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GOVERNMENT COMPETITIVE TEST

UTILITY TACTICAL TRANSPORT AIRCRAFT SYSTEM (UTTAS)

SIKORSKY YUH-60A HELICOPTER

FINAL REPORT

JOHN I. NAGATA
US ARMY
SENIOR PROJECT ENGINEER

GARY L. SKINNER
US ARMY
PROJECT ENGINEER

SAMUEL KOLWEY
US NAVY
PROJECT ENGINEER

ROBERT M. BUCKANIN
US ARMY
PROJECT ENGINEER

CARL F. MITTAG
MAJ, AD
US ARMY
PROJECT OFFICER/PILOT

MICHAEL J. COUMATOS
LCDR, US NAVY
PROJECT PILOT

JOSEPH C. MIESS
CW3, AV
US ARMY
PROJECT PILOT

NOVEMBER 1976

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

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Performance	Airworthiness data

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

The United States Army Aviation Engineering Flight Activity conducted the engineering flight test portion of the Government Competitive Test of the Sikorsky Model YUH-60A utility tactical transport aircraft system (UTTAS) helicopter from 7 April through 17 September 1976. Performance, handling qualities, engine characteristics, and vibration characteristics were evaluated to provide engineering flight test data to the UTTAS Source Selection Evaluation Board for comparison

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20. Abstract

with the UTTAS Request for Proposal, to determine compliance with the applicable paragraphs of the Prime Item Development Specification (PIDS), to provide airworthiness data as a basis for modification/updating of the safety-of-flight release for other Army tests, and to detect and allow for early correction of any aircraft deficiencies or shortcomings. The YUH-60A was tested at Edwards Air Force Base (elevation 2303 feet), California, and at alternate test site elevations of 425, 4120, and 9500 feet. A total of 162 flights were conducted for a total of 156.7 flight hours. The YUH-60A failed to meet the following performance commitments of the PIDS: primary mission gross weight, hover, single-engine takeoff, vertical climb, forward flight climb service ceiling, cruise airspeeds, single-engine maximum airspeed, the alternate endurance mission, and the 3-hour endurance mission. In addition, the 3-hour endurance requirement of the Request for Proposal could not be met. Of the 12 performance commitments of the PIDS that were checked, the YUH-60A could meet only two: turning performance and the vertical and lateral displacement maneuvers. The performance characteristics of the helicopter generally fell below the established design criteria, primarily due to the excess empty weight of the prototype. The YUH-60A exhibited several features which enhance accomplishment of the UTTAS mission. The excellent rotor speed control during normal mission tasks, along with the effective torque-matching capability and automatic turbine inlet temperature limiting of the YT700-GE-700 engine greatly reduced the pilot's power management workload. There was a high degree of pilot confidence in maneuvering the helicopter to the extremes of the flight envelope at mission gross weight. The gust response was heavily damped with the automatic flight control system (AFCS) ON. Aircraft response to sudden engine failures was mild, as was entry into autorotational descent. The aircraft was capable of returning to a landing under simulated instrument meteorological conditions with a complete AFCS failure. A total of 10 deficiencies were noted. The significant deficiencies were (1) the highly undesirable and potentially unsafe characteristics of the roll trim integrator; (2) the high vibration levels in turning flight at alternate gross weight; and (3) the excessive engine/rotor speed transients following large-magnitude collective applications to the opposite extremes of the power demand schedule. A total of 34 shortcomings were noted along with 26 instances of specification noncompliance.

DEPARTMENT OF THE ARMY
HQ, US ARMY AVIATION RESEARCH AND DEVELOPMENT COMMAND
P O BOX 209, ST. LOUIS, MO 63166

DRDAV-EQ

19 AUG 1977

SUBJECT: Directorate for Development and Engineering Position on the Final Report of USAAEFA Project No. 74-06-1, Government Competitive Test Utility Tactical Transport Aircraft System (UTTAS) Sikorsky YUH-60A Helicopter, November 1976

SEE DISTRIBUTION

1. The Directorate for Development and Engineering position on USAAEFA's report is provided herein. Paragraph numbers from the subject report are provided for reference.

a. General:

(1) Paragraph 24: It was the intent of the Army for UTTAS to utilize a design gross weight which was equal to the primary mission gross weight of the aircraft. For added conservation, structural design was accomplished at a gross weight using this same mission loading but with full fuel capacity. Because of the overweight status of the prototype aircraft, this was not achieved in that the mission gross weight exceeded the design gross weight. Redesign to account for this increased empty weight was also not practical within the BED Phase. Based on AVRADCOM established ground rules for the Government Competitive Test (GCT), performance comparisons were made at an adjusted prototype mission gross weight in order to carry the penalty of the overweight condition.

(2) Paragraph 48: For the CAUTION stated, and all other recommended Operator's Manual changes, it should be noted that they apply to the "Prototype Operator's Manual" and may not apply to the production UH-60A.

(3) Paragraph 70: This paragraph justifies not identifying the flight characteristics of the aircraft with AFCS OFF as a shortcoming because it represented a degraded mode of operation. Pilots frequently deliberately disengage the TRM for purposes of enhancing precise maneuvering. For the Sikorsky UTTAS, this action degrades the basic stability characteristics, therefore, the reduced stability with the FAS/TRM OFF is considered a significant shortcoming by this Headquarters. In any event, changes to the flight control system for production eliminate this problem.

SUBJECT: Directorate for Development and Engineering Position on the Final Report of USAAEFA Project No. 74-06-1, Government Competitive Test Utility Tactical Transport Aircraft System (UTTAS) Sikorsky YUH-60A Helicopter, November 1976

(4) Paragraph 76: The precise bank angle control and overall excellent lateral control response characteristics of this aircraft as discussed in this paragraph are considered an enhancing feature of UTTAS by this Headquarters.

(5) Paragraph 104: Evaluation of the Group 8 ECU (part number 635E838G08) was not part of the UTTAS GCT. It was accomplished on the AEFA test aircraft only because the contractor did not have an active flight test ship at the time of the evaluation. Following this evaluation, the Group 8 ECU was no longer used during UTTAS GCT.

b. Deficiencies and Shortcomings:

(1) Paragraph 183a: The production UTTAS system does not incorporate a roll trim integrator, therefore, the problem is eliminated.

(2) Paragraph 183b: The production T700 engine has an improved anti-icing system over that available in YT engines utilized in the GCT. In addition to increased anti-icing capability, a significant improvement in power available with anti-icing "on" is part of the redesign. A further increase in power available is part of the Sikorsky production design. A modulating anti-icing valve for the air induction system restricts bleed air flow at higher ambient temperatures to improve the overall aircraft performance. Together these production changes will eliminate the deficiency noted during GCT.

(3) Paragraph 183c: The production UTTAS incorporates changes in the vibration detuning capability. This problem is addressed in the production changes.

(4) Paragraph 183d: This problem is associated with the Engine Control Unit (ECU) which is being corrected in production with a redesigned ECU.

(5) Paragraph 183e: Production corrections to the AFCS are being incorporated to enhance the reliability and maintainability of the system.

(6) Paragraph 183f: The fuel selector valve has been modified for the production aircraft and the check valve in the crossfeed line has been redesigned to incorporate thermal relief provisions. This redesign will eliminate further occurrences of this deficiency. The production selector valve and modified check valve similar to the production configuration were incorporated in BED aircraft after GCT and found to eliminate the deficiency.

(7) Paragraph 183g: The main transmission chip detector locator panel will be incorporated on the annunciator panel, which will eliminate the deficiency from the production aircraft.

(8) Paragraphs 183h and 183i: These two noted discrepancies require clarification due to the fact that these discrepancies are reported as engine related when they are not. The discrepancies are fuel system related and are not two separate discrepancies as noted. The no. 2 engine could not be started since fuel prime of the engine feed lines was lost. Anytime fuel prime is lost, the engine feed lines must be reprimed (i.e., after fuel system maintenance actions). Secondly, difficulty in priming the no. 2 engine feed lines was experienced. Difficulty in achieving a vapor vent is a misnomer. It is to be stressed that successful priming of the fuel lines was finally achieved by switching from the normal feed mode to the cross-feed mode. A plausible explanation of the inability to prime the no. 2 feed line is that the check valve in the line stuck in the open position allowing the fuel in the feed line to drain back into the tank and, thus, lose prime. With the valve in the open position during fuel priming, fuel would merely be pumped from the prime pump back into the fuel tank rather than filling the no. 2 engine feed line. Trouble shooting by the contractor after the incident revealed that the no. 2 feed line would not hold pressure. The check valve was replaced, since this was an explanation as to the reason for losing fuel prime in the no. 2 feed line. A start at high altitudes was not performed to verify the correction of the deficiency.

(9) Paragraph 183j: This restriction occurred with the seat in the full up and full forward position and was due to the seat cushion thickness. A thinner cushion is being used to solve the problem, however, comfort problems exist. A new cushion is being developed to eliminate the restriction for the production aircraft.

(10) Paragraph 184a thru hh: The shortcomings identified have been eliminated by changes to the production aircraft, except in those cases where the shortcoming is not consistent with UTTAS requirement documentation or the UTTAS Source Selection Evaluation Board (SSEB) accepted a deviation to such requirements.

(a) Examples where the shortcomings are not consistent with UTTAS requirements are:

1 Paragraph 184a: Engine out aural warning is not required.

2 Paragraph 184g: Two VOR/ILS navigation radios are not required.

3 Paragraph 184o: Some avionics will not have preset frequencies since this is not considered a requirement by DA.

(b) One shortcoming which will not be fully corrected but for which a deviation was granted in the production specification is:

Paragraph 184t: For the nose down slope direction only, a deviation from 12° to 9° maximum slope has been granted.

(c) Improvement of the rearward field of view from the flight crew stations is not practical and is not considered a meaningful problem since the aircraft will normally be flown with three crew members and the visibility in question is generally as good as other operational Army helicopters.

(1) Paragraph 184p: This shortcoming is not appropriate since AEFA did not have a conforming aircraft. Additional soundproofing is being added for production, but this fact is not pertinent to the GCT report.

c. Recommendations:

(1) Paragraphs 191 through 201: The term "Operator's Manual" should refer to "Prototype Operator's Manual", since many of these items do not apply to the production UH-60A.

(2) Paragraph 191: This entry is more appropriately classified as a "NOTE" rather than a "CAUTION" and will be so annotated in future revisions to the "Operator's Manual".

2. Because of the overweight status of the prototype aircraft and other simple performance improvements being incorporated into the production design, the performance levels shown in this report are not representative of the UH-60A operational capability. Contractual requirements will result in UTTAS performance being met as specified in the Materiel Need Document. The UTTAS SSEB negotiated corrections to all significant problems encountered during the GCT for the production aircraft. For a complete definition of the production UTTAS configuration and its performance capability, refer to Prime Item Development Specification (PIDS) for UH-60A

DRDAV-FQ
SUBJECT: Directorate for Development and Engineering Position on the Final Report of USAAEFA Project No. 74-06-1, Government Competitive Test Utility Tactical Transport Aircraft System (UTTAS) Sikorsky YUH-60A Helicopter, November 1976

Utility Tactical Transport Aircraft System, Specification Number AMC-CP-2222-S1000B, 1 November 1976, which is available from the UTTAS Project Manager, PO Box 209, St. Louis, Missouri 63166.

FOR THE COMMANDER:

WALTER A. RATCLIFF
Colonel, GS
Director of Development
and Engineering

PREFACE

The engineering flight test portion of the Government Competitive Test was conducted at Edwards Air Force Base, California, and at alternate test sites at Bishop, Coyote Flats, and Bakersfield, California. The test aircraft was maintained by the United States Army Aviation Engineering Flight Activity with backup support provided by the Sikorsky Aircraft Division of United Technologies Corporation. Aircraft test instrumentation was supplied, calibrated, and maintained by Sikorsky Aircraft personnel.

The description drawings and schematics presented in appendixes B, C, and D are used with permission of Sikorsky Aircraft.

DEDICATION

This report reflects the flying and engineering analysis effort conducted by the United States Army Aviation Engineering Flight Activity in participation in the UTTAS competitive fly-off. The YUH-60A test team dedicates this effort to the memory of LTC William R. Horton and CPT Michael A. Hawley, who gave their lives in pursuit of this goal.

TABLE OF CONTENTS

INTRODUCTION

BACKGROUND

1. The utility tactical transport aircraft system (UTTAS) was developed to meet a requirement for increased performance and survivability in a mid intensity combat environment. Particular emphasis has been placed on increased systems reliability and maintainability. The UTTAS, which is to replace the current Army utility helicopter fleet, has a primary mission of transportation of a variety of internal loads in day, night, and instrument meteorological conditions (IMC) with a secondary mission of transporting external loads under visual meteorological conditions (VMC). More specific mission requirements are contained in the Materiel Need document (ref 1, app A).

2. The United States Navy (USN) is interested in the UTTAS as a potential candidate for satisfying their light airborne multipurpose system (LAMPS) MK III extended-range helicopter mission. The USN desired joint participation with the Army during the Army Preliminary Evaluation (APE) and the Government Competitive Test (GCT). The joint effort was agreed upon by the UTTAS and LAMPS MK III Project Managers and formalized in a Memorandum of Understanding (ref 2, app A). Specific tests were included in the APE and GCT to permit valid assessment of the UTTAS for the LAMPS MK III mission. Additionally, UTTAS/shipboard interface testing was completed by the USN test pilots aboard the Fast Frigate USS Paul (FF-1080) in June 1976. An external noise evaluation of the UTTAS was conducted at Edwards Air Force Base by the United States Army Air Mobility Research and Development Laboratory, Ames Directorate, Moffett Field, California.

3. In August 1972, the United States Army Aviation Systems Command (AVSCOM) awarded contracts to the Sikorsky Aircraft Division of United Technologies Corporation and the Boeing-Vertol Company to each build three prototype aircraft and one ground test vehicle. Basic engineering development (BED) tests for the Sikorsky YUH-60A were conducted at Stratford, Connecticut. The United States Army Aviation Engineering Flight Activity (USAAEFA) conducted an APE of the YUH-60A from 23 October through 6 November 1975 (ref 3, app A), as a prerequisite to conducting the GCT.

TEST OBJECTIVES

4. The objectives of the YUH-60A GCT were as follows:

a. To provide engineering flight test data to the UTTAS Source Selection Evaluation Board (SSEB) for comparison with the UTTAS Request for Proposal (RFP) (ref 4, app A).

b. To determine compliance with the applicable paragraphs of the prime item development specification (PIDS) (ref 5, app A).

c. To provide airworthiness data as a basis for modification/updating of the safety-of-flight release (SOFR) (ref 6) for the United States Army Test and Evaluation Command (TECOM) and Operational Test and Evaluation Agency (OTEA) portions of the GCT.

d. To detect and allow for early correction of any aircraft deficiencies or shortcomings.

DESCRIPTION

5. The YUH-60A UTTAS (photo A) is a twin turbine-engine, single-main-rotor configured helicopter designed for transporting cargo, 11 combat troops, and weapons during VMC or IMC. Fixed conventional wheel-type landing gear are provided. The main rotor is four-bladed, with a capability of manual main rotor blade folding. The tail pylon can also be manually folded. A movable horizontal stabilator is located on the lower portion of the tail rotor pylon. The cross-beam tail rotor with composite blades is attached to the right side of the pylon. The plane of the tail rotor is canted 20 degrees upward from the vertical.

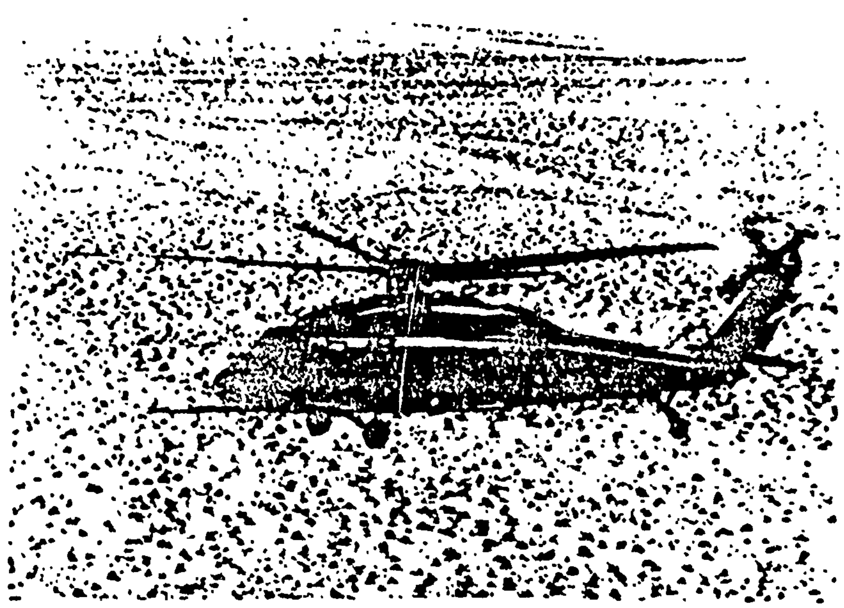

Photo A. YUH-60A Helicopter.

6. The YUH-60A is powered by two YT700-GE-700 turboshaft engines rated at 1536 shaft horsepower (shp) each at sea-level, standard-day conditions. The YT700-GE-700 engine incorporates an emergency lubrication system, a history recorder, erosion indicators, and other diagnostic systems. The YUH-60A mission gross weight for the GCT, based on the SOFR, was 16,790 pounds with an alternate gross weight of 19,930 pounds. General aircraft, flight control, and engine descriptions are contained in appendixes B, C, and D, respectively.

TEST SCOPE

7. The engineering flight test portion of the GCT was conducted at Edwards Air Force Base (2303 feet) and at alternate test sites near Bakersfield (425 feet), Bishop (4120 feet), and Coyote Flats (9500 feet), California, from 7 April through 17 September 1976. A total of 162 flights were conducted on the test aircraft (US Army serial number 73-21651) for a total of 156.7 flight hours. Productive test time was 75.2 hours. A USN test pilot and flight test engineer from the Naval Air Test Center, Patuxent River, Maryland, were assigned as integral members of the APE and GCT teams. Sikorsky Aircraft supplied, calibrated, and maintained the test instrumentation and assisted in aircraft maintenance during the test. Flight restrictions and operating limitations observed during the GCT are contained in the operator's manual (ref 7, app A) and the SOFR. Testing was conducted in accordance with the test plan (ref 8), which was based on the commitments of the PIDS and the requirements of the RFP. Test conditions for the GCT are presented in table 1 and table 2.

TEST METHODOLOGY

8. Established flight test techniques and data reduction procedures were used (refs 9 and 10, app A). The test methods are briefly described in the Results and Discussion section and appendix F of this report. A Handling Qualities Rating Scale (HQRS) (fig. 1, app F) was used to augment pilot comments relative to handling qualities. The 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 critical data parameters was used during certain tests. A detailed listing of the test instrumentation is contained in appendix E.

Table 1. Performance Test Conditions.

Type of Test	Gross Weight (lb)	Center of Gravity (in.)		Density Altitude (ft)	Trim Airspeed (kt)
		Longitudinal	Lateral		
Hover[2]	15,480 to 20,130	352.0 mid	0.4 right	2050 to 11,750	Zero
Takeoff	17,240 to 20,200	347.8 fwd	0.4 right	5180 to 11,000	11 to 62 KTAS[3]
Vertical climb	15,860 to 19,280	347.4 fwd	0.4 right	3080 to 6040	Zero
Forward flight climb	15,840 to 17,890	347.1 fwd	0.4 right	[4]400 to 10,000	71 to 74 KCAS[5]
Level flight[6]	15,940 to 19,880	347.6 fwd	0.4 right	2580 to 11,600	51 to 165 KTAS
Turning	16,790	348.8 fwd	0.4 right	7120	135 KCAS
Vertical and lateral displacement	16,660 to 17,190	346.6 fwd	0.4 right	4870 to 7030	155 KTAS
Autorotation[2]	16,900 to 19,720	347.6 fwd	0.4 right	5050 to 5200	44 to 136 KCAS
Height-velocity	14,370 to 16,950	347.8 fwd	0.4 right	4790 to 5760	Zero to 15 KCAS

[1]Tests were conducted in the normal utility configuration with all doors closed at 258 rpm unless otherwise noted.
[2]Rotor speed was 250 to 271 rpm during hover and 237 to 288 rpm during autorotational descent.
[3]KTAS: Knots true airspeed.
[4]Pressure altitude.
[5]KCAS: Knots calibrated airspeed.
[6]Configuration: Clean (normal utility) with all doors closed, except for level flight performance, which was flown in the following configuration: aft cg, aft cargo doors open, and gunner window removed with M60 guns extended.

[redacted]

Table 2. Handling Qualities Test Conditions.

Type of Test	Gross Weight (lb)	Center-of-Gravity Location (in.)		Density Altitude (ft)	Trim Calibrated Airspeeed (kt)	AFCS[1] Condition
		Longitudinal	Lateral			
Control positions in trimmed forward flight	16,840 to 20,000	346.8 fwd to 361.1 aft	0.4 right	4700 to 8040	44 to 176	AFCS ON
Static longitudinal stability	16,240 to 17,060	359.5 aft to 360.5 aft	0.4 right to 0.6 right	6120 to 7060	30 to 116	FPS, FAS/TRM, pitch bias OFF at 30, 65, and 116 KCAS
Static lateral-directional stability	16,840 to 17,220	359.4 aft	0.4 right	5620 to 6620	45 to 83	AFCS ON
Maneuvering stability	16,460 to 20,160	358.6 aft to 359.6 aft	0.4 right to 0.5 right	6320 to 7100	105 to 143	SAS, FPS, FAS/TRM, pitch bias OFF at 132 and 142 KCAS (turning) and 130 and 143 KCAS (pull-ups/pushovers)
Dynamic stability	16,430 to 16,860	359.2 aft to 359.6 aft	0.4 right	6140 to 6440	76 to 117	SAS, FPS, FAS/TRM, pitch bias OFF at 111 and 117 KCAS
Controllability	15,920 to 17,330	358.8 aft to 359.5 aft	0.4 right	6000 to 11,200	Zero to 116	SAS, FPS, FAS/TRM, pitch bias OFF at zero, 111, 114, and 116 KCAS
Low-speed flight	15,640 to 18,190	346.4 fwd to 359.0 aft	0.4 right to 0.5 right	2440 to 11,200	Zero to 52 KTAS	AFCS ON
Power management	16,660 to 17,230	351.3 mid to 360.0 aft	0.4 right to 0.5 right	4540 to 6450	80 to 145	AFCS ON
Normal mission maneuvers	16,550 to 17,220	358.8 aft to 360.1 aft	0.4 right to 0.5 right	5900 to 7120	138 and 135	AFCS ON
External load evaluation	18,800 to 17,600	359.3 aft	0.5 right	3650 to 6200	Zero to 129	SAS, FPS, FAS/TRM OFF at 65 to 105 KCAS
Instrument flight	17,460 to 16,260	359.4 aft	0.4 right	3460 to 5500	Zero to 115	SAS, FPS, FAS/TRM, pitch bias OFF from zero to 115 KCAS
Aircraft systems failures	16,300 to 17,350	358.0 aft to 358.9 aft	0.4 right to 0.5 right	2960 to 6160	Zero to 168	Pitch bias, FAS actuator, longitudinal, lateral, and directional trim hardovers
Single-engine landings	16,260 to 16,300	360.5 aft	0.5 right	5220	[2]23 to 33	AFCS ON
Vibration	16,900 to 20,000	347.7 fwd to 360.0 aft	0.4 right to 0.5 right	2440 to 7000	80 to 145	AFCS ON

[1]AFCS: Automatic flight control system.
[2]Tail wheel touchdown speed.

[redacted]

RESULTS AND DISCUSSION

GENERAL

9. The performance and handling qualities of the YUH-60A were evaluated under a variety of operating conditions at test sites from near sea level (425 feet) to 9500 feet. The YUH-60A failed to meet the following performance commitments of the PIDS: primary mission gross weight, hover, single-engine takeoff, vertical climb, forward flight climb service ceiling, cruise airspeed (V_{cruise}), single-engine maximum airspeed for level flight (V_H), the alternate endurance mission, and in addition, the 3-hour endurance requirement of the RFP could not be met. Of the 12 performance commitments of the PIDS that were checked, the YUH-60A could meet only two: turning performance and the lateral and vertical displacement maneuvers. The performance characteristics of the helicopter generally fell below the established design criteria, primarily due to the excess empty weight of the prototype. The YUH-60A exhibited several features which enhance accomplishment of the UTTAS mission. The excellent rotor speed control during normal mission tasks, along with the effective torque-matching capability and automatic turbine inlet temperature ($T_{4.5}$) limiting of the YT700-GE-700 engine, greatly reduced pilot workload in the area of power management. There was a high degree of pilot confidence in maneuvering the helicopter to the extremes of the flight envelope at mission gross weight because of low vibrations, excellent controllability, and generally excellent rotor speed control characteristics. The gust response was heavily damped in all axes with the AFCS ON. Aircraft response to sudden engine failure was mild for all flight conditions tested and entry into autorotational descent required minimal pilot effort. The aircraft was capable of returning to a landing under simulated IMC with a complete AFCS failure. A total of 10 deficiencies were noted. These deficiencies were (1) the highly undesirable and potentially unsafe characteristics of the roll trim integrator; (2) significant power losses associated with operation of the engine anti-ice and cockpit heater systems; (3) the high vibration levels in turning flight at 40 to 45 degrees of bank at alternate gross weight for airspeeds of 105 KCAS and greater; (4) excessive engine/rotor speed transients following large-magnitude collective applications to the opposite extremes of the power demand schedule; (5) the poor reliability and maintainability characteristics demonstrated by the AFCS; (6) sticking fuel selector valves; (7) location of the transmission chip detector locator panel; (8) inability to start the YT700-GE-700 engine at high altitudes without successful engine vapor venting; (9) difficulty of achieving successful engine vapor vents at these high altitudes; and (10) inability to achieve full aft longitudinal cyclic and restriction of lateral cyclic movement with the pilot or copilot seat in the full-up and full-forward position. A total of 34 shortcomings were noted along with 26 instances of specification noncompliance.

PERFORMANCE

General

10. Performance flight testing was conducted at test sites of 425, 2303, 4120, and 9500 feet. Performance evaluations included tethered hover, takeoff, vertical and forward flight climb, level flight, turning flight, vertical and lateral displacement, autorotational descent, and height-velocity (H-V) determination. Power available and fuel flow were obtained from the engine model specification corrected for inlet and exhaust losses. A summary of the aircraft performance with a comparison of the commitments of the PIDS and the requirements of the RFP is included in table 3. The YUH-60A met two of the 12 performance commitments that were checked. The performance characteristics of the helicopter generally fell below the established design criteria, primarily due to the excess empty weight of the prototype.

Hover Performance

11. Hovering tests were conducted utilizing the tethered hover method at the conditions listed in table 1. The 5-foot main wheel height in-ground-effect (IGE) and 100-foot wheel height out-of-ground-effect (OGE) tests were conducted at the 2303-foot test site, while OGE only was conducted at the 4120-foot test site and 5-foot IGE only at the 9500-foot test site. A cable tensiometer measured total thrust less gross weight. Variations in coefficient of thrust (C_T) were attained by varying rotor speed from 95 to 103 percent (250 to 271 rpm). Hover test results are presented in figures 1 through 5, appendix G.

12. The standard-day OGE hover ceiling at the primary mission gross weight of 16,853 pounds (para 24) was 8800 feet. At 4000 feet, 35°C day, the OGE hover maximum gross weight was 16,860 pounds at intermediate rated power (IRP) and was 16,193 pounds at 95 percent IRP. The hover performance fails to meet the commitments of paragraph 3.2.1.1.1.1a of the PIDS and the requirements of the RFP, in that the aircraft will not hover OGE at the primary mission gross weight, 4000 feet, 35°C, with 95 percent IRP. The magnitude by which the hover performance failed to meet the commitments of the PIDS is unsatisfactory.

Takeoff Performance

13. Takeoff performance tests were conducted at the conditions listed in table 1 to determine the horizontal distance required to clear a 50-foot obstacle. The technique of level acceleration from a 5-foot hover (left main wheel height) to a constant climb-out airspeed was used. Test results are presented in figures 6 through 11 of appendix G.

14. Accelerations from a hover were initiated by simultaneous application of forward cyclic and increasing collective to obtain IRP. For the conditions where the aircraft could just hover at 5 feet with IRP, acceleration was initiated by slight

Table 3. Performance Summary.

Item	PIDS	RFP	Test Results
Primary mission gross weight	15,850 lb	---	16,853 lb
Hover	16,853 lb	16,853 lb	16,193 lb
Takeoff	14,213 lb	14,213 lb	12,183 lb
Vertical climb	4000 ft	---	2160 ft
	---	4000 ft	2470 ft
Service ceiling	7100 ft	5000 ft	2877 ft
Cruise airspeed	160 KTAS	145 KTAS	138 KTAS
Single-engine level flight airspeed	126 KTAS	100 KTAS	94.5 KTAS
Endurance cruise airspeed	145 KTAS	---	138 KTAS
Endurance	2.3 hr	2.3 hr	2.12 hr
Endurance	3 hr	---	2.95 hr
Endurance	---	3 hr	2.68 hr
Turning airspeed loss	16 KTAS	16 KTAS	5 KTAS
Vertical and lateral displacement	1100 ft	1100 ft	1040 ft vertical
			1100 ft lateral

forward cyclic application only and the aircraft descended slightly during the initial phase of the acceleration. After initial control application was made by the pilot to start the acceleration, the desired power setting was maintained by the copilot. This allowed the pilot to concentrate on controlling aircraft attitude and flight path and resulted in reduced pilot workload during maximum performance takeoffs.

15. A nose-low pitch attitude was required during the initial part of the acceleration. As the aircraft accelerated through translational lift, aft cyclic control and an approximate level attitude were required to prevent the aircraft from descending below the 5-foot wheel height. As the aircraft accelerated further, forward cyclic and a nose-low attitude were again required to maintain wheel height. During accelerations where large excess power was available, the nose-low attitudes (up to 20 degrees nose-down) required to maintain a 5-foot wheel height during the start of the acceleration were uncomfortable initially, but the pilot quickly became used to them. The greater the rate of power application, the more critical was pitch attitude control, particularly for large values of excess power available. As excess power available decreased, pitch attitude control was more easily accomplished.

16. Rotation to the climb-out attitude was initiated approximately 5 knots below the climb-out airspeed. The climb-out airspeed was maintained until 100 feet of altitude was reached. A calibrated pace ground vehicle was used to establish the desired climb-out airspeed and give the rotation command. After several practice takeoffs, the pilot was able to achieve and maintain a desired climb-out airspeed using pitch attitude cues. For airspeeds above 35 knots indicated airspeed (KIAS) the airspeed indicator also provided a useful cue. Below 35 KIAS the airspeed indicator was erratic and unusable.

17. Each acceleration resulted in rotor droop (approximately 3 percent maximum) because of the rapid pull to topping power during rotation to forward flight, with the exception of the condition where IRP was required to hover IGE at 5 feet. For this condition, collective was held fixed during the takeoff. This resulted in a constant rotor speed with a power-required decrease of approximately 6 percent as the aircraft accelerated through translational lift. Rotor droop occurred when IRP was demanded during the collective pulls because of the $T_{4.5}$ limiting feature of the electrical control units (ECU's). As the $T_{4.5}$ limiting temperature (845°C) was approached, a rotor droop was incurred, the magnitude of which depended upon the rate of pull of collective and the final collective position. At $T_{4.5}$ limit temperature, the operating rotor speed (258 rpm) was regained by adjusting the collective position.

18. The PIDS commitment specifies a takeoff capability for the primary mission gross weight less payload (para 3.2.1.1.1.3) using single-engine IRP at 4000 feet pressure altitude and 35°C. The primary mission gross weight less payload was calculated to be 14,213 pounds (para 24). At this gross weight, the aircraft could not hover at 5 feet for a sea-level, 35°C condition, let alone at a 4000 feet, 35°C condition. The calculated gross weights for a 5-foot hover height at 4000 feet, 35°C and sea-level, 35°C conditions were 11,141 and 12,872 pounds, respectively.

Assuming an increased hover capability of 600 to 1000 pounds for a 2-foot hover height, the aircraft still would not be able to hover at the 14,213-pound condition. If the aircraft could hover at 2 feet at the required gross weight, the height loss during translation from hover to forward flight would be approximately 2 to 3 feet and ground contact would occur. The magnitude by which the single-engine takeoff performance failed to meet the PIDS commitment is unsatisfactory.

<u>Vertical Climb Performance</u>

19. Vertical climbs (zero horizontal airspeed) were conducted at the conditions shown in table 1. Three constant C_T vertical climb series were accomplished at the Edwards test site, with the mid C_T repeated at the Bishop test site. The vertical climbs were initiated from an OGE hover using various constant collective control settings throughout the available dual-engine power range. The vertical rate of climb for a given power increment was defined as that portion of the climb after the aircraft had achieved a steady unaccelerated rate-of-climb condition. The nondimensionalized test results are presented in figure 12, appendix G.

20. The calculated hot day vertical climb performance at 95 percent IRP is shown in figure 13, appendix G. This figure shows that at sea-level, 35°C conditions and primary mission gross weight, the maximum vertical rate of climb was 1193 feet per minute (ft/min). At a pressure altitude of 4000 feet, 35°C, 95 percent IRP, and primary mission gross weight, the aircraft would not climb vertically. This performance fails to meet paragraph 3.2 1.1.1.1a of the PIDS and RFP, which specify vertical rates of climb of 550 and 450 ft/min, respectively. The maximum altitudes at which the aircraft will achieve vertical rates of climb of the commitments of the PIDS and the requirements of the RFP are 2160 and 2470 feet pressure altitude, respectively. The magnitude by which the vertical climb performance failed to meet the commitments of the PIDS is unsatisfactory.

<u>Forward Flight Climb Performance</u>

21. Continuous climbs were conducted from near sea level to single-engine service ceiling (altitude at which a 100-ft/min rate of climb is the maximum achievable) to determine the YUH-60A climb capability and associated RFP and PIDS specification compliance. Two continuous single-engine climbs to service ceiling were accomplished at the conditions listed in table 1. Climbs were repeated at reciprocal headings to average the effect of possible wind shear on performance. The climbs were flown at the airspeed schedule for best rate of climb (V_{max} R/C) as determined from the level flight performance. The power schedule used corresponded to IRP at 35°C, based on the engine model specification. Test results are presented in figure 14, appendix G, and were corrected for the increase in equivalent flat plate area (Δf_e) caused by the test instrumentation mounted on the aircraft. Additional climbs were flown to determine power and weight correction factors and are presented in figures 15 and 16.

22. At the primary mission gross weight of 16,855 pounds, the single-engine maximum rate of climb is 315 ft/min at sea level, 35°C. The single-engine service ceiling was determined to be 2877 feet pressure altitude and failed to meet the RFP requirement of 5000 feet by 2123 feet, and the PIDS commitment of 7100 feet (para 3.2.1.1.1.3b) by 4223 feet. The best rate-of-climb airspeed could be satisfactorily maintained (variations of ±2 knots), with minimal pilot compensation during forward flight climbs (HQRS 3). Field of view during these climbs was considered adequate. The magnitude by which the single-engine service ceiling failed to meet the commitments of the PIDS is unsatisfactory.

Level Flight Performance

23. Level flight performance tests were conducted at the conditions listed in table 1 to determine power required and fuel flow for varying airspeeds, density altitudes, and gross weights at the most adverse cg to evaluate the performance of the aircraft and provide the basis for determining the primary mission gross weight. Data were obtained in stabilized level flight at incremental airspeeds from approximately 50 KTAS to V_H. A constant ratio of gross weight to density (W/σ) was maintained by increasing altitude as fuel was consumed. The nondimensional results of these tests are presented in figures 17 and 18, appendix G, and dimensionally in figures 19 through 30. The calculated performance data presented in this section and in figures 31 through 33 were corrected for the increase in Δf_e caused by the test instrumentation mounted externally on the aircraft (ref 11, app A). Level flight performance results are summarized in table 4 and compared with the commitments of the PIDS and the requirements of the RFP.

24. The primary mission gross weight, as used throughout this report, is defined in paragraph 3.2.2.1.5 of the PIDS and RFP as the sum of the payload (11 combat-equipped troops), the mission fuel load for the troop assault primary mission, and the operating empty weight. The primary mission requires an endurance of 2.3 hours at 4000 feet, 35°C, with V_{cruise} of 145 KTAS At maximum continuous power (MCP) the YUH-60A does not meet the 145 KTAS criteria. Therefore, the combat troop-assault mission profile airspeeds and resultant fuel specifics were determined at the airspeed achieved at MCP (138 KTAS). The operating empty weight of 12,113 pounds (ref 12, app A) exceeds the PIDS committed weight of 11,381 pounds by 732 pounds. This was the major contributing factor which resulted in the primary mission gross weight of 16,853 pounds exceeding the PIDS commitment by 1003 pounds. The primary mission gross weight was determined as follows:

Table 4. Level Flight Performance Summary.

Item	PIDS		RFP		GCT Results
	Paragraph	Commitment	Paragraph	Requirement	
Cruise airspeed[1] (KTAS)	3.2.1.1.1.1b	160	3.2.1.1.1.1b	145 to 175	138
Combat troop-assault mission cruise airspeed[1] (KTAS)	3.2.1.1.1.1c	145	3.2.1.1.1.1c	---	138
Maximum endurance[1] (hr)	3.2.1.1.1.1	3	---	---	2.95
Maximum endurance[2] (hr)	---	---	3.2.1.1.1.1	3	2.68
Alternate endurance[2] (hr)	3.2.1.1.1.2	2.3	3.2.1.1.1.2	2.3	2.12
Maximum single-engine level flight airspeed at IRP[1] (KTAS)	3.2.1.1.1.3a	126	3.2.1.1.1.3a	100	95
Primary mission gross weight[1]	3.2.2.1.5	15,850	---	---	16,853

[1] 4000 feet, 35°C.
[2] Sea level, standard day.

Operating empty weight		12,113 lb
Payload (11 combat-equipped troops at 240 lb each)		2,640 lb
Mission fuel:[1]		
20 minutes at MCP	316 lb	
80 minutes at V_{cruise} (MCP)	1,263 lb	
8 minutes at idle power	48 lb	
30 minutes reserve.	473 lb	
		2,100 lb
Primary mission gross weight		16,853 lb

At the primary mission conditions the PIDS specifies a single-engine V_H of 126 KTAS. Under these conditions the single-engine V_H was 95 KTAS and failed to meet the commitments of paragraph 3.2.1.1.1.3a of the PIDS.

25. An alternate endurance at sea-level, standard-day conditions of 2.3 hours for the same mission profile as defined in paragraph 24 at a gross weight equal to the operating empty weight plus payload plus full fuel (17,028 pounds) is specified in paragraph 3.2.1.1.1.2 of the PIDS and RFP. This commitment was not met. For these conditions using full fuel (2275 pounds), the endurance was 2.12 hours. For the aircraft to meet the commitment of 2.3 hours, a fuel load of 2477 pounds would be required.

26. The PIDS also specifies an additional endurance mission of 3 hours at 4000 feet pressure altitude, 35°C, primary mission gross weight, and full fuel (payload off-loaded to account for the increased fuel load). At the fuel weight corresponding to a fuel temperature of 35°C, the reduction in payload would be 140 pounds. The PIDS specifies for this endurance mission that the performance of the aircraft could be reduced, which was interpreted that the V_{cruise} of the cruise segment of the mission profile could be reduced. On this basis, the aircraft could not meet this commitment for any airspeed within the flight envelope of the helicopter. The maximum endurance at these conditions using a cruise airspeed for minimum power required ($V_{min\ pwr}$) in level flight with maximum available fuel was 2.95 hours.

27. The RFP specifies an additional desired 3-hour endurance at sea-level, standard-day conditions and primary mission gross weight on the same basis as was used in paragraph 26. At a fuel temperature of 15°C, the reduction of payload would be 175 pounds. The aircraft does not have sufficient fuel to meet this commitment. The maximum endurance at these conditions using a cruise airspeed for $V_{min\ pwr}$ in level flight with maximum available fuel was 2.68 hours. The magnitude by which the endurance failed to meet the commitments of the PIDS is unsatisfactory.

[1]Calculated from fuel flow based on 5 percent conservatism.

28. Testing with the guns extended and doors and windows open indicated that the equivalent flat plate area of the aircraft would be increased by 9.4 ft^2. This increase in flat plate area will further decrease the calculated endurance (paras 24 through 27).

Turning Performance

29. The turning performance of the aircraft was evaluated to determine the stabilized airspeed loss obtained from the level flight unaccelerated airspeed during a 4-degree-per-second (deg/sec) turn (para 3.2.1.1.2a of the PIDS) at constant altitude and IRP. The test was conducted at mission gross weight by stabilizing in level flight with engine torque settings which correspond to the IRP that a model specification engine installed in the YUH-60A would develop on a 4000 foot, 35°C day. Then, holding altitude and power constant, constant bank angle turns were accomplished. For each subsequent turn the bank angle was increased incrementally to beyond that required for a 4-deg/sec turn rate. The airspeed loss was evaluated for each stabilized turn both to the left and right. Test results are presented in figure 34, appendix G. The stabilized level flight entry airspeed was 150 KTAS for the above-stated conditions. Airspeed loss was 5 knots for turn rates of 4 deg/sec to the right and left and met the commitments of the PIDS, which permits an airspeed loss to a value no less than the airspeed for MCP (138 KTAS). The turning performance characteristics of the YUH-60A helicopter are satisfactory.

Vertical Displacement

30. Vertical displacement maneuvers were conducted at a mid-altitude test site at the conditions listed in table 1. The vertical displacement maneuvers were conducted to accomplish the following: (1) a constant-heading 200-foot vertical displacement within 1100 feet horizontal distance without regard to exit flight path (pull-up); and (2) a constant-heading pull-up and pushover to a flight path paralleling the original flight path, such that the total height change was 200 feet within 1100 to 1300 feet (pull-up with parallel exit). The target entry airspeed for both maneuvers was 150 KTAS. Representative time histories of these maneuvers are presented in figures 35 and 36, appendix G.

31. The constant-heading vertical displacement pull-up was accomplished by rapidly establishing a load factor of 1.5 to 1.6 with aft cyclic control application. The establishment of load factor was easily accomplished by an aft cyclic control movement (1.4 inches) and the use of the cockpit g meter (para 160) (HQRS 3). Maximum pitch attitude during the maneuver was 14 degrees nose-up. Minor left lateral control application was required to maintain near-constant roll attitude, while little directional control movement was required to maintain heading (HQRS 3). Rotor speed control was excellent, decreasing 2 percent (5 rpm), and required no pilot compensation to control during the maneuver (HQRS 2). Airspeed loss was 24 KTAS at the point where the 200-foot vertical displacement was achieved. The forward field of view was restricted by the instrument glare shield as the maximum pitch attitude was reached, but flight path could be maintained by use

of lateral external cues. The aircraft achieved the 200-foot vertical displacement within 1040 feet horizontal distance. The commitment of the PIDS to accomplish this maneuver within 1100 feet was met.

32. The pull-up with parallel exit vertical displacement maneuver required more pilot effort than the pure pull-up maneuver. During the pure pull-up maneuver, a target load factor was used as the aim condition, whereas altitude displacement was the aim condition for the pull-up with parallel exit maneuver. Positive and aggressive control applications by the pilot were required to effect the necessary aircraft flight path changes to attain the maneuver but the confidence in aircraft maneuvering capabilities enabled completion of the maneuver with minimal pilot compensation (HQRS 3). Initiation of the maneuver required rapid aft cyclic control movement (3.4 inches) to achieve rapid flight path displacement, followed by forward cyclic control movement (5.4 inches) to terminate the maneuver in a 200-foot parallel exit. The required control movements resulted in a maximum load factor of 2.25 and a minimum load factor of 0.2. The maximum nose-up pitch attitude reached was 24 degrees and maximum nose-down pitch attitude was 12 degrees. Forward field of view was restricted for the pull-up maneuver because of the high nose-up attitude (para 158). As the pushover was initiated, rapid left lateral control application was required to maintain roll attitude (HQRS 3). Rotor speed changes (±3 percent, ±8 rpm) were within limits and required no pilot compensation to control (HQRS 2). Airspeed decreased 24 KTAS at completion of the parallel exit maneuver. The aircraft achieved the parallel exit with a 200-foot vertical displacement within 1170 feet horizontal distance. Although the pull-up with parallel exit vertical displacement maneuver is more difficult to accomplish than the pure pull-up vertical displacement, pilot confidence in aircraft maneuvering capabilities enabled accomplishment of this maneuver with minimal compensation (HQRS 3). Within the scope of this test, the helicopter was capable of performing the vertical displacement maneuver.

Lateral Displacement

33. Lateral displacement maneuvers were conducted at a mid-altitude test site under the conditions listed in table 1. The lateral displacement maneuvers were conducted to accomplish the following: (1) a constant-altitude 200-foot lateral displacement within 1100 feet horizontal distance without regard to exit flight path (pure turn); and (2) a constant-altitude turn with a turn reversal to parallel the original heading, such that the total lateral displacement was 200 feet within 1100 feet (turn with parallel exit). The target entry airspeed for both techniques was 150 KTAS. Representative time histories of these maneuvers are presented in figures 37 and 38, appendix G.

34. The pure turn maneuvers required greater pilot effort than for the parallel exit maneuvers. The pilot initiated the pure turn maneuvers by rapidly rolling to a 75- to 80-degree bank angle (roll rates were 50 to 54 deg/sec) and near simultaneously applying aft control to achieve load factors of 2.3 to 2.5. The large roll attitudes were required to maintain near-constant altitudes throughout the maneuver. Roll attitudes of less magnitude, combined with the aft control required

to complete the maneuver, resulted in excessive altitude deviation. Rotor speed control was excellent (less than 6 percent deviation) and required no pilot compensation to control throughout the maneuver (HQRS 2). Airspeed loss to achieve a 200-foot displacement was approximately 12 to 17 KTAS, although greater airspeed losses (approximately 40 to 60 KTAS) were incurred to restore the aircraft to a wings-level stabilized condition. The forward field of view during the pure turn maneuver was degraded but flight path over the ground was easily monitored out the side of the aircraft when established in the bank. Complicating both the pure turn and parallel exit maneuvers was the failure of the FAS computer during each maneuver. The force augmentation cues of apparent stick force per g (para 15, app C) were lost and a slight overcontrolling in the pitch axis occurred. The longitudinal control damper was retained following FAS computer failure and prevented the pilot from significant overcontrolling. The FAS computer failures were not attributable to the type of maneuver, as failures also occurred in nonmaneuvering flight, but rather to a malfunction of the vertical gyro. These inadvertent FAS computer shutdowns contributed in part to the considerable pilot compensation required to conduct the pure turn maneuver (HQRS 5). The aircraft was capable of being maneuvered so as to achieve a 200-foot lateral displacement within 1100 feet of horizontal distance without exceeding aircraft limits. The closest limit approached was the normal acceleration limit of 2.7g. Test data showed a maximum load factor of 2.5 and was attributed in part to the overcontrolling resulting from FAS computer shutdowns. The commitments of the PIDS were met.

35. The parallel exit maneuvers required less pilot effort than the pure turn maneuvers. The pilot initiated the maneuver similar to that for the pure turn, but not as much roll attitude (50 to 55 degrees) or load factor (1.5 to 1.75) was required compared to the pure turn. A roll reversal was initiated within 200 feet of lateral horizontal distance so as to establish the aircraft on a parallel heading within 1100 feet. Rotor speed decreased approximately 5 percent at initiation of the maneuver and then increased 2 to 4 percent upon roll-out. Collective control was not required to control rotor speed throughout the maneuver. Airspeed decreased 10 to 15 KTAS at the completion of the parallel exit maneuver. The pilot had to aggressively roll the aircraft for this maneuver, but the confidence in aircraft maneuvering capabilities enabled completion of this exercise with a minimal degree of pilot compensation (HQRS 3). Within the scope of this test, lateral displacement maneuvers were satisfactorily performed.

Autorotational Descent Performance

36. Steady-state autorotational descent performance tests were conducted at both the mission and alternate gross weight conditions listed in table 1. The tests were conducted by retarding the power control levers to the idle position and then stabilizing the aircraft on an airspeed and rotor speed. Descent rates were determined for varying airspeeds at the normal operating rotor speed of 98 percent (258 rpm) and for varying rotor speeds at an airspeed near the airspeed for minimum rate of descent (V_{min} R/D). Test results are presented in figures 39 and 40, appendix G.

37. The minimum rate of descent for both the alternate and mission gross weights was 2180 ft/min and occurred at 73 KCAS. The data indicate that $V_{min\ R/D}$ can vary approximately ±6 KCAS without increasing the rate of descent by more than 1 percent. This is a desirable characteristic, since it allows the pilot to concentrate on landing site selection without increasing rate of descent significantly if the airspeed should vary ±6 KCAS. The airspeed for maximum glide distance ($V_{max\ glide}$) was 108 KCAS and resulted in a 2650-ft/min rate of descent for both configurations. Minimal pilot effort was required to maintain $V_{min\ R/D}$ and $V_{max\ glide}$ (HQRS 3).

38. Rotor speed control during steady-state autorotation required minimal pilot effort to maintain a selected value within ±2 percent (HQRS 3). The fiberoptic display showed excellent trend information as to the rate of change of rotor speed and enhanced the pilot's ability to control rotor speed. The field of view during stabilized autorotational flight was similar to that for level flight, as only 1 to 2 degrees of pitch attitude change (nose-up) were required to transition from level to autorotational flight. The tests to determine the effects of rotor speed on rate of descent were conducted at 71 KCAS for the mission gross weight and 76 KCAS for the alternate gross weight. The rates of descent varied from 1970 ft/min at 90 percent rotor speed (237 rpm) to 2570 ft/min at 110 percent (289 rpm). Within the scope of this test, the autorotational descent characteristics are satisfactory.

Height-Velocity Performance

39. The single-engine H-V performance characteristics were evaluated at the conditions shown in table 1. Testing was accomplished to validate an AVSCOM-provided H-V diagram (ref 13, app A) based on contractor H-V testing. Four airspeed and altitude combinations from below the knee of the H-V diagram were selected as target end points. These end points were hover at 35 feet, 45 feet at 10 KTAS, 70 feet at 15 KTAS, and 125 feet at 20 KTAS. Test gross weight was adjusted to account for test-day variations in pressure altitude and temperature (para 29, app F). The test was accomplished by incrementally increasing altitude at a given airspeed until the selected test point or a limitation was reached. The helicopter was stabilized at the desired condition in dual-engine flight and then one engine was rapidly retarded to ground-idle and the helicopter landed. Figure A shows the AVSCOM H-V diagram and the results of the testing. Figures 41 through 43, appendix G, are time histories of the tested end points.

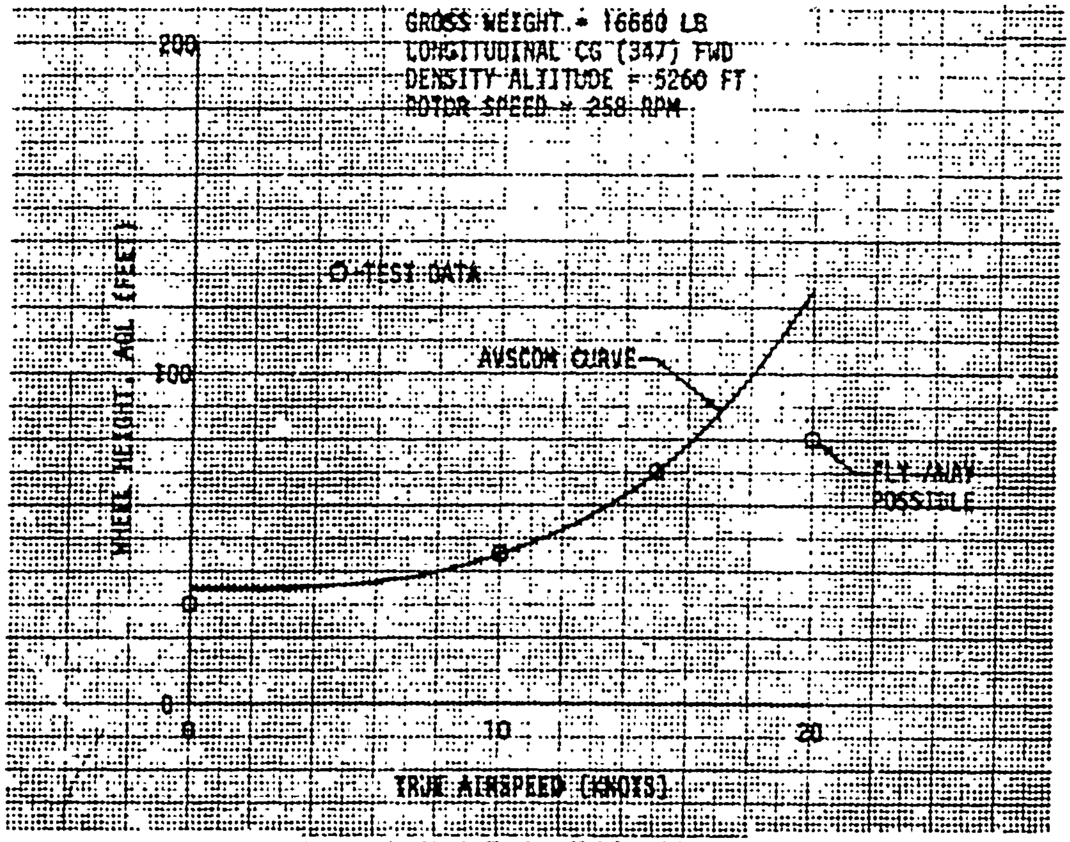

Figure A. Single-Engine Height Velocity.

40. At hover, the helicopter pitch attitude was held essentially level during the altitude build-up to 25 feet above ground level (AGL). For altitudes of 10 feet and higher, the collective control was lowered approximately 10 percent to decrease rotor speed decay, then increased rapidly to cushion the landing. At 20 feet, following simulated engine failure, rotor speed decay was more difficult to minimize (HQRS 5). At 30 feet, the nose was rotated to 21 degrees, nose-down, to gain forward motion and then the aircraft was flared prior to landing. Rotating the helicopter to gain forward motion decreased rotor speed decay at touchdown; however, pilot workload to control rotor speed was still high throughout the maneuvering (HQRS 4). The testing was terminated at 30 feet AGL when the minimum rotor speed (80 percent, 210 rpm) was reached during touchdown. There was relatively little engine noise change or variation in yaw attitude associated with the loss of one engine. The cue used by the pilot to initiate recovery from the simulated engine failure was the low rotor speed warning system, which

activated when rotor speed decreased to 95 percent (250 rpm). For hover heights above 10 feet AGL, the time from engine failure to a rotor speed of 95 percent was approximately 1 second. The delay time (the time from engine cut to collective application) was 1.2 seconds at 30 feet AGL. During the hover H-V testing the helicopter exhibited no degraded flying qualities at low rotor speeds (80 percent, 210 rpm) at touchdown. This excellent characteristic gave the pilot a high degree of confidence in performing the H-V maneuver, such that the low rotor speed limit observed during testing would not operationally be a limiting factor. The excellent aircraft control characteristics at low rotor speed (80 percent) are an enhancing feature. Within the scope of this test, the IGE hover H-V altitude limit for the YUH-60A helicopter is recommended to be 30 feet AGL.

41. The collective control and pitch attitude techniques for the forward airspeed H-V points were essentially the same as the 30-foot hover condition. At the forward airspeed points, rotor speed decay was easier to control as altitude was increased by rotating the aircraft to gain additional airspeed (HQRS 4). The tested H-V performance of the helicopter agreed with the AVSCOM curve at 10 KTAS (45 feet) and 15 KTAS (70 feet). To initiate recovery from the power loss, the pilot used the low rotor speed warning system as the primary cue, since engine noise and yaw attitude changes were minimal. At 10 KTAS, the time from power loss to 95 percent rotor speed was approximately 1 second and was independent of altitude. The time to 95 percent rotor speed at 15 KTAS was longer (1.3 seconds) and also independent of altitude. The delay times were 1.7 and 2.3 seconds at 10 and 15 KTAS, respectively. At the forward airspeed H-V points the helicopter exhibited the same excellent handling qualities at low rotor speed as were observed during the hover H-V testing. At 15 KTAS and altitudes above 30 feet, the collective control technique used prior to touchdown was different than during previous testing. Following collective application to arrest rate of sink, the collective had to be lowered slowly to complete the touchdown. At 20 KTAS and 80 feet, testing was discontinued when a single-engine fly-away capability was achieved, which indicated that the AVSCOM H-V diagram was conservative.

HANDLING QUALITIES

General

42. Stability and control, general operational, mission peculiar, and system re tests were conducted to qualitatively and quantitatively evaluate the handling qualities of the YUH-60A helicopter. Engine and rotor response characteristics were evaluated during simulated engine failure, autorotational entries and recoveries, and during specific power management tests. The excellent rotor speed control during normal mission tasks, along with the effective torque-matching capability and automatic $T_{4.5}$ limiting of the YT700-GE-700 engine, greatly reduced pilot workload in the area of power management. There was a high degree of pilot confidence in maneuvering the helicopter to the extremes of the flight envelope at mission gross weight because of low vibrations, excellent controllability, and generally excellent rotor speed control characteristics. The gust response was heavily

damped in all axes with AFCS ON. Aircraft response to sudden engine failures was mild for all flight conditions and entry into autorotational descent required minimal pilot effort. The aircraft was capable of returning to a landing under simulated IMC with a total AFCS failure. A total of 4 handling qualities-related deficiencies were identified. These include (1) the roll trim integrator displayed highly undesirable and potentially unsafe characteristics; (2) significant power losses associated with operation of both the engine anti-ice and cockpit heater systems; (3) high vibration levels in turning flight at 40 to 45 degrees of bank at alternate gross weight for airspeeds of 105 KCAS and greater: and (4) excessive engine/rotor speed transients following large-magnitude collective applications to the opposite extremes of the power demand schedule. A total of 23 shortcomings were identified. The most significant of these include: the high longitudinal and lateral control force gradients; poor control harmony, which resulted from the unsymmetrical lateral breakout forces; the lack of two VOR/ILS navigation radios; an undesirable buildup of pedal forces caused by the yaw trim integrator; the excessive delay in engagement of the airspeed hold system following trim actuation; lack of an aural engine-out warning device; the restriction to forward field of view caused by nose-up pitch attitudes during landing and rapid decelerations, in addition to the restricted field of view through the chin bubble. The test results were compared with the commitments of the PIDS and 15 instances of handling qualities specification noncompliance were noted.

Control System Characteristics

43. The mechanical characteristics of the AFCS were quantitatively evaluated on the ground with the backup hydraulic pump ON, external electrical power applied to the aircraft, engines and rotor stopped, and the FAS/TRM engaged. The results of the lateral and longitudinal control systems tests, recorded by on-board instrumentation and verified by a hand-held force gage, are presented in figures 44 and 45, appendix G. The flight control system mechanical characteristics are summarized in table 5. Quantitative measurements recorded in flight varied from the ground measurements as a function of the flight condition, due to AFCS inputs. The design force characteristics affecting the total force felt by the pilot are presented in figures 6 through 9, appendix C.

44. The YUH-60A has varying degrees of mechanical coupling which are detailed in paragraphs 11 and 12, appendix C. Cyclic control limits of travel at various collective and pedal positions are presented in figures 46 through 50, appendix G.

45. Longitudinal control system characteristics were further evaluated in flight and are summarized in table 5. Longitudinal control centering was positive throughout the flight envelope. Longitudinal free play was negligible (less than 0.5 percent). The longitudinal friction band varied from 0.75 pound near trim to 3 pounds near the longitudinal control limits. The breakout force (plus friction) was 0.75 pound forward and 1.4 pounds aft. The longitudinal control gradient measured on the ground was 3.25 pounds per inch of control movement forward and aft, which does not include the forces contributed by the AFCS. The gradient actually felt by the pilot in stabilized flight was approximately 7.5 pounds per

inch forward and aft. This gradient was objectionably high and did not include input from the pitch rate gain and the 0.93 pound per inch per second (lb/in./sec) longitudinal damper force (fig. 6, app C). The objectionable longitudinal gradient failed to meet the commitments of paragraph 50.3.2.2 (table 2) of the PIDS, in that the longitudinal gradient in flight was 7.5 pounds per inch (50 percent greater than the 5 pounds allowed). The objectionably high longitudinal control force gradient in flight is a shortcoming.

46. Lateral control system mechanical characteristics are summarized in table 5. The lateral control system friction band near trim was 1.5 to 1.6 pounds. There was approximately 0.5 percent of free play which was not noticeable during flight. Lateral control centering was positive throughout the flight envelope. The lateral control system gradient with no AFCS input was 2 pounds per inch left and right. The lateral gradient felt by the pilot as computed from design force inputs (fig. 8, app C) from the AFCS was 4 pounds per inch left and right. The asymmetrical lateral breakout forces (plus friction) (1.1 pounds left and 1.60 pounds right) and the longitudinal breakout forces (plus friction) were close to the same value and gave the feeling of higher forces required laterally than longitudinally. The higher lateral breakout force (plus friction) required to the right, combined with the pilot's decreased force capability in moving the cyclic control to the right, resulted in a high asymmetrical force feel. The unsymmetrical lateral stick forces, combined with inputs from the roll trim integrator (para 127), increased pilot workload in controlling the aircraft about the roll axis and required extensive pilot compensation (HQRS 5) to maintain the desired roll attitude during hovering tasks. The high lateral breakout forces (plus friction), combined with force inputs from the roll trim integrator, masked longitudinal forces during maneuvering flight, giving a poor indication of stick force per g. The lateral control system characteristics failed to meet the commitments of paragraph 50.3.2.1.1 of the PIDS, in that the right lateral control breakout force (plus friction) was 1.60 pounds (6.6 percent greater than the 1.5 pounds allowed). The lateral control system characteristics failed to meet the commitments of paragraph 50.3.2.2 (table 2), in that the lateral gradient was 4 pounds per inch in flight (33 percent more than allowed). The lateral control breakout forces did not meet the symmetry commitments of paragraph 50.3.2.1.2 of the PIDS, in that the right breakout force was 0.65 pound (or 50 percent) greater than the left breakout force (40 percent more than allowed). The unsymmetrical lateral breakout forces (plus friction) and the high lateral force gradient are shortcomings.

47. Directional control system breakout forces were 5 pounds right and left. Directional control free play was negligible and there were no centering characteristics, due to the pedal microswitch trim function (para 21, app C). The pilot was provided force cues proportional to the rate of control input (para 17, app C). These forces were reduced to zero when the heading hold feature activated. Several occurrences of turn coordinator malfunction, where the coordinator attempted to maintain out-of-balance flight (approximately one ball width), were noted during the test (EPR No. 74-06-1-44). Pedal force required to keep the ball centered increased to an uncomfortable level due to inputs to the turn coordinator from an out-of-null lateral accelerometer. The reliability of the coordinated turn

Table 5. Flight Control System Mechanical Characteristics.[1]

Axis	Direction of Movement	Breakout Force (including friction) (lb)	Limit Control Force[2] (lb)	Average Force Gradient[3] (lb/in.)
Longitudinal	Forward	0.75	16	3.25/*7.5*
	Aft	1.40	15	3.25/*7.5*
Lateral	Left	1.10	7	2.0/*4.0*
	Right	1.60	8	2.0/*4.0*
Directional	Left	5.00	—	—
	Right	5.00	—	—
Collective	Up	3.00	—	—
	Down	1.00	—	—

[1]Measured on the ground, rotors and engines stopped, FAS/TRM engaged, backup hydraulic pump on, and electrical power applied to the aircraft.
[2]Force required to move the control full travel from a mid (50 percent) control position.
[3]Italics: Gradient felt by the pilot in flight.

feature is further discussed in paragraph 128. With the exception of the coordinated turn problem, the directional control system is satisfactory.

48. The collective control system employs an adjustable friction device that enables the pilot to set the desired breakout force (including friction). The breakout forces measured with the friction device OFF were 1 pound down and 3 pounds up. The collective control system characteristics were satisfactory when the adjustable friction device was set to provide a positive force cue. The light collective control system force with friction OFF, coupled with the high degree of collective control effectiveness, resulted in collective overcontrolling or pilot-induced oscillations when attempting to maintain precision hover heights. This situation was aggravated in light-to-moderate turbulence. The following CAUTION should be included in chapter 8 of the operator's manual:

CAUTION

> Collective overcontrolling or pilot-induced oscillations may occur during precision hover tasks with collective control friction OFF. Sufficient collective friction should be set to provide a positive force cue.

49. The control system mechanical characteristics failed to meet the commitments of paragraph 50.3.2.10 of the PIDS, in that limitations in control power exist with certain combinations of control inputs (figs. 46 through 50, app G); however, this characteristic is not considered objectionable, since these combinations of control inputs were never encountered in flight.

50. The lateral and longitudinal control trim system was evaluated during the flight control system evaluation and concurrently during quantitative flight testing and mission maneuver tasks. Evaluation included the FAS/FPS longitudinal and lateral four-way trim rate (beep) trim device and the FPS instantaneous trim release. A detailed description of the trim system functions is contained in paragraphs 18 and 19, appendix C. The FAS/FPS beep trim system was quickly discarded by the pilot as a means to effect precise airspeed and/or attitude trim changes because of undesirable mechanical characteristics. The FAS/FPS had trim system lags of approximately 1.3 and 1.1 seconds for the longitudinal and lateral axes, respectively, and intermittent ratchety control movement (stick jump) was present. Additionally, the FAS/FPS pressure-operated cyclic trim switch did not have positive detents that cued the pilot when the trim switch was engaged. An infinite number of rates between the minimum and maximum were possible, depending on the pressure exerted on the trim switch. The method of determining trim actuation was waiting to detect a force or attitude change. These FAS/FPS trim system mechanical characteristics precluded precise trim changes required during hovering and simulated IMC tasks and required repeated trim actuations to initiate a response, at times exceeding a desired attitude or trim rate change. These undesirable trim system mechanical characteristics degraded pilot confidence in the system and resulted in almost total dependence on the instantaneous trim release to effect required trim changes. The instantaneous trim release was not satisfactory

because trim changes could not be discretely made in a single axis without potentially affecting existing trim positions in other axes. The lateral and longitudinal trim system mechanical characteristics failed to meet the commitments of paragraph 50.3.3.1.1.1 of the PIDS, in that stick jump was present when the trim control system was actuated. The undesirable mechanical characteristics of the pressure-operated beep trim switch are a shortcoming.

Control Positions in Trimmed Forward Flight

51. Control positions in trimmed (ball-centered) forward flight were evaluated for the conditions listed in table 2. Representative test results are presented in figures 51 through 53, appendix G.

52. The variation of longitudinal control position with airspeed during trimmed level flight generally required increasing forward cyclic displacement with increasing airspeed. Minor nonlinearities were noted in the position gradient at the lighter gross weight (forward cg), where a reversal occurred for airspeeds less than 58 KCAS (fig. 51, app G). At the heavy gross weight (fig. 52), the gradient was essentially positive. The control variation with airspeed in autorotation was essentially linear, with increasing forward cyclic displacement required with increasing airspeed (fig. 53). The variation of lateral and directional control was minimal with changes in airspeed for all conditions. The pitch attitude variation was also minimal in level flight, with a maximum variation of from 4 degrees nose-up at 42 KCAS to 3 degrees nose-down at 140 KCAS. In autorotation the pitch attitude did not vary with airspeed (fig. 53). The control positions in trimmed forward flight are satisfactory and meet the commitments of the PIDS.

Static Longitudinal Stability

53. Static longitudinal stability characteristics were evaluated at the conditions listed in table 2. These tests were accomplished by first trimming the aircraft at the desired airspeed, then with collective fixed, the helicopter was displaced from trim and stabilized at incremental airspeeds greater and less than the trim airspeed. Data were recorded at each stabilized airspeed and are presented in figures 54 through 58, appendix G.

54. With AFCS ON, the helicopter exhibited positive control force stability (±5 to 8 knots from trim) for airspeeds greater than 60 KCAS. For larger airspeed increments from trim, the control force stability was neutral. Control position stability was essentially positive about the trim airspeeds above 65 KCAS but the gradients were shallow (less than 0.75 inch for a 15-KCAS change from trim). The force gradient was high for airspeeds about trim and when combined with the shallow position gradient, resulted in poor force-movement cues. At 35 KCAS, longitudinal control force stability was negative for airspeed changes of ±20 KCAS from the trim airspeed. Control force instability was 3 to 4 pounds at airspeeds ±15 KCAS from trim. The magnitude of change of control position was a maximum of 0.5 inch. The control force instability characteristics required considerable pilot compensation to maintain desired pitch attitudes and airspeeds

(HQRS 5). The unstable longitudinal control force stability cues at representative nap-of-the-earth (NOE) airspeeds will degrade mission accomplishment, particularly at night. For maximum rate of climb and minimum rate of descent at 70 KCAS, the longitudinal control force stability characteristics were similar to level flight, although the control position stability exhibited a steeper positive gradient. The level flight static longitudinal stability characteristics did not meet the commitments of paragraph 50.3.3.1.3 of the PIDS, in that control force and control position stability was not positive for all increasing and decreasing airspeed values from trim. Additionally, the magnitude of control force instability in the 15- to 50-KCAS airspeed range was 3 to 4 pounds, exceeding the 1-pound limit of paragraph 50.3.3.1.3. The high longitudinal control force to small control position changes about trim at 60 KCAS and greater, and the unstable longitudinal control force stability at representative NOE airspeeds (40 KCAS and lower) are shortcomings.

55. The AFCS OFF static longitudinal stability characteristics were evaluated at the trim and degraded mode conditions listed in table 2. The control position stability characteristics were similar to AFCS ON, while control force stability characteristics were neutral. Without control force cues and the shallow control position stability gradient, extensive pilot compensation was required to stabilize and maintain a level flight airspeed (HQRS 6). Failure of the FAS computer, as simulated during this test, produced undesirable static longitudinal stability characteristics; however, there was no shortcoming identified because of the degraded mode condition of the AFCS.

Static Lateral-Directional Stability

56. Static lateral-directional stability characteristics were evaluated at the conditions indicated in table 2. Tests were conducted by trimming the aircraft in ball-centered flight at the desired conditions. With collective control fixed, the aircraft was then stabilized at incremental sideslip angles up to limit sideslip on both sides of trim while maintaining a steady heading at the trim airspeed. Test results are presented in figures 59 through 62, appendix G.

57. Static directional stability, as indicated by the variation of directional control position with sideslip, was positive at all test conditions. Directional control variation with sideslip was essentially linear; however, the gradient was shallow at 53 KCAS in level flight and during autorotational descents. Directional control force stability was nonlinear in level, climbing, and descending flight, with a strong discontinuity about trim in autorotational descents (figs. 59 through 62, app G). The increase of approximately 30 pounds right directional control force with an increase in left sideslip angle from trim (fig. 62) was noticeable to the pilot and was probably caused by the yaw trim integrator. These unusual forces would present out-of-trim cues to the pilot and therefore are not objectionable. The directional stability characteristics of the YUH-60A met the commitments of the PIDS and are satisfactory.

58. Dihedral effect, as indicated by the variation of lateral control position with sideslip, was positive and essentially linear in level flight and in autorotational descents. In IRP climbs, left lateral control required for an increase in left sideslip (from 20 to 29 degrees) decreased 0.4 inch (4.3 percent). This negative stability was not objectionable to the pilot. Dihedral effect, as indicated by the variation of lateral control force with sideslip, was nonlinear in level flight at 88 KCAS (fig. 60, app G), in climbs at 75 KCAS (fig. 61), and in autorotational descent at 77 KCAS (fig. 62) with a noticeable discontinuity about trim, especially in IRP climbs. These discontinuities were noticeable during performance of the tests and were probably caused by the roll trim integrator. For the operational pilot these forces would present good out-of-trim cues and therefore are not considered objectionable. The effective dihedral of the YUH-60A met the commitments of the PIDS and is satisfactory.

59. Side-force characteristics, as indicated by the variation of bank angle with sideslip, were positive for all conditions tested except in IRP climbs at 75 KCAS. At these conditions the left roll attitude decreased from 6 degrees to 1 degree left as left sideslip increased from 5 to 20 degrees and then increased again above 20 degrees. This negative side-force characteristic between 5 and 20 degrees sideslip failed to meet the commitments of paragraph 50.3.4.1.7 of the PIDS. The side-force characteristics at 53 KCAS in level flight and at 77 KCAS in autorotation were weak; however, the unusual force characteristics mentioned in paragraphs 57 and 58 make it difficult for the pilot to inadvertently attain these out-of-trim flight conditions. Since it is difficult for the pilot to inadvertently attain this flight condition, this characteristic is not considered a shortcoming. Within the scope of this test, the side-force characteristics of the YUH-60A are satisfactory.

60. The pitch-due-to-sideslip coupling of the aircraft was primarily influenced by lateral acceleration feedback to the stabilator and the pitch-to-yaw mechanical mixing (para 11, app C). Figures 59 and 60, appendix G, show that aft longitudinal control is required for left sideslip and forward longitudinal control for right sideslip. This characteristic does not necessarily hold true during climbs. The pitch-due-to-sideslip coupling is not objectionable.

61. The YUH-60A exhibited inherent sideslip angles of 5 degrees right during ball-centered IRP climbs to 15 degrees left sideslip during ball-centered autorotational descents. The average degree of sideslip during cruising flight was 3 degrees left.

62. Turns with lateral cyclic only were conducted at airspeeds above 83 KCAS with AFCS ON. The turn coordination feature (para 22, app C) maintained a centered ball for left and right turns up to 30 degrees of bank using cyclic only. A lateral control step input sufficient to produce a 30-degree roll displacement in 6 seconds showed no adverse yaw for left and right lateral control inputs.

Maneuvering Stability

63. Maneuvering stability was evaluated in left and right steady-state turns, symmetrical pull-ups and pushovers, and during sudden pull-ups with FPS ON and OFF at the conditions listed in table 2. Steady-state turns were conducted by establishing the desired level flight airspeed and then stabilizing at increasing bank angles while maintaining collective control and airspeed constant. Symmetrical pull-ups and pushovers were conducted by alternately climbing and diving the helicopter to achieve varying normal accelerations (g) while the aircraft was passing through the trim altitude at the desired airspeed. Sudden pull-ups were conducted by rapidly displacing the longitudinal control aft, using various magnitudes of displacement. A control fixture was used to obtain the desired input size. Results of the maneuvering stability tests are presented in figures 63 through 70, appendix G.

64. The stick-fixed maneuvering stability in steady-state turns, as indicated by the variation of longitudinal control position with normal acceleration (g), was essentially neutral for all test conditions (FPS ON and OFF), with the exception of a negative trend at 105 KCAS, alternate gross weight, with FPS ON. The stick-free maneuvering stability, as indicated by the variation of longitudinal control force with g, was positive at all conditions with FPS OFF and neutral to negative for all conditions with FPS ON. The longitudinal control force gradient was 9 to 10 pounds per g in steady-state turns with FPS OFF. The differences in maneuvering stability with FPS ON and OFF were a result of FAS/FPS computer logic for this flight condition. The airspeed and pitch attitude hold functions of the AFCS precluded FAS logic (para 15, app C) from developing apparent stick force per g when in the FPS mode. The neutral to negative control position and force stability, FPS ON, was not considered a problem, as the data reflected the requirements of a unique engineering flight test maneuver where airspeed was required to be maintained within 1 knot from trim. If the airspeed hold function of the FPS was permitted to operate in its tolerance band (3 to 5 knots) during turning flight, the apparent control force stability was neutral and reduced pilot workload during simulated IMC tasks.

65. At mission gross weight, the pilot was able to stabilize on airspeed and at a precise bank angle up to 45 degrees with minimal pilot compensation (HQRS 3). Pilot effort increased for stabilizing at bank angles greater than 45 degrees because of the negative stick-free maneuvering stability characteristics (FPS ON), with greater effort required for right than for left turns. A slight cross-coupling effect (left lateral control required for right roll) occurred for bank angles less than 40 degrees. Unpredicted pitch trim change requirements were experienced at bank angles approaching 60 degrees to the right that precluded precise airspeed and roll attitude control. Extensive pilot compensation was required to stabilize on exact airspeeds while maintaining constant bank angles of 55 degrees or more (HQRS 6). Maintaining exact airspeed at bank angles in excess of 55 degrees is not an operational maneuver; therefore, the high pilot workload reported does not constitute a shortcoming.

66. The helicopter exhibited excessive vibrations in steady turning flight near alternate gross weight (19,200 pounds and above). At 40 to 45 degrees of bank, the vibrations at the pilot seat and instrument panel were so great that interpretation of the flight instruments was extremely difficult. The pilot felt extremely uncomfortable maneuvering the aircraft at these conditions and such vibrations would probably deter operational pilots from this flight regime. The high vibration levels in turning flight at alternate gross weight are further discussed in paragraph 151.

67. Symmetrical pull-up and pushover tests showed positive stick-fixed and stick-free maneuvering stability with FPS ON or OFF. The longitudinal control force gradient during pull-ups varied from a high of 21 pounds per g at the higher airspeeds tested (130 KCAS and greater, FPS OFF) to 10 pounds per g at 106 KCAS (FPS ON). The longitudinal control push forces during pushovers varied from approximately 5 pounds per g at 106 KCAS (FPS ON) to 16 pounds per g at 143 KCAS (FPS OFF). Longitudinal control forces during sudden pull-ups were approximately 16 to 17 pounds per g at 136 KCAS (FPS ON). The longitudinal control force during the symmetrical pull-ups and pushovers and during sudden pull-ups provided the pilot with adequate normal acceleration cues. For symmetrical pull-ups the data showed that cg normal acceleration did not always increase with time until maximum acceleration was achieved. For sudden pull-ups, the cg normal acceleration data showed an initial peak corresponding to the maximum pitch acceleration developed and then the normal acceleration would continue to increase uniformly as pitch rate developed. This discontinuity was not apparent to the pilot. Rotor speed control was good throughout the conduct of these maneuvers, requiring no pilot compensation to maintain rotor speed within the transient limits (HQRS 2). Maximum excursions from the trimmed rotor speed (98 percent) were a 3.5-percent increase (101.5 percent) during a 2.25g pull-up and an 8.5-percent increase (106.5 percent) during a -0.25g pushover. Roll-due-to-pitch coupling was evident during the pull-up and pushover maneuvers. A pitch-up resulted in right roll and pitch-down produced a left roll. This coupling was easily compensated for by the pilot and did not detract from the maneuvering capabilities of the helicopter. A slight yaw due to the roll coupling was also evident during these maneuvers. It was noticed that the ball was one to two widths from this center position when maneuvering at the high (2.0g) and low (0.0g) values. The sideslip excursions were well within flight limits for these conditions. Minimal pilot effort to compensate for yaw-due-to-roll coupling was required when maneuvering the helicopter (HQRS 3).

68. The commitments of paragraph 50.3.3.1.4 of the PIDS were not met, in that the YUH-60A did not exhibit positive cyclic control force or position versus normal acceleration in steady turning flight with FPS ON. This is not considered a problem, however, as the commitments were met with FPS OFF, which would be the normal AFCS mode of operation in tactical maneuvering situations. Also, the stick force per g gradient at 130 KCAS exceeded the maximum specified of 20 pounds per g by 1 pound. Additionally, paragraph 50.3.3.1.4.1 was not met in that the ratio of maximum longitudinal control force to peak normal acceleration for sudden

pull-ups was not always greater than the ratio obtained in steady acceleration. All other maneuvering stability commitments were met. The maneuvering stability characteristics are satisfactory.

Dynamic Stability

69. The short-term and long-term longitudinal dynamic stability characteristics of the YUH-60A helicopter were evaluated at the conditions listed in table 2. Gust response characteristics were simulated in all control axes by making single-axis 1-inch control inputs which were held for approximately 0.5 second, and by releases from steady-heading sideslips during lateral-directional tests. The longitudinal long-term response was evaluated by releasing the controls from stabilized off-trim airspeed conditions. The dynamic stability tests were conducted controls-free for the axis evaluated because of the attitude-hold characteristics of the AFCS. The trimmed attitude was maintained in the other axes.

70. The short-term response characteristics of the helicopter with AFCS ON were essentially deadbeat, as shown in figures 71 through 74, appendix G. The deadbeat short-term characteristics were also evident for all axes during flight through moderate turbulence. The pilot was able to correct for attitude disturbances in level flight and hover with minimal effort (HQRS 3). The short-term characteristics were also evaluated with AFCS OFF and a representative time history is presented in figure 74. In this mode, pilot-induced oscillations occurred about all axes during all flight phases due to the loss of rate damping and attitude hold. Aircraft control during precision hovering and simulated IMC tasks with AFCS OFF required extensive pilot compensation (HQRS 6). Since AFCS OFF is a degraded mode of operation, these flight characteristics were not identified as a shortcoming. The AFCS OFF handling qualities were significantly degraded at airspeeds greater than 100 KIAS. At 75 to 80 KIAS pilot workload was minimized (para 123). The short-period response characteristics met the commitment of the PIDS and are satisfactory. The following CAUTION should be included in chapter 4 of the operator's manual:

CAUTION

> Intentional flight in instrument meteorological conditions (IMC) should not be conducted with an AFCS failure that renders the FAS/FPS and/or SAS inoperative. If failure should occur during IMC flight, reduce airspeed to 75 to 80 KIAS to minimize pilot workload.

71. Additional lateral-directional gust response characteristics were evaluated by releases from steady-heading sideslips. The lateral-directional oscillatory responses were highly damped. During flight through light-to-moderate turbulence the aircraft exhibited deadbeat lateral-directional response characteristics which permitted flight and hover in turbulent conditions with no pilot compensation required (HQRS 2). The lateral-directional gust response characteristics met the commitments of the PIDS and are satisfactory.

72. Spiral stability characteristics were evaluated at 117 KCAS by establishing 5 degrees of roll attitude with pedal only or by lateral cyclic only and then returning the controls to trim and observing the resultant roll attitude with controls fixed. The tests were conducted with FPS OFF and with FPS and FAS/TRIM OFF for both the pedal and lateral cyclic-only inputs. The spiral stability characteristics were positive. The aircraft returned to the trimmed level attitude for releases from 5 degrees of right bank. Releases from left bank showed positive stability through trim and resulted in stabilized roll attitudes of up to 30 degrees of right bank. Although the bank angle was halved in less than 10 seconds, the positive spiral stability characteristics were not objectionable. The commitments of paragraph 50.3.4.4 of the PIDS were not met, in that the bank angle from a steady 5-degree banked turn to the left was halved in 4 seconds, failing to meet the commitment by 6 seconds (60 percent).

73. The long-term characteristics (AFCS ON) showed positive damping and a return-to-trim airspeed within one cycle for both higher and lower entry airspeeds. Figure 75, appendix G, is representative of the long-term characteristics, AFCS ON. The airspeed and pitch attitude hold characteristics of the FPS precluded development of the long term throughout the forward flight regime (60 KCAS and greater). The long-term characteristics at 117 KCAS, AFCS OFF, were unconventional, in that a drift in apparent trim airspeed was noted for both higher and lower entry airspeeds. The slower entries resulted in a steadily decreasing trim airspeed which was 78 KCAS when recovery was initiated. The long term was easily excited with AFCS OFF and required extensive pilot compensation to maintain trim airspeed and pitch attitude (HQRS 6), but is not considered a problem because of the degraded AFCS mode. This contributed to the pilot workload reported in paragraph 70. The long-term characteristics of the helicopter met the commitments of the PIDS and are satisfactory.

__Controllability__

74. Controllability tests were conducted at the conditions shown in table 2. Step control inputs of varying magnitudes were made in the longitudinal, lateral, and directional axes, utilizing control fixtures to obtain the desired input size. Representative results are presented in figures 76 through 82, appendix G. The GCT controllability test results were identical to APE I data with the exception of lateral control response and sensitivity. These lateral control system characteristics were changed due to a 17-percent increase in the lateral SAS gain that had been implemented by Sikorsky Aircraft prior to commencement of the GCT. Summary plots for control response (deg/sec/in.) and control sensitivity (deg/sec^2/in.) for each control axis are presented in figures 83 and 84. Representative time histories of control inputs are presented in figures 85 through 88.

75. The longitudinal control response with AFCS ON was essentially constant (7 to 10 deg/sec/in.) at 73 and 111 KCAS and was unchanged from the data gathered during APE I. The longitudinal control sensitivity was varied from 13 to 18 deg/sec^2/in. at 73 and 111 KCAS, respectively. The longitudinal controllability

characteristics of the helicopter showed angular acceleration response within 0.2 second following a step control input and the angular velocity was concave downward within 0.2 second after the start of the maneuver and remained concave downward until the attainment of maximum angular velocity. These characteristics permitted precise pitch attitude control throughout the flight envelope. Pitch-to-roll coupling was evident to the pilot throughout the flight regime but did not contribute to an increase in pilot workload. Within the scope of this test, the longitudinal controllability commitments of the PIDS were met and are satisfactory.

76. The lateral control response, AFCS ON, was 10 to 13 deg/sec/in. of control displacement for 77 and 116 KCAS, respectively. The lateral control sensitivity was an average of 30 deg/sec^2/in. for both airspeeds for the above conditions. Aircraft response to lateral control step inputs produced a rapid roll in the proper direction with no perceptible lag in attitude change. There was no perceptible yaw associated with the lateral control inputs, nor were any reversals in rolling velocity noted throughout the range of control inputs. Precise bank angles could be commanded during all phases of maneuvering flight with no tendencies to overcontrol (HQRS 2). The pilot gained immediate confidence to maneuver the aircraft in roll to the extremes of the flight envelope (para 35). Aircraft control was easily accomplished in the roll axis during representative mission maneuvers. The lateral controllability characteristics met the commitment of the PIDS and are satisfactory.

77. Directional control response, AFCS ON, was approximately 12 deg/sec/in. and sensitivity was 20 deg/sec^2/in. at 77 and 114 KCAS. Hover directional controllability tests conducted at the high-altitude test site (11,200-foot density altitude) showed an average control response of 10 deg/sec/in. and control sensitivity of approximately 15 deg/sec^2/in. for both left and right directional inputs. Figures 80 through 82, appendix G, are representative test results. The data show significant increases in yaw rates that were generated compared to APE 1 test results at lower test altitudes. Yaw SAS saturation occurred during all step control inputs in a hover. This resulted in the secondary increases in yaw acceleration and rate that occurred. The times to 63 percent of steady-state yaw rate were not computed as the maximum yaw rate was not obtained before recovery. During precision hovering and low-speed tasks, the pilot was able to control yaw attitude with minimal compensation (HQRS 3) because of the directional control effectiveness characteristics. The directional control system characteristics met the commitment of the PIDS and are satisfactory. During normal maneuvering of the helicopter, the pilot was able to command desired yaw rate and attitude responses during hover and low-speed flight with minimal pilot effort (HQRS 3).

78. Collective control system characteristics were evaluated by making collective step control inputs (up) at 77 and 114 KCAS. The aircraft's vertical control response to a collective control input produced 63 percent of initial peak value of normal acceleration within 0.35 second for all cases. There was no control force coupling in the cyclic controls upon movement of the collective. The collective control system characteristics met the commitments of the PIDS and are satisfactory.

79. Controllability characteristics of the aircraft with AFCS OFF (SAS, FAS/TRM, FPS) are presented in figures 77, 79, 80, and 82, appendix G. The increases in rate response (AFCS OFF) reflect the loss of damping that was provided by the SAS. The pilot was required to reenter the control loop and provide his own measure of damping to arrest the rates being developed for all axes. The high level of control response, coupled with the long times to steady state due to the loss of rate damping, was particularly evident when flying through light-to-moderate turbulence. There was a tendency to overcontrol the aircraft in all axes and, combined with the cross-coupling characteristics, extensively increased pilot workload during precision hovering and simulated IMC tasks (para 123).

Ground Handling Characteristics

80. The ground handling characteristics of the YUH-60A helicopter, which included starting, systems checks, taxiing, and engine/rotor shutdown, were evaluated concurrently with other tests. The starting and shutdown procedure for the YUH-60A is simple and straightforward, with many of the sequences automatic. The main engines could be started simultaneously under most conditions. Main rotor engagement and disengagement was easily accomplished in winds up to 25 knots, which was the maximum experienced during the GCT. During simultaneous dual-engine starts above 30°C, however, the APU would flame out and cause either hung or aborted starts on one of the engines. The following NOTE should be included in chapter 3 of the operator's manual:

NOTE

> Simultaneous dual-engine starts should not be attempted when the outside air temperature is above 30°C, due to the possibility of APU flameout.

81. Ground taxiing of the helicopter was easily accomplished on paved level surfaces. Forward motion required only minor collective and forward cyclic control manipulations. Desired taxi direction was easily maintained without the use of wheel brakes and 360-degree turns were easily accomplished in either direction in gusting winds up to 25 knots. At no time during the ground taxi maneuvers did the directional control contact the control stops.

82. Detracting from the overall desirable ground handling characteristics was the difficulty experienced in unlocking and locking the tail wheel. The pilot had to manipulate the directional controls to relieve pressure on the tail wheel locking pin to effect engagement or disengagement. When difficulties were experienced in unlocking the tail wheel, an excessive ground roll was usually required prior to achieving an unlocked condition. As a result of manipulating the directional control while unlocking the tail wheel, the external manual lock lever would at times become positioned such that locking the tail wheel from the cockpit was not possible until the helicopter had been brought to a stop and a crewman had reset the manual lock latch. The difficulty in unlocking and locking the tail wheel is a shortcoming. The frequent inability to lock the tail wheel from the cockpit is a shortcoming.

83. A pressure refueling and defueling system permits complete refueling and defueling of both fuel tanks from one point on the left side of the helicopter. The design capabilities of this system are a refueling rate of 300 gallons per minute at a refueling nozzle pressure of 55 psig. The system was tested during the USN shipboard evaluation (para 2) and enabled refueling within 2 minutes from a near-empty fuel loading. The single-point pressure refueling system expedites ground turnaround of the helicopter and allows for hot refueling from a single source. The single-point pressure refueling and defueling system is an enhancing characteristic.

Takeoff and Landing Characteristics

84. Takeoff and landing characteristics were evaluated with other tests throughout the test program. Normal takeoffs and landings, steep angle and vertical approaches and takeoffs, and jump takeoffs were evaluated.

85. Normal takeoffs and landings were easily accomplished (HQRS 3). Steep or vertical landings were difficult due to the restricted field of view through the chin bubble (para 158) (HQRS 6). Jump takeoffs were accomplished by rapidly increasing collective control (approximately 1 second) to the maximum dual-engine torque, accompanied by the required cyclic and directional control manipulations to lift off and transition to forward flight. The application of collective caused a 2- to 3-percent transient rotor speed droop which returned essentially to trim within 1 second. Pilot effort was not required to control rotor speed during jump takeoffs (HQRS 2). The jump takeoff characteristics are satisfactory.

Slope Landing Evaluation

86. The slope landing and takeoff capabilities of the YUH-60A helicopter were evaluated at a mid cg and mission gross weight in winds of less than 3 knots. Vertical landings and takeoffs were performed on surveyed soil-stabilized slopes of 12 and 15 degrees. Control margins, aircraft attitudes when on a slope at flat pitch, and ability to maintain positive control during landings and takeoffs were investigated. The control margin remaining during landings and takeoffs, and aircraft attitude for each slope orientation, are presented in table 6.

87. The technique employed during landing was essentially the same for each slope orientation tested. Parking brakes were on and the tail wheel locked during the testing. Coordinated cyclic, collective, and directional control movements were required until the helicopter was firmly positioned on the slope. For left, right, and nose-downslope landings, the tail wheel made ground contact first, increasing pilot effort for roll control until the main landing gear contacted the ground. Nose-upslope landings required less effort for roll attitude control because the main landing gear contacted the ground first. For all slope landing orientations tested, the wheel brakes were effective in maintaining position on the slope after the collective was lowered. The aircraft attitudes on all but a 15-degree nose-upslope

Table 6. Slope Landing and Takeoff Control Margins.

Slope[1] (deg)	Aircraft Attitude (deg)	Minimum Control Margin Remaining[2] (in.)					
		Longitudinal		Lateral		Directional	
		Landing	Takeoff	Landing	Takeoff	Landing	Takeoff
12 R	12 right	4.4 aft	4.0 aft	2.7 right	2.9 right	2.0 left	1.8 left
12 NU	15 nose-up	3.9 fwd	3.0 fwd	3.8 left	3.0 left	2.0 left	2.0 left
12 L	12.1 left	4.2 aft	4.3 aft	2.0 left	1.6 left	2.1 right	2.0 right
12 ND	13 nose-down	0.0 aft	0.0 aft	4.5 left	3.5 left	2.2 left	2.0 left
15 R	18 right	4.2 aft	4.0 aft	0.5 right	1.0 right	1.8 left	1.5 left
15 NU	16 nose-up	2.0 fwd	2.2 fwd	3.2 left	3.0 left	1.9 left	1.9 left

[1] Aircraft orientation:
ND: Nose-downslope.
NU: Nose-upslope.
L: Left gear upslope.
R: Right gear upslope.

[2] Full travel:
Longitudinal: 9.2 inches.
Lateral: 10.7 inches.
Directional: 9.0 inches.

were measured at the aircraft leveling plate with an inclinometer. The difference between aircraft attitude and slope angle was due to differential compression of the gear struts.

88. For the 12-degree slope, adequate control margin remained in all axes for all aircraft-slope orientations, except for the nose-downslope orientation. The aircraft could not be held stationary on the slope just prior to main gear-ground contact with full aft longitudinal control applied. With 100 percent aft cyclic control applied, the aircraft translated downslope if vertical descent was stopped just prior to main gear touchdown. When a positive rate of descent was maintained to solid main gear touchdown, the aircraft was capable of landing nose-downslope with approximately 2 feet of downslope translation. The ability to maintain positive control to the ground during a 12-degree downslope landing was demonstrated during the APE; however, only one stage of the two-stage tail gear strut had been pressurized. Pressurizing the second stage of the tail gear strut caused the aircraft to sit at a more nose-low attitude and resulted in an increase in aft cyclic required during nose-downslope landings. Vertical takeoff from a 12-degree nose-downslope while maintaining positive control was impossible and with 100 percent aft cyclic control applied, the aircraft translated downslope. The commitments of paragraph 50.3.7.7.2 of the PIDS were not met, in that the helicopter could not land under positive control on a 12-degree nose-downslope. The inability to positively control the aircraft to the ground on a 12-degree nose-downslope is a shortcoming. The following NOTE should be included in chapter 7 of the operator's manual:

NOTE

During nose-down landings and takeoffs on 12-degree slopes, the aircraft will translate downslope. The pilot must ensure that adequate obstacle clearance is available prior to landing.

89. While on a 12-degree slope in the nose-upslope orientation, the transmission oil pressure was at the 25- to 30-psi range. This is at the top of the red, bottom of the yellow caution minimum pressure range, indicating that the limit of the nose-upslope landing capability had been attained.

90. On a 15-degree slope, a right main gear upslope landing required minimal pilot compensation (HQRS 3). A nose-upslope landing on a 15-degree slope required considerable pilot effort to attain a gradual touchdown of the tail gear (HQRS 4), due to instability in pitch attitude control with only the main gear on the ground. As soon as tail gear touchdown was achieved the transmission oil pressure dropped to 10 psi. Shortly after the oil pressure dropped, the transmission oil pressure caution light illuminated. The aircraft was picked up to a hover and the transmission oil pressure returned to normal. The CHIP MAIN XMSN light then illuminated. An attempt was made to check the location of the chip but the undesirable location of the chip locator panel (para 161) precluded rapid identification of the affected module, and identification could not be made until the aircraft wwas set down. The YUH-60A slope landing capabilities met the

commitments of the PIDS for a 15-degree slope. The YUH-60A should be restricted to a maximum of 12 degrees nose-upslope landing due to the inability of the transmission oil pumps to maintain adequate oil pressure while in the nose-upslope orientation on slopes greater than 12 degrees.

91. During slope operations vertical clearance between the main rotor tip path plane and the ground is extremely reduced on the upslope side of the helicopter. During testing on a 15-degree slope, the tip path to ground clearance was approximately 3 feet on the upslope side. This tip path clearance is a hazard to personnel in close proximity to the helicopter during slope operations. The following WARNING should be included in chapter 8 of the operator's manual:

WARNING

> During cross-slope and nose-up slope operations, vertical clearance between the main rotor tip path plane and the ground is extremely reduced on the upslope side of the helicopter. Personnel must be warned not to approach the helicopter from the upslope direction or depart from the helicopter in the upslope direction.

Low-Speed Flight Characteristics

92. The low-speed flight characteristics of the YUH-60A were evaluated at the Edwards Air Force Base and Coyote Flats test sites at the conditions listed in table 2. These tests were accomplished to determine control margins and handling characteristics in low-speed flight, with simulated wind conditions from various relative azimuths. The testing at Edwards also included an evaluation with an empty CONEX sling load (1580 pounds). A ground pace vehicle was used as an airspeed reference for the conduct of all the tests, except at Coyote Flats, at airspeeds above 35 KTAS, where a radar speedgun was used as a reference. Surface wind conditions were 3 knots or less. The low-speed flight test data are presented in figures 89 through 96, appendix G.

93. During steady low-speed flight adequate control margins remained, with little variation between the conditions tested. The minimum longitudinal, lateral, and directional control margins observed during the tests were at the Coyote Flats test site. The minimum longitudinal control margin was 1.7 inches (18 percent) aft longitudinal control remaining during left sideward flight at 52 KTAS (fig. 91, app G). At 40 KTAS, 120 degrees relative azimuth, 2.7 inches (25 percent) of left lateral control remained (fig. 96). The minimum control margin incurred during the tests was the left pedal control during right sideward flight at 35 KTAS (fig. 91). The control margin remaining was approximately 1.3 inches (22 percent). During testing the maximum control excursions noted were at the 120-degree relative azimuth at Edwards (no sling load) and Coyote Flats. The transient values were maximum in the longitudinal and lateral cyclic control but

sufficient control margin existed at all airspeeds. The critical azimuth, the wind azimuth at which the minimum control margin remaining occurred, was determined to be 90 degrees.

94. The variation of control positions in low-speed flight showed varying gradients in all axes, depending on the direction and magnitude of the relative wind azimuth; however, these variable gradients did not present a control problem. The only control position gradient noticeable to the pilot was the abrupt change in lateral control position required when passing through translational lift both in forward and rearward flight (figs. 90 and 92, app. G). Longitudinal control position was nonlinear and very shallow in rearward flight to 40 KTAS and positive and essentially linear in forward flight to approximately 30 KTAS with a reversing, but not objectionable, gradient from 32 to approximately 55 KTAS. The same longitudinal trends were exhibited at Coyote Flats and with the sling load. The undesirable force characteristics of the cyclic control system (paras 45 and 46) degraded precise low-speed translational flight in the YUH-60A However, once the helicopter was stabilized in low-speed flight (in any azimuth), pilot effort was significantly reduced. Only minimal control manipulations were required to maintain the helicopter at the desired stabilized condition (HQRS 3). The handling qualities of the YUH-60A helicopter during low-speed flight and hover in steady winds from any azimuth are satisfactory and met the commitments of the PIDS.

Power Management

95. The power management characteristics of the YUH-60A helicopter were investigated at the conditions listed in table 2 and during representative mission maneuver evaluations. Evaluations were conducted to determine engine acceleration, deceleration, and lag characteristics; static and transient droop characteristics; rotor speed control; torque-sharing; engine stability; beep system characteristics; emergency power management; and an evaluation of a Group 8 ECU. Power management evaluations during mission maneuver tests included jump takeoffs, rapid accelerations and decelerations, terrain-avoidance maneuvers, and autorotational recoveries. Test results are presented in figures 97 through 99, appendix G.

96. The tests to determine engine acceleration and rotor droop during collective pulls to IRP from single-engine autorotational flight showed excessive delays in engine response, which resulted in excessive rotor droop for collective inputs as slow as 0.67 in./sec for over 4 inches of control travel. The fastest collective input evaluated was 1.3 in./sec and resulted in main rotor droop to 89 percent (229 rpm) and approximately 6 seconds from the time of this maximum droop was required to return to operating rotor speed. Engine gas generator speed (N_G) required from 1.5 to 4.5 seconds to accelerate to an IRP setting for the fast and slow inputs, respectively. The times required for a 95-percent power change from flight-idle to IRP varied from 10 to 12 seconds, and the time required to stabilize at 100 percent power change ranged from 18 to 22 seconds from initiation of the collective pull for the fast and slow conditions, respectively. Similar engine/rotor response characteristics were noted during dual-engine recoveries from

autorotational flight. The engine deceleration characteristics for a 95-percent power reduction showed times of 7 to 3.1 seconds for collective input rates varying from 1.1 to 3.15 in./sec, respectively. The corresponding times for a 100-percent power change ranged from 9 to 6.1 seconds. The maximum rate of collective reduction was limited by the minimum normal acceleration flight envelope limits. The maximum rotor overspeed was to 104 percent for all conditions. The excessive engine/rotor speed transients which occurred following large-magnitude collective applications to the opposite extreme of the power demand schedule are a deficiency which should be corrected prior to production.

97. Rotor speed control during the majority of mission maneuvers was excellent. No pilot compensation was required to manage rotor speed during the UTTAS maneuvers (para 111), vertical and lateral displacements (paras 31 and 34), during mission maneuvers (para 106), and jump takeoffs (para 85). The excellent rotor speed control during normal mission tasks is an enhancing characteristic.

98. Engine stability was good during all tests and no appreciable lags were observed when operating within the extremes of the power demand schedule. The pilot was able to maintain positive control of altitude within ±0.5 foot by use of the collective with less than ±0.5-inch control movements. There were no objectionable lags in engine response during precise hovering tasks. The YUH-60A employed a single beep switch on each collective control to vary engine and rotor speed. The system was rarely required once the operating rpm had been established prior to takeoff. When needed, minimal pilot compensation was required to vary or reset engine and rotor speed (HQRS 3).

99. Engine torque-matching characteristics were qualitatively assessed throughout the GCT. The maximum torque difference noted throughout the static and dynamic tests was 5 percent, which occurred predominantly at the lower power settings. At power settings of 80 percent and above, the torque settings were precisely matched. The effective torque-matching capability of the YT700-GE-700 engines is an enhancing feature.

100. The YT700-GE-700 engine incorporates a $T_{4.5}$ temperature limiting feature which prevents inadvertent overtemperature operation (para 7a, app D). This feature reduced pilot workload considerably with respect to power management tasks, since the pilot did not have to meticuously monitor temperature when operating at or near the engine temperature limits. The automatic $T_{4.5}$ limiting feature of the YT700-GE-700 engine is an enhancing characteristic.

101. The emergency power management characteristics were qualitatively evaluated in a variety of representative flight and power conditions. Emergency or manual control of the YT700-GE-700 engines in the YUH-60A helicopter is obtained by moving the desired engine power control lever to the ECU lockout position (full forward). After gaining manual control of the engine, the power lever is retarded to manually adjust engine power to the desired level. Once manual

control of the engine was obtained, matching of the manually-controlled engine torque to the other (governed) engine was easily accomplished and required little crew coordination.

102. The engine anti-ice and cockpit heater systems were not reevaluated during the GCT; however, the results from APE I remain valid. The power losses associated with operating both systems were approximately 28 percent of available power throughout the flight envelope. This significant power loss will degrade accomplishment of mission tasks when performing operations which require near-maximum power. The significant power loss associated with operation of the engine anti-ice and cockpit heater systems is a deficiency which should be corrected prior to production.

103. The engine manufacturer redesigned the ECU in response to engine-related problem areas (para 7, app D) identified during APE I. Although not a specific part of the UTTAS GCT, two redesigned ECU's (Group 8) were supplied to USAAEFA for installation and testing (in cooperation with the airframe manufacturer) in the helicopter prior to further modification of remaining prototype engines. Thirteen tests were conducted to compare differences between the originally-installed Group 7 ECU and the Group 8 ECU. A complete description of the tests is contained in reference 14, appendix A. Figures 100 through 102, appendix G, show representative time histories of the testing.

104. There were no significant differences between the functioning of the Groups 7 and 8 ECU's for jump takeoffs, one-engine-out governing, uncompensated response characteristics (collective fixed, load/power change required due to cyclic or pedal input), and normal engine/rotor governing during typical maneuvers. The control of rotor speed was slightly improved with the Group 8 ECU at test-day V_H and the never-exceed airspeed (V_{NE}). Rotor speed transient overshoot from rapid collective reductions from maximum power was decreased with the Group 8 ECU. The Group 7 transient rotor overspeed with a 29-percent (2.6-inch) collective reduction was 8 percent (21.6 rpm) (fig. 100, app G). With the Group 8 ECU installed, the transient rotor overspeed was 6 percent (15 rpm) (fig. 101). For both ECU's, the time to return to a stabilized rotor speed was essentially the same. The transient rotor speed droop characteristics during autorotational recovery were significantly improved with the Group 8 ECU. During a 3/4-in./sec collective rate input (fig. 102), rotor speed drooped from 112 percent (294 rpm) to 97 percent (255 rpm) and then stabilized at 98 percent. For a comparable collective rate input, the Group 7 ECU transient rotor speed droop was to approximately 88 percent (231 rpm). To achieve the improved transient rotor speed droop characteristics, the no-load engine gas producer (N_G) and power turbine (N_P) speeds with the Group 8 ECU were increased. The N_G was increased from 71 to 77 percent and N_P from 102 percent (normal operating range) to 110 percent (precautionary range) for the Groups 7 and 8 ECU's, respectively. The method of improving the transient droop characteristics may not be satisfactory, since the effects on the engine of increased N_P are unknown. The $T_{4.5}$ limiting characteristics were significantly degraded with installation of the Group 8 ECU. When operating at the power-available limits,

the normal $T_{4.5}$ limiting occurs at 845°C, ±5 degrees. With the Group 7 ECU, transient overshoot of $T_{4.5}$ occurred before the limiting feature functioned. This characteristic allowed rapid power application to limit and still provided overtemperature protection. The $T_{4.5}$ limiting with the Group 8 ECU began to function at approximately 805°C $T_{4.5}$, causing premature rotor speed droop and the limit temperature was not reached for 15 to 20 seconds. The premature limiting of $T_{4.5}$ temperature in effect reduced the power available to the pilot by eliminating the capability to rapidly apply power to its available limit. Because of the unknown effects on the engine with the no-load N_p at 110 percent, and the unsatisfactory $T_{4.5}$ limiting characteristics, the Group 8 ECU's were removed and the Group 7 ECU's reinstalled.

Normal Mission Maneuvers

105. A limited quantitative and qualitative evaluation of mission maneuvering characteristics was accomplished at the conditions shown in table 2. Aircraft agility and maneuverability were assessed during accelerations and decelerations from and to a hover, lateral accelerations from a hover, and during pop-ups and bob-ups with the FPS OFF.

106. Forward flight accelerations were conducted from a stabilized OGE hover by simultaneously increasing power to IRP and varying pitch attitude as required to maintain constant altitude during acceleration to V_H. Acceleration to V_H required approximately 45 seconds. During acceleration from a hover, achieving and maintaining IRP was easily accomplished with minimal initial rotor speed droop (approximately 2 percent transient). Deceleration was initiated from V_H by simultaneously reducing collective at a rate necessary to control rotor speed and varying pitch attitude to maintain constant altitude. During both acceleration and deceleration altitude was easily held constant (HQRS 3). The required nose-high attitude (20 degrees) resulted in a restricted forward field of view caused by the instrument panel glare shield, which required the pilot to move his head and body to the right to see through the chin bubble. The restricted forward field of view during high nose-up attitudes is a shortcoming. The acceleration and deceleration characteristics of the YUH-60A met the commitments of the PIDS and are satisfactory.

107. Left and right lateral accelerations were accomplished from a stabilized 50-foot hover. The helicopter was rapidly rolled in the desired direction of flight, collective was increased to obtain IRP, and the aircraft accelerated while constant altitude was maintained. The helicopter rapidly accelerated in either direction with minimal pilot effort required to execute the maneuver (HQRS 3). Lateral acceleration characteristics are satisfactory.

108. The pop-up maneuver was initiated from level flight at 50 feet AGL at an airspeed of 50 KIAS. The maneuver was executed by bringing the helicopter to a climbing flare with termination in a 150-foot OGE hover and then performing a rapid 180-degree change of direction and a level transition to forward flight.

The nose-high pitch attitude required during the deceleration restricted the pilot's forward field of view and required him to rely on peripheral vision to maintain position (para 106). The pop-up maneuver was accomplished with minimal pilot effort (HQRS 3).

109. Bob-up maneuvers to 100 feet AGL were initiated from a stabilized 20-foot IGE hover, using various rates and magnitudes of collective input. Desired position over the ground was maintained by manipulation of directional and cyclic controls. The bob-up maneuver was satisfactorily accomplished with minimal pilot effort (HQRS 3).

UTTAS Maneuvers

110. The UTTAS maneuvers were evaluated at entry airspeeds of 149 to 153 KTAS at density altitudes of 5900 to 7120 feet. The objective of this evaluation was to assess the helicopter's capability to achieve the following aim conditions: (1) from an entry airspeed of 150 KTAS, attain a sustained load factor of 1.75 within 1 second in a symmetrical pull-up; maintain a minimum load factor of 1.75 for 3 seconds with an airspeed loss at the end of 3 seconds not to exceed 30 KTAS; (2) from an entry airspeed of 150 KTAS, attain a sustained load factor of 0.0 within 1 second in a symmetrical pushover and maintain the 0.0 minimum load factor for 2 seconds; and (3) evaluate the pushover from a level flight entry as well as a continuation of the pull-up maneuver. Figures 103 through 105, appendix G, are representative time histories of the UTTAS maneuvers. Table 7 presents a summary of the significant data quantified during the tests.

111. The pull-up maneuver was accomplished by initially attaining a target load factor greater than the required load factor of 1.75 (fig. 103, app G). A target load factor of 2.0 resulted in achieving the specification conditions of the maneuver. Rapid aft cyclic control was required to establish the load factor quickly. Once established at the load factor, positive control force cues enabled the required condition to be easily maintained for the required 3.0 seconds (HQRS 3). Airspeed reduction was well within the 30-knot maximum loss criteria. Flight control movements to maintain roll and yaw attitude were minimal (HQRS 3). Rotor speed slightly increased (+1.5 percent from trim) during the maneuver and required no corrective action (HQRS 2). The pushover maneuver required load factor was easily reached by attaining a lesser target load factor (fig. 104). The maneuver was initiated by a rapid downward collective movement. Pitch attitude change and airspeed gain were minimized by slight initial aft cyclic control movement. Once reached, the 0.0 required load factor was easily maintained for the required 2 seconds. Yaw attitude change during the pushover maneuver was 7 degrees right with directional controls fixed. Roll attitude was easily maintained within limits by use of right lateral control application (HQRS 3). Rotor speed increased 3 percent from trim momentarily and required no corrective action (HQRS 2). The commitments of the PIDS were met for the pull-up and pushover UTTAS maneuvers.

Table 7. UTTAS Maneuver Data Summary.[1]

Maneuver	Entry True Airspeed (kt)	Time[2] (sec)	Normal Acceleration[3] (g)	True Airspeed Change[4] (kt)	Maximum Attitude During Maneuver[5] (deg)			Maximum Control Displacement During Maneuver (in.)		
					Pitch	Roll	Yaw	Longitudinal	Lateral	Directional
Pull-up	149	1.0	1.85	-9	43 nose-up	5 right	4 right	2.6 aft	0.8 left	0.4 left
Pushover	149	0.8	-0.1	-1	11 nose-down	6 left	7 right	0.7 aft	1.1 right	Zero
Pull-up and pushover[6]	153	0.9	1.80	-8	39 nose-up	7 right	3 right	2.6 aft	0.6 left	0.4 left
	145	3.2	Zero	[7]-26	10 nose-down	2 left	12 right	1.1 aft	1.6 right	Zero

[1]Average test conditions: Gross weight 16,817 pounds; longitudinal cg 358.8 aft; lateral cg 0.5 right; density altitude 5900 to 7120 feet.
[2]Time from maneuver initiation to achieving the sustained load factor.
[3]Normal acceleration sustained for 3 seconds (pull-up) and 2 seconds (pushover).
[4]Airspeed gain/loss at end of sustained normal acceleration period.
[5]Attitude at end of 3-second (pull-up) or 2-second (pushover) sustained normal acceleration period.
[6]Pull-up maneuver followed by pushover.
[7]Airspeed continued to decrease during the pushover portion of the maneuver (24-knot airspeed loss from initiation of the pushover).

112. The pull-up, pushover combination maneuver was accomplished using the same techniques as the individually performed maneuvers, and resulted in essentially the same basic characteristics (fig. 105, app G). Entry airspeed for the pushover was lower because of airspeed loss during the pull-up. The time to reach a 0.0 load factor was longer because of starting the pushover from a load factor of 1.75. To reach the 0.0 load factor the collective control was fully lowered and forward cyclic was required (HQRS 3). The collective-to-roll coupling effects were greater, as evidenced by the amount of right lateral control used to control roll attitude (HQRS 3). Rotor speed variation from trim during the pull-up was +1 percent and during the pushover was +2 and -3 percent and required no pilot corrective action (HQRS 2). Within the scope of this test, the UTTAS maneuvers and the pull-up and pushover combination maneuver can be satisfactorily accomplished.

External Load Evaluation

113. The handling qualities of the YUH-60A helicopter were qualitatively evaluated with an external sling load at the conditions shown in table 2. Low-speed (para 92) and forward flight tests to an airspeed of 65 KCAS were performed with an empty CONEX (1580 pounds). Additionally, forward flight tests to an airspeed of 129 KCAS and a load release from hover were conducted with a high-density load (auxiliary power unit (APU) container ballasted to 3090 pounds). The forward flight tests were conducted AFCS ON and OFF and simulated gust response characteristics were evaluated in the longitudinal and lateral axes.

114. The CONEX sling load was extremely unstable in flight. The load oscillated laterally and twisted and untwisted at all forward flight airspeeds. These load motions were felt in the cockpit in the roll axis of the helicopter, but no unusual flight control applications were required to control the aircraft (HQRS 3). The high-density load was significantly more stable in flight than the CONEX, with little or no oscillation. Flight control activity in all flight conditions was normal for the aircraft at this gross weight condition (HQRS 3). The control inputs to simulate gust response excited both loads, with excitation generally damping out within 3 cycles. No difference in load excitation was noted with AFCS ON or OFF for the simulated gust responses. The helicopter was not noticeably affected by the load excitation and gust response characteristics were essentially the same as those with no load.

115. Load acquisition required ground guidance, since the hook could not be seen from the cargo compartment. The field of view forward and through the chin bubble was inadequate (para 158) and degraded rapid and precise load acquisition. The hover sling load release (high-density) was conducted with the controls held fixed in excess of 5 seconds following release. No abrupt or extreme aircraft attitude changes were observed. Following release, yaw rate reached a maximum of 4 deg/sec after 7 seconds and normal acceleration increased 0.2g within 0.5 second. One unintentional load release occurred during the test. The CONEX was released in level flight at 65 KCAS with AFCS OFF. No abrupt or extreme aircraft attitude changes were observed following the release. The cause of the release was an unintentional actuation by the pilot of the cyclic grip normal

load release switch. The design of the cargo hook release system is such that arming of the normal and emergency release systems is accomplished by one switch (cargo release SAFE-ARM switch). The arming of the normal load release system when arming the emergency release system is a shortcoming. The normal acceleration following load release in hover failed to meet the commitments of paragraph 50.3.1.8.5 of the PIDS by increasing to 1.2g. This is greater than 10 percent of the design gross weight positive limit load factor (1.8) by 11 percent. Within the scope of the test, the external sling load handling qualities are satisfactory.

Instrument/Night Flight Characteristics

116. Instrument and night flight characteristics were qualitatively evaluated at the conditions shown in table 2. Instrument takeoffs, basic enroute instrument tasks, and instrument landing system (ILS) approaches were conducted with the command instrument system (flight director) ON and OFF. A limited instrument evaluation was conducted with AFCS OFF. The night evaluation encompassed representative tasks throughout the flight envelope, including hover and low-speed flight. Aircraft external and internal lighting and navigational avionics were qualitatively evaluated throughout the tests.

117. Instrument flight tasks performed with the flight director OFF required normal pilot instrument cross-check techniques to monitor five flight instruments (airspeed indicator, altimeter, horizontal situation indicator (HSI), vertical situation indicator (VSI), and vertical speed indicator). As flight task complexity increased, pilot workload increased, with a corresponding decrease in flight path accuracy. Tasks requiring a change in flight condition, such as transition from level flight to climb or descent with turns, resulted in degraded accuracy in holding constant other flight parameters not required to be changed (HQRS 3). With the flight director ON, pilot cross-check was reduced to three instruments (HSI, VSI, and altimeter) and for stabilized flight conditions (level flight) pilot cross-check was reduced to one instrument (VSI). This was possible because of the centralized commanded functions of airspeed, heading, and collective control (vertical speed/altitude) provided in the VSI. The centralized display of commanded airspeed, heading, and collective control significantly reduced pilot workload and increased flight path accuracy compared to instrument flight with the flight director OFF (HQRS 2). The improved instrument flight characteristics provided by the commanded flight parameters of the flight director are an enhancing characteristic.

118. During the instrument evaluation four unsatisfactory characteristics associated with the flight director were noted. These characteristics were related to commanded and advisory information displayed to the pilot which could have been corrected as a maintenance function had the proper technical expertise been available. The first unsatisfactory characteristic was observed during ILS approaches when the flight director automatically switched to the deceleration mode. The deceleration mode provided the pilot a commanded slowing of airspeed to 50 KIAS while still furnishing proper information to maintain the glide path and localizer of the ILS. The commanded slowing of airspeed occurred at 750 feet AGL and

resulted in significantly increasing the total approach time and the time to reach approach altitude minimums. The premature commanded slowing of the aircraft at 750 feet AGL in the deceleration mode of the flight director is a shortcoming.

119. The go-around mode of the flight director provided the pilot commanded information to initiate a missed approach. This information resulted in heading and airspeed and collective commands to achieve 80 KIAS and a 500-ft/min rate of climb. The go-around mode was selected by the pilot through a cyclic grip button. When activated there was no advisory information provided to the crew that the go-around mode had been selected. This characteristic is unsatisfactory. Additionally, the improperly set initial commands of airspeed and collective (go-around mode selected at 50 KIAS) caused the pilot to overcontrol the cyclic control and resulted in a momentary airspeed gain to 90 KIAS prior to reaching a stabilized rate of climb. The lack of advisory information and the improperly set airspeed and collective commands of the go-around mode of the flight director are shortcomings.

120. With the flight director in the vertical speed mode, the pilot can select a vertical speed on the instantaneous vertical speed indicator which will be displayed as commanded collective on the VSI. Once a vertical speed was selected, the improperly set collective symbol required the pilot to satisfy the command several times before a stabilized vertical speed was reached. This characteristic resulted in excessive time to reach a desired vertical speed. The excessive time to reach a selected vertical speed in the vertical speed mode of the flight director is a shortcoming.

121. There is only one VHF navigation receiver (VOR/ILS) installed in the YUH-60A helicopter. When two different VHF navigation stations were required to fix airway intersections and approach fixes, crew workload was significantly increased because of the requirement to cross-tune the one installed VHF navigation radio. Additionally, when cross-tuning the VHF navigation radio, commanded navigation provided to the flight director was interrupted for the time the radio was off the primary navigation frequency. Two VHF navigation radios would significantly reduce crew workload during instrument flight and allow more efficient and complete utilization of the flight director. The lack of two VOR/ILS navigation radios is a shortcoming.

122. The HSI has two bearing pointers, but only the No. 2 pointer was used to display either ADF or VOR navigation information. The No. 1 bearing pointer could not be used with any installed navigation radio. The inability to use the No. 1 bearing pointer of the HSI is a shortcoming.

123. Simulated instrument flight was conducted with AFCS OFF. Airspeeds greater than 100 KIAS required the pilot to intensely monitor the VSI to maintain aircraft attitude (HQRS 6). At 75 to 80 KIAS pilot workload was reduced but the pilot was still required to closely monitor the flight instruments to prevent

aircraft attitude deviations (HQRS 4). All normal instrument tasks could be performed at the 75- to 80-knot airspeeds. The simulated instrument flight characteristics with AFCS OFF were not identified as a shortcoming because of the degraded mode and the CAUTION of paragraph 70 applies.

124. The internal cockpit lighting and lighting controls were evaluated on the ground and in flight. All lighting and lighting controls were satisfactory, with the exception of the secondary lights/cockpit floodlight controls. The lights (red and white) were mounted between the pilot and copilot and were controlled by two switches and two rotary dimmer controls. One switch functioned as a bright/dim control and the other switch selected red or white light or remote. When the red/white/remote switch was in the REMOTE position, each rotary dimmer control could be used to turn on and control the intensity of the respective red or white light. The secondary lights/cockpit floodlight controls caused confusion because of their dissimilar method of operation. The functions of these multiple controls should be consolidated and simplified. The multiple secondary lights/cockpit floodlight controls and their dissimilar method of operation are a shortcoming.

125. The external aircraft lighting systems were evaluated by the test aircraft crew and by the crew of the chase aircraft. The configuration and variable-intensity control of the external formation lights permitted quick recognition of relative bearing and proximity by the chase aircraft for most rear-intercept azimuths. Additionally, the formation lights made the task of formation flying by the chase aircraft easier. The configuration and readily identifiable aircraft formation lights are an enhancing characteristic.

<u>Automatic Flight Control System Characteristics</u>

126. The AFCS was evaluated throughout the test program. The AFCS was evaluated for suitability in IMC, fly-through capability during mssion tasks, system reliability, pilot interface, and system failure characteristics. Quantitative and qualitative evaluations were conducted with various AFCS modes (FPS, FAS, SAS) selected or failed.

127. Aircraft attitude changes with full AFCS ON were made by moving a flight control out of detent and either retrimming to a new attitude/heading reference or returning the control to the trim detent at the completion of the maneuver. When maneuvering the aircraft through the AFCS attitude hold, the magnitude of longitudinal and lateral control forces was objectionable (paras 45 and 46). The most objectionable characteristic was the interaction of the roll trim integrator in the AFCS. The lateral control force buildup as a result of the roll trim integrator was proportional with time for bank angles of less than 5 degrees and could result in a maximum of 8.5 pounds of control force. Flights during simulated IMC, where small heading corrections were made, required the pilot to constantly retrim because of the build-up of lateral control forces. Even when attempting to maintain level flight, inadvertent lateral control inputs (particularly where a slight pressure was applied over a period of time) resulted in lateral control force buildups that required retrimming by use of the force trim release. This

retrimming also resynchronized the longitudinal and airspeed-hold logic and resulted in significant increases in pilot workload when conducting IMC tasks. The cyclic beep trim was not utilized to trim out the roll integrator forces, as this would establish a trim reference error over a period of time. The roll trim integrator also degraded the pilot's capabilities during hovering and NOE tasks. Hovering in crosswinds, making attitude corrections, or when traversing laterally resulted in lateral control force buildups from the action of the roll trim integrator. The pilot had to constantly retrim the aircraft to relieve the lateral forces. This would cause loss of the trimmed attitude reference and is unsatisfactory for night NOE tasks as well as for IMC tasks. If the pilot were to become disoriented while these lateral control forces were increasing, he might think that he had displaced the aircraft from trim. Allowing the lateral control to return to a zero force in an attempted recovery could result in unusual attitudes which might preclude recovery. The highly undesirable and potentially unsafe characteristics of the roll trim integrator are a deficiency which should be corrected prior to further government tests.

128. The yaw trim integrator was designed to assist the pilot in maintaining balanced flight and to provide a coordinated turn feature for airspeeds above 60 KIAS (para 22, app C). If the pilot miscoordinated the ball, a force proportional to sideslip would be felt in the directional controls. Unless the pilot was extraordinarily careful in allowing the pedal to travel with the force (easing foot pressure), the force would build up to such magnitude that the helicopter could not be retrimmed directionally without disengaging and reengaging the FPS on the AFCS control panel. This problem was further complicated by the unreliability of the accelerometers to maintain a null position in the system. Migration of this null resulted in constant force buildups in the directional controls, even when trimmed in balanced flight. The pilot lost confidence in the FPS mode of the AFCS because of the increased workload required to conduct precise forward flight tasks associated with IMC flight The undesirable characteristics of the yaw trim integrator are a shortcoming.

129. The airspeed hold system is functional whenever airspeed is 60 KIAS or greater and FPS is engaged. Activation of either the beep or instantaneous trim synchronizes the airspeed hold and an 18-second delay is incurred prior to regaining this function. This excessive delay contributed to the moderate pilot compensation required to trim to desired airspeeds. This system degraded the pilot's capabilities in trimming to precise airspeeds, particularly in simulated IMC. Once engaged, the airspeed hold system reduced pilot workload under most conditions; however, the effort required to establish an airspeed is excessive. The excessive delay in engagement of the airspeed hold system following trim actuation is a shortcoming.

130. The AFCS could be totally or selectively engaged/disengaged throughout the flight envelope with no noticeable transients in any axis. AFCS monitor information was provided on the master caution panel (SAS, FAS/TRM, FPS, and PITCH BIAS fail lights) and through ON/OFF switches on the AFCS control panel. The information relating AFCS status was confusing with regard to FAS/TRM or FPS status. Engaging the FAS switch would remove the FAS and FPS lights from the master caution panel, while the FPS ON/OFF switch would still be in the

disengaged mode. Conversely, when disengaging the FPS selectively, there was no CAUTION light indicating a disengaged condition, but the switch would show FPS OFF. This condition is inconsistent with the intent of caution panel information. The exact status of the AFCS and not an interpretation of lights or switch positions is required. The potential of having the FPS disengaged without a corresponding CAUTION light is a shortcoming.

Aircraft Systems Failures

131. Aircraft systems failures were investigated at the conditions outlined in table 2. Simulated failures of an engine, the AFCS, and the hydraulic and trim systems were conducted.

Simulated Engine Failure Characteristics:

132. Sudden engine failures were induced from dual-engine to single-engine flight and from stabilized single-engine flight to autorotation by rapidly retarding the desired power control lever to the idle stop. The response of the helicopter to sudden single-engine failures was evaluated during hover, forward flight, and in dives at airspeeds up to 167 KCAS. Flight controls were held fixed for 2 seconds following the power loss, or until activation of the low rotor speed warning light plus 1 second, or when an aircraft angular rate of 5 deg/sec within 0.3 second was reached. Figure 106, appendix G, is a representative time history of a simulated engine failure. Table 8 summarizes the delay times for the dual-engine to single-engine failures.

133. Aircraft response to a sudden single-engine failure from stabilized dual-engine flight at all conditions was mild. Primary reaction of the helicopter was a left yaw which was a maximum of 6 degrees at 167 KCAS. Slight left roll and nose-up pitch accompanied the sudden engine failure but was not discernible to the pilot as a cue. The primary pilot cue of a failure at level flight airspeeds from 76 to 118 KCAS was the change in color from green to red of the engine power turbine speed indication on the pilot display unit (PDU) fiberoptic instrument. As engine power turbine speed decreases through 95 percent, the fiberoptic display goes from green to red. Although the minimum rotor speed achieved for the engine failure at 76 KCAS was 96 percent (252 rpm), the PDU momentarily displayed a low (red) power turbine speed warning. At airspeeds above 118 KCAS, primary pilot cues of engine failure were left yaw and the PDU display. These cues would enable the pilot to identify the failure and take corrective action in a timely manner at all flight conditions and times, with the exception of low-powered descents. The helicopter response to a single-engine failure was controllable, requiring no rapid or objectionable cyclic or directional control manipulations. For level flight airspeeds at 76, 86, and 108 KCAS, the rotor speed droop was within the power-on limits of 95 percent rotor speed following acceleration of the unfailed engine and required no collective control application. The remaining test conditions required 5 to 20 percent of collective reduction (depending on airspeed) after rotor speed had decreased past the warning limit value (95 percent rotor speed). The mild aircraft response characteristics following

Table 8. Single-Engine Failure Characteristics from Dual-Engine Flight.

Flight Condition	Calibrated Entry Airspeed (kt)	Collective Control Delay Time[1] (sec)	Minimum Rotor Speed (percent/rpm)
Level flight	76	Note[2]	96/252
Level flight	86	Note[2]	95/250
Level flight	108	Note[2]	95/250
Level flight	118	4.6	94/247
Level flight	129	1.2	94/247
Level flight	146	1.5	89/234
Dive (IRP)	167	1.7	89/234

[1]Elapsed time from throttle chop to initiation of collective reduction.
[2]Collective reduction not required.

single-engine failure from dual-engine flight were desirable from a controllability and handling qualities standpoint, but could contribute to late recognition of an engine failure in some flight regimes. During a single-engine failure in a low-power descent, the aircraft attitude response cues were very mild, the change in engine noise was not always perceptible, and there was virtually no rotor speed decay. The YUH-60A incorporates an engine-out warning light system (master caution panel) which is activated by low gas generator speed (55 percent N_G) and would possibly be the only cue of engine failure during a low-powered descent. This warning system does not incorporate an aural warning signal. During mission tasks where the pilot's attention is directed outside the cockpit (such as a low-powered descent into a landing zone), the visual engine-out warning could go unheeded and the pilot would not have adequate cues generated from other sources. The lack of an aural engine-out warning device is a shortcoming.

134. Simulated sudden single-engine failures from stabilized single-engine flight and subsequent entry into autorotation were conducted. Pilot workload in maintaining aircraft control and ease of establishing stabilized autorotational flight were evaluated, as well as steady-state autorotational handling qualities. Figures 107 through 109, appendix G, are representative time histories of engine failure, autorotational entry, and autorotational descent. Table 9 summarizes delay times and pilot action for engine failure from single-engine flight.

Table 9. Engine Failure Characteristics From Single-Engine Flight.

Flight Condition	Calibrated Entry Airspeed (kt)	Collective Control Delay Time[1] (sec)	Minimum Rotor Speed (percent/rpm)
Level flight	12	2.2	91/239
Level flight	53	1.5	86/226
Level flight	77	1.6	86/226
Level flight	96	1.2	84/221

[1]Elapsed time from throttle chop to initiation of collective reduction.

135. Similar aircraft characteristics were noted for the single-engine failures from single-engine flight as were noted for the single-engine failures from dual-engine flight (para 133), although left yaw was more pronounced. The critical control was the collective for all test conditions and the minimum available time for pilot reaction (1.2 seconds) occurred at 96 KCAS (maximum airspeed tested). At this airspeed, the entry into autorotational flight required a moderate reduction (1.64 in./sec) of the collective to control rotor speed. Although initiation of collective control was not delayed for 2 seconds following the simulated engine failure, nor had 1 second elapsed following activation of the low engine speed warnings, the time required to regain the minimum allowed power-off rotor speed (90 percent rotor speed) was not considered excessive. At 96 KCAS the maximum time required to regain minimum steady-state autorotational rotor speed from time of collective reduction was 5.9 seconds, with an altitude loss of 40 feet. These times could have been shortened if the collective had been lowered at a faster rate. There were no unusual aircraft response characteristics following the engine failure and pilot effort was minimal to enter and establish autorotational flight for all airspeeds at 70 KCAS and above (HQRS 3). The engine failure at 53 KCAS required moderate pilot compensation to enter and establish autorotational flight (HQRS 4) because of the increased pilot effort required to establish V_{min} R/D from the slow entry conditions. For all test conditions the maximum attitude change was 10 degrees in yaw and there was no altitude loss in the first 2 seconds following the engine failures.

136. Maintaining aircraft control throughout the autorotational descent required minimal pilot effort (HQRS 3). The pilot was able to establish and maintain a desired airspeed and rotor speed with minimum control requirements. Coordinated turns of up to 30 degrees of bank in either direction required moderate pilot compensation (HQRS 4) because of the increased effort to maintain rotor speed. During one descent, the aircraft developed a tail shake similar to that reported during APE I but of less magnitude. The tail shake was not objectionable. The commitments of paragraph 50.3.6.1.1 of the PIDS were not met, in that collective control motion could not be delayed 1 second after activation of the low engine speed warning system. All other applicable specification commitments were met. Within the scope of this evaluation, the engine failure and autorotational characteristics from stabilized single-engine flight are satisfactory. Because of the rapid rotor speed decay, caution should be exercised to ensure that collective control is rapidly and fully reduced when performing autorotational entries from single-engine flight or following dual-engine failures. The following CAUTION should be included in chapter 8 of the operator's manual:

CAUTION

> In the event of an engine failure during single-engine flight or dual-engine failure, rotor speed will rapidly decay to below the minimum power-off limit (90 percent rotor speed). Rapid reduction of collective control to full-down will be required to arrest rotor speed decay and regain suitable autorotational rotor speed.

Automatic Flight Control System Failures:

137. Automatic flight control system failures were simulated in a hover and in forward flight at the conditions listed in table 2. Engagement and disengagement and single-axis hardover conditions were conducted for the various AFCS modes. The FPS, FAS/TRM, and SAS were selectively engaged and disengaged at all test conditions. Single-axis hardovers were simulated utilizing a hardover box that had the capability of inducing the following malfunctions: pitch bias and FAS actuator hardover, and runaway trim for the longitudinal, lateral, and directional trim systems. Representative time histories for the AFCS failures are presented in figures 110 through 113 of appendix G.

138. Abrupt disengagement of either the FPS, FAS/TRM, or SAS did not result in any immediate aircraft response at any of the test conditions. A complete failure of the SAS did cause aircraft divergence from trim due to the loss of rate damping provided by the system, but was easily controlled by the pilot. When engaging either mode of the AFCS, there were no switching transients that required more than 0.25 inch of control input to maintain attitude.

139. Single-axis hardovers were conducted in a hover, in right sideward and rearward flight, and in forward flight at airspeeds from 77 to 168 KCAS. The pitch bias and FAS hardovers were the mildest, since the FAS computer would shut down within 1 second after initiation of the failure. The attitude hold feature of the AFCS (FPS mode) worked to regain the pitch trim reference following pitch bias hardovers and pilot compensation was not a factor in recovery (HQRS 2). The FAS hardovers caused greater pitch excursions (12 degrees nose-up attitude change) partly because of the loss of the attitude hold feature. If the pilot reentered the control loop prior to 3 seconds elapsed time, the attitude deviations were not excessive. Minimal pilot compensation was required to control the hardover and regain trim conditions (HQRS 3). The pitch rates generated by pitch bias and FAS hardovers were approximately 4 deg/sec for all conditions. The pilot has monitor information on the master caution panel for either of these failures.

140. The control trim hardovers (runaways) were more severe than pitch bias and FAS failures. The longitudinal trim runaways were not excessive, however, in that the pitch rates developed never exceeded 8 deg/sec. This gave the pilot adequate cues of the failure and allowed rapid corrective action to be taken. The longitudinal force buildup for 100 percent trim runaway was a maximum of 12 pounds fore and aft and was easily overcome by the pilot. The lateral trim runaways generated high roll rates: from 12 deg/sec in a hover to 37 deg/sec at 141 KCAS at 3 seconds following initiation of the failure. The maximum bank angles within the 3-second period ranged from 46 to 56 degrees at 141 KCAS and required 8 to 10 pounds of lateral force to regain level flight. Moderate pilot compensation was required to recover the aircraft during lateral trim runaways (HQRS 4) because of the rapidly developing roll rates and resultant roll attitudes. Roll trim runaways were also evaluated in turning flight at bank angles up to 30 degrees at 119 KCAS with failures into the bank. The roll rates at the end of 3 seconds were 18 to 19 deg/sec and required the pilot to enter the control

loop no later than 3 seconds following failure, because of the large roll attitudes developed (56 degrees maximum). The directional trim runaways generated yaw rates from 8 deg/sec at 141 KCAS to a maximum of 16 deg/sec in a hover. The pilot was able to control directional trim runaways with minimal pilot compensation (HQRS 3). The maximum directional control forces resulting from 100 percent trim runaway were 58 to 75 pounds. Although the directional control forces were high, landings were conducted with failures of the trim system requiring minimal pilot compensation (HQRS 3). The roll and yaw rates developed from trim runaways exceed the 10-deg/sec commitment after 3 seconds of paragraph 50.3.2.7.1 of the PIDS by the amounts shown in table 10. The high roll rate developed following lateral control trim system failure is a shortcoming. All other AFCS failure characteristics were satisfactory within the scope of this evaluation.

Table 10. Single-Axis Hardover Test Results.[1]

Flight Condition (KCAS)	Pitch Bias Hardover (deg/sec)	Pitch FAS Hardover (deg/sec)	Longitudinal Trim Hardover (deg/sec)	Lateral Trim Hardover (deg/sec)	Directional Trim Hardover (deg/sec)
Zero	4	4	8	12	16
77	4	4	8	[2]20	14
119	4	4	8	26	8
141	4	4	6	37	8
168	-	-	4	—	—

[1]Data presented represent controls-free pitch, roll, and yaw rates developed at 3 seconds following the simulated hardover.
[2]Roll rate of 20 deg/sec developed within 2 seconds.

Hydraulic Systems Failures:

141. Hydraulic systems failures were investigated at the conditions listed in table 2. The failures were simulated by turning off various combinations of the first and second stage servos, hydraulic backup pump, SAS control circuit breaker, FAS/TRM control, FPS control, and yaw and collective boost control.

142. At all airspeeds tested, there were no objectionable aircraft responses when single systems were failed. Reactivation of the failed system likewise produced no adverse aircraft response. Deactivation of the first stage servo at V_H resulted in the nose of the aircraft yawing right approximately 3 degrees. This yaw occurred

because of the time required for the backup pump to activate and power the second stage tail rotor servo. With single system hydraulic failures, continued safe flight, approach, and landing was easily accomplished (HQRS 3). The single system hydraulic failure characteristics of the YUH-60A are satisfactory and meet the commitments of the PIDS.

143. Dual system (first stage and backup) failures were simulated by pulling the backup pump circuit breaker and turning the first stage servo control off. At 75 and 119 KCAS there was only a slight right yaw; however, at 145 KCAS the aircraft yawed right approximately 10 degrees, attaining 15 degrees left sideslip, and stabilized. Attempts to return to balanced flight resulted in application of increased pedal force (approximately 30 pounds), with only 10 percent directional control remaining and approximately 5 degrees decrease in the sideslip angle. There are no PIDS commitments for dual hydraulic system failures; however, under certain conditions loss of the second stage hydraulics and loss of the backup system would be possible. If a rupture should occur in the system downstream of the No. 2 transfer module, the No. 2 system would be depleted of hydraulic fluid. Under these conditions the backup pump would activate, and the backup system would be depleted. The following systems would be lost if the No. 2 and backup systems were depleted: second stage primary servos; SAS, FAS, pitch FPS functions; collective and yaw boost; and APU accumulator charge capability (except manual). The hydraulic system should be redesigned to preclude depletion of the backup hydraulic system in this manner.

Single-Engine Landings

144. Single-engine landings were conducted to a level paved surface at the conditions listed in table 2. The single-engine condition was simulated by placing one engine in the idle position and conducting successively slower roll-on type landings to determine minimum touchdown speed and stopping distance. Aircraft limits established by USAAEFA to assure safety during this test, in addition to those presented in reference 6, appendix A, were: a minimum rotor speed of 90 percent (237 rpm) and maximum gear loads of 10,000 and 6500 pounds for the main and tail gear, respectively. The test results are presented in table 11. Representative time histories are presented in figures 114 and 115 of appendix G.

145. The single-engine technique utilized was to flare the aircraft to arrive at a ground speed of approximately 25 knots while maintaining enough height above the ground to preclude tail wheel touchdown. Once the tail wheel contacted the ground and the nose started to fall through, the pilot maintained a flare attitude while applying collective to cushion the main gear touchdown. This method resulted in the differences in touchdown speeds of the main and tail gear shown in table 9. These touchdown speeds represent a maximum capability within the constraints of the main gear limits imposed on this test. The 10,000-pound limits represent 62.5 percent of the recommended main gear limits (16,000 pounds), and as can be seen in table 10, these limits were reached or exceeded on the right gear for

Table 11. Single-Engine Landing Characteristics.

Touchdown True Airspeed for Tail/Main Landing Gear (kt)	Landing Distance for Tail/Main Landing Gear[1] (ft)	Gear Loads (lb)			Minimum Rotor Speed[2] (percent/rpm)
		Right	Left	Tail	
33/15	182/162	10,800	7450	2400	---
25/15	157/109	9550	7700	2800	92.5/243
23/12	129/98	10,400	8000	2400	93.5/246
23/12	123/82	8950	7300	2850	91.0/239
24/14	101/86	10,500	9750	2000	88.0/231

[1]Distances are from tail wheel touchdown and main gear touchdown to main gear stop.
[2]Rotor speed at point of main gear touchdown.

the majority of touchdowns. The minimum touchdown speed capability of the aircraft during single-engine landings can be expected to be reduced from the test results with greater main landing gear load tolerances.

146. The high nose attitude (16 to 19 degrees) required in the flare decreased the field of view to a point where the intended touchdown point was masked from the pilot throughout the last 25 to 35 yards of the approach. This decreased field of view required moderate pilot compensation to perform minimum ground speed single-engine landings (HQRS 4). The single-engine landings were conducted on a level paved surface at the mid-altitude test site, although landings could have as easily been conducted on smooth level unprepared surfaces. The single-engine landing commitments of the PIDS were met and are satisfactory.

STRUCTURAL DYNAMICS

Vibration Characteristics

147. Vibration characteristics were qualitatively evaluated throughout the test and quantitatively evaluated during the level flight performance, low-speed flight, and maneuvering stability tests and were recorded at the conditions listed in table 2. Vibration sensors were installed at the following fuselage stations: pilot and copilot seat pans, right-hand shroud of the cockpit instrument panel, and cabin floor at the aircraft cg. Additional sensors were located as indicated in appendix E. Vibration data test results are presented in figures 116 through 138, appendix G.

148. Representative level flight and IRP dive vibration data are presented in figures 116 through 119, appendix G. In level flight, the vibration characteristics at mission and alternate gross weight were similar and qualitatively assessed as very low. As shown in figure 116, the pilot seat vertical and longitudinal accelerations at the 4-per-rotor-revolution (4/rev) frequency (17.2 Hz) were below 0.05g to V_{cruise} (approximately 138 KCAS). This specification commitment of the PIDS was not met at the copilot seat or instrument panel for airspeeds at V_{cruise} and less, with maximum values of 0.18 to 0.2g being experienced. These higher measured vibration levels were not considered significant, however, as these increases were barely perceptible to the crew. The lateral 4/rev vibrations were generally higher than the longitudinal and vertical for all level flight conditions but did not alter the qualitative assessment of an overall low vibration level. Center-of-gravity vibration data for all axes (fig. 118) show accelerations of less than 0.05g at airspeeds between 90 and 140 KCAS. At airspeeds above and below these values, increases to 0.2g (vertical axis) were noted, while the majority of the vibrations in the other axes show accelerations at 0.1g or below. The 4/rev vibration levels above 138 KCAS for all locations and all axes (except copilot vertical) showed gradual acceleration increases with increases in airspeed, with a maximum value of 0.46g occurring in the lateral axis at the instrument panel for 160 KCAS (IRP dive), 17,240 pounds, and 7600 feet density altitude. The vertical 4/rev vibration was 0.42g for this condition and the information on the instrument panel was difficult to interpret. These values occurred within 6 knots of the V_{NE} of

166 KCAS for the test conditions. These moderate increases in vibration levels were not considered objectionable, as they provided a cue to the pilot of approaching V_{NE}. The vibration characteristics in forward flight failed to meet paragraph 3.2.1.1.5.1.3 of the PIDS, in that the instruments were difficult to read at 160 KCAS; and paragraph 3.2.1.1.5.1.4, in that vibration acceleration of 0.05g in all axes was not met at the fundamental main rotor frequency (4/rev, 17.2 Hz) for airspeeds up to V_{cruise} and 0.2g from V_{cruise} to V_{NE}. Within the scope of this test, the vibration levels of the YUH-60A helicopter in level flight at the mission and alternate gross weights are satisfactory.

149. The variance in the 4/rev vibration levels with rotor speed was determined in level flight at 140 KCAS. Figures 120 through 123 of appendix G represent test results of single-amplitude vibratory acceleration versus rotor speed. From 98 percent (258 rpm) to 94.5 percent (248 rpm), the trend shows a relatively constant g value of vertical acceleration. With increasing rotor speed above 258 rpm, the vibration levels significantly increased. A maximum value of 0.47g (vertical) was recorded at the copilot station at 269.7 rpm (102.5 percent). The increase in 4/rev vibration with increasing rotor speed was readily felt by the pilot but did not degrade the handling qualities of the aircraft, as these higher rotor speeds were rarely achieved throughout the majority of mission maneuvers.

150. In low-speed flight the highest 4/rev vibration levels occurred between 30 and 40 KTAS. Figures 124 through 130 of appendix G are representative test results. The highest recorded acceleration (0.32g) occurred at the cg (vertical axis) at 35 KTAS in both forward and rearward flight. Vibration levels of 0.2g and higher were observed for the vertical and lateral axes. The vibration levels were generally higher in rearward flight than for other conditions. Although generally higher than the vibration levels in forward flight, the magnitude experienced in the low-speed flight regime did not degrade flying qualities. The commitments of paragraph 3.2.1.1.5.1.4 of the PIDS were not met, in that vibration acceleration exceeded 0.05g from 30 knots rearward to V_{cruise}.

151. Figures 131 through 138, appendix G, contain vibration test results at mission and alternate gross weight during maneuvering flight. The vibration levels were markedly different between mission and alternate gross weight at these conditions. At 130 KCAS and a gross weight of 17,460 pounds, the highest 4/rev vibration level was 0.34g laterally at the pilot seat, which occurred at 1.8g normal acceleration. The vertical and longitudinal vibrations at this condition were 0.23 and 0.03g, respectively. Similar test results were noted for the remaining sensor locations. These low vibrations throughout the maneuvering envelope at mission gross weight allowed the pilot to fly the aircraft at higher normal acceleration levels without apprehension. Vibration levels in turning flight at alternate gross weight, however, were excessive (para 66). The data for all stations (figs. 135 through 138) show large increases in vertical and longitudinal vibration. The pilot felt step increases in the vibration levels at 1.3 to 1.4g normal acceleration (40 to 45 degrees of bank). Additionally, the cockpit star load indicator (pitch link load measurement) showed step increases in this flight regime. Star load indications remained at the high end of the caution area for steady turns at

30 degrees of bank and progressed to the red warning area at bank angles of 45 degrees and greater. High vibrations were experienced in steady turning flight at normal acceleration values significantly within the allowable g envelope for the conditions tested. At 45 degrees of bank, the vibrations at the pilot seat and instrument panel were so great that interpretation of the flight instruments was extremely difficult. Although the vibration data trend with load factor shows discontinuities at these higher normal accelerations, there is reason to suspect the quantitative information obtained because the pilot had difficulty in stabilizing at the higher load factors for sufficient time to obtain reliable data. The commitments of paragraph 3.2.1.1.5.1.4 of the PIDS were not met, in that the vibration levels exceeded 0.05g at the fundamental main rotor passage to V_{cruise} and exceeded the 0.2g limit from V_{cruise} to V_{NE}. The high vibration levels in turning flight at alternate gross weight are a deficiency. Further testing should be conducted in maneuvering flight at alternate gross weight to determine if potential restrictions to the flight envelope should be imposed by pitch link load structural limits.

HUMAN FACTORS

Cockpit Evaluation

152. The YUH-60A cockpit was evaluated throughout the test program. An avionics evaluation was conducted during the instrument/night flight evaluation.

153. The cockpit doors are secured in the closed position by lugs extending into a latch when the door release handle is in the locked position. If the door is closed with the door handle locked, the securing lugs knock the latch out of position, precluding subsequent securing of the doors without mechanically readjusting the latch. Under these conditions, damage to the lugs and/or the latch is probable. The inability to close the cockpit doors with the release handle in the locked position without causing damage to the door locking mechanism is a shortcoming.

154. The spring detent that holds the door in the open position was weak and did not hold the door open in light, gusty wind (approximately 10 knots). The inability of the spring detent to hold the door open in light, gusty winds is a shortcoming.

155. The pilot and copilot seats are well-padded, and comfortably conform to body contour. The seat adjustment range is 5 inches vertically and 4.37 inches fore and aft. The seats are easily adjusted; however, for tall pilots (95 percentile) the seats do not adjust sufficiently aft and down for a comfortable pilot position. In the full-up and forward position the seat position precludes attainment of full aft cyclic. The forward edge of the seat cushion is contoured to accommodate the crotch strap of the pilot restraint. With the cyclic aft in this contour, lateral movement of the cyclic is restricted. The inability to achieve full aft cyclic and

the restriction of lateral cyclic control are a deficiency that should be corrected prior to production. The lack of adequate seat adjustment range aft and down is a shortcoming.

156. The pilot/copilot restraint system is a five-strap design with the normal lap belt and shoulder harness arrangement, with the addition of a crotch strap. The design incorporates a central release mechanism which, when rotated, releases all five straps. Simulated emergency egress was rapid and unencumbered. The YUH-60A pilot/copilot restraint system is enhancing to flight crew safety.

157. The copilot collective telescopes in to permit rapid normal and emergency egress. The telescoping copilot collective control is an enhancing characteristic.

158. Rearward visibility is restricted by a side structural bulkhead just to the rear of the pilot and copilot seats. The field of view through the chin bubble is restricted by the directional pedals, cockpit floor, and the limited plexiglass area provided. This restricts the field of view during nose-high attitude maneuvers, steep approaches, and while descending into confined areas. The restricted field of view aft and through the chin bubble are shortcomings.

159. The instrument panel is a one-piece unit with dual essential flight instruments (pilot and copilot) tilted forward from the vertical at an angle that reduces parallax. The engine and transmission instruments, centrally mounted between the pilot and copilot flight instruments, consist of columns of vertical fiberoptic lighted instruments. The $T_{4.5}$, torque, and fuel vertical instruments are supplemented by digital readouts below their respective columns. The intensity of the instrument lights is controlled by an electric eye which varies lighting intensity in accordance with existing ambient light. Dual essential flight instruments and individual master warning panels, each containing #1 ENG OUT, #2 ENG OUT, FIRE, and LOW ROTOR RPM, are located directly in front of the pilot and copilot. The warning panels provided readily perceived instantaneous warning to the pilot of these parameters. The excellent design features of the instrument panel, the lighted fiberoptic instruments with digital readouts, the automatically controlled lighting intensity, the dual essential flight instruments, and the individual master warning panels are enhancing features.

160. The cockpit g meter installed for the GCT proved to be a useful instrument, since the aircraft has the capability of developing significant load factors. The g meter is particularly useful when the pilot desires to operate at or near the extremes of the normal acceleration envelope. During the evaluation of the helicopter specific engineering tests, such as the vertical displacement and UTTAS maneuvers (paras 31 and 111), required maneuvering to precise load factors. Other tests, such as the lateral displacement maneuver and maneuvering stability (paras 34 and 64) required maneuvering to the extremes of the normal acceleration envelope. These tests were all successfully and safely completed because use of the cockpit g meter gave the pilot the necessary confidence and quantitative data to precisely attain the desired load factor. During operational use, a cockpit g meter is necessary to allow the pilot to confidently maneuver the aircraft

throughout the flight envelope without exceeding it. The cockpit g meter allowed precise, safe maneuvering of the YUH-60A helicopter within the normal acceleration envelope and is an enhancing feature which should be incorporated in production aircraft.

161. The main transmission assembly is protected by five separate chip detectors (one in the main transmission, one in each accessory module, and one in each input module). If any one of the five detectors is activated, the CHIP MAIN XMSN light on the caution panel is illuminated. The warning system also incorporates a chip locator panel which designates which of the five detectors has activated, thereby identifying the contaminated area. However, the panel is located forward of the pilot's left leg on the right side of the center console (under the instrument panel) and cannot be readily seen in flight. The information displayed by the hidden panel would be imperative to the pilot in determining what emergency action to take (land immediately, shut down an engine, continue to a more suitable landing site, etc). The location of the main transmission chip detector locator panel is a deficiency.

162. The FAS computer has a maintenance panel incorporating eight malfunction flags which indicate which portion of the FAS computer has a malfunction. The panel is located inside a black box behind the center console with a cover secured by dzus fasteners. The panel is inaccessible to the pilots in flight and if the FAS or FPS ON-OFF switch is cycled so that the failed portion is regained, the flag disappears and identification of the malfunctioning portion of the FAS computer is lost. The inaccessible location of the FAS computer maintenance panel is a shortcoming.

163. The parking brake is set by pulling up on the lock handle and depressing the toe brakes. The parking brake advisory light located on the caution advisory panel is activated when the parking brake handle is in the UP position. This light only indicates that the handle is up and not that the brakes are set. The lack of positive parking brake advisory information is a shortcoming.

164. The fuel selector valves often stuck in the direct or cross-feed position, which precluded closing of the fuel valve. In the event of a fire in the engine compartment it may be impossible to close the fuel valves and arm the engine compartment fire extinguishers. The sticking fuel selector valves are a deficiency which must be corrected prior to production.

165. An external intercom system is provided which permits voice contact with the crewmen during all startup and shutdown sequences and permits the crewmen complete walkaround capability. This is an enhancing feature.

166. The AC and DC primary bus circuit breakers are located on an overhead panel located aft of the pilot's and copilot's heads. It is impossible to see the panels without considerable head rotation (aft and up), which can induce vertigo and/or spatial disorientation. In the event of emergency action requiring a circuit

breaker to be pulled, the pilot and/or copilot must dissociate his attention from the instrument panel and the flight path to operate the circuit breakers. The inaccessibility of the AC and DC overhead circuit breaker panels is a shortcoming.

167. The AN/ARC-114 VHF-FM, AN/ARC-115 VHF-AM, and AN/ARC-116 VHF-AM radios each have a continuous guard channel monitor capability, but do not have a guard transmit switch to allow for immediate transmission on the guard frequency. To transmit on guard, the frequency must be manually tuned. This action requires excessive time and division of attention in event of an emergency. The radios do not have a preset frequency selection and each desired frequency must be manually tuned. The lack of a guard transmit switch and preset frequency selection on the UHF and VHF radios is a shortcoming.

Noise Evaluation

168. The noise level in the cockput during flight was objectionably high. The noise level at the engineer station (forward center portion of the cabin) was higher than at the pilot station. When the bleed air was turned on, the noise level at both stations increased considerably. The USAAEFA aircraft did not have soundproofing due to unique instrumentation requirements, but noise reduction with the proposed production soundproofing is not anticipated to be sufficient. The high internal noise level is a shortcoming.

169. The APU system provides pneumatic power for main engine starting and cabin heating and electrical power for ground and emergency in-flight electrical operations. Maintenance personnel and flight crew experienced extreme discomfort when working around the aircraft with the APU running. Personnel were virtually unable to communicate with one another unless on the intercom or by shouting in the person's ear. An external power cart was utilized more frequently throughout the GCT than the APU for ground systems checkout because of the noise discomfort. The APU, however, can be expected to be used more frequently in an operational environment. The high noise level of the APU is a shortcoming.

RELIABILITY AND MAINTAINABILITY

170. Reliability and maintainability of the YUH-60A were continuously monitored throughout the GCT. Deficiencies and shortcomings were submitted as Equipment Performance Reports (EPR's) when identified and are listed in appendix H. Some of the more significant items are discussed in this section.

171. Numerous failures of various components of the AFCS were encountered throughout the GCT. The characteristics of the AFCS varied, and often malfunctions/failures could not be duplicated from one flight to the next. During one flight longitudinal stick chatter was experienced during the first half of the flight and then disappeared. The problem could not be duplicated later, and the cause could not be determined. A total of 15 EPR's were submitted on AFCS components that failed. However, numerous discrepancies occurred which the

contractor could temporarily clear by adjusting various components of the AFCS. Component failures were so numerous that it is doubtful if the present system would be maintainable in a field environment. The poor reliability and maintainability of the AFCS is a deficiency which should be corrected prior to production.

172. During high-altitude tests at Coyote Flats (9500-foot pressure altitude), engine start problems were encountered when the engines were deliberately shut down (EPR No. 74-06-1-27). Normal starting procedures as outlined in the operator's manual do not require vapor venting (priming), although at this altitude the engines could not be started without vapor venting. All initial start attempts without vapor venting resulted in hung starts (N_G stable at 40 percent, $T_{4.5}$ at 670°C). Any subsequent start attempts (without vapor venting) resulted in an inability to achieve a light-off. The requirement to vapor vent prior to starting an engine at altitudes of 9500 feet or greater degrades mission effectiveness and reaction time, and has serious implications for achieving a quick start in a flight at these altitudes. Achieving a successful vapor vent at Coyote Flats was also a problem. The first engine to be started (either left or right) could be vapor vented after 3 to 4 minutes of priming. Venting at lower altitudes (4120 feet) normally required less than 1 minute. Once the first engine was started, achieving a vapor vent on the second engine was difficult and required 5 to 8 minutes of priming. Positive determination of the problem was not possible; however, there are two possible reasons. First, a possibly faulty check valve was installed in the fuel system that maintains fuel in the line from the tank to the engine. This valve was changed after testing was completed at the high-altitude test site, but it appeared to be free and not sticking when visually inspected. If this valve had been defective it could explain the requirement to vapor vent prior to achieving a successful start at the high-altitude site. Secondly, the vapor venting (fuel prime) system uses a pressure pump to supply pressurized fuel to the engine fuel control units. Both engines' fuel control units are supplied from a single line. After the first engine is running, this pressurized fuel is being fed into a system where one engine is sucking fuel and it is possible that the prime system pressure is being suctioned off into the running engine, causing the prime system for the second engine to be considerably less effective. The inability to start the YT700-GE-700 engines at high altitude without first achieving a successful vapor vent is a deficiency which should be corrected prior to production. The difficulty of the fuel system to achieve successful vapor vents at high altitude is a deficiency which should be corrected prior to production.

SUBSYSTEMS TESTS

Engine Performance

173. The uninstalled YT700-GE-700 engine is rated at 1536 shp at the intermediate (30-minute) limit and 1250 shp for maximum continuous operation under sea-level, standard-day conditions. These ratings are based on $T_{4.5}$ temperature and an output shaft speed of 20,000 rpm (263 rpm main rotor speed).

Installed in the YUH-60A, this engine will produce a maximum of 1516 shp at the IRP limit and 1236 shp at the MCP limit under sea-level, standard-day conditions at zero airspeed and an operational main rotor speed of 258 rpm (19,602 engine output shaft rpm), based on the GE engine model specification (AMC-CP-2222-02000). These installed powers were obtained by applying the installation losses discussed in the following paragraph to the engine model specification.

174. Engine induction system temperature and pressure characteristics were determined during the GCT. Results of the inlet survey are presented in figures 139 through 142, appendix G. At zero airspeed, the temperature rise was determined to be approximately 1°C for the left engine and 1.5°C for the right engine. The left engine inlet temperature fluctuated considerably in the airspeed region between 50 and 80 KTAS. The measured values indicate a temperature rise of approximately 3°C at 50 KTAS, decreasing to 1°C at 80 KTAS. Beyond 80 KTAS the temperature rise essentially followed the 100 percent ram recovery temperature. The temperature rise for the right engine was displaced above the 100 percent ram recovery temperature by approximately 1°C at 50 KTAS, 0.5°C at 110 KTAS, and 1°C at approximately 165 KTAS. The engine inlet pressure ratio, defined as the ratio of total inlet pressure to ambient static pressure, increased with increasing airspeed beyond translational lift for both engines. At zero airspeed, a pressure increase of 0.24 percent for the left engine and 0.17 percent for the right engine was indicated. The left and right engines indicated a pressure increase of 4.44 and 4.24 percent, respectively, at 165 KTAS. Exhaust system characteristics were supplied by Sikorsky Aircraft and are presented in figure 143, appendix G.

175. Based on the inlet and exhaust losses of the left (best) engine and the reduction in operational main rotor speed, IRP and MCP available were reduced in hover by 20 and 14 shp, respectively, compared to an uninstalled engine, for sea-level, standard-day conditions, as noted in paragraph 173. At 165 KTAS, IRP available was reduced from 1589 shp for the uninstalled specification engine to 1584 shp for the left engine. The installation power loss per engine, based on an average of both engines, at IRP at sea-level, standard-day conditions, was 23 shp at hover and 11 shp at 165 KTAS.

176. Intermediate rated power and MCP available and fuel flow at the conditions used for the contractual compliance analysis within this report are presented in figures 144 through 150, appendix G.

177. Maximum power available for the YT700-GE-700 engine is limited by four different parameters. These parameters are measured gas temperature ($T_{4.5}$), gas generator speed (N_G), referred N_G, and fuel flow. These parameters limit the power as a function of engine inlet temperature and pressure. At sea level, these limits occur approximately as follows: N_G above 35°C; $T_{4.5}$, 7 to 35°C; fuel flow, 7 to -9°C; and referred N_G below -9°C (fig. 144, app G). The fuel flow limit is present up to an altitude of approximately 700 feet. At an altitude of 700 feet, the change from the $T_{4.5}$ and referred N_G limits occurs at approximately 0.5°C. The temperatures where these limits change generally decrease slightly with

increasing pressure altitudes. Additionally, the temperatures where one limit supersedes another limit can be significantly different for various engines. This is especially true for the change between the $T_{4.5}$ and N_G limits which, for some engines, appear to be as low as 15°C. The significance of this to the pilot is that when operating at moderate to high ambient temperatures, one engine may be at the $T_{4.5}$ limit and the other engine unable to reach the $T_{4.5}$ limit due to an N_G limiter in the ECU. This might precipitate unnecessary maintenance action to attempt to restore power to the low engine. Further, when operating at colder ambient temperatures (0°C and below), the engines may develop considerably less power than would be expected. The above power-available degradation could be significant during takeoff or when attempting to terminate an approach with a hover at conditions where the aircraft performance is marginal. For these reasons, the following NOTES should be included in chapter 7 of the operator's manual:

NOTE

When operating at ambient temperatures in excess of 15°C, one or both engines may not be able to reach $T_{4.5}$ limit due to an N_C limiter in the ECU.

NOTE

At ambient temperatures below approximately 0°C, the maximum power available decreases with decreasing temperature approximately 0.25 percent per degree.

178. Power losses for the drive train and accessory losses were determined by comparing the total engine shp to the sum of main and tail rotor horsepower for similar conditions. The results of these comparisons for hovering flight are presented in figure 151, appendix G. Data obtained in forward flight substantiated these results. The power required of the drive train and accessories was approximately 130 shp.

179. Four YT700-GE-700 engines were required during the GCT phase. Referred engine parameters were used to compare the test engines with the model specification engine. Data of referred $T_{4.5}$, fuel flow, N_G, and compressor discharge static pressure versus referred shp are presented in figures 152 through 167, appendix G. Plots of referred fuel flow and compressor discharge static pressure versus referred shp displayed the least data scatter and therefore would be considered the parameters of most value for comparison purposes. Generally, all four engines produced less horsepower when compared to the installed engine model specification.

<u>Airspeed Calibration</u>

180. The airspeed system for the YUH-60A was calibrated using the trailing bomb and pace aircraft methods. The ship's airspeed calibration is presented in figures 168 and 169, appendix G. The ship's airspeed system is satisfactory.

181. The pilot and copilot airspeed systems were evaluated for the effects of sideslip. Sideslip was observed to have little effect on either the pilot or copilot airspeed systems for sideslip excursions up to 8 degrees either side of trim. At 83 KIAS the pilot airspeed system was more accurate in right sideslip, while in left sideslip the copilot airspeed system was more accurate. Generally, left or right sideslips up to 8 degrees were possible before observing airspeed differences greater than 2 knots between the two systems. The pilot and copilot airspeed system characteristics with sideslip variation are satisfactory.

CONCLUSIONS

GENERAL

182. The Sikorsky YUH-60A helicopter represents a significant improvement over present Army utility helicopters in the areas of handling qualities, crashworthiness, and general design features. However, the performance characteristics of the helicopter generally fell below the established design criteria.

a. The following enhancing characteristics were identified:

(1) The excellent aircraft control at low rotor speed (80 percent, 237 rpm) (para 40).

(2) The single-point pressure refueling and defueling system (para 83).

(3) Excellent rotor speed control during normal utility mission tasks (para 97).

(4) Effective torque-matching capability of the YT700-GE-700 engines (para 99).

(5) Automatic $T_{4.5}$ limiting feature of the YT700-GE-700 engine (para 100).

(6) Improved instrument flight characteristics provided by the commanded flight parameters of the flight director (para 117).

(7) The configuration of and readily identifiable aircraft formation lights (para 125).

(8) Pilot and copilot restraint system (para 156).

(9) The telescoping copilot collective control (para 157).

(10) Excellent design features of the instrument panel (para 159).

(11) The cockpit g meter allows precise, safe maneuvering to the helicopter load factor limits (para 160).

(12) The external intercom system (para 165).

b. There was a high degree of pilot confidence in maneuvering the helicopter to the extremes of the flight envelope at mission gross weight because of low vibrations, excellent controllability, and generally excellent rotor speed control characteristics (para 42).

c. The light collective control system breakout forces with adjustable friction OFF, coupled with the high degree of collective control effectiveness, resulted in collective pilot-induced oscillations when attempting to maintain precision hover heights, but could be corrected by the addition of pilot-adjusted collective friction (para 48).

d. The helicopter nose-upslope landing capability was not possible beyond 12 degrees because main transmission oil pressure decreased below the minimum allowable (para 89).

e. The Group 8 ECU did not satisfactorily correct engine-related problem areas attributable to the Group 7 ECU (para 104).

f. Ten deficiencies and 34 shortcomings were noted.

DEFICIENCIES AND SHORTCOMINGS

183. The following deficiencies were identified:

a. Highly undesirable and potentially unsafe characteristics of the roll trim integrator (para 127).

b. Significant power loss associated with operation of the engine anti-ice and cockpit heater systems (para 102).

c. High vibration levels in turning flight at alternate gross weight (para 151).

d. Excessive engine/rotor speed transients which occurred following large-magnitude collective application to the opposite extremes of the power demand schedule (para 96).

e. Poor reliability and maintainability of the AFCS (para 171).

f. Sticking fuel selector valves (para 164).

g. Location of the main transmission chip detector locator panel (para 161).

h. Inability to start the YT700-GE-700 engines at high altitudes without first achieving a successful vapor vent (para 172).

i. Difficulty in achieving a successful fuel system vapor vent at high altitude (para 172).

j. The inability to achieve full aft cyclic and the restriction of lateral cyclic control (para 155).

184. The following shortcomings were identified:

a. Lack of an aural engine-out warning device (para 133).

b. The high longitudinal control force gradient in flight (para 45).

c. The unsymmetrical lateral breakout forces (para 46).

d. The high lateral control force gradient (para 46).

e. Restricted forward field of view during high nose-up attitudes (para 106).

f. The arming of the normal load release switch when arming the emergency release system (para 115).

g. The lack of two VOR/ILS navigation radios (para 121).

h. Undesirable characteristics of the yaw trim integrator (para 128).

i. Excessive delay in engagement of the airspeed hold system following trim actuation (para 129).

j. Restricted field of view through the chin bubble (para 158).

k. The inaccessibility of the AC and DC overhead circuit breaker panels (para 166).

l. The high longitudinal control force to small control position change about trim at 60 KCAS and greater (para 54).

m. Unstable longitudinal control force stability at representative NOE airspeeds (40 KCAS and slower) (para 54).

n. The inability to use the No. 1 bearing pointer of the HSI (para 122).

o. The lack of a guard transmit switch and preset frequency selection on the UHF and VHF radios (para 167).

p. High internal noise level in the cockpit (para 168).

q. The undesirable mechanical characteristics of the pressure-operated beep trim switch (para 50).

r. High roll rates developed following lateral trim control system failure (para 140).

s. The inability to close the cockpit doors with the release handles in the locked position without causing damage to the door locking mechanism (para 153).

t. Inability to positively control the aircraft to the ground on a 12-degree nose-downslope (para 88)

u. Frequent inability to lock the tail wheel from the cockpit (para 82)

v. The premature commanded slowing of the aircraft at 750 feet AGL in the deceleration mode of the flight director (para 118).

w. The lack of advisory information when the flight director system was in the go-around mode (para 119)

x. The improperly set airspeed and collective commands of the go-around mode of the flight director (para 119)

y. The excessive time to reach a selected vertical speed in the vertical speed mode of the flight director (para 120).

z. Multiple secondary lights/cockpit floodlight controls and their dissimilar method of operation (para 124)

aa. Potential of having the FPS disengaged without a corresponding caution light (para 130)

bb. The poor cockpit door position detent characteristics (para 154)

cc. The lack of adequate seat adjustment range aft and down (para 155)

dd. Restricted field of view rearward, caused by the side structural bulkhead located to the rear of the pilot and copilot seats (para 158)

ee. Inaccessible location of the FAS computer maintenance panel (para 162)

ff. Lack of positive parking brake advisory information (para 16[illegible]

gg. High noise level of the APU (para 169).

hh. Difficulty in unlocking and locking the tail wheel (para 82)

SPECIFICATION COMPLIANCE

185. Within the scope of these tests, the YUH-60A helicopter failed to meet the following commitments of the PIDS:

a. Paragraph 3.2.1.1.1.1a - The helicopter could not hover OGE at the primary mission gross weight, 4000 feet pressure altitude, 35°C, and 95 percent IRP (para 12).

b. Paragraph 3.2.1.1.1.1a - The aircraft could not climb vertically 550 ft/min (RFP para 3.2.1.1.1.1a, 450 ft/min) at a pressure altitude of 4000 feet, 35°C, 95 percent IRP, and primary mission gross weight (para 20).

c. Paragraph 3.2.1.1.1.1b - V_{cruise} at a pressure altitude of 4000 feet, 35°C, MCP, and primary mission gross weight was 138 KTAS (22 KTAS less than required) (RFP para 3.2.1.1.1.1b, 7 KTAS less than required) (para 24).

d. Paragraph 3.2.1.1.1.1c - The aircraft could not perform the combat troop-assault primary mission profile at the specified airspeed of 145 KTAS (paras 24 and 26).

e. Paragraph 3.2.1.1.1.2 - The aircraft endurance for the alternate endurance mission was 2.12 hours (0.18 hour less than required) (paras 23 and 25).

f. Paragraph 3.2.1.1.1.3a - The single-engine V_H at 4000 feet, 35°C, and IRP, was 94.5 KTAS (31.5 knots less than required) (RFP para 3.2.1.1.1.3a, 5.5 knots less than required) (para 24).

g. Paragraph 3.2.1.1.1.3b - The single-engine service ceiling (100-ft/min rate of climb) at 35°C was 2877 feet (4223 feet less than required) (RFP para 3.2.1.1.1.3b, 2123 feet less required) (para 22).

h. Paragraph 3.2.1.1.1.3d - The helicopter could not take off using single-engine IRP at 4000 feet pressure altitude and 35°C at primary mission gross weight less payload (para 18).

i. Paragraph 3.2.1.1.5.1.3 - The instruments on the instrument panel were difficult to read at 160 KCAS because of vibration (para 148).

j. Paragraph 3.2.1.1.5.1.4 - The 4/rev vibration levels exceeded 0.5g at airspeeds from 30 KTAS rearward to V_{cruise} and 0.2g from V_{cruise} to V_{NE} (paras 148 and 150).

k. Paragraph 3.2.2.1.5 - The primary mission gross weight was 16,853 pounds (1003 pounds greater than specified) (paras 23 and 24).

l. Paragraph 50.3.1.8.5 - The normal acceleration following load release in hover increased to 1.2g, 11 percent over the allowed 10 percent of the positive limit load factor (1.8) (para 115).

m. Paragraph 50.3.2.1.1 (Table I) - The right lateral breakout force (plus friction) was 1.60 pounds (6.6 percent greater than specified (para 46).

n. Paragraph 50.3.2.1.2 - The right lateral breakout force was 0.65 pound, 50 percent greater than the left breakout force (para 46).

o. Paragraph 50.3.2.2 - The longitudinal force gradient in flight was 7.5 lb/in. (para 45).

p. Paragraph 50.3.2.2 - The lateral force gradient in flight was 4 lb/in. (para 46).

q. Paragraph 50.3.2.7.1 - Roll and yaw rates developed from trim runaways exceed the 10-deg/sec commitments after 3 seconds by a maximum of 27 and 6 deg/sec, respectively (para 140).

r. Paragraph 50.3.2.10 - Limitation in control power exists with various combinations of control inputs (para 49).

s. Paragraph 50.3.3.1.1 1 - Stick jump was present when the trim control system was actuated (para 50).

t. Paragraph 50.3.3.1.3 - The level flight control force and control position static stability was not positive for all increasing and decreasing airspeed values from trim. The magnitude of control force instability in the 15- to 50-KCAS airspeed range was 3 to 4 pounds, exceeding the 1-pound limit (para 54).

u. Paragraph 50.3.3.1.4 - The control force and position versus normal acceleration in steady-state turning flight was neutral to negative with FPS ON. The stick force per g gradient at 1. [illegible] KCAS exceeded the maximum specified of 20 pounds per g by 1 pound (para [illegible]8).

v. Paragraph 50.3.3.1.4.1 - The ratio of maximum longitudinal control force to peak normal acceleration for sudden pull-ups was not always greater than the ratio obtained in steady acceleration (para 65).

w. Paragraph 50.3.4.1.7 - The side-force characteristics at 73 KCAS IRP climb and sideslip angles from 5 to 20 degrees left sideslip were negative (para 59)

x. Paragraph 50.3.4.4 - The bank angle from a steady 5-degree banked turn to the left was halved in 4 seconds, 6 seconds (50 percent) less than the specified 10 seconds (para 72).

y. Paragraph 50.3.6.1.1 - The collective control could not be delayed 1 second after activation o[illegible] the low engine sp[illegible]d warning system following an engine failure (para 136).

z. Paragraph 50.3.7.7.2 - The aircraft was unable to land on a 12-degree nose-downslope with positive aircraft control (para 88).

RECOMMENDATIONS

186. The deficiency reported in paragraph 183a should be corrected prior to further government testing.

187. The deficiencies reported in paragraph 183b through j should be corrected prior to production.

188. The shortcomings reported in paragraph 184 should be corrected.

189. The single-engine IGE hover H-V altitude should be 30 feet AGL (para 40).

190. The following CAUTION should be included in chapter 8 of the operator's manual (para 48).

CAUTION

> Collective overcontrolling or pilot-induced oscillations may occur during precision hover tasks with collective control friction OFF. Sufficient collective friction should be set to provide a positive force cue.

191. The following CAUTION should be included in chapter 4 of the operator's manual (para 70).

CAUTION

> Intentional flight in instrument meteorological conditions (IMC) should not be conducted with an AFCS failure that renders the FAS/FPS and/or SAS inoperative. If failure should occur during IMC flight reduce airspeed to 75 to 80 KIAS to minimize pilot workload.

192. The foll[illegible]g NOTE should be included in chapter 3 of the operator's manual (par[illegible]

NOTE

> Simultaneous dual-engine starts should not be attempted when the outside air temperature is above 30°C, due to the possibility of APU flameout.

193. The following NOTE should be included in chapter 7 of the operator's manual (para 88).

NOTE

> During nose-down landings and takeoffs on 12-degree slopes, the aircraft will translate downslope. The pilot must ensure that adequate obstacle clearance is available prior to landing.

194. The YUH-60A helicopter should be restricted to a maximum of 12 degrees nose-upslope landing due to the inability of the transmission oil pumps to maintain adequate oil pressure while in the nose-upslope orientation on slopes greater than 12 degrees (para 90).

195. The following WARNING should be included in chapter 8 of the operator's manual (para 91).

WARNING

> During cross-slope and nose-upslope operations, vertical clearance between the main rotor tip path plane and the ground is extremely reduced on the upslope side of the helicopter. Personnel must be warned not to approach the helicopter from the upslope direction or depart from the helicopter in the upslope direction.

196. The following CAUTION should be included in chapter 8 of the operator's manual (para 136).

CAUTION

> In the event of an engine failure during single-engine flight or dual-engine failure, rotor speed will rapidly decay to below the minimum power-off limit (90 percent rotor speed). Rapid reduction of collective control to full-down will be required to arrest rotor speed decay and regain suitable autorotational rotor speed.

197. The hydraulic system should be redesigned to preclude depletion of the backup system for certain failures of the No. 1 or No. 2 system (para 143).

198. Further testing should be conducted in maneuvering flight at alternate gross weight to determine if potential restrictions to the flight envelope should be imposed by pitch link load structural limits (para 151).

199. The following NOTE should be included in chapter 7 of the operato s manual (para 177).

NOTE

When operating at ambient temperatures in excess of 15°C, one or both engines may not be able to reach $T_{4.5}$ limit due to an N_G limiter in the ECU.

200. The following NOTE should be included in chapter 7 of the operator's manual (para 177).

NOTE

At ambient temperatures below approximately 0°C, the maximum power available decreases with decreasing temperature approximately 0.25 percent per degree.

APPENDIX A. REFERENCES

1. Materiel Need Document, Action Control No. 10705, February 1972.

2. Memorandum of Understanding, 30 September and 1 October 1975, subject: Memorandum of Understanding, UTTAS and LAMPS Project Managers.

3. Advance copy, Final Report, USAAEFA, Project No. 74-03-1, *Army Preliminary Evaluation I, Utility Tactical Transport Aircraft System (UTTAS), Sikorsky YUH-60A Helicopter*, February 1976.

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7. Operator's Manual, Sikorsky Aircraft Division, No. SA4047-25, *Army Model YUH-60A Helicopter*, 1 December 1975, with Change 2, 15 March 1976.

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11. Letter, AVSCOM, DRSAV-EQA(A), 2 July 1976, subject: UTTAS Instrumentation Drag Accountability.

12. Report, Sikorsky Aircraft Division, SER 70099, *YUH-60A Weight and Balance GCT Aircraft*, 31 March 1976.

13. Letter, AVSCOM, DRSAV-EQ, 13 July 1976, subject: UTTAS Government Competitive Test, Height-Velocity (H-V) Diagram, USAAEFA Project No. 74-06-1.

14. Letter, General Electric Company, GCT-3, 11 May 1976, subject: Field Use of Improved ECU (G08) for UTTAS GCT Engines.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The Sikorsky YUH-60A UTTAS is a twin turbine-engine, single-main-rotor helicopter capable of transporting cargo, 11 combat troops, and weapons during day, night, visual, and instrument conditions. A complete description of the aircraft is contained in the operator's manual (ref 4, app A). Major features of the helicopter are described below and shown in photos 1 through 4.

AIRFRAME

2. The airframe, as shown in figure 1, includes the cockpit, mid-fuselage section (cabin), transition section, tail cone, and tail pylon. The fuselage primary structure is aluminum alloy and is semimonocoque in construction.

3. The cockpit basic structure consists of four longitudinal beams that start at the nose and extend back to the transition section. The cockpit nose serves as the avionics compartment. The cockpit roof is one-piece fiberglass. The windshields are shatter-resistant, scratchproof glass with electrically operated windshield wipers. The windshields are electrically heated for defogging and anti-icing. The jettisonable cockpit doors are made of aluminum and contain a sliding window. Dual controls and duplicate flight instruments are provided. Cockpit drawings are presented in figures 2 and 3. Circuit breaker panels are installed above and behind the pilot's and copilot's heads. The engine and transmission instrument displays contain miniature lamps that light up to show the value of the sensed parameter. The lamplight is carried to the instrument face by fiberoptics which light vertical strips and digital displays.

4. The mid-fuselage section includes the crew chief/gunner stations and troop/cargo compartment. The compartment will accommodate 11 troops plus the crew chief/gunner. Provisions are made for four litters. The compartment is 82 inches wide at the floor, 92 inches wide at seat level, 108 inches long, and 54 inches high. Cargo tie-down rings of 5000 pounds capacity are provided in the floor and along the side frames. Floor load limit is 300 pounds per square foot (lb/ft^2) in the cargo area and 75 lb/ft^2 in the crew area. An aft sliding cargo door providing an opening of 69 inches by 54 inches is on each side of the troop/cargo compartment. The GCT test aircraft troop/cargo compartment contained instrumentation and cg control equipment.

5. The transition section connects the mid-fuselage section with the tail cone. This section contains two 178-gallon fuel tanks and two equipment compartments (75-pound limit capacity) which are accessible from inside the troop/cargo compartment.

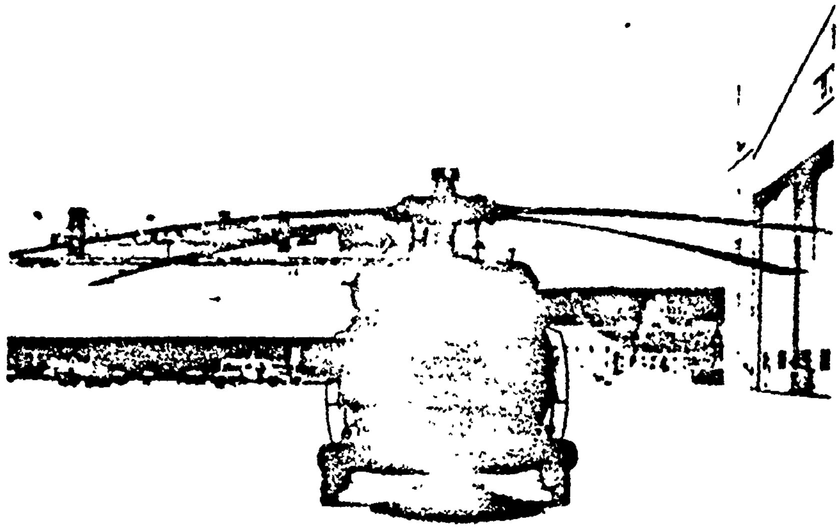

Photo 1. Front View, YUH-60A Helicopter.

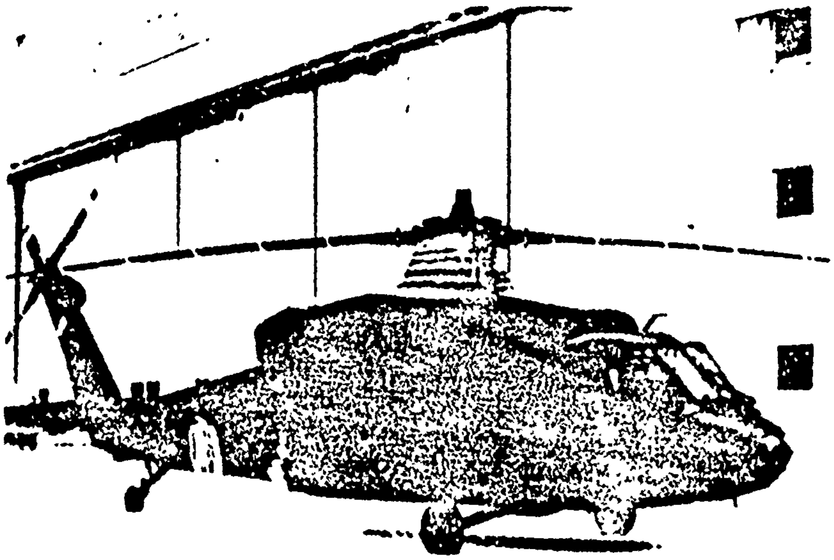

Photo 2. Right Quarter View, YUH-60A Helicopter.

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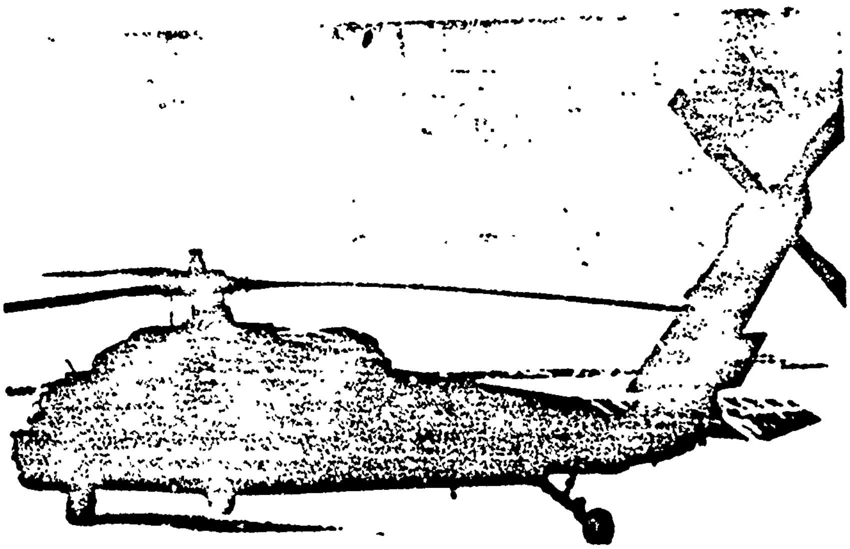


Photo 3. Left Three-Quarter View, YUH-60A Helicopter.

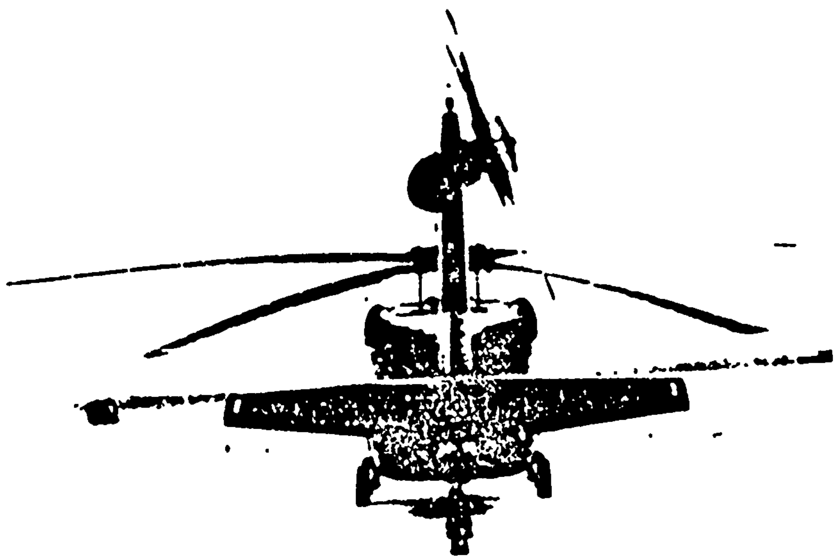


Photo 4. Rear View, YUH-60A Helicopter.

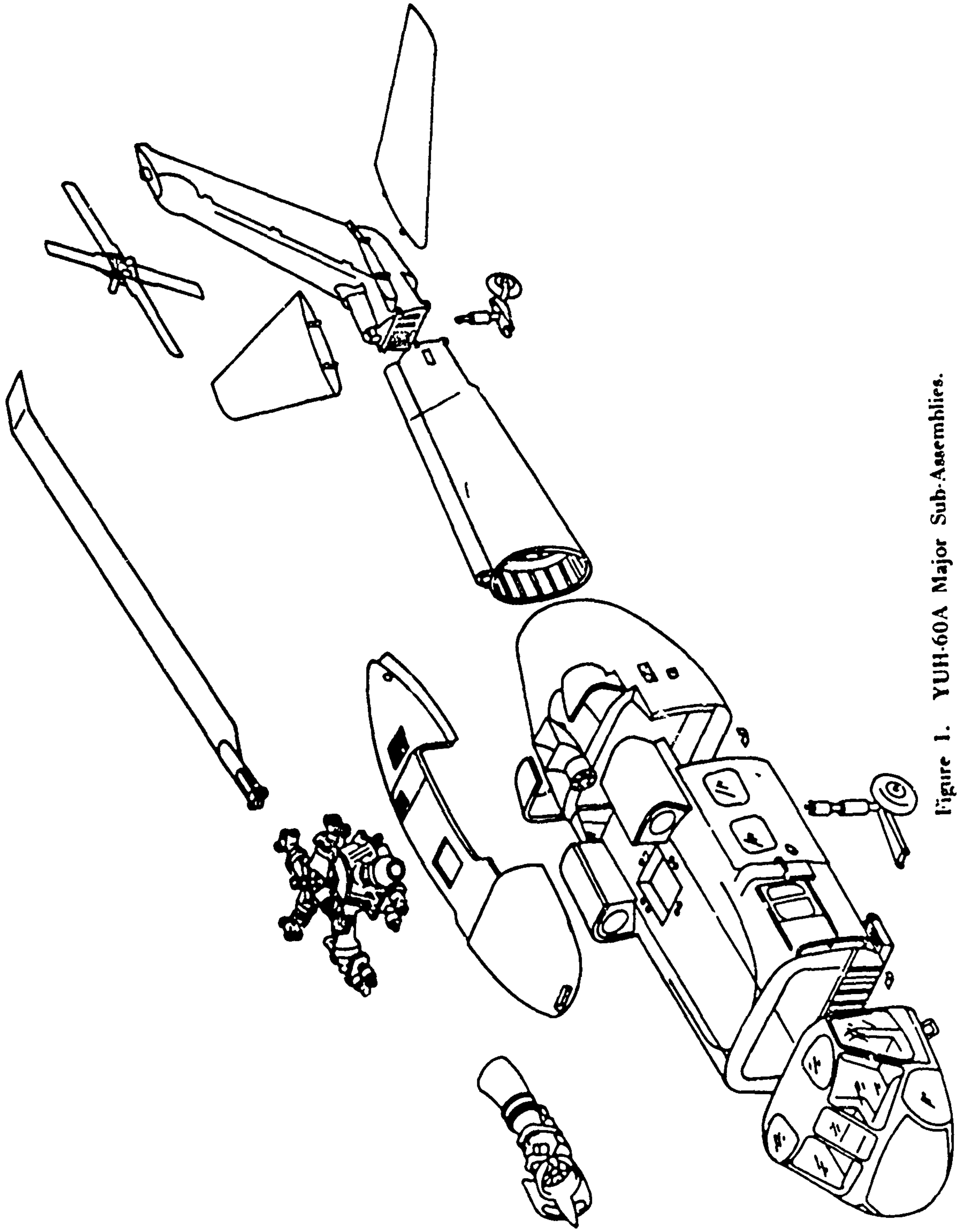
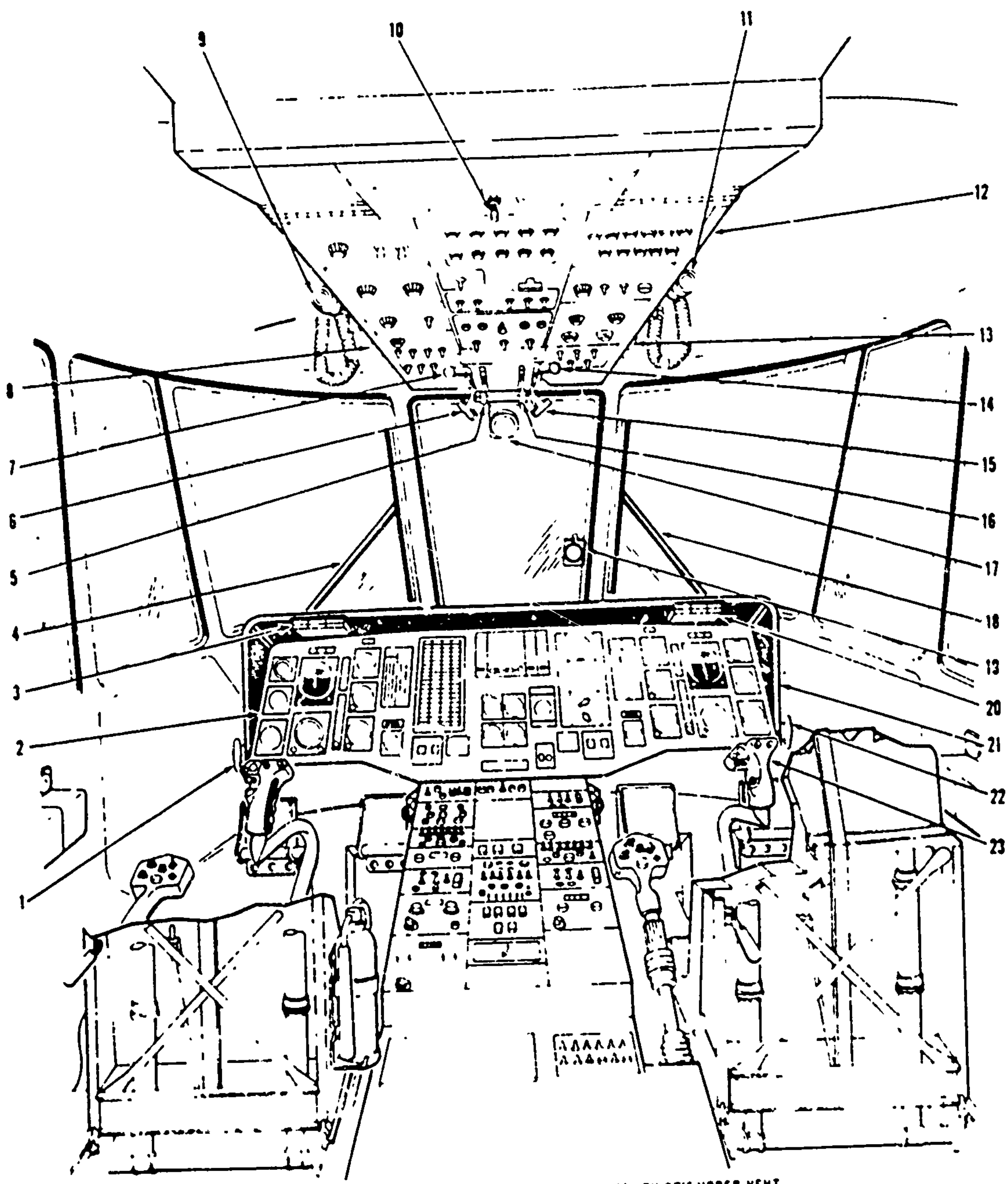
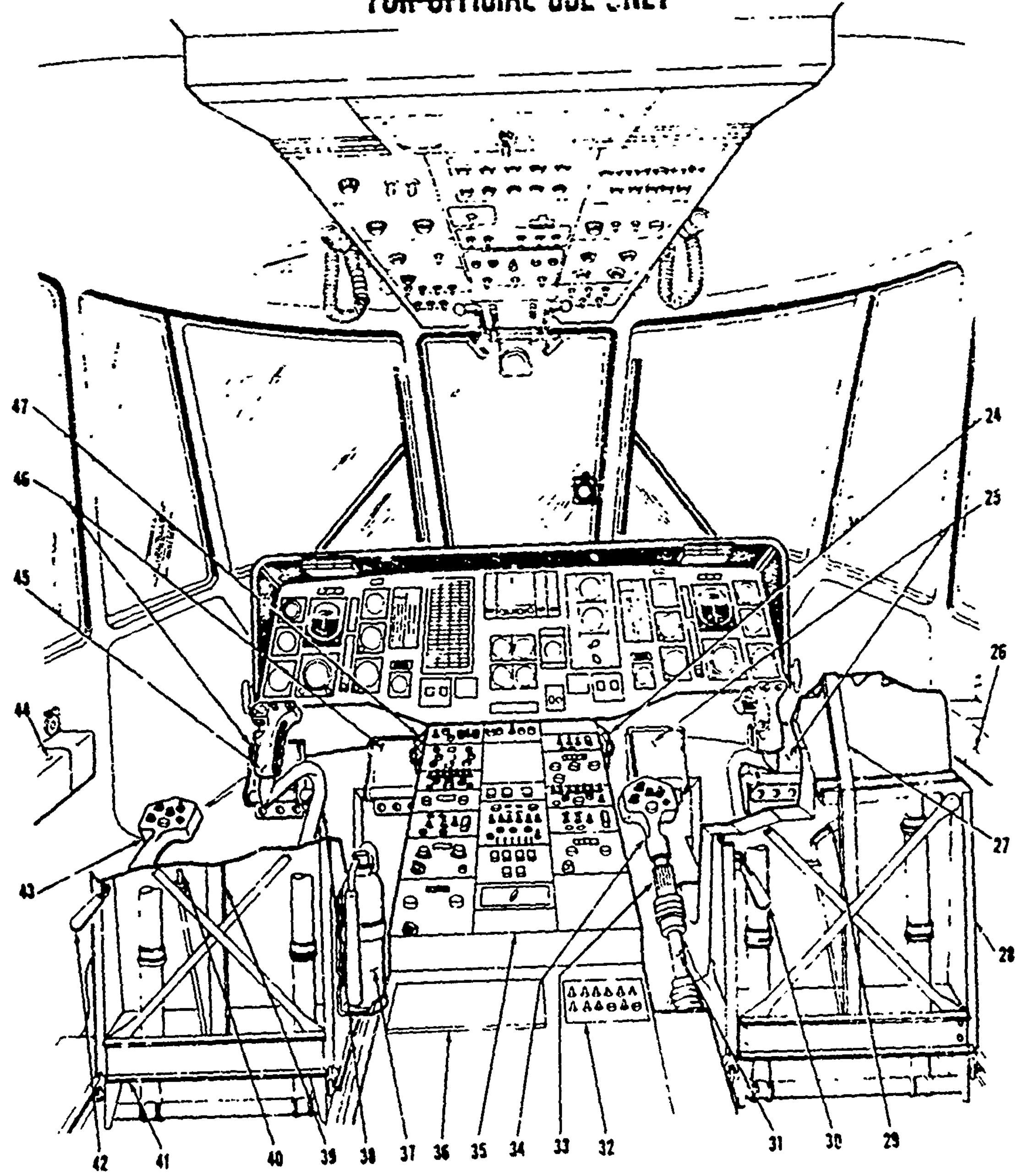


Figure 1. YUH-60A Major Sub-Assemblies.



1. COPILOT'S PEDALS ADJUST LEVER
2. INSTRUMENT PANEL
3. MASTER WARNING PANEL
4. COPILOT'S WINDSHIELD WIPER
5. NO. 1 ENGINE POWER CONTROL LEVER
6. NO. 1 ENGINE FIRE T HANDLE
7. NO. 1 ENGINE FUEL SHUTOFF LEVER
8. COPILOT'S UPPER VENT
9. COPILOT'S UTILITY LIGHT
10. COCKPIT FLOOD LIGHT CONTROL
11. PILOT'S UTILITY LIGHT
12. UPPER CONSOLE
13. PILOT'S UPPER VENT
14. NO. 2 ENGINE FUEL SHUTOFF LEVER
15. NO 2 ENGINE FIRE T HANDLE
16. NO 2 ENGINE POWER CONTROL LEVER
17. OUTSIDE AIR TEMPERATURE GAGE
18. PILOT'S WINDSHIELD WIPER
19. MAGNETIC STANDBY COMPASS
20. PILOT'S MASTER WARNING PANEL
21. GLARE SHIELD
22. PILOT'S PEDAL ADJUST LEVER
23. PILOT S CYCLIC STICK

Figure 2. Cockpit Schematic.



24. PILOT'S LOWER VENT
25. PILOT'S ANTI TORQUE ROTOR PEDALS
26. PILOT'S FIRST AID KIT
27. PILOT'S SHOULDER HARNESS
28. PILOT'S SEAT
29. PILOT SEAT EMERGENCY VERTICAL RELEASE
30. PILOT SEAT TILT RELEASE
31. PILOT'S COLLECTIVE STICK
32. LOWER CONSOLE CIRCUIT BREAKER PANEL
33. COLLECTIVE FRICTION CONTROL
34. PILOT'S COLLECTIVE STICK GRIP
35. LOWER CONSOLE
36. DATA CASE
37. FIRE EXTINGUISHER
38. COPILOT'S CYCLIC STICK
39. COPILOT'S SHOULDER HARNESS
40. COPILOT SEAT EMERGENCY VERTICAL RELEASE
41. COPILOT'S SEAT
42. COPILOT SEAT TILT RELEASE
43. COPILOT'S COLLECTIVE STICK GRIP
44. COPILOT'S FIRST AID KIT
45. COPILOT'S CYCLIC STICK GRIP
46. COPILOT'S ANTI TORQUE PEDALS
47. COPILOT'S LOWER VENT

Figure 3. Cockpit Schematic.

6. The tail cone connects the transition section, tail rotor and stabilator controls, and the tail rotor drive shaft to the tail pylon. The tail pylon provides mounting points for the intermediate gearbox, tail gearbox, tail rotor, and stabilator. An operational description of the horizontal stabilator is contained in appendix C. The tail pylon leading edge has a built-in FM antenna and is hinged to allow inspection of the pylon tail rotor drive shaft. The pylon surface is cambered to unload the tail rotor in cruising flight. The tail cone incorporates a hinge just above the tail wheel which allows folding of the pylon for air transport.

MAIN ROTOR AND SWASHPLATE

7. The articulated four-bladed main rotor consists of a one-piece titanium rotor hub splined to the main rotor shaft, a blade retention assembly, elastomeric bearings, blade dampers, adjustable control rods, swashplate assembly, rotating scissors, four titanium-spar main rotor blades, and a bifilar vibration absorber. The main rotor has 7 degrees of precone and 7 degrees of prelag. The main rotor blades are a modified SC 1095 airfoil with slight camber and drooped leading edge, 18 degrees of negative twist, and aft swept tips. An indicator is installed on each blade at the root trailing edge to visually indicate when blade spar structural integrity (internal pressure) is lost. The blades are attached to the rotor head by two quick-release expandable bolts. All blades can be folded to the rear and downward along the tail cone. A gust lock prevents the blades from rotating when the helicopter is parked. The gust lock is designed to withstand the ground-idle power output of one engine. A caution light on the caution/advisory panel is provided to show when the gust lock is engaged.

TAIL ROTOR

8. The four-bladed crossbeam tail rotor is composite in construction, and contains no bearings. Blade flap and pitch change motion are provided by deflection of the flexible graphite fiberglass spar. The spar is a continuous member running from the tip of one blade to the tip of the opposite blade. The tail rotor is attached to the tail gearbox on the right side of the pylon and the plane of the rotor is canted 20 degrees (lift vector up from horizontal).

LANDING GEAR

9. The main landing gear, mounted on each side of the helicopter, incorporates a two-stage oleo strut which will absorb landing gross weight loads of 15,850 pounds up to 15 feet per second. The main landing gear wheels incorporate single disc brakes that are toe-operated by the pilot through master cylinders located on the pedals. The tail gear is below the rear section of the tail cone and has a two-stage shock strut. The tail wheel swivels 360 degrees and can be locked in the trail position.

POWER TRAIN

10. The power train consists of two YT700-GE-700 engines, a main transmission, intermediate gearbox, tail gearbox, and connecting drive shafts. Power from the engines is transmitted to the main transmission through input modules.

11. The main transmission is mounted on top of the cabin fuselage just forward of and between the two engines. The main transmission is made of magnesium, has a 3-degree forward tilt, and is modular in construction, consisting of two input modules, the main module, and two accessory modules.

12. Two interchangeable input modules transfer power from the engines to the main module and provide the first gear reduction from 20,000 to 5867 rpm (fig 4). The input modules are mounted on the left and right front of the main module and support the front of the engines. The input modules contain an input pinion and gear, a free-wheel unit, and an accessory module. The input bevel pinion gear and quill shaft drives a combining gear which drives the main module. Ramp and roller-type (sprague) free-wheel clutches allow engine disengagement from the transmission during autorotation or in event of a nonoperating engine.

13. One accessory module is mounted on the forward section of each input module. Each accessory module provides mounting and drive for an AC electrical generator and a hydraulic pump package. A rotor speed tachometer sensor is mounted on the left accessory module only. The accessory module will continue to be driven by the main rotor with engine disengagement.

14. The main module supports the input modules and provides for a reduction in speed from the input module rotor speed of 5867 rpm to the rotor speed of 263 rpm (100 percent N_R), and the tail rotor drive shaft speed of 4200 rpm.

VIBRATION ABSORBERS

15. Table 1 lists the vibration absorber locations, weights, and the frequency that the absorber dampens.

ENGINES

16. A description of the YT700-GE-700 engines is contained in appendix D.

FUEL SYSTEM

17. A separate suction fuel system is provided for each engine. The fuel systems consist of two interchangeable, crashworthy, ballistic-resistant tanks, self-sealing lines from the main fuel tanks, firewall mounted selector valves, prime/boost pump and engine driven suction pumps. Selector valves on the engine control quadrant

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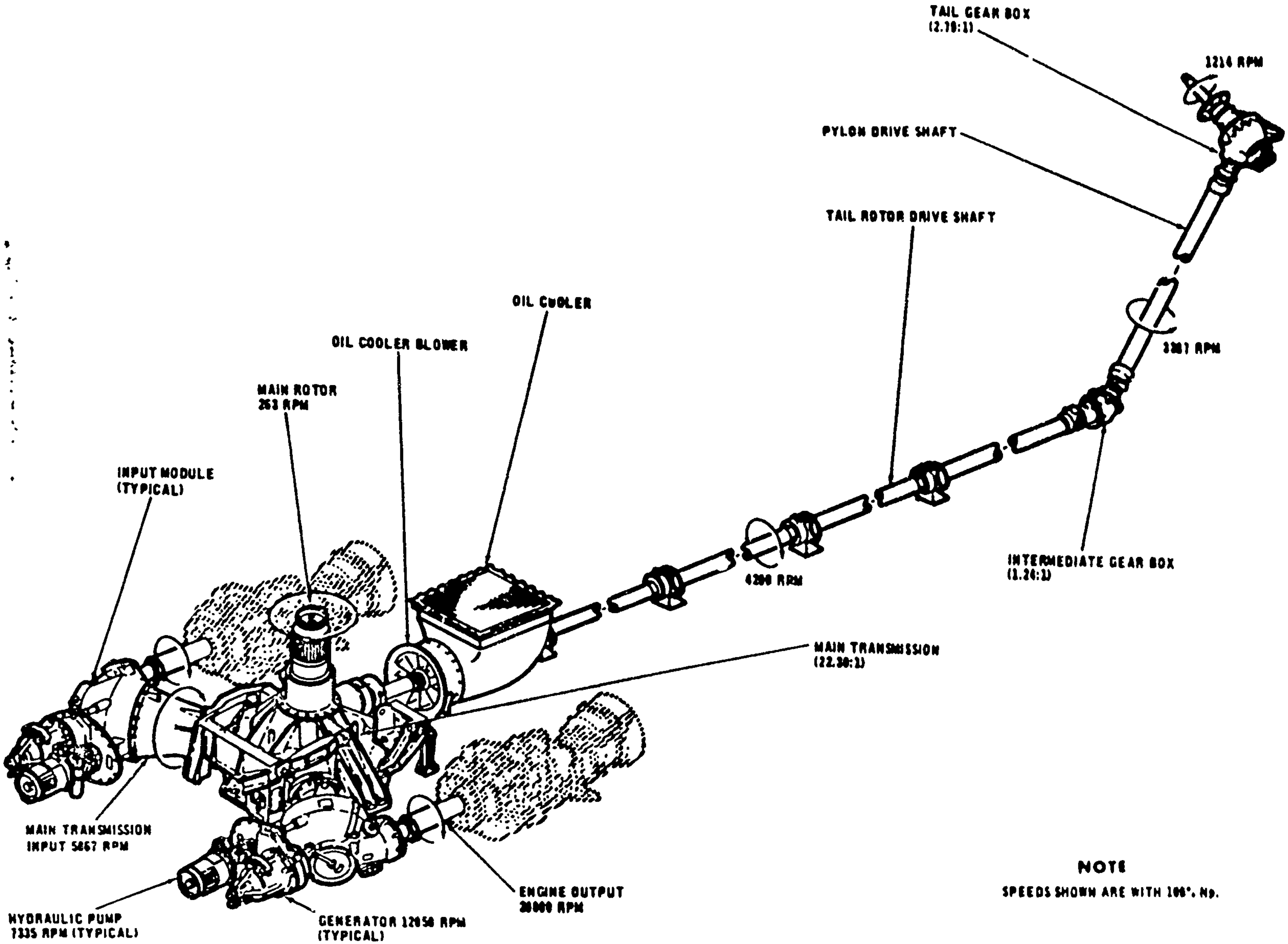

Figure 4. YUH-60A Drive Train.

Table 1. Vibration Absorber Location.

Location[1]	Frequency	Weight[2] (lb)	Water Line (in.)	Fuselage Station (in.)
Nose bay	4/rev (17.2 Hz)	99.6	212.0	176.5
Transmission overhead (fwd)	4/rev (17.2 Hz)	98.9	272.5	317.2
Transmission overhead (aft)	4/rev (17.2 Hz)	97.1	272.5	363.3
Nose bay	4/rev (12.9 Hz) vertical	22.1	209.0	192.4
Bifilar	4/rev (12.9 Hz) in-plane	174.0	320.5	340.9

[1]Buttline for all absorbers was 0.0 inch.
[2]Total weight devoted to vibration damping is 491.7 pounds.

permit operation of either engine from either fuel tank. The prime pump primes all fuel lines if prime is lost and acts as an APU boost for APU operation above 8000 feet MSL.

Fuel Tanks

18. Both fuel tanks are crashworthy, self-sealing and interchangeable, with a designed fuel capacity of 177 US gallons each. Internal fuel quantity is continuously displayed by a gaging system consisting of a probe in each tank, a dual channel fuel quantity signal conditioner, a fuel quantity indicator, and a dual channel low level warning system. The tank probes are connected to the fuel quantity gage signal conditioner. The signal conditioner provides a low voltage to the tank probes and converts the tank probes' high current to a DC voltage that is proportional to fuel quantity. The signal is then sent to a fuel quantity indicator provided for each engine. A separate total fuel quantity indicator numerically displays the total quantity of fuel on board.

Fuel System Controls

19. Each fuel system has a selector valve which is manually operated through the selector lever on the overhead engine control quadrant. An emergency handle on each side of the quadrant is arranged so that pulling the handle engages the fuel selector lever, bringing it to the OFF position. The fuel selector levers are connected to the fuel selector valves with flexible push-pull cables. Each lever can be actuated to three positions: OFF; direct (DIR); and cross-feed (XFD). With the levers positioned at OFF, the control valves are closed, allowing no fuel flow to the engines. With the levers in the DIR position, the selector valves are opened, providing fuel flow for each engine from its individual tank. At the XFD position, the engine is connected to the opposite tank. Any tank can feed any one or both engines. A low-level sensor on each probe provides signals to the dual channel control unit, which activates two low-level warning lights on the caution-advisory panel indicating #1 FUEL FLOW or #2 FUEL FLOW when the fuel level decreases to 170 to 190 pounds.

Engine Fuel Prime System

20. A toggle switch on the upper console controls the fuel boost/prime pump. The pump is used to prime the fuel system to the main engines when required and to provide boost fuel pressure to the APU for APU starts above 8000 feet MSL. Advisory panel indication is displayed during operation of the pump by a light marked PRIME BOOST PUMP ON.

Refueling/Defueling

21. There are three methods to refuel the aircraft: gravity, pressure, and closed circuit. A pressure refueling and defueling system provides completed refueling and defueling of both tanks from one point on the left side of the helicopter. Pressure refueling lines accept a system flow rate of 300 gpm at a refuel nozzle pressure

of 55 psig. The pressure refueling adapter and closed circuit adapter are combined in one refueling receptacle. The adapters are independent units operating in parallel. with a common discharge port. Closed circuit refueling uses the pressure refueling system and its components. The closed circuit refueling adapter delivers 110 gpm of fuel at 15 psig. A tank-full automatic shut-off valve is float-operated. Gravity fueling is done through filler caps on each side of the fuselage for the respective tanks. The tanks can be individually defueled through a valve in each tank sump.

ELECTRICAL SYSTEM

General

22. A schematic of the electrical system is presented in figures 5 and 6. Alternating current is the primary source of electrical power. The primary electrical system consists of two independent systems and an APU backup system. Each of the three systems is individually capable of supplying power to aircraft electrical equipment. An electric power priority feature allows either the No. 1 or No. 2 main generator to automatically preempt the APU generator, which in turn automatically preempts external power. Primary DC power is obtained from two converters, with the battery as a secondary DC power source. A subsystem feeds two independent AC primary buses and an AC essential bus. A portion of each AC primary bus load is converted to 28 VDC by unregulated 200-ampere AC/DC converters. The 28-VDC is distributed by two independent DC primary buses and a DC essential bus. The primary power sources are 115-VAC, 3-phase, 400-Hz, 30/45 KVA brushless oil spray-cooled generators.

Primary Alternating Current Power

23. Each primary power system contains a 30/45-KVA generator mounted on and driven by the transmission accessory gearbox module, a current transformer, a generator control unit, generator controller, and current limiter, all of which are interchangeable. The No. 1 and No. 2 system outputs are applied to the No. 1 and No. 2 AC primary buses, respectively. The system also contains an underfrequency cut-out system. The underfrequency cut-out system is not operational when the helicopter is airborne because of a cut-out switch located on the left main landing gear drag beam.

Auxiliary Alternating Current Power

24. Auxiliary AC power (backup) is provided by a 115-VAC, 3-phase, 400-Hz 20/30-KVA air-cooled generator mounted on and driven by the APU, a current transformer, and a generator control unit. If the primary AC generators are not operating, the auxiliary AC power output will be supplied to the No. 2 primary AC bus and then through the current limiter to the No. 1 primary bus. The AC essential bus will obtain power from the No. 1 primary bus based on design logic.

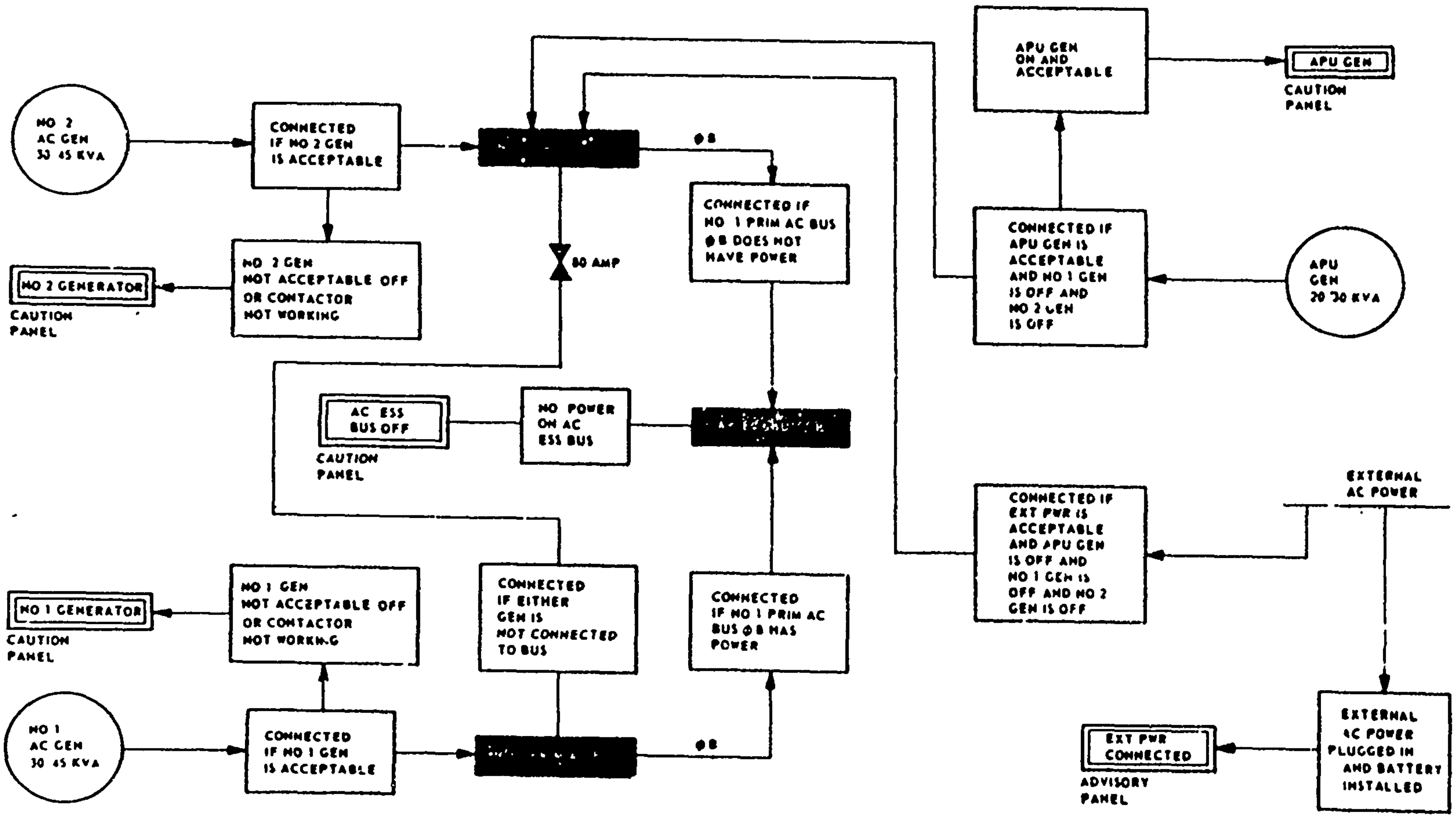

Figure 5. AC Power System Block Diagram.

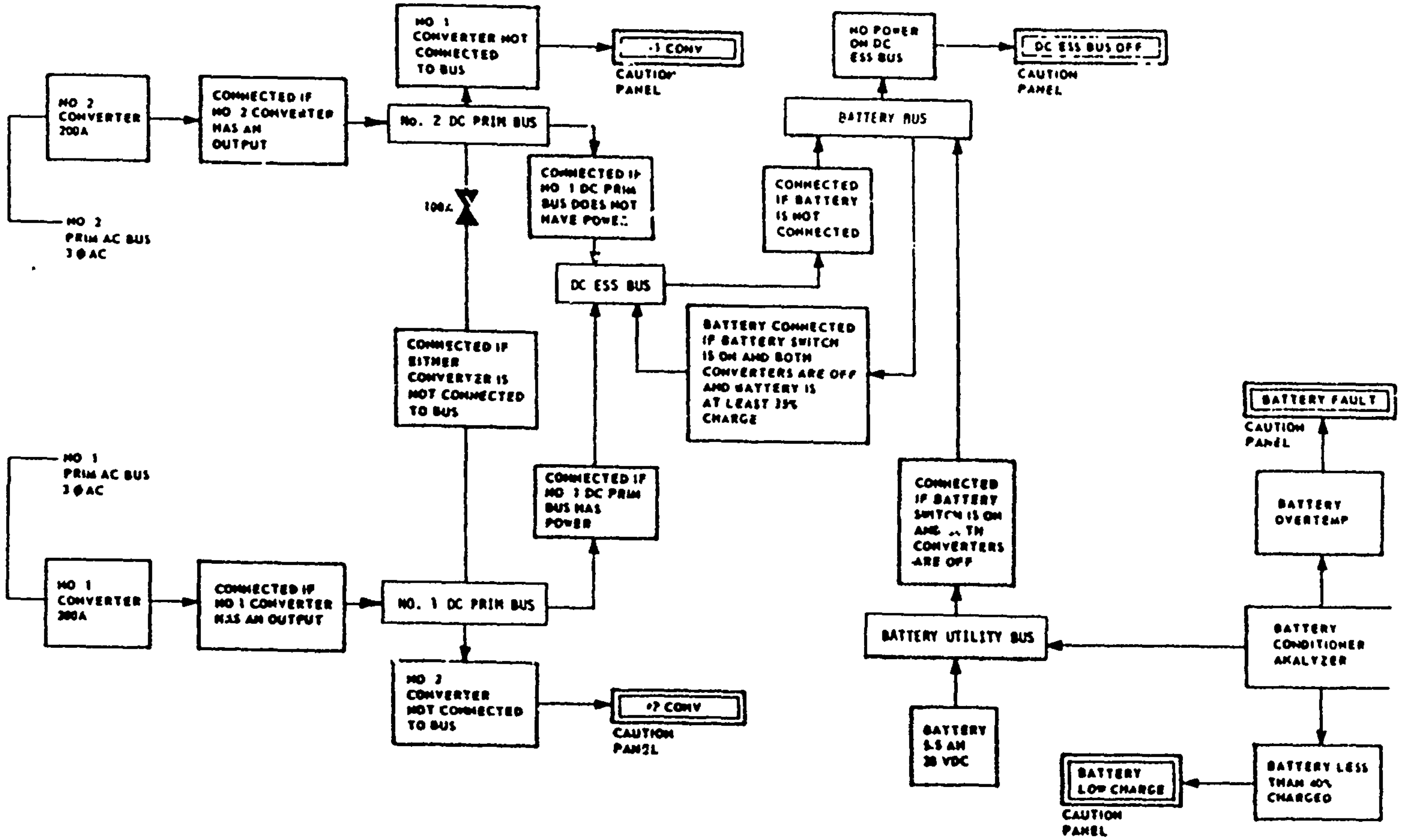

Figure 6. DC Power System Block Diagram.

Direct Current System

25. Two 200-ampere converters, each normally powered by the No. 1 and No. 2 AC primary buses, respectively, convert AC power into DC power and reduce it to 28 VDC. If one converter's output is lost, the load will be transferred to the operating system.

26. A charger/analyzer system restores the 24-VDC, 5.5 ampere-hour, 20-cell battery charge and determines the condition of the battery. The system charges the battery whenever AC power is available and the battery switch is ON. The charging current automatically reduces to a trickle charge when the battery becomes fully charged. The analyzer system monitors battery charge and lights a caution light when the charge lowers to 40 percent (±5) of battery capacity. If the battery charge continues to lower, at 35 percent (±5) of battery capacity the DC essential bus will be disconnected from the battery. At that point the battery can provide two APU starts. Another analyzer circuit monitors battery temperature. When the internal temperature reaches 70.1°C, a caution light will illuminate. When a battery fault occurs, the charger/analyzer should automatically disconnect the battery from the charging circuit. As a backup, the increasing temperature may be stopped by turning the battery switch off.

External Power

27. An external power connector on the right side of the helicopter accepts a ground source of AC power. The system is controlled by a switch on the upper console marked EXT PWR RESET OFF-ON. External power will be introduced into the system, if acceptable (correct frequency and voltage), and if no other generating source is operating, and the external power control switch is ON.

FIRE PROTECTION SYSTEM

28. The engines and APU are monitored by infrared (IR) radiation sensing units and protected by a high rate of discharge fire extinguishing system. The detection system consists of five IR sensing flame detectors, three control amplifiers, and a test panel. Two detectors are installed in each main engine compartment and one detector is in the APU compartment. In case of fire, the detectors react to the IR radiation and send a signal to one of the three control amplifiers, which in turn signals the fire warning assembly. A light in the proper "T" handle, as well as the master fire warning lights, will go on (fig. 7). There are three "T" handles, one for each main engine and one for the APU. Actuation of a "T" handle arms the fire extinguishing system and shuts off fuel. The extinguishing agent is discharged by activating a toggle switch located under the "T" handle. The detector system automatically resets itself when the detectors cease to sense radiation. A test switch on the fire warning panel sends a test signal through the system to illuminate the fire warning lights and verify proper system operation up to, but not including, the sensing units.

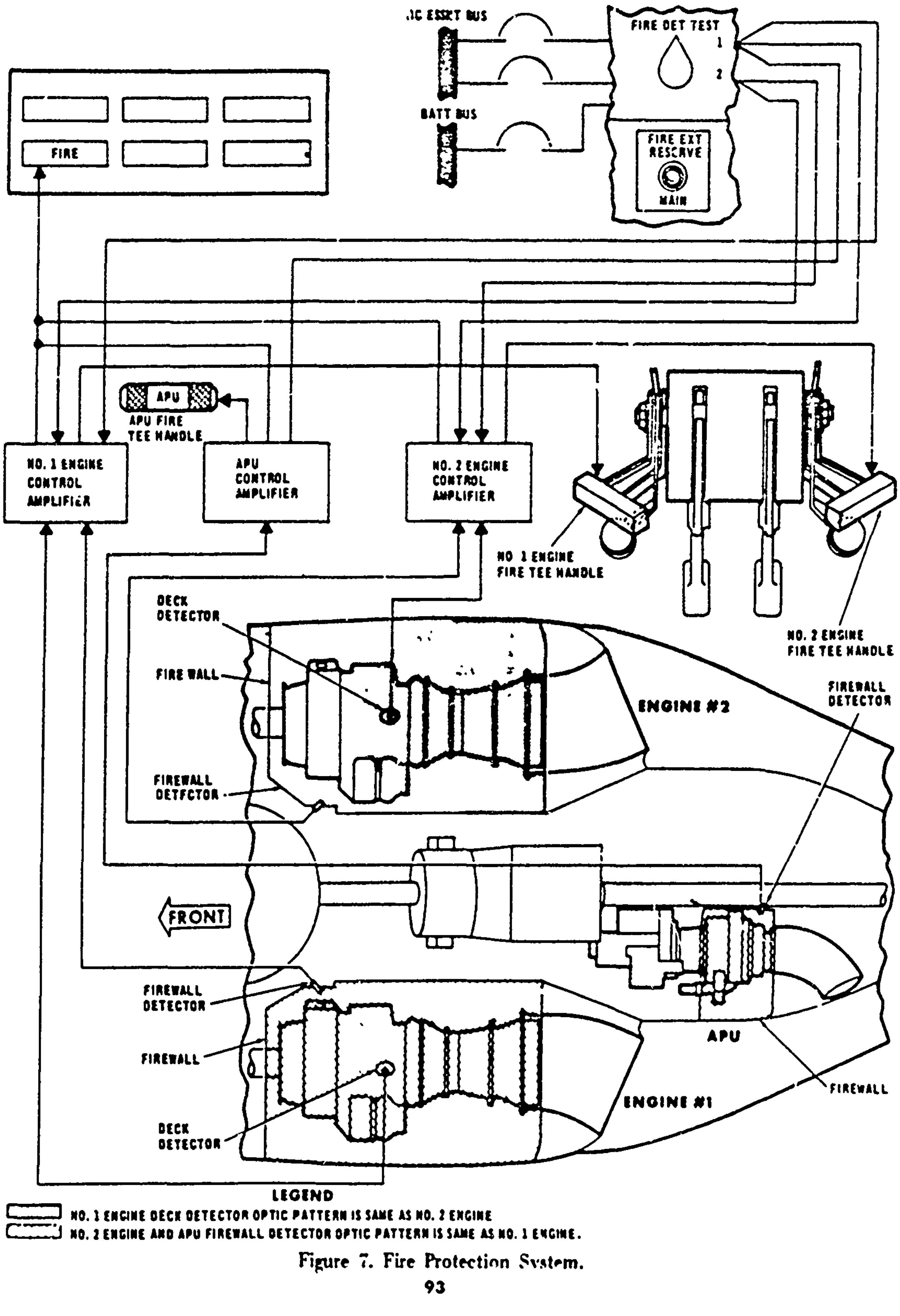

Figure 7. Fire Protection System.

29. The fire extinguishing system provides a main and reserve capability to either main engine compartment or APU compartment. The containers are mounted above the upper deck behind the right engine compartment. Both containers have dual outlets, each with its own firing mechanism (fig. 8). A crash-actuated system is incorporated into the fire extinguisher system which, upon an impact of 10g or more in any direction, automatically fires both extinguishers into both engine compartments.

PITOT-STATIC SYSTEM

30. The YUH-60A has a dual pitot-static system. The No. 1 system (roof-mounted just aft of the copilot station) provides ram pressure to the copilot airspeed indicator and the No. 2 system (roof-mounted just aft of the pilot station) provides ram pressure to the pilot airspeed indicator. The static sources for the two systems are interconnected. Each system has separate airspeed sensors which provide information to the FAS computer and the automatic stabilator control. Primary information for the stabilator control and the FAS computer is taken from the No. 2 system. Airspeed information for the stabilator is cross-checked with airspeed information from the No. 1 system and if a limit error is exceeded, the automatic stabilator control feature shuts itself down.

WEIGHT AND BALANCE

31. A typical loading is shown in table 2.

Table 2. Primary Mission Gross Weight Calcuation.

Item	Gross Weight (lb)
Empty weight (SOFR)	11,182
Trapped fluids (SOFR)	42
Crew	725
Weapons and ammunition	164
Operating weight, empty	12,113
Useful load, including fuel (maximum fuel load 2380 lb)	4740
Primary mission gross weight	16,853

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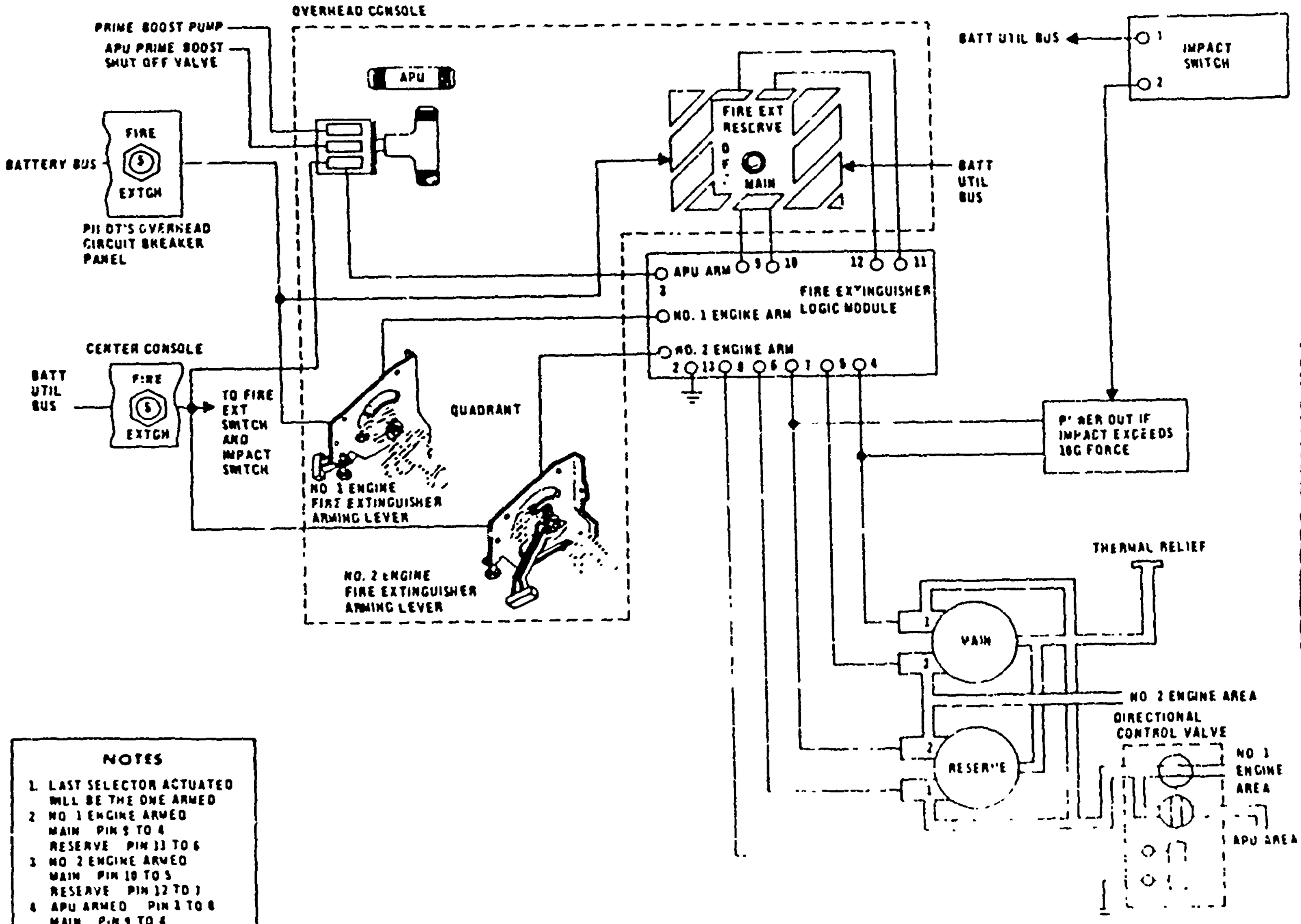

Figure 8. Fire Extinguisher System.

BASIC AIRCRAFT INFORMATION

32. Additional aircraft descriptive data are shown in the following listing.

Main Rotor

Diameter	53.67 ft
Chord	1.73 ft
Disc area	2262.3 ft^2
Number of blades	4
Solidity ratio	.0821
Disc loading (16,853-lb GCT design gross wt)	7.45 lb/ft^2
Blade twist	-18 deg
Airfoil section	SC 1095 (modified)
Flapping angle	-6 to +25 deg
Hub precone angle	7 deg
Blade prelag angle	7 deg
Design rotor speed (100 percent)	263 rpm
Design tip speed	739.1 ft/sec
Mass moment of inertia	7089 slug-ft^2
Blade tip sweep	20 deg

Tail Rotor

Diameter	11 ft
Chord	0.81 ft
Number of blades	4
Solidity ratio	0.188
Blade twist	-18 deg
Airfoil	SC 1095
Design rotor speed (100 percent)	1214 rpm
Design tip speed	699.2 ft/sec
Cant angle	20 deg

Vertical Tail Pylon

Span	8.17 ft
Chord, average	56.9 in.
Area	38.74 ft^2
Leading edge sweep angle	42 deg
Airfoil section	NACA 0021 with 7 deg tab on lower portion

Horizontal Stabilator

Span	172 in.
Mean aerodynamic chord	37.67 in.
Area	45 ft^2
Trailing edge incidence angle	-8 to +34 deg
Airfoil section	NACA 0012

Landing Gear Width

(No load)	108.5 in.

Weights

Empty weight (GCT)	11,185 lb
Primary mission gross weight (GCT SOFR)	16,790 lb
Primary mission gross weight (calculated from GCT)	16,853 lb
Alternate gross weight	19,930 lb

Engines (two YT700-GE-700)

Intermediate shp at standard, sea level (30-minute limit)	1536 shp/engine
Maximum continuous shp	1250 shp/engine
Design shaft speed	20,000 rpm
Speed reduction gearboxes:	
Engine input	20,000 rpm
Reduction gearbox output	5867 rpm

Main Transmission (Dual-Engine)

Shaft horsepower rating (10 sec)	3768
Shaft horsepower rating (2 min)	3210
Shaft horsepower rating (continuous)	2791
Design input shaft speed (100%)	5867 rpm
Design mast speed	263 rpm

Main Transmission (Single-Engine)

Shaft horsepower rating (10 sec)	1887
Shaft horsepower rating (2 min)	1605
Shaft horsepower rating (continuous)	1543

Tail Rotor Shaft Horsepower Limits

Caution level	330 at 100% rotor speed
Maximum continuous	420 at 100% rotor speed
Transient (10 sec)	720 at 100% rotor speed

Flight Airspeeds

V_{NE} at sea level (design gross weight)	185 KCAS
Sideward flight	40 KTAS
Rearward flight	35 KTAS

Main Rotor Speeds

Design minimum:	
Power on	250 rpm
Power off	237 rpm
Design maximum:	
Power on	315 rpm
Power off	315 rpm
Design operational (100%)	263 rpm
Normal operational (98%)	258 rpm

Tail Rotor Speeds

Design minimum:	
Power on	1154 rpm
Power off	1094 rpm
Design maximum:	
Power on	1454 rpm
Power off	1454 rpm
Design operational	1214 rpm
Normal operational	1191 rpm

APPENDIX C. FLIGHT CONTROL DESCRIPTION

GENERAL

1. The Sikorsky YUH-60A UTTAS utilizes conventional helicopter cyclic, collective, and directional controls powered by a triply redundant 3050-psi hydraulic system. Control inputs are transferred from the cockpit to the rotor blades by mechanical linkages and hydraulic servos. Outputs from the cockpit controls are carried by mechanical linkage through the pilot-assist servos/actuators to a mixing unit. The mixing unit combines, sums, and couples the cyclic, collective, and yaw inputs and provides proportional output signals to the main and tail rotor controls. There is an AFCS comprised of five basic subsystems: SAS, FAS/TRM, FPS, stabilator, and pitch bias actuator. The hydrofluidic SAS provides short-term rate damping in the pitch, roll, and yaw axes. An integrated FAS/TRM system provides directional and cyclic control trim reference and control force functions. The FPS provides attitude hold functions for the pitch, roll, and yaw axes and incorporates an airspeed hold function. The stabilator system is electromechanically operated and is designed to improve handling qualities throughout the flight envelope. A pitch bias actuator is incorporated to improve static and dynamic stability in the longitudinal axis.

HYDRAULIC SYSTEM

General

2. The YUH-60A has two separate hydraulic systems: first stage and second stage, and incorporates a third hydraulic pump/reservoir capable of powering either the first- or second-stage systems if required. The first- and second-stage pump modules are driven by the main gearbox and supply pressure to the flight control servos whenever the main rotor is turning. The first-stage pump module supplies 3050 psi to the first stage of the three main rotor (primary) servos and to the first stage of the tail rotor servo. The second-stage pump module supplies 3050 psi to the second stage of the three main rotor (primary) servos, 3050 psi to the collective and yaw boost servos and the SAS actuators, and 600 psi to the FAS servo and the fluidic sensor controllers. The electrically operated backup hydraulic pump supplies emergency hydraulic power to the No. 1 and/or No. 2 hydraulic systems and to the second stage of the tail rotor servo at 3050 psi. This system can also supply hydraulic power to all servos during ground checkout and recharges the APU accumulator. Electrical power to drive the backup pump module motor is supplied by either of the two gearbox-driven generators, by the APU-driven generator, or by external AC power for ground operations. The electric motor driving the backup pump module is automatically activated by either a low-pressure sensing switch in the No. 1 and No. 2 pump modules, by the APU start accumulator switch, or by the manual switch in the cockpit. A simplified hydraulic system schematic is presented in figure 1.

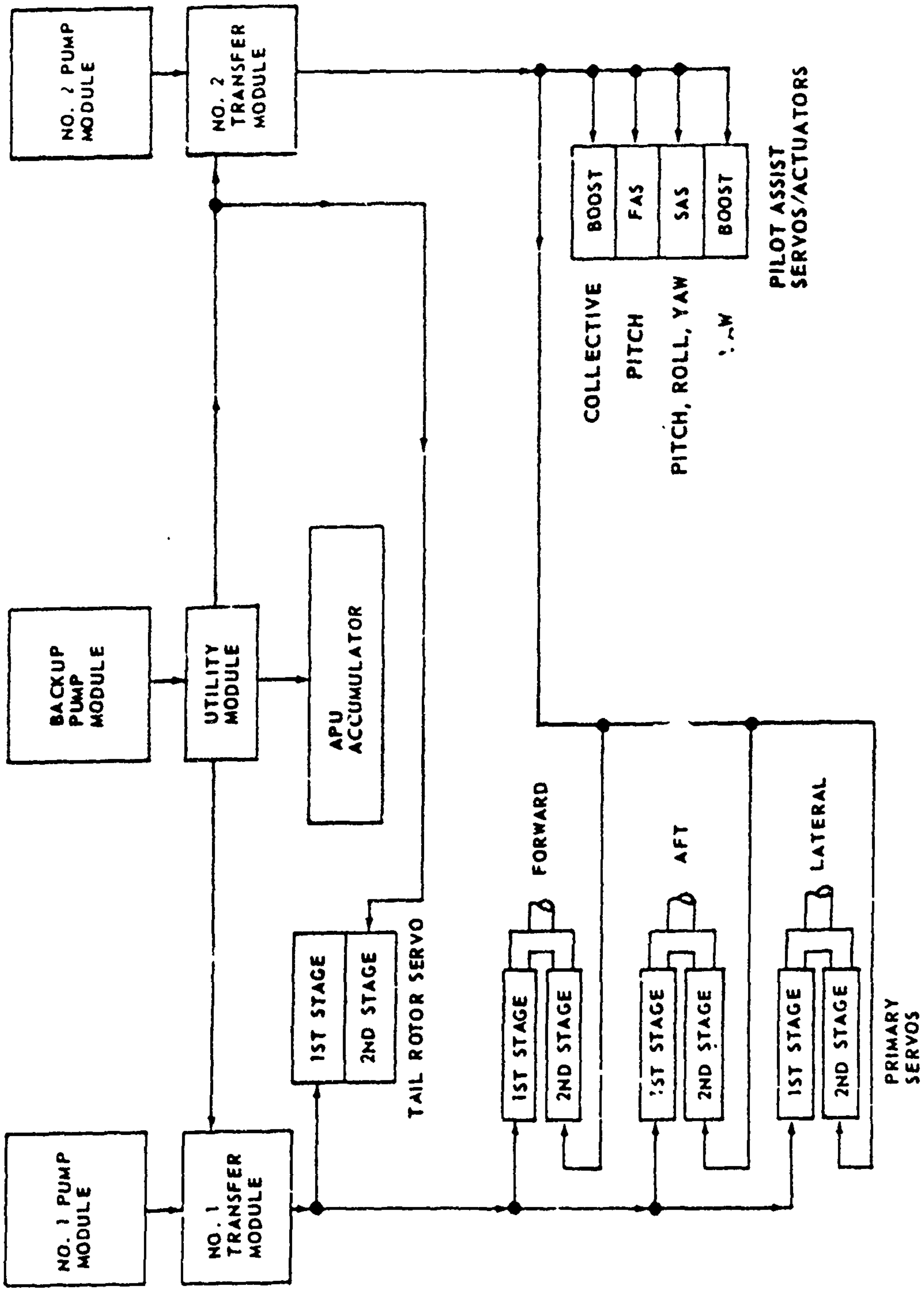

Figure 1. Hydraulic System Simple Block Diagram.

Pump Modules

3. The three hydraulic system pump modules each contain a pressure compensated variable delivery pump (rated at 3050 psi ±50), a spring-loaded reservoir, a visual fluid level indicator at the reservoir, pressure and return line filters with visual filter contamination indicators, and disposable filter elements common to both pressure and return filters. There is a check valve at the pump outlet to prevent reverse flow when the system is pressurized from an external source. The three self-contained hydraulic pumps, reservoirs, and component assemblies are identical and interchangeable. External hydraulic power can be connected at either of the three pump modules to pressurize the respective hydraulic systems.

Transfer Modules

4. The hydraulic system transfer modules are control devices that allow the hydraulic system to be pressurized from either of two independent power sources, while at the same time providing complete isolation between the two power sources. If a pressure differential greater than 1000 psi exists between the No. 1 system and the backup system, a hydraulically controlled transfer valve actuates to block the No. 1 system while simultaneously connecting the backup hydraulic system to the main servos. A servo shutoff valve at the pressure outlet of the transfer module is deenergized open to maintain servo operation in case of total loss of electrical power. An electrical interlock circuit assures pressure at the other stage of the main servos following a drop in pressure of 2000 psi ±50. This circuit is electrically connected in series with the shutoff valve in the other system. Loss of pressure in one system opens the contacts of that pressure switch, which interrupts power to the shutoff valve in the other servo system and prevents the other system from being deactivated. The No. 2 transfer module is similar to the No. 1 system. The primary and pilot-assist servos of the second-stage hydraulic system receive emergency power from the backup system through the No. 2 transfer module in the same manner as the No. 1 system.

FLIGHT CONTROL SYSTEM

General

5. The flight control system consists of longitudinal, lateral, collective, and directional control subsystems. Control inputs are transferred from the cockpit to the control surfaces by mechanical linkage and hydraulic servos. The pilot and copilot controls are dual but have separate paths to a combining linkage for each control axis. The output from the cockpit controls is carried by mechanical linkage through the pilot-assist and boost servos to the mixing unit. The mixing unit combines, sums, and couples the cyclic, collective, and directional control inputs. The tail rotor controls for the GCT configuration utilize a cable system back to the tail rotor servo. A flight controls schematic is presented in figure 2.

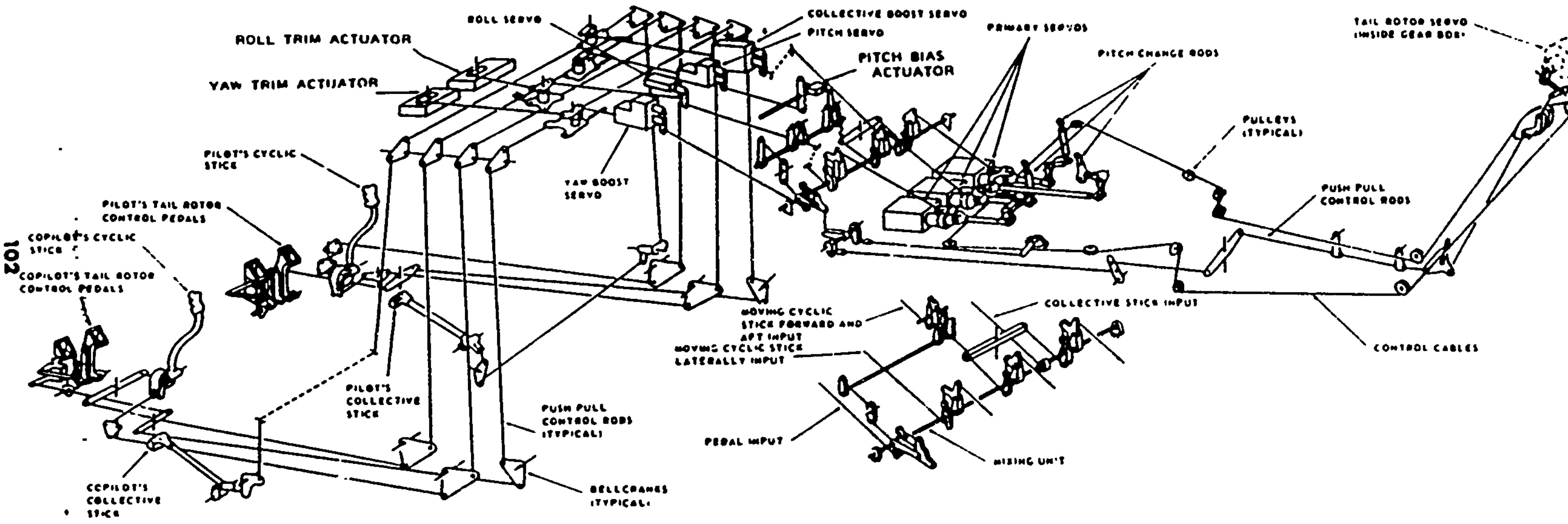

Figure 2. Flight Controls Schematic.

6. Helicopter flight control is achieved by action of hydraulically operated main rotor primary servos for the fore, aft, and lateral axes, and by a tail rotor servo in the yaw axis. Each of the three main rotor servos has two independent stages, with either stage having the capability to react to aerodynamic forces. The tail rotor servo incorporates a two-stage servo. Supplementary to the main rotor (primary) and tail rotor servos are boost servos for the collective and yaw channels which hydraulically boost control inputs from the cockpit controls to the mixing unit. The AFCS actuators are the FAS actuator and pitch bias actuator serving the pitch axis, electromechanical trim actuators serving the roll and yaw axes, and SAS actuators serving the pitch, roll, and yaw axes. The SAS, FAS, mixing unit, and servos are all above the cabin and forward of the main transmission. A flight control block diagram is presented in figure 3.

Main Rotor Primary Servos

7. The first- and second-stage primary servos are powered by the No. 1 and No. 2 hydraulic systems, respectively. The No. 1 hydraulic system powers the first stage of the tail rotor servo. The backup hydraulic pump powers the first- and second-stage primary servos in the event of failures in one or both of the main hydraulic power sources and powers the second-stage tail rotor servo if there is a pressure loss in the No. 1 hydraulic system. Each stage of the primary servos is capable of reacting to the maximum aerodynamic flight loads. Should a stage become inoperative, a bypass valve within the depressurized stage will open, preventing a hydraulic lock. All aerodynamic loads are then reacted by the other stage. If the input pilot valve to the servo becomes jammed, internal sleeves displace, which allows bypass to occur. Electrical interlocks powered by the No. 1 and No. 2 DC primary bus prevent both flight controls from being turned off simultaneously. The first- and second-stage servo shutoff controls are on the pilot and copilot collective control grips.

Boost Servos

8. The aircraft has identical boost servos acting in the yaw and collective control axes. The purpose of these boost servos is to aid the pilot by reducing control system friction forces. Each servo employs a piston and housing, rotary input, servo valve with jam override of 20 pounds of force, and a bypass valve to eliminate hydraulic lock with the servo depressurized and integral input/feedback linkage. Loss of boost servo pressure will result in increased directional and/or collective control forces.

Automatic Flight Control System Actuators

9. In addition to the boost servos, there are AFCS actuators and their controls that are designed to improve flying qualities and ease pilot workload. There are fluidic sensor controllers that provide an output proportional to aircraft angular rate. This output results in SAS actuator movement to provide short-term stability in the pitch, roll, and yaw axes. A FAS servo provides artificial force-feel proportional to FAS computer input signals and supplies stabilizing signals in series

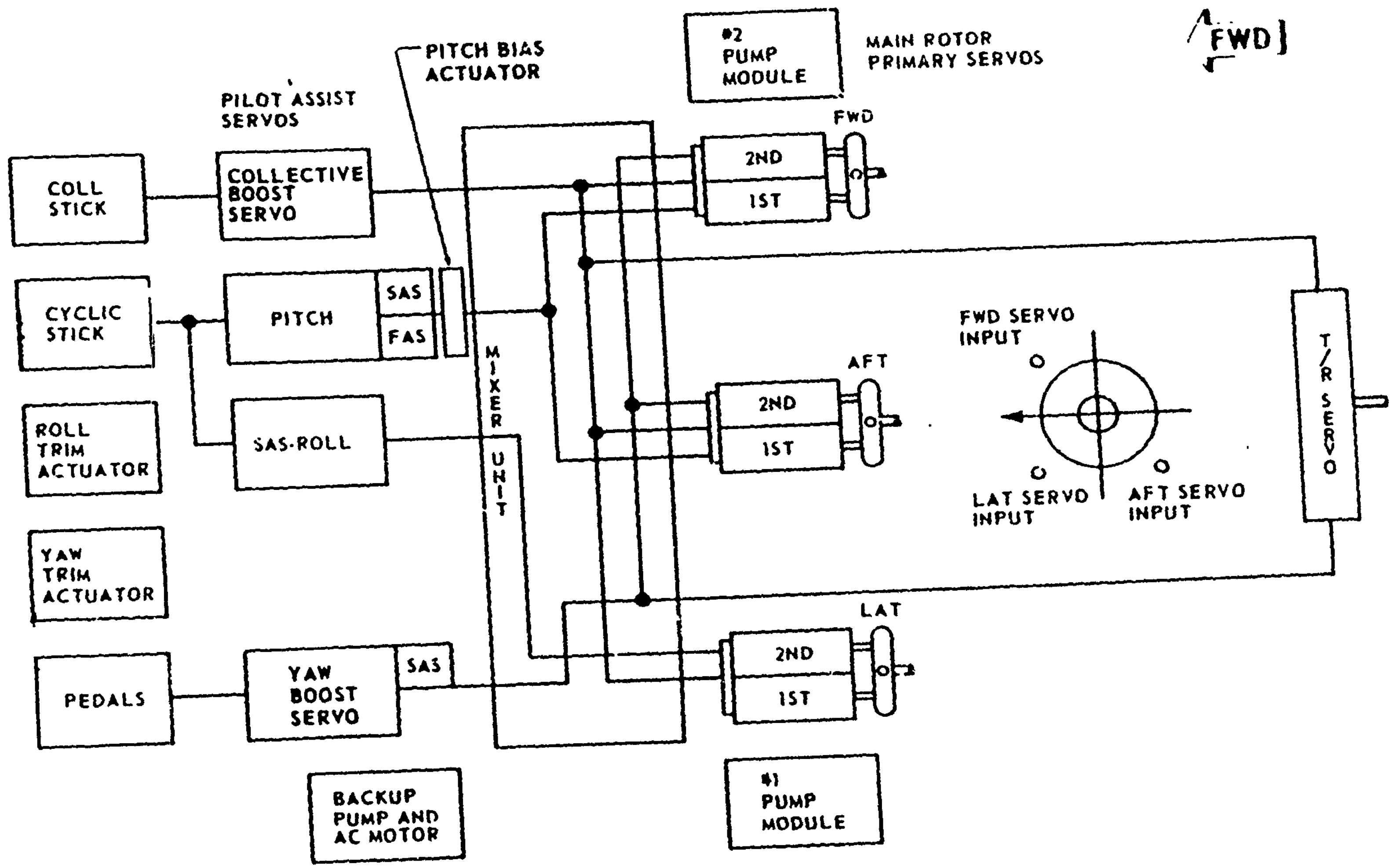

Figure 3. Flight Control Block Diagram.

with pilot inputs. Longitudinal static stability is improved through the action of a pitch bias actuator that works as a function of pitch attitude, rate, and airspeed signals.

Cyclic Control

10. Longitudinal and lateral control of the helicopter is accomplished by either of two cyclic controls, through pushrods, to the main rotor. Longitudinal inputs move the forward and aft primary servos, while lateral inputs move the lateral primary servo. The cycli. control grip (fig. 4) contains buttons and switches to control or operate the radio systems, intercom system, FAS trim and release systems, panel light system, and cargo hook release system.

Directional Control

11. Directional control is accomplished by either of two sets of directional control system pedals, through pushrods, to the yaw boost servo and to the mixing unit. Primary directional system control is then accomplished by pushrods to the tail rotor servo. The GCT configuration incorporated a dual tail rotor servo, with the first stage powered by the No. 1 hydraulic power system and the second stage by the backup hydraulic pump. Mechanical pitch to yaw coupling is effected at the mixing unit. The magnitude of mechanical coupling is presented in table 1. Electronic coupling of yaw due to collective is set at 2.2 degrees of tail rotor pitch increase per inch increase of collective from zero to 40 KIAS. This value decreases linearly to zero at 0.037 deg/kt/in. from 40 to 100 KIAS.

Table 1. Control Motion and Coupling.

Control Input[1] (in.)	Main Rotor Pitch Increase (deg)			Tail Rotor Pitch Increase (deg)
	Pitch	Roll	Yaw	Yaw
Pitch, 9.1	28.3	—	—	—
Roll, 10.2	—	16.0	—	—
Yaw,[2] 6.0	9.25	—	—	33
Collective,[3] 9.0	4.64	1.65	16.0	19.2

[1]Maximum control travel available.
[2]Left directional control input to produce indicated pitch due to directional control coupling.
[3]Up collective input to produce indicated pitch, roll, and yaw due to collective control coupling.

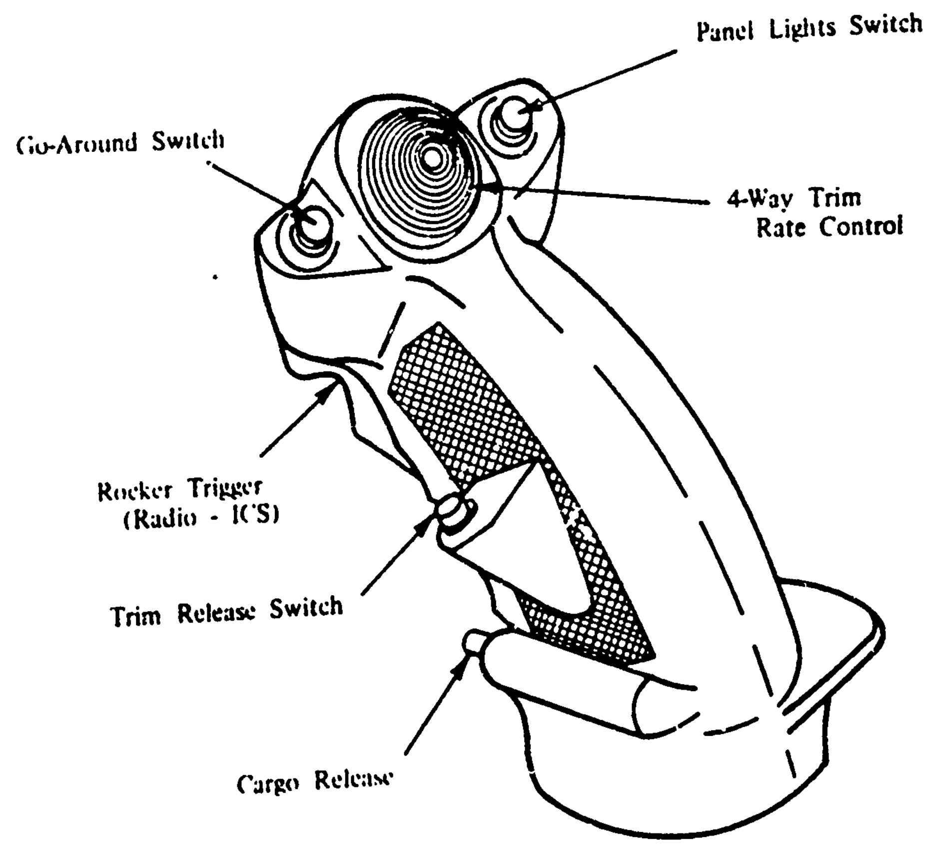

Figure 4. Cyclic Control.

Collective Control

12. Collective pitch control is accomplished by two cockpit pitch levers which change the pitch of the main rotor blades through control rods to the collective boost servo and then to the mixing unit by additional control rods. Collective mechanical coupling to yaw, pitch, and roll is effected in the mixing unit and the values are presented in table 1. A friction device on the pilot lever mechanically adjusts collective control friction. The copilot collective control telescopes to improve access to his seat. Each collective control has a grip with switches to control the following systems: landing light control, searchlight control, engine power turbine speed, first- and second-stage main rotor (primary) servo systems controls, and the cargo hook emergency release switch. Figure 5 presents a detailed drawing of the collective control grip.

AUTOMATIC FLIGHT CONTROL SYSTEM

General

13. The AFCS is an electrohydromechanical system designed to provide improved static and dynamic stability for the helicopter. The YUH-60A AFCS is composed of five subsystems: SAS, FAS, FPS, pitch bias actuator, and stabilator. Connections to the flight control system are accomplished through hydraulic actuators for SAS and FAS and through electromechanical trim actuators and the FAS servo for FPS. The electromechanical pitch bias actuator derives position information from the FAS computer. The variable angle incidence stabilator is controlled through dual electromechanical actuators and is scheduled as a function of information received from various sensors.

Stability Augmentation System

14. The SAS is a dynamic rate stabilization system designed to provide rate damping of the helicopter. The SAS incorporates an all-hydraulic system composed of flueric rate sensors, fluidic amplifiers and filters, servo valves, and actuators. The SAS modules are provided for the yaw, longitudinal, and lateral channels. All three SAS actuators receive their hydraulic power supply through the pilot-assist module of the hydraulic system. The SAS actuator consists of two major components: a fluidic controller and a servo valve that controls a spring-centered hydraulic piston. The fluidic sensor/controller produces a differential pressure proportional to helicopter attitude change rates to drive the servo valve. The servo valve causes a differential pressure across the actuator which will move the output in a direction that opposes the original aircraft direction of motion. There is no electrical power required for operation of the fluidic SAS controllers. The SAS actuators provide series inputs limited mechanically to ±10 percent of control travel. The SAS is normally ON whenever the rotor is turning and the hydraulic pumps are supplying pressure. The SAS may be disengaged by pulling the SAS ON circuit breaker located on the pilot DC circuit breaker panel. When this circuit

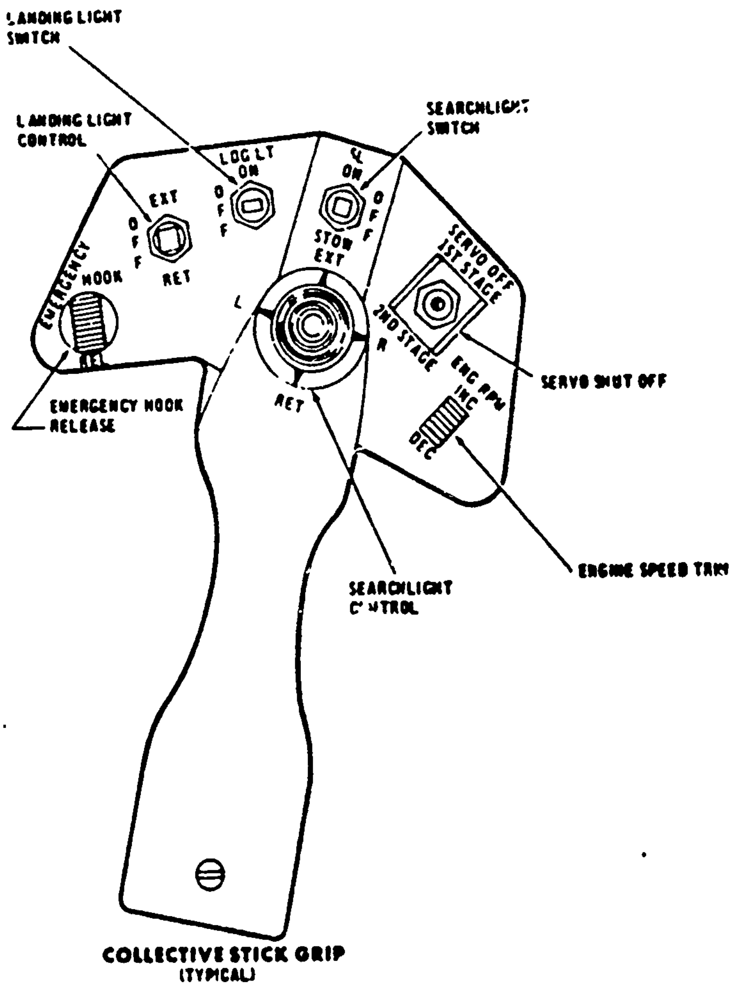

Figure 5. Collective Control Grip.

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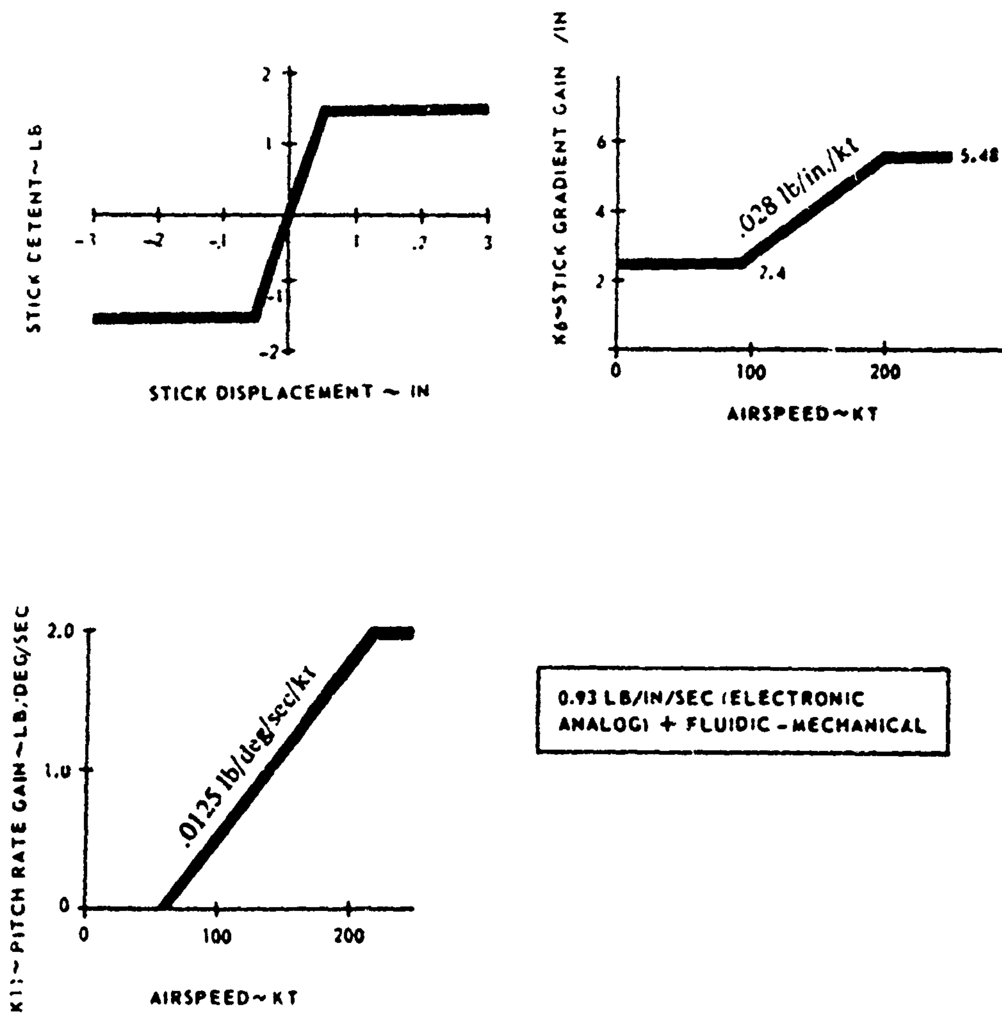
breaker is pulled, the SAS actuators are driven to their mid stroke position. If loss of electrical power is experienced with SAS OFF it will automatically come back on.

Force Augmentation System/Trim

15. The FAS provides a constant maneuvering force as a function of pitch rate and airspeed, longitudinal and directional control damping forces, and cyclic control forces from trim. The FAS consists of a computer, input sensors (trim rate controls, airspeed sensing, rate gyros, and control position sensors), FAS actuator, and the flight controls panel. When the airspeed signal is greater than 60 knots, input sensor signals are applied to the FAS computer, which computes pitch rate times airspeed to provide a designed maneuvering force of approximately 9 lb/g. Pitch rate is derived from rate gyros within the computer and an airspeed sensor signal is used within the computer to set gains. Cyclic control rate damping is designed to provide 0.93 lb/in./sec, which is an electronic analog function of the computer. Additional damping forces will be felt as a function of the fluidic and mechanical inertia of the flight control system. The control displacement forces are derived from position sensors and are a constant 2.4 lb/in. up to 90 KIAS, increasing to 5.48 lb/in. at 200 KIAS. The FAS provides trim hold for the longitudinal, lateral, and directional controls. The steady-state FAS forces in the longitudinal cyclic are effective for control displacements of 0.5 inch or more from trim. Figure 6 summarizes FAS longitudinal forces in graphic form.

16. The FAS computer program self-tests itself every 0.1 second and if requirements are not met will automatically shut itself down. The FAS actuator reverts to a damping mode when shut off to limit control overshoot following a system shutdown. The FAS servo authority is 100 percent, with override capabilities by the pilot at approximately 20 pounds maximum force The four-way trim switch located on the cyclic control grip is used as a control rate controller in the FAS mode of AFCS operation. The trim system rate (percent of control movement per second in the FAS mode and degree of attitude change per second in the FPS mode) is a function of thumb force on the trim button and is a different value between the longitudinal and lateral cyclic. Table 2 lists the trim rates for the FAS and FPS modes.

17. Engagement of the FAS/TRM button (fig. 12, para 25c) activates the pitch, roll, and yaw trim systems, which maintains cyclic and directional control position. The directional and lateral control forces are developed in the electromechanical yaw and roll trim actuators. The longitudinal force is developed by the electrohydromechanical FAS actuator in conjunction with the FAS computer. The longitudinal and lateral gradient and damping control forces are removed when engaging the trim release button on the cyclic grip (fig. 4) at airspeeds below 60 KIAS. Above 60 KIAS, the longitudinal damping force is retained when the trim release is engaged. The directional control damping function is independent of the FPS switch position. Directional control trim position and centering force is maintained whenever the FAS/TRM button is engaged and the pedal microswitches are not depressed. By placing the feet on the directional controls



FAS LONGITUDINAL FORCE $= q \times K11 + \Delta$ STICK $\times$ K6 ± 1.5

WHERE $q = \dfrac{1091\,(N_z - 1)}{V_{(KTS)}}$

Figure 6. FAS Longitudinal Force.

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Table 2. Cyclic Trim System Rates.

	Cyclic Movement in FAS Mode (percent/sec)		Attitude Change in FPS Mode (deg/sec)	
	A[1]	B[2]	A[1]	B[2]
Longitudinal	4.0	11.0	2.5	7.6
Lateral	2.5	7.5	4.6	14.0

[1]Trim system rate is constant for forces of 2.6 pounds or less.
[2]Trim system rate is constant for forces greater than 2.6 pounds.

and depressing one or more of the pedal microswitches, the trim position and centering force is removed; however, pedal damping is retained. The pedals may then be moved to the desired position and released. Electronic collective-to-yaw coupling is provided, with the greatest gain at zero to 40 KIAS, decreasing linearly to zero at 100 KIAS (para 11). Electronic collective-to-roll coupling is provided in the FAS/TRM mode, whose gain is 0.0912 degree left lateral pitch change per inch of up-collective control.

Flight Path Stabilization System

18. The FPS system works in conjunction with the FAS to provide pitch and roll attitude hold, airspeed hold, heading hold, and coordinated turn. The FPS system uses signals from the FAS sensor/computer to generate the required feedback control signal and is engaged by a switch on the flight controls panel. The desired pitch and roll attitude of the helicopter may be established in one of the following ways:

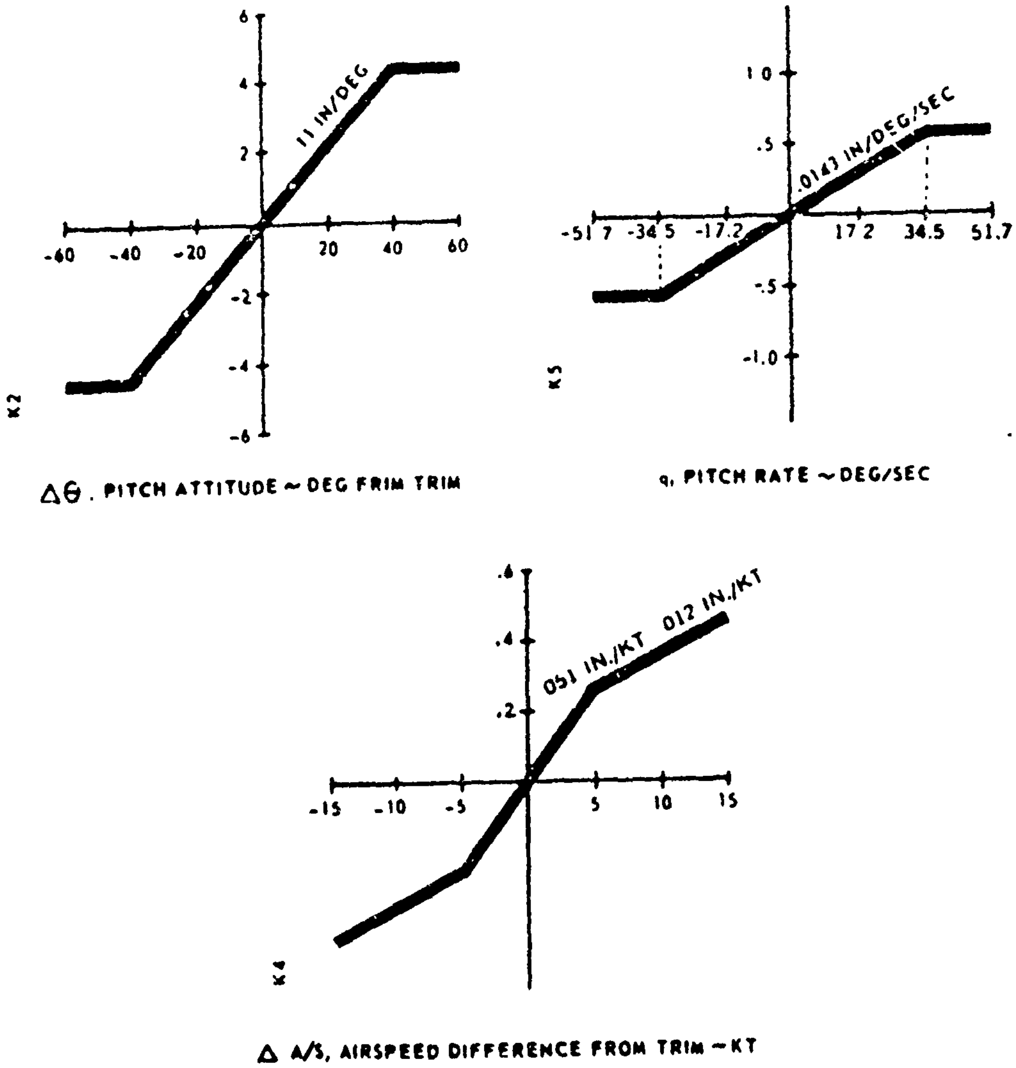
a. Beep the trim rate control to slew the reference to a desired attitude.

b. Depress the trim release button on the pilot/copilot cyclic grip, manually fly the helicopter to the desired trim condition, and release the trim release button.

c. Override the cyclic control forces to establish the desired trim condition and then neutralize the control forces by means of the trim rate control. The trim reference attitude, once established, is automatically maintained until changed by the pilot.

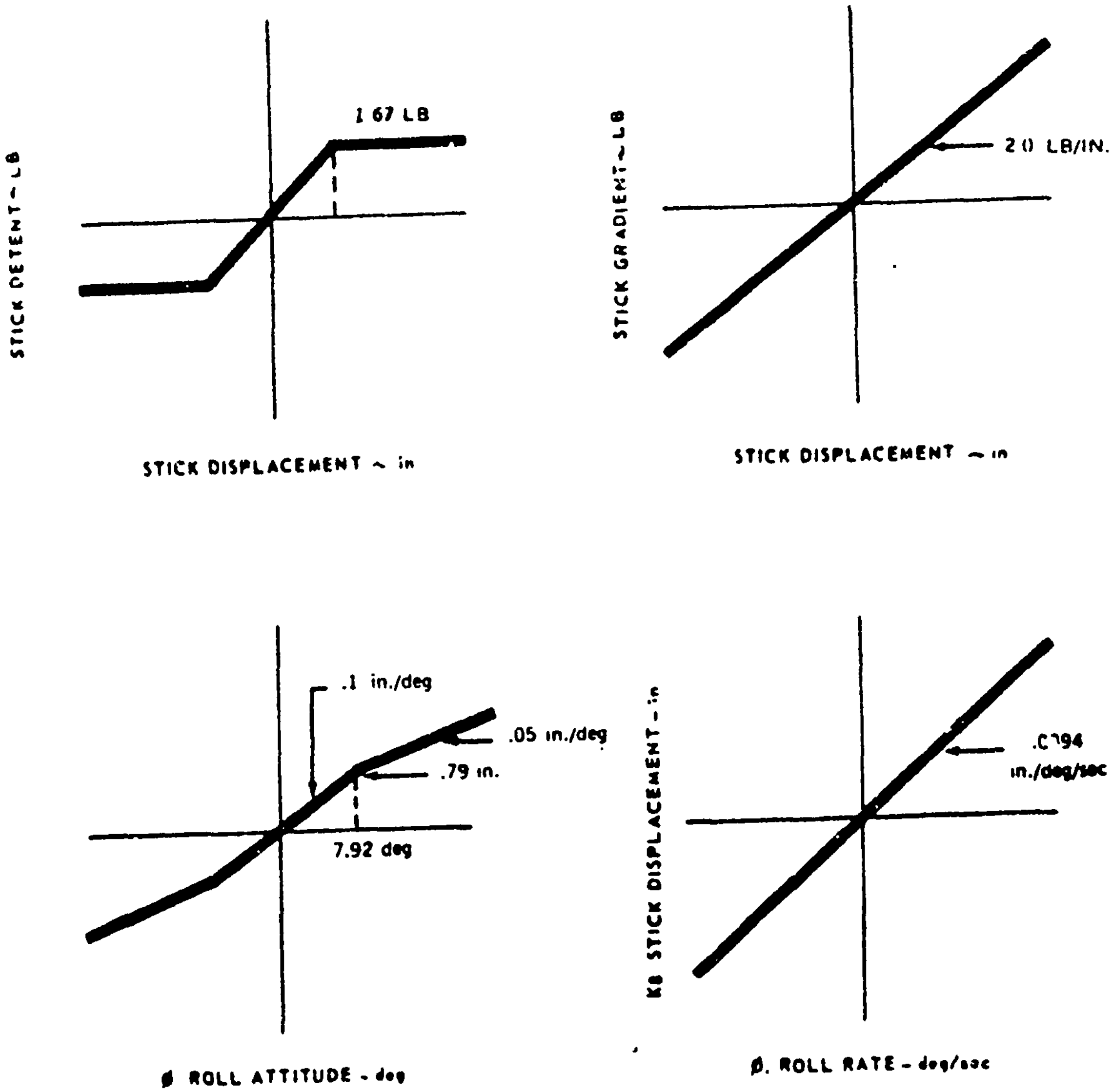
19. Pitch attitude and airspeed hold are accomplished through the FAS computer with outputs to the pitch bias actuator and the FAS actuator. At airspeeds greater than 60 KIAS, the pitch axis seeks to maintain the airspeed at which the trim is established by variation of pitch attitude. When pitch attitude is changed by means of the trim rate control or trim release, there is an 18-second delay from the time that the trim control input is removed until the new reference airspeed is acquired. This is designed to prevent premature engagement of the airspeed hold when effecting longitudinal trim changes. An electromechanical trim actuator provides roll attitude hold functions. A roll trim integrator is synchronized when in the FPS mode of AFCS operation and is designed to maintain zero bank angle throughout the flight envelope. Additionally, the roll trim integrator reduces the left lateral trim requirements as the aircraft is accelerated from a hover throughout the forward flight envelope. The design longitudinal and lateral control forces with FPS engaged are presented in figures 7 and 8, respectively.

20. The yaw axis of the FPS provides both heading hold and coordinated turns. Heading hold is accomplished by first maneuvering the helicopter with feet on the pedals and establishing the desired heading. The yaw attitude hold is then established by removing the feet from the directional controls. The compass attitude signal is compared with the yaw rate gyro signal to provide directional control



FPS LONGITUDINAL FORCE = $\left[\Delta \text{STK} + \Delta\Theta \times K2 + q \times K5 + \Delta \text{A/S} \times K4\right] K6\, f(\text{A/S}) \pm 1.5$

Figure 7. FPS Longitudinal Control Force.



FPS LATERAL FORCE = $\left[\Delta STK + K9 + K8 \times \dot{\phi}\right] 2.0 + 1.67$

FAS ONLY ENGAGED = $\Delta STK \times 1.2 + 1.67$

Figure 8. Lateral Control Force.

through the electromechanical yaw trim actuator. To change heading, the pilot activates one or both directional control microswitches, trims up on the desired heading, and removes feet from the directional controls.

21. The coordinated turn feature of the FPS is functional at airspeeds greater than 60 KIAS and is engaged in turns when the pilot's feet are on the directional controls. A lateral accelerometer senses sideslip and derived roll rate is utilized to effect pedal motion to maintain a centered ball at entry, during, and recovery from turning flight. If the pilot miscoordinates the helicopter either during turns or in straight and level flight, a force build-up proportional to sideslip will be sensed at the directional controls. The directional control forces with FPS engaged are summarized in figure 9.

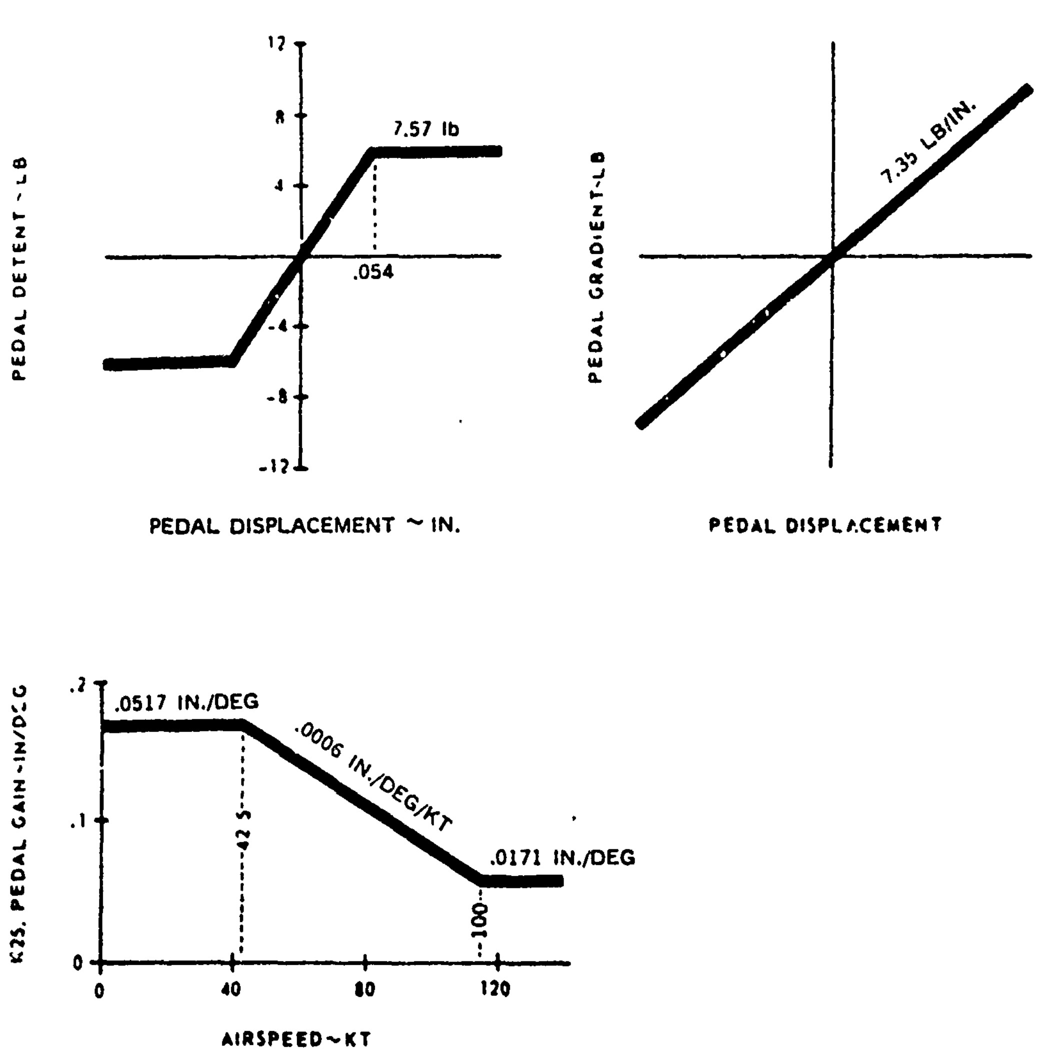
Pitch Bias Actuator

22. An electromechanical pitch bias actuator is built into the longitudinal control system and receives airspeed, pitch attitude, and pitch rate inputs from the FAS computer. These inputs are utilized to improve the static (slope of control position versus airspeed) and dynamic longitudinal stability. The position of this actuator is then monitored by the FAS computer. The pitch bias actuator is a series actuator that has a 15 percent control authority, which in the event of failure can mean a loss of up to 1.38 inches of total longitudinal control travel. Whenever the longitudinal bias actuator output differs from its commanded output by a predetermined threshold, the power is removed from the actuator and the PITCH BIAS FAIL caution light will go on. The actuator will maintain its last output by virtue of a jackscrew construction and will not recenter. Operation of the pitch bias actuator is independent of FAS/TRM or FPS ON-OFF select position.

Stabilator

23. The stabilator system on the YUH-60A helicopter is composed of an unswept, tapered 45-square-foot stabilator and associated control linkage. The stabilator is hinged about its aerodynamic center and is controlled by a dual electromechanical system. The stabilator schedule is a function of airspeed, collective position, pitch rate, and lateral acceleration. A schematic of the stabilator control system is presented in figure 10. The limits of travel are 34 degrees trailing edge down at zero forward airspeed and 100 percent collective to 10 degrees trailing edge up at zero collective and airspeeds in excess of 165 knots. Lateral cg acceleration and pitch rate are secondary input parameters used to desensitize the longitudinal axis to sideslip and improve dynamic stability. The stabilator may be operated in an automatic or manual mode and each mode is electronically independent. Figure 11 is representative of the YUH-60A stabilator control panel.

24. The stabilator automatic control system is dual throughout to provide fault detection and fail-safe shutdowns. Whenever the positions of the two actuators differ by more than 10 degrees of stabilator angle or 4 deg/sec rate of actuator travel, the system will automatically shut down. The stabilator may be reengaged anywhere in the flight envelope.



FPS ENGAGED

PEDAL FORCE $= (\Delta_{PED} + \Delta\psi \times K25\ (A/S))\ 7.35 + 7.57$

FAS ONLY ENGAGED

PEDAL FORCE $= \Delta_{PED} \times 7.35 + 7.57$

Figure 9. Directional Control Force.

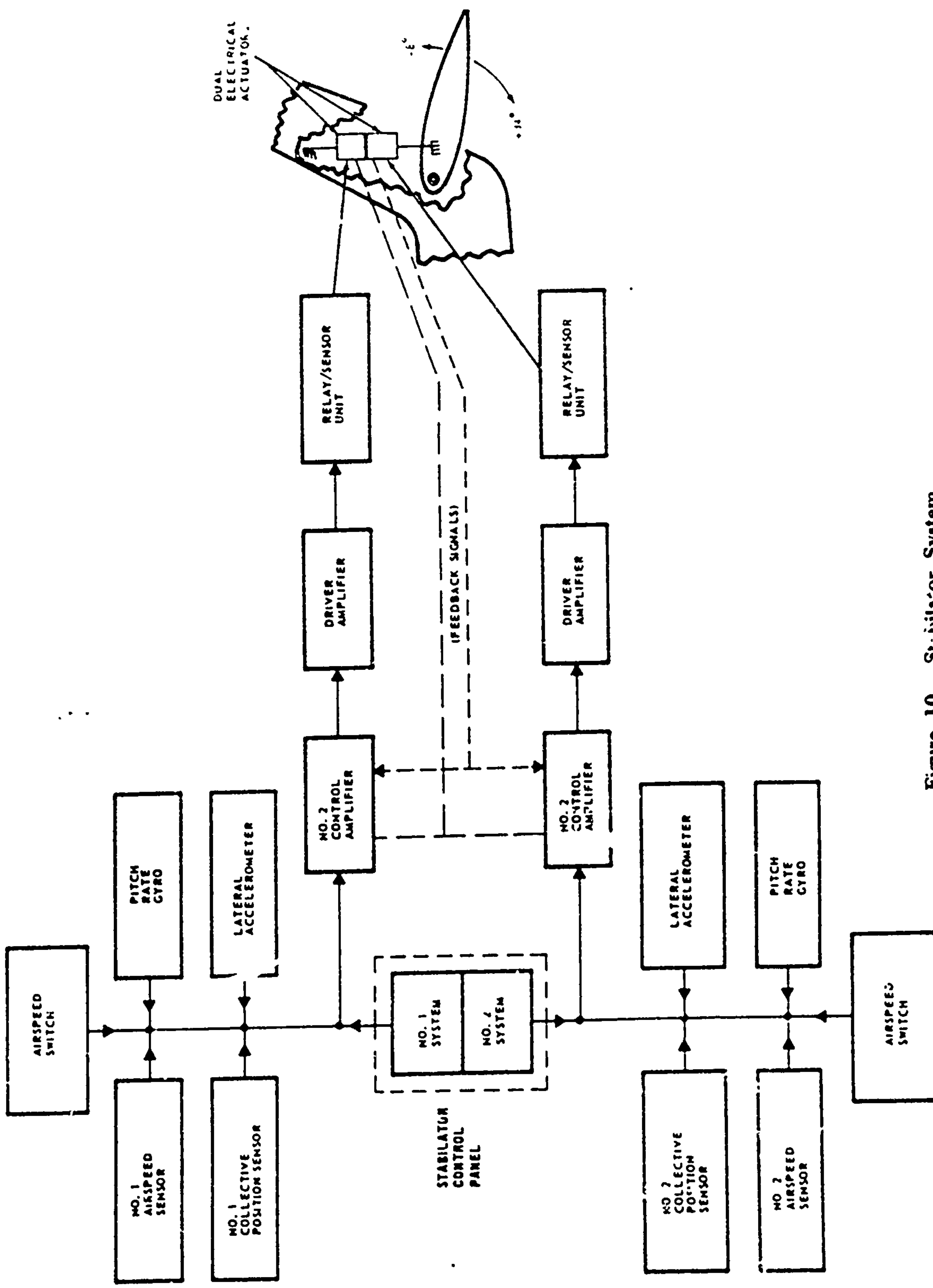


Figure 10. Stabilator System.

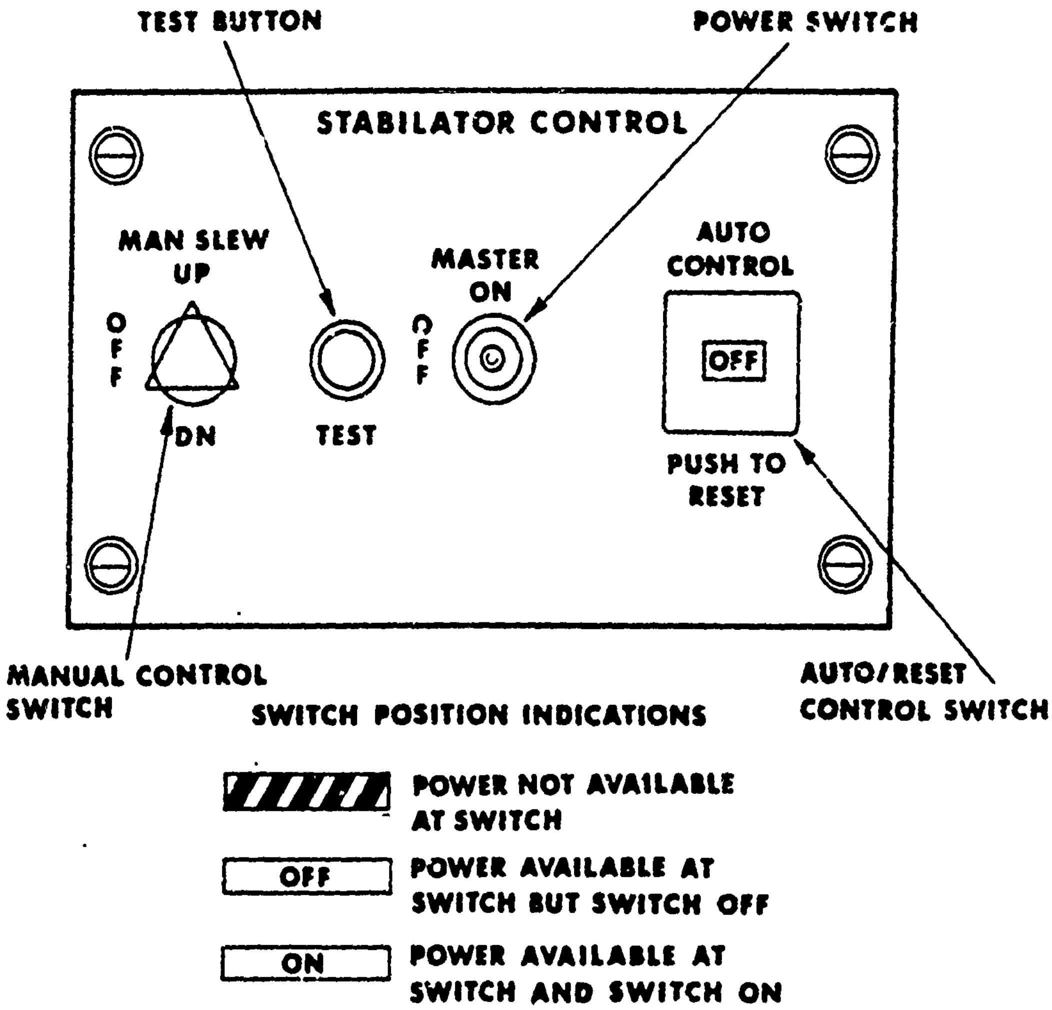


Figure 11. Stabilator Control Panel.

AUTOMATIC FLIGHT CONTROL SYSTEM CONTROL PANEL

25. The AFCS is designed to be flown with all systems operating for all mission tasks. The AFCS control panel consists of the following system-engage switches:

a. BOOST - This switch engages/disengages the collective and yaw boost servos.

b. FAS/TRM - This switch engages the FAS computer and trim hold functions of the cyclic and directional controls. The FAS must be engaged to have FPS engaged.

c. FLT PATH STAB - Activation of this switch provides the attitude, heading, and airspeed hold functions of the FPS. The proposed AFCS control panel is presented in figure 12.

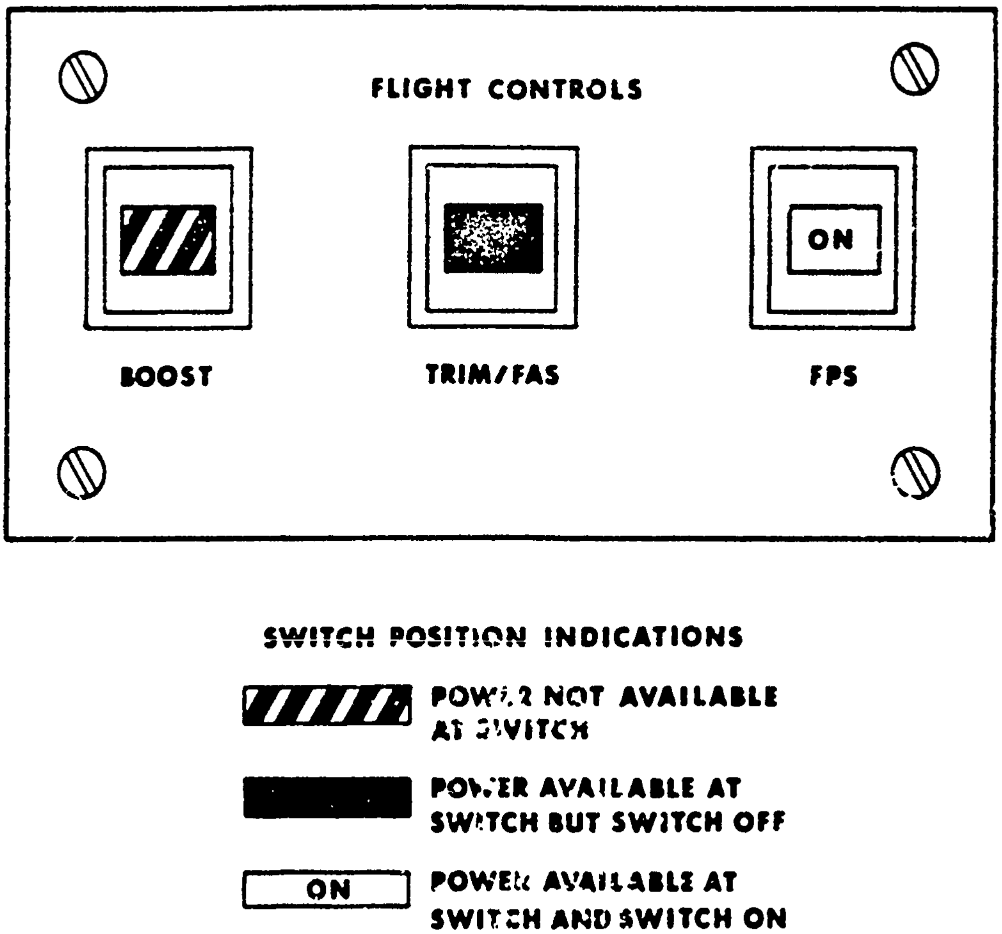


Figure 12. AFCS Control Panel.

APPENDIX D. ENGINE DESCRIPTION

GENERAL

1. The primary power plant for the YUH-60A helicopter is the General Electric YT700-GE-700 front drive turboshaft engine, rated at 1536 shp (sea level, standard day). The engines are mounted in nacelles on either side of the main transmission. The basic engine consists of four modules: a cold section, a hot section, a power turbine, and an accessory section. 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. In order to reduce sand and dust erosion and foreign object damage (FOD), an integral particle separator operates when the engine is running. The YT700-GE-700 engine also incorporates a history recorder which records total engine events. Pertinent engine data are shown below.

Model	YT700-GE-700
Type	Turboshaft
Rated power	1536 shp, sea-level, standard-day
Compressor	5 axial stages, 1 centrifugal stage
Variable geometry	Inlet guide vanes, stages 1 and 2 stator vanes
Combustion chamber	Single annular chamber with axial flow
Gas generator stages	2
Power turbine stages	2
Direction of rotation	Clockwise
Weight (dry)	400 lb
Length	47 in.
Maximum diameter	25 in.
Fuel	MIL-T-5624 JP-4 or JP-5
Lubricating oil	MIL-L-7808 or MIL-L-23699
Electrical power requirements for history recorder and Np overspeed protection	40W, 115VAC, 400 Hz
Electrical power requirements for anti-ice valve, fuel filter bypass indication, oil filter bypass indication, and magnetic chip detector	1 amp, 28VDC

Engine Modules

2. The engine consists of four separate modules, which are described in the following subparagraphs. Right and left side views of the engine are presented in figure 1.

a. The cold section module includes an inertial inlet particle separator incorporating an engine-driven blower mounted on the accessory gearbox. This module also includes the transonic six-stage compressor and the output shaft assembly which interfaces with the helicopter transmission shaft. The compressor has five axial stages and one centrifugal stage. The axial section is transonic, with variable inlet guide vanes and variable first- and second-stage stator vanes. Operation of the compressor variable geometry components is discussed in paragraph 10.

b. The hot section module contains an axial-type annular combustor. The combustor liner is cooled by air inpingement and air film. The two-stage gas generator turbine assembly is also included in the hot section module.

c. The power turbine module includes the power turbine, exhaust frame, the shaft, and sump assembly. The power turbine rotor has two stages with uncooled, shrouded tips. The power turbine shaft rotates inside of the gas generator rotor shaft and extends to the front of the engine. The power turbine shaft contains a torque sensor tube that mechanically displays the total twist of the shaft. A diagram of the engine torque system is shown in figure 2. 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 phase angle between the reference teeth on the two shafts is picked up by the torque/overspeed sensor. Power turbine speed (Np) is also picked up from these teeth by the Np sensor, which is mounted in the same location.

d. The accessory section module includes the top-mounted accessory drive gearbox and a number of line replaceable units (LRU's). An LRU is an item authorized to be removed and replaced with an interchangeable item. The LRU's mounted on the aft side of the accessory gearbox include the hydromechanical control unit (HMU), sequence valve, particle separator blower, and engine starter. The LRU's mounted on the forward side include the fuel filter, lube oil cooler, fuel boost pump, chip detector, lube and scavenge pump, bypass sensor, and alternator stator. The housing of the engine lube and scavenge pump and cored passages for oil and fuel are integrally cast into the gearbox, reducing external lines and fittings.

Engine Subsystems

3. The engine has four basic subsystems: lubrication, fuel, electrical, and air. Other subsystems of the engine include the variable geometry linkage assembly and internal washing system.

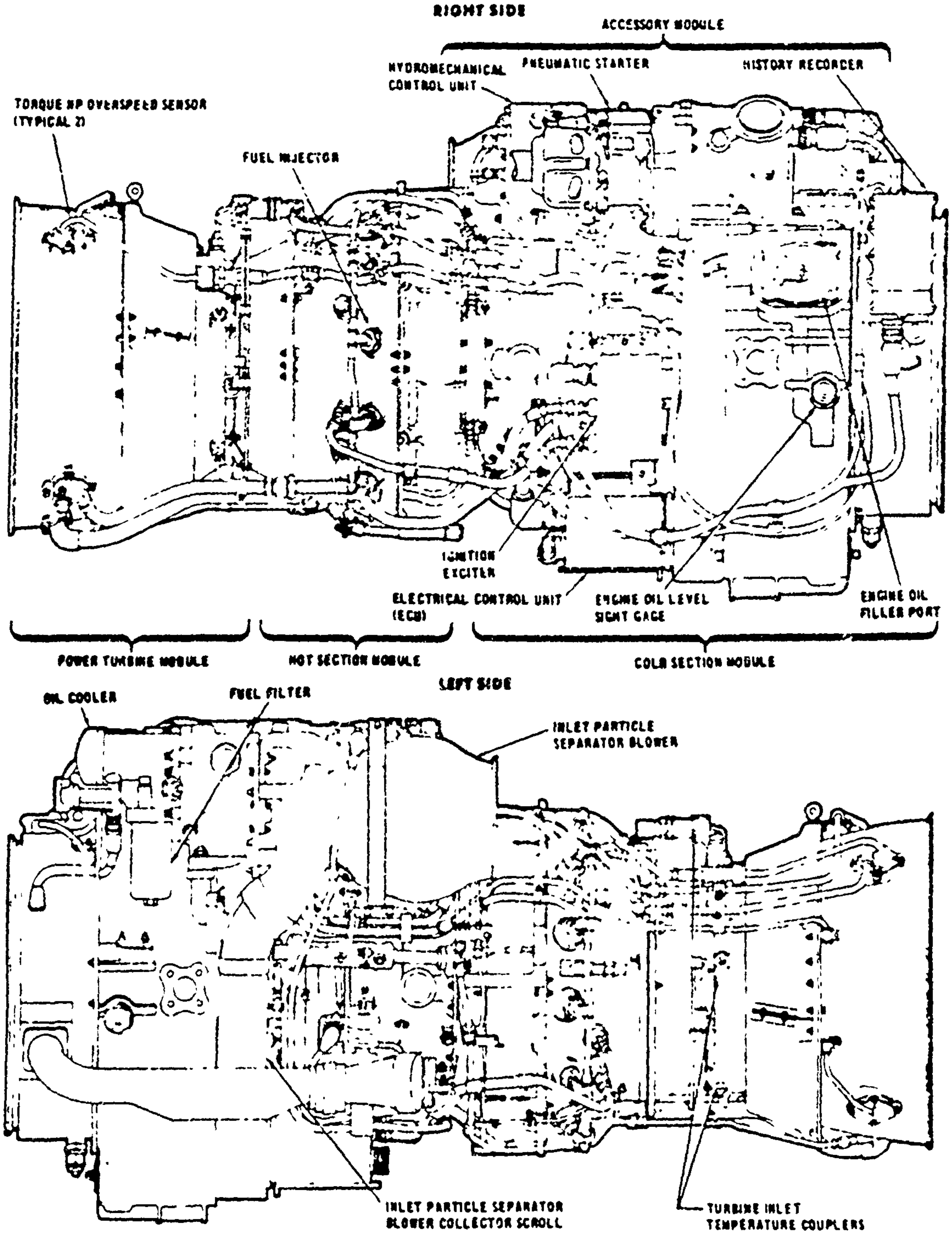


Figure 1. YT700-GE-700 Engine.

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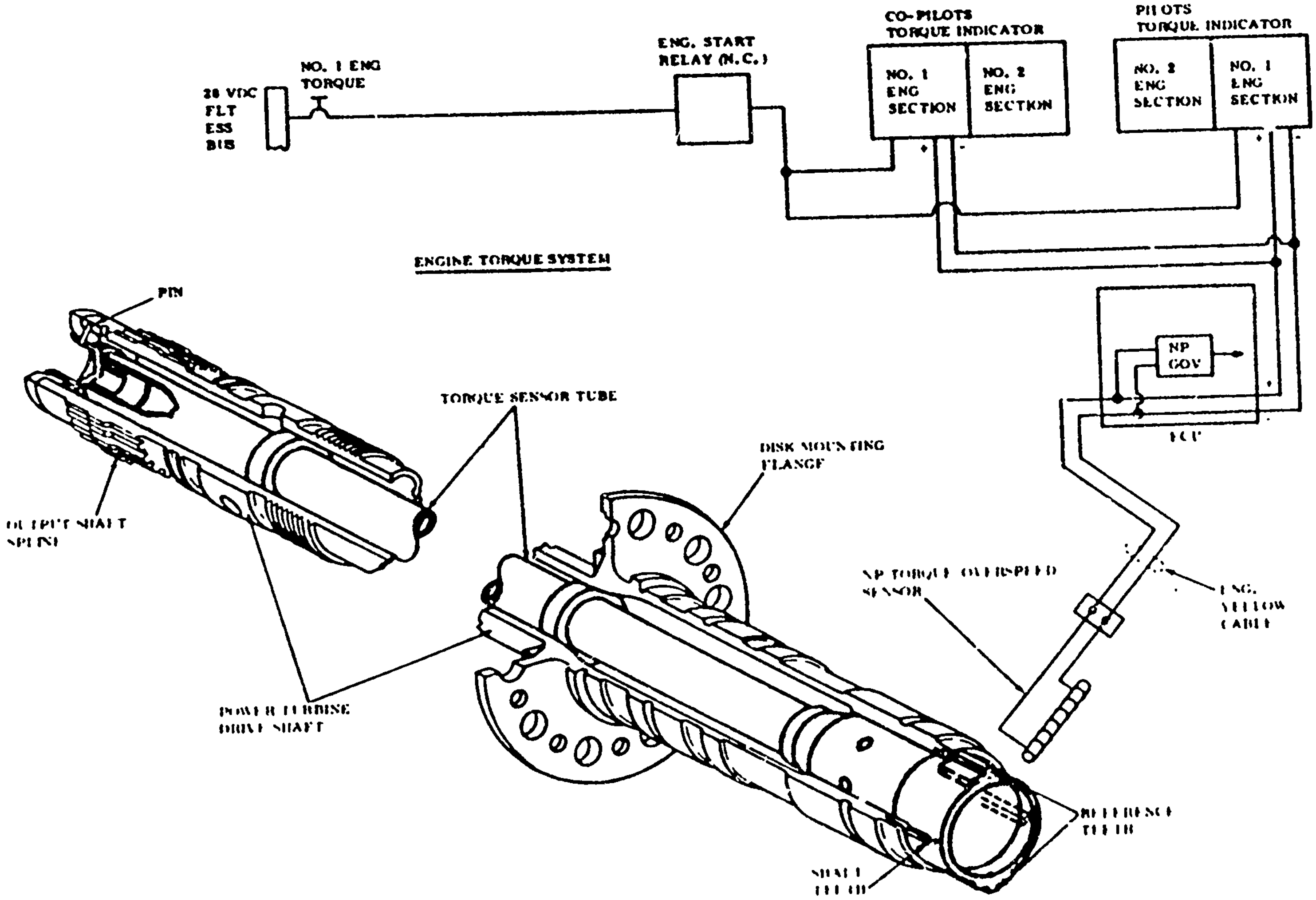

Figure 2. Engine Torque System.

Lubrication System:

4. The lubrication system is a self-contained, pressurized, recirculating dry sump system. The system consists of an integral oil tank, lube supply and scavenge pump, scavenge screens, oil filter, oil filter impending bypass and indicator, oil filter bypass valve and switch, chip detector, oil sampling port, sight gages, gravity-fill port with screen, oil cooler, pump cold-starting relief valve, emergency oil reservoirs, and sump distribution systems. The system is capable of supplying and scavenging oil, emergency bearing lubrication, and filtering and monitoring the condition of the oil. A schematic of the lubrication system and bearing and sump location is shown in figure 3. Oil from the supply tank is pumped through the filter and through cored passages in the accessory gearbox, where the flow divides to the A, B, and C sumps in the engine and the accessory drive gearbox. Scavenged oil flows through the scavenge screens, chip detector, the fuel-oil cooler, the engine inlet scroll vanes, and returns to the supply tank. An emergency system provides oil mist to lubricate the bearings if the primary oil system fails (fig. 4). Small integral oil reservoirs, located in each sump, are kept full during normal operation by the oil pump. Oil from these reservoirs passes through the oil mist nozzles to provide at least 6 minutes of lubrication.

Fuel System:

5. The fuel system consists of the engine-driven boost pump, filter, HMU, sequence valve, primer and main fuel manifolds, primer nozzles, and main fuel injectors. A schematic of the fuel system is presented in figure 5. Fuel from the tank passes through the engine-driven boost pump, a reusable filter, and the HMU high-pressure pump. High-pressure fuel is diverted to the wash filter, which supplies finely filtered fuel for the HMU computing servos. The metering pressure regulator valve and the metering valve respond to a signal from the HMU to schedule the required amount of fuel to the engine. The fuel not diverted to the HMU or through the high-pressure bypass valve flows through a metering valve (controlled by the HMU), through a shutoff valve, a pressurizing valve, and the oil cooler to a sequence valve. The sequence valve has four functions. First, it schedules fuel to the primer nozzles and main fuel injectors for starting and engine operation. Second, it purges fuel from the primer nozzles by directing compressor discharge (P_3) air through them after primer nozzle shutoff; this prevents coking of the nozzles. Third, it drains fuel from the main fuel manifold on shutdown to prevent coking. Fourth, it has a bypass valve for power turbine overspeed protection. To accomplish the fourth function the sequence valve contains a solenoid valve which is actuated by a power turbine overspeed signal controlled by the ECU. Operation of the solenoid valve causes some of the fuel coming from the HMU to be diverted from the combustion section back to the inlet of the HMU. The fuel flow to the combustor is transiently reduced to a level which reduces power turbine speed to prevent destructive overspeed. Once power turbine speed is reduced below the ECU overspeed reference valve, the signal to the solenoid ceases and the engine control system governs power turbine speed normally. The overspeed system also contains a cockpit test function (fig. 6) which permits the system to be checked while the engine is running.

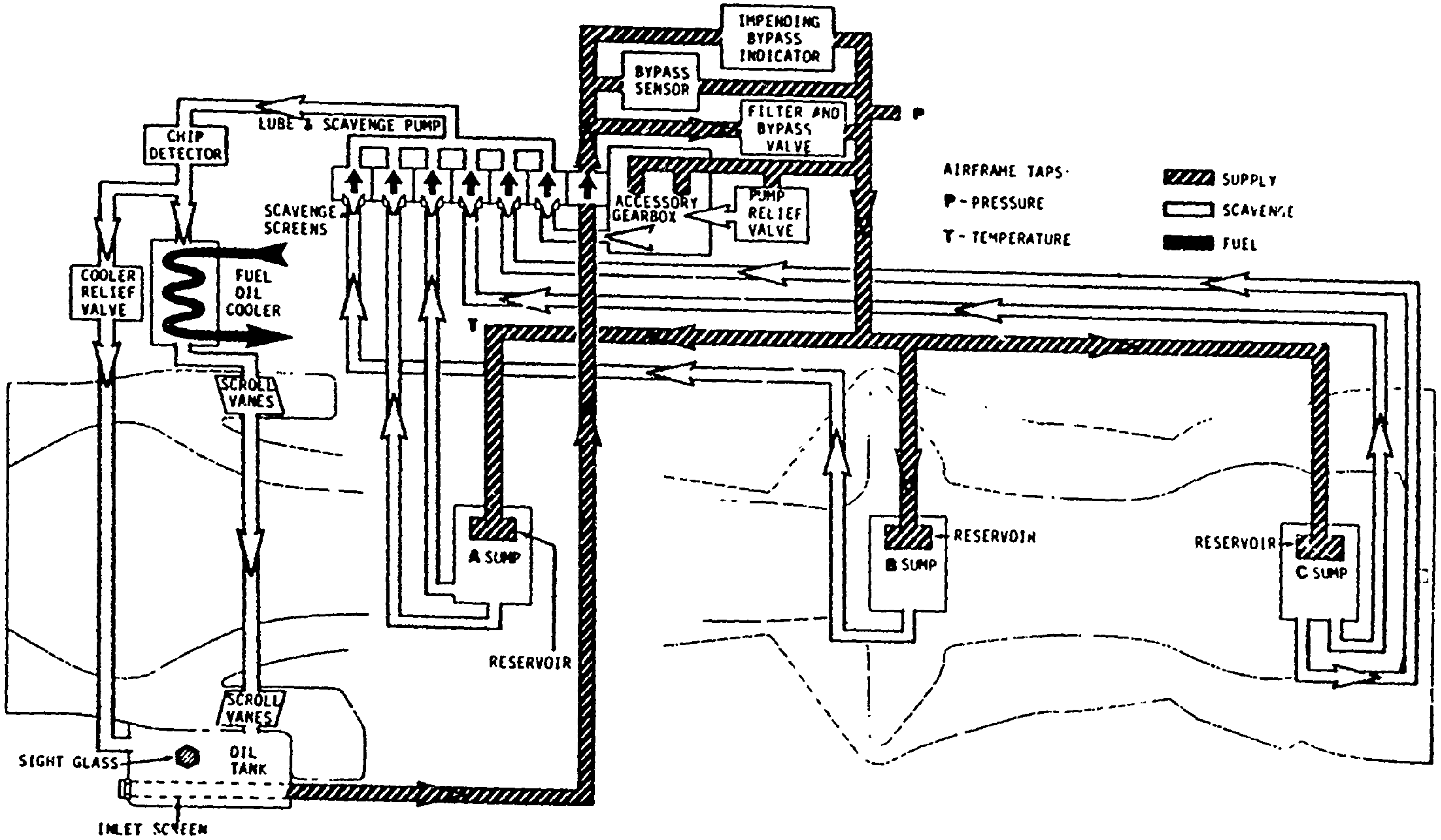

Figure 3. Oil System Schematic.

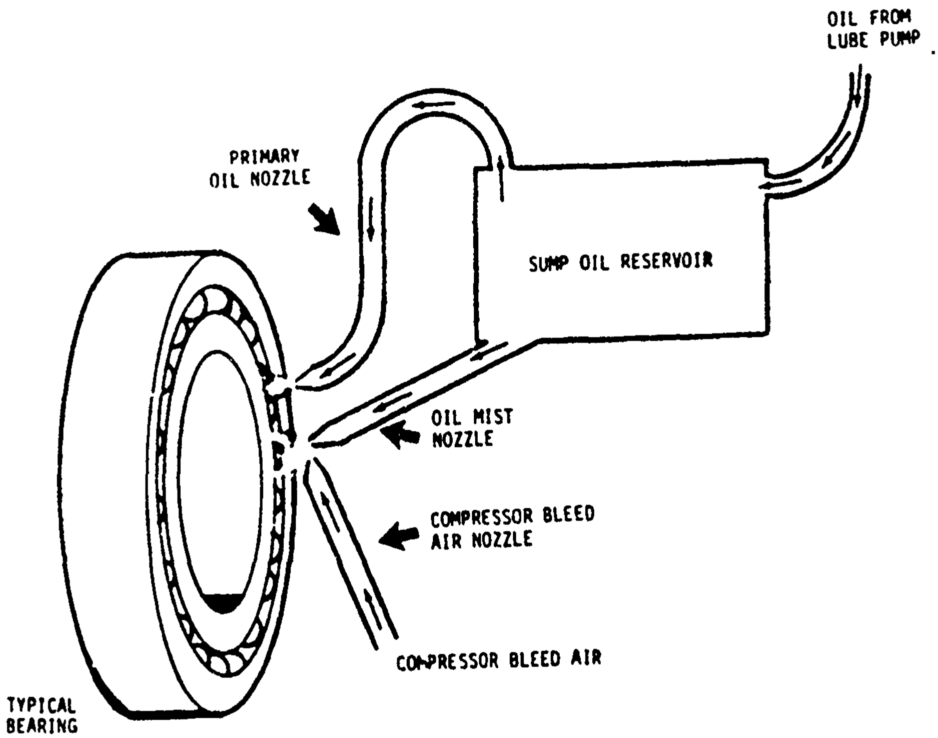

Figure 4. Emergency Oil System.

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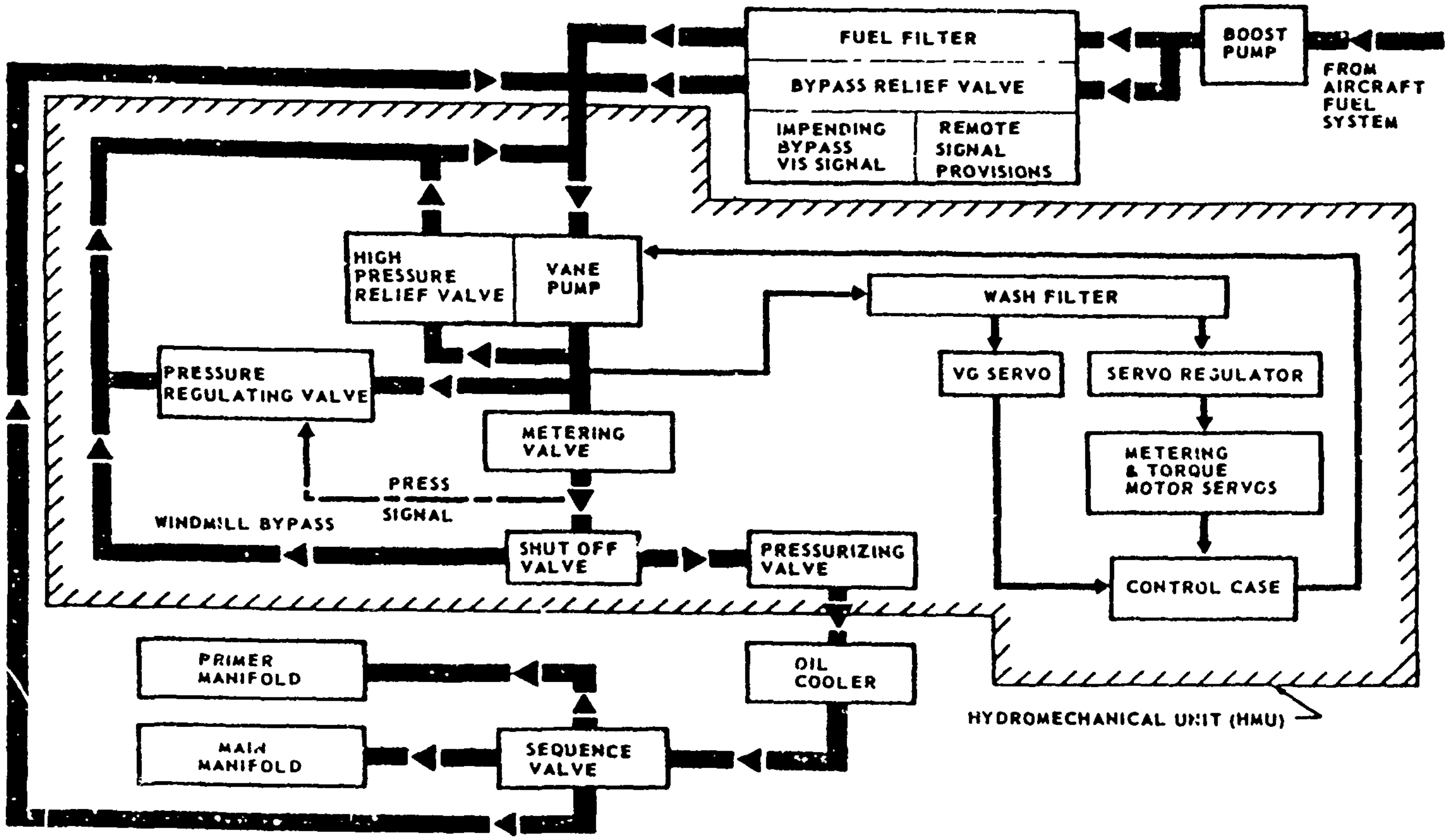

Figure 5. Fuel System Schematic.

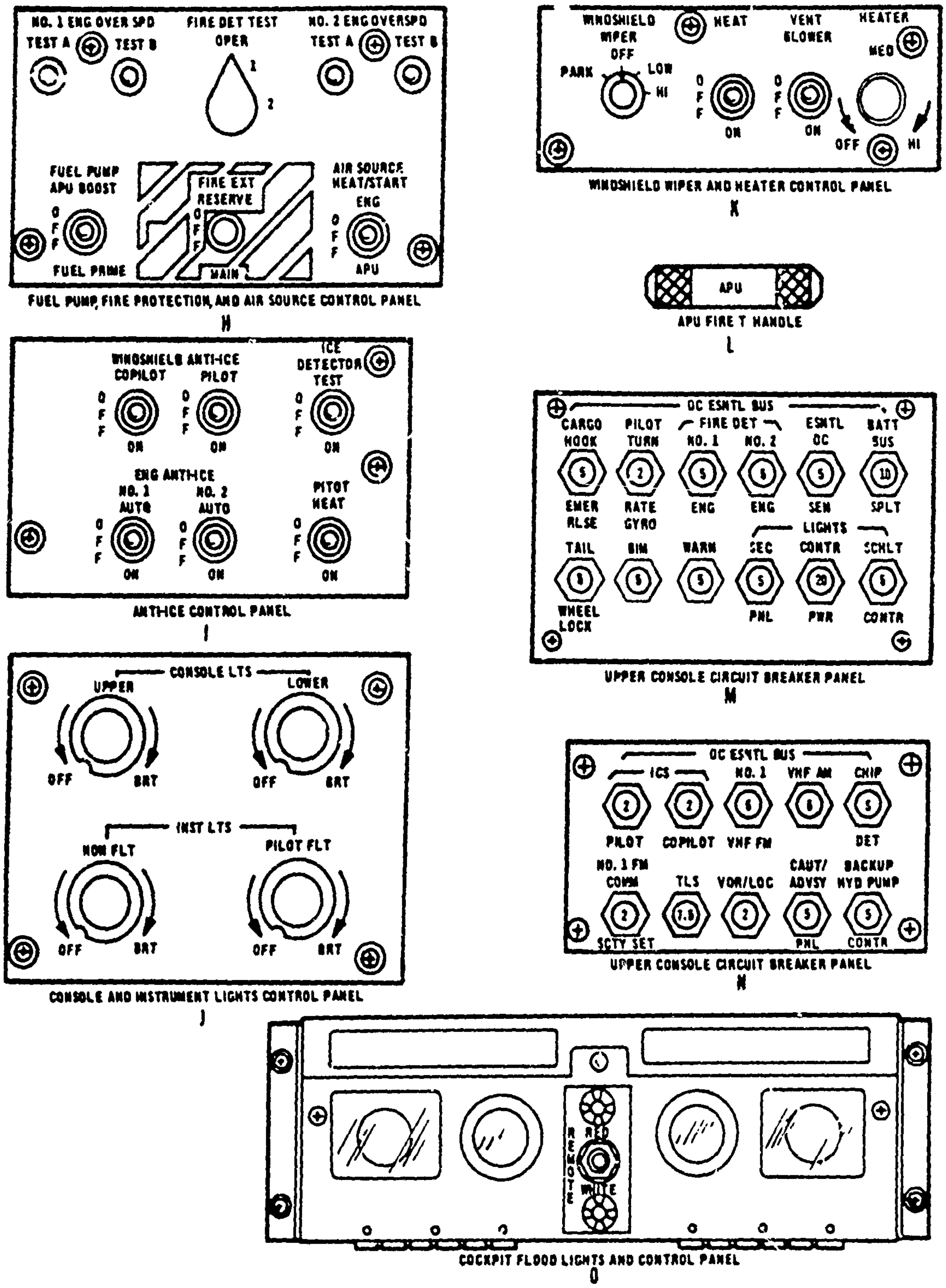

Figure 6. Fire Protection System.

6. Control of fuel to the combustion system is accomplished by the HMU. The HMU contains a vane-type high-pressure pump and a variable geometry actuator. The HMU receives two separate linkage inputs from the cockpit. One input is provided by a linkage connected from the cockpit to the power-available spindle (PAS) on the HMU. A third input is an electrical signal from the Np demand circuit, which is cockpit-mounted. This source provides a signal to the ECU. The ECU, in turn, provides a signal to the HMU which provides for constant Np over the full power range. The HMU also responds to inputs from compressor inlet temperature (T_2) and the compressor discharge pressure (P_3).

Electrical System:

7. The engine electrical system has five components: the ECU, alternator, ignition system, $T_{4.5}$ thermocouple harness, Np sensor, and torque/overspeed sensor. Their functions are described in the following subparagraphs.

a. The ECU provides engine control functions, conditioned signals for the engine history recorder and cockpit indications, and test points to a ground connector for electrical and engine system diagnostics. The following control functions are provided: constant Np governing to within ±1 percent of sensed Np; $T_{4.5}$ temperature limiting with an accuracy of ±5°C; Np overspeed protection completely independent of the normal Np governor; and load sharing on torque to within ±5 percent of intermediate rated power torque. It also provides the following noncontrol signals: Np to the cockpit; torque signal to the cockpit; $T_{4.5}$ signal to the cockpit; $T_{4.5}$ signal to the history recorder for overtemperature events and time-temperature integration; $T_{4.5}$ signal to ground units for engine diagnostics; and DC power and 400-Hz power to the history recorder. The ECU is air-cooled by air passing through the engine inlet scroll and is mounted on vibration isolators. The model ECU used for the majority of the GCT was a Group 7 ECU. Several flights were flown using Group 8 ECU's (para 106, Results and Discussion). The Group 8 ECU's were designed to correct the problems listed below.

(1) Slow governor response on $T_{4.5}$ transient overshoots resulting in longer than desired temperature excursions over 840°C.

(2) Insufficient stability margin for all hardware combinations: HMU, ECU, $T_{4.5}$ harness, and engine.

(3) Saturation of the Np governor, producing Np transient overshoot.

(4) Transient Np droop on rapid load increase from zero power.

(5) Load change unrelated to collective change caused Np excursions.

(6) The lower authority limit of the Np governor integrator had insufficient trim range to allow sufficient power reduction to maintain isochronous governing.

(7) The upper authority limit of the Np governor integrator had insufficient trim range to increase power for isochronous governing with single-engine operation.

(8) Insufficient stability margin in autorotation, resulting in limit cycle oscillation Np.

(9) As Np recovers from transient droop, it overshoots the desired speed.

(10) Following an Np error the system is too slow in returning to the desired speed.

(11) High-frequency noise in the torque signal, acting through load sharing, causes Np to run over reference.

b. The alternator is gearbox-mounted and has three separate windings. Winding No. 1 supplies the ignition exciter. Winding No. 2 supplies power to the ECU and to the primary Np overspeed circuit. Winding No. 3 supplies the N_G cockpit signal.

c. The ignition system is a noncontinuous AC-powered capacitor discharge system. It includes an ignition exciter, two igniter plugs, and two ignition leads. The ignition duty cycle is 2 minutes ON, 3 minutes OFF, 2 minutes ON, and 23 minutes OFF.

d. The $T_{4.5}$ thermocouple harness is a five-probe dual-immersion harness using chromel-alumel functions to provide signals to the ECU for $T_{4.5}$ limiting and cockpit indication. Each of the five probes is individually wired to a multipin connector, allowing diagnostic checks to be made for open or grounded elements.

e. The Np, Np overspeed, and torque sensing are provided by two identical variable reluctance pickups. The Np sensor provides a basic Np signal to the ECU. The torque and overspeed sensor senses power turbine torque and provides a speed signal to the separate Np overspeed protection system.

8. The electrical system components are engine-mounted and self-contained. The power sources and the components they power are as follows:

a. Alternator winding No. 1 - Ignition exciter.

b. Alternator winding No. 2 - Electrical control unit and primary Np overspeed circuit.

c. Alternator winding No. 3 - N_G cockpit signal.

d. Airframe 400-Hz, 115-VAC - History recorder and backup Np overspeed circuit.

e. Airframe 28-VDC - Anti-icing valve, oil filter bypass, fuel filters bypass, and magnetic chip detector.

Air System:

9. The air system performs the following functions: cools the turbine section and provides anti-icing air, seal pressurization, sump venting, airframe bleed air requirements, and compressor discharge pressure (P_3) signal to the HMU. These functions are described in the following subparagraphs.

a. Compressor discharge leakage air is used to cool the stage 1 and stage 2 nozzles. Air leaking from the centrifugal compressor at the diffuser is ducted through the mid frame and into the nozzle vanes. The air cools the vanes and exits through the holes in the vane airfoils.

b. Anti-icing is achieved by a combination of noncontinuous axial compressor discharge bleed air and continuous heat rejection from the air-oil cooler, which is an integral part of the scroll vanes. The compressor discharge bleed air anti-ices the swirl frame and swirl vanes. Control of anti-icing air is provided by a solenoid-operated anti-icing valve which is actuated by a cockpit switch. The switch is fail-safe, in that when electrical power is supplied to the anti-ice valve solenoid, the anti-icing air is turned off. When power is off the valve is open.

c. Seal pressurization prevents oil loss from sumps by controlled air flow. It prevents hot gases, dust, and moisture from entering sumps and provides air for the emergency oil system.

Variable Geometry Linkage Assembly:

10. The compressor variable geometry components consist of a fuel-driven actuator integral with the HMU; a crankshaft with the necessary links to attach to the actuating rings of the inlet guide vanes; first- and second-stage variable vanes; and the anti-icing and start bleed valve. Rotation of the crankshaft by the HMU actuator translates to circumferential movement of the actuator rings, which results in synchronized opening or closing of the variable vanes and opening or closing of the anti-icing and start bleed valve.

Engine Internal Washing System:

11. The engine wash manifold, integral with the swirl frame, has jets aimed at the compressor inlet annulus in the front frame. The wash manifold fitting is located at the 6 o'clock position on the swirl frame.

Pneumatic Start System

12. The pneumatic start system uses an air turbine-type engine start motor. System components include the APU, APU bleed air shutoff valve, engine start motor, start control valve, external start connector and check valve, controls, and ducting. A schematic of the bleed air control system is presented in figure 7. Three air sources may provide air for engine starts: the APU, engine cross bleed, or ground air source. Starts are accomplished in part through an electrically operated start

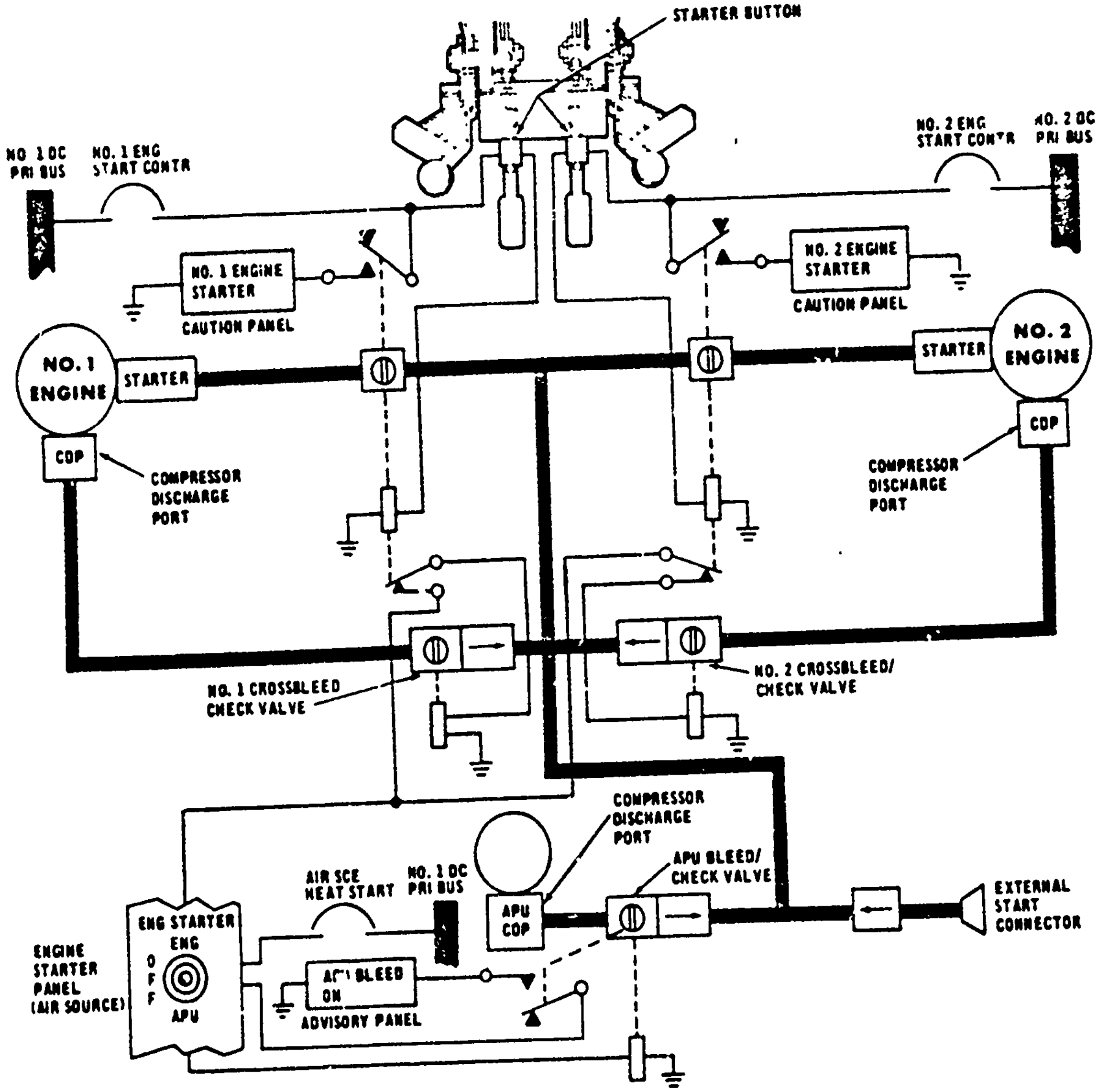

Figure 7. Bleed Air Control Schematic.

control valve that provides regulated air flow to the pneumatic start system. Pressure regulation prevents an overtorque situation when starts are conducted at high bleed air pressures. The start control valve is designed so that downstream pressure builds gradually to prevent impact damage of the engine starter pad. The pneumatic starter turbine wheel drives the engine through a gearbox and a slip clutch. The slip clutch prevents overtorque of the engine drive and eliminates possible malfunctions of the starter shear section. A retractable jaw in the starter engages an engine jaw during starts. A starter speed switch wired to the start control valve terminates the start cycle when cutoff speed is reached. The cross-bleed shutoff valves provide ON-OFF control of bleed flow between engines and check reverse flow. Both valves are opened by selection of the ENG position with air source during cross-bleed starting. The check feature prevents flow into the engine bleed ports from the APU or external air supplies or from the opposite engine, should an extreme power hence bleed pressure mismatch occur when the operating engine is providing bleed air for heating. The external start connector is on the left side of the fuselage. It is the attachment point for a bleed air line from an external source for engine starting or helicopter heating on the ground. The connector contains a check valve to prevent engine or APU bleed air from being vented.

Cockpit Engine Controls

13. There are four sets of cockpit controls, mechanical and electrical, which control engine functions. They are the power available levers (PAL), the collective pitch levers, the collective engine trim switches, and the engine emergency-off handles (figs. 8 and 9).

14. The PAL's are mechanically linked to the PAS's on the hydromechanical fuel control, which controls fuel shutoff, ground-idle, normal flight power range, and fuel vapor vent and ECU lockout. The various positions of the PAS are listed below.

Zero to 5 degrees	Shutoff
23.5 to 28.5 degrees	Ground-idle
115 to 130 degrees	Normal flight power range
127 to 130 degrees	Fuel vapor vent and ECU lockout

15. Each collective pitch lever mechanically connects to the load demand spindle (LDS) on the hydromechanical fuel control. Contained on the collective control head is an engine trim switch which is electrically linked to the ECU. The switches provide a limited rotor speed adjustment.

16. The engine emergency arming ("T") handles (fig. 8) stopcock the fuel control levers and arm the aircraft fire extinguishers when they are pulled to the rear. To activate either fire extinguisher, move the fire extinguisher switch to either MAIN or RESERVE as required.

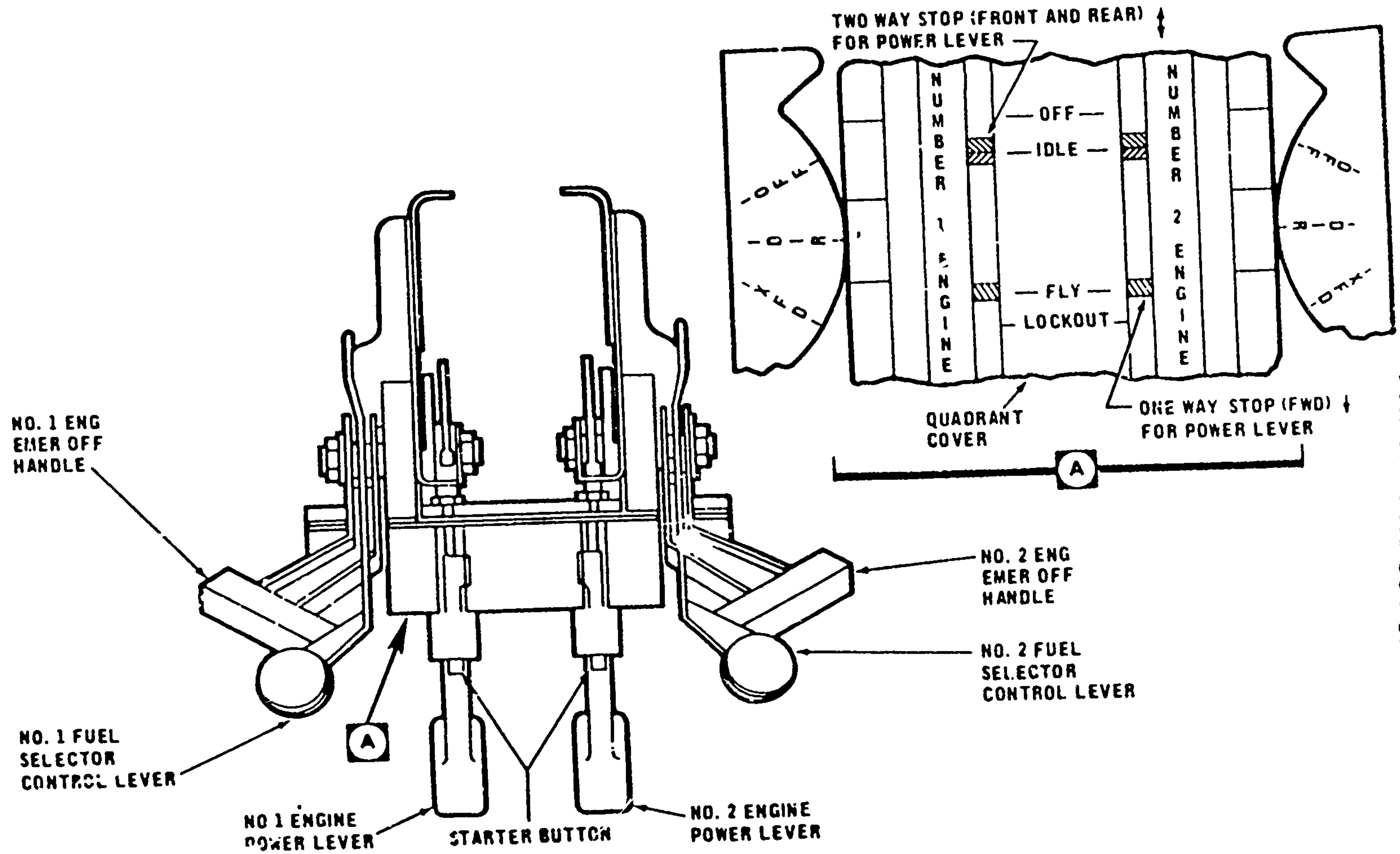

Figure 8. Engine Control Quadrant.

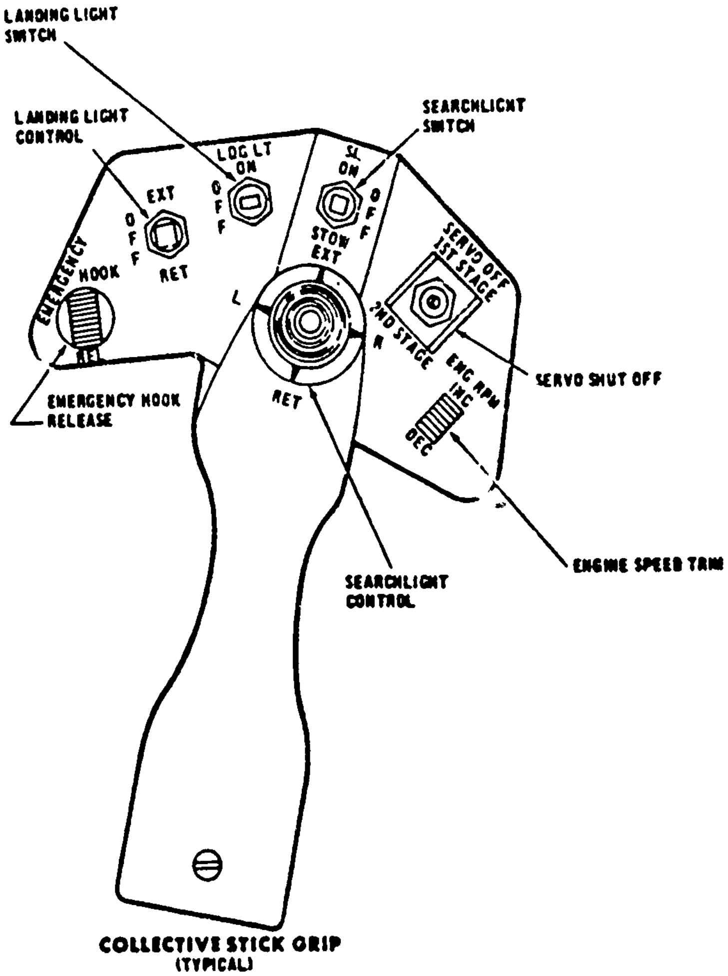

Figure 9. Control Stick Grip.

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Automatic Governing Characteristics

17. In general, the HMU provides for gas generator control in the areas of acceleration limiting, stall and flameout protection, N_G control, rapid response to power demand, and variable geometry actuation. The electrical unit trims the HMU to satisfy the requirements of the load so as to maintain rotor speed and load sharing and also to limit engine turbine inlet temperature. The basic control and governing functions of these two units are outlined in table 1 and schematically shown in figures 10 and 11.

18. Control of the gas generator is by a droop N_G governor in the HMU. The N_G governor reference is set by the PAS angle and modified mechanically by the LDS angle and trimmed electrically by an input from the ECU through an electrical-mechanical interface device called a "torque motor." After the PAS is set at 120 degrees with the PAL's, the load demand signal is then provided through the LDS. As the LDS is reduced from its maximum setting with a reduction of collective pitch, the desired N_G is reset down from the prevailing PAS schedule to provide a gas generator response. This reset schedule is then trimmed by the ECU to satisfy the Np and load control functions established by the ECU. This results in a zero steady-state Np error. The electrical trimming signal is a result of the Np governor, load-sharing circuit, and $T_{4.5}$ limiter, as combined in the ECU. The signal causes a resetting of the "collective compensation" curve, as shown in figure 12. In response to the resulting N_G reference, the HMU operates as a conventional gas generator power control and reschedules fuel flow within preset limits to obtain a reference N_G.

Manual Governing Characteristics

19. Failure of the ECU causes automatic Np governing to become inoperative. The Np overspeed system and LDS reset still remain operable. During this failure mode, the engine is controlled by use of the PAS and LDS. Without an electrical input, the HMU reverts to a power control. This power control is a droop control of N_G to a value called for by the PAS position as reset by the LDS input. The PAS position is set by manually adjusting the PAL.

20. In the event the ECU torque motor fails to a lower engine power, the torque motor can be mechanically deactivated. By advancing the PAS past intermediate to 130 degrees, the ECU interface is deactivated and locked-out. Engine power is then controlled through the PAS by manually adjusting the PAL's. Since this deactivation of the ECU interface is at the HMU torque motor, it does not affect any other ECU functions such as torque computation, Np overspeed protection, or signal generation. It does deactivate Np governing, $T_{4.5}$ limiting, and load sharing, which all normally act through the torque motor.

Table 1. Fuel and Control System Functional Split.

Hydromechanical Unit	Electrical Control Unit
Fuel pumping	HMU trimming of N_G governor as determined by:
Fuel flow metering	Isochronous N_P governing
Acceleration and deceleration limiting as a function of $\frac{W_f}{P_3}$, N_G, and T_2	$T_{4.5}$ limiting
N_G limiting	Load sharing on torque
Variable geometry positioning as a function of $N_G/\sqrt{\theta}$	N_P reference input from cockpit
Electrical unit signal acceptance to trim N_G governor	Redundant N_P overspeed limit
Starting fuel scheduling	Cockpit signal generation of N_P, $T_{4.5}$, and torque
Collective compensation through LDSA	Recorder power and signal supply
Electrical unit disable function through PAS	
PAS control with electrical unit inoperative	
Engine shutdown on N_P o/s signal from ECU	

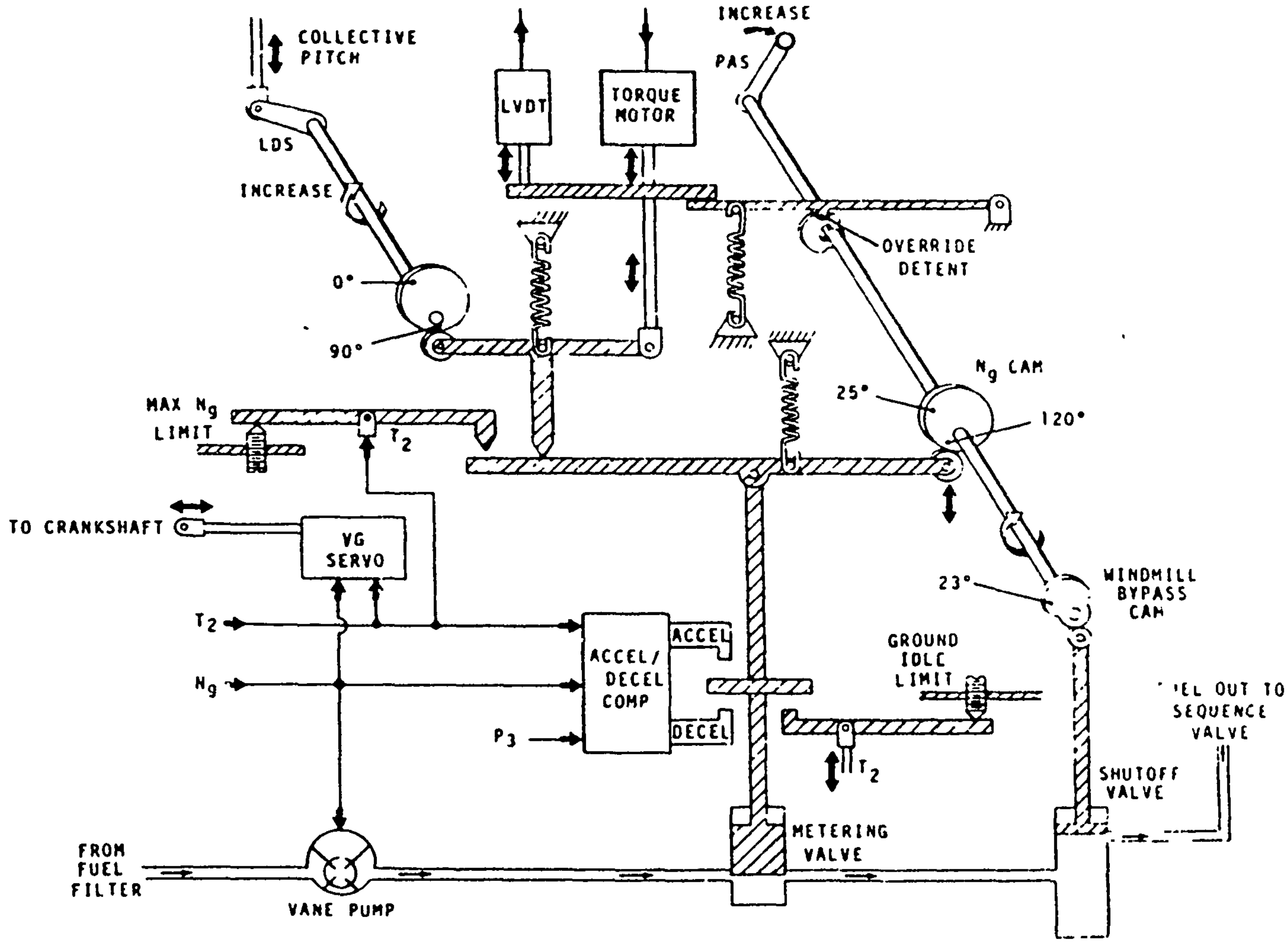


Figure 10. HMU Schematic.

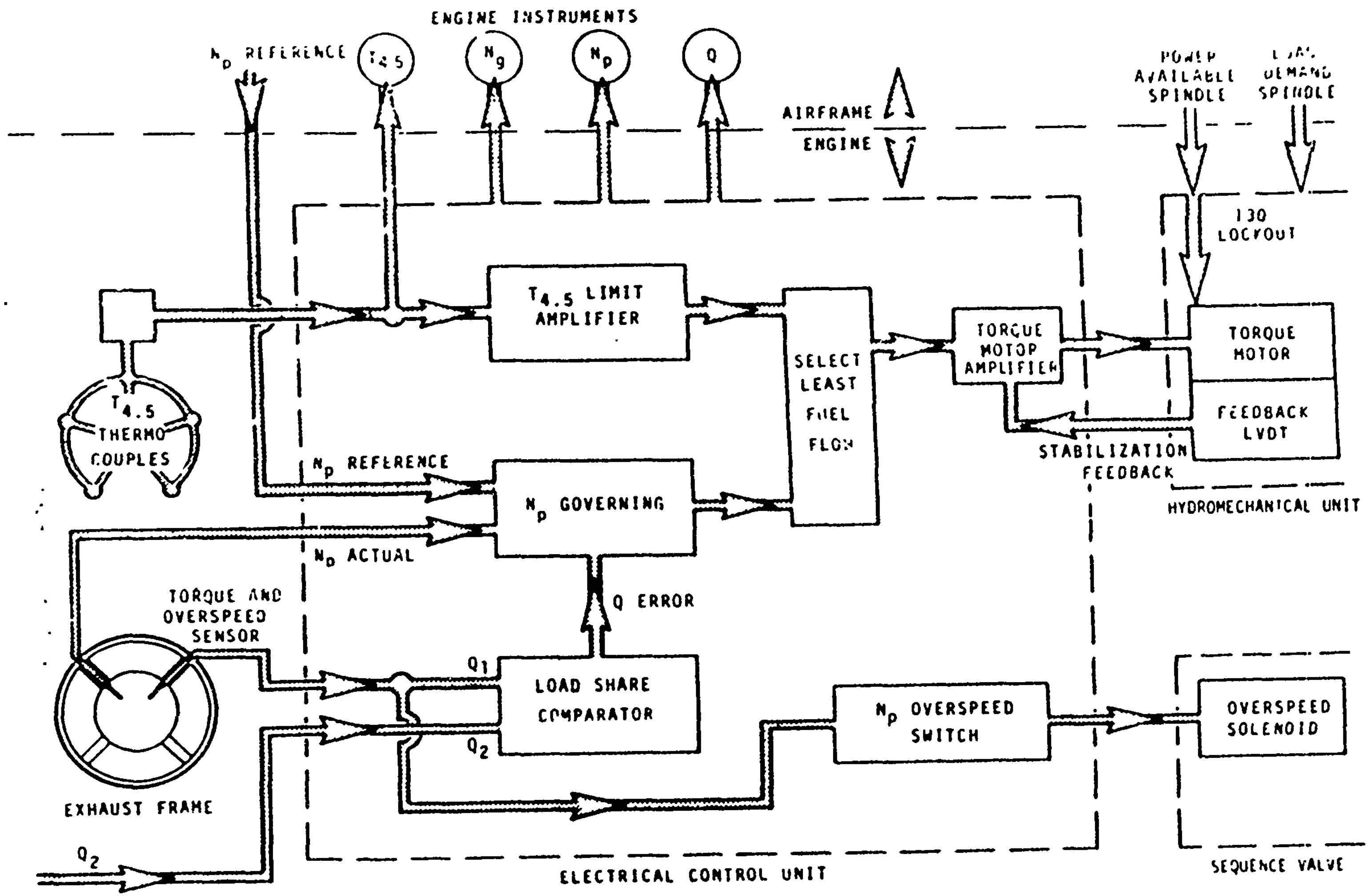


Figure 11. ECU Schematic.

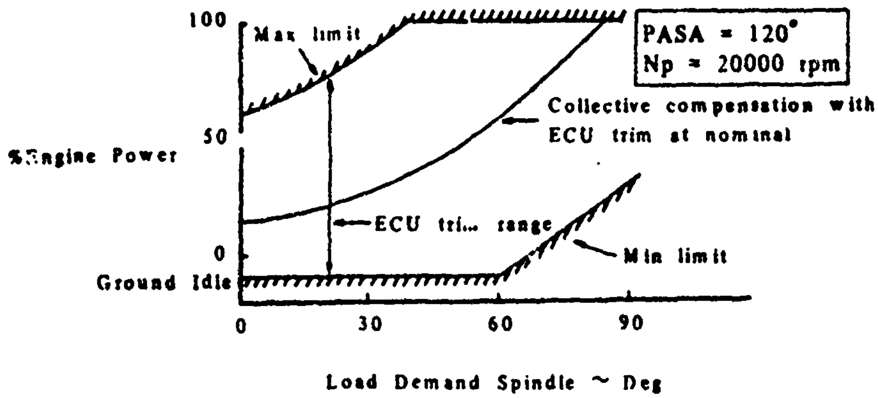


Figure 12. Collective Compensation Curve.

Auxiliary Power Unit

21. The International Harvester-built Model T62T-40-1 APU consists of a gas turbine power section, reduction gear drive, and appropriate controls and accessories. The gas turbine power section is a single-spool gas turbine using a single centrifugal compressor and a single-stage radial inflow turbine. The APU is hydraulically started with hydraulic fluid from an hydraulic accumulator. Ignition is accomplished by a separate pressure atomizing fuel nozzle and a spark plug. Once ignition and combustion of the main fuel is completed, the spark plug and start fuel are turned off. A purge valve allows compressor discharge pressure to bleed through the start fuel nozzle to continuously keep it clean for the next start. A pinion gear supported by three planetary reduction gears (5.1:1) reduces turbine shaft speed from 61,248 rpm to an output speed of 12,000 rpm. The APU is lubricated by an accessory-driven lubricating pump which circulates oil from the gearbox sump to three jets spraying the high-speed pinion gear. The resulting splash and oil mist fills the gearbox cavity and passes through the hollow shaft, lubricating the remaining gears and bearings. Pertinent information concerning the APU is listed below.

Rated engine speed	61,248 rpm
Exhaust gas temperature	1220°F ±10
Weight (dry)	92 lb (approx)
Output shp	90
Fuel consumption at rated power	150 lb/hr (approx)
Reduction gear and accessories:	
Input speed (rated)	61,248 rpm
Output speed (rated)	12,000 rpm
Fuel control assembly	4200 rpm
Fuel	JP-4, JP-5
Lubricating oil	MIL-L-7808 or MIL-L-23699
Oil pump pressure at rated speed	15 to 40 psi
Components and systems:	
Compressor	Single-stage, centrifugal flow
Turbine	Single-stage, radial flow
Combustor	Annular type

22. The APU controls are located on the overhead console and consist of a control switch and a fire annunciator/fuel shutoff "T" handle. The control switch has positions marked OFF, RUN, and START. Placing the switch to START energizes the APU hydraulic start valve to the open position to release the APU accumulator fluid to the hydraulic start motor. Placing the switch to RUN energizes the automatic start sequence resetting the sequencing system to a known reference state, and energizing the APU start bypass valve. The APU fire/fuel-off "T" handle warns the pilot and copilot of fire in the APU compartment.

APPENDIX E. INSTRUMENTATION

GENERAL

1. The test instrumentation was installed, calibrated, and maintained by Sikorsky Aircraft. A test boom with a swiveling pitot-static tube and angle of attack and sideslip vanes were installed at the nose of the aircraft. Equipment required only for specific tests was installed when needed and is discussed in the section on special equipment. Data were obtained from calibrated instrumentation and displayed or recorded as indicated below.

Pilot Panel

Airspeed (boom)
Altitude (boom)
Altitude (radar)
Rate of climb (boom)
Rotor speed
Engine torque*
Gas generator speed*
Control position:
  Longitudinal
  Lateral
  Directional
  Collective
Horizontal tail position
Center-of-gravity normal acceleration
Angle of sideslip
Attitude gyro*

Copilot Panel

Instrumentation controls
Event switch
Control fixtures
Airspeed*
Rotor speed
Engine torque*
Free air temperature*
Attitude gyro*
Fuel used (totalizer)*
Run number

*Ship's system

Engineer Panel

Instrumentation controls
Free air temperature
Fuel remaining
Time code display
Run number
Center-of-gravity control panel

2. Data parameters recorded on board the aircaft included the following:

Digital (PCM) Data Parameters

Airspeed (ship's system)
Airspeed (boom)
Altitude (boom)
Altitude (radar)
Rate of climb (boom)
Total air temperature (boom)
Rotor speed
Gas generator speed**
Power turbine speed**
Fuel used
Engine fuel flow**
Engine output shaft torque**
Engine inlet total temperature**
Engine inlet total pressure**
Engine air particle separator inlet total pressure**
Engine inlet guide vane actuator position**
Compressor discharge static pressure**
Fuel control discharge pressure**
Engine beep actuator position (ECU)**
Main rotor shaft torque
Tail rotor shaft torque
Control position:
    Longitudinal cyclic
    Lateral cyclic
    Directional
    Collective
    Engine condition lever
Stability augmentation position:
    Longitudinal
    Lateral
    Directional

**Both engines

Mixer input position:
- Longitudinal
- Lateral
- Directional

Primary servo output position:
- Forward
- Lateral
- Aft

Control force:
- Longitudinal
- Lateral
- Directional

Aircraft attitude:
- Pitch
- Roll
- Yaw

Aircraft angular velocity:
- Pitch
- Roll
- Yaw

Aircraft angular acceleration:
- Pitch
- Roll
- Yaw

Center-of-gravity acceleration:
- Longitudinal
- Lateral
- Vertical

Angle of sideslip
Angle of attack
Landing gear vertical load (all gear)
Stabilator angle of incidence
Tether cable tension
Tether cable angle
Ballast cart location
Low-airspeed system
Vibration (accelerometers):
- Pilot seat vertical
- Pilot seat lateral
- Pilot seat longitudinal
- Copilot seat vertical
- Copilot seat lateral
- Copilot seat longitudinal
- Center-of-gravity vertical
- Center-of-gravity lateral
- Center-of-gravity longitudinal

Right instrument panel vertical
Right instrument panel lateral
Forward avionics compartment vertical
Forward avionics compartment lateral
Forward avionics compartment longitudinal
Left aft cabin vertical
Right aft cabin vertical
Left forward cabin vertical
Pilot cyclic control vertical
Left pilot pedal longitudinal
Pilot heel rest vertical
Main transmission vertical
Main transmission lateral
Main transmission longitudinal
Tail rotor gearbox vertical
Tail rotor gearbox lateral
Tail rotor gearbox longitudinal
No. 1 engine exhaust frame vertical
No. 1 engine front frame vertical
No. 1 engine front frame lateral
No. 2 engine exhaust frame lateral
No. 2 engine front frame vertical
No. 2 engine front frame lateral

3. Critical parameters necessary for certain tests were telemetered to a ground station for real time monitoring.

<u>Airspeed Calibration</u>

4. The calibration of the airspeed system was accomplished by determining the existing airspeed position error of the standard ship's system and the test nose boom in level, climbing, diving, and autorotational flight. The two systems were calibrated in climbs and descents utilizing a trailing bomb as a standard and an AH-1G pace aircraft for level flight. An F-51D pace aircraft was utilized to calibrate the system during high-speed level flight and IRP descent airspeeds (110 KCAS and greater). The test helicopter was stabilized in ball-centered flight in 10-knot increments at a constant rotor speed of 258 rpm. The test nose boom system was used as the source for all airspeed data reduction. It was determined that three individual mathematical curve fits of the data were necessary to adequately define the nose boom position error in level/diving, climbing, and autorotational flight. The position error of the boom airspeed system is presented for level/diving flight and climbs and autorotations in figures 1 and 2.

FIGURE 1
BOOM SYSTEM AIRSPEED CALIBRATION IN LEVEL FLIGHT
YUH-60A USA S/N 73-21651

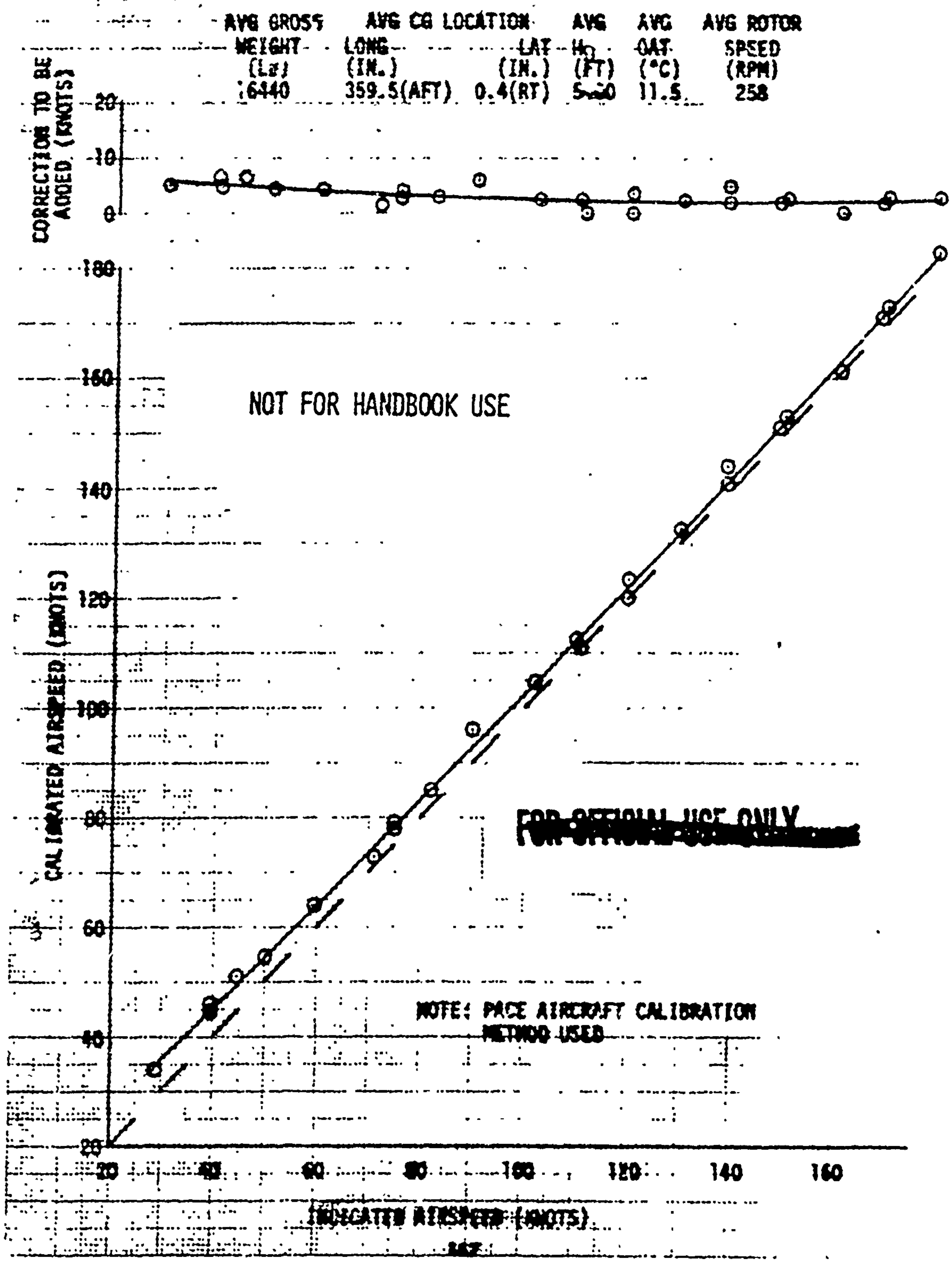
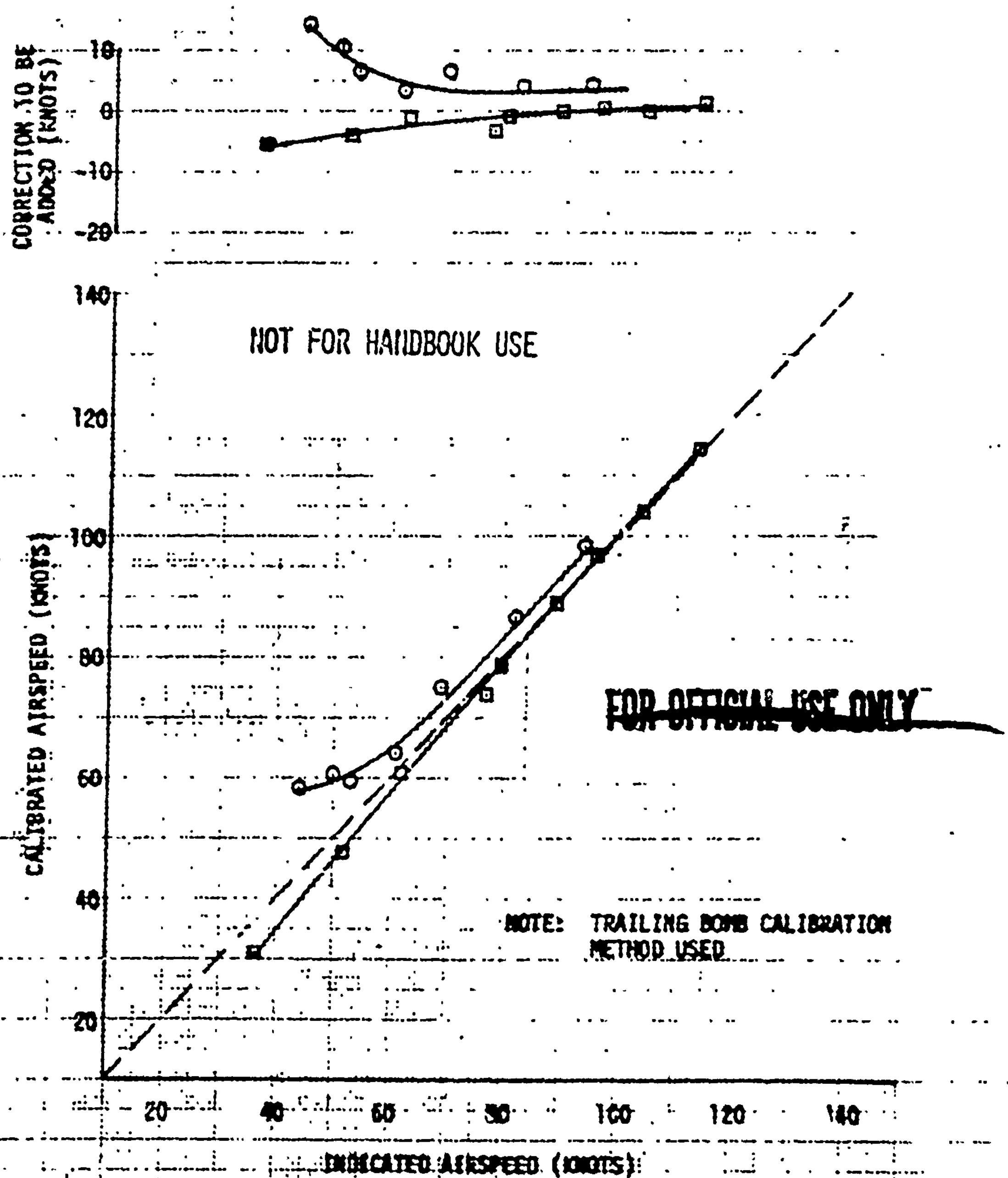


FIGURE 2

BOOM SYSTEM AIRSPEED CALIBRATION IN CLIMBING AND AUTOROTATIONAL FLIGHT

YUH-60A USA S/N 73-21651

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	16900	359.8(AFT)	0.4(RT)	4900	12.0	258	AUTO
□	16860	359.7(AFT)	0.4(RT)	5100	12.5	258	CLIMB



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SPECIAL EQUIPMENT

Low Airspeed System

5. The low airspeed system used in this evaluation is manufactured by Marconi-Elliott Avionics Systems Ltd. Rochester, Kent, England. The system is designed to provide airspeed information in the longitudinal, lateral, and vertical axes of the helicopter. The system consists of a multiaxis omnidirectional probe which senses pitot and static pressure and angular data; an air data computer; longitudinal and lateral airspeed indicators; and a vertical speed indicator. For the YUH-60A, the sensing probe was mounted on the right side of the aircraft (photo 1) and between the crew door and gunner's window. The airspeed and vertical speed indicators were mounted on the glare shield between the pilot and copilot. A USAAEFA-designed electronic interface box was used to take calculated longitudinal and lateral airspeed data from the air data computer and display longitudinal and lateral airspeed on the airspeed and heading command bars of the vertical situation indicator (VSI). The information displayed on the VSI was used to maintain zero horizontal airspeed during the vertical climb tests.

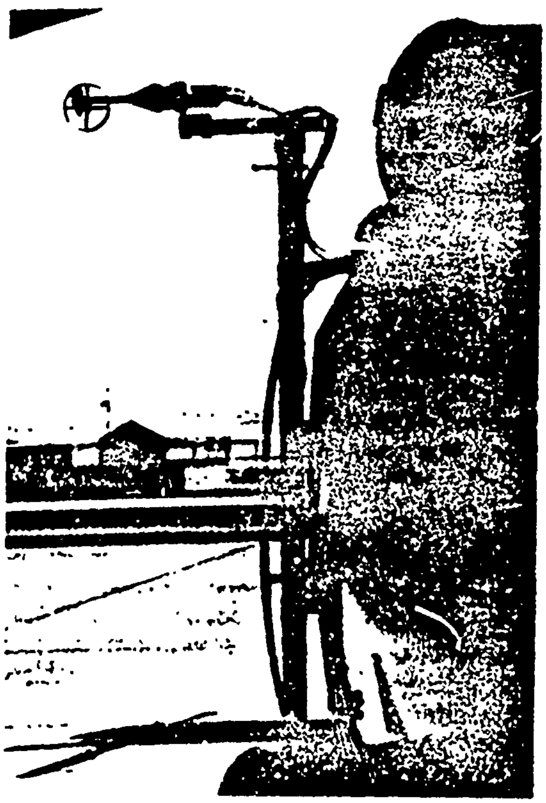


Photo 1. Low-Airspeed System.

Digital Recording Observation Instrument

6. The digital recording observation instrument (ROI) is an optical tracking device manufactured by Keuffel and Esser Company, Morristown, New Jersey. The ROI is designed to continuously gather elevation and azimuth data, both of which are electronically sensed and printed on paper tape as a target is tracked. Two ROI's were used during the H-V testing to track the aircraft from entry through touchdown. Additionally, the ROI's were used during vertical climb, takeoff performance, and vertical displacement testing. The data derived from the ROI recordings were vertical height, vertical rate of descent, horizontal distance, and horizontal airspeed.

Weather Station

7. A portable weather station, consisting of an anemometer, sensitive temperature gage, and altimeter, was used to record wind speed, wind direction, ambient temperature, and pressure altitude.

Digital Doppler Traffic Radar

8. The digital doppler traffic radar (speedgun) is a hand-held battery-operated device with a digital readout manufactured by CMI, Inc, Chanute, Kansas. During single-engine landing tests, the speedgun was used to record aircraft ground speed at touchdown. The speedgun was also used during the low-speed tests at Coyote Flats.

Movie and Video Cameras

9. A 16mm movie camera and a video tape camera were used simultaneously throughout testing. Flights were recorded on film and video tape to provide the test team with a means of reviewing test techniques. The video tape was especially useful because it could be replayed immediately after flying. Specific data parameters were not recorded with this equipment.

Load Cell

10. A calibrated load cell incorporating a cockpit readout of cable tension and load position (cable angle) was installed and maintained by Sikorsky. During the tethered-hover tests, the load cell measured cable tension and transmitted this information to the cockpit indicator in pounds; the data were also recorded on magnetic tape. The load position indicator provided the pilot with longitudinal and lateral displacement con... information for centering the aircraft over the ground anchoring (tie-down) point.

Flight Analyzer

11. A special ground camera, a Fairchild Flight Analyzer, is designed to record aircraft motion by taking a series of pictures while simultaneously recording relative elapsed time. The camera was located at surveyed distances from the runways and the photographs used to quantify data gathered during the H-V, takeoff, and vertical displacement tests.

Ground Pace Vehicles

12. Sedans, station wagons, and military jeeps were used as ground pace vehicles throughout the testing. The speedometers of these vehicles were calibrated by a speedgun. The pace vehicles were used to establish precise airspeed during the H-V and takeoff performance tests and low-airspeed handling qualities tests. Additionally, they were used to quantify touchdown airspeed during the single-engine landing tests.

APPENDIX F. TEST TECHNIQUES AND DATA ANALYSIS METHODS

TEST TECHNIQUES

1. Conventional test techniques were used on both the performance and handling qualities tests. Performance testing was conducted in coordinated (ball-centered) flight instead of zero sideslip. The basic techniques for each test are described in the Results and Discussion section of this report. Performance test techniques are amplified, where necessary, in this appendix in the following paragraphs. Detailed descriptions of all test techniques are contained in references 9 and 10, appendix A.

DATA ANALYSIS METHODS

General

2. Performance data were evaluated using the methods described in Army Materiel Command Pamphlet AMCP 706-204 (ref 9, app A). Handling qualities data were evaluated using standard test methods described in Naval Air Test Center Flight Test Manual FTM No. 101 (ref 10).

Aircraft Weight and Balance

3. The aircraft was weighed in the instrumented configuration with all fuel drained prior to, during, and at the completion of the GCT. The results of these weighings were 13,121 pounds with the longitudinal cg located at FS 351.9, 13,200 pounds with cg not determined, and 13,185 pounds with a cg at FS 352.1, respectively. The small differences in the weighings (0.5 percent) can be attributed to the multitude of component and instrumentation changes that occurred to the prototype aircraft during the GCT phase. A fuel cell calibration was accomplished during the first weighing. Measured total fuel volume by gravity fueling methods was 350 gallons. The fuel weight for each test flight was determined prior to engine start and after engine shutdown by using a calibrated external sight gage to determine fuel volume and by measuring specific gravity. Fuel used in flight was recorded by a sensitive fuel-used system and verified with the pre- and postflight sight gage readings. Aircraft cg was controlled by a trimmable ballast system which automatically maintained a constant cg while fuel was burned. The trimmable ballast system was comprised of a cart (2900-pound capacity) attached to the cabin floor by rails and driven by an electric screw jack with a total excursion of 70.5 inches. The cart was prepositioned at a desired station on the ground. In flight, the fuel-used information required for movement of the cart was compensated by a potentiometer on the control panel. The potentiometer setting was determined by the engine start gross weight of the aircraft and the weight of the cart.

4. The helicopter loading was controlled during specific weight-critical tests by adding ballast between data points. These tests and the respective loading tolerance W/σ were as follows.

a. Takeoff performance (±75 pounds).

b. Vertical climbs (±75 pounds).

c. Height-velocity (±200 pounds).

d. Vertical and lateral displacement (±300 pounds).

e. Low-speed flight (±200 pounds).

Nondimensional Method

5. Helicopter performance may be generalized through the use of nondimensional coefficients. The test results obtained at specific test conditions may be used to define performance at conditions not tested. The following nondimensional coefficients were used.

a. Coefficient of power (C_P).

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

b. Coefficient of thrust (C_T).

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

c. Advance ratio (μ).

$$\mu = \frac{V_T\ (1.6889)}{\Omega R} \tag{3}$$

d. Advancing blade tip Mach number (M_{tip}).

$$M_{tip} = \frac{(V_T) + (\Omega R)}{a} \tag{4}$$

Where:

SHP = Engine output shp (both)

ρ = Air density (slug/ft^3)

A = Main rotor disc area (ft^2)

Ω = Main rotor angular velocity (radians/sec)

R = Main rotor radius (ft)

Thrust = Gross weight (lb) during free flight in which there is no acceleration or velocity component in the vertical direction; tether load must be added in the case of tethered hover

V_T = True airspeed (kt)

a = Speed of sound (kt)

True airspeed (V_T) was calculated using calibrated airspeed (V_{cal}) and density ratio as follows.

$$V_T = V_{cal}/\sqrt{\sigma} \tag{5}$$

This nondimensional method is useful only where the effects of compressibility and blade stall are not significant. During this evaluation, minor trends indicating an influence by compressibility and blade stall were noted. However, it was beyond the scope of this test to separate and define their effect.

Shaft Horsepower Required

6. The engine output shaft torque was determined by use of the engine torquemeter (para 2c, app D). This torquemeter was calibrated in a test cell by the engine manufacturer. The output from the engine torquemeter was recorded on the on-board data recording system. The output shp was determined from the engine's output shaft torque and rotational speed by the following equation:

$$SHP = \frac{(2\pi)\ (N_P)\ (Q)}{33{,}000} \tag{6}$$

Where:

Np = Engine output shaft rotational speed (rpm)

Q = Engine output shaft torque (ft-lb)

Tail Rotor Performance

7. During the hover performance tests, tail rotor performance parameters were also recorded. Terms in equations 1 and 2 which apply to the main rotor were replaced by tail rotor parameters for nondimensionalized tail rotor performance. Antitorque system output torque was measured at the input shaft to the intermediate gearbox. The horsepower loss due to the intermediate and 90-degree gearboxes was determined by Sikorsky Aircraft to be 1 shp. In order to determine shp at the tail rotor, this value was subtracted from the measured value at the input to the intermediate gearbox. The terms redefined are as follows.

SHP = Tail rotor shaft horsepower (SHP_{TR})

A = Tail rotor disc area (ft^2)

Ω = Tail rotor angular velocity (radians/sec)

R = Tail rotor radius (ft)

Thrust = Tail rotor thrust (lb)

Tail rotor shaft horsepower was determined from the following equation.

$$SHP_{TR} = \left\{ \frac{(2\pi) \left[(N_P)\ (4.7619) \right] (Q_{TR})}{33,000} \right\} - 1 \qquad (7)$$

Where:

Q_{TR} = Tail rotor torque (ft-lb)

8. The tail rotor thrust for hover was determined by making two assumptions. These assumptions were necessary since sufficient information was not available and tail rotor thrust could not be measured directly during the evaluation. The first assumption was that all directional moments to maintain stabilized hover would be generated by the antitorque tail rotor. This assumption neglected any possible restoring directional moments that could be derived from rotor downwash and recirculating airflow over the fuselage, tail boom section, and empennage. The second assumption was that the temperature of the air mass flow passing through the tail rotor was not significantly influenced by the engine exhaust gases.

Tail rotor thrust was determined from the following equation.

$$THRUST = \frac{QMR}{l_T} \qquad (8)$$

Where:

Q_{MR} = Main rotor shaft torque (ft-lb)

l_T = Perpendicular distance between center lines of main and tail rotor shafts = 32.567 feet

Hover

9. Hover performance was obtained by the tethered hover technique. Additional free flight hover data were accumulated to verify the tethered hover data. All hover tests were conducted in winds of less than 3 knots. Tethered hover consists of restraining the helicopter to the ground by a cable in series with a load cell. An increase in cable tension, measured by the load cell, had the same effect on hover performance as increasing gross weight. Free-flight hover tests consisted of stabilizing the helicopter at a desired height using the radar altimeter as a height reference. Atmospheric pressure, temperature, and wind velocity were recorded from a ground weather station. All hovering data were reduced to nondimensional parameters of C_P and C_T (equations 1 and 2, respectively), and grouped according to wheel height. A line was faired through each set of data. Summary hovering performance was then calculated from these nondimensional plots and the power available in figure 144, appendix G, at selected ambient conditions and power settings.

10. Nondimensional tail rotor performance and directional control position were used to determine tail rotor horsepower and directional control margins as a function of wheel height. All antitorque data were recorded simultaneously with the hover performance data of the evaluation.

Takeoff

11. Takeoff performance was determined using the level acceleration technique (paras 14 through 17, Results and Discussion). Takeoff tests were conducted to obtain curves of climb-out airspeed versus distance required to clear a 50-foot obstacle. Each curve was obtained by conducting a series of takeoffs using various climb-out airspeeds. During each series, ballast was added as necessary to maintain the desired excess power available conditions as fuel was consumed and ambient temperature varied. A ground-operated Fairchild Flight Analyzer was used to produce a photographic record of time, horizontal distance, and vertical displacement for each takeoff. The climb-out airspeed range varied from just below minimum achievable to the maximum practical airspeed (approximately 50 KIAS). All takeoff tests were performed in winds of 3 knots or less.

12. The excess power method of takeoff analysis was used. For data analysis purposes, power required was the actual horsepower during the stabilized 5-foot hover phase of the takeoff test maneuver. Likewise, power available was the horsepower at the 50-foot wheel height point during the climb phase of the takeoff

test maneuver. Calculation of ΔCp for data presentation purposes was based on the operational main rotor speed of 258 rpm. The excess power was then determined as follows.

$$\Delta C_p = C_p \text{ available} - C_p \text{ required at 5 ft} \qquad (9)$$

13. Distance to clear a 50-foot obstacle was obtained by plotting a time history of height and distance from the Fairchild plate and by reading the horizontal distance at the 50-foot wheel height point. The climb-out airspeed was determined from the height-distance time history by calculating the horizontal and vertical velocities and then determining the resultant velocity along the flight path. The climb-out airspeed was corrected for wind by adding the headwind component to the value of the horizontal aircraft velocity. The takeoff distance was plotted versus climb-out true airspeed for each takeoff. Takeoff performance was then summarized by combining all individual takeoff plots three-dimensionally as distance required to clear a 50-foot obstacle versus ΔCp and climb-out true airspeed. Fairings for this three-dimensional plot were based upon the collective influence of each takeoff plot, including the influence of data where a 50-foot wheel height was unobtainable due to the combination of ΔCp and climb-out airspeed. All dimensional takeoff performance was derived from this summary plot. The Cp required to hover at a wheel height of 5 feet was determined by calculating the C_T at the takeoff conditions (weight, density altitude, and rotor speed) and entering the 5-foot hover curve at this C_T to obtain the corresponding Cp. The Cp available was determined using the power available from the engine model specification corrected for installation losses at the atmospheric conditions.

Vertical Climb

14. The vertical climb technique used was to stabilize in a 100-foot OGE hover based on the radar altimeter and then to increase engine power by a predetermined increment of engine torque. Various increments of engine torque up to the engine topping li · ts were used. An Elliott low-airspeed system was used to provide cues of longitu... lateral translation. The Elliott low-airspeed system was accurate to within 1 ... horizontal airspeed. Each vertical climb was flown at a predetermined C_T wi... ...or speed held constant at 258 rpm. Ballast was added as fuel burned off or ...rature varied. Tests were conducted in ambient wind conditions of 3 knots or less.

15. The climb rates were measured after the aircraft was stabilized in unaccelerating vertical climbing flight by differentiating the output of a radar altimeter with respect to time. Vertical climb performance was determined by using equation... ...defined equation 3, and as follows:

$$\mu = \frac{V_v}{\Omega R} \qquad (10)$$

a. Vertical velocity ratio (VVr).

$$VVr = \frac{Vv}{\left(V_{tip}\right)\left(\frac{C_T}{2}\right)^{1/2}} \tag{11}$$

b. Generalized excess power coefficient (variation from hover) (ΔCp gen).

$$\Delta C_P \text{ gen} = \frac{C_{P_C} - C_{P_H}}{\left(\left(\frac{C_T}{2}\right)^3\right)^{1/2}} \tag{12}$$

Where:

Vv = Vertical velocity (ft/sec)

V_{tip} = Main rotor tip speed (ft/sec)

C_{PC} = Coefficient of power for climb

C_{PH} = Coefficient of power for hover

The vertical climb performance data were presented in nondimensional terms from which vertical climb capability at the PIDS conditions was extracted.

Forward Flight Climb

16. The forward flight climb tests were conducted at a constant rotor speed and a predetermined power and airspeed schedule. The power schedule represented single-engine IRP available at 35°C based on the engine model specification. The climb airspeed schedule was determined by the airspeed corresponding to the minimum power required in level flight. Two continuous single-engine climbs to service ceiling were accomplished at reciprocal headings to average the effect of wind shear on climb performance. The data were corrected from test-day conditions to 35°C at all altitudes; from test-day gross weight to a constant primary mission gross weight of 16,853 pounds; and for any deviations from the climb airspeed and power schedules. Test rate of climb was determined from pressure altitude variation with time and corrected for altimeter error caused by nonstandard temperature using the following equation.

$$R/C_T = \left(\frac{dH_P}{dt}\right)\left(\frac{T_t}{T_s}\right) \tag{13}$$

Where:

R/C_T = Tapeline rate of climb (ft/min)

$\frac{dH_p}{dt}$ = Slope of pressure altitude versus time curve at a given pressure altitude (ft/min)

T_t = Test ambient air temperature at the pressure altitude at which the slope is taken (°K)

T_s = Standard ambient air temperature at the pressure altitude at which the slope is taken (°K)

Tapeline rate of climb and test shaft horsepower were referenced to test day density altitude. All corrections to rate of climb were applied with reference to density altitude. Power corrections were made by the following equation.

$$\Delta R/C_p = \frac{(K_p)(\Delta SHP)(33{,}000)}{GW_t} \tag{14}$$

Where:

Kp = Power correction factor

$\Delta SHP = SHP_s - SHP_t$

Where:

SHP_s = shp available at 35°C ambient temperature from model specification engine installed

SHP_t = Test shp measured

GW_t = Test gross weight (lb)

Weight corrections were made by the following equation:

$$\Delta R/C_w = (K_w)(SHP_s)(33{,}000)\left[\frac{GW_t - GW_s}{(GW_s)(GW_t)}\right] \tag{15}$$

Where:

K_W = Weight correction factor

GW_s = Standard gross weight (lb)

17. Two additional series of climbs were conducted to achieve the power and weight correction factors. These factors were used to correct the continuous climb data from test to desired conditions, as mentioned in the previous paragraph. One series of climbs (Kp) was flown at a constant aim gross weight from altitudes near sea level up to 10,000 feet at various power settings. This series of climbs was corrected to a constant gross weight. The other series of climbs (K_W) was flown dual-engine with the total power equal to the single-engine IRP at 35°C at various gross weights, from altitudes near sea level to each respective climb service ceiling. This series of climbs was corrected for deviations from the aim power schedule. Corrected test results of rate of climb versus shp and gross weight are presented in figures 15 and 16, appendix G. A constant value of 0.74 was determined for K_P, while K_W was found to vary as a function of gross weight throughout the altitude range tested. Power and weight factors were determined from the following equations.

$$K_P = \left(\frac{\Delta R/C}{\Delta SHP}\right)\left(\frac{GW_t}{33,000}\right) \quad (16)$$

$$K_W = \frac{R/C_2 - R/C_1}{(SHP)(33,000)}\left[\frac{(GW_1)(GW_2)}{GW_1 \quad GW_2}\right] \quad (17)$$

Where:

GW_1 = Gross weight at R/C_1 from R/C versus GW curve

GW_2 = Gross weight at R/C_2 from R/C versus GW curve

18. Power corrections were applied for variations in airspeed from the climb airspeed schedule determined from the level flight performance data. Any deviations from this minimum power airspeed were corrected by equation 14.

Where:

ΔSHP = Difference in test shp measured and minimum power required for level flight at test conditions

19. A power correction was applied for the increased airframe drag due to instrumentation (para 25), again using equation 14.

Where:

ΔSHP = Difference in test shp measured and power required for level flight without airframe-mounted instrumentation

20. The standard rate of climb at 35°C ambient temperature was finally determined by the summarized equation:

$$R/C_s = R/C_T + \Delta R/C_P + R/C_w + \Delta R/C_{A/S} + \Delta R/C_{INSTR} \qquad (18)$$

Where:

$\Delta R/C_{A/S}$ = Rate of climb difference due to deviation from climb airspeed schedule (ft/min)

$\Delta R/C_{INSTR}$ = Rate of climb difference due to instrumentation drag

The corrected R/C_S, SHP_S, corrected V_{cal}, and calculated time to climb, nautical air miles traveled and V_T parameters with reference to density altitude, were plotted as a function of pressure altitude at 35°C temperature conditions (fig. 14, app G).

<u>Level Flight Performance</u>

21. Level flight speed-power performance was determined by using equations 1, 2, and 3. Each speed power was flown at a predetermined C_T with rotor speed held constant. To maintain gross weight to density ratio (W/σ) constant, altitude was increased as fuel was consumed. Test-day level flight power was corrected to standard-day conditions by assuming that the test-day dimensionless parameters, C_{P_t}, C_{T_t}, and μ are independent of atmospheric conditions. Consequently, the standard-day dimensionless parameters C_{P_s}, C_{T_s}, and μ_s are identical to C_{P_t}, C_{T_t}, and μ_t, respectively. From equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \left(\frac{\rho_s}{\rho_t}\right) \qquad (19)$$

Where:

t = Test day

s = Standard day

22. Curves defined by the power required as a function of airspeed were plotted as C_P versus μ for a constant value of C_T. These curves were then joined by lines of constant μ to form a carpet plot. The reduction of this carpet plot into a family of curves, C_T versus C_P, for a constant μ value allows determination of the power required as a function of airspeed for any value of C_T.

23. The specific range (NAMPP) data were derived from the test level flight power required and fuel flow. The NAMPP curves presented on the summary plots were obtained from the power and airspeed from the level flight carpet plot and the fuel flow from the engine model specification, corrected for installation losses, for the particular conditions. The following equation was used for determination of NAMPP.

$$NAMPP = \frac{V_T}{W_f} \qquad (20)$$

Where:

V_T = True airspeed

W_f = Fuel flow (lb/hr)

The PIDS compliance endurance missions were determined based on 5 percent conservatism applied to the fuel flow.

24. The tip Mach number of the advancing blade during level flight was determined from equation 4.

25. Changes in the equivalent flat plate area (Δf_e) due to change in aircraft cg were calculated from the following equation.

$$\Delta f_e = \frac{2(\Delta C_p)(A)}{\mu^3} \qquad (21)$$

Where:

ΔC_p = Change in coefficient of power

Summary level flight performance plots for standard-day and US Army hot day presentations were corrected for the effects of instrumentation drag. An equivalent Δf_e of 1.41 ft^2 was used with an assumed rotor propulsive efficiency of 0.85, which resulted in an effective Δf_e of 1.66 ft^2 (ref 11, app A). The additional power required determined from the instrumentation drag was subsequently subtracted from the power measured for level flight.

Turning Performance

26. Turning performance tests were conducted to determine these characteristics at V_H. The tests were accomplished by stabilizing in level flight at the engine torque settings which corresponded to IRP that a model specification engine would develop at 4000 feet, 35°C day conditions. While altitude and power were held constant, the bank angle was incrementally increased to beyond that required for

a standard rate turn in order to obtain data to determine the airspeed loss for a standard rate turn. Quantitative values of turn rate, altitude, and airspeed were recorded for these tests. After completion of the test series, the initial data points were repeated to determine the effect of a lesser gross weight on test results.

Autorotational Descent Performance

27. Autorotational descent performance data were acquired at variations in stabilized airspeeds with constant rotor speed and variations in rotor speed with constant airspeed. The tapeline rates of descent were calculated by the following equation.

$$\text{R/D tapeline} = \left(\frac{dH_p}{dt}\right)\left(\frac{T_t}{T_s}\right) \qquad (22)$$

Where:

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

T_t = Test ambient air temperature (°K)

T_s = Standard ambient air temperature (°K)

Height-Velocity Performance

28. The limited-scope evaluation of H-V performance consisted of verifying four contractor-demonstrated points (below the knee of the contractor-supplied H-V curve) as modified by AVSCOM. The tests were conducted by a simulated failure of one engine from dual-engine operation. Test-day gross weight was adjusted to account for variations in pressure altitude and temperature because of the inability to adjust topping power on the unfailed engine. Aircraft height above the ground was determined using the aircraft radar altimeter. Aircraft airspeed was established by utilizing a calibrated pace car.

29. Each data point was telemetered to a remote ground station in addition to being recorded on the on-board aircraft instrumentation system. Telemetry information on rotor speed decay and landing gear loads was utilized to assist the pilot in developing a qualitative assessment for verification of the H-V curve.

Vertical and Lateral Displacement

30. Vertical and lateral displacement tests were conducted to quantify the agility of the helicopter. The tests were conducted by stabilizing in coordinated steady-heading level flight at 150 KTAS, then rapidly displacing the helicopter vertically or laterally a distance of 200 feet, as described in paragraphs 27 and 30 of the Results and Discussion section, while determining the horizontal distance

along the entry flight path required to complete the maneuver. The Fairchild camera was utilized to determine vertical distance. Lateral displacements were determined on a measured course utilizing runway borders and ground observers.

Vibration

31. Vibration data were analyzed using a Spectral Dynamics Model SD301B real time spectral analyzer. The analyzer converted the data from the time domain (acceleration as a function of time) to the frequency domain (acceleration as a function of frequency). The data were analyzed using two frequency ranges: zero to 100 Hz and zero to 500 Hz. Frequency resolution was 0.2 Hz for the 100-Hz range and 1 Hz for the 500-Hz range. In order to minimize random variation in acceleration amplitude, the data were averaged over an 8-second time interval, using a Spectral Dynamics Model 302B ensemble averager.

Engine Performance

32. Engine temperature and pressure inlet characteristics were obtained from an inlet ring consisting of 48 probes positioned four per rake, 30 degrees apart, attached to the air inlet of each engine. The probes were manifolded or electrically combined into one average reading for temperature and pressure, respectively. The pressure probes were referenced to the test boom static source system to provide a differential pressure measurement. The temperature probes measured temperature directly.

33. Engine inlet temperature rise is based on the total temperature at the probe minus the boom total temperature corrected to ambient conditions, as follows.

$$\Delta T = T_{T1} - T_{SO} \tag{23}$$

Where:

T_{T1} = Engine inlet total temperature (°C)

T_{SO} = Ambient temperature (°C)

34. Pressure ratio determined at the engine inlet is based on the total pressure at the probe divided by the test boom static pressure corrected for installation position error to ambient conditions as follows.

$$\frac{P_{T1}}{P_{SO}} = \frac{\Delta P_1 + P_{SB}}{P_{SO}} \tag{24}$$

Where:

$\Delta P_1 + P_{SB} = P_{T1}$, engine inlet total pressure (lb/in.^{2})

Where:

$\Delta P_1 = P_{T1} - P_{SB}$, differential pressure (lb/in.^{2})

P_{SB} = Measured static pressure (boom system) (lb/in.^{2})

35. Exhaust system characteristics were supplied by Sikorsky Aircraft (fig. 143, app G) as exhaust duct static pressure rise coefficient versus power turbine discharge swirl angle.

36. Main transmission and drive train power losses were determined by comparing the total engine shp to the total rotor horsepower, as follows.

$$\Delta HP = ESHP - RHP \tag{25}$$

Where:

ESHP = Total engine shaft horsepower (both)

RHP = Main rotor horsepower plus tail rotor horsepower

37. The referred terms of the engine parameters were used to compare the test engines with the model specification engine. Data on SHP, measured gas temperature ($T_{4.5}$), fuel flow, gas generator speed (N_G), and compressor discharge static pressure were referred as follows.

a. Referred SHP (RSHP):

$$RSHP = SHP/\delta_1\sqrt{\theta_1} \tag{26}$$

b. Referred gas temperature (RGAST):

$$RGAST = \left[\frac{T_{4.5} + 273.15}{(\theta_1)^{0.96}}\right] - 273.15 \ (^\circ C) \tag{27}$$

c. Referred fuel flow (RWF):

$$RWF = \frac{W_f}{(\delta_1)(\theta_1)^{0.55}} \ (lb/hr) \tag{28}$$

d. Referred gas generator speed (RN_G):

$$RN_G = \frac{N_G}{(\theta_1)^{0.50}} \quad (\%) \tag{29}$$

e. Referred compressor discharge static pressure (RCDSP):

$$RCDSP = \frac{CDP + P_{SO}}{\delta_1} \quad (lb/in.^2) \tag{30}$$

Where:

$$\delta_1 = \frac{\Delta P_1 + P_{SB}}{14.697}$$

$$\theta_1 = \frac{T_{T1} + 273.15}{288.15}$$

$T_{4.5}$ = Turbine inlet temperature (°C)

W_f = Engine fuel flow (lb/hr)

N_G = Gas producer speed referenced to 44,700 rpm (percent)

$CDP + P_{SO}$ = Compressor discharge static pressure (lb/in.^{2})

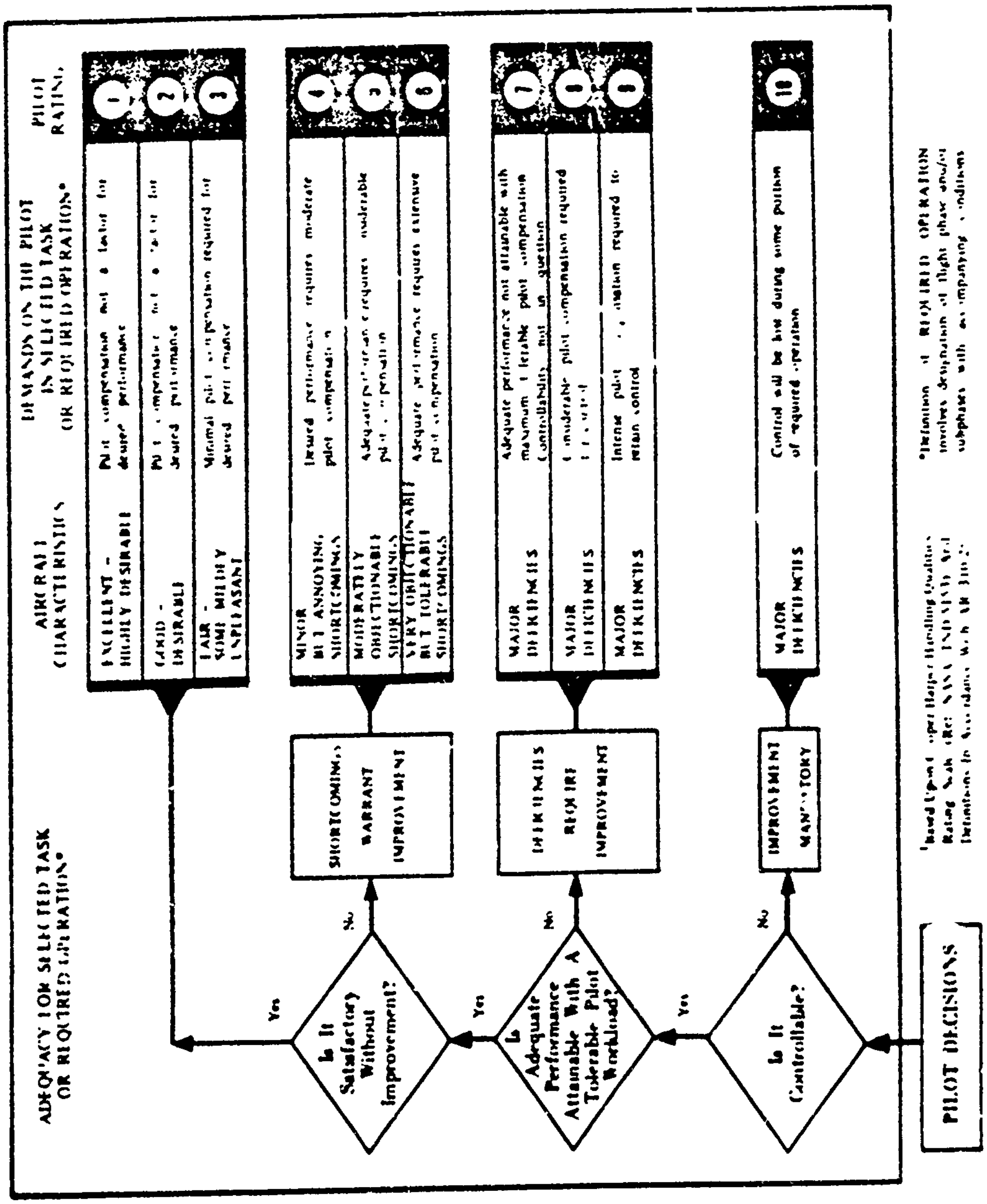


Figure 1. Handling Qualities Rating Scale.

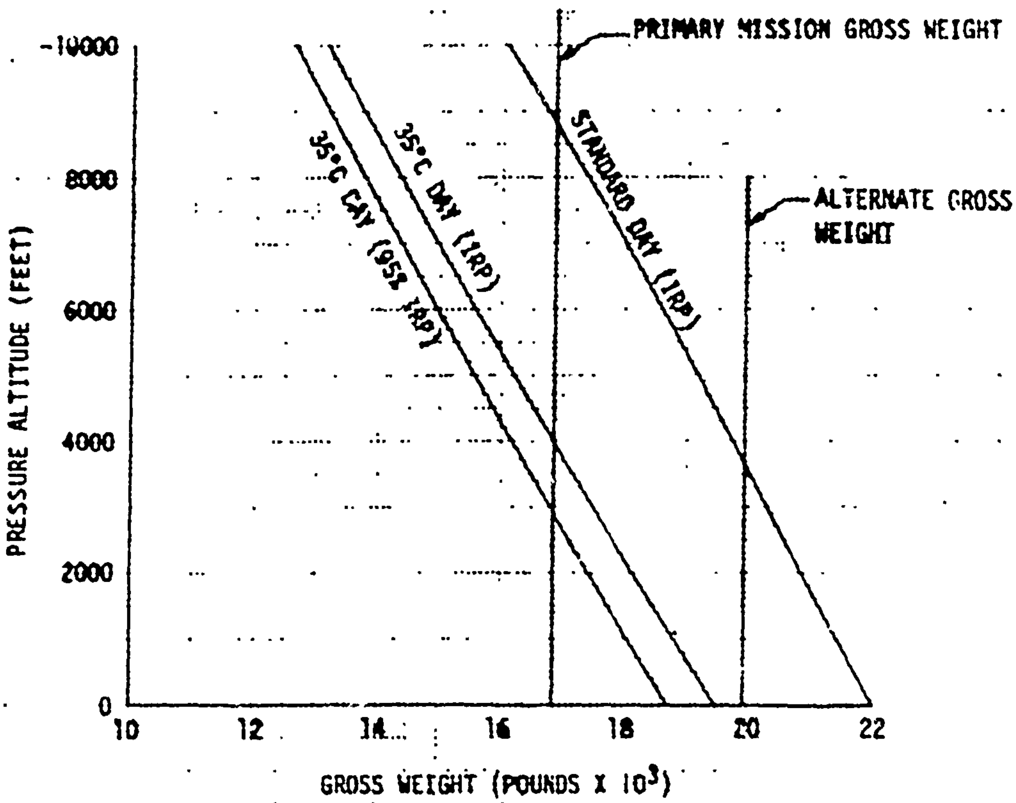
APPENDIX G. TEST DATA

INDEX

FIGURE 1
SUMMARY HOVER PERFORMANCE
YUH-60A USA S/N 73-21651

ROTOR SPEED = 258
INTERMEDIATE (30 MINUTE LIMIT) POWER AVALIABLE
WHEEL HEIGHT = 100 FEET (OGE)

NOTES: 1. POWER AVALIABLE OBTAINED FROM FIGURE 144.
2. HOVER DATA DERIVED FROM FIGURE 3.
3. WINDS LESS THAN 3 KT



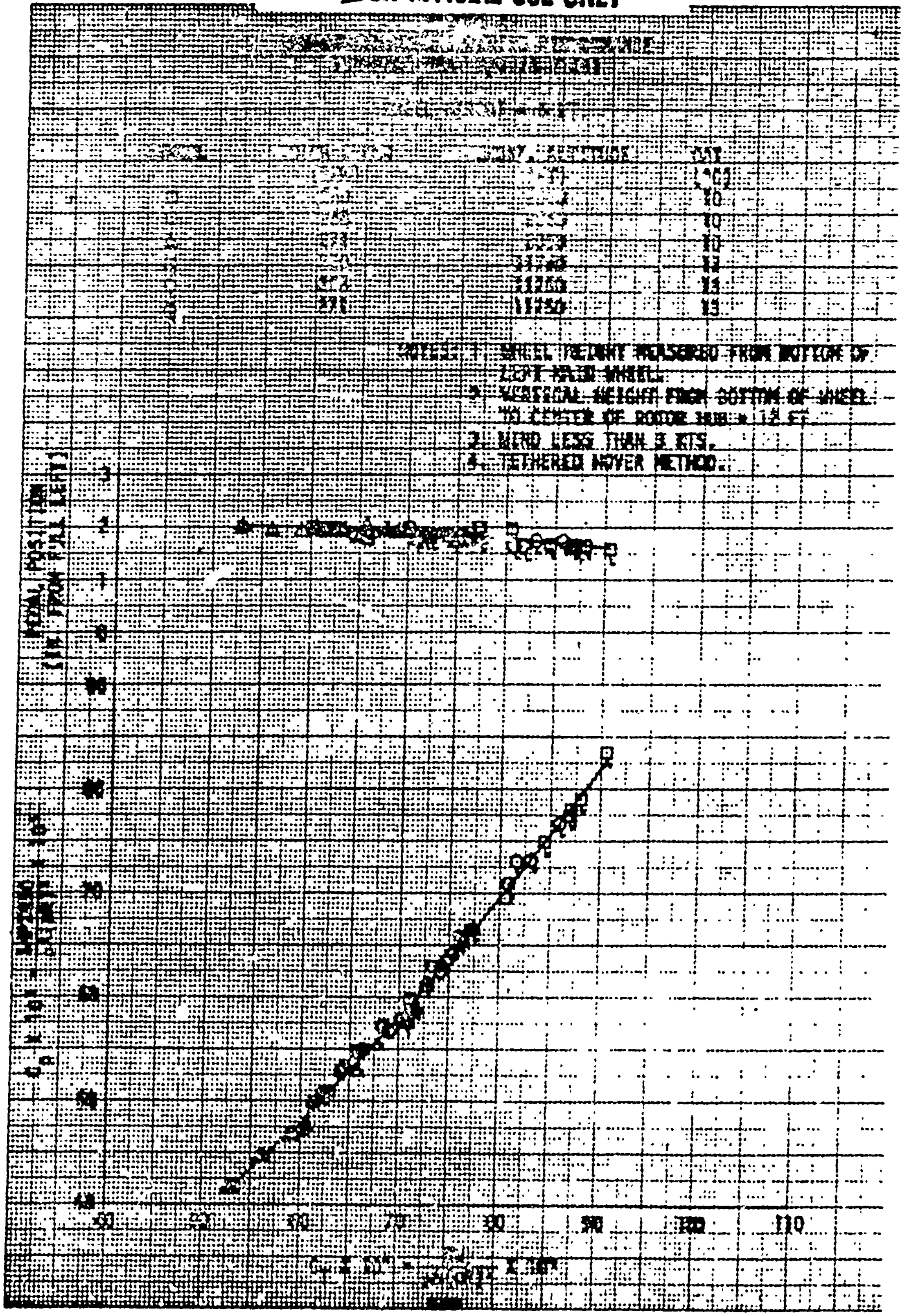

NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL.
2. VERTICAL HEIGHT FROM BOTTOM OF WHEEL TO CENTER OF ROTOR HUB = 12 FT.
3. WIND LESS THAN 3 KTS.
4. TETHERED HOVER METHOD.
PEDAL POSITION (IN. FROM FULL LEFT)
3
2
1
0
40
50
60
70
80
90
100
110

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FIGURE [illegible]
NON-DIMENSIONAL HOVERING PERFORMANCE
YUH-60A USA S/N 73-21651
WHEEL HEIGHT = 100 FT

SYMBOL	ROTOR SPEED (RPM)	DENSITY ALTITUDE (FT)	OAT (°C)
□	250	2570	11
○	258	2570	11
△	271	2570	11
□ (flagged)	250	5550	20
○ (flagged)	258	5550	20
△ (flagged)	271	5550	20

NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL.
2. VERTICAL HEIGHT FROM BOTTOM OF WHEEL TO CENTER OF ROTOR HUB = 12 FT.
3. WINDS LESS THAN 3 KTS.
4. TETHERED HOVER METHOD.

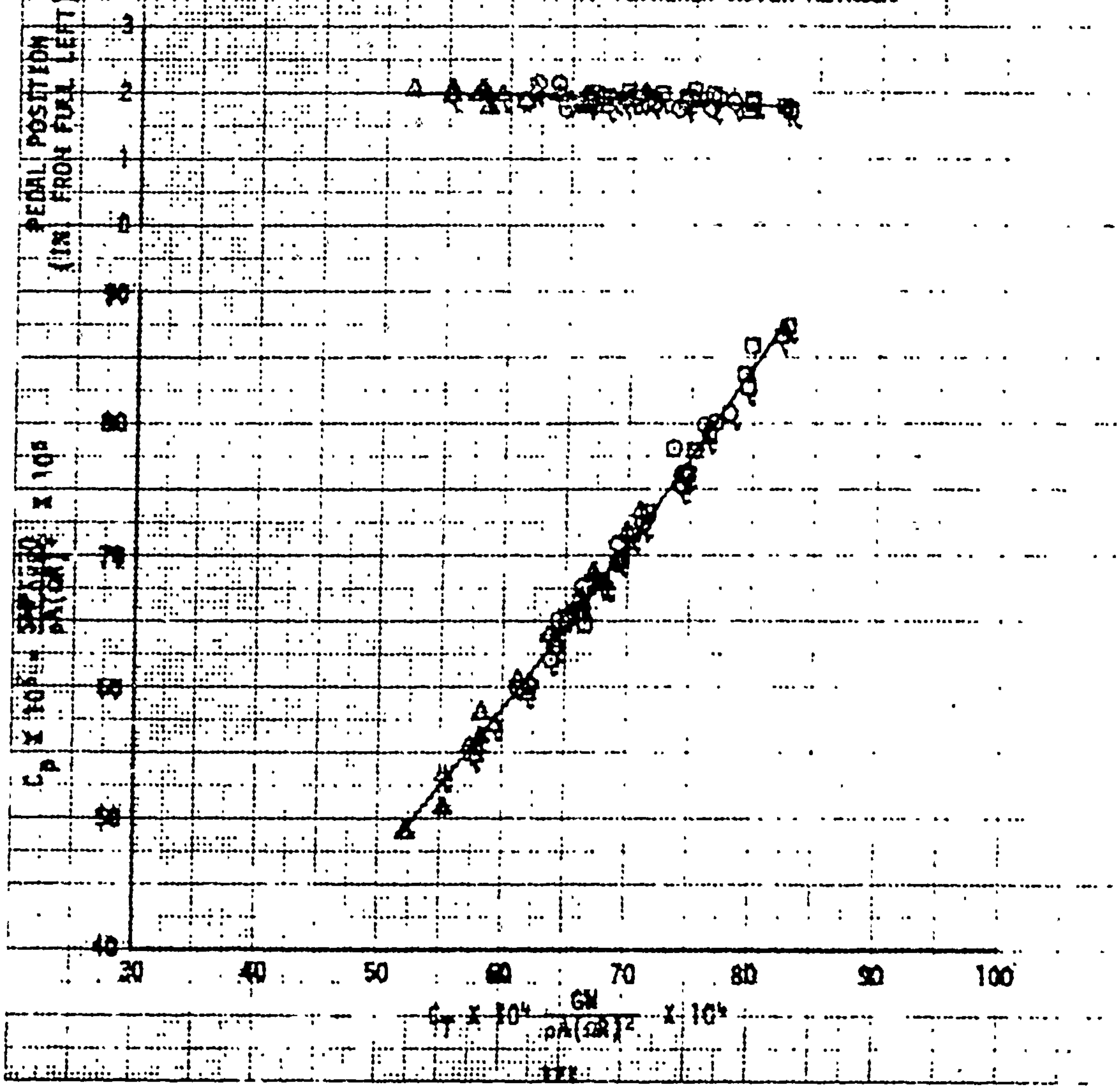

FIGURE 4
NON-DIMENSIONAL TAIL ROTOR PERFORMANCE
YUH-60A USA S/N 73-21651

WHEEL HEIGHT = 5 FT

SYMBOL	ROTOR SPEED (RPM)	DENSITY ALTITUDE (FT)	OAT (°C)
□	250	2050	10
○	258	2050	10
△	271	2050	10

NOTES:
1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL.
2. VERTICAL HEIGHT FROM BOTTOM OF WHEEL TO CENTER OF ROTOR HUB = 12 FT.
3. WIND LESS THAN 3 KTS.
4. TETHERED HOVER METHOD.

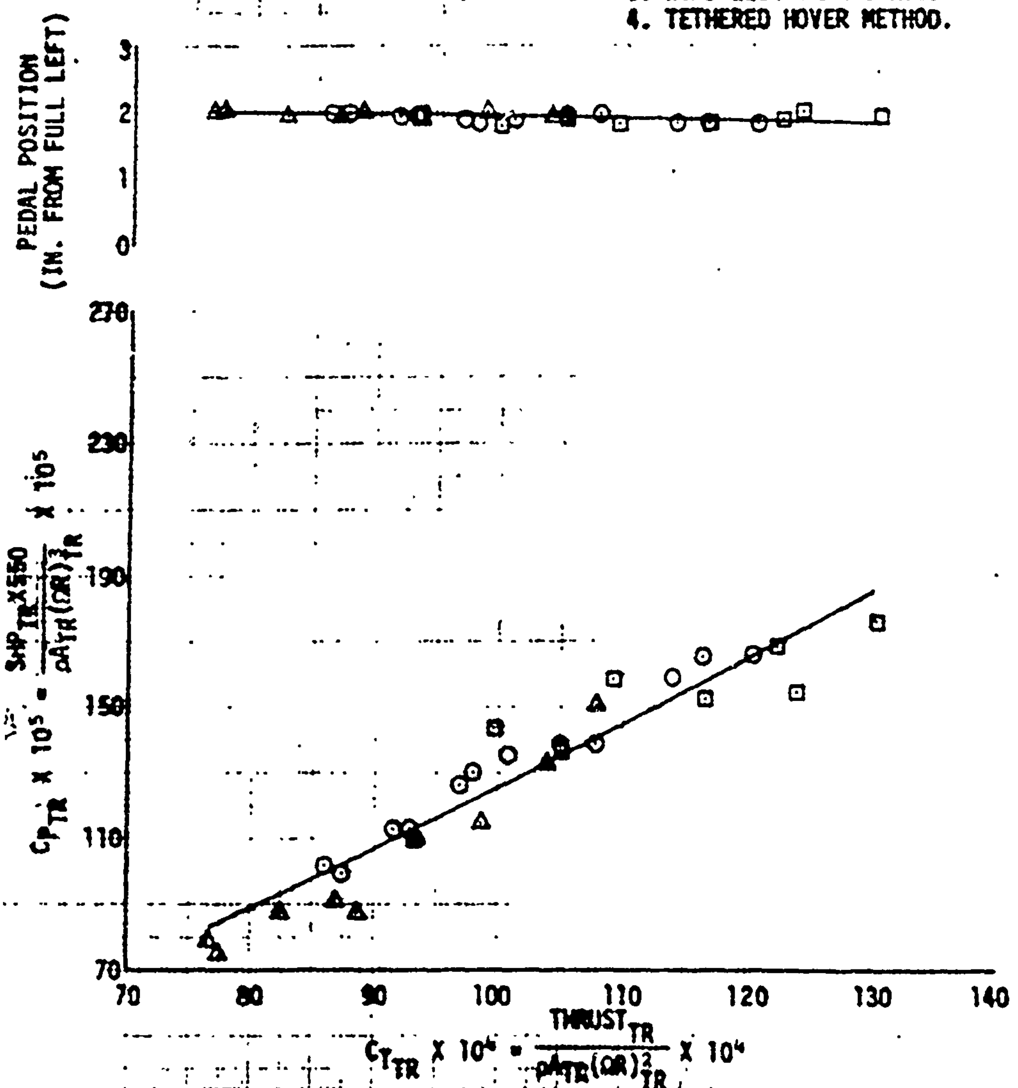

FIGURE 5
NON-DIMENSIONAL TAIL ROTOR PERFORMANCE
YUH-60A USA S/N 73-21651

WHEEL HEIGHT = 100 FT

SYMBOL	ROTOR SPEED (RPM)	DENSITY ALTITUDE (FT)	OAT (°C)
□	250	2570	11
○	258	2570	11
△	271	2570	11

NOTES: 1. WHEEL HEIGHT MEASURED FROM BOTTOM OF LEFT MAIN WHEEL.
2. VERTICAL HEIGHT FROM BOTTOM OF WHEEL TO CENTER OF ROTOR HUB = 12 FT.
3. WINDS LESS THAN 3 KTS.
4. TETHERED HOVER METHOD.

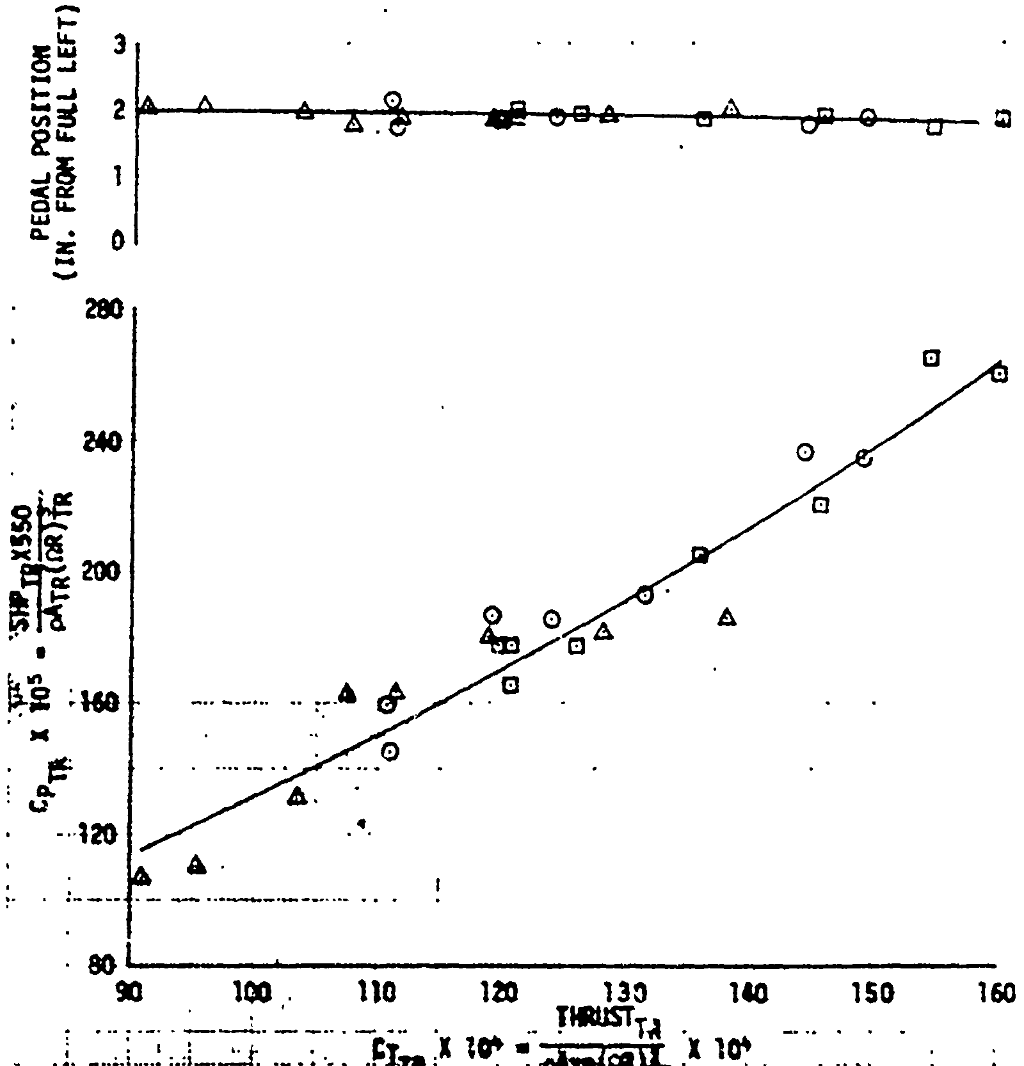

FIGURE 6
TAKEOFF PERFORMANCE
YUH-60A USA S/N 73-21651

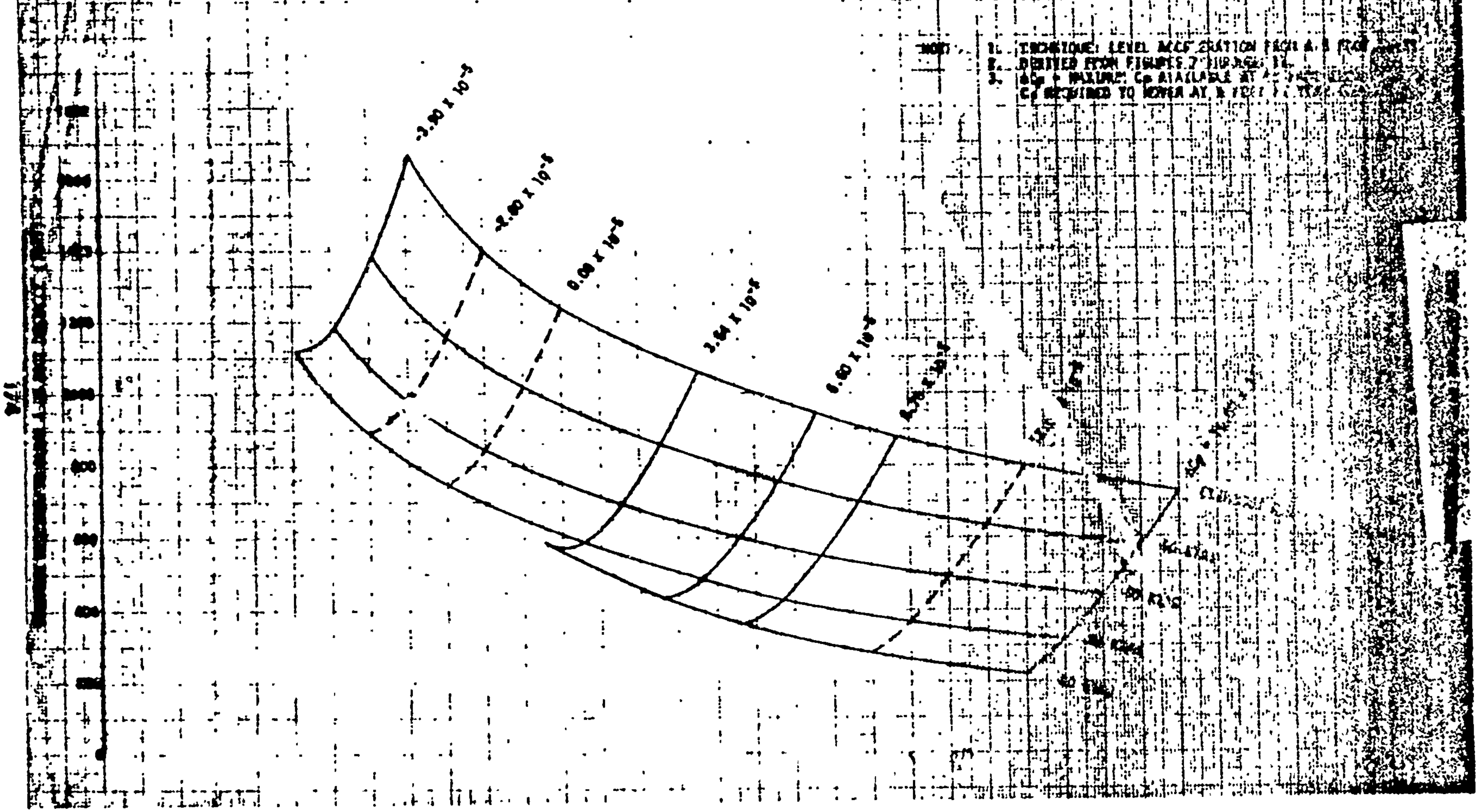

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FIGURE [illegible]

TAKEOFF PERFORMANCE

YUH-61A USA S/N 73-21657

ENGINE S/N's 207213, 207273

AVG GROSS WEIGHT (LB)	AVG C.G. LOCATION LONG (IN.)	AVG C.G. LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $C_T \times 10^4$	CONFIGURATION
[illegible]	[illegible]	[illegible]	[illegible]	10.0	258	[illegible]	NORMAL UTILITY

NOTES: 1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 5 FOOT HOVER.
2. WINDS LESS THAN 3 KNOTS.
3. FAIRING OBTAINED FROM FIGURE 6.

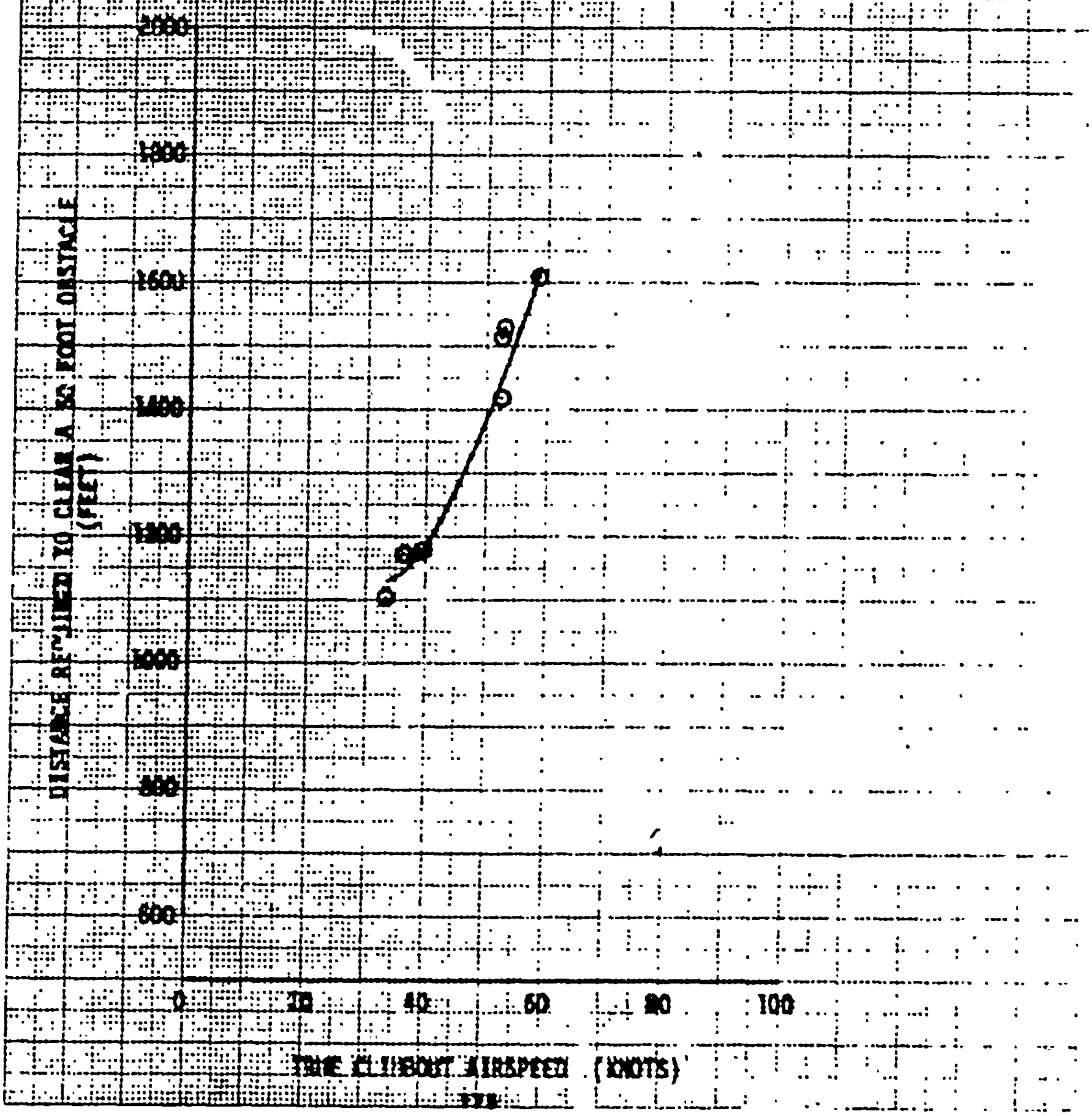

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FIGURE [illegible]

[illegible] PERFORMANCE

([illegible] USA S/N 73-21[illegible])

ENGINE S/N's 207213, 207270

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG HD (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	$C_T \times 10^4$	CONFIGURATION
20200	347.9(FWD)	0.4(RT)	5520	20.0	258	3.54	[illegible] UTILITY

NOTES: 1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 5 FOOT HOVER.
2. WINDS LESS THAN 3 KNOTS.
3. FAIRING OBTAINED FROM FIGURE 6.

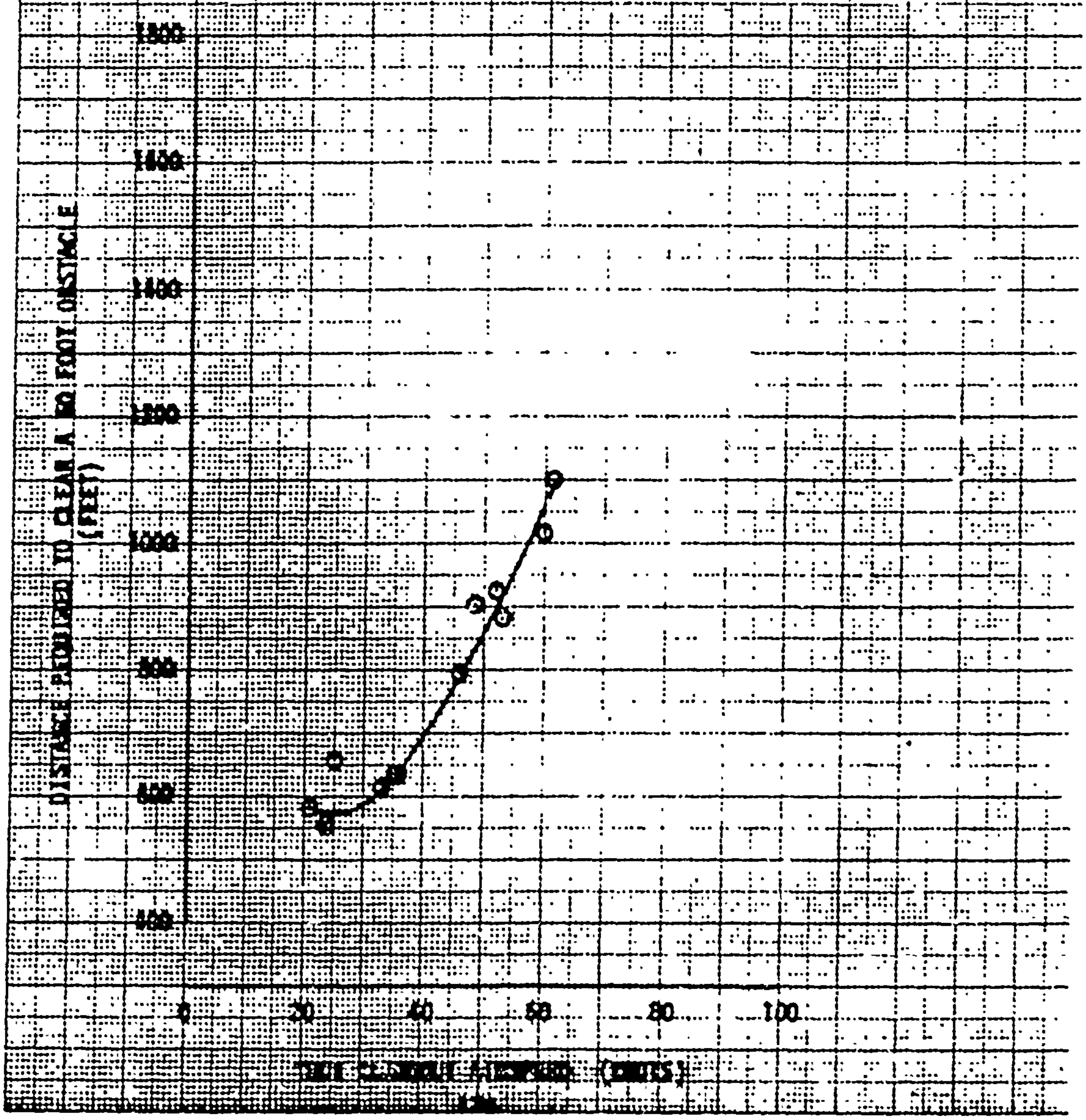

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FIGURE [illegible]

TAKEOFF PERFORMANCE

[illegible] USA S/N [illegible]

ENGINE S/N'S [illegible]

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG [illegible]	CONFIGURATION
20140	[illegible] (FWD)	2.4(RT)	6020	18.0	258	[illegible].60	NORMAL UTILITY

NOTES:
1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 5 FOOT HOVER.
2. WINDS LESS THAN 3 KNOTS.
3. FAIRING OBTAINED FROM FIGURE 6.

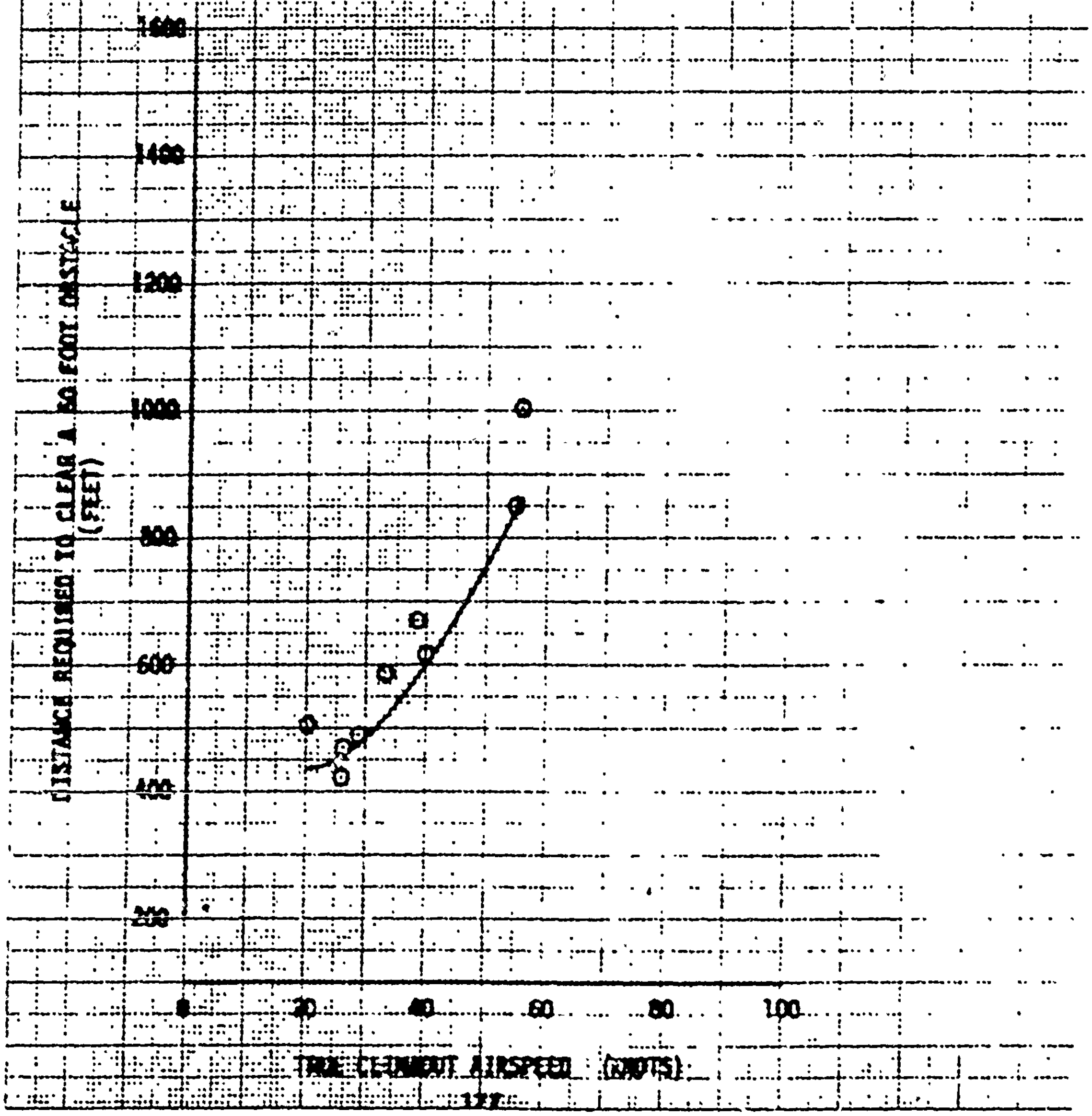

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FIGURE [illegible]

[illegible]

[illegible]

ENGINE S/N's 207213, 207270

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG Hd (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG [illegible]	CONFIGURATION
[illegible]	347.[illegible]	[illegible]	[illegible]	[illegible]	253	[illegible]	NORMAL UTILITY

NOTES: 1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 5 FOOT HOVER.
2. WINDS LESS THAN 3 KNOTS.
3. FAIRING OBTAINED FROM FIGURE [illegible].

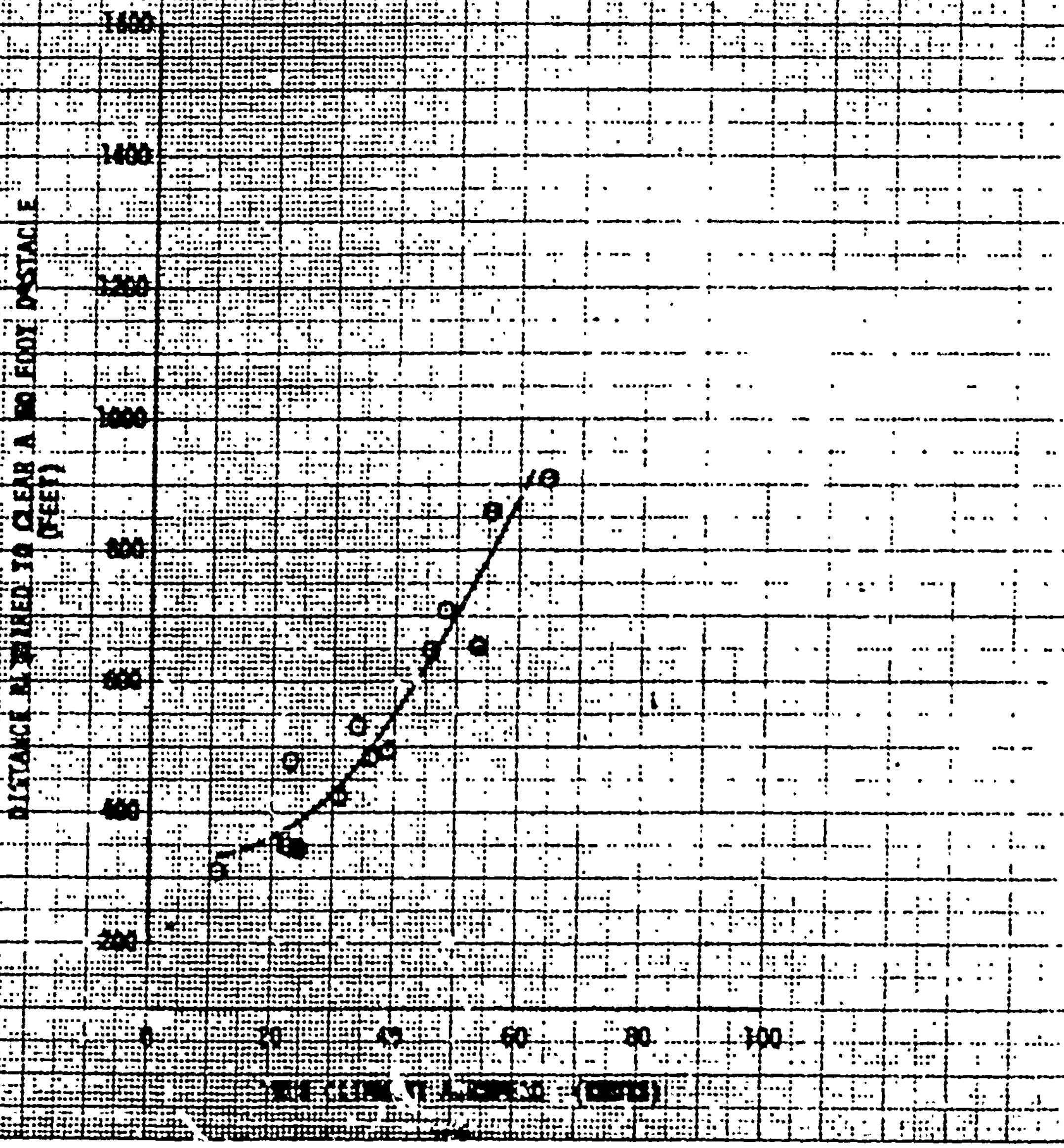

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FIGURE 11

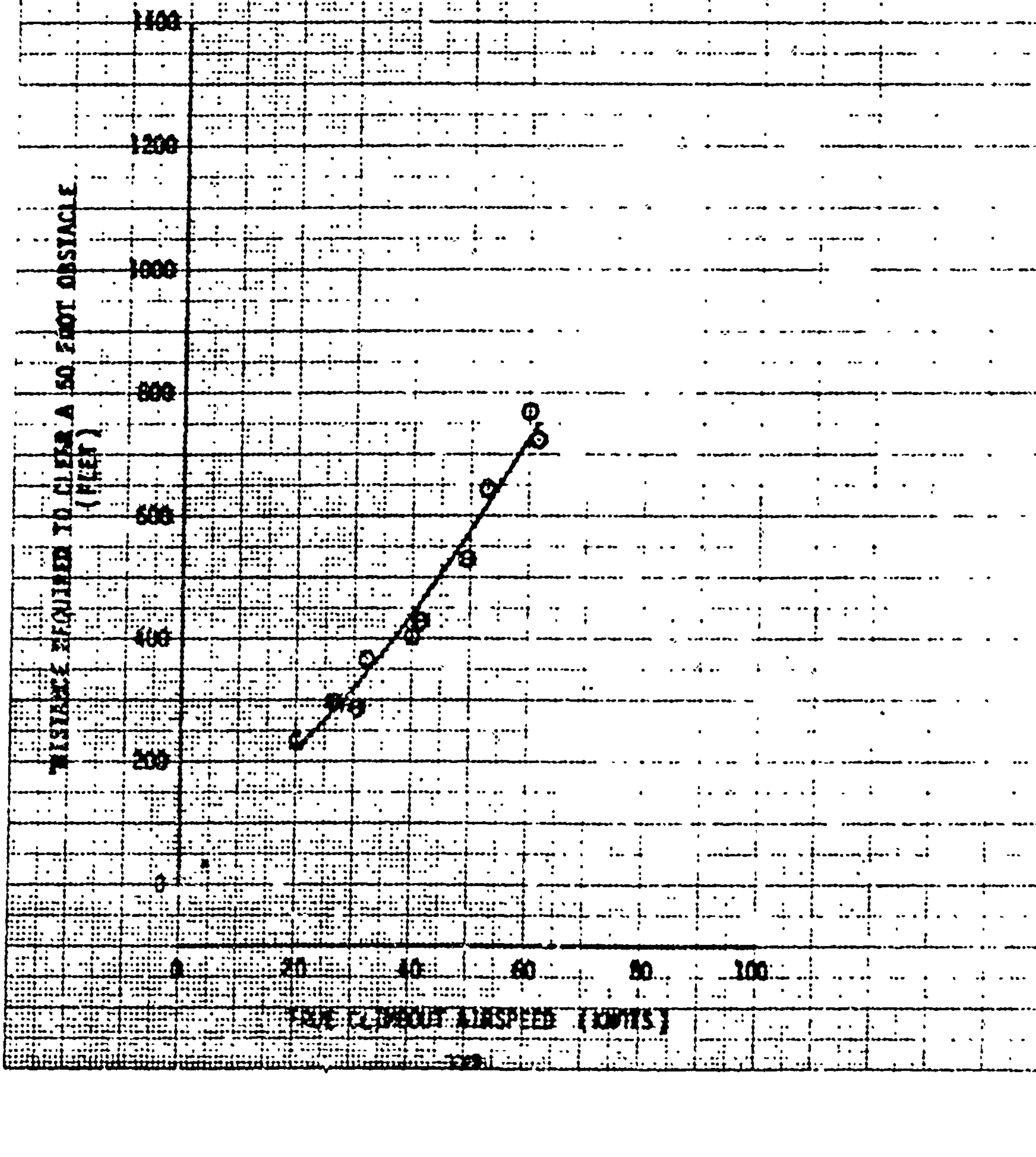
TAKEOFF PERFORMANCE

UH-1H USA S/N 72-21651

ENGINE S/N's 207273, 207270

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG $\Delta C_P \times 10^5$	CONFIGURATION
17240	137.3(FWD)	0.4(RT)	6260	25.5	25B	16.00	NORMAL UTILITY

NOTES: 1. DATA OBTAINED DURING LEVEL ACCELERATION FROM A 5 FOOT HOVER.
2. WINDS LESS THAN 3 KNOTS.
3. FAIRING OBTAINED FROM FIGURE 8.



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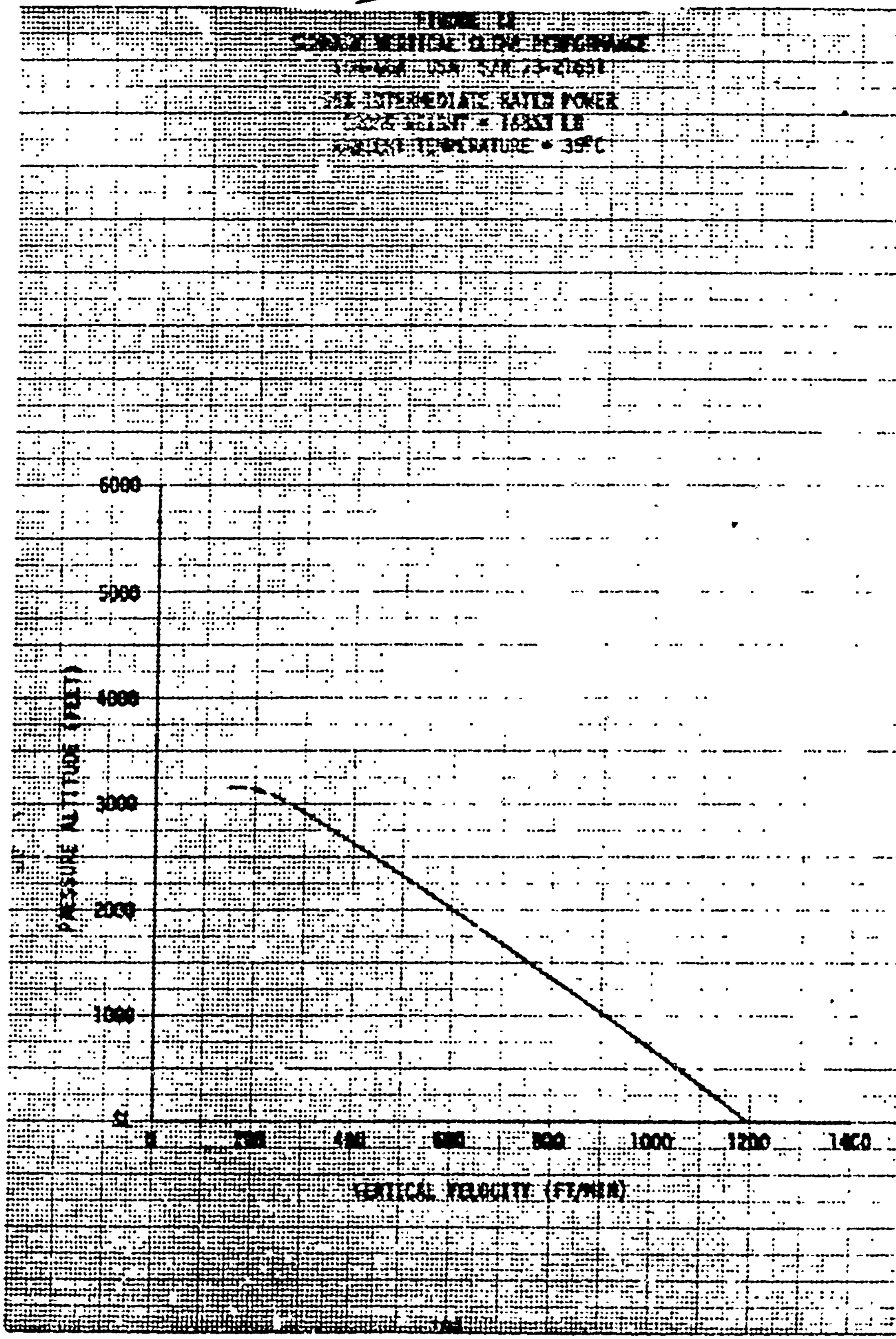
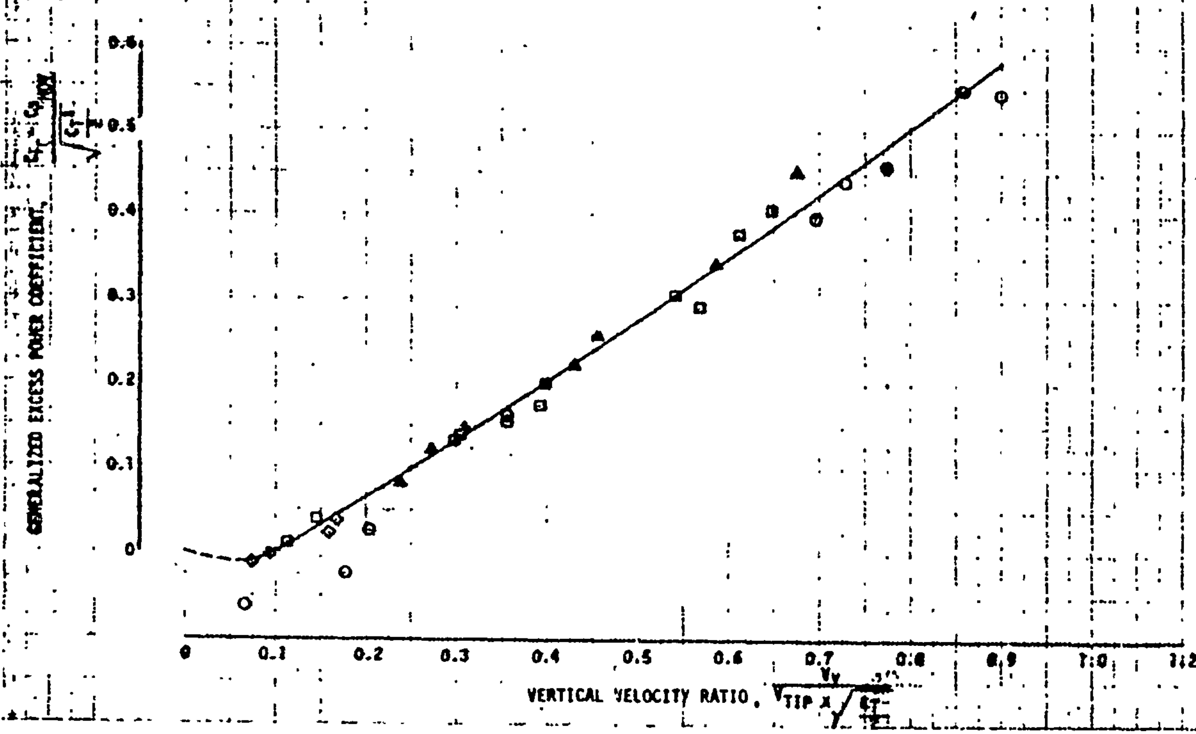


FIGURE 13
NON-DIMENSIONAL VERTICAL CLIMB PERFORMANCE
YUH-60A USA S/N 73-21651

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENSITY ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
⊖	15860	347.2(FWD)	0.5(RT)	3580	13.0	258	0.006245	NORMAL UTILITY
◻	16540	347.5(FWD)	0.5(RT)	4020	20.5	258	0.006597	NORMAL UTILITY
▲	15560	347.1(FWD)	0.4(RT)	6040	19.0	258	0.006597	NORMAL UTILITY
◇	19280	347.7(FWD)	0.4(RT)	3080	13.5	258	0.007478	NORMAL UTILITY



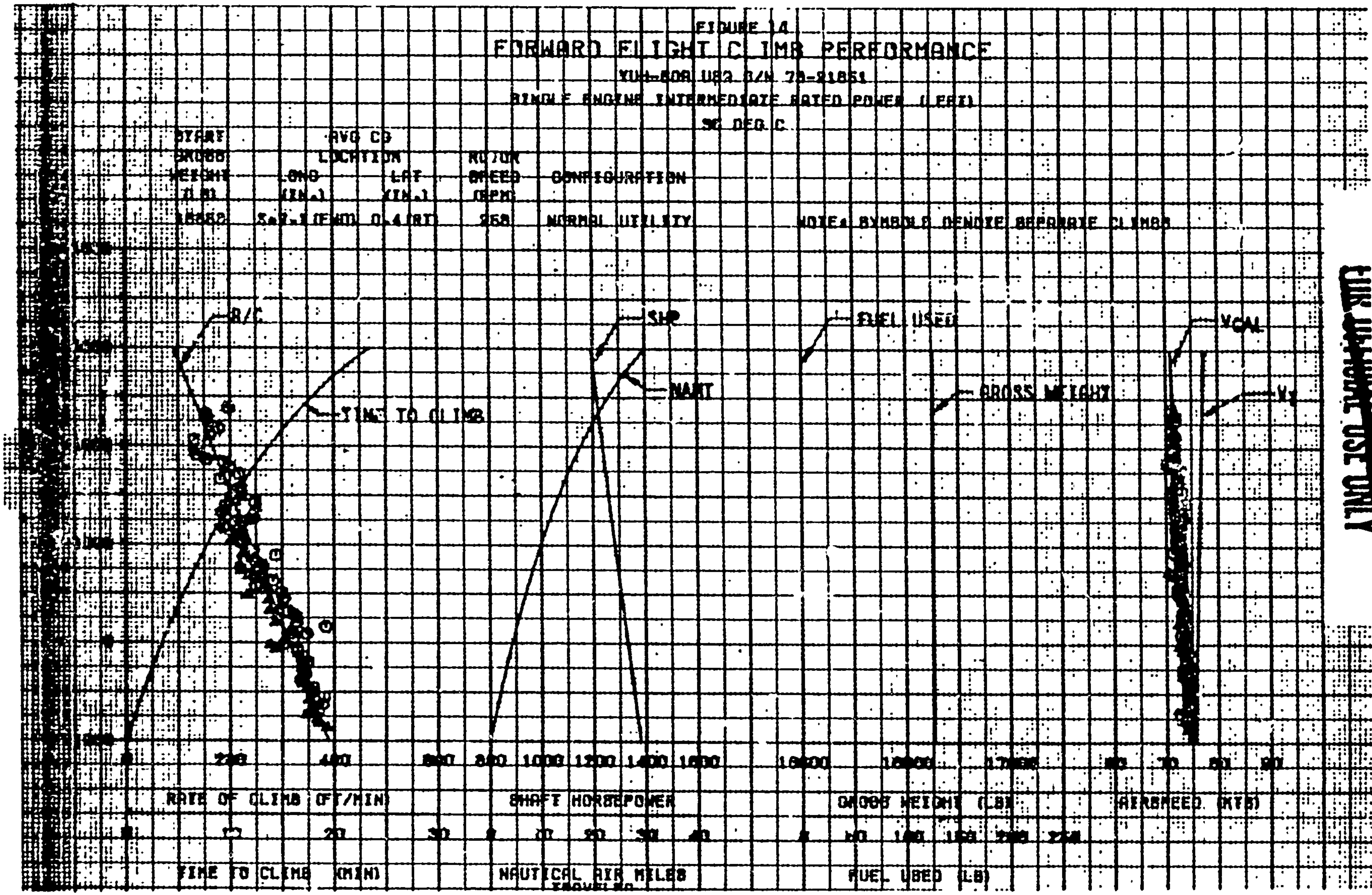


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FIGURE 15

VARIATION IN RATE OF CLIMB AS A FUNCTION OF GROSS WEIGHT

YUH-60A USA S/N 73-21651

SYM	H_p (FT)	ROTOR SPEED (RPM)	SHP
○	1000	258	1384
□	2000	258	1345
◇	3000	258	1304
△	4000	258	1265
▽	5000	258	1228

NOTES: 1. TEST CONDUCTED AT BEST RATE OF CLIMB SPEED AS DETERMINED FROM FIGURES 17 AND 18.

2. $K_W = \frac{(R/C_2 - R/C_1)(GW_1 \times GW_2)}{SHP \times 33000\ (GW_1 - GW_2)}$

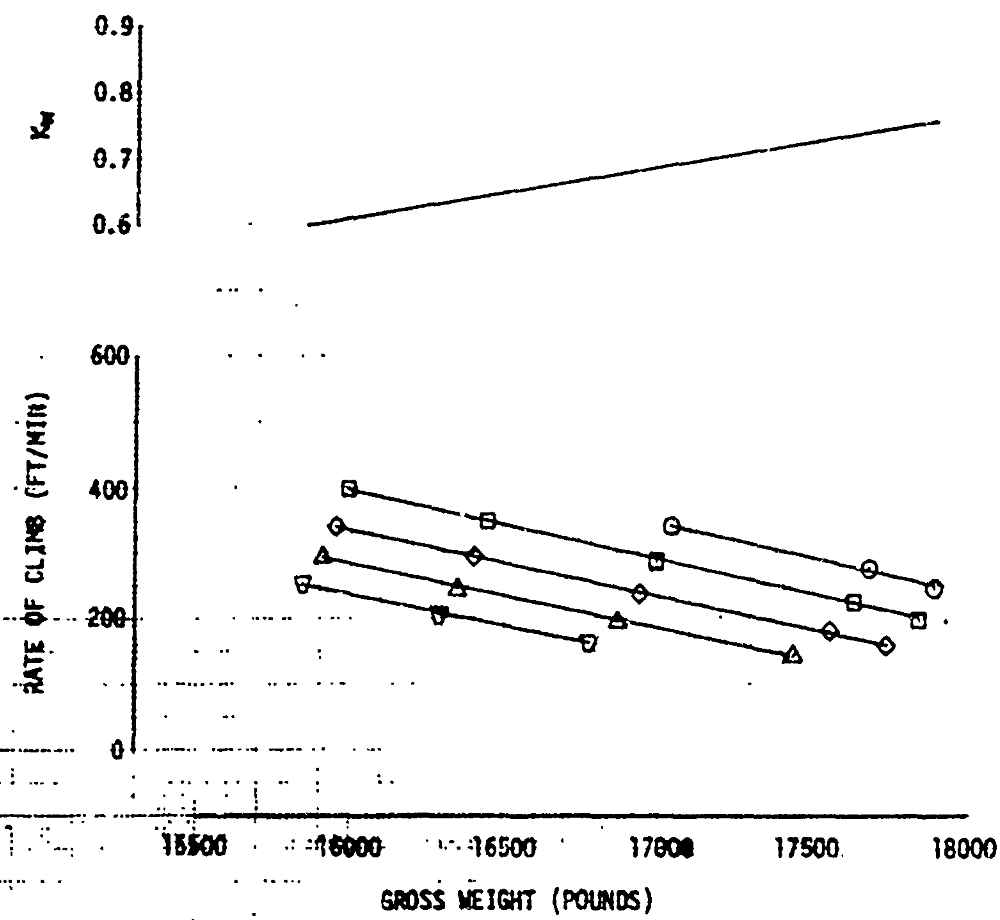

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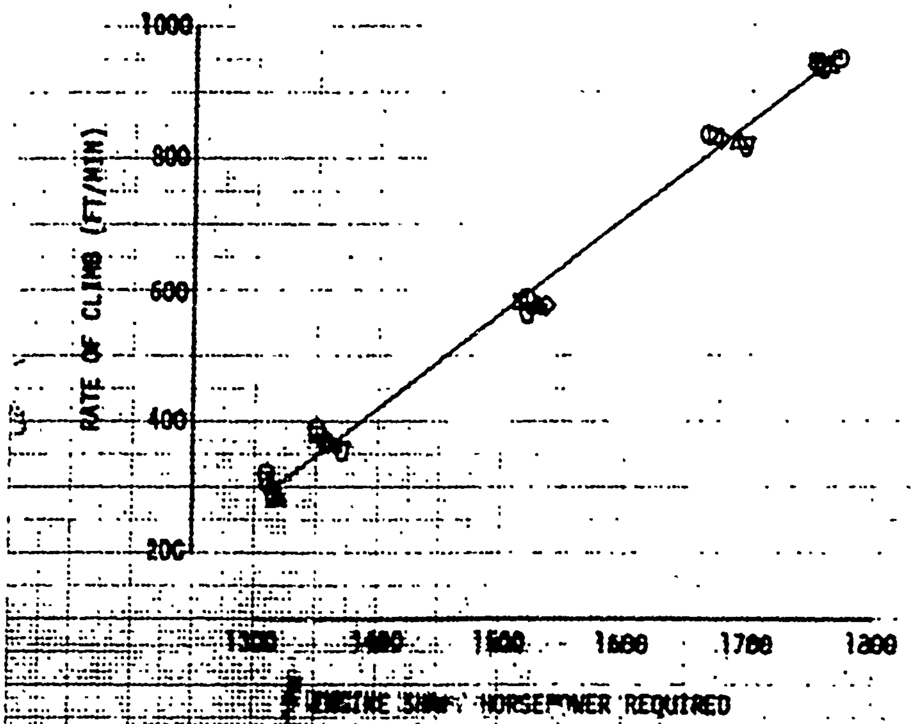
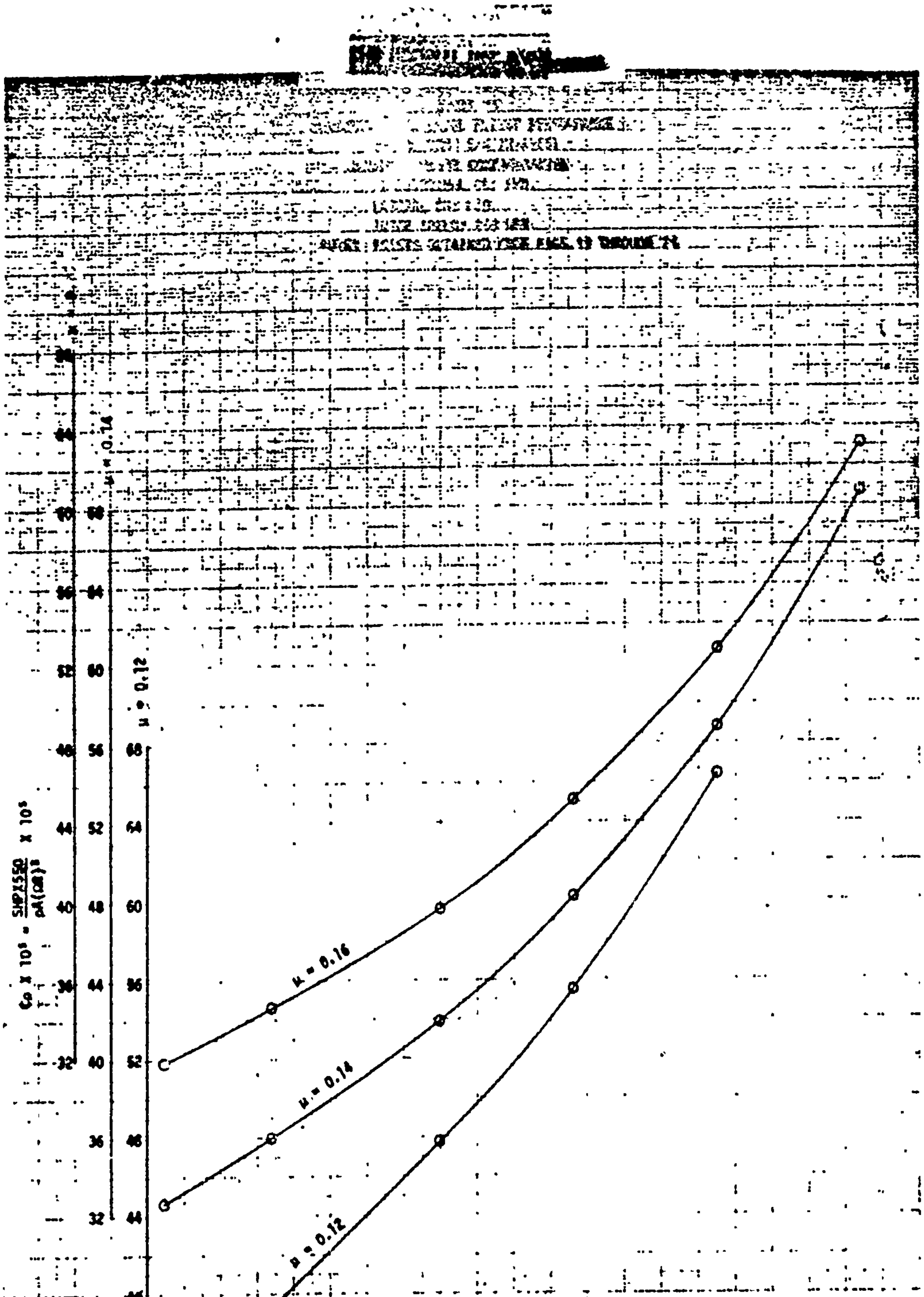
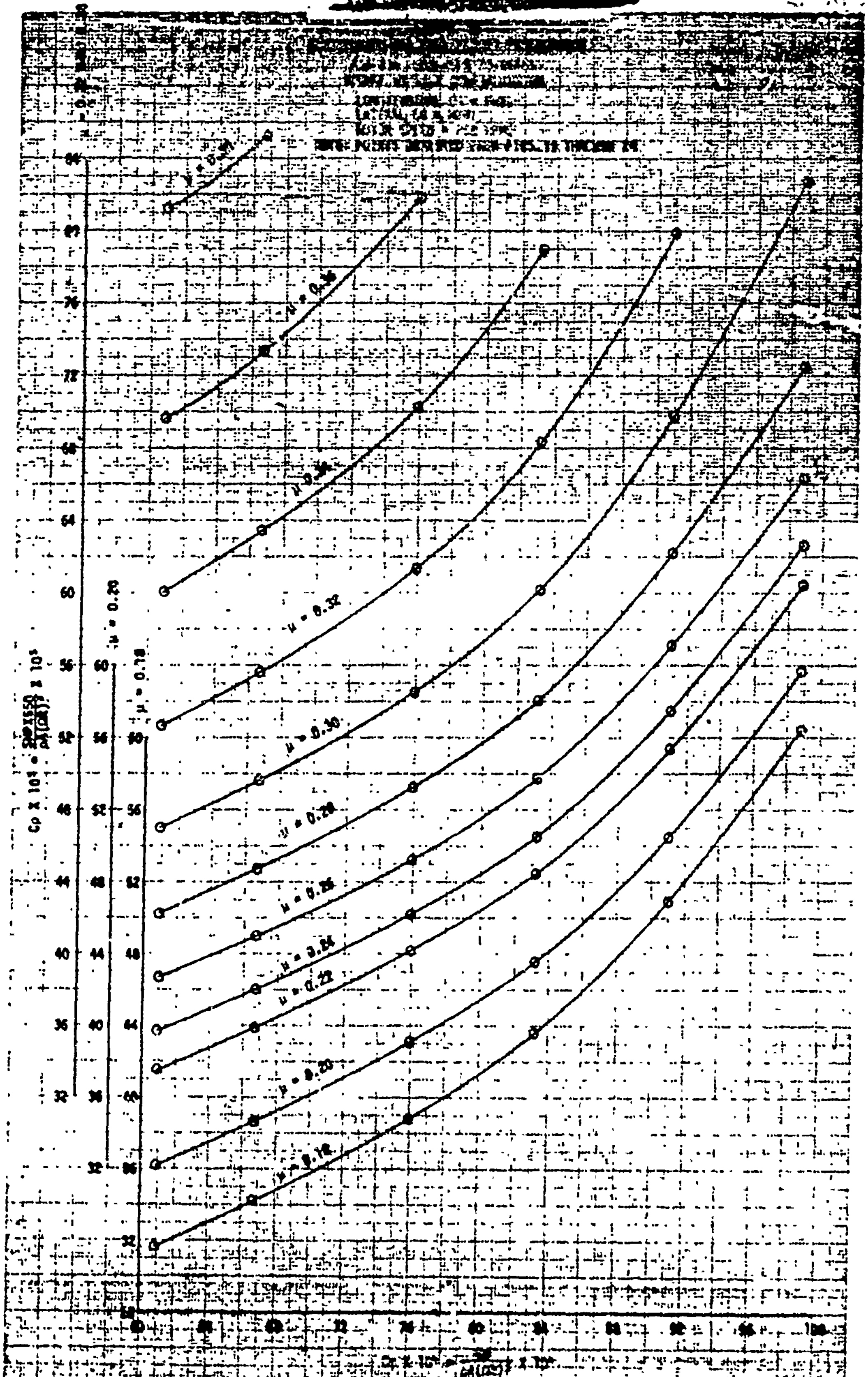
FIGURE 16
VARIATION IN RATE OF CLIMB AS A FUNCTION OF SHAFT HORSEPOWER
YUH-60A USA S/N 73-21651

SYM	H_D (FT)	ROTOR SPEED (RPM)	GROSS WEIGHT (LB)
⊙	1000	258	16790
◻	2000	258	16790
◇	3000	258	16790
△	4000	258	16790
▽	5000	258	16790

NOTES: 1. TEST CONDUCTED AT BEST RATE OF CLIMB SPEED AS DETERMINED FROM FIGURES 17 AND 18.
2. $K_p = \left[\frac{\Delta R/C}{\Delta SHP} \times \frac{GW}{33000}\right] = 0.74$




μ = 0.16
μ = 0.14
μ = 0.12



μ = 0.32
μ = 0.30
μ = 0.28
μ = 0.26
μ = 0.22
μ = 0.20
μ = 0.18

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FIGURE 19
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15940	347.5(FWD)	0.4(RT)	2580	14.0	258	0.006090	NORMAL UTILITY

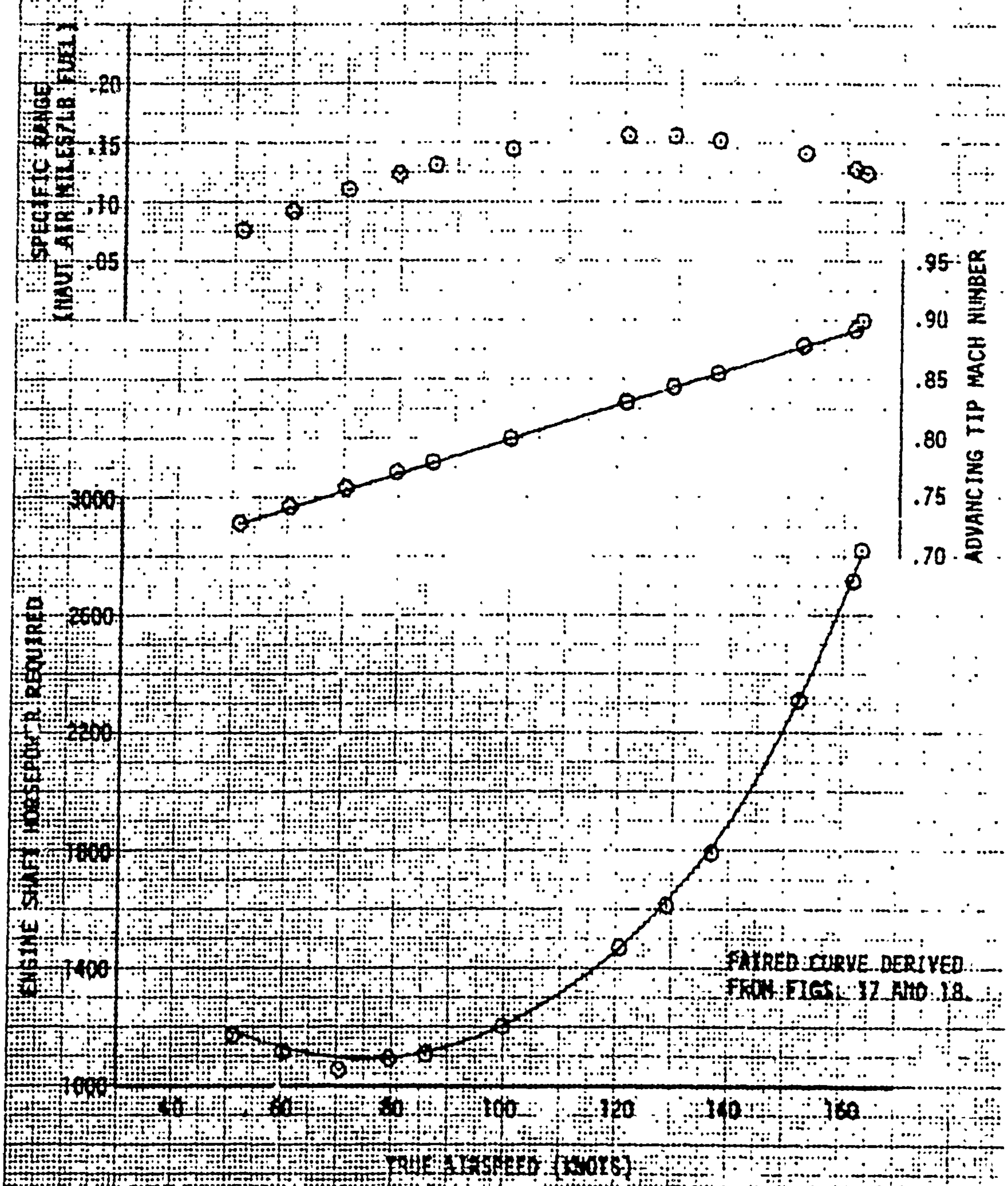

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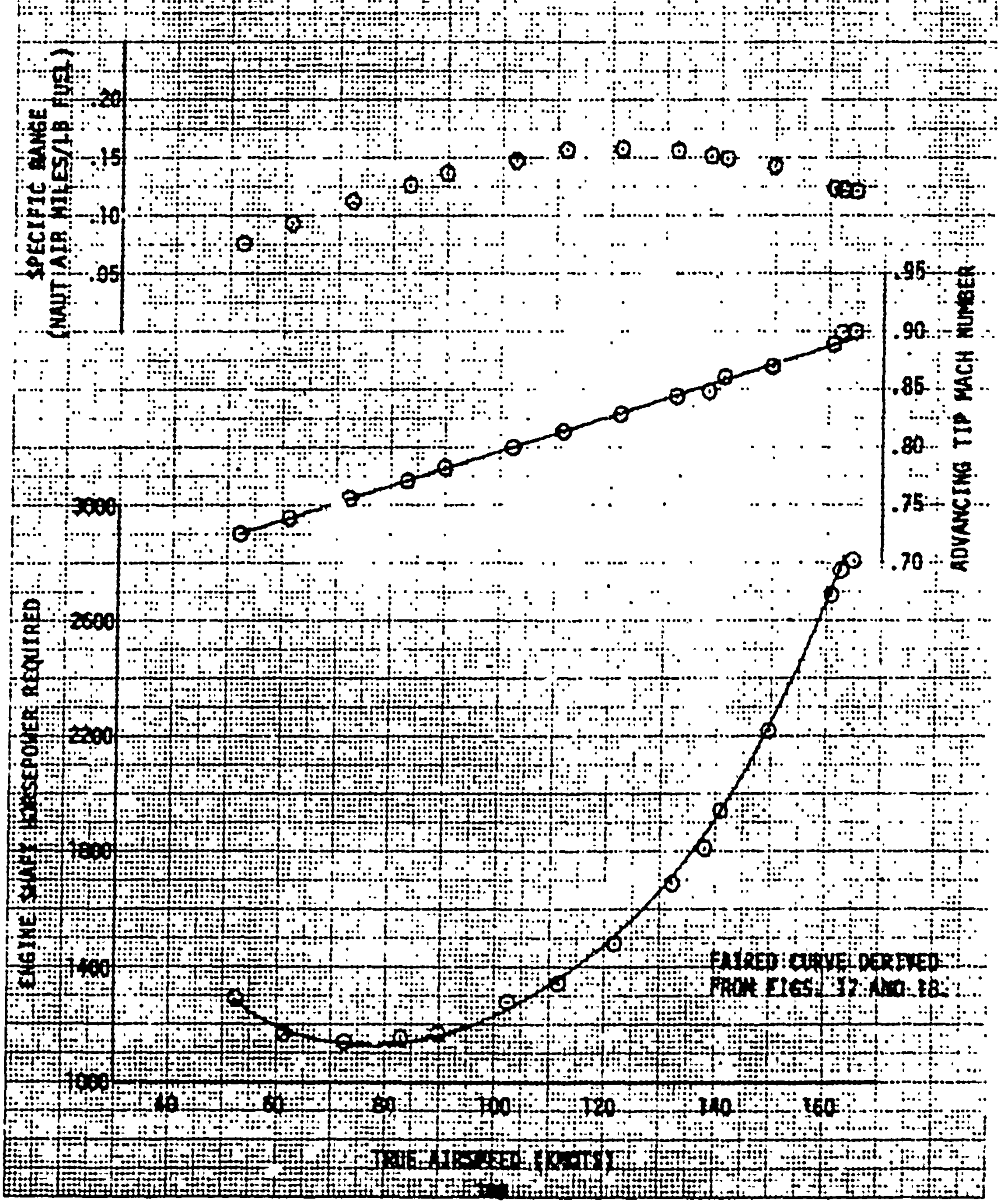
FIGURE 20
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

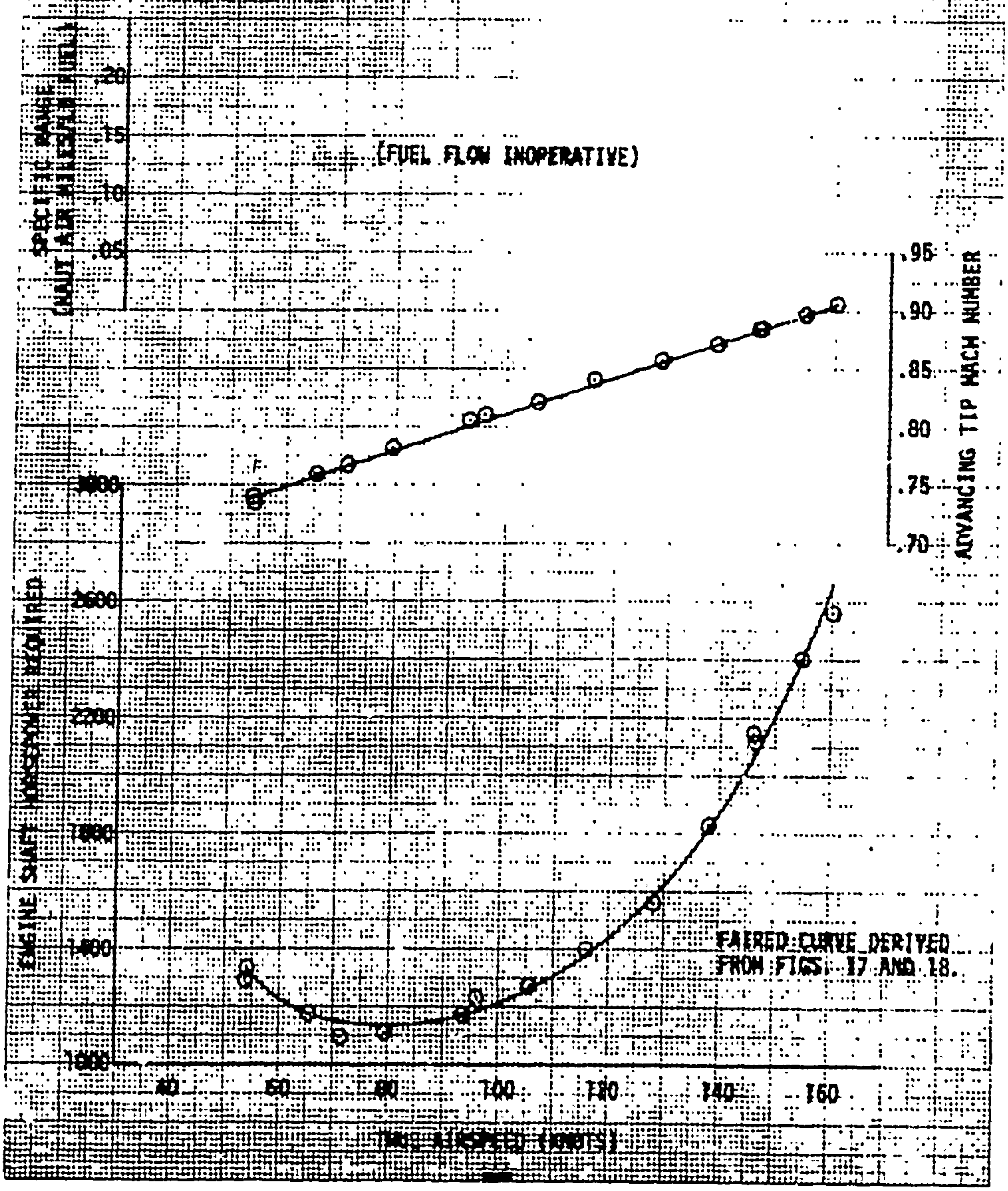
AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15740	347.7(FWD)	0.4(RT)	4000	18.0	258	0.006674	NORMAL UTILITY



~~FOR OFFICIAL USE ONLY~~

[illegible]

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
[illegible]	[illegible](FWD)	[illegible](RT)	8040	6.0	258	0.007595	NORMAL UTILITY



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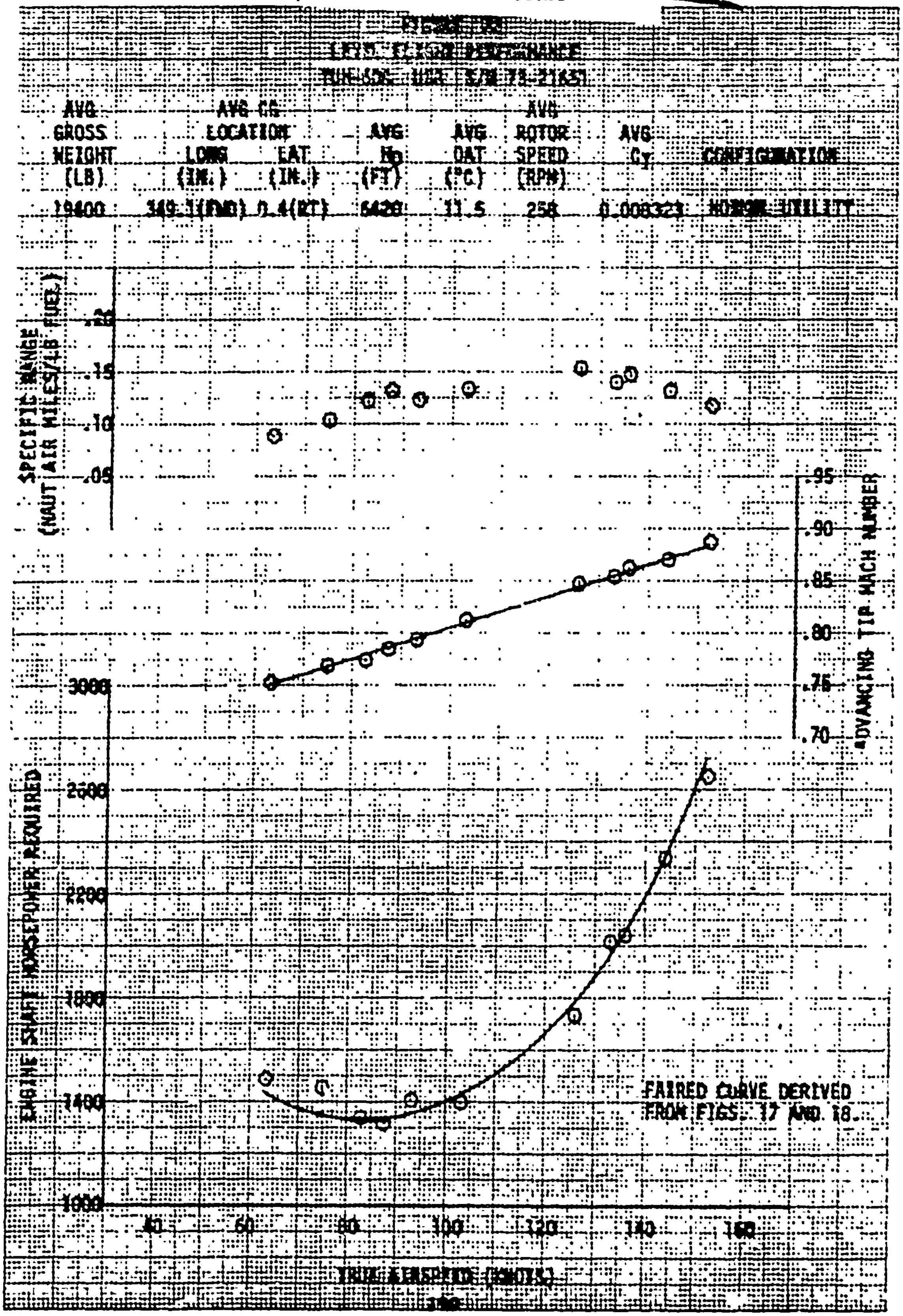

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FIGURE 23
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-216

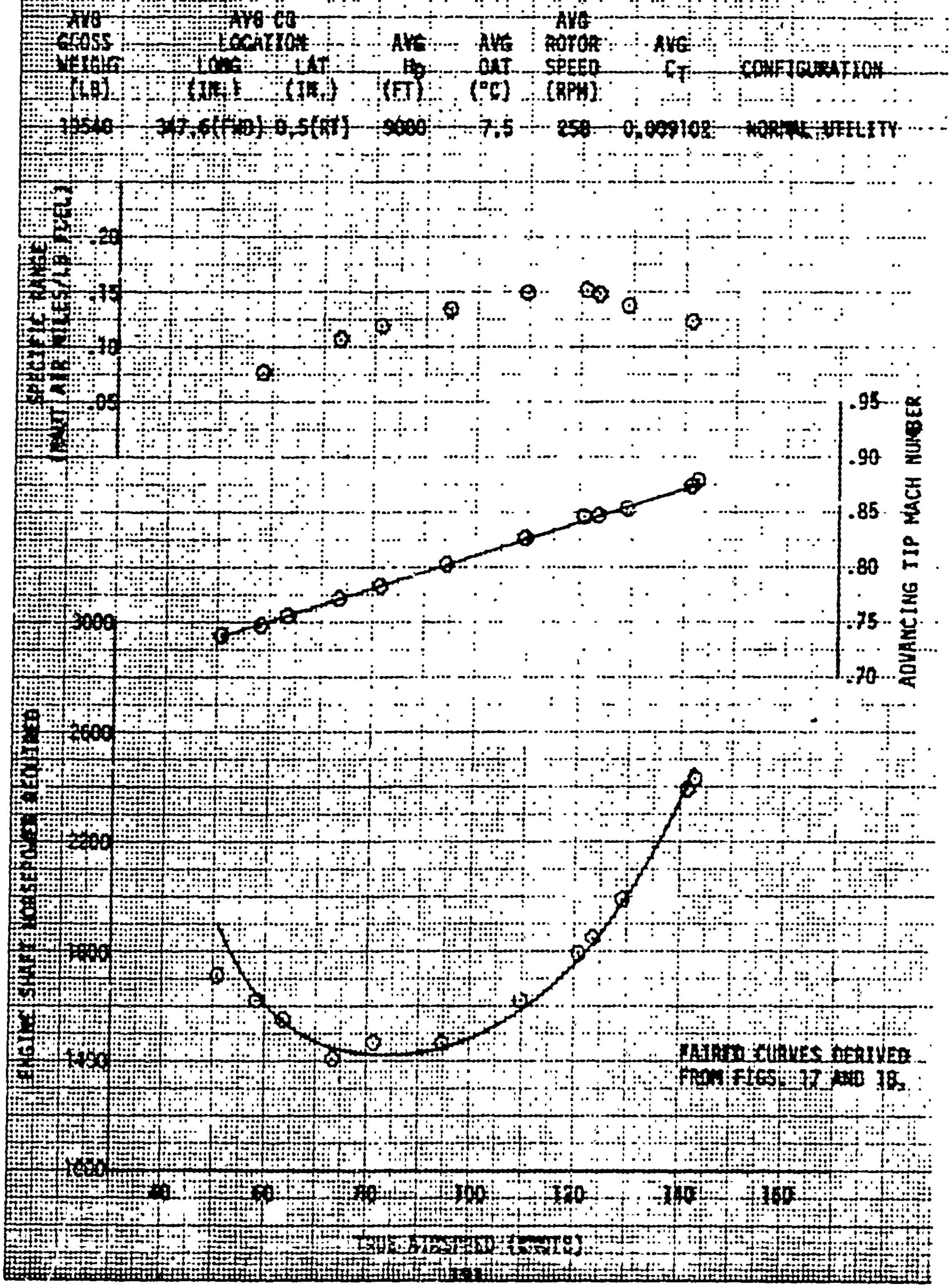

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FIGURE 28
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19600	347.6(FWD)	0.5([illegible])	11600	1.0	258	0.009884	NORMAL UTILITY

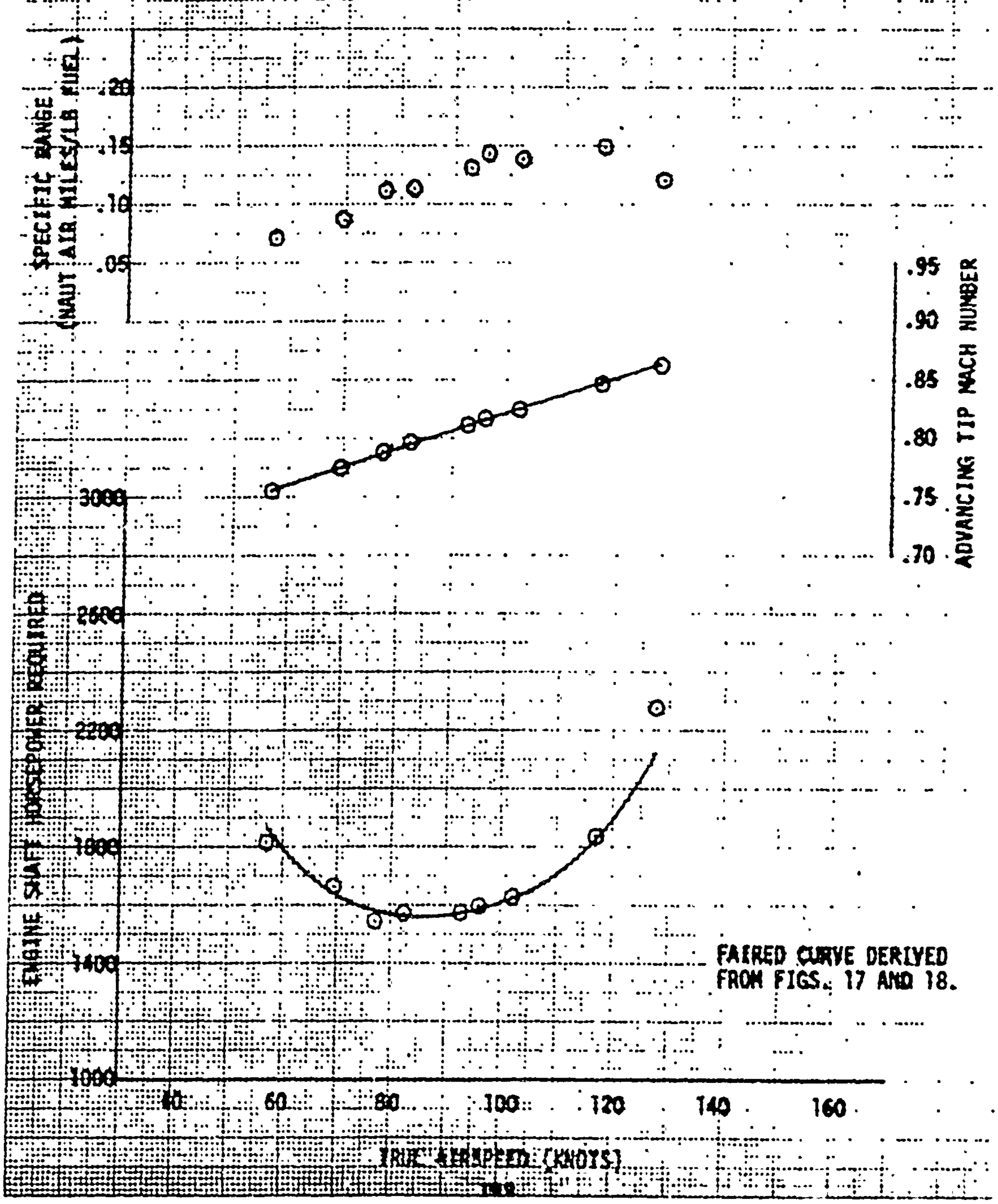

FIGURE 26
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
15960	346.8(FWD)	0.4(RT)	9660	15.0	258	0.007569	NORMAL UTILITY

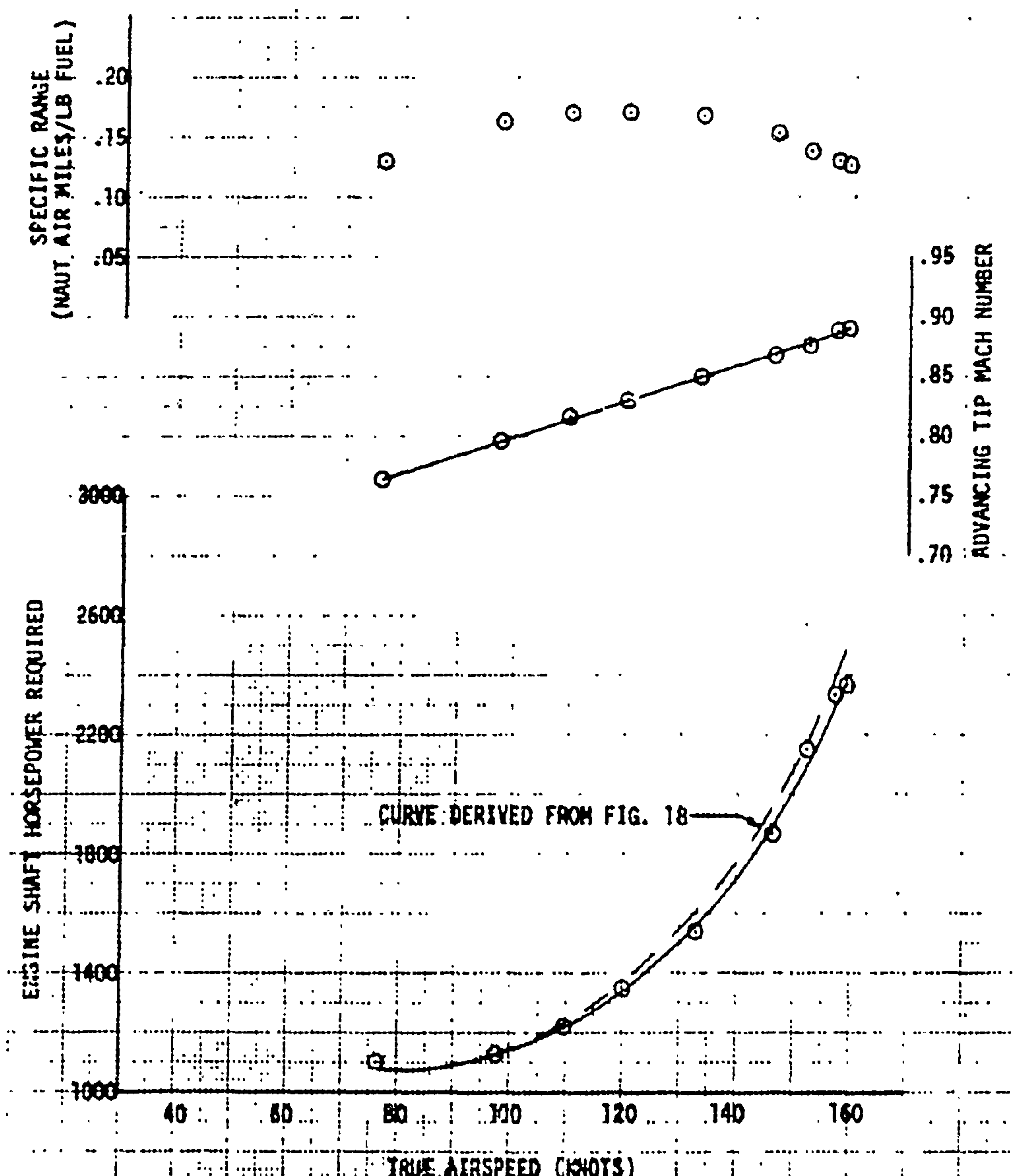

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FIGURE 2[illegible]

LEVEL FLIGHT PERFORMANCE

YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
17820	347.2(FWD)	0.4(RT)	7700	21.5	258	0.007953	NORMAL UTILITY

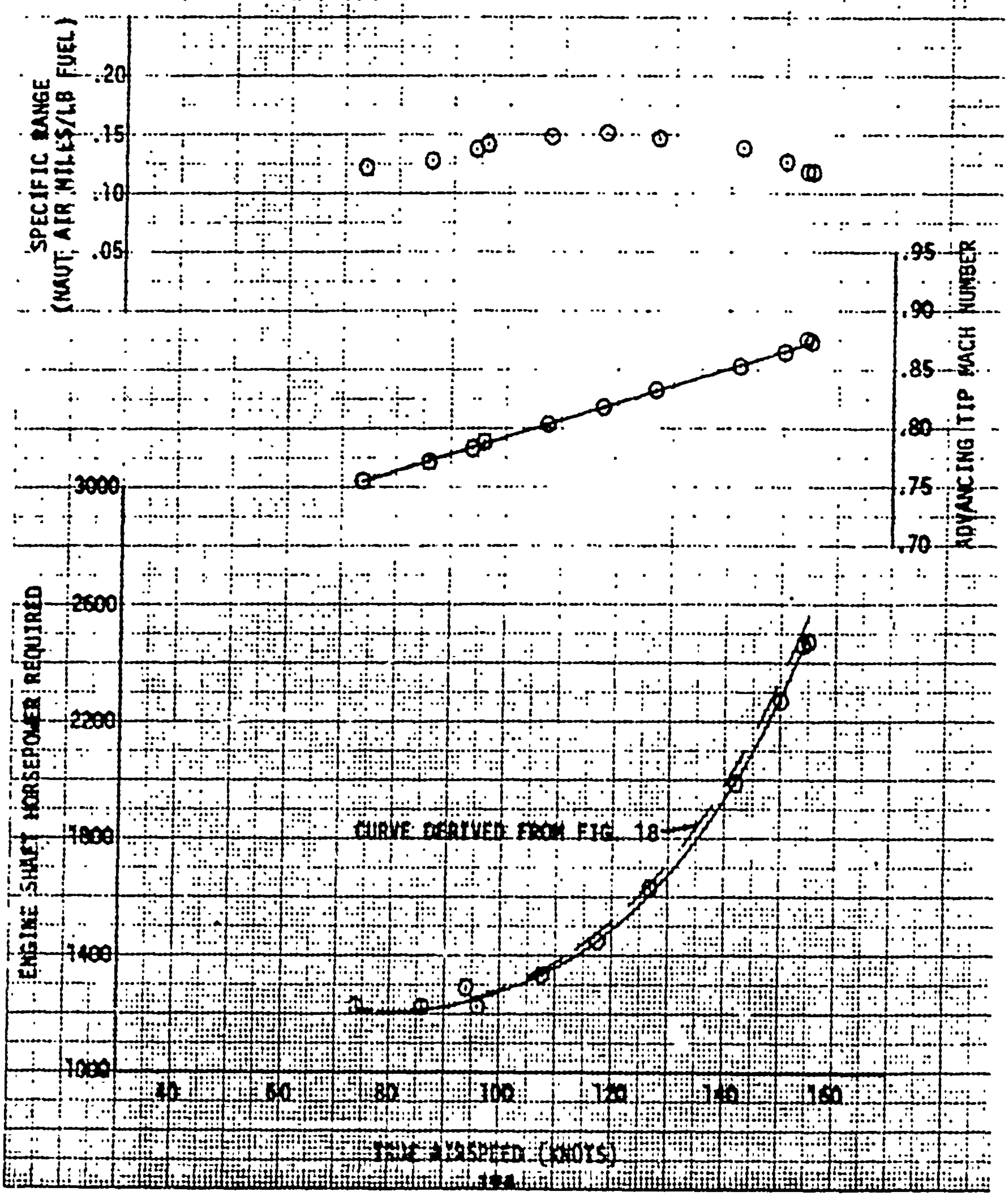

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FIGURE 27
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
16620	360.0(AFT)	0.4(RT)	4480	15.0	258	0.006722	NORMAL UTILITY

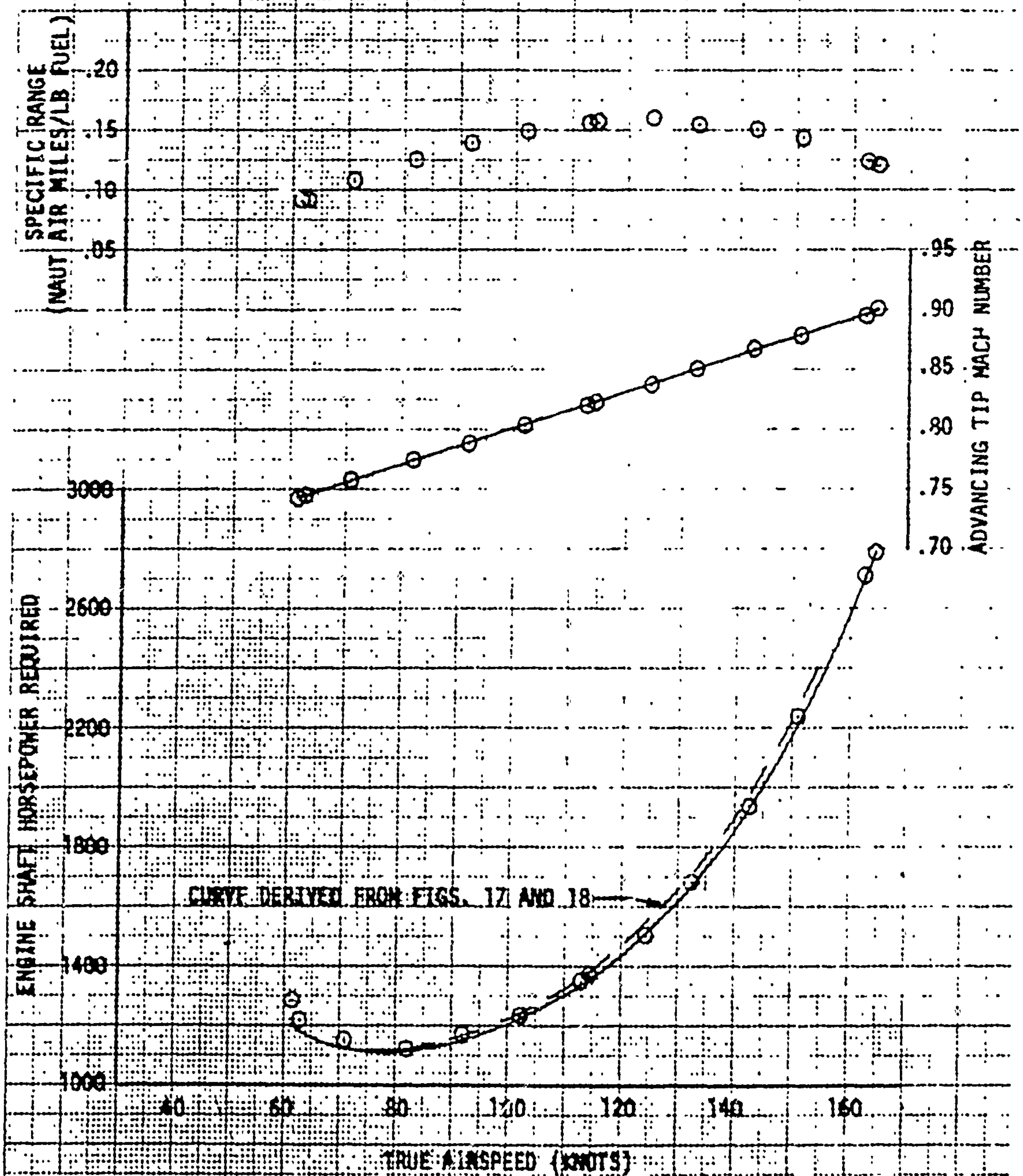

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FIGURE 28
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
16600	359.5(AFT)	0.3(RT)	8000	20.0	258	0.007477	NORMAL UTILITY

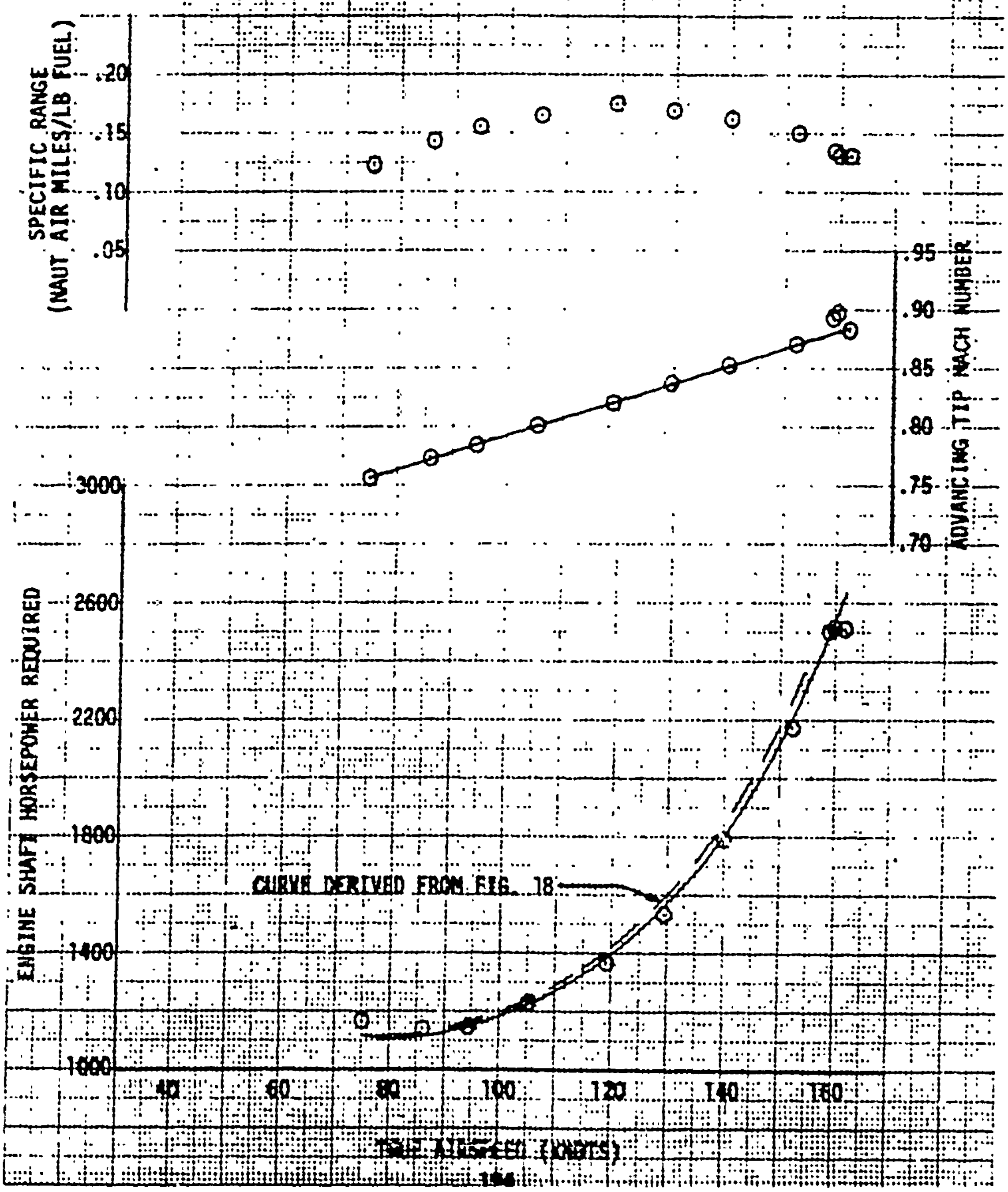

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FIGURE 29

LEVEL FLIGHT PERFORMANCE

YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
17100	348.3(FWD)	0.4(RT)	3940	18.5	258	0.006805	CARGO DOORS OPEN M-60 GUNS EXTENDED

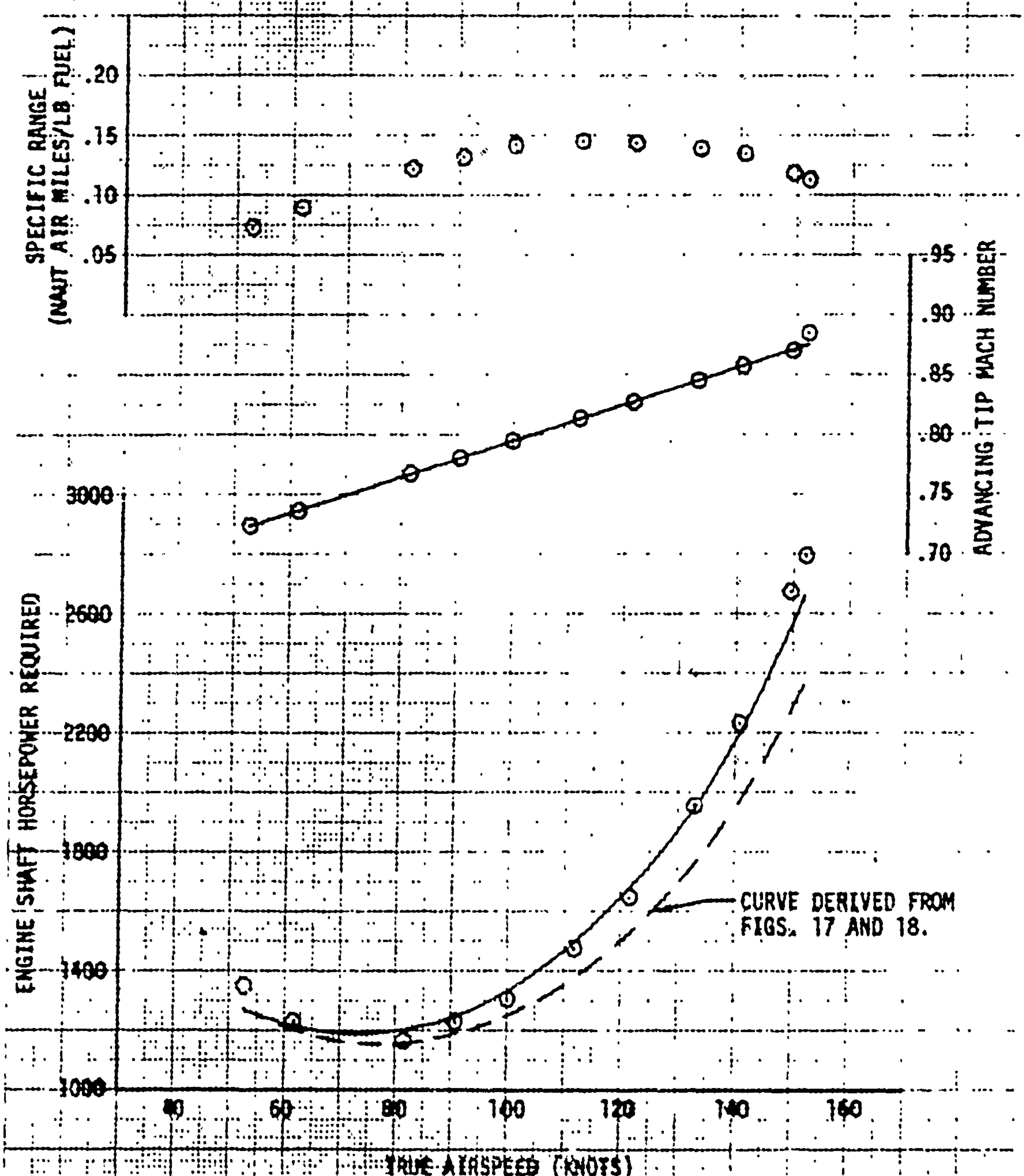

FIGURE 30
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
19880	347.8(FWD)	0.5(RT)	8480	16.5	258	0.009089	CARGO DOORS OPEN M-60 GUNS EXTENDED

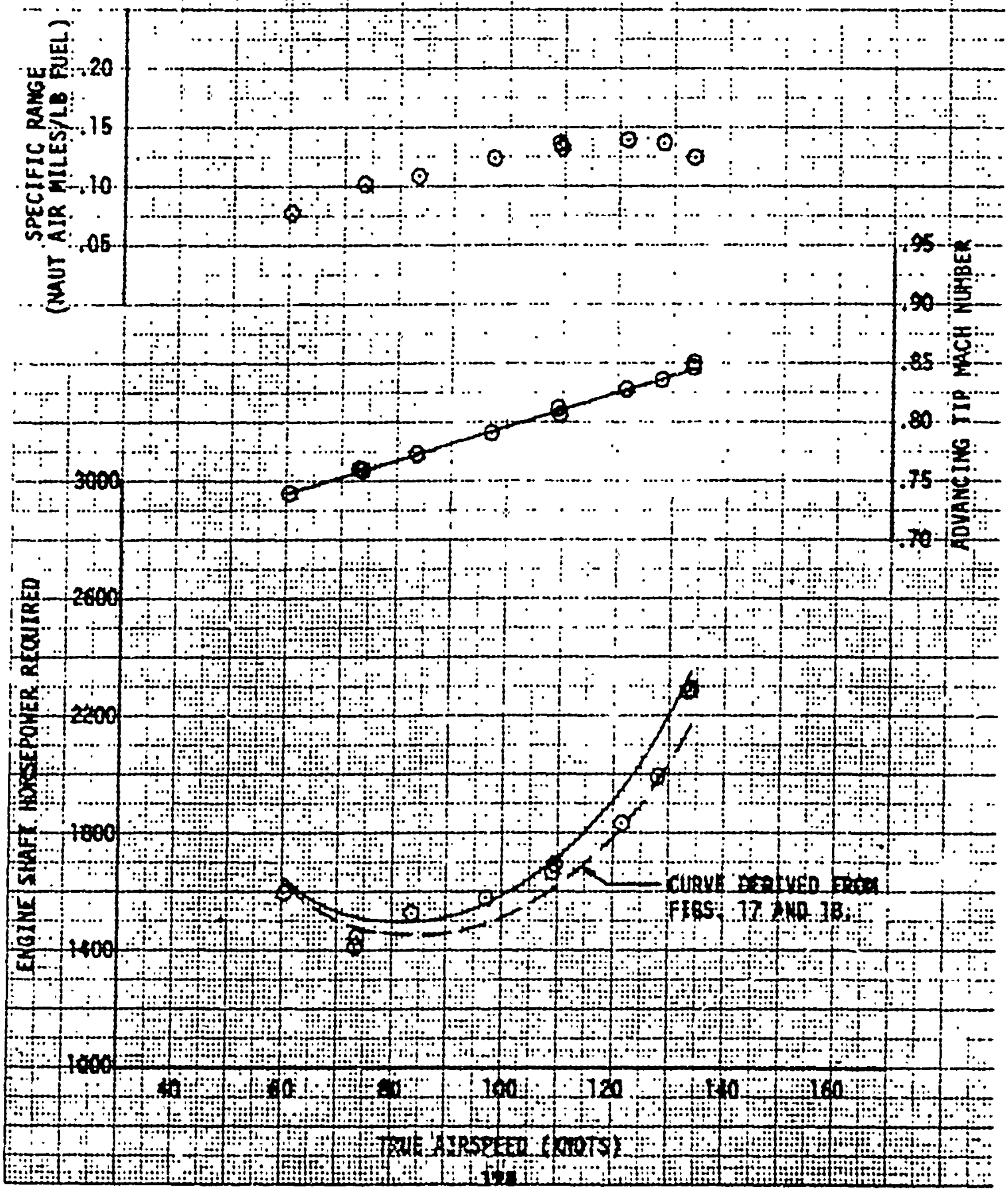

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FIGURE 31
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	C.G. LOCATION LONG (IN.)	C.G. LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	C_T	CONFIGURATION
16853	347(FWD)	0.4(RT)	7325	35	258	.007388	NORMAL UTILITY

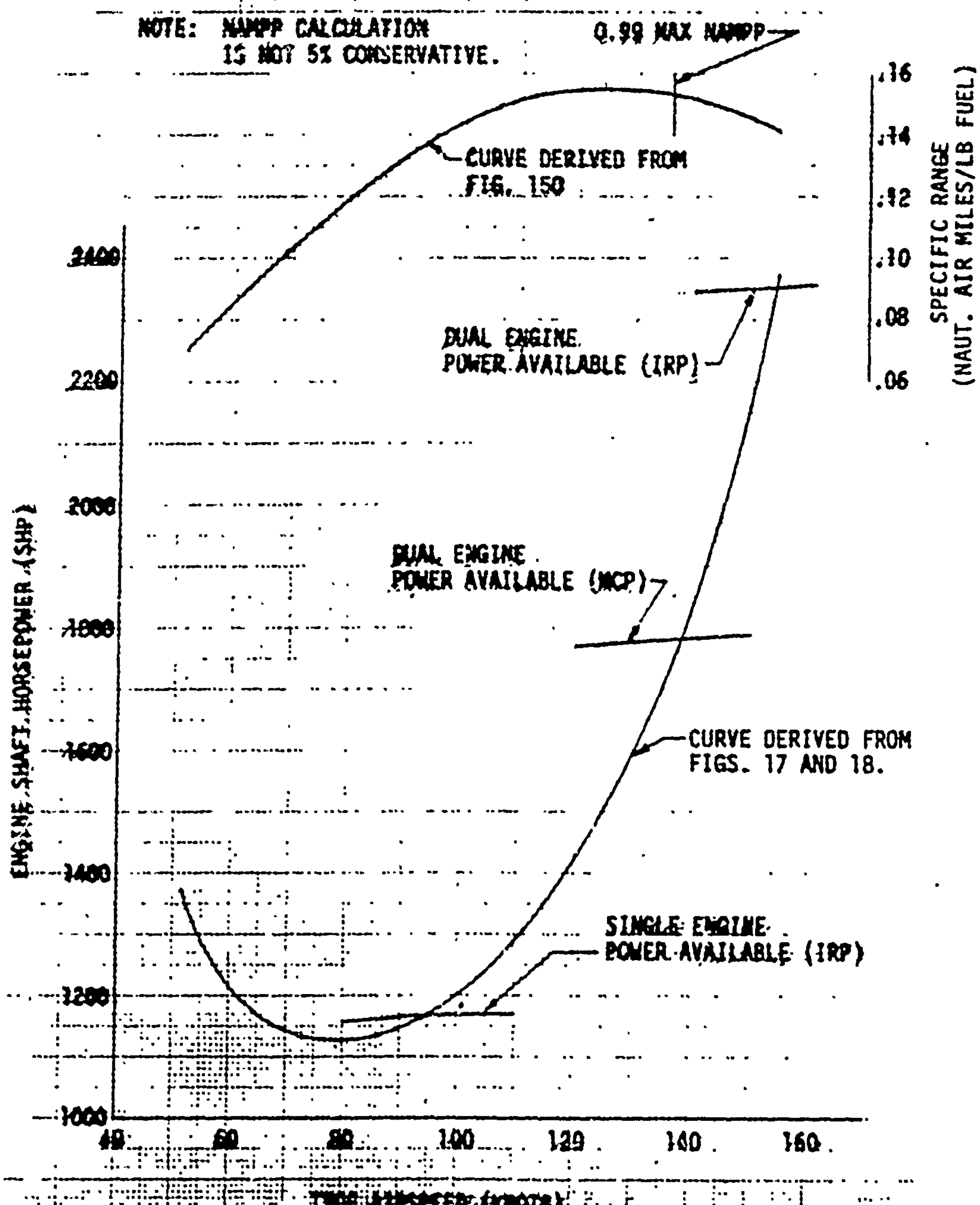

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FIGURE [illegible]
LEVEL FLIGHT PERFORMANCE
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	C.G. LOCATION LONG (IN.)	C.G. LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	C_T	CONFIGURATION
17028	347(FWD)	0.4(RT)	SEA LEVEL	15.0	258	.006028	NORMAL UTILITY

NOTE: NAMPP CALCULATION IS NOT 5% CONSERVATIVE.

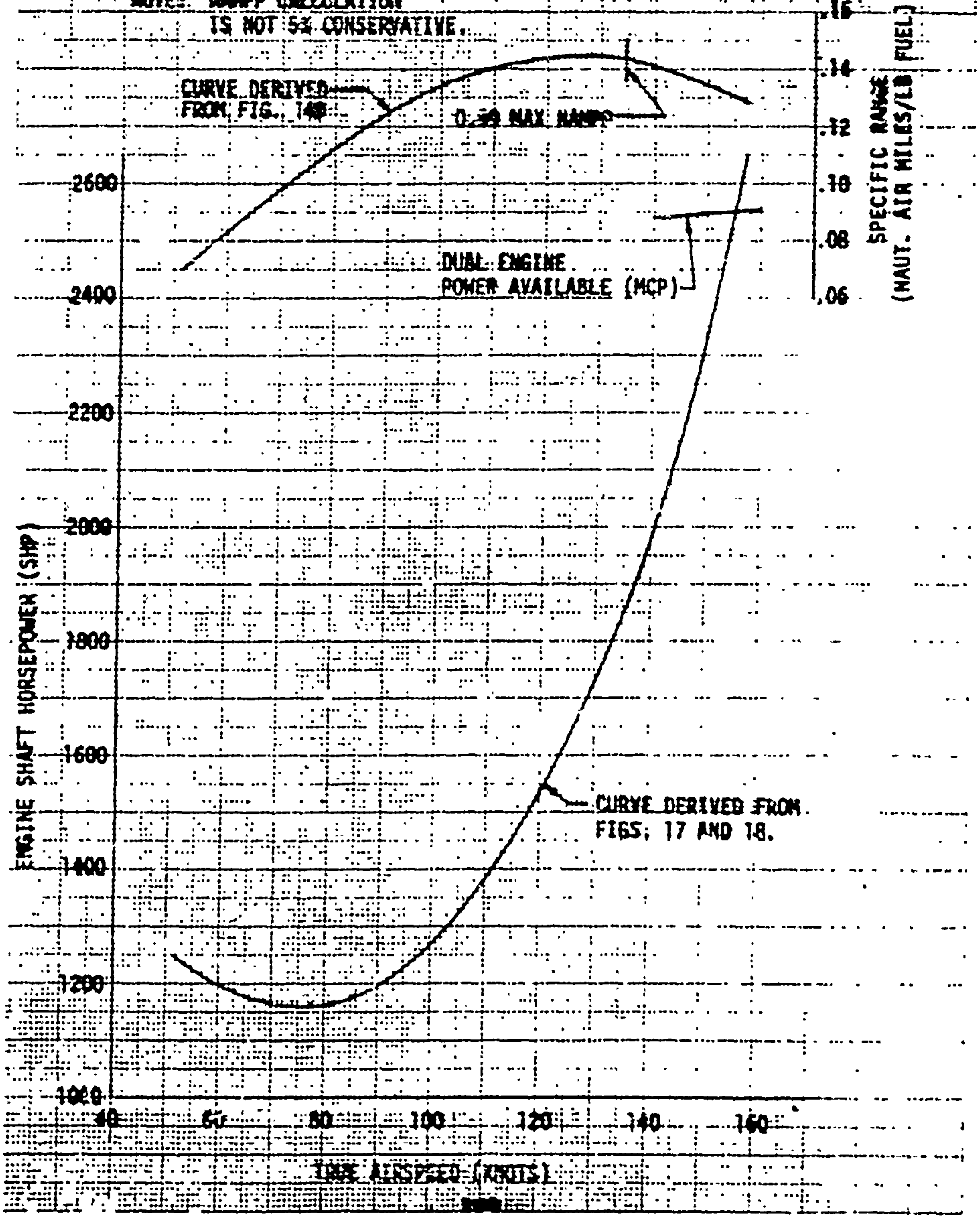

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FIGURE 33.
LEVEL FLIGHT FUEL FLOW SUMMARY
YUH-60A USA S/N 73-21651

STANDARD DAY

C.G. LOCATION LONG (IN.)	C.G. LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (°C)	ROTOR SPEED (RPM)	CONFIGURATION
347(FWD)	0.4(RT)	SEA LEVEL	15.0	258	NORMAL UTILITY

NOTE: DEPICTED FUEL FLOW IS NOT 5% CONSERVATIVE

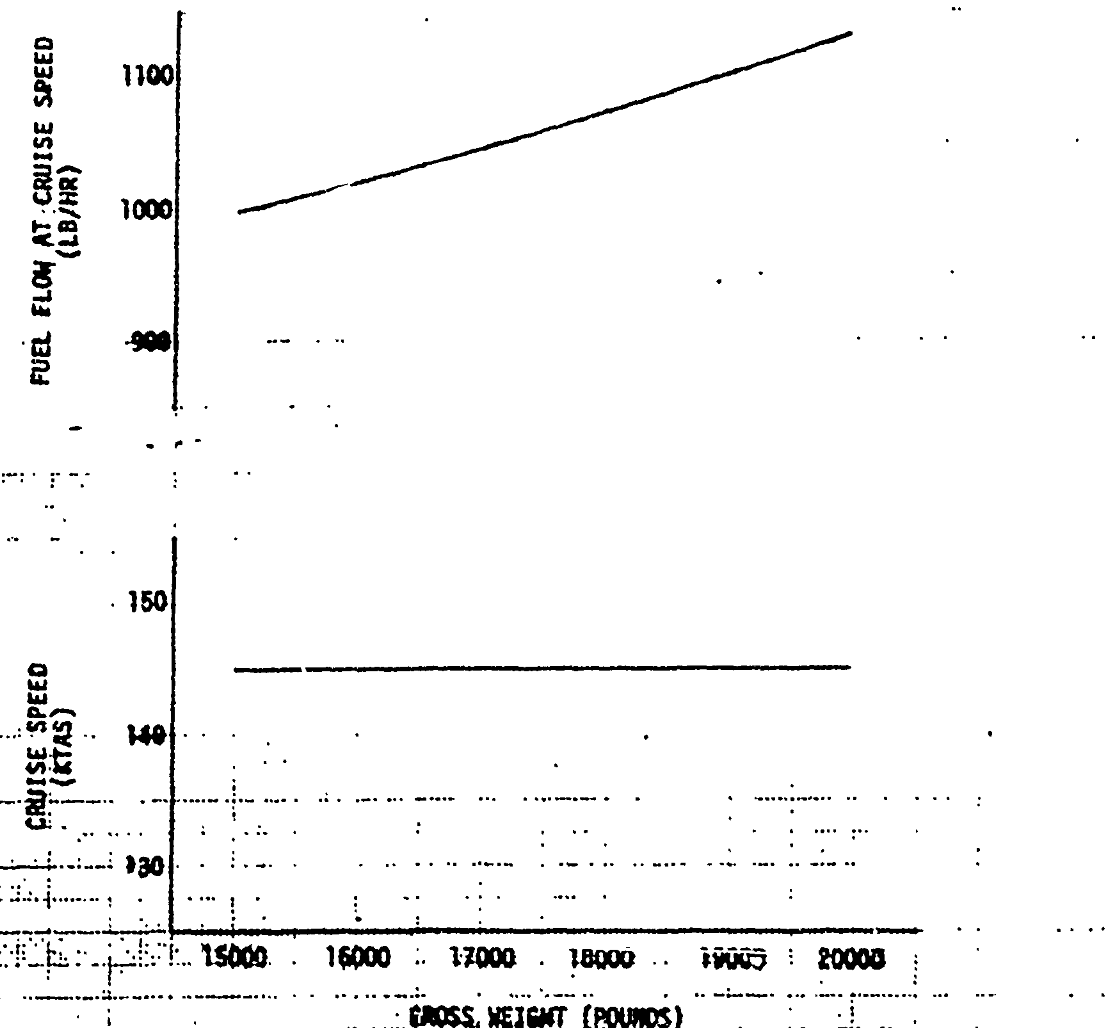

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FIGURE 34
TURNING PERFORMANCE
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	LAT (IN.)	AVG Hp (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	TRIM TRUE AIRSPEED (KTS)
16790	348.8 (FWD)	0.4 (RT)	7120	13.5	259	150

NOTE: ○ DENOTES RIGHT TURN
□ DENOTES LEFT TURN

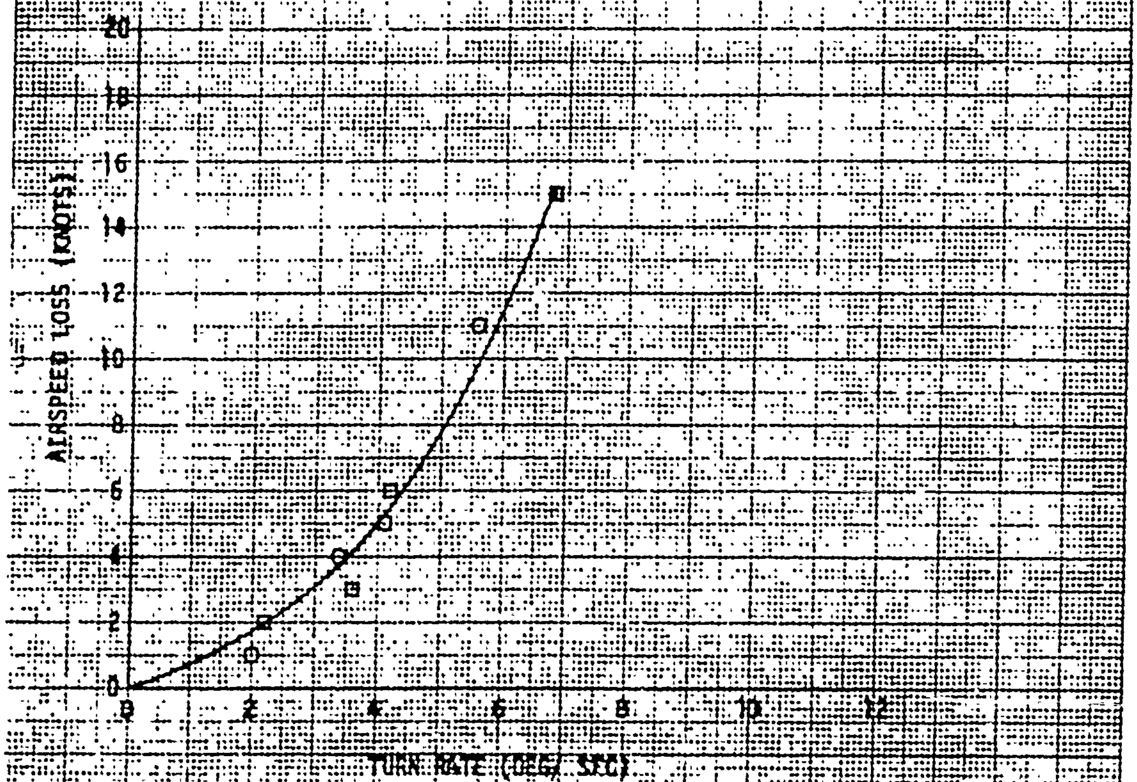
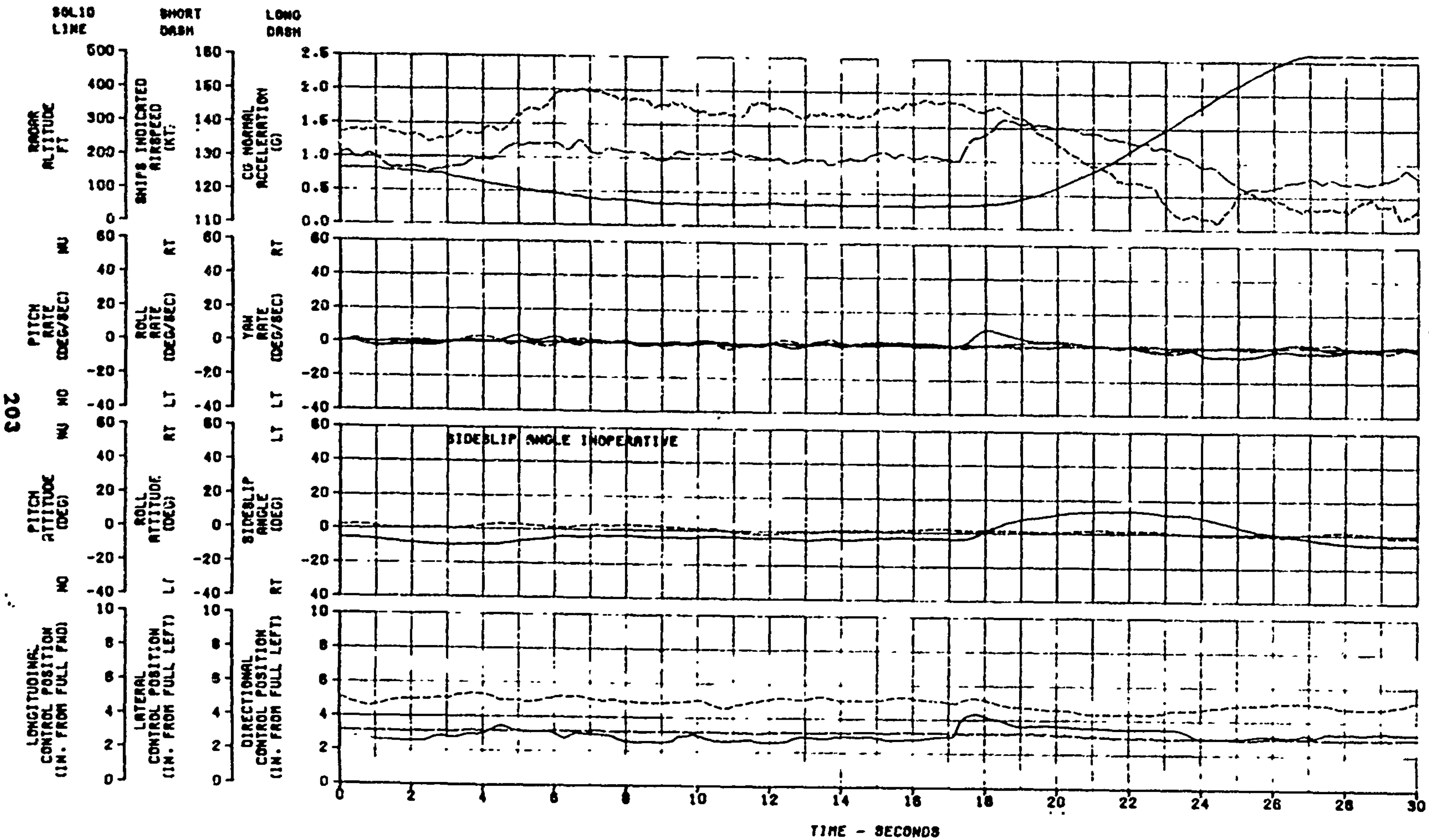


FIGURE 35

VERTICAL DISPLACEMENT (PULL UP)

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
17190	347.0 (FWD)	.4 (RT)	6300	25.0	258	ON



[redacted]

FIGURE 36

VERTICAL DISPLACEMENT (PARALLEL EXIT)

YUH-60A USA S/N 73-21651

GROSS WEIGHT	CG LOCATION LONG	CG LOCATION LAT	DENSITY ALTITUDE	OAT	TRIM ROTOR SPEED	AFCS CONDITION
(LB)	(IN.)	(IN.)	(FT)	(DEG C)	(RPM)	
16660	348.4 (FWD)	.4 (RT)	6300	25.0	260	ON

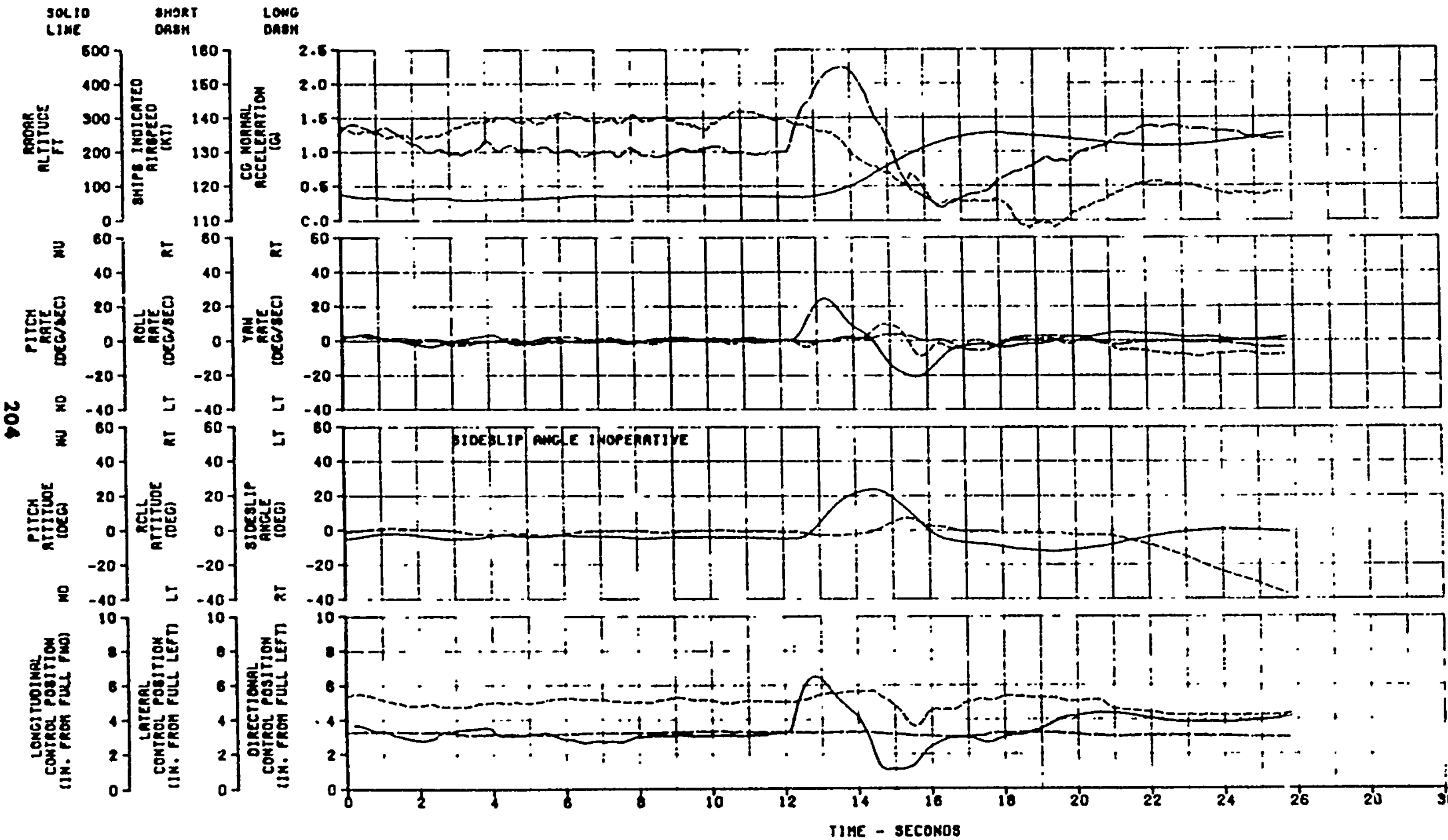
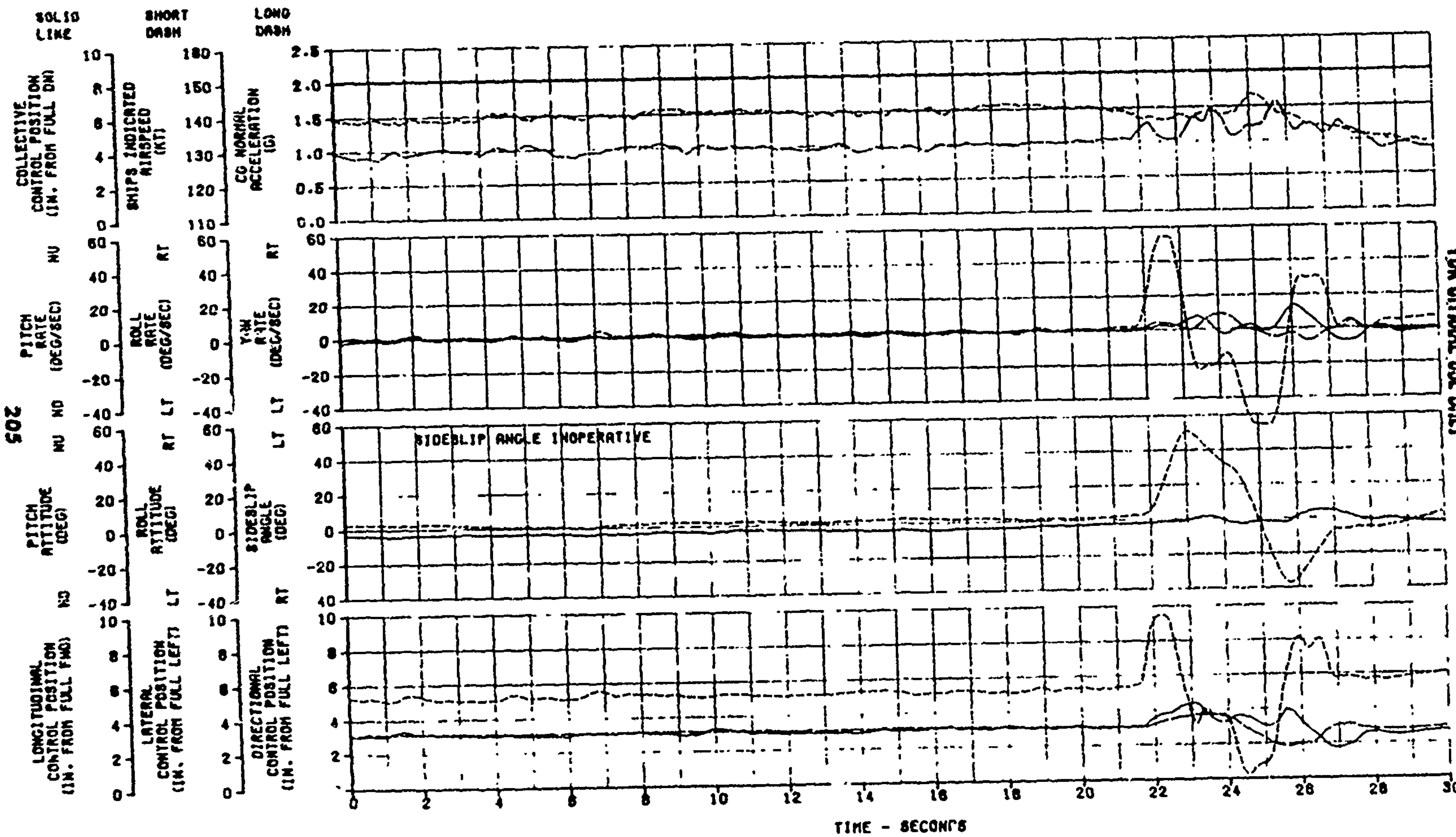


FIGURE 37
LATERAL DISPLACEMENT (PARALLEL EXIT)
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
16990	346.8 (FWD)	.4 (RT)	5950	26.5	260	ON



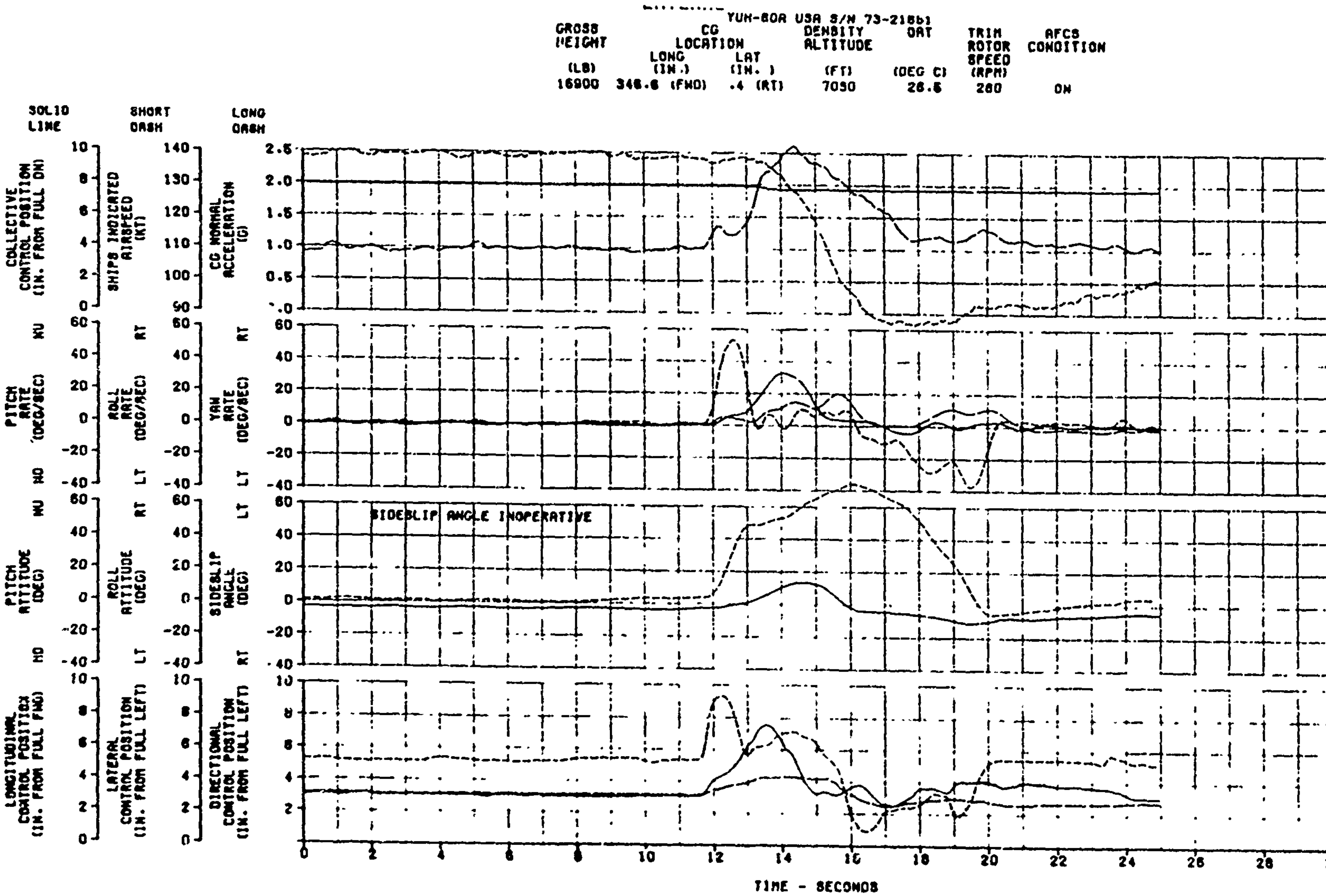

YUH-60A USA S/N 73-21651
GROSS WEIGHT (LB) 16900
CG LOCATION LONG (IN.) 346.6 (FWD) LAT (IN.) .4 (RT)
DENSITY ALTITUDE (FT) 7050
OAT (DEG C) 26.5
TRIM ROTOR SPEED (RPM) 260
AFCS CONDITION ON
SOLID LINE
SHORT DASH
LONG DASH
COLLECTIVE CONTROL POSITION (IN. FROM FULL DN)
SHIPS INDICATED AIRSPEED (KT)
CG NORMAL ACCELERATION (G)
PITCH RATE (DEG/SEC) NU ND
ROLL RATE (DEG/SEC) RT LT
YAW RATE (DEG/SEC) RT LT
PITCH ATTITUDE (DEG) NU ND
ROLL ATTITUDE (DEG) RT LT
SIDESLIP ANGLE (DEG) LT RT
SIDESLIP ANGLE INOPERATIVE
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
TIME - SECONDS

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FIGURE [illegible]
AUTOROTATIONAL PERFORMANCE
YUH-60A USA S/N 73-21651

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
□	16500	347.7(FWD)	0.4(RT)	5200	14.0	258	0.006928	NORMAL UTILITY
○	19720	347.5(FWD)	0.4(RT)	5050	9.5	258	0.007884	NORMAL UTILITY

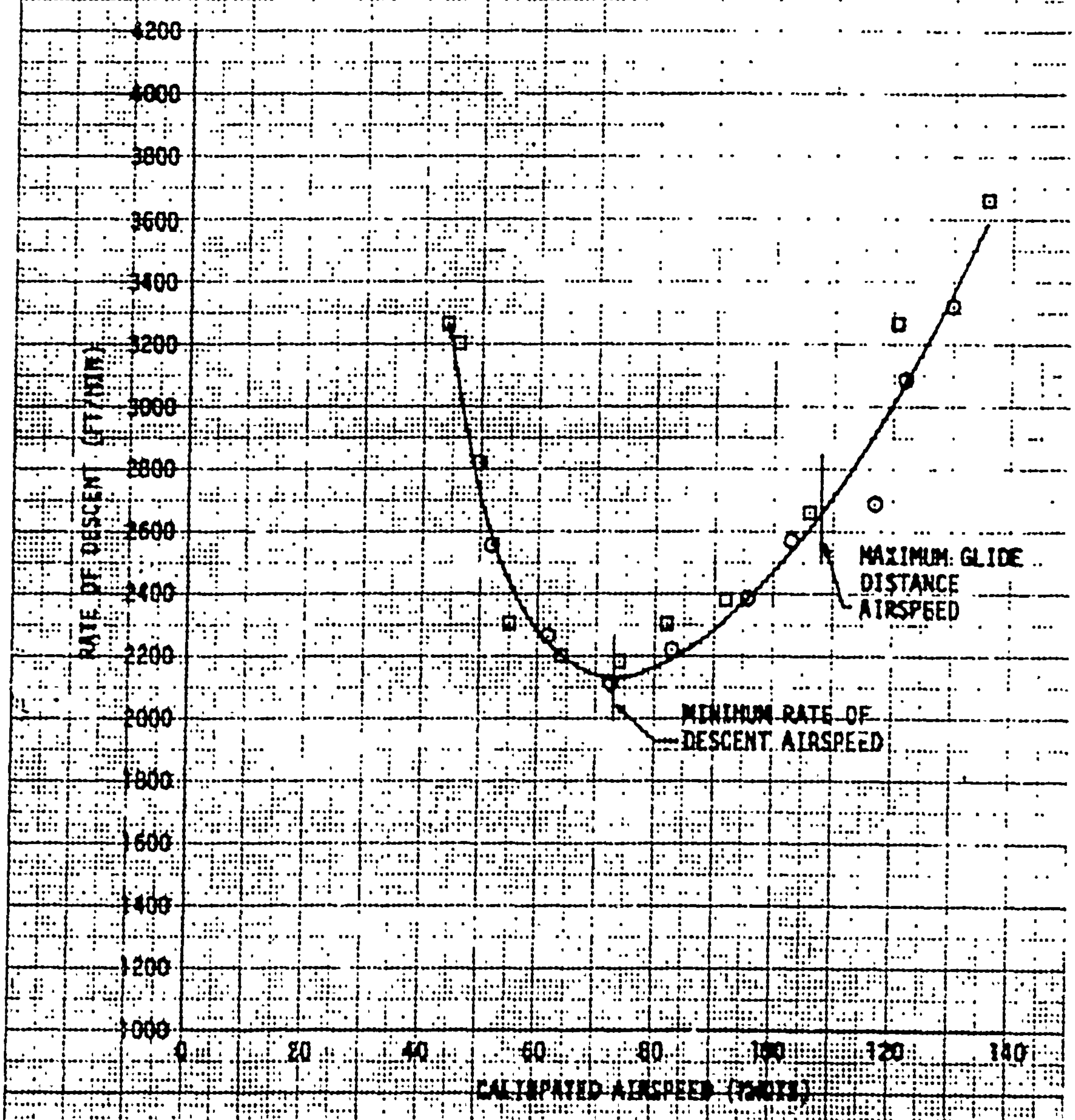

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FIGURE 40
AUTOROTATIONAL PERFORMANCE
YUH-60A USA S/N 73-21651

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	TRIM CALIBRATED AIRSPEED (KT)	CONFIGURATION
□	16760	347.7(FWD)	0.4	5200	14.0	71	NORMAL UTILITY
○	19160	347.5(FWD)	0.4	5050	9.5	76	NORMAL UTILITY

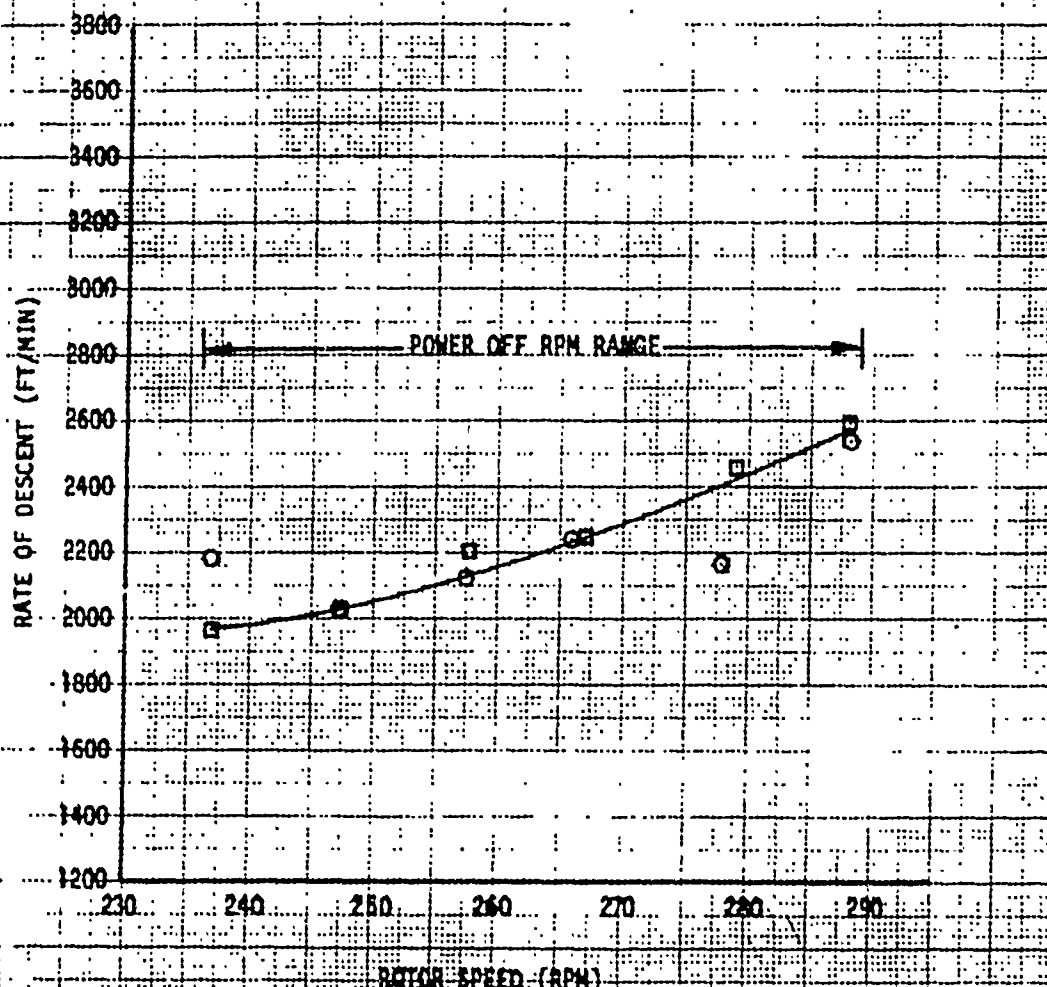
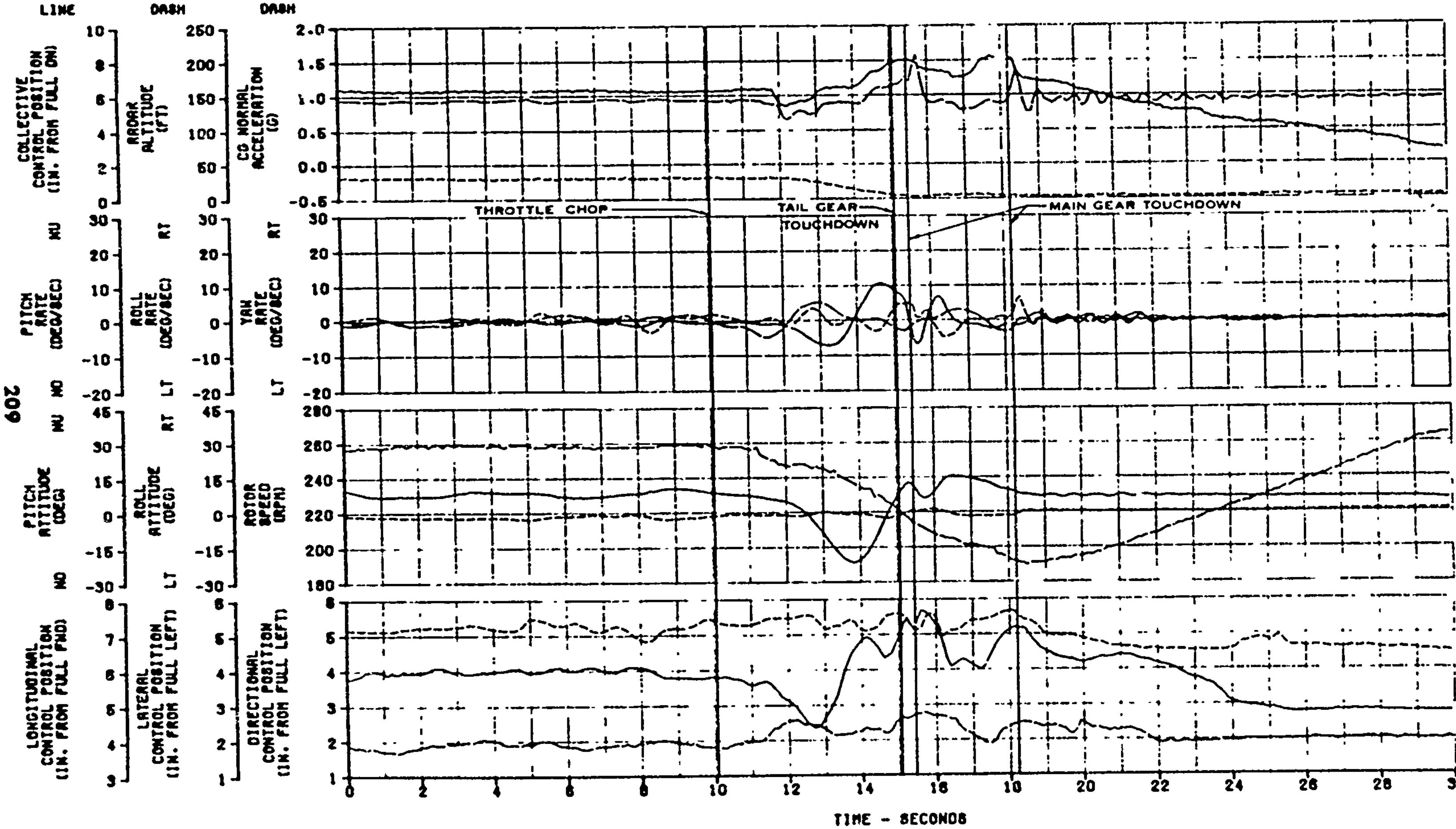


FIGURE 41
HEIGHT-VELOCITY TEST
YUH-60A USA S/N 73-21651

ENTRY TRUE AIRSPEED (KT)	GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
0	16960	347.8(FWD)	0.4(RT)	4790	14.6	263	ON

NOTE: NO 2 ENGINE FAILED FROM STABILIZED DUAL ENGINE FLIGHT



[redacted]

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FIGURE 42
HEIGHT-VELOCITY TEST
YUH-60A USA S/N 73-21651

ENTRY TRUE AIRSPEED (KT)	GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
10	16660	347.1 (FWD)	0.4 (RT)	5300	19.0	258	ON

NOTE: NO 2 ENGINE FAILED FROM STABILIZED DUAL ENGINE FLIGHT

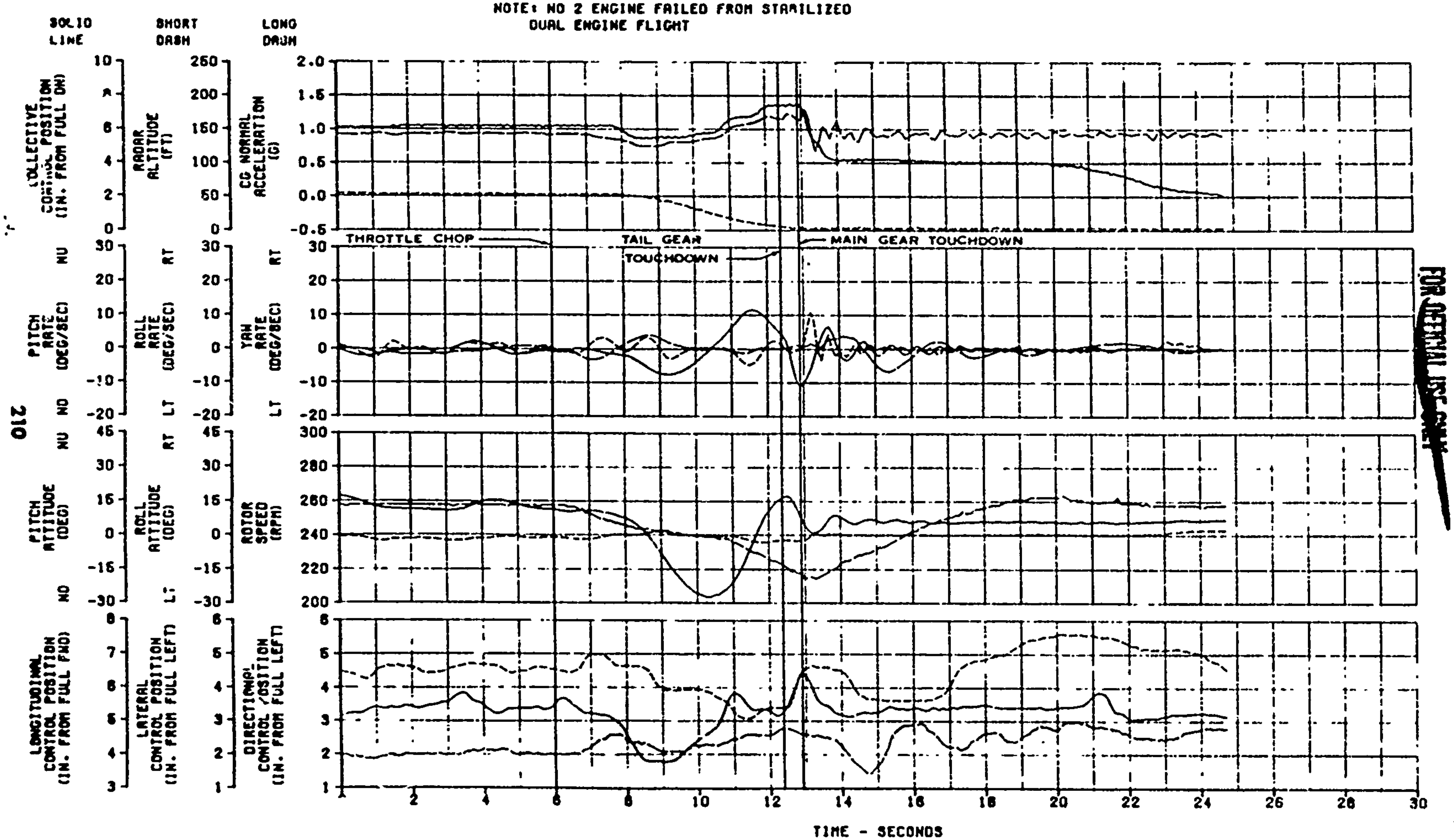

FIGURE 43
HEIGHT-VELOCITY TEST
YUH-60A USA S/N 73-21651

ENTRY TRUE AIRSPEED (KT)	GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
15	16600	347.8 (FWD)	0.4 (RT)	5760	23.5	258	ON

NOTE: NO 2 ENGINE FAILED FROM STABILIZED DUAL ENGINE FLIGHT

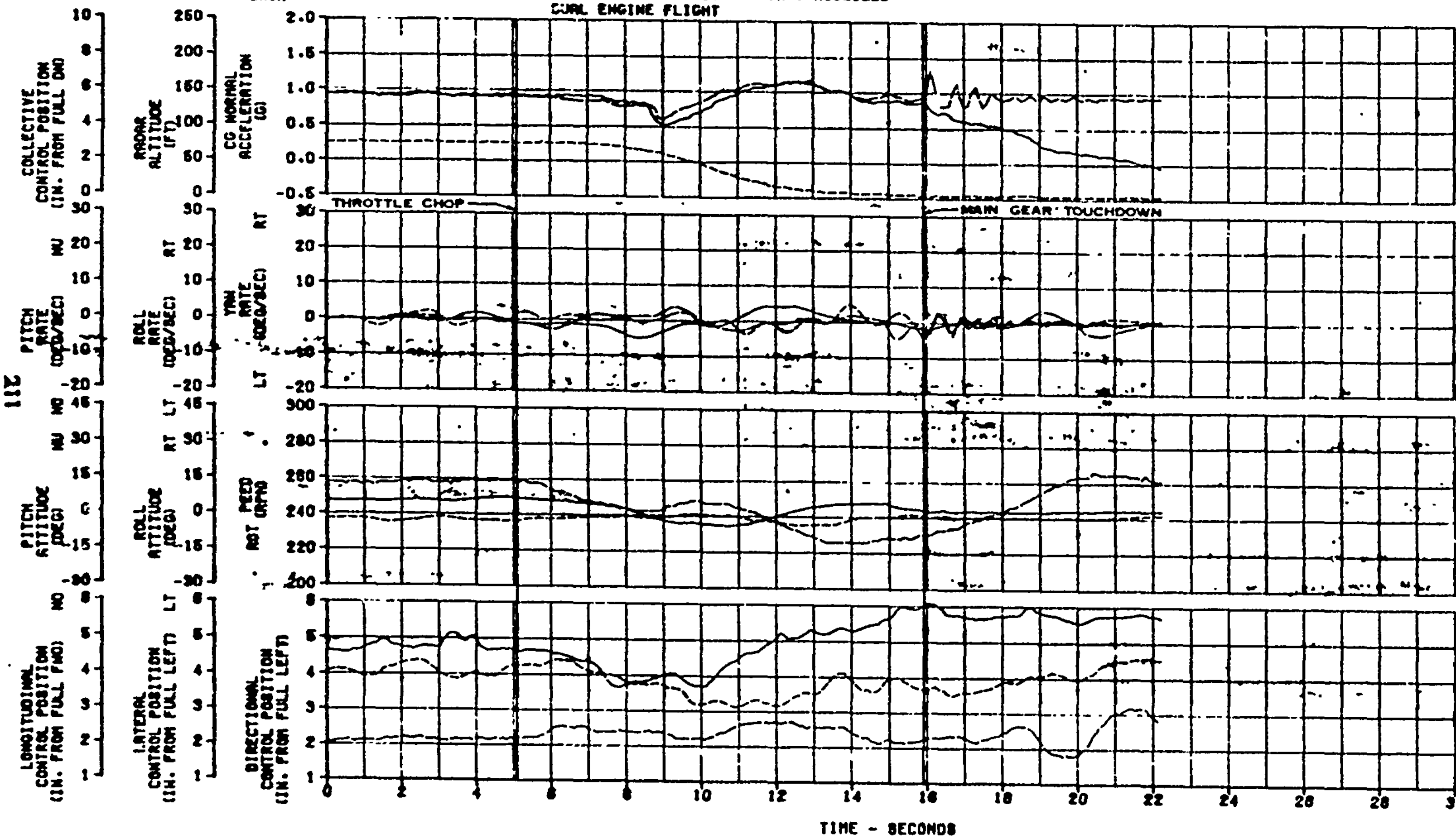

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FIGURE 44
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YUH-60A USA S/N 73-21651

NOTES:
1) ROTOR STATIC
2) FORCES AND POSITIONS MEASURED AT CENTER OF CONTROL GRIP
3) HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
4) NO. 1 AND NO. 2 BOOST SYSTEMS ON
5) FAS/TRN ON
6) LATERAL CONTROL POSITION = 4.5 IN. FROM FULL LEFT
7) COLLECTIVE CONTROL POSITION = 5.1 IN. FROM FULL DOWN
8) DIRECTIONAL CONTROL POSITION = 1.3 IN. FROM FULL LEFT

AVERAGE FRICTION BAND (NEAR TRIM)
AFT = 1.75 LB
FWD = 1.0 LB

AVERAGE FORCE GRADIENT (NEAR TRIM)
AFT = 3.25 LB/IN.
FWD = 3.25 LB/IN.

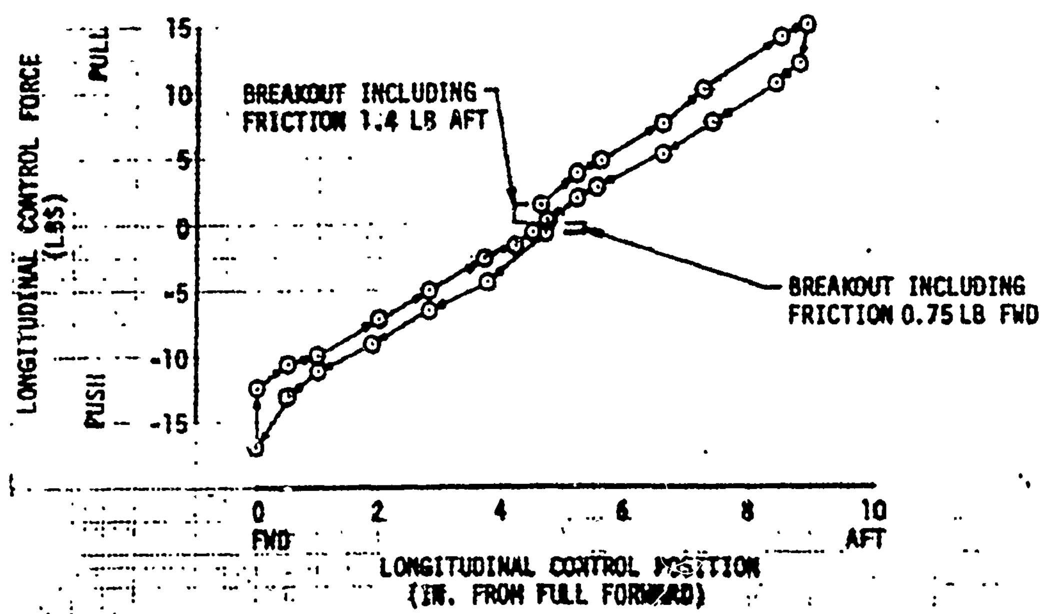

FIGURE 45
LATERAL CONTROL SYSTEM CHARACTERISTICS
YUH-60A USA S/N 73-21651

NOTES:
1) ROTOR STATIC
2) HYDRAULIC AND ELECTRICAL POWER PROVIDED BY EXTERNAL POWER UNITS
3) NO. 1 AND NO. 2 BOOST SYSTEMS ON
4) FAS/TRM ON
5) LONGITUDINAL CNTRL POSN = 4.6 IN. FROM FULL FWD
6) FORCES AND POSITIONS MEASURED AT CENTER OF CONTROL GRIP
7) COLLECTIVE CONTROL POSITION = 5.1 IN. FROM FULL DOWN
8) DIRECTIONAL CONTROL POSITION = 1.3 IN. FROM FULL LEFT

AVERAGE FRICTION BAND (NEAR TRIM)
LT = 1.5 LB
RT = 1.6 LB

AVERAGE FORCE GRADIENT (NEAR TRIM)
LT = 2.0 LB/IN.
RT = 2.0 LB/IN.

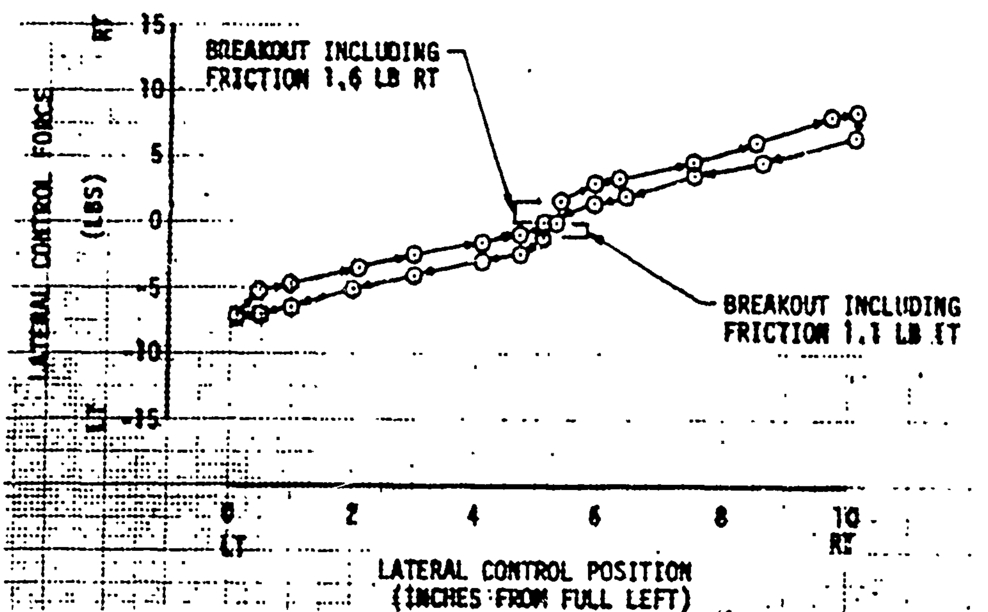

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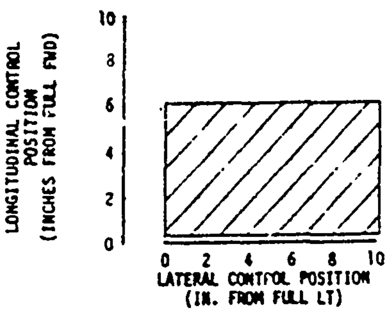
FIGURE NO. [illegible]

LIMITS OF CYCLIC CONTROL TRAVEL

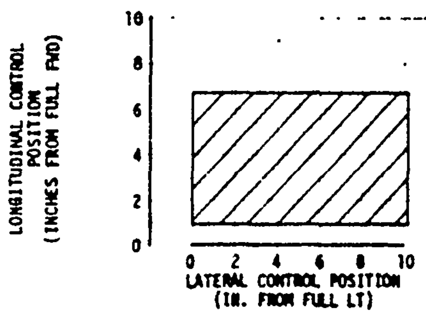
YUH-60A USA S/N 73-21651

NOTES:
1) ROTOR STATIC
2) CONTROL POSITIONS MEASURED AT CENTER OF GRIP
3) HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
4) NO. 1 AND NO. 2 BOOST SYSTEMS ON
5) HATCHED AREAS DENOTE LIMIT OF CYCLIC CONTROL TRAVEL
6) COLLECTIVE CONTROL POSITION FULL DOWN
7) TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 IN.
8) TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 IN.

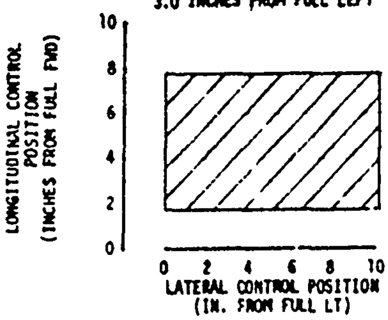
DIRECTIONAL CONTROL POSITION
0 INCHES FROM FULL LEFT



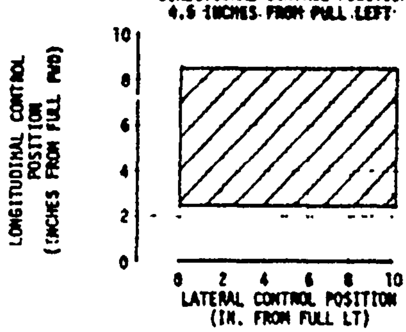
DIRECTIONAL CONTROL POSITION
1.4 INCHES FROM FULL LEFT



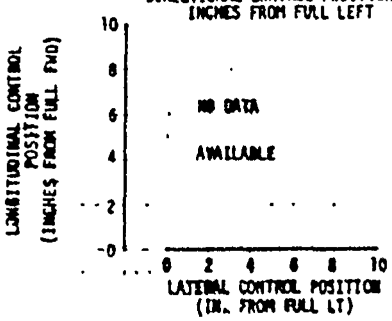
DIRECTIONAL CONTROL POSITION
3.0 INCHES FROM FULL LEFT



DIRECTIONAL CONTROL POSITION
4.5 INCHES FROM FULL LEFT



DIRECTIONAL CONTROL POSITION
INCHES FROM FULL LEFT



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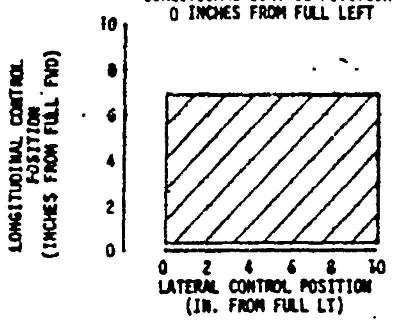
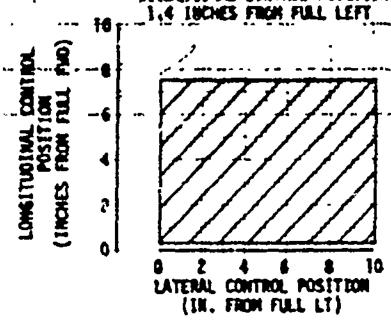
LIMITS OF CYCLIC CONTROL TRAVEL
YUH-60A USA S/N 73-21651

NOTES: 1) ROTOR STATIC
2) CONTROL POSITIONS MEASURED AT CENTER OF GRIP
3) HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
4) NO. 1 AND NO. 2 BOOST SYSTEMS ON
5) HATCHED AREAS DENOTE LIMIT OF CYCLIC CONTROL TRAVEL
6) COLLECTIVE CONTROL POSITION 2.8 IN. FROM FULL DOWN
7) TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 IN.
8) TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 IN.

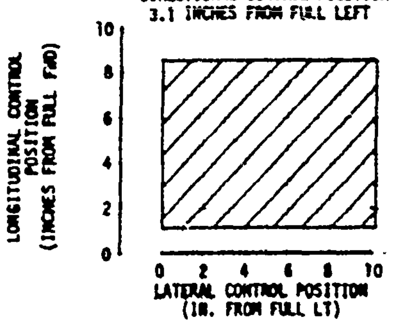
DIRECTIONAL CONTROL POSITION
0 INCHES FROM FULL LEFT



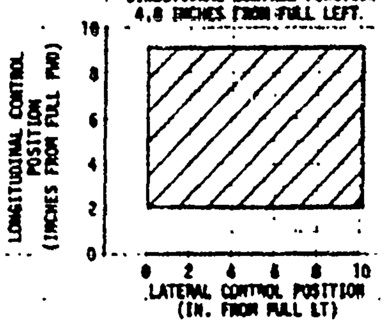
DIRECTIONAL CONTROL POSITION
1.4 INCHES FROM FULL LEFT



DIRECTIONAL CONTROL POSITION
3.1 INCHES FROM FULL LEFT



DIRECTIONAL CONTROL POSITION
4.8 INCHES FROM FULL LEFT



DIRECTIONAL CONTROL POSITION
5.1 INCHES FROM FULL LEFT

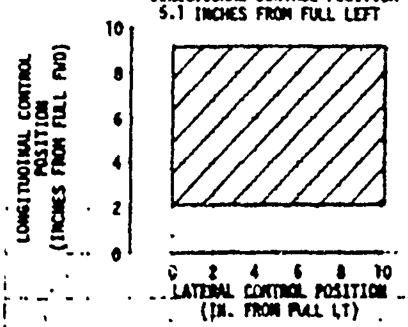
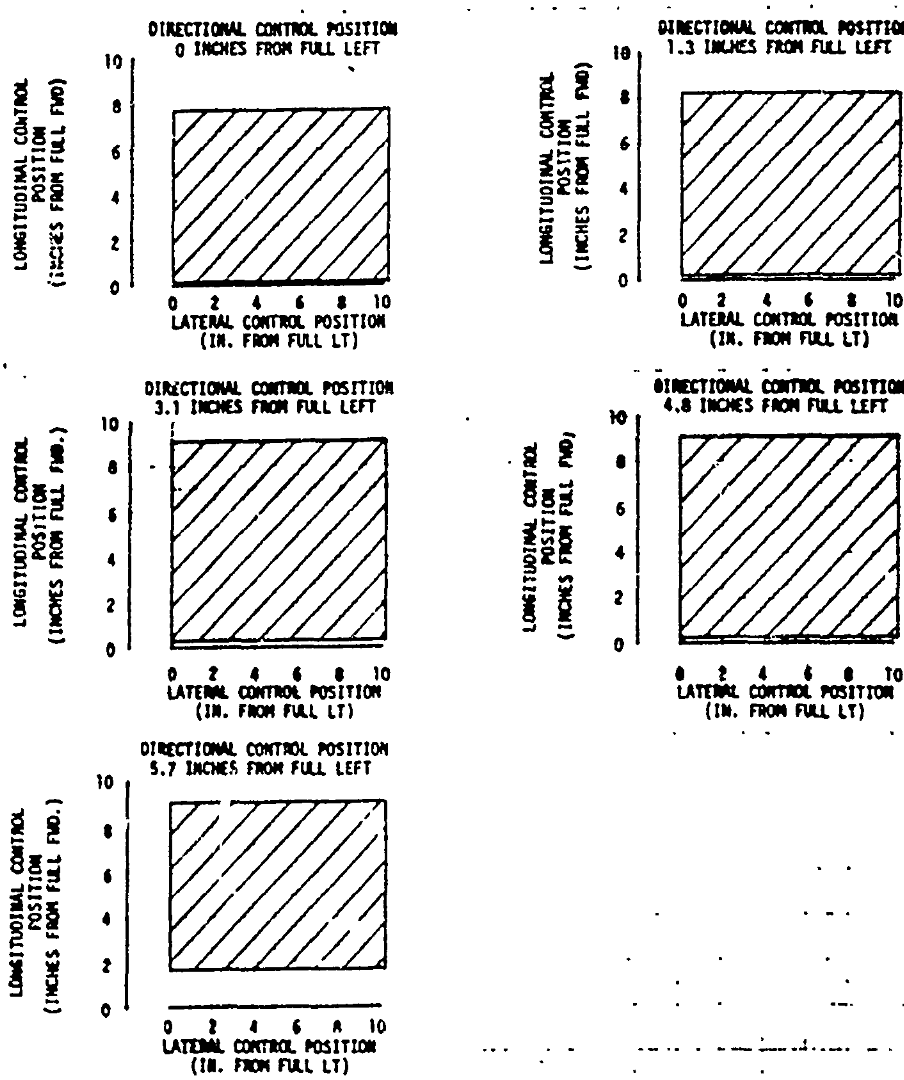


FIGURE NO.

LIMITS OF CYCLIC CONTROL TRAVEL
YUH-60A USA S/N 73-21651

NOTES:
1) ROTOR STATIC
2) CONTROL POSITIONS MEASURED AT CENTER OF GRIP
3) HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
4) NO. 1 AND NO. 2 BOOST SYSTEMS ON
5) HATCHED AREAS DENOTE LIMIT OF CYCLIC CONTROL TRAVEL
6) COLLECTIVE CONTROL POSITION 4.6 IN. FROM FULL DOWN
7) TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 IN.
8) TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 IN.



[redacted]

FIGURE [illegible]
LIMITS OF CYCLIC CONTROL TRAVEL
YUH-60A USA S/N 73-21651

NOTES:
1) ROTOR STATIC
2) CONTROL POSITIONS MEASURED AT CENTER OF GRIP
3) HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
4) NO. 1 AND NO. 2 BOOST SYSTEMS ON
5) HATCHED AREAS DENOTE LIMIT OF CYCLIC CONTROL TRAVEL
6) COLLECTIVE CONTROL POSITION 6.9 IN. FROM FULL DOWN
7) TOTAL DIRECTIONAL CONTROL TRAVEL = 6.0 IN.
8) TOTAL COLLECTIVE CONTROL TRAVEL = 9.0 IN.

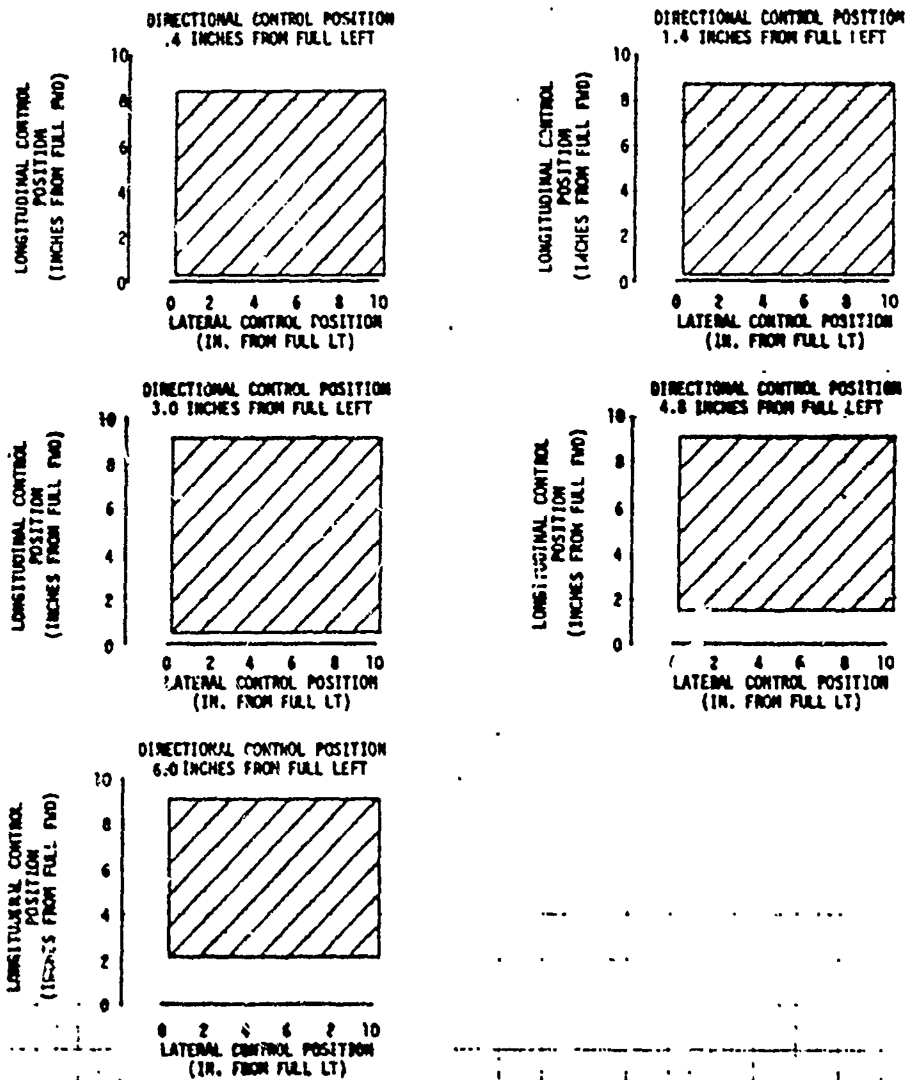


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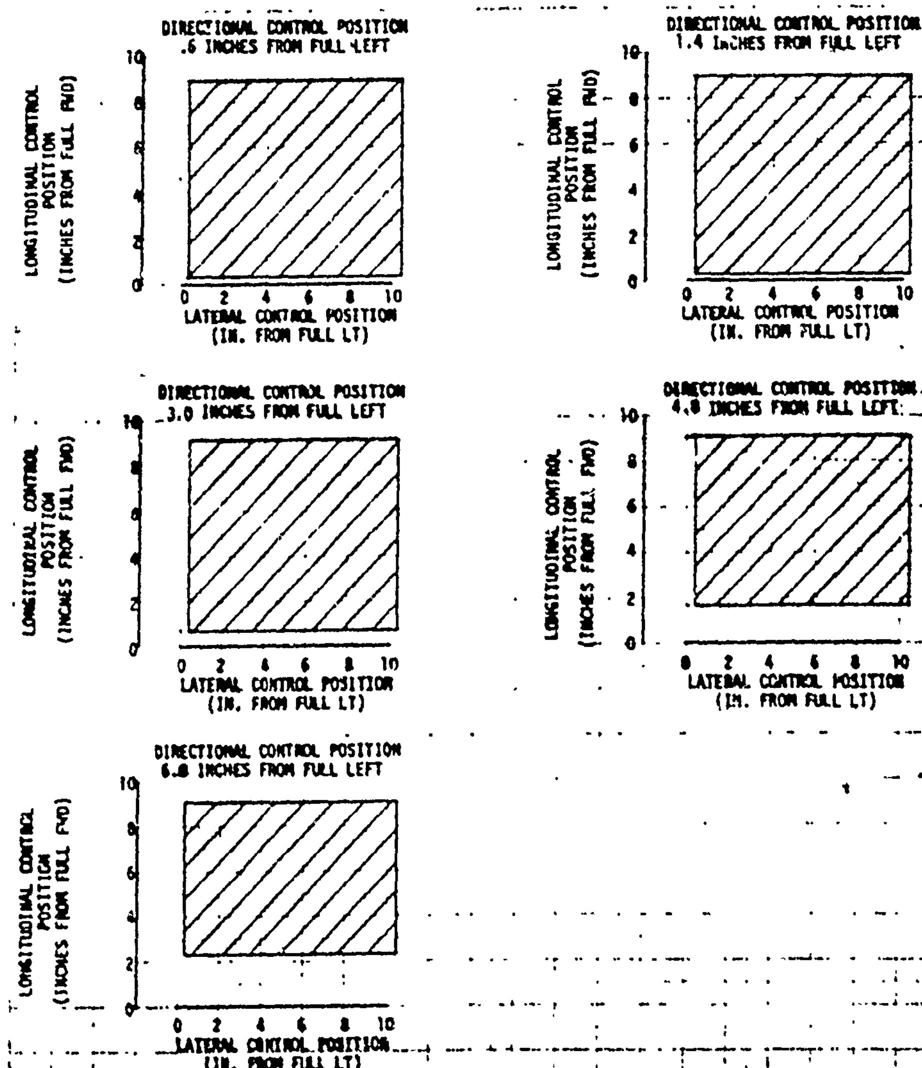
FIGURE [illegible]

LIMITS OF CYCLIC CONTROL TRAVEL
YUH-60A USA S/N 73-21651

NOTES: 1) ROTOR STATIC
2) CONTROL POSITIONS MEASURED AT CENTER OF GRIP
3) HYDRAULIC AND ELECTRIC POWER PROVIDED BY GROUND POWER UNITS
4) NO. 1 AND NO. 2 BOOST SYSTEMS ON
5) HATCHED AREAS DENOTE LIMIT OF CYCLIC CONTROL TRAVEL
6) COLLECTIVE CONTROL POSITION 9.0 IN. FROM FULL DOWN (FULL UP)
7) TOTAL DIRECTIONAL CONTRAL TRAVEL = 6.0 IN.



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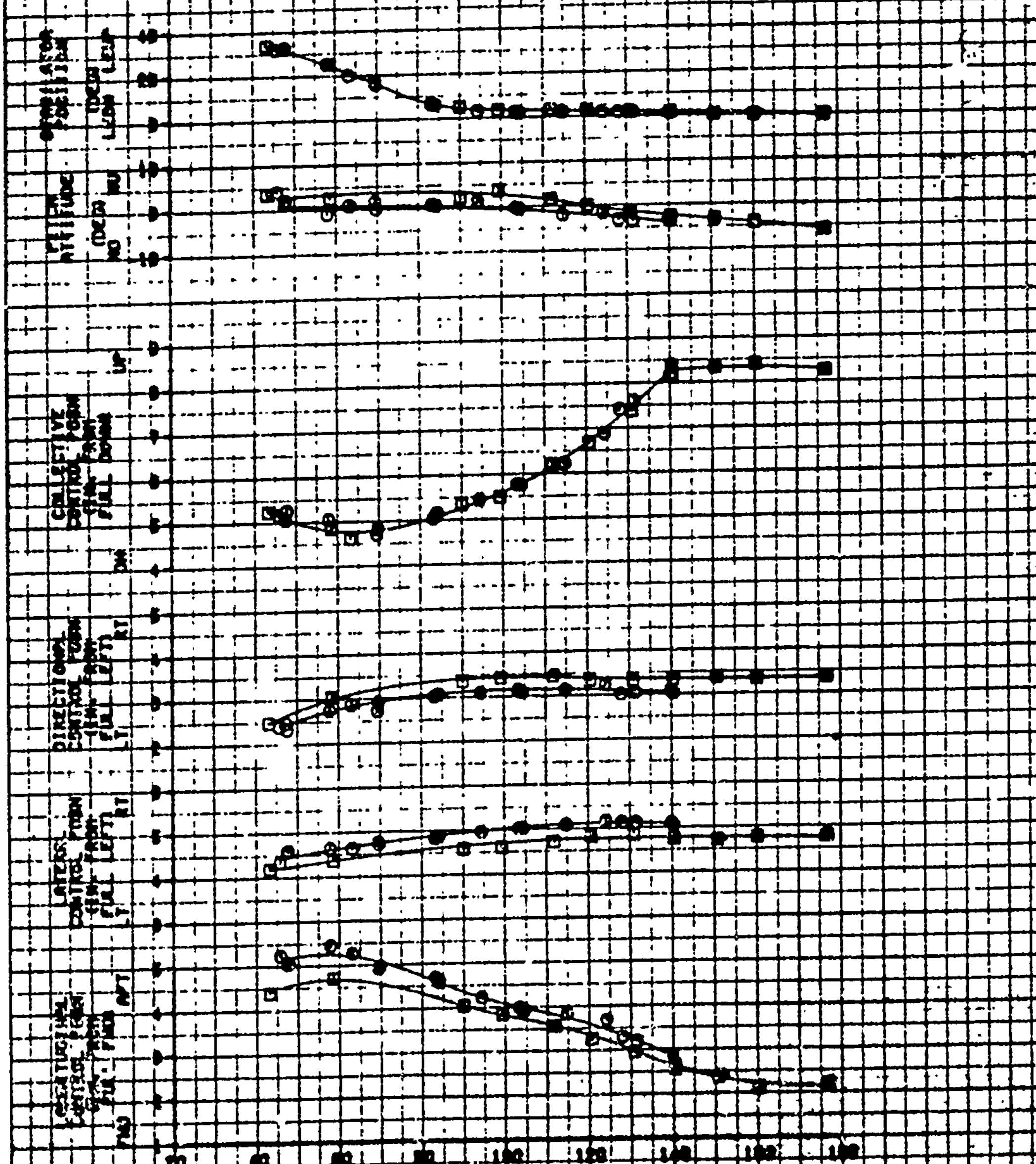
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

NOTE: SHADED SYMBOLS DENOTE DIVING FLIGHT AT INTERMEDIATE RATED POWER



CALIBRATED AIRSPEED ~ KNOTS

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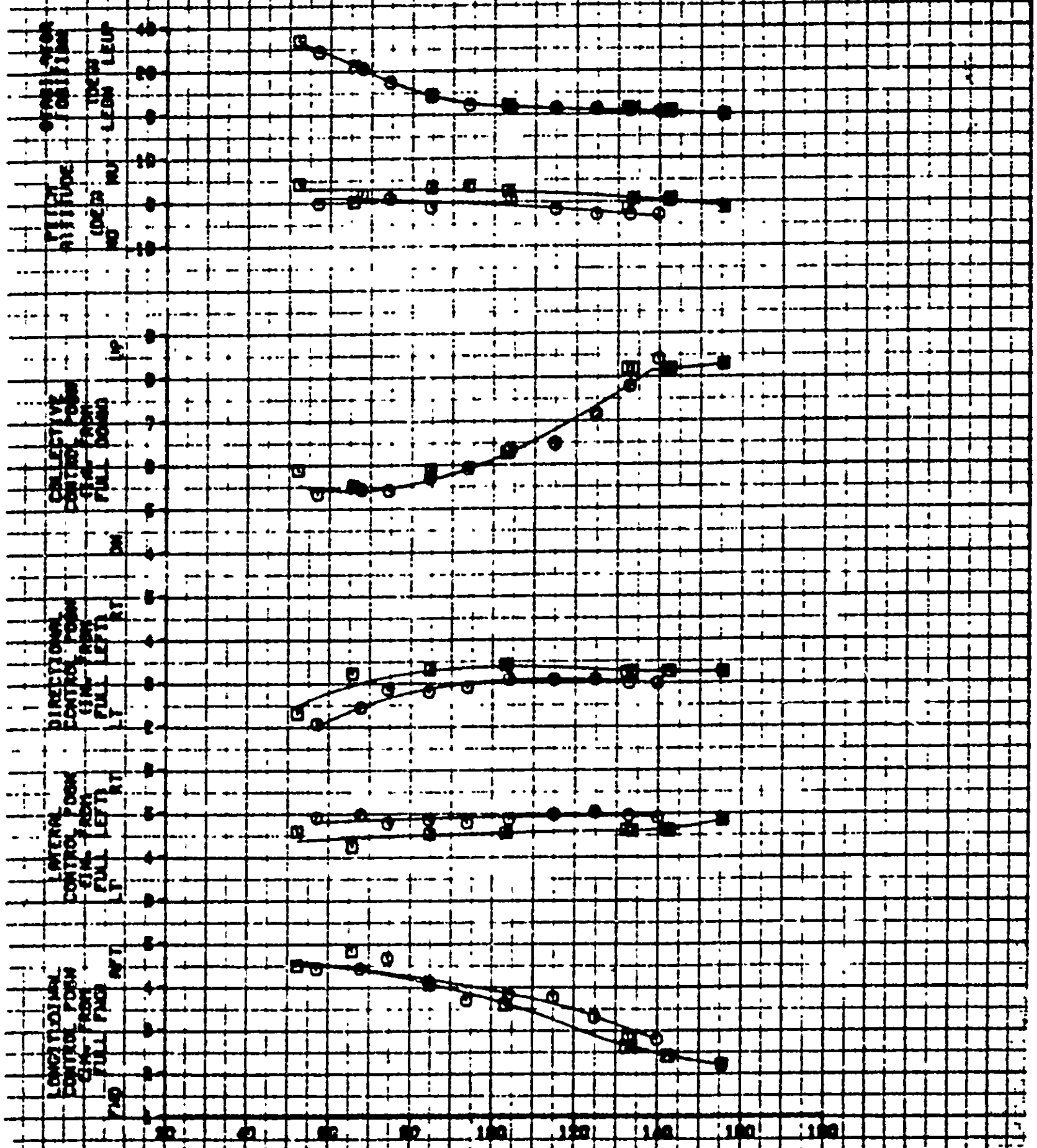
FIGURE 103

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

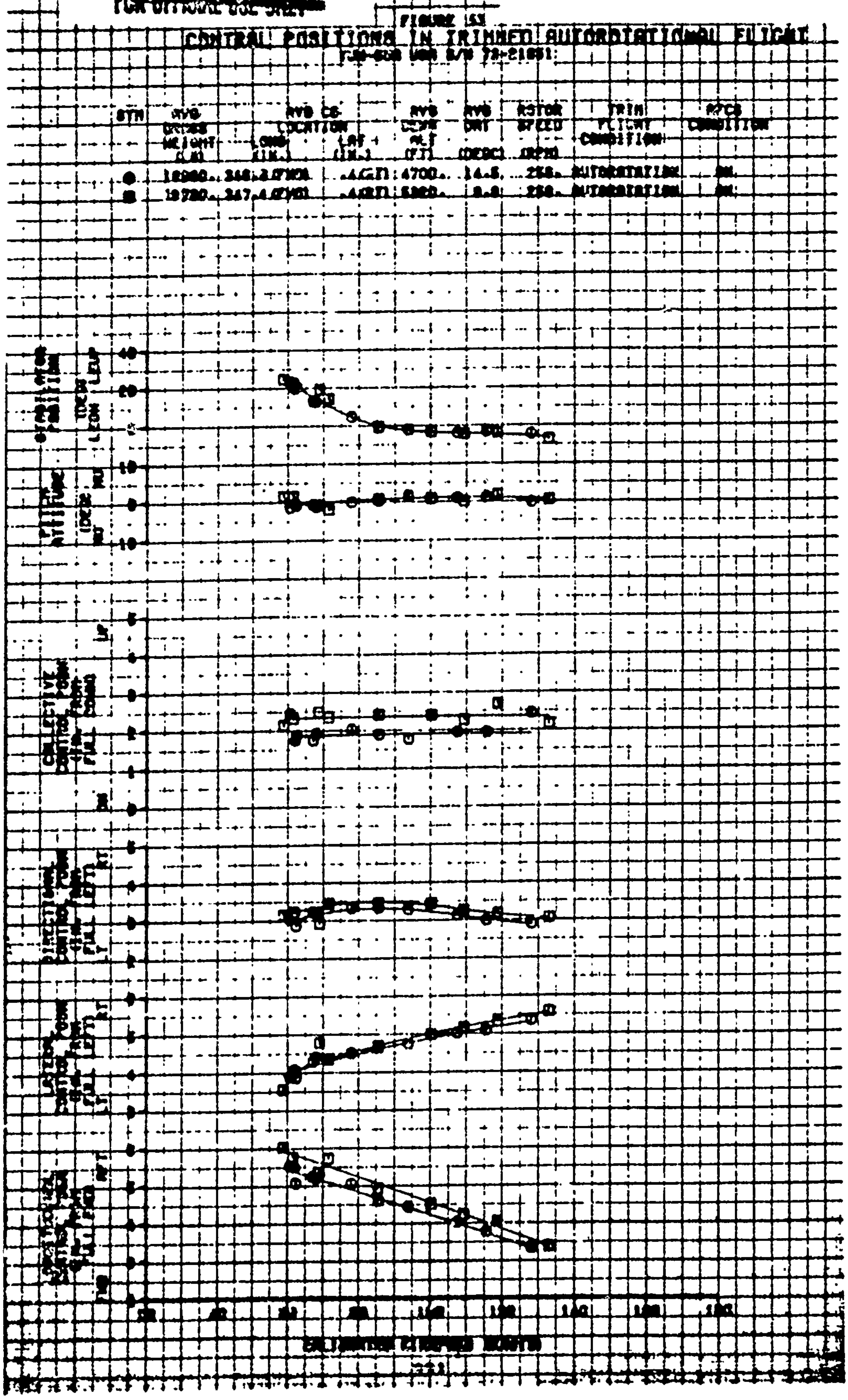
YUH-60A USA S/N 73-21651

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
○	15820.	348.9(FWD)	.4(RT)	2280.	11.5	228.	LEVEL	ON
□	20000.	361.1(AFT)	.4(RT)	3640.	12.0	237.	LEVEL	ON

NOTE: SHADED SYMBOLS DENOTE DIVING FLIGHT AT INTERMEDIATE RATED POWER

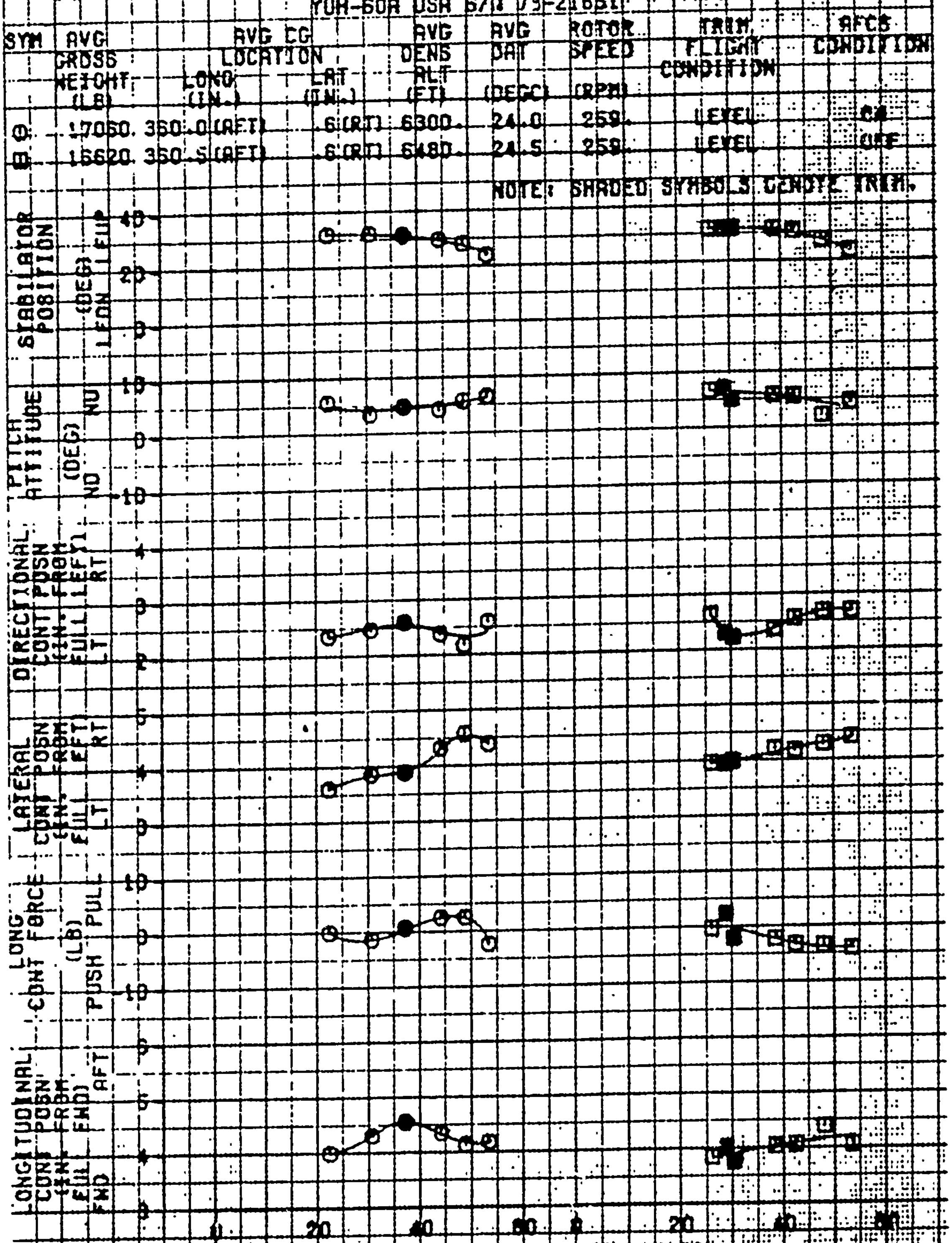


FOR OFFICIAL USE ONLY

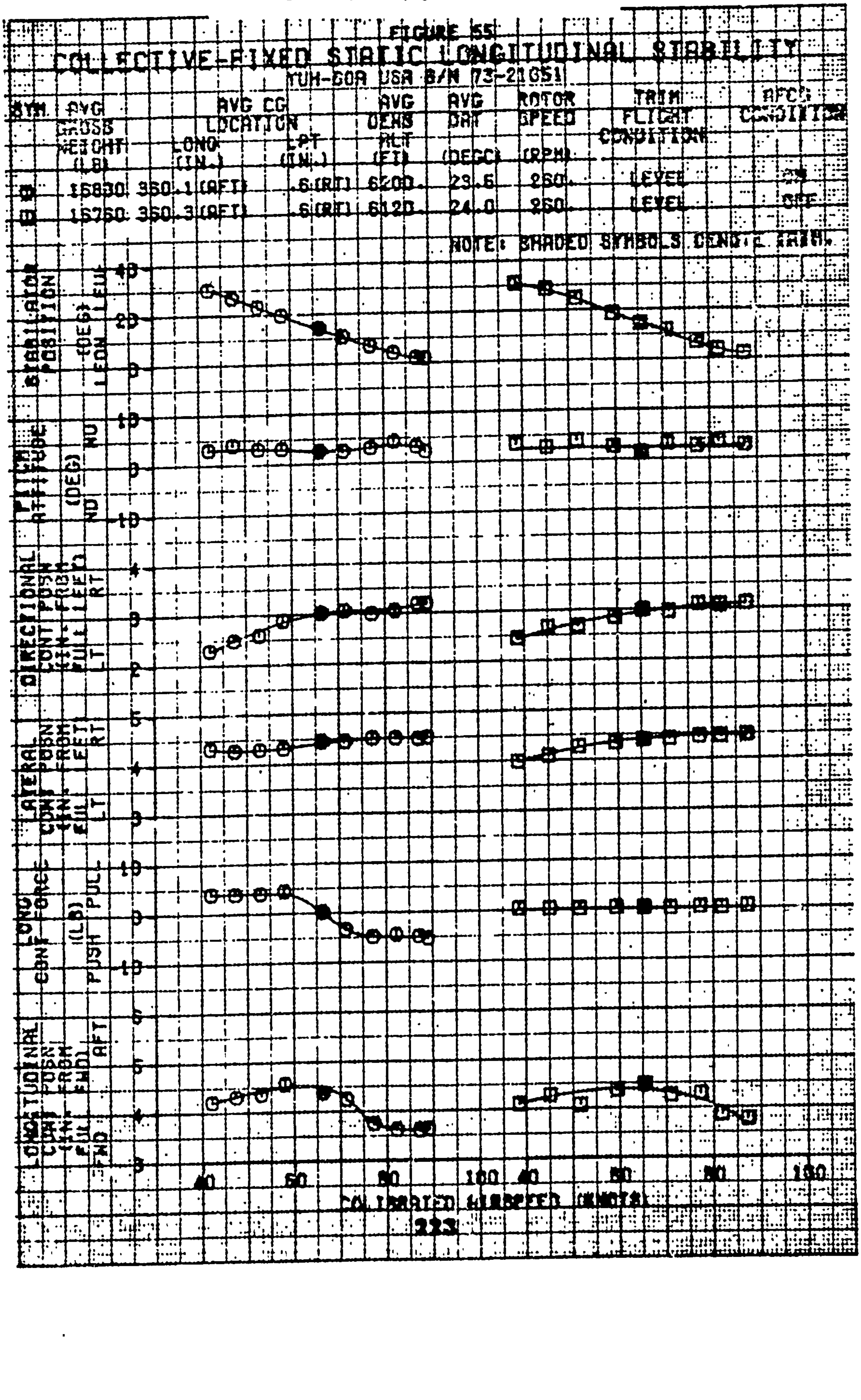


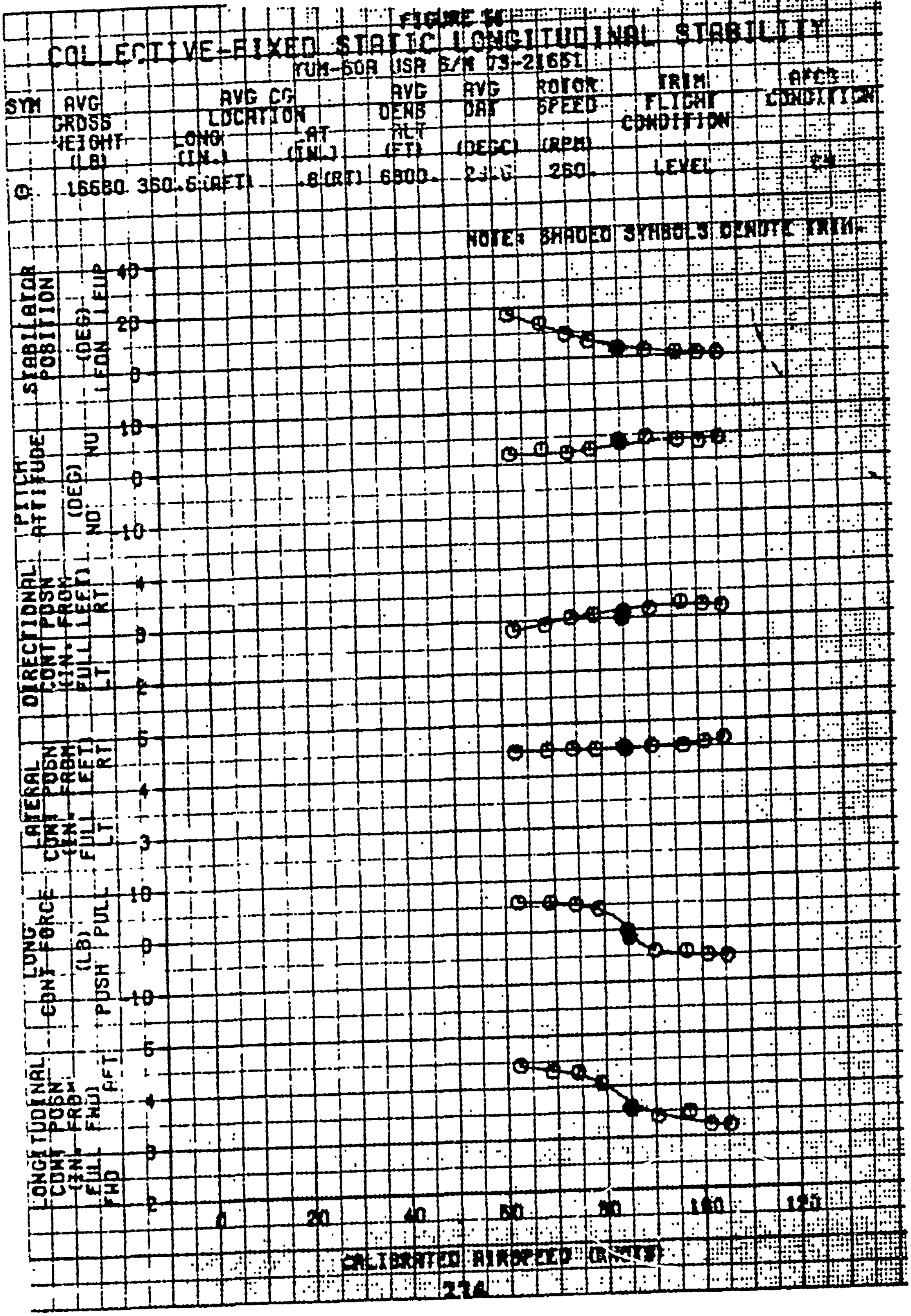
~~FOR OFFICIAL USE ONLY~~

FIGURE 51
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YUH-60A USA S/N 73-21651

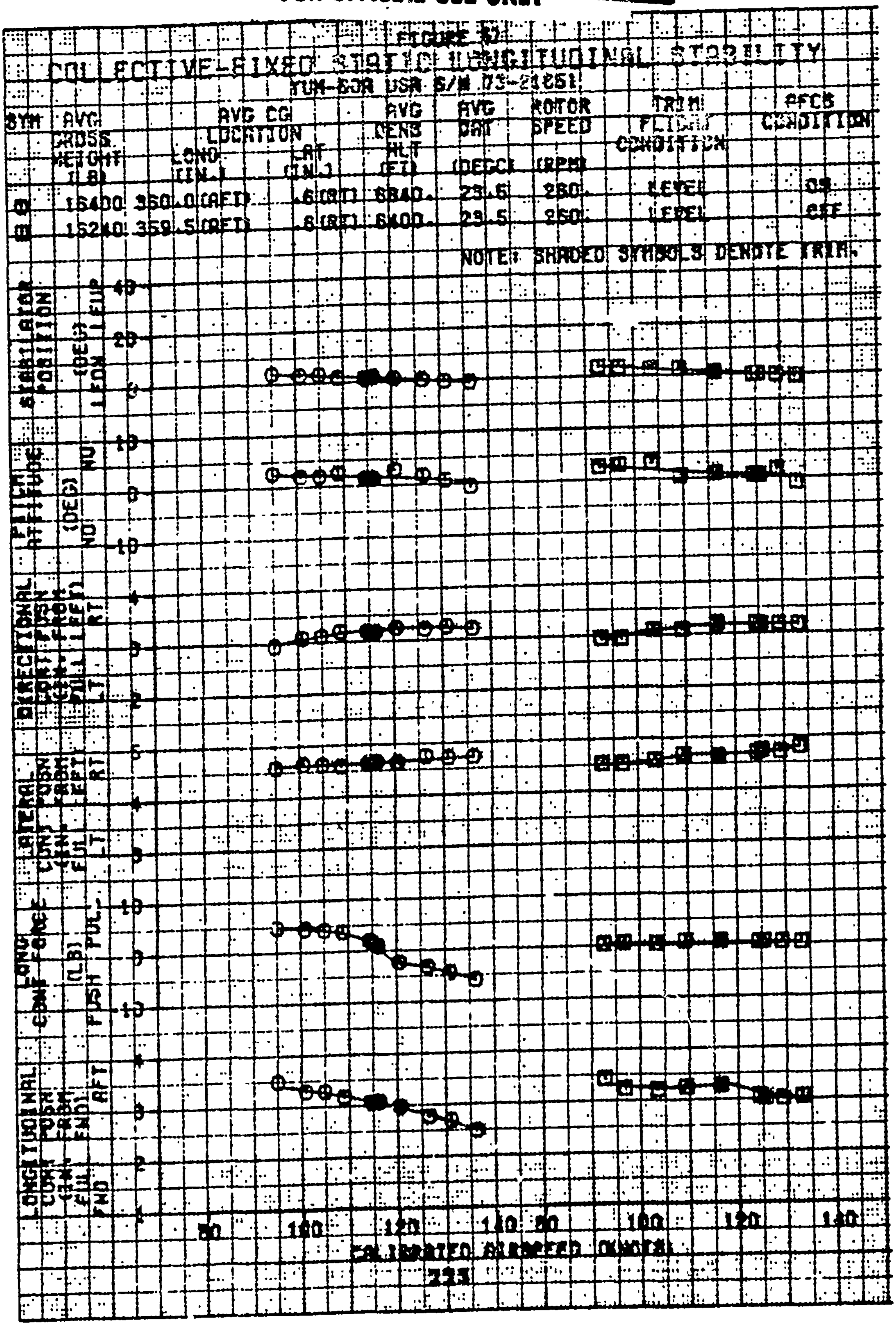


~~FOR OFFICIAL USE ONLY~~

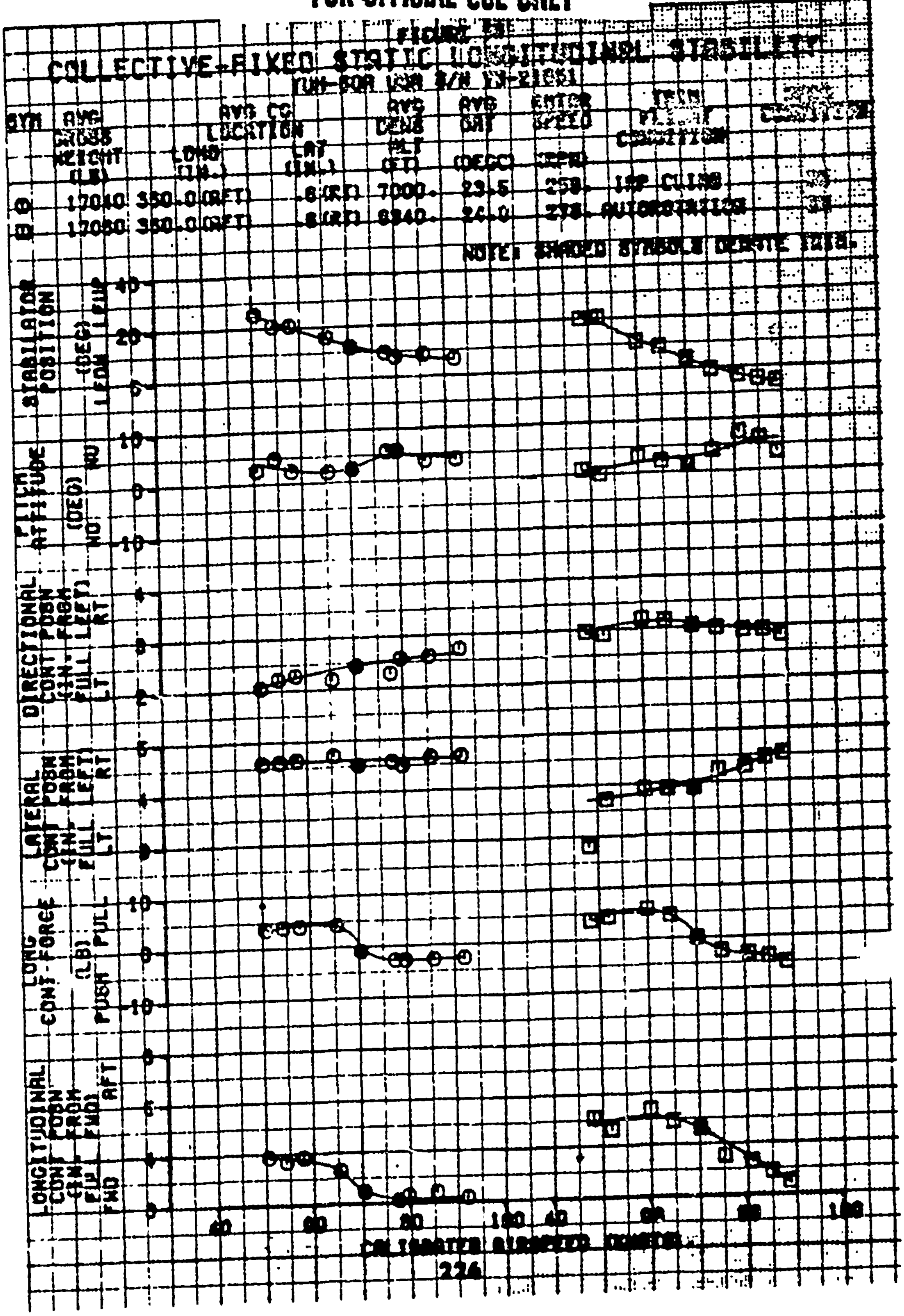



COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YUH-60A USA S/N 73-21651
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM FLIGHT CONDITION
AFCS CONDITION
16580
360.6(AFT)
.8(RT)
6800
260
LEVEL
NOTE: SHADED SYMBOLS DENOTE TRIM.
STABILATOR POSITION (DEG) LEDN LEUP
PITCH ATTITUDE (DEG) ND NU
DIRECTIONAL CONT POSN (IN. FROM FULL LEFT) LT RT
LATERAL CONT POSN (IN. FROM FULL LEFT) LT RT
LONG CONT FORCE (LB) PUSH PULL
LONGITUDINAL CONT POSN (IN. FROM FULL FWD) FWD AFT
40
20
0
10
0
-10
4
3
2
5
4
3
10
0
-10
5
4
3
2
0
20
40
60
80
100
120
CALIBRATED AIRSPEED (KNOTS)

~~FOR OFFICIAL USE ONLY~~



FOR OFFICIAL USE ONLY



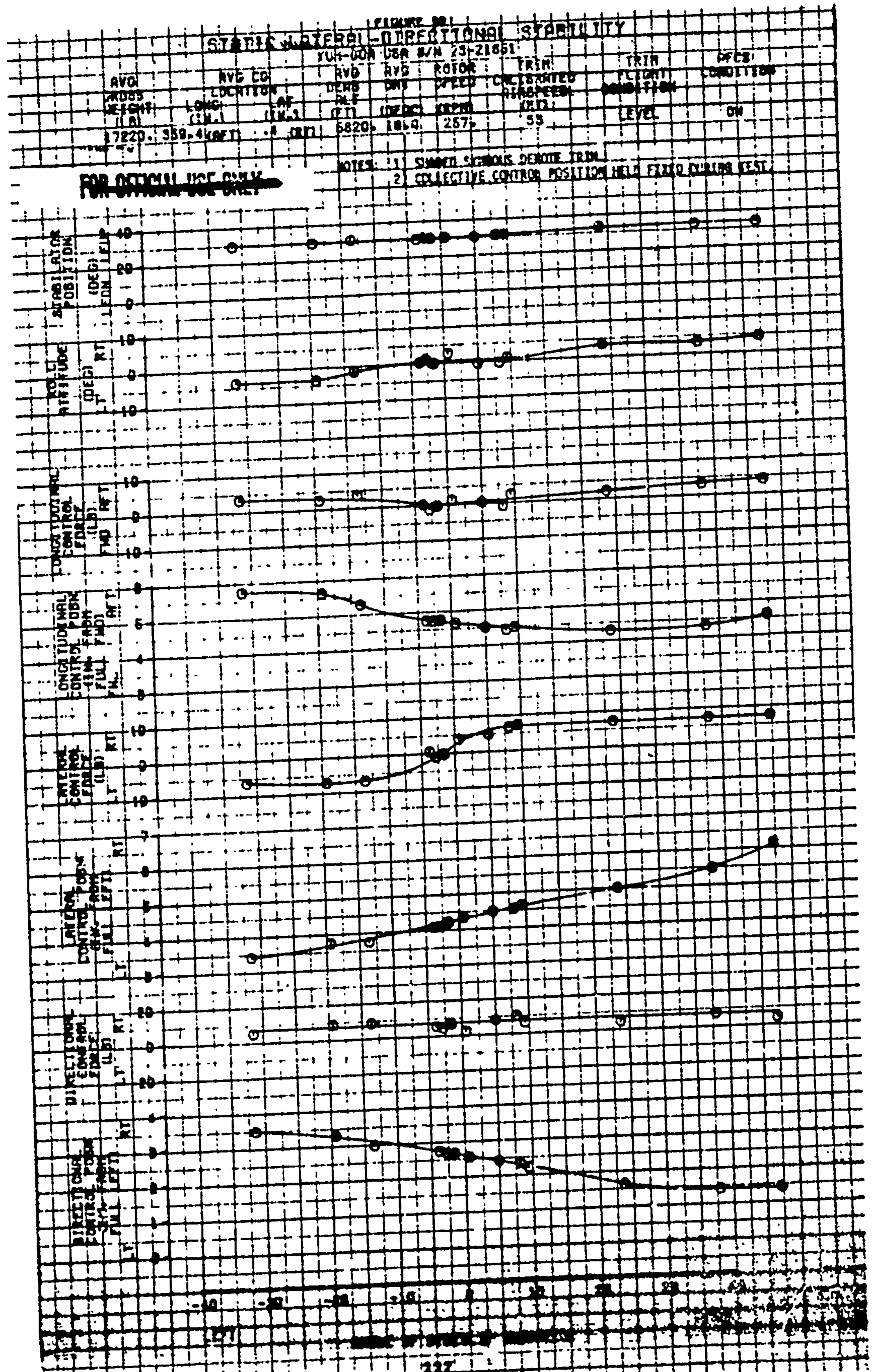


FIGURE 90
STATIC LATERAL-DIRECTIONAL STABILITY
YUH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB) 17220.
AVG CG LOCATION LONG (IN.) 359.4(AFT)
LAT (IN.) .4 (RT)
AVG DENS ALT (FT) 6820
AVG OAT (DEG C) 18.0
ROTOR SPEED (RPM) 257
TRIM CALIBRATED AIRSPEED (KT) 53
TRIM FLIGHT CONDITION LEVEL
AFCS CONDITION ON
NOTES: 1) SHADED SYMBOLS DENOTE TRIM.
2) COLLECTIVE CONTROL POSITION HELD FIXED DURING TEST.
FOR OFFICIAL USE ONLY
STABILATOR POSITION (DEG) LEDN
ROLL ATTITUDE (DEG) LT RT
LONGITUDINAL CONTROL FORCE (LB) FWD AFT
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD) AFT
LATERAL CONTROL FORCE (LB) LT RT
LATERAL CONTROL POSN (IN. FROM FULL LT) RT
DIRECTIONAL CONTROL FORCE (LB) LT RT
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT) LT RT
LFT
237

FIGURE 40

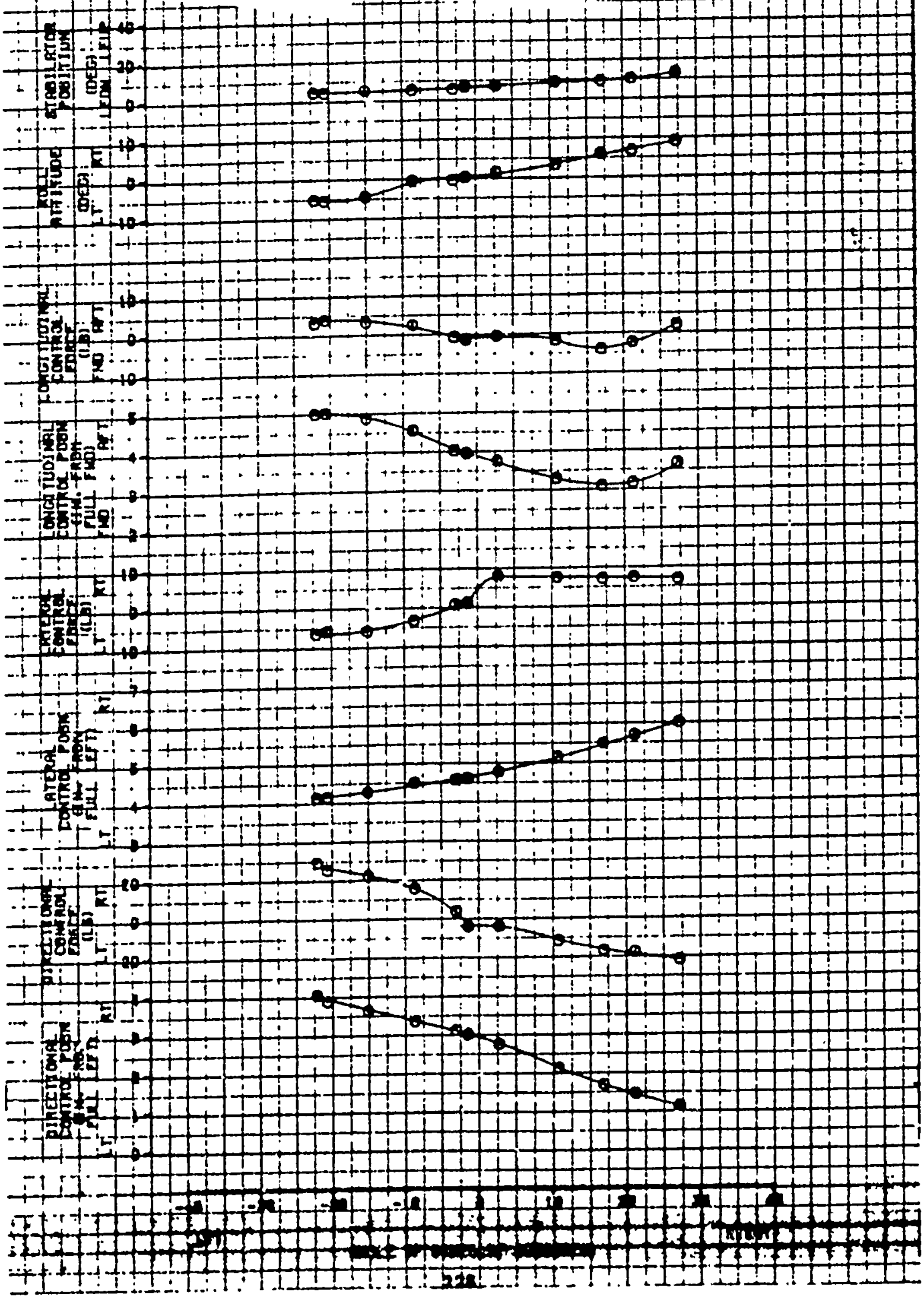
STATIC LATERAL-DIRECTIONAL STABILITY

YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG. (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	TRIM FLIGHT CONDITION	SAS CONDITION
16840	358.4(AFT)	.4 (RT)	8080	17.0	257	80	LEVEL	ON

FOR OFFICIAL USE ONLY

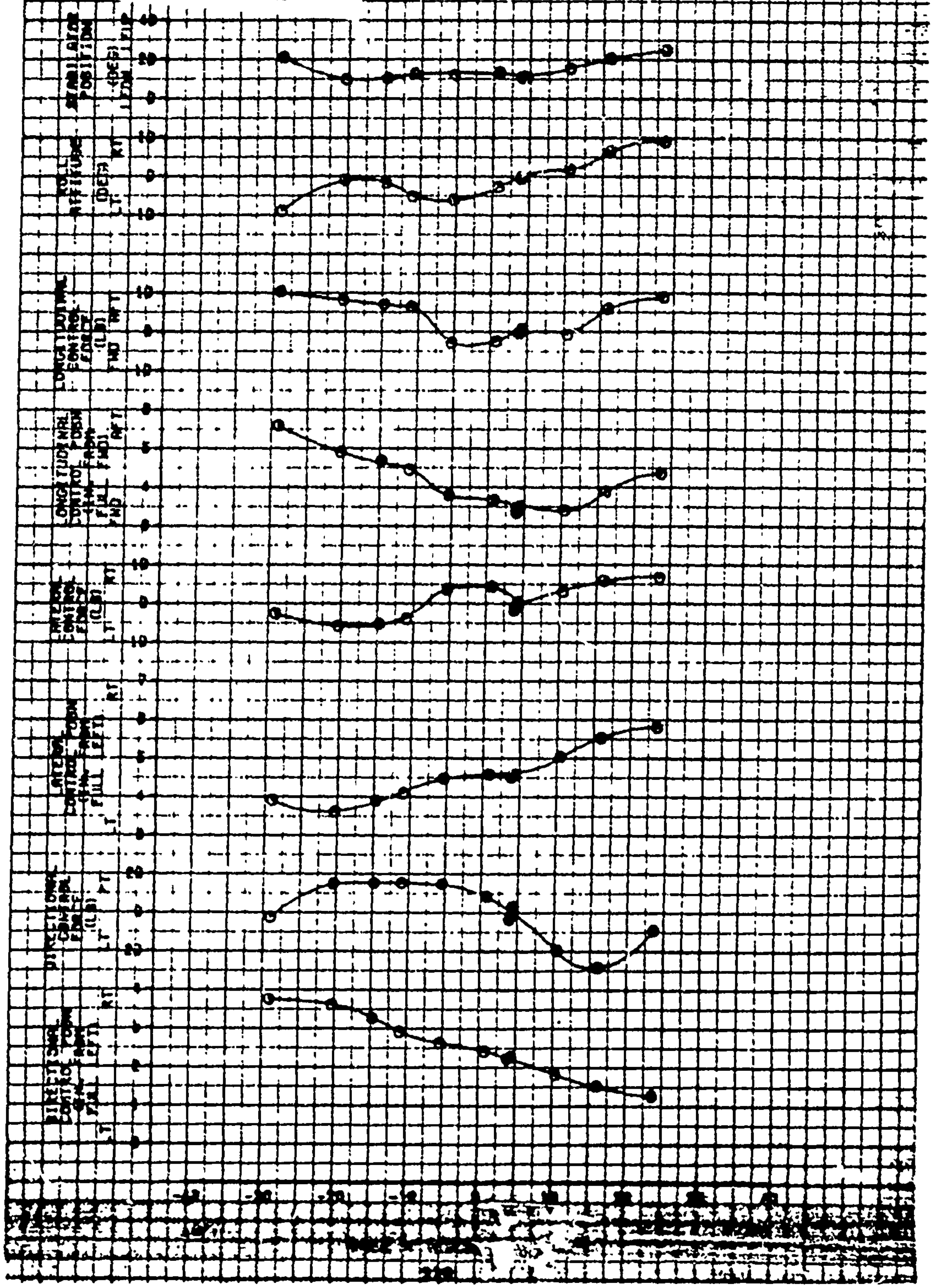
NOTES: 1) SHADED SYMBOLS DENOTE TRIM.
2) COLLECTIVE CONTROL POSITION HELD FIXED DURING TEST.

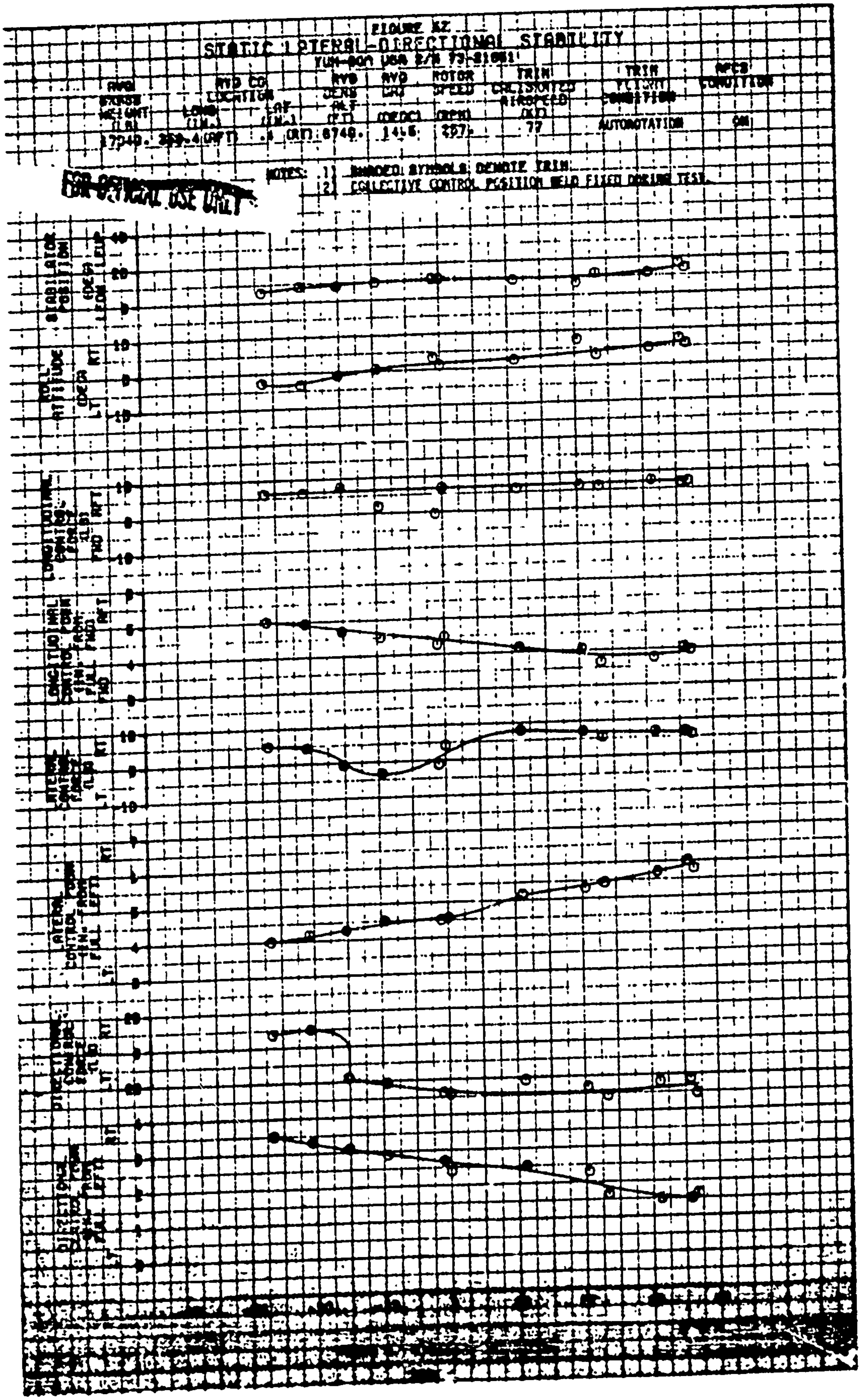


STATIC LATERAL-DIRECTIONAL STABILITY

FOR OFFICIAL USE ONLY

NOTES: (1) SHADED SYMBOLS DENOTE TRIM
(2) COLLECTIVE CONTROL POSITION HELD FIXED DURING TEST.



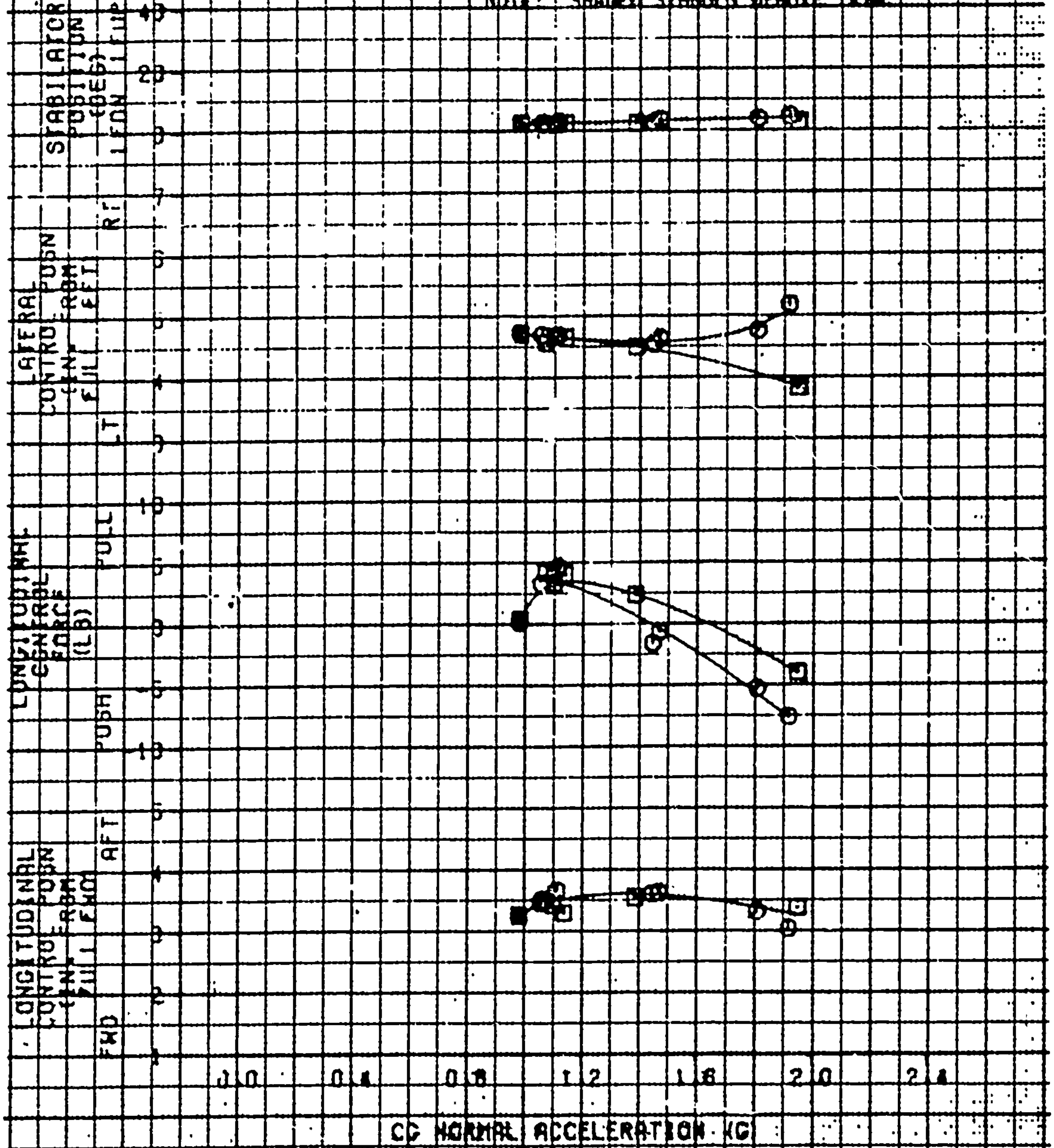

STATIC LATERAL-DIRECTIONAL STABILITY
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CALIBRATED AIRSPEED (KT)
TRIM FLIGHT CONDITION
SCAS CONDITION
17240.
(AFT)
(RT)
8740.
14.6
297.
77
AUTOROTATION
ON
NOTES: 1) SHADED SYMBOLS DENOTE TRIM.
2) COLLECTIVE CONTROL POSITION HELD FIXED DURING TEST.
FOR OFFICIAL USE ONLY

FOR OFFICIAL USE ONLY

FIGURE 53
MANEUVERING STABILITY
YUH-60A USA S/N 73-21651
TRIM CONDITION: 115 KCAS, PLF

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SFCS CONDITION
○	16460.	358.7 (AFT)	4 (RT)	6780.	15.5	259.	RIGHT TURN	ON
□	16300.	358.8 (AFT)	4 (RT)	6900.	15.5	258.	LEFT TURN	ON

NOTE: SHADED SYMBOLS DENOTE TRIM



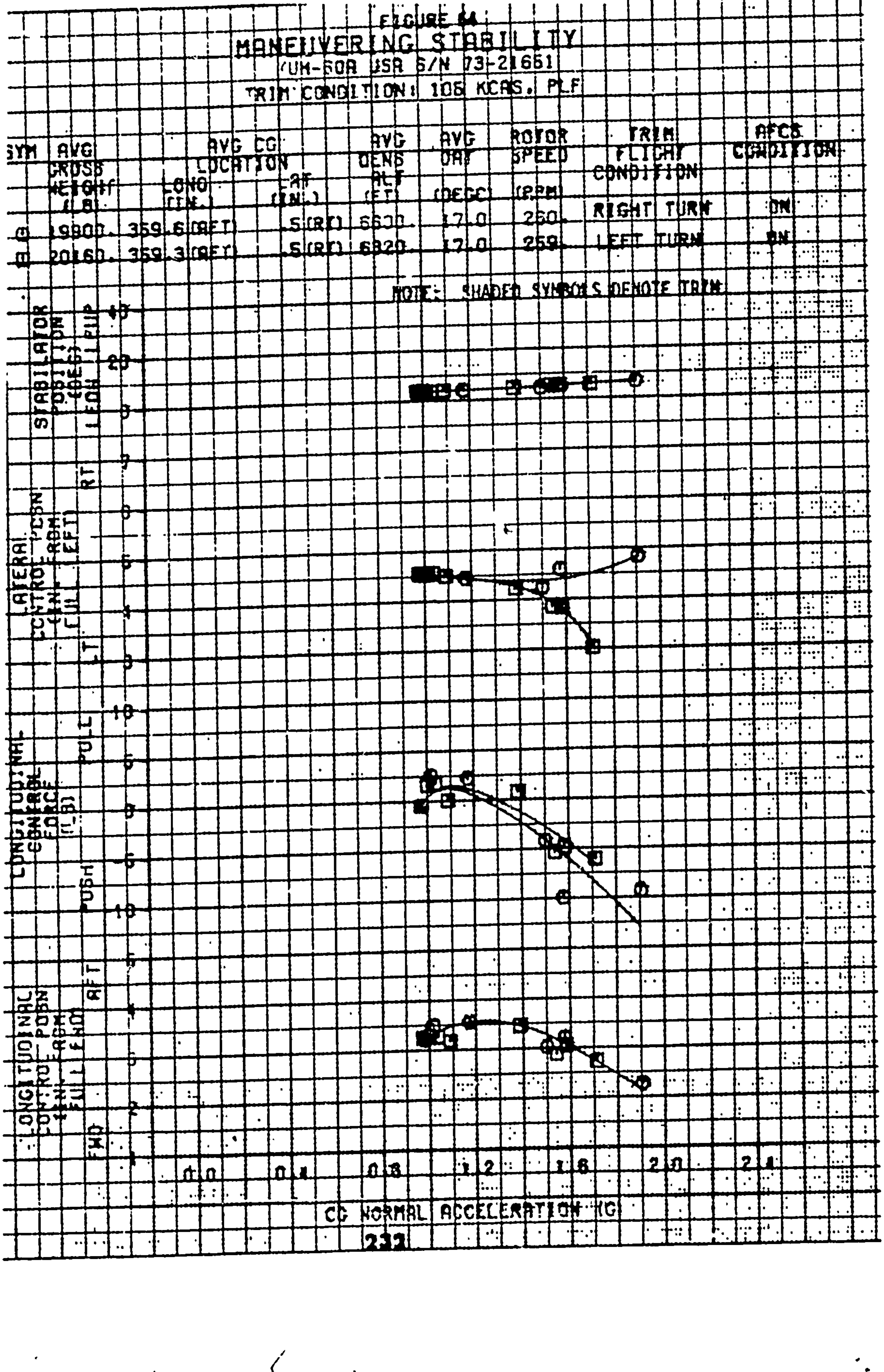


FIGURE 64
MANEUVERING STABILITY
UH-60A USA S/N 73-21651
TRIM CONDITION: 106 KCAS, PLF
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM FLIGHT CONDITION
AFCS CONDITION
19800. 359.6(AFT) .5(RT) 6630. 17.0 260. RIGHT TURN ON
20160. 359.3(AFT) .5(RT) 6820. 17.0 259. LEFT TURN ON
NOTE: SHADED SYMBOLS DENOTE TRIM
STABILATOR POSITION (DEG)
LATERAL CONTROL POSN FROM FULL LEFT (IN.)
RT
LT
LONGITUDINAL CONTROL FORCE (LB)
PULL
PUSH
LONGITUDINAL CONTROL POSN FROM FULL FWD (IN.)
AFT
FWD
0.0
0.4
0.8
1.2
1.6
2.0
2.4
CG NORMAL ACCELERATION (G)

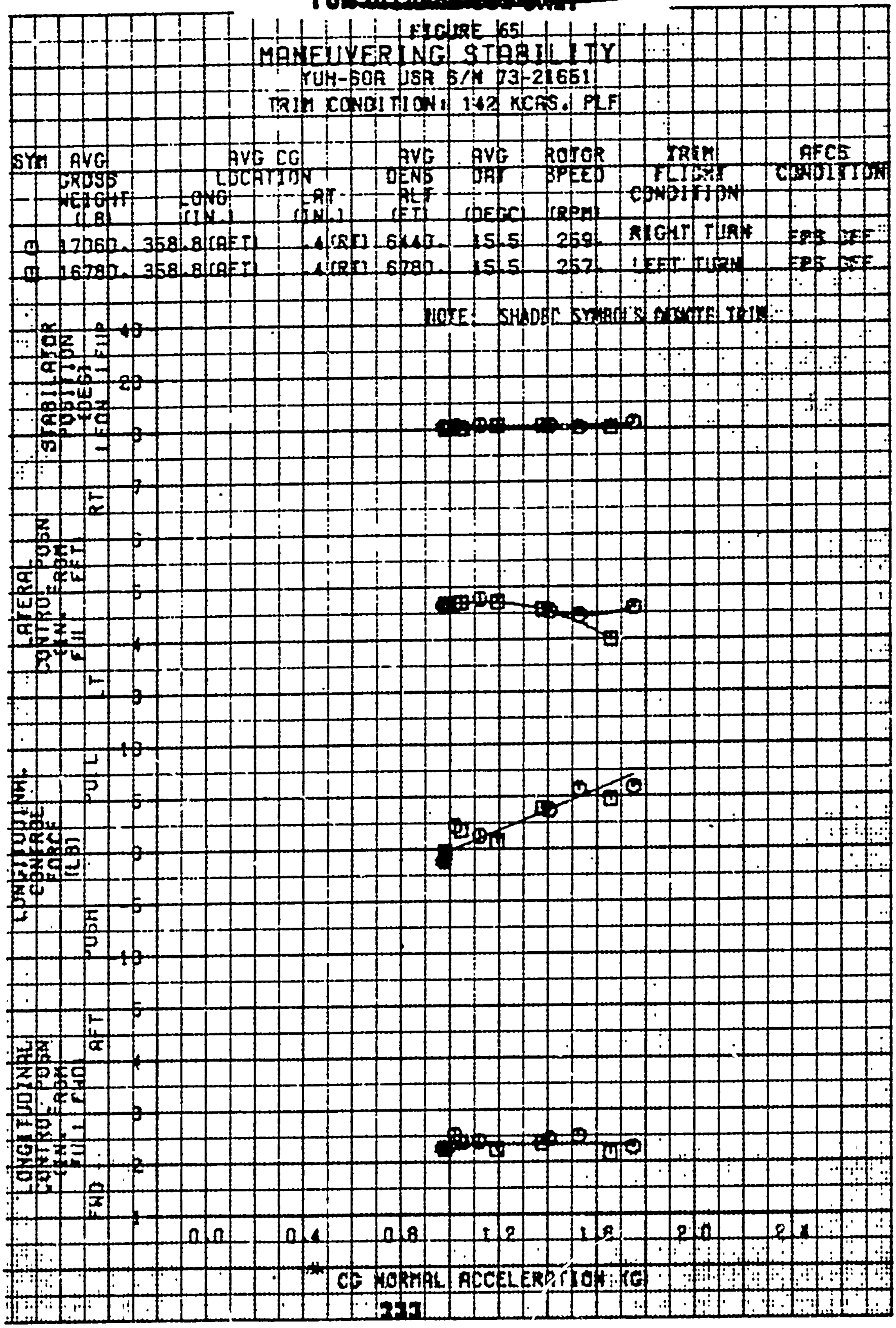


FIGURE 65
MANEUVERING STABILITY
YUH-60A USA S/N 73-21651
TRIM CONDITION: 142 KCAS, PLF
SYM
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN)
LAT (IN)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM FLIGHT CONDITION
AFCS CONDITION
17060. 358.8 (AFT) .4 (RT) 6440. 15.5 259 RIGHT TURN FPS OFF
16780. 358.8 (AFT) .4 (RT) 6780. 15.5 257 LEFT TURN FPS OFF
NOTE: SHADED SYMBOLS DENOTE TRIM
STABILATOR POSITION (DEG)
TED
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
RT
LT
LONGITUDINAL CONTROL FORCE (LB)
PULL
PUSH
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
AFT
FWD
0.0
0.4
0.8
1.2
1.6
2.0
2.4
CG NORMAL ACCELERATION (G)

FIGURE 56

MANEUVERING STABILITY

YUH-60A USA S/N 73-21651

TRIM CONDITION: 132 KCAS, PLF

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
◻	19720.	359.7(AFT)	.5(RT)	7100.	16.5	280.	RIGHT TURN	FPS OFF
◻	19820.	359.5(AFT)	.4(RT)	6780.	15.5	258.	LEFT TURN	FPS OFF

NOTE: SHADED SYMBOLS DENOTE TRIM.

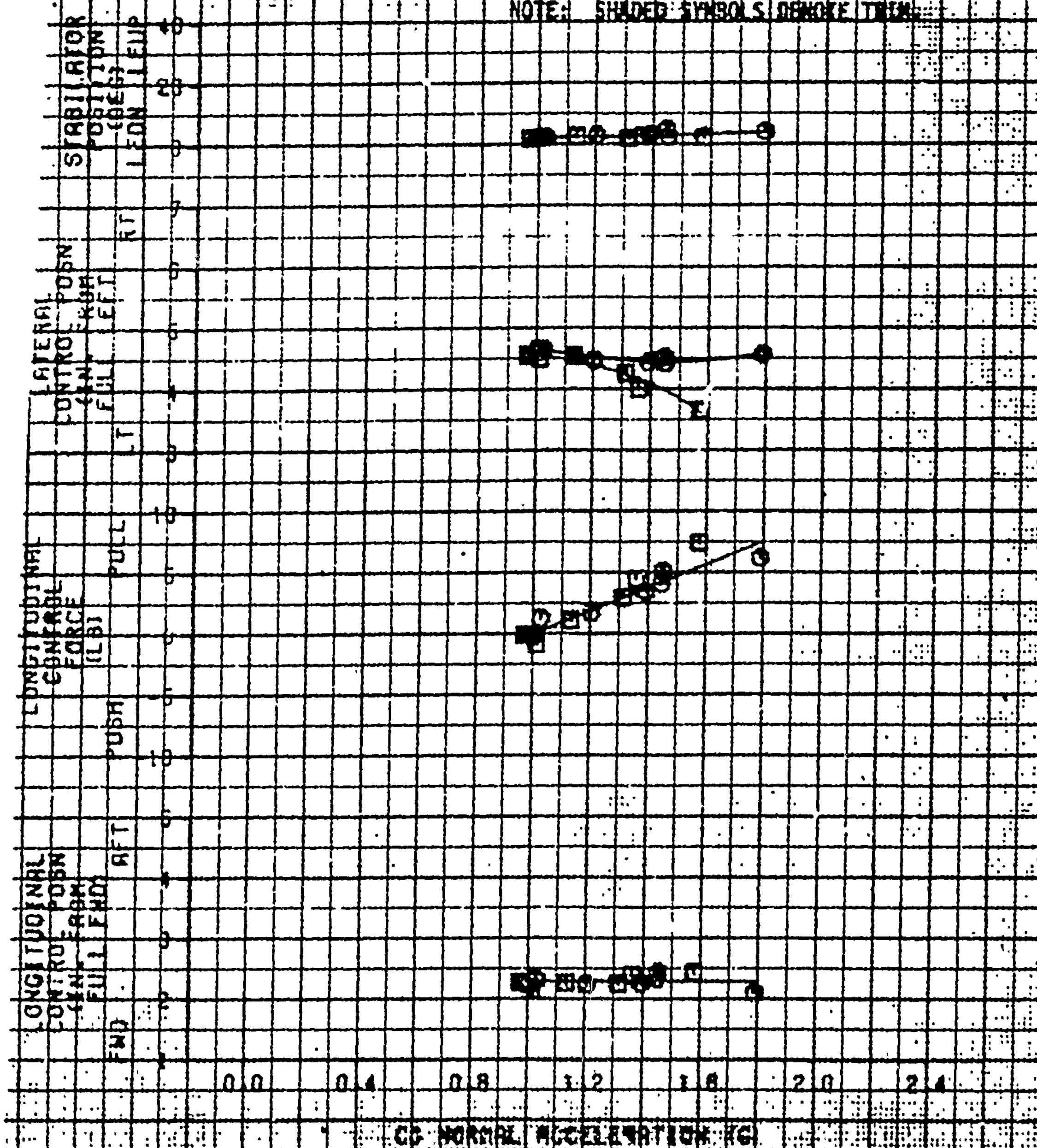


FIGURE 67
MANEUVERING STABILITY
YUH-60A USA S/N 73-21651
TRIM CONDITION: 114 KCAS, PLF

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
○	16550	358.9 (AFT)	0.4 (RT)	6420	16.5	260	SYM PULL UP	ON
□	16420	358.9 (AFT)	0.4 (RT)	6440	16.0	260	SYM PUSH OVER	ON

NOTE: SHADED SYMBOLS DENOTE TRIM.

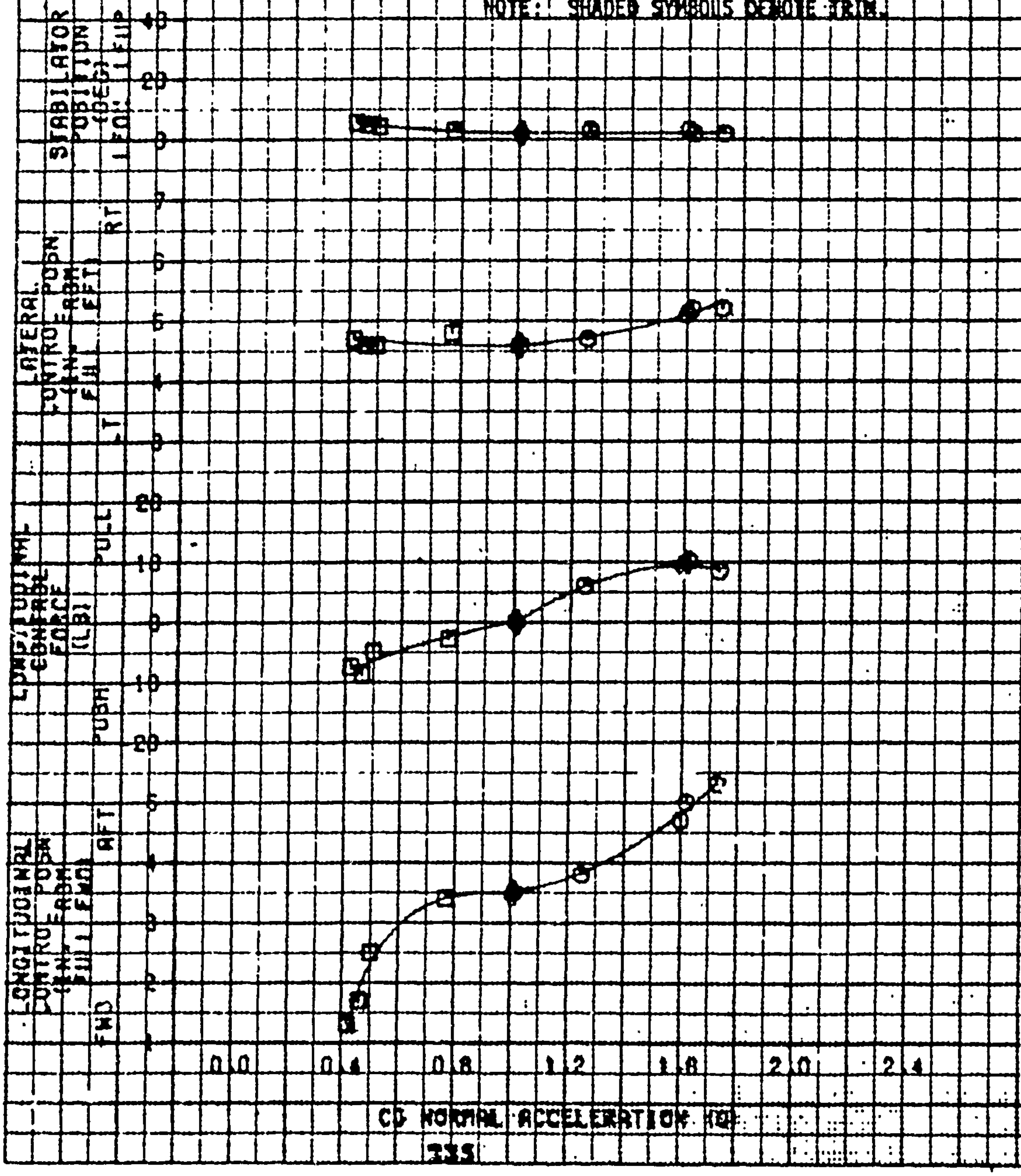


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FIGURE 68
MANEUVERING STABILITY
YUH-60A USA S/N 73-21651
TRIM CONDITION: 106 KCAS, PLF

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
○	20080	359.2 (AFT)	0.4 (RT)	6160	18.0	258	SYM PULL UP	ON
□	19860	359.5 (AFT)	0.4 (RT)	6100	18.0	258	SYM PUSH OVER	ON

NOTE: SHADED SYMBOL DENOTES TRIM

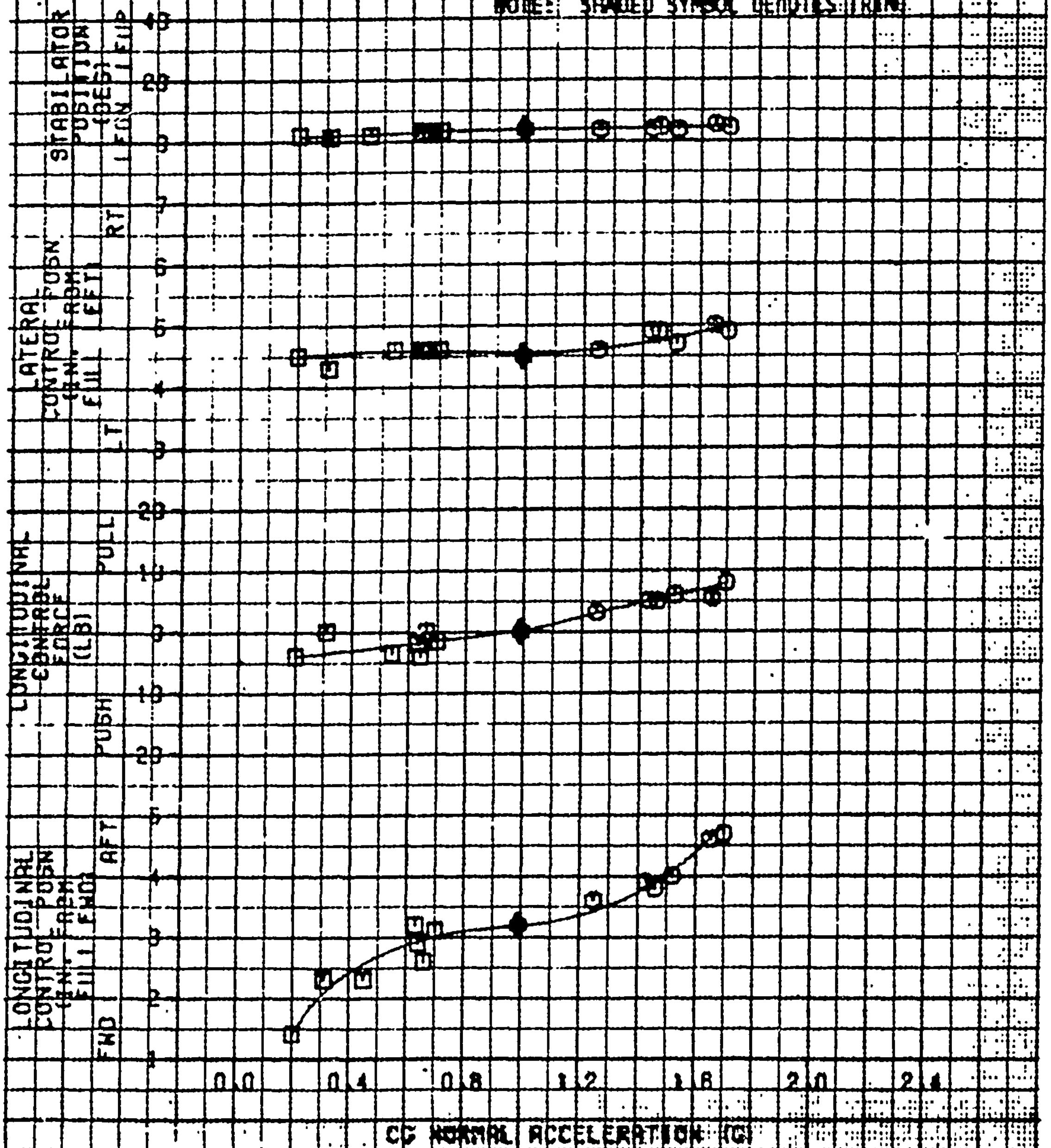

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FIGURE 69

MANEUVERING STABILITY

YUH-60A USA S/N 73-21651

TRIM CONDITION: 143 KCAS, RLF

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFCS CONDITION
○	16960	358.9 (AFT)	0.4 (RT)	6760	15.0	257	SYM PULL UP	FPS OFF
□	16790	358.9 (AFT)	0.4 (RT)	5780	16.5	250	SYM PUSH OVER	FPS OFF

NOTE: SHADED SYMBOL DENOTES TRIM.

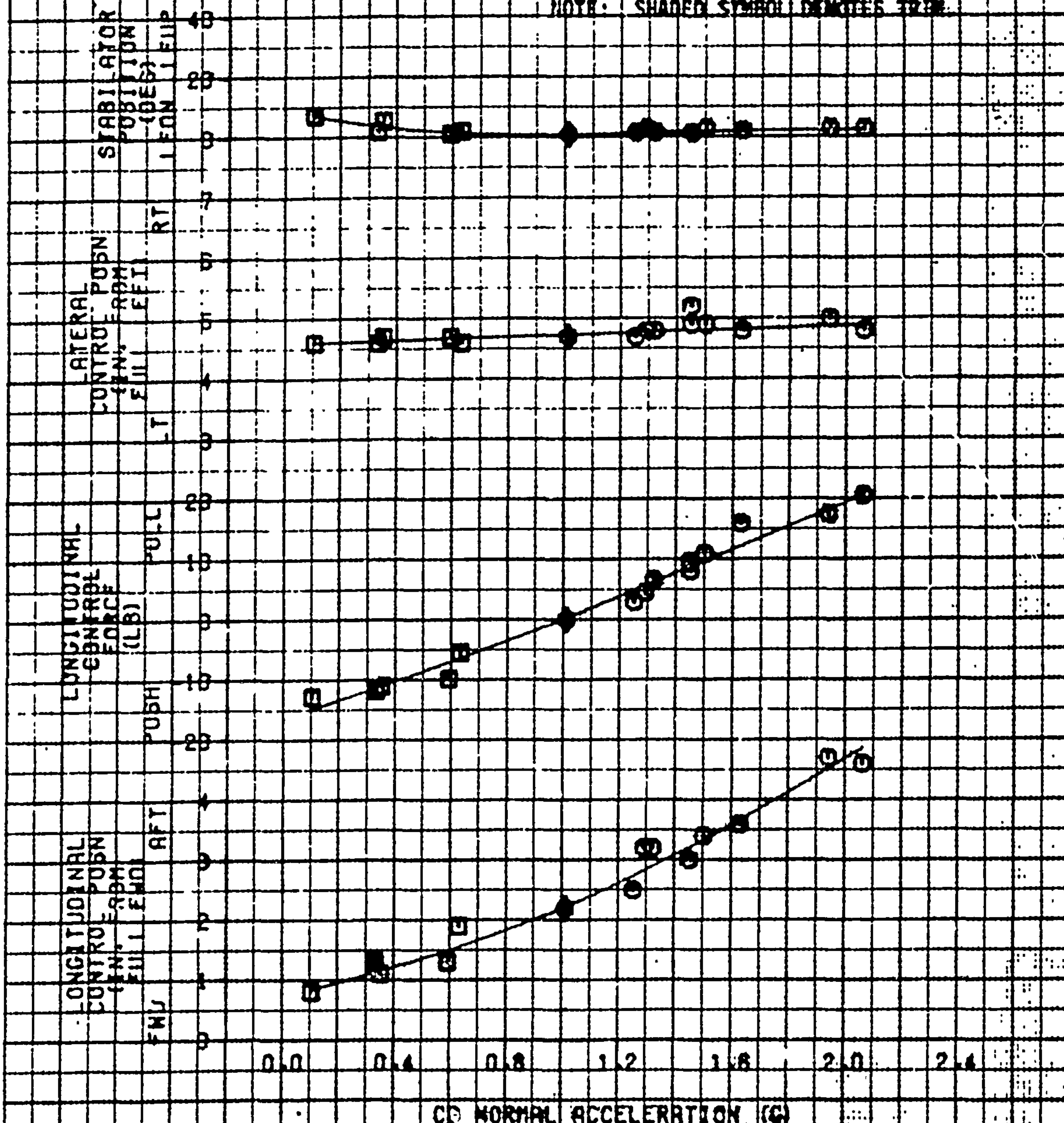

FIGURE 70
MANEUVERING STABILITY
YUH-60A USA S/N 73-21651
TRIM CONDITION: 130 KCAS, PLF

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	AFC CONDI
○	19620	359.8 (AFT)	0.4 (RT)	5800	18.0	258	SYM PULL UP	FPS
□	19400	360.0 (AFT)	0.4 (RT)	5740	18.0	259	SYM PUSH OVER	FPS

NOTE: SHADED SYMBOL DENOTES TRIM.

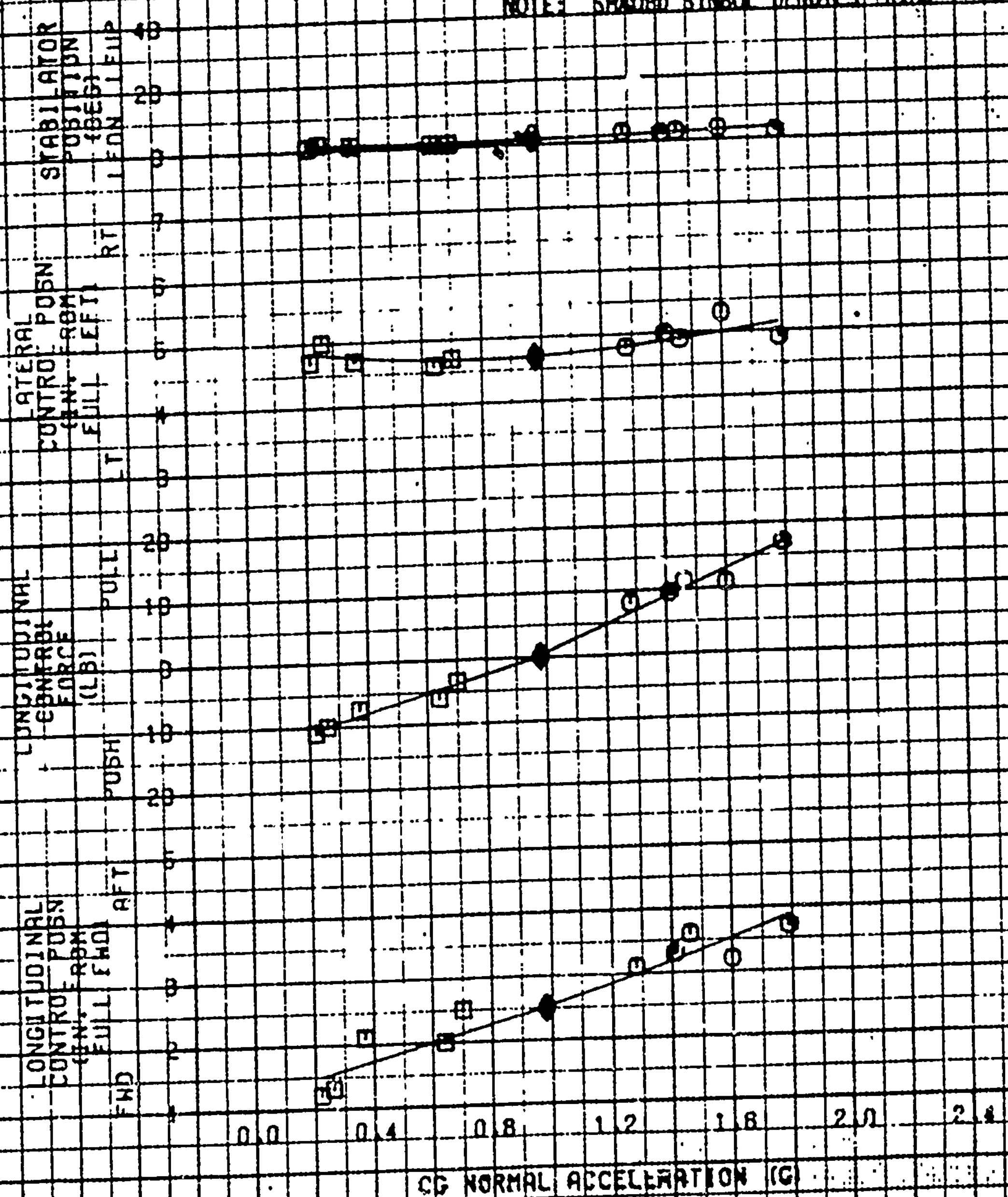

FIGURE 71

AFT LONGITUDINAL PULSE

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16610	359.6 (AFT)	0.4 (RT)	8420	30.0	258	115	LEVEL	ON

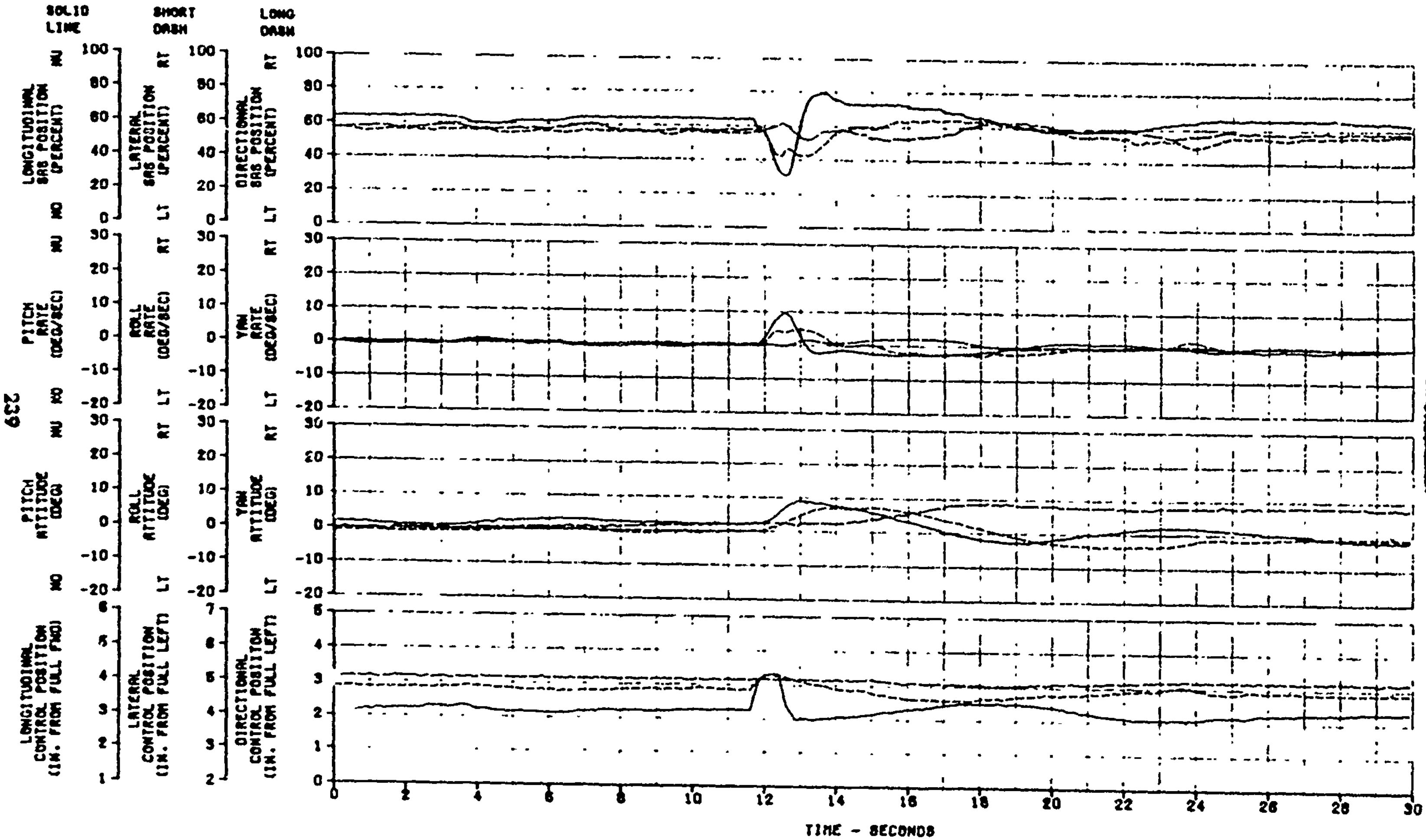

FIGURE 72
RIGHT LATERAL PULSE
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16470	359.4 (AFT)	0.4 (RT)	6440	29.5	258	115	LEVEL	ON

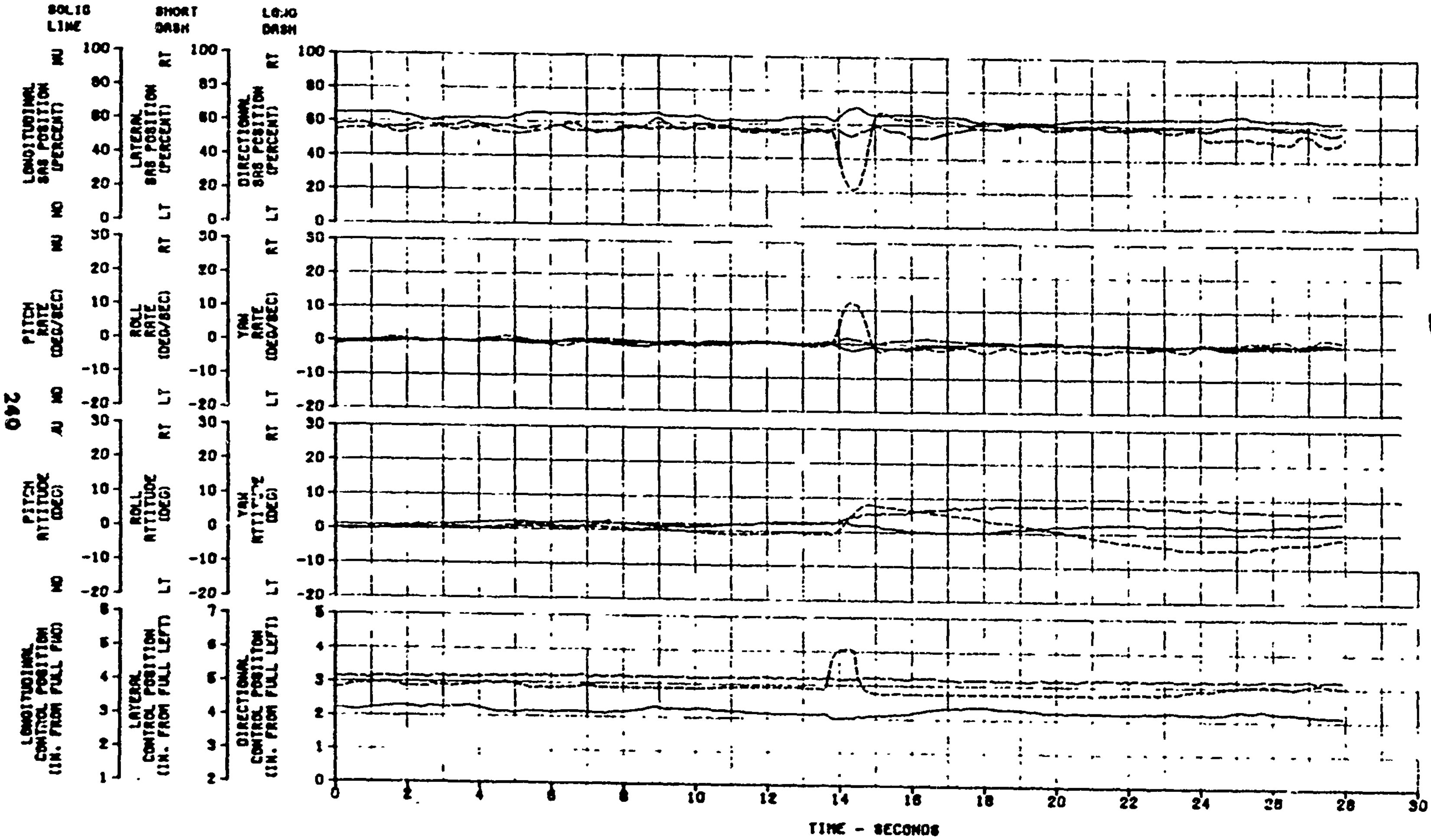

FIGURE 73

LEFT DIRECTIONAL PULSE

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16860	359.5 (AFT)	0.4 (RT)	6320	30.0	258	76	LEVEL	ON

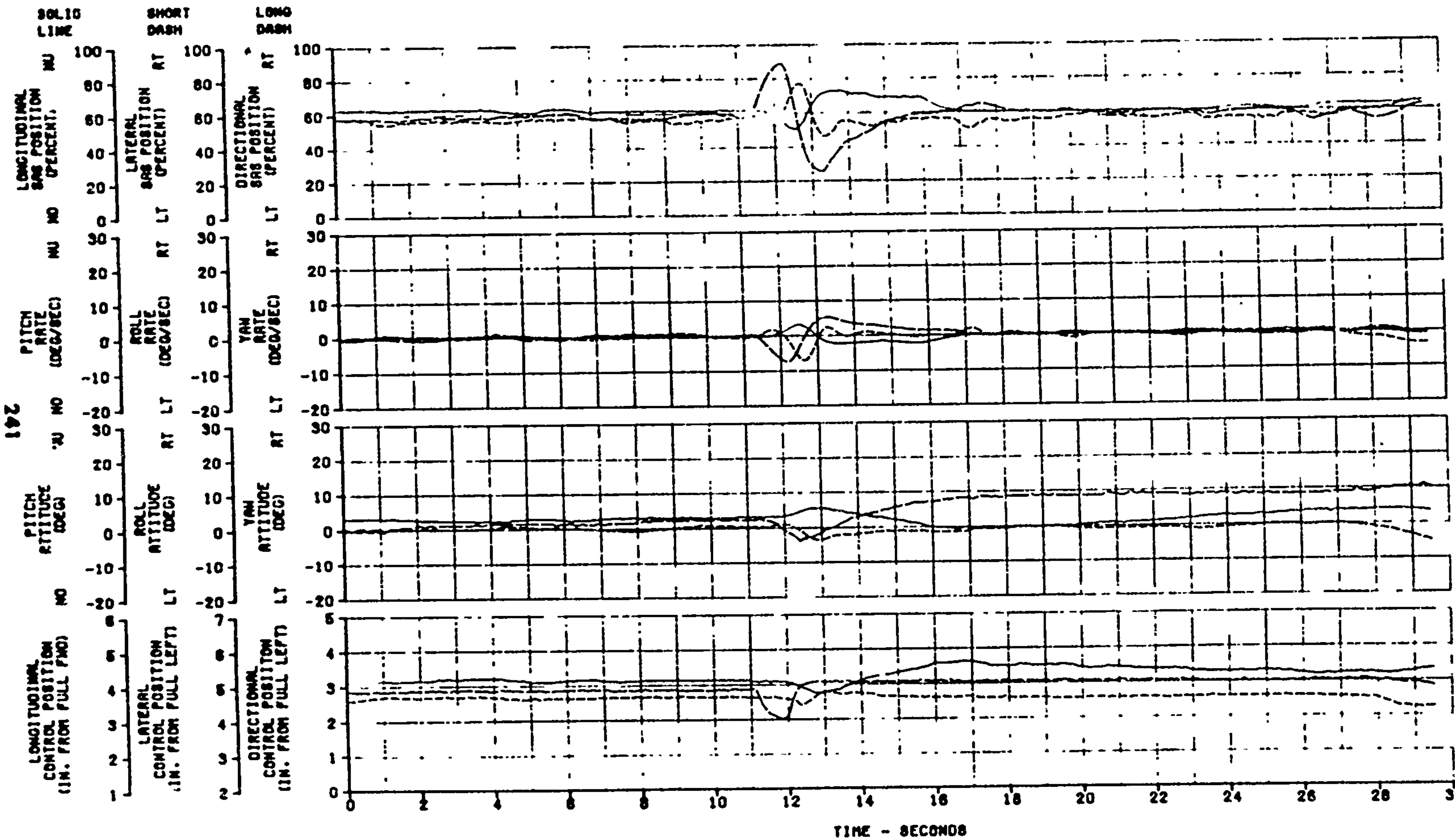

FIGURE 74
LEFT LATERAL PULSE
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16630	359.5 (AFT)	0.4 (RT)	6140	25.5	257	111	LEVEL	OFF

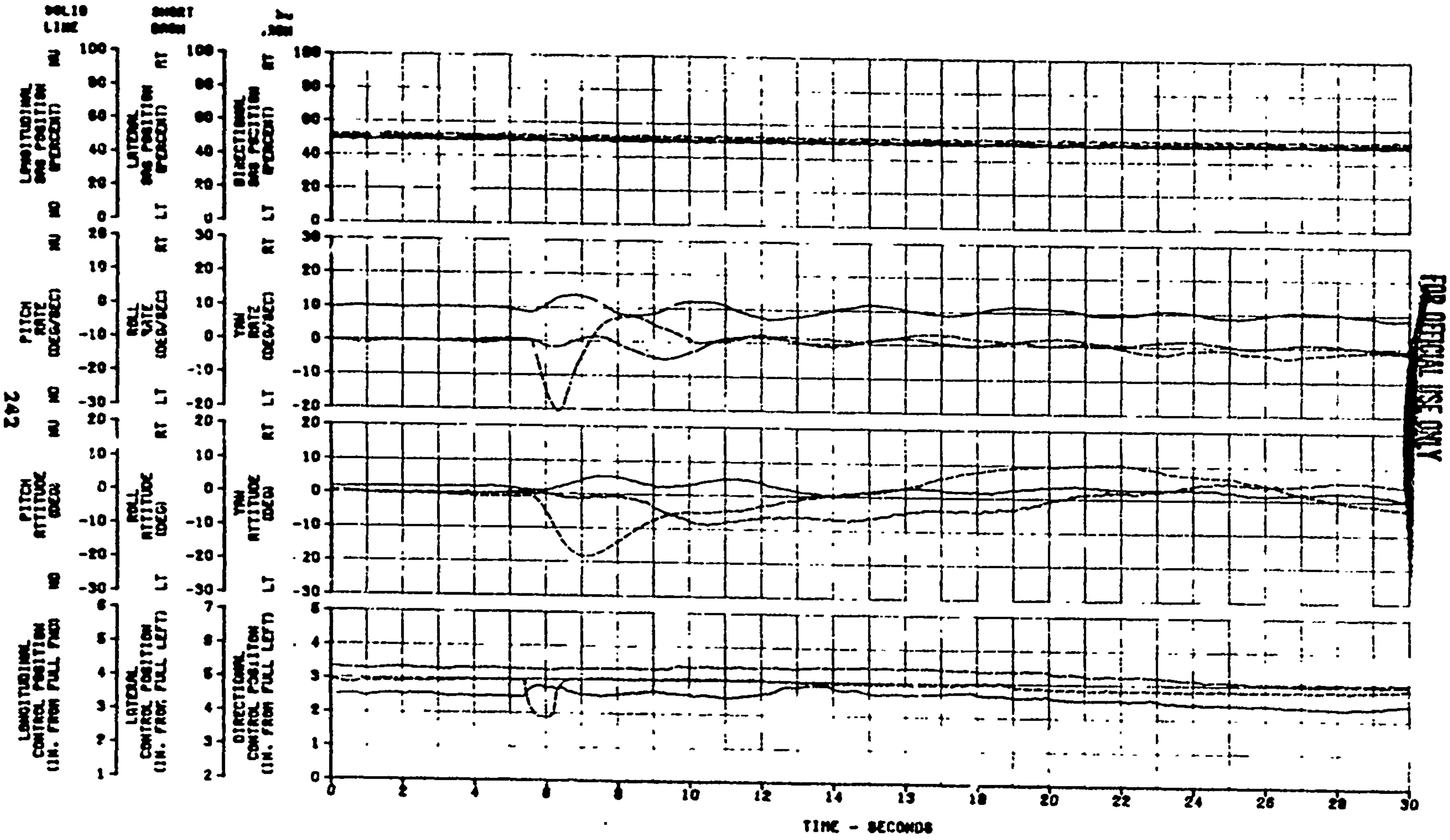

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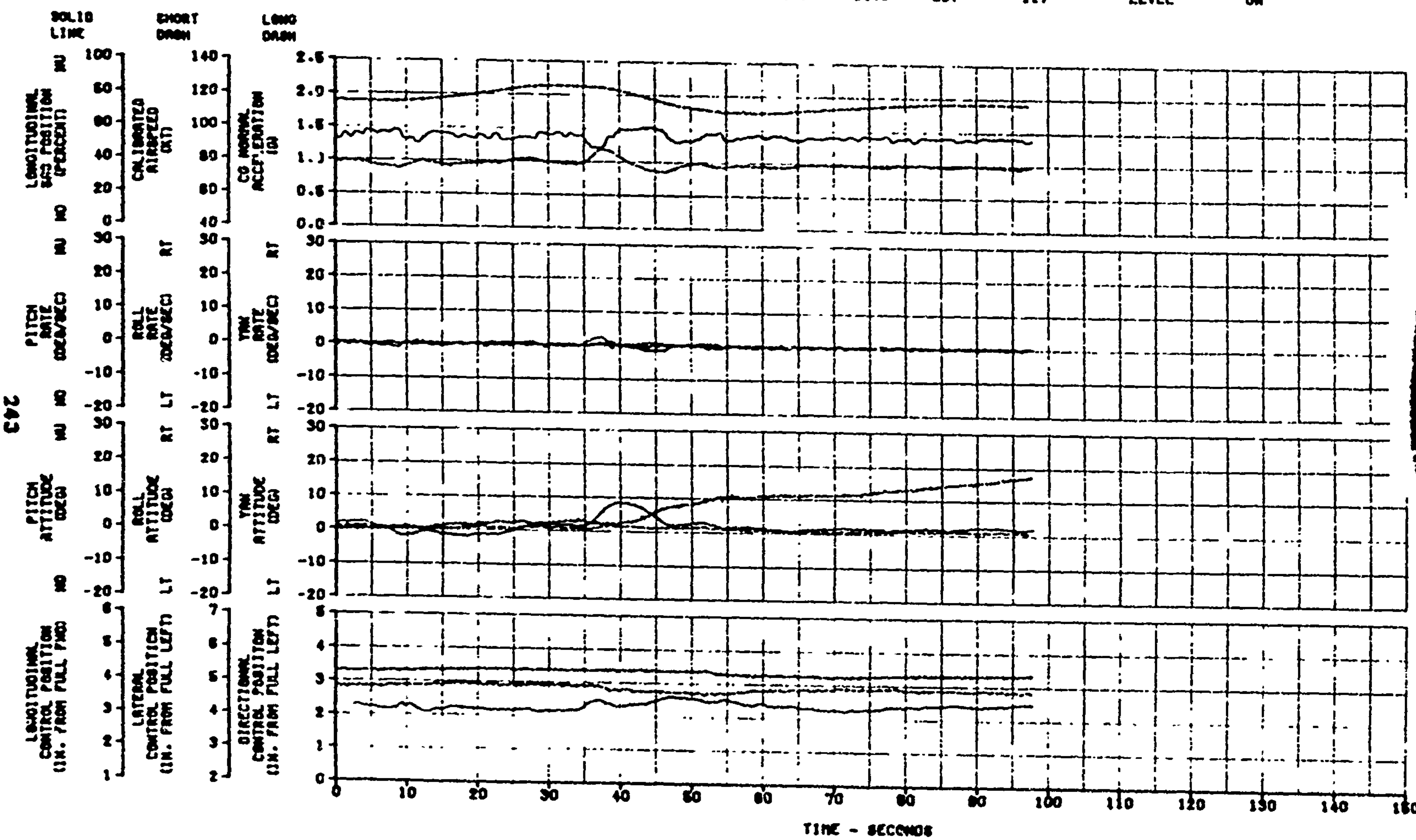
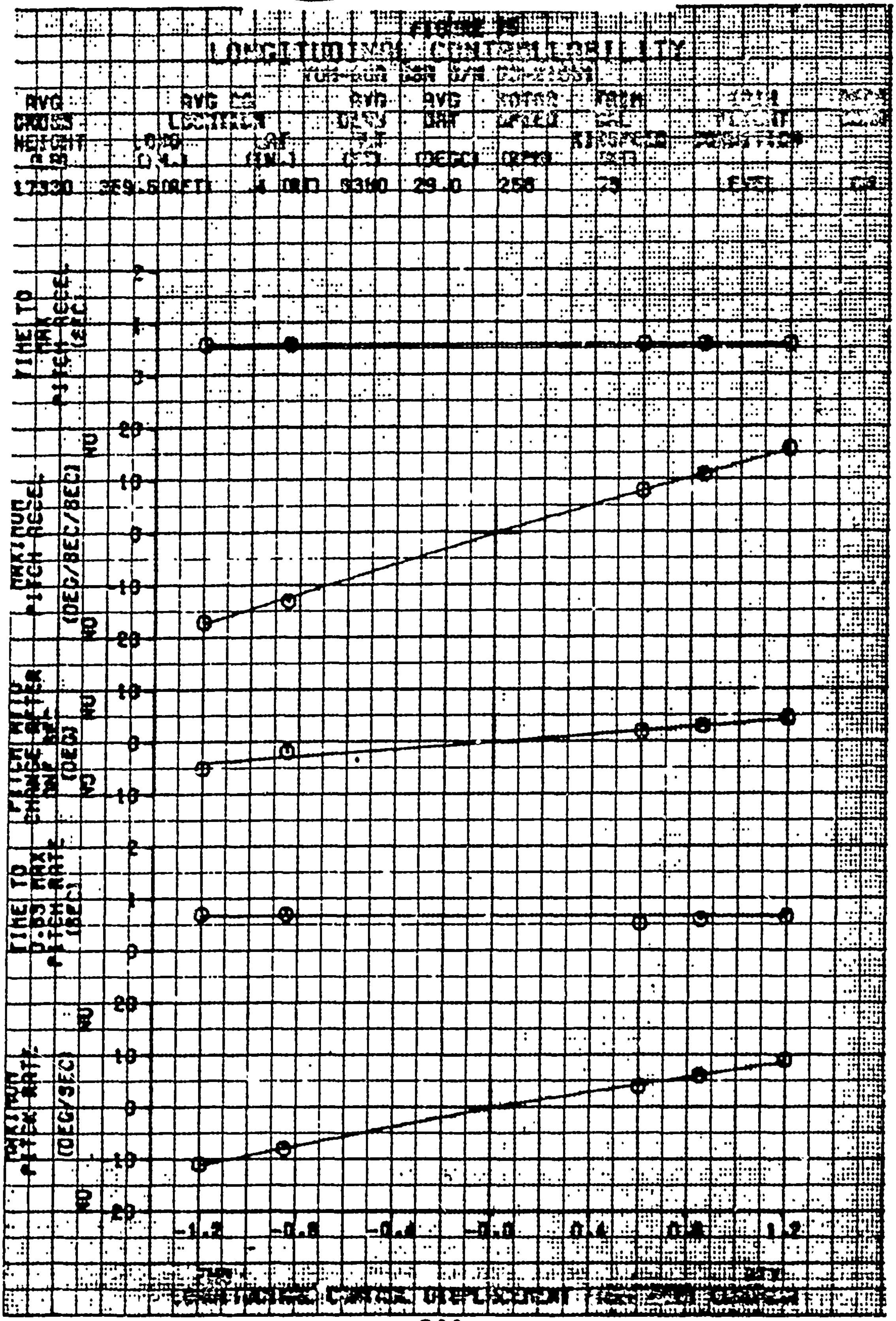
FIGURE 75

LONGITUDINAL LONG TERM RESPONSE

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16430	358.2(AFT)	0.4(RT)	6160	25.5	257	117	LEVEL	ON




AVG
GROSS
WEIGHT
(LB)
AVG CG
LOCATION
AVG
AVG
ROTOR
SPEED
(RPM)
17330
258
TIME TO
(SEC)
MAXIMUM
PITCH ACCEL
(DEG/SEC/SEC)
(DEG)
TIME TO
0.63 MAX
PITCH RATE
(SEC)
MAXIMUM
PITCH RATE
(DEG/SEC)
ND
NU
20
10
0
-10
-20
-1.2
-0.8
-0.4
0.0
0.4
0.8
1.2

[redacted]

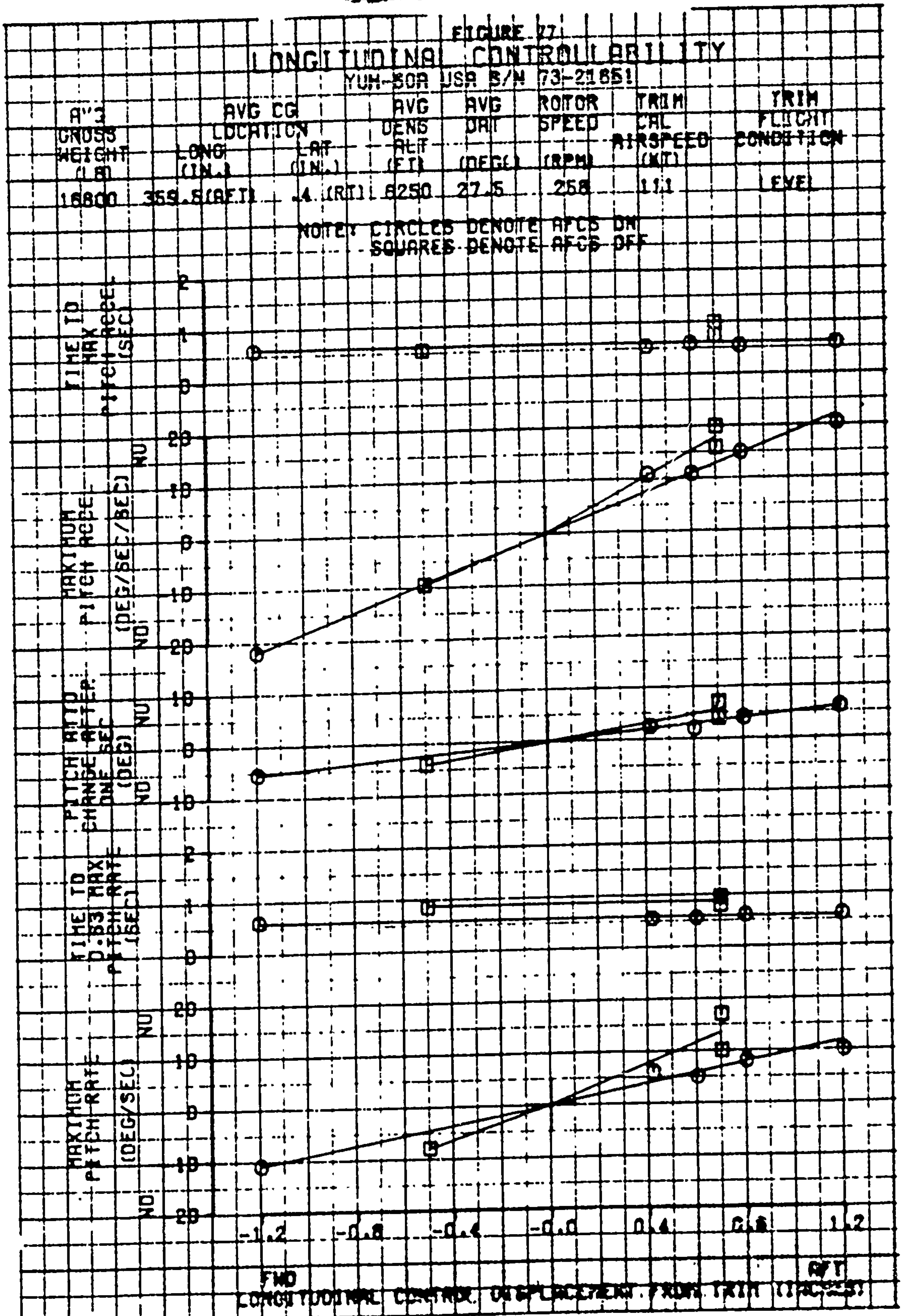


FIGURE 27
LONGITUDINAL CONTROLLABILITY
YUH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
16800
355.8(AFT)
.4 (RT)
6250
27.5
258
111
LEVEL
NOTE: CIRCLES DENOTE AFCS ON
SQUARES DENOTE AFCS OFF
TIME TO MAX PITCH ACCEL (SEC)
MAXIMUM PITCH ACCEL (DEG/SEC/SEC)
PITCH ATTD CHANGE AFTER ONE SEC (DEG)
TIME TO 0.63 MAX PITCH RATE (SEC)
MAXIMUM PITCH RATE (DEG/SEC)
NU
ND
-1.2
-0.8
-0.4
-0.0
0.4
0.8
1.2
FWD
AFT
LONGITUDINAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

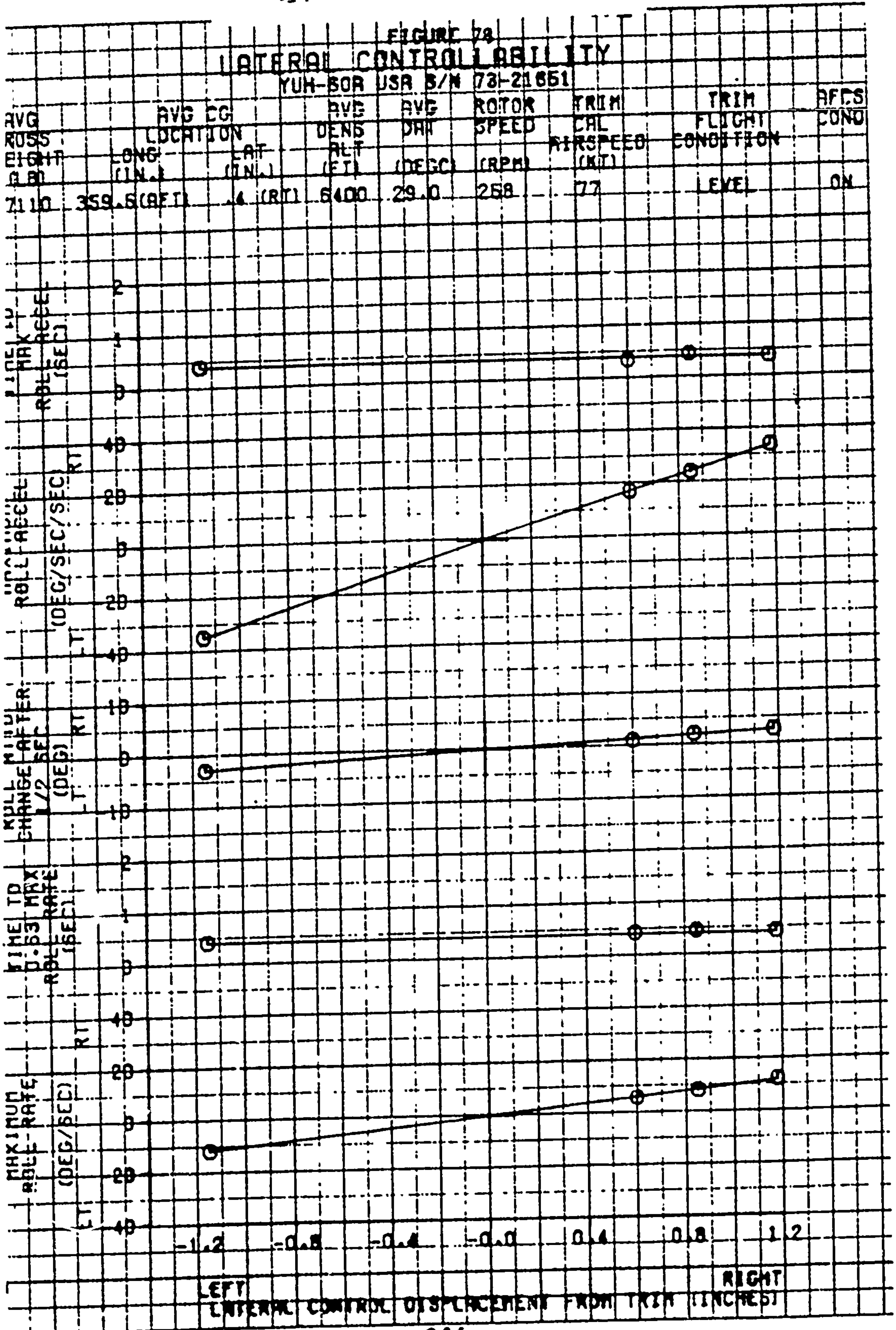


FIGURE 78
LATERAL CONTROLLABILITY
YUH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
AFCS COND
7110
359.6(AFT)
.4 (RT)
6400
29.0
258
77
LEVEL
ON
TIME TO MAX ROLL ACCEL (SEC)
MAXIMUM ROLL ACCEL (DEG/SEC/SEC)
ROLL ATTITUDE CHANGE AFTER 1/2 SEC (DEG)
TIME TO 0.63 MAX ROLL RATE (SEC)
MAXIMUM ROLL RATE (DEG/SEC)
RT
LT
-1.2
-0.8
-0.4
-0.0
0.4
0.8
1.2
LEFT
RIGHT
LATERAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

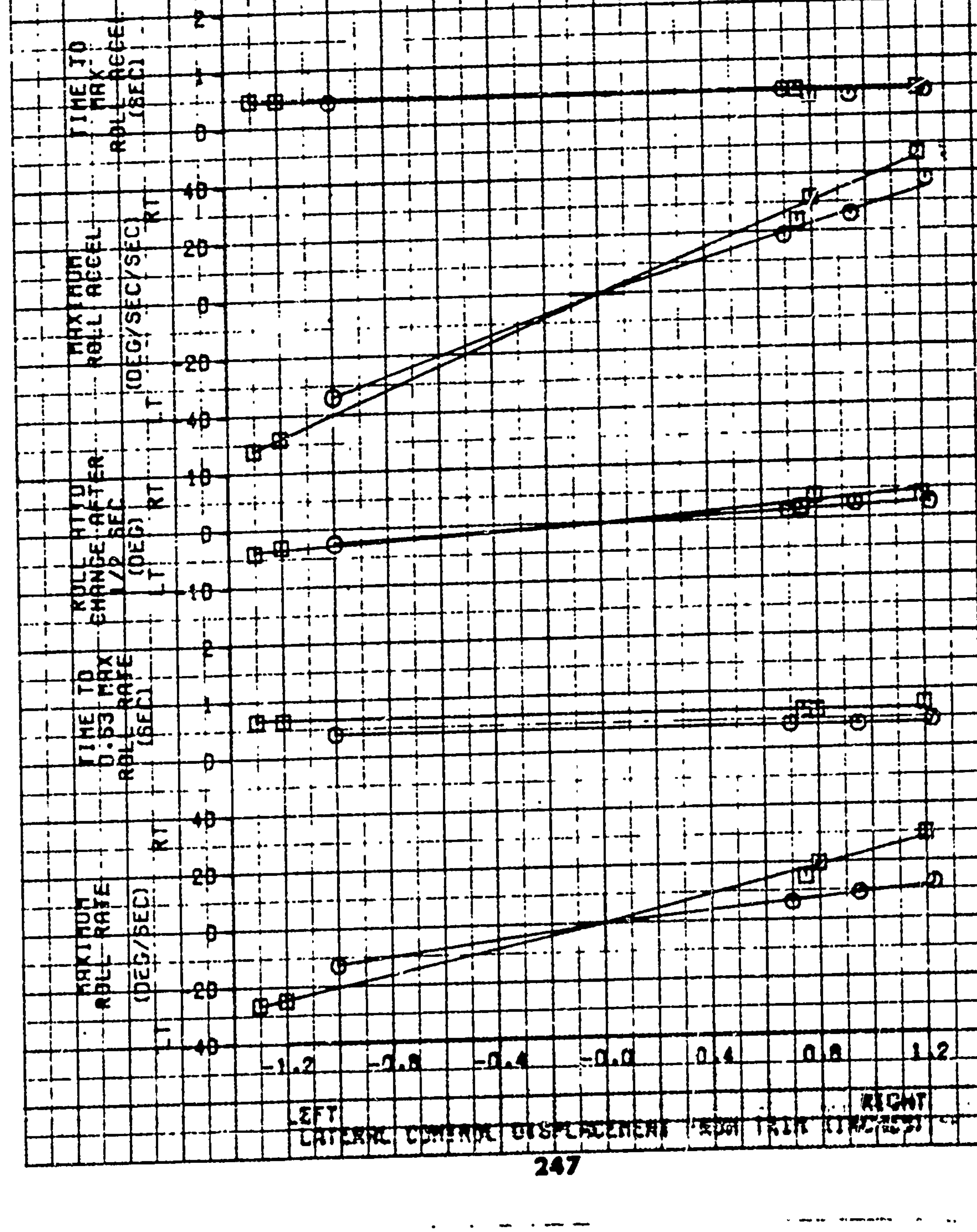
FIGURE 79

LATERAL CONTROLLABILITY

YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CAL AIRSPEED (KT)	TRIM FLIGHT CONDITION
16660	359.5(AFT)	.4 (RT)	6220	27.5	257	116	LEVEL

NOTE: CIRCLES DENOTE AFCS ON
SQUARES DENOTE AFCS OFF



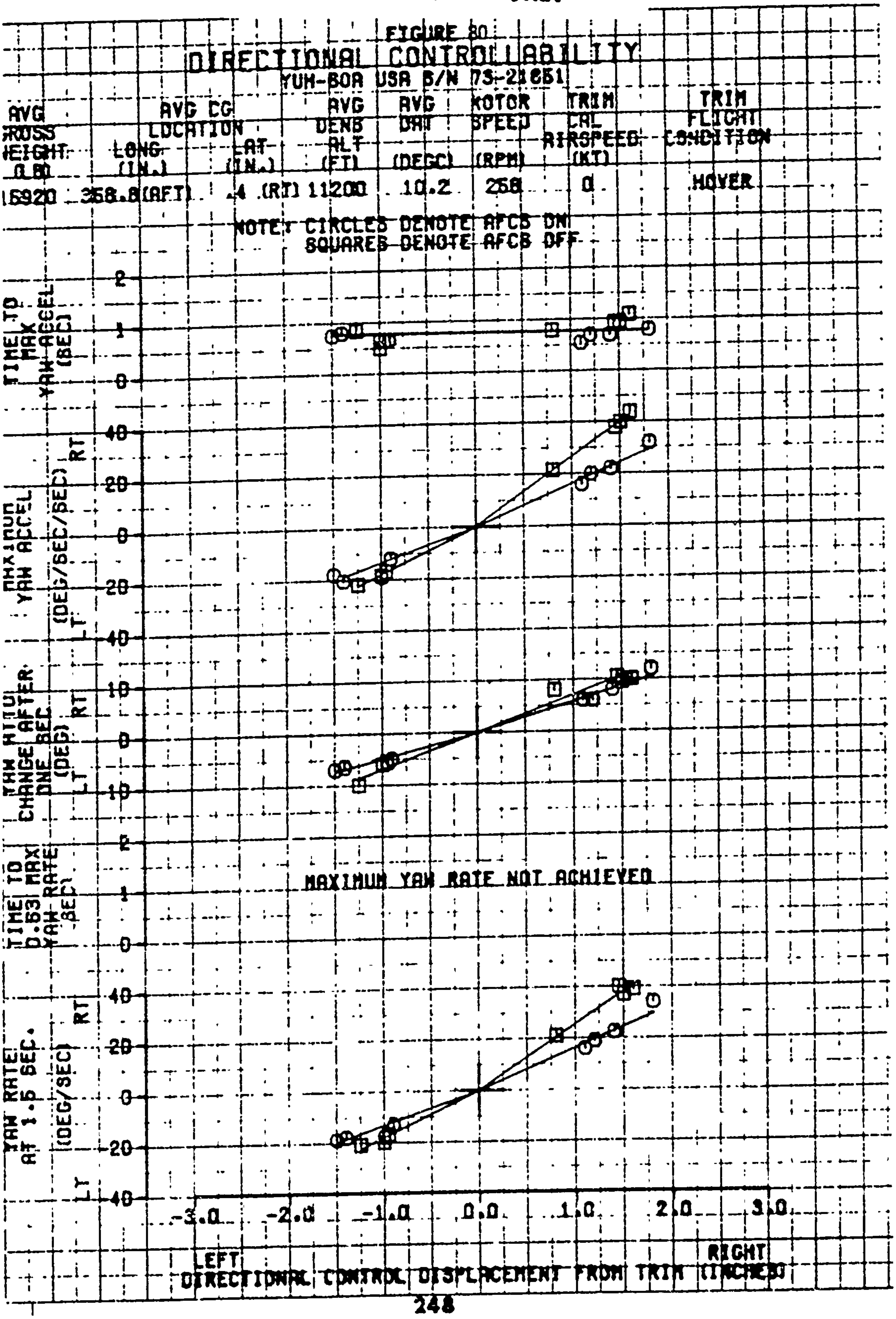


FIGURE 80
DIRECTIONAL CONTROLLABILITY
YUH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
15920
358.8(AFT)
.4 (RT)
11200
10.2
258
0
HOVER
NOTE: CIRCLES DENOTE AFCS ON
SQUARES DENOTE AFCS OFF
TIME TO MAX YAW ACCEL (SEC)
MAXIMUM YAW ACCEL (DEG/SEC/SEC)
YAW ATTITUDE CHANGE AFTER ONE SEC (DEG)
TIME TO 0.63 MAX YAW RATE (SEC)
MAXIMUM YAW RATE NOT ACHIEVED
YAW RATE AT 1.5 SEC. (DEG/SEC)
RT
LT
LEFT
RIGHT
DIRECTIONAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

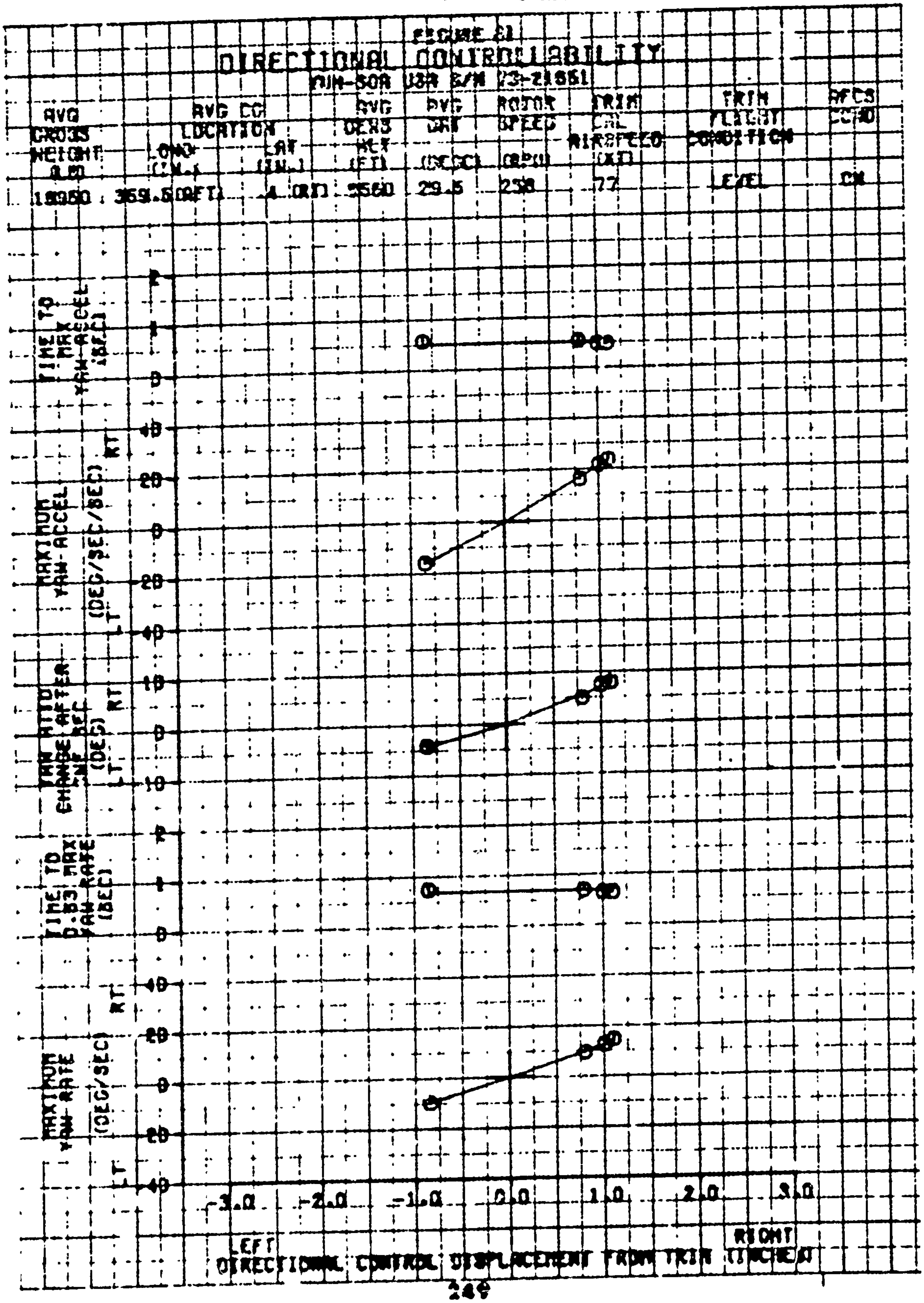


FIGURE 21
DIRECTIONAL CONTROLLABILITY
MH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
AFCS CONFIG
18080
359.5(AFT)
.4 (RT)
5560
29.5
258
77
LEVEL
ON
TIME TO MAX YAW ACCEL (SEC)
MAXIMUM YAW ACCEL (DEG/SEC/SEC)
LT
RT
YAW ATTD CHANGE AFTER 1 SEC (DEG)
TIME TO 0.63 MAX YAW RATE (SEC)
MAXIMUM YAW RATE (DEG/SEC)
-3.0
-2.0
-1.0
0.0
1.0
2.0
3.0
LEFT
RIGHT
DIRECTIONAL CONTROL DISPLACEMENT FROM TRIM (INCHES)

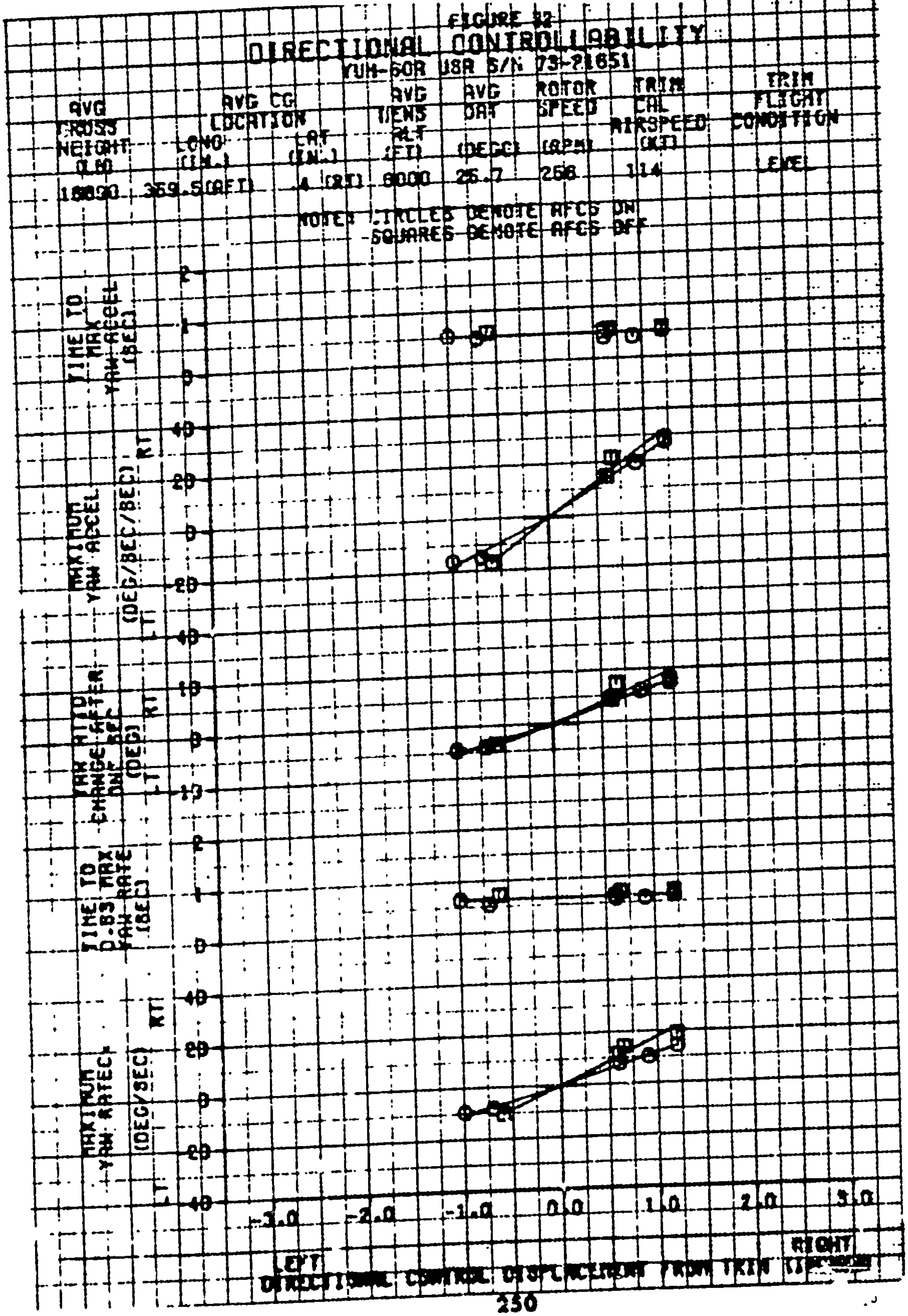


FIGURE 32
DIRECTIONAL CONTROLLABILITY
YUH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
TRIM CAL AIRSPEED (KT)
TRIM FLIGHT CONDITION
369.5(AFT)
.4 (RT)
6000
25.7
258
114
LEVEL
NOTE: CIRCLES DENOTE AFCS ON
SQUARES DENOTE AFCS OFF
TIME TO MAX YAW ACCEL (SEC)
MAXIMUM YAW ACCEL (DEG/SEC/SEC)
RT
LT
YAW ATTITUDE CHANGE AFTER ONE SEC (DEG)
TIME TO 0.63 MAX YAW RATE (SEC)
MAXIMUM YAW RATE (DEG/SEC)
-3.0
-2.0
-1.0
0.0
1.0
2.0
3.0
LEFT
RIGHT
DIRECTIONAL CONTROL DISPLACEMENT FROM TRIM (IN.)

FIGURE 83
SUMMARY CONTROL RESPONSE
YUH-60A USA S/N 73-21651

NOTE: 1) DATA DERIVED FROM FIGURE 75 THROUGH 82
2) RATES DENOTE LEFT OR FORWARD INPUT
3) SHADED SYMBOLS DENOTE AFCS OFF

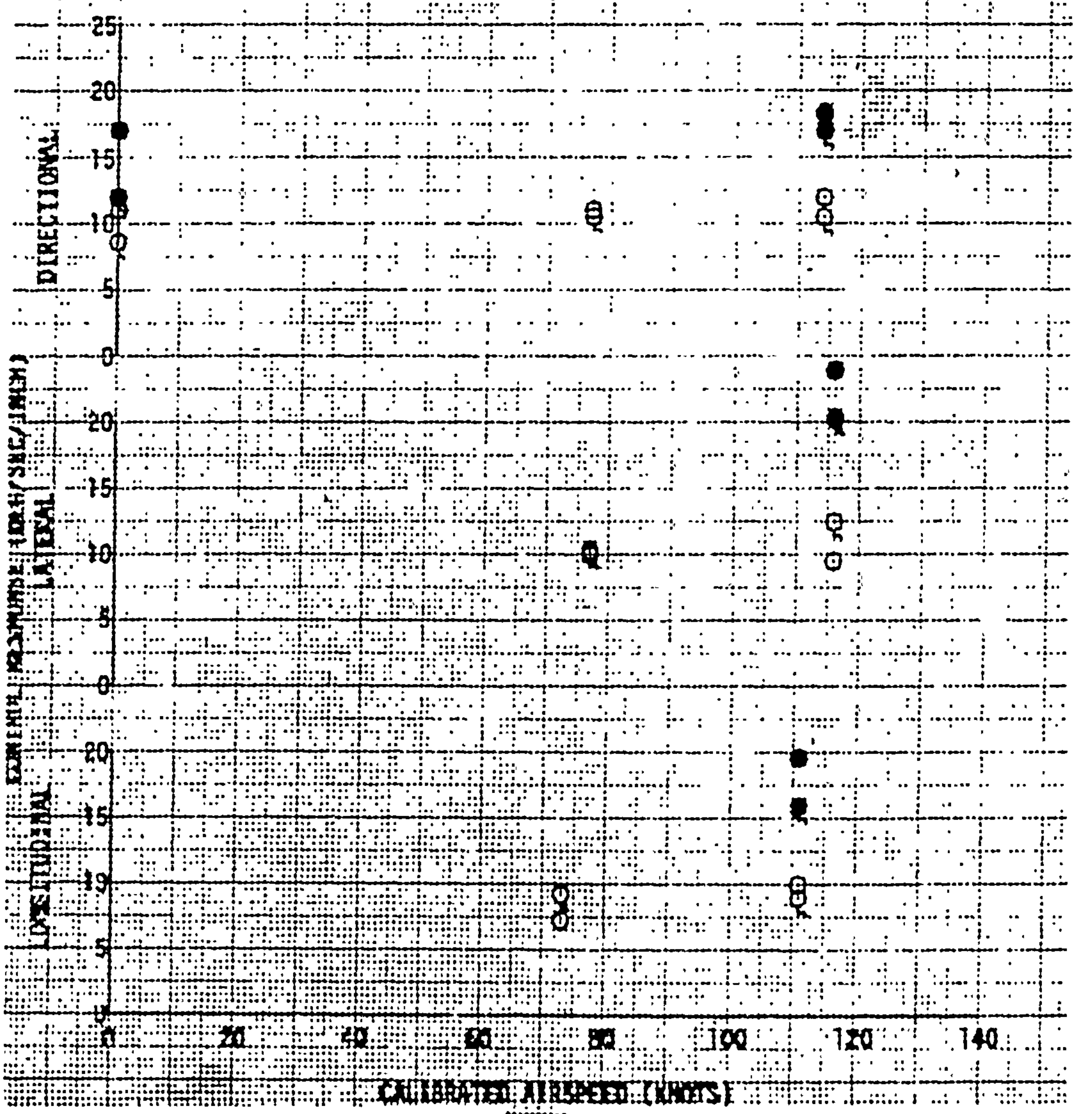


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FIGURE [illegible]
SUMMARY CONTROL SENSITIVITY
YUH-60A USA S/N 73-21651

NOTE 1) DATA DERIVED FROM FIGURE 76 THROUGH 82
2) TAILS DENOTE LEFT OR FORWARD INPUT
3) SHADED SYMBOLS DENOTE AFCS OFF

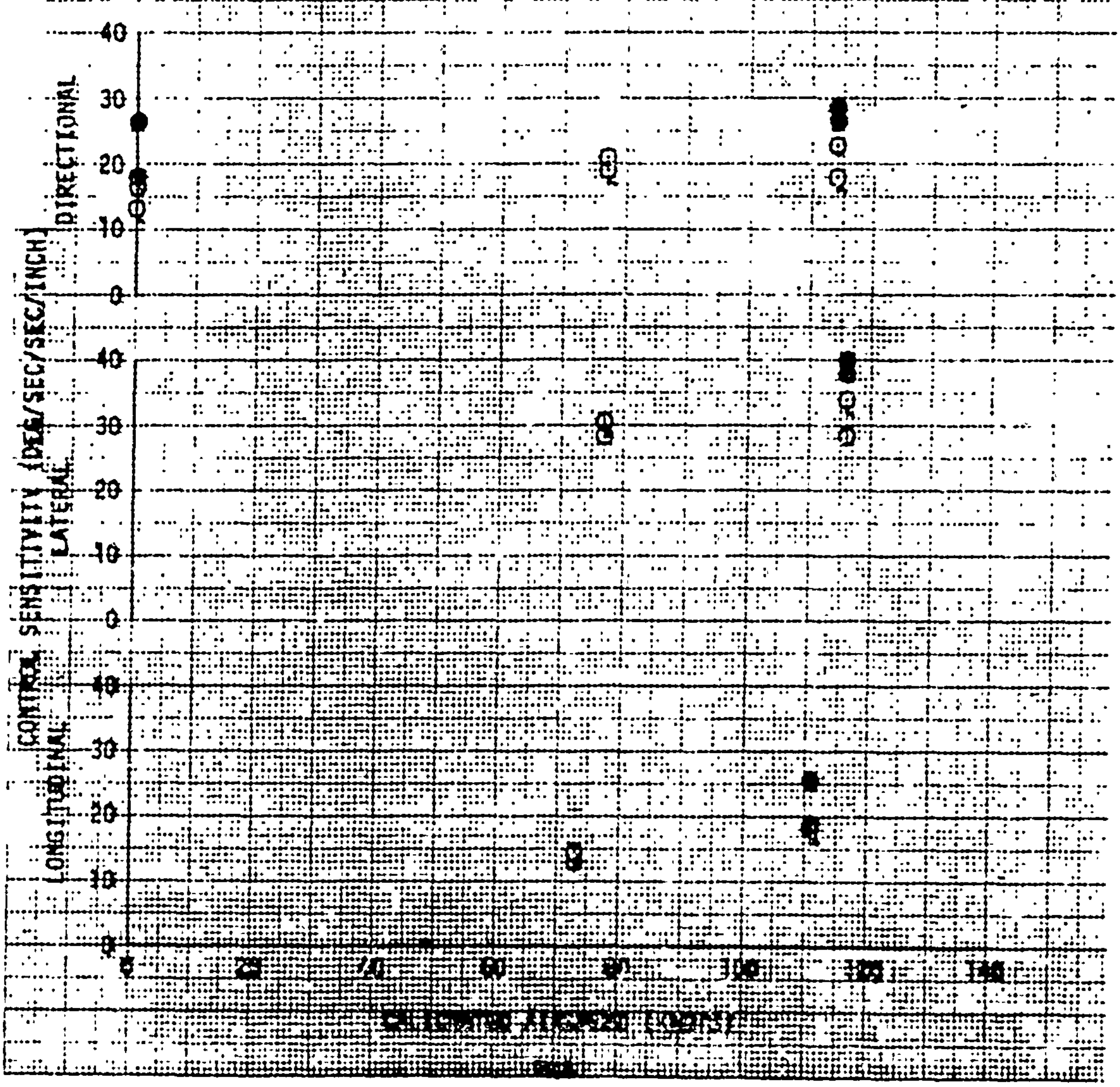

FIGURE 85
AFT LONGITUDINAL STEP
YUH-60A USA S/N 73 21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16650	359.8 (AFT)	0.4 (RT)	6550	29.0	[illegible]	115	LEVEL	ON

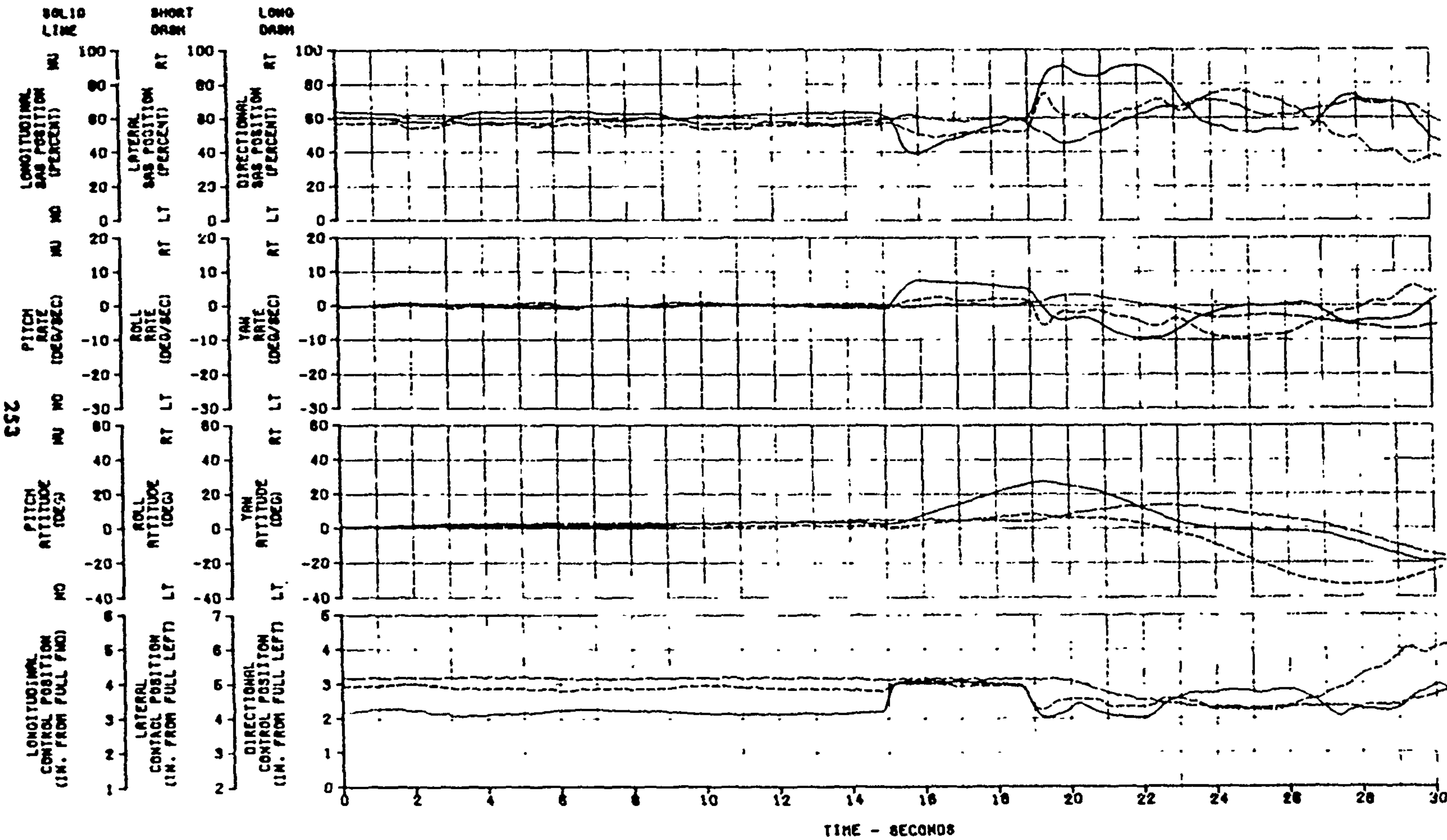

FIGURE [illegible]

RIGHT LATERAL STEP

YUH-60A USA S/N 73-21661

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16490	359.5 (AFT)	0.4 (RT)	6440	30.0	258	115	LEVEL	ON

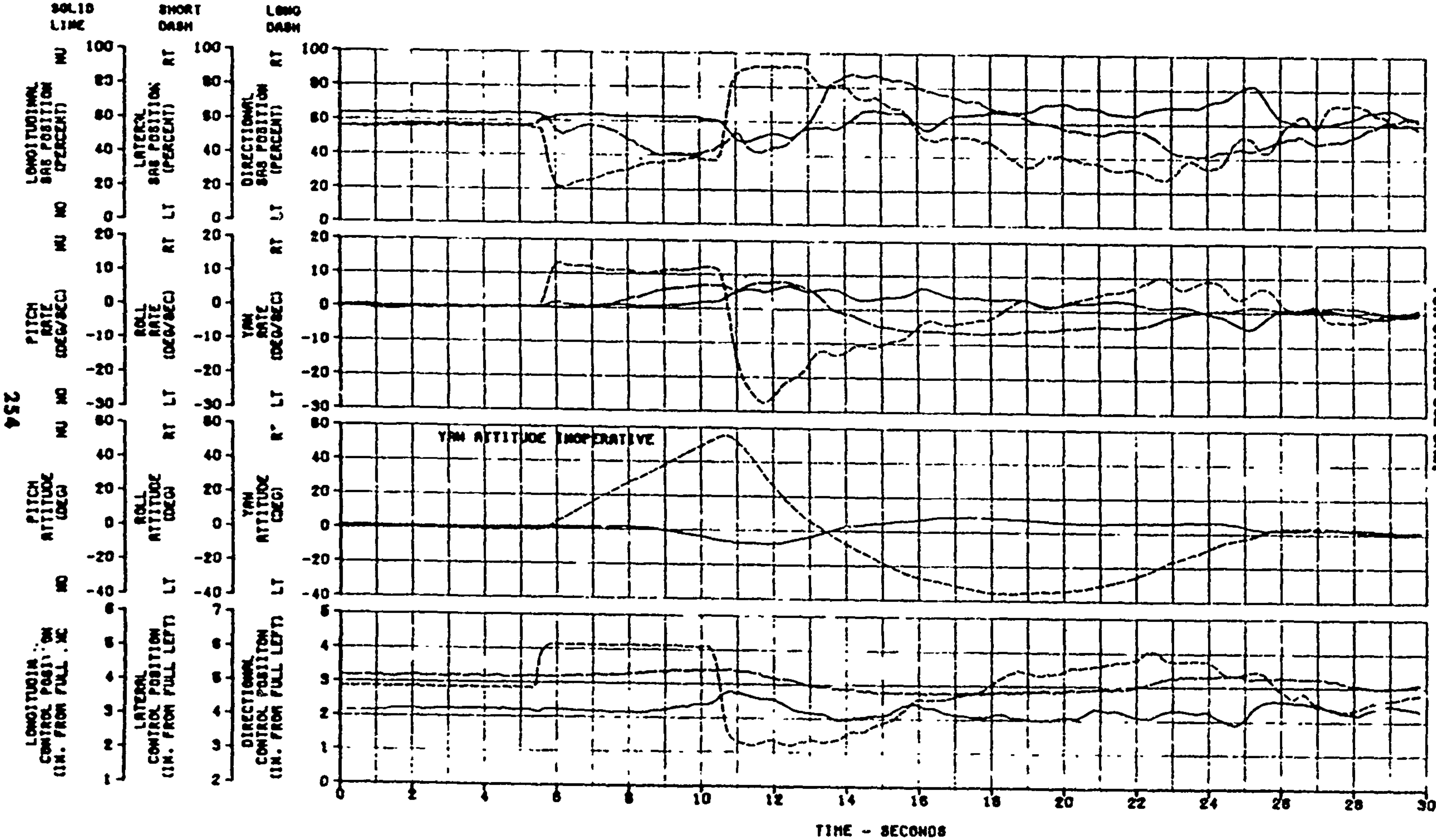

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FIGURE 87
RIGHT DIRECTIONAL STEP
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	(KT)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
17160	359.5 (AFT)			6080	26.5	257	114	LEVEL	ON

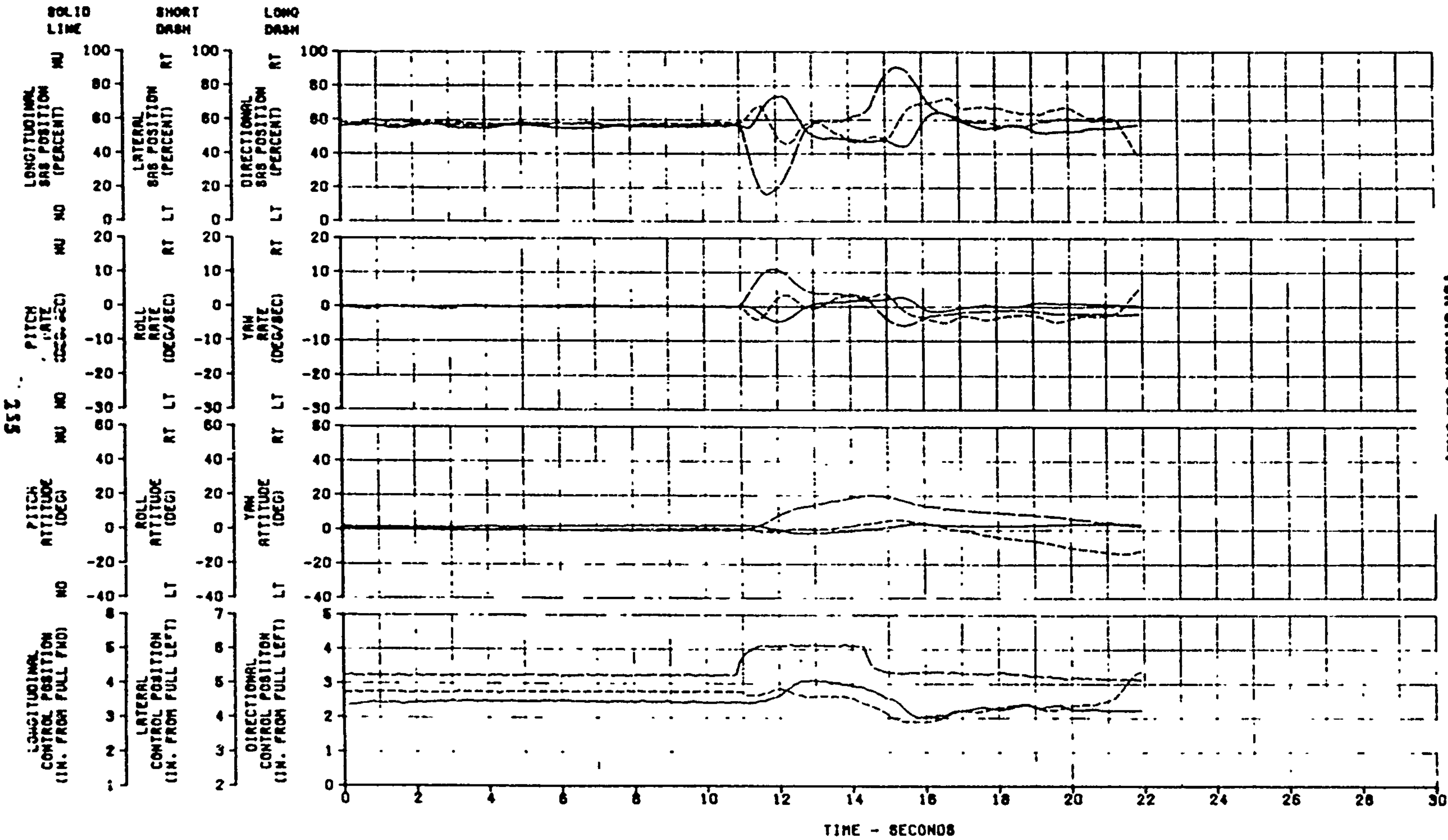

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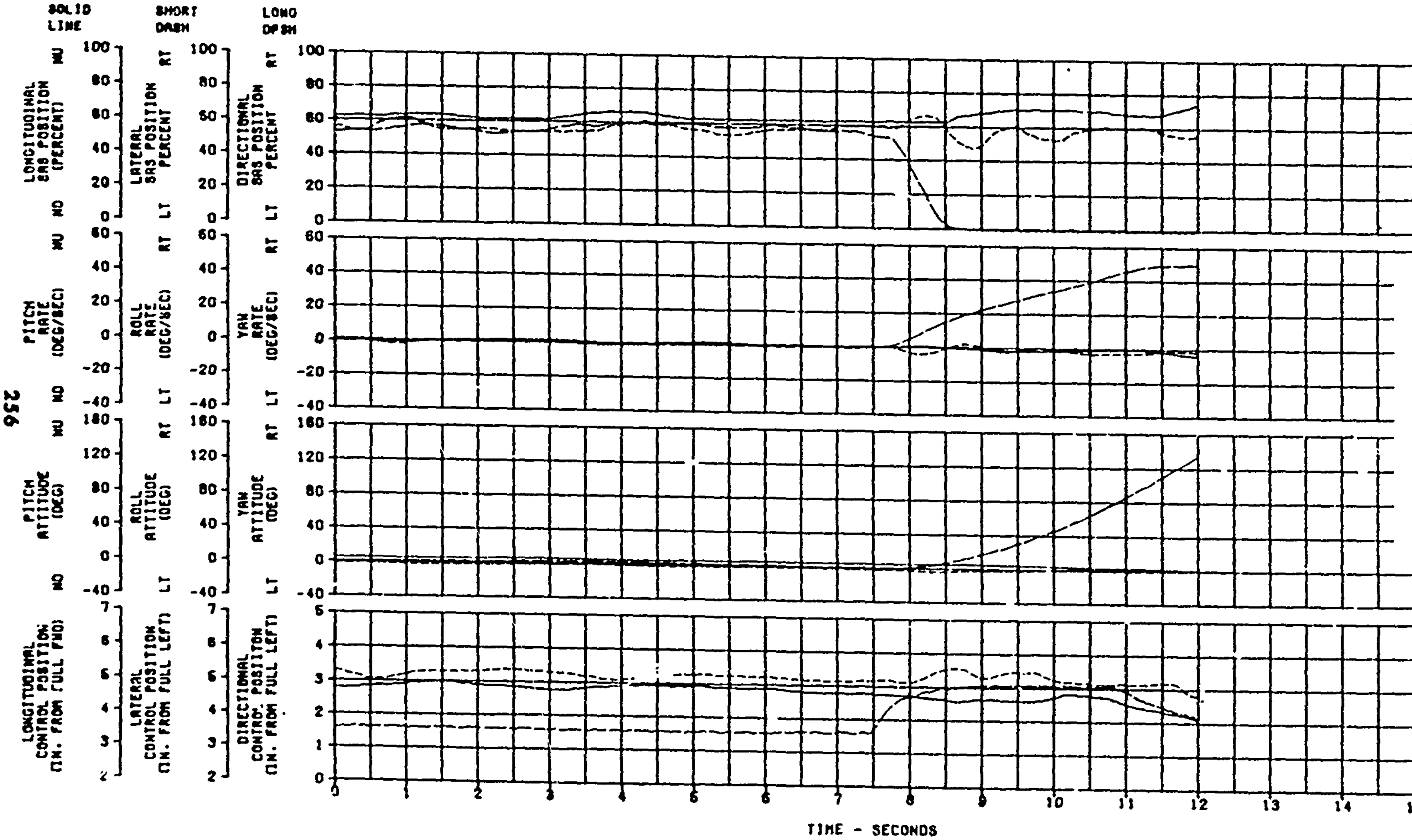
FIGURE 88

RIGHT DIRECTIONAL STEP

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION	AFCS CONDITION
16060	359.3 (AFT)	0.4 (RT)	1110	10.0	260	0	HOVER	ON



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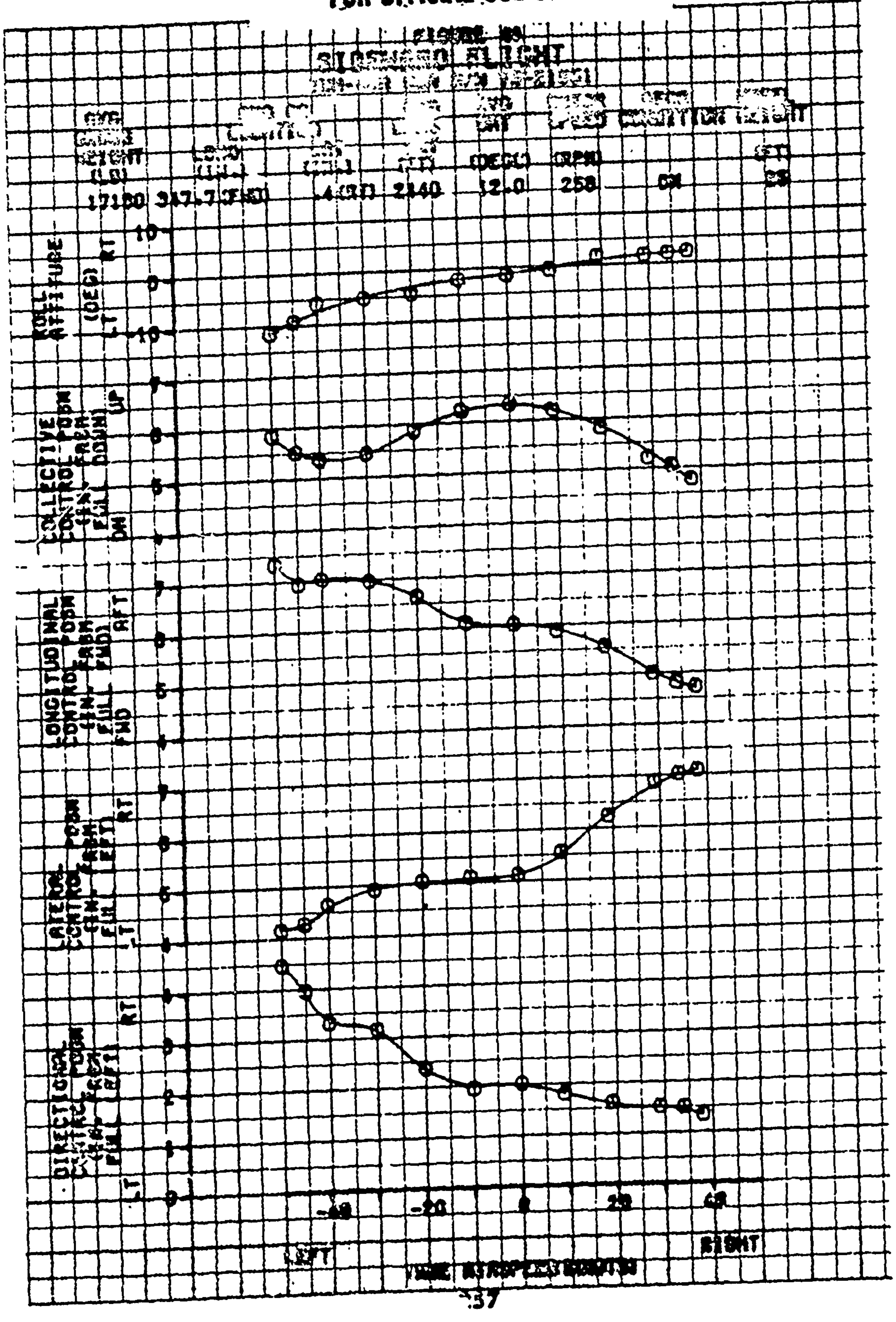


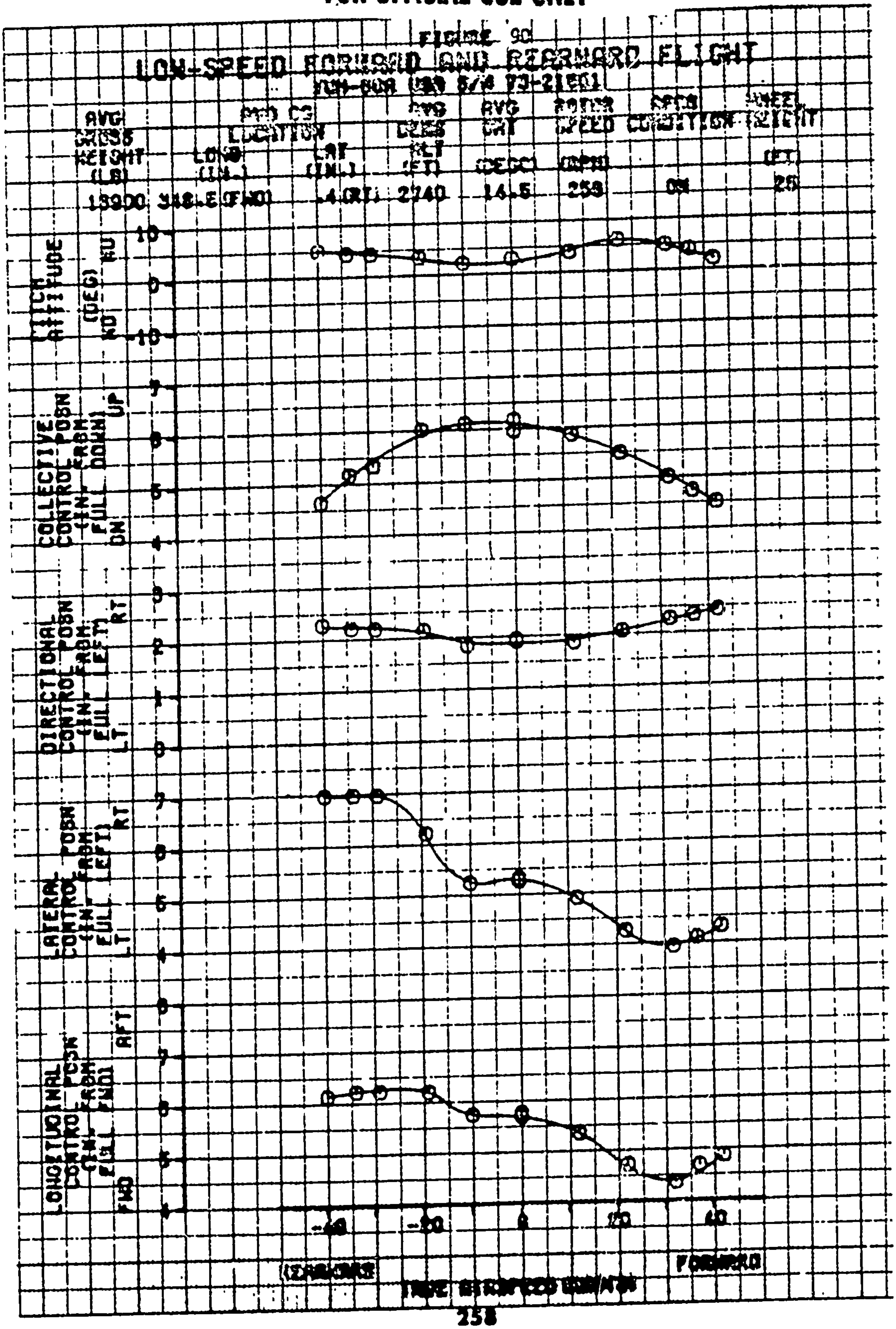


FIGURE 90
LOW-SPEED FORWARD AND REARWARD FLIGHT
YUH-60A USA S/N 73-21651
AVG GROSS WEIGHT (LB) 13900
AVG CG LOCATION LONG (IN.) 348.6 (FWD)
LAT (IN.) .4 (RT)
AVG DENS ALT (FT) 2740
AVG OAT (DEG C) 14.5
ROTOR SPEED (RPM) 258
FLIGHT CONDITION OGE
WHEEL HEIGHT (FT) 25
PITCH ATTITUDE (DEG)
COLLECTIVE CONTROL POSN (IN. FROM FULL DOWN)
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
LATERAL CONTROL POSN (IN. FROM FULL LEFT)
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD)
REARWARD
FORWARD
TRUE AIRSPEED (KNOTS)

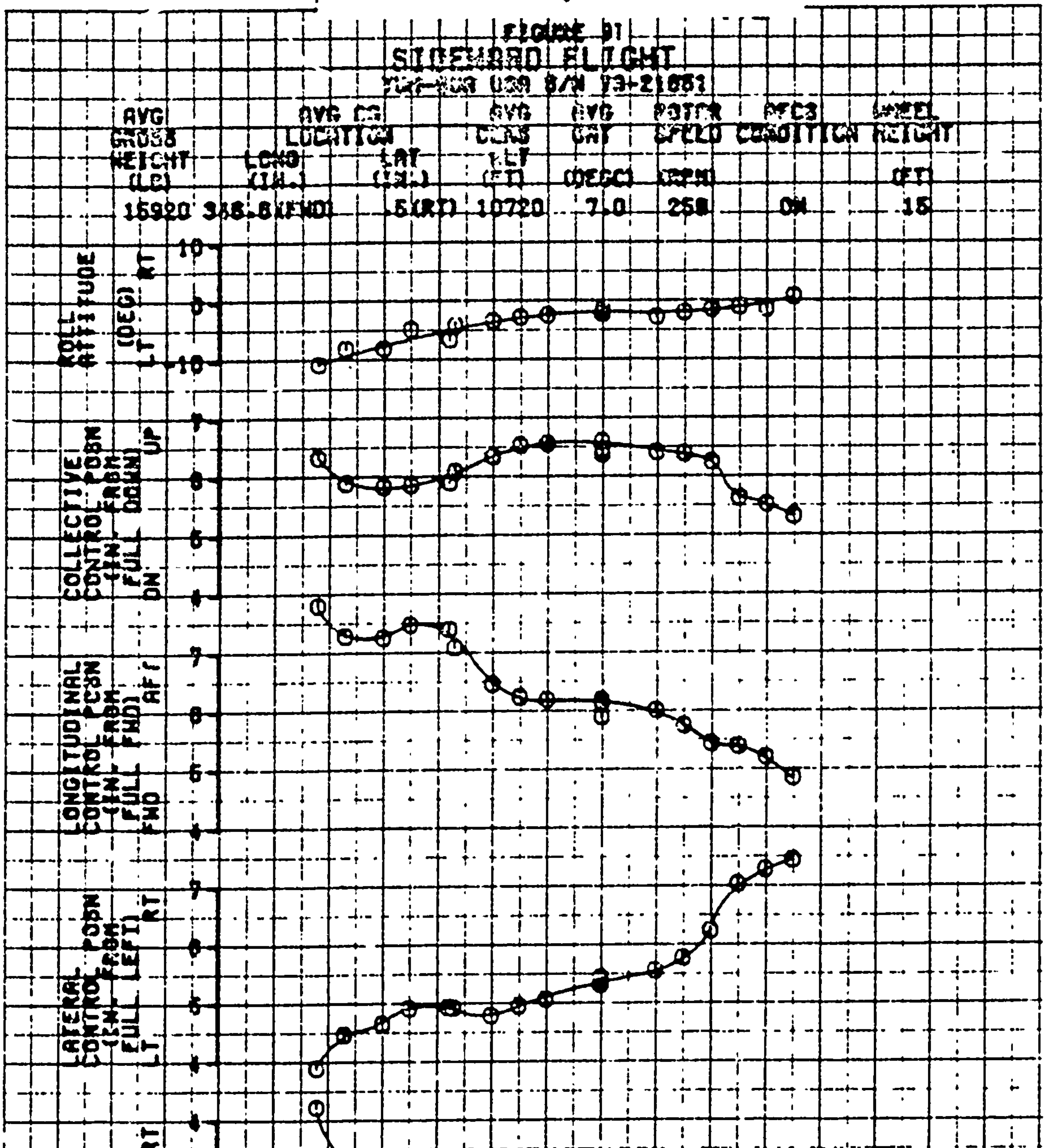
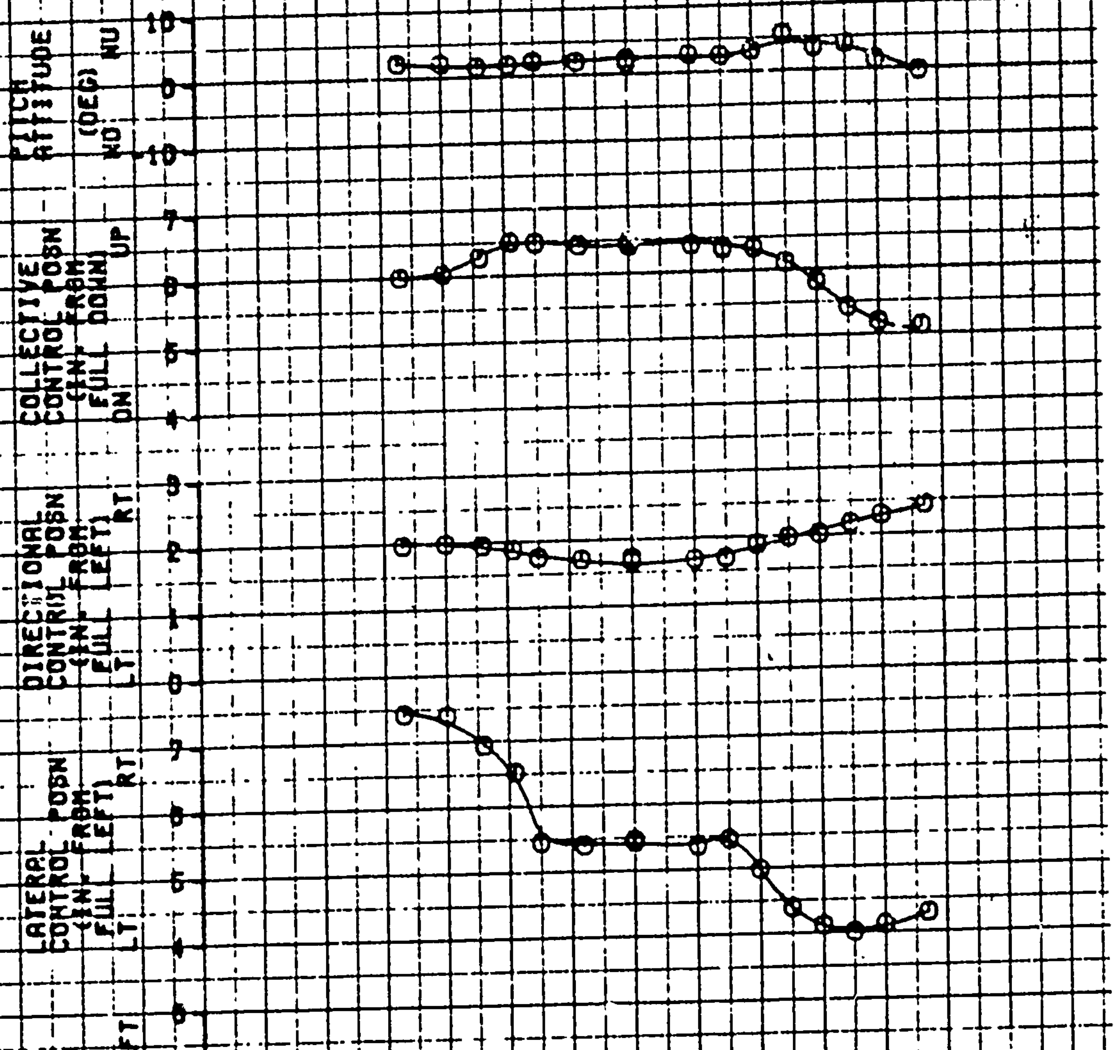


FIGURE 91
SIDEWARD FLIGHT
AVG GROSS WEIGHT (LB)
AVG CG LOCATION
LONG (IN.)
LAT (IN.)
AVG DENS ALT (FT)
AVG OAT (DEG C)
ROTOR SPEED (RPM)
SCAS CONDITION
WHEEL HEIGHT (FT)
15920
348.8(FWD)
.5(RT)
10720
7.0
258
ON
15
ROLL ATTITUDE (DEG)
LT
RT
COLLECTIVE CONTROL POSN (IN. FROM FULL DOWN)
DN
UP
LONGITUDINAL CONTROL POSN (IN. FROM FULL FWD)
FWD
AFT
LATERAL CONTROL POSN (IN. FROM FULL LEFT)
LT
RT
DIRECTIONAL CONTROL POSN (IN. FROM FULL LEFT)
LT
RT
-40
-20
0
20
40
LEFT
RIGHT
TRUE AIRSPEED (KNOTS)

FIGURE 32
LOW-SPEED FORWARD AND REARWARD FLIGHT
YUH-60A USA S/N 73-21651

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	AFCS CONDITION	WHEEL HEIGHT (FT)
15830	346.6 (FWD)	.5 (RT)	10780	7.0	259	ON	15



[redacted]

FIGURE [illegible]
CONTROL POSITIONS IN [illegible] WIND AZIMUTH
[illegible]

SYMBOL	TRUE AIRSPEED (KT)	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	C_T	ROTOR SPEED (RPM)	AFCS CONDITION	WHEEL HEIGHT (FT)
○	10	17070	343.8(FWD)	0.4	2570	13.0	.008475	257	ON	25
△	10	15820	346.4(FWD)	0.5	11033	9.0	.007810	258	ON	15
□	10	[illegible]	300.2(AFT)	0.4	[illegible]	17.0	.007086	257	ON	50

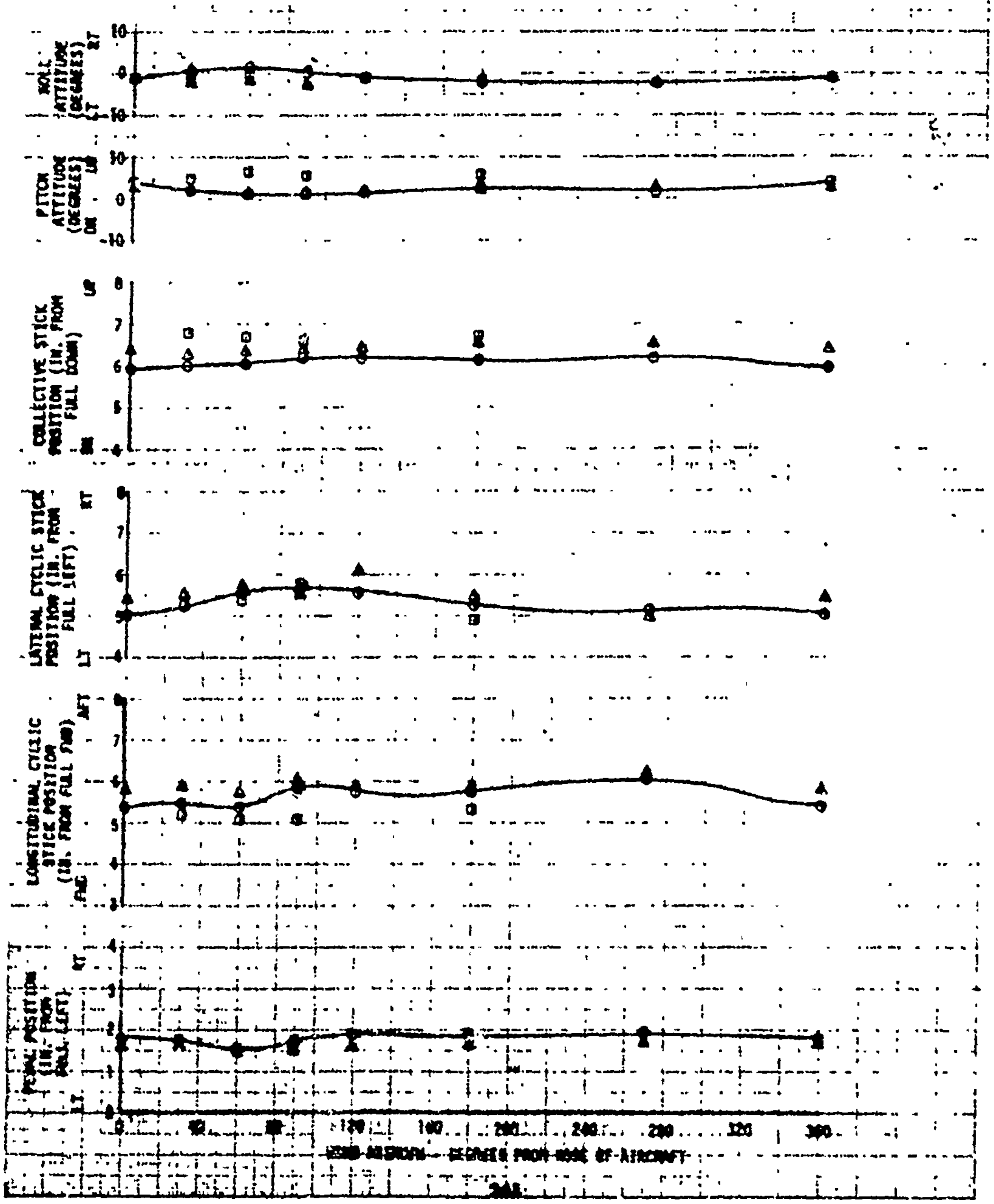


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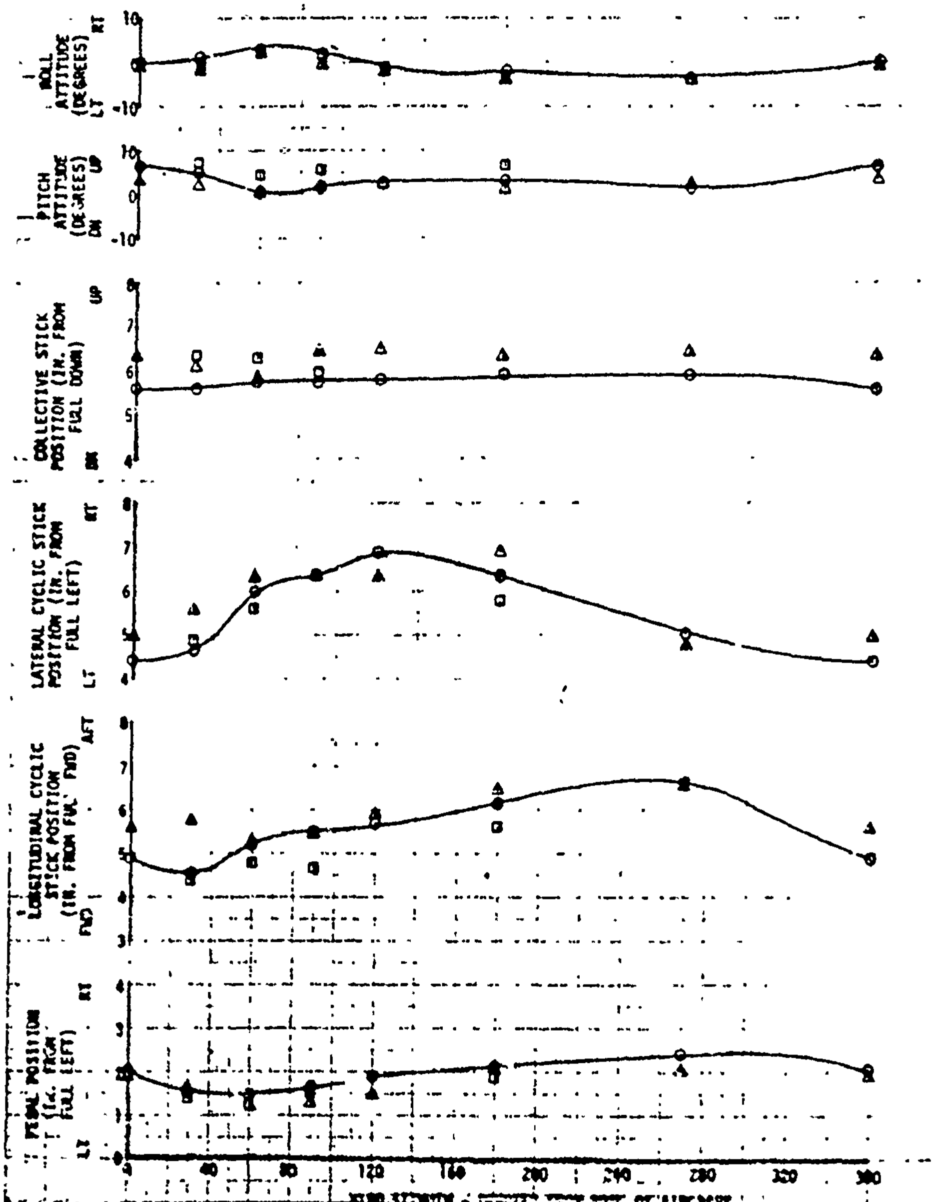
FIGURE [illegible]

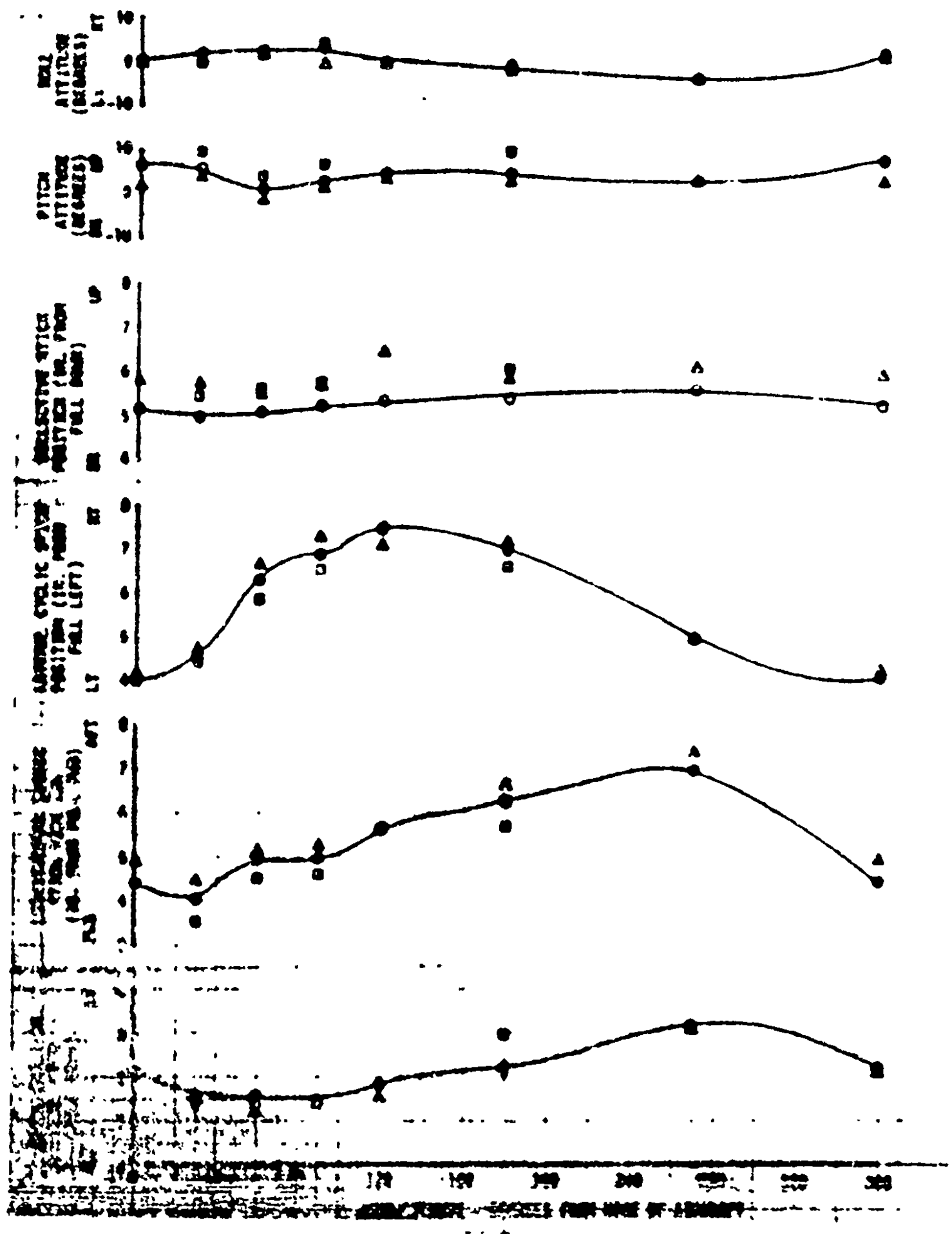
CONTROL POSITIONS VERSUS RELATIVE WIND AZIMUTH

[illegible]

SYMBOL	TRUE AIRSPEED (KT)	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	C_T	ROTOR SPEED (RPM)	AFCS CONDITION	WHEEL HEIGHT (FT)
○	20	17000	342.0(FWD)	0.4	2550	13.0	.006485	258	ON	25
△	20	15820	346.4(FWD)	0.5	[illegible]	10.8	.00[illegible]	259	ON	15
□	20	18080	358.2(AFT)	0.4	3020	17.2	.007043	257	ON	50



SYMBOL	TRUE AIRSPEED (KT)	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	C_T	ROTOR SPEED (RPM)	AFCS CONDITION	MODEL HEIGHT (FT)
●	30	17300	395.0(FWD)	0.0	2900	13.0	.006490	258	ON	25
▲	30	16460	395.4(FWD)	0.5	11700	10.0	.007740	259	ON	15
■	30	16630	388.2(AFT)	0.4	3020	17.0	.007005	257	ON	50



[redacted]

SYMBOL	TRUE AIRSPEED (KT)	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	C_T	ROTOR SPEED (RPM)	AFCS CONDITION	WHEEL HEIGHT (FT)
●	40	[illegible]	[illegible](FWD)	[illegible]	[illegible]	13.0	[illegible]	[illegible]	ON	25
▲	40	[illegible]	376.6(FWD)	2.5	[illegible]	5.0	[illegible]	[illegible]	[illegible]	15
■	40	[illegible]	[illegible](AFT)	0.4	[illegible]	17.0	[illegible]	257	[illegible]	80

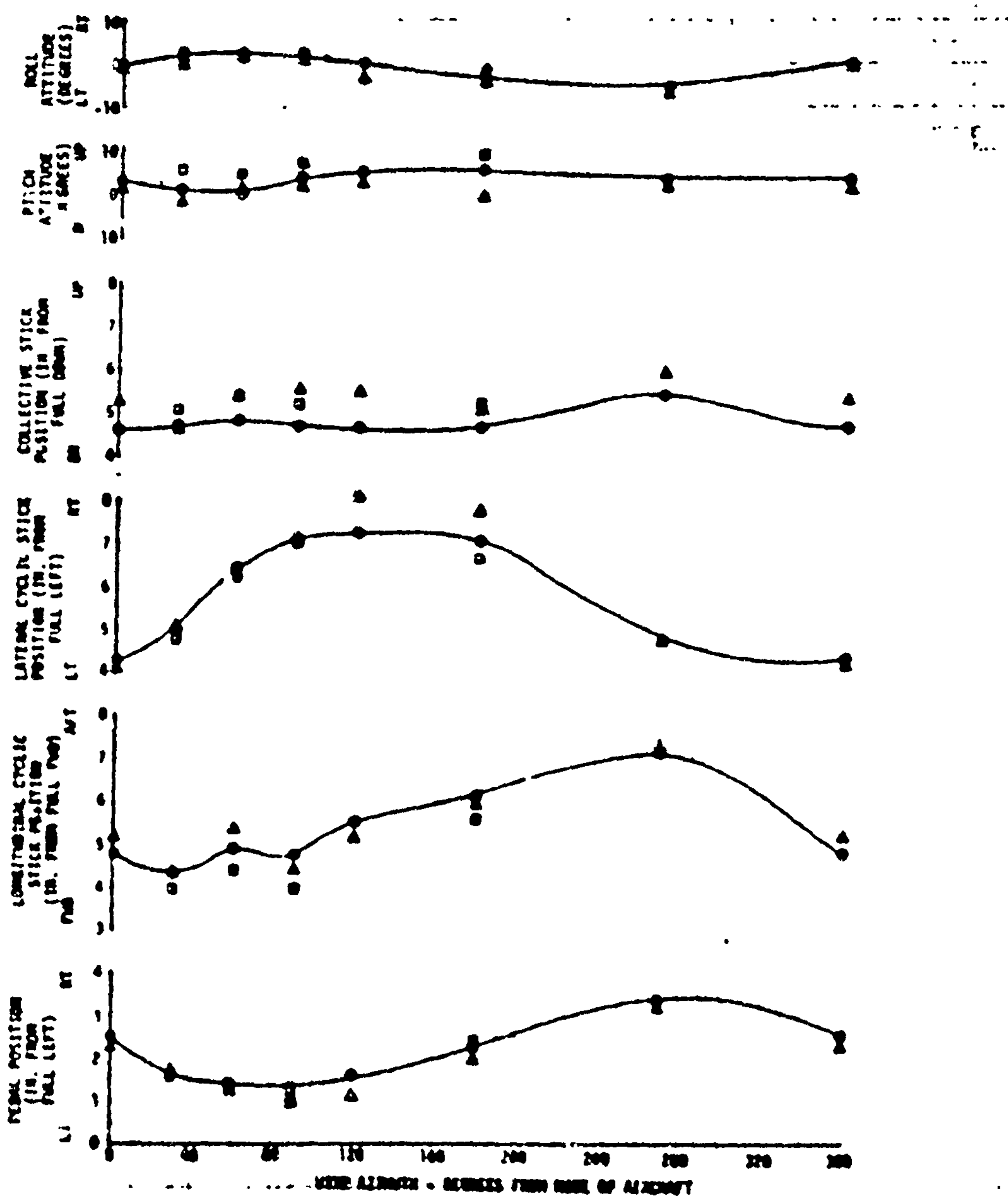
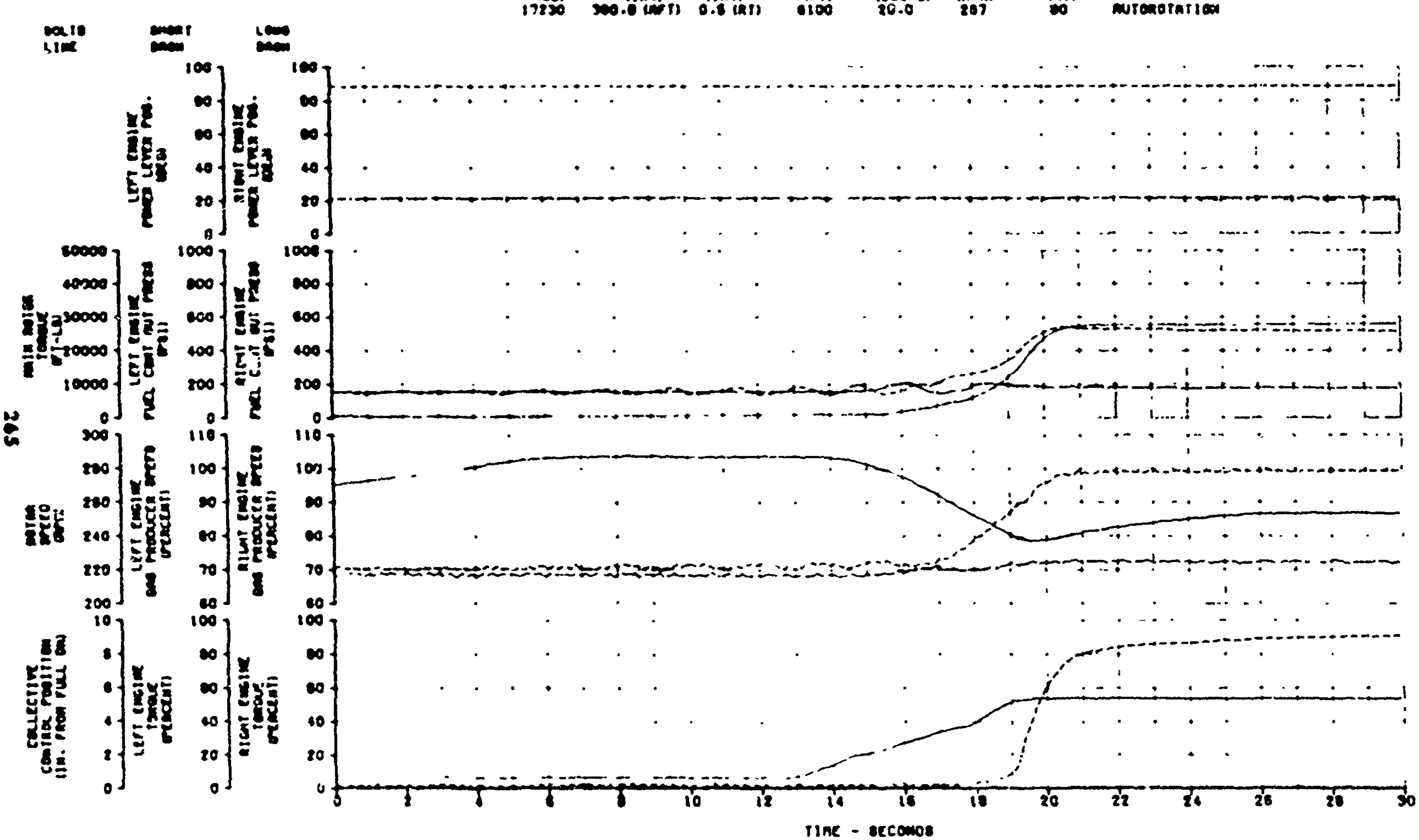


FIGURE 97

ENGINE ACCELERATION PERFORMANCE

YUH-60A USA S/N 73-21651
LEFT ENGINE S/N 207213 / RIGHT ENGINE S/N 207217

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
17230	360.8 (AFT)	0.5 (RT)	6100	20.0	257	90	AUTOROTATION



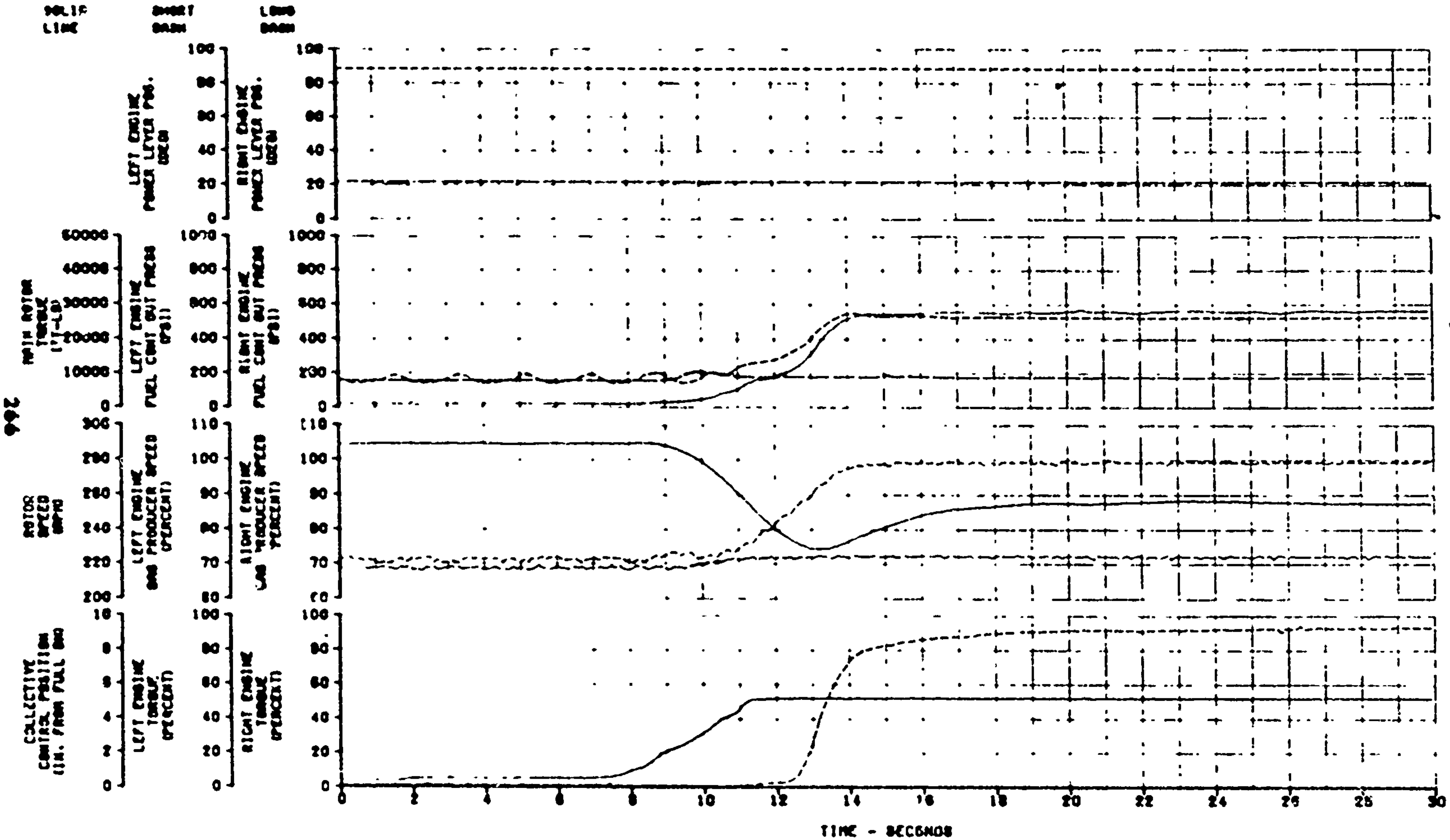
[redacted]

FIGURE 98

ENGINE ACCELERATION PERFORMANCE

YUH-60A USA S/N 73-21651
LEFT ENGINE S/N 207213 / RIGHT ENGINE S/N 207217

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
17150	360.3 (AFT)	0.5 (RT)	5710	20.5	258	80	AUTOROTATION



[redacted]

FIGURE 99
ENGINE DECELERATION PERFORMANCE
YUH-60A USA S/N 73-21651
LEFT ENGINE S/N 207213 / RIGHT ENGINE S/N 207217

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
16840	360.0 (AFT)	0.5 (RT)	6450	18.5	258	140	LEVEL

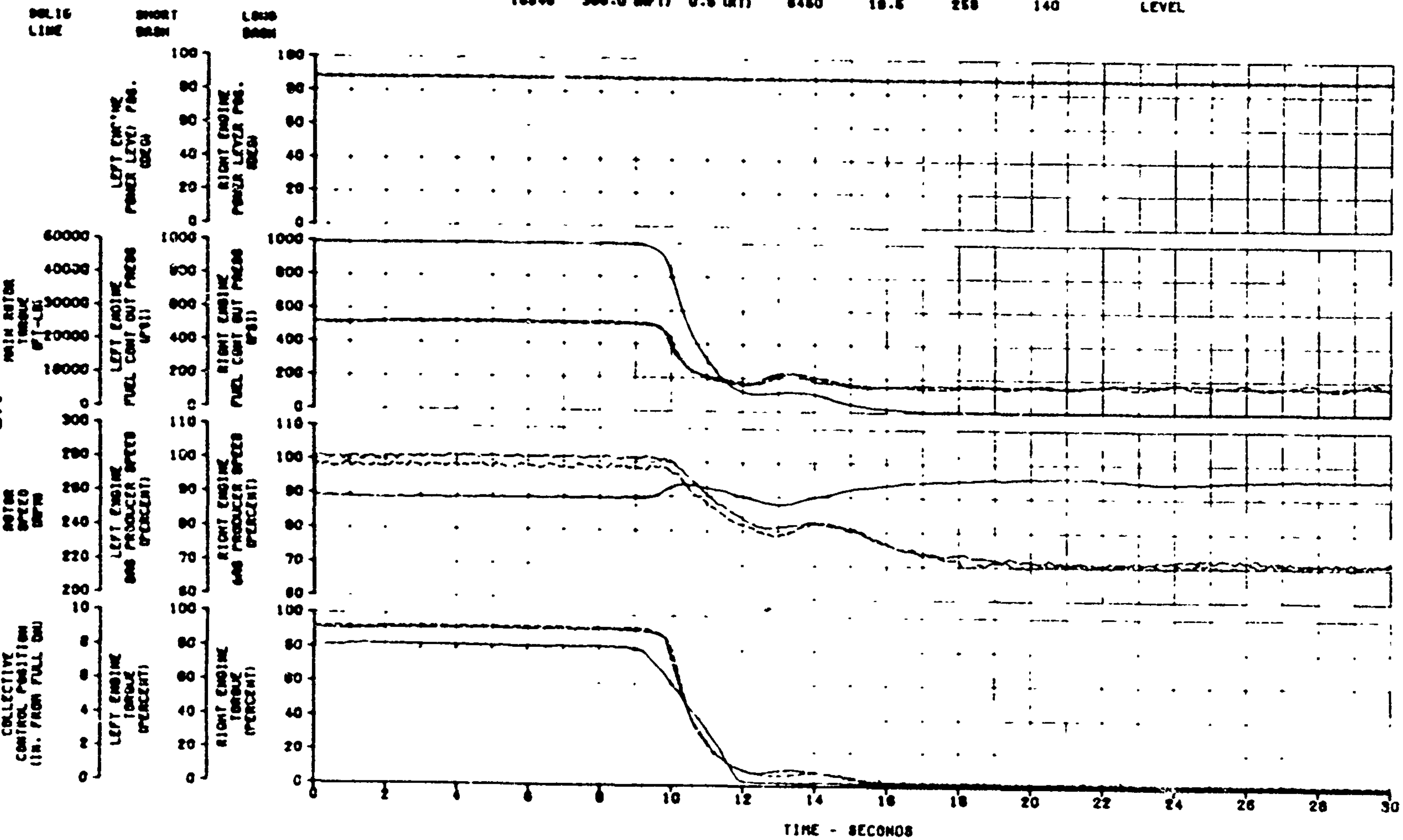


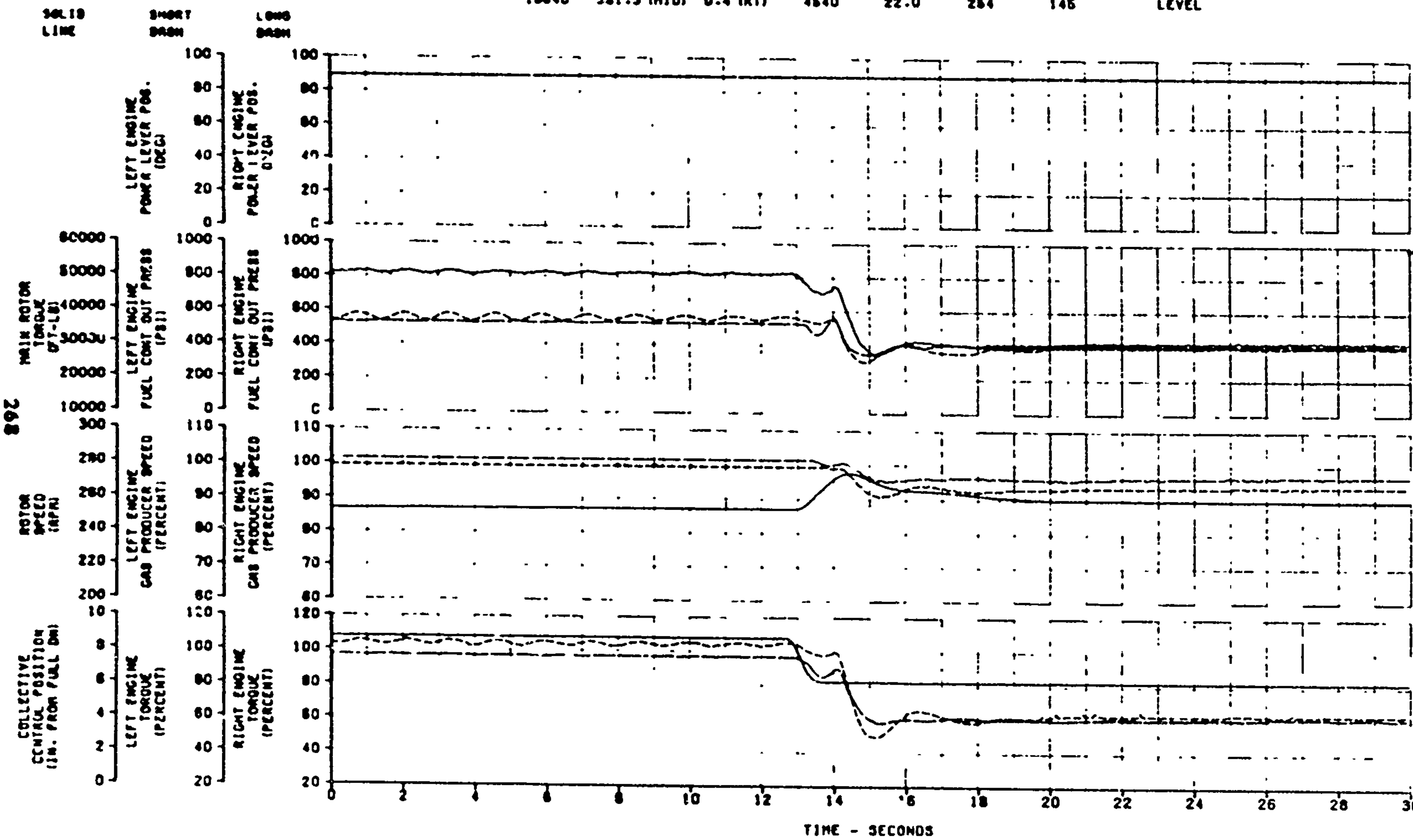
FIGURE 100

GROUP 7 ELECTRICAL CONTROL UNIT CHARACTERISTICS

YUH-60A USA S/N 73-21651

LEFT ENGINE S/N 207213 / RIGHT ENGINE S/N 207217

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
16840	351.3 (MID)	0.4 (RT)	4540	22.0	254	145	LEVEL



[redacted]

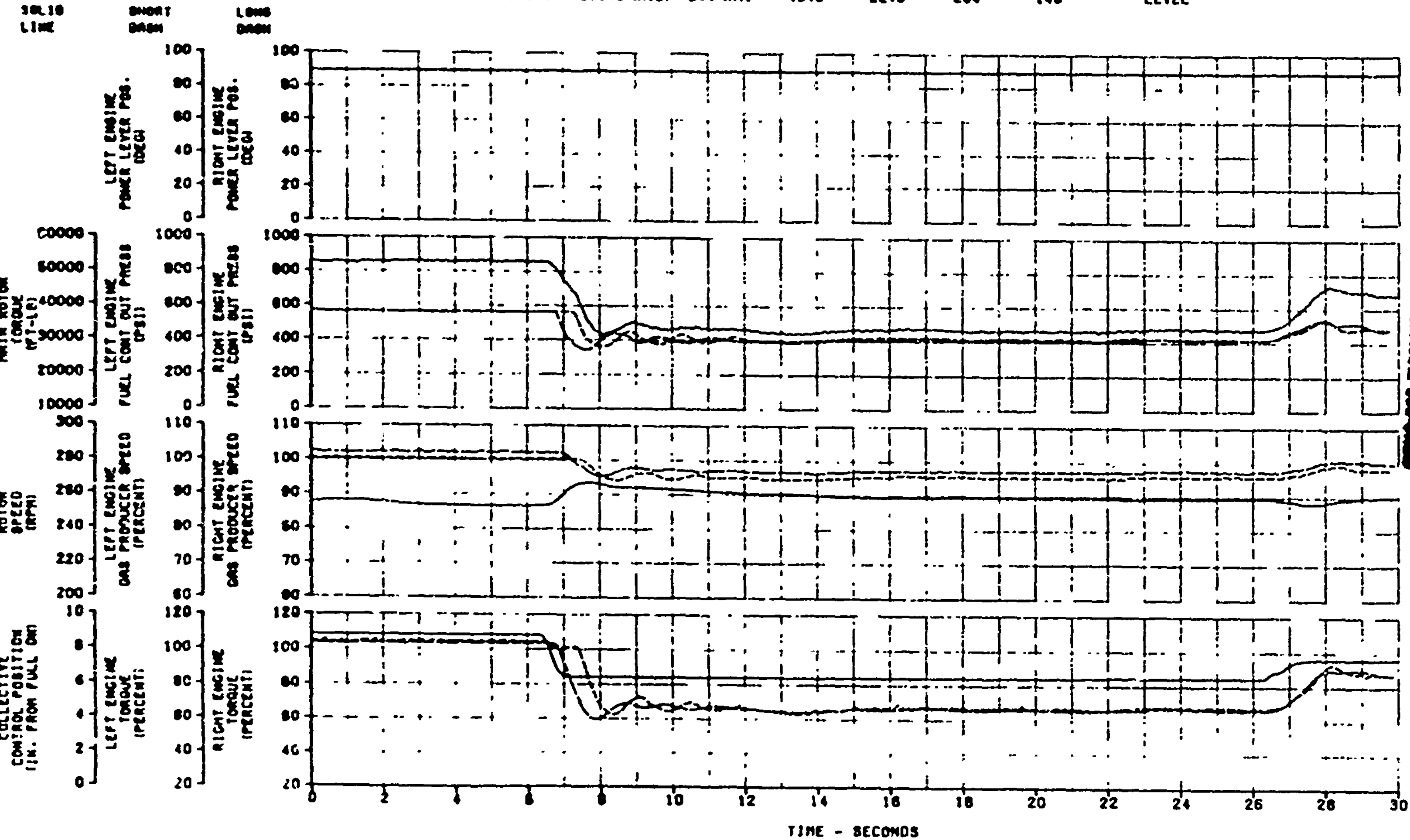
FIGURE 101

GROUP 8 ELECTRICAL CONTROL UNIT CHARACTERISTICS

YUH-60A USA S/N 73-21651

LEFT ENGINE S/N 207213 / RIGHT ENGINE S/N 207217

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
16840	351.3 (MID)	0.4 (RT)	4540	22.0	254	145	LEVEL



[redacted]

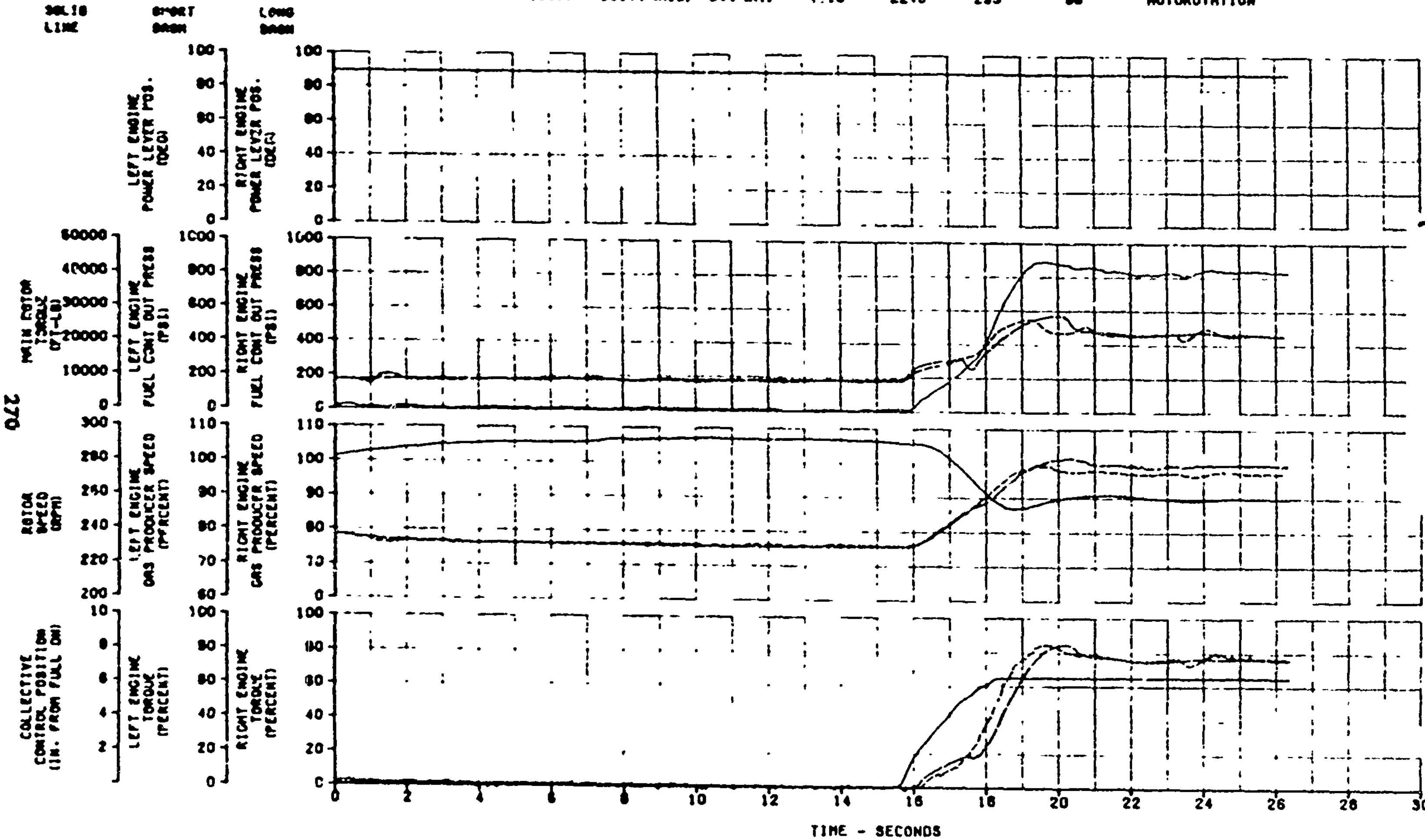
FIGURE 102

GROUP 8 ENGINE ACCELERATION PERFORMANCE

YUH-60A USA S/N 73-21651

LEFT ENGINE S/N 207213 / RIGHT ENGINE S/N 207217

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
16660	361.4 (MID)	0.4 (RT)	4710	22.0	293	96	AUTOROTATION



[redacted]

FIGURE 103
UTTAS MANEUVER (PULL-UP)

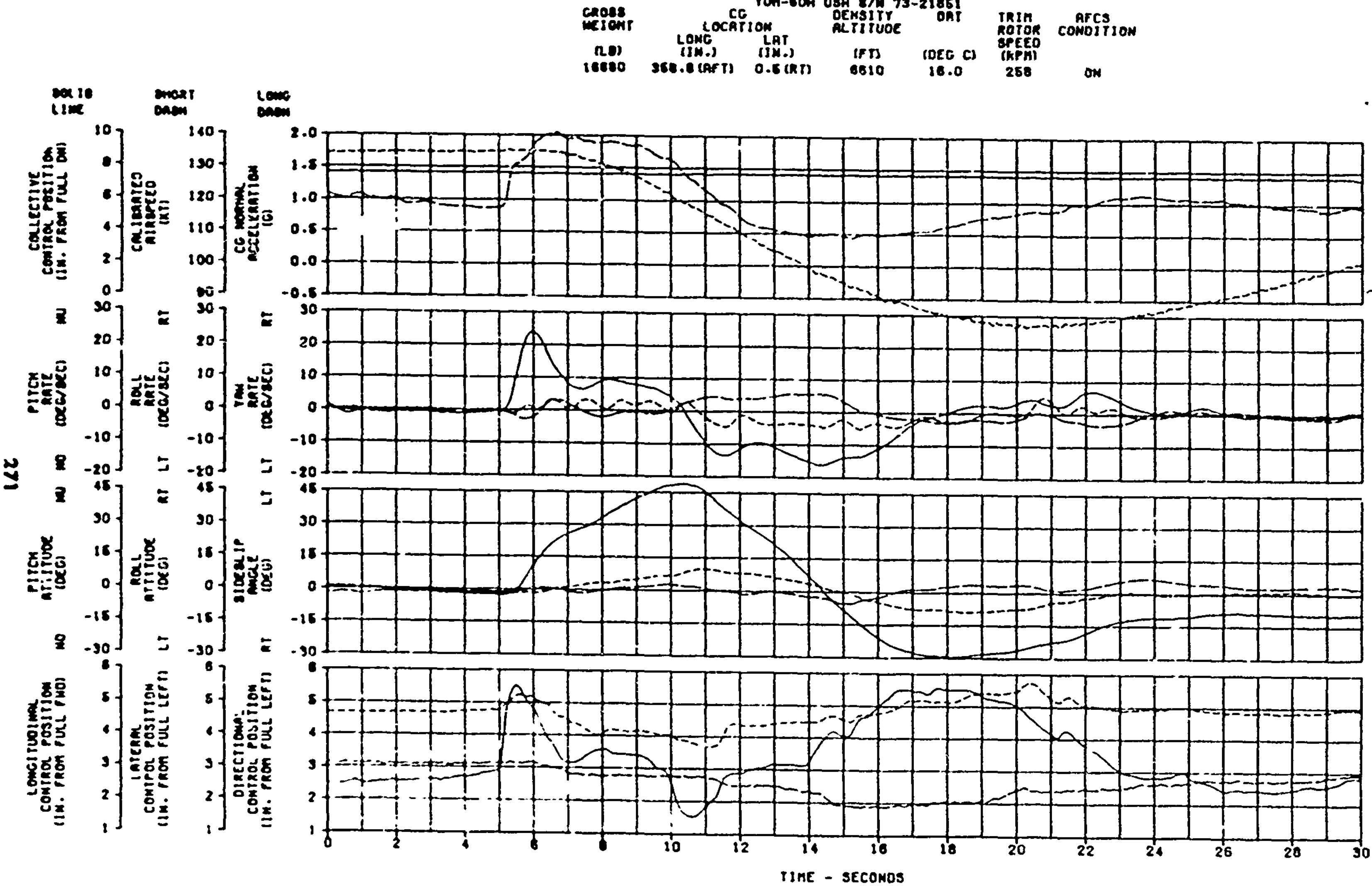


FIGURE 104
UTTAS MANEUVER (PUSHOVER)
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
16550	358.8 (AFT)	.5 (RT)	5900	17.0	258	ON

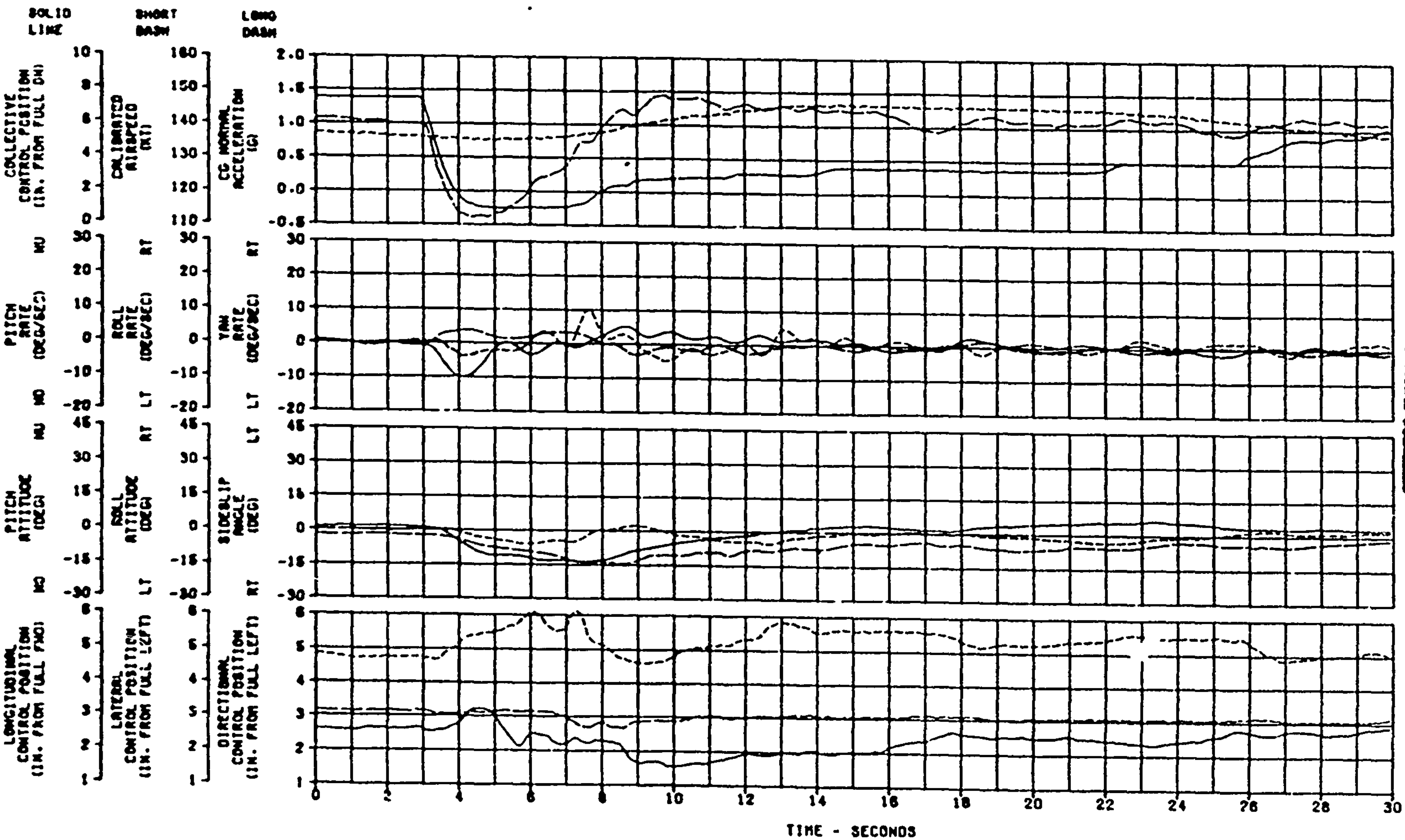


FIGURE 105

UTTAS MANEUVER (PULL-UP AND PUSHOVER)

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
17220	360.1(AFT)	0.4(RT)	7120	28.0	253	ON

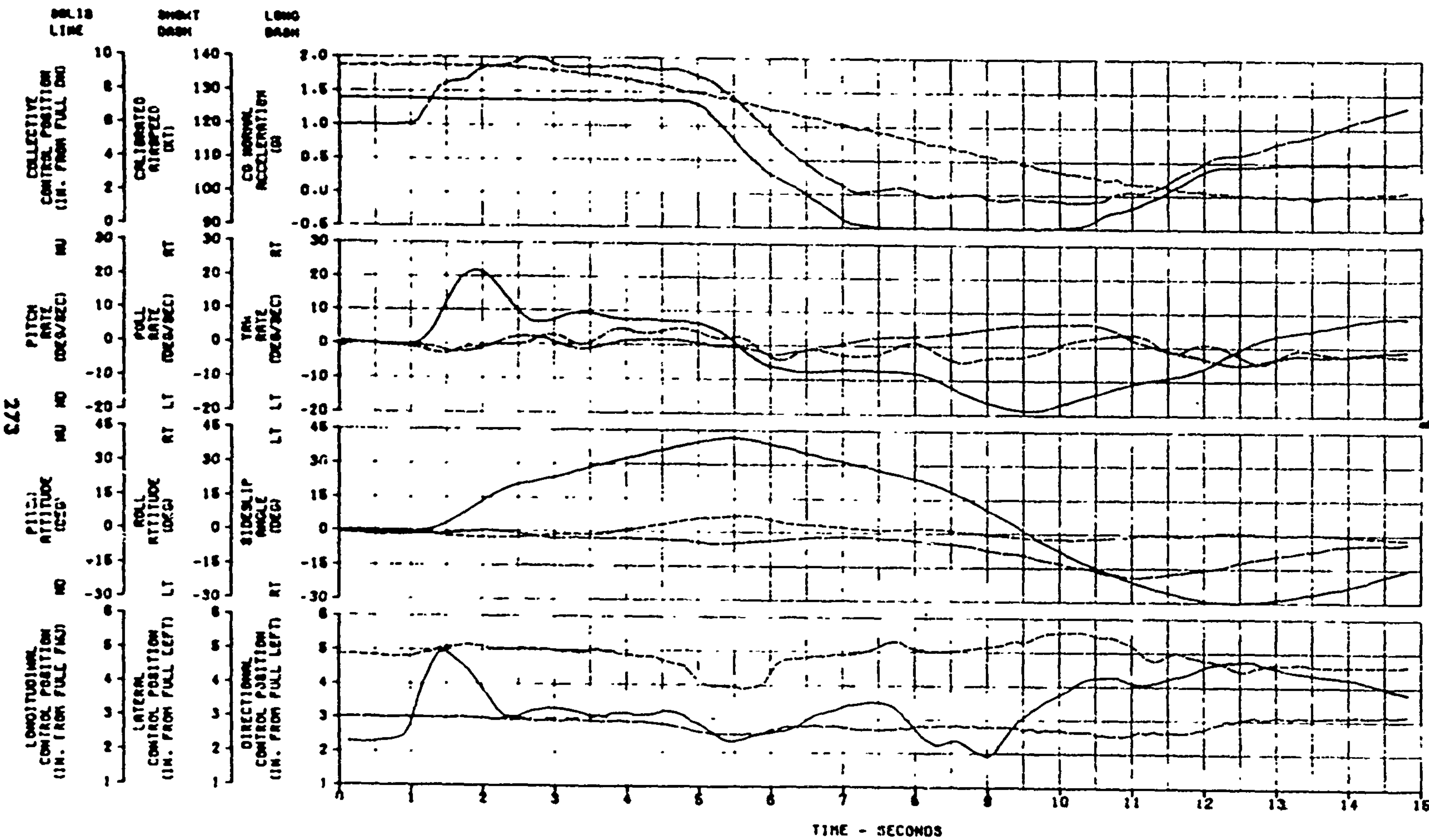


FIGURE 106
SIMULATED ENGINE FAILURE
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY CALIBRATED AIRSPEED (KT)	ENTRY ENGINE TORQUE (PERCENT)	ENTRY FLIGHT CONDITION	AFCS CONDITION
16890	358.6 (AFT)	.6 (RT)	6000	17.0	166	98/97	LEVEL	ON

NOTE: NO 1 ENGINE FAILED FROM STABILIZED DUAL ENGINE POWERED LEVEL FLIGHT

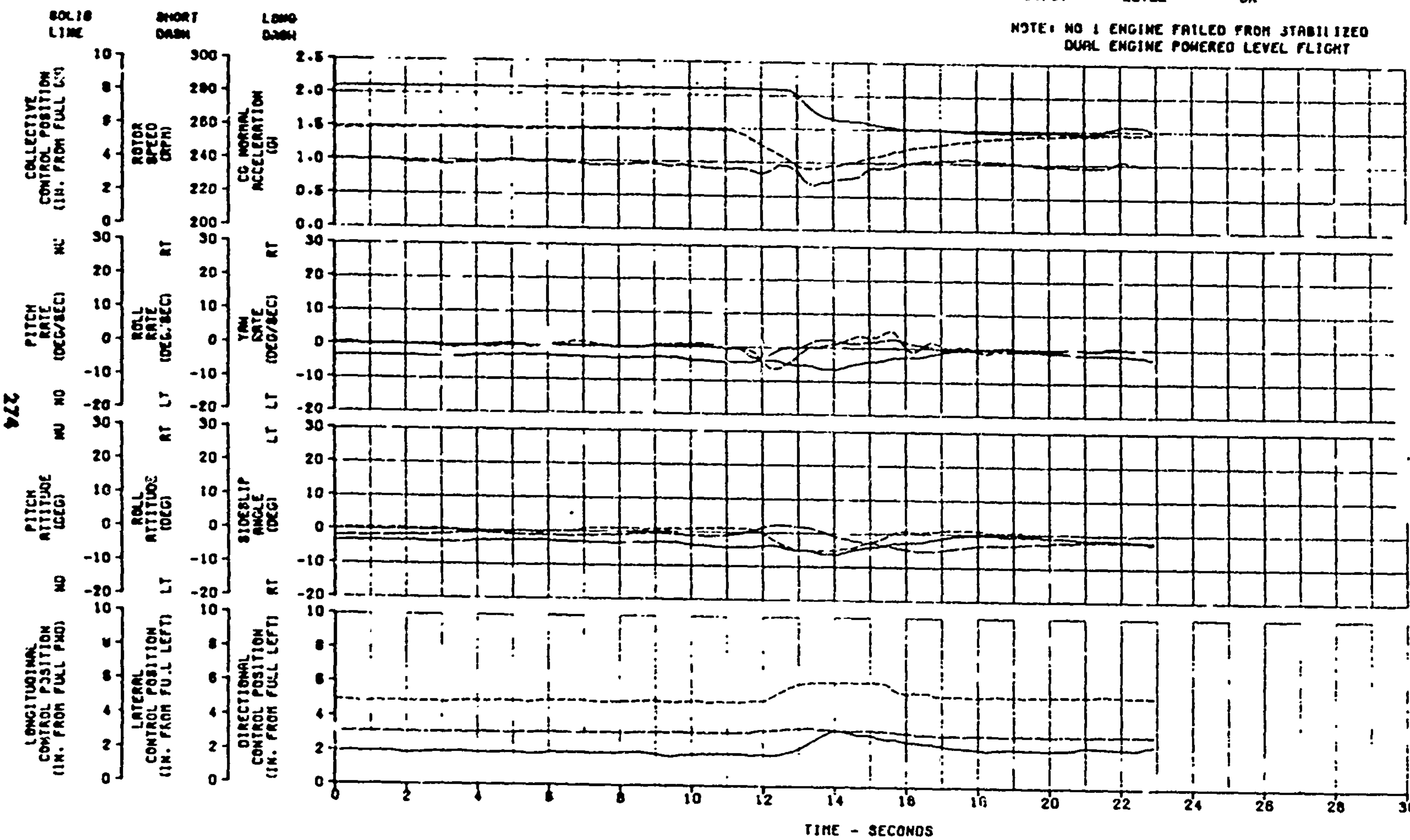


FIGURE 107
SIMULATED ENGINE FAILURE
YUH-60A USA S/N 73-21661

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY CALIBRATED AIRSPEED (KT)	ENTRY ENGINE TORQUE (PERCENT)	ENTRY FLIGHT CONDITION	AFCS CONDITION
16630	358.8 (AFT)	.4 (RT)	6160	16.0	53	0/91	LEVEL	ON

NOTE: NO 2 ENGINE FAILED FROM STABILIZED SINGLE ENGINE POWERED LEVEL FLIGHT

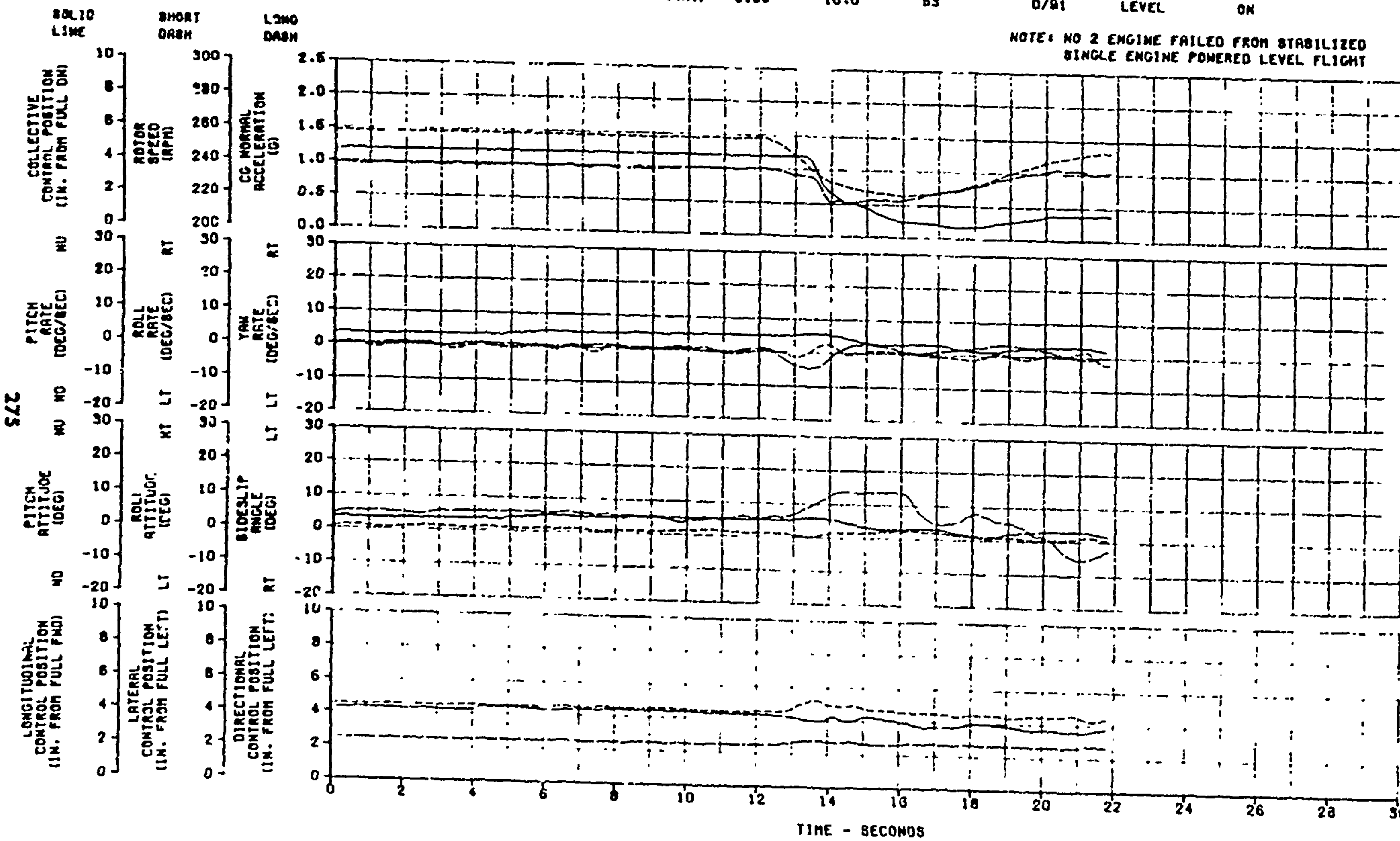


FIGURE 108

SIMULATED ENGINE FAILURE

YUH-60A USA S/N 73-21651

[redacted]

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY CALIBRATED AIRSPEED (KT)	ENTRY ENGINE TORQUE (PERCENT)	ENTRY FLIGHT CONDITION	AFCS CONDITION
16490	358.8 (AFT)	.4 (RT)	6090	17.0	77	45/46	LEVEL	ON

NOTE: NO 2 ENGINE FAILED FROM STABILIZED SINGLE ENGINE POWERED LEVEL FLIGHT

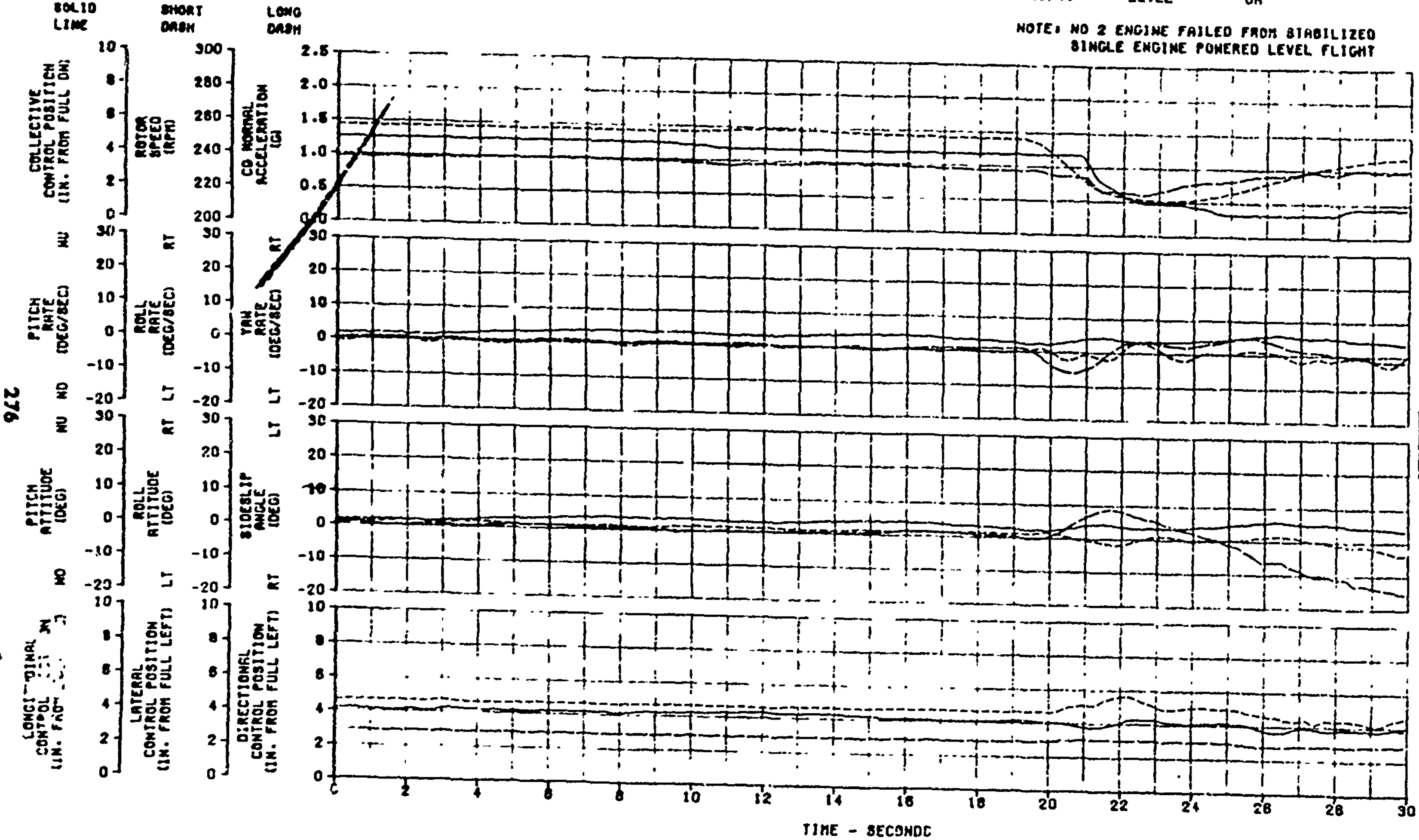


FIGURE 1C9

SIMULATED ENGINE FAILURE

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY CALIBRATED AIRSPEED (KT)	ENTRY ENGINE TORQUE (PERCENT)	ENTRY FLIGHT CONDITION	AFCS CONDITION
16420	358.8(AFT)	.4(RT)	4670	17.0	96	0/93	LEVEL	ON

NOTE: NO 2 ENGINE FAILED FROM STABILIZED SINGLE ENGINE POWERED LEVEL FLIGHT

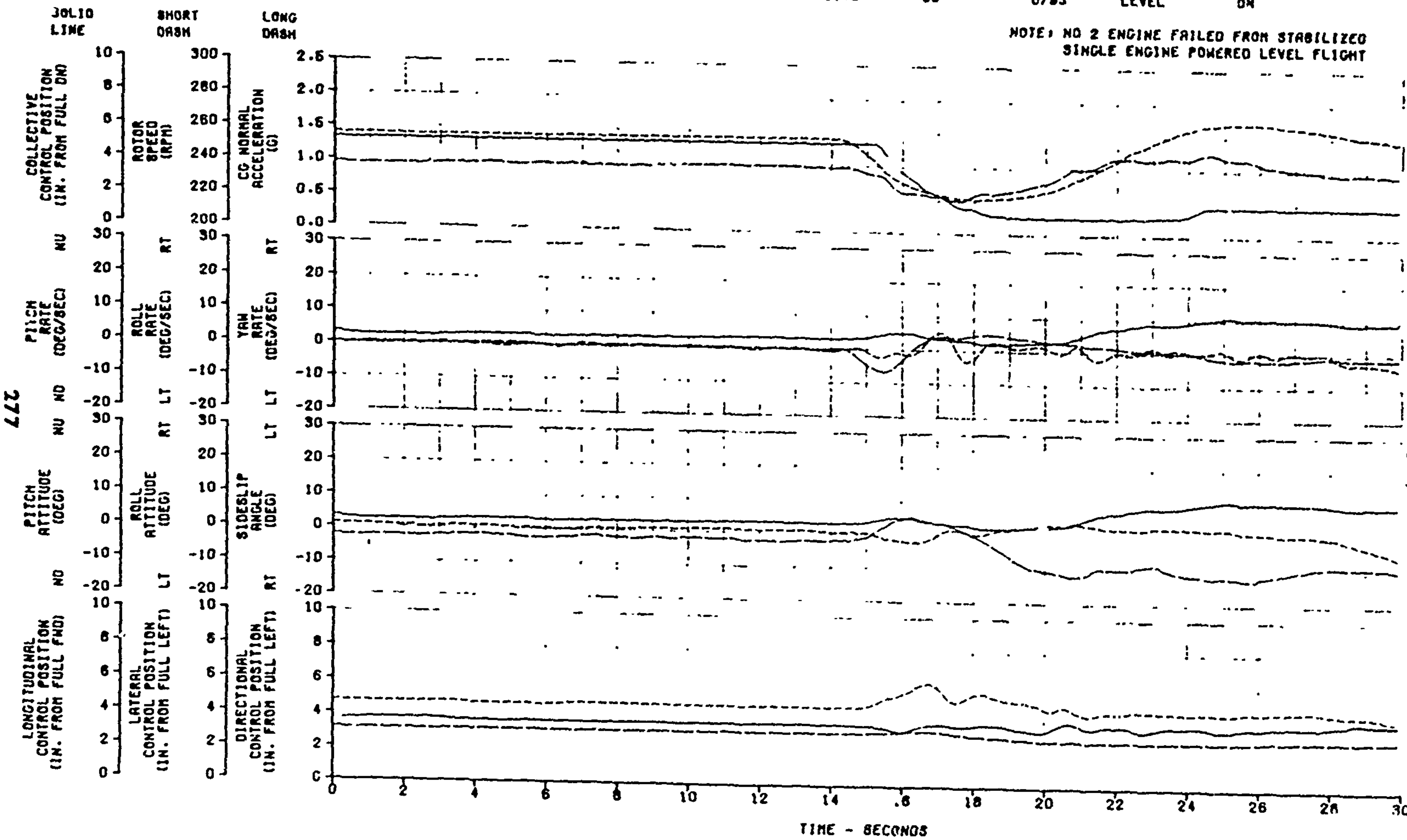
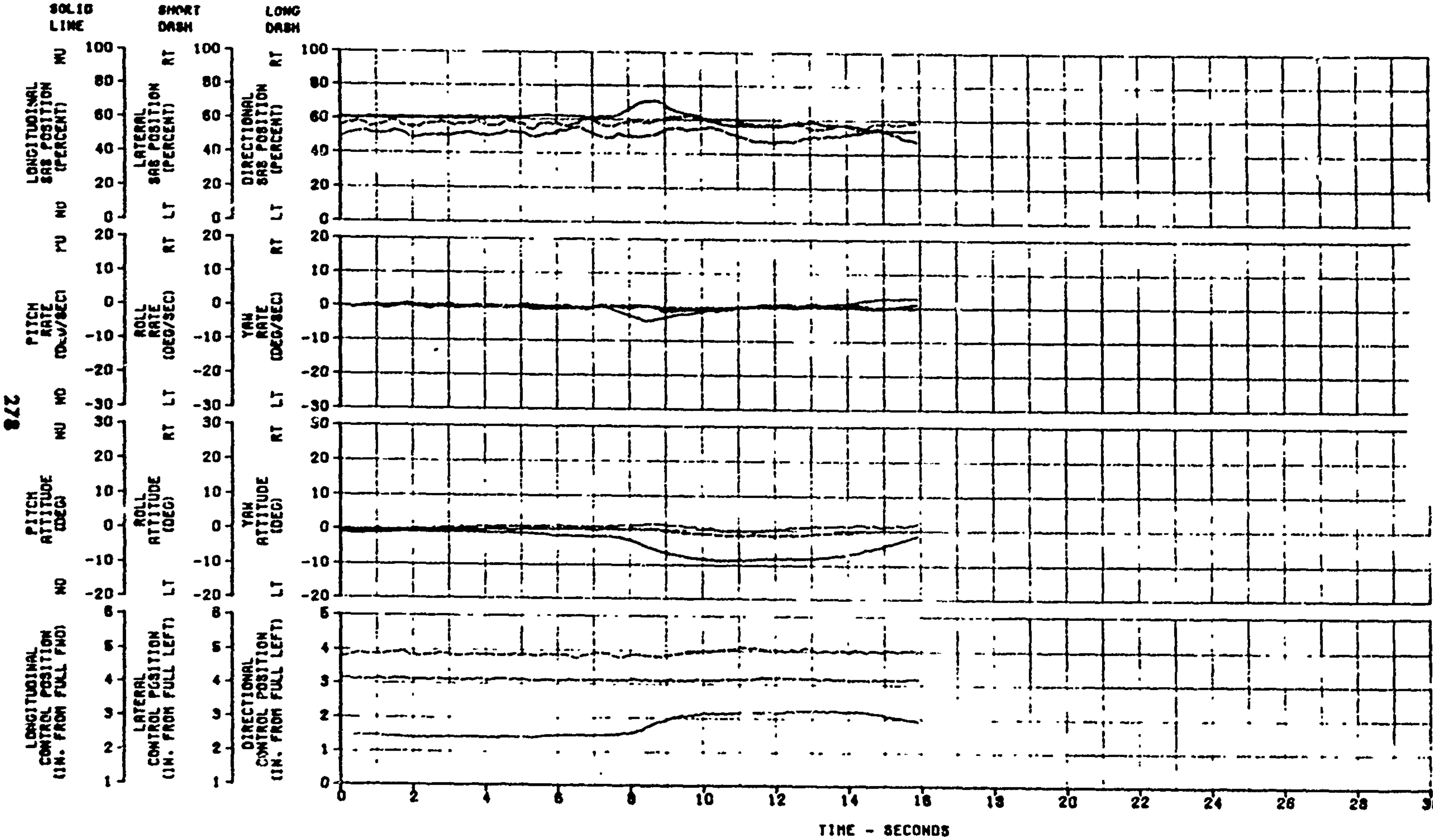


FIGURE 110

AFCS FAILURE (PITCH BIAS ACTUATOR)

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
16470	358.1 (AFT)	0.4 (RT)	6060	23.0	257	144	LEVEL



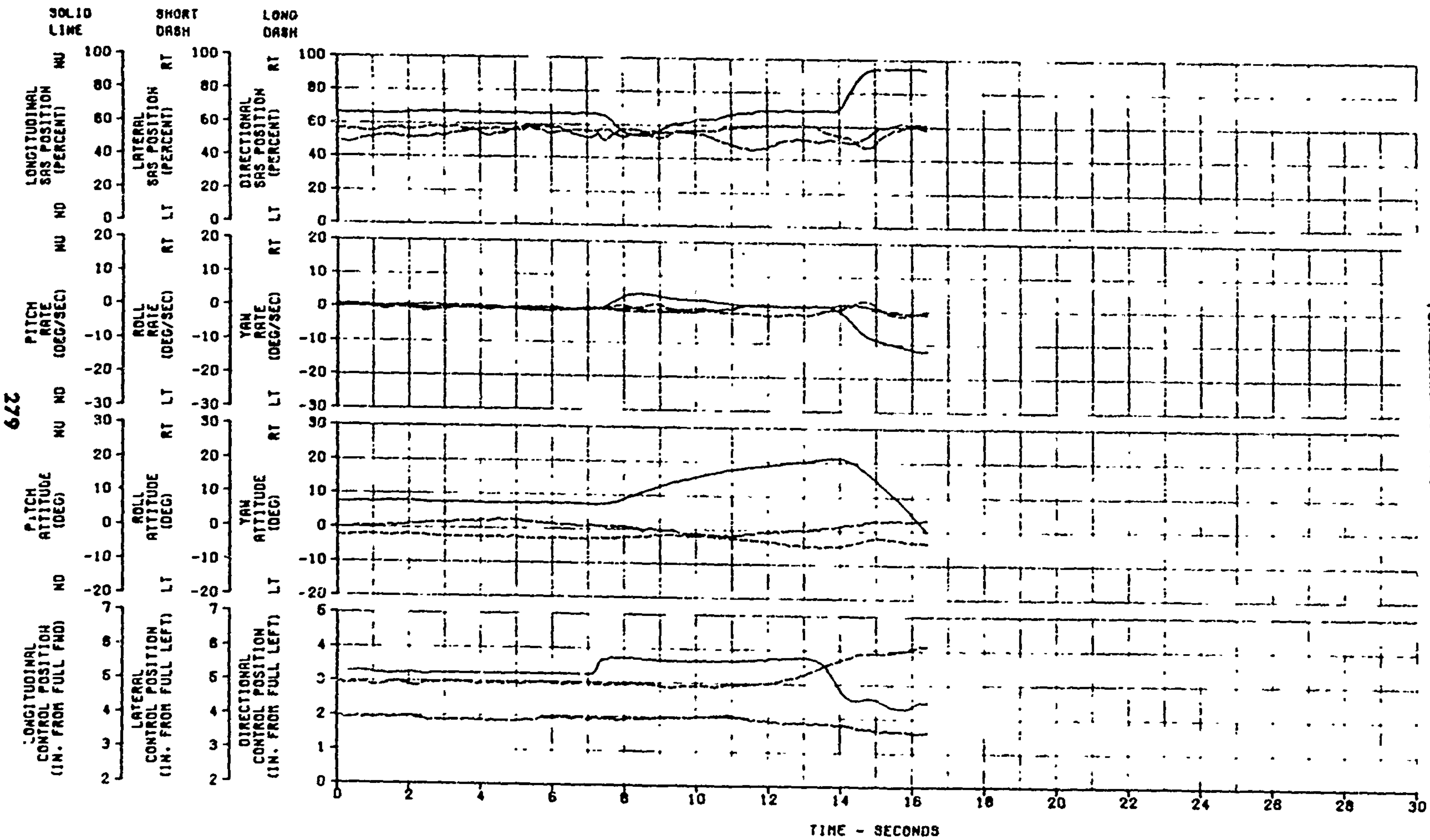
[redacted]

FIGURE 111

AFCS FAILURE (FAS HARDOVER)

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
17360	368.9 (AFT)	0.4 (RT)	3060	17.5	257	20	LEVEL



[redacted]

[redacted]

FIGURE 112
AFCS FAILURE (LATERAL TRIM RUNAWAY)
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
16300	358.0 (AFT)	0.4 (RT)	6030	23.5	257	141	LEVEL

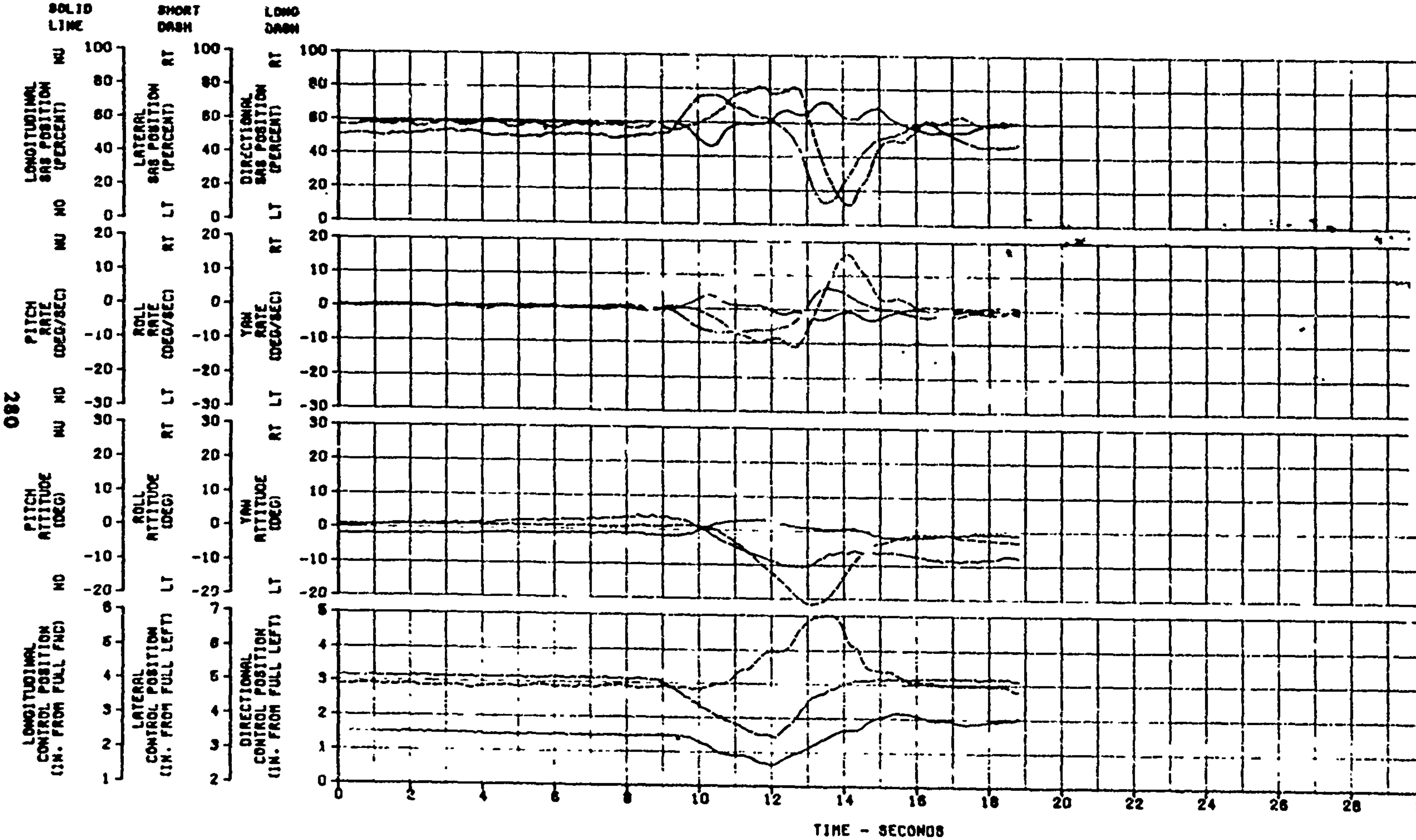


FIGURE 113

AFCS FAILURE (DIRECTIONAL TRIM RUNAWAY)

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
17230	358.8 (AFT)	0.4 (RT)	2960	17.0	258	0	HOVER

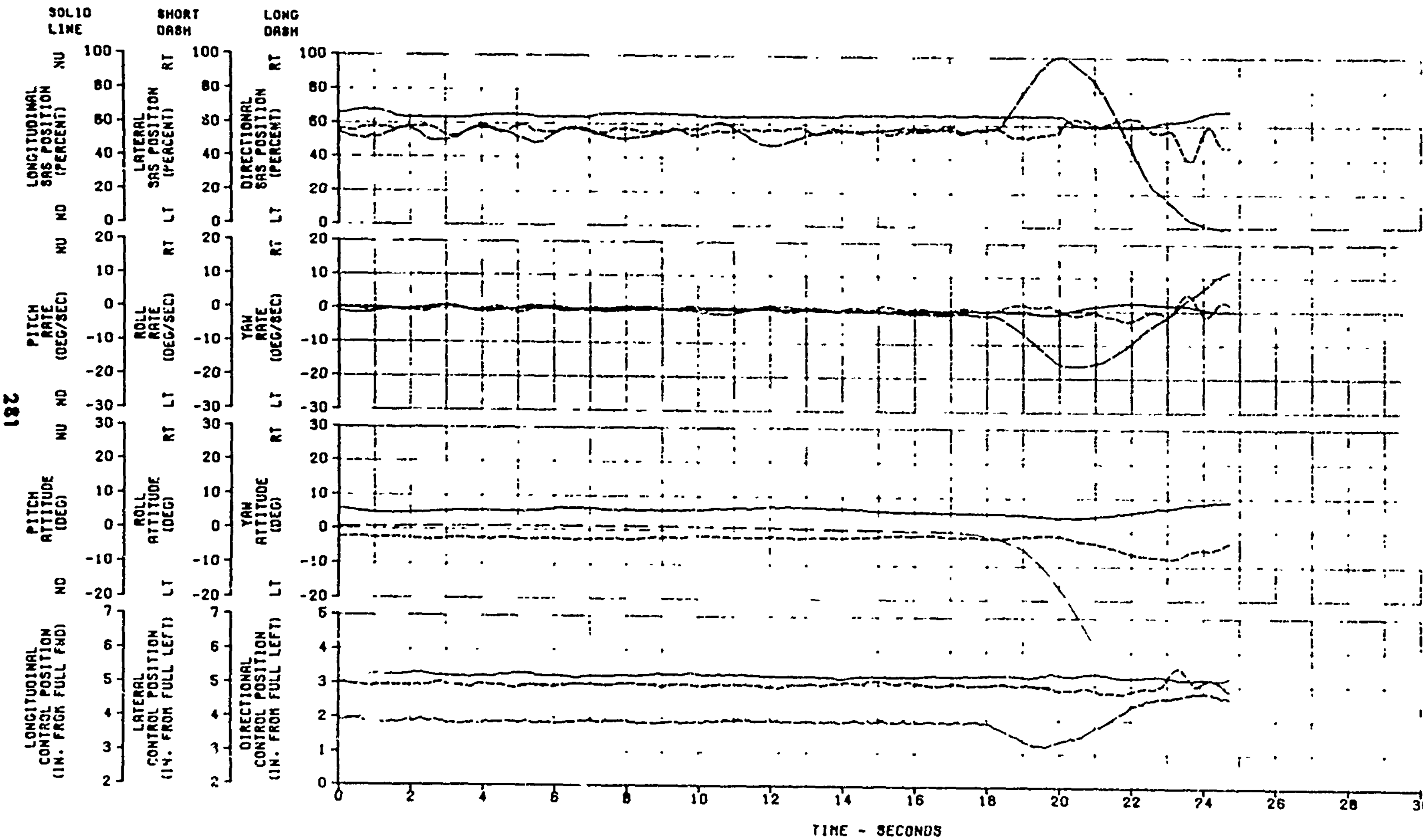
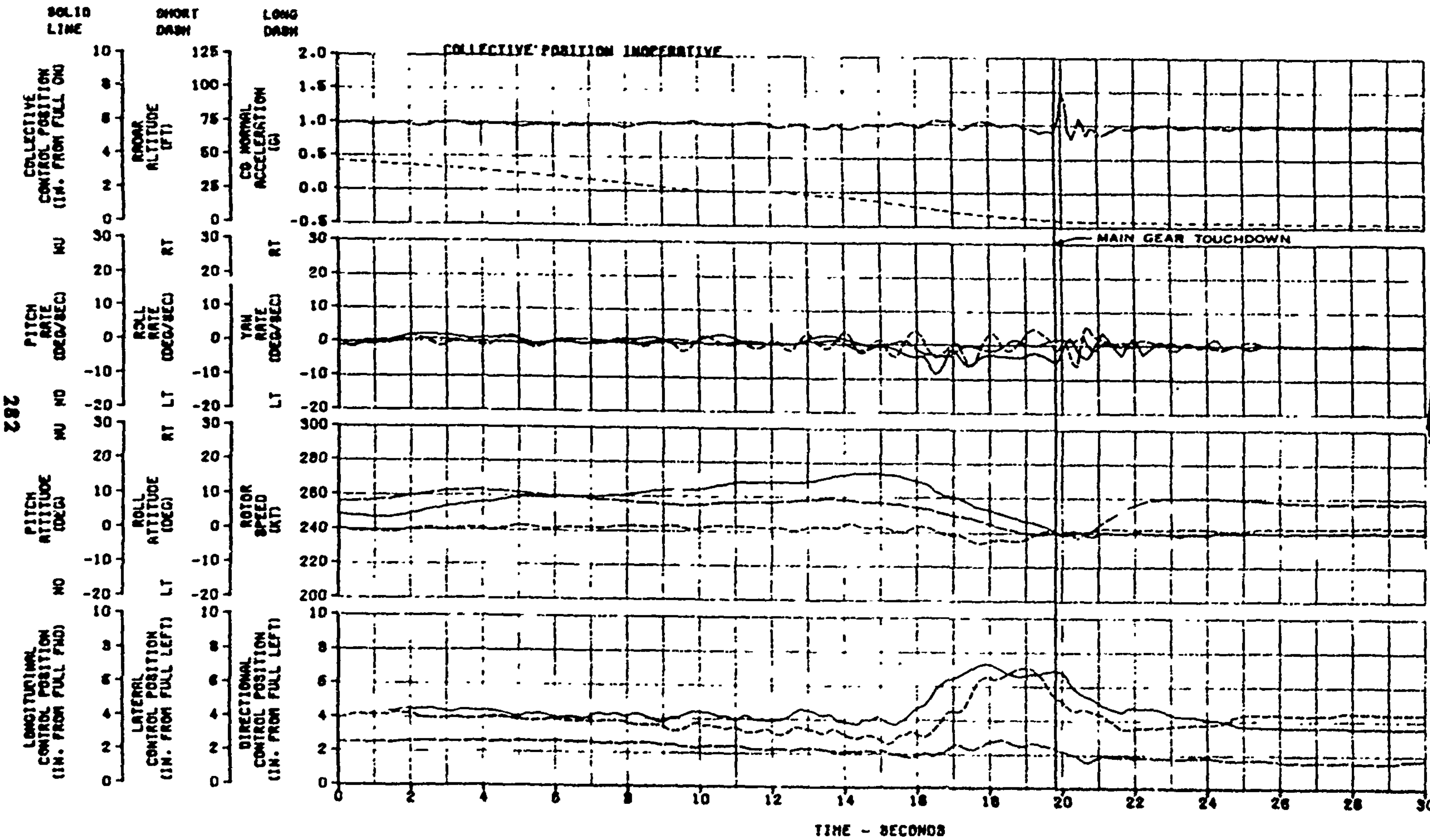


FIGURE 114

SINGLE ENGINE LANDING

YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
16300	360.5 (AFT)	.5 (RT)	5220	17.5	259	ON



[redacted]

FIGURE 115
SINGLE ENGINE LANDING
YUH-60A USA S/N 73-21651

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	AFCS CONDITION
16260	360.5 (AFT)	.5 (RT)	5220	17.6	259	ON

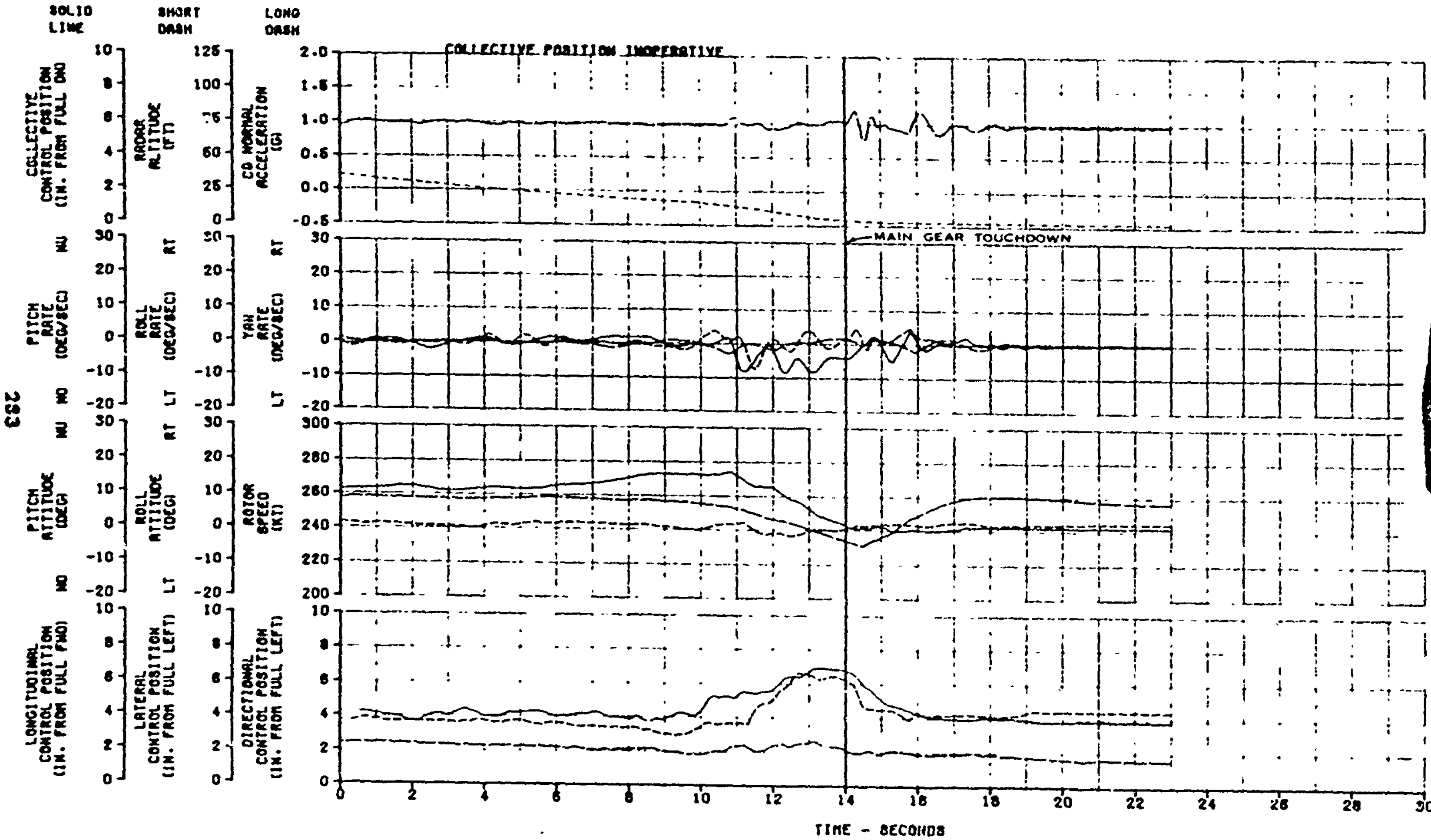


FIGURE 116
VIBRATION CHARACTERISTICS
YUH-60A USA S/N 73-21651
PILOT STATION
4/REV (17.2 Hz)

SYMBOL	AVG GROSS WEIGHT (LB)	LOCATION LONG (IN.)	LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITIONS
○	17240	359.5(AFT)	0.4(RT)	7600	11.0	257	LEVEL AND IRP DIVE
△	20000	360.0(AFT)	0.4(RT)	6640	12.0	258	LEVEL AND IRP DIVE

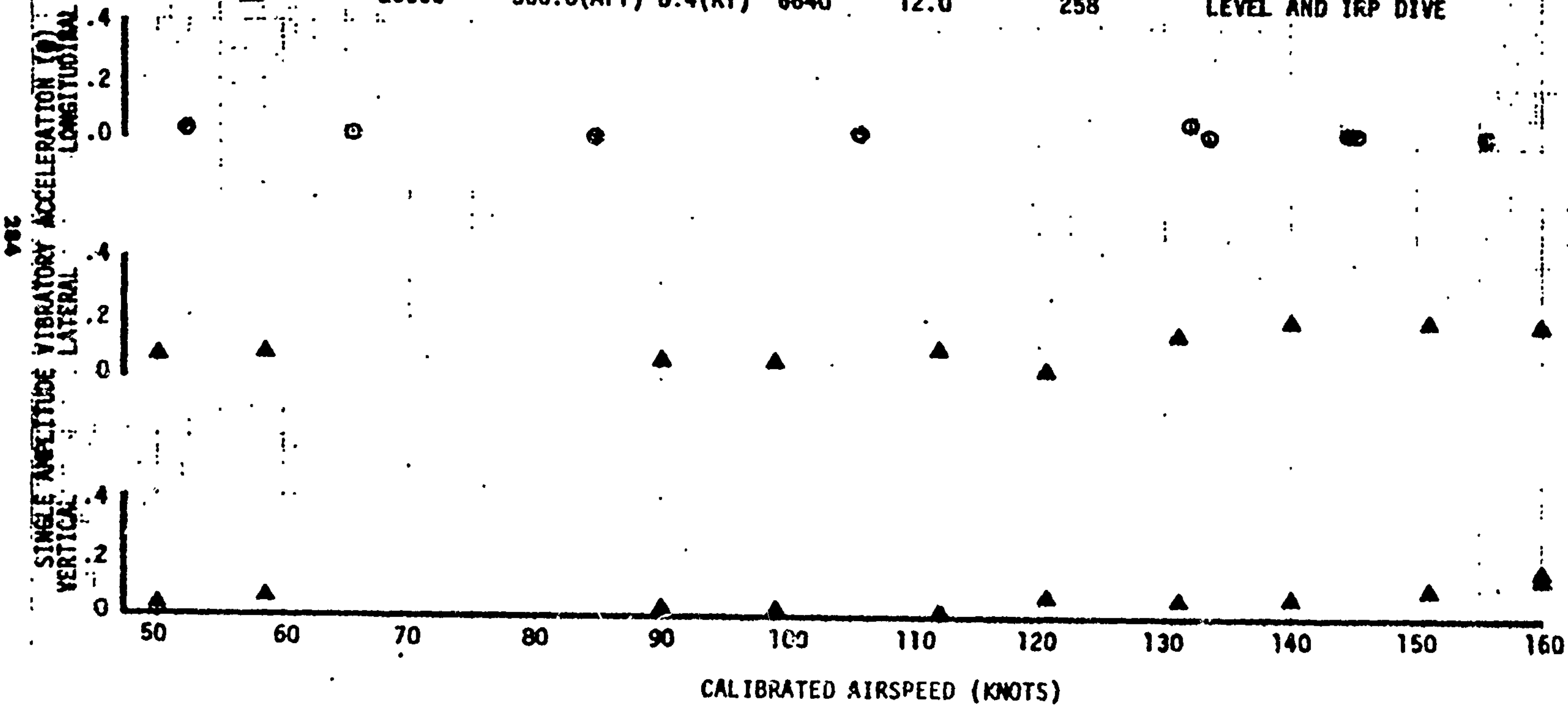


FIGURE 117
VIBRATION CHARACTERISTICS
YUH-60A USA S/N 73-21651
COPILOT STATION
4/REV (17.2Hz)

SYMBOL	AVG GROSS WEIGHT (LB)	LOCATION LONG (IN.)	LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITIONS
○	17240	359.5(AFT)	0.4(RT)	7600	11.0	257	LEVEL AND IRP DIVE
△	20000	360.0(AFT)	0.4(RT)	6640	12.0	258	LEVEL AND IRP DIVE

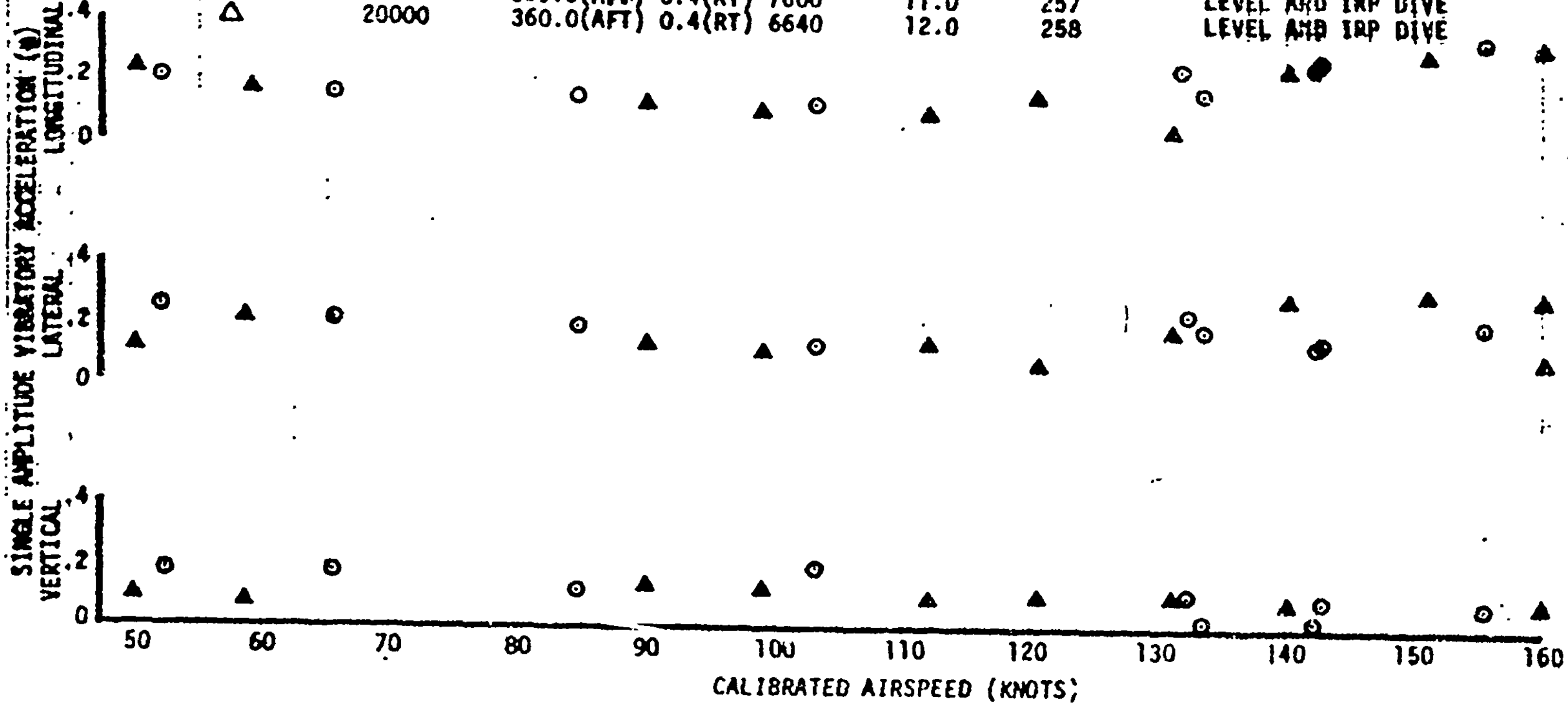


FIGURE 113

VIBRATION CHARACTERISTICS

YUH-60A USA S/N 73-21651

CENTER OF GRAVITY

4/REV (17.2 Hz)

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITIONS
○	17250	359.5(AFT)	0.4(RT)	7600	11.0	257	LEVEL AND IRP DIVE
△	23000	360.0(AFT)	0.4(RT)	6640	12.0	258	LEVEL AND IRP DIVE

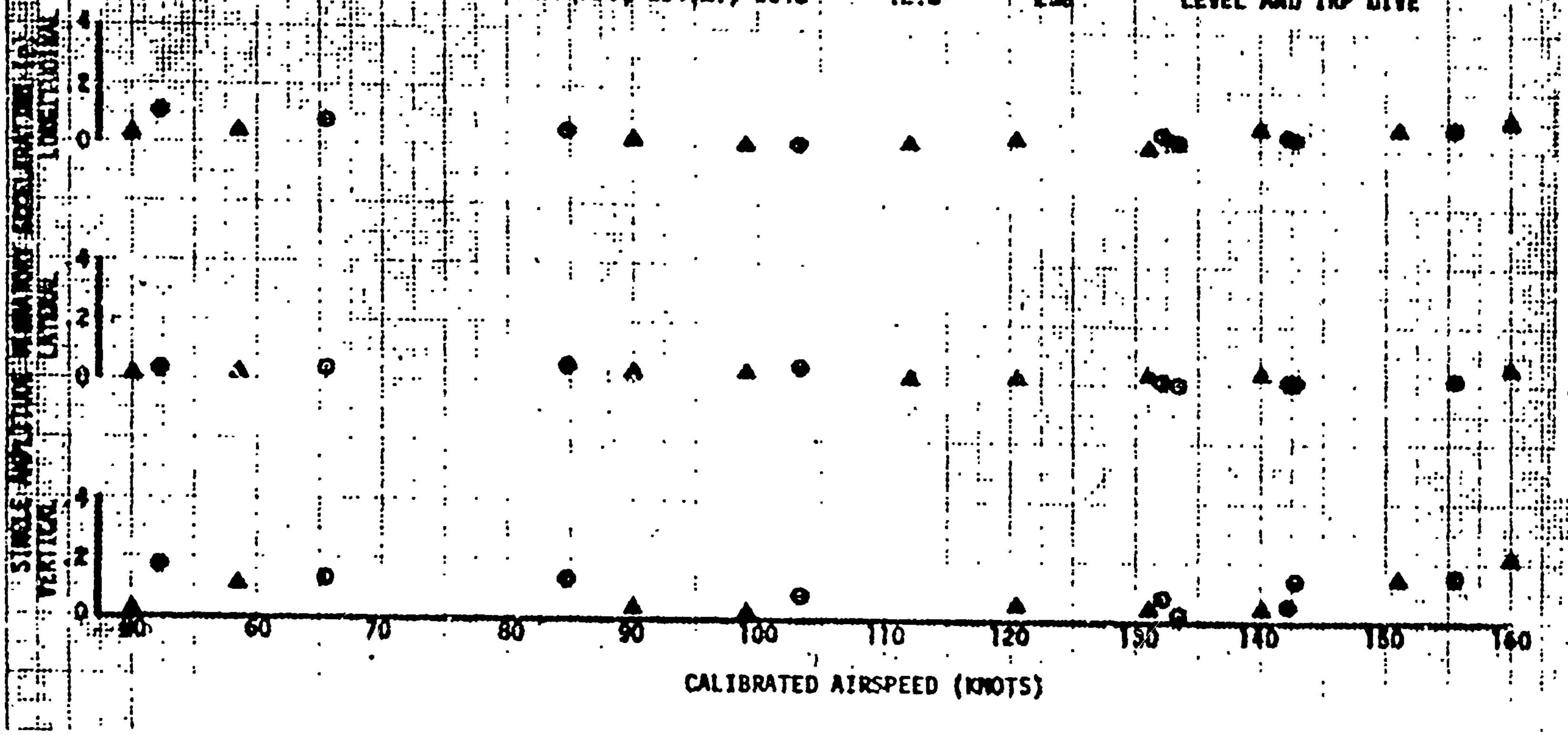
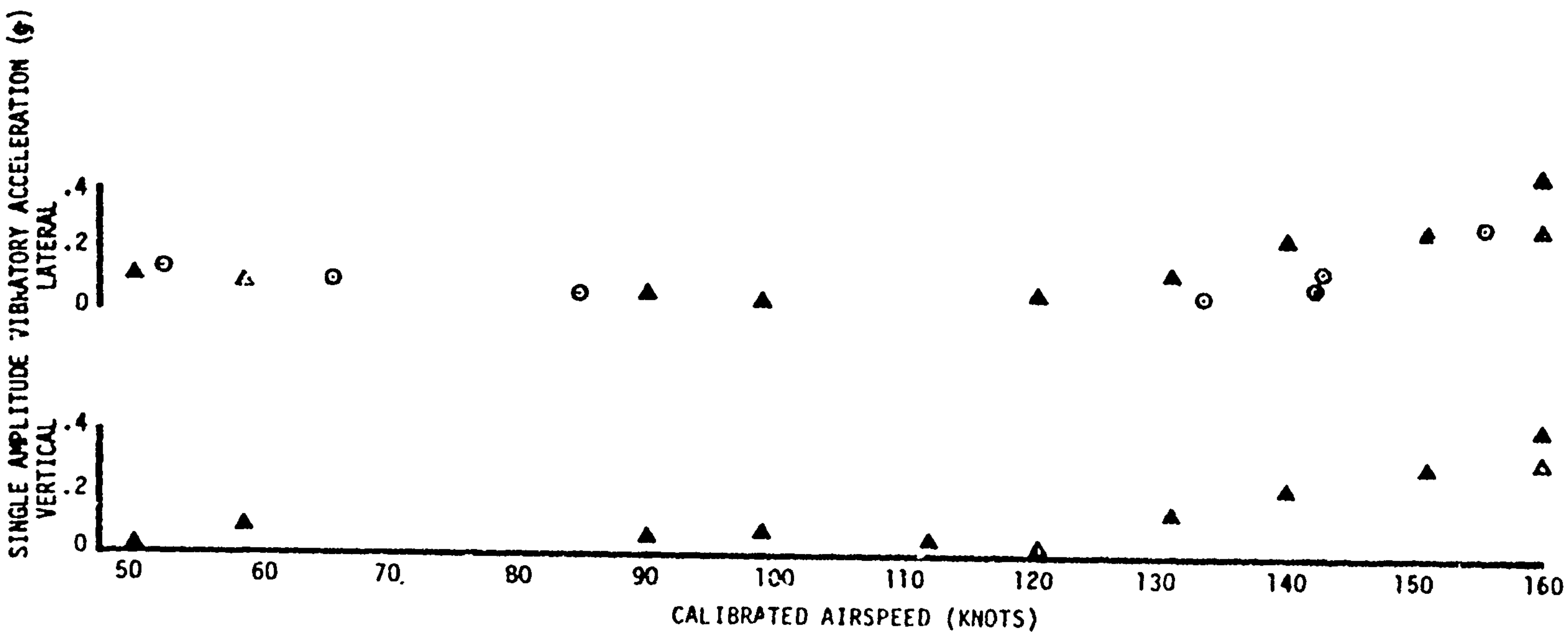


FIGURE 119
VIBRATION CHARACTERISTICS
YUH-60A USA S/N 73-21651
INSTRUMENT PANEL (RIGHT SIDE)
4/REV (17.2 Hz)

SYMBOL	AVG GROSS WEIGHT (LB)	LOCATION LONG (IN.)	LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	FLIGHT CONDITIONS
○	17240	359.5(AFT)	0.4(RT)	7600	11.0	257	LEVEL AND IRP DIVE
△	20000	360.0(AFT)	0.4(RT)	6640	12.0	258	LEVEL AND IRP DIVE



[redacted]

FIGURE 120
VIBRATION CHARACTERISTICS
YUH-60A USA S/N 73-21651
PILOT SEAT
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	TRIM CALIB. AIRSPEED (KTS)	FLIGHT CONDITION
17040	359.5(AFT)	0.4(RT)	7570	10.0	140	LEVEL FLIGHT

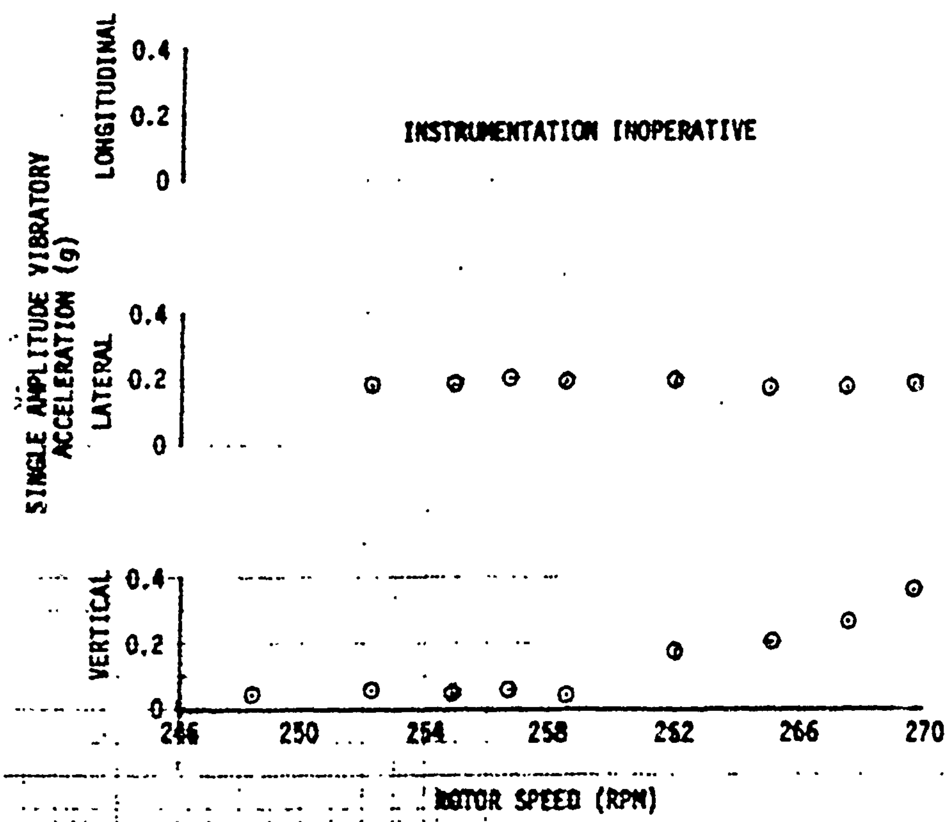
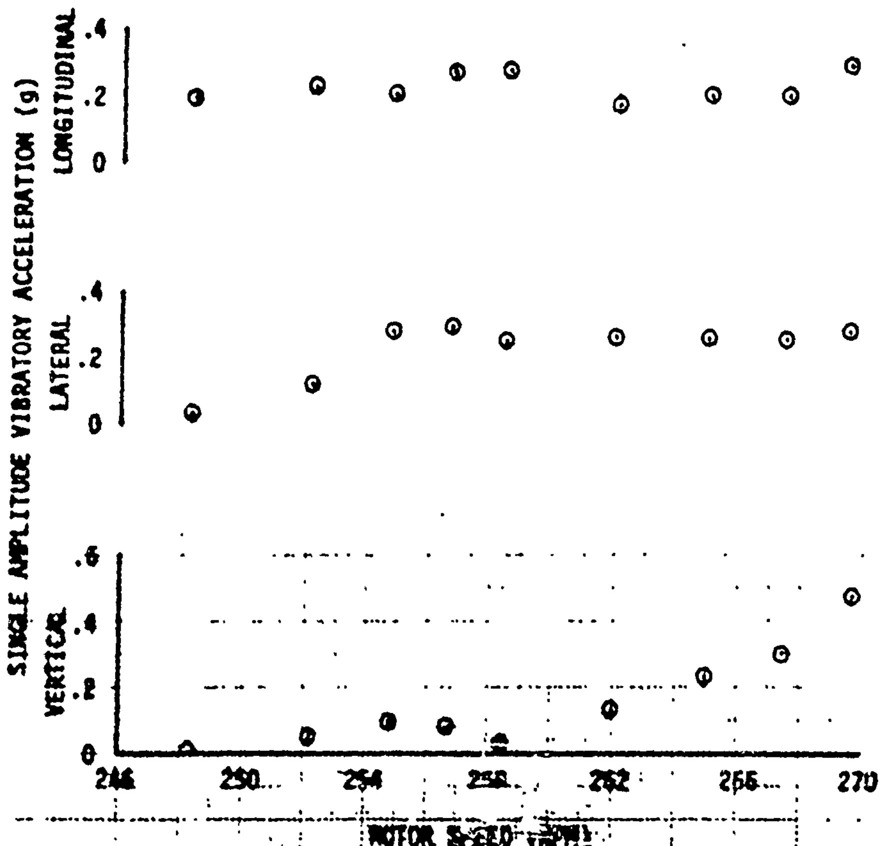


FIGURE 121
VIBRATION CHARACTERISTICS
YUH-60A USA S/N 73-21651
COPILOT SEAT
4/REV (17.2 Hz)

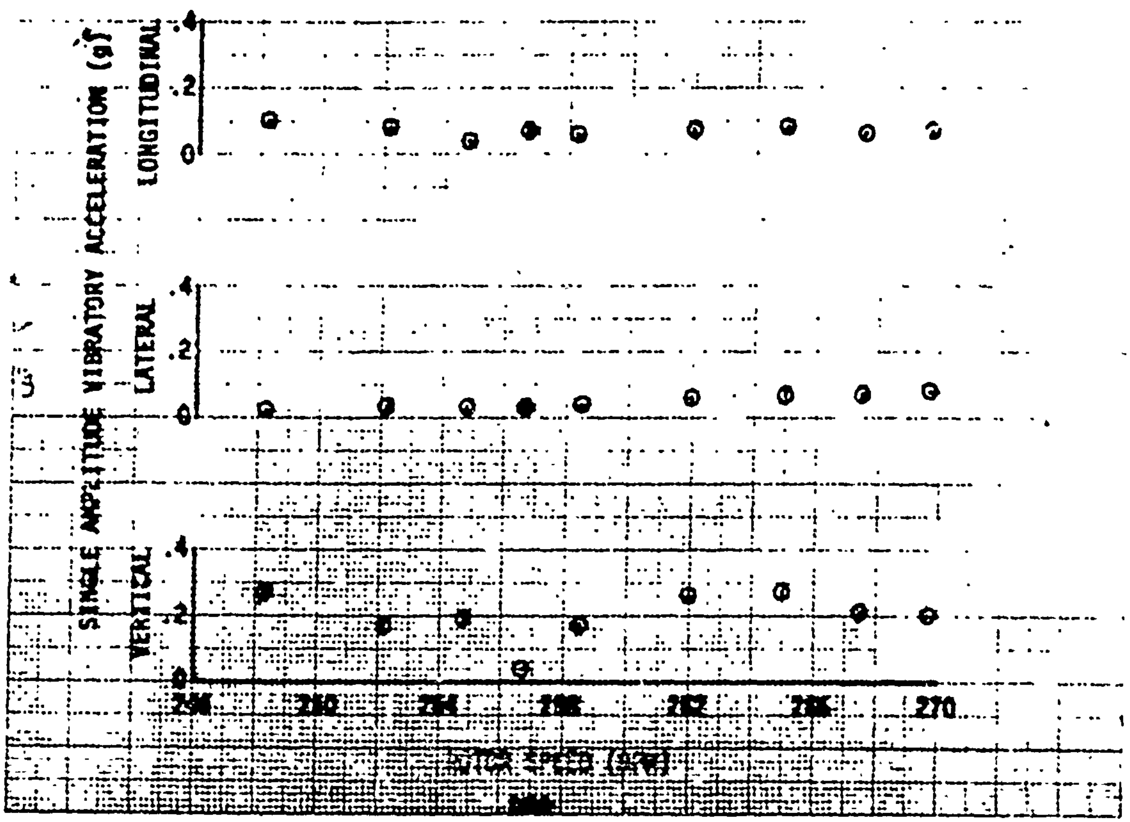
AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	TRIM CALIB. AIRSPEED (KTS)	FLIGHT CONDITION
17040	359.5(AFT)	0.6(RT)	7570	10.0	140	LEVEL FLIGHT



~~FOR OFFICIAL USE ONLY~~

FIGURE [illegible]
VIBRATION CHARACTERISTICS
[illegible] (2165)
[illegible]
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	TRIM CALIB. AIRSPEED (KTS)	FLIGHT CONDITION
17040	359.5(AFT)	[illegible]	[illegible]	10.0	140	LEVEL FLIGHT



[redacted]CIAL USE ONLY

FIGURE 123[illegible]

VIBRATION CHARACTERISTICS

[illegible] USA S/N 72-21651

INSTRUMENT PANEL (RIGHT SIDE)

4/REV (17.2 HZ)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	TRIM CALIB. AIRSPEED (KTS)	FLIGHT CONDITION
17040	359.5(AFT)	0.4(RT)	7570	10.0	140	LEVEL FLIGHT

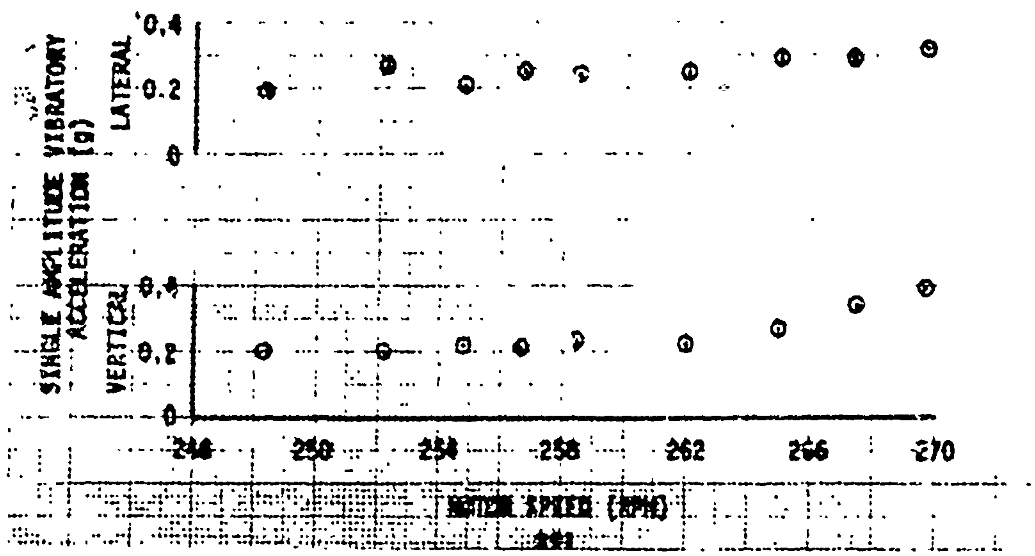


FIGURE 124
VIBRATION CHARACTERISTICS IN LOW-SPEED FORWARD AND REARWARD FLIGHT
YUH-60A S/N 73-21651
PILOT STATION
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
16900	348.5(FWD)	.4(RT)	2740	14.5	259

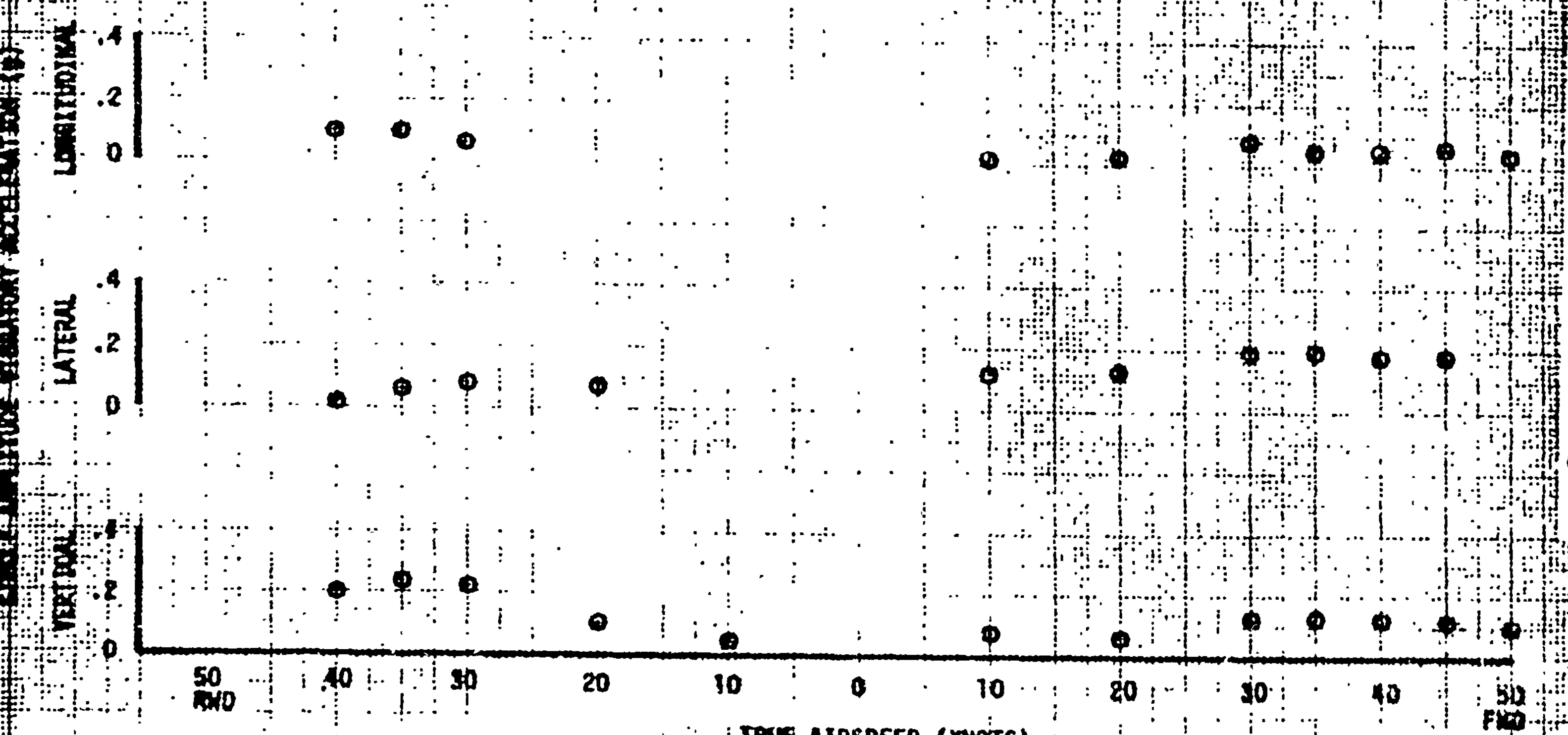
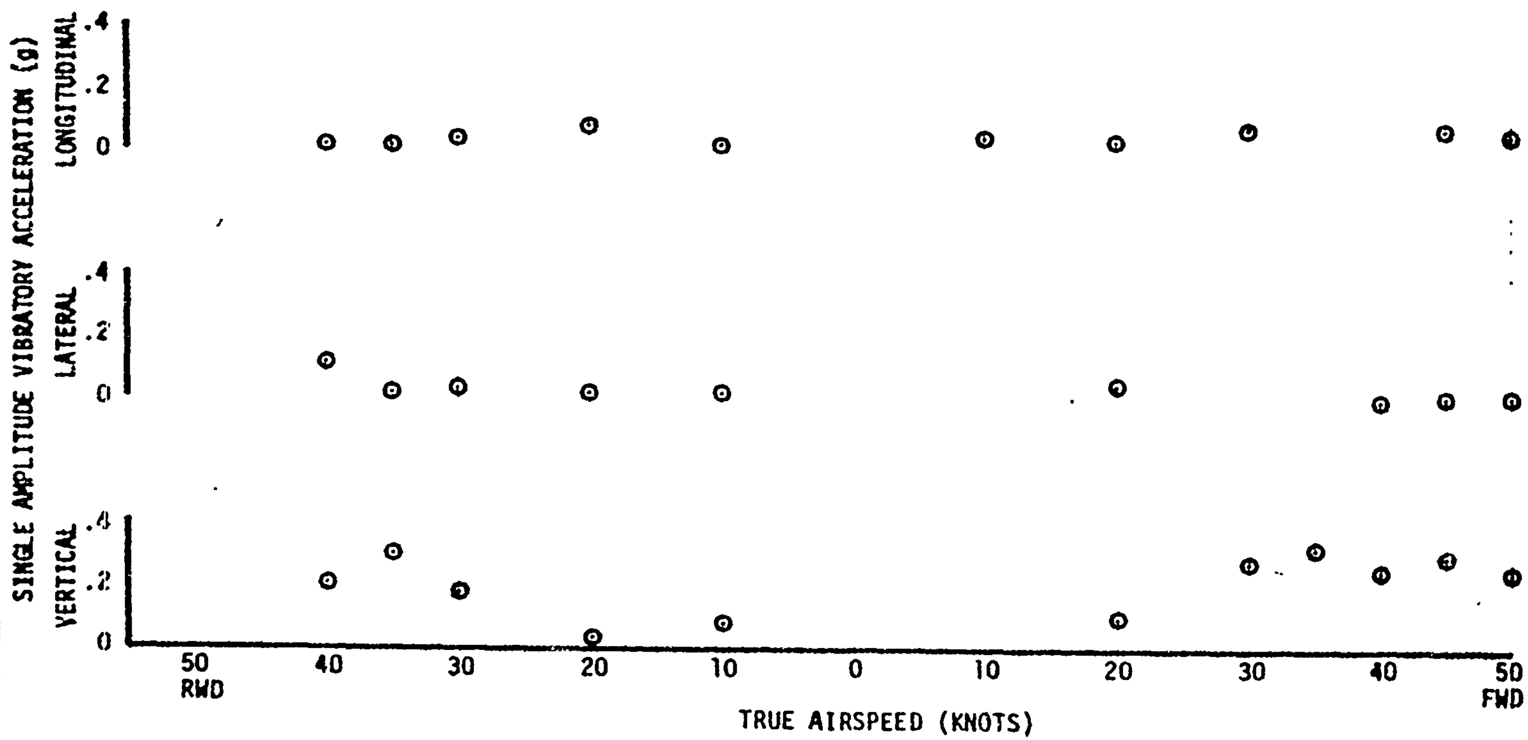


FIGURE 125
VIBRATION CHARACTERISTICS IN LOW-SPEED FORWARD AND REARWARD FLIGHT
YUH-60A S/N 73-21651
CENTER OF GRAVITY
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
16900	348.5(FWD)	.4(RT)	2740	14.5	259



[redacted]

FIGURE 126
VIBRATION CHARACTERISTICS IN LOW-SPEED FORWARD AND REARWARD FLIGHT
YUH-60A S/N 73-21651
INSTRUMENT PANEL (RIGHT SIDE)
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
16900	348.5(FWD)	.4(RT)	2740	14.5	259

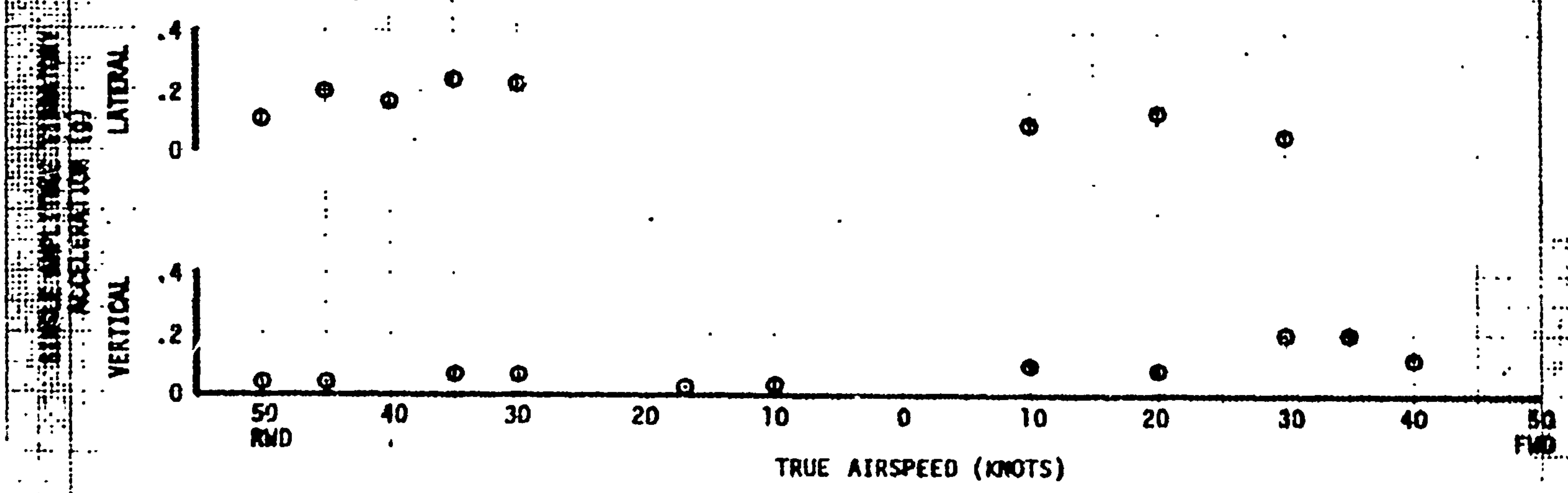


FIGURE 127
VIBRATION CHARACTERISTICS IN SIDEWARD FLIGHT
YUH-60A S/N 73-21651
PILOT STATION
4/REV (17.2Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
17180	347.7(FWD)	.4(RT)	2440	12.0	258

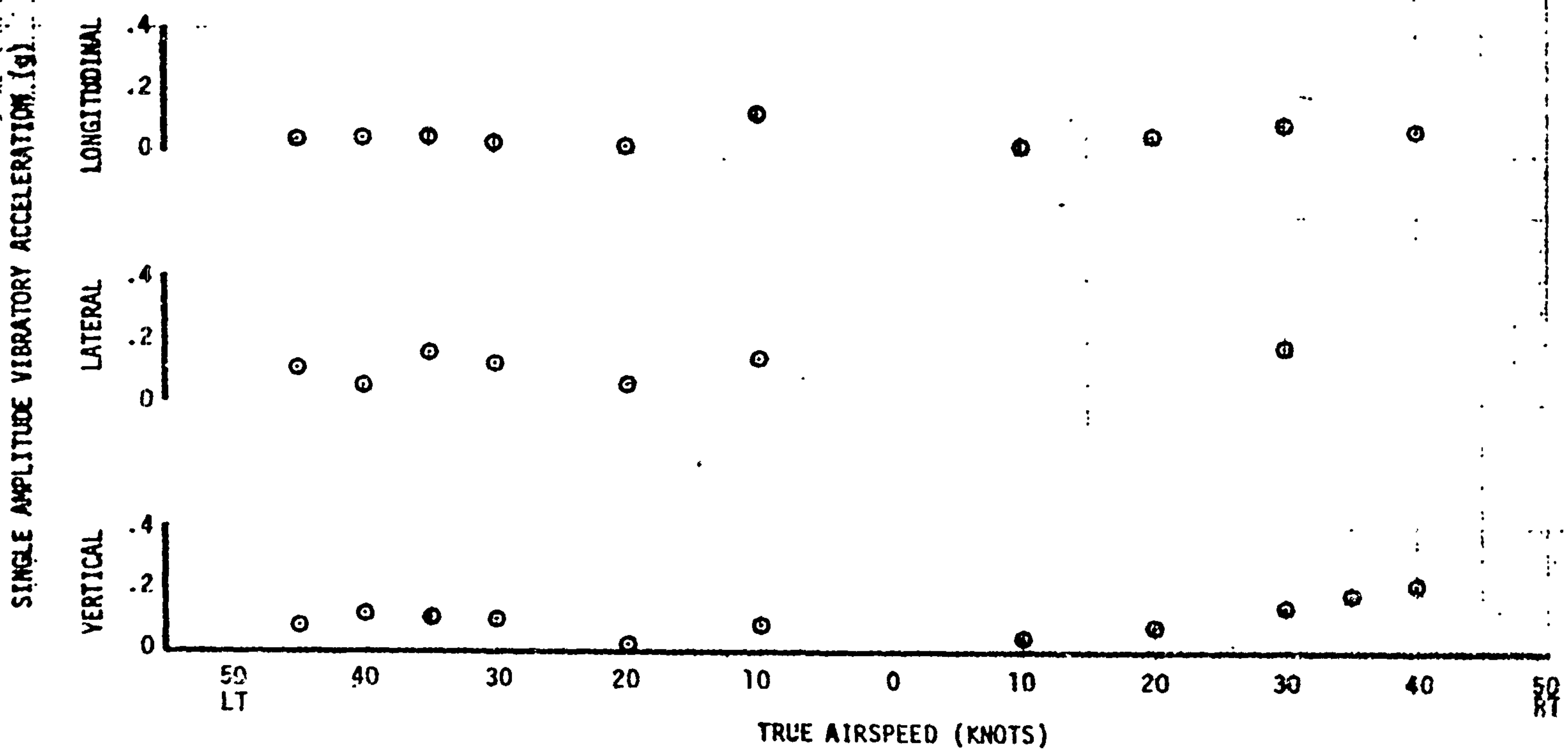


FIGURE 128
VIBRATION CHARACTERISTICS IN SIDEWARD FLIGHT
YUH-60A S/N 73-21651
COPILOT STATION
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_p (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
17180	347.7(FWD)	.4(RT)	2440	12.0	258

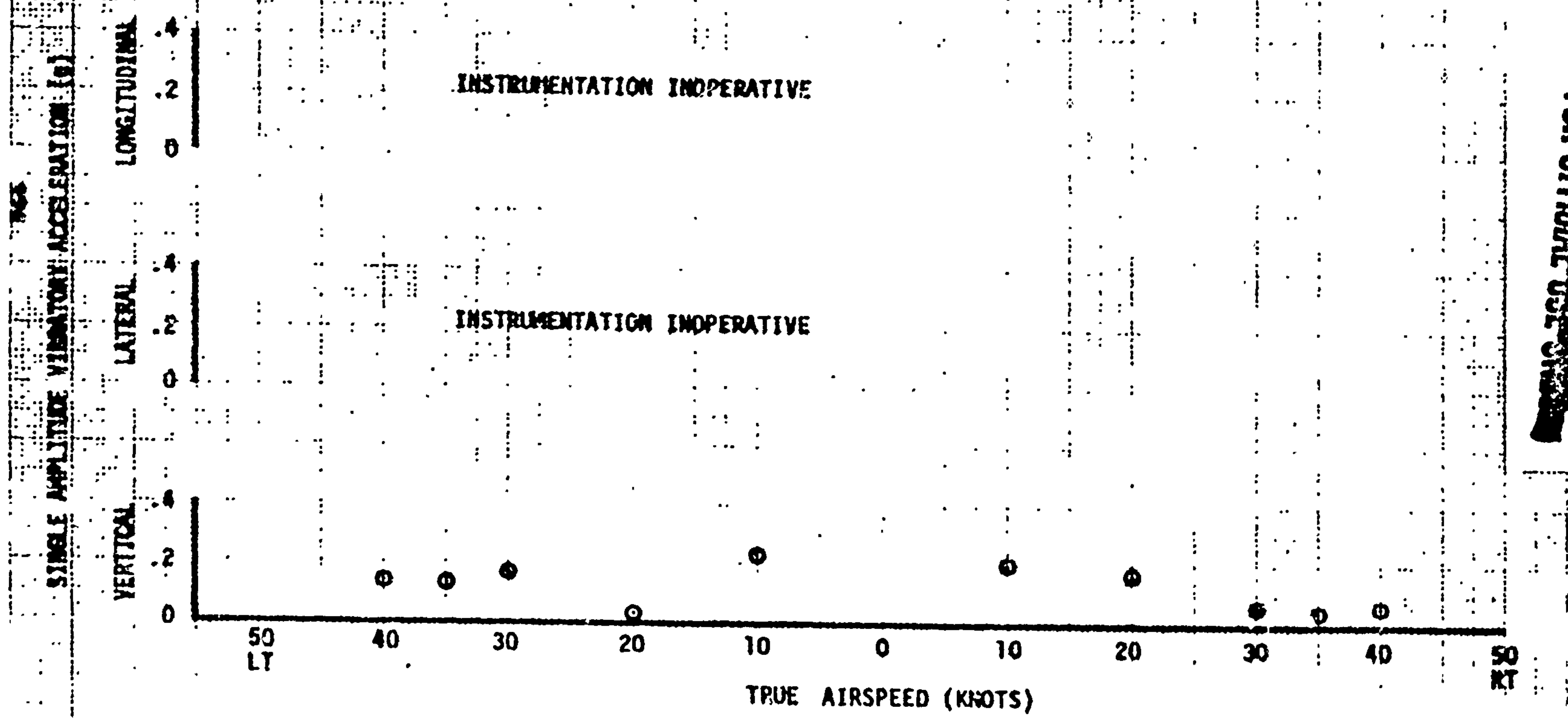


FIGURE 129
VIBRATION CHARACTERISTICS IN SIDEWARD FLIGHT
YUH-60A S/N 73-21651
CENTER OF GRAVITY
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
17180	347.7(FWD)	.4(RT)	2440	12.0	258

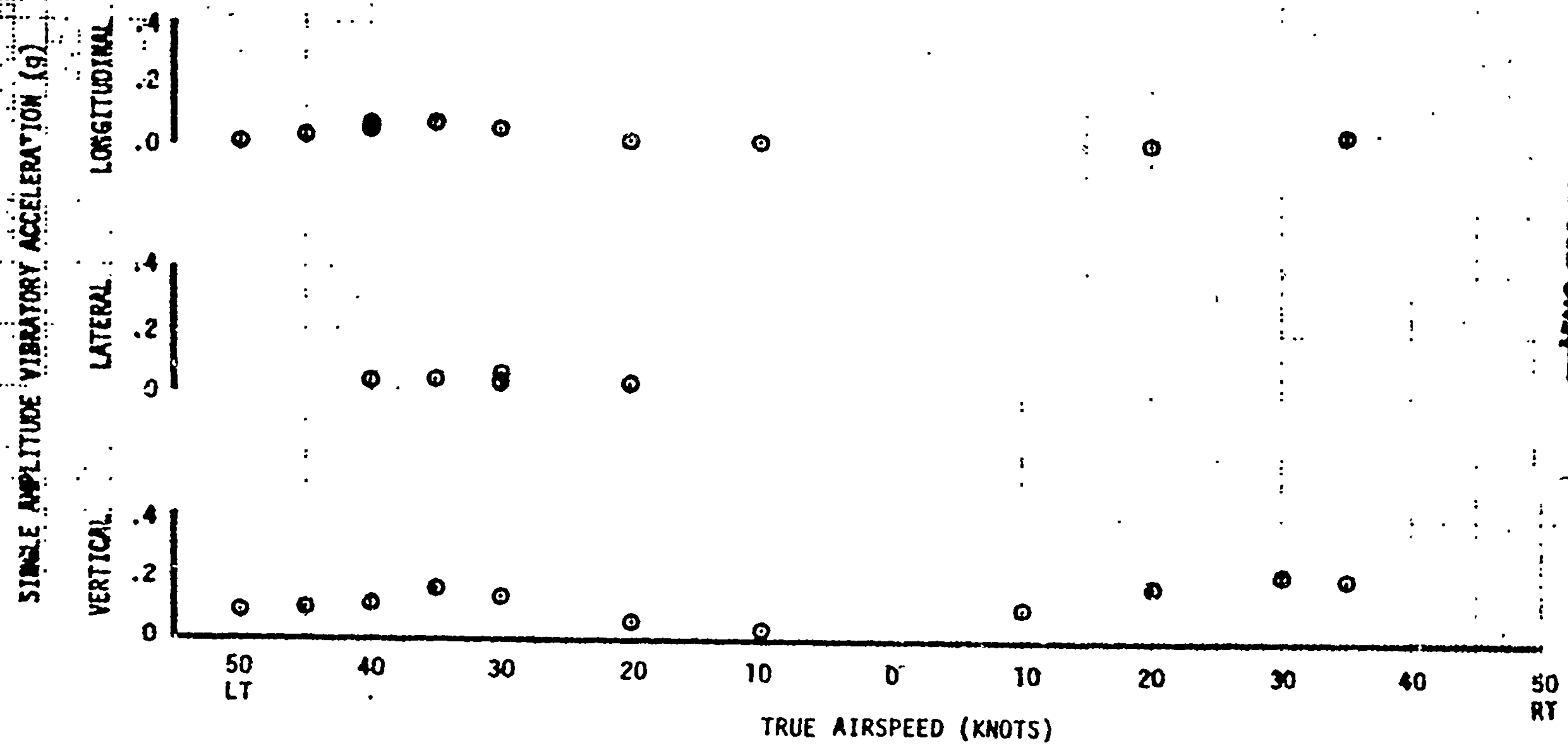


FIGURE 130
VIBRATION CHARACTERISTICS IN SIDEWARD FLIGHT
YUH-60A S/N 73-21651
INSTRUMENT PANEL (RIGHT SIDE)
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)
17180	347.7(FWD)	.4(RT)	2440	12.0	258

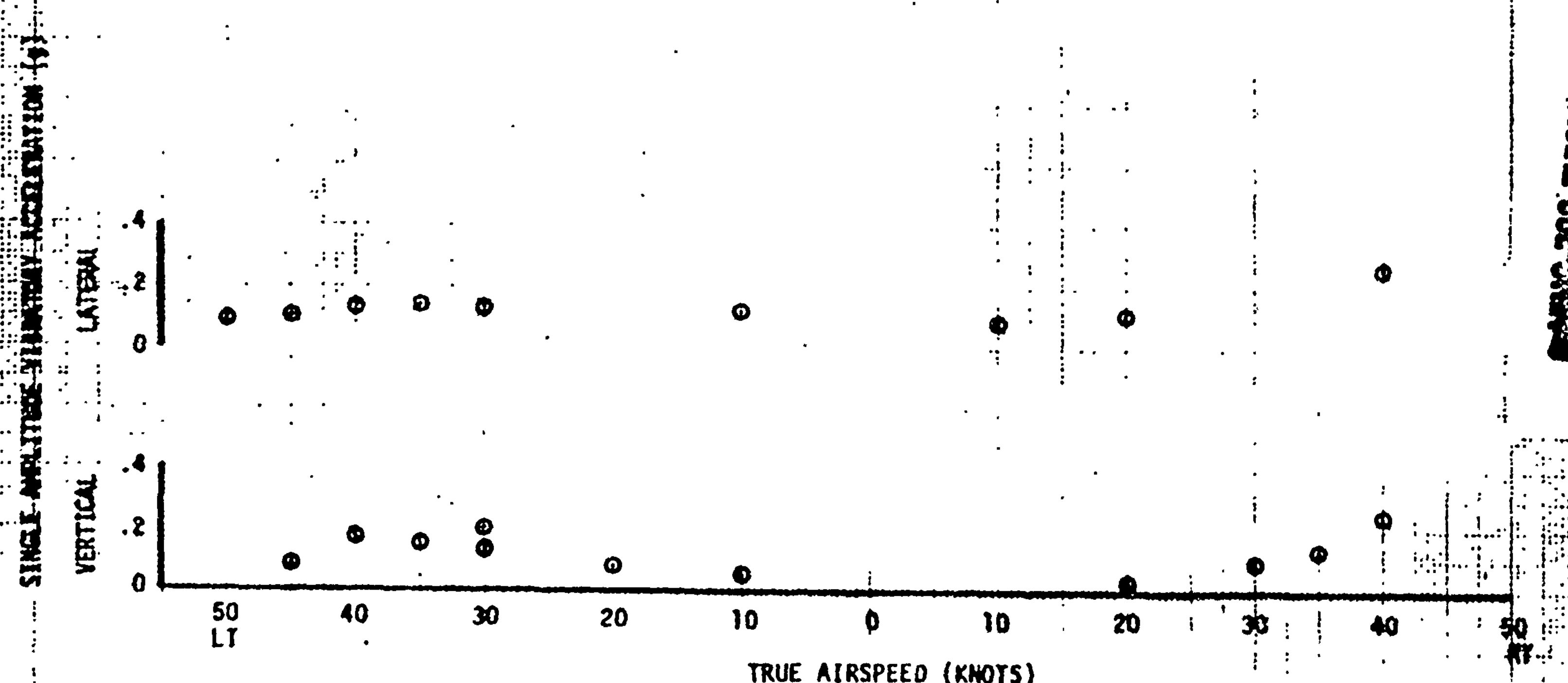


FIGURE I38
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
PILOT STATION
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
17460	359.5(AFT)	0.4(RT)	7000	18.0	258	130	TURNING FLIGHT

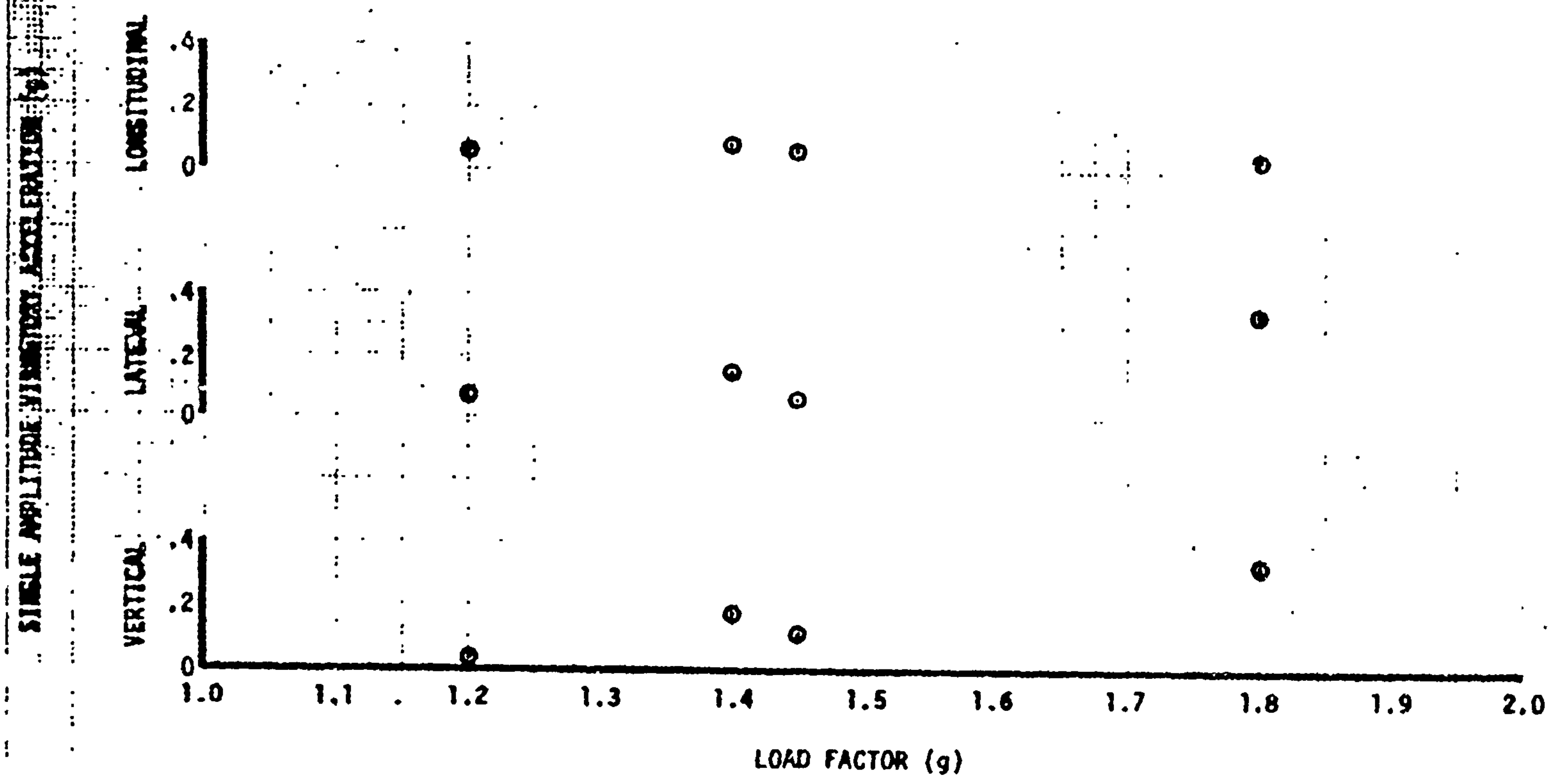


FIGURE 132
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
COPILOT STATION
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
17450	359.5(AFT)	0.4(RT)	7000	18.0	258	130	TURNING FLIGHT

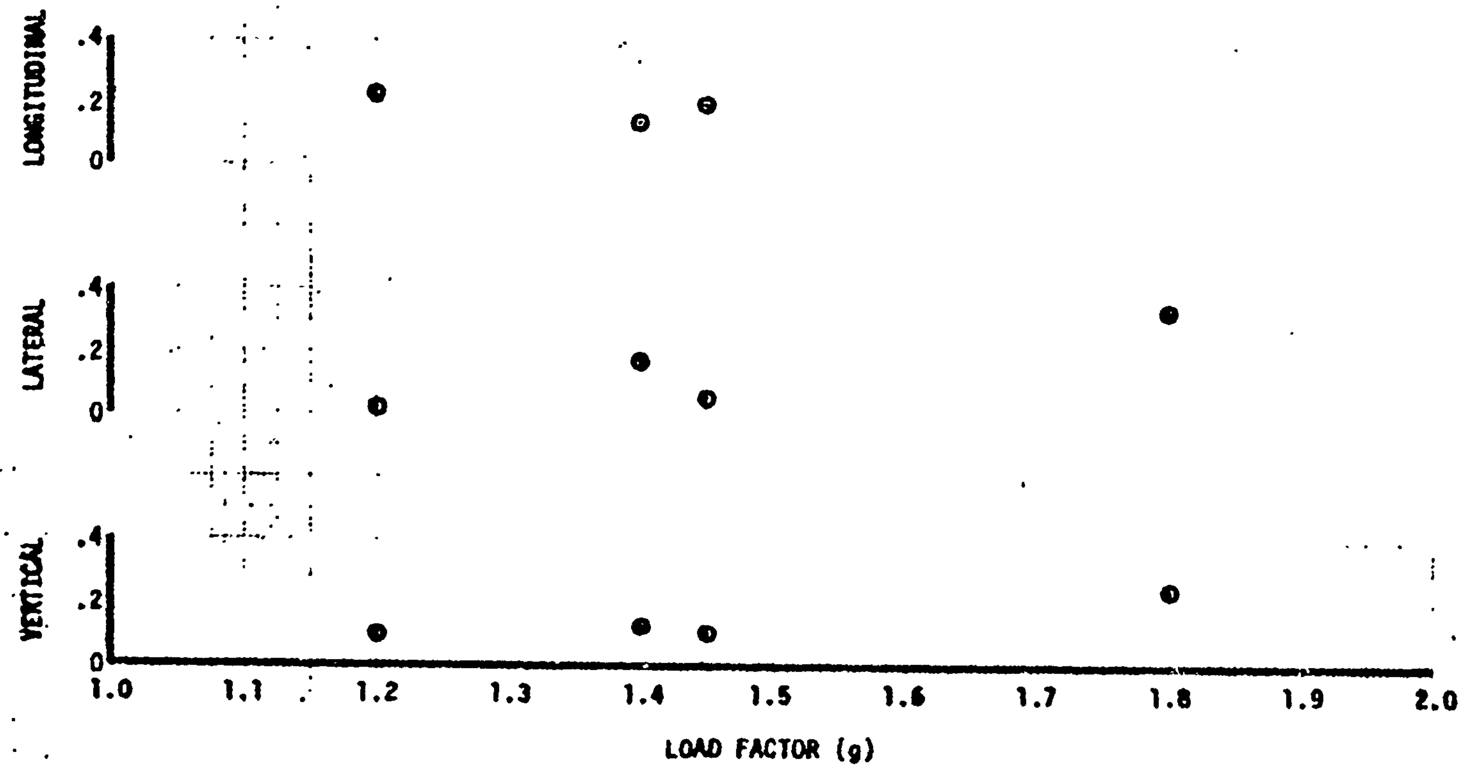


FIGURE 133
VIBRATION CHARACTERISITCS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
CENTER OF GRAVITY
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
17460	359.5(AFT)	0.4(RT)	7000	18.0	258	130	TURNING FLIGHT

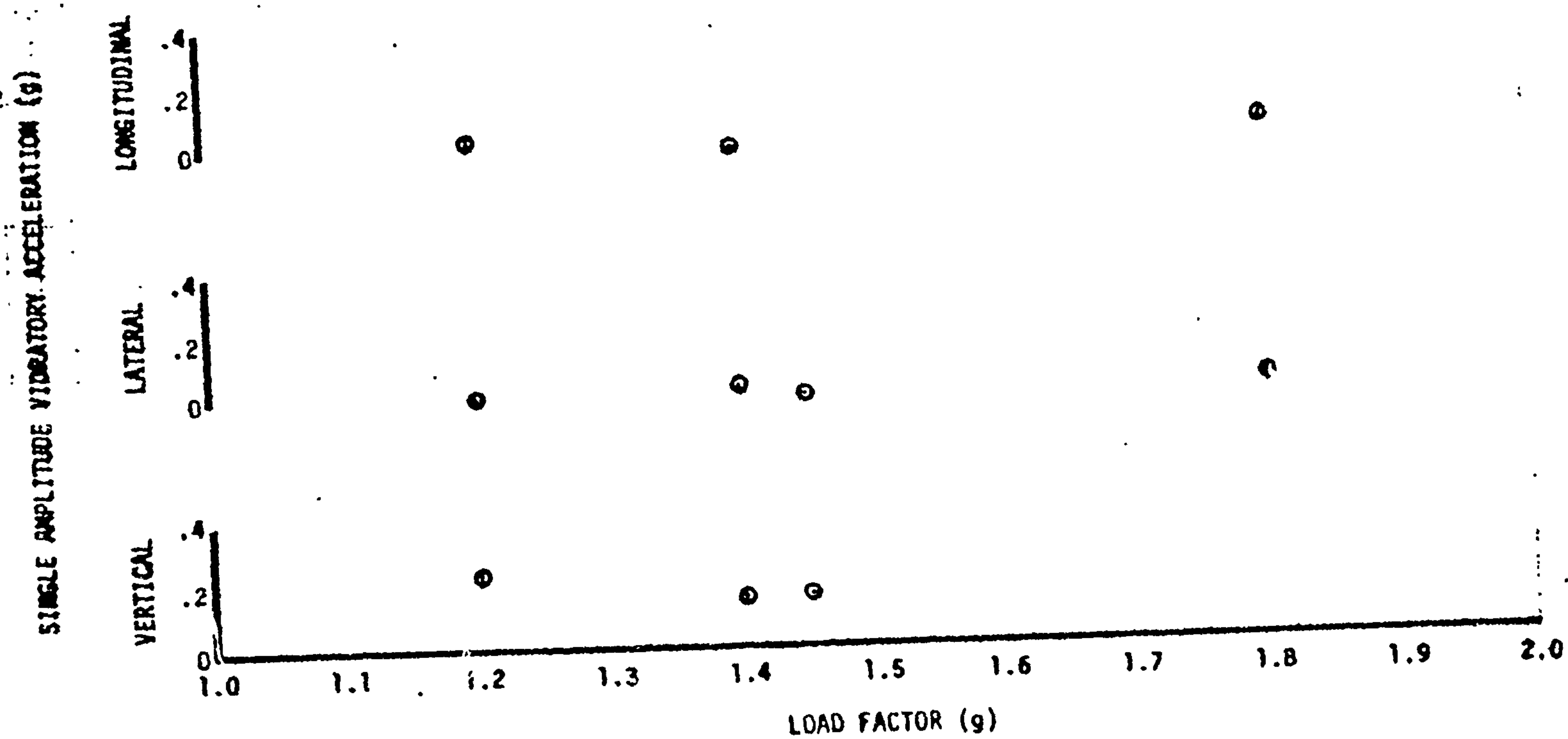


FIGURE 134
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
INSTRUMENT PANEL (RIGHT SIDE)
4/REV (17.2 Hz)

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
17460	359.5(AFT)	0.4(RT)	7000	18.0	258	130	TURNING FLIGHT

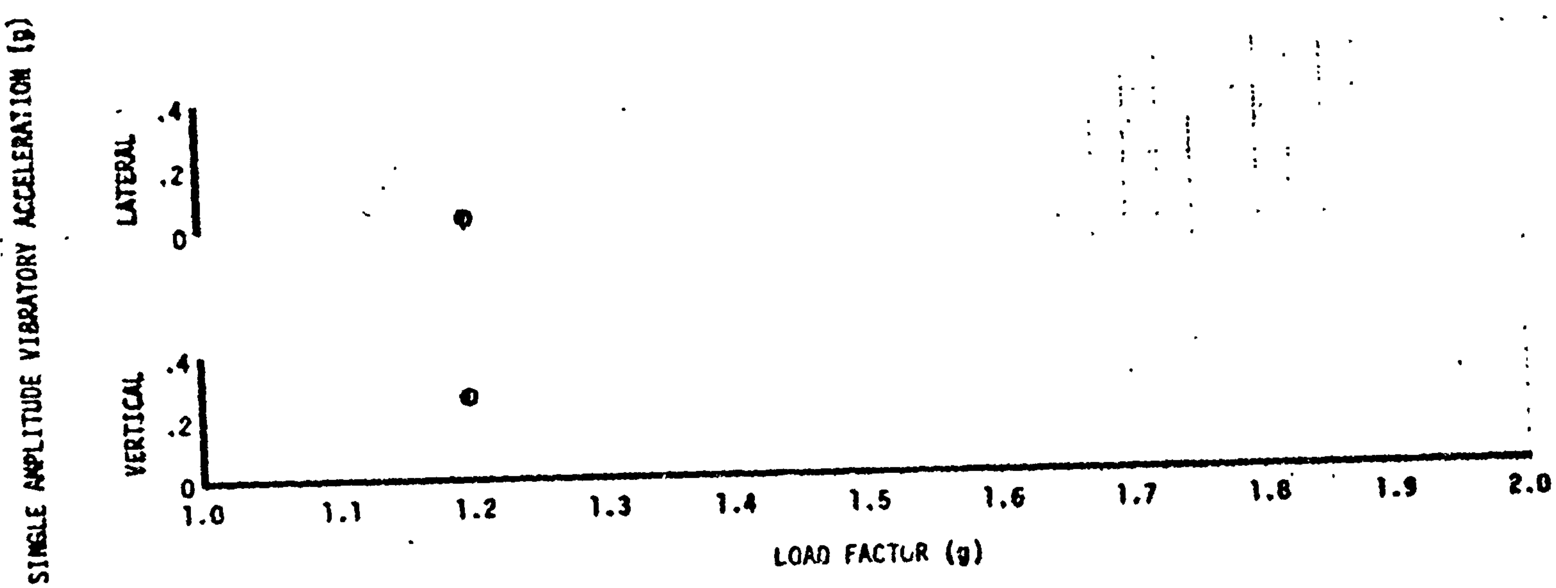


FIGURE 135
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
PILOT STATION
4/REV (17.2 Hz)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
○	20160	359.3(AFT)	0.5(RT)	6320	17.0	259	105	LT TURN
△	19500	359.6(AFT)	0.5(RT)	6600	17.0	266	105	RT TURN

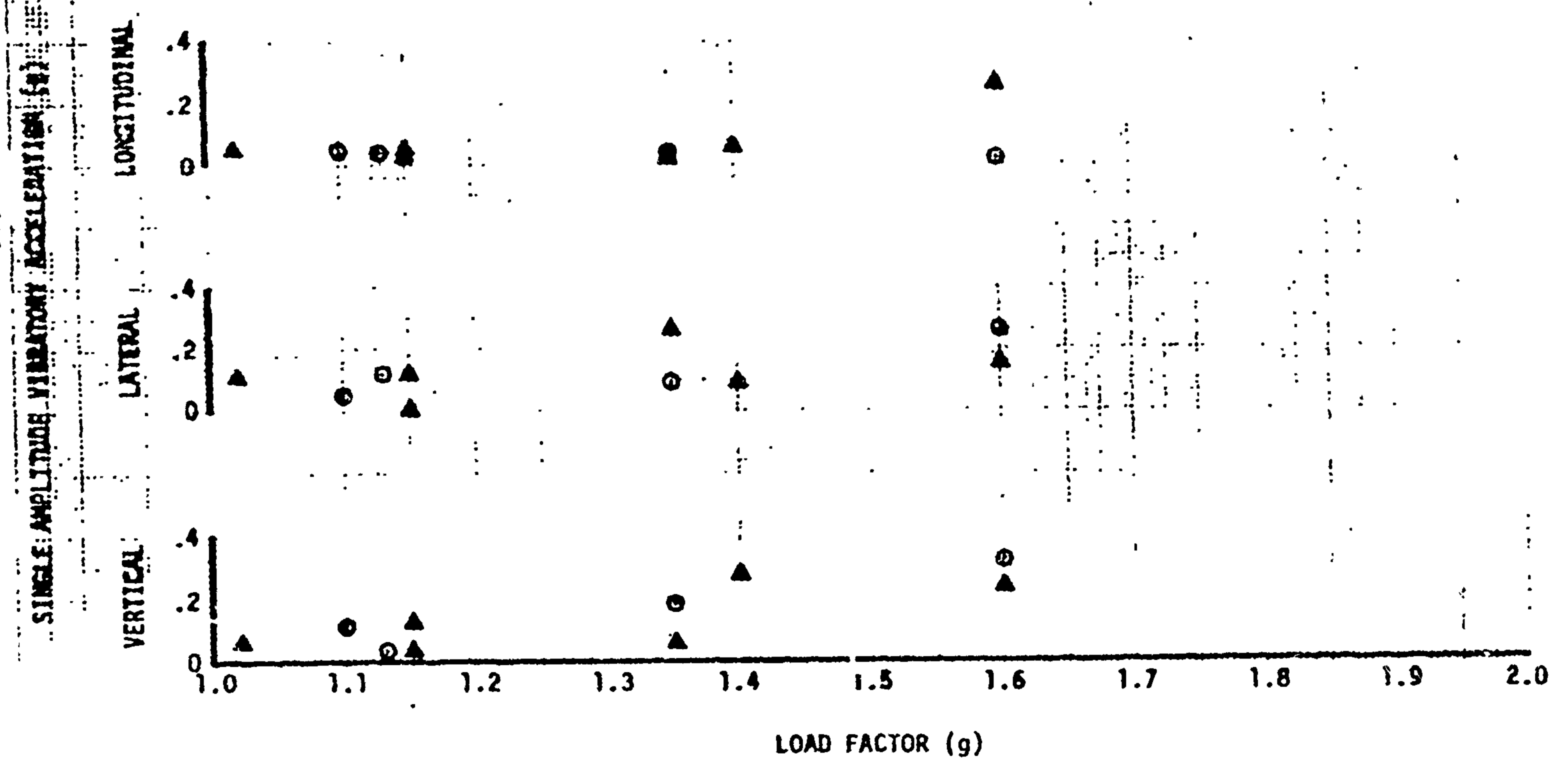
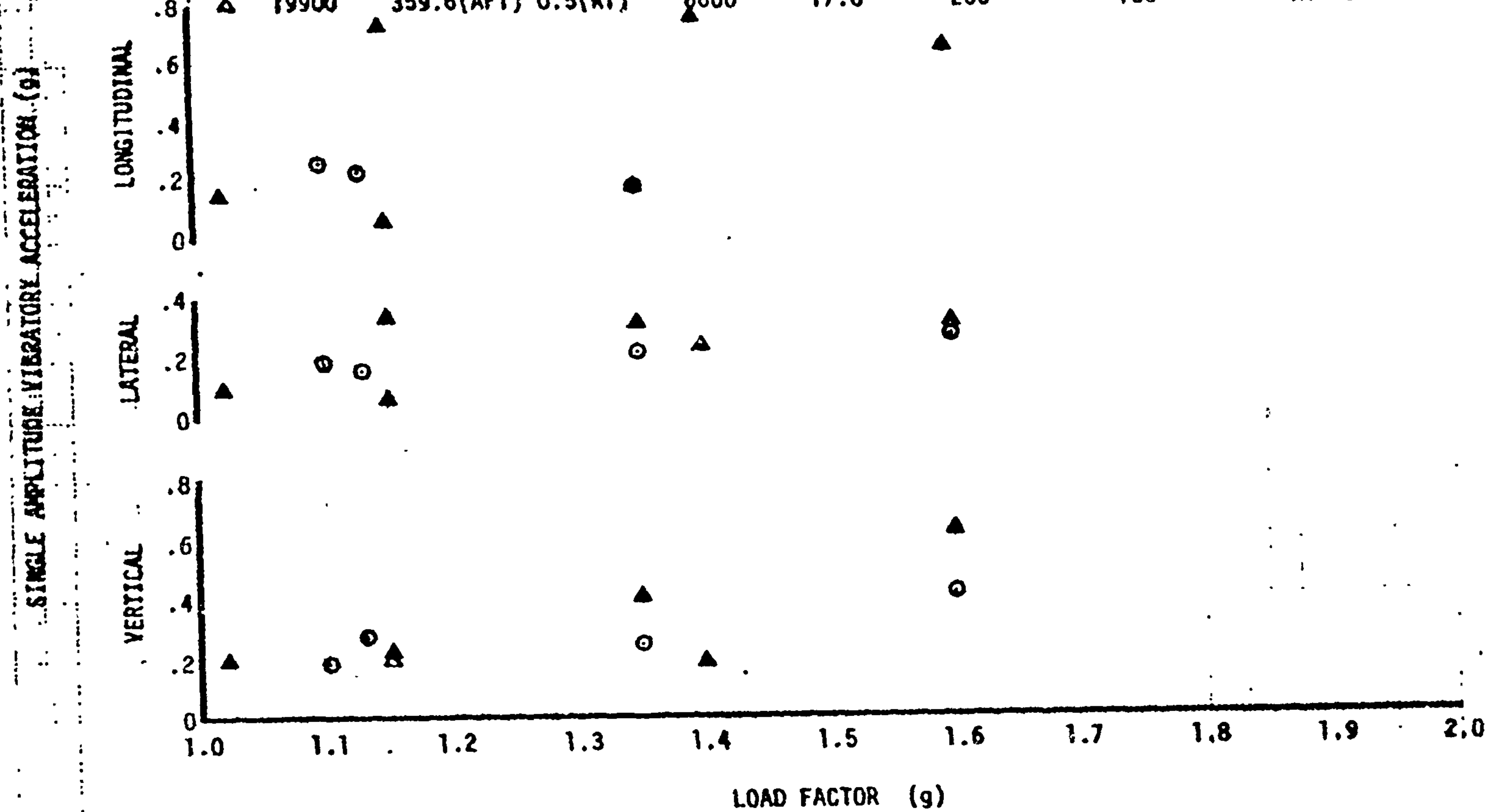


FIGURE 136
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
COPILOT STATION
4/REV (17.2 Hz)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
O	20160	359.3(AFT)	0.5(RT)	6320	17.0	259	105	LT TURN
△	19900	359.6(AFT)	0.5(RT)	6600	17.0	266	105	RT TURN



[redacted]

FIGURE 137
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
CENTER OF GRAVITY
4/REV (17.2 Hz)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
○	20160	359.3(AFT)	0.5(RT)	6320	17.0	259	105	LT TURN
△	19900	359.6(AFT)	0.5(RT)	6600	17.0	266	105	RT TURN

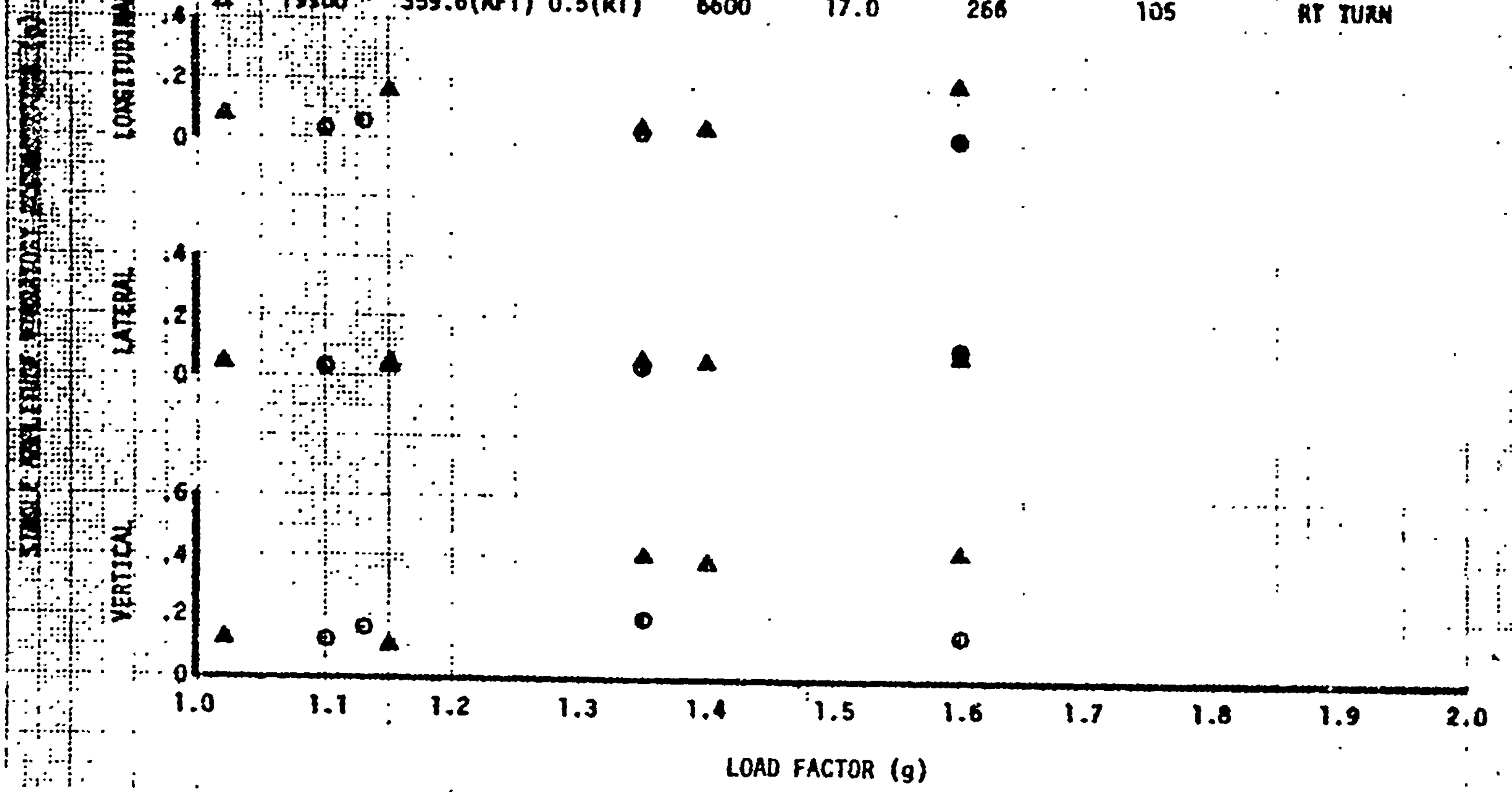


FIGURE 138
VIBRATION CHARACTERISTICS IN TURNING FLIGHT
YUH-60A USA S/N 73-21651
INSTRUMENT PANEL (RIGHT SIDE)
4/REV (17.2 Hz)

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION
○	20160	359.3(AFT)	0.5(RT)	6320	17.0	259	105	LT TURN
△	19900	359.6(AFT)	0.5(RT)	6600	17.0	266	105	RT TURN

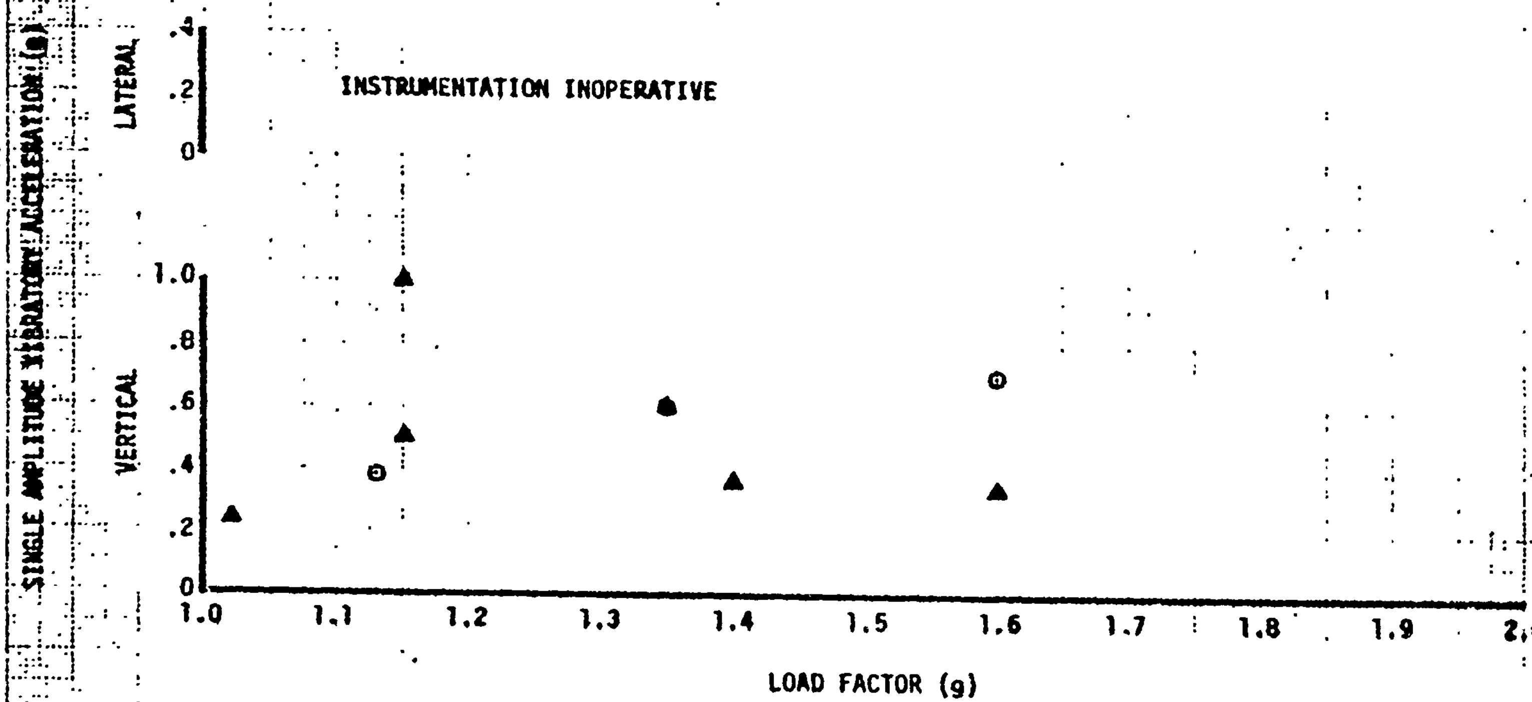
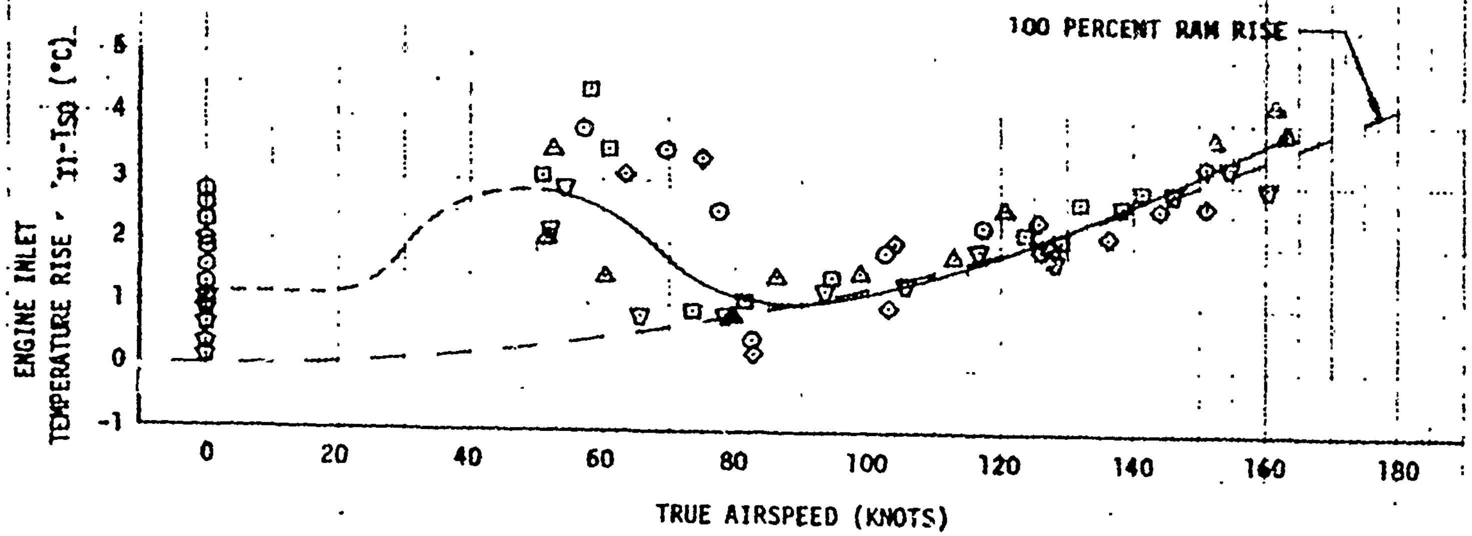


FIGURE 139
ENGINE INLET TEMPERATURE RISE
YUH-60A USA S/N 73-21651
LEFT ENGINE

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
⊙	19600	347.6(FWD)	0.5(RT)	11600	1.0	258	LEVEL
⊡	19540	347.6(FWD)	0.5(RT)	9080	7.5	258	LEVEL
◇	19400	349.1(FWD)	0.4(RT)	6420	11.5	258	LEVEL
▽	16840	349.4(FWD)	0.4(RT)	8040	6.0	258	LEVEL
△	15940	347.6(FWD)	0.4(RT)	2580	14.0	258	LEVEL
○	18000	352.0(MID)	0.5(RT)	2570	11.0	258	HOVER
☆	16500	352.0(MID)	0.5(RT)	5550	20.0	258	HOVER



[redacted]

FIGURE 140
ENGINE INLET TEMPERATURE RISE
YUH-60A USA S/N 73-21651
RIGHT ENGINE

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	19600	347.6(FWD)	0.5(RT)	11600	1.0	258	LEVEL
□	19540	347.6(FWD)	0.5(RT)	9080	7.5	258	LEVEL
◇	19400	349.1(FWD)	0.4(RT)	6420	11.5	258	LEVEL
▽	16840	349.4(FWD)	0.4(RT)	8040	6.0	258	LEVEL
△	15940	347.6(FWD)	0.4(RT)	2580	14.0	258	LEVEL
○	18000	352.0(MID)	0.5(RT)	2570	11.0	258	HOVER
⬠	16500	352.0(MID)	0.5(RT)	5550	20.0	258	HOVER

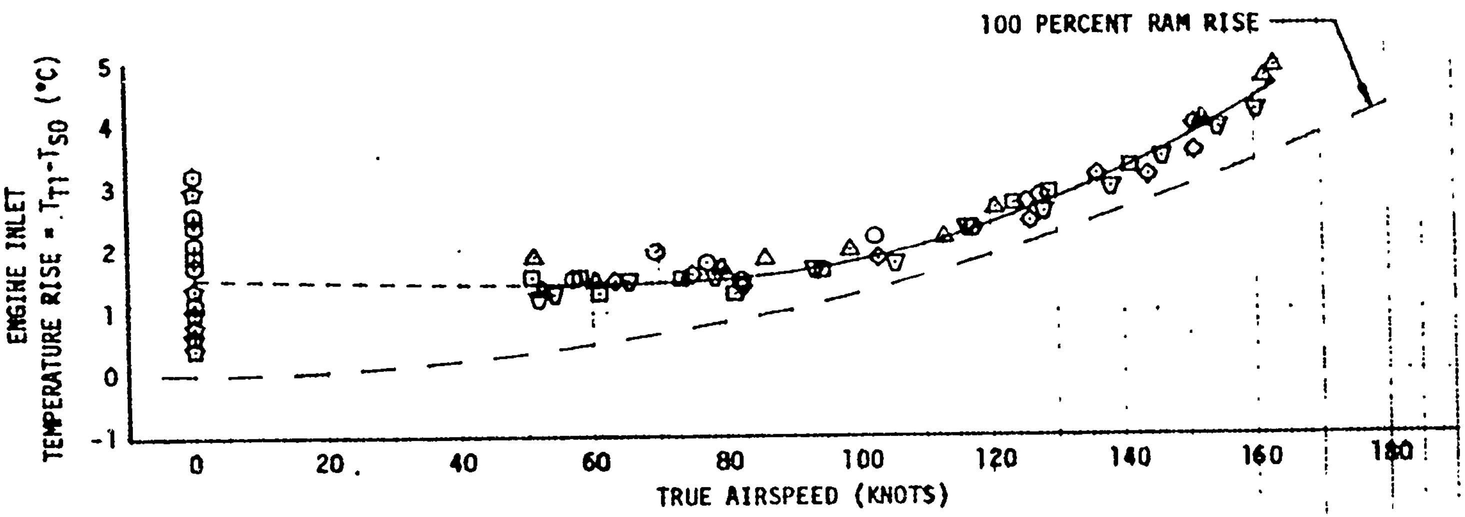


FIGURE 141
ENGINE INLET PRESSURE RATIOS
YUH-60A USA S/N 73-21651
LEFT ENGINE

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG H_D (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	17900	347.2(FWD)	0.4(RT)	7620	22.0	258	LEVEL
□	16140	347.5(FWD)	0.5(RT)	7100	27.5	258	LEVEL
◇	15640	347.2(FWD)	0.5(RT)	11300	18.5	258	LEVEL
△	15700	347.2(FWD)	0.5(RT)	9000	24.0	258	DIVE
○	16340	347.5(FWD)	0.5(RT)	7040	29.0	258	HOVER
◇	15430	347.1(FWD)	0.5(RT)	7000	31.5	258	HOVER

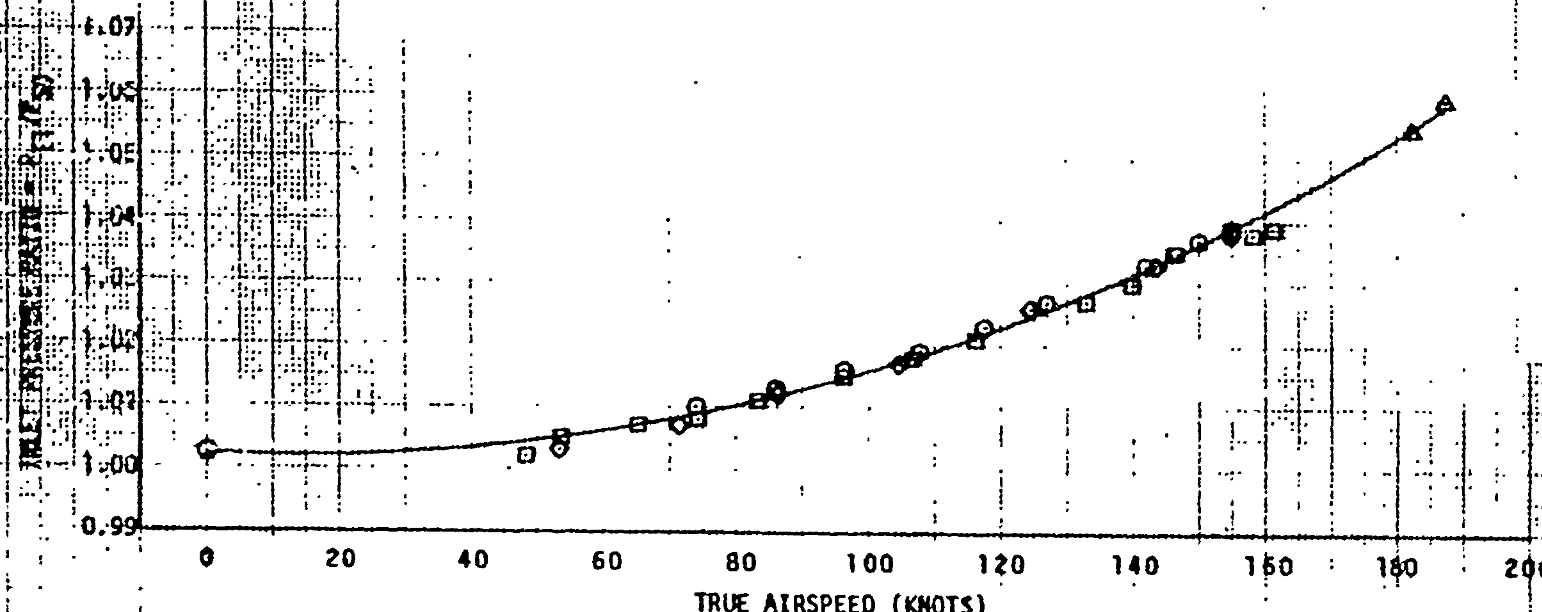


FIGURE 142
ENGINE INLET PRESSURE RATIOS
YUH-60A USA S/N 73-21651
RIGHT ENGINE

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG H_{D} (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	17900	347.2(FWD)	0.4(RT)	7620	22.0	258	LEVEL
□	16140	347.5(FWD)	0.5(RT)	7100	27.5	258	LEVEL
◇	15640	347.2(FWD)	0.5(RT)	11300	18.5	258	LEVEL
△	15700	347.2(FWD)	0.5(RT)	9000	24.0	258	DIVE
○	16340	347.5(FWD)	0.5(RT)	7040	29.0	258	HOVER
☆	15430	347.1(FWD)	0.5(RT)	7000	31.5	258	HOVER

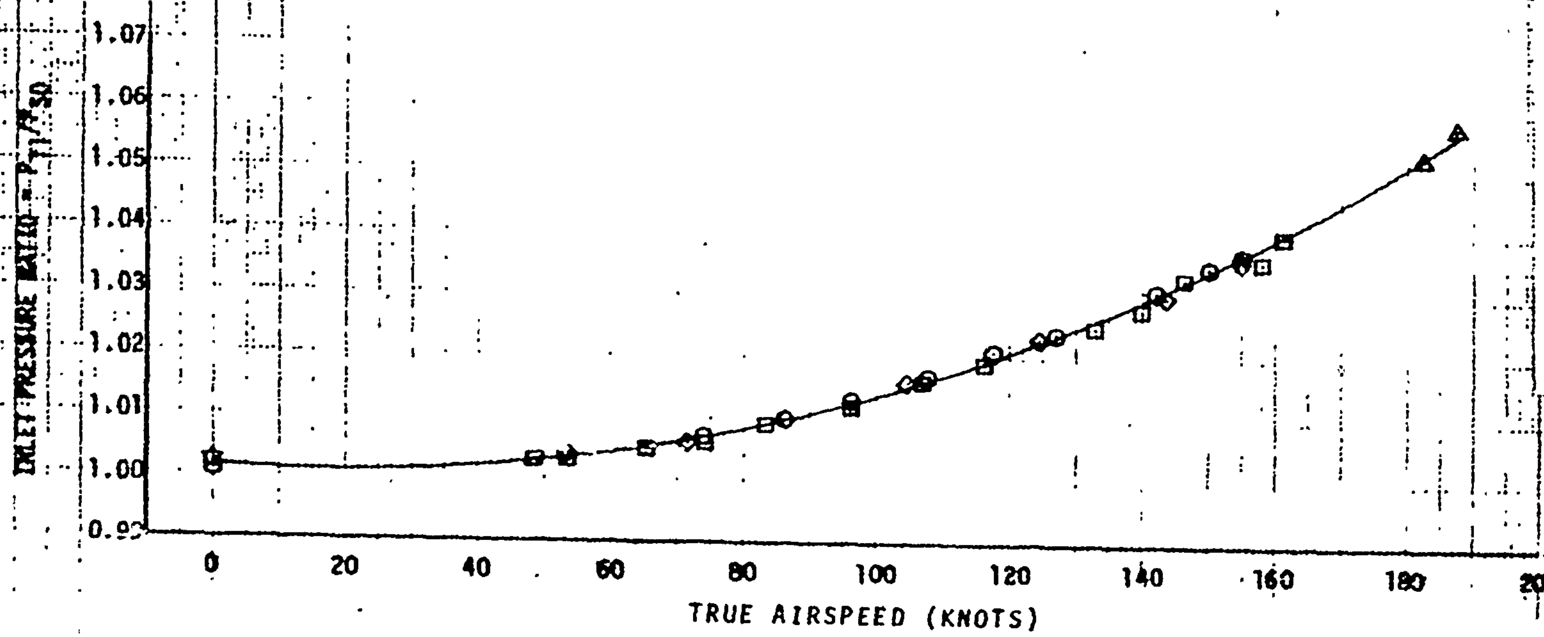
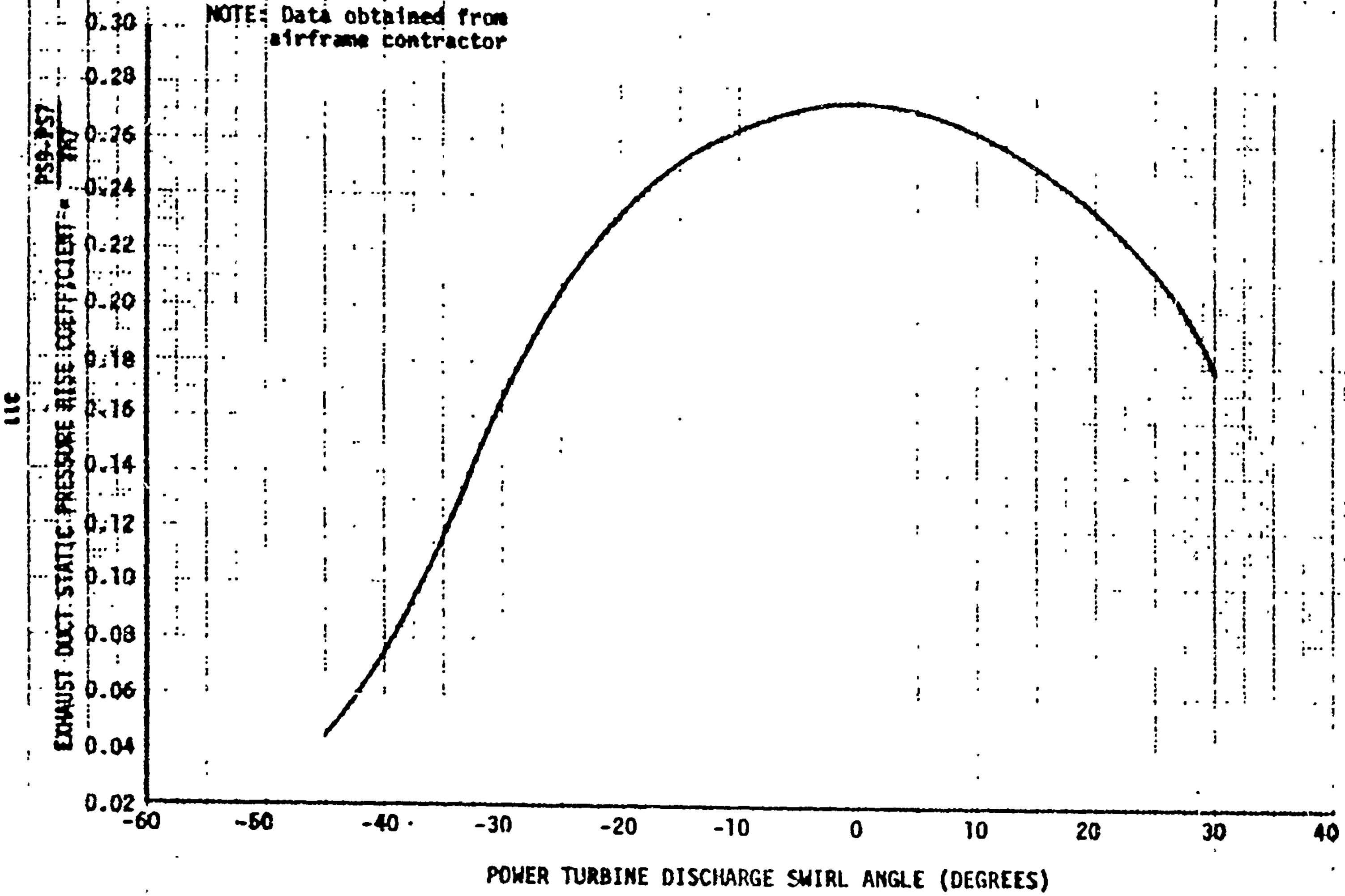


FIGURE 143

EXHAUST SYSTEM CHARACTERISTICS

YUH-60A YT700-GE-700



[redacted]

FIGURE 144

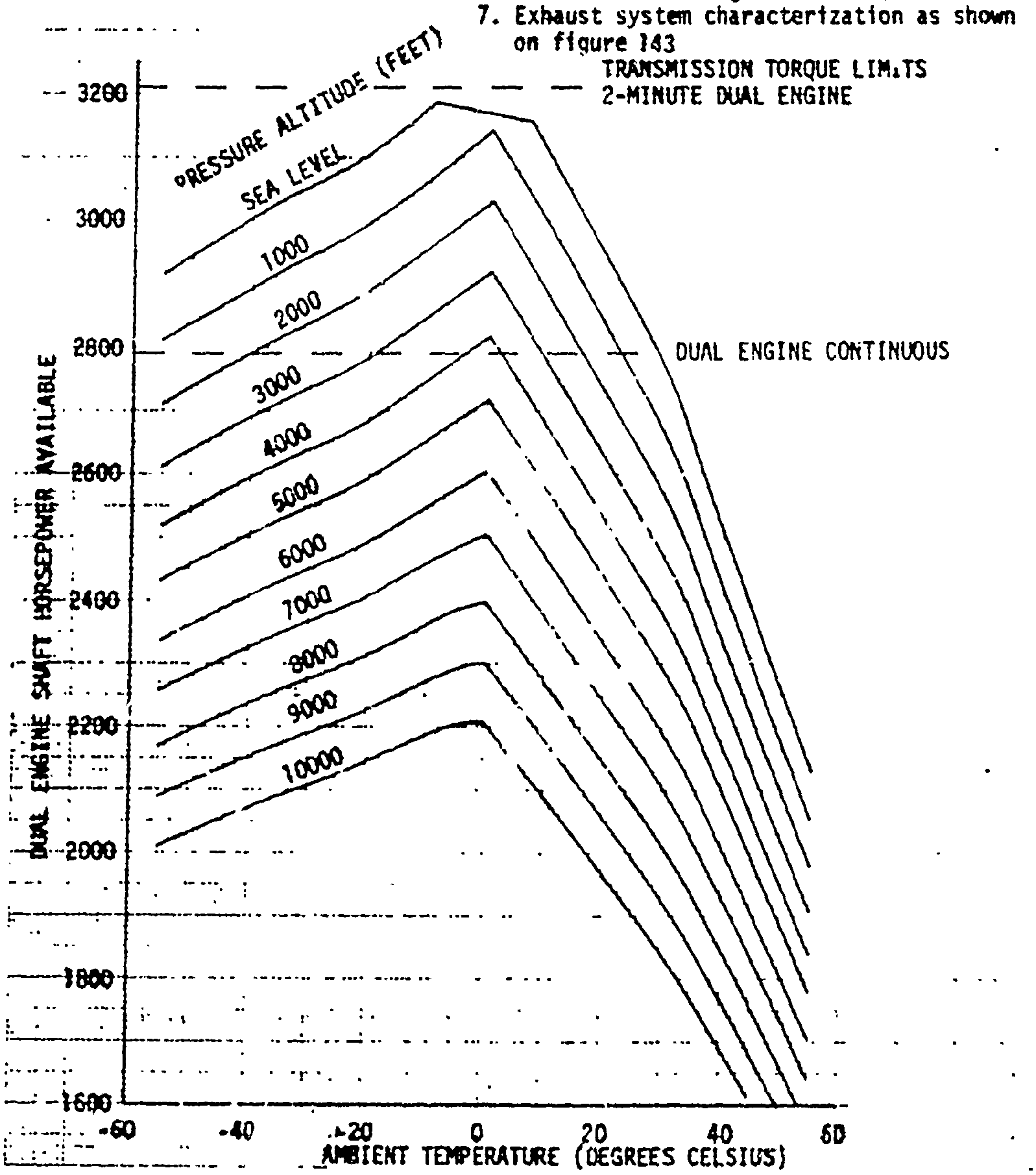
INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YUH-60A USA S/N 73-21651 YT700-GE-700 BOTH ENGINES

19,602 OUTPUT SHAFT (258 ROTOR) RPM ZERO KNOTS TRUE AIRSPEED

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Engine inlet temperature rise
   Left engine = 1.15°C, Right engine = 1.53°C
2. Engine inlet pressure ratios
   Left engine = 1.0024, Right engine = 1.0017
3. Gas producer horsepower extracted = zero
4. Customer bleed air = zero
5. Engine anti-ice off
6. Fuel lower heating value = 18,300 BTU/lb
7. Exhaust system characterization as shown on figure 143



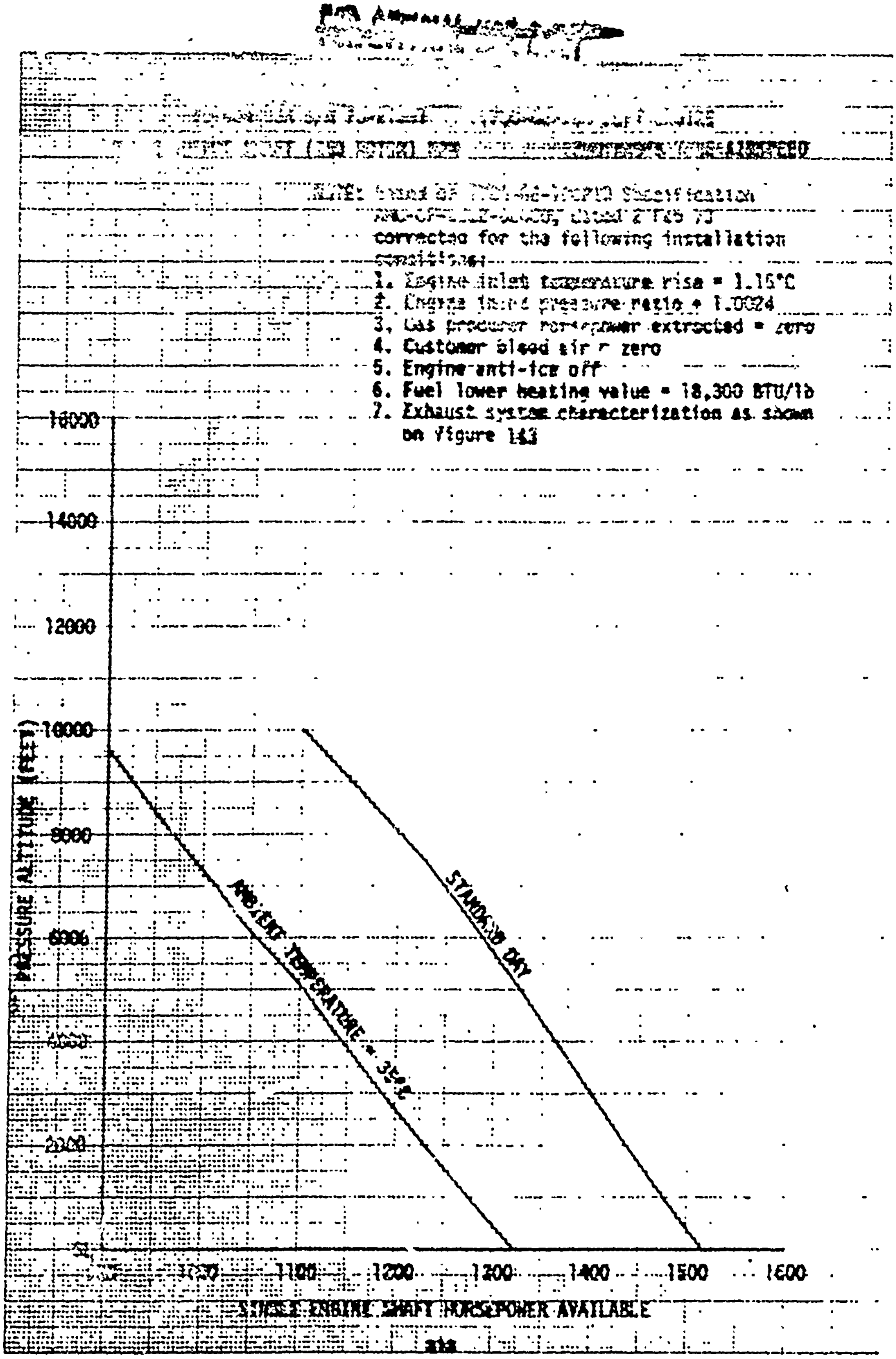

corrected for the following installation
1. Engine inlet temperature rise = 1.15°C
2. Engine inlet pressure ratio = 1.0024
3. Gas producer horsepower extracted = zero
4. Customer bleed air = zero
5. Engine anti-ice off
6. Fuel lower heating value = 18,300 BTU/lb
7. Exhaust system characterization as shown on Figure 143
16000
14000
12000
10000
8000
6000
4000
2000
PRESSURE ALTITUDE (FEET)
STANDARD DAY
AMBIENT TEMPERATURE = 35°C
1100
1200
1400
1500
1600
SINGLE ENGINE SHAFT HORSEPOWER AVAILABLE

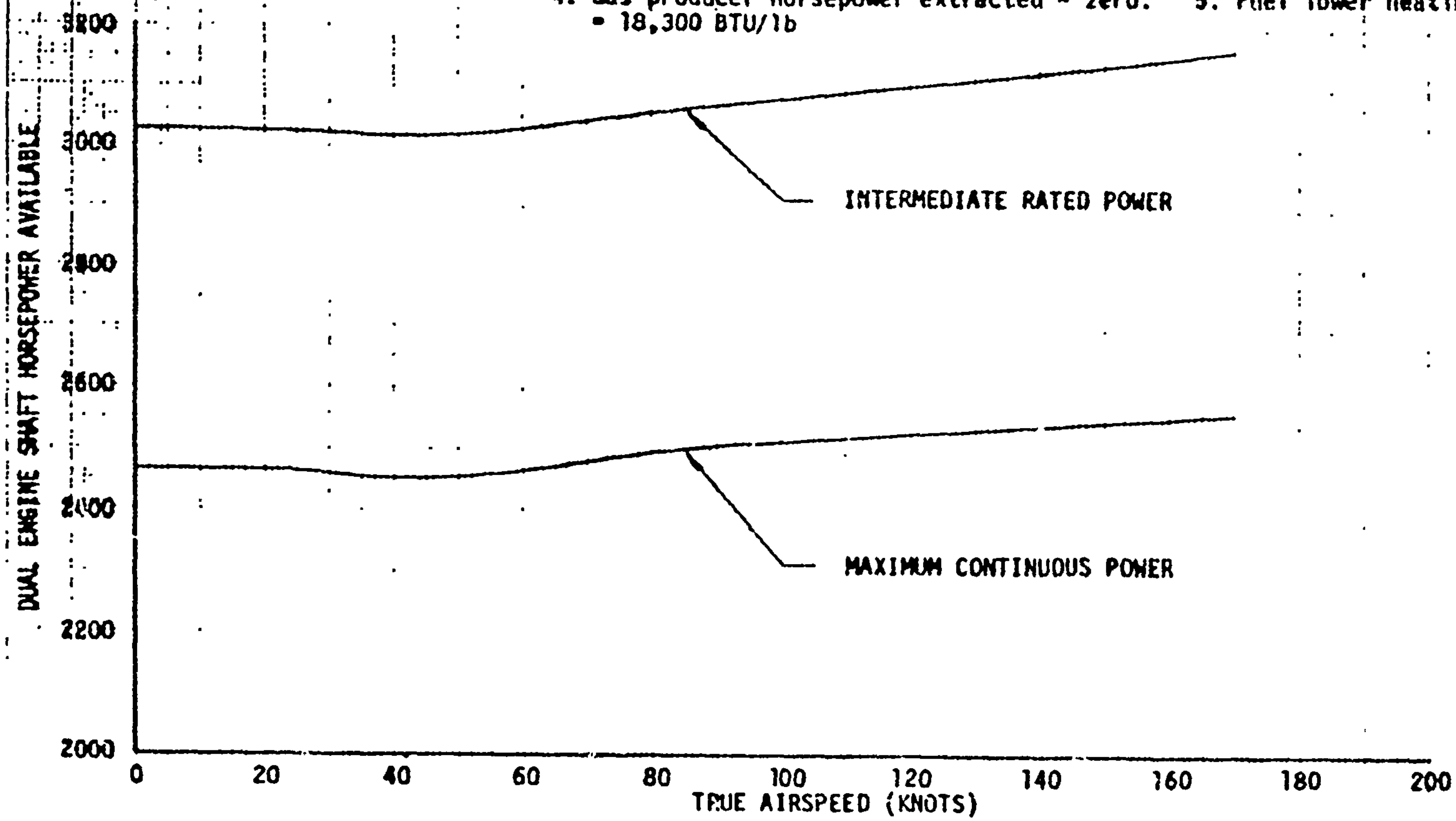
FIGURE 146

POWER AVAILABLE VERSUS AIRSPEED

YUH-60A USA S/N 73-21651 YT700-GE-700 BOTH ENGINES

19,602 OUTPUT SHAFT (258 ROTOR) RPM STANDARD DAY PRESSURE ALTITUDE = SEA LEVEL

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:
1. Inlet and exhaust characteristics obtained from figures 139 through 143
2. Customer bleed air = zero. 3. Engine anti-ice off
4. Gas producer horsepower extracted = zero. 5. Fuel lower heating value = 18,300 BTU/lb



[redacted]

FIGURE 147

POWER AVAILABLE VERSUS AIRSPEED

YUH-60A USA S/N 73-21651 YT700-GE-700 BOTH ENGINES

19,602 OUTPUT SHAFT (258 ROTOR) RPM AMBIENT TEMPERATURE = 35°C PRESSURE ALTITUDE = 4000 FEET

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:
1. Inlet and exhaust characteristics obtained from figures 139 through 143
2. Customer bleed air = zero 3. Engine anti-ice off
4. Gas producer horsepower extracted = zero 5. Fuel lower heating value = 18,300 BTU/lb

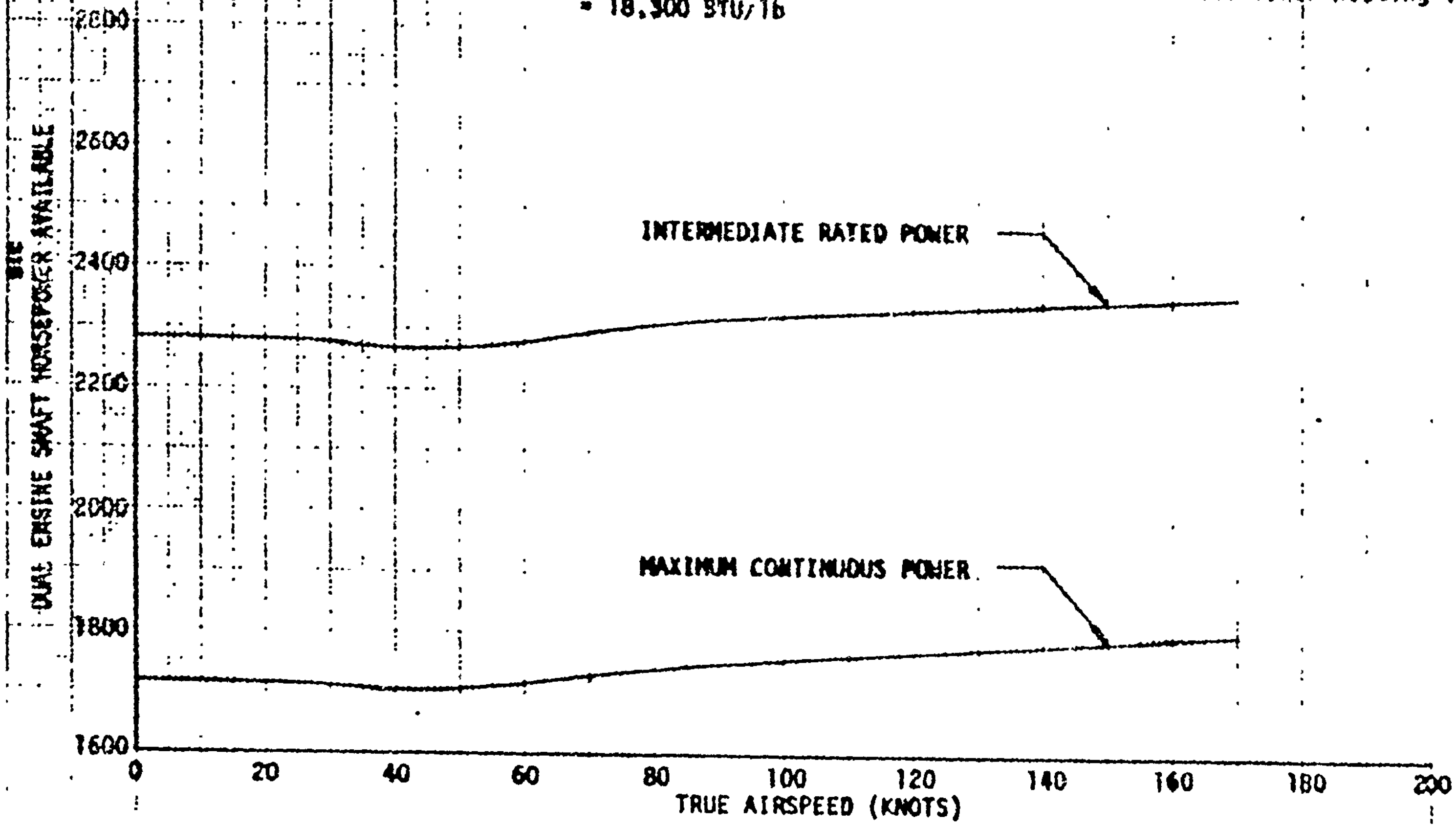


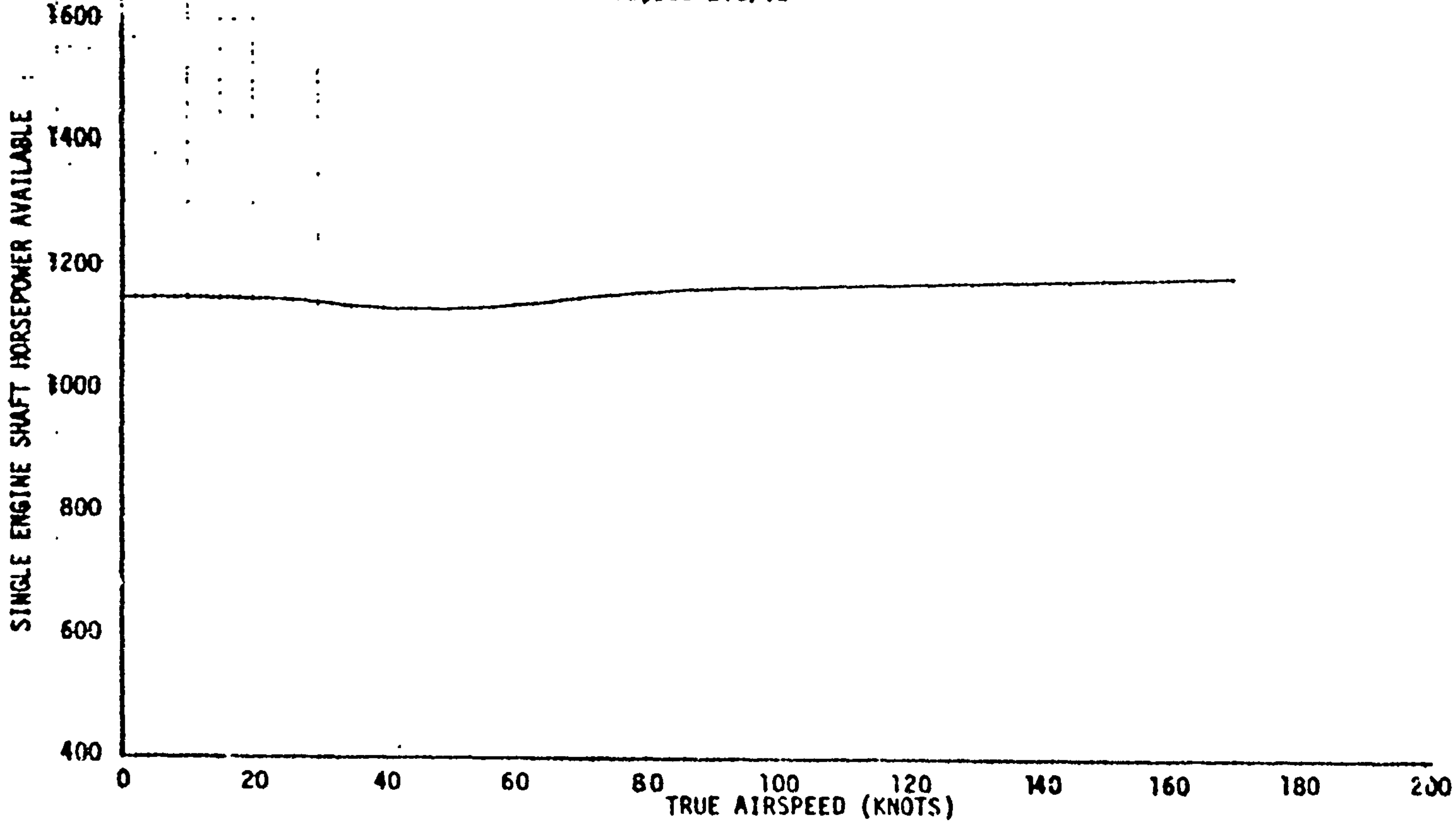
FIGURE 148

INTERMEDIATE (30-MINUTE LIMIT) POWER VERSUS AIRSPEED

YUH-60A USA S/N 73-21651 YT700-GE-700 LEFT ENGINE

19,602 OUTPUT SHAFT (258 ROTOR) RPM AMBIENT TEMPERATURE = 35°C PRESSURE ALTITUDE = 4000 FEET

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:
1. Inlet and exhaust characteristics obtained from figures 139, 141, and 143
2. Customer bleed air = zero. 3. Engine anti-ice off
4. Gas producer horsepower extracted = zero. 5. Fuel lower heating value = 18,300 BTU/lb



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FOR OFFICIAL USE ONLY

FIGURE 149

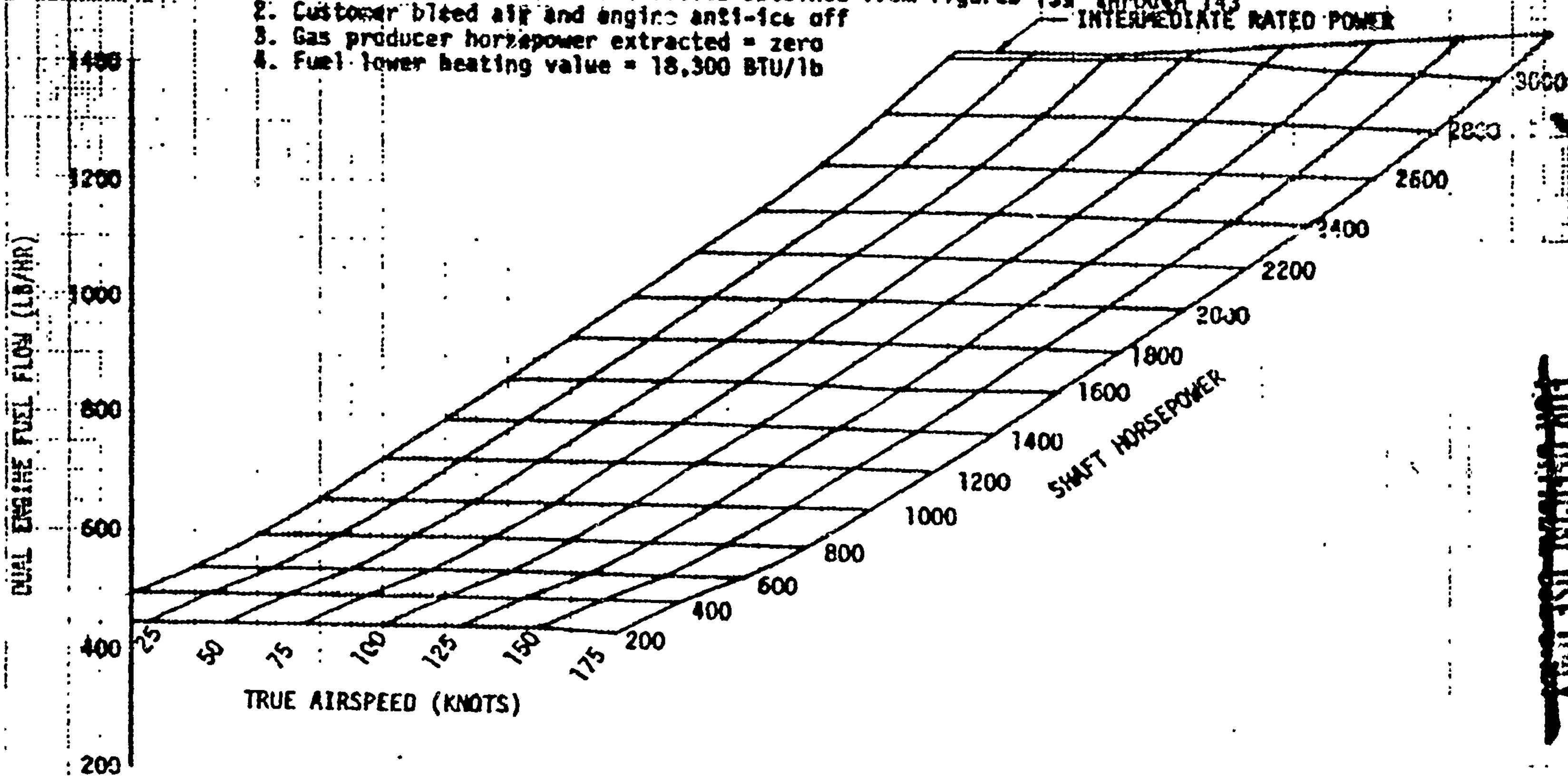
DUAL ENGINE SPECIFICATION FUEL FLOW

YUH-60A USA S/N 73-21651 YT700-GE-700 BOTH ENGINES

19,602 OUTPUT SHAFT (258 ROTOR) RPM STANDARD DAY PRESSURE ALTITUDE = SEA LEVEL

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Inlet and exhaust characteristics obtained from figures 139 through 143
2. Customer bleed air and engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb



[redacted]

FIGURE 150

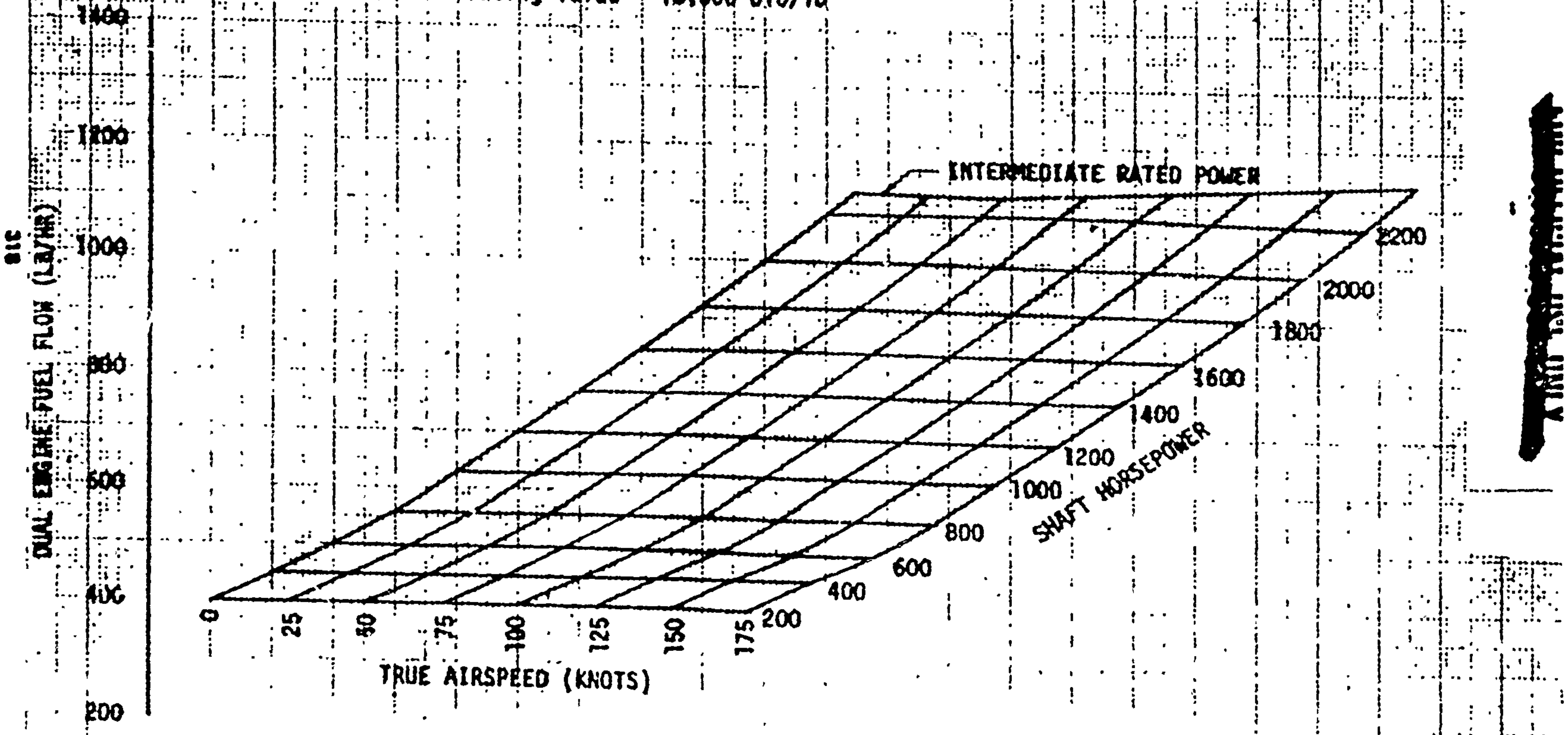
DUAL ENGINE SPECIFICATION FUEL FLOW

YUH-60A USA S/N 73-21651 YT700-GE-700 BOTH ENGINES

19,602 OUTPUT SHAFT (258 ROTOR) RPM AMBIENT TEMPERATURE = 35°C PRESSURE ALTITUDE = 4000 FEET

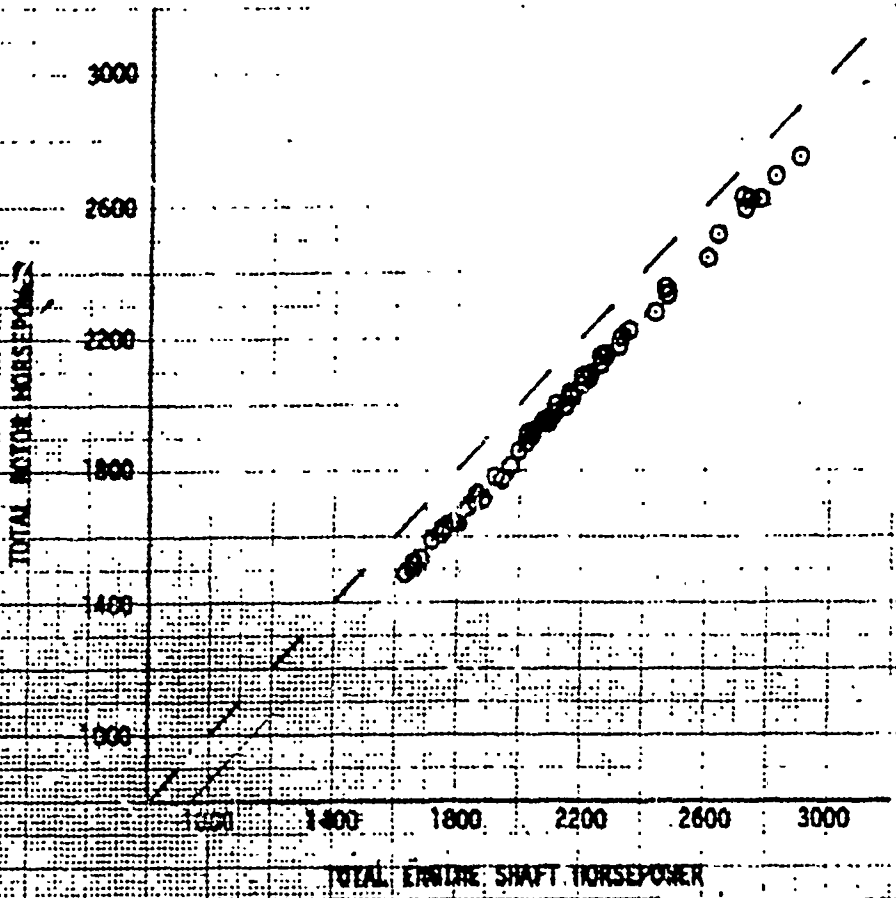
NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Inlet and exhaust characteristics obtained from figures 139 through 143
2. Customer bleed air and engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb



T700-GE-700 ENGINES
LEFT ENGINE S/N 207211
RIGHT ENGINE S/N 207226

NOTE: DATA DEPICTED WAS OBTAINED DURING HOVERING FLIGHT



[redacted]

FIGURE 152

REFERRED POWER AND GAS TEMPERATURE

AIRCRAFT YUH-60A S/N 73-21651 ENGINE YT700-GE-700 S/N 207211

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20,000	SL	15.0	OFF	0		PID SPEC AMC-CP-2222-02000
—●—	20,000	220	15.5	OFF	0	0	TEST CELL CALIB, 9 Feb 76
■	19,000	190	13.6	OFF	0	0	TEST CELL CALIB, 9 Feb 76
◆	21,000	140	15.0	OFF	0	0	TEST CELL CALIB, 9 Feb 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

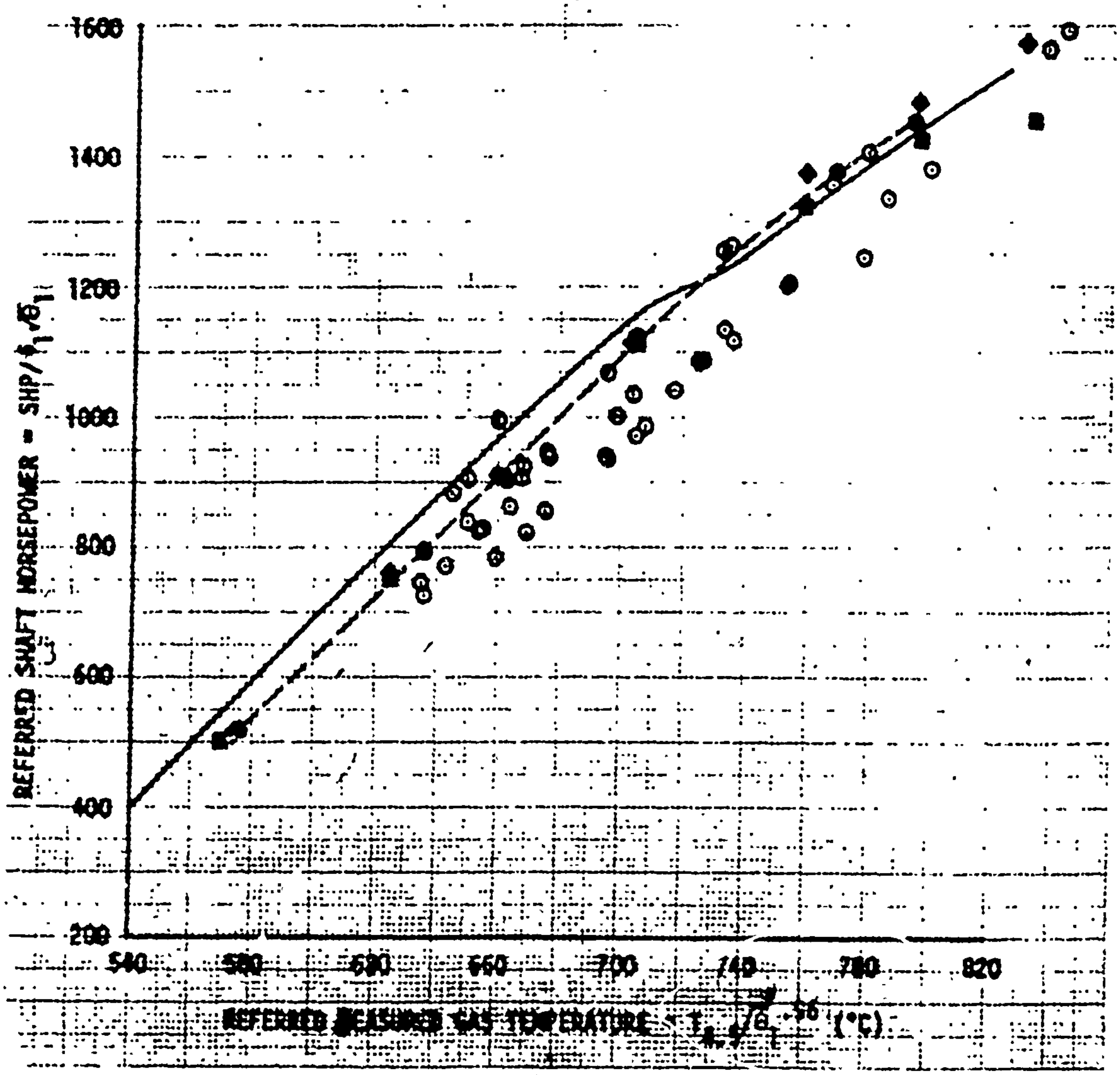


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FIGURE [illegible]

AIRCRAFT [illegible] ENGINE [illegible]-700 S/N 207211

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRESS ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI-ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	[illegible]	SL	15.0	OFF	0		PIG SPEC AMC-CP-2222-02000
—◆—	[illegible]	220	15.5	OFF	0	8	TEST CELL CALIB, 9 Feb 76
■	[illegible]	180	13.5	OFF	0	8	TEST CELL CALIB, 9 Feb 76
◆	[illegible]	150	15.0	OFF	0	8	TEST CELL CALIB, 9 Feb 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

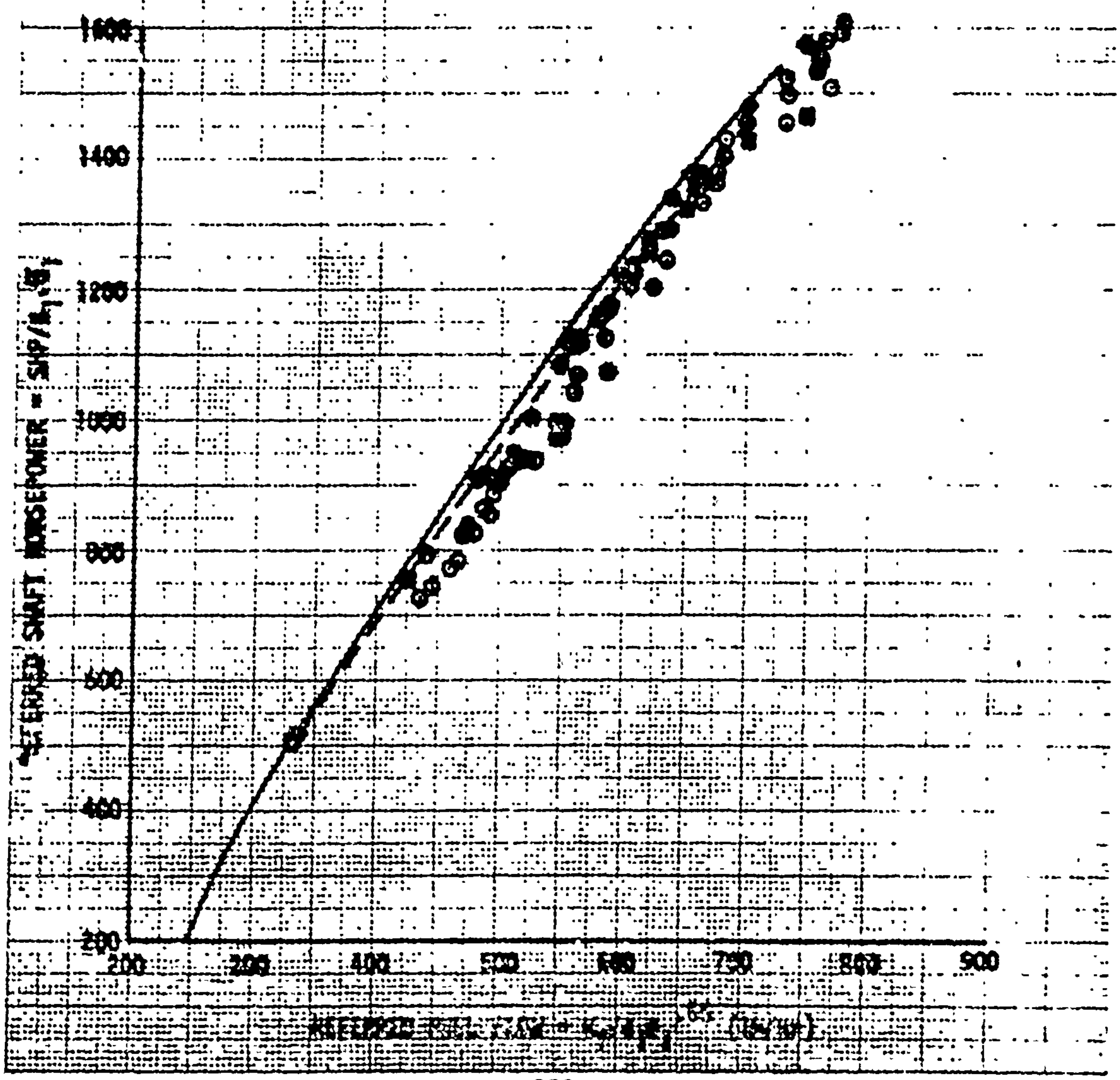


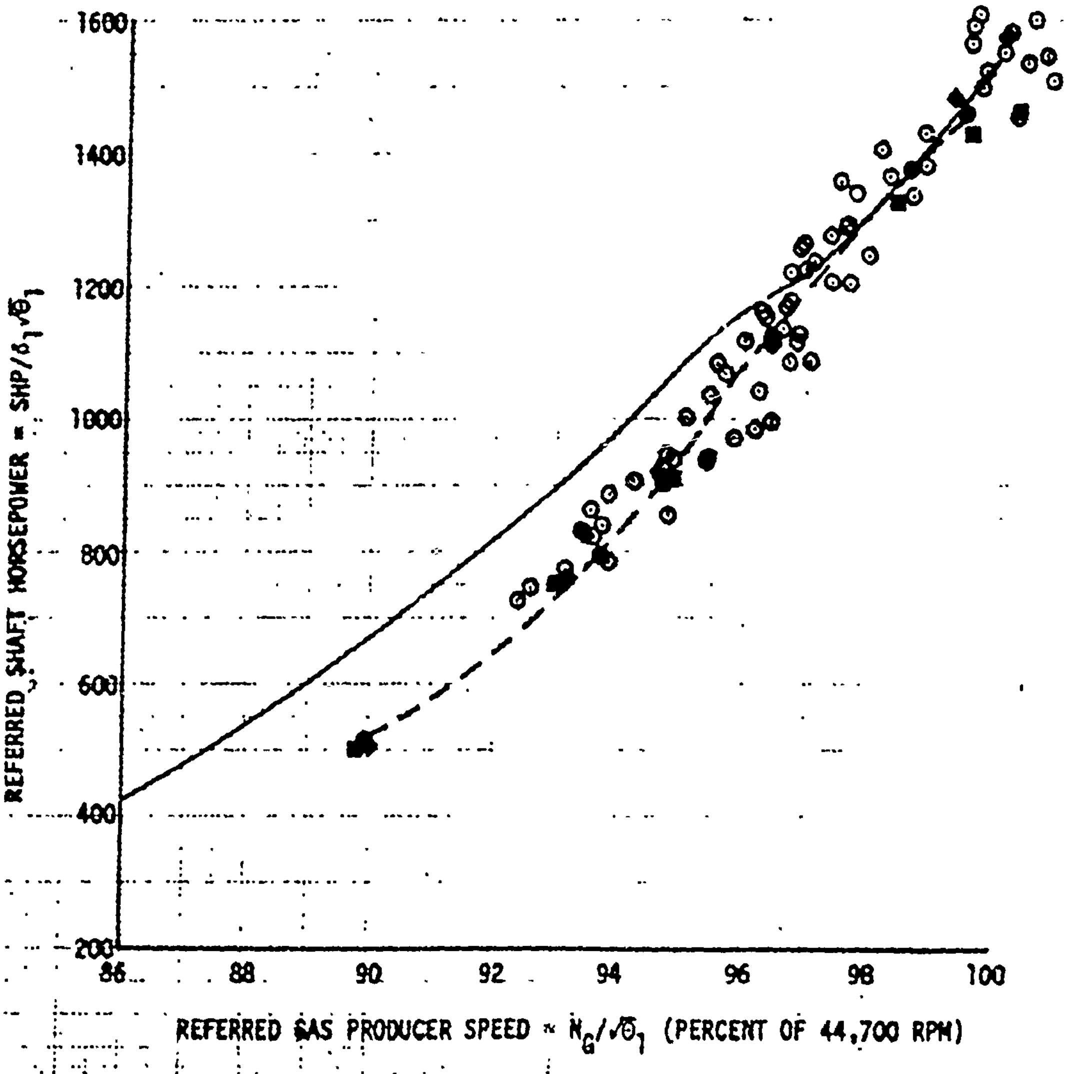
FIGURE 154

REFERRED POWER AND GAS PRODUCER SPEED

AIRCRAFT YUH-60A S/N 73-21651 ENGINE T700-GE-700 S/N 207211

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI-ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20,000	SL	15.0	OFF	0		PID SPEC AMC-CP-2222-02000
—●—	20,000	220	15.5	OFF	0	0	TEST CELL CALIB, 9 Feb 76
■	19,000	180	13.5	OFF	0	0	TEST CELL CALIB, 9 Feb 76
◆	21,000	140	15.0	OFF	0	0	TEST CELL CALIB, 9 Feb 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA



S/N 107211

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FIGURE 158
REFERRED POWER AND GAS TEMPERATURE

AIRCRAFT YUH-60A S/N 73-21651 ENGINE T700-GE-700 S/N 207228

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20,000	SL	15.0	OFF	0		PID SPEC AMC-CP-2222-02000
—◆—	20,000	-400	13.5	OFF	0	0	TEST CELL CALIB, 14 Feb 76
■	19,000	-480	13.5	OFF	0	0	TEST CELL CALIB, 14 Feb 76
◆	21,000	-560	15.0	OFF	0	0	TEST CELL CALIB, 14 Feb 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

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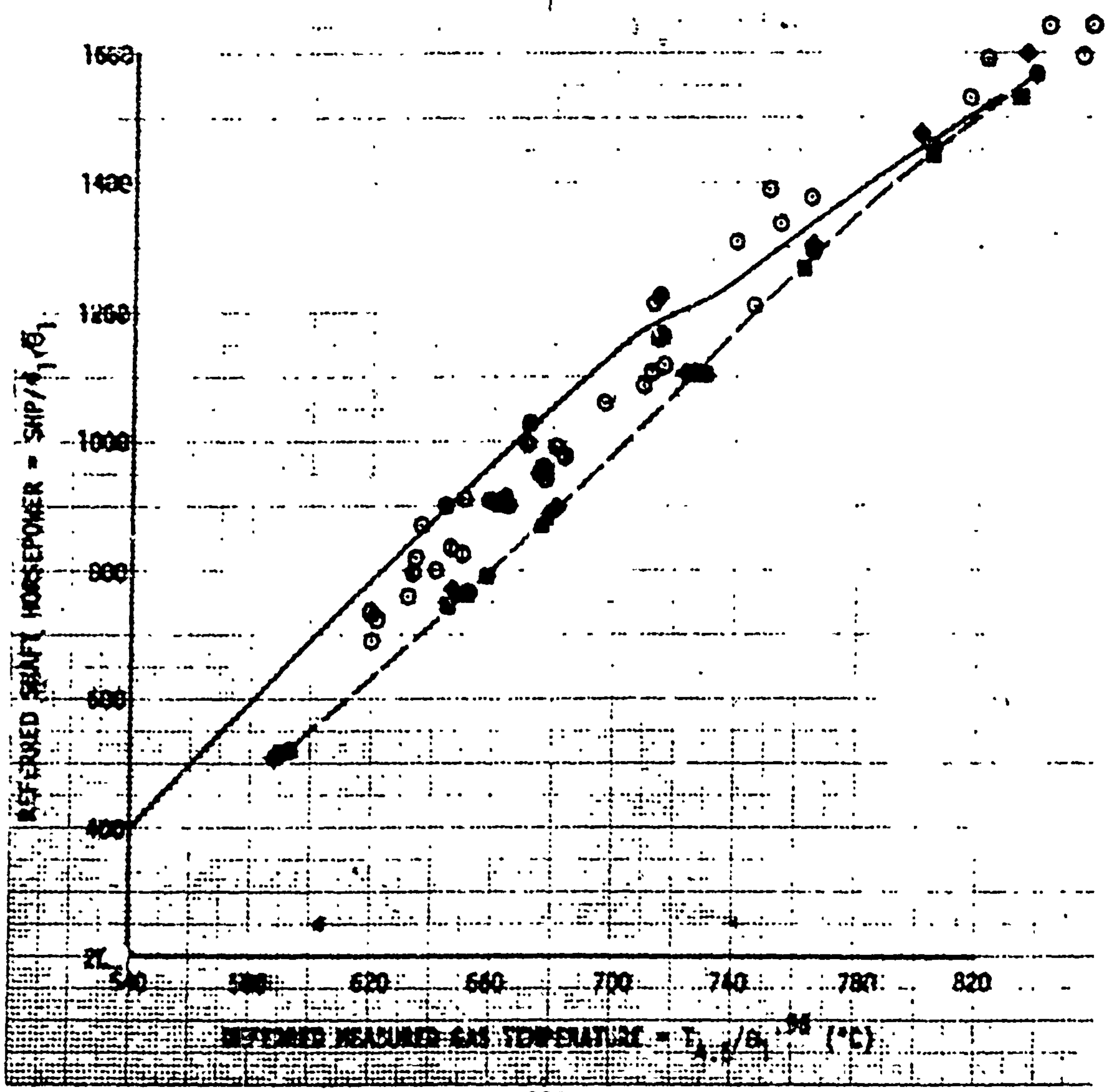
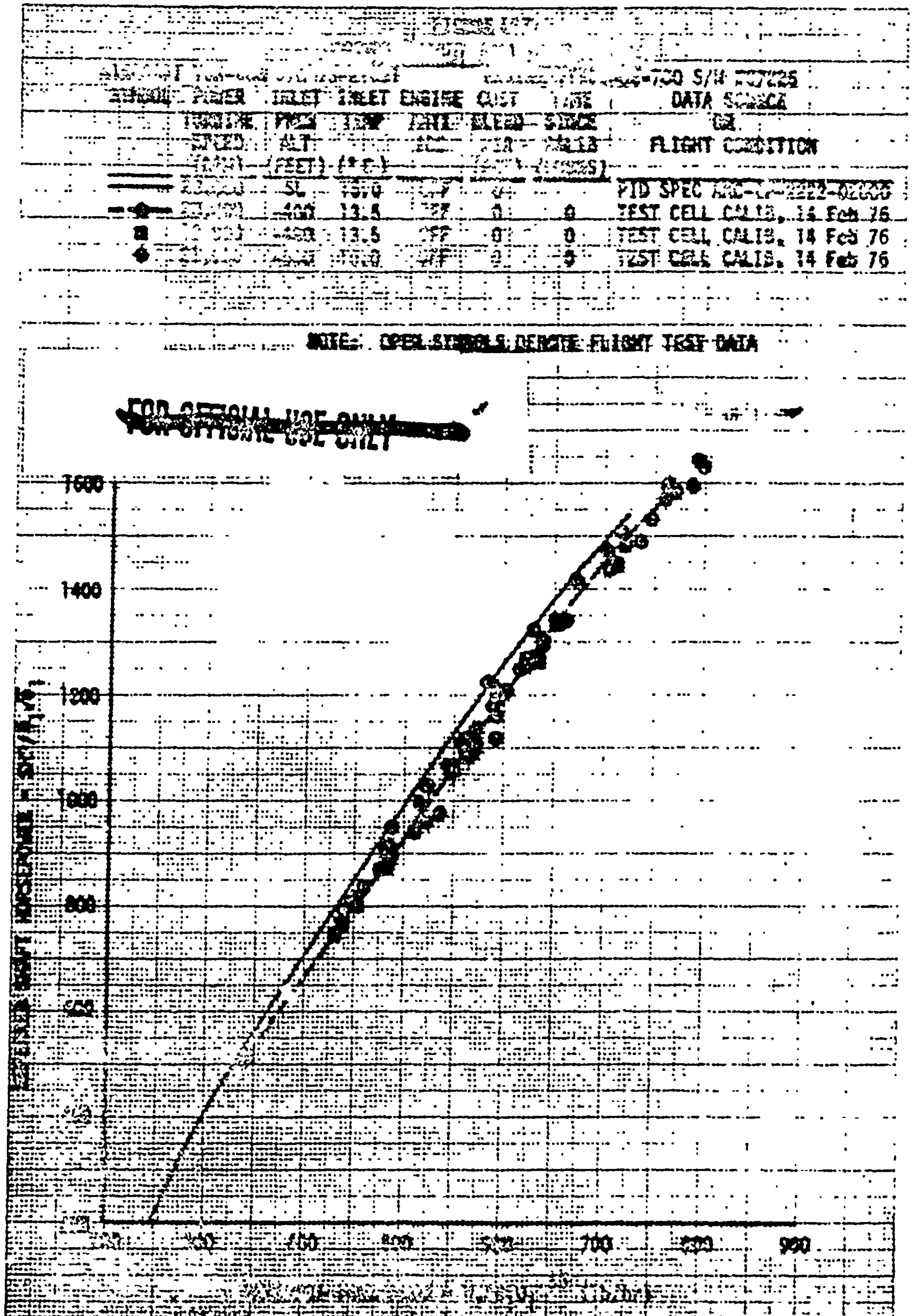



DATA SOURCE
OR
FLIGHT CONDITION
TEST CELL CALIB, 14 Feb 76
TEST CELL CALIB, 14 Feb 76
TEST CELL CALIB, 14 Feb 76
NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA
FOR OFFICIAL USE ONLY
1600
1400
1200
1000
800
700
900

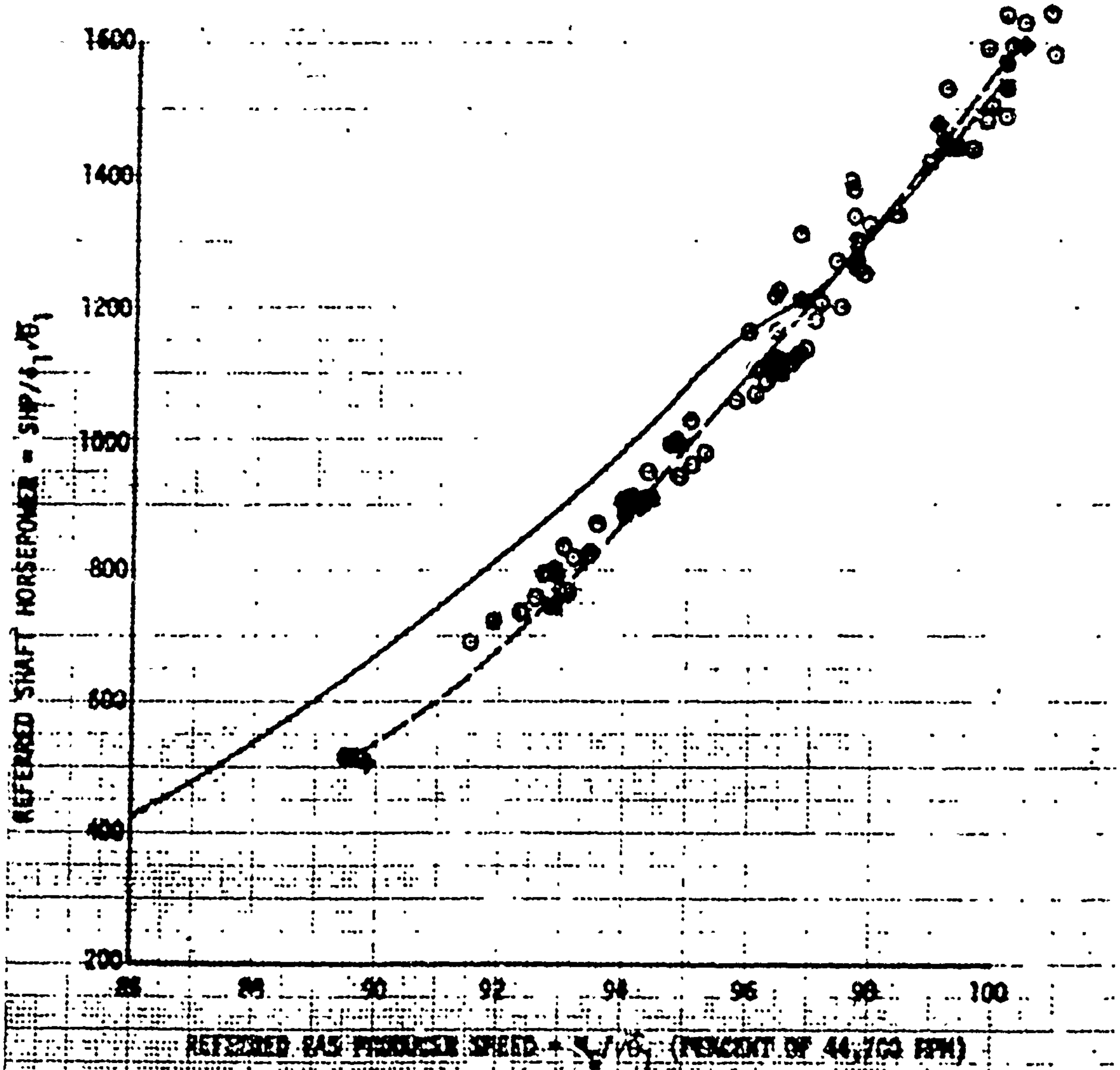
REFERRED POWER AND GAS PRODUCER SPEED

AIRCRAFT YUH-60A S/N 73-21651 ENGINE YT700-GE-700 S/N 207226

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20,000	SL	15.0	OFF	0		PID SPEC ANC-CP-2222-02000
—●—	20,000	-400	13.5	OFF	0	0	TEST CELL CALIB, 14 Feb 76
■	19,000	-480	13.5	OFF	0	0	TEST CELL CALIB, 14 Feb 76
◆	21,000	-560	15.0	OFF	0	0	TEST CELL CALIB, 14 Feb 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

[redacted]



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AIRCRAFT YUH-61A S/N [illegible] ENGINE [illegible] S/N [illegible]

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20,000	SL	15.0	OFF	0		PID SPEC AMC-CP-2222-02000
—●—	20,000	-100	15.0	OFF	0	0	TEST CELL CALIB, 20 Mar 76
◧	19,000	-160	16.5	OFF	0	0	TEST CELL CALIB, 20 Mar 76
◆	21,000	-140	17.5	OFF	0	0	TEST CELL CALIB, 20 Mar 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

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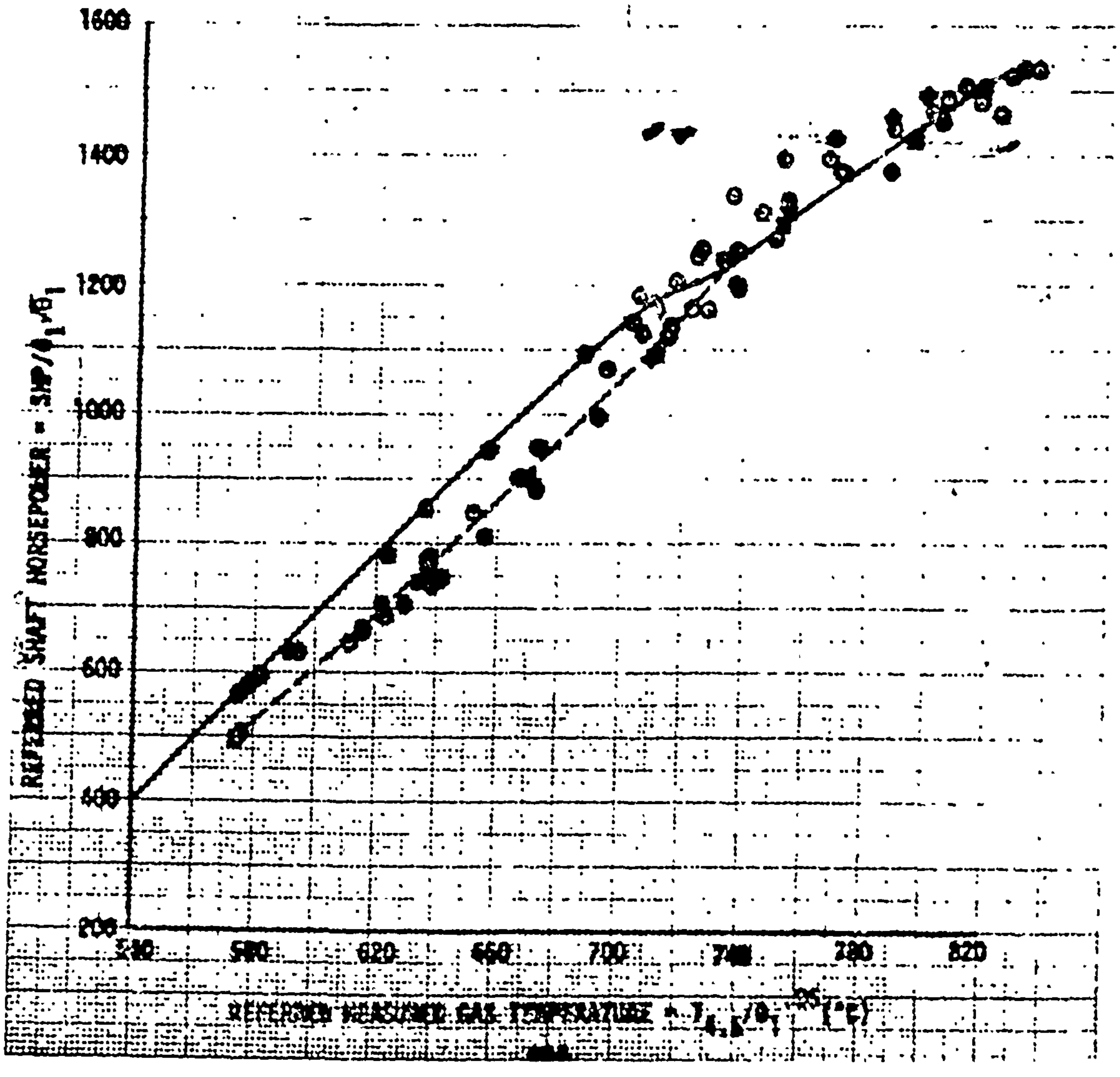

						FLIGHT CONDITION
[illegible]	[illegible]	[illegible]	[illegible]	0		[illegible] SPEC [illegible]-CP-2222-G3000
[illegible]	[illegible]	35.0	[illegible]	0	0	TEST CELL CALIB, 20 Mar 76
[illegible]	[illegible]	36.5	[illegible]	0	0	TEST CELL CALIB, 20 Mar 76
[illegible]	[illegible]	17.5	[illegible]	0	0	TEST CELL CALIB, 20 Mar 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

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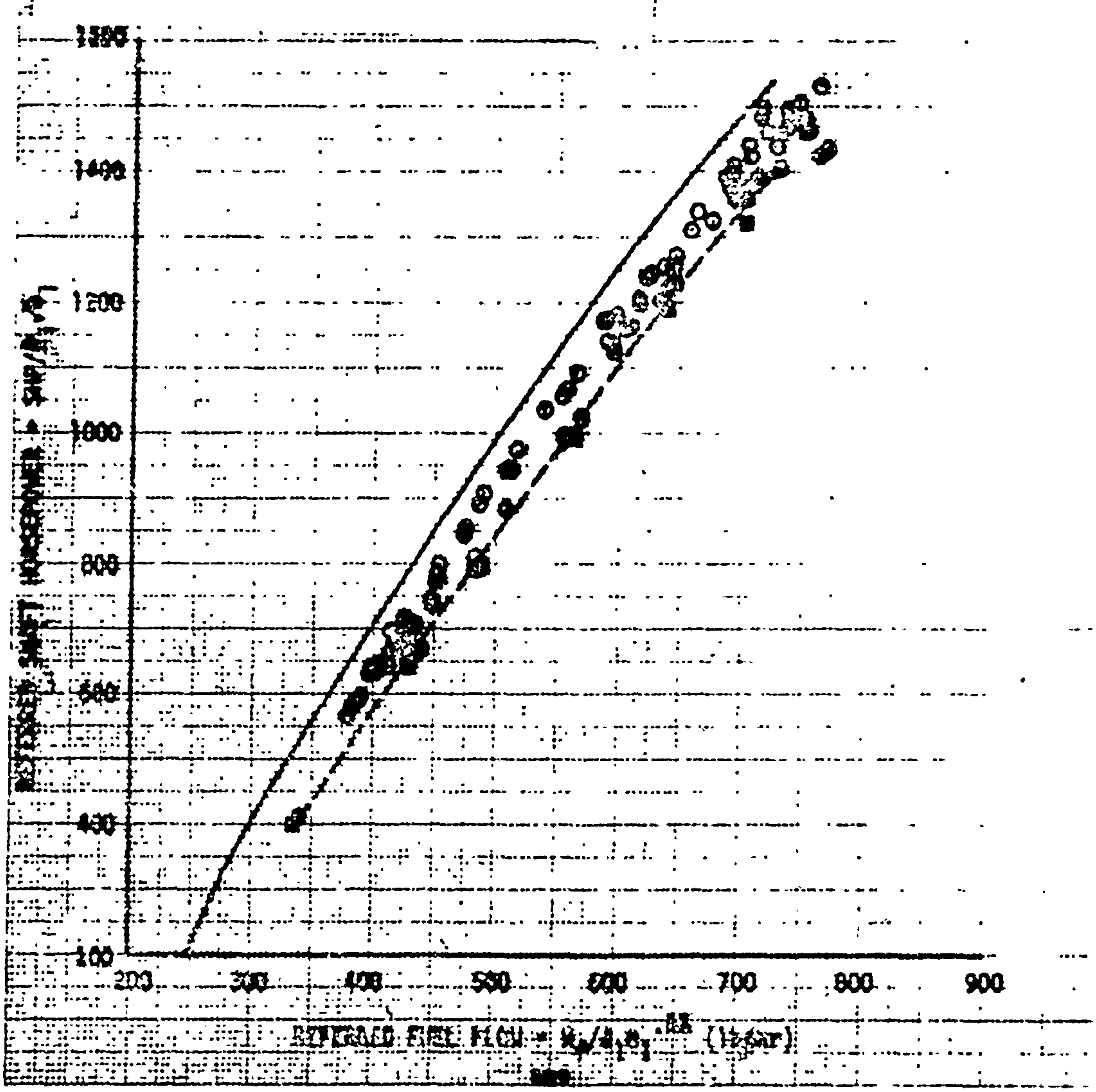

SYMBOL	POWER TURBINE SPEED (RPM)	PRESS ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	[illegible]	DATA SOURCE OR FLIGHT CONDITION
———	[illegible]	SL	[illegible]	OFF	0		[illegible]
—●—	20,000	-180	15.0	OFF	0	0	TEST CELL CALIB, 20 Mar 76
■	19,000	-180	16.5	OFF	0	0	TEST CELL CALIB, 20 Mar 76
●	21,000	-180	17.5	OFF	0	0	TEST CELL CALIB, 20 Mar 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

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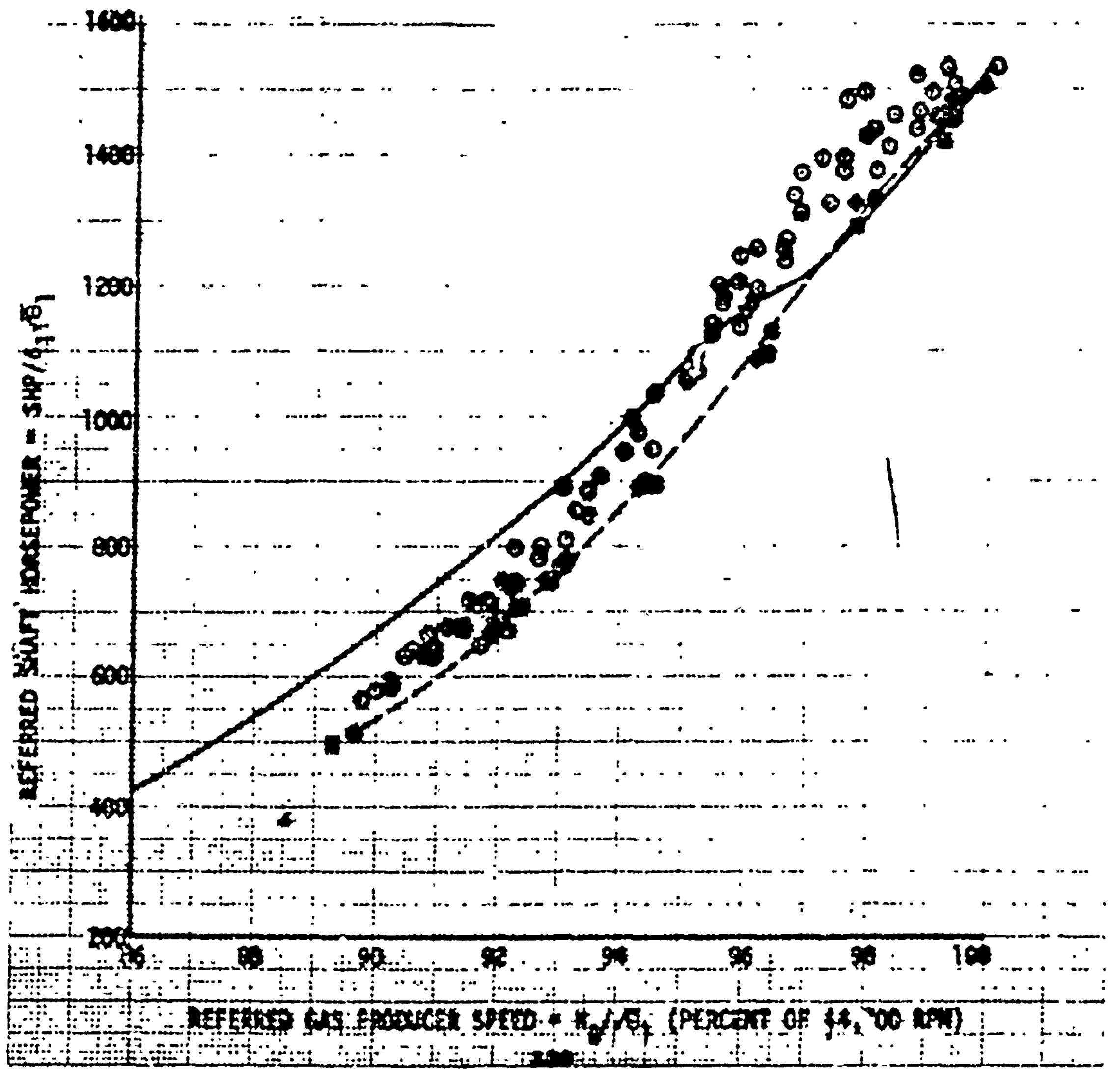

FIGURE 163

[illegible] POWER AND [illegible] DISCHARGE PRESSURE

AIRCRAFT [illegible] [illegible] S/N 207213

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FIGURE [illegible]

[illegible]

AIRCRAFT [illegible] ENGINE [illegible] S/N 207220

SYMBOL	POWER TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	20,000	SL	[illegible]	OFF	0	---	PID SPEC [illegible]-2222-[illegible]
—◆—	20,000	-20	22.5	OFF	0	0	TEST CELL CALIB, 30 Apr 76
■	19,600	-60	21.0	OFF	0	0	TEST CELL CALIB, [illegible] Apr 76
◆	21,000	-100	19.5	OFF	0	0	TEST CELL CALIB, 30 Apr 76

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

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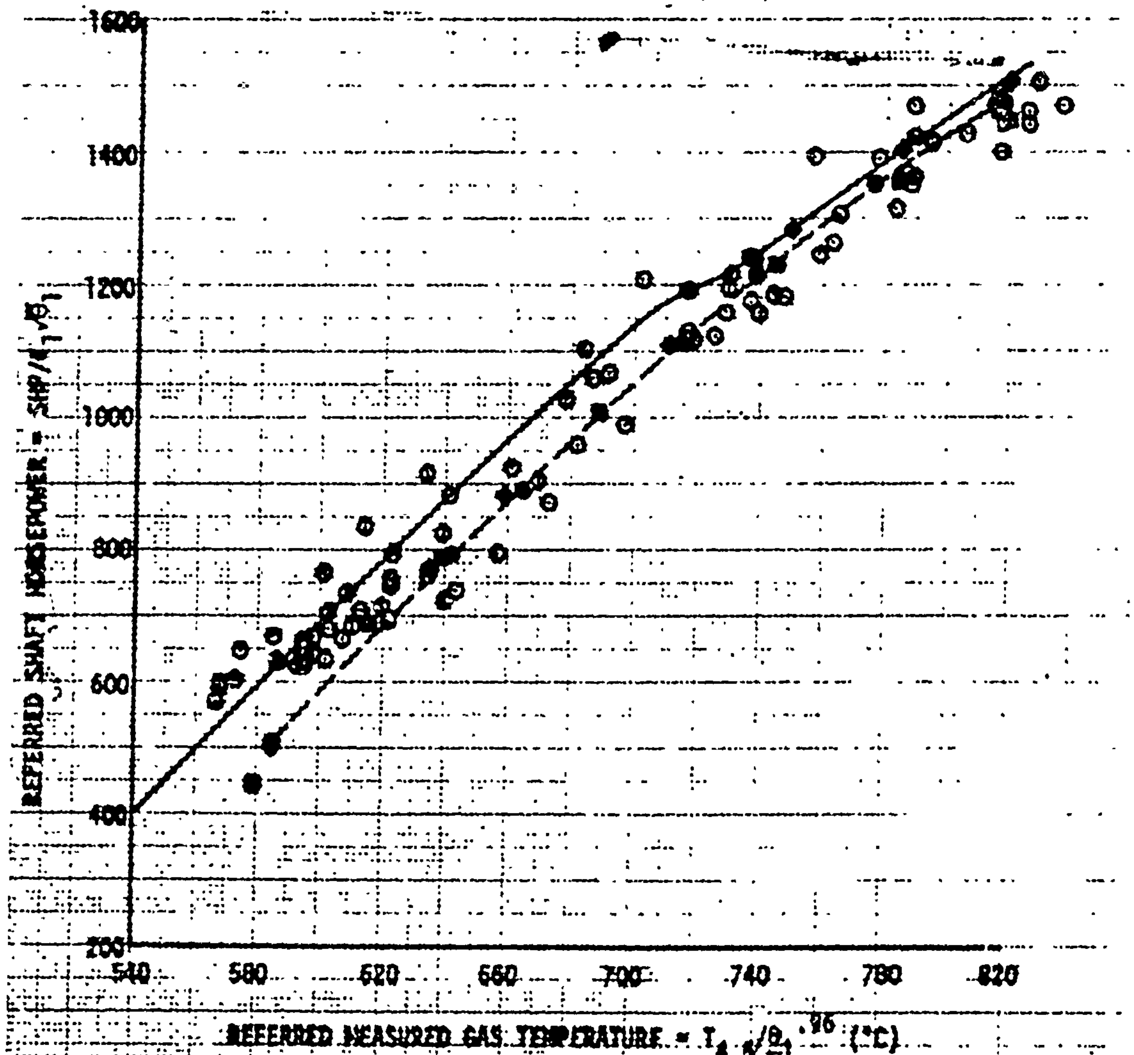

AIRCRAFT [illegible] ENGINE [illegible]

SYMBOL	TURBINE SPEED (RPM)	INLET PRES ALT (FEET)	INLET TEMP (°C)	ENGINE ANTI ICE	CUST BLEED AIR (PCT)	TIME SINCE CALIB (HOURS)	DATA SOURCE / FLIGHT CONDITION
(solid line)	[illegible]	[illegible]	15.0	OFF	0		[illegible]
(half-filled circle)	20,000	-20	22.5	OFF	0	0	TEST CELL CALIB. [illegible]
(square)	19,000	-60	21.0	OFF	0	0	TEST CELL CALIB. 30 Apr [illegible]
(filled circle)	21,000	-100	13.5	OFF	0	0	TEST CELL CALIB. [illegible]

NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA

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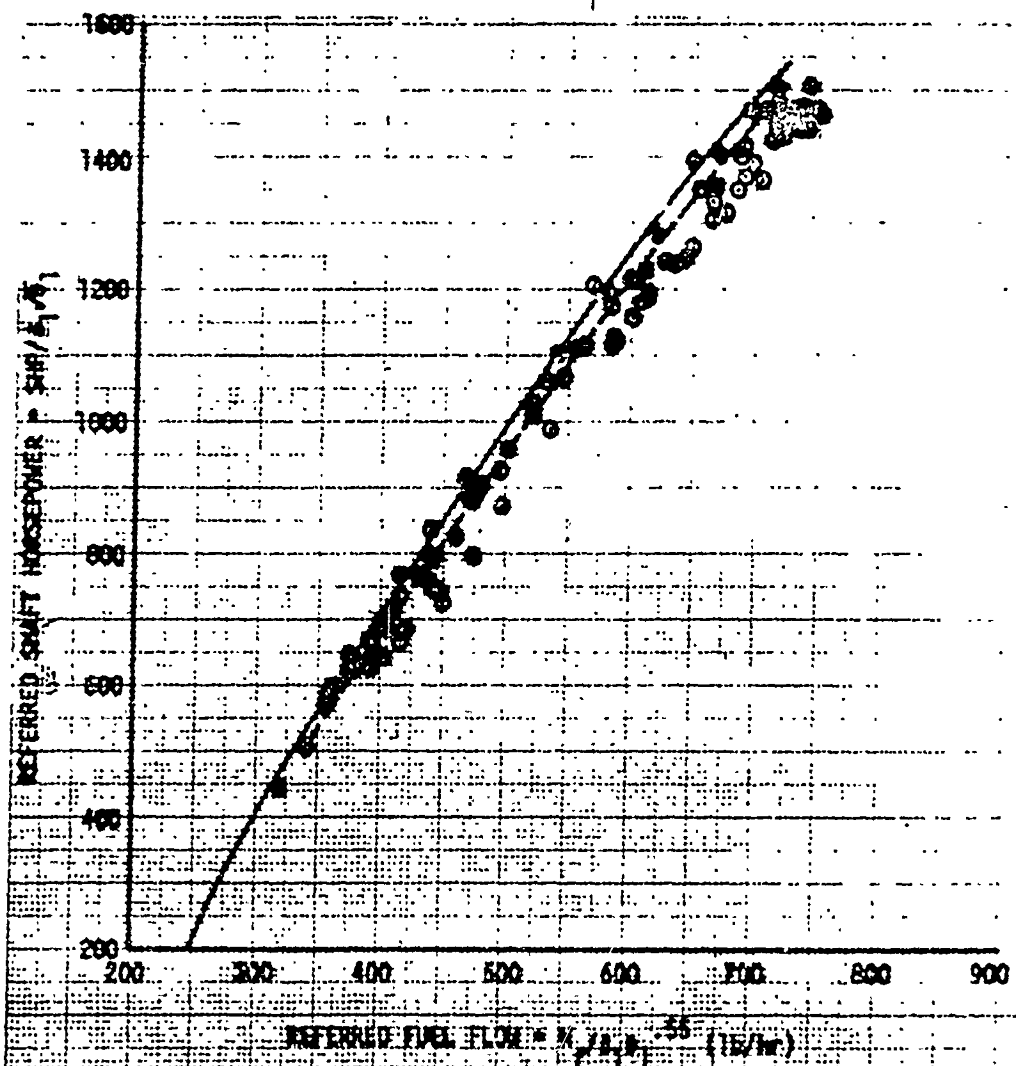

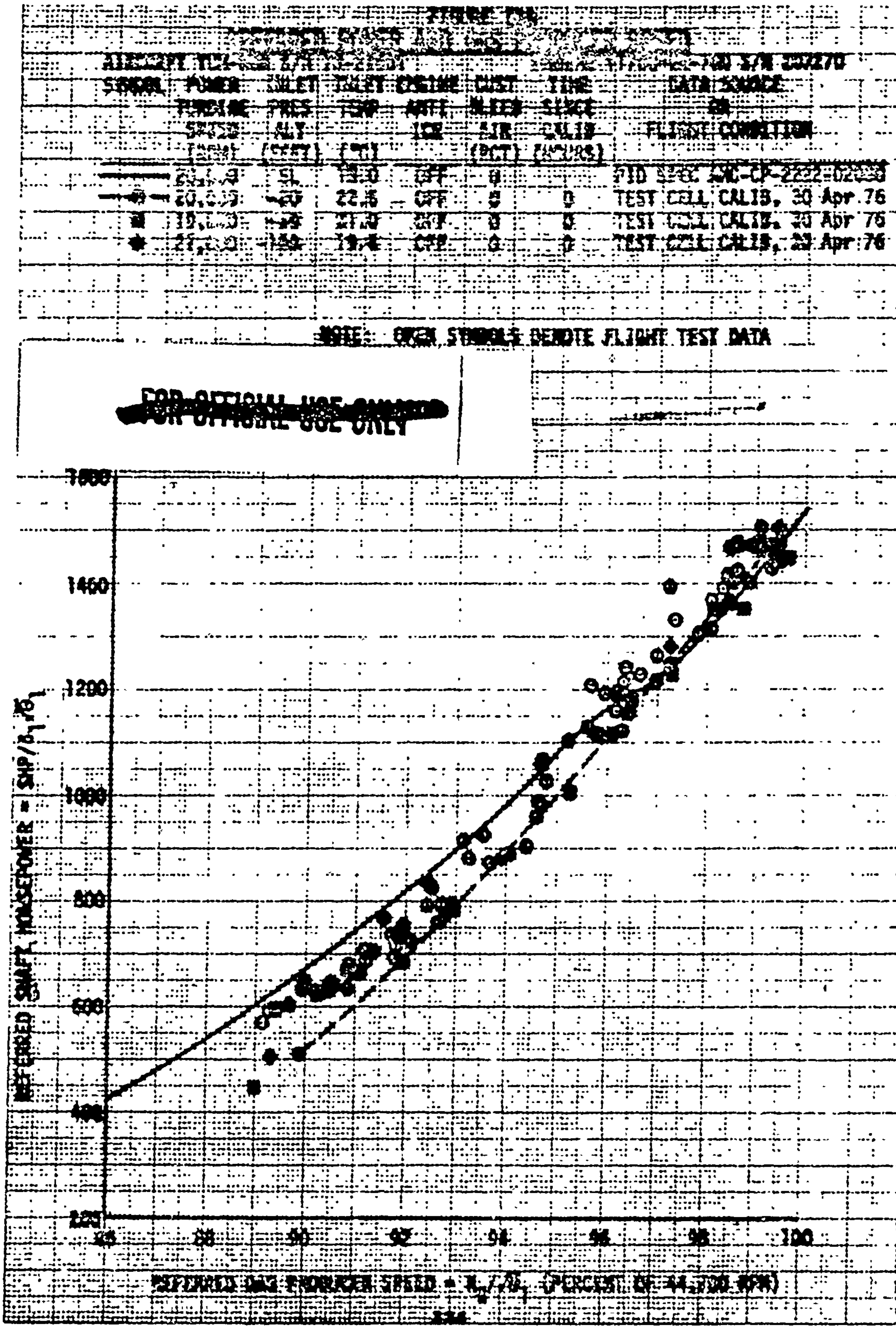

SYMBOL
DATA SOURCE
FLIGHT CONDITION
TEST CELL CALIB, 30 Apr 76
NOTE: OPEN SYMBOLS DENOTE FLIGHT TEST DATA
FOR OFFICIAL USE ONLY
1600
1400
1200
1000
800
600
400
200
86
88
90
92
94
96
98
100
REFERRED SHAFT HORSEPOWER

FIGURE 157

AIRCRAFT YUH-60A S/N 73-21651 ENGINE YT700-GE-700 S/N 207270

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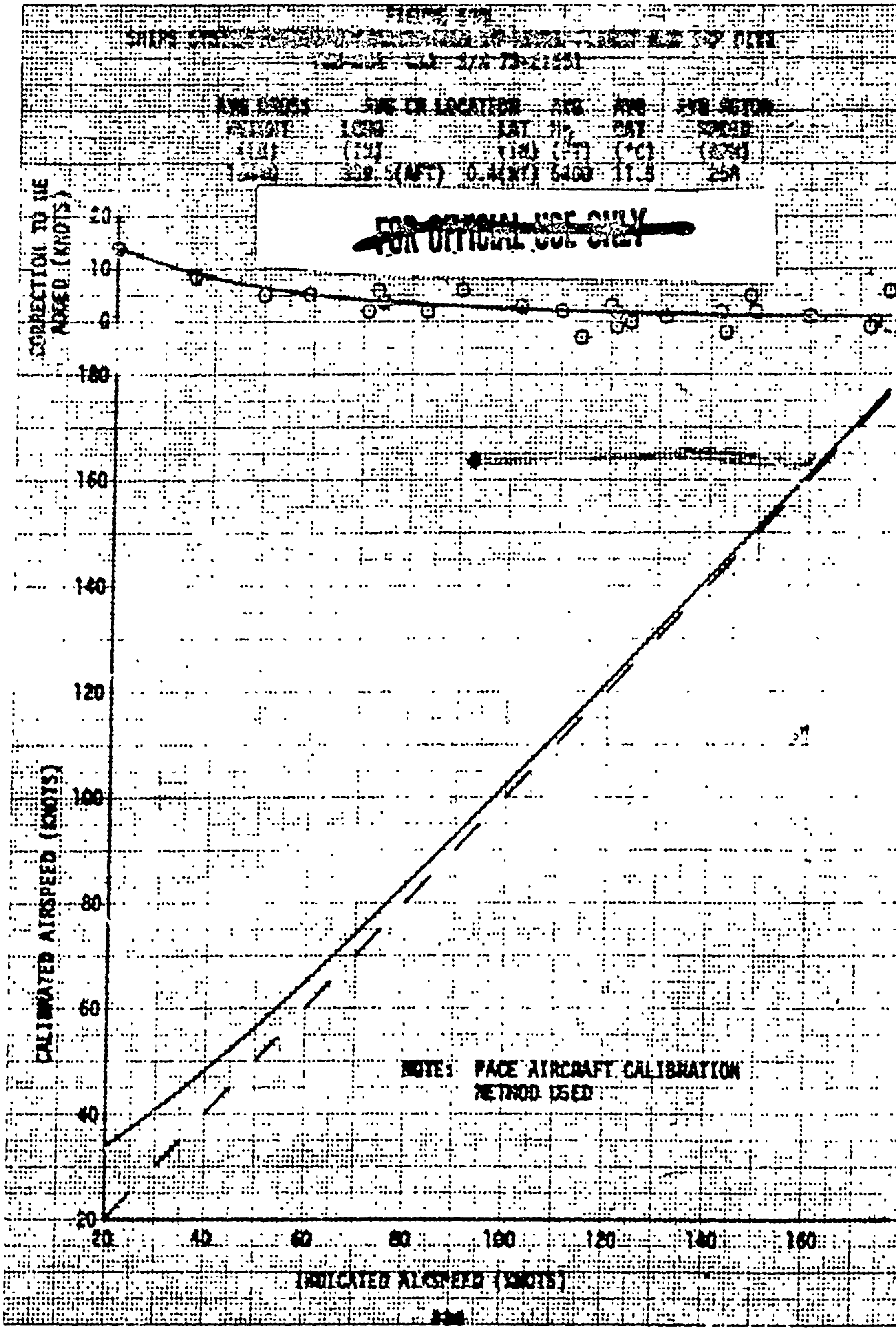

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CORRECTION TO BE ADDED (KNOTS)
20
10
0
CALIBRATED AIRSPEED (KNOTS)
180
160
140
120
100
80
60
40
20
NOTE: PACE AIRCRAFT CALIBRATION
METHOD USED
20
40
60
80
100
120
140
160
INDICATED AIRSPEED (KNOTS)

FIGURE 129

[illegible] IN CLIMBING AND AUTOROTATIONAL FLIGHT

[illegible] S/N 71-21[illegible]

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (IN)	AVG Hd (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION
○	[illegible]	359.8(AFT)	0.4(RT)	[illegible]	12.0	258	AUTO
□	10360	359.7(AFT)	0.4(RT)	5100	12.5	258	CLIMB

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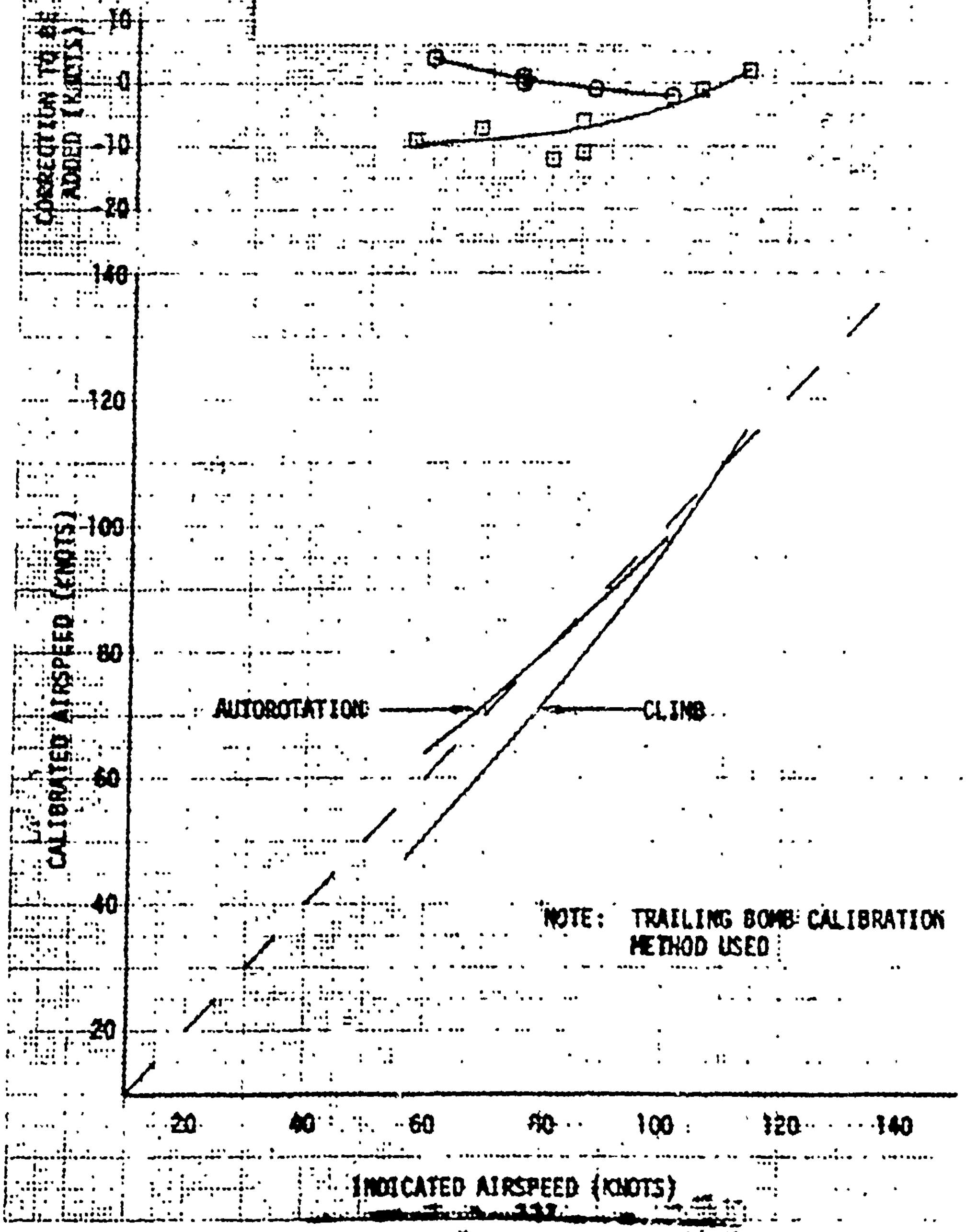

APPENDIX H. EQUIPMENT PERFORMANCE REPORTS

The following EPR's were submitted during the Model YUH-60A GCT.

EPR No.	Date Submitted	Descriptive Title
74-06-1-1	28 April 1976	Engine flame-out during overspeed check
74-06-1-2	28 April 1976	Sticking engine fuel selector valves
74-06-1-3	28 April 1976	Requirement to replace the No. 1 engine due to metal chips
74-06-1-4	29 April 1976	Requirement to replace the No. 2 engine due to metal chip
74-06-1-5	3 May 1976	Failure of the starter-coupling jaws
74-06-1-6	7 May 1976	Improper load sharing of the No. 1 engine
74-06-1-7	11 May 1976	False indications of hydraulic filter impending by-pass
74-06-1-8	8 June 1976	Inaccessible location of the transmission chip locator
74-06-1-9	8 June 1976	Inoperative starter
74-06-1-10	8 June 1976	Cracks in No. 1 engine exhaust assembly
74-06-1-11	8 June 1976	Failure of the lower droop stop spring
74-06-1-12	15 June 1976	Weak LDS cable housing supports
74-06-1-13	8 June 1976	Lack of access panels for the No. 2 tail rotor drive shaft bearing
74-06-1-14	8 June 1976	Inaccessibility of intermediate and tail rotor gearbox oil sight gages

[redacted]

74-06-1-15	8 June 1976	Locking of cabin door securing lugs prior to flight
74-06-1-16	8 June 1976	Lack of access to the vertical fin steps
74-06-1-17	28 June 1976	The probability of starter damage by the oil drain tube
74-06-1-18	28 June 1976	Drainage of fuel from the exhaust assembly into the engine deck area
74-06-1-19	8 June 1976	Drainage of engine oil into the cabin roof and into the cabin area
74-06-1-20	28 June 1976	Failure of APU to start
74-06-1-21	8 June 1976	Chafing hydraulic lines
74-06-1-22	28 June 1976	Loss of pressure from MRB spar
74-06-1-23	29 June 1976	Cyclic control longitudinal oscillation
74-06-1-24	28 June 1976	Ratchety longitudinal and lateral beep trim
74-06-1-25	28 June 1976	Improper operation of the horizontal stabilator
74-06-1-26	30 June 1976	Failure of the tail wheel locking pin to seat and unseat
74-06-1-27	1 July 1976	Intermittant successful vapor venting at Hp of 9500 feet
74-06-1-28	21 July 1976	Failure of the No. 2 engine to start at 9500 feet Hp
74-06-1-29	21 July 1976	Electrical short in the No. 1 engine power lever
74-06-1-30	21 July 1976	Leaking seal between the No. 1 generator and the generator control electrical connection
74-06-1-31	21 July 1976	Leaking hydraulic brake check valve

74-06-1-32	21 July 1976	Leaking hydraulic pressure switch
74-06-1-33	13 August 1976	Lack of inspection access for the upper tail gear shock strut
74-06-1-34	6 August 1976	Failure of the MRB damper bearing rod end
74-06-1-35	14 August 1976	Cracking of MRB tip caps
74-06-1-36	14 August 1976	Elastomeric bearing rubber-concentric ring separation
74-06-1-37	14 August 1976	Lack of inspection access to the No. 1 hanger bearing
74-06-1-38	14 August 1976	Lack of dust cover for directional pedals
74-06-1-39	14 August 1976	Inadequate engine oil drain
74-06-1-40	14 August 1976	Lack of tie down for the forward two main rotor blades
74-06-1-41	14 August 1976	Replace inadequate gust lock with a rotor brake
74-06-1-42	23 August 1976	Bonding separation between leading edge skin and the spar
74-06-1-43	8 August 1976	Accumulation of chips at the transmission main sump during landing nose up on a 15 degree slope
74-06-1-44	8 September 1976	Malfunction of the lateral accelerometer
74-06-1-45	7 September 1976	In-flight longitudinal control oscillation
74-06-1-46	1 September 1976	Broken and cracked bifilar assembly bushings
74-06-1-47	7 September 1976	Failure of the roll and yaw servo amplifiers
74-06-1-48	7 September 1976	Force slop in the longitudinal FAS servo

74-06-1-49	7 September 1976	Failure of the FPS vertical gyro
74-06-1-50	7 September 1976	Failure of the roll trim actuator
74-06-1-51	7 September 1976	Longitudinal cyclic control oscillation
74-06-1-52	7 September 1976	High roll SAS gain
74-06-1-53	7 September 1976	Self-exciting roll oscillation and ratchety lateral beep trim
74-06-1-54	7 September 1976	Bending in the tail rotor pedals
74-06-1-55	7 September 1976	Failure of the FAS Computer
74-06-1-56	8 September 1976	Runaway FAS trim
74-06-1-57	25 September 1976	Fuel boost pump switch design
74-06-1-58	7 September 1976	Pass of lateral trim
74-05-1-59	25 August 1976	Inability to disconnect the main transmission chip detector from its electrical connection

[redacted]

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