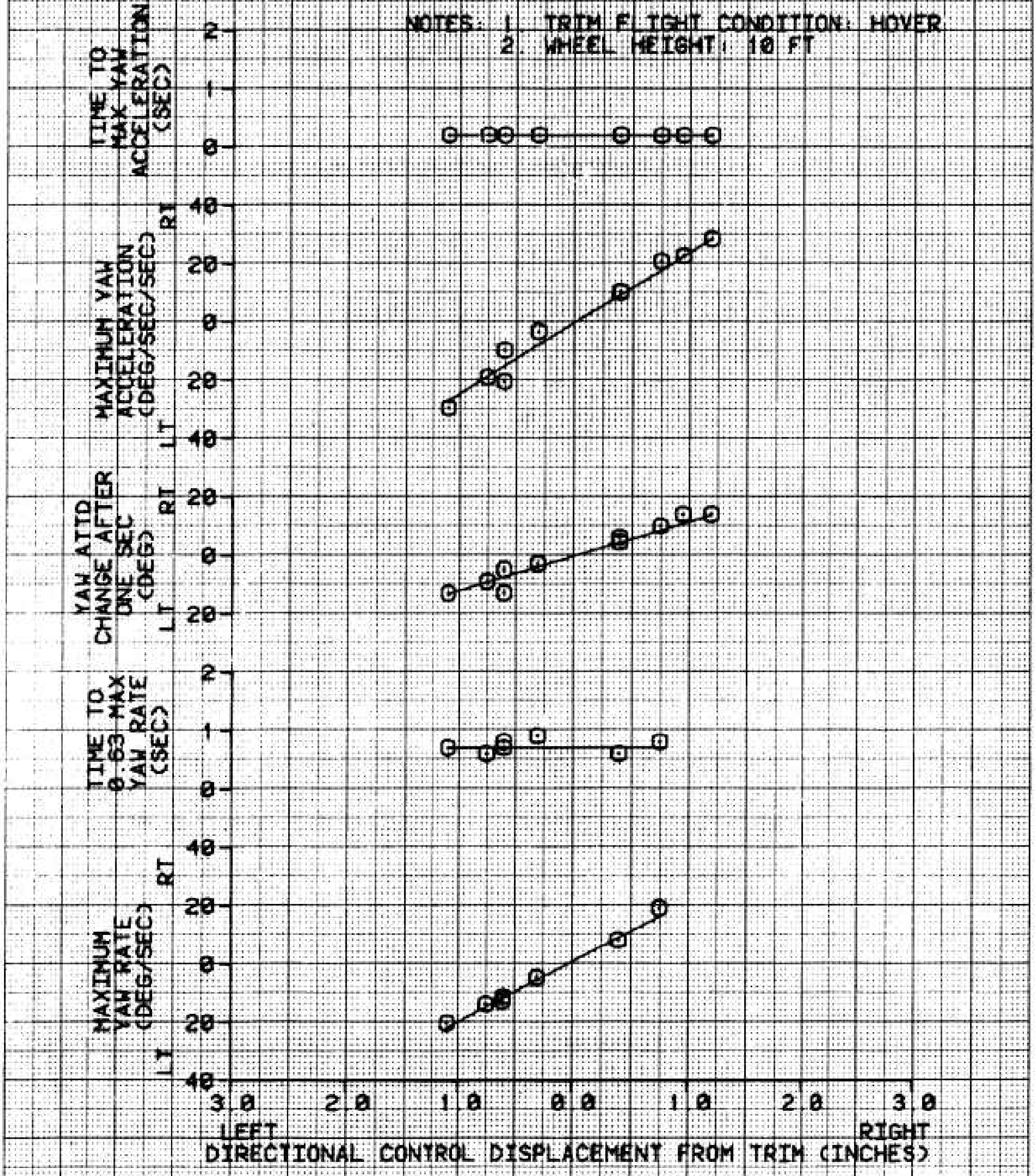
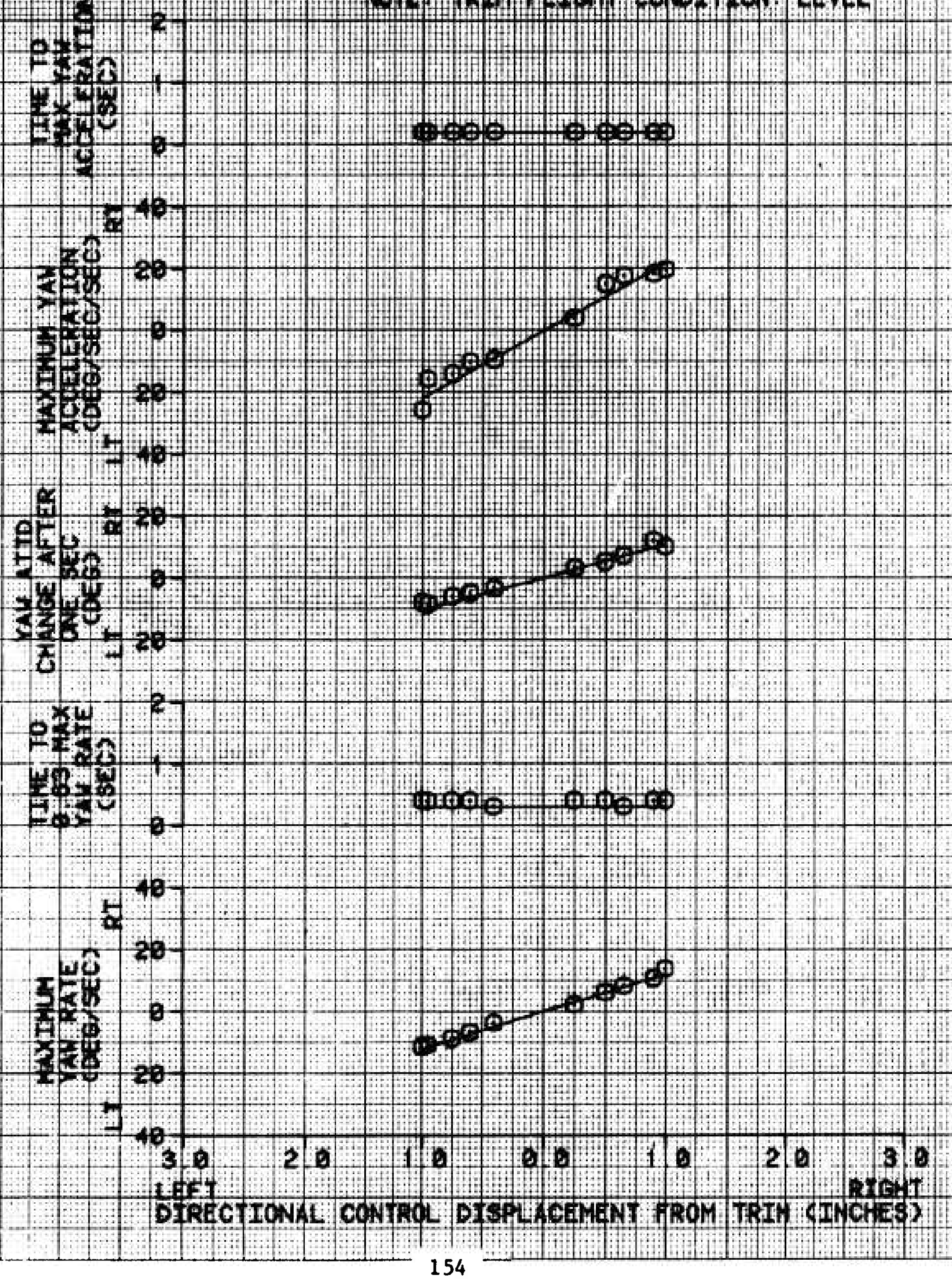


FIGURE 44
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3106W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG CALIBRATED AIRSPEED (KTS)
17160	237.3(AFT)	0.3 LT	6000	21.0	287	80

NOTE: TRIM FLIGHT CONDITION: LEVEL



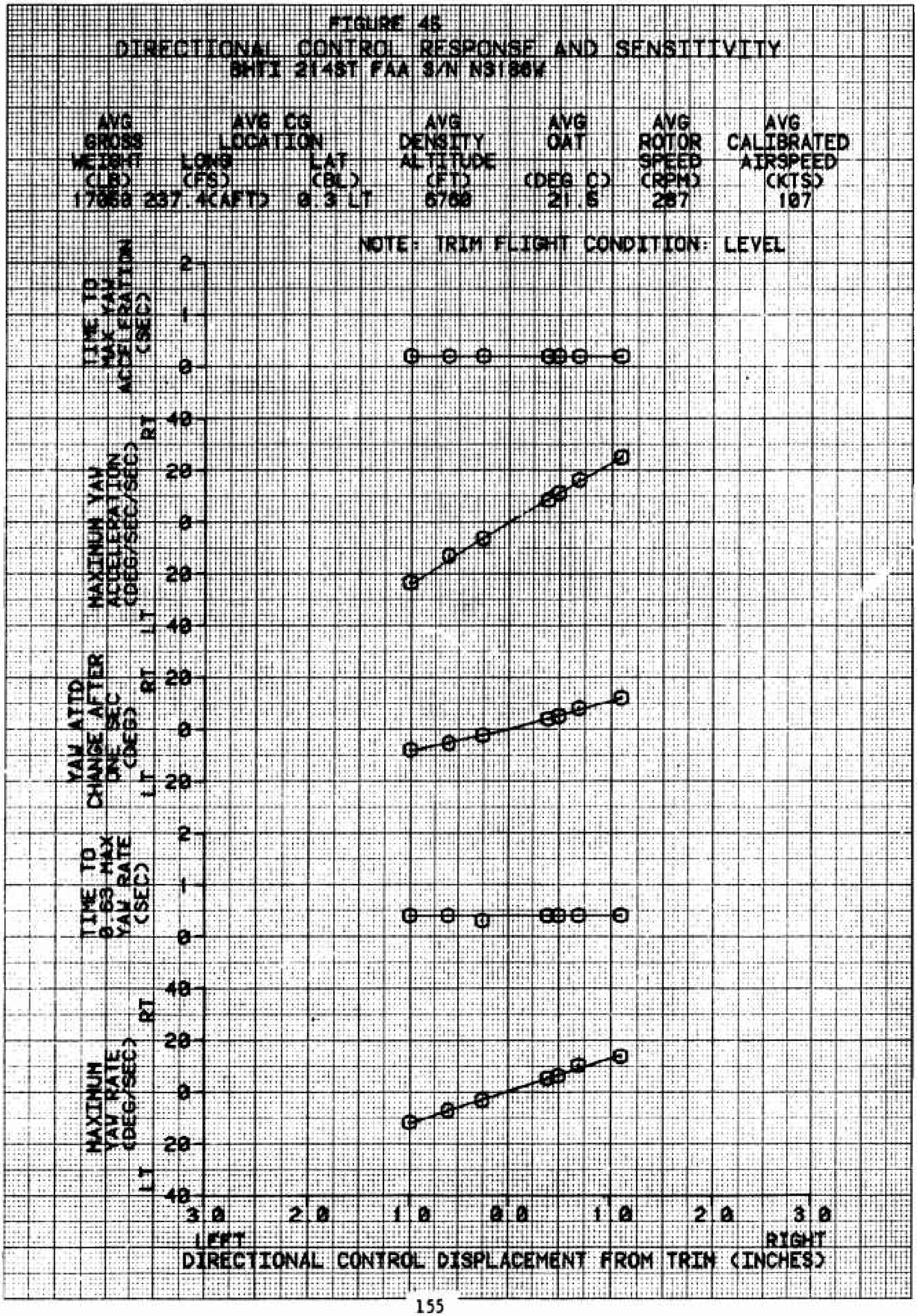


FIGURE 45
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
BHTI 214ST FAA S/N N3186W

FIGURE 46
LEFT DIRECTIONAL STEP
BHTI 214ST FAA S/N N3106W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
17200	237.5(AFT)	0.3 LT	7420	20.0	287	60	LEVEL

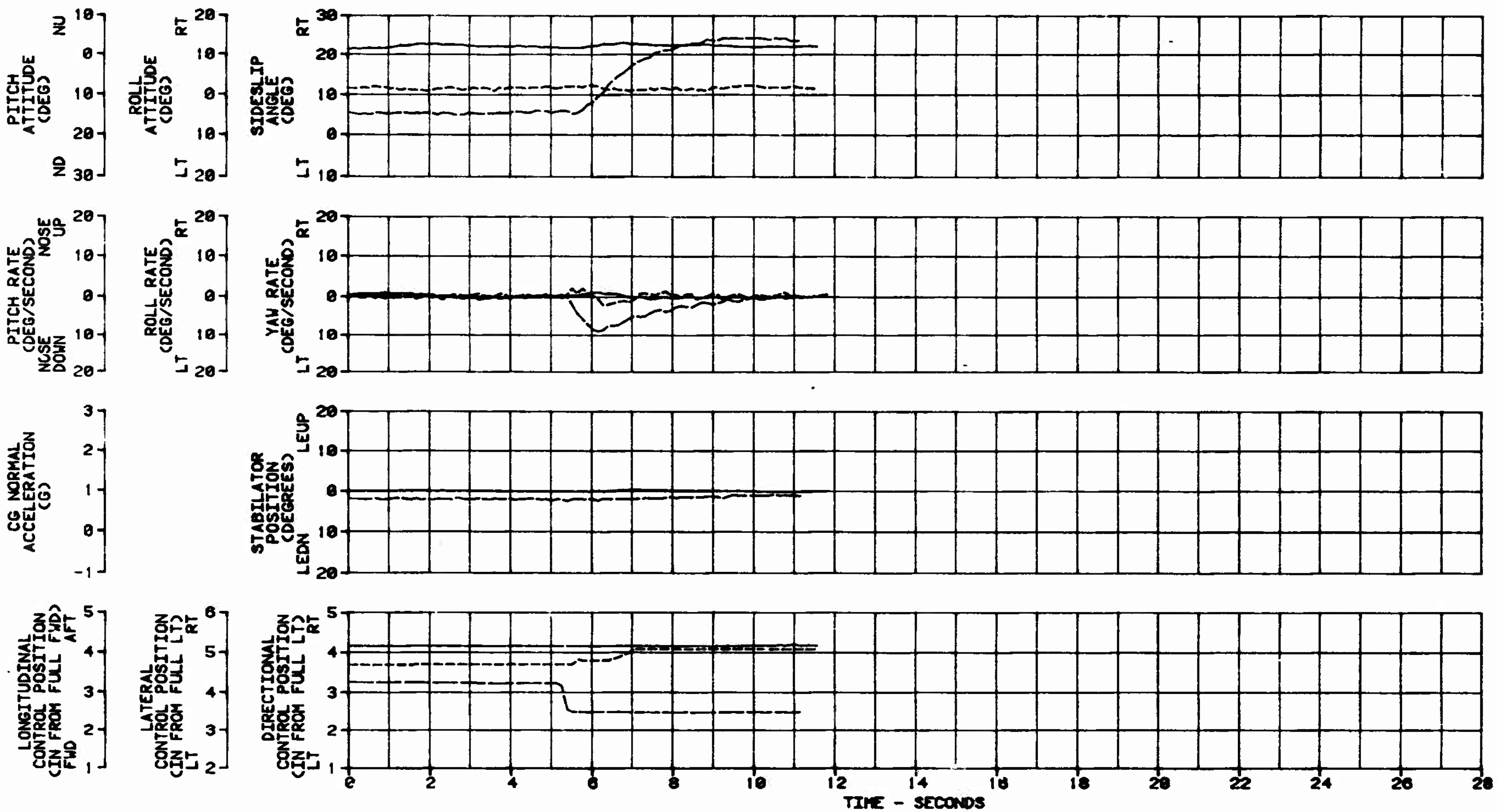
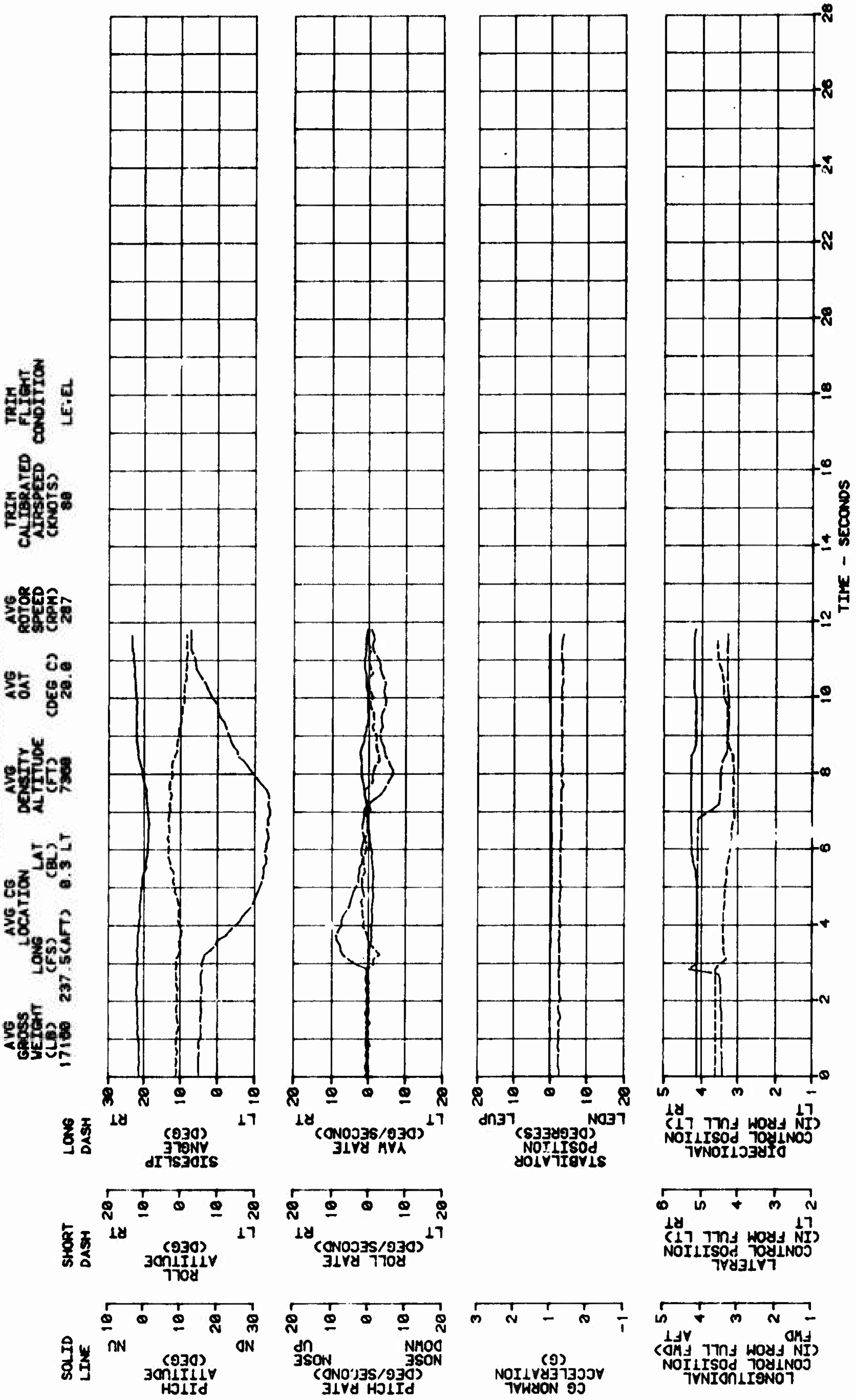
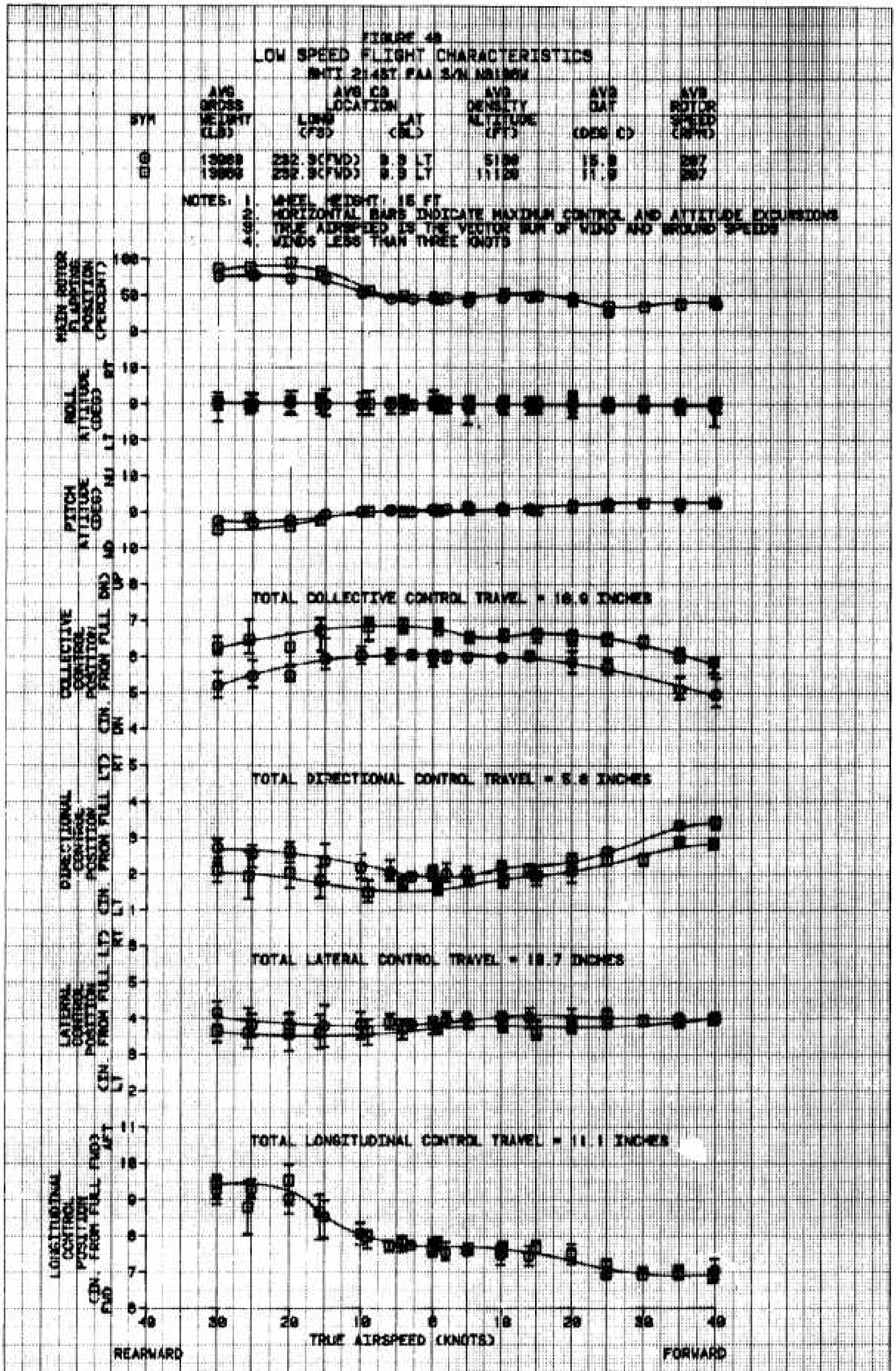


FIGURE 47
RIGHT DIRECTIONAL STEP
BHTI 214ST FAA S/N N3180W




LOW SPEED FLIGHT CHARACTERISTICS
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (FS)
LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
SYM
NOTES:
1. WHEEL HEIGHT: 15 FT
2. HORIZONTAL BARS INDICATE MAXIMUM CONTROL AND ATTITUDE EXCURSIONS
3. TRUE AIRSPEED IS THE VECTOR SUM OF WIND AND GROUND SPEEDS
4. WINDS LESS THAN THREE KNOTS
MAIN ROTOR FLAPPING POSITION (PERCENT)
ROLL ATTITUDE (DEG)
PITCH ATTITUDE (DEG)
TOTAL COLLECTIVE CONTROL TRAVEL
COLLECTIVE CONTROL POSITION (IN. FROM FULL DN)
TOTAL DIRECTIONAL CONTROL TRAVEL
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL LATERAL CONTROL TRAVEL
LATERAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL LONGITUDINAL CONTROL TRAVEL = 11.1 INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
TRUE AIRSPEED (KNOTS)
REARWARD
FORWARD

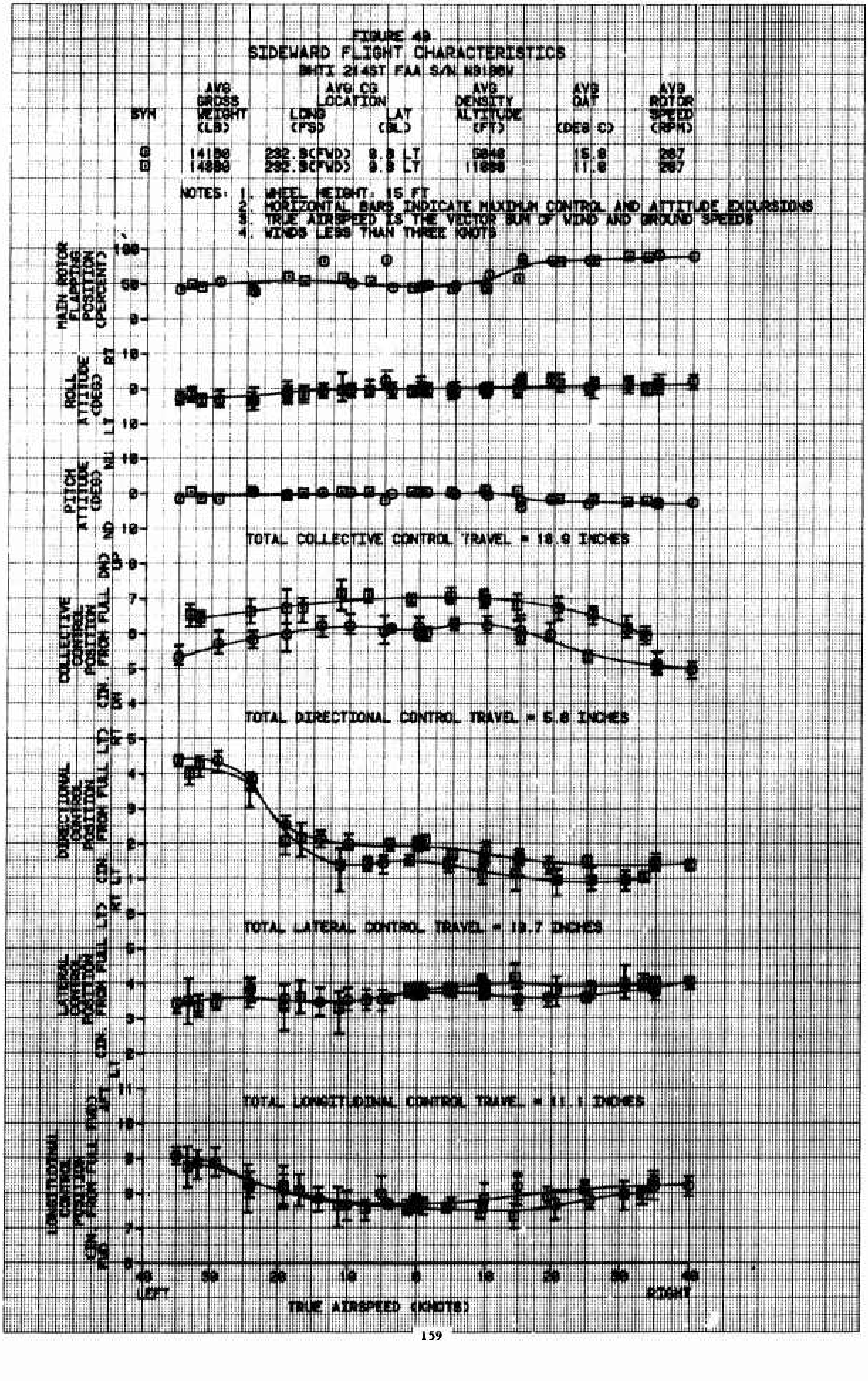


FIGURE 49
SIDEWARD FLIGHT CHARACTERISTICS
BHTI 214ST FAA S/N N918BW
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION LONG (FS) LAT (BL)
AVG DENSITY ALTITUDE (FT)
AVG OAT (DEG C)
AVG ROTOR SPEED (RPM)
NOTES: 1. WHEEL HEIGHT: 15 FT
2. HORIZONTAL BARS INDICATE MAXIMUM CONTROL AND ATTITUDE EXCURSIONS
3. TRUE AIRSPEED IS THE VECTOR SUM OF WIND AND GROUND SPEEDS
4. WINDS LESS THAN THREE KNOTS
MAIN ROTOR FLAPPING POSITION (PERCENT)
ROLL ATTITUDE (DEG)
PITCH ATTITUDE (DEG)
TOTAL COLLECTIVE CONTROL TRAVEL = 10.9 INCHES
COLLECTIVE CONTROL POSITION (IN. FROM FULL DN)
TOTAL DIRECTIONAL CONTROL TRAVEL = 5.8 INCHES
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL LATERAL CONTROL TRAVEL = 10.7 INCHES
LATERAL CONTROL POSITION (IN. FROM FULL LT)
TOTAL LONGITUDINAL CONTROL TRAVEL = 11.1 INCHES
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
LEFT
RIGHT
TRUE AIRSPEED (KNOTS)

FIGURE 50
LOW SPEED FLIGHT CHARACTERISTICS
BHTI 214ST FAA S/N N3180W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG TRUE AIRSPEED (KTS)
14100	232.3(FWD)	0.3 LT	5100	18.0	287	15

NOTE: AIRCRAFT RELATIVE WIND AZIMUTH = 270 DEG

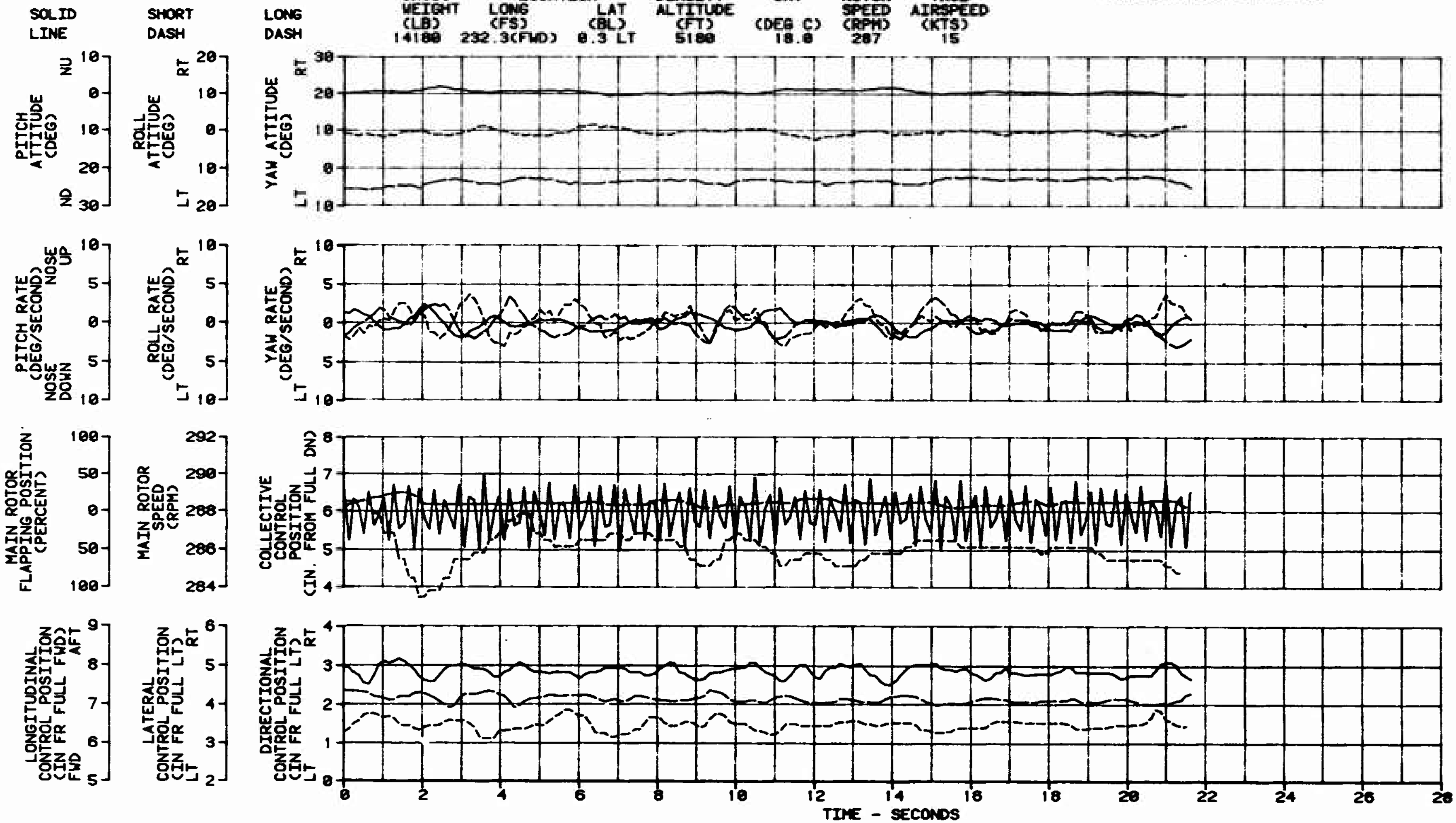


FIGURE 51
LOW SPEED FLIGHT CHARACTERISTICS
BHTI 214ST FAA S/N N3186W

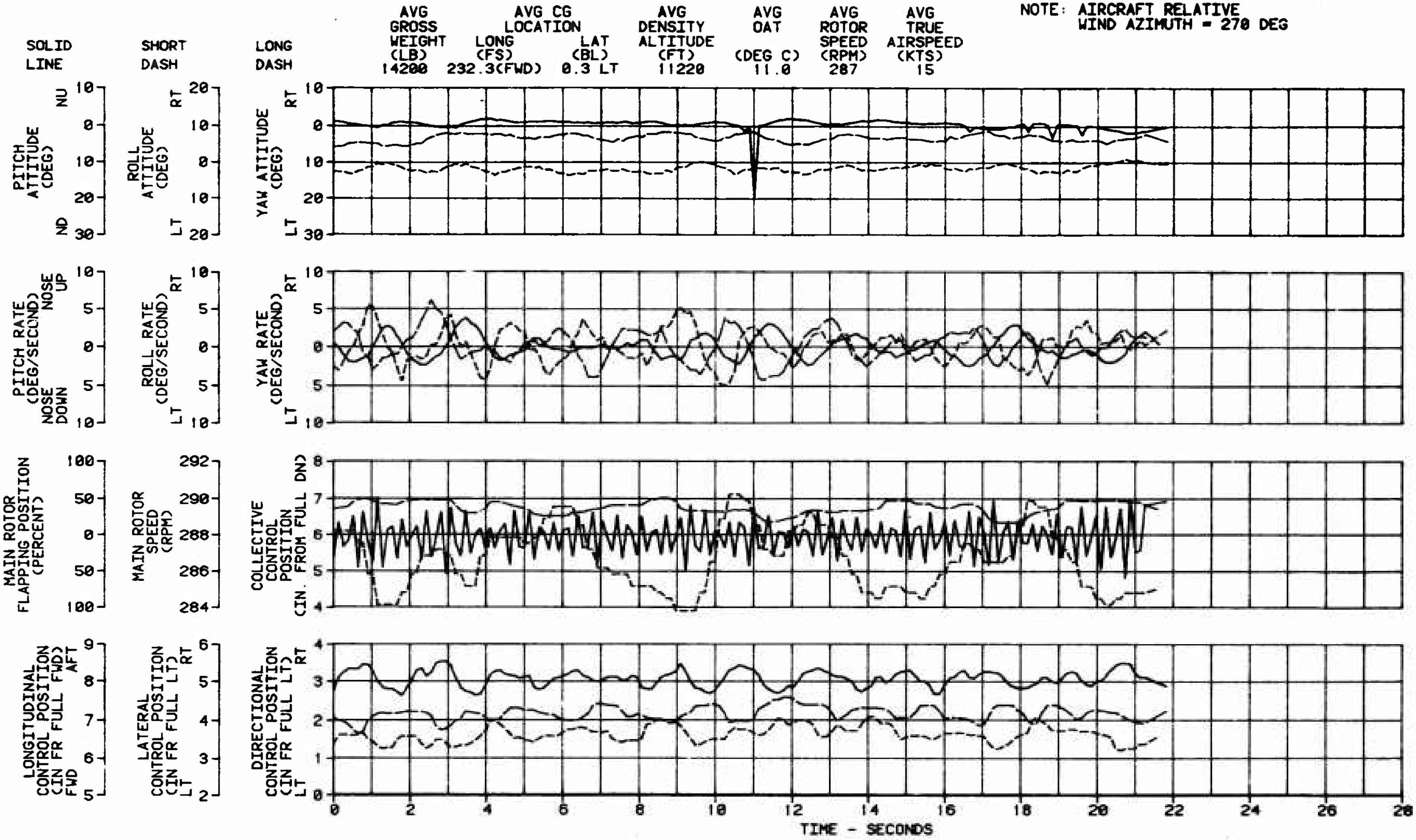


FIGURE 52
LOW SPEED FLIGHT CHARACTERISTICS AT VARIOUS RELATIVE WIND AZIMUTHS
BHTI 214ST FAA S/N N3186W
TRUE AIRSPEED: 25 KNOTS

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)
⊙	13580	232.3(FWD)	0.3 LT	5040	15.0	287
⊡	13600	232.3(FWD)	0.3 LT	11100	11.0	287

NOTES:
1. WHEEL HEIGHT: 15 FT
2. TRUE AIRSPEED IS THE VECTOR SUM OF WIND AND GROUND SPEEDS
3. WINDS LESS THAN THREE KNOTS
4. RELATIVE WIND AZIMUTH MEASURED COUNTERCLOCKWISE FROM AIRCRAFT NOSE

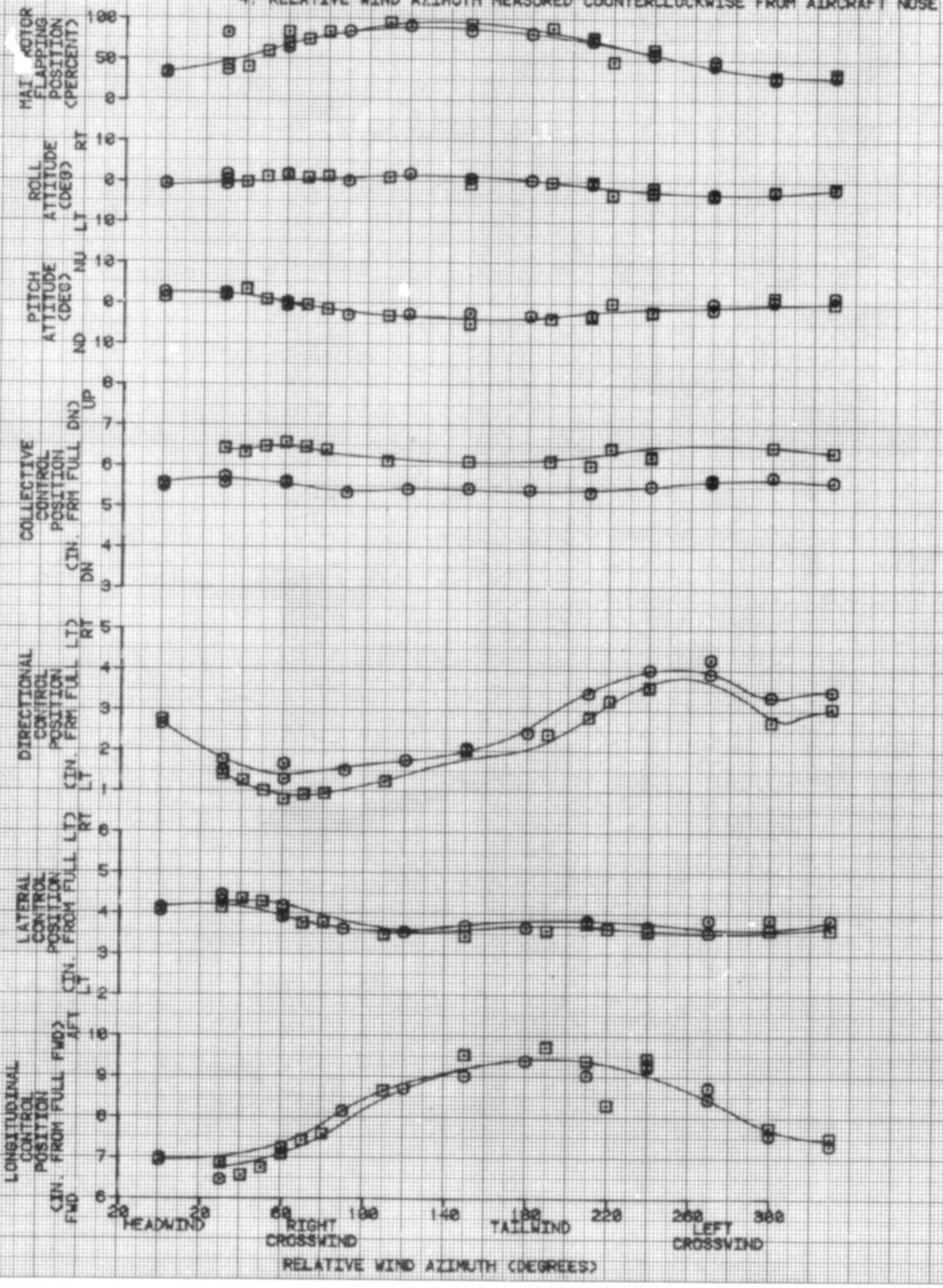


FIGURE 53
LOW SPEED FLIGHT CHARACTERISTICS
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG TRUE AIRSPEED (KTS)
13500	232.3(FWD)	0.3 LT	5220	18.0	287	25

NOTE: AIRCRAFT RELATIVE WIND AZIMUTH = 60 DEG

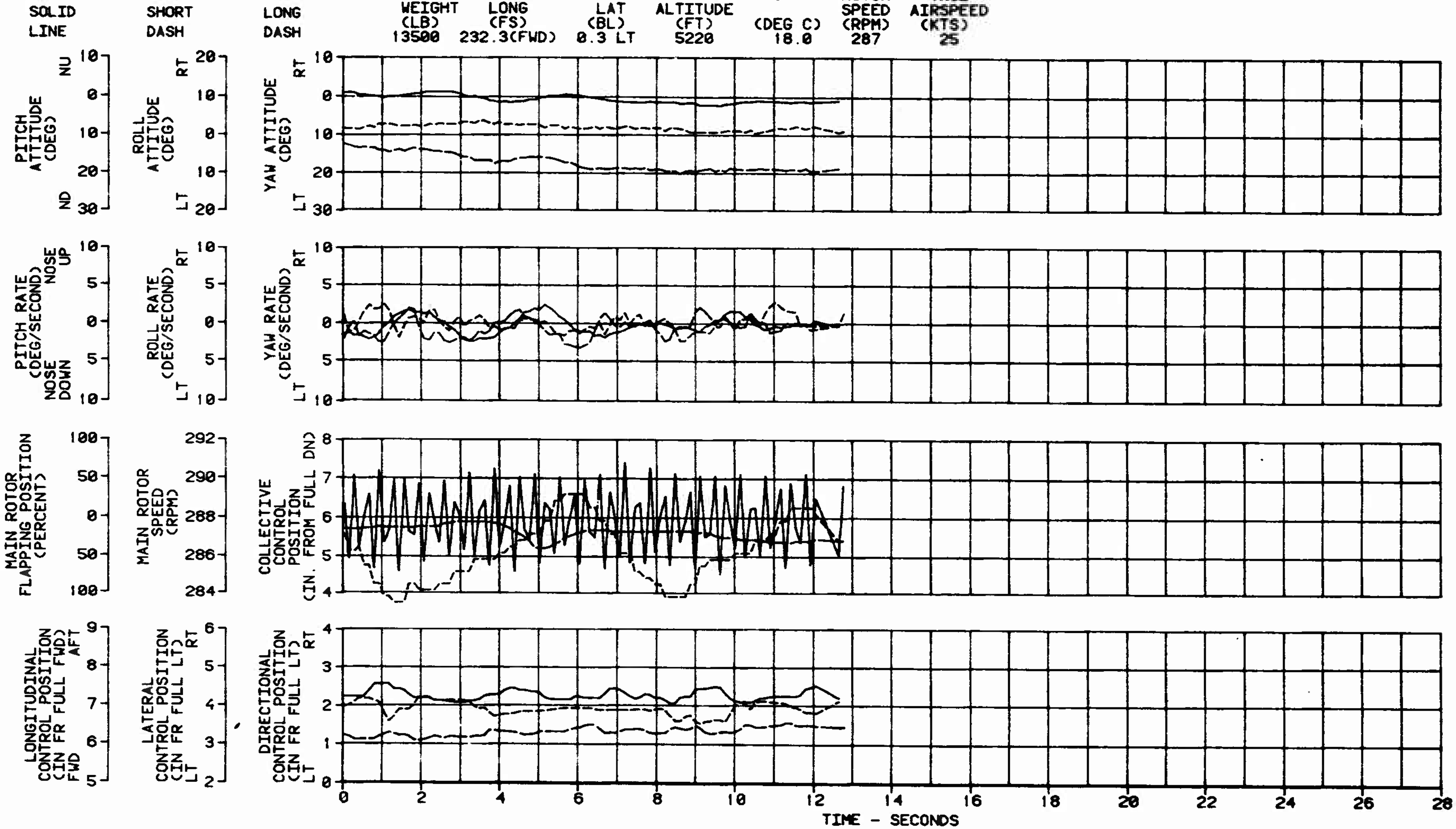


FIGURE 54
LOW SPEED FLIGHT CHARACTERISTICS
BHTI 214ST FAA S/N N3180W

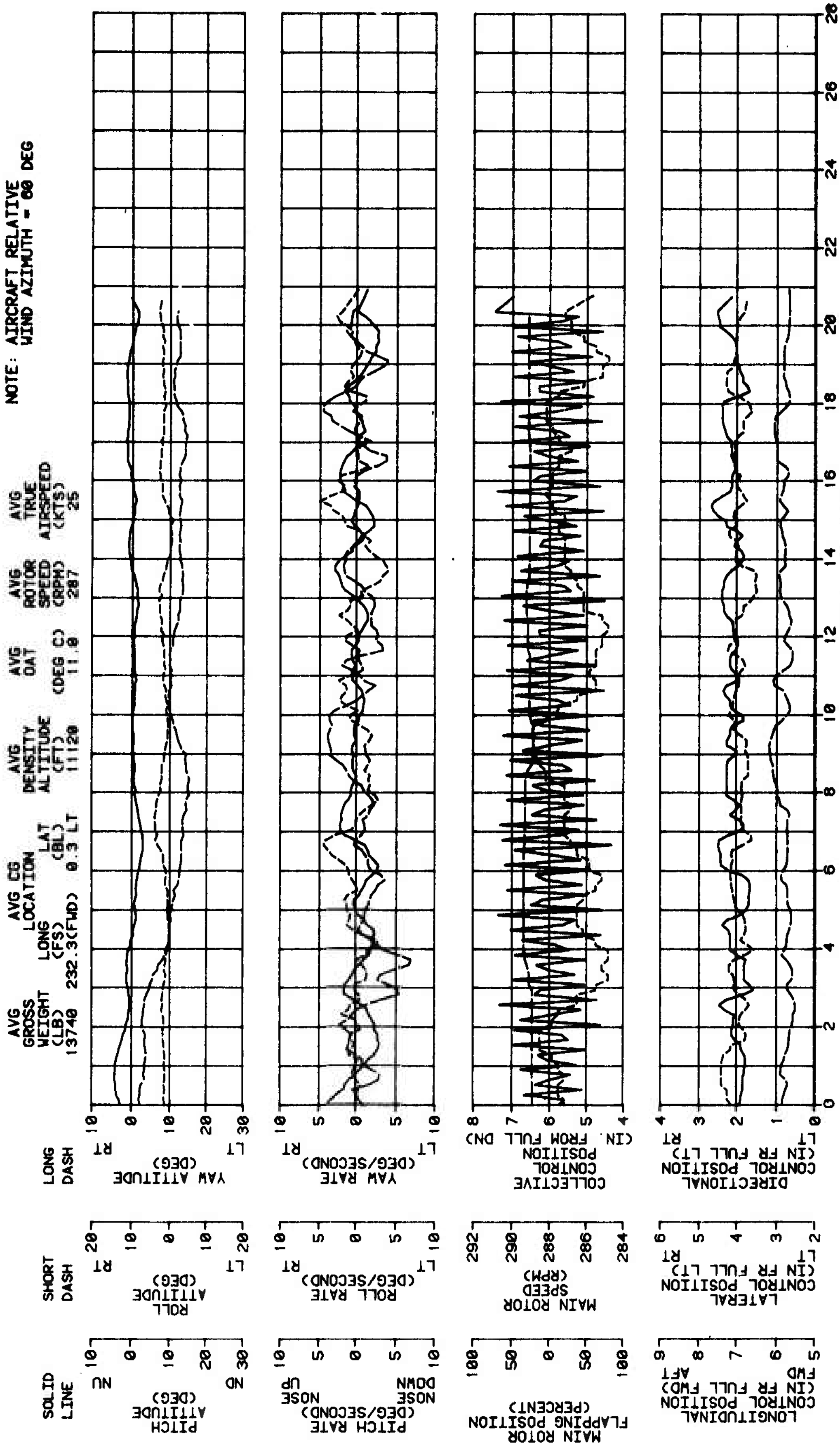


FIGURE 55
ROTOR SPEED DROOP CHARACTERISTICS
BHTI 214ST FAA S/N N3180W

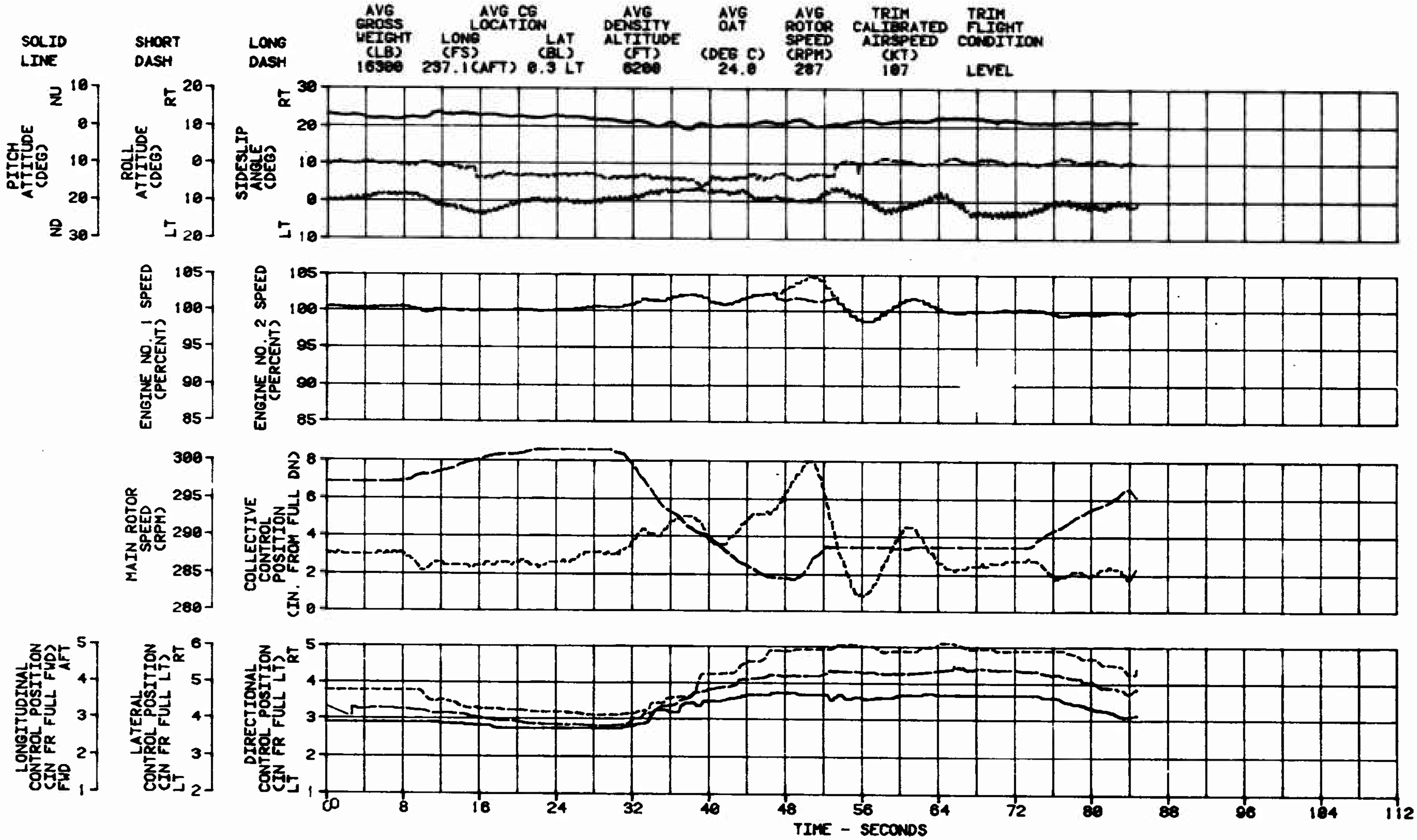


FIGURE 58
SIMULATED ENGINE FAILURE
BHTI 214ST FAA S/N N3186W

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KNOTS)	TRIM FLIGHT CONDITION
15560	232.3(FWD)	0.3 LT	6320	18.0	287	111	LEVEL

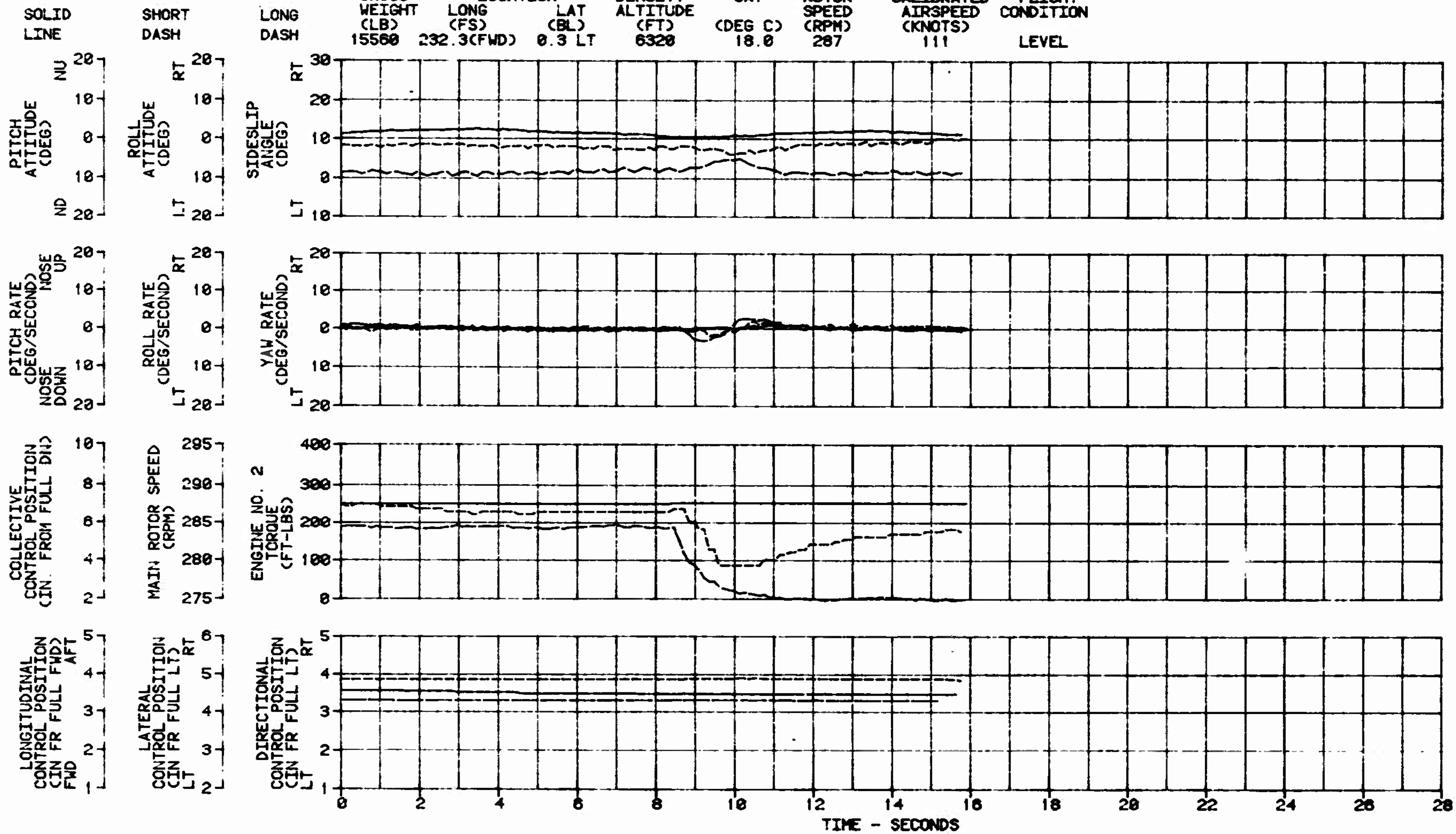


FIGURE 57
SIMULATED ROLL SCAS FAILURE
BHTI 214ST FAA S/N N3186W

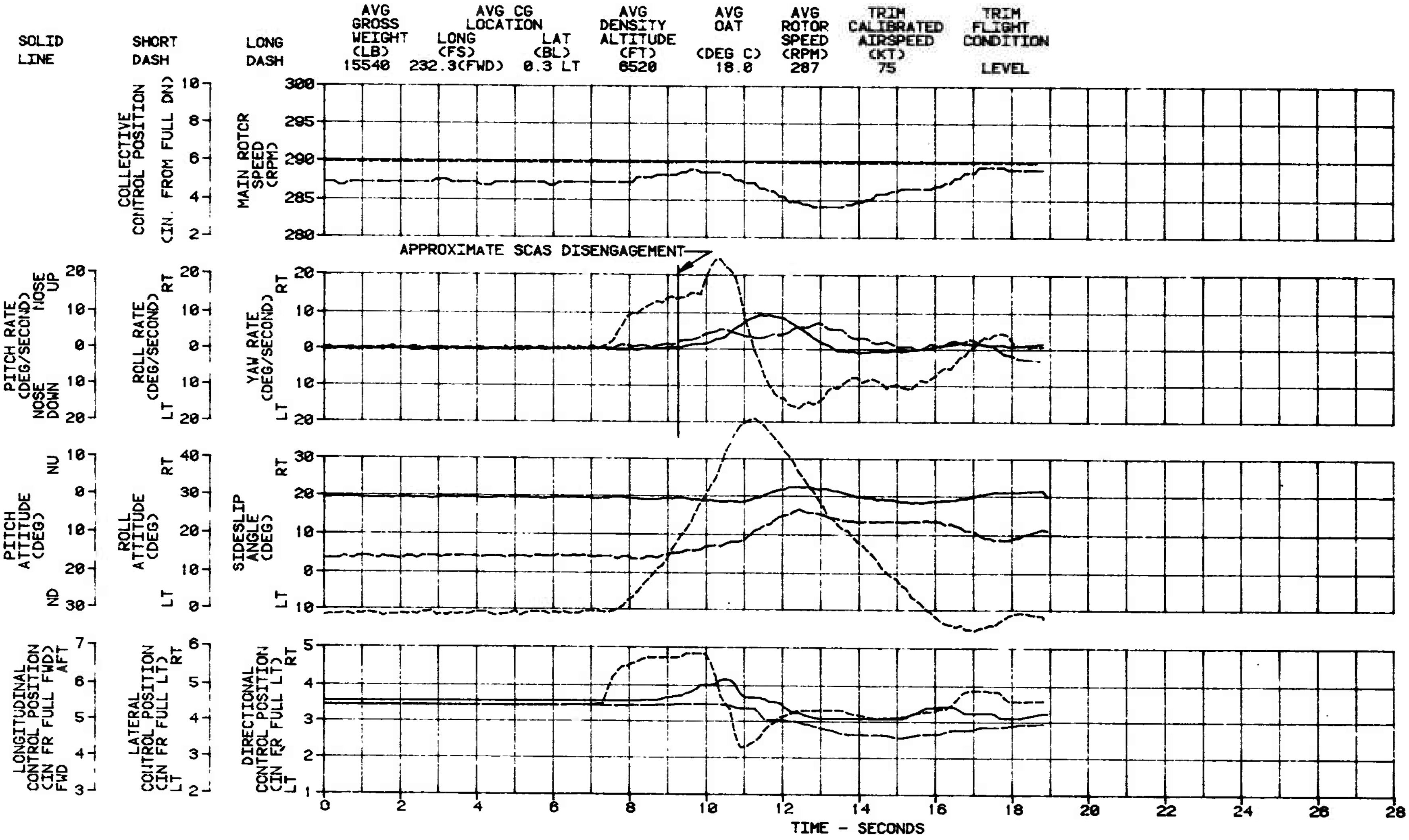


FIGURE 58
SHIPS SYSTEM (COPILOT) AIRSPEED CALIBRATION IN LEVEL FLIGHT
BHTI 214ST FAA S/N N3188W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	CALIBRATION METHOD
⊙	14100	243.5 AFT	0.3 LT	4340	25.5	287	PACE
⊡	13930	238.9 MID	0.3 LT	7500	20.0	287	TRAILING BOMB
△	17110	236.4 AFT	0.3 LT	7120	21.5	287	TRAILING BOMB

NOTES: 1. BALL-CENTERED FLIGHT
2. FLIGHT CONDITION: LEVEL
3. DASHED LINES DENOTE AIRSPEED CALIBRATION PUBLISHED IN FAA APPROVED ROTORCRAFT FLIGHT MANUAL BHT-214ST-FM

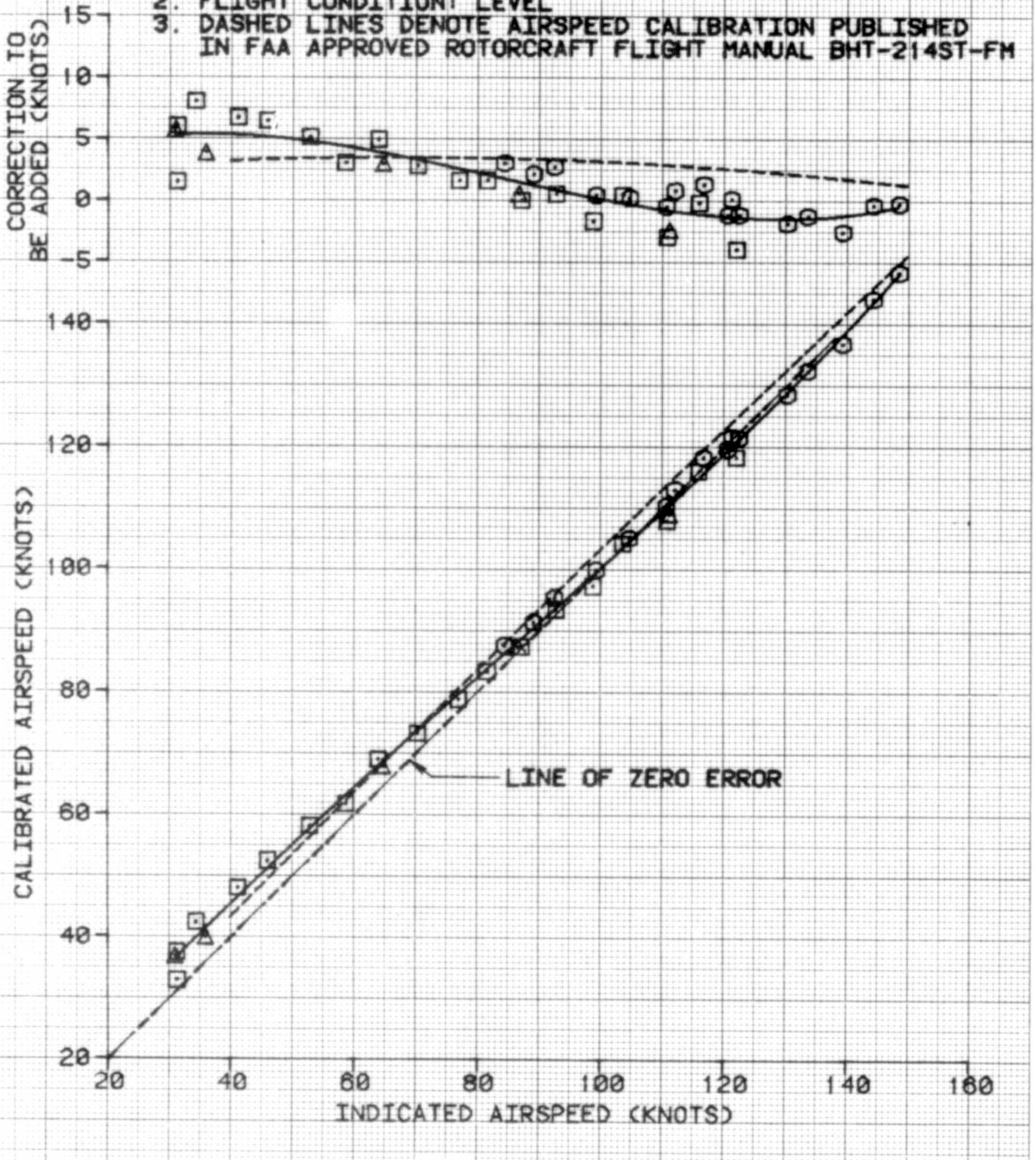


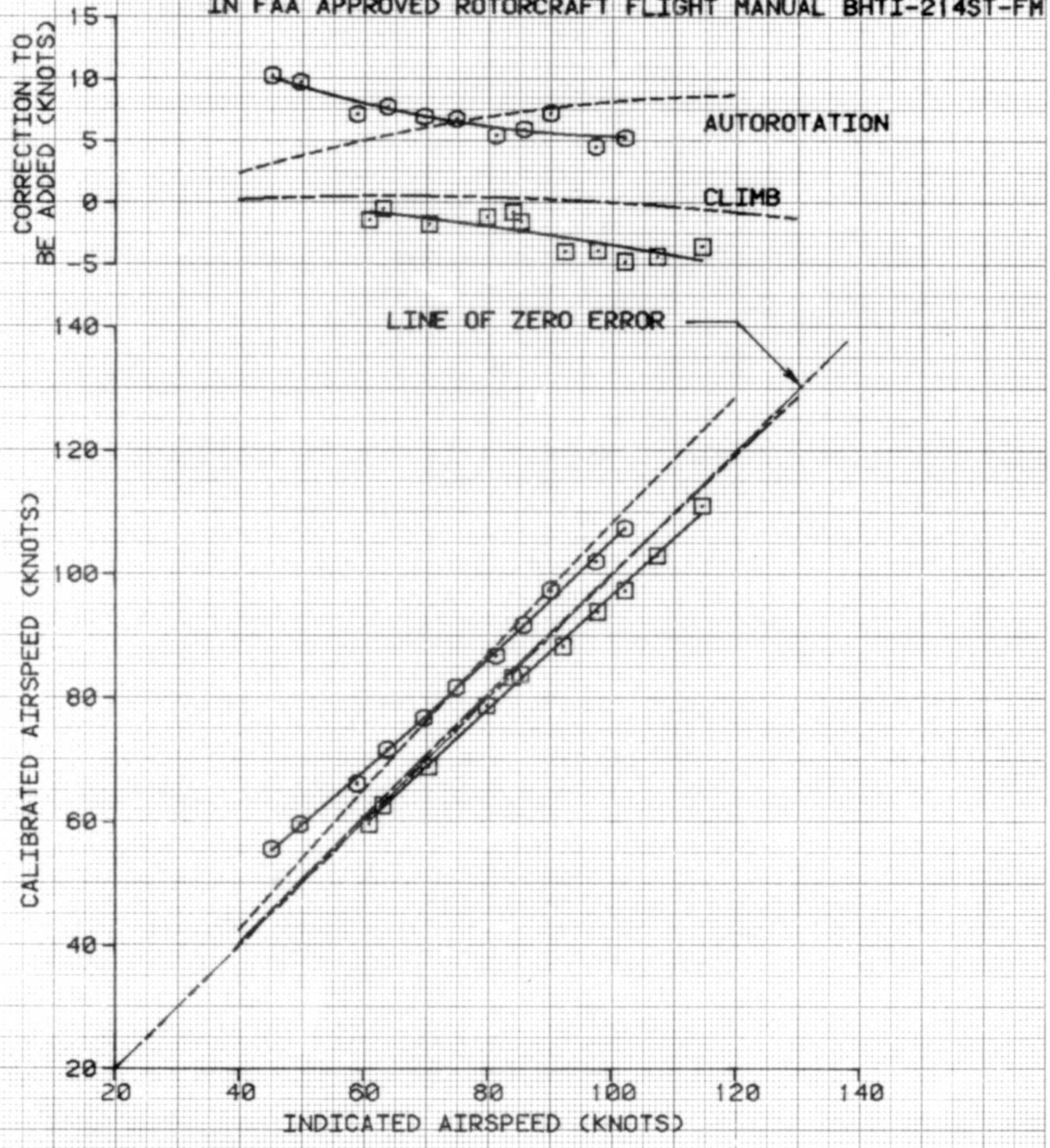
FIGURE 59

SHIPS SYSTEM (COPILOT) AIRSPEED CALIBRATION IN CLIMBS AND DESCENTS

BHTI 214ST FAA S/N N3186W

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
⊙	17380	236.7 AFT	0.3 LT	8380	20.0	287	AUTOROTATION
⊡	17300	236.5 AFT	0.3 LT	8300	20.0	287	CLIMB

NOTES: 1. TRAILING BOMB METHOD
2. BALL-CENTERED FLIGHT
3. DASHED LINES DENOTE AIRSPEED CALIBRATION PUBLISHED IN FAA APPROVED ROTORCRAFT FLIGHT MANUAL BHTI-214ST-FM



APPENDIX F. GLOSSARY

AARS	Attitude/Altitude Retension System
AC	Alternating Current
ADC	Air Data Computer
ADF	Automatic Direction Finding
ADI	Attitude Direction Indicator
AFCS	Automatic Flight Control System
app	appendix
AARS	Attitude/Altitude Retention System
AVSCOM	United States Army System Command
A&FC	Airworthiness and Flight Characteristics
cg	center of gravity
BHTI	Bell Helicopter Textron, Inc.
CMT	Control Motion Tranducer
Cp	Coefficient of Power
C_T	Coefficent of Thrust
DC	Direct Current
deg	degree or degrees
deg/sec	degrees per second
degree/sec/sec	degrees per second per second
DME	Distance Measuring Equiptment
ECU	Electrical Control Unit
FAA	Federal Aviation Administration
FBW	Fly-By-Wire
fig.	figure
fpm	feet per minute
FS	Fuselage Station (inches)
ft	foot or feet
fwd	forward
g	normal acceleration
GE	General Electric
HMU	Hydromechanical Unit
HSI	Horizontal Situation Indicator
HQRS	Handling Qualities Rating Scale
IGE	In-Ground Effect
ILS	Instrument Landing System
IMC	Instrument Meteorological Conditions
in.	inch or inches
INV	inverter
in./g	inches per normal acceleration
in./kt	inches per knot
IVSI	Instaneous Vertical Speed Indicator
KCAS	Knots Calibrated Airspeed
KIAS	Knots Indicated Airspeed
KTAS	Knots True Airspeed
lb	pound or pounds
lb/in	pounds/inch
lt	left
max	maximum

MGT	Measured Gas Temperature
min	minimum
mm	milimeter
MSL	Mean Sea Level
NAV	navigation
Ng	gas producer speed
N_p	power turbine speed
N_R	main rotor speed
OAT	Outside Air Temperature
OGE	Out-of-Ground Effect
para	paragraph
PAS	Power Available Spindle
PCM	Pulse Code Modulation
PIO	Pilot Induced Oscilation
PTA	Parallel Trim Actuator
PWR	power
RNAV	Area Navigation
ref	reference
rt	right
rpm	revolutions per minute
SCAS	Stability Control Augmentation System
shp	shaft horsepower
S/N	Serial Number
TEDN	Trailing Edge Down
USAAEFA	United States Army Aviation Engineering Flight Activity
vac	Volts Alternating Current
VMC	Visual Meteorological Conditions
$V_{min\ R/D}$	Airspeed for minimum rate of descent
V_{NE}	Never-Exceed Velocity
VOR	Very high frequency Ommni Range
VRS	Vibration Rating Scale
1/rev	once per revolution
W/σ	Weight to density ratio
%	percent
#	number

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