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DEVELOPMENT TEST I

ADVANCED ATTACK HELICOPTER COMPETITIVE EVALUATION

BELL YAH-63 HELICOPTER

FINAL REPORT

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Development Test I of the YAH-63 was conducted utilizing two prototype aircraft (serial numbers (SN) 74-22246 and 74-22247). Aircraft SN 246 was tested between 28 July and 5 October 1976 at Edwards Air Force Base, California (2303 feet elevation), and test sites near Bishop, California (4228 feet and 9500 feet). Aircraft SN 247 was tested between 7 August and 3 September 1976 at Edwards Air Force Base. A total of 99 flights for 100.7 hours (66.6 productive hours) were flown

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20. Abstract

on the two aircraft. Performance evaluation included hover, vertical climbs, forward flight climbs, level flight, autorotational descent, lateral acceleration, and vertical displacement. The YAH-63 failed to meet most of the performance requirements of the systems specification. Numerous envelope limits were imposed during the test which would be unacceptable for an operational aircraft. The helicopter exhibited several features which will enhance accomplishment of the attack helicopter mission. The excellent field of view and forward visibility afforded by the front cockpit pilot station, along with outstanding airspeed control at low airspeeds, provide an excellent nap-of-the-earth capability. The excellent handling qualities in rearward flight will enhance bob-up target acquisition and tracking tasks in downwind conditions. The chip verification system allows an in-flight check on transmission chips and could save a mission which would otherwise be aborted. The ordnance jettison panel allows easy and rapid selection and jettison of external stores. The airspeed and rotor speed control in autorotation are excellent. Nine deficiencies were identified during the tests. At airspeeds greater than 100 knots indicated airspeed, the handling qualities were unsatisfactory for flight in instrument meteorological conditions. The excessive transient rotor speed droop following a rapid power demand from a low-power condition limited the aircraft's ability to perform a quick-stop maneuver. Aircraft control following a stability and control augmentation system failure at airspeeds in excess of 100 knots calibrated airspeed is extremely difficult because of a divergent oscillation about all three axes. The XM188 weapon system failed to fire repeatedly during these tests. These failures would severely limit the combat effectiveness of the helicopter. Inadequate control margins in three flight regimes seriously degrade aircraft handling qualities in those regimes. The requirement for the pilot to manually tune radios during nap-of-the-earth flight creates an unacceptable workload. A total of 59 shortcomings were noted along with 20 instances of specification noncompliance.

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DEPARTMENT OF THE ARMY
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DRDAV-EQ

SUBJECT: Directorate for Development and Engineering Position on the Conclusions and Recommendations of the Final Report on USAAEFA Project No. 74-07-1, Development Test I, Advanced Attack Helicopter, Competitive Evaluation, Bell YAH-63 Helicopter, December 1976

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1. The purpose of this letter is to document the Directorate of Development and Engineering position on the subject test report. It must be recognized that the Bell Helicopter Textron YAH-63 was not selected for continued engineering development to meet the Army's Advanced Attack Helicopter (AAH) requirements, however as a part of the selection process, the AAH Source Selection Board (SSEB) did negotiate with the manufacturer proposed corrections to all discrepancies (except for GFE) found during these tests with which the SSEB agreed were significant problems. In addition, the resulting configuration would have had a significantly lighter empty weight and many other features which would improve aircraft performance. Details of these corrections are no longer important and are therefore not included within this letter. However, some areas raising fundamental technical issues are discussed by paragraph numbers from the subject report.

a. Para 97. The SCAS monitor system contained in this aircraft should be considered an enhancing feature. While it was not specifically required of the specification, it represents one of the most significant safety improvements for SCAS systems that the Army has yet seen.

b. Paras 143b, 143d, 143i and 144hh. The AAH is the first attempt by the Army to obtain an attack helicopter capable of operating under IMC and other adverse weather conditions, such as in moderate ice. Therefore the flight characteristics under these conditions during normal operation and/or with various modes of the SCAS inoperative are extremely important. Contract emphasis must be placed on these flying qualities to provide staying power on the battlefield under adverse conditions.

c. Para 143h. As with other Army air items, some avionics will not have preset frequencies since this is not considered a requirement by

MAY 9 1978

DRDAV-EQ
SUBJECT: Directorate for Development and Engineering Position on the Conclusions and Recommendations of the Final Report on USAAEFA Project No. 74-07-1, Development Test I, Advanced Attack Helicopter, Competitive Evaluation, Bell YAH-63 Helicopter, December 1976

DA. However, this position is being reconsidered due to the significant nap-of-the-earth operation of the attack helicopter.

d. Para 143s. The YAH-63 did not possess adequate sideslip characteristics (increasing bank angle with sideslip) but did meet PIDS requirements. This suggests that a minimum value derivative should be contained in future specification requirements rather than just a requirement for a positive variation.

e. Para 144b. The rotor speed fluctuations which continually activated the RPM warning light during day NOE flight should be considered a deficiency in that it continually detracted the pilots attention back into the cockpit. This diversion is more significant than that associated with the absence of pre-tuned radios because of the difference in frequency of occurence.

f. Para 144h. The YAH-63 brake system meets MIL Spec requirements. However, redesign to the crew station would make brake application easier and alleviate the problems associated with ground handling.

g. Para 144nn. While the vertical baffles were not properly optimized, this concept represents one of the best approaches to reducing reflections within the cockpit that we have seen to date. This problem of reducing internal reflections from both internal and external sources is complicated by the rear flat glass canopy which is an attempt to significantly reduce external glint, therefore visual detection by the enemy. Industry should be encouraged to continue development of several approaches to the solution of this problem.

2. Because of the overweight status of the prototype aircraft and other simple performance improvements planned for incorporating into the production design, the performance levels shown in this report are not representative of the potential operational capability of the AH-63.

FOR THE COMMANDER:

Walter A. Ratcliff

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

INTRODUCTION

BACKGROUND

1. On 22 June 1973, the United States Army Aviation Systems Command (AVSCOM), since redesignated the Army Aviation Research and Development Command (AVRADCOM), awarded a Phase I engineering development contract to Bell Helicopter Textron (BHT). The contract required BHT to design, develope, fabricate, and test two advanced attack helicopter (AAH) prototypes, and one ground test vehicle (GTV), designated YAH-63. The YAH-63 made its first flight on 1 October 1975. The United States Army Aviation Engineering Flight Activity (USAAEFA) was tasked to prepare a test plan (ref 1, app A) for the conduct of Development Test I (DT I) of the BHT prototypes. The first YAH-63 was delivered to Edwards Air Force Base, California on 10 July 1976, followed by the second aircraft on 2 August 1976.

TEST OBJECTIVES

2. The objectives of DT I were as follows:

a. To provide engineering data to the AAH Source Selection Evaluation Board (SSEB) for comparison with the systems specification for the AAH (ref 2, app A).

b. To provide engineering data for determining compliance with the BHT systems specification for the YAH-63 (ref 3, app A).

c. To provide airworthiness data as a basis for updating the safety-of-flight release (SOFR) for Operational Test I (OT I).

DESCRIPTION

3. The YAH-63 is a two-place, tandem-seat, twin-engine helicopter with two-bladed main and antitorque rotors. The wheel-type tricycle landing gear incorporates a kneeling feature. The helicopter is powered by two General Electric YT700-GE-700 turboshaft engines. The YAH-63 incorporates a 30mm gatling gun in an integral chin turret and also is capable of firing 2.75-inch folding fin aircraft rockets (FFAR) and tube-launched, optically-tracked, wire-guided (TOW) missiles from wing mounting stations. A detailed description of the aircraft and the flight control system is contained in appendixes B and C, respectively. Appendix D contains a detailed description of the YT700-GE-700 engines used during these tests. Mission gross weight of the YAH-63 is 16,054 pounds.

TEST SCOPE

4. Development Test I of the YAH-63 was conducted utilizing two prototype aircraft (serial numbers (SN) 74-22246 and 74-22247). Aircraft SN 246 was tested between 28 July and 5 October 1976 at Edwards Air Force Base, California (2303 feet elevation), and test sites near Bishop, California (4228 and 9500 feet). Aircraft SN 247 was tested between 7 August and 3 September 1976 at Edwards Air Force Base. A total of 99 flights for 100.7 hours (66.6 productive hours) were flown on the two aircraft. Pilots from the Operational Test and Evaluation Agency (OTEA) flew in the copilot seat as often as possible to prepare for OT I. The test aircraft were equipped with special instrumentation which was installed, calibrated, and maintained by the contractor. The aircraft was also maintained by the contractor. Tests were flown in accordance with the restrictions contained in the SOFR (refs 4 and 5, app A). Test results were compared to the requirements of the Army and BHT systems specifications. The vertical agility of the helicopter was evaluated based on a modified vertical displacement maneuver defined in reference 6.

5. The tests were conducted in the following three external wing store configurations: (1) clean: no rocket pods or TOW missile launchers installed, outboard stores pylons installed, inboard pylons removed; (2) 8-TOW: two TOW missile lauchers installed on each outboard wing store station, outboard pylons installed, inboard pylons removed; and (3) 76-rocket: one M200A1 rocket pod on each wing stores station, pylons installed on inboard and outboard stations. During some hover performance tests ballast cans were installed on the stores stations. The shape of these cans simulated the M200A1 rocket pod. The XM188 weapon was in the straight-ahead stowed position for all tests except weapons firing. Performance and handling qualities test conditions are detailed in table 1.

TEST METHODOLOGY

6. Standard test techniques were used during these tests and are briefly described in the Results and Discussion section and Appendix F of this report. Trim conditions for all tests were in coordinated (ball-centered) flight. A Handling Qualities Rating Scale (HQRS) was used during evaluation of mission tasks (fig. 1, app F). Flight test data were obtained from sensitive calibrated test instrumentation and standard ship's system indicators displayed on the instrument panel and recorded on magnetic tape. A list of the test instrumentation is presented in appendix E. Test techniques and data analysis methods are described in appendix F.

Table 1. Test Conditions.[1]

Type of Test	Gross Weight (lb)	Longitudinal Center-of-Gravity Location (in.)	Density Altitude (ft)	Trim Calibrated Airspeed (kt)
Hover performance[2]	16,270 to 18,560	295.7 (mid) to 296.7 (mid)	2300 to 2320	Zero
	14,160 to 19,110	292.9 (fwd) to 298.1 (aft)	4060 to 5640	
	14,550 to 15,020	294.0 (fwd) to 296.6 (mid)	10,660 to 11,240	
Vertical climb performance	15,000 to 17,200	295.0 (fwd) to 297.0 (mid)	5520 to 6300	Zero
Forward flight climb performance	15,100 to 16,400	293.5 (fwd)	4000 to 5200	68 to 72
Level flight performance	14,900 to 16,480	292.5 (fwd) to 293.0 (fwd)	5440 to 6880	[3]37 to 147
	15,580 to 16,180	291.3 (fwd) to 298.6 (aft)	7400 to 9500	[3]39 to 145
	16,580 to 17,200	293.2 (fwd) to 293.6 (fwd)	10,140 to 10,960	[3]39 to 116
Autorotational descent performance	15,740 and 16,020	293.2 (fwd) and 293.5 (fwd)	6340 and 6300	61 to 94
Lateral accelerations	16,020	295.8 (mid)	5320	Stabilized hover
Vertical displacement	16,240	294.8 (fwd)	5080	[3]140
Control positions in trimmed forward flight	14,660 to 16,140	291.3 (fwd) to 298.8 (aft)	6420 to 7960	35 to 136
	16,180	293.5 (fwd)	9500	
Static longitudinal stability	15,540 to 16,200	298.1 (aft) to 298.7 (aft)	6320 to 7180	44 to 110
Static lateral-directional stability	15,820 to 16,300	298.2 (aft) to 298.8 (aft)	5980 to 7640	40 to 125
Maneuvering stability	14,960 to 16,420	298.3 (aft) to 299.0 (aft)	5700 to 7360	58 to 122
Dynamic stability	15,540 to 16,340	298.1 (aft) to 298.8 (aft)	5380 to 6340	87 to 127
Controllability	15,840 to 16,340	298.4 (aft) to 298.7 (aft)	3520 to 3760	Zero
	15,640 to 16,520	298.1 (aft) to 298.7 (aft)	5320 to 7260	Zero to 127
	15,780	297.3 (aft)	10,840	Zero to 127
Takeoff and landing characteristics	15,000	298.7 (aft)	4300	Zero to 125
Low-speed flight characteristics	15,420 to 16,260	294.5 (fwd) to 295.3 (fwd)	10,760 to 11,520	[4]Zero to 45
	16,120 to 16,140	295.8 (mid)	3440 to 4440	
Mission maneuvering characteristics	16,000	295.8 (mid)	4000	Zero to 140
	14,400	298.5 (aft)	7000	
Weapons firing	17,600	297.5 (aft)	5700	Zero, 90, 120
Instrument flight operations and night visibility	16,100	298.7 (aft)	6500	45 to 126
Aircraft systems failures	16,440 to 16,500	299.3 (aft) to 299.4 (aft)	6060 to 6200	82 to 116
Simulated single-engine failures and autorotational entries	14,640 to 16,120	298.2 (aft) to 298.7 (aft)	8280 to 9080	73 to 80

[1]Rotor speed: 276 rpm (272 to 289 rpm during autorotational descent and 272 to 279 rpm during hover performance). Configuration: 8-TOW, except for level flight performance, which was flown clean and 8-TOW; hover, which was flown in the 76-rocket configurstion; and weapons firing, which was flown clean and 76-rocket configuration. Stability and control augmentation system (SCAS) ON unless specified OFF.
[2]Free flight hover technique at wheel heights of 5 and 100 feet.
[3]Knots true airspeed (KTAS).
[4]Forward, rearward, and sideward flight (zero to 10 KTAS to the right at the high altitude).

RESULTS AND DISCUSSION

GENERAL

7. Performance and handling qualities of the YAH-63 helicopter were evaluated at high-altitude and low-altitude test sites. Two instrumented test aircraft were used. Performance of the YAH-63 failed to meet most of the requirements of the systems specification. Numerous envelope limits were imposed during this test which would be unacceptable for an operational aircraft. Several aircraft characteristics were found to enhance the capability to perform the attack helicopter mission. The excellent field of view and forward visibility afforded by the front cockpit pilot station, along with outstanding airspeed control at low airspeeds, provide an excellent nap-of-the-earth (NOE) capability. The excellent handling qualities in rearward flight will enhance bob-up target acquisition tasks in downwind conditions. The chip verification system allows an in-flight check on transmission chips and could save a mission which would otherwise be aborted or at least delayed. The ordnance jettison panel allows easy and rapid selection and jettison of external stores. Nine deficiencies were identified during the evaluation. Of these, the most significant were the internal reflection of external light sources on the canopy during night flight; the unsatisfactory handling qualities for flight in instrument meteorological conditions (IMC) at airspeeds greater than 100 knots; the excessive transient rotor speed droop following a rapid power demand from a low-power condition; the divergent oscillation about all three axes at airspeeds greater than 100 knots with SCAS OFF; and the repeated failure of the XM188 weapon system. A total of 59 shortcomings and 20 instances of specification noncompliance were noted.

PERFORMANCE

General

8. Performance testing was conducted with aircraft SN 74-22246 at test site elevations of 2302, 4228, and 9500 feet. Performance evaluations included hover, vertical climbs, forward flight climbs, level flight, autorotational descent, lateral acceleration, and vertical displacement. Most of the systems specification performance requirements are at mission gross weight (16,054 pounds for the YAH-63) on a hot day (35°C), with primary mission external stores. At these conditions, the helicopter has an out-of-ground-effect (OGE) hover ceiling and single-engine service ceiling of 4700 feet and 2070 feet pressure altitude, respectively. At the same gross weight and temperature and 4000 feet pressure altitude, the aircraft has a maximum level flight airspeed of 142 KTAS but did not meet the vertical climb requirement at 0.95 percent intermediate rated power (IRP) (the aircraft could not hover at these conditions). The minimum autorotational rate of descent was 2250 ft/min at 61 knots indicated airspeed (KIAS) and a rotor speed of 276 rpm. Power available for all performance specification compliance

calculations was based on the YT700-GE-700 prime item development specification (AMC-CP-2222-02000) using induction and exhaust losses measured during these tests. As shown in table 2, the YAH-63 failed to meet most of the mission performance requirements of the systems specification.

Table 2. Performance Specification Compliance.[1]

Test[2]	Specification Requirement	YAH-63 Performance
Vertical climb[3] at 0.95 IRP	450 to 500 ft/min	Could not hover
Single-engine service ceiling	5000-ft pressure altitude	2070 ft
Level flight airspeed at MCP[3,4]	145 to 175 KTAS	122 KTAS
Maximum level flight airspeed[3]	150 KTAS	142 KTAS
Single-engine level flight at IRP[3]	90 KTAS	Single-engine level flight not possible
Endurance at sea level	2.5 hours	2.38 hours
Lateral accelerations[3]	0.25g	0.35g left, 0.48g right

[1]Army systems specification.
[2]All results were at mission gross weight (16,054 pounds) and 35°C, except endurance at sea level, 15°C, and operating weight plus primary mission payload and maximum internal fuel.
[3]Pressure altitude 4000 feet.
[4]Maximum continuous power.

Hover Performance

9. Hover performance testing was accomplished at both the high-altitude and low-altitude test sites at the conditions listed in table 1. The aircraft fuel load, ballast, and rotor speed were varied to obtain data at a wide range of thrust coefficients. The OGE hover performance summary (fig. 1, app G) shows a hover ceiling of 4730 feet pressure altitude on a 35°C day at mission gross weight. The in-ground-effect (IGE) (5-foot wheel height) hover ceiling at mission gross weight on a 35°C day was 7250 feet pressure altitude. Figures 2 and 3 present nondimensional hovering performance data for 5-foot and 100-foot wheel heights, respectively. Nondimensional tail rotor performance is presented in figure 4.

Climb Performance

Vertical:

10. Unaccelerated vertical climbs were made between 200 to 500 feet above ground level (AGL) at various constant collective control settings at the conditions shown in table 1. A radar altimeter and recording observation instruments (ROI) were used to measure rates of climb. A detailed description of the test techniques and data analysis methods used is contained in appendix F. The minimum vertical rate of climb for a given power increment was defined and the results are presented in figures 5 through 8, appendix G.

11. At mission gross weight, hot-day conditions (4000 feet, 35°C), and 95 percent IRP, the maximum vertical rate of climb was calculated to be less than zero. The vertical climb performance requirement of paragraph 3.2.1.1.1.1a of the systems specification (ref 2, app A) of 450 ft/min was not met.

Forward Flight:

12. The single-engine climb performance of the YAH-63 was determined at the conditions listed in table 1. Two continuous climbs were conducted (one on each engine) using an airspeed versus altitude schedule determined from level flight performance tests. Correction factors for gross weight (K_W) and power (Kp) variation were determined from sawtooth climbs and were applied to the continuous climb data (app F). Figures 9 and 10, appendix G, present the Kp and K_W data. The continuous climb test results were then corrected to hot-day (constant 35°C) conditions and are presented in figure 11, appendix G. The hot-day single-engine service ceiling (*ie*, altitude at which maximum rate of climb equals 100 ft/min) was 2070 feet pressure altitude at mission gross weight. This service ceiling fails to meet the 5000-foot requirement of paragraph 3.2.1.1.1.3b of the systems specification.

Level Flight Performance

13. Level flight performance tests were conducted at the conditions listed in table 1 to determine power required and fuel flow as functions of airspeed. In

addition, specific range, long-range cruise airspeed (V_{cruise}), endurance airspeed (airspeed at minimum fuel flow) and maximum airspeed for level flight (V_H) at MCP were determined. Data were obtained in stabilized level flight at incremental airspeeds from 40 KTAS to V_H in the clean and 8-TOW configurations. A constant ratio of gross weight to air density ratio (W/σ) was maintained by increasing altitude as fuel was consumed. The results of these tests are presented nondimensionally in figures 12 through 15, appendix G, and dimensionally in figures 16 through 25. Aircraft specific range, maximum endurance and V_{cruise} for the 8-TOW configuration are summarized in figures 26 and 27.

14. The increase in equivalent flat-plate area (Δf_e) from the clean to 8-TOW configuration was a constant 3.6 ft^2 (assuming a propulsive efficiency of unity). One 8-TOW level flight performance test was conducted at an aft center of gravity (cg). Changing cg from forward to aft decreased Δf_e by 4.6 ft^2. Figure A presents level flight power required versus airspeed for the mission gross weight and 8-TOW configuration at a 4000-foot pressure altitude at 35°C. As shown in figure A, V_H at MCP is 122 KTAS, which fails by 23 knots to meet the 145-KTAS requirement of paragraph 3.2.1.1.1.1b of the systems specification. Figure A also indicates that single engine level flight is not possible under these conditions. Therefore, the 90-KTAS single-engine level flight requirement of paragraph 3.2.1.1.1.3a was not met. The mission profile specified in paragraph 3.2.1.1.1.1c requires a 6-minute level flight segment at 150 KTAS at the conditions of figure A. The V_H of the YAH-63 at those conditions was 142 KTAS. Therefore, the mission profile specified could not be performed. Table 3 summarizes the endurance specified in paragraph 3.2.1.1.1.1d. The YAH-63 endurance of 2.38 hours failed by 0.12 hours to meet the 2.5 hour requirement.

15. Mission gross weight for AAH is defined in paragraph 3.2.2.1.5 of the systems specification as ". . . operating weight plus primary mission payload and primary mission fuel as defined in paragraph 3.2.1.1.1.1c . . ." Table 4 shows the calculation of mission gross weight for the YAH-63. Calculation of the primary mission fuel load is shown in table 5. As discussed in paragraph 13, the YAH-63 could not maintain level flight at 150 KTAS. Therefore, V_H was used in calculating fuel burned during that segment of the primary mission profile of paragraph 3.2.1.1.1.1c. Mission gross weight of the YAH-63 is 16,054 pounds.

Maneuvering Performance

Lateral Acceleration:

16. The lateral acceleration performance of the YAH-63 helicopter was determined at the conditions listed in table 1. The lateral acceleration maneuver is illustrated in figure B. The test was conducted at bank angles up to 50 degrees to the right and 38 degrees to the left. Performance data were recorded with ground operated space positioning equipment and on-board instrumentation. Data reduction methods are described in appendix F. Lateral flight performance data are presented in figure 28, appendix G.

Figure A. Level Flight Performance

NOTES: 1. Design gross weight and 8-TOW configuration.
2. Forward center of gravity.
3. Pressure altitude = 4000 feet.
4. OAT = 35°C.
5. Rotor speed = 276 rpm.

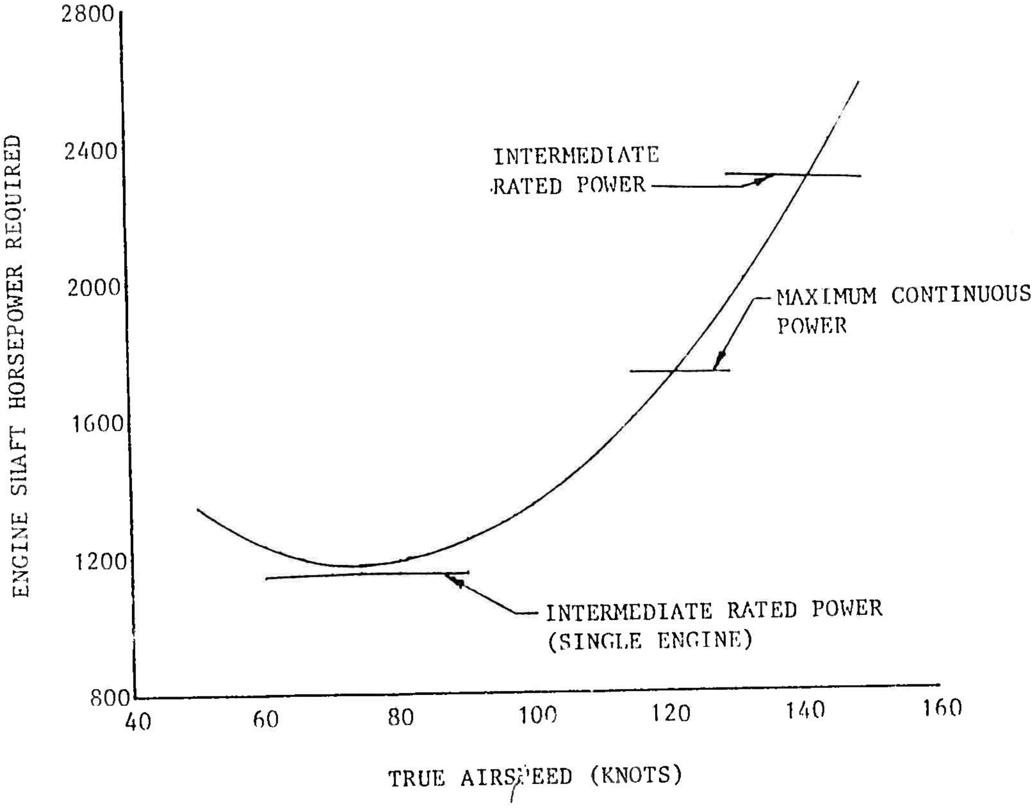

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Table 3. YAH-63 Endurance.[1]

Item	Time (min)	Flight Condition	Fuel Flow (lb/hr)	Fuel Used (lb)
1	8	Maximum continuous power	1245	166
2	16	Hover OGE at ESGW[2]	1136	303
	16	Hover OGE at ESGW minus one-half expendable ordnance	1102	294
3	15	150 KTAS at ESGW with mission stores	1370	342
	15	150 KTAS at ESGW minus one-half expendable ordnance	1342	336
4	15	Maximum endurance airspeed at ESGW	729	182
	15	Maximum endurance airspeed at ESGW minus one-half expendable ordnance	709	177
5	10	80 KTAS at ESGW with mission stores	733	122
	10	80 KTAS at ESGW minus one-half expendable ordnance	713	119
6	[3]24	Reserve at maximum range airspeed with mission stores at ESGW minus all expendable ordnance and fuel used in items 1 through 5	820	327
Total	144 (2.38 hours)			[4]2368

[1]Data corrected for instrumentation drag ($\Delta f_e = 1\ ft^2$). Fuel flow based on 5 percent conservatism. Sea-level, standard-day atmospheric conditions.
[2]ESGW: Engine start gross weight (operating weight plus primary mission payload and maximum internal fuel (16,762 pounds)).
[3]The systems specification requires 30 minutes. Fuel remaining limits the reserve to 24 minutes.
[4]Total usable fuel.

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Table 4. YAH-63 Mission Gross Weight.

Item	Weight (lb)
Weight empty per SOFR	12,188
Unusable fuel per SOFR	64
Engine oil per SOFR	29
Crew	500
Fixed useful load per SOFR (8-TOW missile tubes, 2-TOW missile launchers, gun, 2 stores pylons, gun camera, and IR decoy flares)	557
Operating weight	13,338
Primary mission (expendable ordnance) payload (8-TOW missiles and 728 30mm rounds)	1056
Primary mission fuel for 1.9 hours	1660
Mission gross weight	16,054

Table 5. Primary Mission Fuel.[1]

Item	Time (min)	Flight Condition	Fuel Flow (lb/hr)	Fuel Used (lb)
1	8	Maximum continuous power	923	123
2	19	80 KTAS at MGW[2] with mission stores	729	231
	19	80 KTAS at MGW minus one-half expendable ordnance	706	224
[3]3	3	142 KTAS at MGW with mission stores	1168	58
	3	143.8 KTAS at MGW minus one-half expendable ordnance	1166	58
4	16	Hover OGE at MGW	1146	306
	16	Hover OGE at MGW minus one-half expendable ordnance	1103	294
5	30	Reserve at maximum range airspeed at MGW minus all expendable ordannce and fuel used in items 1 through 4	731	366
Total	114 (1.9 hours)			1660

[1]Data corrected for instrumentation drag ($\Delta f_e = 1\ ft^2$). Fuel flow based on 5 percent conservatism.
[2]MGW: Mission gross weight.
[3]Maximum level flight airspeed attainable at IRP.

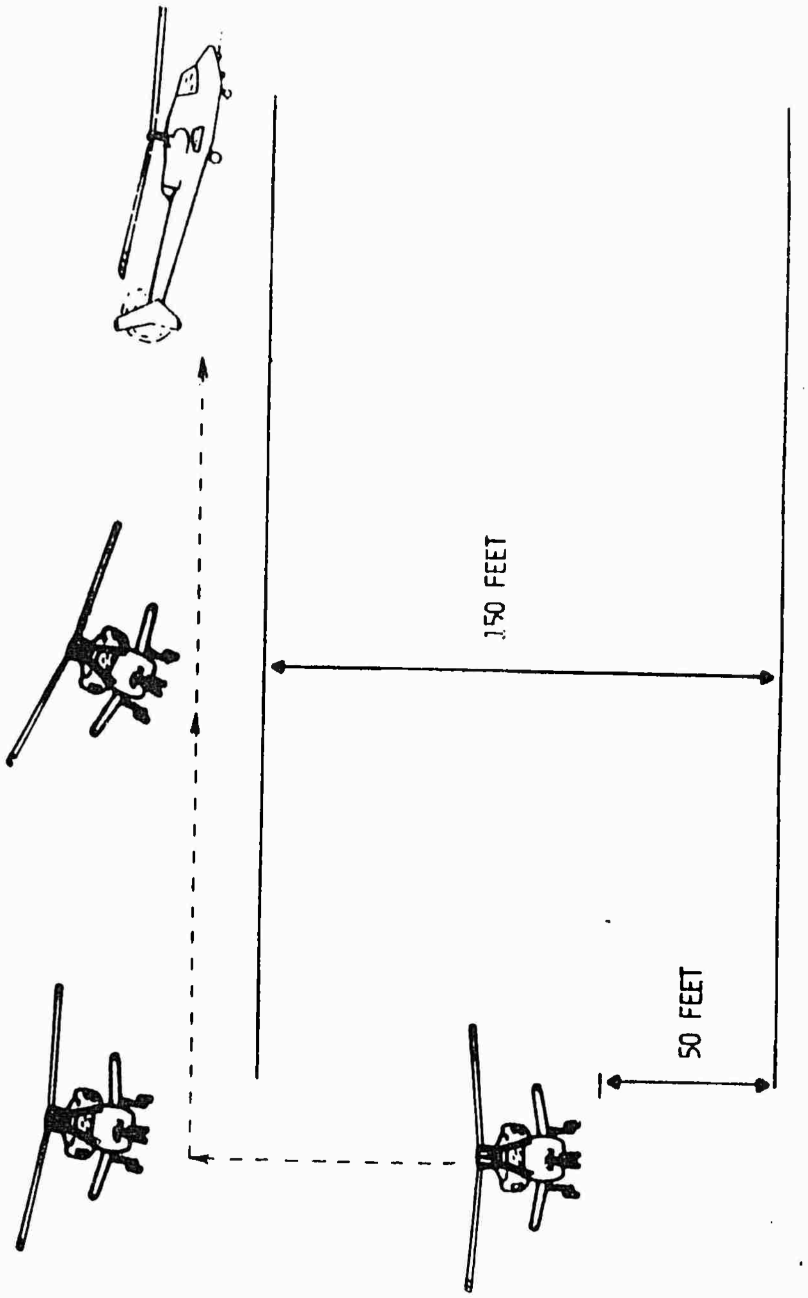

Figure B. Lateral Acceleration.

17. Average right lateral accelerations to 35 KTAS were in excess of the 0.25g minimum requirement of the systems specification. In right accelerations, full left pedal was occasionally insufficient to maintain aircraft heading. During this maneuver, tail rotor shaft horsepower (shp) in excess of the continuous limit (457 shp) was recorded. Additionally, the tail rotor shp approached the transient limit of 685 shp. However, peak shp could not be determined because of limited instrumentation range (fig. 29, app A). In left lateral accelerations, the maximum bank angle achieved was 38 degrees. Twenty-four degrees of left bank angle were required to obtain 0.25g average acceleration to 35 KTAS. Maximum lateral accelerations achieved were 0.35g left and 0.48g right. Further tests should be conducted to determine the maximum tail rotor shp in right sideward accelerations. The inadequate directional control margin in right sideward accelerations is further discussed in paragraph 58. The inability of the aircraft to maintain heading does not meet the requirements of paragraph 3.2.1.1.1.4d of the systems specification.

Vertical Displacement:

18. The vertical displacement performance of the YAH-63 helicopter was evaluated at the conditions listed in table 1. The test was conducted to determine if a 200-foot change in vertical height could be achieved within 1300 feet horizontal distance without losing more than 30 KTAS forward airspeed. The vertical displacement maneuvers consisted of a cyclic pull-up from a level flight airspeed of 140 KTAS (fig. C). An entry airspeed of 140 KTAS was used rather than the 150 KTAS specified in reference 1, appendix A, because of aircraft limitations. Performance data were recorded with ground operated space positioning equipment and on-board instrumentation. The results of these tests are presented in figures 30 and 31, appendix G.

19. During the test, a maximum peak normal acceleration of 1.82g was achieved with a vertical displacement of 200 feet within a horizontal distance of 1150 feet. The airspeed loss at 1.82g was 33 KTAS. As indicated in figure 30, appendix G, a 200-foot vertical displacement can be achieved within a 1168-foot distance with a load factor of approximately 1.8 with an airspeed loss of less than 30 KTAS. At load factors less than 1.72, more than 1300 feet were required to gain 200 feet of altitude. The vertical displacement performance of the YAH-63 helicopter is satisfactory.

20. There was a tendency for the aircraft to roll right during these maneuvers. Figure 31, appendix G, shows that a right bank angle of 11 degrees developed in spite of a large left lateral control input. This rolling tendency was not objectionable. At load factors in excess of 1.8 in pull-ups, excessive vibrations were noted. These excessive vibrations are further discussed in paragraph 114.

Autorotational Descent Performance

21. Autorotational descent performance tests were conducted at the conditions listed in table 1. To determine the airspeed for minimum rate of descent (V_{min} R/D), rotor speed was held constant at 275 rpm and data were obtained

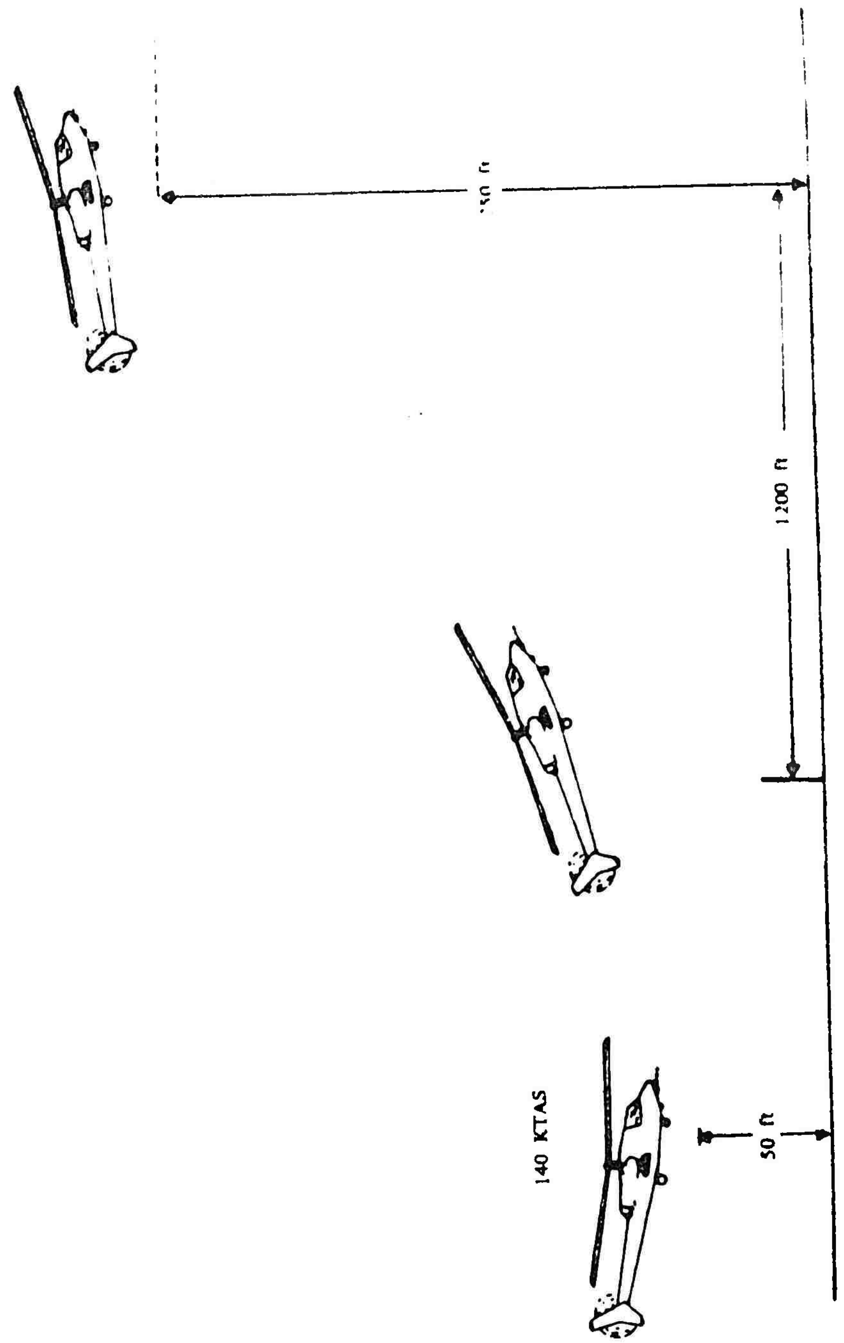

Figure C. Vertical Displacement Maneuver.

at incremental stabilized airspeeds from 28 to 82.5 KCAS. After V_{min} R/D was determined, another series of descents was conducted at that airspeed at incremental rotor speeds from 271.8 to 289.5 rpm. The results of these tests are presented in figures 32 and 33, appendix G.

22. The minimum rate of descent was 2250 ft/min at the V_{min} R/D of 61 KIAS. Figure 27, appendix G, also indicates that rate of descent does not vary significantly for airspeeds within ±5 knots of trim. Pilot comments indicated that control of airspeed in autorotation was easily maintained within a ±5 knot tolerance. The rotor speed for minimum rate of descent was 276 rpm (normal power-on operating speed). Rotor speed was easily maintained within ±3 rpm and such rotor speed fluctuations caused only minor changes in rate of descent.

23. The airspeed for maximum glide distance is beyond the 90-KCAS limit of the SOFR and could not be determined from these tests. Operationally, 90 KCAS should be used as the airspeed for maximum glide distance until further testing defines the maximum glide airspeed.

HANDLING QUALITIES

General

24. Handling qualities of the YAH-63 were evaluated using both aircraft at the high-altitude and low-altitude test sites. Numerous envelope limits were imposed during this test which would be unacceptable for an operational aircraft. The helicopter exhibited several features which will enhance accomplishment of the attack helicopter mission. The excellent field of view and forward visibility afforded by the front cockpit pilot station, along with outstanding airspeed control at low airspeeds, provide an excellent NOE capability. The excellent handling qualities in rearward flight will enhance bob-up target acquisition and tracking tasks in downwind conditions. Airspeed and rotor speed control in autorotation are excellent. Seven deficiencies were identified during the tests. At airspeeds greater than 100 KIAS, handling qualities were unsatisfactory for flight in IMC. The excessive transient rotor speed droop following a rapid power demand from a low-power condition limited the aircraft's ability to perform a quick-stop maneuver. Aircraft control following a SCAS failure at airspeeds in excess of 100 KCAS is extremely difficult because of a divergent oscillation about all three axes. The XM188 weapon system failed to fire repeatedly during these tests. These failures would severely limit the combat effectiveness of the helicopter. Inadequate control margins in three flight regimes seriously degrade aircraft handling qualities in those regimes. The requirement for the pilot to manually tune radios during NOE flight creates an unacceptable workload. A total of 23 handling qualities shortcomings were noted along with 13 instances of specification noncompliance.

Control System Characteristics

25. Pilot station control breakout forces, force versus position gradients, and range of travel were determined during ground tests with rotors stationary, SCAS OFF, and force feel system ON. Hydraulic and electrical power were provided by ground power units and all three hydraulic systems were pressurized. Airspeed effects were simulated by applying pressure to the Q sensor representing airspeeds of 50, 100, and 150 knots. Control forces were measured at the center of the cyclic and collective grips and at the hinge point of the toe brakes. The design of the cyclic force feel system (app C) is such that the cyclic forces measured during these ground tests should be representative of those experienced in flight. The collective and directional forces in flight might vary from those measured on the ground because in-flight vibrations may reduce friction. Data from these tests are presented in figures 34 through 41, appendix G, and summarized in tables 6 and 7.

26. Longitudinal and lateral control force gradients increased with increasing airspeed and were within the limits specified in the systems specification at all airspeeds. However, pilot comments indicated that cyclic forces were excessive. These comments probably resulted from the high breakout forces rather than the force gradients. These high cyclic breakout forces also contributed to the difficulty in establishing a precise trim. Control centering was satisfactory. Directional control breakout forces and force gradients are satisfactory. Collective control breakout forces, presented in table 7, are satisfactory. The longitudinal and lateral control breakout forces exceeded the maximum limit specified in paragraph 10.3.2.1.1 of the systems specification by 1.5 pounds. Additionally, the cyclic and directional forces failed to meet the requirements of paragraph 10.3.2.1.2, in that the breakout forces were not symmetrical about trim. This asymmetry in cyclic and directional breakout about trim is a shortcoming. The high cyclic control breakout forces constitute a shortcoming.

27. Cyclic control forces could be trimmed to zero either by use of the trim release button (releasing cyclic and directional forces simultaneously) or by use of the single-axis beep trim switch. No beep trim was provided for the directional control. Trimming forces to zero by use of the beep trim switch was difficult. It could not be determined if this difficulty resulted from trim rates (0.5 in./sec longitudinal and 0.85 in./sec lateral) or from the systems overshooting the desired position after the switch was released. The unsatisfactory operation of the cyclic beep trim system is a shortcoming. The trim release button functioned satisfactorily, although its poor location caused difficulties which are discussed in paragraph 78. A small stick jump was sometimes associated with activation of this button during accelerations or other maneuvers requiring large trim changes.

28. The cyclic force feel system amplified lateral vibrations in the pilot cyclic control. These vibrations were most apparent in forward flight at 300 to 400 foot-pounds (ft-lb) of transmission torque and airspeeds of 70 to 90 KIAS. When the pilot was not touching the control, the top of the cyclic grip oscillated 2 inches either side of trim. The vibrations were easily eliminated when the pilot lightly gripped the cyclic control, and were reduced in amplitude when the force

Table 6. Pilot Station Force Feel System Characteristics.

Control Axis	Q Sensor Airspeed (kt)	Breakout Force[1] (lb)	Gradient (lb/in.)	Specification Limits[2]			
				Breakout Force[1] (lb)		Gradient (lb/in.)	
				Minimum	Maximum	Minimum	Maximum
Longitudinal	Zero	2.5 aft, 3.0 fwd	1.1 aft, 0.8 fwd	0.5	1.5	0.5	5.0
	50	3.0	1.2 aft, 1.0 fwd				
	100	2.5 aft, 3.0 fwd	1.9 aft, 1.3 fwd				
	150	2.5 aft, 3.0 fwd	2.4 aft, 1.8 fwd				
Lateral	Zero	2.5 right, 2.0 left	0.5	0.5	1.5	35 to 60% of longitudinal	35 to 60% of longitudinal
	50	2.0	0.6 right 0.5 left				
	100	3.0 right, 2.0 left	0.7				
	150	2.5 right, 2.0 left	0.9 right, 0.8 left				
Directional	NA	4.0 right, 5.0 left	4.5 right, 4.7 left	3.0	7.0	2.0	10.0

[1](including friction).
[2]Army systems specification.

Table 7. Pilot Station Collective Control Breakout Forces[1].

Collective Position (% from full down)	Breakout Forces (lb)		Specification Limits[2] (lb)
	Up	Down	
Zero	6	–	1 to 10
25	7	7	
50	8	7	
75	9	7	
100	–	2	

[1]Including friction.
[2]Army systems specification.

feel system was disengaged. The oscillations of the unattended pilot cyclic control are excessive and represent a shortcoming.

Control Positions in Trimmed Forward Flight

29. Control positions in trimmed forward flight were evaluated at the conditions listed in table 1. Figures 42 through 45, appendix G, present the results of this test.

30. The longitudinal control position gradient in level flight indicates stability at all airspeeds above 60 KCAS and slight instability at lower airspeeds. The stability at high airspeeds reduced pilot effort when changing airspeeds (HQRS 3). The slight instability at low airspeeds was not objectionable. Longitudinal control margin was inadequate at high level flight airspeeds. Figure 45, appendix G, shows that the longitudinal control margin at 132 KCAS at an aft cg location (FS 198.6) was approximately 5 percent (0.5 inch). This inadequate forward longitudinal control margin at high airspeeds is a deficiency. Longitudinal cyclic control margin will not produce the 15 degrees per second (deg/sec) angular rate required by paragraph 10.3.3.1 of the systems specification.

31. Lateral control position changes with airspeed and power were minimal and satisfactory. Above 60 KCAS, the total directional control motion required to maintain balanced flight as airspeed increased was less than 1 inch. These small directional trim shifts at higher airspeeds resulted in minimal pilot effort required to maintain balanced flight. A directional control trim shift of 0.75 to 1 inch between 45 and 55 KCAS in the 8-TOW configuration was objectionable, in that considerable pilot effort was required to maintain balanced flight with small airspeed changes within that range. This directional trim shift between 45 and 55 KCAS is a shortcoming.

32. Pitch attitude change with airspeed was nearly linear and always in the proper direction (more nose-down with increasing airspeed). Attitude change with trim airspeed contributed to a reduction in pilot workload to change trim airspeed.

Static Longitudinal Stability

33. Collective-fixed static longitudinal stability characteristics were evaluated at the conditions listed in table 1 in level flight, climbs, and descents. The helicopter was trimmed at the desired airspeed in steady-heading, ball-centered flight. With the collective control held fixed, the helicopter was stabilized at incremental airspeeds greater and less than the trim airspeed. Data from these tests are presented in figures 46 through 50, appendix G.

34. The variation of longitudinal control position with airspeed was nonlinear and indicated neutral to unstable static longitudinal stability near trim at all conditions except climbs at a trim airspeed of 70 KCAS and level flight at 80 KCAS. Figure 47, appendix G, indicates stability for 10 knots above the level flight trim airspeed of 80 KCAS. This stability did not help in maintaining a trim airspeed,

however, because of nonlinearity of the longitudinal control position variation with airspeed. A problem noted during these tests, and influenced by the inadequate static longitudinal stability, was the inability to stabilize at an airspeed with control forces trimmed to zero. Allowing the aircraft to react to zero cyclic control forces at airspeeds below 100 KCAS resulted in a mildly divergent long-period oscillation (para 51). The requirement to continually counter the oscillation tendency with longitudinal control inputs will significantly accelerate pilot fatigue. Mission tasks such as cross-country or instrument flight should require minimum pilot attention for airspeed control. The inadequate static longitudinal stability at airspeeds for instrument and cross-country flight is a shortcoming. The variation of longitudinal control position and force with airspeed failed to meet the collective-fixed static stability requirements of paragraphs 10.3.3.2.7 and 10.3.4.1 of the systems specification.

35. The weak static longitudinal stability at lower airspeeds (below 60 KIAS) was considered favorable, since retrimming was not required for the airspeed changes required for NOE flight. This weak stability contributes to ease in accurately and rapidly changing airspeed during NOE flight, which is an enhancing characteristic further discussed in paragraph 77.

Static Lateral-Directional Stability

36. Static lateral-directional stability characteristics were evaluated at the conditions listed in table 1 in level flight, climbs, and descents. Sideslips were increased incrementally left and right from the trim sideslip condition. Collective control position, heading, airspeed, and trim were held constant. The results of these tests are presented in figures 51 through 55, appendix G.

37. Static directional stability, as evidenced by the variation of directional control position with sideslip, was positive. The gradient of directional control position with sideslip was approximately linear and steepened with increasing airspeed in level flight. In descending flight, static directional stability was weak about trim and became stronger with increasing sideslip angles from trim. Static directional stability is satisfactory.

38. Dihedral effect, as indicated by the variation of lateral control position with sideslip, was positive. The dihedral effect increased as airspeed or power was increased but decreased at the large sideslip angles. The dihedral effect is satisfactory.

39. Side force, as indicated by the variation in bank angle with sideslip, was weak at sideslip angles near trim. Because of the weak side forces about trim, the pilot was unable to detect small sideslip excursions. Varying sideslip angles will affect rocket firing accuracy. Sideslip excursions also degraded instrument flight capability because of the large pitch with sideslip coupling in this aircraft, as discussed in the next paragraph. While entering turns during IMC flight, large sideslip excursions can occur with no indication to the pilot. When this happens, airspeed control is difficult (para 87). Weak side forces near trim constitute a shortcoming.

40. Large longitudinal trim shifts with sideslip were encountered during these tests. Forward longitudinal control displacement and force were required to balance the nose-up pitching moment resulting from right sideslip, and an aft longitudinal control displacement and force were required to balance the nose-down pitching moment resulting from left sideslip. These trim shifts increased with increasing airspeed. At 13 degrees right sideslip at 125 KCAS in level flight, the forward longitudinal stop was contacted. The excessive pitching moment with sideslip is unsatisfactory and is a shortcoming. The inadequate forward longitudinal control margin in right sideslips at high airspeeds is a deficiency. The longitudinal trim change due to sideslip did not meet the requirements of paragraphs 10.3.3.1 and 10.3.4.5 of the systems specification.

Maneuvering Stability

41. Maneuvering stability characteristics were evaluated under the conditions listed in table 1. The variation of longitudinal control position and control force with normal acceleration was determined by trimming the aircraft in coordinated level flight at a desired trim airspeed and then stabilizing at incremental bank angles in steady turns, both left and right. Collective control and airspeed were held constant and the helicopter was allowed to descend during the maneuver. Data were recorded at each stabilized bank angle. The results of the maneuvering stability evaluation in turning flight are presented in figures 56 through 61, appendix G.

42. Stick-fixed stability was stable (increased aft control position with increased load factor) and essentially linear at all conditions tested. Stick-free stability was also stable at all conditions evaluated, *ie*, an increased pull force was required to increase load factor. Control position and force characteristics, summarized in table 8, are satisfactory. The average longitudinal control force versus normal acceleration gradient was less than the 6.0 lb/g minimum specified in paragraph 10.3.6.1.2 of the systems specification, except during right turns at 60 KCAS.

43. The load factors presented in figures 56 through 61, appendix G, were limited by the ability of the pilot to remain within the test criteria: no sideslip or airspeed variation and data taken within the test band of 6000 to 8000 feet density altitude. At the rate of descent experienced during 1.8g turns, the pilot had approximately 45 seconds to stabilize bank angle, sideslip, and airspeed, and remain within the allowable density altitude band. Although 1.8g turns are the maximum presented during the low-speed maneuvering stability tests, higher load factors were obtained but did not meet the test criteria for stabilized data points. At higher airspeeds (120 KCAS), the SOFR limit load factor (2.5g at 7000 feet) was easily attained. However, because of the limitations for stabilized data points mentioned previously, the maximum load factor presented is 2.15. The maneuvers were not limited by control power or vibration at the test conditions.

44. At the higher airspeed and bank angle combination, there was a significant increase in the 2-per-rotor-revolution (2/rev) and 4/rev vibration levels in the lateral and longitudinal axes. The vibrations at 120 KCAS at load factors in excess of

Table 8. Maneuvering Flight Summary.[1]

Calibrated Airspeed (kt)	Gross Weight[2]	Direction of Turn	Longitudinal (in./g)	Control Gradient (lb/g)
58	Light	Left	4.3	5.8
59	Light	Right	4.9	6.6
97	Light	Left	1.7	3.1
95	Light	Right	2.4	4.6
121	Light	Left	0.9	1.8
121	Light	Right	1.0	2.2
61	Heavy	Left	3.7	5.0
61	Heavy	Right	5.0	7.0
97	Heavy	Left	2.3	4.1
96	Heavy	Right	2.7	5.0
122	Heavy	Left	1.0	1.4
121	Heavy	Right	1.1	3.3

[1]Constant-collective, constant-airspeed stabilized turns at approximately 6500 feet density altitude; average longitudinal cg 298.6 inches (aft); lateral cg 0.2 left; 8-TOW configuration; rotor speed 276 rpm.

[2]Average gross weight: Light - 15,240 pounds; heavy - 16,060 pounds.

2 were extremely uncomfortable to the pilot. The excessive 2/rev and 4/rev vibration levels at high airspeeds and load factors are further discussed in paragraph 114.

Dynamic Stability

45. The longitudinal and lateral-directional dynamic stability characteristics were evaluated in an OGE hover and in forward flight with SCAS ON and OFF. Tests were conducted at the conditions listed in table 1. Data from these tests are presented in figures 62 through 64, appendix G.

46. Short-term gust response characteristics were obtained by rapidly displacing the desired control 1 inch from trim for a duration of 0.5 second and returning the control to the trim position while recording subsequent aircraft response. Test results are summarized in table 9. A time history of a longitudinal pulse input is presented in figure 62, appendix G. The short-period response of the helicopter was deadbeat below 90 KCAS. However, as airspeed was increased above 90 KCAS there was a tendency for a coupled short-period response in all three axes when individually excited. This short-period response above 90 KCAS will increase pilot workload required to maintain precise attitude in turbulent conditions and is a shortcoming. The short-term response characteristics of the helicopter failed to meet the requirements of paragraph 10.3.4.2.1d of the systems specification, in that small-amplitude, short-period residual oscillations affect mission capability.

47. Lateral-directional oscillation (Dutch-roll) characteristics were evaluated by releases from steady sideslips in level flight at 90 KCAS, SCAS ON and OFF, and at 120 KCAS, SCAS ON. For these conditions, there was no evidence of a significant lateral-directional oscillation. In all cases, roll attitude returned to trim with no overshoot. Sideslip returned to trim initially and in all cases stabilized within 2 degrees of trim.

48. When SCAS was disengaged in trimmed, coordinated level flight at 120 KCAS, the helicopter demonstrated a divergent oscillation in all three axes (fig. 63, app G). This response will induce excessive pilot workload, and possibly vertigo, during recovery from a loss of SCAS at higher airspeeds under IMC. This divergent oscillation in pitch, roll, and yaw contributes to the deficiency discussed in paragraph 96.

49. Spiral stability was evaluated in level flight at 90 KCAS, SCAS ON and OFF, and at 120 KCAS, SCAS ON. Bank angles were established by using directional control only and then returning the directional control to trim. From a 10-degree right bank angle at 90 KCAS, SCAS ON and OFF, the aircraft demonstrated neutral spiral stability. From a 10-degree left bank under the same conditions, the spiral mode was weakly convergent, with a time to half amplitude of approximately 8 seconds. At 120 KCAS, SCAS ON, right wing low, spiral stability was weakly convergent (time to half-amplitude, 10 seconds) and left spiral stability was neutral. The spiral stability characteristics were satisfactory.

Table 9. Short-Term Response[1]

Calibrated Airspeed (kt)	Axis Excited	Period (sec)	Damping Ratio	Cycles to One-Half Amplitude
Hover	All	-	Deadbeat	-
55	All	-	Deadbeat	-
90	Lateral	-	Deadbeat	-
90	Longitudinal	-	Note[2]	-
90	Directional	-	Note[2]	-
120	Left lateral	-	Note[2]	-
120	Right lateral	-	Note[2]	-
120	Forward longitudinal	-	Note[2]	-
120	Aft longitudinal	4.25	0.29	0.36
120	Directional	-	Deadbeat	-

[1]Approximate test conditions: Mission gross weight, aft cg, 8-TOW configuration, SCAS ON, level flight.
[2]One overshoot observed during controls-fixed delay prior to recovery.

50. Adverse/complementary yaw characteristics were evaluated at 90 KCAS, SCAS ON and OFF, and at 120 KCAS, SCAS ON, using cyclic-only turns and roll reversals. No adverse or complementary yaw response was observed.

51. Longitudinal long-term response characteristics were evaluated in level flight. Test conditions and results are summarized in table 10. At 87 KCAS, the long-period response was lightly damped with SCAS OFF and divergent with SCAS ON (3.68 cycles to double amplitude). A time history of this divergent long-period response is presented in figure 64, appendix G. Of particular interest is that at 87 KCAS, the long-period oscillations (SCAS ON and OFF) began with no control inputs. At 120 KCAS, SCAS ON, a 10-knot disturbance from trim was required to excite the long-term response. In all cases, it was readily recognized and easily corrected, using either outside references or cockpit instruments. However, it is anticipated that airspeed excursions generated by the long-term response in IMC will increase pilot workload. The long-term response characteristics are a shortcoming and fail to meet the requirements of paragraph 10.3.4.2.1e of the systems specification in that they will be objectionable during IMC flight.

Table 10. Longitudinal Long-Term Response Characteristics[1]

Flight Condition	Excitation	Period (sec)	Damping Ratio
87 KCAS, SCAS ON	None	29.0	-.03
87 KCAS, SCAS OFF	None	21.5	.22
120 KCAS, SCAS ON	10 knots from trim	Deadbeat	-

[1]Approximate test conditions: Mission gross weight, aft cg, 8-TOW configuration.

52. Coupling between collective control inputs and roll attitude was observed following rapid 1-inch inputs to the collective at 90 KCAS, SCAS ON. With all other controls fixed, a 1-inch up-collective input produced a right roll of 16 degrees in 5 seconds; a 1-inch down-collective input produced a left roll of 13 degrees in 5 seconds. Although the resultant roll rates were mild, they will increase pilot workload during instrument flight. The excessive rolling moments created by collective input are a shortcoming.

Controllability

53. Controllability tests were conducted to evaluate the control power, response, and sensitivity characteristics of the aircraft. Controllability was measured in terms of aircraft attitude displacements (control power), angular velocities (control response), and angular accelerations (control sensitivity) about an aircraft axis

following a rapid step control input of a measured size. Following the input all controls were held fixed until the maximum rate was reached or recovery action was necessary. The magnitude of the inputs was varied by using an adjustable rigid control fixture. Controllability tests were conducted under the conditions listed in table 1. Controllability characteristics are shown in figures 65 through 80, appendix G.

54. Longitudinal controllability characteristics are presented in figures 65 through 69, appendix G. Control power (pitch attitude change after 1 second following a 1-inch input) varied from 2.6 degrees at 95 KCAS with a forward input to 4.1 degrees at 126 KCAS with both forward and aft inputs. Longitudinal control response varied from a minimum of 8 degrees per second per inch (deg/sec/in.) of control displacement at 95 KCAS with a forward cyclic input to 13 deg/sec/in. with a forward input in a hover. Control sensitivity was the highest in a hover (12 deg/sec^2/in.) with a forward input and at a minimum with a forward input at 95 KCAS (9.1 deg/sec^2/in.). The longitudinal controllability characteristics permitted smooth, precise control of aircraft attitude and airspeed at a hover and in low-speed forward flight.

55. Lateral controllability characteristics are presented in figures 70 through 74, appendix G. Control response in roll was low, with the maximum response occurring at 95 KCAS with a right cyclic input (12.4 deg/sec/in.). Roll control response in a hover was 10.5 deg/sec/in. with both left and right lateral control inputs. The low control response resulted in pilot comments that the aircraft felt sluggish in roll. The low control response in roll is a shortcoming.

56. Directional controllability characteristics are presented in figures 75 through 80, appendix G. Maximum directional control power, response, and sensitivity occurred in a hover. Pilots have lauded the ease of maintaining directional control while hovering. The combination of excellent directional and longitudinal controllability characteristics resulted in an extremely responsive aircraft in low-speed flight, which was substantiated by pilot comments during NOE flying.

Lateral Acceleration

57. Lateral acceleration maneuvers were accomplished from a vertical maximum performance climb under the conditions listed in table 1 and discussed in paragraph 16. A representative time history of the lateral acceleration maneuver is presented in figure 29, appendix G.

58. Desired bank angle was easily acquired and maintained throughout the maneuver. Pitch control required constant correction, resulting in a pitch attitude oscillation. The most difficult aspect of the maneuver was maintaining a heading perpendicular to the flight path. As the aircraft accelerated in sideward flight, directional control required to maintain heading increased. As depicted in figure 29, appendix G, less than 6 seconds after initiating the right lateral acceleration the directional control was against the left stop. Even with the control against the stop, the right yaw increased. Recovery from the maneuver was accomplished by

adding directional control in the direction of flight, allowing the aircraft to continue in forward flight. The inadequate directional control margin in right lateral accelerations is a deficiency and fails to meet the requirements of paragraph 10.3.9.1.2 of the systems specification.

Ground Handling Characteristics

59. Ground handling characteristics were evaluated throughout the test program. Taxi operations were confined to paved surfaces and a dry lake bed in winds up to 15 knots. It was possible to initiate forward motion using cyclic only (to within 20 percent of full forward). However, considerable main rotor flapping was experienced and airframe vibration level was high. These vibrations could be reduced by increasing collective to initiate forward movement. Toe brakes were required to stop the helicopter, since airframe vibrations became uncomfortable as the cyclic was displaced aft of its neutral position. To achieve full braking action, the pilot was required to remove his feet from the rudder pedals, shift them to the top of the pedals, and then rotate his feet forward to a position approaching the horizontal. Additionally, excessive force on the brakes was required to stop the aircraft. During braking, directional control was difficult (HQRS 5). The awkward foot movement required for toe brake application and the marginal effectiveness of the brakes are shortcomings.

60. Taxi turns at normal taxi speed (fast walk) with neutral cyclic generally resulted in an uncomfortable outside-wing-down attitude (2 to 3 degrees). The extent to which the wing dipped was a function of taxi speed and oleo servicing (*ie*, the higher the oleo pressure, the faster the turn could be made without producing an excessive wing-low attitude). Lateral cyclic, when applied in the direction of the turn to compensate for the wing-low condition, tended to increase airframe vibrations. As a result, with underserviced oleos, pilot workload during taxi turns was extremely high; with properly serviced oleos, workload was reduced considerably (HQRS 3). Precise directional control during changes in heading required negligible pilot effort.

61. Normal procedures require the nose wheel to be unlocked for all taxi turns and locked during takeoff and landing and when the helicopter is parked. To unlock the nose wheel, the pilot must reach down and forward past the cyclic to push in the nose wheel lock handle on the console. To move the handle, the pilot must first have the helicopter in forward motion and then must vary heading with the rudder pedals while applying pressure to the handle. The awkward procedure required to unlock the nose wheel produces an excessive workload during the initial phases of taxiing and is a shortcoming.

Takeoff and Landing Characteristics

62. Takeoff and landing characteristics were qualitatively evaluated throughout the test. Operations were conducted in surface winds varying from calm to maximum gusts of 25 knots. The force feel system (FFS) and SCAS were ON and nose gear locked for the test. The evaluation included lift-off to and touchdown

from a hover, normal takeoffs and landings, simulated confined area takeoffs and landings, and running takeoffs and landings.

63. The lift-off to and touchdown from a hover were characterized by small changes in control position and helicopter attitude. The hover attitude for the aft cg configuration and calm winds was 2 degrees left wing down and 2 degrees nose-up. A soft touchdown from a hover often resulted in uneven depression or sticking of the main gear oleos. A stuck oleo resulted in bank attitudes up to 5 degrees and made ground taxi slightly more difficult. Even compression of the oleos could be obtained by rapidly decreasing the collective once the helicopter was light on the gear. This technique was normally adequate to cause simultaneous compression of the oleos but also resulted in main rotor speed as high as 103 percent. The lift-off to a hover often resulted in uneven extension of the oleos, thereby causing slight roll oscillations. Uneven compression/extension of the oleos on consecutive takeoffs and landings was normally an indication that the gear needed servicing. The sticking and uneven compression/extension of the main gear oleos was annoying during takeoffs and landings, increased pilot workload during ground taxi, and is a shortcoming.

64. Normal transitions to forward flight from a hover were characterized by a pitch-down tendency as the helicopter accelerated through translational lift. Aft cyclic and a slight increase in power were required to prevent the helicopter from descending. Once through translational lift, forward longitudinal movement was required to continue the acceleration. At high density altitudes a left lateral input of approximately 1 inch was also required. As airspeed was increased to 60 KIAS power was reduced, and right pedal was required. The large directional control shift noted during a normal takeoff in the 8-TOW configuration is a shortcoming (para 31).

65. Normal transitions from forward flight to a hover in the 8-TOW configuration were characterized by a pitch-up tendency between 55 and 45 KCAS. This characteristic is evidenced by the control position plots for trim forward flight (figs. 43 and 45, app G). The deceleration prior to this airspeed required a gradual aft movement of the longitudinal control position. During deceleration through 55 KCAS a forward longitudinal control input was required to counter the pitch-up tendency. This forward position of the cyclic was maintained until the helicopter had decelerated through translational lift, after which aft cyclic was required to continue to decelerate to a hover. The longitudinal control reversal to counter the change in helicopter attitude during approach to a hover was less than 1/2 inch; however, it resulted in an increase in pilot workload during the landing, prevented a smooth deceleration to a hover, and is a shortcoming.

66. Confined area takeoffs and landings were simulated by executing vertical climbs with a transition to forward flight and steep approaches to a hover. The vertical climb portion of the takeoff required only the addition of power while maintaining the hover attitude. The transition from vertical climb to forward flight required a smooth application of forward cyclic to start the forward acceleration. Minimal pilot compensation was required during this transition to prevent the

helicopter from descending (HQRS 3). The steep approach was similar to the normal approach, in that forward cyclic was required between 55 and 45 KCAS to prevent the tail from dipping.

67. Running takeoffs and landings were made from a hard-surfaced runway. Running takeoffs were made by increasing the collective sufficiently to establish a forward roll and then allowing the aircraft to fly off the ground. Lift-off speed was varied from 20 to 45 knots. The helicopter control position and attitude changes were negligible and pilot compensation was not a factor (HQRS 2). Running landings were made by flying the helicopter to the ground and touching down at airspeeds below 50 KIAS. Helicopter attitude during the approach was held constant and collective was used to control the descent rate. Pilot compensation was not a factor during the touchdown phase of the running landing (HQRS 2). Once the helicopter was on the ground, the feet had to be repositioned to operate the toe brakes. This contributes to the shortcoming discussed in paragraph 59.

Low-Speed Flight Characteristics

68. Sideward, rearward, and low-speed forward flight tests were conducted at approximately 4000- and 11,000-foot density altitudes at the conditions listed in table 1. Tests were conducted in winds of 3 knots or less at a wheel height of 10 feet. Data are presented in figures 81 through 84, appendix G. A ground pace vehicle was used as an airspeed reference. The tests were greatly complicated at 11,000 feet because of insufficient engine power margin. The engine measured gas temperature ($T_{4.5}$) limit was reached at about 20 KTAS in rearward and left sideward flight. To obtain left sideward flight airspeeds in excess of 20 KTAS, it was necessary to accelerate to the desired airspeed in forward flight and then make a pedal input to turn the aircraft into sideward flight. To obtain rearward flight airspeeds in excess of 20 KTAS, the aircraft was allowed to descend from a high hover as it accelerated rearward. In both sideward and rearward flight, power required for level flight decreased with increasing airspeed above 20 KTAS, and stabilized data at higher airspeeds could be obtained. At the 4000-foot density altitude, adequate power margins were available at all test conditions.

69. Adequate control margins in all axes were available throughout the tests at both altitudes (greater than 10 percent of full travel remaining). However, it should be noted that right sideward flight at the high-altitude site was limited to 10 KTAS by the SOFR. This is an unacceptable limit for operational aircraft. Also, only 1 inch of aft longitudinal cyclic travel remained at 45 KTAS in left sideward flight at the high-altitude site. Although this margin was considered adequate during these tests (cg location 2.9 inches aft of the forward limit), at a forward cg location there may be inadequate longitudinal control remaining. A similar large aft longitudinal control displacement is required in left sideward flight at the lower altitude. Although considered adequate during these tests, the aft longitudinal control margin in left sideward flight will not produce the 15 deg/sec angular rate required by paragraphs 10.3.3.1 and 10.3.9.1.1 of the systems specification.

70. Large control trim shifts about all axes were encountered at translational lift airspeed (approximately 20 KTAS). The magnitude and abruptness of the trim shifts varied somewhat with altitude and direction of flight. The abrupt trim shifts would greatly increase pilot workload to hover in a gusty wind of approximately 20 knots. The longitudinal trim shifts in left sideward and low-speed forward flight were particularly annoying. Trim shifts as large as 1.3 inches were encountered (fig. 81, app G). The abrupt longitudinal trim shifts in left sideward and low-speed forward flight are a shortcoming.

71. Handling qualities in rearward flight were excellent. Airspeed and attitude control were easily accomplished to the limit rearward airspeed of 45 KTAS. During pedal turns when recovering from rearward flight, aircraft attitudes and yaw rate were easily controlled. The excellent handling qualities in rearward flight will significantly enhance the ability to accomplish tasks when hovering in downwind conditions (*eg*, air taxiing and target acquisition with a tail wind).

Power Management

72. The in-flight power management of the YAH-63 was qualitatively evaluated throughout the test. The YT700 control system was designed to automatically control the functions of fuel metering, compressor bleed and variable geometry control, power modulation for rotor speed control, and engine overspeed protection. The system also incorporates control features for torque matching and overtemperature protection. Pilot control is provided by the power available spindle (PAS), load demand spindle (LDS), and the rotor speed reference input (trim wheel). The PAS function is similar to that of the engine condition lever on previous helicopters, *ie*, to adjust the power available from the engine. The LDS position is a function of collective pitch and provides collective compensation to reduce rotor transient droop. The rotor speed reference input is a single electrical potentiometer on the pilot collective control that adjusts both engine speeds simultaneously. The specific problems of power management noted during this evaluation were the difficulty in precisely selecting the desired rotor speed, the poor rotor speed governing characteristics, and the slow engine response rates that allowed excessive rotor speed droop.

73. The difficulty associated with selecting a desired rotor speed was noted throughout the test. The engine response to an adjustment of the engine trim wheel was immediate; however, the resulting rotor speed oscillations required several seconds to damp sufficiently to determine if the desired rotor speed was obtained. This problem was especially objectionable during single-engine failure tests when the operational engine was at or near topping power. The procedure for an engine failure includes the adjustment of rotor speed to 279 rpm (101 percent) for single-engine flight. Following a single-engine failure, the pilot must make multiple adjustments of the engine trim to obtain the desired rotor speed. The difficulty in selecting the desired rotor speed (HQRS 5) is a shortcoming (para 104).

74. Poor rotor speed governing characteristics were noted throughout the test, but were most objectionable during maneuvering flight. The requirement for rapid

and frequent collective adjustments during NOE flight resulted in continuous rotor speed fluctuations outside the normal operational range. Although the fluctuations were within the transient rotor speed limitations, they actuated the rotor speed warning light. During NOE flight, the pilot's attention must be concentrated almost exclusively outside the helicopter; therefore, the illumination of the rotor speed warning light causes division of the pilot's attention, which may jeopardize safety or mission effectiveness. The poor governing characteristics that allow rotor speed fluctuations outside the normal operational range are a shortcoming.

75. The slow engine response rate was evident during maneuvers requiring high power demands from low power conditions. This was especially evident during quick stops from high-speed flight where the collective was completely lowered during the flare to minimize the stopping distance. As flare effectiveness was lost, a rapid power application was required to prevent excessive descent rates. This rapid power demand resulted in transient rotor speeds as low as 90 percent. Preliminary test results indicated that pilot techniques necessary to prevent rotor droop would result in excessive stopping distances and seriously affect the combat effectiveness of the helicopter. Excessive transient rotor speed droop following a rapid power demand from a low-power condition is a deficiency. This problem is further discussed in paragraphs 81 and 136.

Mission Maneuvering Characteristics

76. The mission maneuvering capability of the YAH-63 was evaluated by conducting NOE flight, contour flight, high-speed low-level flight, bob-up maneuvers, accelerations, and decelerations. The tests were conducted at the conditions listed in table 1.

77. Nap-of-the-earth flights, contour flights, and high-speed low-level flights were evaluated over rolling desert terrain with sparse vegetation and over mountainous, wooded terrain. The field of view (area free of obstructions) from the forward cockpit was much improved over the rear seat of present attack helicopters. The improved field of view resulted in better depth perception and was instrumental in attaining maximum capability from the helicopter and maximum utilization of terrain for cover and concealment. Visibility was also improved, in that canopy distortion and reflections on the canopy were greatly reduced. Additionally, the weak static longitudinal stability at the lower airspeeds (para 35), coupled with the improved depth perception, enhanced the pilot's ability to control airspeed and to maneuver. The resulting pilot confidence reduced the stress and fatigue of NOE, contour, and low-level flight. The improved field of view and visibility from the forward cockpit and the ability to accurately and rapidly control airspeed during NOE flight are enhancing characteristics that will significantly improve the mission maneuvering capability of the helicopter.

78. During the NOE, contour, and low-level flight evaluation, airspeed was varied from below translational lift to never-exceed airspeed (V_{NE}). The large longitudinal control shift at translational lift and the large shifts from forward to lateral flight resulted in very high cyclic control forces, especially during NOE flight. As an

example, from 30 KTAS forward flight, a rapid right pedal turn would effectively put the helicopter in left lateral flight. The longitudinal control shift during this maneuver required an aft cyclic input of 2.5 inches. The forces resulting from such large cyclic movements were in excess of 5 pounds and prolonged operation against such forces was fatiguing. To relieve these high forces, the force trim release button (photo 1) had to be depressed at all times. Disengagement of the FFS switch on the automatic flight control system (AFCS) panel (photo 2) resulted in much higher forces. The uncomfortable position of the trim release button under the right thumb position on the cyclic grip, and the requirement to constantly depress the button, resulted in rapid fatigue of the right hand. Additionally, the same thumb was required to operate the intercom and radio transmit buttons, and therefore the constant depression of the trim release button interfered with necessary radio and intercom operations. The lack of a feature to relieve all control forces without the requirement to hold the trim release button down accelerated pilot fatigue, interfered with radio/intercom operations, and is a shortcoming.

79. Bob-up maneuvers were evaluated to determine the ability to unmask, simulate target acquisition, and remask. The favorable helicopter stability, directional control (para 56), and pilot field of view and visibility in a high OGE hover permitted target acquisition with minimum pilot effort. Additionally, the vertical climbs and descents for unmasking and masking were easily accomplished with small control trim changes (HQRS 2). Settling with power was not encountered.

80. Maximum acceleration was evaluated to determine the helicopter dash capability. The dash was initiated from a stopped position with wheels on the ground. The dash was performed by rapidly increasing collective to maximum power while maintaining low-level flight over a 3000-meter straight course. Power had to be reduced between 2000 and 3000 meters to prevent airspeeds in excess of V_{NE}. Helicopter control during the dash was easily maintained (HQRS 2). Torque tended to decrease slightly as airspeed increased and several increases in collective position were required to maintain maximum power; however, satisfactory performance required only minimal compensation (HQRS 3).

81. Quick stops were evaluated from 65 and 115 KIAS to determine handling qualities during this mission task. Constant altitude was maintained during the deceleration. Two problems were noted during the quick-stop maneuvers that increased pilot workload and prevented maximum deceleration of the helicopter. The initiation of the quick stop, especially from the high airspeed, resulted in a tendency for the rotor to overspeed. A rapid reduction in collective, coupled with a rapid flare, resulted in rotor speeds in excess of the allowable transient power-on limit (105 percent). To prevent rotor overspeed, collective was reduced more slowly and the flare was gradual. This technique was adequate to prevent rotor overspeed; however, it resulted in a significant increase in stopping distance. The termination of the quick stop resulted in the second problem of excessive transient rotor speed droop. Once the effectiveness of the flare was lost, a rapid application of power was required to prevent a descent and to further decelerate. The acceleration of the engines was too slow to meet the power demands. As a result, rotor speed drooped below the allowable transient power-on limit

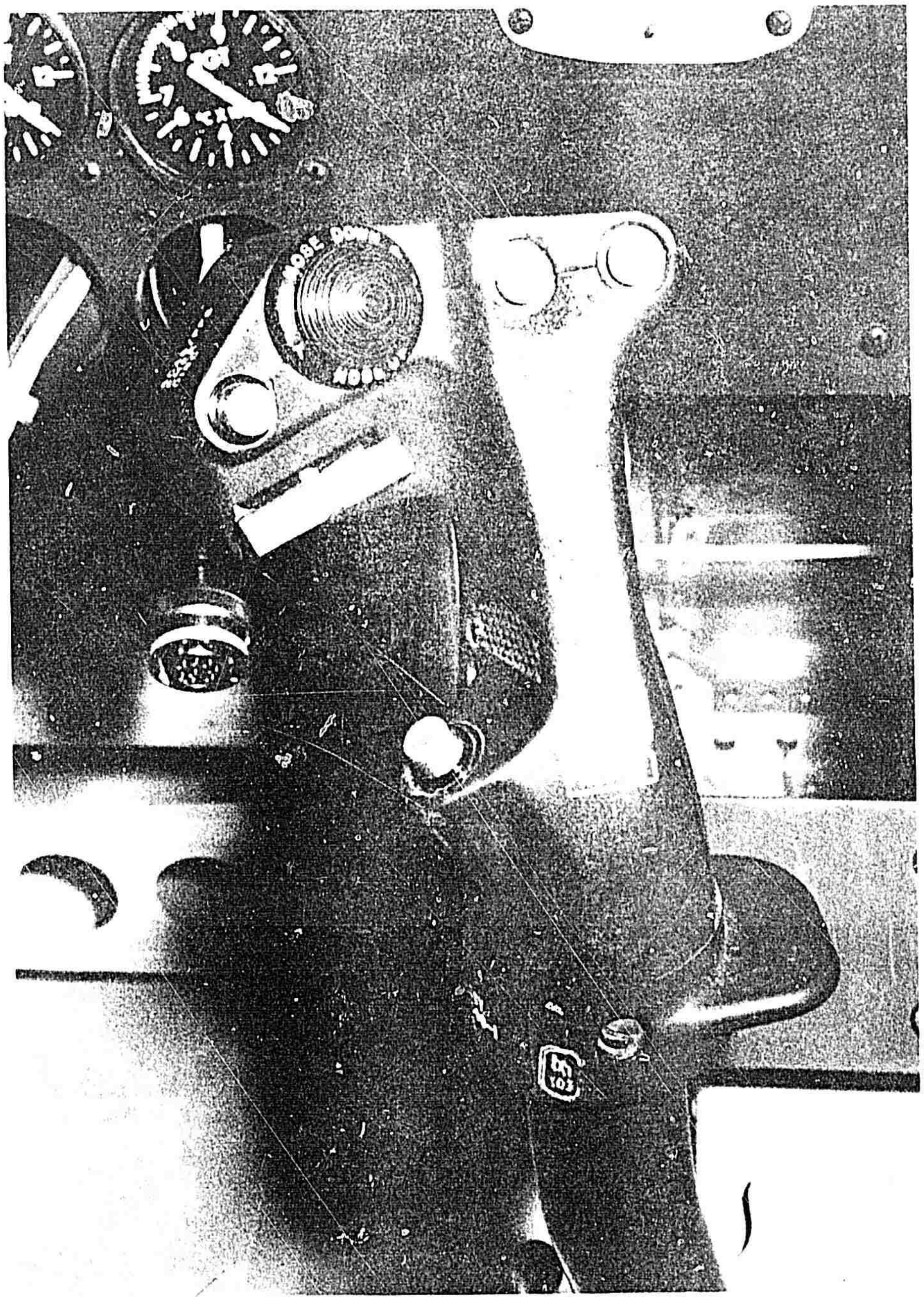

Photo 1. Pilot Cyclic Grip

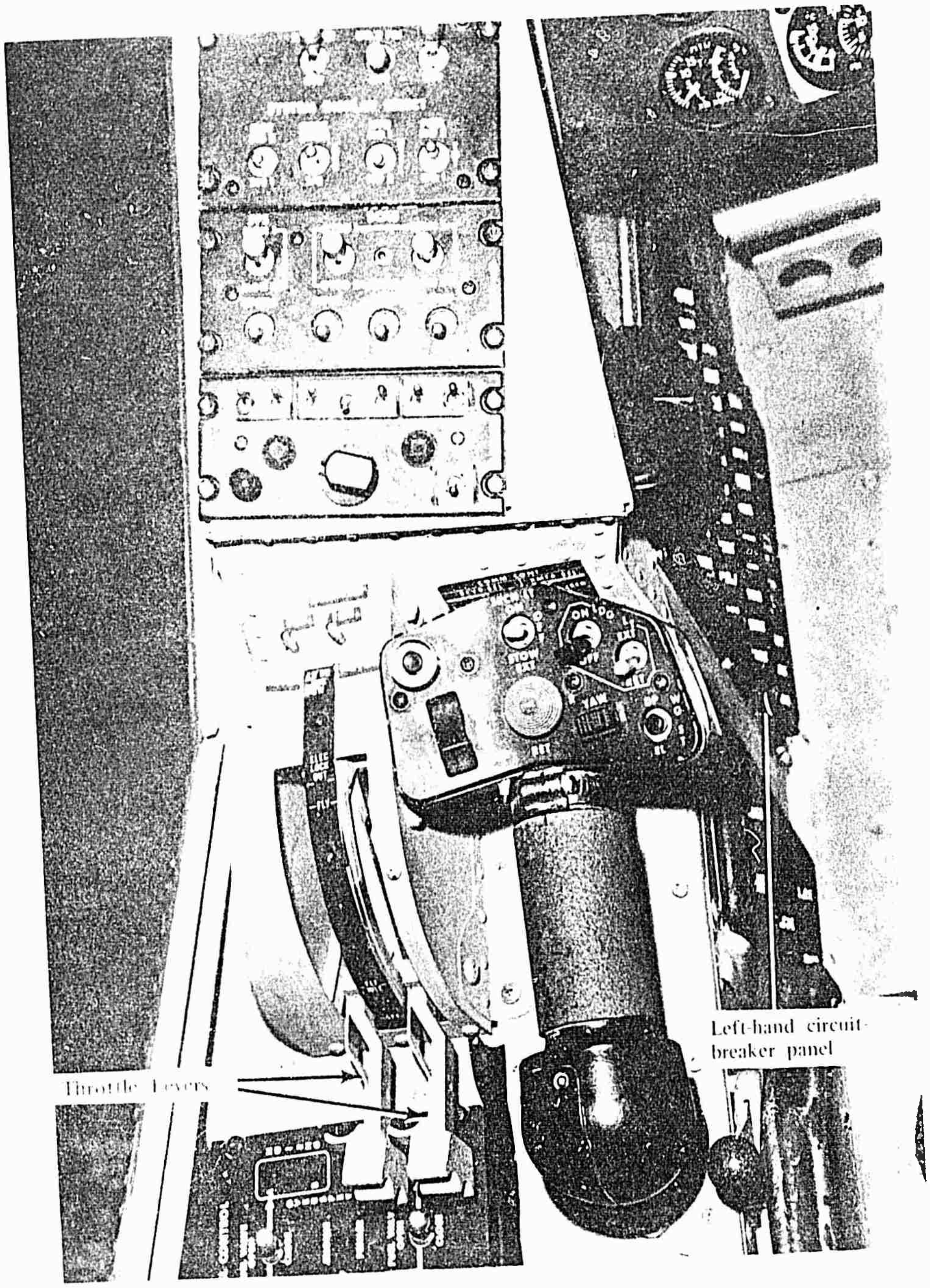

Photo 2. Left Side Pilot Cockpit.

(94 percent). To prevent rotor speed from drooping, a gradual application of power was required to anticipate the loss of flare effectiveness. This technique also resulted in an increased stopping distance. The excessive transient rotor speed droop following a rapid power demand from a low-power condition is a deficiency.

Weapons Firing

General:

82. Aircraft SN 74-22246 was used to conduct the weapons firing tests. The weapons fired were the XM188 (30mm) weapon system and FFAR. The firing was conducted under the conditions listed in tables 11 and 12. The TOW missile weapon system was not installed on the test aircraft.

XM188 Weapon System Firing:

83. During the XM188 firing tests, 348 rounds of 30mm ammunition were fired. Except when abnormal stoppages occurred, all data for each firing condition were collected using single 25-round bursts. The firing tests produced very low noise and airframe vibration levels. Only in the 90-degree left azimuth position was any significant aircraft response to weapons firing observed. With the weapon in this side firing position, the initial rounds of the 25-round burst caused a slight roll and yaw response which were easily controlled under all test conditions (HQRS 3). No noticeable aircraft response resulted from the abrupt slewing of the turret to its firing position. Weak gun-gas odor was detected during all firing conditions in which the gun barrels were elevated. The environmental control system (ECS) was ON throughout the tests and the noticeable gas odor dissipated within 15 seconds. Firing stoppages were experienced five times during the tests. The first and fifth resulted from a fouled gun drive motor. The absence of a fairing to protect the gun from dust and debris seriously degraded gun reliability while operating from unprepared sites. The second stoppage resulted when the feeder delinker solenoid failed to disengage, the third resulted from a slipping case ejector clutch, and the fourth occurred when a link jammed in the feeder delinker. The repeated failure of the XM188 weapon system is a deficiency which will severely degrade mission effectiveness.

Folding Fin Aircraft Rocket Firing:

84. Thirty-two rockets (8 rockets from each of the four M200A1 pods) were fired in a ripple mode at 90 KIAS in level flight without any significant effects on aircraft handling qualities. Although a slight deceleration was detected, it was not objectionable. Under all other test conditions, rockets were fired with little, if any, noticeable aircraft response. Eight rockets were fired in a ripple mode with the pods depressed 20 degrees and the aircraft hovering OGE. A similar firing was accomplished with the pods elevated 16 degrees. Aircraft response was mild and OGE hover was easily maintained with minimal pilot effort under both test conditions (HQRS 3).

Table 11. Firing Conditions.
XM188 (30mm Nose Turret)

Indicated Airspeed (kt)	Gun Azimuth (deg)	Gun Elevation (deg)	Ammunition Expended (rnd)
Hover	Zero	Zero	25
Hover	Zero	10 up	25
40	Zero	Zero	25
40	Zero	10 up	25
40	Zero	45 down	25
120	Zero	Zero	25
120	Zero	10 up	25
120	Zero	45 down	30[1]
Hover	90 left	Zero	30[1]
Hover	90 left	10 up	24[2]
40	90 left	Zero	14[3]
40	90 left	10 up	25
40	90 left	45 down	25
120	90 left	Zero	25

[1]Abnormal stoppage at 5 rounds on first attempt; point refired.
[2]Abnormal stoppage at 24 rounds.
[3]Abnormal stoppage at 14 rounds.

Table 12. Firing Conditions, Aerial Rocket Subsystem. (2.75-inch FFAR)

Indicated Airspeed[1] (kt)	Rockets Fired	Rockets Per Firing Pulse	Firing Pulses	Interval Between Pulses (millisec)	Launcher Position (deg)
90	4	1	4	170	Zero
90	4	2	2	170	Zero
90	8	4	2	170	Zero
90	32	4	8	70	Zero
120	2	2	1	170	Zero
120	2	2	1	170	Zero
120	4	2	2	170	Zero
120	8	4	2	170	Zero
Hover[2]	1	1	1	NA	Zero
Hover[2]	1	1	1	NA	Zero
Hover[2]	2	1	2	170	Zero
Hover[2]	3	1	[3]3	170	Zero
Hover[2]	4	2	2	170	20 down
Hover[2]	8	4	2	170	20 down
Hover[2]	4	2	2	170	16 up
Hover[2]	8	4	2	170	16 up

[1]Flight path was level for all test conditions.
[2]OGE.
[3]4 pulses were intended to fire 4 rockets, but one rocket was bad.

Instrument Flight

85. The instrument flight capability of the YAH-63 helicopter was evaluated throughout the test. Because no navigational equipment was installed in the prototype helicopters, the instrument flight evaluation was limited to helicopter handling qualities during simulated basic instrument flight. Pilot workload was evaluated during instrument flight tasks such as airspeed and altitude control, standard-rate turns, climbs and descents, and radar controlled approach. Several characteristics noted throughout the development testing contributed to poor handling qualites during simulated instrument flight and are discussed individually in the following paragraphs.

86. Precise altitude and airspeed control during level flight was difficult due to the weak to unstable static longitudinal stability noted in paragraph 34 and the long-term oscillations noted in paragraph 51. The effects of the undamped long-term oscillation below 100 KIAS resulted in increasing airspeed and altitude excursions from trim unless damped by the pilot. Although the pilot could control the long-term oscillation with moderate effort (HQRS 4), the added workload would hasten pilot fatigue. The undamped long-term oscillation was not apparent at airspeeds above 100 KIAS; however, other problems encountered at higher airspeeds are discussed in succeeding paragraphs.

87. A significant change in longitudinal trim position as a result of changes in sideslip angles contributed to difficulty in stabilizing airspeed during turns. The sideslip excursions resulted in pitching moments, which caused poor airspeed control during the turn. The longitudinal trim change required due to sideslip was more significant at higher airspeeds and correspondingly, the difficulty in maintaining airspeed and attitude increased with airspeed. At airspeeds above 100 KIAS the pitch oscillations caused by sideslip excursions, coupled with the high cyclic control forces (para 78), resulted in flight conditions conducive to pilot-induced oscillations. Climbs and descents during turns did not significantly increase pilot workload.

88. The high control forces (para 78) were objectionable during turns, climbs, and descents unless the aircraft was retrimmed for the flight condition. For instrument flight, it is preferable to establish aircraft trim at the level flight condition and not change the trim setting for turns or climbs and descents. Control forces increased and became more objectionable as airspeed was increased. Additionally, the high breakout forces resulted in overcontrol and contributed to the pilot-induced oscillation tendency.

89. Loss of SCAS at high airspeeds resulted in divergent oscillations (para 48). Control of the helicopter under such failure conditions was very difficult (para 96) and would be particularly difficult in IMC.

90. Vibration (drumming) of the flat-plate canopy (all panels) was noted throughout the test and contributed to pilot discomfort and fatigue during cross-country and IMC flights. The predominant frequency was qualitatively

determined to be a 2/rev frequency. Not only could the vibrations be physically felt, but the flicker of reflections and noise generated by the vibrations were objectionable. The flicker of reflections was also within the frequency band (4 to 20 Hz) that contributes to flicker vertigo (ref 7, app A). Drumming of the flat-plate canopy is a shortcoming.

91. All the helicopter radios are located in the forward (pilot) cockpit on the side consoles. To tune the radios or to position the transmitter select knob, the pilot was required to look down and to the side. Additionally, the radios do not have preset channel capability, and therefore, the time to change frequencies was excessive. The communications radios and the transponder were also located on different sides of the cockpit, requiring a shift of hands on the cyclic as well as a shift in head position should a transponder change be required. The position of the head and the shift from one side to the other to tune the radios and transponder were conducive to vertigo. The requirement to manually tune radios during NOE flight is a deficiency. The undesirable location of the communications radio, transponder control panels, and transmitter select knob is a shortcoming.

92. The short-period gust response of the helicopter was deadbeat at airspeeds below 90 KCAS with SCAS ON. However, as airspeed was increased above 90 KCAS there was a tendency for a coupled short-period response in all three axes (para 46). This short-period response will increase pilot workload during tasks requiring precise attitude control during turbulent conditions. Since some turbulence should be expected during IMC, especially during the approach phase, the short-period gust response contributes to the poor instrument flight characteristics of the helicopter.

93. The vertical baffles installed on the pilot instrument panel restricted visibility of several flight and engine instruments. This problem is further discussed in paragraph 124n.

94. The characteristics listed in the preceding paragraphs individually and collectively degrade the instrument flight capability of the YAH-63 helicopter. Below 100 KIAS the instrument flight capability is marginal and above 100 KIAS the handling qualities and SCAS failure response are degraded beyond a safe operational degree. The YAH-63 handling qualities at airspeeds above 100 KIAS were unsatisfactory for IMC flight and constitute a deficiency.

Aircraft Systems Failures

95. Failures of the SCAS, FFS, and hydraulic systems were evaluated. Two types of SCAS failures were evaluated: loss of SCAS in all axes (disengagement) and hardover failure in one axis. The two types of force feel failures were complete failure of force (disengagement) and runaway trim failure. The two types of hydraulic failures evaluated were single-system and dual-system failures.

96. Complete loss of SCAS was evaluated by turning off the SCAS with the SCAS control switch. Pilot workload required to fly the aircraft following a complete

SCAS failure was a function of airspeed. At an airspeed of 100 KCAS or less, flight in visual reference conditions could be safely conducted, with the major objectionable characteristics being a coupled oscillation in the pitch, roll, and yaw axes. At airspeeds in excess of 100 KCAS the oscillation was divergent and damping decreased as airspeed increased. At 130 KCAS, intense pilot compensation was required to control the aircraft. The divergent oscillation about the pitch, roll, and yaw axes at greater than 100 KCAS with SCAS disengaged is a deficiency (para 48).

97. During one flight the yaw SCAS failed twice because of a malfunction in a rate gyro. The SCAS monitor system in both cases disengaged the yaw SCAS. The first cue to the pilot that yaw SCAS had failed was the yaw SCAS caution light on the caution panel. No aircraft response associated with yaw SCAS failure was noted. In a separate incident, when transfer of aircraft control in a hover was made from copilot to pilot, the copilot accidentally disengaged the SCAS and FFS by pushing the SCAS disengage switch on the cyclic. The pilot was immediately aware of loss of SCAS by the oscillation associated with complete loss of SCAS. A slow run-on landing was made from the hover (HQRS 5).

98. Hardover SCAS failures were evaluated by introducing 100 percent hardover signal into the SCAS with a pulser box (electronic test device) which allowed selection of any control axis and direction of failure. SCAS hardovers produced no adverse aircraft reactions. Following failure of single yaw, roll, or pitch SCAS, pilot workload to maintain control was negligible. Flight in visual reference conditions could be continued following a SCAS hardover in any one axis. Time histories of representative SCAS hardovers are shown in figures 85 through 87, appendix G. The lateral SCAS hardovers failed to meet the requirements of paragraph 10.3.2.7.1 of the PIDS in that, within 3 seconds, the roll rate exceeded 10 deg/sec. However, the roll rate returned to zero prior to any recovery action by the pilot. The single axis SCAS hardover characteristics as tested are satisfactory.

99. Complete loss of force feel was simulated by disengaging the force feel control switch on the AFCS panel (photo 2). Runaway failures were evaluated in each trim axis by beeping the trim to full travel in each direction and holding cyclic in the initial trim position. Simulated complete FFS failures were qualitatively conducted in level flight. When the FFS was disengaged, there was a slight stick jump and a corresponding aircraft reaction from which recovery was easily made (HQRS 3). Runaway trim failures were evaluated in level flight at 90 KCAS. From the trimmed flight condition, the beep trim in the lateral and longitudinal axes was motored to maximum extension while the pilot maintained the controls at the trimmed position. Control forces encountered were high and pilot fatigue would be a factor should the FFS not be disengaged. When the beep trim was motored in a lateral direction the FFS automatically disengaged. Stick jump, although noticeable, was not excessive in the disengagement. Force feel failure characteristics under the conditions tested were satisfactory.

100. Hydraulic systems failures were simulated by turning off either one or two of the three hydraulic systems in the aircraft. The SOFR would not allow

flight with the No. 2 hydraulic system (PC2) and the utility system off at the same time. With the PC2 and utility hydraulic systems OFF, yaw control will be seriously affected.

101. Single hydraulic failure of either the No. 1 hydraulic system (PC1) or PC2 system resulted in minimal degradation in aircraft handling qualities. Utility hydraulics OFF was characterized by a right yaw input and some tendency for pilot-induced oscillations in pitch and roll, resulting in the greatest increase in pilot workload of any single hydraulic system failure (HQRS 4). Continued safe flight was possible following a single failure of the PC1 or PC2 hydraulic systems. Loss of the utility hydraulic system caused the loss of SCAS and FFS, which seriously degraded handling qualities (para 96). The loss of FFS following a single hydraulic system failure does not meet the requirement of paragraph 10.3.2.6.6 of the systems specification.

102. Dual hydraulic failure was simulated by turning off hydraulics in either the PC1 and PC2 or the PC1 and utility systems. With PC1 and PC2 hydraulic systems inoperative, aircraft control tasks were considerably more difficult due to the loss of SCAS and FFS, but continued safe flight with visual attitude reference was possible at airspeeds below 100 KIAS. Aircraft control was improved when airspeed was reduced to a range of 70 to 90 KIAS (HQRS 6). As stated earlier, the combination failure of the PC2 and utility hydraulic systems was not tested. Under the conditions tested, the dual hydraulic failure characteristics are satisfactory.

Simulated Single-Engine Failure from Dual-Engine Flight

103. The response of the helicopter to sudden single-engine failures was evaluated at the conditions listed in table 1. All tests were initiated in stabilized wings-level flight with SCAS and FFS ON. Engine failure was initiated by rapidly retarding one throttle to the idle position. Flight controls were held fixed for a minimum of 2 seconds. A representative time history is presented as figure 88, appendix G.

104. Aircraft response following a single-engine sudden power loss was mild. The most well-defined airframe response was a left yaw of approximately 5 degrees. Yaw rates developed at the end of the 2-second controls-fixed delay time were less than 5 deg/sec. Pitch and roll rates and attitude changes were negligible. Delay times were shortest (2 seconds) in cases where topping power on the remaining engine was insufficient to maintain rotor speed. For these conditions (125-KCAS level flight and 80-KCAS MCP climb) recovery was initiated because of low rotor speed. During the MCP climb and at V_H, rotor speed decayed at approximately 10 rpm/sec following power loss. In all test conditions, aircraft control was easily maintained but establishing single-engine operating rotor speed was difficult (HQRS 5) due to rotor speed fluctuations. These fluctuations (±4 rpm) lasted in excess of 15 seconds following initial repositioning of the trim wheel. The difficulty in selecting a desired rotor speed is a shortcoming (para 73). Aircraft response following sudden single-engine power loss from dual-engine flight was satisfactory.

105. During low-power descents, the primary cue to engine failure was the engine-out warning light. Improper PAS rigging resulted in inadvertent engine shutdown during a low-power descent. The engine-out light provided adequate warning that engine shutdown had occurred.

Simulated Engine Failure from Single-Engine Flight

106. Aircraft reactions following a sudden engine failure from single-engine flight were evaluated to determine the adequacy of pilot cues and to evaluate recovery techniques required to establish autorotation. All tests were conducted with SCAS and FFS ON. Tests were conducted at the conditions listed in table 1. All tests were initiated from stabilized wings-level flight. Sudden engine failure was simulated by rapidly retarding the throttle to the idle position. Following the simulated engine failure, all flight controls were held fixed until limits of aircraft attitude or rotor speed were reached. Figures 89 and 90, appendix G, represent time histories of autorotational entries.

107. The initial reaction of the aircraft following simulated engine failure from single-engine flight was, for all test conditions, an immediate left yaw (5 to 10 degrees). The yaw rate developed during the controls-fixed delay was mild (approximately 8 deg/sec) and varied slightly as a function of engine torque at engine failure. A slight pitch-up tendency was barely recognizable and virtually no roll was encountered. As rotor speed decayed past approximately 254 rpm (92 percent), a well-defined airframe shudder was encountered which persisted until collective was lowered and recovery initiated.

108. The engine failure indications sensed by the pilot following sudden engine failure from single-engine flight were mild and indistinct. The immediate left yaw was the most well-defined reaction, but it may easily be masked by moderately gusty wind conditions. The low rpm audio warning is activated at 97 percent rotor speed, and provides the primary warning of engine failure. Changes in engine and rotor noise were too subtle to provide a distinct engine failure warning for the pilot.

109. The time available for pilot recognition and reaction to sudden engine failure from single-engine flight (delay time) was evaluated. For all test conditions, aircraft attitude changes and rates developed were mild. Controlling rotor speed decay with collective was the primary pilot action required. For single-engine level flight at 75 KCAS and heavy gross weight, a delay of approximately 1.4 seconds was achieved and a minimum rotor speed of 243 rpm (88 percent) resulted. Qualitative evaluation of aircraft reactions under other test conditions indicates that delay times in excess of 1 second will result in minimum rotor speeds approaching 240 rpm (minimum transient limit). Time available for pilot recognition and reaction to sudden engine failure from single-engine flight was satisfactory.

110. The recovery technique following sudden engine failure from single-engine flight required moderate pilot effort and was similar for all test conditions. A

smooth but rapid lowering of the collective was adequate to control rotor speed decay. Slight forward cyclic followed by a mild cyclic flare reduced rate of descent and checked rotor speed decay rate. Minimum normal acceleration was 0.3g when recovery controls were applied, and average altitude loss prior to stabilizing at desired airspeed and rotor speed in autorotational flight was approximately 1000 feet. During recovery, rotor speed control was complicated by the fact that the rotor speed needle does not reach the scale on the triple tachometer. This requires the pilot to devote excessive time to reading the gauge during the initial phase of rotor speed stabilization (HQRS 5). The poor design of the rotor speed gauge is distracting and can interfere with timely identification of a landing zone and a subsequent safe autorotational landing. The short rotor speed needle which does not reach the scale on the triple tachometer is a shortcoming.

111. The rotor speed decay rate varied from 9 to 18 rpm/sec (3 to 7 percent/sec), depending on gross weight and single-engine torque at the time of simulated engine failure. Airspeed in the range tested had no apparent effect on decay rates. Minimum rotor speeds from 245 rpm (89 percent) to 254 rpm (92 percent) were common for delay times of approximately 1 second. The lowest rotor speed encountered during the tests was 243 rpm (88 percent) following a 1.4-second delay at 75 KCAS at heavy gross weight. Rotor speed decay was more quickly arrested at the higher gross weight. Rotor speed was easily controlled by a combination of cyclic and collective under all test conditions. Once the aircraft was stabilized in autorotation, the excellent rotor speed and airspeed control greatly enhanced the handling qualities in autorotation.

VIBRATION CHARACTERISTICS

112. Vibration characteristics were evaluated throughout the test program. Particular emphasis was placed on evaluating the data recorded during level flight performance, maneuvering stability, and weapons firing tests. The flight conditions during those tests are listed in table 1. Transducer locations are shown in appendix E. Thirty-one accelerometers were installed in aircraft SN 74-22246 and eight in aircraft SN 74-22247. The vibration data are presented in figures 91 through 133, appendix G.

113. Figures 91 through 121, appendix G, indicate that vibration amplitudes at all locations during level flight were low at airspeeds less than 120 KCAS. The 2/rev vibration amplitudes were above the 0.05g limit of paragraph 3.2.11.5.1.3 of the systems specification for some airspeed/transducer location combinations, but were generally less than 0.1g. At airspeeds greater than 120 KCAS, the 2/rev and 4/rev longitudinal axis vibration levels at some locations (fig. 93, app G) increased very rapidly with airspeed. This trend correlated well with pilot comments. Some high-speed test points were aborted due to excessive vibration. The generally excessive vibration amplitudes at high airspeeds are a shortcoming.

114. Figures 122 through 130, appendix G, present representative data gathered during steady turns at various load factors. Of significance is the rapid

increase in vibration levels at high load factors (particularly at the 2/rev and 4/rev frequencies in the lateral and longitudinal axes). As discussed in paragraphs 20 and 44, the vibration levels at a combination of high load factors and airspeeds were objectionable. The increase in vibration levels with load factor and airspeed is particularly significant if the airspeed and maneuvering capabilities are to be increased. During these tests, airspeeds and load factors were less than 150 KCAS and 2.3, respectively. The excessive vibration amplitudes at high load factors above 90 KCAS are a shortcoming.

115. Vibration data were gathered at gross weights from 14,500 to 17,050 pounds in three configurations (clean, 8-TOW, and 76-rocket), and at both forward and aft cg locations. No appreciable change in vibration levels was noted at the crew stations or aircraft cg as a result of gross weight or configuration changes. However, vibration levels were significantly increased at an aft cg, particularly at the pilot station longitudinal axis at the 4/rev frequency (fig. 131, app G). The increased level of vibration at an aft cg is a shortcoming.

116. Excessive vibratory motion of the pilot collective control, the seat-mounted and door-mounted armor, and flat-plate canopy gave crewmembers the impression of a higher level vibration environment than actually existed. The distraction caused by canopy drumming was discussed in paragraph 90. The collective vibration motion was generally lateral and required that the pilot keep his hand on the control to damp the vibration. Motion of the armor, when combined with low sun angles coming from aft of the aircraft, produced a flicker on the instrument panel which was distracting and seriously interfered with the pilot's instrument scan. The excessive vibration of the pilot collective control and the seat-mounted and door-mounted armor in both cockpits is a shortcoming.

117. Vibration data were gathered during firing of the XM188 weapon. The data indicated an increase in amplitude across the frequency band with little periodic content. The gun firing vibration amplitudes were not considered excessive by the flight crew. The maximum amplitude of gun firing-induced vibration was less than 0.4g throughout the frequency spectrum of zero to 500 Hz at all locations. Figures 132 and 133, appendix G, show a comparison of vibration levels during nonfiring and firing conditions at the pilot instrument panel longitudinal axis. This location showed the greatest effects of gun firing-induced vibration. The data substantiated pilot comments that vibration caused by weapons firing was not objectionable.

HUMAN FACTORS

Aircraft Preflight Inspection

118. The exterior preflight inspection procedure for the YAH-63 starts with the front cockpit and progresses around the aircraft in a clockwise direction, terminating with the aft cockpit. The exterior inspection as described in the preliminary operator's checklist was adequate and easy to perform except for the following areas:

a. The external canopy jettison system does not have safety warnings or instructions on system operations. The absence of safety markings and operating instructions creates a safety hazard and is a shortcoming.

b. Inspection of the transmission, accessory gearbox, and ECS compressor oil levels cannot be adequately accomplished because of poor sight gauge location and design. The difficulty in determining transmission, accessory gearbox, and ECS oil levels is a shortcoming.

c. The hydraulic fluid lines to the main landing gear wheel brakes were routed on the underside of the gear struts. This location increases the likelihood of damage and loss of braking. The undesirable routing of the wheel hydraulic brake lines is a shortcoming.

d. Inspection of the hydraulic compartment requires the use of a flashlight to check hydraulic fluid levels and to inspect the compartment area. There is a light located in the hydraulic compartment; however, the light is too small and the switch must be held in the ON (depressed) position. The hydraulic systems sight gauges are not located in clear view and precise interpretation of fluid level is difficult. Excessive hydraulic fluid was found in the bottom of the hydraulic compartment because of inadequate drains from the hydraulic compartment. The inadequate lighting in the hydraulic compartment is a shortcoming. The difficulty in interpreting hydraulic fluid gauge readings is a shortcoming. Inadequate drainage for the hydraulic compartment is a shortcoming.

e. During exterior inspection of the aircraft, excessive oil leakage from the engines was noted. The excessive oil leakage from the YT700-GE-700 engine is a shortcoming.

f. Inspection of the main rotor head and transmission area requires that the operator climb on the engine cowling. The pilot had to use the top edge of the cowling for climbing because of the lack of integral handholds. The use of the top edge of the cowling for a handhold resulted in numerous cracks. The lack of handhold provisions for climbing to the top of the aircraft is a shortcoming.

Engine Starting and Shutdown Procedures

119. Engine starting and shutdown procedures for the YAH-63 are simple and minimum pilot effort is required to start the aircraft. Once the engines are started, severe vibrations can be encountered if the flight controls are not precisely and properly positioned. At low rotor speeds, as in start-up and shutdown, the vibrations encountered with displaced controls are excessive. The requirement to precisely locate the flight controls for starting and shutdown is a shortcoming.

120. The rotor brake allows engine start-up without engaging the rotor system and rotor blade stoppage shortly after engine shutdown. The use of the rotor brake will permit operation in high wind conditions and will reduce shutdown time. The incorporation of a rotor brake is an enhancing characteristic.

Cockpit Evaluation

121. The cockpits of aircraft SN 74-22246 and 74-22247 were in engineering flight test configurations. Many mission-essential items in the pilot crew position were not installed or were replaced by special test instrumentation. The copilot crew position was designed primarily for a flight test engineer for this phase of development. A qualitative evaluation of the pilot crew position was conducted throughout the test program. Four enhancing characteristics, one deficiency, and 19 shortcomings were noted.

122. The enhancing characteristics were as follows:

a. The copilot main landing gear brake handle (photo 3) provides an alternate means for slowing or stopping the helicopter during ground operations. Brake pressure, applied simultaneously to both main landing gear brakes, is proportional to the extent to which the handle is pulled. Steady braking can be applied without the adverse effect on directional control frequently associated with the use of the pilot toe brakes. This is particularly desirable following high-speed (*eg*, 50 knots) touchdowns for autorotational landings and hydraulic system failure.

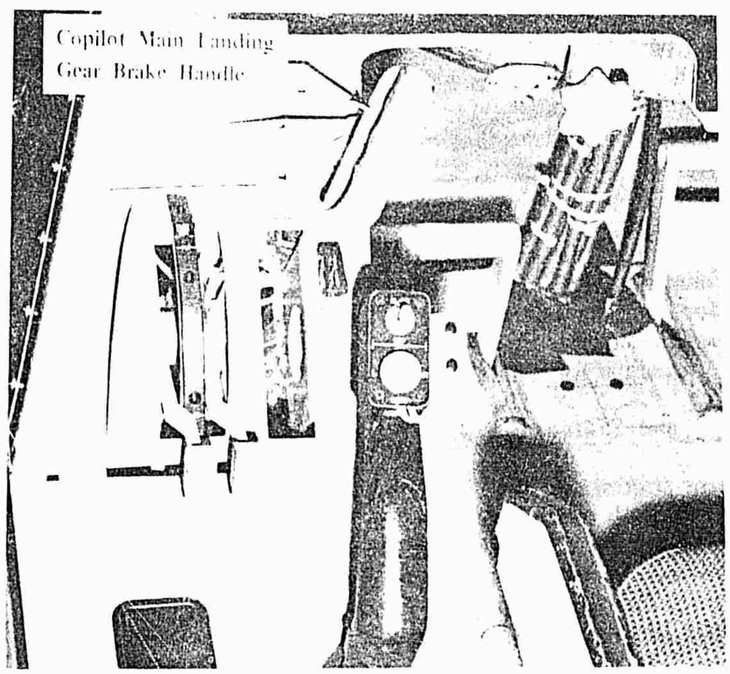

Photo 3. Copilot Main Landing Gear Brake Handle.

b. The pilot ordnance jettison panel consists of five individually actuated press-to-arm lights and one covered EMER JETT button which will jettison all stores when depressed. When one or more of the lights is illuminated, the pilot can jettison the selected stores by depressing the shrouded button on his collective. The jettison system is highly visible, easily set up, readily activated, and allows the pilot to select any combination of stations for emergency jettison according to the loading scheme and flight condition. The jettison system also ensures symmetrical jettison of wing stores.

c. The pilot right-hand console is equipped with a chip verification button. This button, when depressed momentarily, activates a burn-off feature on the chip detector plugs in the main, accessory, and tail rotor gearboxes. If an inconsequential piece of metal has caused the transmission chip light to illuminate, it will be burned off, the light will extinguish, and the mission, which would otherwise have been aborted, can be continued.

d. The emergency intercom system provides an intercockpit communications capability in the event of primary intercom failure. This capability will permit mission continuation, which would not otherwise be possible with intercom failure because of the physical isolation of the pilot from the copilot.

123. The deficiency noted was the requirement to manually tune radios during NOE flight. This requirement increases pilot workload while changing frequencies. In NOE flight, the pilot will be unable to divert his attention from the flying task for the length of time required to tune the radios.

124. The 19 shortcomings noted were as follows:

a. The unsatisfactory location of the engine throttle levers (photo 2, para 78) and the extreme stiffness in the throttle control mechanism. The pilot is required to remove his left hand from the collective to coordinate throttle position with collective changes during engine electrical control unit (ECU) lockout operations. The close proximity of the No. 1 engine throttle lever to the canopy makes power modulation difficult during entry to, and operations in, ECU lockout. The stiffness in the throttle control mechanism results in excessive pilot workload during manual throttle operations. The YAH-63 design does not incorporate a means to rapidly retard throttle without removing the hand from the collective. Such a feature should be incorporated.

b. The difficulty in closing the canopy door and verifying that it is secure without outside assistance. The helicopter is not capable of safe flight operations when either canopy door is not properly secured.

c. The copilot's restricted forward field of view and distorted visibility. His visibility is mildly distorted by a combination of the canopy above and forward of the pilot station and the turnover bulkhead (ballistic shield) separating pilot and copilot stations. During the evaluation, the bulkhead was easily scored and pock-marked. As a result, the copilot's visibility was decreased. Ground reflections in the overhead canopy panels are disorienting. A slight film of dust on the overhead canopy totally obscured the copilot's forward view on one occasion, and he was required to use the side panels for forward visibility. The field of view is restricted by the canopy support structure and the pilot's helmet.

d. The unsatisfactory design of the cyclic grip (photo 1, para 78). An unnatural reach of the thumb is required to activate the radio/intercom transmit switch and the slow trim switch. It is impossible to trim and transmit or transmit and fire wing stores simultaneously.

e. The undesirable location of the communications radios, transponder control panels and transmitter select knob. A downward and rearward movement of the pilot's head is required for visual reference to the radios or the transmitter select knob. This disrupts the pilot's reference to the instrument panel and external visual cues and is conducive to vertigo during periods of reduced visibility. Additionally, the pilot must change hands on the cyclic to operate the transponder because it is located in the right-hand console.

f. The lack of an adequate feature to prevent inadvertently placing the rotor brake handle in the rotor hold/emergency stop position. The hold (emergency stop) position of the rotor brake handle can be inadvertently achieved during rotor shutdown. Placing the handle in this position while the rotor is turning may cause damage.

g. The lack of a rotor brake ON warning light. The position of the brake handle is out of the pilot's normal field of view and he should be provided with a more visible indication that the rotor brake is engaged.

h. The inaccessibility of the left (photo 2) and right-hand circuit breaker panels in the pilot cockpit. To reach the most remote circuit breakers in flight requires considerable forward motion which can result in cyclic interference.

i. The difficulty in reading lighted segments of the pilot and copilot (photo 4) caution segment panels. Under bright ambient light conditions, recognition of displayed information is difficult. To interpret displayed information on his caution panel, the pilot must remove his right hand from the cyclic to shield panel lights. The copilot's view of his caution panel is partially obstructed by his cyclic control.

j. The location of the fuel transfer light. It is out of the pilot's normal scan pattern and is easily overlooked unless the pilot makes a conscious effort to monitor the fuel transfer operation. There is no fuel transfer light in the copilot cockpit.

k. The inadequate storage space in either cockpit. The map cases are too small to accommodate navigation publications and the helicopter log book at the same time.

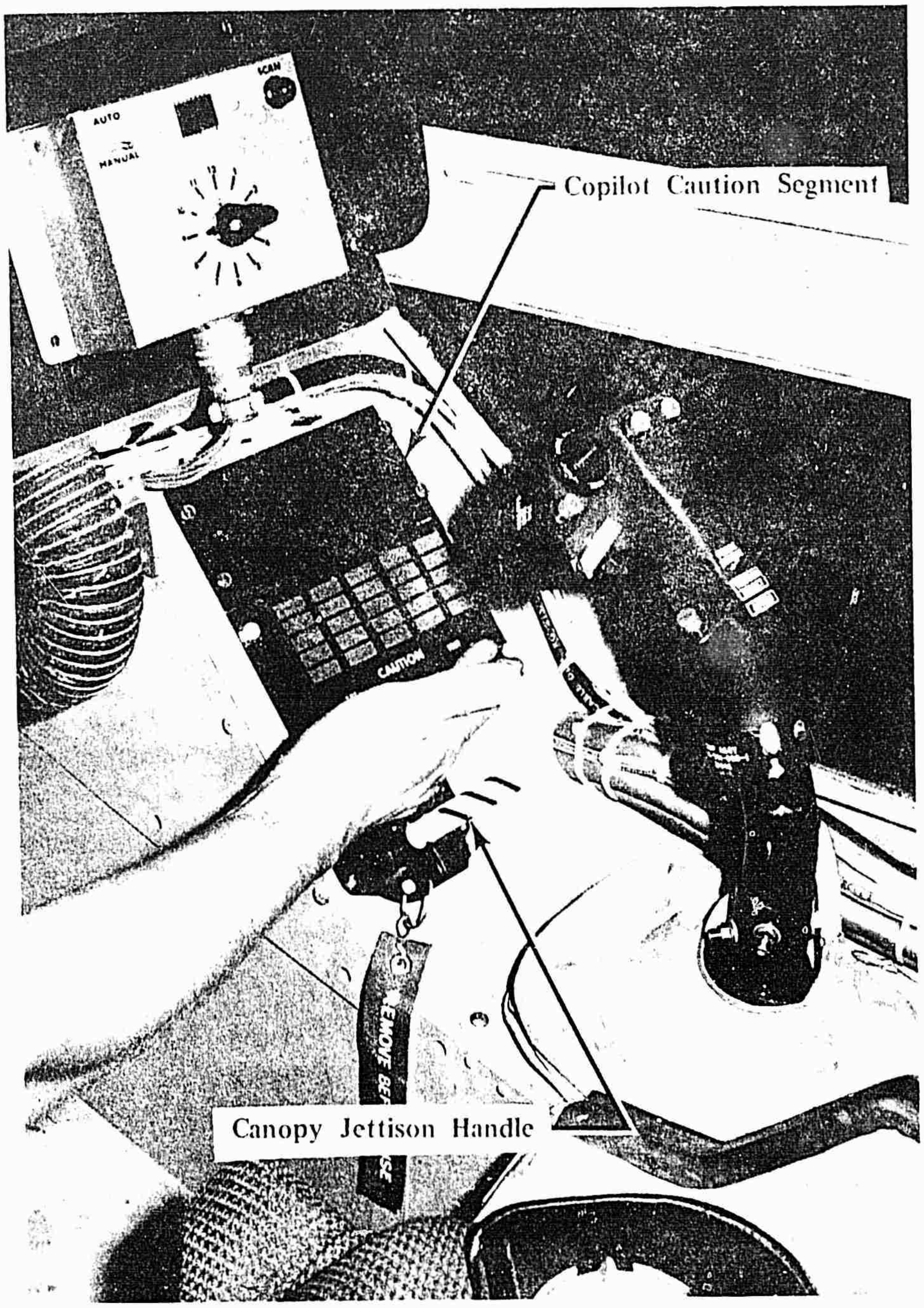

Photo 4. Copilot Cockpit.

l. The inappropriate location of the infrared (IR) BLOWER segment light. It is located on the caution panel and signals an IR BLOWER drive shaft failure when lighted. Since the emergency procedure for shaft failure is to land immediately, the light should be repositioned on the warning panel with other lights which signal emergencies which require immediate action.

m. The limited accessibility of the copilot canopy jettison handle (photo 4). It is positioned just forward of cyclic control. In the event that it becomes necessary to activate it there may be interference between the cyclic and the jettison handle.

n. The restricted visibility of the pilot instrument panel. The baffles which partition his instrument panel for the purpose of reducing instrument glare during night flight seriously limit the pilot's ability to scan his instruments using peripheral vision. He is required to deliberately move his head to read many of the instruments which are not directly in front of him. This is distracting and the time required to maintain cockpit scan becomes excessive.

o. The pilot cannot adjust seat back tilt without assistance. A capability to adjust tilt in flight could assist in reducing pilot fatigue.

p. The lack of an ENG OIL TEMP/PRESS caution light in either pilot or copilot cockpits.

q. The inability of the pilot and copilot to pass mission-related items between cockpits.

r. Lack of advisory and complete caution panels in the copilot cockpit of aircraft SN 74-22246. The pilot and copilot crew positions should be equipped with identical complete warning, caution, and advisory panels.

s. A loud background noise in the pilot and copilot headsets of aircraft SN 74-22247. The noise emanates from the Grimes light and constantly oscillates in amplitude. The intensity is increased considerably when the pilot or copilot transmits on either radio or intercom. The sound is distracting, accelerates pilot fatigue, and should be eliminated.

Night Visibility

125. The night evaluation of the YAH-63 helicopter was conducted to determine the adequacy of night lighting and the effects of reflections from internal and external light sources on the canopy. Both cockpits were evaluated on a dark night over areas of densely and sparsely lighted areas. Altitude was varied from low level (200 to 500 feet AGL) to 5000 feet AGL.

126. The reflection of internal lights in the pilot (forward) cockpit was primarily from the left and right console panels. These reflections were only slightly visible in the forward canopy and in both side canopies. With the console panels dimmed the reflections were not so intense as to reduce the field of view or

visibility. The reflection of instrument and panel lights (when dimmed) on the canopy was minimal and should not degrade the night operational capability of the helicopter.

127. The reflection of internal lights in the copilot/gunner (rear) cockpit was primarily from the pilot instrument panel. The reflections were visible in the forward one-third of the pilot side canopy. Since the copilot's forward visibility is primarily through these canopies, the reflection of internal lights limits forward visibility from the rear cockpit and is a shortcoming.

128. The internal reflection from external light sources was similar in both cockpits. In areas of sparse ground lights, the field of view and visibility were good; however, in areas of dense ground lights the reflections in the flat-plate canopy significantly restricted visibility. The ground lights located to either side of the helicopter were reflected off the opposite side canopy. During level flight, the reflections restricted the pilot's and copilot/gunner's lateral visiblity; however, the reflections did not restrict forward visibility. The restriction to lateral visibility was more serious at low altitude because of the increased intensity of the reflection and the position of the reflection on the canopy opposite to the actual light source. During banked turns, the reflections were most objectionable, since they significantly restricted field of view and visibility in the direction of the turn. At slower airspeeds, where canopy vibrations were minimal, the reflections were easily confused with actual light sources. At higher airspeeds, the increase in canopy vibrations resulted in a blurring of the reflections. The blurring prevented confusion with actual light sources; however, the blurred reflection caused a greater reduction in field of view and visibility. In addition to the reflection of ground light sources on the side canopy, visibility during right turns was further reduced by the glare of the right wing (green) position light on the underside of the main rotor blade. The left wing (red) position light did not cause a similar glare during left turns. The position light dim feature was not operational for this evaluation. During landings and takeoffs from a lighted runway, the runway lights were vividly reflected in the side canopy. These reflections had a disorienting effect and could jeopardize safety during reduced visibility operations. In the battlefield environment, the inability of the pilot to immediately detect or determine the source of ground light (*ie*, ground fire) significantly reduces the survivability of the YAH-63 during night operations. The internal reflection of external light sources on the canopy during night flight is a deficiency that must be corrected.

129. The use of the landing light during an approach to landing caused no problem when flying from the forward cockpit; however, from the rear cockpit the glare in the pilot overhead canopy from the landing light seriously reduced the copilot/gunner's forward visibility. This reduction in visibility, in addition to the lack of depth perception from the rear cockpit, results in considerable pilot effort when executing an approach (HQRS 5). The glare in the overhead canopy from the landing light severely reduced the copilot's forward visibility and is a shortcoming.

130. The vertical baffles installed on the pilot instrument panel to reduce reflections resulted in the partial loss of visibility of several flight and engine instruments. From a center cockpit eye position, all instruments were visible; however, when the head was leaned during a turn or for radio tuning, partial loss of instrument visibility was noted. The partial loss of instrument visibility resulted in a break in the routine instrument scan pattern. The partial loss of visibility of flight and engine instruments due to the vertical baffles is a shortcoming.

131. The instrument panel gauges are equipped with integral lighting that enhances the reading of the gauge and allows for a lower intensity level for gauge monitoring. However, the limit markings are decals on the exterior of the instrument glass. This results in instrument markings that are visible only as a faint shadow with no color distinction. Since a quick scan of engine instruments often depends upon the gauge reading relative to a known or marked value, the lack of discernible instrument markings slows the scan pattern. Difficulty in interpreting limit markings on the instruments during night flight is a shortcoming.

SUBSYSTEMS TESTS

Engine Performance Characteristics

132. For checking compliance of the YAH-63 with performance requirements of the systems specification (AMC-CP-2222-02000, 2 Feb 73), engine power available and fuel flow data from the YT700-GE-700 engine specification were used. These data are presented in figures 134 through 141, appendix G, and have been adjusted for the installed engine inlet and exhaust losses determined during these tests. The inlet temperature/pressure and exhaust pressure data are presented in figures 142 through 149. Transmission and accessory losses are shown in figure 150.

133. Referred engine characteristics data gathered during these tests and during the engine calibrations are compared to the requirements of the engine specification in figures 151 through 166, appendix G. Figure 151 shows that the inlet guide vane schedule on engine SN 207260 shifted by 3 to 6 degrees after approximately 38 hours of flight. This shift resulted in a reduction in power available of more than 100 shp. A similar shift was noted on engine SN 207211 (figs. 155 and 156), but the shift was apparent only on one flight. The guide vane schedule shift on SN 207260 was apparently in a direction to reduce the compressor stall margin at high altitude. Severe compressor stalls were encountered at altitude on SN 207279, apparently because of the guide vane schedule (fig. 159). The engine was changed as a result (EPR 74-07-1-7, app II). Engine SN 207260 was also on the aircraft during the high-altitude tests but did not have compressor stalls. The high-altitude tests were done prior to the shift in guide vane schedule on engine SN 207260.

134. A third problem was encountered as a result of operating in a high dust environment. During the weapons firing tests, repeated takeoffs and landings were

made on a dry lake bed. During each takeoff and landing the aircraft was surrounded by a cloud of very fine dust. After several such operations, one engine began to encounter minor compressor stalls during starts. The engine continued to operate, however, during the entire day of firing tests. Prior to further flights, the engine was washed and several unsuccessful start attempts were made. On the last of these attempts, the engine compressor seized and the engine was subsequently changed (EPR 74-07-1-8, app H).

135. The YT700-GE-700 engine controls were designed to provide a constant power turbine speed regardless of changing power demands on the engines. The hydromechanical unit (HMU) and ECU were designed to provide this isochronous governing, as well as other functions (app D). To accomplish this governing, the HMU makes gross changes in fuel flow as a function of LDS position. The LDS is positioned by the collective control through mechanical linkage. Fine tuning of fuel flow is provided by the ECU.

136. The transient rotor speed excursions observed during these tests appear to be the result of two problems with engine controls. One problem, an improper design feature of the ECU, contributes to the large transient rotor speed droop following a rapid power demand from low power conditions (paras 75 and 81). This problem is illustrated by fuel flow traces in figure 167, appendix G. Following the collective increase, the fuel flow initially increases, then levels off, then continues to increase. This leveling off of fuel flow is caused by the ECU, which senses that the power turbine speed is already at 100 percent when collective is applied. The ECU cuts back the fuel flow to prevent an overspeed. Without the leveling off of fuel flow, the transient rotor speed droop would be less.

137. Figure 167, appendix G, also illustrates the second problem, a malfunctioning of either the HMU or the linkage between the collective control and the LDS. The problem shows up as a delay in the initial fuel flow increase following collective application. In this instance, the delay is nearly 1 second, which contributes to the excessive rotor speed droop. The fact that fuel flow does not rapidly respond to collective control position changes also contributes to rotor speed fluctuations following small power changes (para 74).

138. The HMU problem was further investigated during ground tests with one engine at flight-idle and the other in ECU lockout and delivering power to the rotor to maintain 100 percent rotor speed. In this condition, fuel flow to the engine in ECU lockout should follow the collective control position with no discernible delay. Figure 168, appendix G, shows more than a 1-second delay between collective increase and fuel flow increase. Then, as collective is lowered, fuel flow remains at the increased values. Figure 169 shows a collective increase with a different engine operating in ECU lockout. The fuel flow remains essentially constant during this entire test. These ECU lockout tests indicated either a malfunction of the HMU or binding or free play in the linkage between the collective control and the LDS. Checks by GE and BHT personnel failed to determine the cause. Because this problem was found on all three engines checked, a linkage problem is suspected. Also, it should be noted that rotor speed fluctuations

following small power changes were not a problem on aircraft SN 74-22246 during the first month of the tests. This may indicate that wear in the linkage was the cause.

Airspeed Calibration

1 9. Calibration of the ship's airspeed system was accomplished during level flight performance tests in the clean configuration. No calibration was accomplished in climbing or descending flights. A calibrated pace aircraft was used as a speed reference. Data from this calibration are presented in figure 170, appendix G.

140. The YAH-63 has two ship's airspeed systems. One utilizes the left-hand pitot tube and the other uses the right-hand one (photo 2, app B). Both systems use the same two static parts located on either side of the fuselage. During these tests, only the right-hand system was operational. The airspeed system functioned satisfactorily in level and climbing flight. In autorotation, however, airspeed indications were unreliable below approximately 50 KCAS. The unreliable airspeed indication in autorotation is a shortcoming.

CONCLUSIONS

GENERAL

141. The YAH-63 helicopter exhibits excellent potential for the AAH mission. The helicopter's handling qualities at airspeeds less than 100 knots are a significant improvement when compared to current attack helicopters. The handling qualities were particularly well suited to flight in the NOE environment. Numerous envelope limits were imposed during this test which would be unacceptable for an operational aircraft. Improvement of the helicopter's performance and correction of the deficiencies and shortcomings identified during this evaluation would make the YAH-63 an excellent attack helicopter.

a. The following YAH-63 characteristics were found to be particularly enhancing for the attack helicopter role:

(1) The field of view and forward visibility from the pilot station (forward cockpit) during NOE flight (para 77).

(2) The ability to accurately and rapidly control airspeed during NOE flight (paras 35 and 77).

(3) The excellent handling qualities in rearward (*ie*, downwind hovering) flight (para 71).

(4) The chip verification system (para 122c).

(5) The pilot ordnance jettison panel (para 122b).

(6) The copilot main landing gear brake handle (para 122a).

(7) The excellent rotor speed and airspeed control in autorotational descents (para 111).

(8) The incorporation of a rotor brake (para 120).

(9) The emergency intercom system (para 122d).

142. A total of nine deficiencies and 59 shortcomings were noted.

DEFICIENCIES AND SHORTCOMINGS

143. The following deficiencies were identified during this evaluation:

a. The internal reflection of external light sources on the canopy during night flight (para 128).

b. The unsatisfactory handling qualities for flight in IMC at airspeeds greater than 100 KIAS (para 94).

c. The excessive transient rotor speed droop following a rapid power demand from a low-power condition (paras 75 and 81).

d. The divergent oscillation about the pitch, roll, and yaw axes at airspeeds greater than 100 KCAS with SCAS OFF (paras 48 and 96).

e. The repeated failure of the XM188 weapon system (para 83).

f. The inadequate directional control margin in right lateral accelerations (para 58).

g. The inadequate forward longitudinal control margin at high airspeeds (para 30).

h. The requirement to manually tune radios during NOE flight (paras 91 and 123).

i. The inadequate forward longitudinal control margin in right sideslips at high airspeeds (para 40).

144. The following shortcomings were identified:

a. The unsatisfactory design of the cyclic grip (para 124d).

b. The poor engine governing characteristics that allow rotor speed fluctuations outside the normal operating range (para 74).

c. The lack of an ENG OIL TEMP/PRESS caution light in either cockpit (para 124p).

d. The lack of a feature to relieve all control forces without holding the force trim release button down or interfering with radio/intercom communications (para 78).

e. The unsatisfactory location and extreme stiffness of engine throttle levers (para 124a).

f. The requirement to precisely locate the flight controls during engine start and shutdown (para 119).

g. The awkward foot movement required for toe brake application (para 59).

h. The marginal effectiveness of the brakes (para 59).

i. The undesirable location of the communications radio, transponder control panels, and transmitter select knob (paras 91 and 124e).

j. Drumming of the flat-plate canopy (para 90).

k. The generally excessive vibration amplitudes at high airspeeds (para 113).

l. The abrupt longitudinal trim shifts in left sideward and low-speed forward flight (para 70).

m. The excessive pitching moment with sideslip (para 41).

n. The excessive oscillation of the pilot unattended cyclic control (para 28).

o. The short-period response about all axes at airspeeds above 90 KCAS (para 46).

p. The reflection on the canopy of internal cockpit lights which limit forward visibility from the aft cockpit (para 127).

q. The glare in the overhead canopy from the landing light, which severely limits the copilot's forward visibility (para 129).

r. The location of the fuel transfer light (para 124j).

s. The weak side force near trim (para 39).

t. The asymmetry in cyclic and directional breakout forces about trim (para 26).

u. The large directional control trim shift between 45 and 55 KCAS during normal takeoffs in the 8-TOW configuration (paras 31 and 64).

v. The high cyclic control breakout forces (para 26).

w. Difficulty in reading lighted segments of the pilot and copilot caution segment panels (para 124i).

x. The copilot's restricted forward field of view and distorted visibility (para 124c).

y. The unsatisfactory operation of the cyclic beep trim system (para 27).

z. The low control response in roll (para 55).

aa. The longitudinal control reversal during approach to a hover (para 65).

bb. The difficulty in selecting a desired rotor speed (paras 73 and 104).

cc. The excessive vibration amplitudes at high load factors above 90 KCAS (para 114).

dd. The inappropriate location of the IR BLOWER segment light (para 124l).

ee. The difficulty in interpreting limit markings on the instruments during night flight (para 131).

ff. The poor design of the rotor speed gauge (para 110).

gg. The increased level of vibration at an aft cg (para 115).

hh. The inadequate static longitudinal stability at airspeeds for instrument and cross-country flights (para 34).

ii. The awkward procedure required to unlock the nose wheel (para 61).

jj. The sticking and uneven compression/extension of main landing gear oleos (para 63).

kk. The excessive vibration of the pilot collective control and the seat-mounted and door-mounted armor in both cockpits (para 116).

ll. The longitudinal long-term response characteristics (para 51).

mm. The excessive rolling moments created by collective inputs (para 52).

nn. The partial loss of visibility of flight and engine instruments due to the vertical baffles on the instrument panel (paras 124n and 130).

oo. The lack of an adequate feature to prevent inadvertently placing the rotor brake handle in the rotor hold/emergency stop position (para 124f).

pp. The lack of a rotor brake ON warning light (para 124g).

qq. The inaccessibility of left- and right-hand circuit breaker panels in the pilot cockpit (para 124h).

rr. The inadequate storage space in either cockpit (para 124k).

ss. The limited accessibility of the copilot canopy jettison handle (para 124m).

tt. The lack of a capability to adjust seat back tilt in flight (para 124o).

uu. The difficulty in closing the canopy door and verifying it is secure without outside assistance (para 124b).

vv. The inability to pass mission-related items between cockpits (para 124q).

ww. The excessive oil leakage from the YT700-GE-700 engines (para 118e).

xx. The difficulty in determining transmission, accessory gearbox, and ECS oil levels (para 118b).

yy. The difficulty in interpreting hydraulic fluid gauge readings (para 118d).

zz. The undesirable routing of the wheel hydraulic brake lines (para 118c).

aaa. Absence of external canopy jettison system safety markings and operating instructions (para 118a).

bbb. The inadequate drainage for the hydraulic compartment (para 118d).

ccc. The inadequate lighting in the hydraulic compartment (para 118d).

ddd. The lack of handhold provisions for climbing to the top of the aircraft (para 118f).

eee. Lack of an advisory panel and complete caution panel in the copilot cockpit of aircraft SN 74-22246 (para 124r).

fff. The loud background noise in the headsets of aircraft SN 74-22247 caused by the rotating beacon (para 124s).

ggg. The unreliable airspeed indication in autorotation (para 140).

SPECIFICATION COMPLIANCE

145. Within the scope of these tests, the YAH-63 failed to meet the following requirements of the systems specification:

a. Paragraph 3.2.1.1.1.1a - The aircraft could not hover at specified conditions and therefore failed to meet the 450 ft/min vertical rate of climb requirement.

b. Paragraph 3.2.1.1.1.1b - The 122 KTAS cruise airspeed at MCP failed to meet the 145 KTAS requirement by 23 knots (para 14).

c. Paragraph 3.2.1.1.1.1c - The 142 KTAS level flight airspeed obtained failed to meet the 150 KTAS requirement by 8 knots (para 14).

d. Paragraph 3.2.1.1.1.1d - The 2.38 hours endurance did not meet the 2.5 hours requirement by 0.12 hour (para 13).

e. Paragraph 3.2.1.1.1.3a - The YAH-63 cannot maintain level flight on one engine at the required conditions and, therefore, did not meet the 90 KTAS single-engine level flight airspeed requirement (para 14).

f. Paragraph 3.2.1.1.1.3b - The hot-day single-engine service ceiling of 2070 feet pressure altitude failed to meet the 5000-foot requirement by 2930 feet (para 12).

g. Paragraph 3.2.1.1.1.4d - Although the YAH-63 meets the lateral acceleration requirements of the paragraph, the inability to maintain aircraft heading during the maneuver failed to meet the requirement for tracking and fire control (para 19).

h. Paragraph 3.2.11.5.1.3 - The 2/rev vibration amplitudes exceeded the 0.05g limit at some tranducer locations in some flight regimes (para 114).

i. Paragraph 10.3.2.1.1 - Longitudinal and lateral control breakout forces exceed the maximum allowed by 1.5 pounds (para 26).

j. Paragraph 1).3.2.1.2 - The cyclic and directional control breakout forces were not symmetrical about trim (para 26).

k. Paragraph 10.3.2.6.6 - Loss of the utility hydraulic system results in deactivation of the cyclic FFS. This is in direct violation of the requirement (para 101).

l. Paragraph 10.3.2.7.1 - Roll rates in excess of 10 deg/sec were encountered during 3-second controls-fixed delays following lateral SCAS hardover failures (para 98).

m. Paragraph 10.3.3.1 - The forward longitudinal control margin at high speed in level flight and in right sideslips failed by 10 deg/sec and 15 deg/sec, respectively, to meet the 15 deg/sec requirement (paras 30 and 40). Additionally, inadequate directional control margin in right lateral accelerations failed to meet the requirement by 15 deg/sec (para 58). The aft longitudinal control margin in left sideward flight will not produce the 15 deg/sec requirement of the paragraph (para 69).

n. Paragraph 10.3.3.2.7 and 10.3.4.1 - The collective-fixed longitudinal control position and force gradients with airspeed failed to meet the static stability requirements (para 34).

o. Paragraph 10.3.4.2.1d - The short-term response failed to meet the requirements, in that small-amplitude, short-period residual oscillations exist which will affect mission capability (para 46).

p. Paragraph 10.3.4.2.1c - The long-term response characteristics are objectionable and, therefore, do not meet the requirements (para 51).

q. Paragraph 10.3.4.5 - The longitudinal trim change with sideslip exceeded the limits set in this paragraph (para 4[illegible]

r. Paragraph 10.3.6.1.2 - The average longitudinal control force with normal acceleration gradient was less than the 3.0 lb/g minimum specified except during right turns at approximately 60 KCAS (para 42).

s. Paragraph 10.3.9.1.1 - The aft longitudinal control margin in left sideward flight will not produce the 15 deg/sec angular rate requirement of this paragraph (para 69).

t. Paragraph 10.3.9.1.2 - Inadequate directional control margin in right lateral accelerations prevented the YAH-63 from meeting the requirements of this paragraph (para 58).

RECOMMENDATIONS

146. The deficiencies identified during this evaluation must be corrected for the YAH-63 to safely perform the attack helicopter mission (para 143).

147. The shortcomings identified should be corrected in the next development phase of the YAH-63 (para 144).

148. Further tests should be conducted to determine maximum tail rotor slip in right lateral accelerations (para 17).

149. During autorotational descents, 90 KCAS should be used as the airspeed for maximum glide distance (para 23).

APPENDIX A. REFERENCES

1. Test Plan, USAAEFA, Project No. 74-07 C4, *Development Test I, Advanced Attack Helicopter Competitive Evaluation, YAH-63 Helicopter, YAH-64 Helicopter*, May 1976.

2. Systems Specification, United States Army Materiel Command, AMC-SS-AAH-10000A, "Systems Specification for Advanced Attack Helicopter", 1 July 1973.

3. Systems Specification, Bell Helicopter Company, AMC-SS-AAH-B10000, "Systems Specification for Advanced Attack Helicopter," June 1973.

4. Letter, AVSCOM, DRSAV-EQ, 31 May 1976, subject: Safety-of-Flight Release for the YAH-63 Government Competitive Test, with revisions R-1 dated 30 June 1976, R-2 dated 30 July 1976, R-3 dated 16 August 1976, and R-4 dated 25 September 1976.

5. Message, AVSCOM, DRSAV-EQ, 091630Z 10 September 1976, subject: One Engine Inoperative Climb Performance of YAH-63.

6. Message, AVSCOM, DRSAV-EQ, 161815Z, August 1976, subject: AAH Competitive Test.

7. Training Circular, Headquarters Department of the Army, TC 1-28, "*Rotary Wing Night Flight*", February 1976.

8. Letter, AVSCOM, DRSAV-EQP, 26 August 1976, subject: GCT Exhaust System Performance Losses using the T700 Engine Deck No. 73004.

APPENDIX B. AIRCRAFT DESCRIPTION

GENERAL

1. The YAH-63 is a two-place advanced attack helicopter manufactured by BHT and powered by two General Electric YT700-GE-700 engines rated at 1536 shp maximum at sea level on a standard day. The cockpit provides tandem seating with the pilot in the forward seat and the copilot/gunner in the aft, elevated seat. The short wings provide for mounting of the main landing gear and hardpoints for external stores.

2. Distinctive features of the YAH-63 include the two-bladed, semirigid, seesaw-type main rotor; the two-bladed flex beam tail rotor; the "T" configured tail containing the vertical fin and upper and lower horizontal stabilizer; and kneeling, tricycle-type landing gear. The two engines are installed aft of the main transmission and on top of the fuselage, parallel to and 30 inches to either side of the aircraft center line. The cockpit canopy is a seven-plane, flat-plate design equipped with a linear explosive canopy removal system.

MAIN ROTOR

3. The main rotor is a two-bladed, semirigid, seesaw type with preconing, underslinging, and flapping restraint springs. The main rotor blades are dual spar with a visual inspection system to provide for dectection of disbonds and fatigue cracks in the blade spar structure. Cyclic control is accomplished by tilting the swashplate and collective control is accomplished by raising/lowering the swashplate. The mast is equipped with helical splines that allow lowering of the mast and main rotor blade for air transportability. Principal main rotor characteristics are tabulated below.

Number of blades	2
Diameter	51 ft, 6 in.
Blade chord (constant)	3 ft, 6.6 in.
Blade twist	-12.4 deg
Hub precone angle	4 deg, 15 min
Geometric disc loading, at 16,054 lb	7.71 lb/ft^2
Airfoil section designation and thickness	FX69HL083 8.3 percent thick
Normal main rotor speed	276 rpm

TAIL ROTOR

4. The tail rotor is a two-bladed, flex beam, skewed-flapping axis, pusher-type design. The tail rotor rotating control system is a push-pull axial input type activated

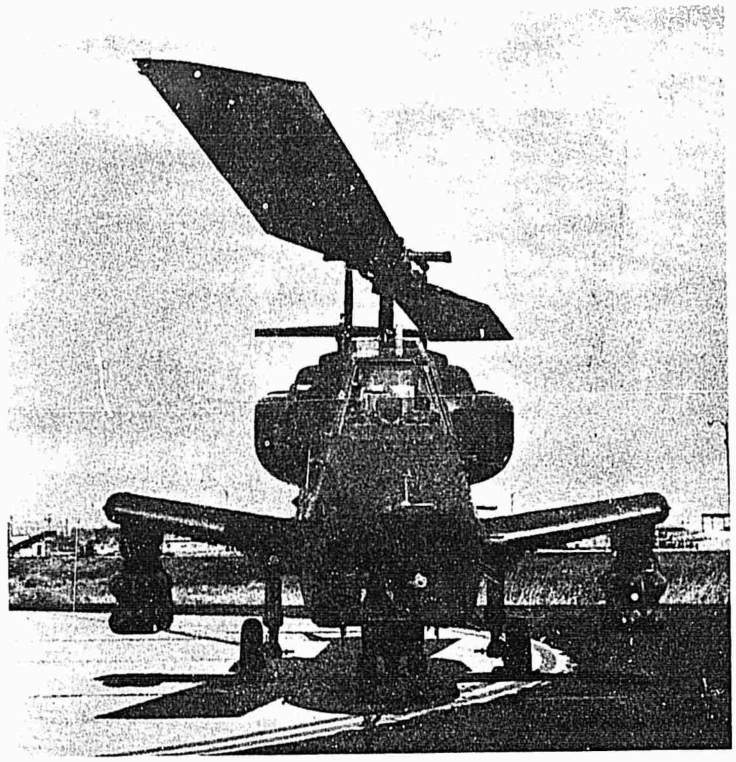

Photo 1. Front View.

Photo 2. Front Quartering View.

Photo 3. Rear View.

Photo 4. Left Side View.

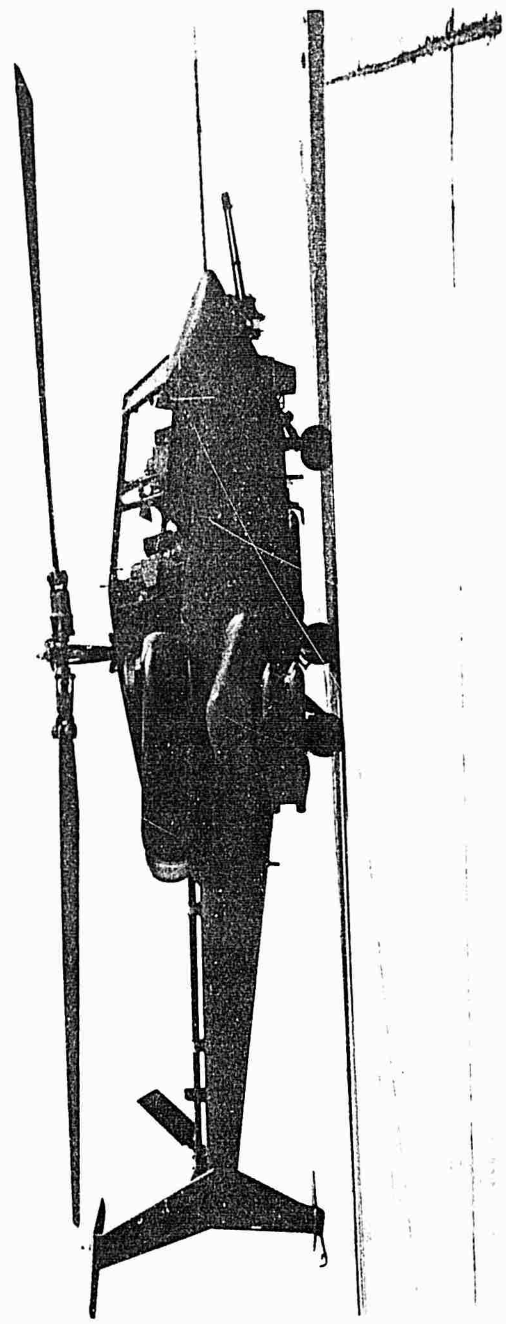

Photo 5. Right Side View.

by dual inputs from fixed controls. Principal tail rotor characteristics are tabulated below.

Number of blades	2
Diameter	9 ft, 6 in.
Blade chord	1 ft, 5 in.
Blade twist	-10 deg
Hub preconing angle	Zero deg
Airfoil section designation and thickness	Bell droop snoot with conventional tip 10.9 percent thick
Normal operating speed	1446 rpm

WING

5. The aircraft has a fixed cantilever wing designed primarily to support the main landing gear and external stores. The wing is a 24-percent-thick modified Clark-Y airfoil section consisting of an integral fuselage section and removable (for transportability) outboard sections. Two aluminum main spars support a lockup structure of webs and aluminum skin. The forward and aft spars are connected to the mid fuselage at fuselage stations (FS) 275 and 306.5, respectively. Principal wing characteristics are tabulated below.

Span, maximum	18 ft, 2 in.
Span, wing outer sections removed	9 ft, 0.7 in.
Chord:	
At root	5 ft, 1.75 in.
At tip	4 ft, 2.5 in.
Airfoil section designation and thickness	Modified Clark-Y
Incidence:	
At root	Zero deg
At tip	Zero deg
Dihedral	11 deg

TAIL

6. The aircraft incorporates an I-tail, the principal features of which are listed below.

Horizontal, Upper

Span	8 ft
Chord:	
At root	2 ft, 6 in.
At tip	1 ft, 6 in.
Airfoil section designation and thickness	NACA 65

Incidence of flat (top) surface	4.5 deg
Sweep of leading edge	14 deg
Dihedral of flat (top) surface	Zero deg
Aspect ratio	4.0

Horizontal, Lower

Span	5 ft
Chord:	
At root	1 ft, 10 in.
At tip	1 ft, 2 in.
Airfoil section designation	Inverted Clark-Y
Incidence of flat (top) surface	5 deg
Sweep of leading edge	15 deg
Dihedral of flat (top) surface	Zero deg
Aspect ratio	3.3

Vertical Fin

Area	33.8 ft^2
Airfoil section designation and thickness	NACA 65_3-418
Incidence of flat surface	3 deg right
Sweep of leading edge:	
Upper	35 deg
Lower	30 deg
Aspect ratio	3.37

GROSS WEIGHT

7. The following gross weights were obtained from the SOFR and test data.

Empty gross weight	12,188 lb
Design gross weight	16,054 lb
Maximum allowable	19,260 lb

FUEL

8. Fuel is contained in two internal fuel tanks. Maximum fuel load is 365 gallons (2368 pounds usable).

APPENDIX C. FLIGHT CONTROL DESCRIPTION

GENERAL

1. The flight control system is of the mechanical hydraulically boosted irreversible type with conventional controls in the forward cockpit (pilot station). The controls in the aft cockpit (copilot/gunner station) consist of conventional antitorque pedals and collective control and a side-arm cyclic control. The main rotor cyclic and collective controls have redundant mechanical links from the two cockpits to the controls mixer located at the aft of the avionics bay. The mixer assembly is completely redundant except for the collective input arm. The output of the mixer assembly is hydraulically boosted by three independent hydraulic systems to control six power actuators. Controls from the mixer to the swashplate are redundant, but controls above the swashplate are not redundant.

2. Directional control is maintained through redundant cables and tubes from both cockpits. Two power actuators operate redundantly to control the tail rotor. Toe brakes for the main gear are provided only for the pilot.

FLIGHT CONTROLS RIGGING

3. Rigging of the primary flight controls on the two prototype aircraft was the same for this evaluation. The following are limits of swashplate travel measured in angular displacement from a plane perpendicular to the main rotor mast:

Cyclic Control Position	Swashplate Position
Full aft	8° 30' (aft tilt)
Full forward	13° 45' (fwd tilt)
Full left	7° 58' (left tilt)
Full right	6° 8' (right tilt)

4. The main rotor mast was tilted 45' aft and 1° 23' left with respect to the fuselage. With the aircraft on the ground, the aft swashplate tilt was limited to 6° 29' with respect to the mast. Microswitches on the main landing gear activated this auxiliary stop. This approximate 2-degree restriction in swashplate travel is to help prevent main rotor contact with the tail rotor drive shaft. Collective position did not affect the cyclic control travel. Collective rigging is as follows.

Collective Position	Average Main Rotor Blade Angle
Full up	16.6 deg
Full down	-0.5 deg

5. The limits of tail rotor collective blade angle are listed below.

Blade angle at full right pedal	-11° 16'
Blade angle at full left pedal	32° 10'
Total blade angle change	43° 26'

STABILITY AND CONTROL AUGMENTATION SYSTEM

6. The aircraft incorporates a 3-axis electrohydraulic SCAS. The SCAS provides angular rate damping about the pitch, roll, and yaw axes, as well as control quickening for cyclic and directional controls. The SCAS actuators are downstream of the FFS in the flight control system and therefore SCAS imputs are not normally felt at the cockpit controls. The SCAS inputs are "washed out" with time so the actuators will not remain at the limit of their travel during sustained maneuvers or following large trim changes.

7. In each of the pitch and roll axes, a single SCAS actuator is controlled by an electronic control unit which senses aircraft angular rate from a gyro and cyclic control motion from motion transducers. The unit makes appropriate control inputs through the actuator. Although this is a nonredundant system, protection from system hardover failures is provided by a monitor system. The monitor system consists of transducers and an electronic control unit which are identical to those of the SCAS. However, the electronic control unit sends signals to an electronic model of the actuator rather than to the actuator itself. The monitor system compares the motion of the actuator to the output of the model. If the two disagree, the monitor disengages the failed axis and centers the actuator.

8. The yaw axis of the SCAS does have two parallel SCAS systems and two monitor systems. This does not provide redundancy, however, because a failure in either of the yaw SCAS systems or either of the monitor systems will cause disengagement of the entire yaw axis of the SCAS.

9. The authority of each axis of the SCAS is presented below as a percentage of full travel of the primary flight control in that axis.

Pitch	±12.1 percent
Roll	±14.5 percent
Yaw	±12.5 percent

10. Microswitches on the main landing gear cause the longitudinal SCAS to travel to its forward limit when the aircraft is on the ground. This function of the SCAS is to help prevent main rotor contact with the tail rotor drive shaft. A failure of the microswitch could cause a hardover-type SCAS failure in flight. Therefore, a two-position switch in the cockpit is provided to disable the microswitch circuit.

FORCE FEEL SYSTEM

11. The FFS is incorporated to provide proper cyclic and directional control breakout forces and control force versus displacement gradients. The cyclic FFS senses control force and displacement from trim and adjusts the force by use of a hydraulic actuator. The system also senses airspeed and adjusts the cyclic force versus displacement gradients as functions of airspeeds. A monitor system will disengage the FFS if a failure is detected. The directional FFS consists of a spring cartridge with a magnetic brake release.

12. The forces on all three axes can be trimmed to zero simultaneously by use of a push button on the cyclic grip. Longitudinal and lateral forces may be trimmed to zero individually by use of a four-position beeper switch on the cyclic control grip. No such function is provided in the directional system.

APPENDIX D. ENGINE DESCRIPTION

GENERAL

1. The primary power plant for the YAH-63 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 aircraft. 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, 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 input 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, 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). 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. Another subsystem of the engine is the variable geometry linkage assembly.

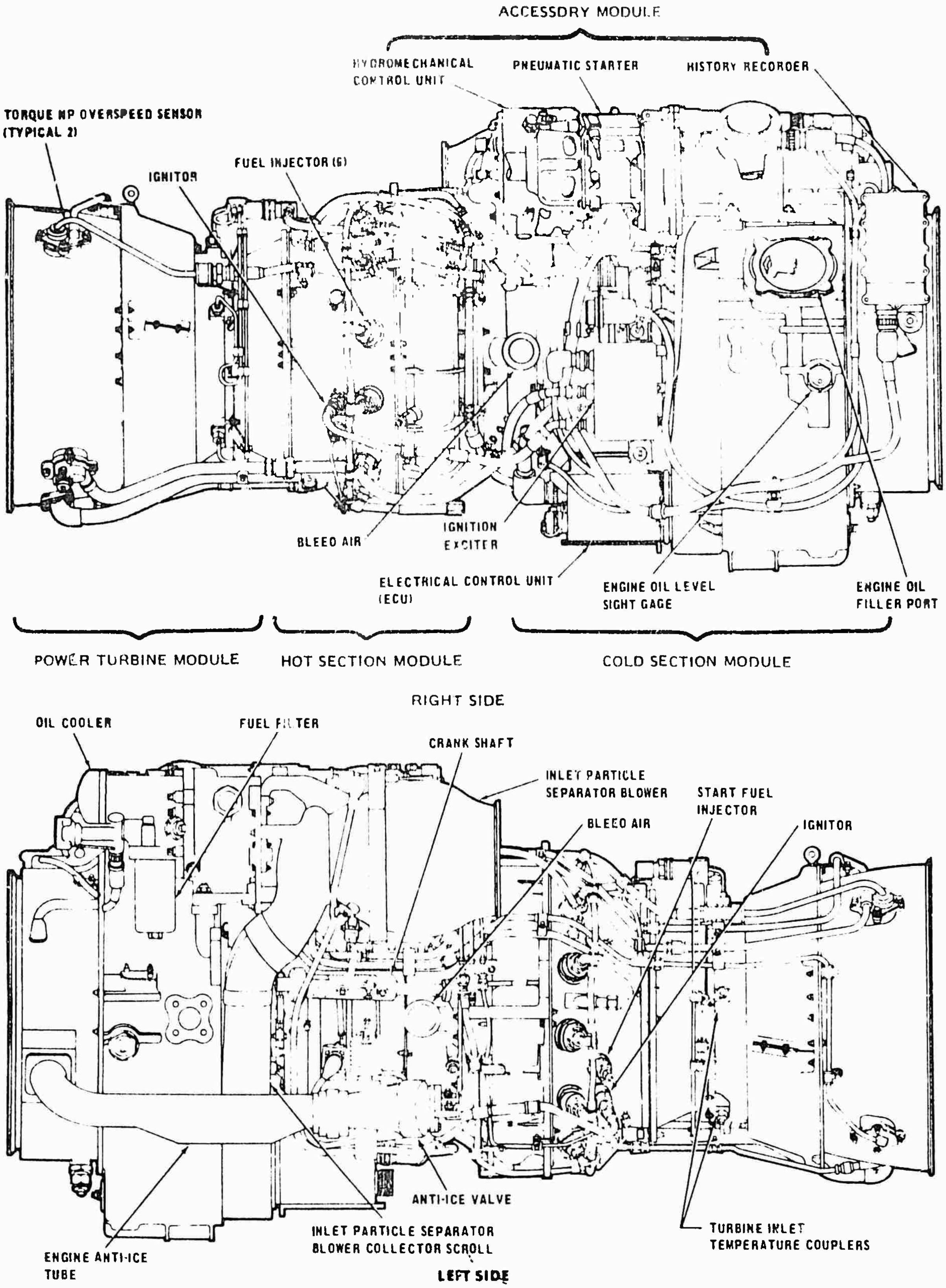

Figure 1. YT700GE-700 Engine.

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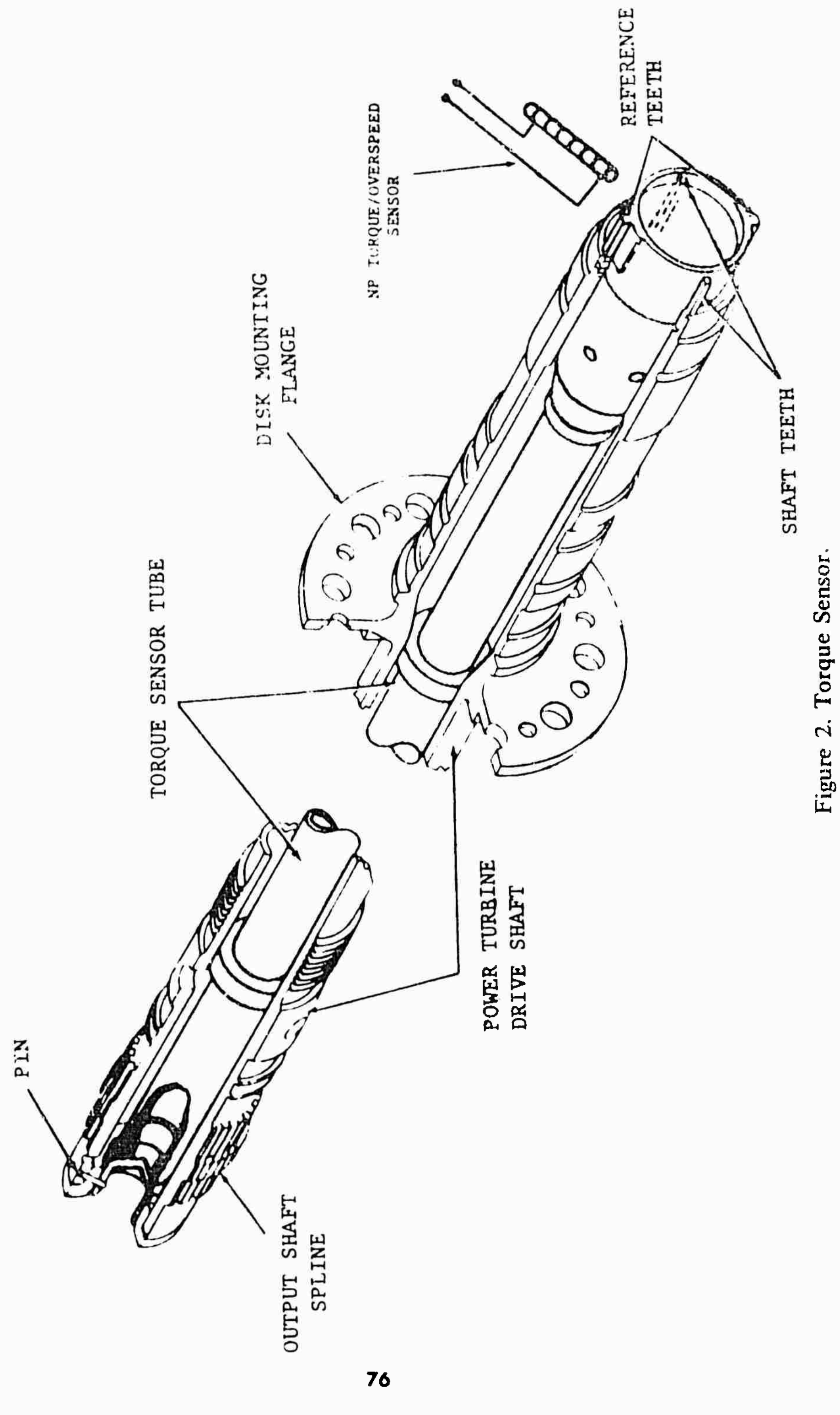

Figure 2. Torque Sensor.

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 (P3) 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 which permits the system to be checked while the engine is running.

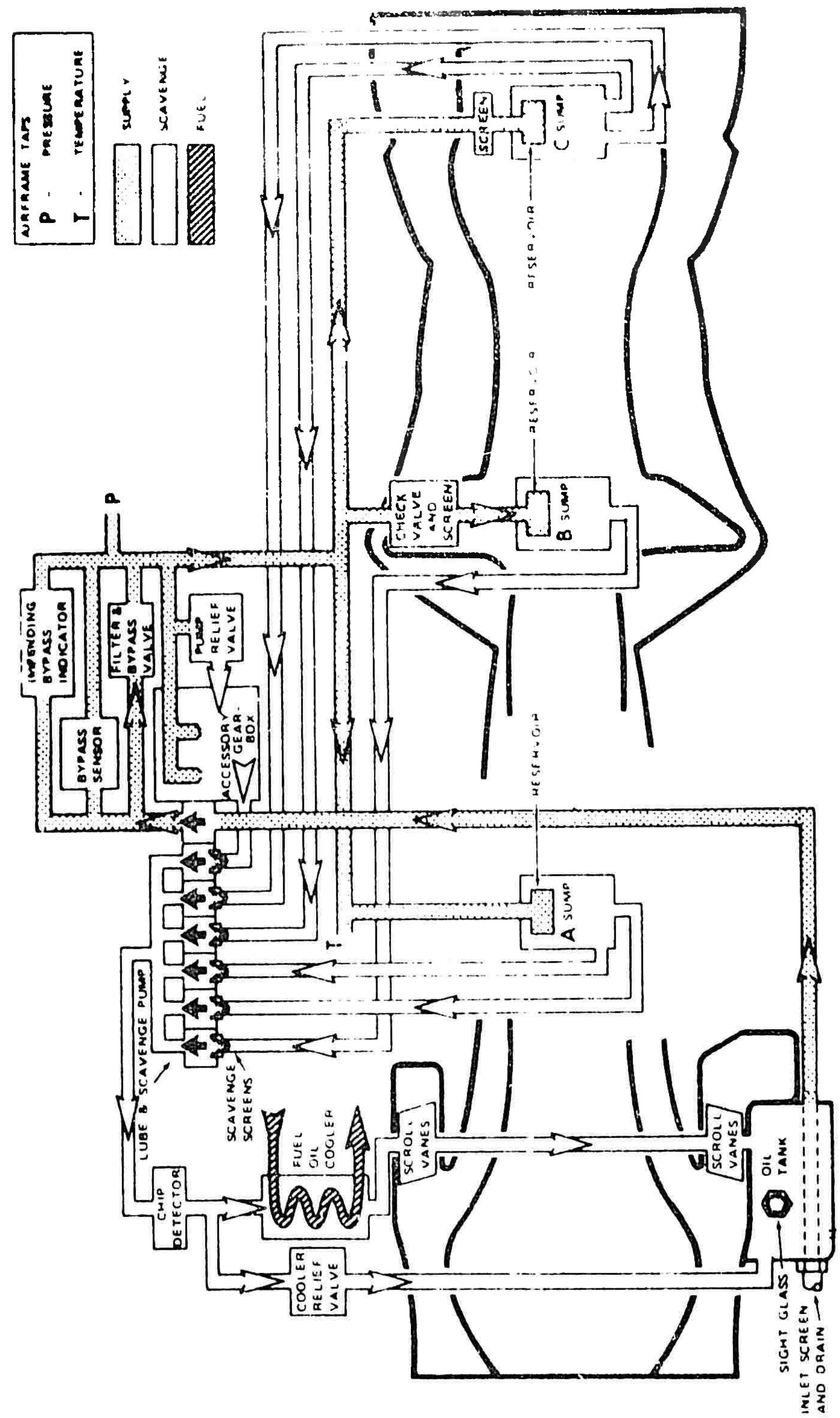

Figure 3. Oil System Schematic.

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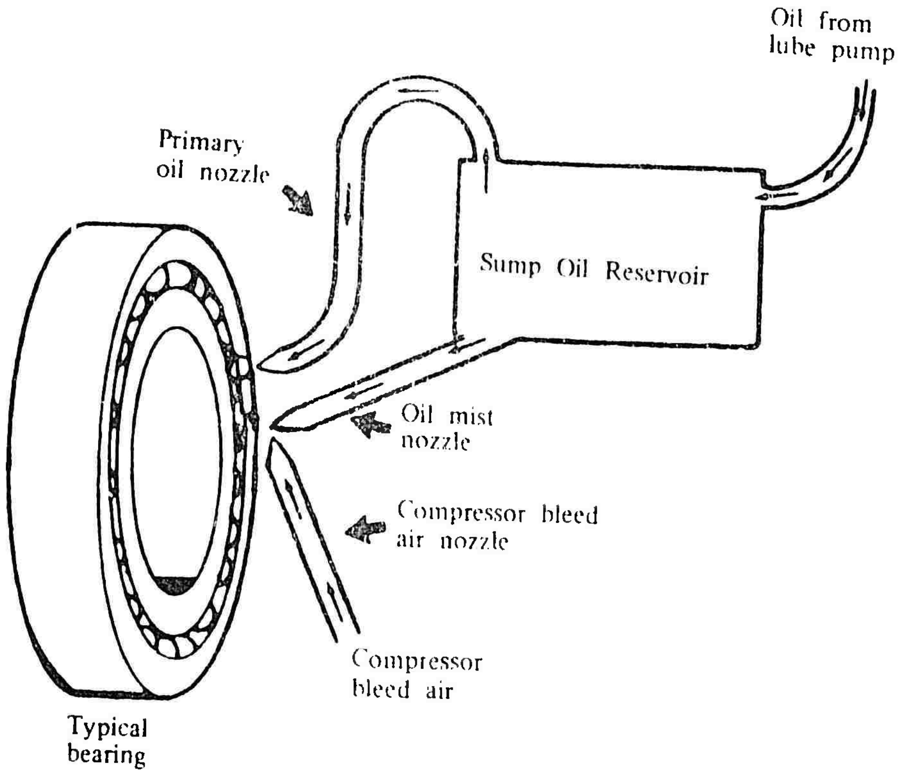


Figure 4. Emergency Oil System.

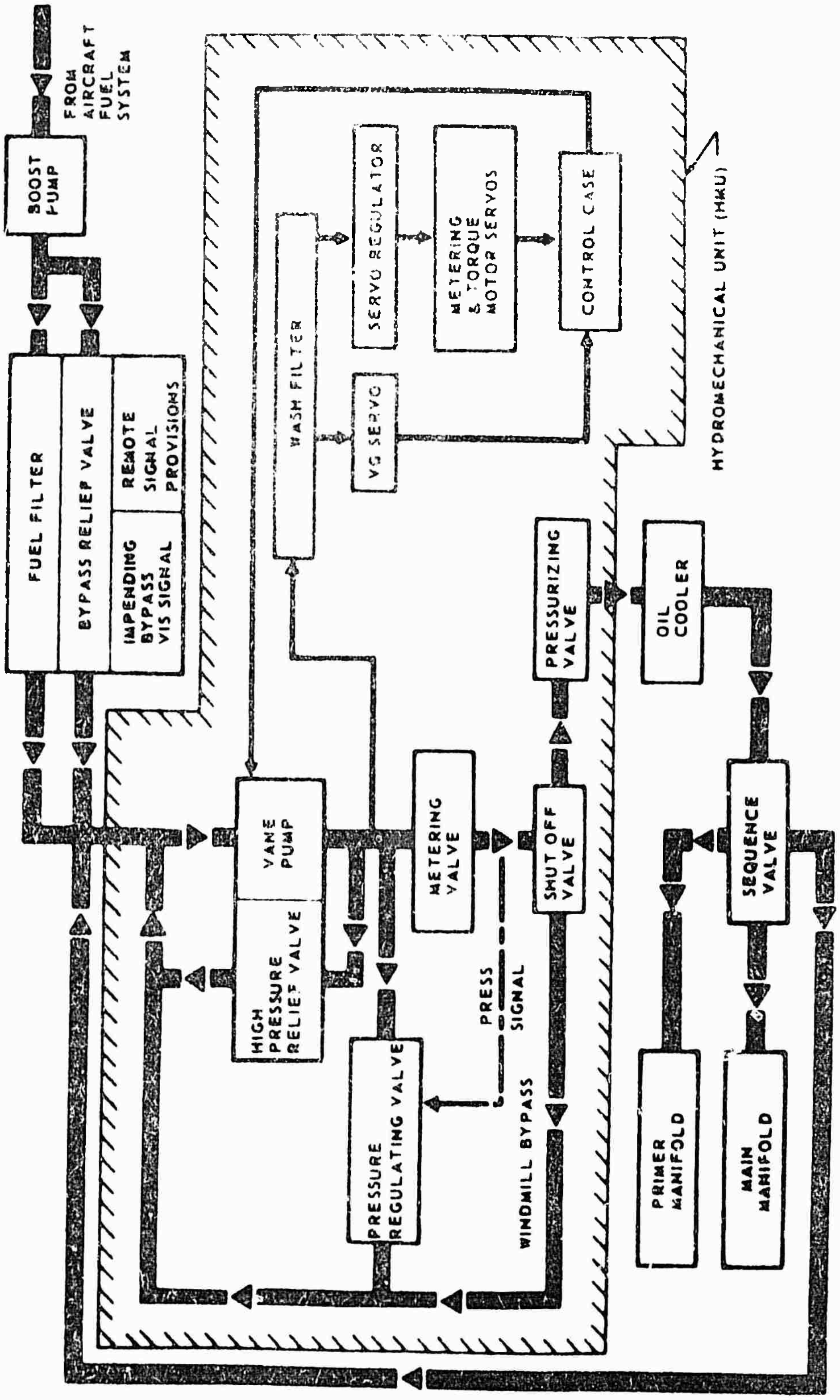


Figure 5. Fuel System Schematic

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 is controlled by mechanical linkage from the PAS and collective control and by the ECU. 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 speed governing; $T_{4.5}$ temperature limiting; Np overspeed protection completely independent of the normal Np governor; and load sharing on torque. It also provides the following noncontrol signals: Np speed 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.

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.

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.

Pneumatic Start System

11. 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. 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 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. A 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 right 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

12. There are four sets of cockpit controls, mechanical and electrical, which control engine functions. They are the power available spindle (PAS), the collective pitch controls, and the rotor speed reference input (trim wheel).

13. The PAS's are mechanically linked to the hydromechanical fuel control, which controls fuel shutoff, ground-idle, normal flight power range, 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	ECU lockout

14. 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 wheel which is electrically linked to the ECU. The wheel provides a limited rotor speed adjustment.

15. The engine emergency arming T-handles 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.

Automatic Governing Characteristics

16. In general, the HMU provides for gas generator control in the areas of acceleration limiting, stall and flameout protection, gas generator speed 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 6 and 7.

17. 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's are set at 120 degrees, the load demand signal is provided through the LDS. As the LDS is reduced from its maximum setting with a reduction of collective pitch, the desired gas generator speed 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 speed 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 8. In response to the resulting N_G speed 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

18. 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 manually.

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

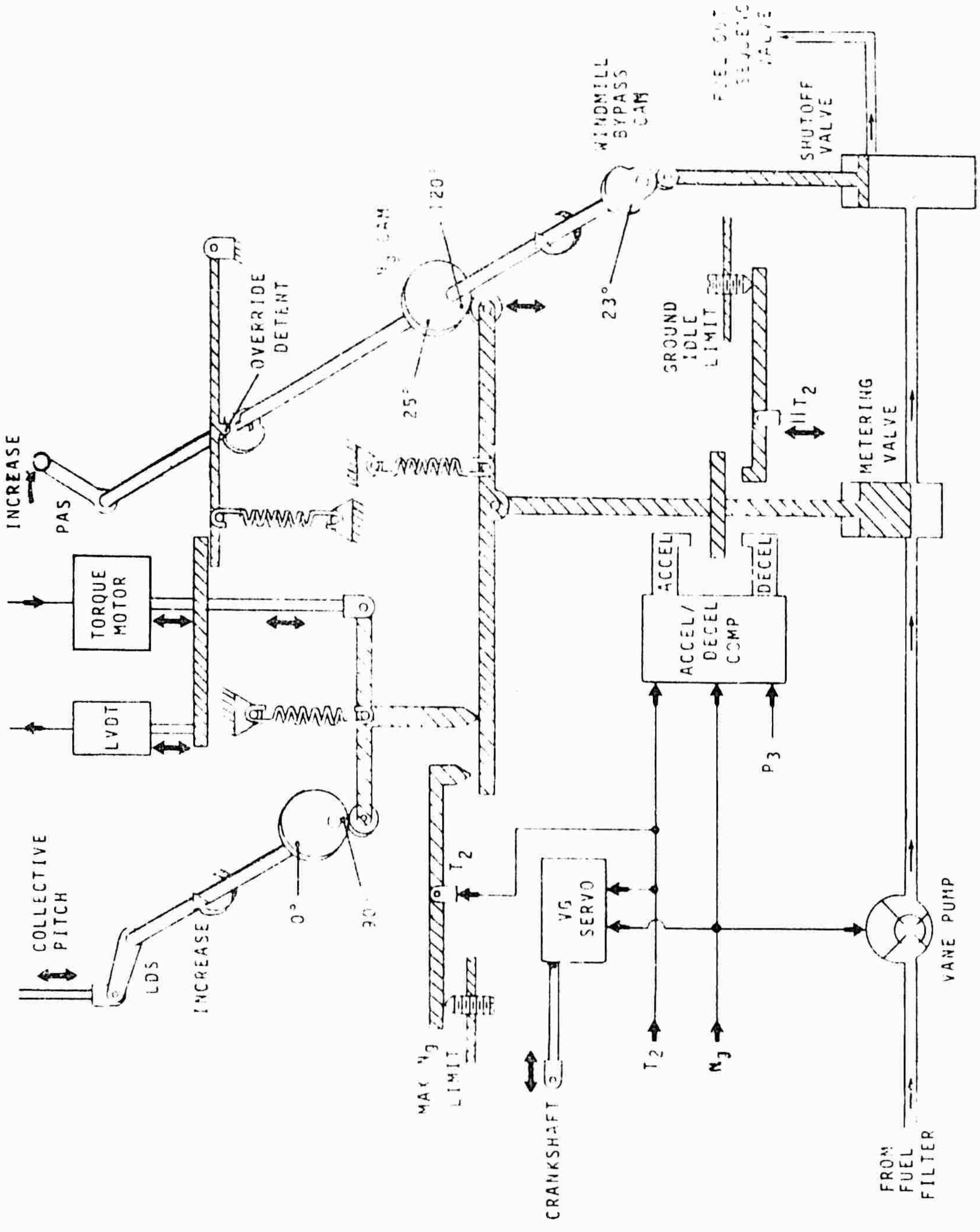


Figure 6. HMU Schematic Diagram

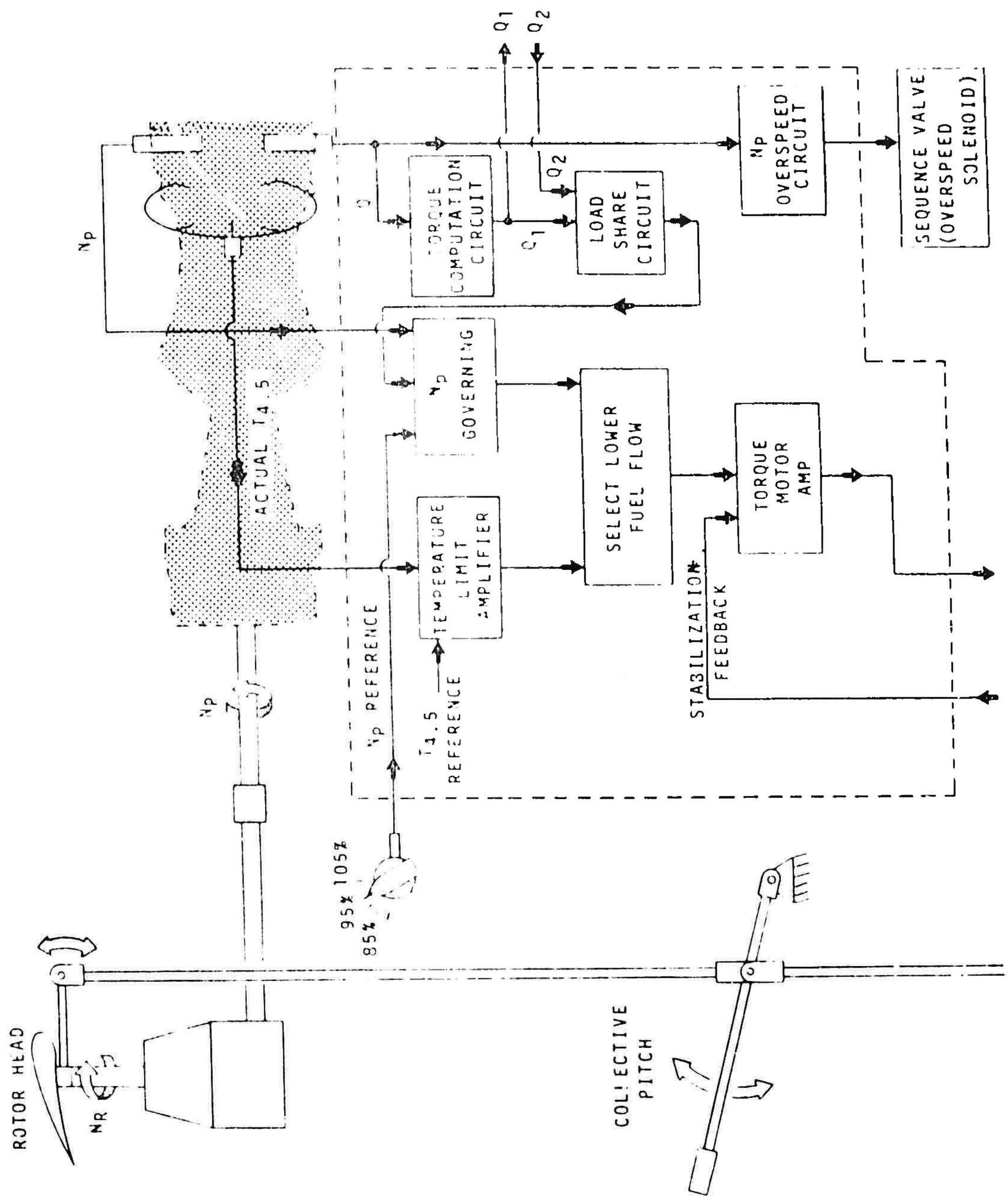


Figure 7. ECU Schematic Diagram

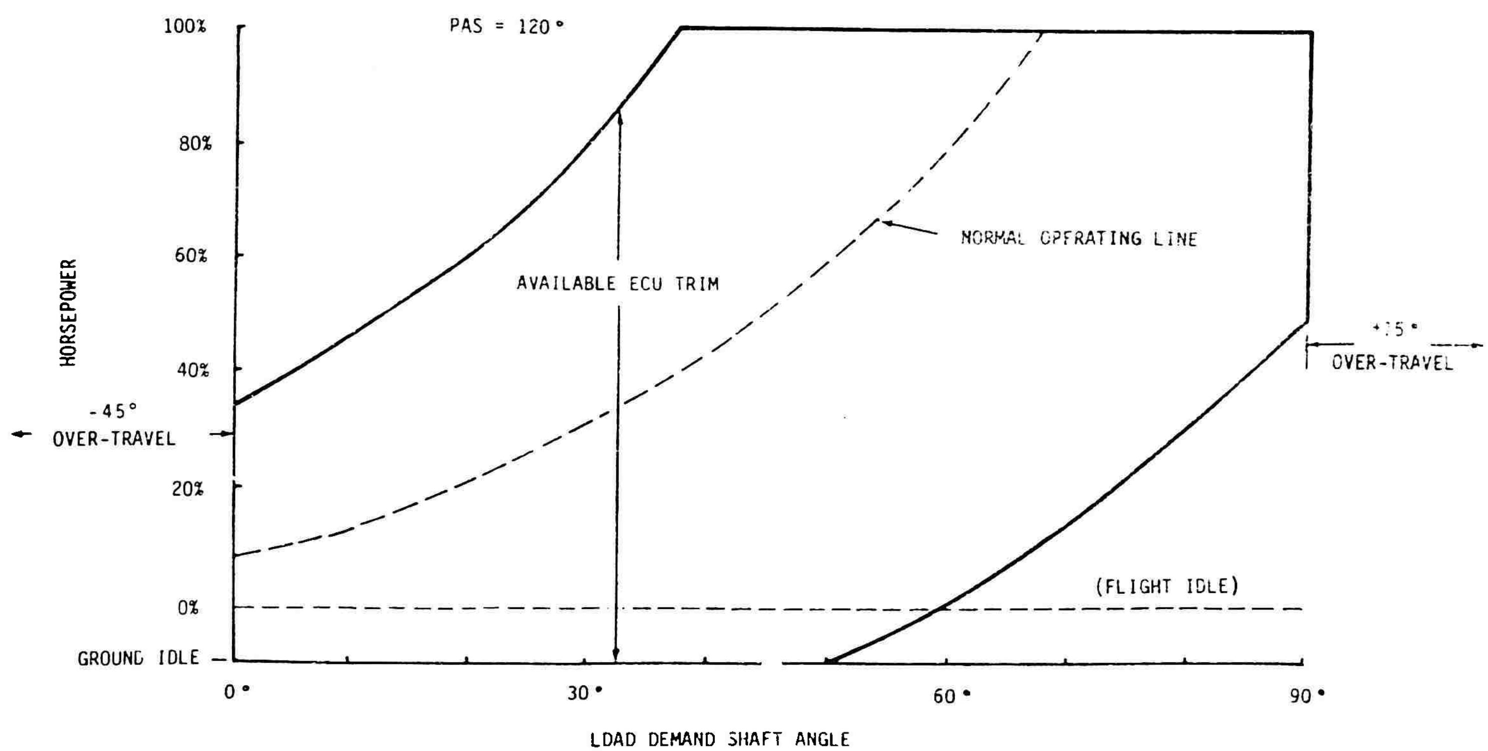


Figure 8. Collective Compensation Curve.

APPENDIX E. INSTRUMENTATION

1. Test instrumentation was istalled, calibrated, and maintained by BHT. A test airspeed boom with a swiveling pitot-static head was installed at the nose of each ship. The following parameters were measured with calibrated instrumentation and were displayed or recorded as indicated. The parameters were measured on both aircraft unless otherwise noted.

Pilot Station

Event switch

Pilot Panel

Airspeed (boom)
Altitude (boom)
Altitude (radar)[1]
Rate of climb (boom)
Rotor speed
Engine torque[2]
Measured gas temperature ($T_{4.5}$)[2]
Gas generator speed[1]
Control position:
- Longitudinal
- Lateral
- Directional
- Collective

Center-of-gravity normal acceleration
Angle of sideslip
Low airspeed

Copilot/Engineer Station

Instrumentation controls and lights
Event switch
Control fixtures
Time code display

Copilot/Engineer Panel

Airspeed (ship's system)
Airspeed (boom)
Altitude (boom)

[1] Performance aircraft only.
[2] Both engines.

Rate of climb (boom)
Rotor speed
Engine torque[1]
Measured gas temperature (T4.5)[1]
Gas generator speed[1]
Free air temperature
Angle of sideslip
Fuel used (totalization)
Time of day
Correlation counter

<u>Magnetic Tape Recorder</u>

Airspeed (ship's system)[2]
Airspeed (boom)
Altitude (boom)
Altitude (radar)[2]
Free air temperature
Rotor speed
Gas generator speed[1]
Power turbine speed[1]
Fuel used[1]
Fuel temperature (at flowmeters)[1]
Engine fuel flow[1]
Engine output shaft torque[1]
Measured gas temperature (T4.5)[1]
Engine inlet temperature[1,2]
Engine inlet total pressure[1,2]
Engine exhaust system static pressure[1,2]
Variable geometry stator No. 1 position[1,2]
Compressor discharge static pressure[1,2]
Engine air particle separator discharge static pressure[1,2]
Fuel control discharge pressure[1,2]
Main rotor shaft torque
Tail rotor shaft torque
Time of day (time code generator)
Pilot event
Engineer event
Correlation counter
Control position:
- Longitudinal cyclic
- Lateral cyclic
- Collective
- Engine condition lever[1]

[1] Both engines.
[2] Performance aircraft only.

Flight control augmentation positions:
- Longitudinal
- Lateral
- Directional

Control forces:
- Longitudinal cyclic
- Lateral cyclic
- Pedal

Aircraft attitudes:
- Pitch
- Roll
- Yaw

Aircraft angular velocities:
- Pitch
- Roll
- Yaw

Aircraft angular accelerations:
- Pitch
- Roll
- Yaw

Normal acceleration (at instrument panel and cg)
Angle of sideslip
Angle of attack
Main rotor blade angle
Main rotor flapping angle
Tail rotor blade angle
Tail rotor flapping angle
Eight vibration accelerations on both ships
24 additional vibration accelerations on SN 74-22246

2. Locations of the vibration accelerometers are shown in tables 1 and 2.

Table 1. Vibration Transducer Locations, Aircraft [illegible] 76-22256

Description	Axis	Fuselage Station	Buttline	Water Line
Pilot seat	Vertical	171	2 L	66
Pilot seat	Lateral	171	2 L	66
Pilot seat	Longitudinal	171	2 L	66
Pilot instrument panel (right)	Vertical	153	11.5 R	75.5
Pilot instrument panel (right)	Lateral	153	11.5 R	75.5
Pilot instrument panel (right)	Longitudinal	153	11.5 R	75.5
Pilot instrument panel (left)	Vertical	153	11.5 L	91.5
Pilot instrument panel (left)	Lateral	153	11.5 L	91.5
Pilot instrument panel (left)	Longitudinal	153	11.5 L	75.5
Pilot collective	Vertical	164	11 L	71
Pilot collective	Lateral	164	11 L	71
Pilot heel rest	Vertical	148	3.5 R	60.5
Pilot heel rest	Lateral	148	3.5 R	60.5
Copilot seat	Vertical	236	1.5 L	78
Copilot seat	Lateral	236	1.5 L	78
Copilot seat	Longitudinal	236	1.5 L	78
Copilot instrument panel (center)	Vertical	216	3 L	91.5
Copilot instrument panel (center)	Lateral	216	3 L	91.5
Copilot instrument panel (center)	Longitudinal	216	3 L	91.5
Avionics bay (left)	Vertical	227	28 L	53.5
Avionics bay (left)	Lateral	227	28 L	53.5
Avionics bay (left)	Longitudinal	227	28 L	53.5
Avionics bay (right)	Vertical	226.5	28 R	53.5
Avionics bay (right)	Lateral	226.5	28 R	53.5
Avionics bay (right)	Longitudinal	226.5	28 R	53.5
Center of gravity	Vertical	288	Zero	74
Center of gravity	Lateral	288	Zero	74
Center of gravity	Longitudinal	288	Zero	74
Wing (left hand)	**Vertical**	**293**	**93 L**	**56**
Top of transmission	**Vertical**	**288**	**Zero**	**100**
90° gearbox	**Vertical**	**662**	**3 L**	**109**

Table 2. Vibration Transducer Locations.
Aircraft SN 74-22246

Description	Axis	Fuselage Station	Buttline	Water Line
Pilot seat	Vertical	183	1.5 R	64
Pilot seat	Lateral	183	1.5 R	64
Copilot seat	Vertical	231	1.5 L	77
Copilot seat	Lateral	231	1.5 L	77
Center of gravity	Vertical	280.5	2.0 R	74
Center of gravity	Lateral	280.5	2.0 R	74
Pilot instrument panel	Vertical	153.5	10.5 L	74
Pilot instrument panel	Lateral	153.5	10.5 L	74

APPENDIX F. TEST TECHNIQUES AND DATA ANALYSIS METHODS

INTRODUCTION

1. This appendix contains some of the test techniques and data reduction and analysis methods used to evaluate the YAH-63. The topics discussed include hover, tail rotor performance, climb, level flight performance, and specific range.

GENERAL

2. The helicopter performance test data were generalized by use of nondimensional coefficients and were such that the effects of compressibility and blade stall were not separated and defined. The following nondimensional coefficients were used to generalize the hover, level flight, and climb test results obtained during this flight test program.

a. Coefficient of power (C_P):

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

b. Coefficient of thrust (C_T):

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

c. Advance ratio (μ):

$$\mu = \frac{1.6889\ V_T}{\Omega R}$$

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

$$M_{tip} = \frac{1.6889\ V_T + (\Omega R)}{a}$$

Where:

shp = Engine output shaft horsepower

550 = Conversion factor (ft-lb/sec/shp)

ρ = Air density (slug/ft^3)

A = Main rotor disc area (ft^2)

Ω = Main rotor angular velocity (radian/sec)

R = Main rotor radius (ft)

W = Aircraft gross weight (lb)

1.6889 = Conversion factor (ft/sec/kt)

V_T = True airspeed (kt)

a = Speed of sound (ft/sec)

SHAFT HORSEPOWER REQUIRED

3. The engine output shaft torque was determined by measuring the torsional strain of the engine output shaft. From the engine manufacturer's dynamic calibration, the engine's shaft torsional strain was related to the applied torque. The shaft strain measuring system was electrical and its output was recorded on the on-board data recording system and displayed on cockpit indicators in units of foot-pounds. The output shp was determined from the engine's output shaft torque and rotational speed by the following equation:

$$shp = \frac{2\pi \times N_P \times Q}{33000}$$

Where:

Np = Engine output shaft rotational speed (rpm)

Q = Engine output shaft torque (ft/lb)

33,000 = Conversion factor (ft-lb/min/shp)

TAIL ROTOR PERFORMANCE

4. During the hover performance tests, tail rotor performance parameters were also recorded. Terms in equations 1, 2, and 3 were replaced by tail rotor parameters to nondimensionalized tail rotor performance as follows:

shp = Tail rotor shaft horsepower

A = Tail rotor disc area (ft^2)

Ω = Tail rotor angular velocity (radian/sec)

R = Tail rotor radius (ft)

T = Tail rotor thrust (lb)

Q = Tail rotor torque (ft-lb)

Approximate tail rotor thrust was calculated from the following equation:

$$T = \frac{Q_{MR}}{l_t}$$

Where:

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

l_t = Perpendicular distance between center lines of main and tail rotor shafts in feet

VERTICAL CLIMB PERFORMANCE

5. Vertical climb tests were conducted in ambient wind conditions of 3 knots or less. The data were recorded with the helicopter in a stabilized vertical climb between 200 and 500 feet AGL. The C_T was maintained approximately constant by adding ballast between each climb. Rotor speed was maintained at 276 rpm. The climb rates were measured by means of a radar altimeter and two ROI's. The raw data were reduced to referred and generalized parameters as shown below.

$$VV_R = V_v/\Omega R\sqrt{CT/2}$$

$$\Delta CP_{GEN} = (CP_c - CP_h)\ /.707CT^{3/2}$$

$$V_{v_R} = V_v/\sqrt{\theta}$$

$$shp_R = shp/\delta\sqrt{\theta}$$

$$GW_R = GW/\delta$$

Where:

VV_R = Vertical velocity ratio

V_v = Vertical velocity

ΔCP_{GEN} = Generalized excess power coefficient from hover

CP_c = Coefficient of power for climb

CP_h = Coefficient of power for hover

V_{vR} = Referred vertical speed

shp_R = referred shp

FORWARD FLIGHT CLIMBS

6. Single-engine service ceiling was determined by conducting two continuous climbs, one on each engine, and correcting the rate-of-climb data for variations in power and gross weight. The climb airspeed versus altitude schedule used during these climbs corresponded to the airspeeds for minimum power required in level flight and were obtained from the level flight performance data. Sawtooth climbs were flown to determine the power correction factor (Kp) and the weight correction factor (Kw).

7. 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:

$$R/C_T = \frac{dH_p}{dt} \times \frac{T_t}{T_s}$$

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 which the slope is taken (°K)

At the test density altitude, rate of climb was corrected for differences between power output during the test and specification power available, using the following equation:

$$\Delta R/C_P = K_P \times \frac{\Delta shp}{GW_t} \times 33{,}000$$

The rate of climb was also corrected for differences between the test gross weight and the desired (standard) gross weight, using the following equation:

$$\Delta R/C_W = K_W \times shp_s \times 33{,}000 \left(\frac{1}{GW_s} - \frac{1}{GW_t}\right)$$

Where:

$\Delta R/C_p$ = Rate of climb difference due to power difference (ft/min)

K_p = Coefficient of power correction

Δshp = Difference in standard shp available and test shp measured

GW_t = Test gross weight (lb)

K_W = Coefficient of weight correction

shp_S = Standard shp obtained from model specification engine

GW_S = Standard gross weight (lb)

$\Delta R/C_W$ - Rate of climb difference due to weight difference (ft/min)

The standard rate of climb was finally determined by the following equation:

$$R/C_s = R/C_T + \Delta R/C_p + \Delta R/C_w$$

Where:

R/C_S = Final rate of climb standard (ft/min)

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

$\Delta T/C_p$ = Rate of climb difference due to power difference (ft/min)

$\Delta R/C_W$ = Rate of climb difference due to weight difference (ft/min)

LEVEL FLIGHT PERFORMANCE AND SPECIFIC RANGE

8. 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 the ratio of gross weight to air density ratio (W/σ) constant, altitude was increased as fuel was consumed.

9. 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:

$$shp_s = shp_t \times \frac{\rho_s}{\rho_t}$$

Where:

t = Test day

s = Standard day

10. Specific range was calculated using level flight performance curves and the specification installed engine fuel flow characteristics.

$$NAMPP = \frac{V_T}{W_f}$$

Where:

NAMPP = Nautical air miles per pound of fuel

V_T = True airspeed (kt)

W_f = Fuel flow (lb/hr)

11. Changes in the equivalent flat plate area (Δf_e) for various aircraft configurations were calculated by the following equation:

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

Where:

Δf_e = Change in flat plate area (ft^2)

ΔC_p = Change in coefficient of power

A = Main rotor disc area (ft^2)

AUTOROTATIONAL DESCENT PERFORMANCE

12. 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 expression:

$$\text{R/D tapeline} = \frac{dH_P}{dt} \times \frac{T_t}{T_s}$$

Where:

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

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

T_t = Test ambient air temperature (°K)

T_s = Standard ambient air temperature (°K)

ENGINE INLET CHARACTERISTICS

13. The engine inlet temperature and pressure characteristics were determined by the following formulas:

a. Temperature rise:

$$\Delta T = T_1 - T_a$$

Where:

ΔT = Temperature difference (°K)

T_1 = Engine inlet total temperature (°K)

T_a = Ambient temperature (°K)

b. Pressure ratio:

$$\frac{P_{T_1}}{P_a} = \frac{P_{T_1} - P_a}{P_a} + 1$$

Where:

P_{T_1} = Engine inlet total pressure (in. of mercury)

P_a = Ambient pressure (in. of mercury)

ENGINE EXHAUST LOSSES

14. Engine exhaust loss was determined by measuring the average static pressure at the exhaust station (P_{S7}), power turbine measured gas temperature ($T_{4.5}$), power turbine speed (N_p), and inlet pressure and temperatures (P_{T1} and T_1). The procedures for reduction of the test data were provided by AVSCOM in reference 8, appendix A. Figures 1 and 2 of reference 8 provided the referred velocity head at station 7 ($V_{H}7/\delta 1$) and exhaust swirl angle (Γ_7) as a function of referred measured gas temperature ($T_{4.5}/\theta_1^{.96}$) and referred power turbine speed ($N_p/\sqrt{\theta 1}$). The exhaust duct pressure recovery coefficient (PS9D7Q) was computed by equation 22 and presented as a function of Γ_7 in figures 148 and 149, appendix D. This curve was input to the YT 700-GE-700 engine deck No. 73004 when determing specification power available.

$$PS9D7Q = \frac{(P_{S0} - P_{S7})/\delta 1}{V_{H}7/\delta 1} + .4070\sin^2\Gamma_7$$

DRIVE TRAIN LOSSES

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

$$\Delta HP = ESHP - RHP$$

Where:

ESHP = Total engine shaft horsepower (both)

RHP = Main rotor horsepower plus tail rotor horsepower

SHAFT HORSEPOWER AVAILABLE AND SPECIFICATION FUEL FLOW

16. Shaft horsepower available and specification fuel flow were obtained from the General Electric PIDS, AMC-CP-2222-02000. Inlet and exhaust losses used were derived from figures 143 through 150, appendix G.

17. 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, and gas generator speed (N_G) were referred as follows.

a. Referred SHP (RSHP):

$$RSHP = \frac{SHP}{(\delta_1)(\theta_1)^{0.50}} \text{ (SHP)}$$

b. Referred gas temperature (RGAST):

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

c. Referred fuel flow (RWF):

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

d. Referred gas generator speed (RN_G):

$$RN_G = \frac{N_G}{(\theta_1)^{0.50}} \text{ (\%)}$$

Where:

$$\delta_1 = \frac{P_{T1}}{14.697}$$

$$\theta_1 = \frac{T_1}{288.15}$$

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

W_f = Engine fuel flow (lb/hr)

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

AIRSPEED CALIBRATION

18. The boom and ship's standard pitot static system were calibrated by using the pace aircraft method to determine the airspeed position error. Calibrated airspeed (V_{cal}) was obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}) and position error (Δ_{pc}).

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc}$$

19. Equivalent airspeed was used to reduce the flight test data, as it is a direct measure of the free stream dynamic pressure (q).

$$V_e = V_{cal} + \Delta V_c$$

Where:

ΔV_c is the compressibility correction, $q = 0.00339\ V_c^2$

20. True airspeed (V_t) was calculated from the equivalent airspeed and density ratio.

$$V_t = \frac{V_e}{\sqrt{\sigma}}$$

Where:

σ = Density ratio $(\frac{\rho}{\rho_0})$ where ρ_0 is the density at sea level on a standard day

WEIGHT AND BALANCE

21. Prior to testing, the aircraft gross weight and longitudinal and lateral cg were determined by using calibrated scales. The longitudinal cg was calculated by a summation of moments about a reference datum line (FS 0.0). The aircraft was weighed empty in the clean configuration, which included instrumentation minus all munitions and fuel.

HANDLING QUALITIES RATING SCALE

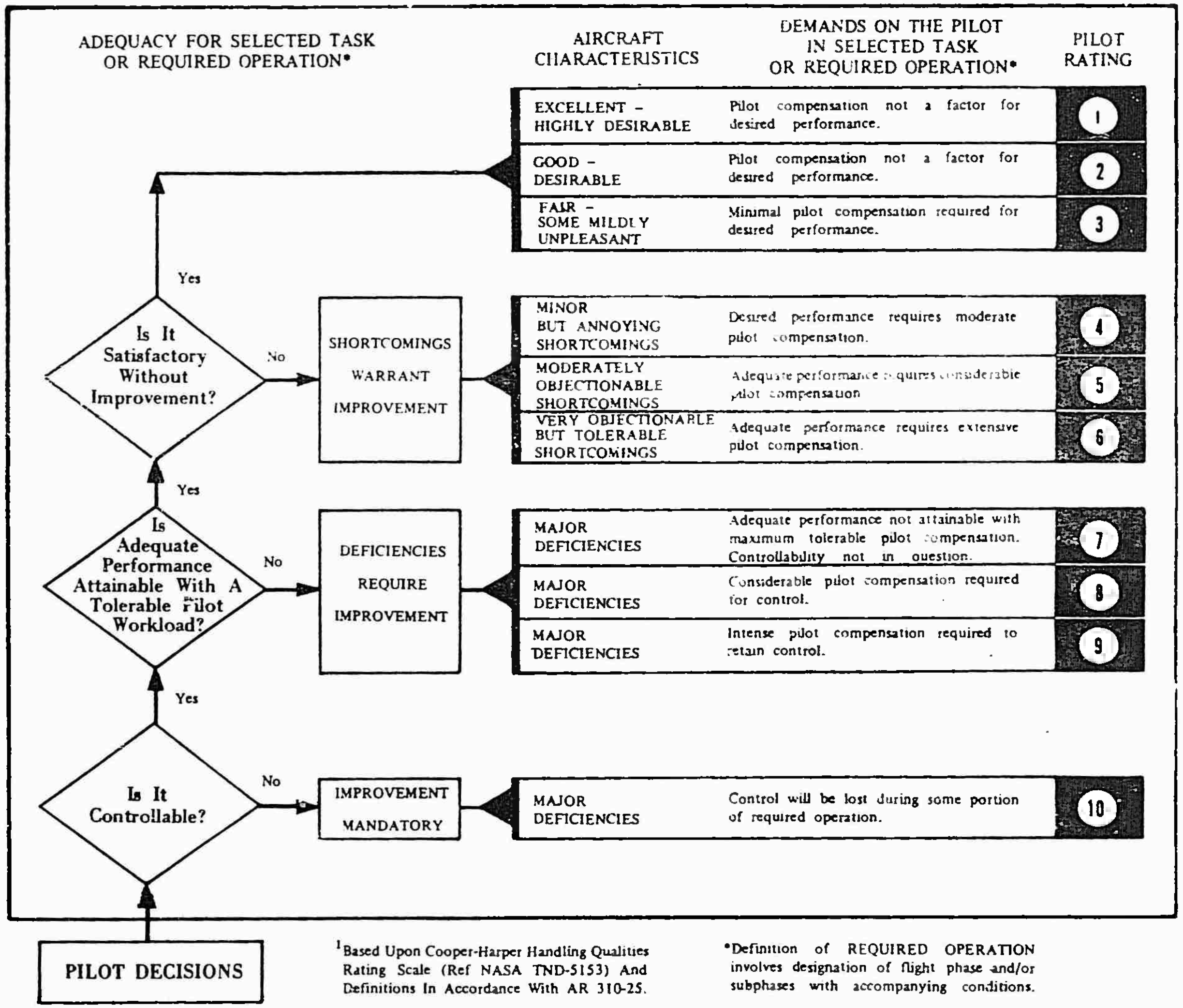
22. An HQRS was used to quantify pilot assessments of workload during mission tasks and is shown as figure 1.

SPACE POSITIONING

23. Time history data for lateral accelerations, vertical displacements, and vertical climbs were recorded on airborne magnetic tape and from cockpit instruments. Additional data were recorded on space-positioning equipment during the maneuvers. Time history data were also obtained from the following equipment:

a. Radar altimeter: Height AGL and vertical rate of descent.

b. Two recording observation instruments (ROI): Azimuth and elevation of the aircraft were printed continuously on paper during each data run. The following



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

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

Figure 1. Handling Qualities Rating Scale.

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equations were used to reduce these data to a workable form and figure 2 illustrates the geometric relationship of the aircraft to the ROI.

Horizontal displacement:

$$dx = \frac{x_1 \text{ TAN } \alpha_2 + y_1 \text{ TAN } \alpha_1 \text{ TAN } \alpha_2}{1 + \text{TAN } \alpha_1 \text{ TAN } \alpha_2}$$

$$dx = (y_1 + dy) \text{ TAN } \alpha_1$$

$$A_1 = \sqrt{dx^2 + (dy + y_1)^2}$$

$$A_2 = \sqrt{dy^2 + (dx + x_1)^2}$$

Elevation:

$$dz_1 = A_1 \text{ TAN } \beta_1$$

$$dz_2 = A_2 \text{ TAN } \beta_2$$

$$dz = \frac{dz_1 + dz_2}{2}$$

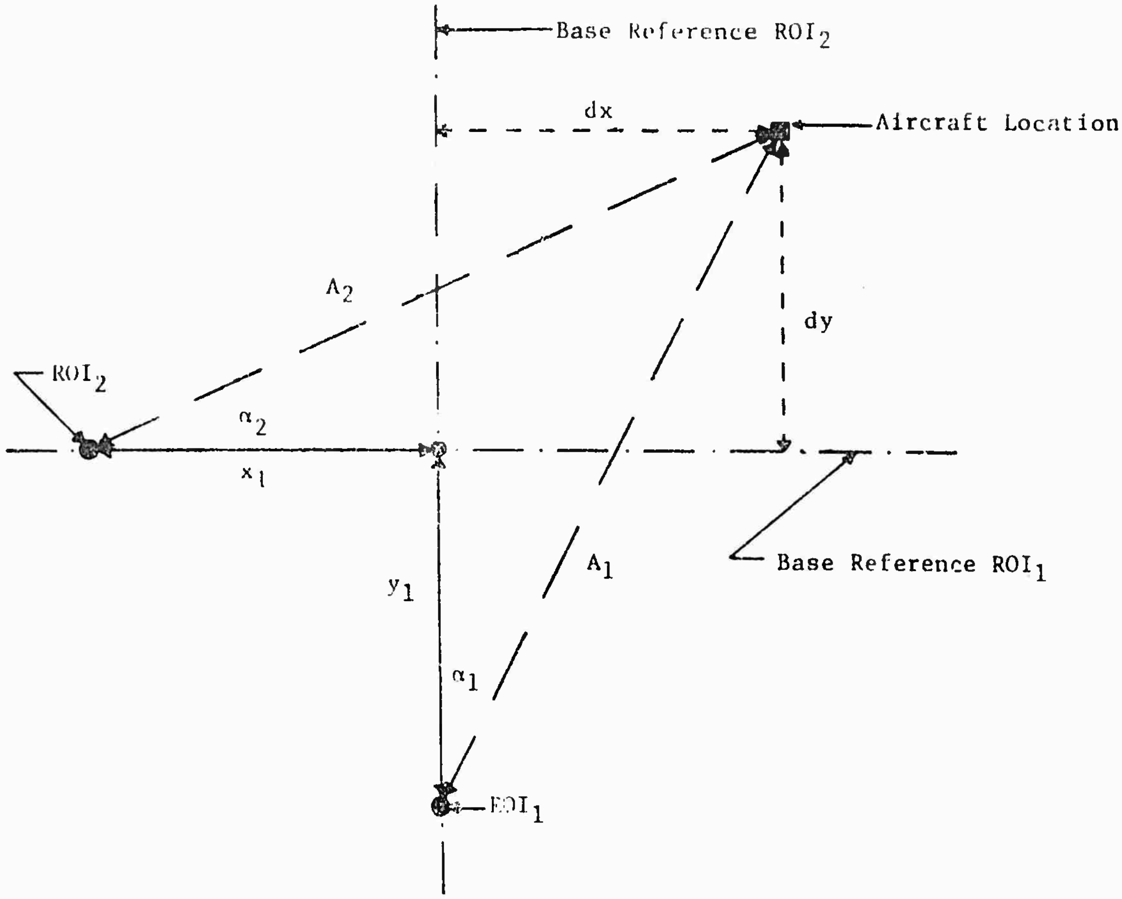
Where:

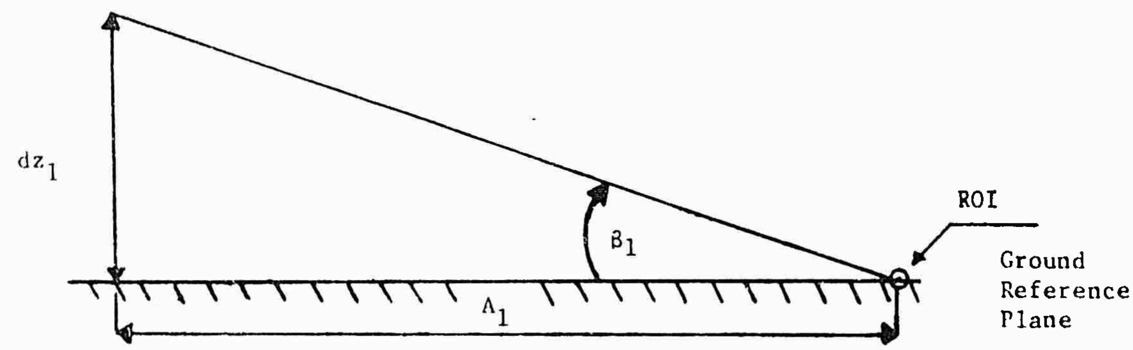
dx and dy = Horizontal displacement of aircraft from base reference

dz = Elevation of aircraft from ground reference plane

A_1 and A_2 = Horizontal displacement of aircraft from ROI's

x_1 and x_2 = Distance of ROI offset from base reference





Base reference line: An imaginary line located at a convenient distance from the aircraft and ROI's.

Figure 2.

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α_1 and α_2 = Horizontal displacement angles

β_1 and β_2 = Vertical displacement angles

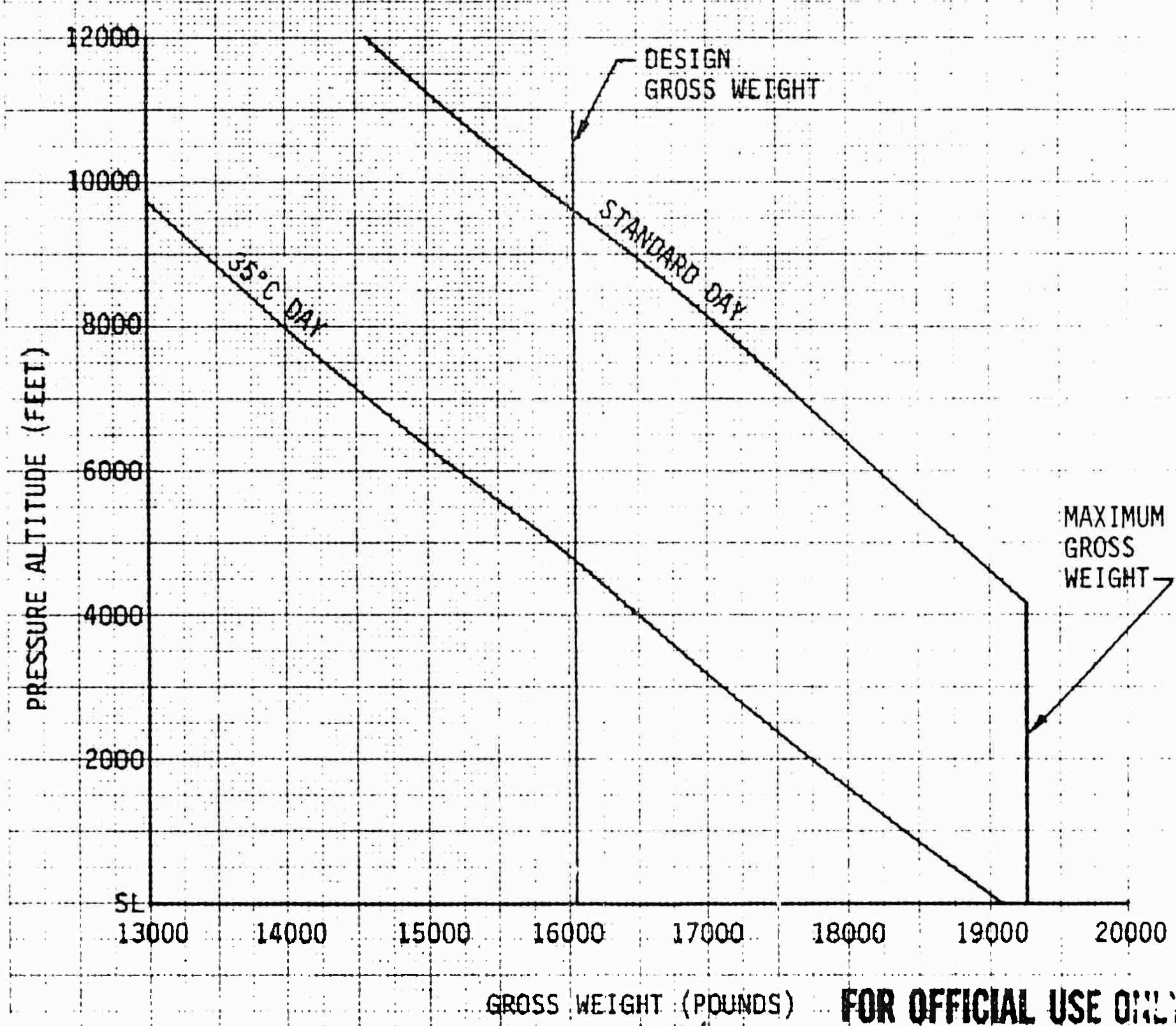
Rate of ascent and horizontal velocity were obtained by taking the first time derivatives of the resulting displacements.

APPENDIX G. TEST DATA

INDEX

FIGURE 1
SUMMARY HOVER PERFORMANCE
OUT OF GROUND EFFECT
YAH-63 USA S/N 74-22246
INTERMEDIATE RATED POWER

NOTES: 1. SHP BASED ON FIGURES 138 AND 139.
2. ROTOR SPEED = 276 RPM.



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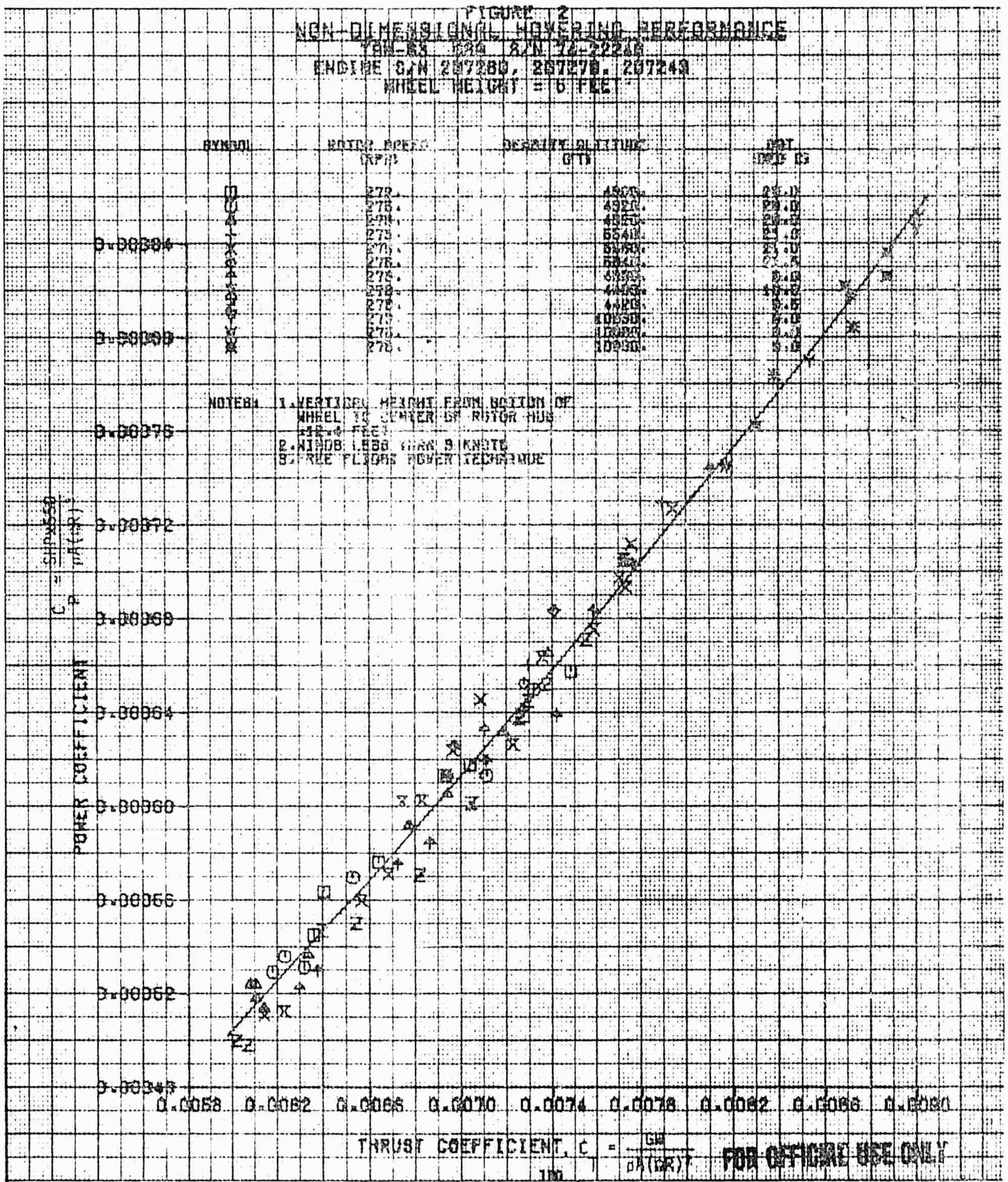
FIGURE 2
NON-DIMENSIONAL HOVERING PERFORMANCE
ENGINE S/N 207280, 207270, 207243
WHEEL HEIGHT = 6 FEET
SYMBOL
ROTOR SPEED (RPM)
DENSITY ALTITUDE (FT)
NOTES:
1. VERTICAL HEIGHT FROM BOTTOM OF WHEEL TO CENTER OF ROTOR HUB
3. FREE FLIGHT HOVER TECHNIQUE
POWER COEFFICIENT
0.00384
0.00380
0.00376
0.00372
0.00368
0.00364
0.00360
0.00356
0.00352
0.00348
0.0058
0.0062
0.0066
0.0070
0.0074
0.0078
0.0082
0.0086
0.0090
THRUST COEFFICIENT
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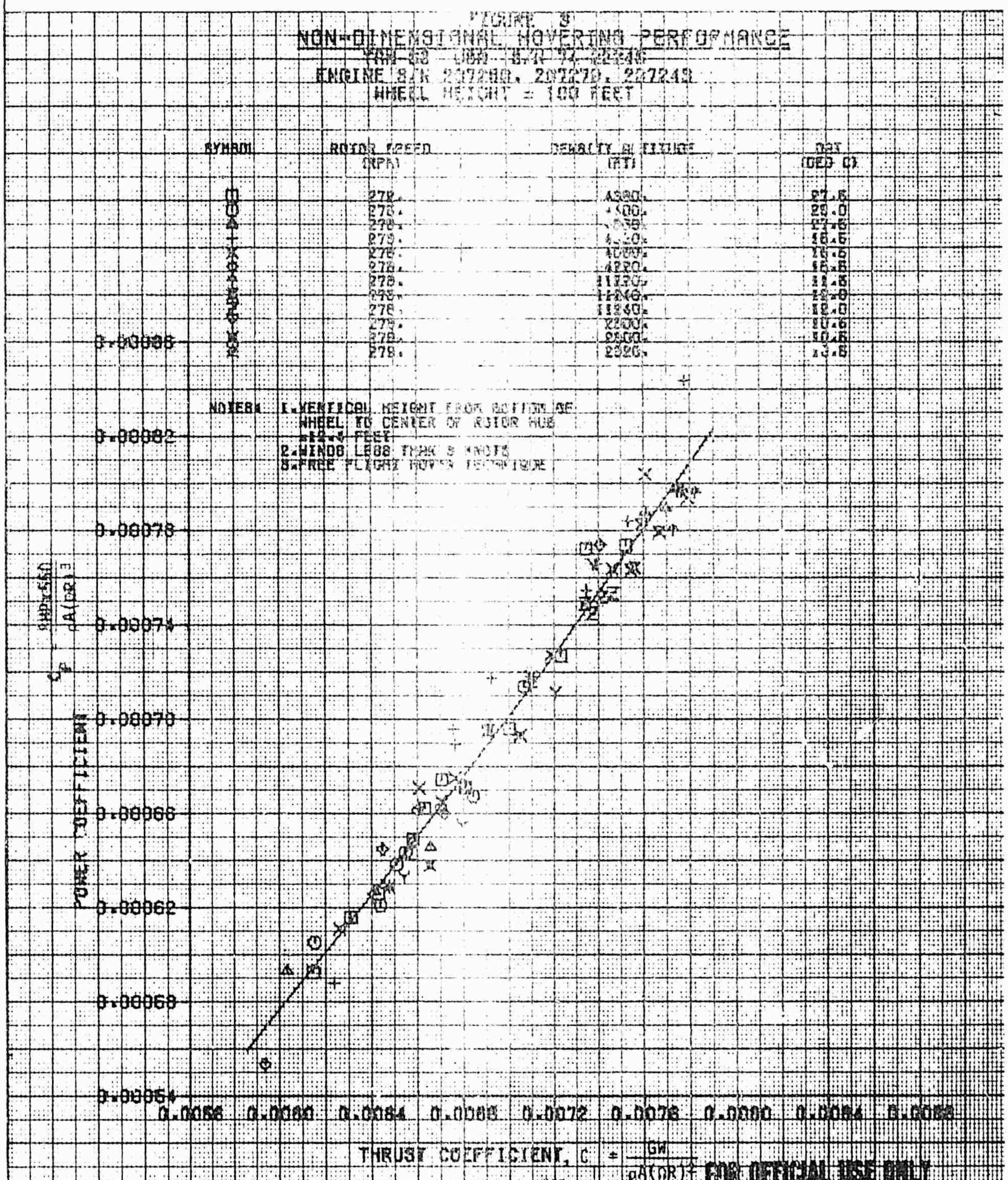


FIGURE 9
NON-DIMENSIONAL HOVERING PERFORMANCE
ENGINE S/N 207280, 207270, 207249
WHEEL HEIGHT = 100 FEET
SYMBOL
ROTOR SPEED (RPM)
DENSITY ALTITUDE (FT)
OAT (DEG C)
278.
278.
278.
278.
278.
278.
278.
278.
278.
278.
278.
278.
4390.
4220.
11220.
11240.
2200.
2320.
27.6
28.0
16.6
15.6
15.8
11.6
12.0
10.6
10.6
NOTES: 1. VERTICAL HEIGHT FROM BOTTOM OF WHEEL TO CENTER OF ROTOR HUB
2. WINDS LESS THAN 3 KNOTS
3. FREE FLIGHT HOVER TECHNIQUE
0.00086
0.00082
0.00078
0.00074
0.00070
0.00066
0.00062
0.00058
0.00054
POWER COEFFICIENT
0.0056
0.0060
0.0064
0.0068
0.0072
0.0076
0.0080
0.0084
0.0088
THRUST COEFFICIENT

FIGURE 9
NON-DIMENSIONAL TAIL ROTOR PERFORMANCE
YAH-63, USA S/N 74-22246
ENGINE S/N 207278, 207280

SYMBOL	ROTOR SPEED (RPM)	DENSITY ALTITUDE (FT)	OAT (DEG C)	WHEEL HEIGHT (FT)
⊡	272	4300	28.0	5
⊡	279	4200	18.0	5
△	275	4320	28.0	5
+	275	4280	18.0	5
X	278	4320	28.0	5
◇	279	4300	10.5	5
◇	273	4020	14.5	100
X	273	2300	10.5	100
Z	228	4960	16.0	100
Y	276	2300	10.5	100
X	278	5000	16.5	100
✱	279	2320	10.5	100

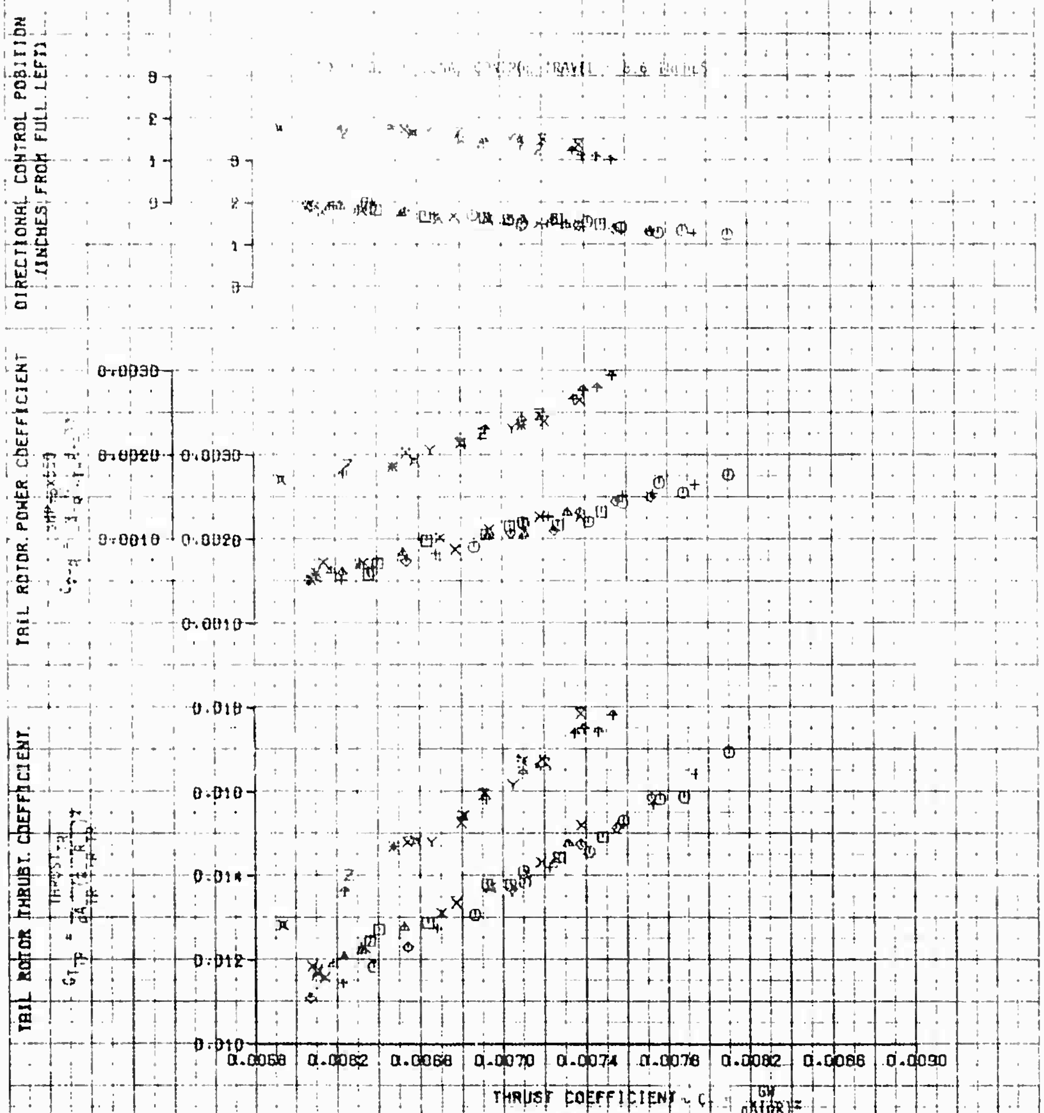
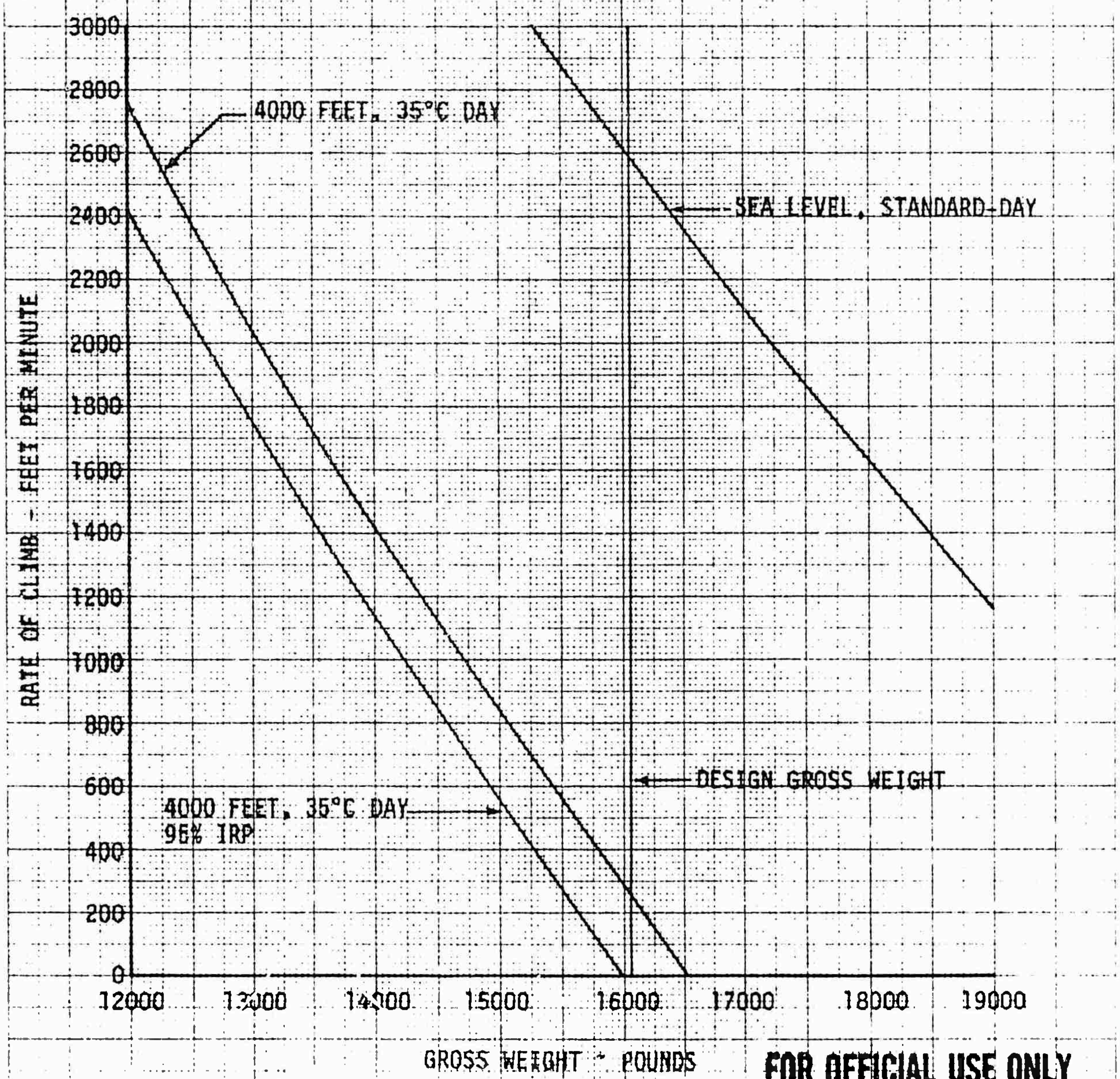


FIGURE 5

VERTICAL CLIMB PERFORMANCE SUMMARY

YAH-63 USA S/N 74-22246

NOTES:
1. MAXIMUM RATE OF CLIMB AT INTERMEDIATE RATED POWER (IRP) UNLESS OTHERWISE NOTED.
2. ROTOR SPEED = 276 RPM.
3. SPECIFICATION POWER BASED ON T700-GE-700 PID SPECIFICATION AMC-CP-2222-02000, 2 Feb 73.
4. RATE OF CLIMB CALCULATED FROM FIGURE 6.



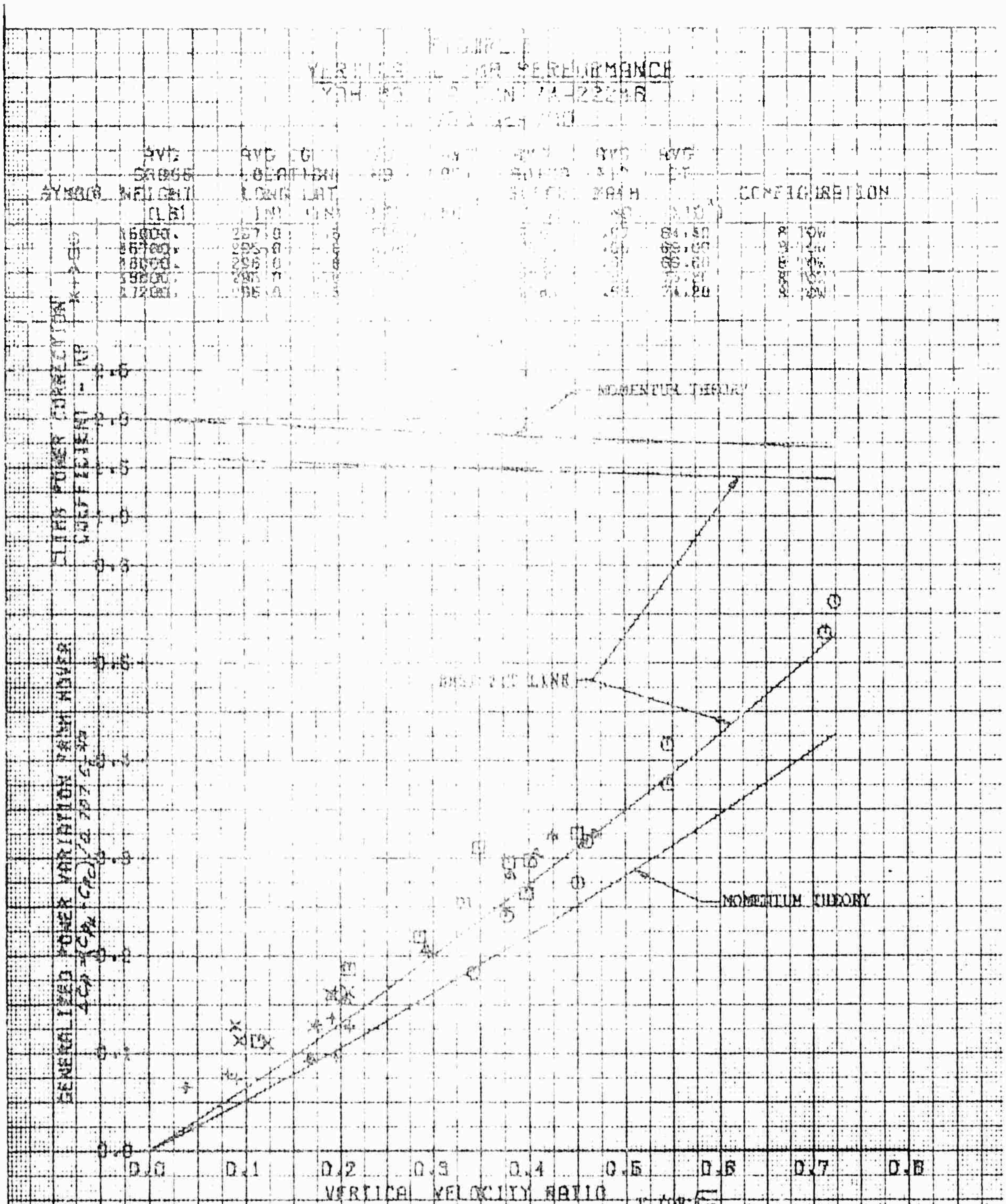


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FIGURE 7
VERTICAL CLIMB PERFORMANCE
YAH-63 USA S/N 74-22246
YT 700-GE-700

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG HD (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG TIP MACH NO.	AVG CT (X10⁴)	CONFIGURATION
○	16000.	297.0	-.6	5620.	14.	276.	.67	64.44	8 TOW
□	16700.	296.0	-.6	4300.	25.	275.	.66	69.67	8 TOW
△	16000.	296.0	-.6	5070.	22.	276.	.66	69.63	8 TOW

NOTE: DATA FAIRINGS BASED ON BEST-FIT LINES ON FIGURE 6.

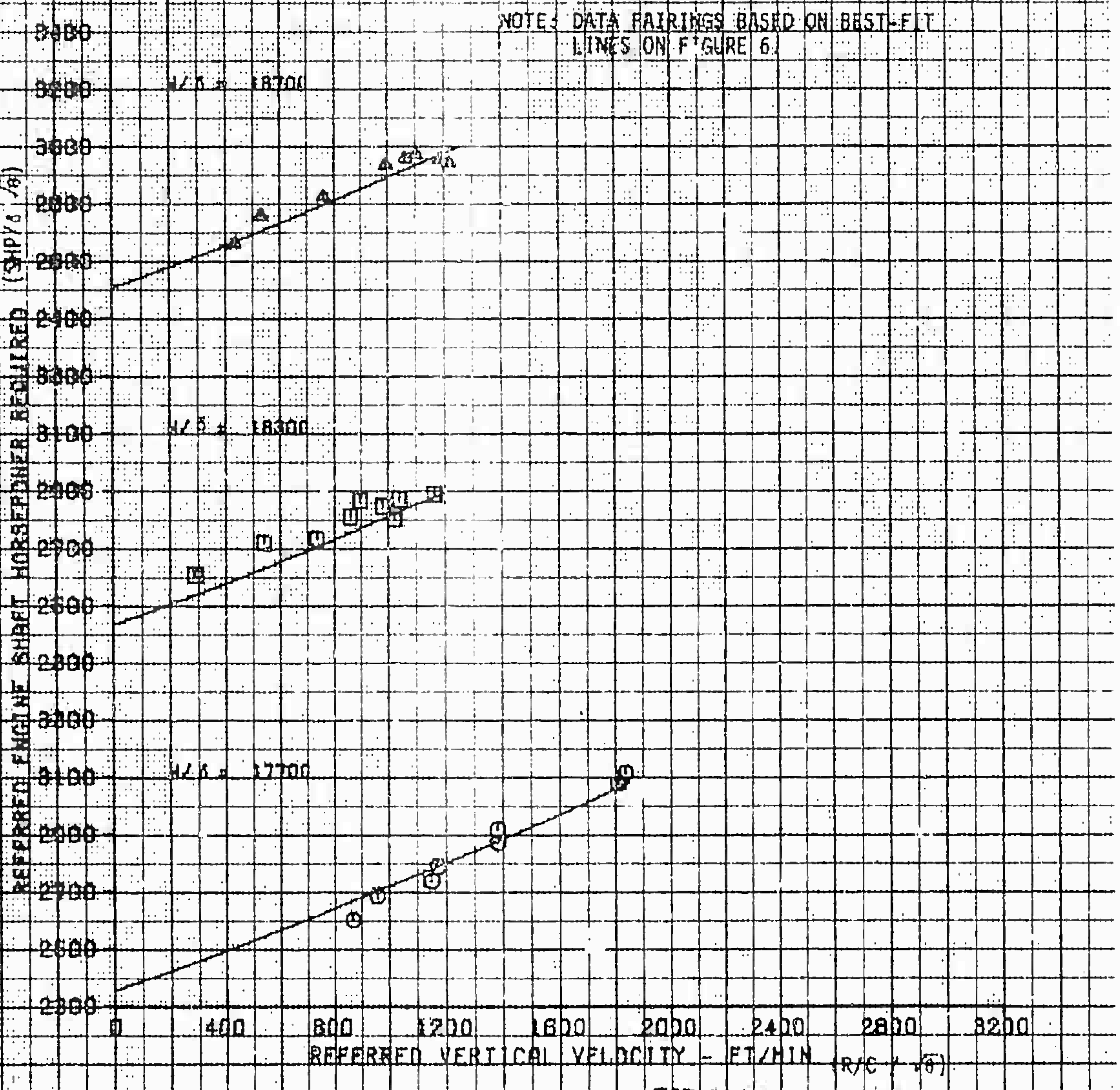

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FIGURE 8
VERTICAL CLIMB PERFORMANCE
YAH-63 USA S/N 74-22246
YT 700-GE-700

SYMBOL	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN)	AVG CG LOCATION LAT (IN)	AVG HD (FT)	AVG OAT (DEG C)	AVG ROTOR SPEED (RPM)	AVG TIP MACH NO.	AVG CT (X10^4)	CONFIGURATION
+	18600.	206.0	-.8	5960.	23.	276.	.68	73.18	8 TOW
⌅	17900.	205.0	-.8	6030.	20.	276.	.68	74.23	8 TOW

NOTE: DATA FAIRINGS BASED ON BEST-FIT LINES ON FIGURE 6.

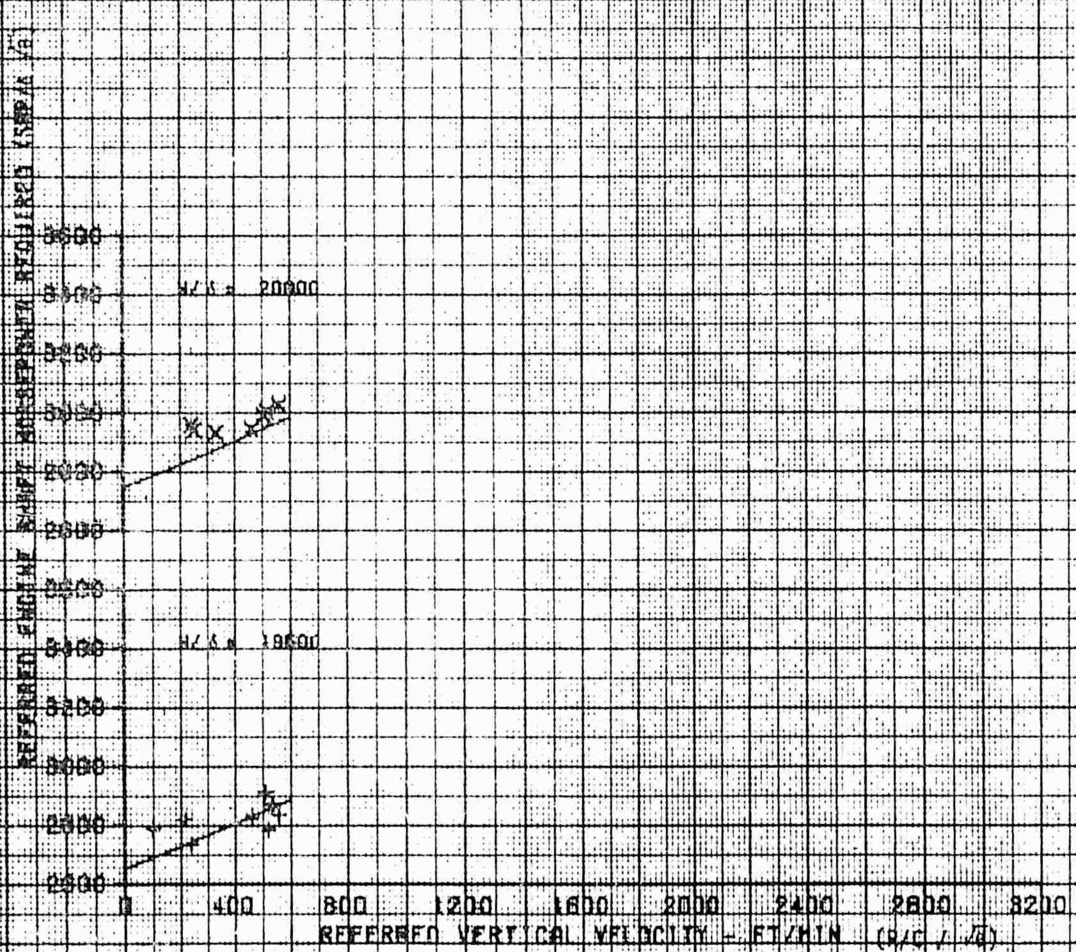


FIGURE 9
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF HORSEPOWER
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	AVG C_T	TRIM CALIBRATED AIRSPEED (KTS)
16080	293.6(FWD)	-0.1(LT)	6980	16.0	276	0.00723	76

NOTES:
1. 8 TOW CONFIGURATION
2. $K_p = \frac{\Delta R/C}{\Delta SHP} \times \frac{GW}{33000}$
3. $K_p = 0.801$

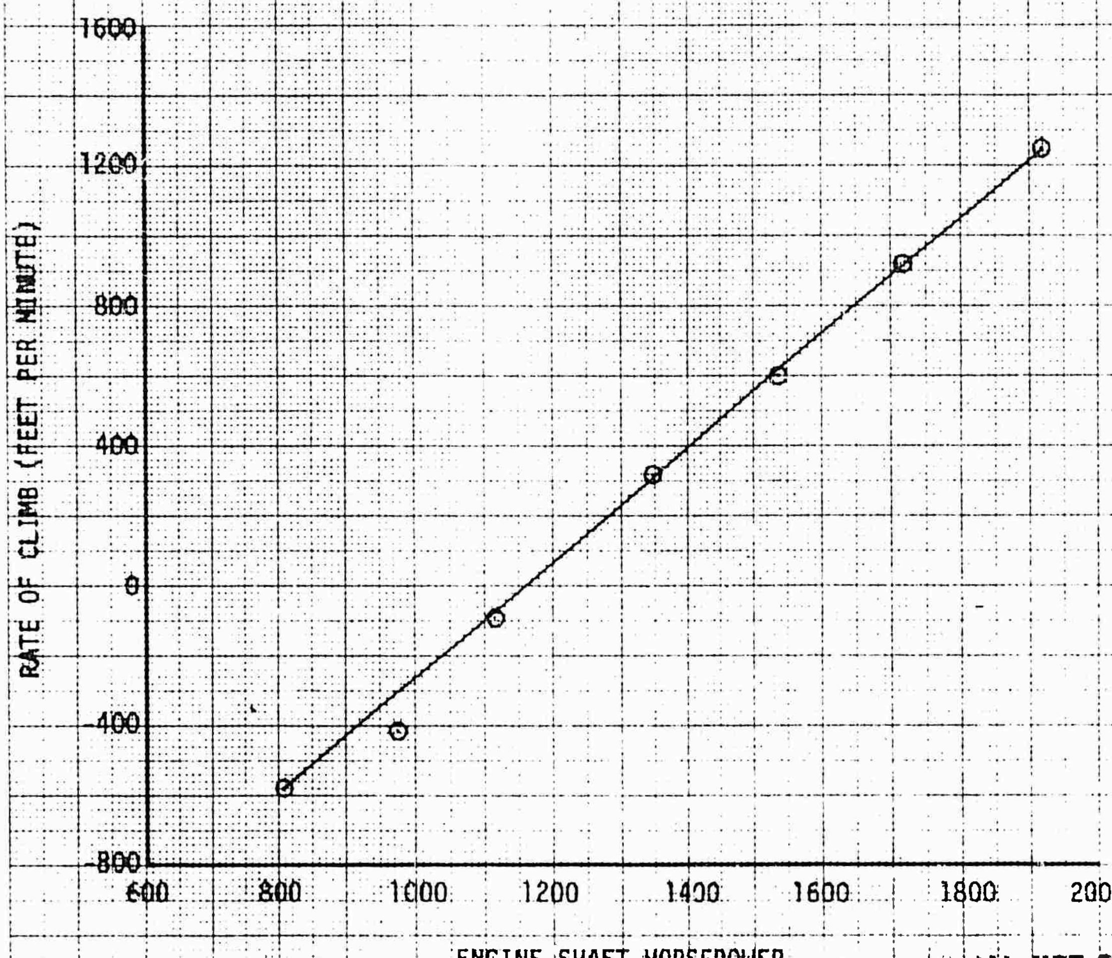
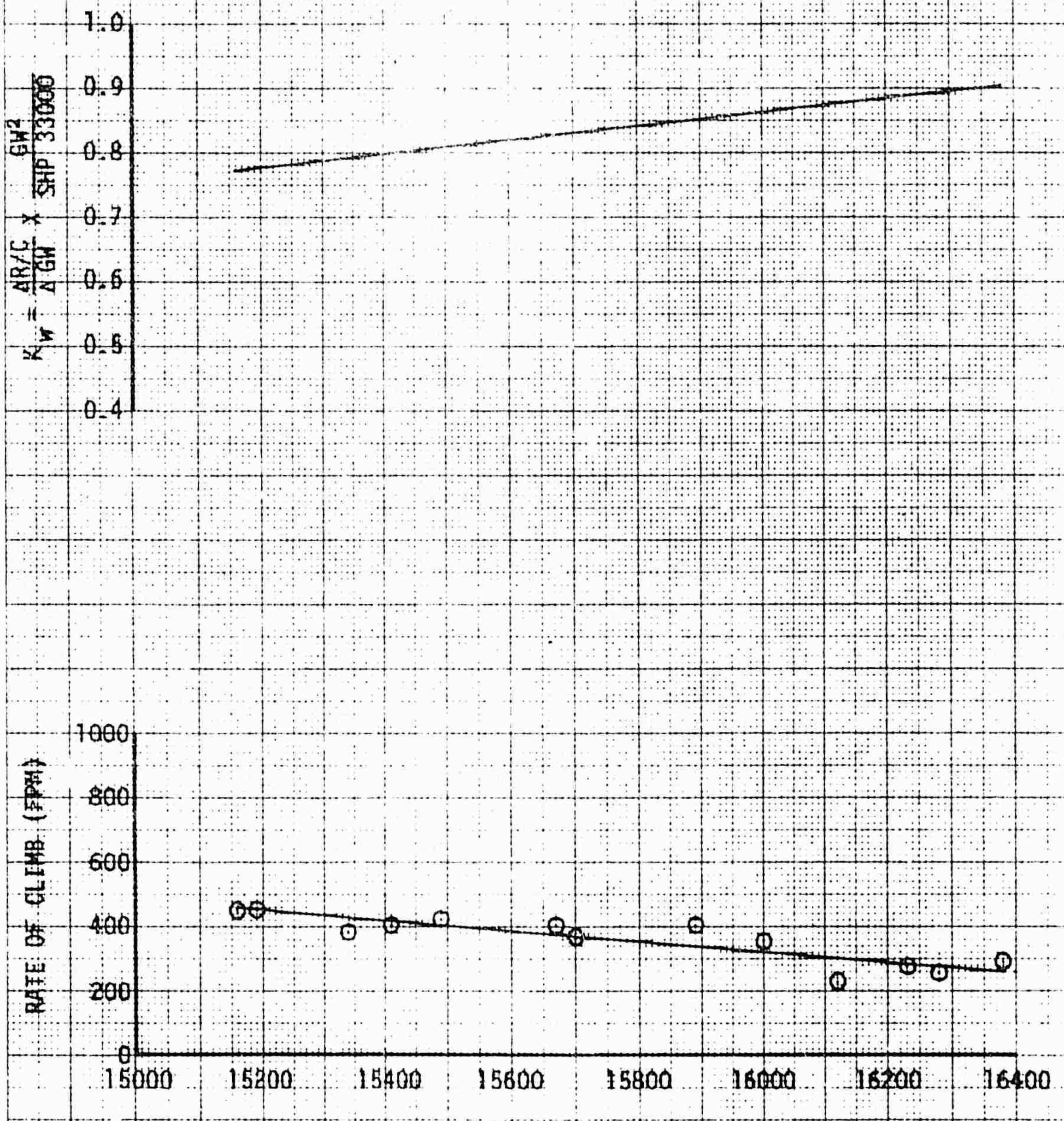


FIGURE 10
VARIATION IN RATE OF CLIMB
AS A FUNCTION OF GROSS WEIGHT
YAH-63 USA S/N 74-22246

AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KTS)	SHAFT HORSEPOWER
293.7(FWD)	-0.1(LT)	4460	17.0	276	74	1327

NOTE: B-TOW CONFIGURATION



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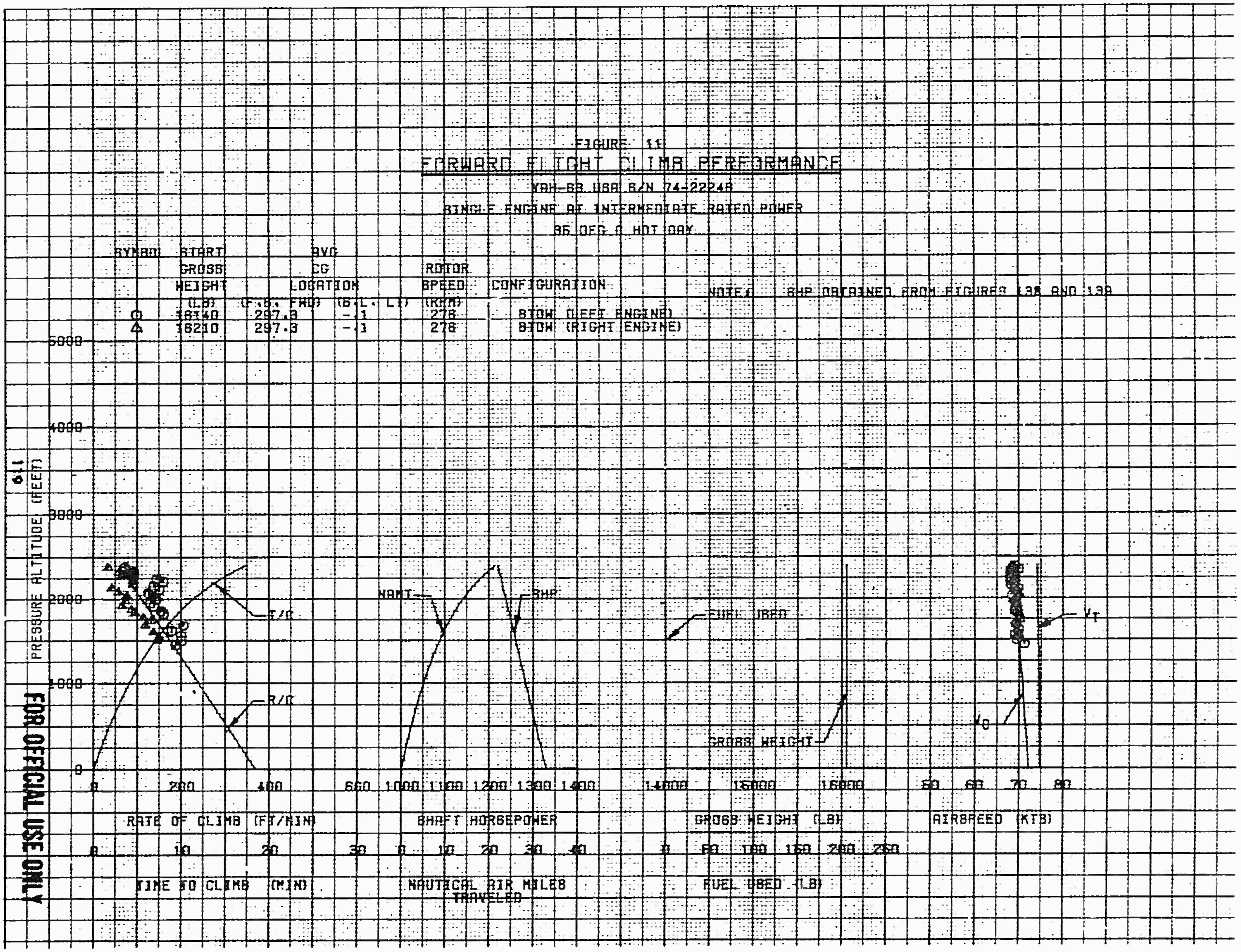
FIGURE
FORWARD FLIGHT CLIMB PERFORMANCE
YAH-63 USA S/N 74-22248
SINGLE ENGINE AT INTERMEDIATE RATED POWER
35 DEG C HOT DAY
SYMBOL START GROSS WEIGHT (LB) | AVG CG LOCATION (F.S. FWD) (B.L. LT) | ROTOR SPEED (RPM) | CONFIGURATION
○ 16140 297.3 -.1 276 STOW (LEFT ENGINE)
△ 16210 297.3 -.1 276 STOW (RIGHT ENGINE)
NOTE: SHP OBTAINED FROM FIGURES 138 AND 139
PRESSURE ALTITUDE (FEET)
T/C
R/C
NAMT
SHP
FUEL USED
GROSS WEIGHT
V_T
V_C
RATE OF CLIMB (FT/MIN)
TIME TO CLIMB (MIN)
SHAFT HORSEPOWER
NAUTICAL AIR MILES TRAVELED
GROSS WEIGHT (LB)
FUEL USED (LB)
AIRSPEED (KTS)

FIGURE 12
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
ENGINE S/N's 207260, 207279

NOTES:
1. LONGITUDINAL CG = FWD.
2. LATERAL CG = MID.
3. ROTOR SPEED = 276 RPM.
4. POINTS OBTAINED FROM FIGS 16 THROUGH 18.
5. CONFIGURATION = CLEAN.

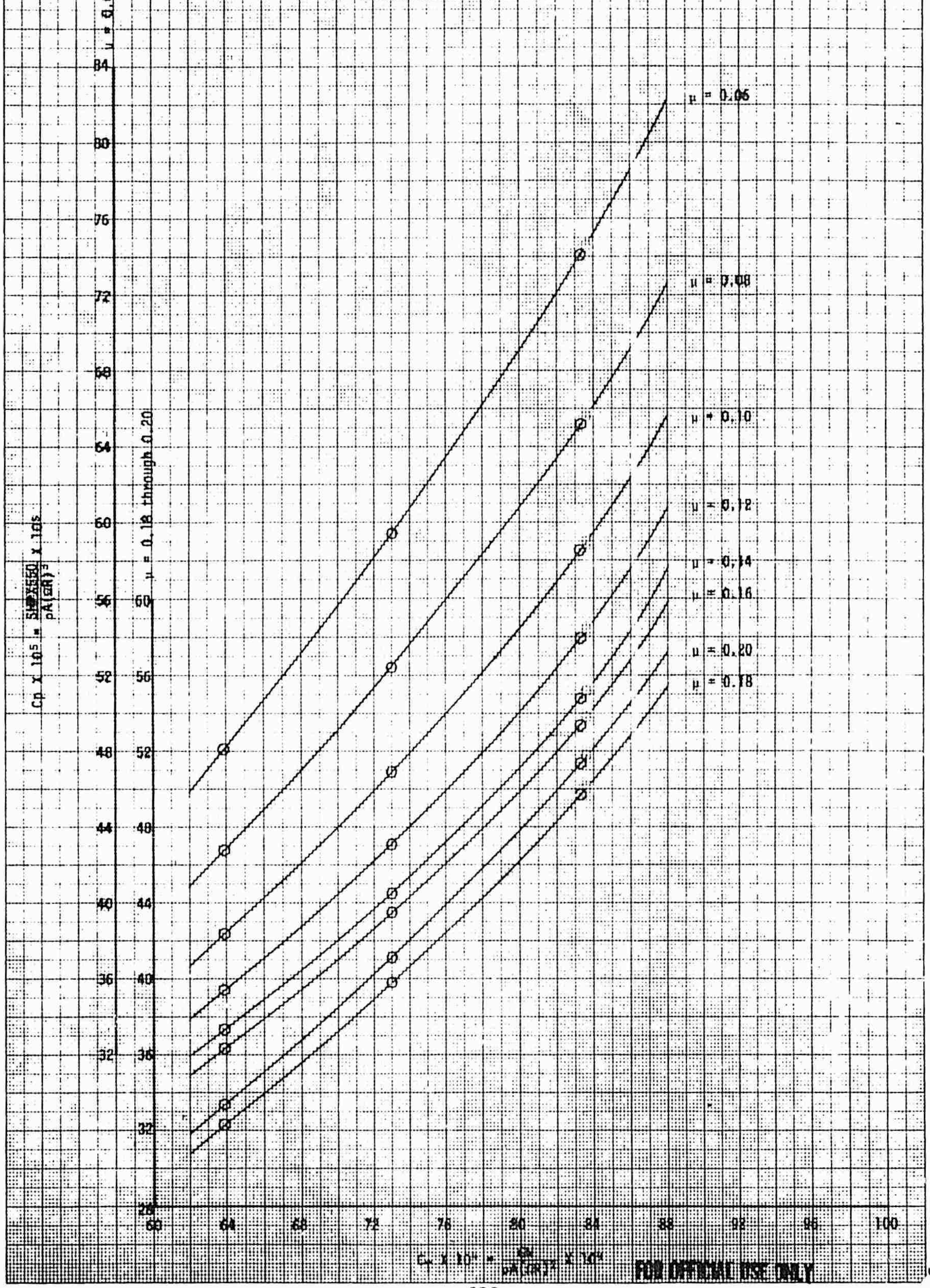

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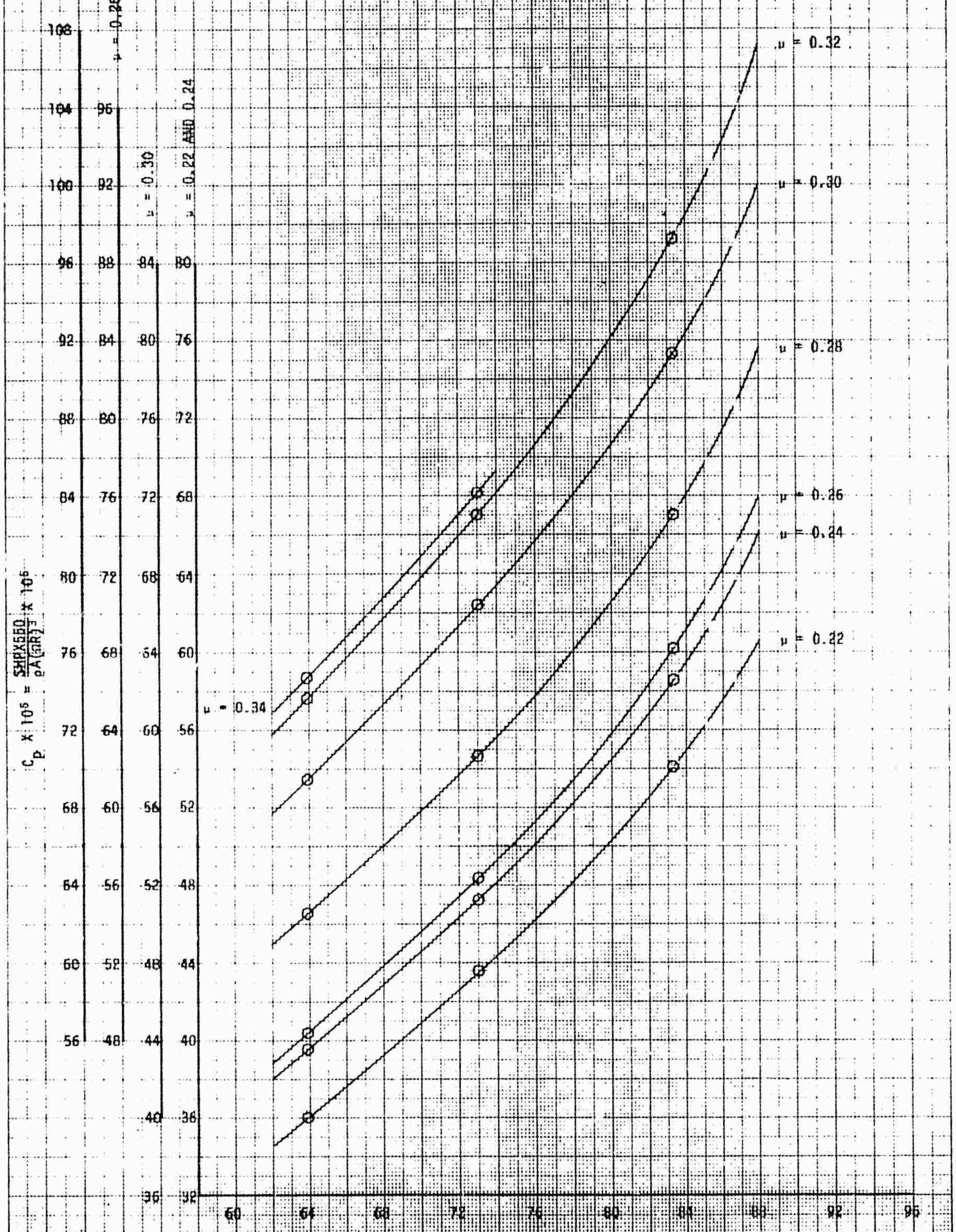
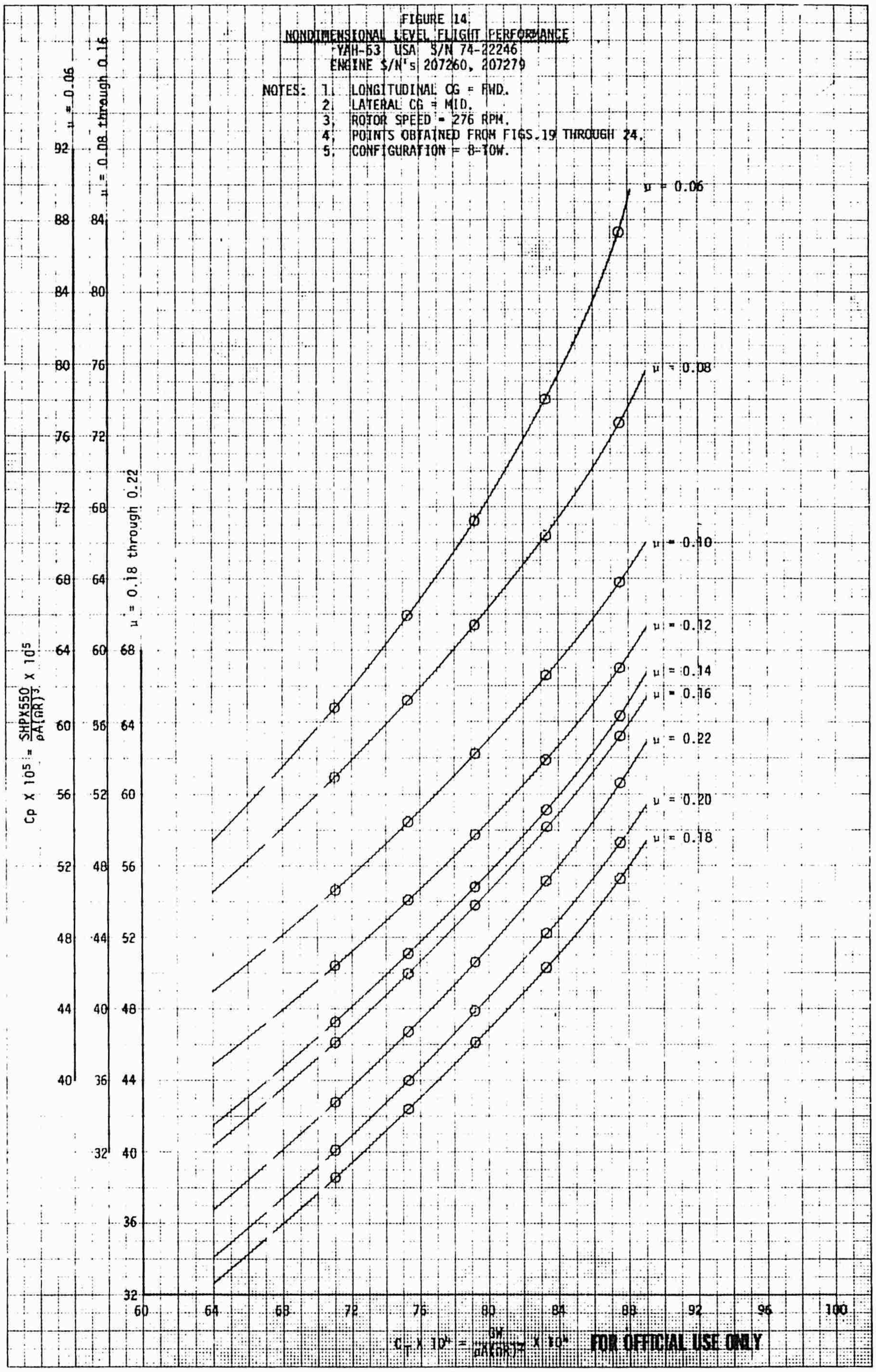
FIGURE 15
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
ENGINE S/N's 207260, 207279

NOTES: 1. LONGITUDINAL CG = FWD.
2. LATERAL CG = MID.
3. ROTOR SPEED = 276 RPM.
4. POINTS OBTAINED FROM FIGS. 16 THROUGH 18.
5. CONFIGURATION = CLEAN.





FOR OFFICIAL USE ONLY

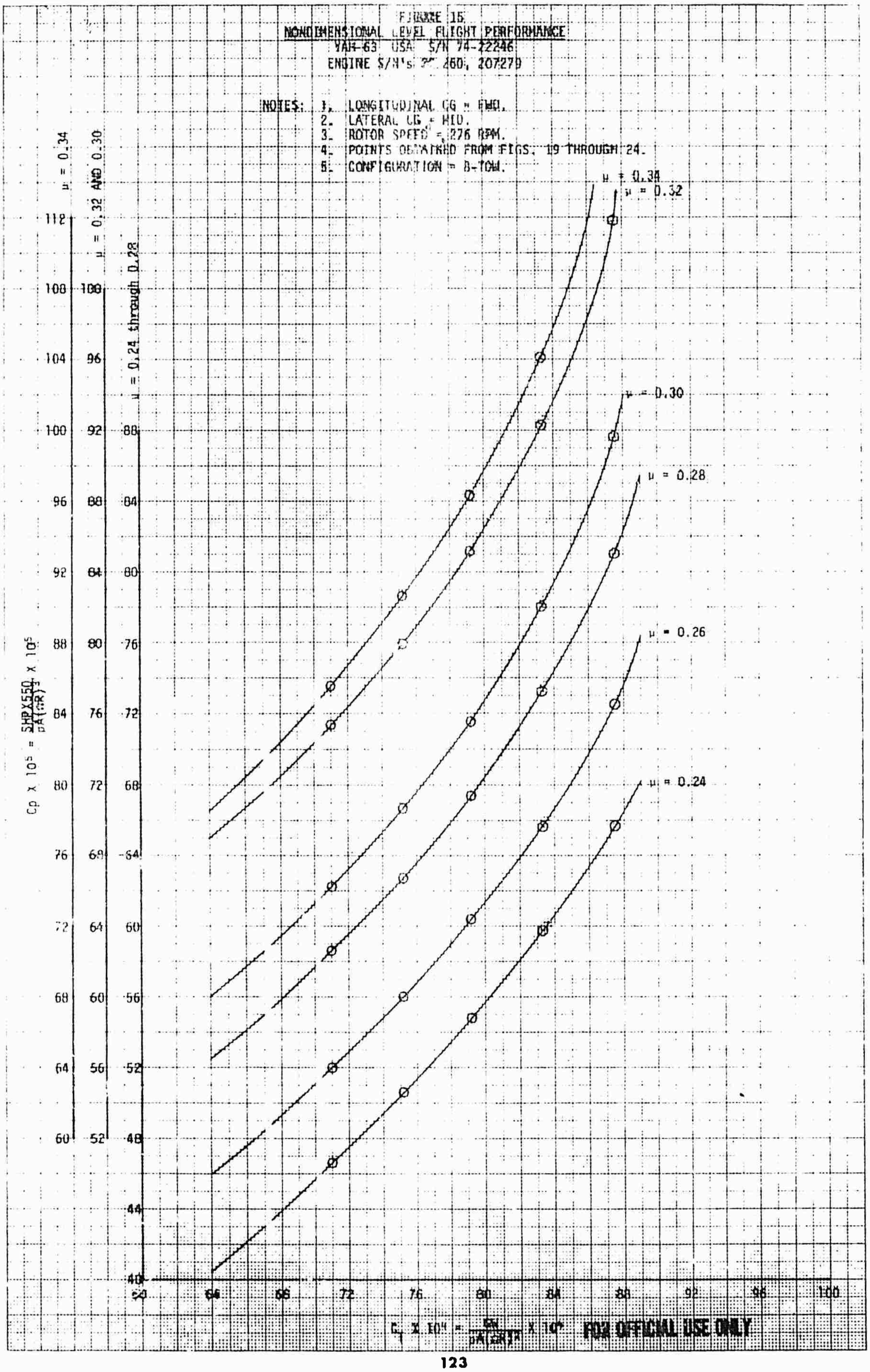


FIGURE 15
NONDIMENSIONAL LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
NOTES: 1. LONGITUDINAL CG = FWD.
2. LATERAL CG = MID.
3. ROTOR SPEED = 276 RPM.
4. POINTS OBTAINED FROM FIGS. 19 THROUGH 24.
5. CONFIGURATION = B-TOW.
μ = 0.34
μ = 0.32
μ = 0.30
μ = 0.28
μ = 0.26
μ = 0.24
μ = 0.34
μ = 0.32 AND 0.30
μ = 0.24 through 0.28
$C_P \times 10^5 = \frac{SHP \times 550}{\rho A (\Omega R)^3} \times 10^5$
$C_T \times 10^4$
FOR OFFICIAL USE ONLY

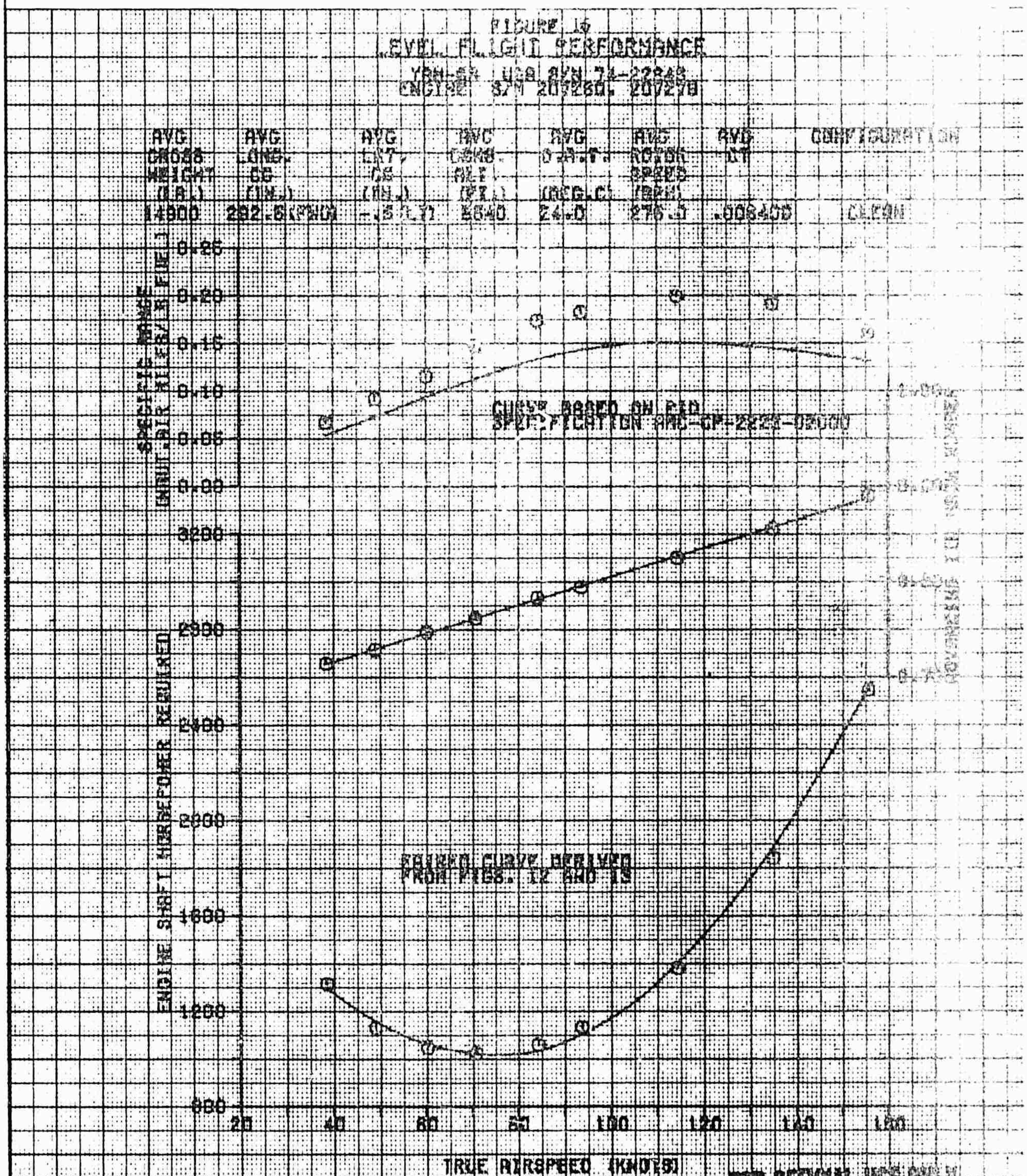


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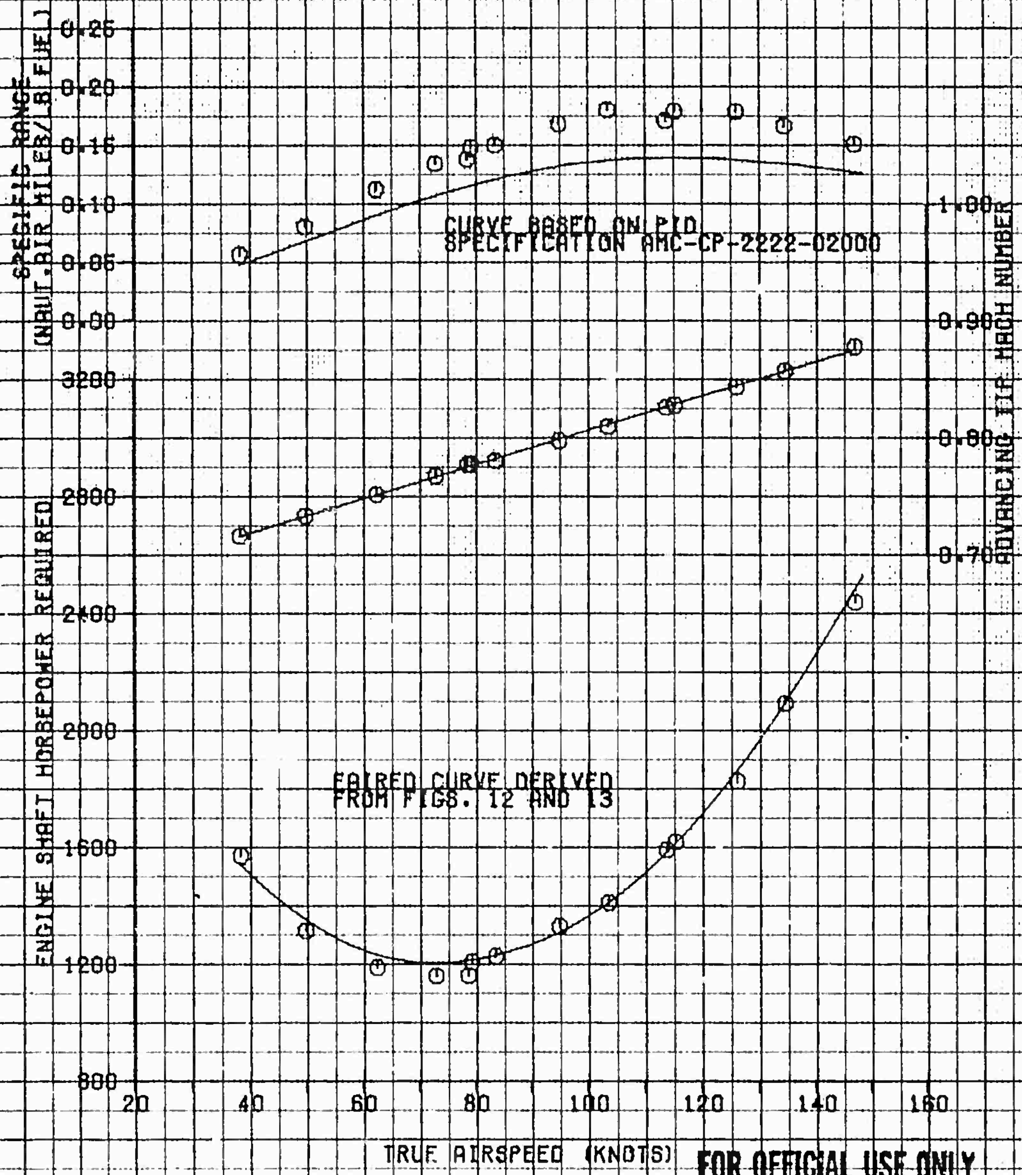
FIGURE 17
LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
ENGINE S/N 207280, 207279

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16480	298.0 (FWD)	-.5 (LT)	6480	23.0	276.0	.007295	CLEAN



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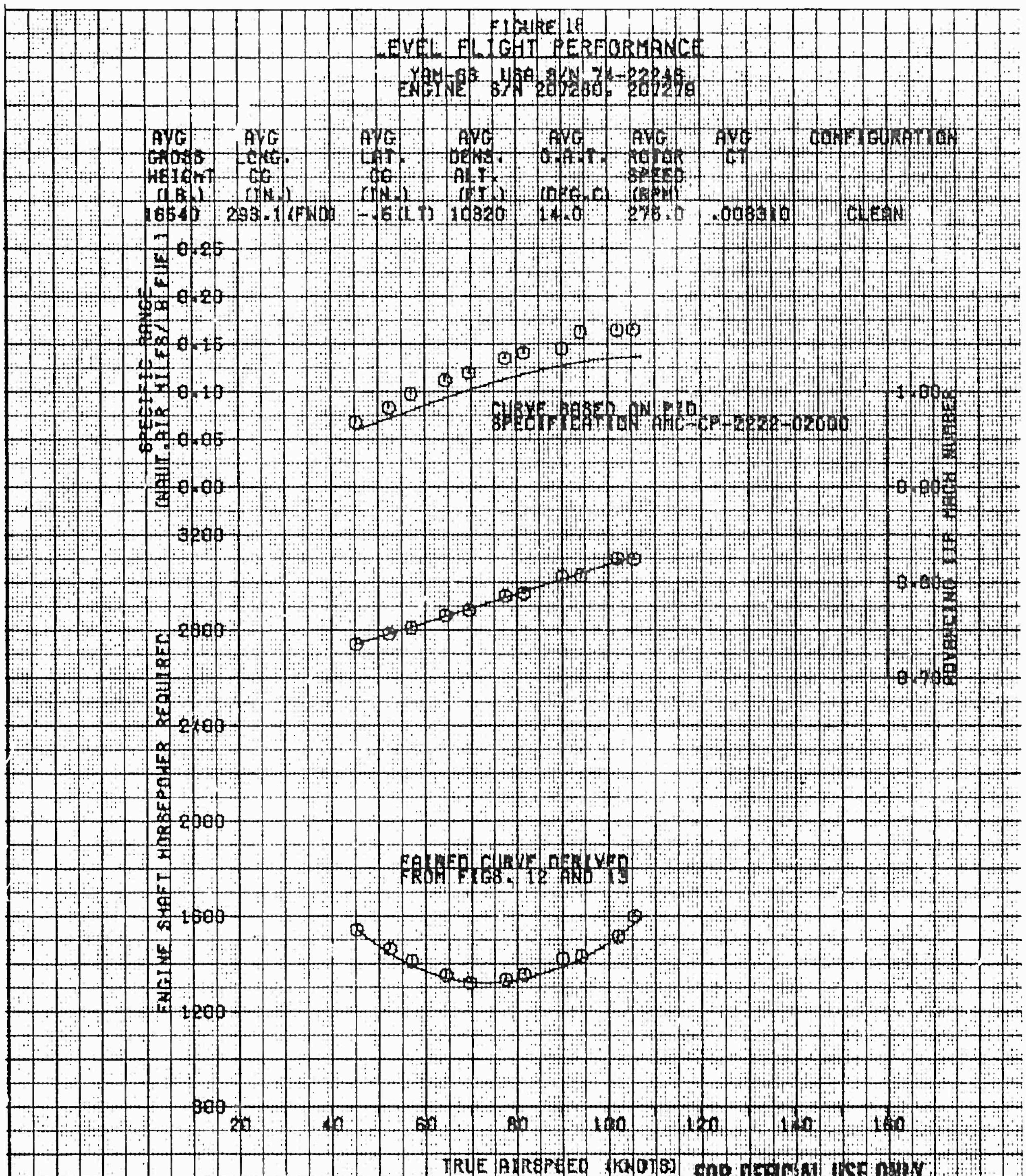
FIGURE 18
LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
ENGINE S/N 207280, 207278
AVG GROSS WEIGHT (LB)
AVG LONG. CG (IN.)
AVG LAT. CG (IN.)
AVG DENS. ALT. (FT)
AVG O.A.T. (DEG.C)
AVG ROTOR SPEED (RPM)
AVG CT
CONFIGURATION
16540
293.1 (FWD)
-.6 (LT)
10820
14.0
276.0
.008310
CLEAN
SPECIFIC RANGE (NAUT. AIR MILES/LB FUEL)
0.25
0.20
0.15
0.10
0.05
0.00
CURVE BASED ON PID SPECIFICATION AMC-CP-2222-02000
ADVANCING TIP MACH NUMBER
1.00
0.90
0.80
0.70
ENGINE SHAFT HORSEPOWER REQUIRED
3200
2800
2400
2000
1600
1200
800
FAIRED CURVE DERIVED FROM FIGS. 12 AND 13
20
40
60
80
100
120
140
160
TRUE AIRSPEED (KNOTS)

FIGURE 19
LEVEL FLIGHT PERFORMANCE
YUH-6? USA S/N 7?-22246
ENGINE S/N 207280, 207279

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16720	292.9 (FWD)	-.1 (LT)	87??	22.0	275.0	.007100	8-TOW

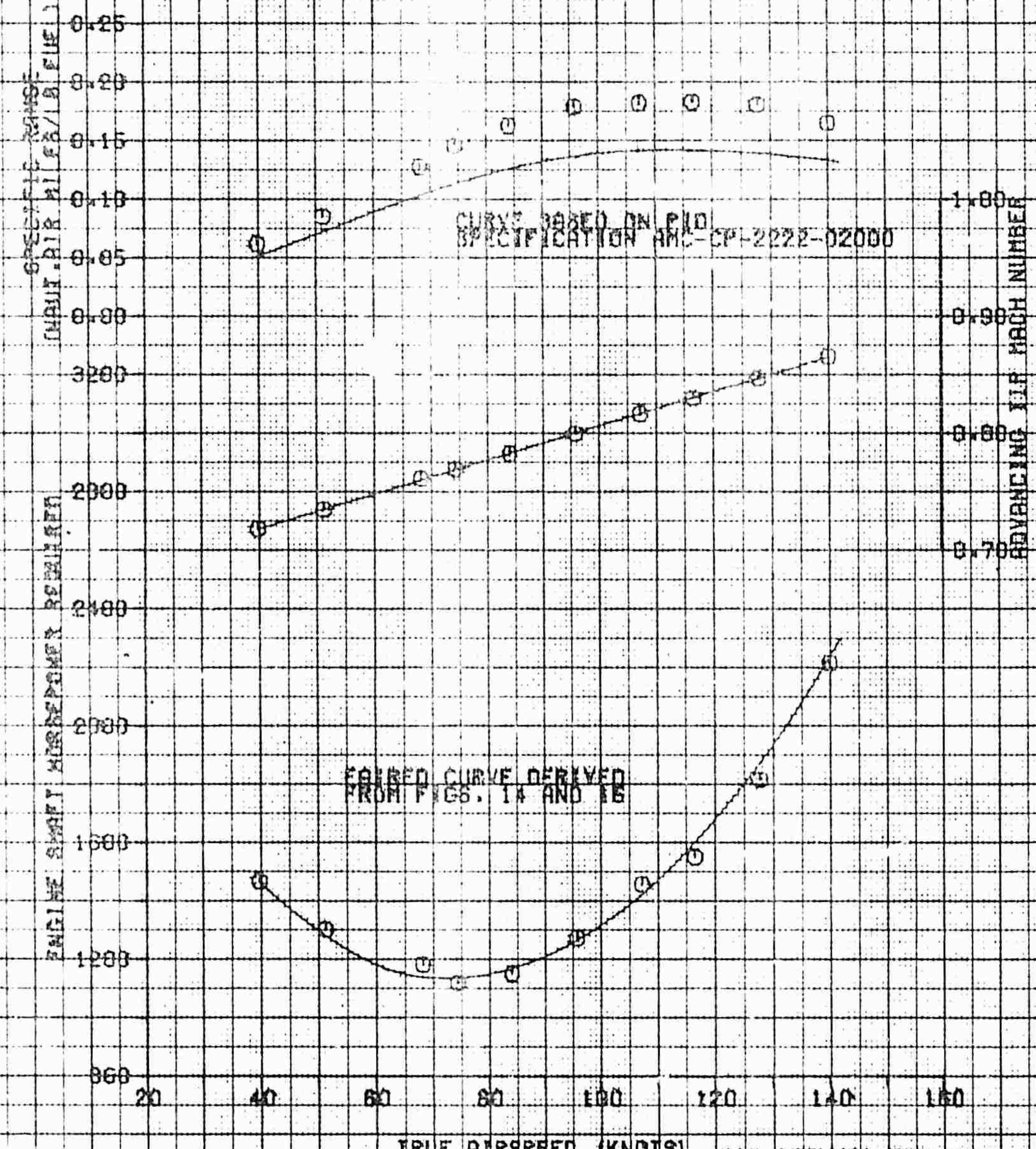

FIGURE 20
LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
ENGINE S/N 200260, 200273

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
15760	292.9(FWD)	-.5(LT)	7360	12.6	276.0	.007225	8-TOW

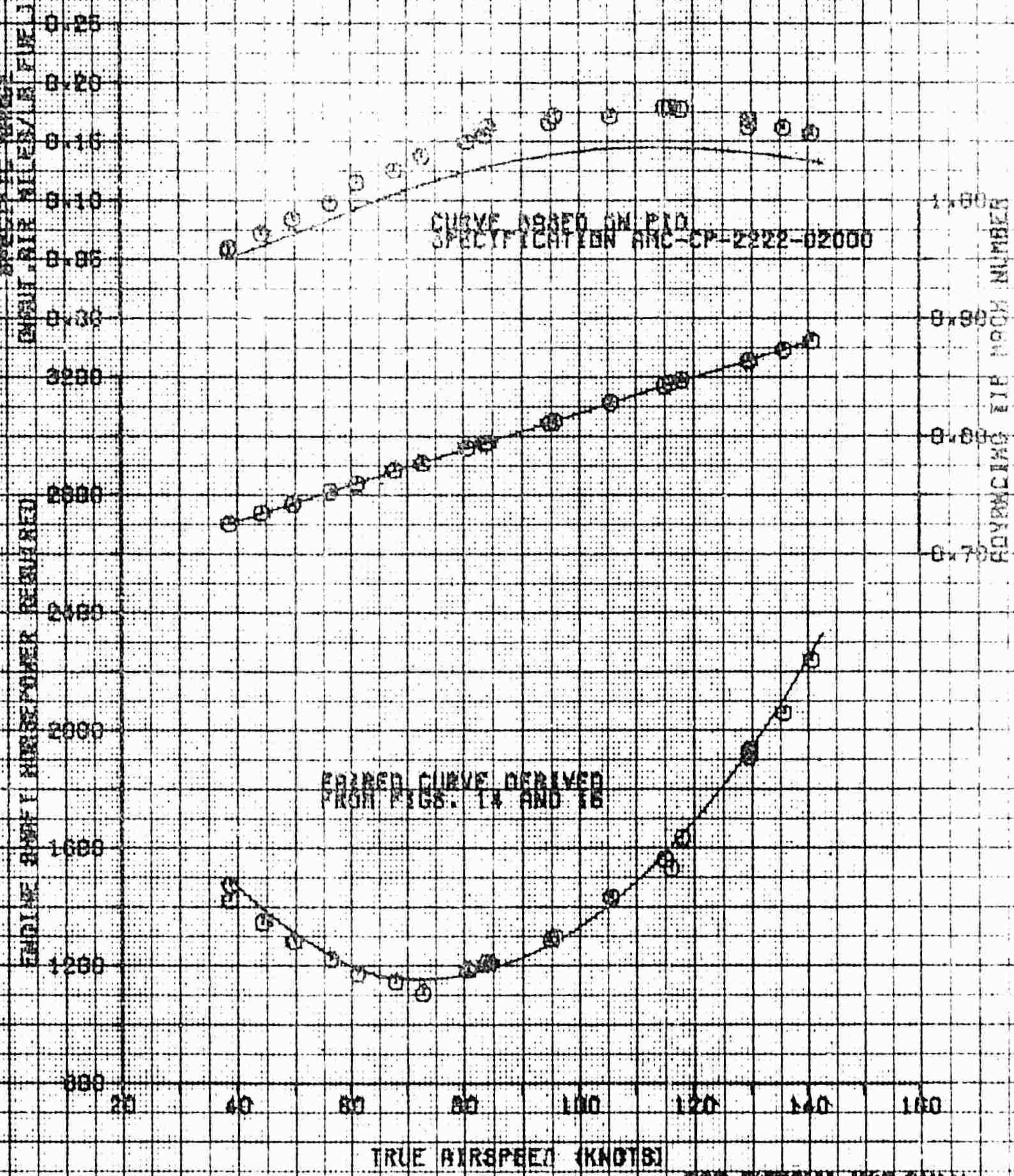

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FIGURE 21
LEVEL FLIGHT PERFORMANCE
YAH-64 USA S/N 74-22246
ENGINE S/N 207260, 207279

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16140	291.3 (FWD)	-.1 (LT)	7900	17.6	275.0	.007519	S-TSH

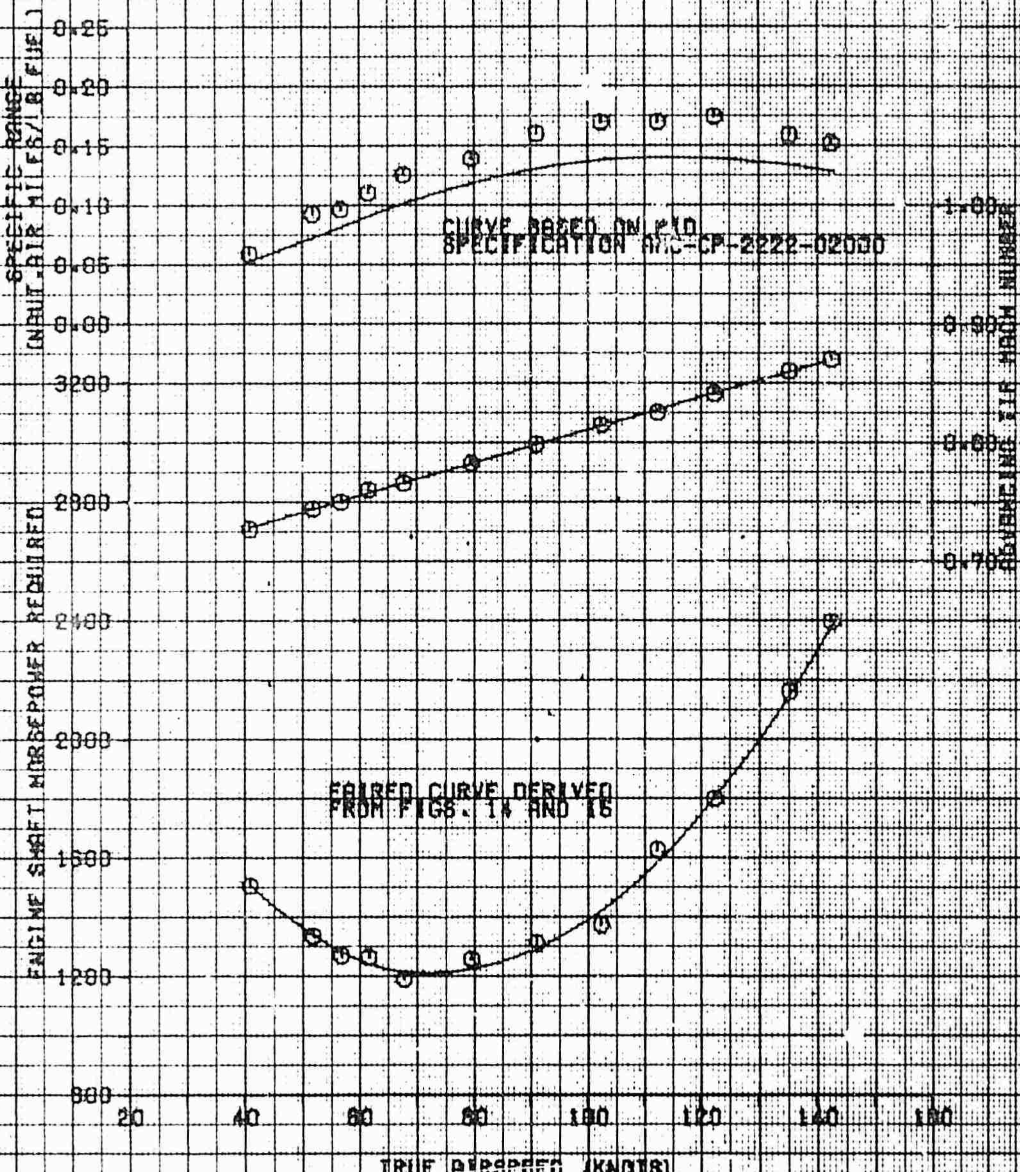
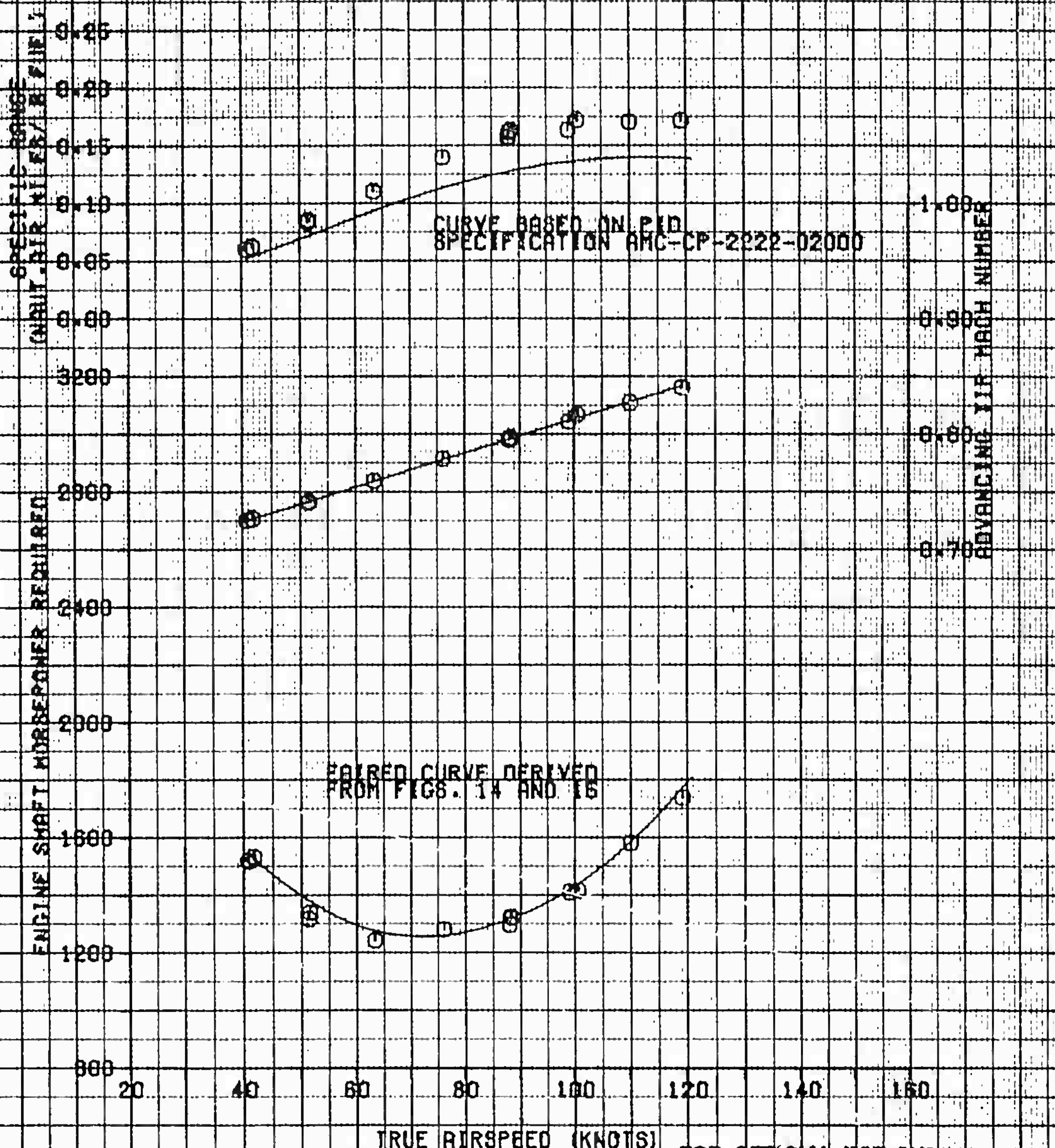


FIGURE 22
LEVEL FLIGHT PERFORMANCE
YUH-60 USA S/N 73-22248
ENGINE S/N 207260, 207279

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
16160	358.6(FWD)	-.1(LT)	9520	16.0	275.0	.007917	8-TOW



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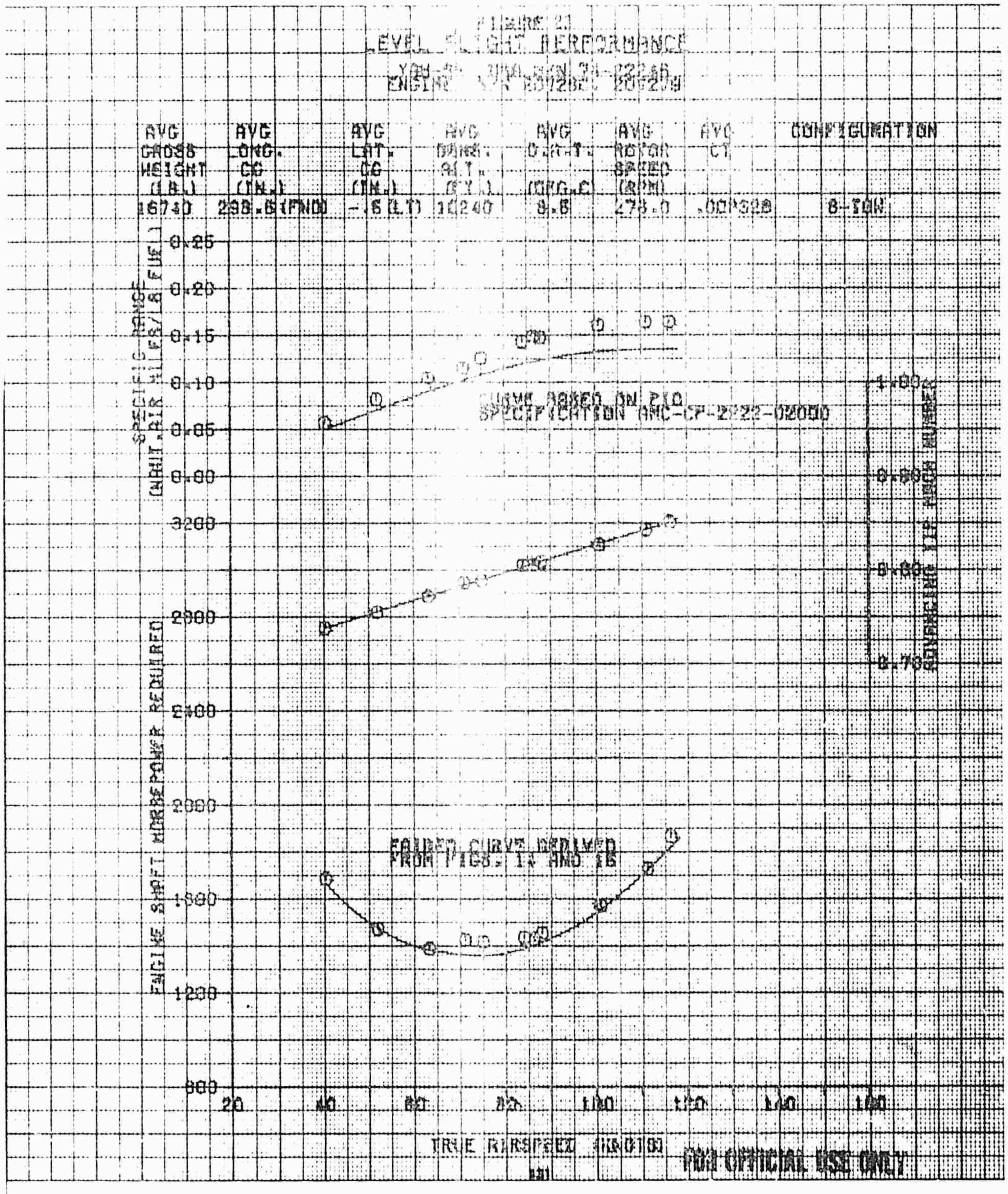

FOR OFFICIAL USE ONLY

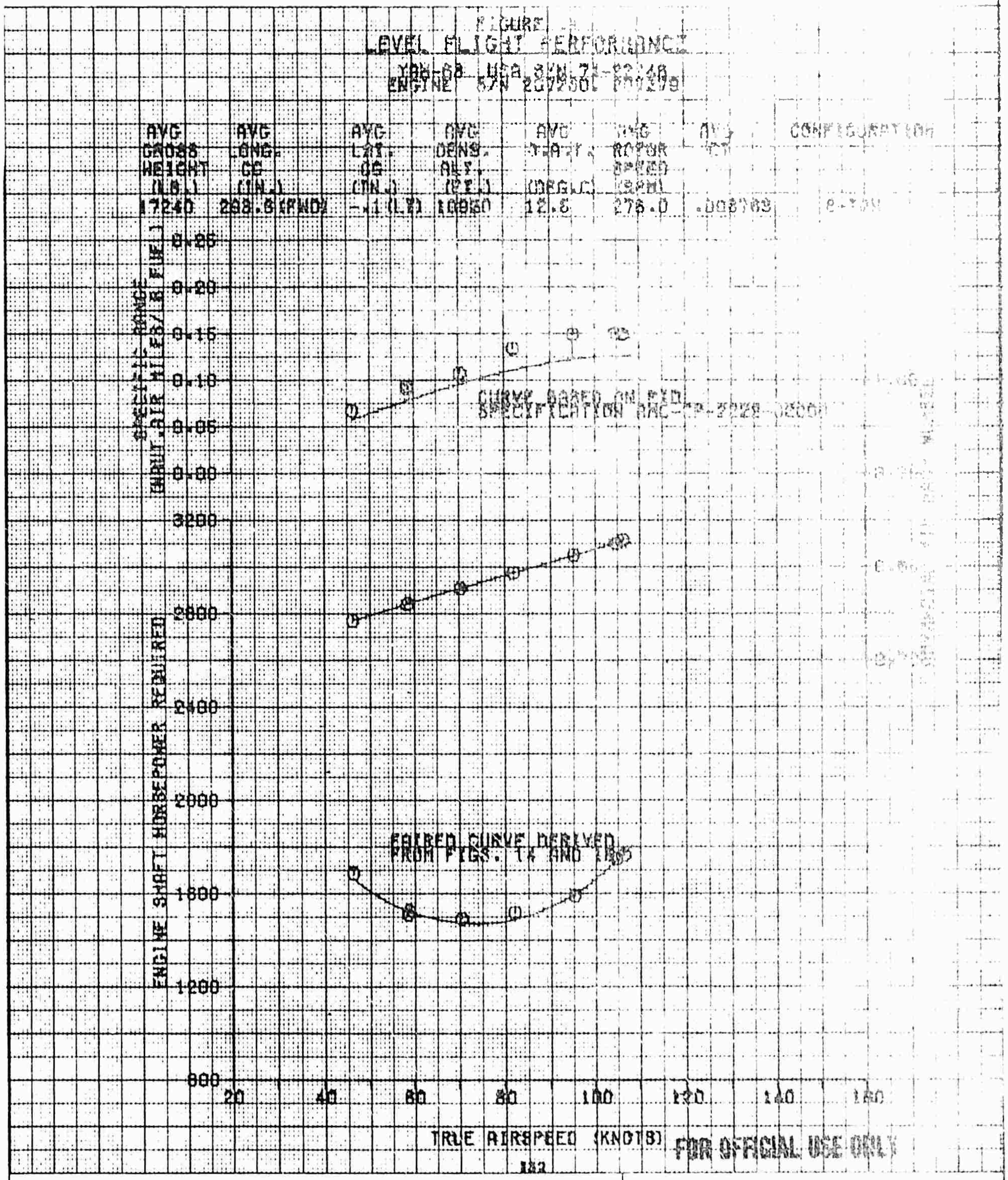

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FIGURE 25
LEVEL FLIGHT PERFORMANCE
YAH-63 USA S/N 74-22246
ENGINE S/N 207260, 207211

AVG GROSS WEIGHT (LB.)	AVG LONG. CG (IN.)	AVG LAT. CG (IN.)	AVG DENS. ALT. (FT.)	AVG O.A.T. (DEG.C)	AVG ROTOR SPEED (RPM)	AVG CT	CONFIGURATION
15520	298.5 (AFT)	-.1 (LT)	7560	10.0	276.0	.007102	8-TOW

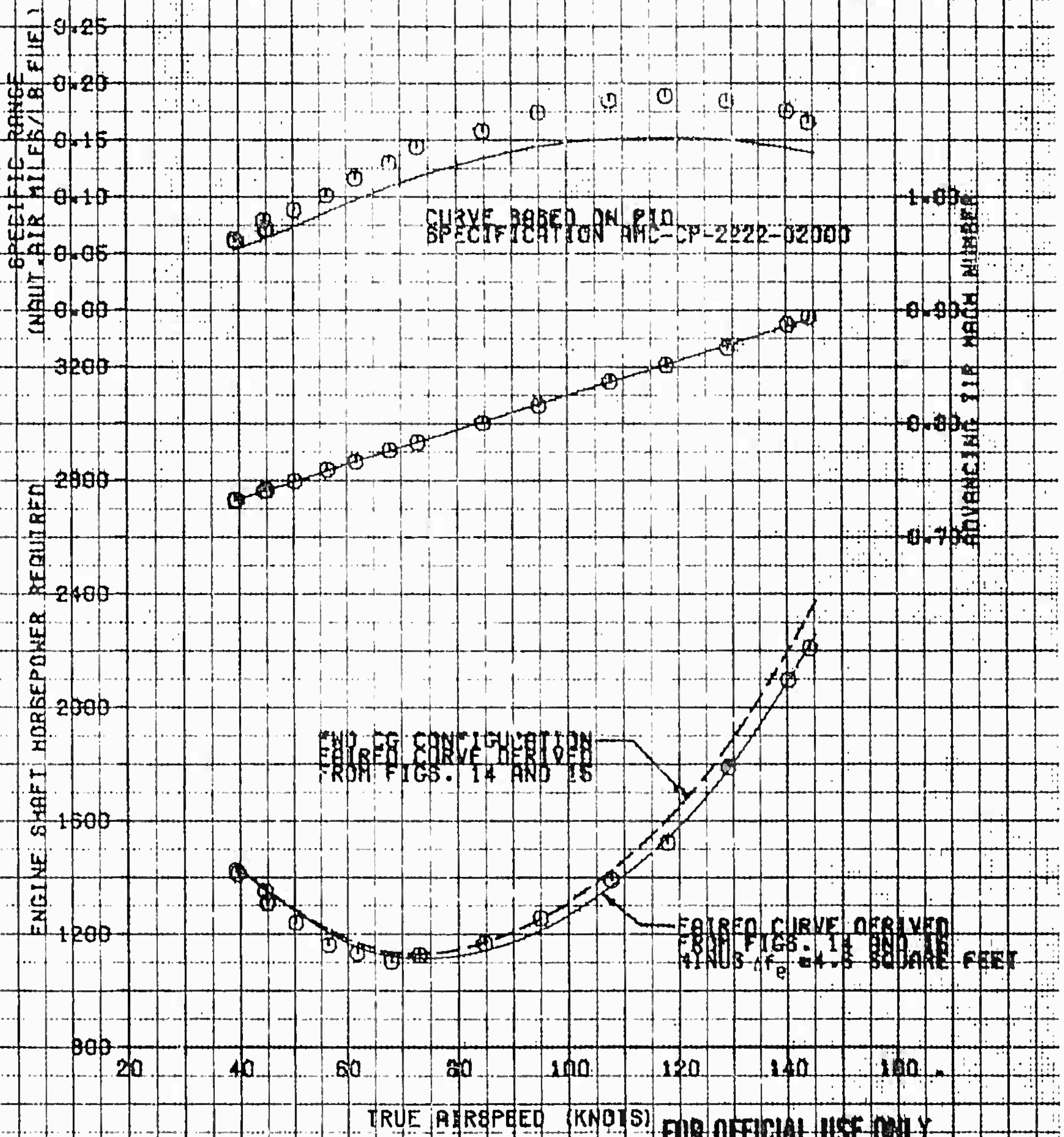

FIGURE 26
LONG-RANGE SUMMARY
YAH-63 USA S/N 74-22246

ROTOR SPEED = 275 RPM
FWD CENTER OF GRAVITY
8-TOW CONFIGURATION

NOTE: 1. FUEL FLOW IS 5 PERCENT CONSERVATIVE.
2. STANDARD DAY CONDITIONS UNLESS OTHERWISE NOTED.

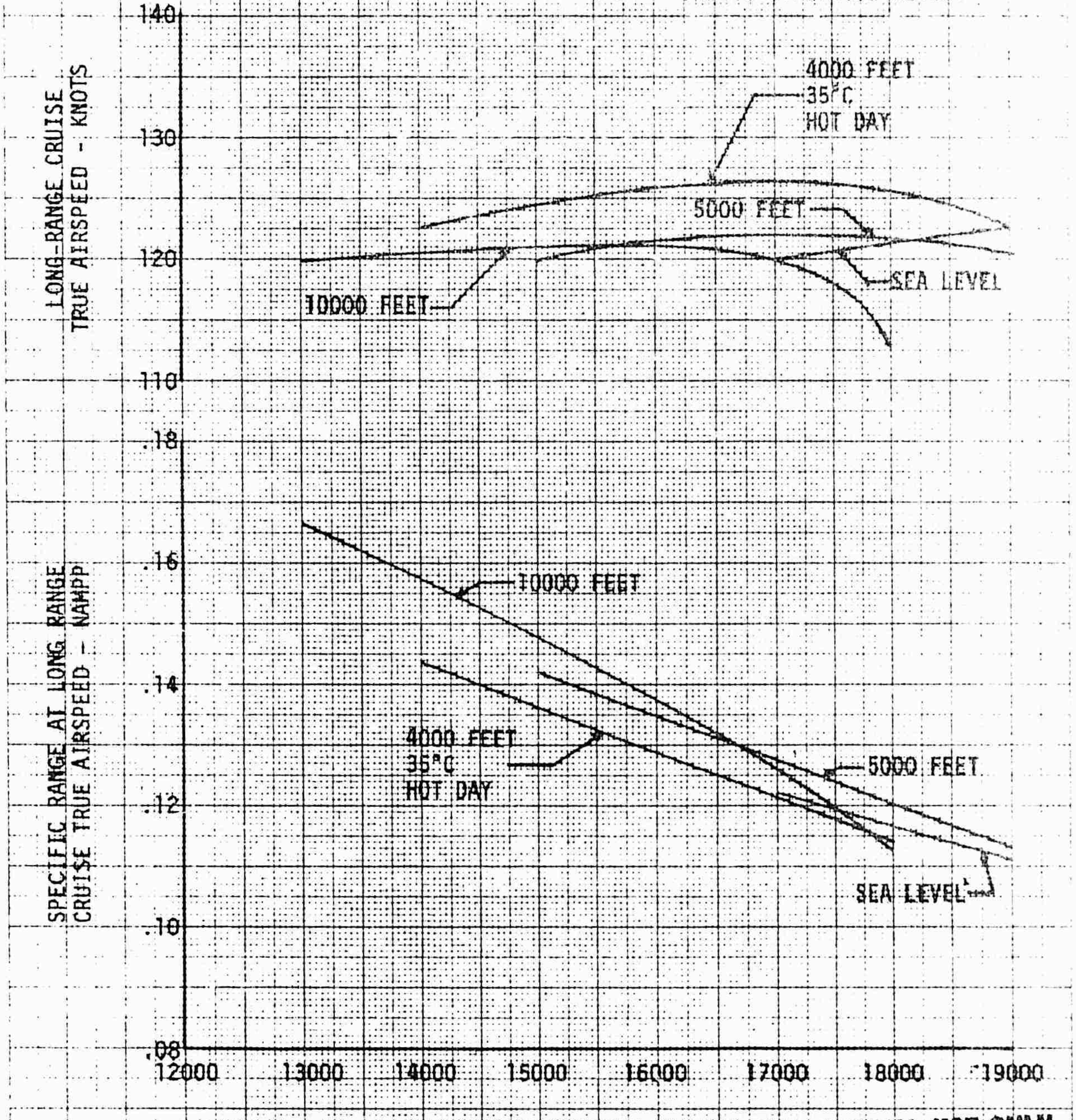

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FIGURE 27
MAXIMUM ENDURANCE
YAH-63 USA S/N 74-22246

ROTOR SPEED = 276 RPM
FWD CENTER OF GRAVITY
8-TOW CONFIGURATION

NOTES: 1. FUEL FLOW IS 5 PERCENT CONSERVATIVE.
2. STANDARD DAY CONDITIONS UNLESS OTHERWISE NOTED.

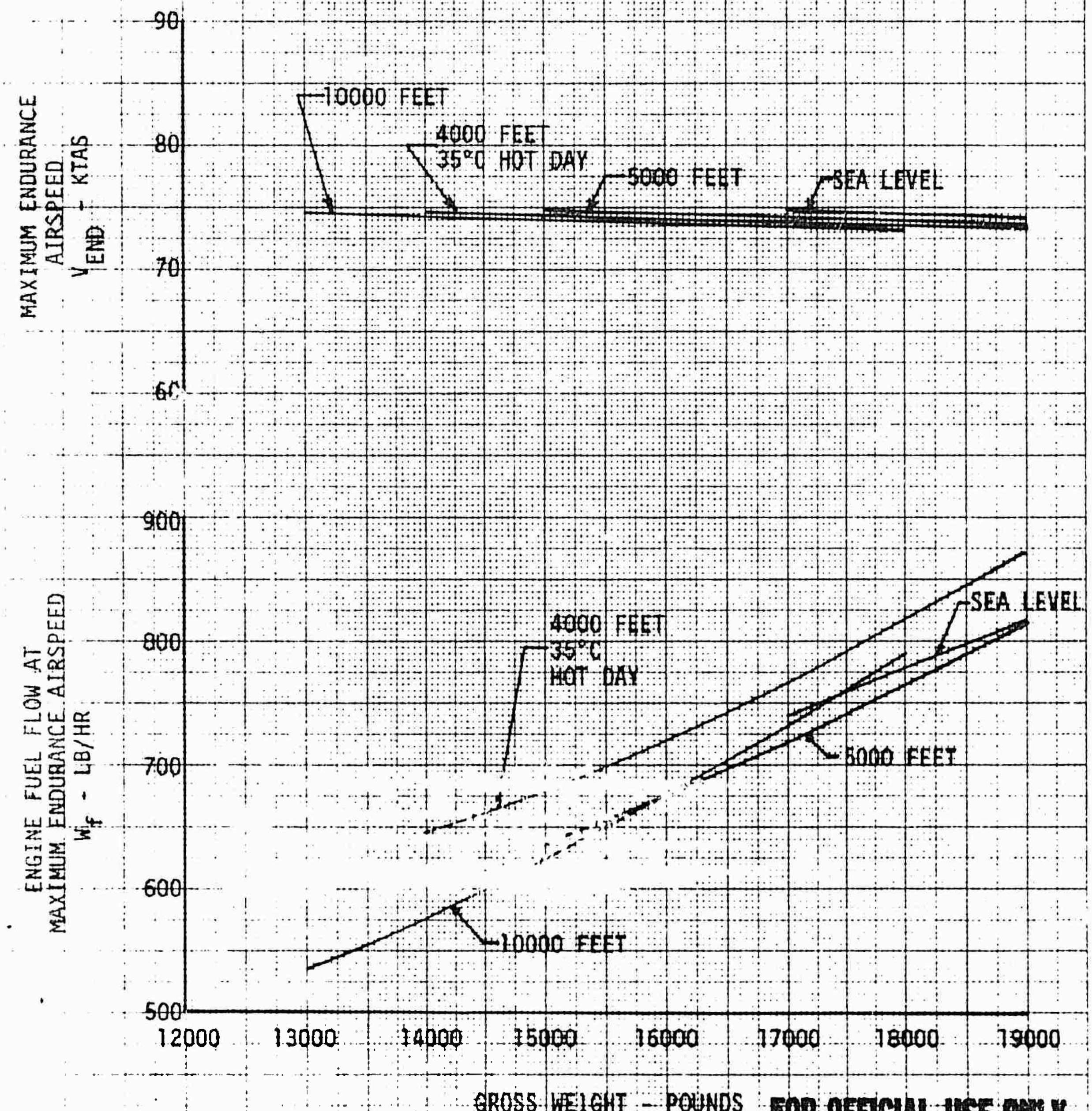

FIGURE 28
LATERAL ACCELERATIONS
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	CONFIGURATION
16020	295.8(MID)	-.1(LT)	5320	15.1	276	B-TOW

NOTES: 1. CIRCLES DENOTE LEFT ACCELERATION.
2. SQUARES DENOTE RIGHT ACCELERATION.

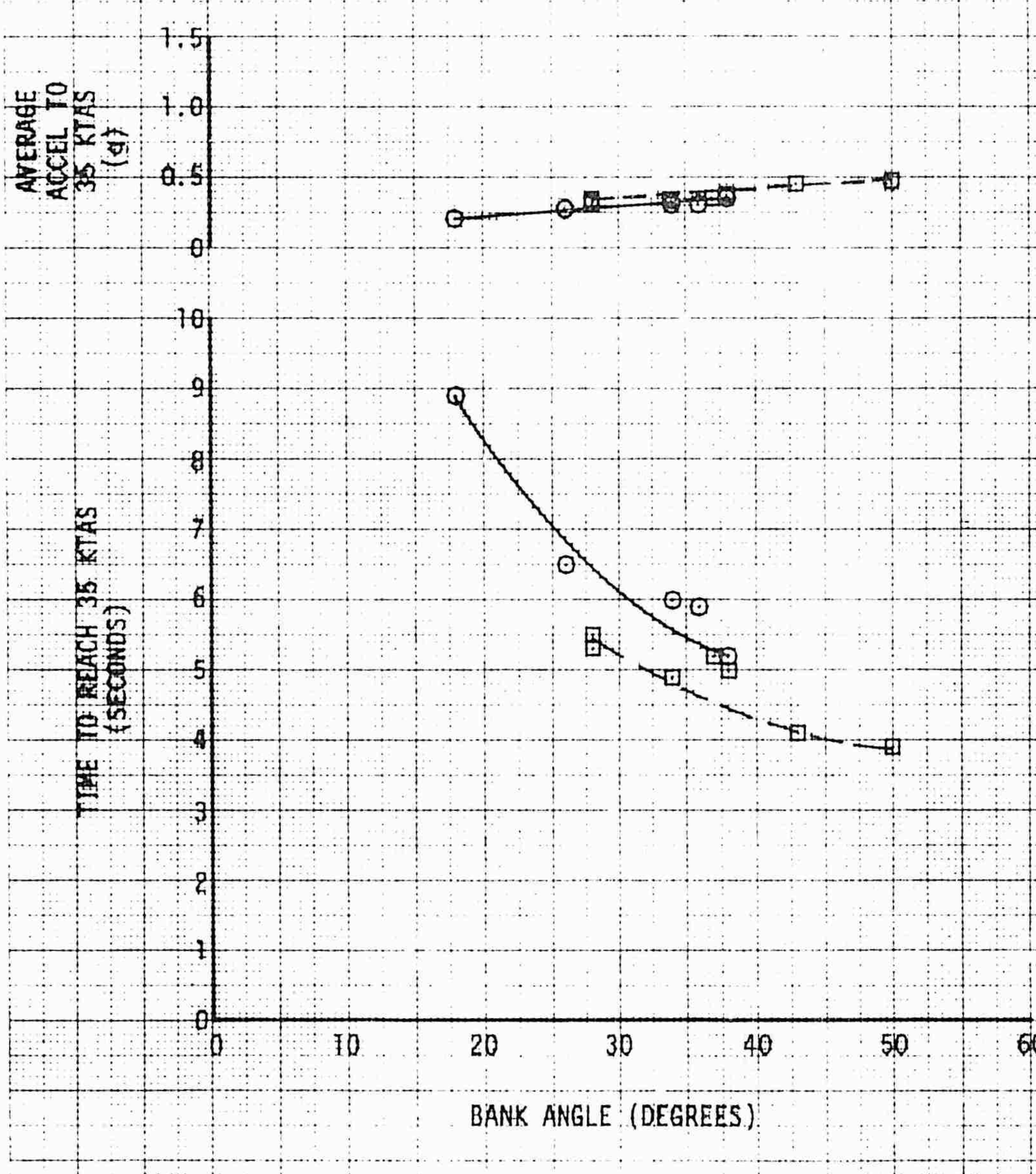

FIGURE 29
LATERAL ACCELERATION
YAH-63 USA S/N 74-22246

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	SCAS CONDITION
16040	295.6 (MID)	-.1 (LT)	5260	14.5	276	ON

NOTE: 8-TOW CONFIGURATION

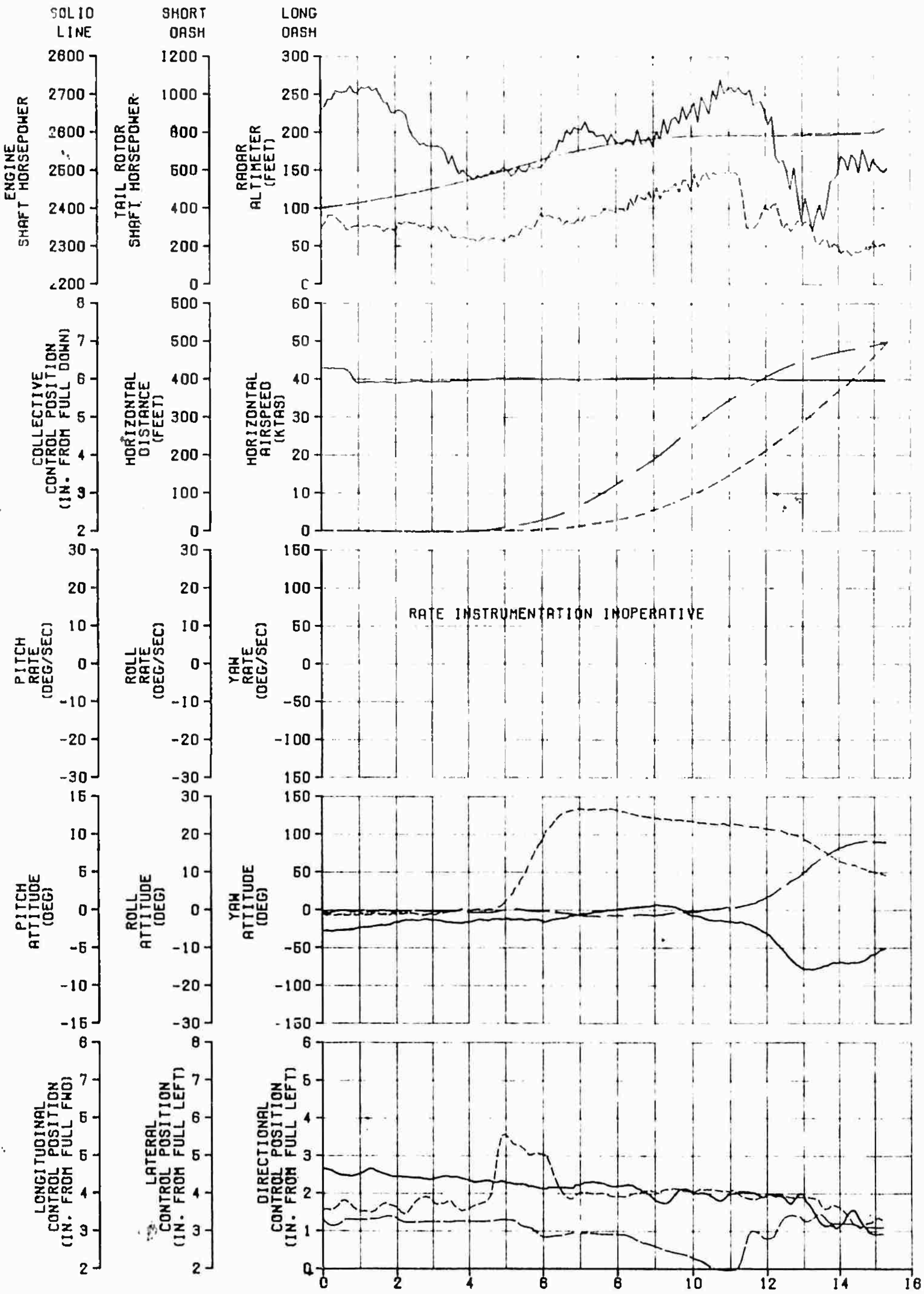

FIGURE 30
VERTICAL DISPLACEMENT
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	ENTRY TRUE AIRSPEED (KTS)	AVG C_T	CONFIGURATION
16240	294.8(MID)	0.5(LT)	5080	15.0	276	140	0.006889	8 TOW

NOTES: 1. HORIZONTAL DISTANCE IS DISTANCE REQUIRED TO GAIN 200 FEET IN ALTITUDE FROM ENTRY HEIGHT.
2. AIRSPEED LOSS IS THE CHANGE FROM ENTRY AIRSPEED TO THE AIRSPEED AT THE POINT WHERE THE AIRCRAFT HAS GAINED 200 FEET IN ALTITUDE.

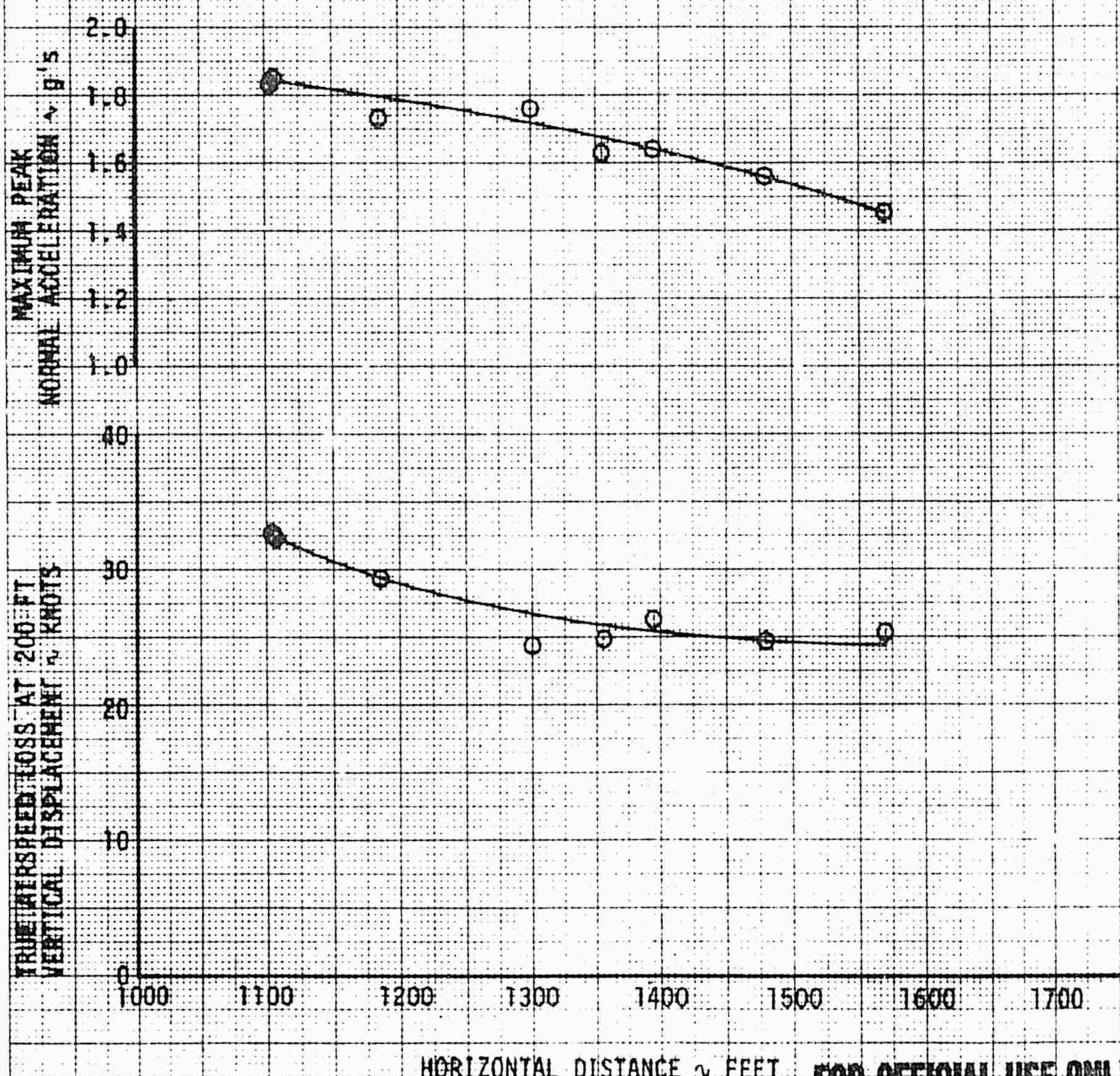

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FIGURE 31

VERTICAL DISPLACEMENT

YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION (FS)	AVG CG LOCATION (BL)	ENTRY DENSITY ALTITUDE (FT)	ENTRY OAT (°C)	ROTOR SPEED (RPM)	ENTRY AIRSPEED (KTAS)	CONFIGURATION
15140	294.7(FWD)	-.1(LT)	4920	15.5	277	141	8-TOW

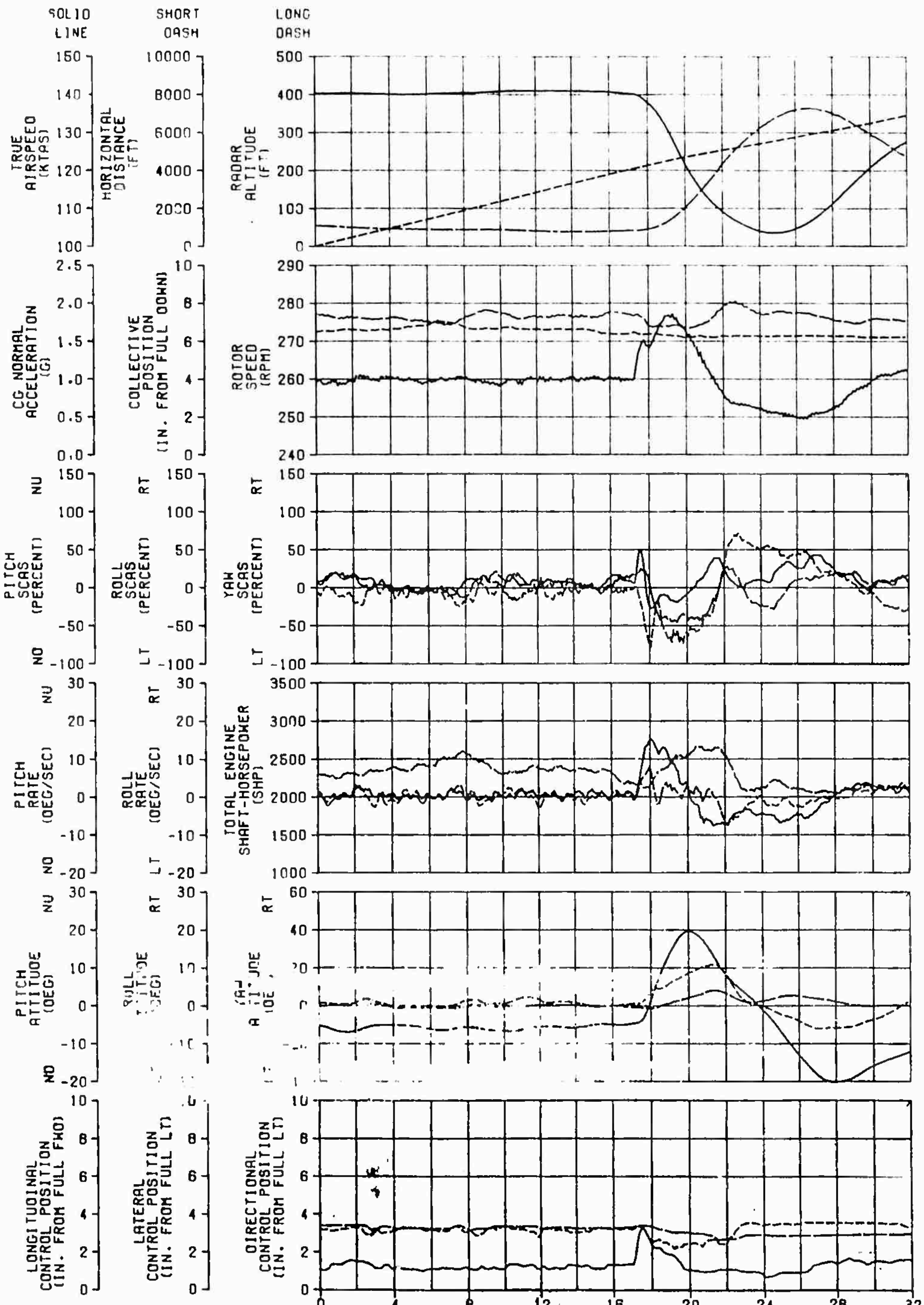

FIGURE 32
AUTOROTATIONAL PERFORMANCE
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	AVG C_T	CONFIGURATION
16020	293.5(FWD)	-.1(LT)	6300	16.0	275	.00711	B-TOW

NOTE: POSITION ERROR IS UNAVAILABLE IN AUTOROTATION.

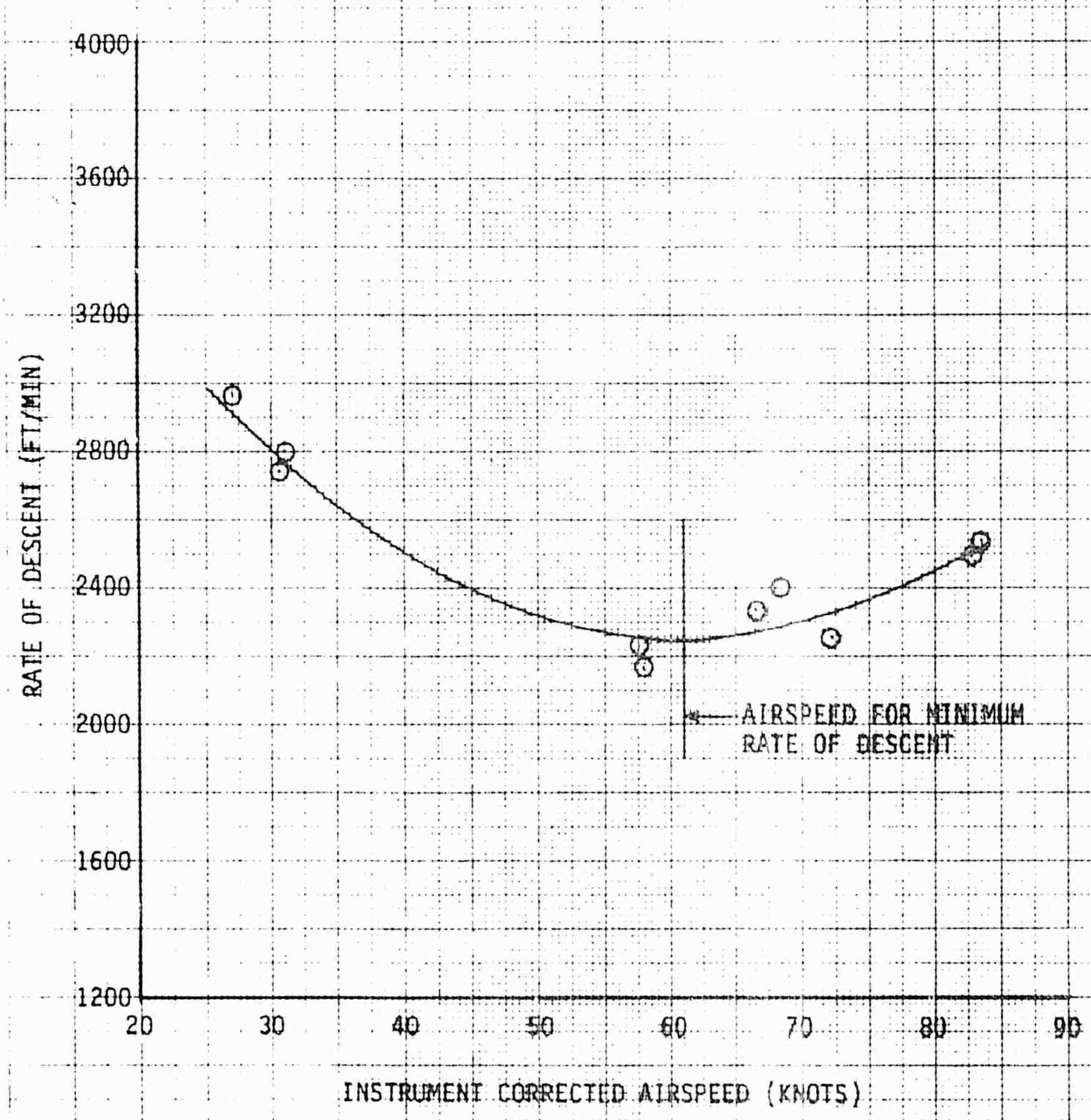

FIGURE 33
AUTOROTATIONAL PERFORMANCE
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG INST CORR AIRSPEED (KNOTS)	CONFIGURATION
15740	293.2(FWD)	-.1(LT)	6340	16.0	64	8-TOW

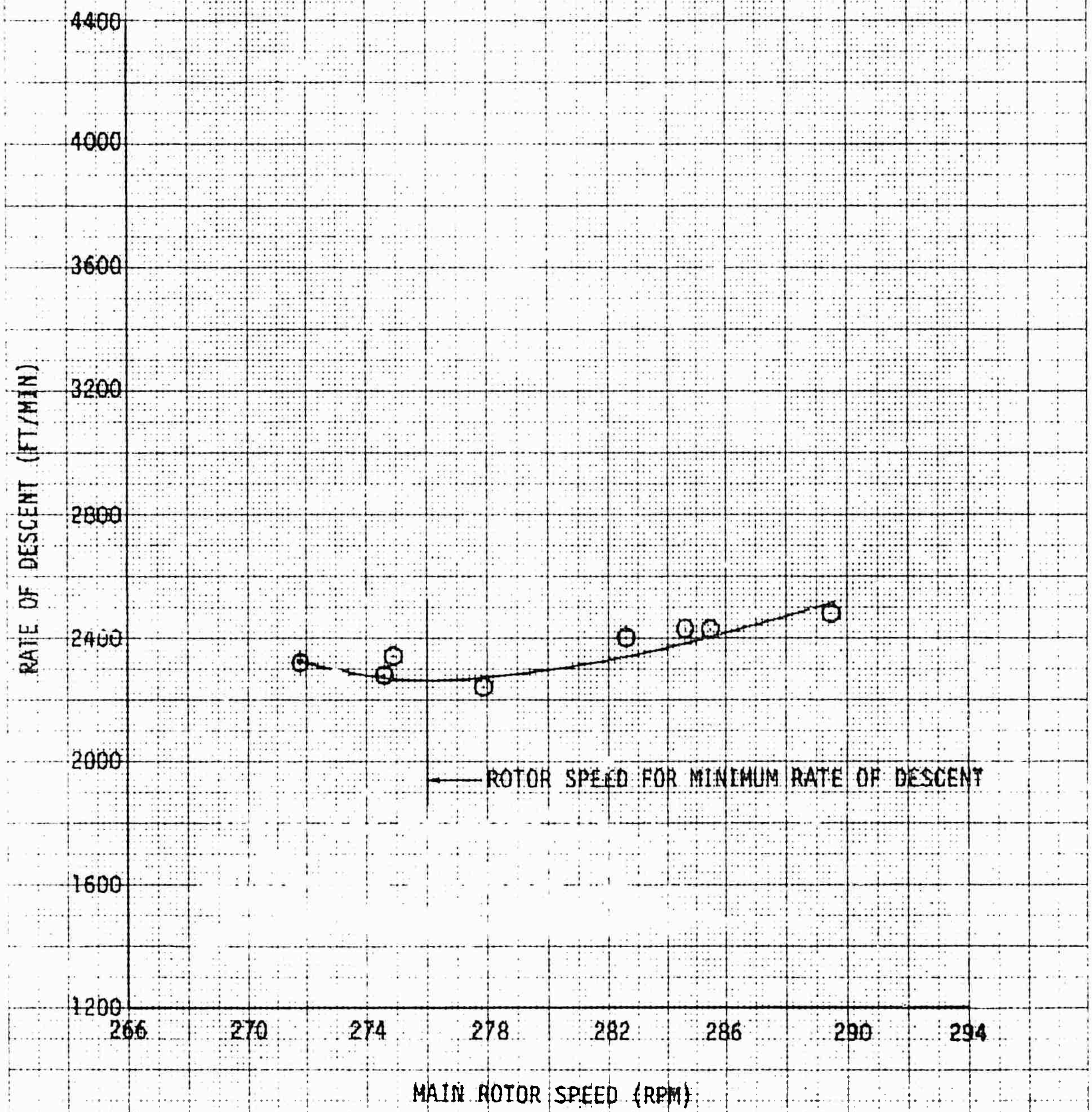

FIGURE 34
LIMITS OF CYCLIC CONTROL TRAVEL
YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. CONTROL POSITION MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. COLLECTIVE CONTROL POSITION: FULL DOWN.
6. CYCLIC TRIM POSITIONS: 3.8 INCHES FROM FULL FWD
   3.7 INCHES FROM FULL LEFT
7. DIRECTIONAL CONTROL POSITION: 3.3 INCHES FROM FULL LEFT.
8. CYCLIC CONTROL PATTERN DOES NOT CHANGE WITH A CHANGE OF COLLECTIVE CONTROL POSITION.

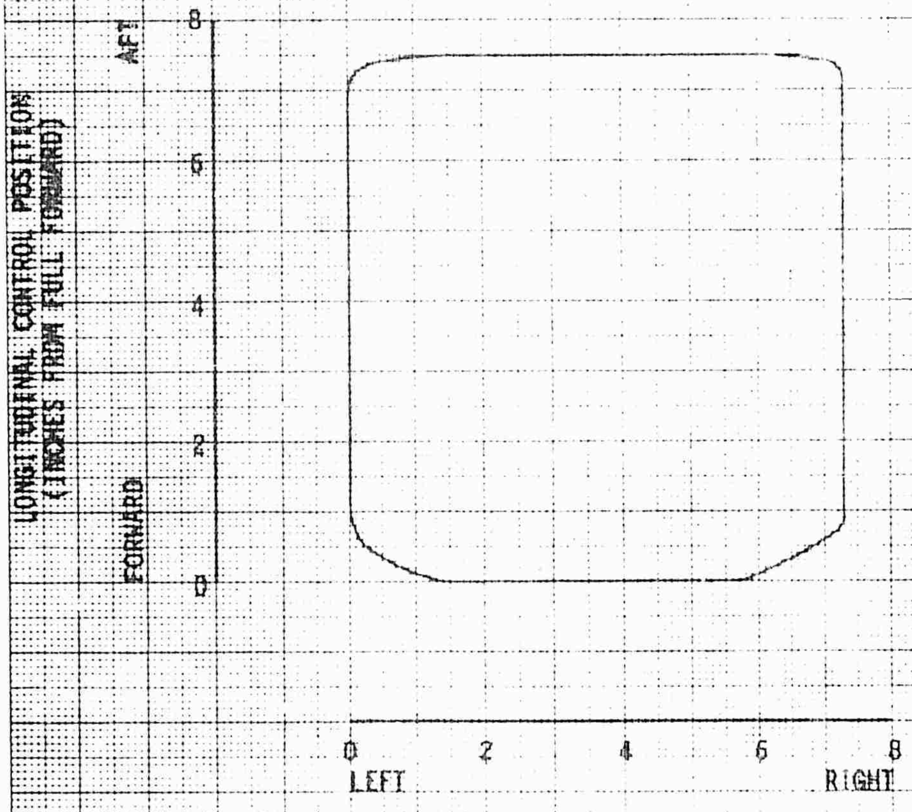

FIGURE 35
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-63 USA S/N 74-22248

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL LONGITUDINAL CONTROL TRAVEL = 7.5 INCHES.
6. LATERAL CONTROL POSITION = 3.7 INCHES FROM FULL LEFT.
7. FORCE TRIM ON, CYCLIC FRICTION AT ZERO.
8. Q-SENSOR PRESSURIZED TO SIMULATE 0 KNOTS.
9. SCAS OFF.

AVERAGE FRICTION BAND (NEAR TRIM)
AFT = 0 LB
FWD = 0 LB

○ INCREASING FORCE
□ DECREASING FORCE

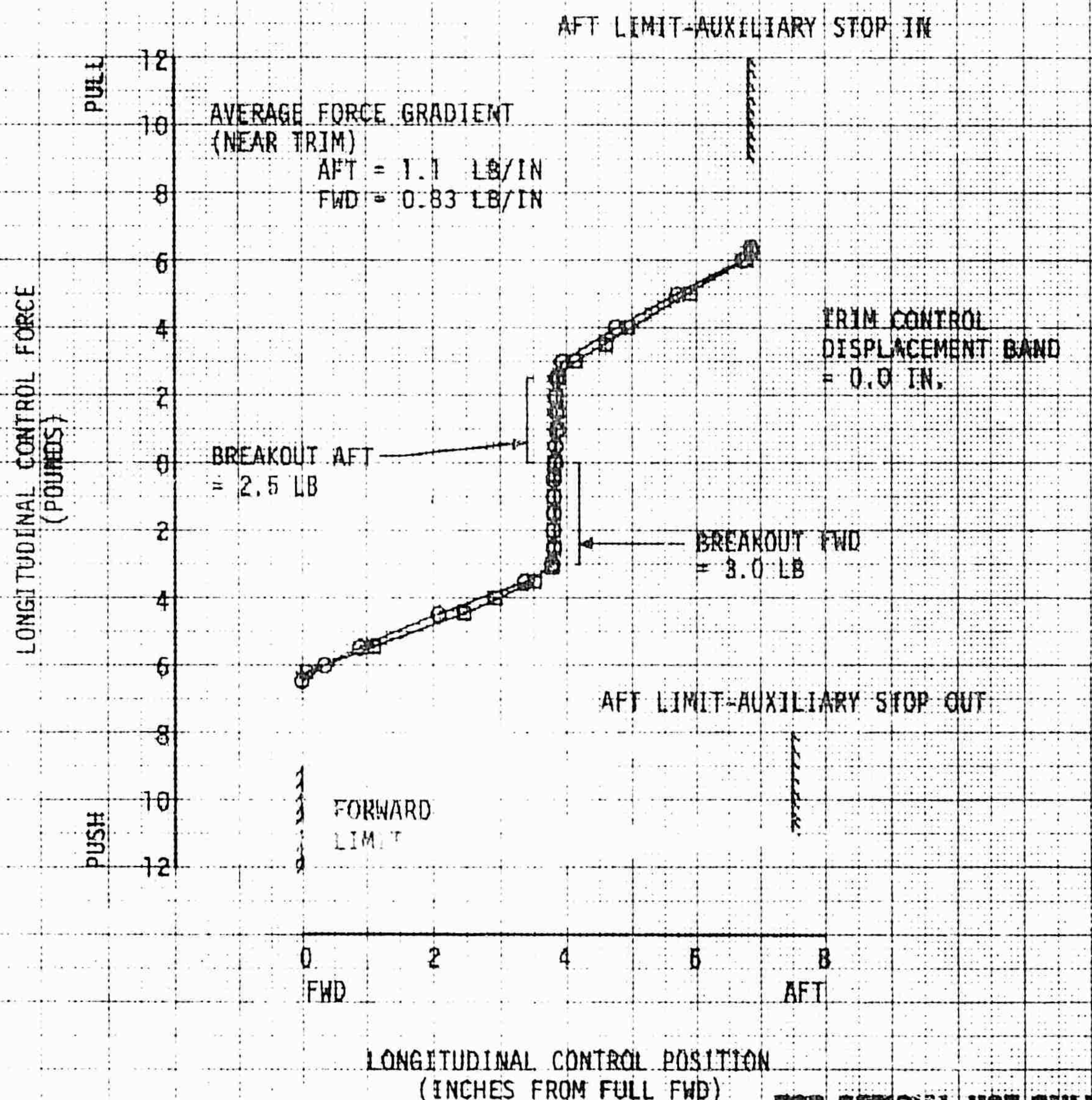

FIGURE 36
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL LONGITUDINAL CONTROL TRAVEL = 7.5 INCHES.
6. LATERAL CONTROL POSITION = 3.7 INCHES FROM FULL LEFT.
7. FORCE TRIM ON, CYCLIC FRICTION AT ZERO.
8. Q-SENSOR PRESSURIZED TO SIMULATE 50 KNOTS.
9. SCAS OFF.

AVERAGE FRICTION BAND (NEAR TRIM)
AFT = 0 LB
FWD = 0 LB

○ INCREASING FORCE
⊟ DECREASING FORCE

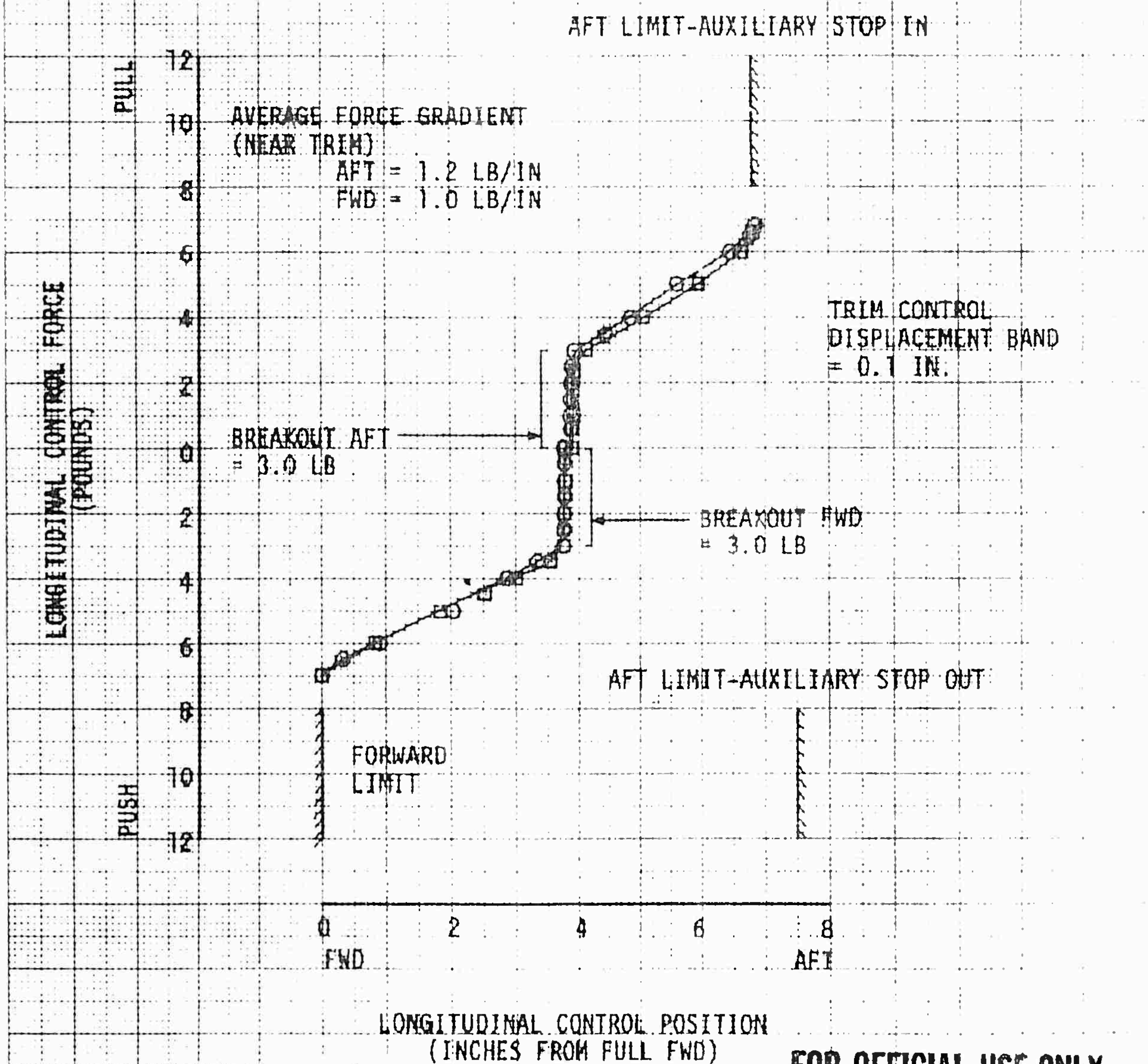

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FIGURE 37

LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS

YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL LONGITUDINAL CONTROL TRAVEL = 7.5 INCHES.
6. LATERAL CONTROL POSITION = 3.7 INCHES FROM FULL LEFT.
7. FORCE TRIM ON, CYCLIC FRICTION AT ZERO.
8. Q-SENSOR PRESSURIZED TO SIMULATE 100 KNOTS.
9. SCAS OFF.

AVERAGE FRICTION BAND (NEAR TRIM)
AFT = 0 LB
FWD = 0 LB

○ INCREASING FORCE
⊡ DECREASING FORCE

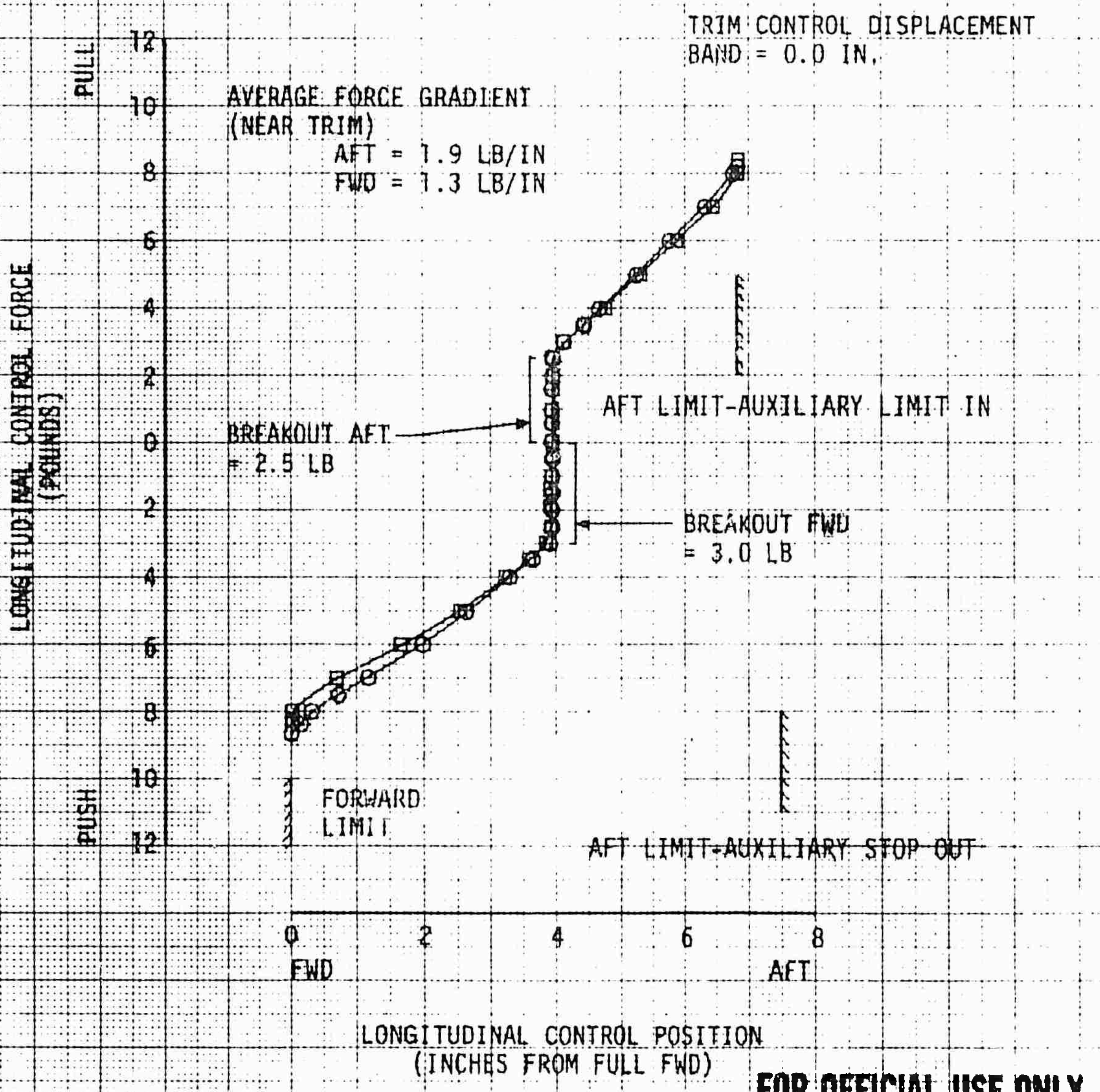

FIGURE 38
LONGITUDINAL CONTROL SYSTEM CHARACTERISTICS
YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL LONGITUDINAL CONTROL TRAVEL = 7.6 INCHES.
6. LATERAL CONTROL POSITION = 3.7 INCHES FROM FULL LEFT.
7. FORCE TRIM ON, CYCLIC FRICTION AT ZERO.
8. Q-SENSOR PRESSURIZED TO SIMULATE 150 KNOTS.
9. SCAS OFF.
10. TRIM CONTROL DISPLACEMENT BAND = 0.2 INCHES.

AVERAGE FRICTION BAND
(NEAR TRIM)
AFT = 0 LB
FWD = 0 LB

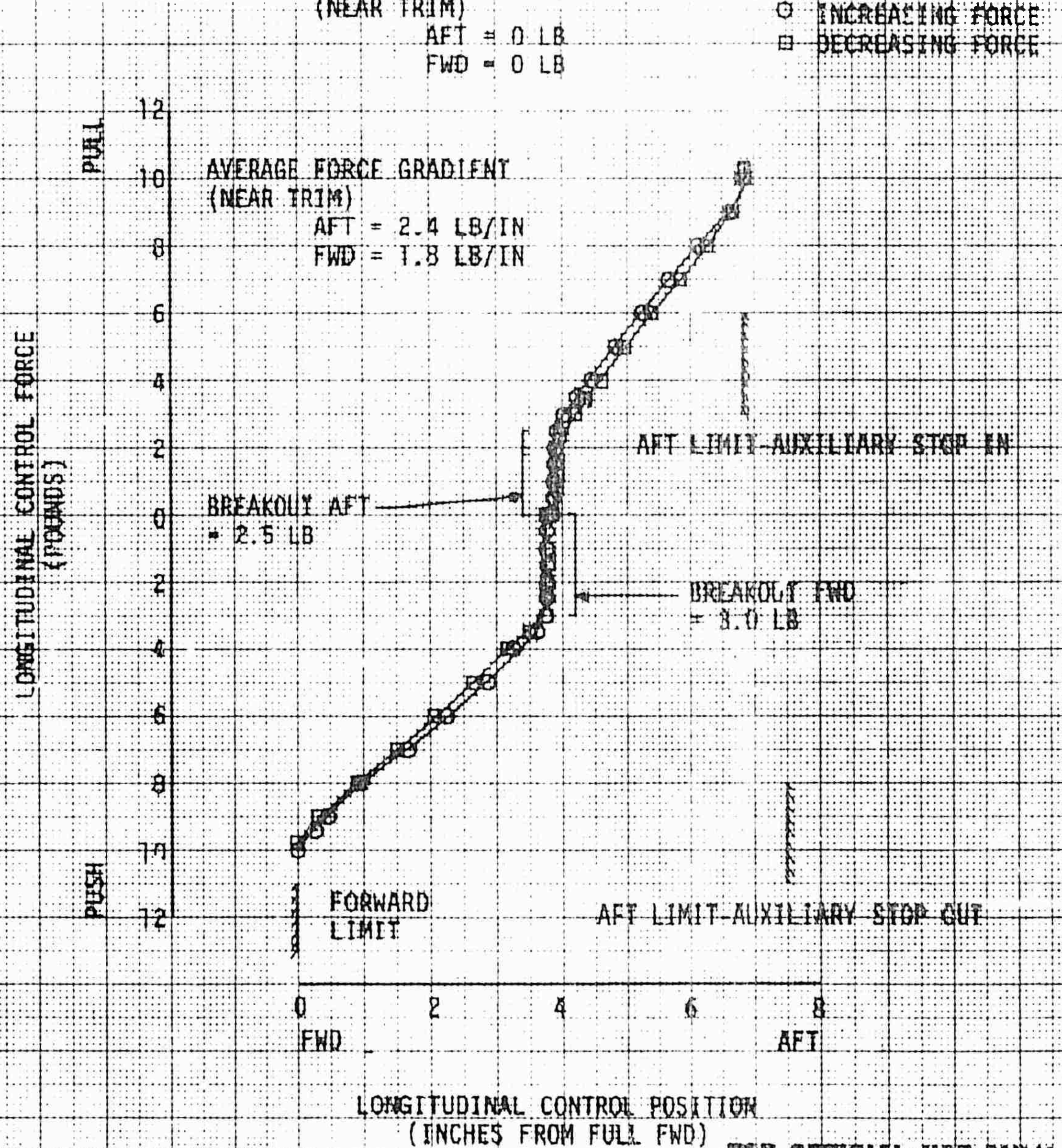

FIGURE 39

LATERAL CONTROL SYSTEM CHARACTERISTICS

YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL LATERAL CONTROL TRAVEL = 7.3 INCHES.
6. LONGITUDINAL CONTROL POSITION = 3.8 INCHES FROM FULL FORWARD.
7. FORCE TRIM ON, CYCLIC FRICTION AT ZERO.
8. TRIM CONTROL DISPLACEMENT BAND = 0 INCHES.

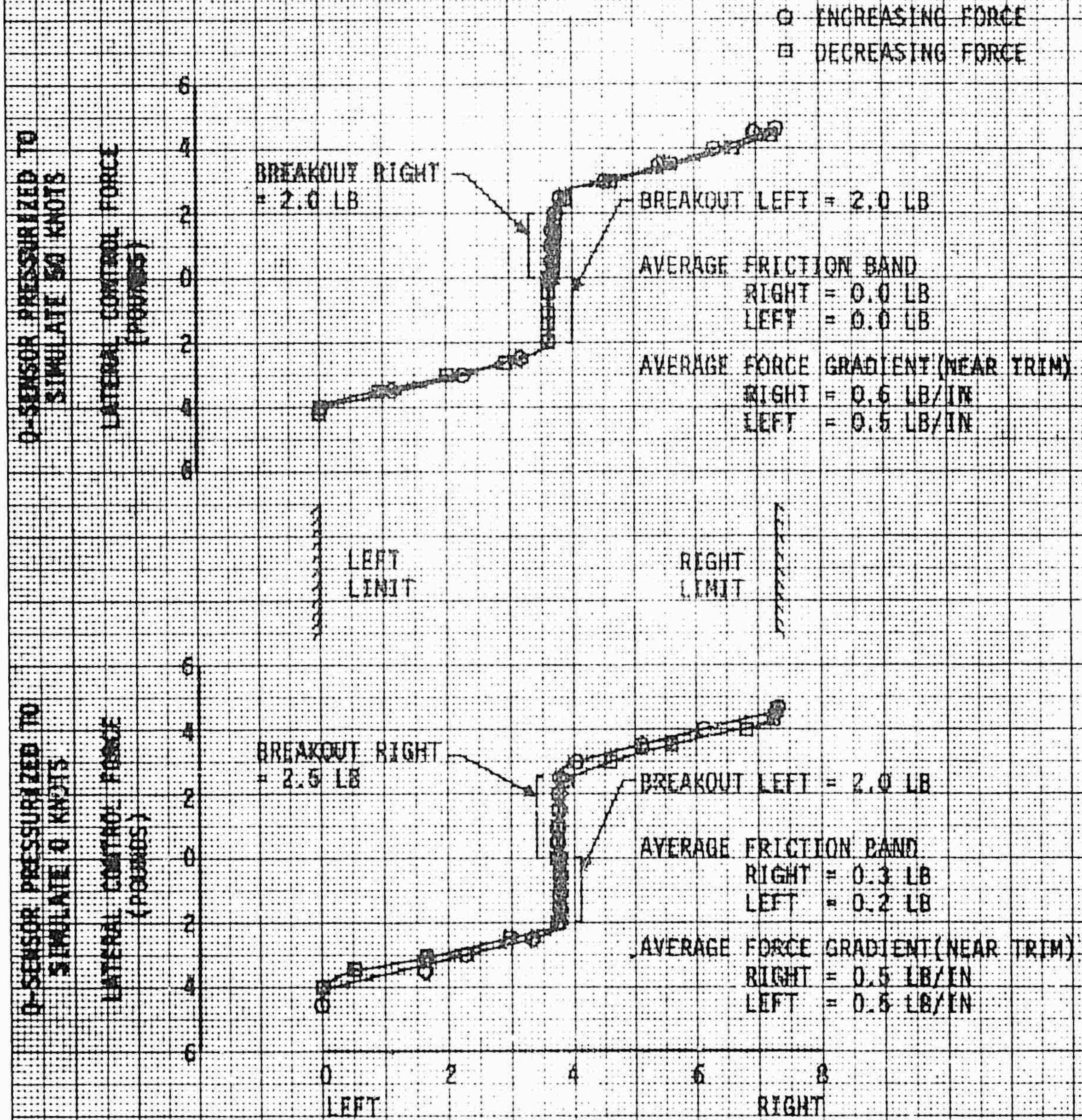

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FIGURE 40

LATERAL CONTROL SYSTEM CHARACTERISTICS

YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. FORCES AND POSITIONS MEASURED AT CENTER OF GRIP.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL LATERAL CONTROL TRAVEL = 7.3 INCHES.
6. LONGITUDINAL CONTROL POSITION = 3.8 INCHES FROM FULL FORWARD.
7. FORCE TRIM ON, CYCLIC FRICTION AT ZERO.

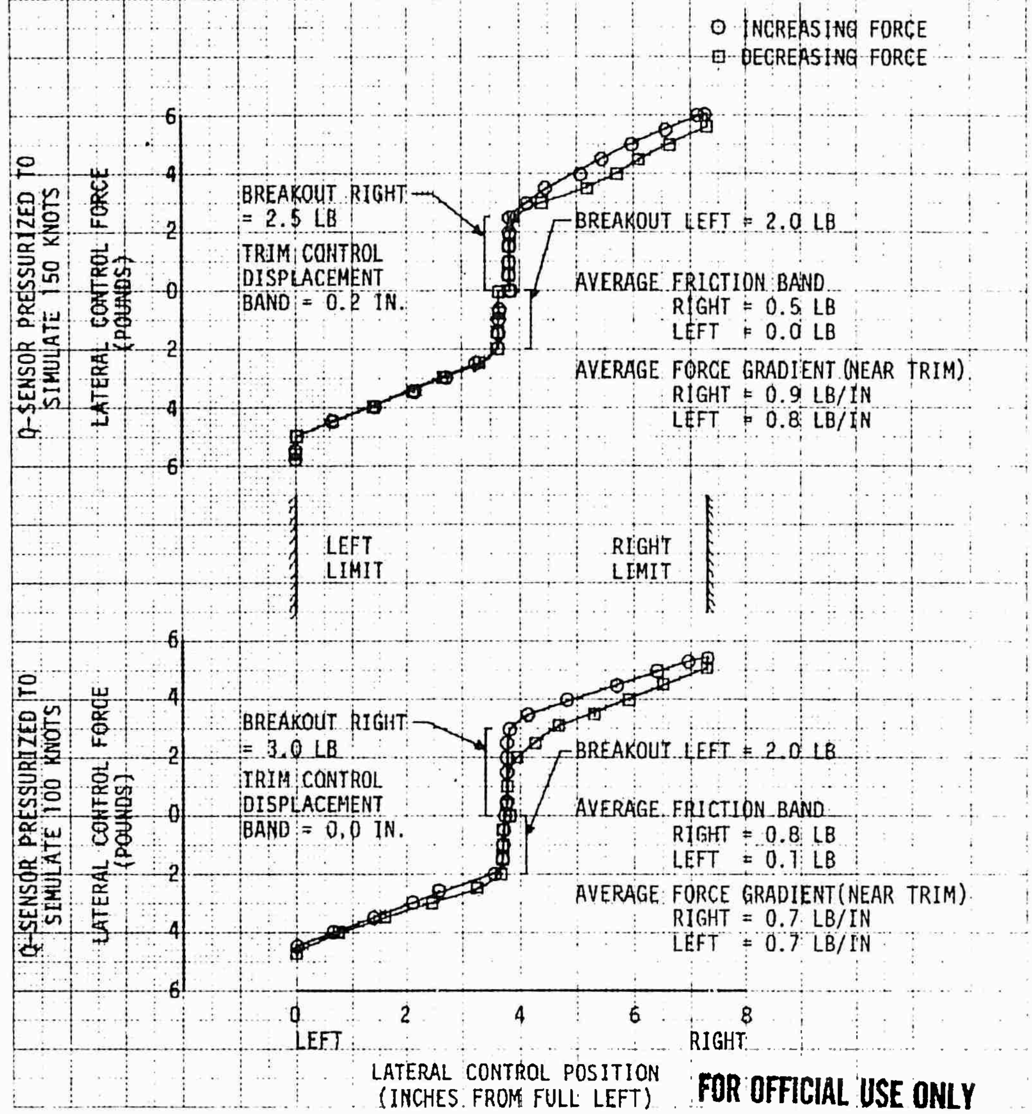

FIGURE 41

DIRECTIONAL CONTROL SYSTEM CHARACTERISTICS

YAH-63 USA S/N 74-22246

NOTES:
1. ROTORS STATIC.
2. FORCES MEASURED AT THE DIRECTIONAL CONTROL.
3. HYDRAULIC AND ELECTRICAL POWER PROVIDED BY GROUND POWER UNITS.
4. NO. 1, 2 AND UTILITY BOOST SYSTEMS ON.
5. TOTAL DIRECTIONAL CONTROL TRAVEL = 6.6 INCHES.
6. FORCE TRIM ON.
7. TRIM CONTROL DISPLACEMENT BAND = 0.9 INCHES.

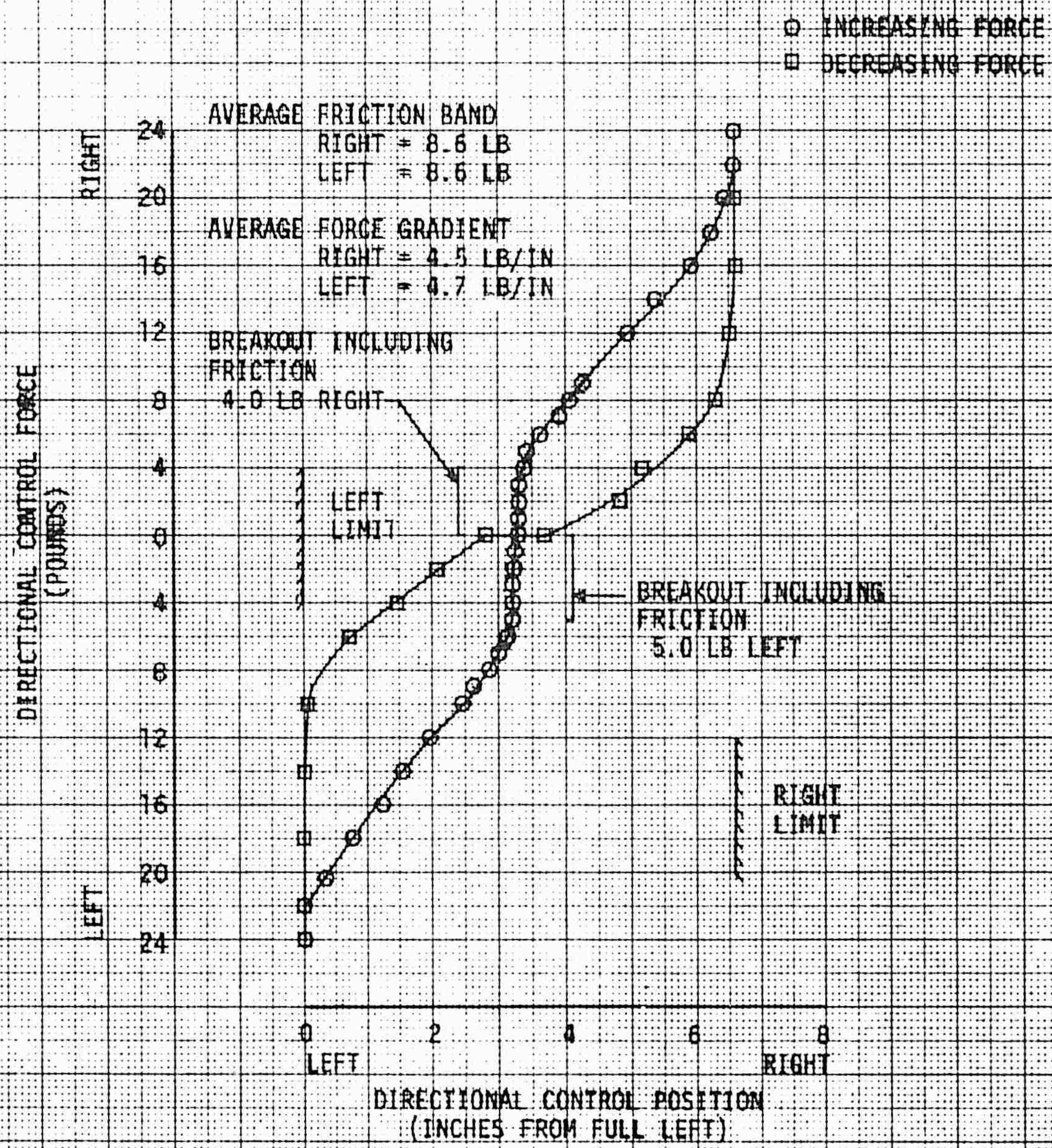

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FIGURE 42
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (IN.)	AVG CG LOCATION LAT (IN.)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	CONFIGURATION	SCAS CONDITION
16480.	293.0(FWD)	-.6(LT)	6480.	23.0	276.	CLEAN	ON

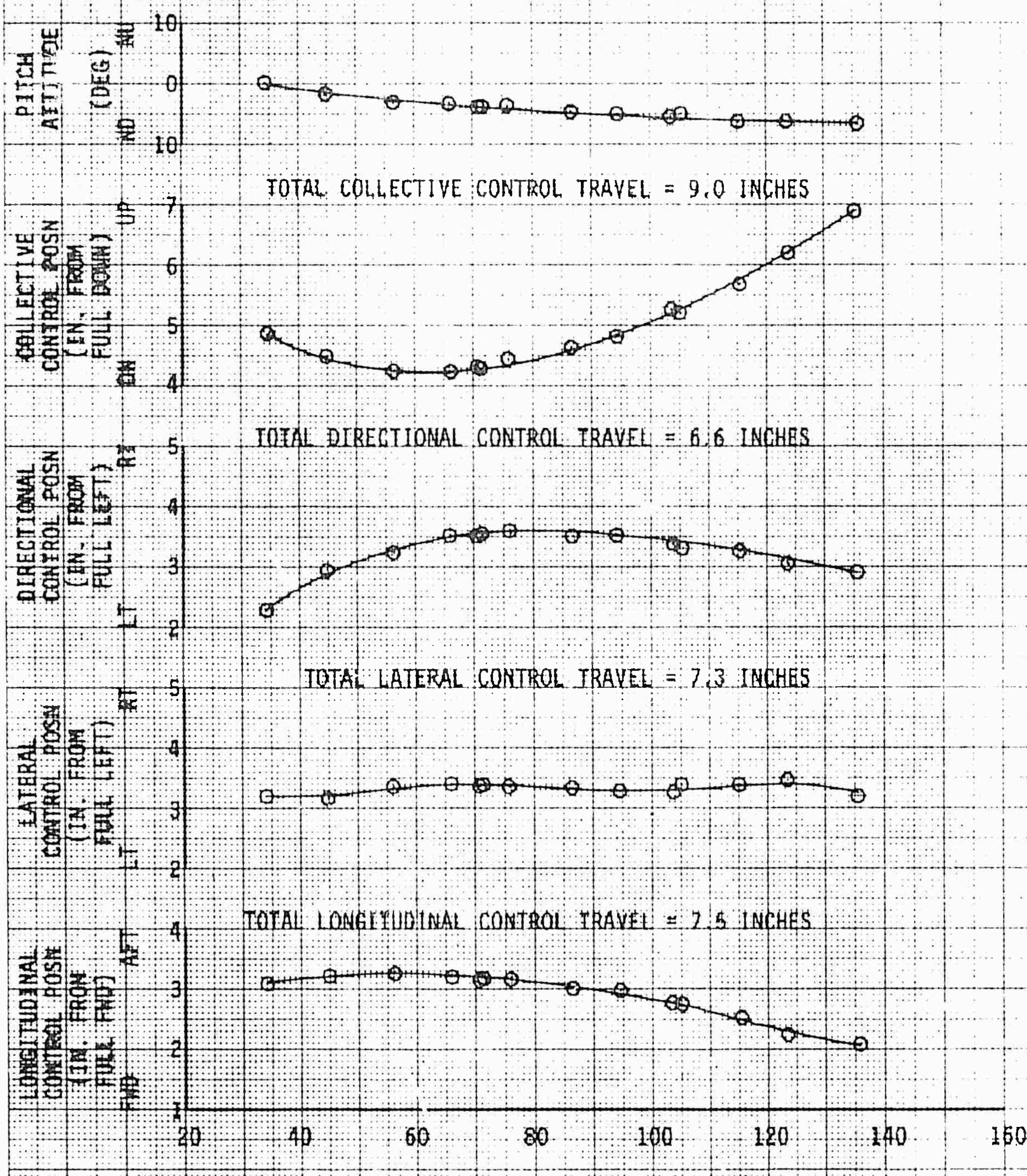

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FIGURE 43
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	CONFIGURATION	SCAS CONDITION
15700	292.9(FWD)	-.6(LT)	6880	22.0	275	8 TOW	ON

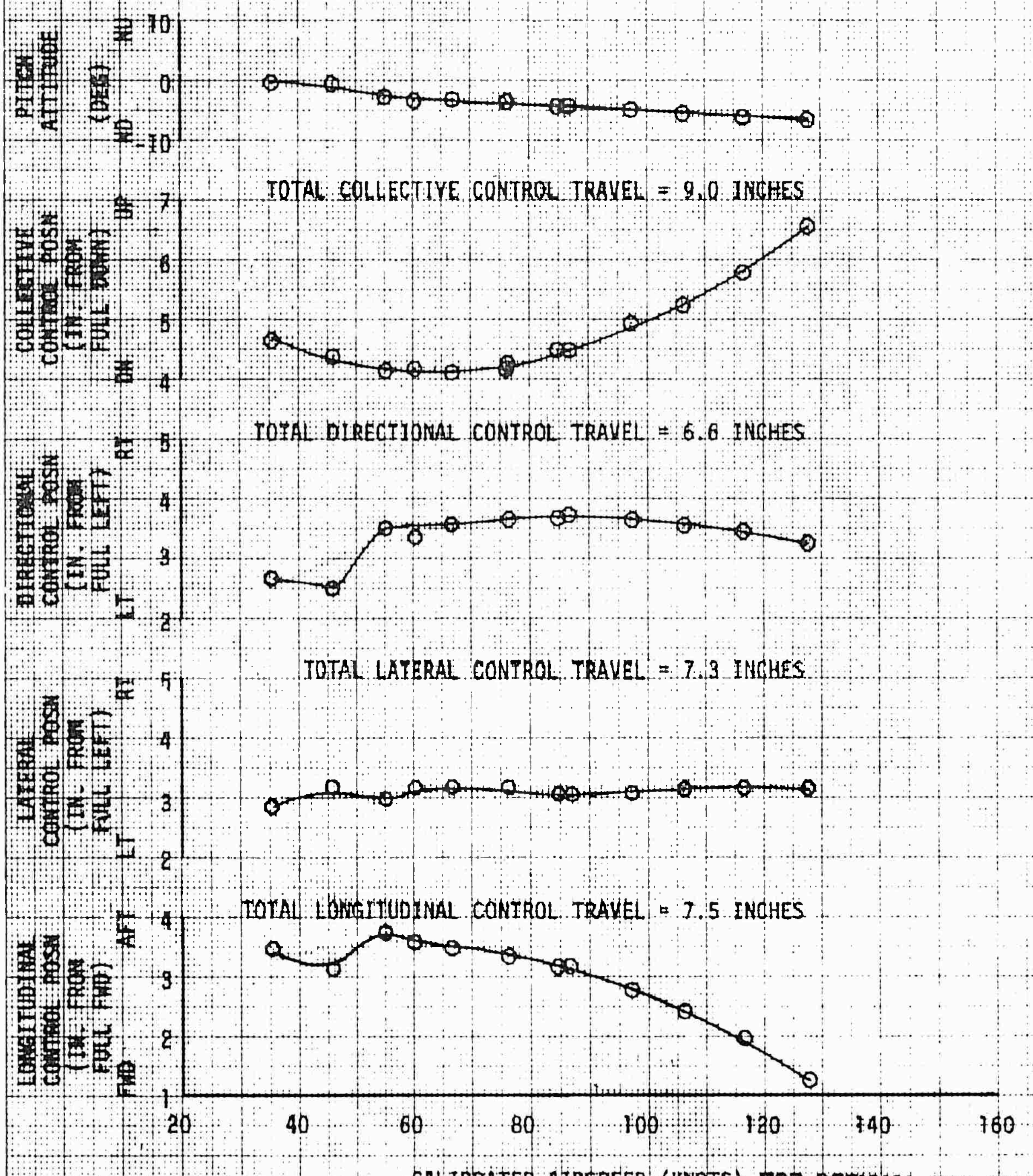

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FIGURE 44
CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	CONFIGURATION	SCAS CONDITION
14660	298.8(AFT)	-.6(LT)	6920	15.5	277	CLEAN	ON

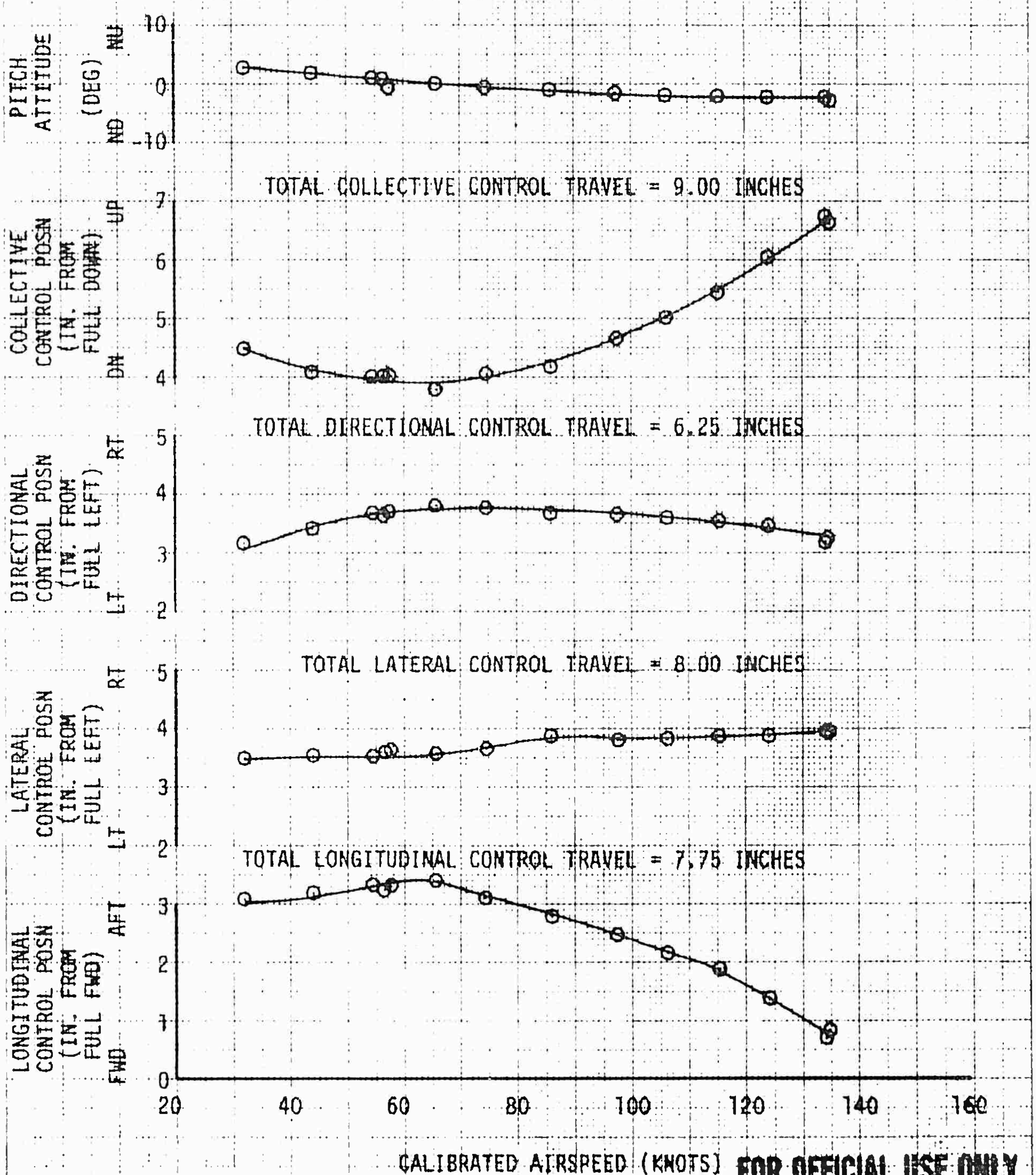

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FIGURE 45

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	CONFIGURATION	SCAS CONDITION
15580	298.6(AFT)	-0.1(LT)	7580	11.5	276	8-TOW	ON

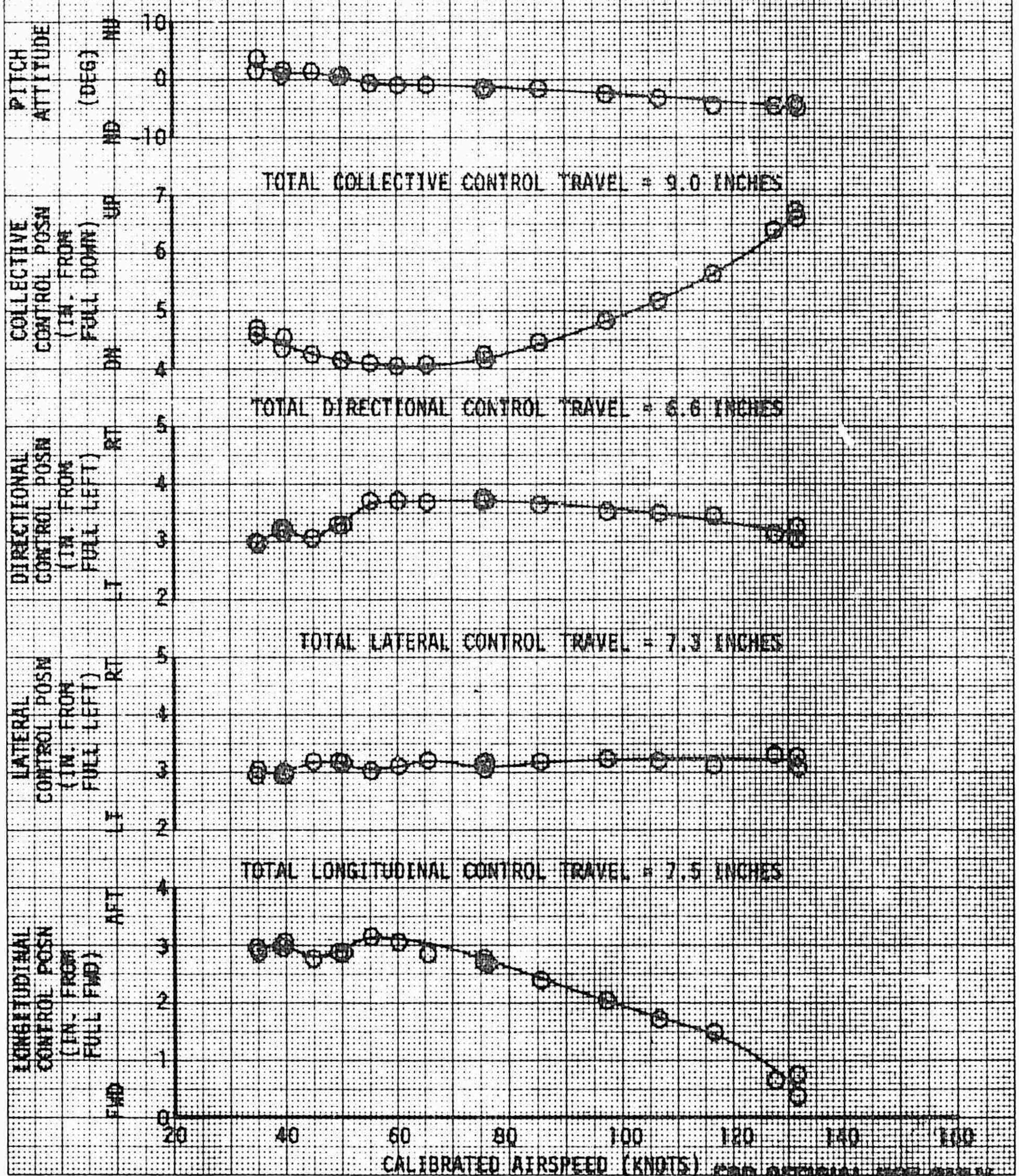

FIGURE 46
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
16140	298.6(AFT)	-.3(LT)	6660	18.5	277	LEVEL	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. B TOW CONFIGURATION.

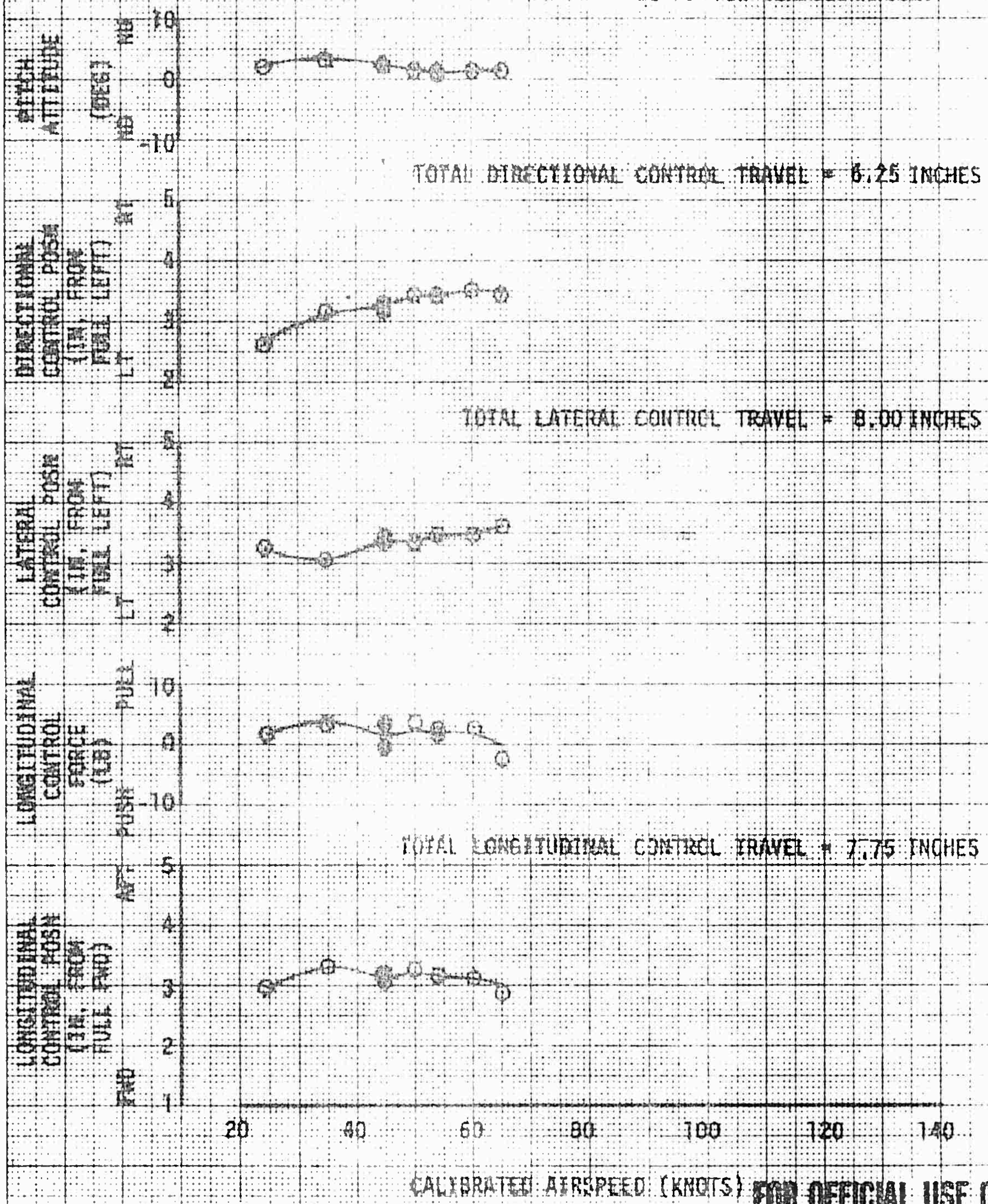

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FIGURE 47
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
15840	298.5(AFT)	-.3(LT)	6320	19.0	277	LEVEL	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION

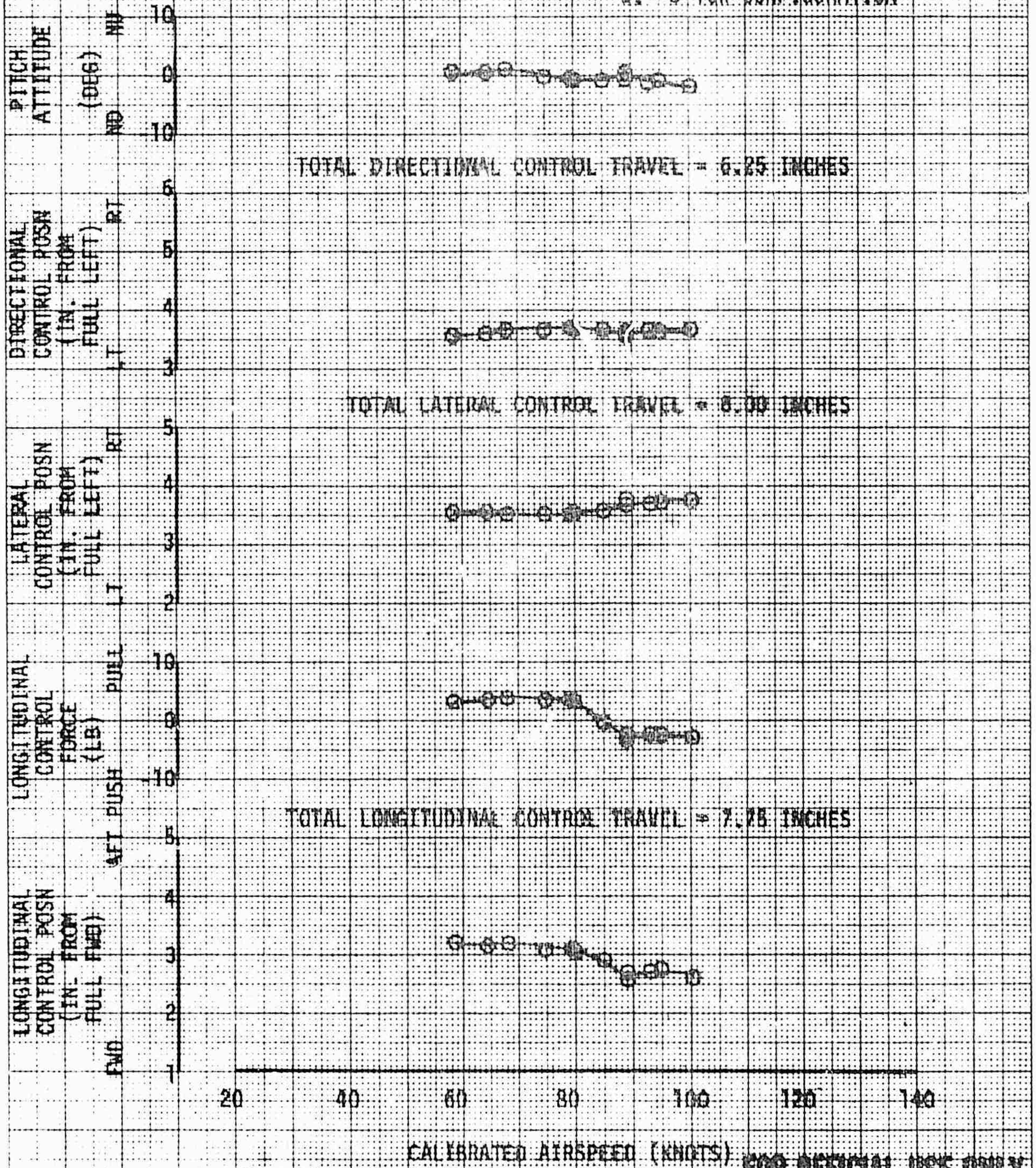

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FIGURE 48

COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY

YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
16540	298.1(AFT)	-.3(LT)	6440	19.0	277	LEVEL	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION.

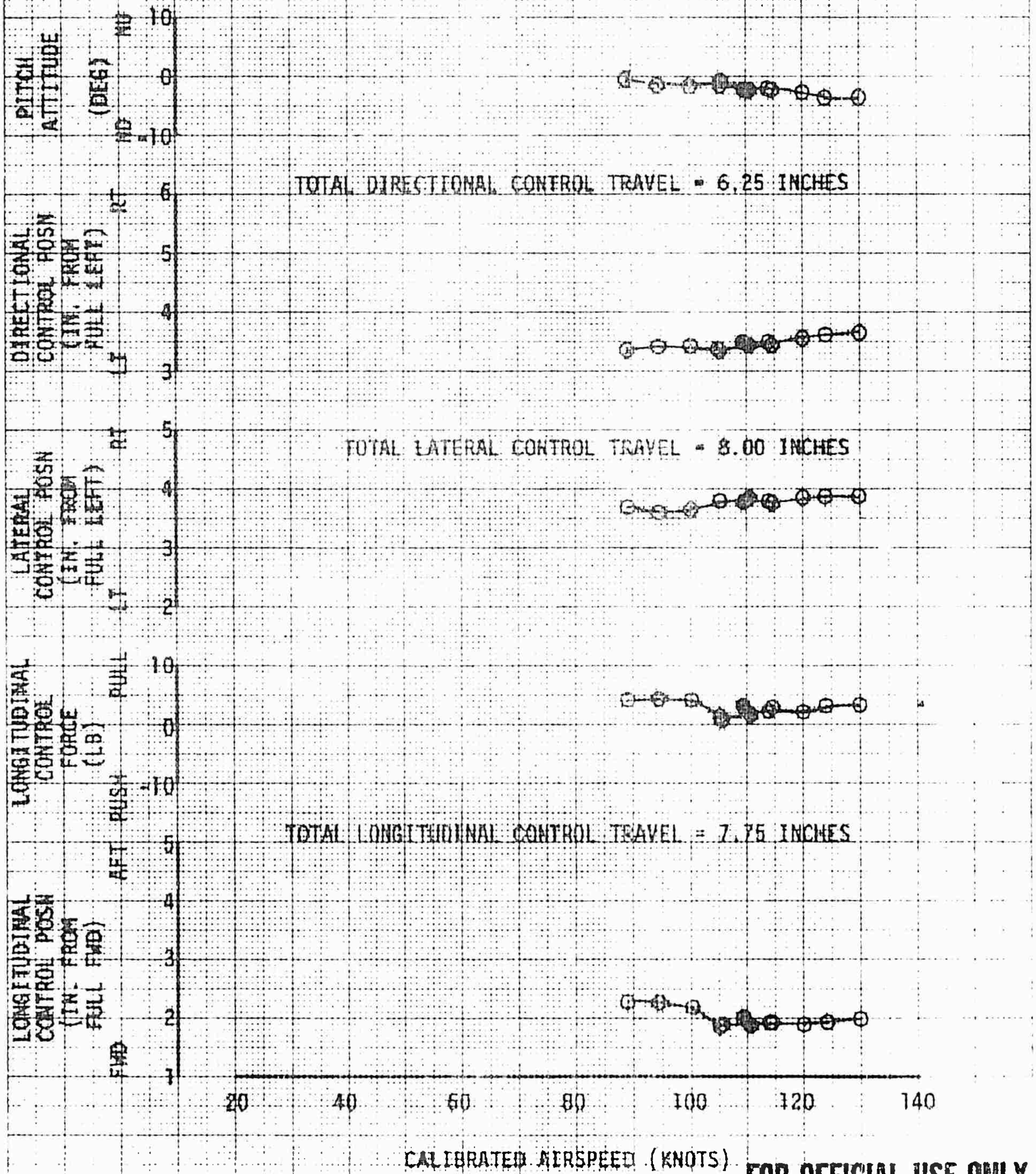

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FIGURE 49
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
16120	298.7(AFT)	-.3(LT)	7180	20.0	277	CLIMB	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION.
3. 600 FT-LB TORQUE CLIMB.

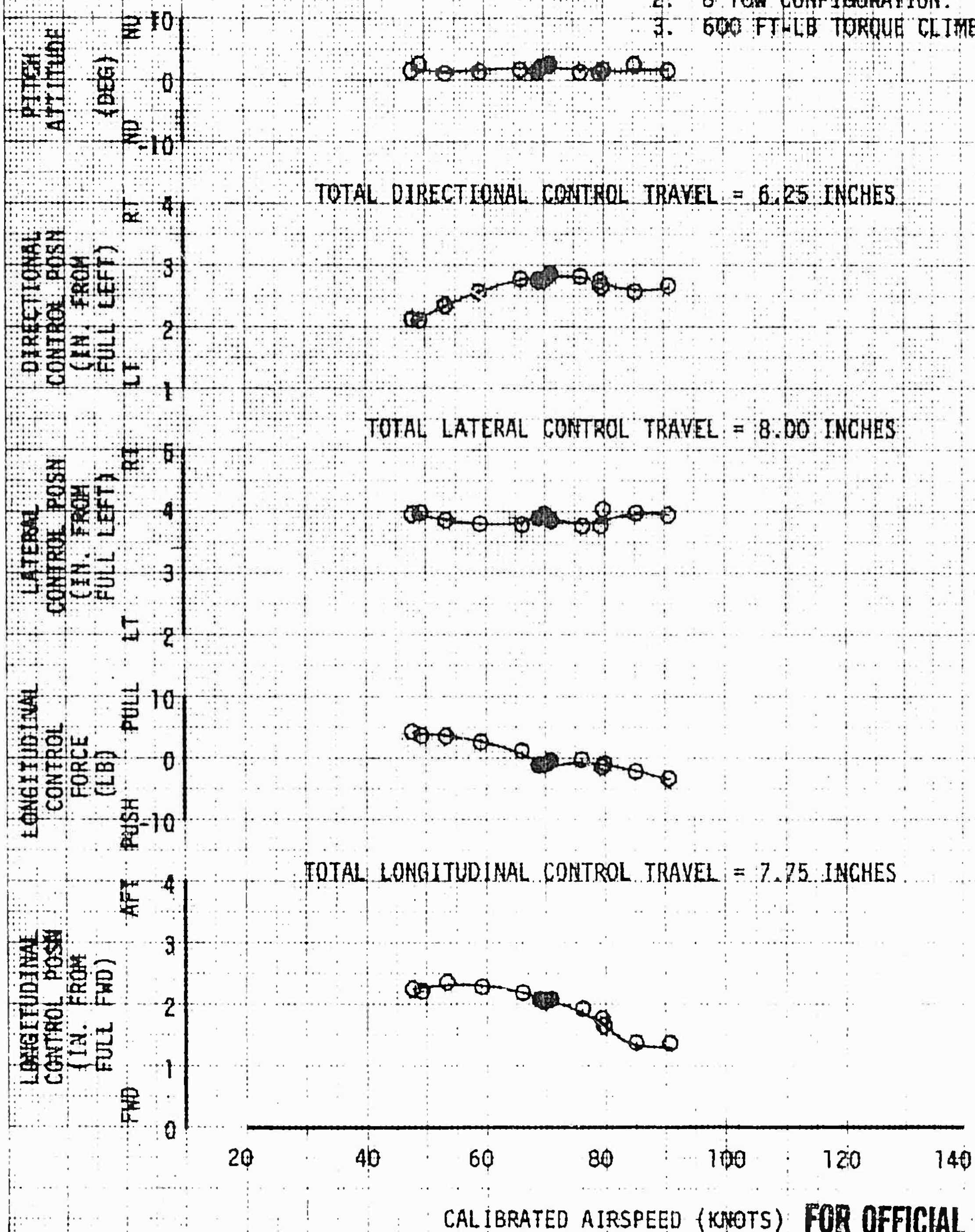

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FIGURE 50
COLLECTIVE-FIXED STATIC LONGITUDINAL STABILITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
16200	298.7(AFT)	-.3(LT)	6840	20.0	276	DESCENT	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION
3. 1000 FPM DESCENT.

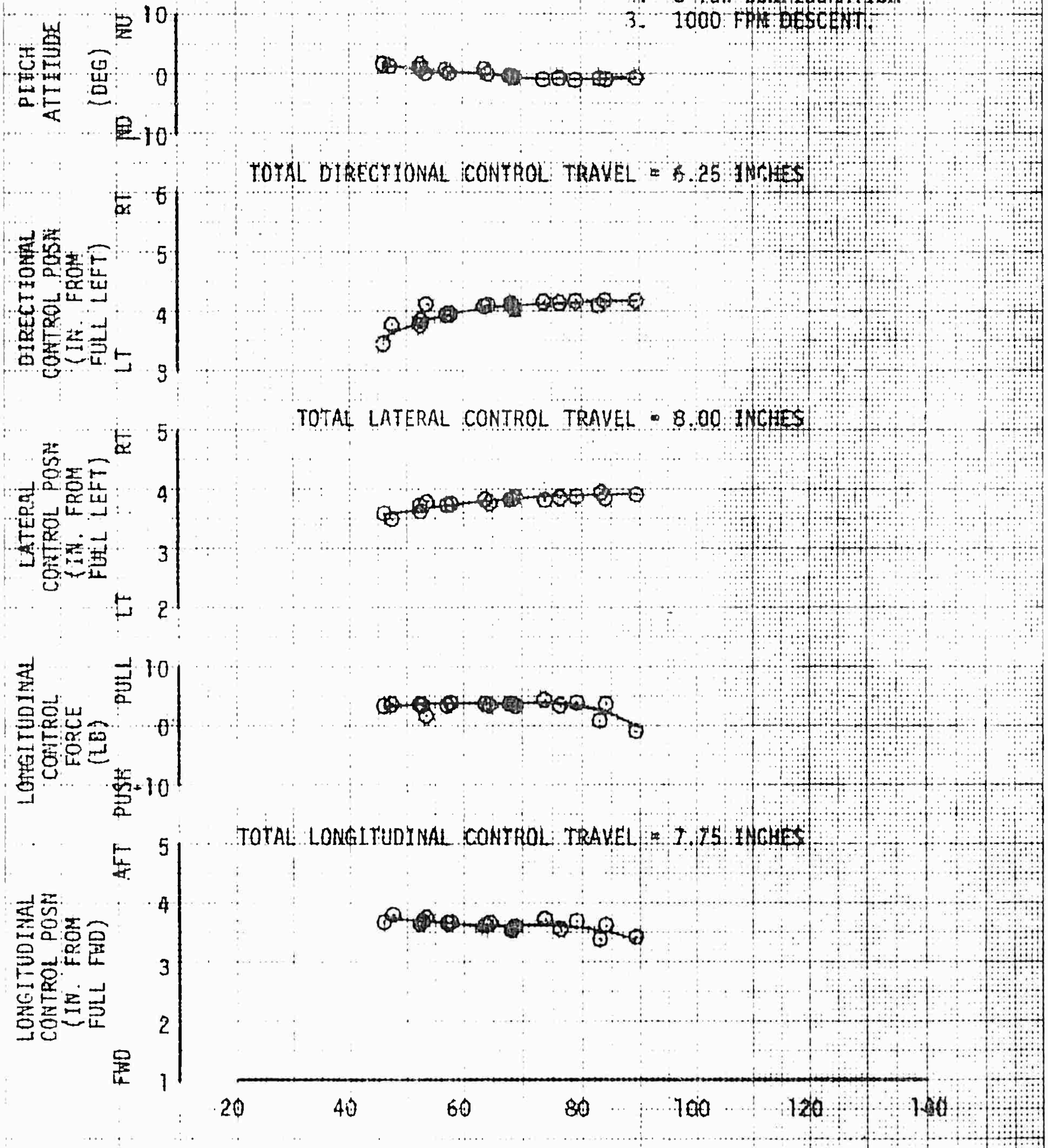

FIGURE 51
STATIC LATERAL-DIRECTIONAL STABILITY
YAH-63 USA S/N 74-22247
TRIM CONDITION: LEVEL FLIGHT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
15940.	298.6(AFT)	-.3 (LT)	7000.	20.5	276.	40	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. 8 TOW CONFIGURATION

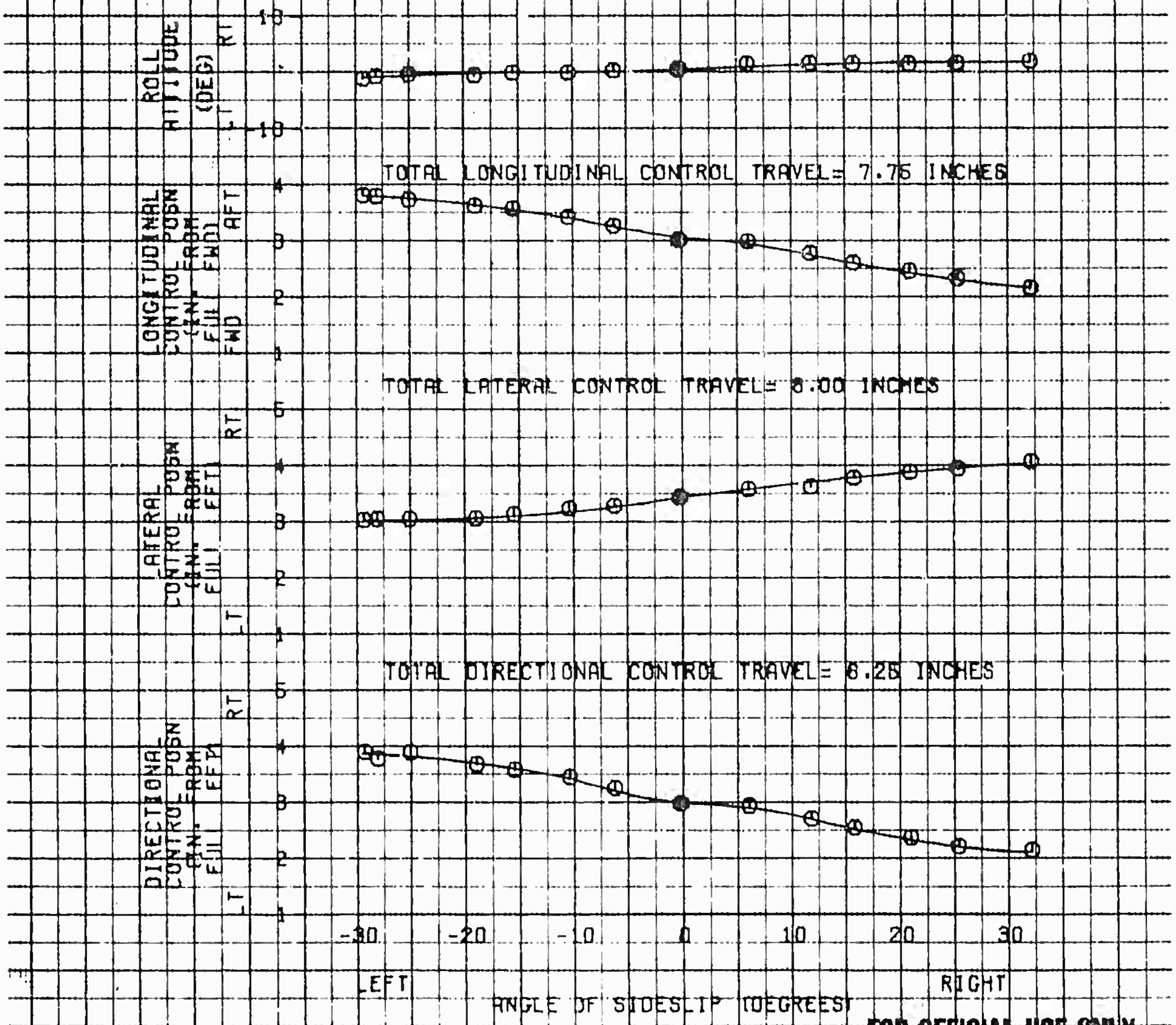

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FIGURE 52

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-63 USA S/N 74-22247

TRIM CONDITION: LEVEL FLIGHT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
16300.	298.8(AFT)	-.3 (LT)	6980.	20.0	276.	85.	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. 8 TOW CONFIGURATION

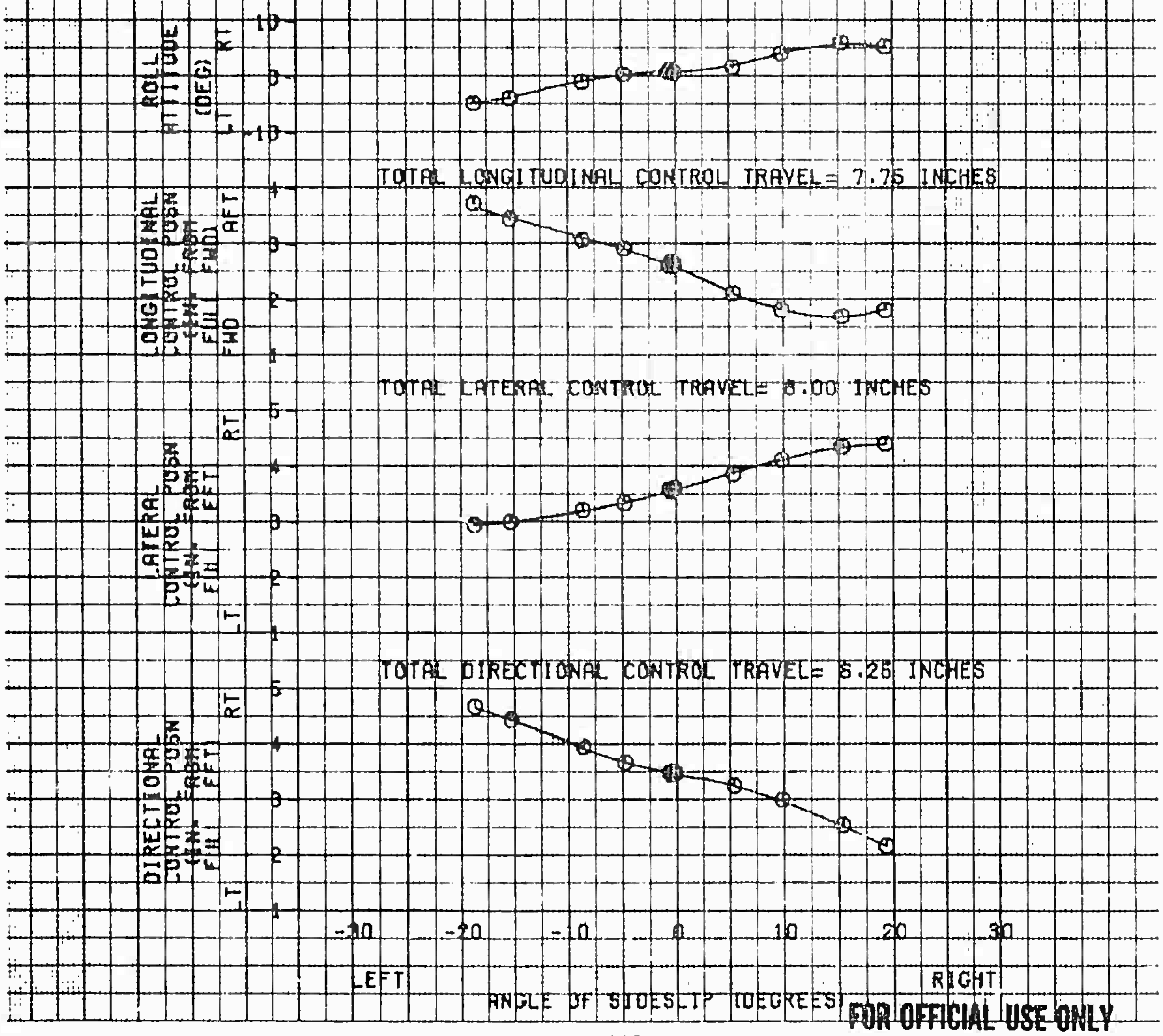

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FIGURE 53

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-63 USA S/N 74-22247

TRIM CONDITION: LEVEL FLIGHT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS.)	AVG CG LOCATION LAT (BL.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
16220.	298.8(AFT)	-.3(LT)	5980.	11.5	277.	125.	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. 8 TOW CONFIGURATION

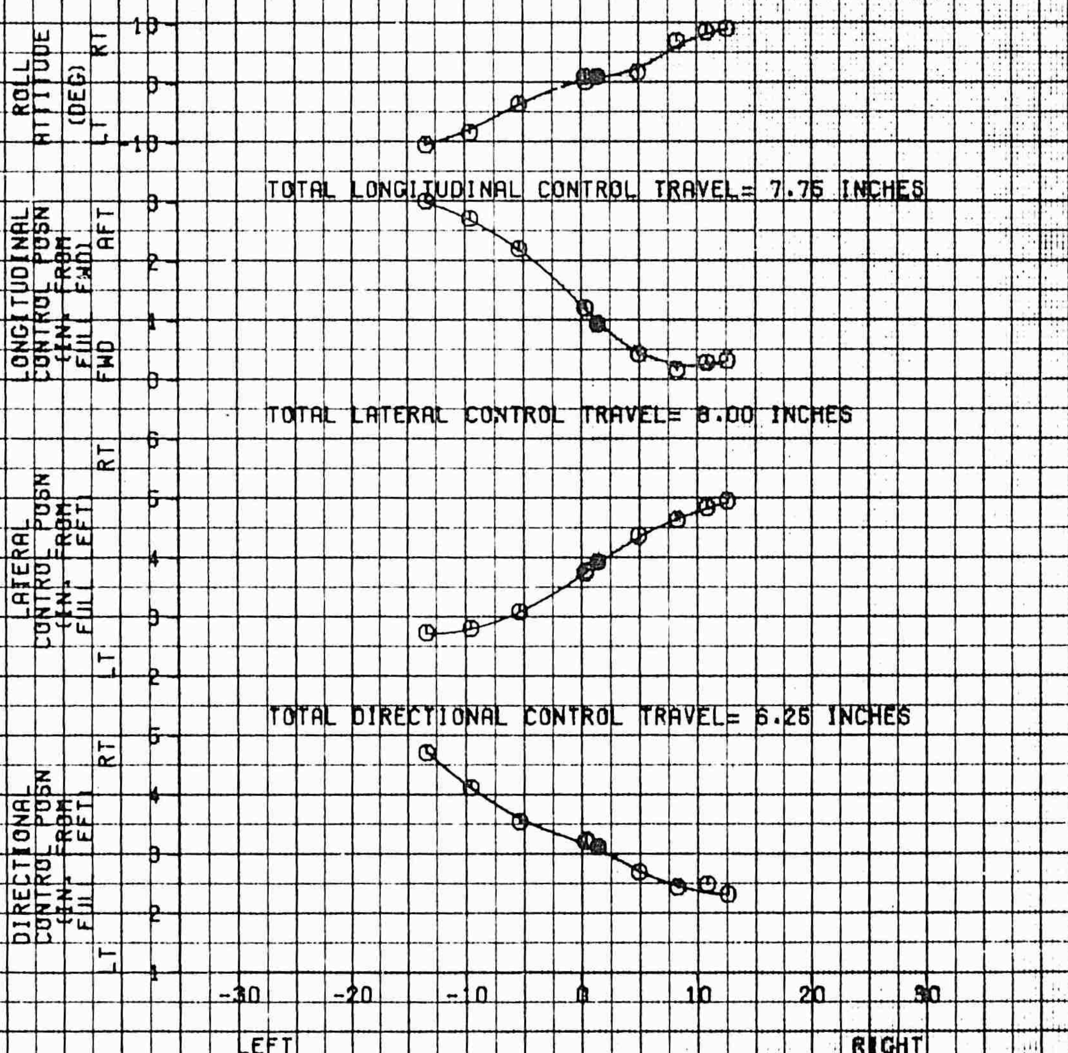

FIGURE 54

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-63 USA S/N 74-22247

TRIM CONDITION: CLIMBING FLIGHT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
15820.	298.2(AFT)	-.3 (LT)	7640.	12.5	276.	70.	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. 8 TOW CONFIGURATION
3. 600 FT-LB TORQUE CLIMB

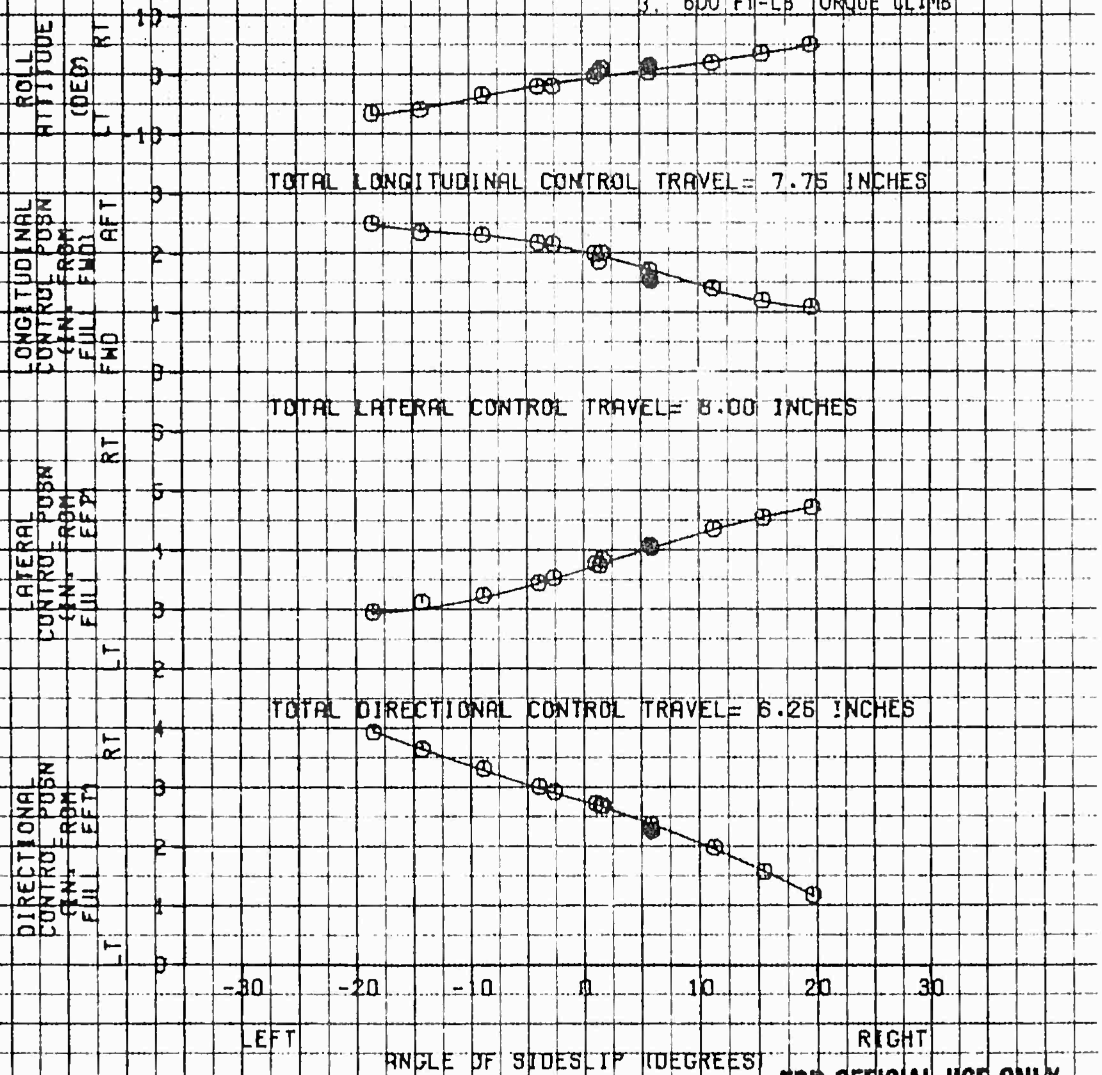

FIGURE 56

STATIC LATERAL-DIRECTIONAL STABILITY

YAH-63 USA S/N 74-22247

TRIM CONDITION: DESCENDING FLIGHT

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS.)	LAT (BL.)	AVG DENS ALT (FT)	AVG OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
18020.	298.3(AFT)	-.3(LT)	7020.	13.0	277.	68.	ON

NOTES: 1. SHADED SYMBOLS DENOTE TRIM
2. 8 TOW CONFIGURATION
3. 1000 FPM DESCENT

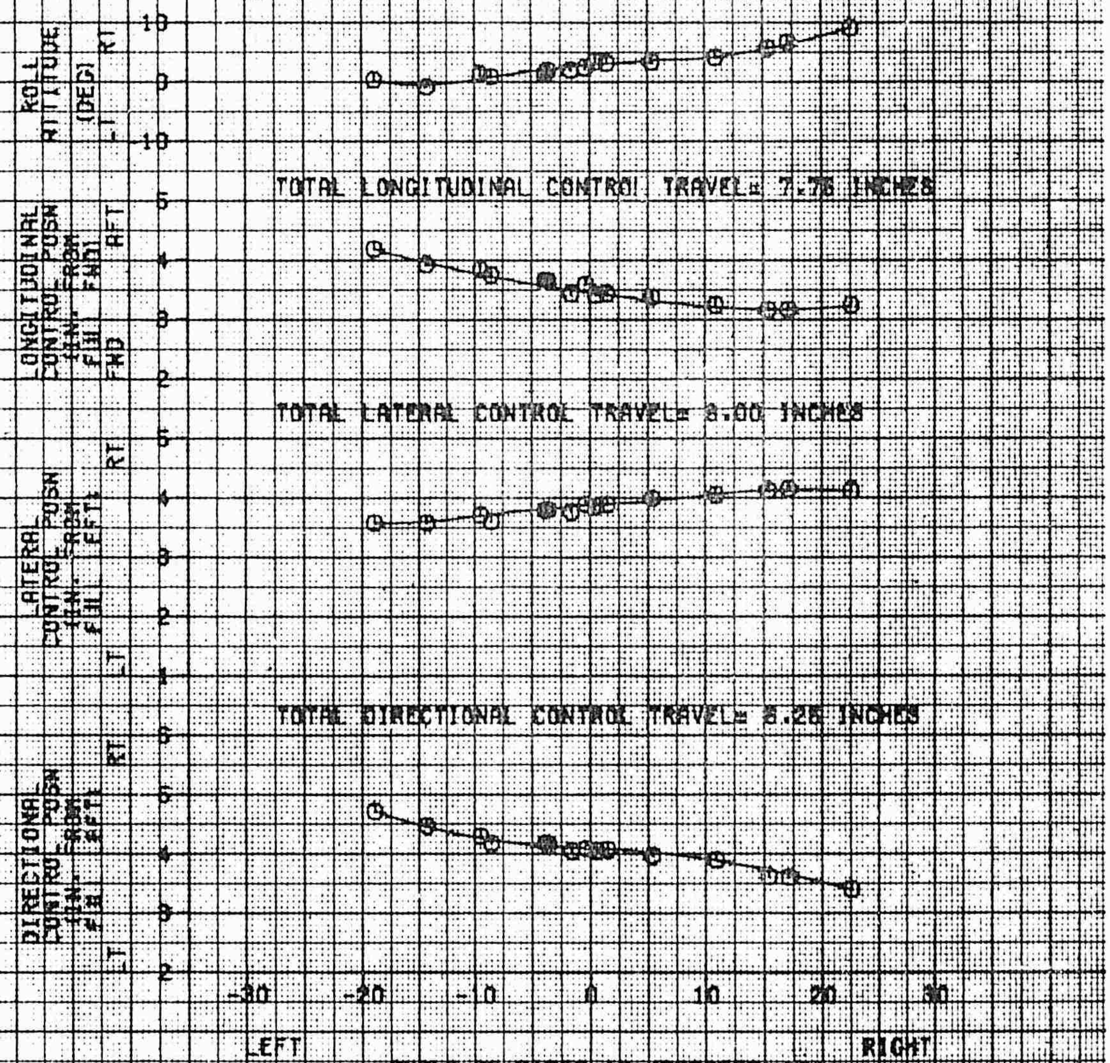

FIGURE 56
MANEUVERING STABILITY
YAH-63 USA S/N 74-22247

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB A/S (KT)	FLIGHT CONDITION
○	15300	298.7(AFT)	-0.2(LT)	7120	21.5	276	58	LT TURN
□	15520	299.0(AFT)	-0.2(LT)	6860	21.5	277	59	RT TURN

NOTE: 8-TOW CONFIGURATION

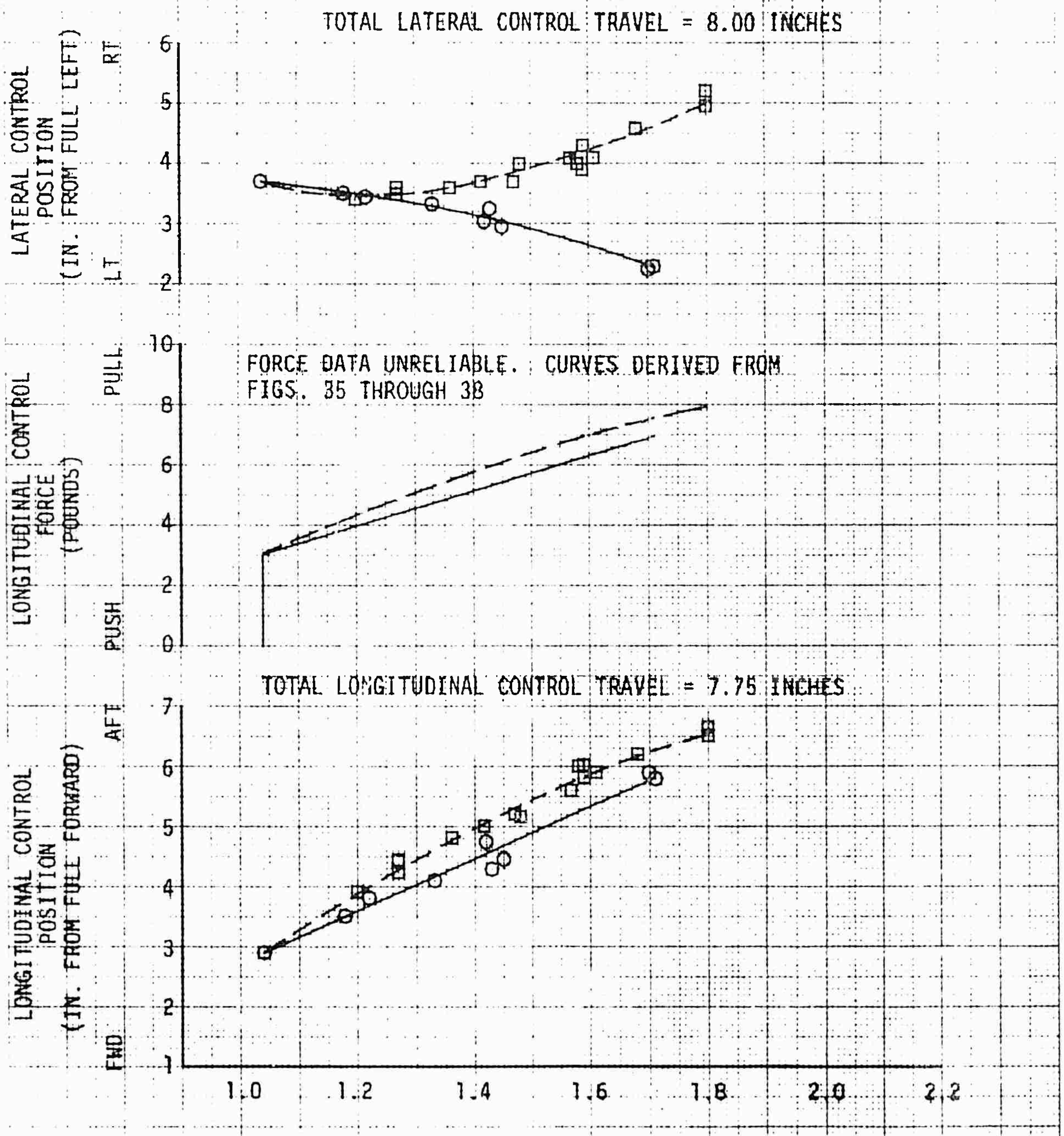

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FIGURE 57
MANEUVERING STABILITY
YAH-63 USA S/N 74-22246

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB A/S (KT)	FLIGHT CONDITION
○	16420	298.6(AFT)	-0.1(LT)	6600	11.0	276	61	LT TURN
□	15980	298.4(AFT)	-0.1(LT)	7360	10.5	276	61	RT TURN

NOTE: 8-TOW CONFIGURATION

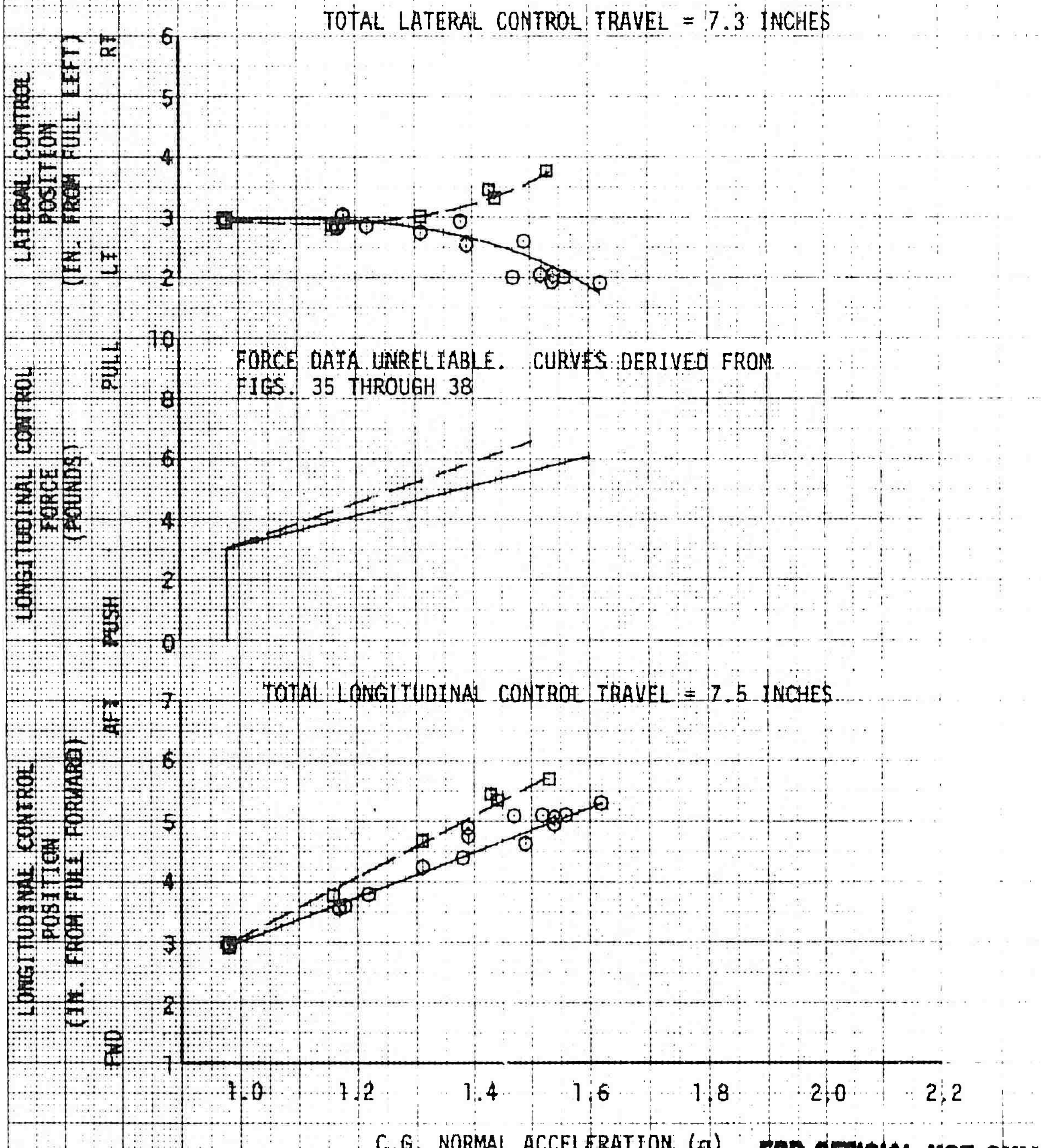

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FIGURE 58
MANEUVERING STABILITY
YAH-63 USA S/N 74-22247

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB A/S (KT)	FLIGHT CONDITION
-○-	14960	298.3(AFT)	-0.2(LT)	6040	20.5	277	97	LT TURN
-□-	15410	298.8(AFT)	-0.2(LT)	5780	21.0	277	95	RT TURN

NOTE: 8-TOW CONFIGURATION

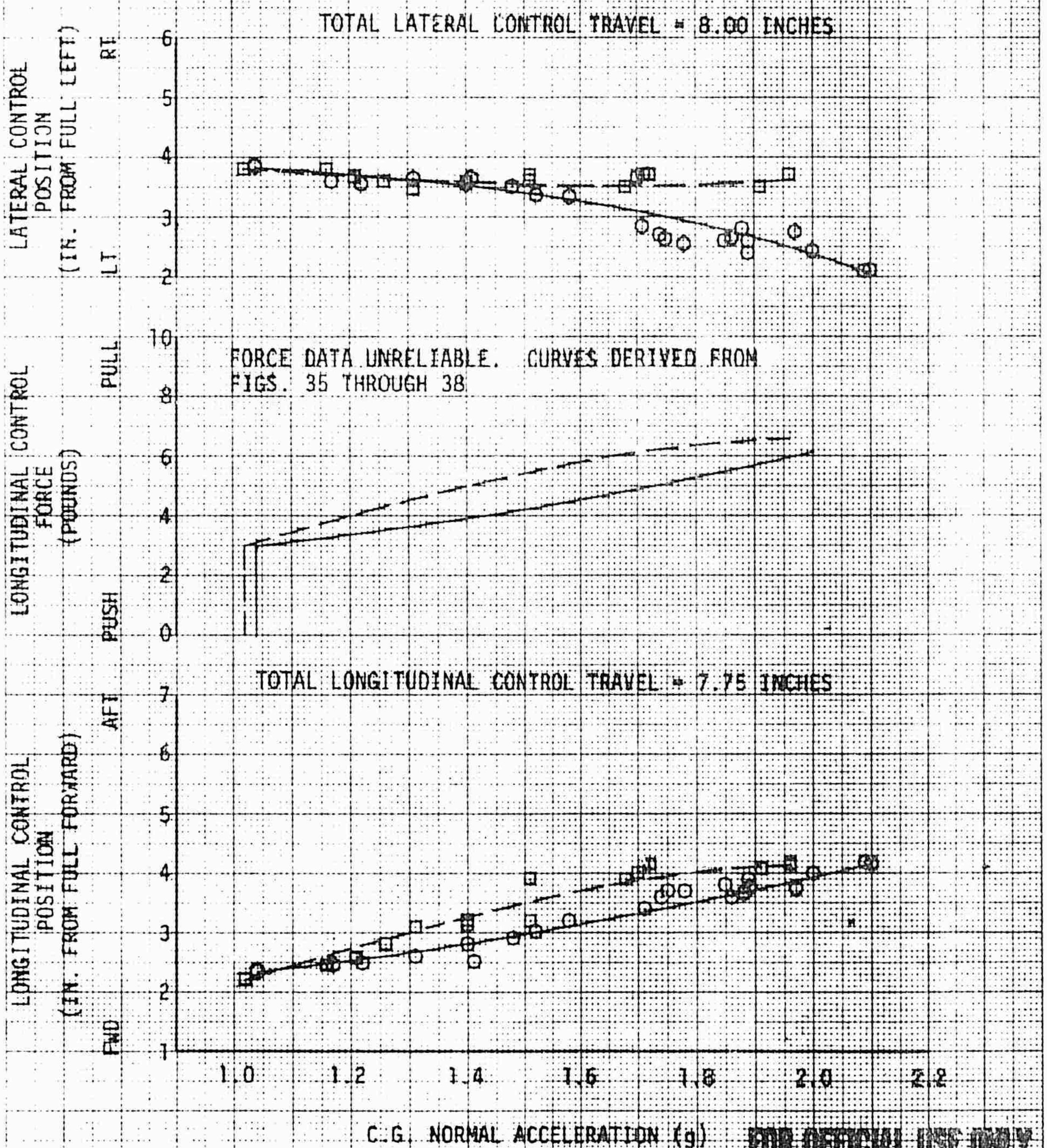

FIGURE 59
MANEUVERING STABILITY
YAH-63 USA S/N 74-22246

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB A/S (KT)	FLIGHT CONDITION
○	16000	298.4(AFT)	-0.1(LT)	6400	15.5	276	97	LT TURN
□	16340	298.8(AFT)	-0.2(LT)	5700	13.5	277	96	RT TURN

NOTE: 1. SQUARES DENOTE DATA GATHERED ON AIRCRAFT S/N 74-22247
2. 8-TOW CONFIGURATION

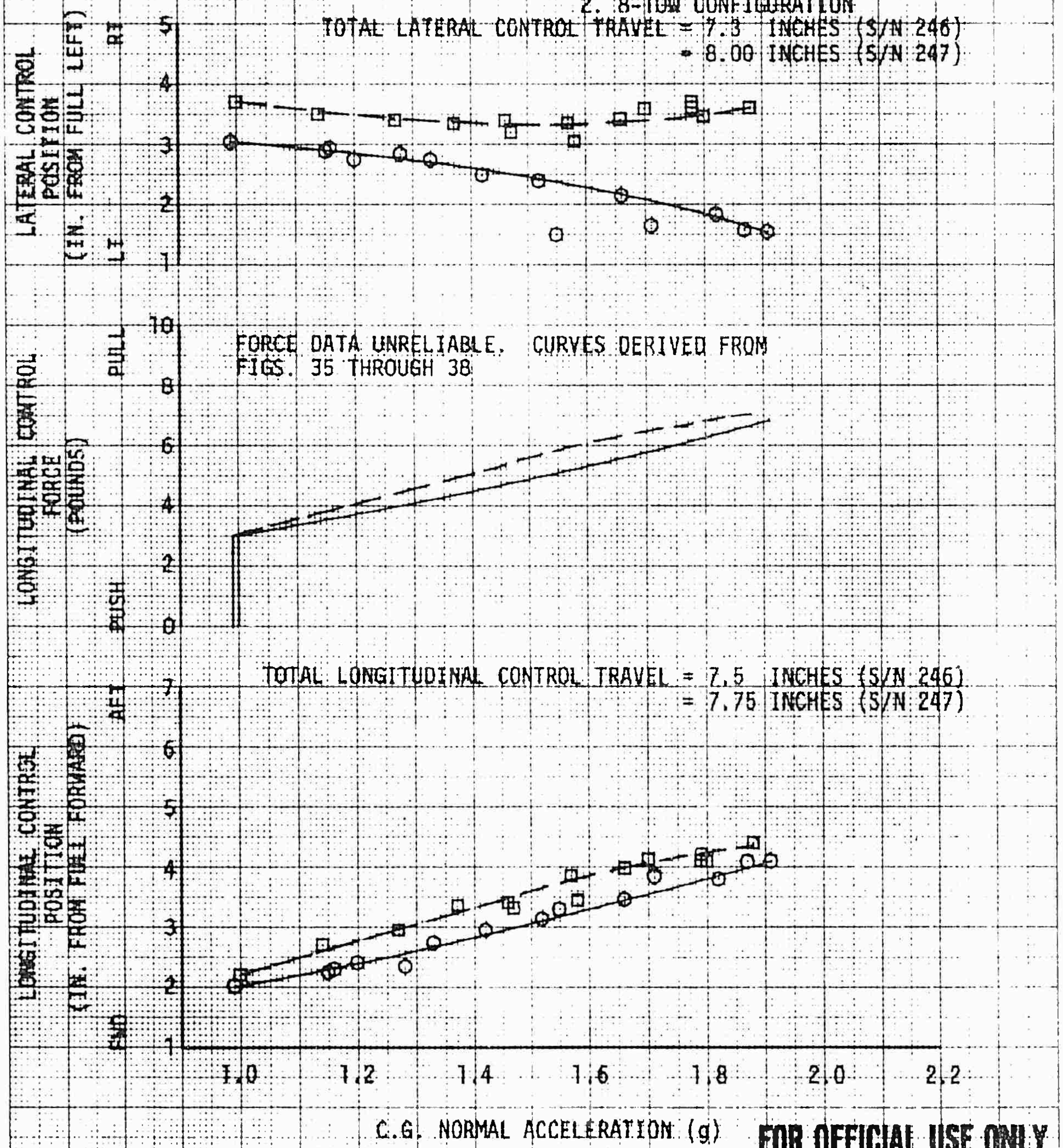

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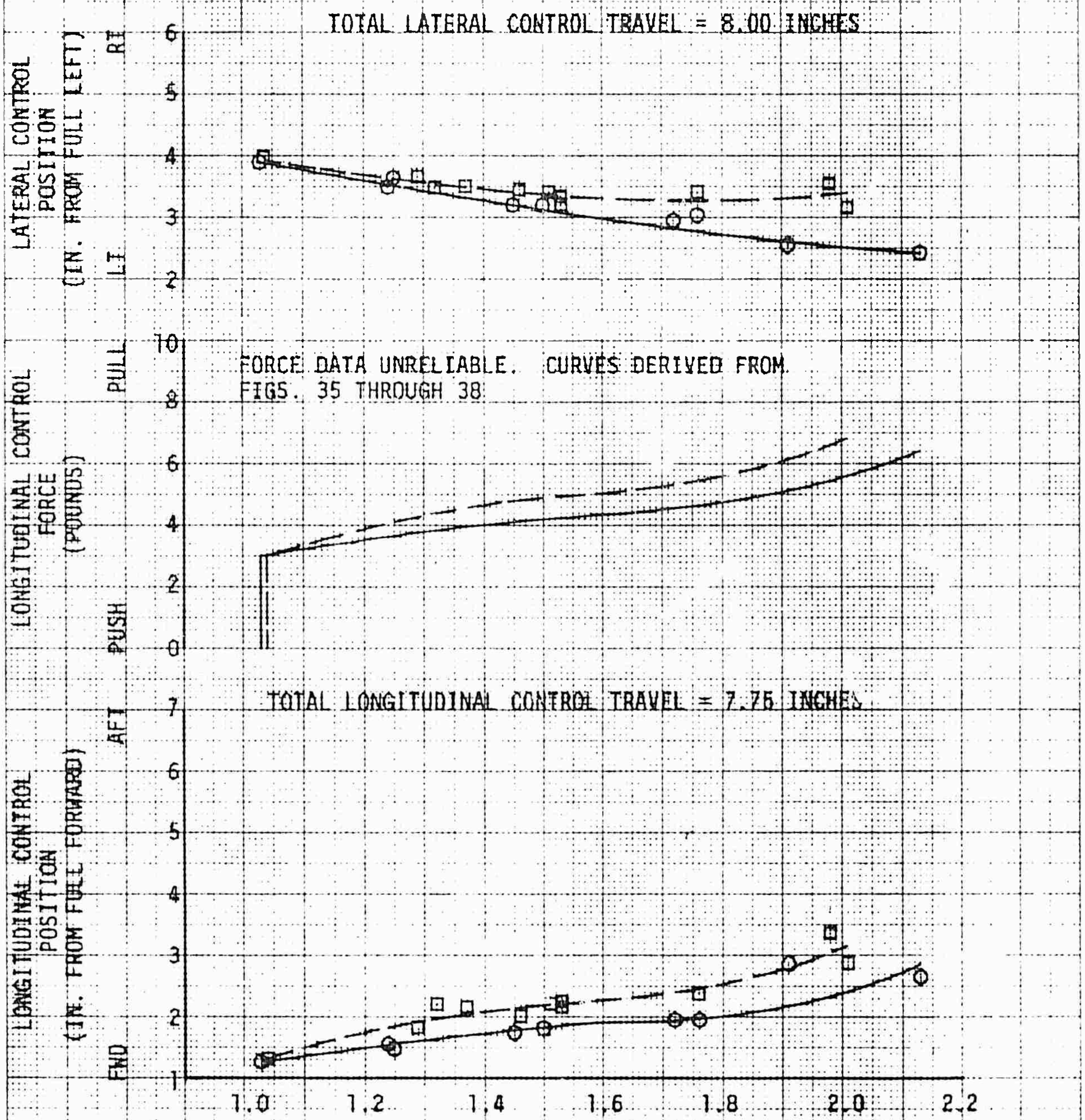
FIGURE 60
MANEUVERING STABILITY
YAH-63 USA S/N 74-22247

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB A/S (KT)	FLIGHT CONDITION
—○—	15180	298.7(AFT)	-0.2(LT)	7000	22.5	277	121	LT TURN
—□—	15500	298.6(AFT)	-0.2(LT)	6820	23.0	277	121	RT TURN

NOTE: 8-TOW CONFIGURATION



C.G. NORMAL ACCELERATION (g)

FIGURE 61
MANEUVERING STABILITY
YAH-63 USA S/N 74-22247

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB A/S (KT)	FLIGHT CONDITION
○	16020	230.7(AFT)	-0.2(LT)	6640	13.0	277	122	LT TURN
□	15780	230.4(AFT)	-0.2(LT)	6000	13.0	277	121	RT TURN

NOTE: 8-TOW CONFIGURATION

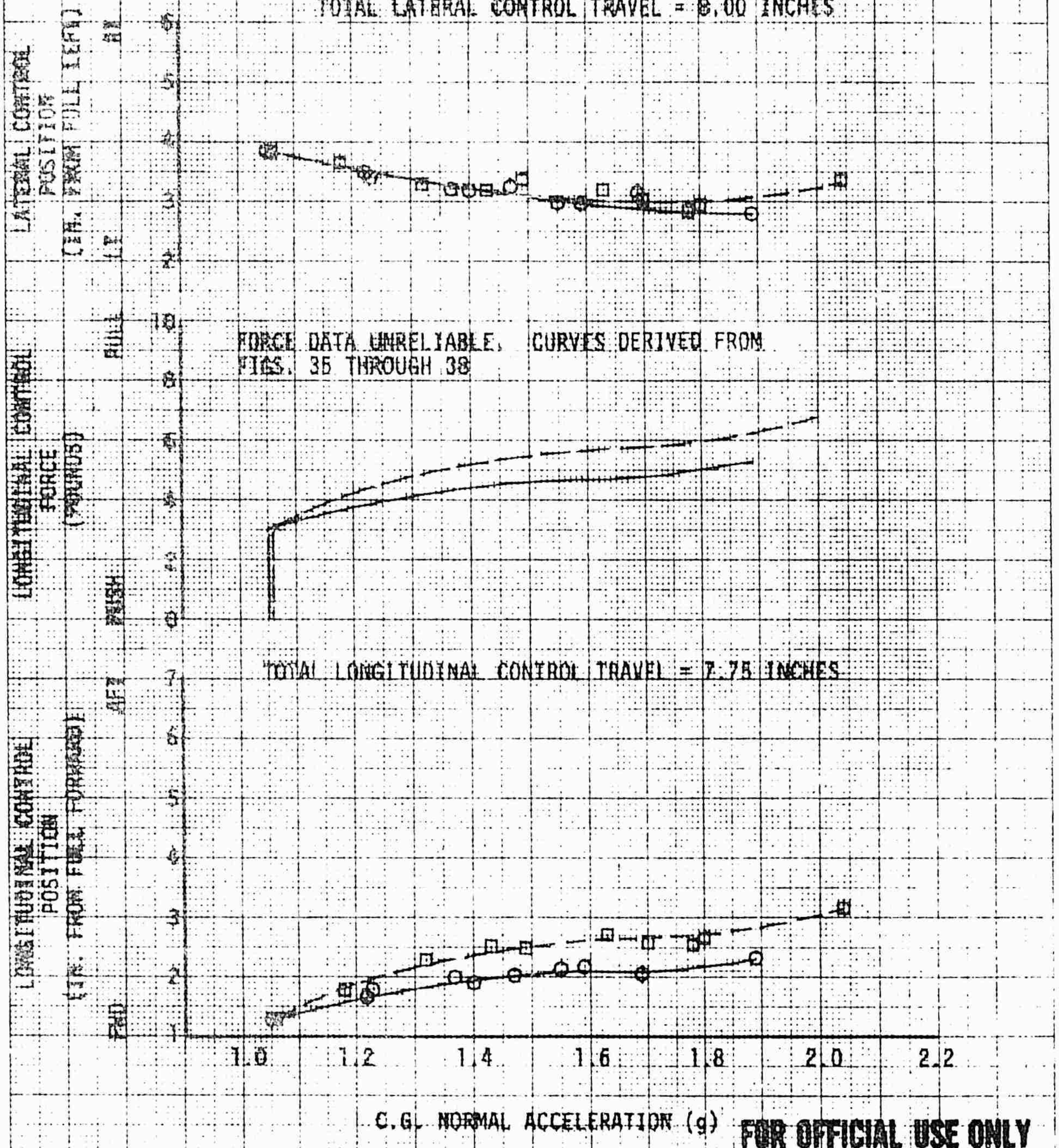


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FIGURE 62

LONGITUDINAL SHORT PERIOD RESPONSE

YCH-53 USA S/N 74-22247

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
16160	298.4 (AFT)	-.2 (LT)	6020	25.5	276	90	ON

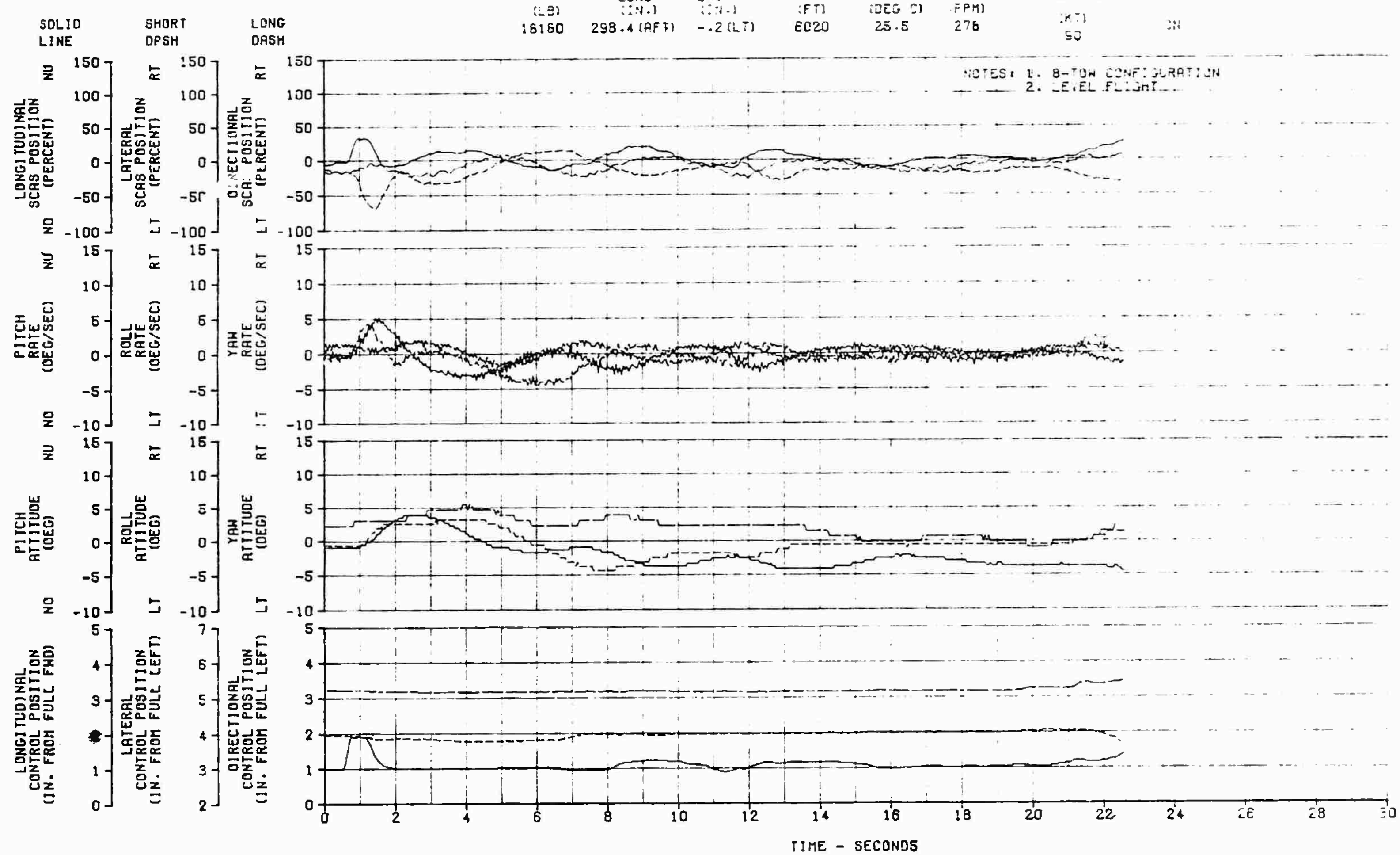

FIGURE 63

SCAS OFF RESPONSE

YAH-63 USA S/N 74-22247

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	ENTRY CALIBRATED AIRSPEED (KT)	SCAS CONDITION
15540	298.1 (AFT)	-.2 (LT)	6340	25.0	278	127	OFF

NOTES: 1. 8-TOW CONFIGURATION
2. LEVEL FLIGHT

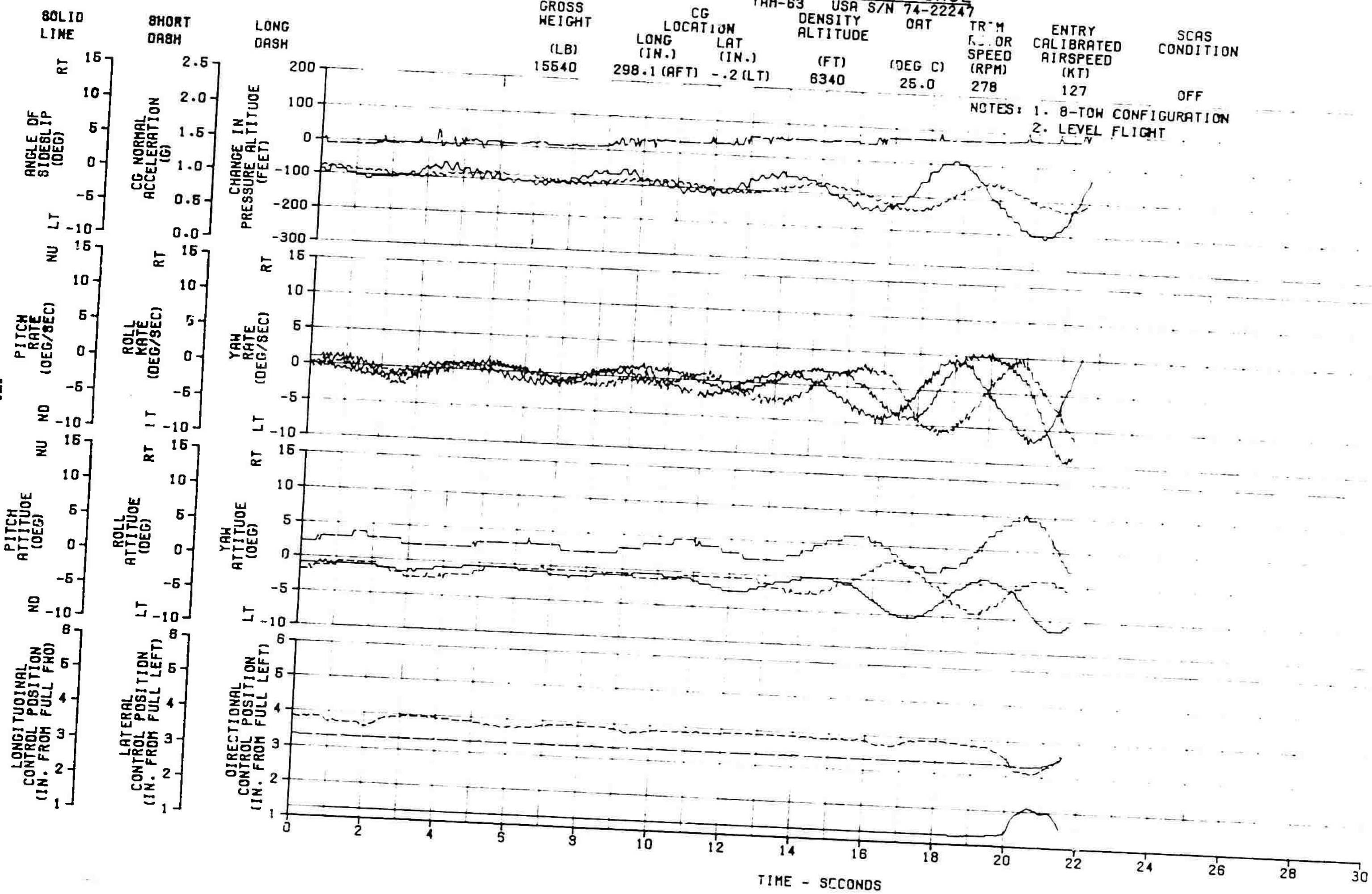

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FIGURE 54

LONGITUDINAL LONG PERIOD RESPONSE

YAH-63 USA S/N 74-22247

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	ENTRY CALIBRATED AIRSPEED (KT)	SCAS CONDITION
15340	298.8 (AFT)	-.2 (LT)	5380	22.0	277	97	ON

NOTES: 1. 8-TOW CONFIGURATION
2. LEVEL FLIGHT

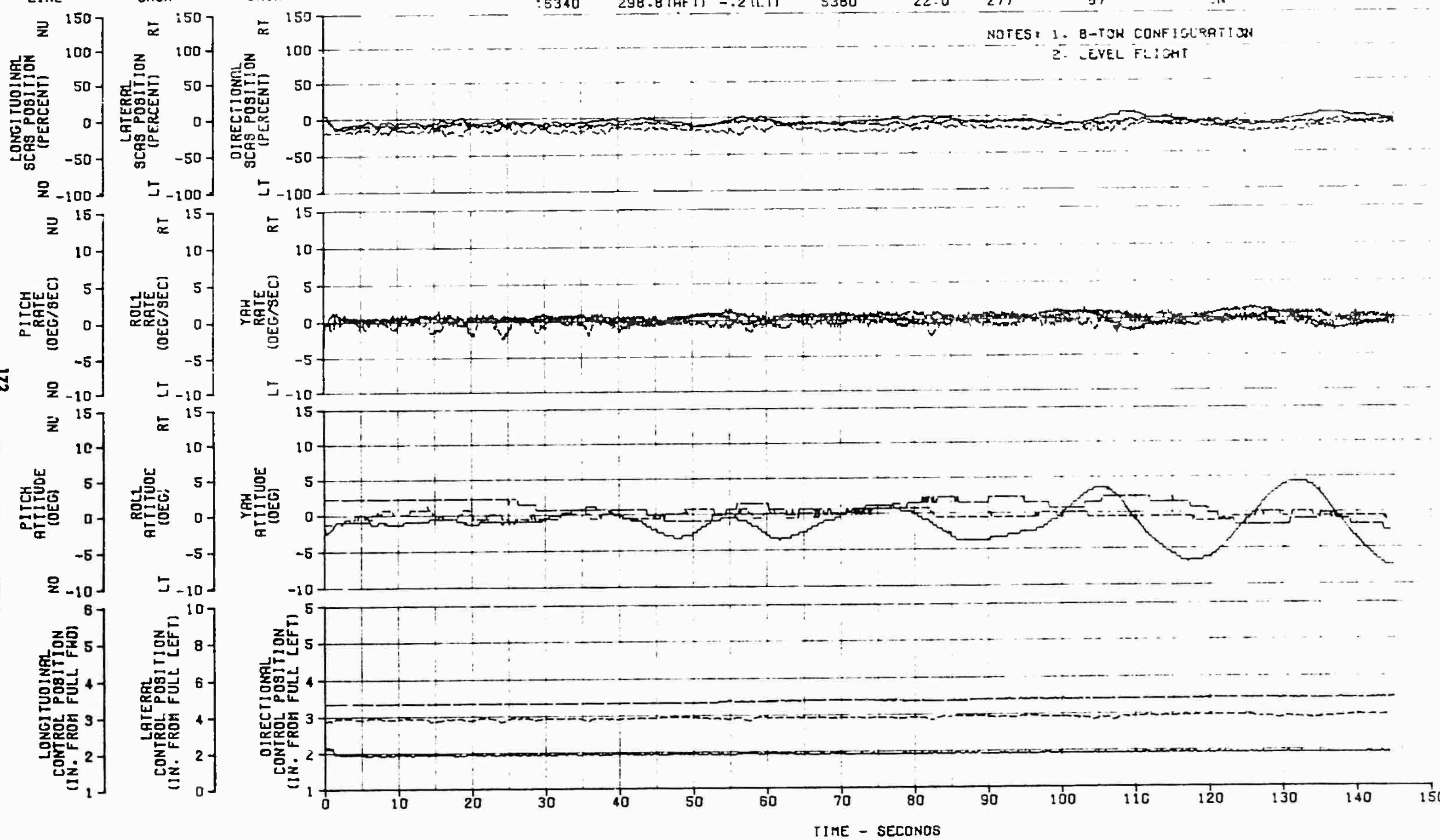

FIGURE 65

SUMMARY LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22246

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
○	16340	298.7(AFT)	-.2(LT)	3760	22.0	276	HOVER	ON
□	16040	298.4(AFT)	-.1(LT)	5880	10.5	276	LEVEL	ON
△	16080	298.7(AFT)	-.2(LT)	6940	27.0	277	LEVEL	ON
⬡	16240	298.4(AFT)	-.2(LT)	6080	26.0	277	LEVEL	ON

NOTES:
1. CURVE DERIVED FROM FIGURES 66 THROUGH 69
2. TOTAL LONGITUDINAL CONTROL TRAVEL = 7.75 INCHES.
3. SHADED SYMBOL DENOTES A/C 74-22246.
4. FLAGGED SYMBOLS DENOTE FORWARD INPUT.
5. 8-TOW CONFIGURATION.
6. A/C 74-22246 LONGITUDINAL CONTROL TRAVEL = 7.5 INCHES.

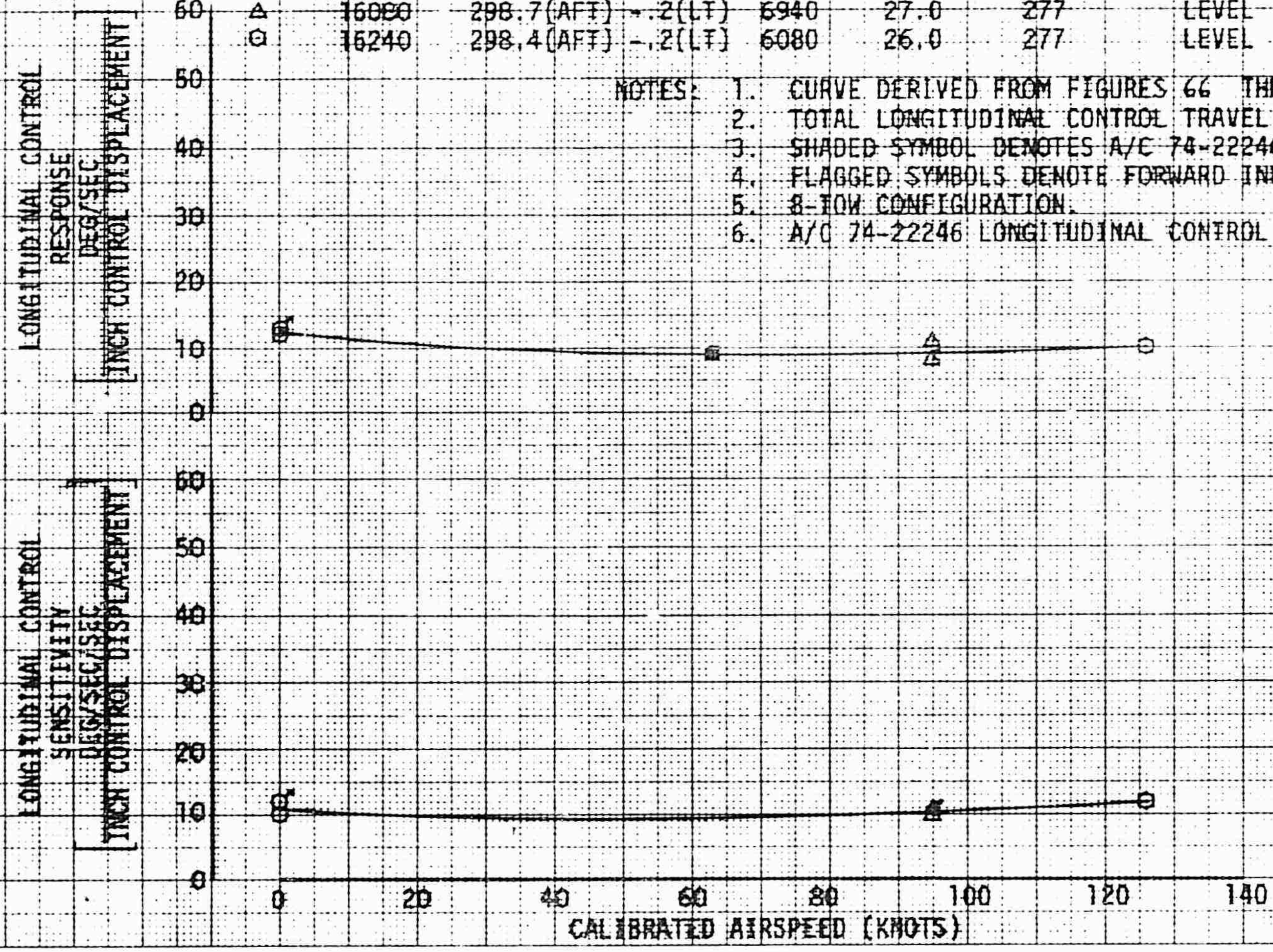

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FIGURE 66
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16340	298.7(AFT)	-.2(LT)	3760	22.0	276	0	HOVER	ON

NOTE: 8 TOW CONFIGURATION.

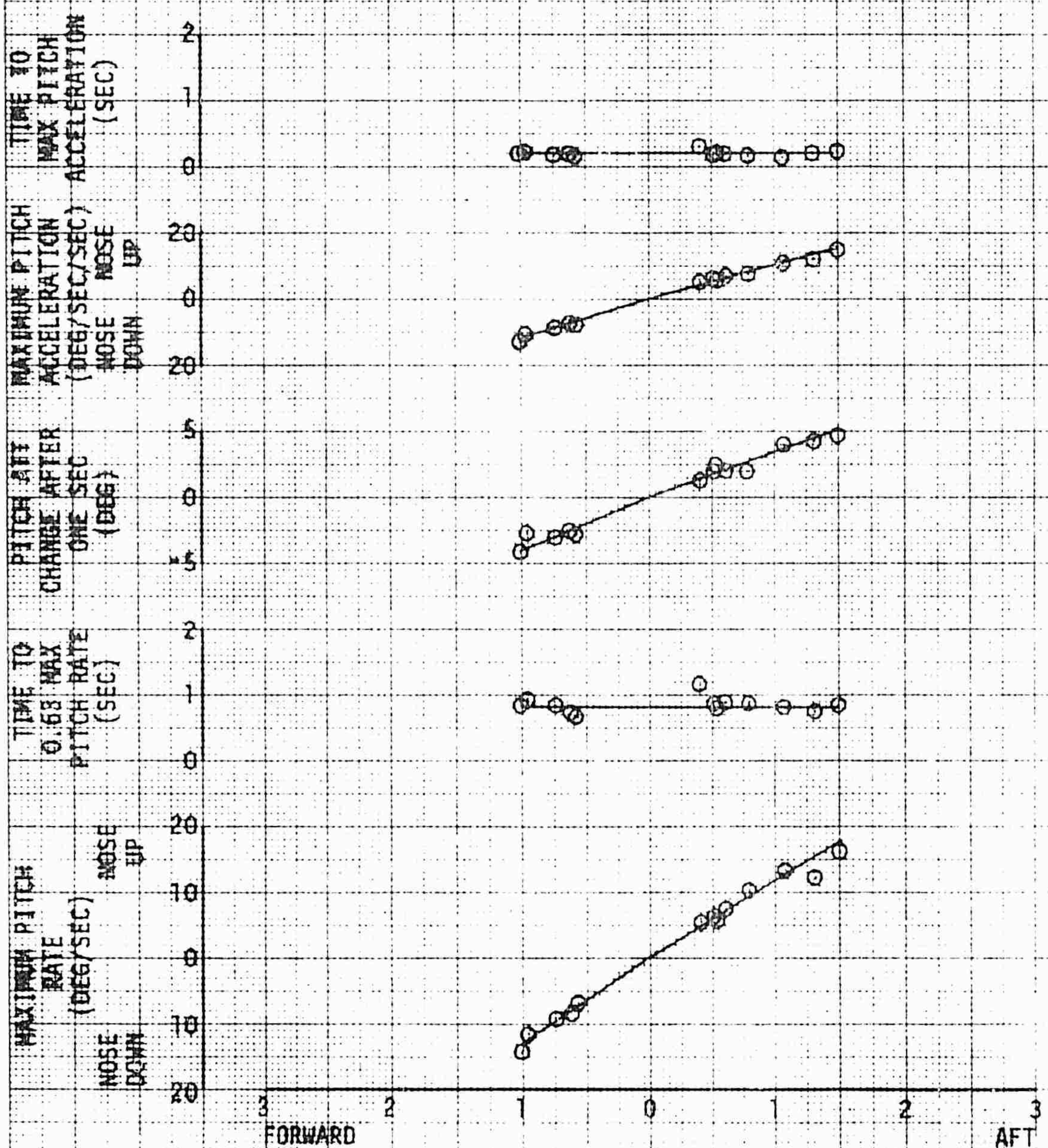

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FIGURE 67

LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16040	298.4(AFT)	-.1(LT)	5880	10.5	276	63	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

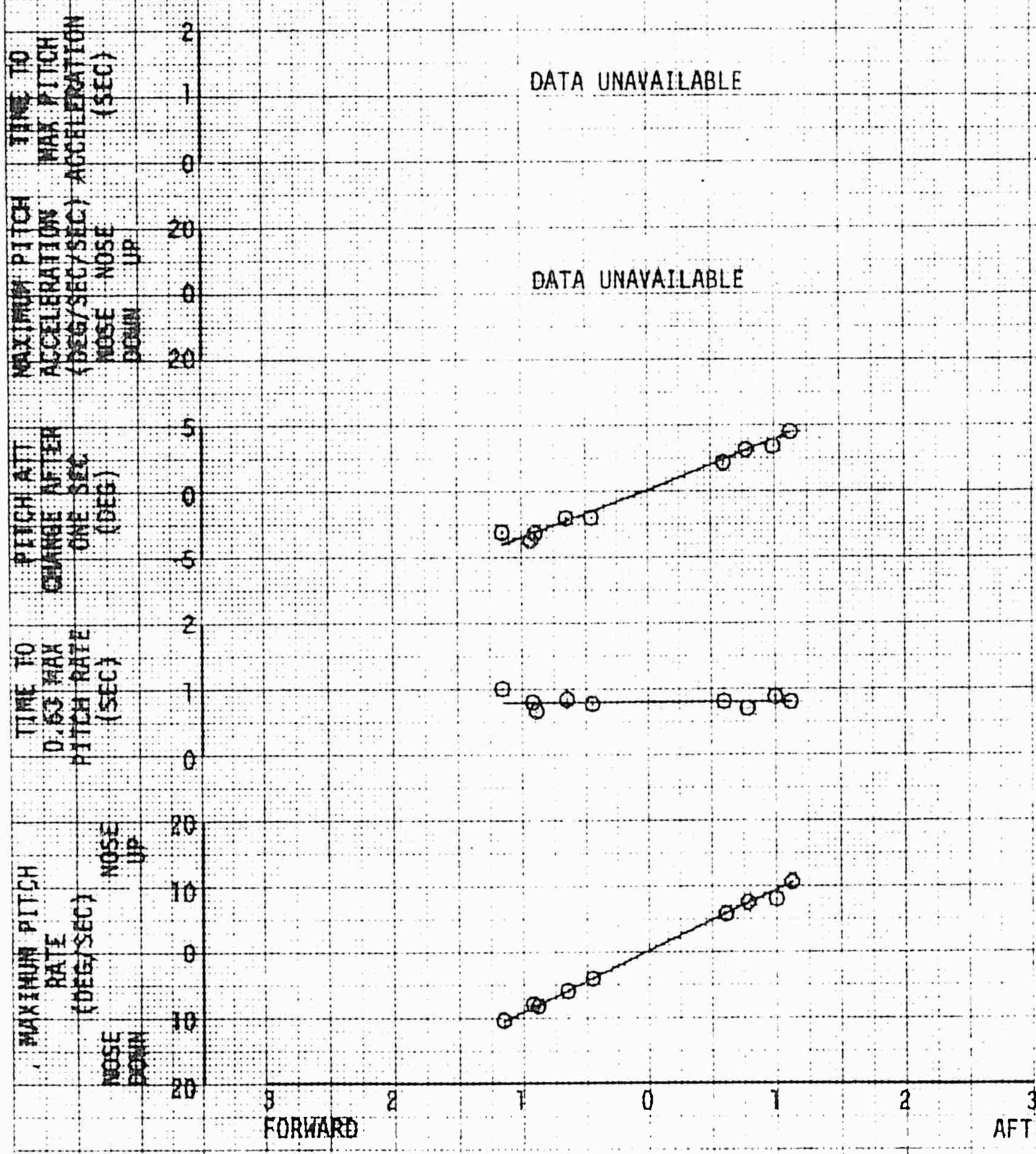

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FIGURE 68
LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16080	298.7 (AFT)	-.2 (LT)	6940	27.0	277	96	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

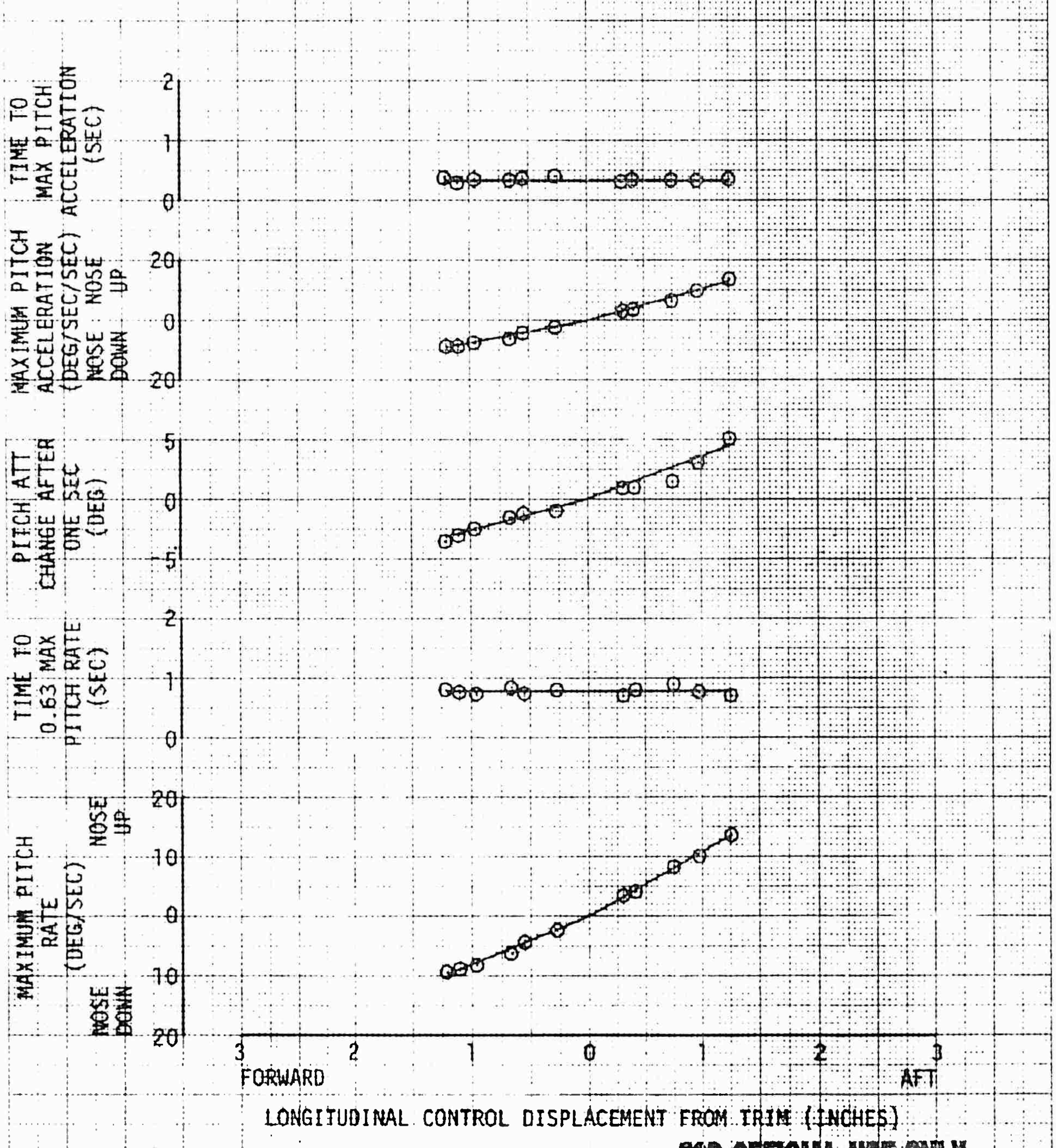

FIGURE 69

LONGITUDINAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16240	298.4(AFT)	-.2(LT)	6080	26.0	277	126	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

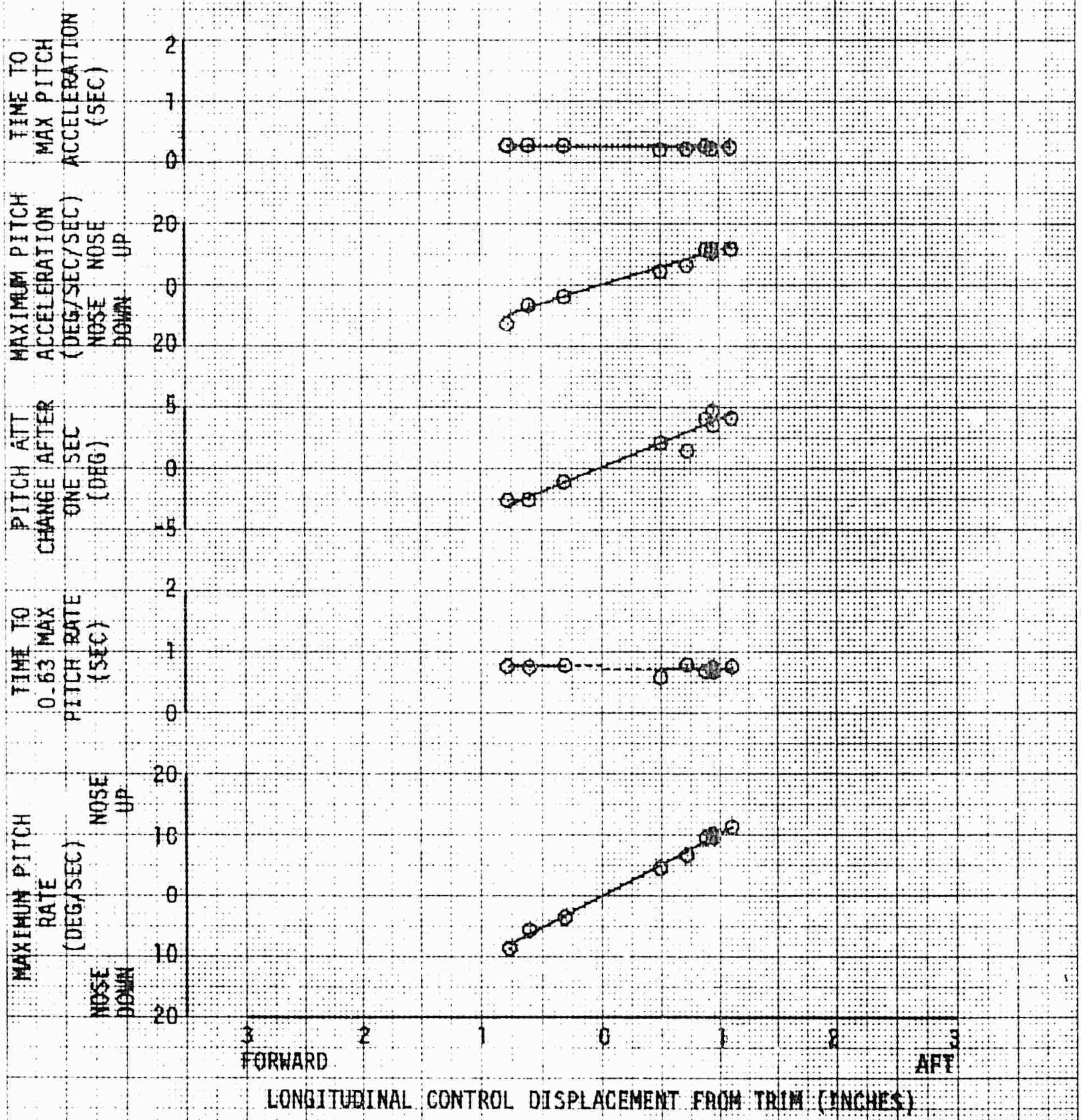

FIGURE 70
SUMMARY LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
○	16040	298.5(AFT)	+.2(LT)	3580	21.0	277	HOVER	ON
□	16440	298.6(AFT)	-.1(LT)	5900	11.0	276	LEVEL	ON
△	15780	298.1(AFT)	-.2(LT)	6640	23.5	279	LEVEL	ON
○	16520	298.5(AFT)	-.2(LT)	6180	25.5	276	LEVEL	ON

NOTES:
1. CURVE DERIVED FROM FIGURES 71 THROUGH 74
2. TOTAL LATERAL CONTROL TRAVEL = 8.00 INCHES.
3. SHADED SYMBOL DENOTES A/C 74-22246.
4. FLAGGED SYMBOLS DENOTE LEFT INPUT.
5. 8 TOW CONFIGURATION.
6. A/C 74-22246 LATERAL CONTROL TRAVEL = 7.3 INCHES.

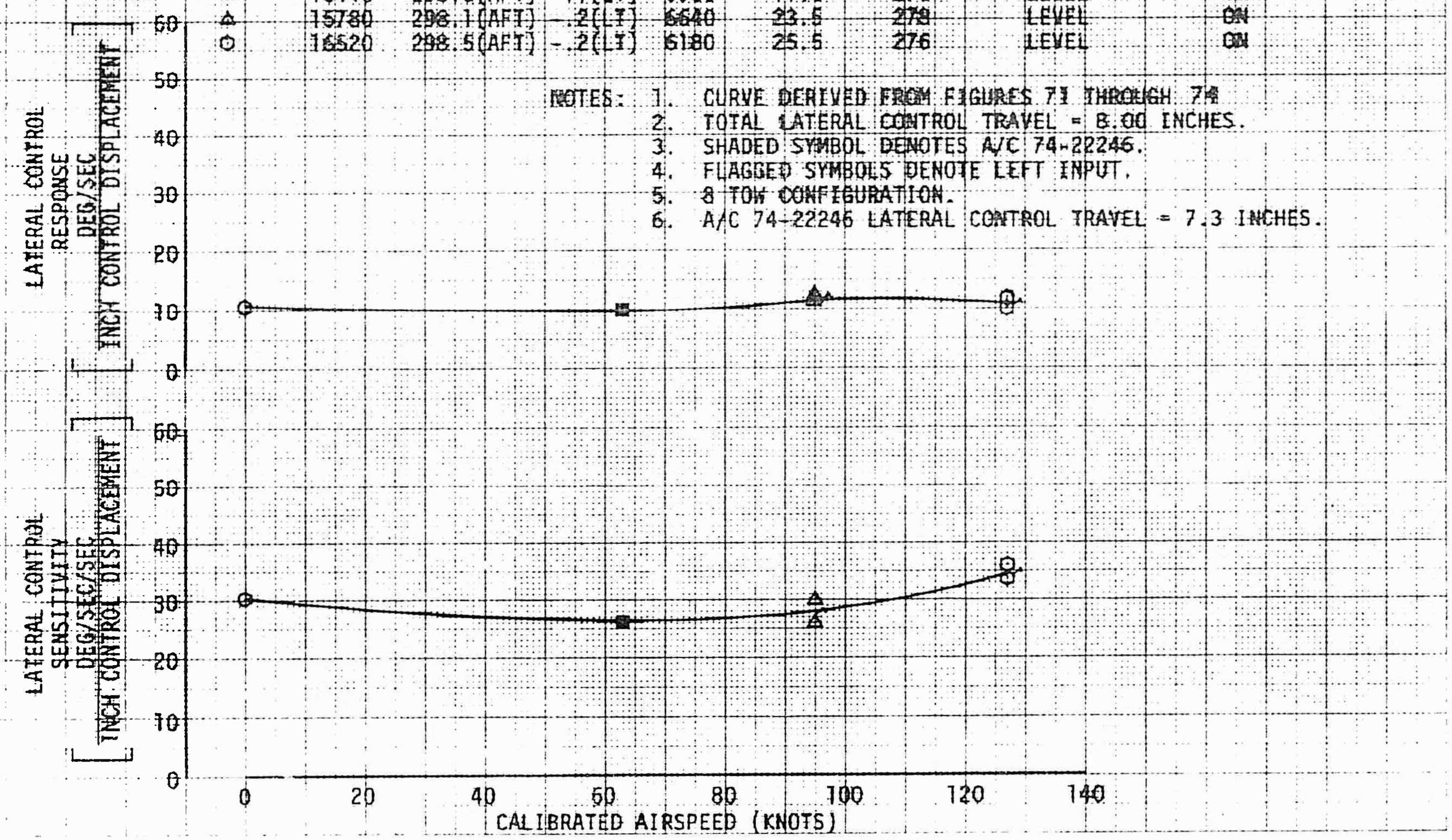

FIGURE 71
LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16040	298.5(AFT)	-.2(LT)	3580	21.0	277	0	HOVER	ON

NOTE: 8 TOW CONFIGURATION.

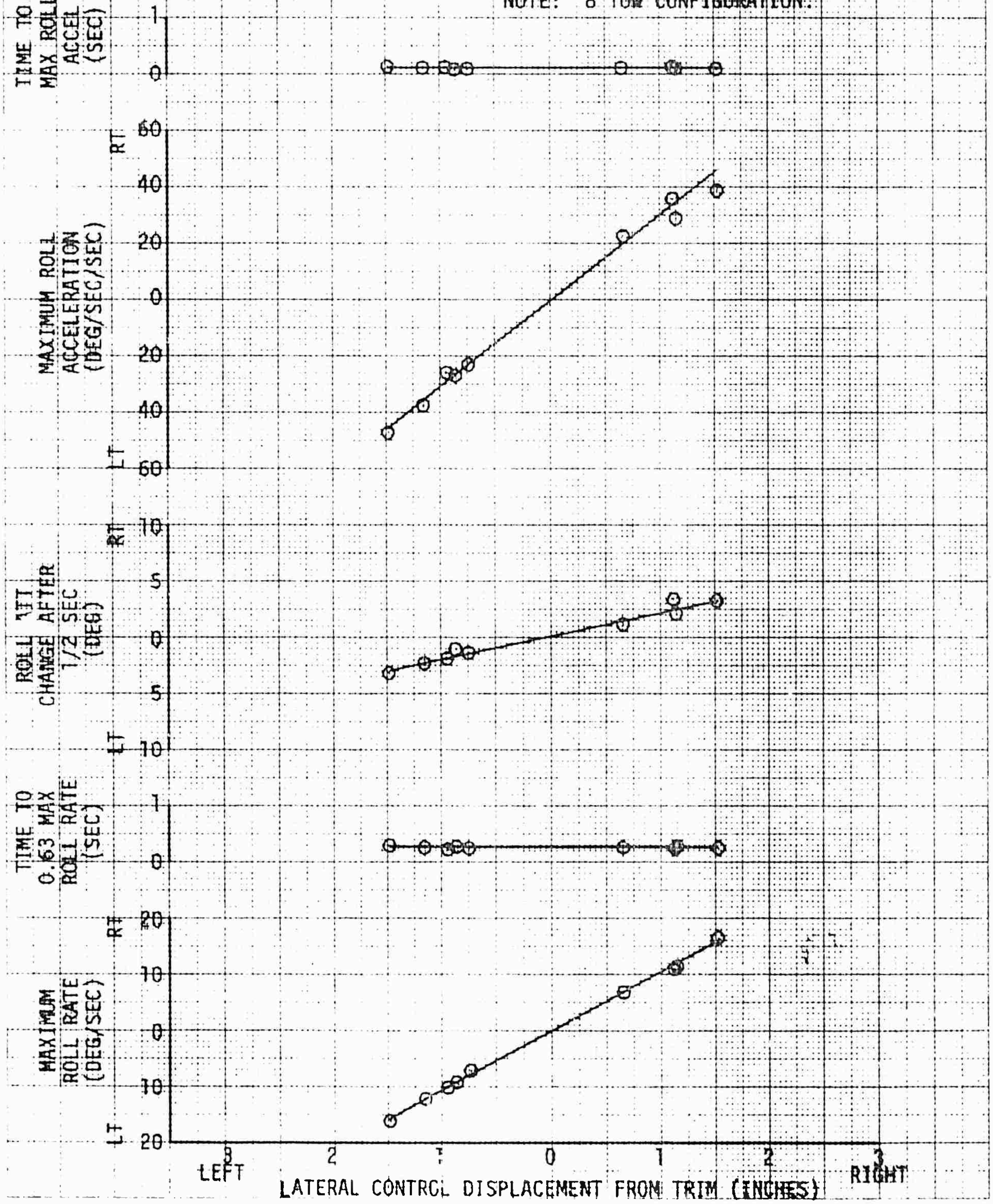

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FIGURE 72
LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16440	298.6(AFT)	-.1(LT)	5900	11.0	276	63	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

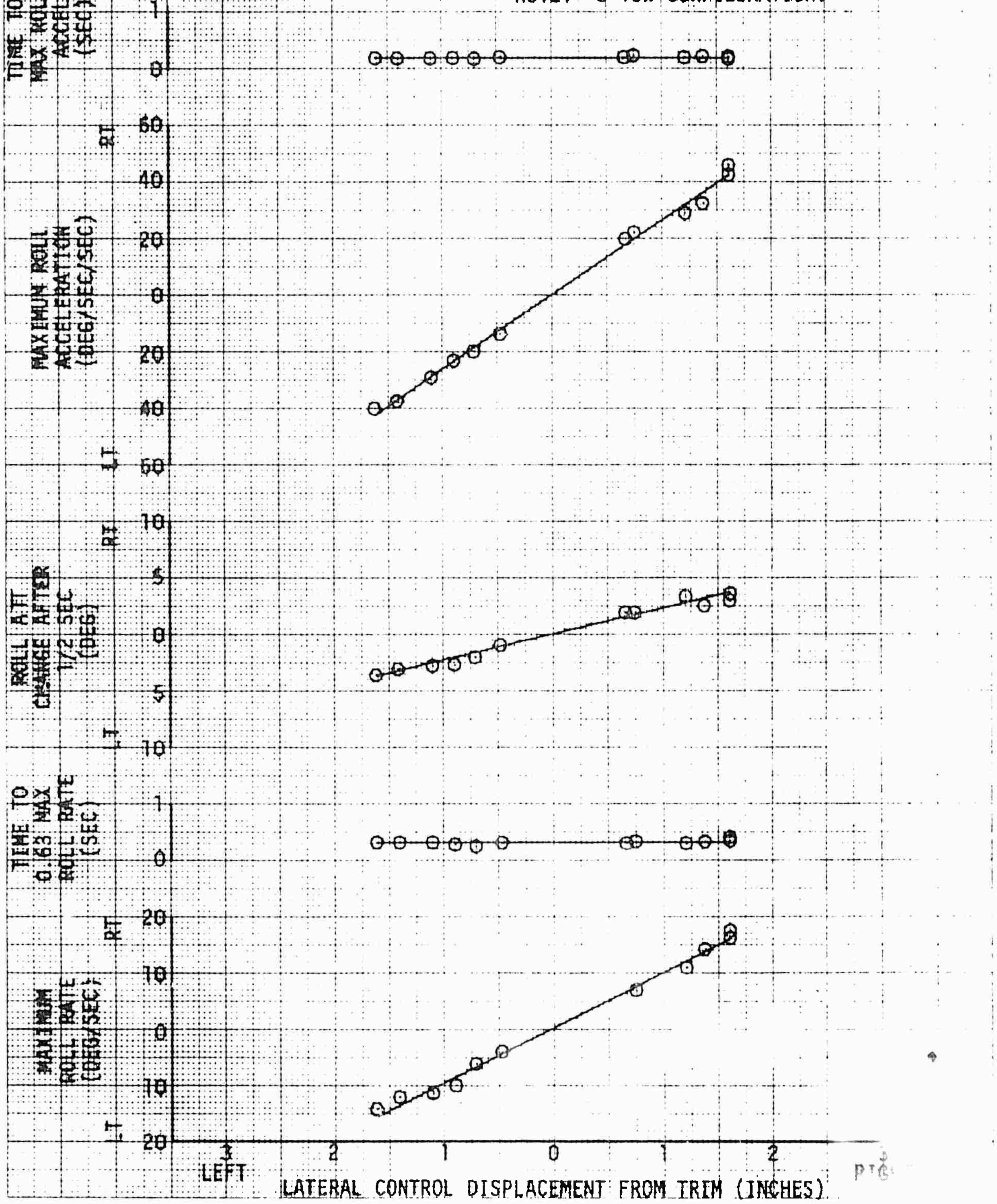

FIGURE 73
LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
15780	298.1(AFT)	-.2(LT)	6640	23.5	278	95	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

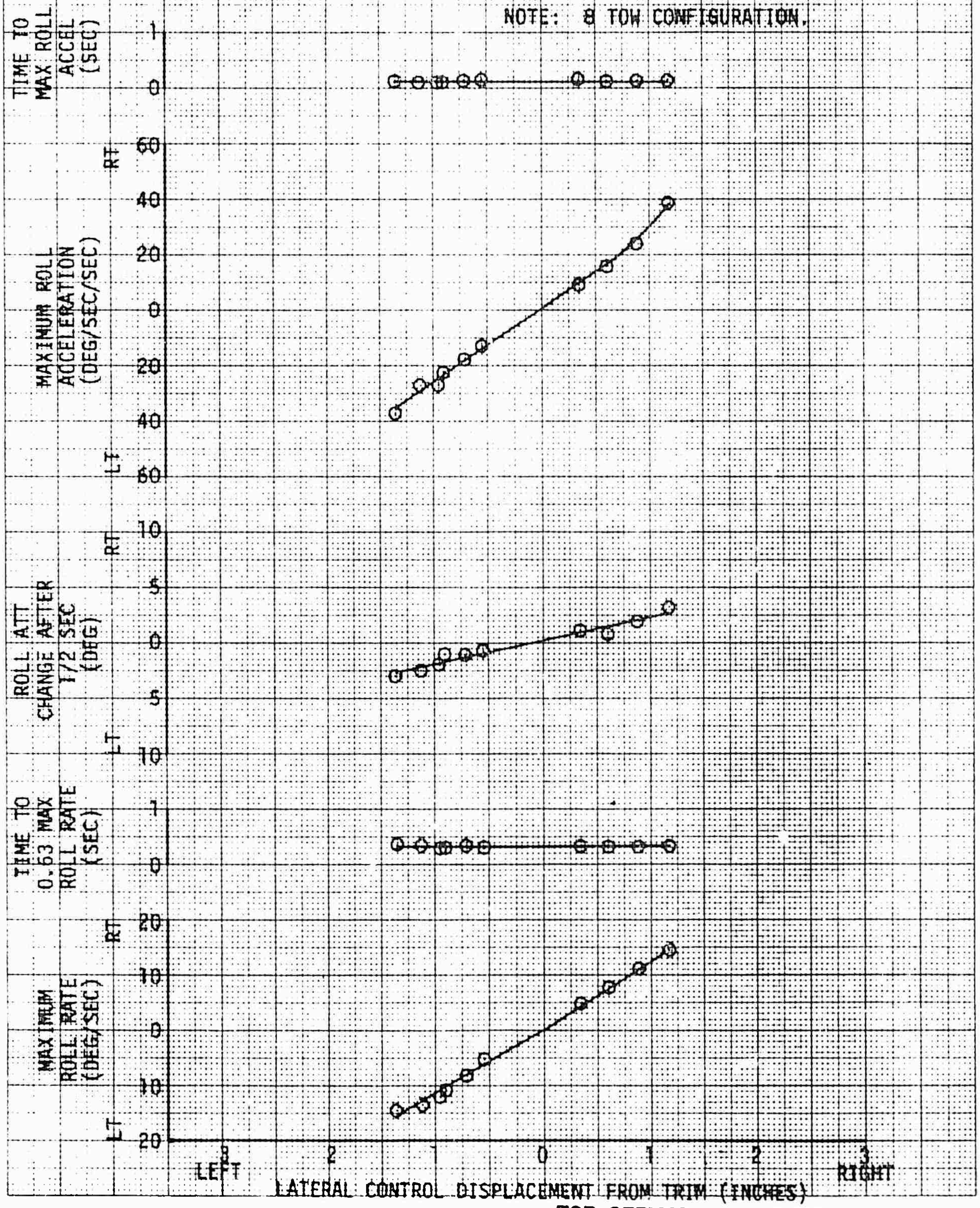

FIGURE 74
LATERAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	TRIM FLIGHT CONDITION	SCAS CONDITION
16520	298.5(AFT)	.2(LT)	6180	25.5	276	127	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

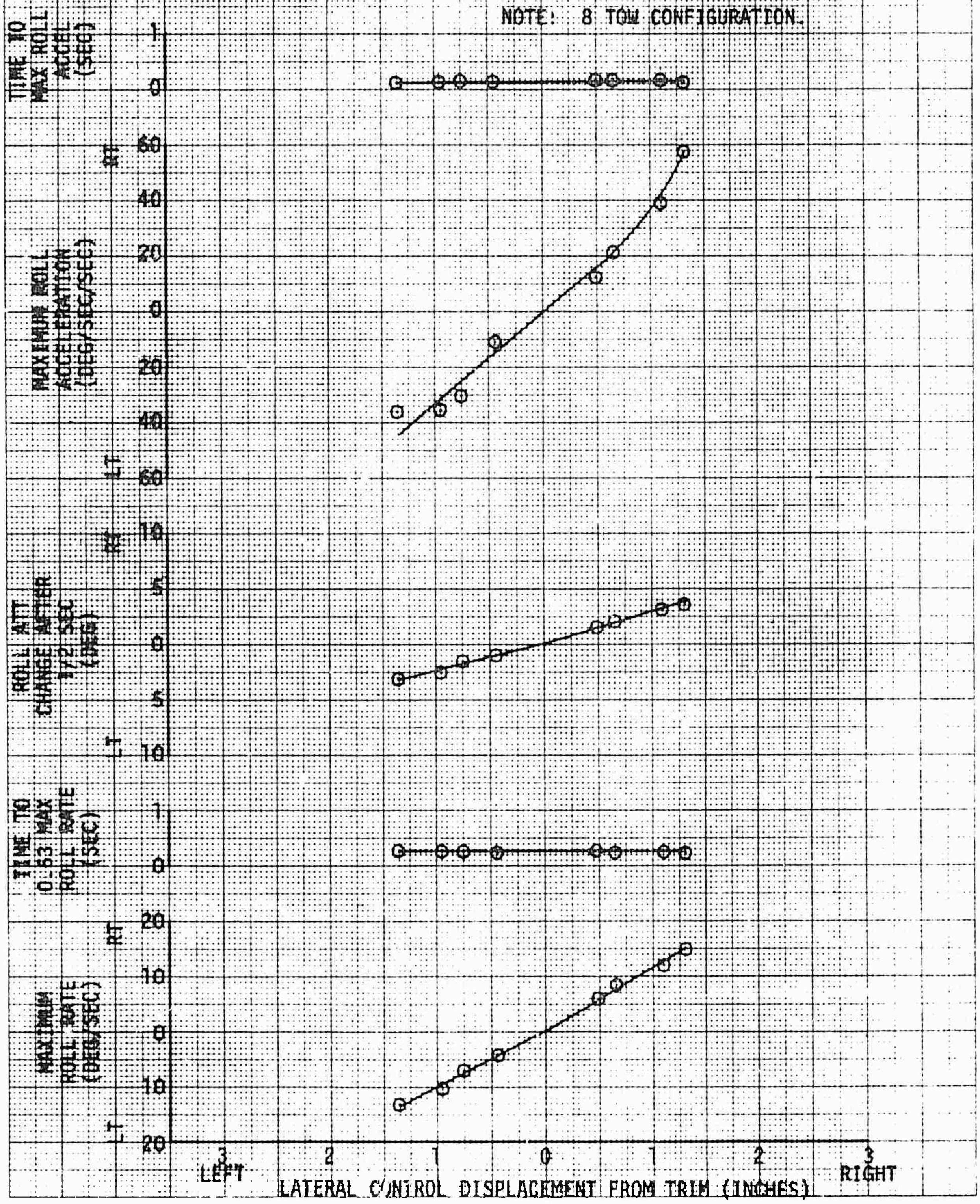

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FIGURE 75

SUMMARY DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22247

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM FLIGHT CONDITION	SCAS CONDITION
⊖	15540	298.4(AFT)	-.2(LT)	3520	21.5	276	HOVER	ON
◻	16260	298.5(AFT)	-.1(LT)	5320	11.5	276	LEVEL	ON
△	15640	298.3(AFT)	-.2(LT)	7260	27.0	277	LEVEL	ON
○	16000	298.3(AFT)	-.2(LT)	5940	25.0	277	LEVEL	ON
◇	15780	297.3(AFT)	-.1(LT)	10840	9.0	274	HOVER	ON

NOTES:
1. HOVER RESPONSE RATE DETERMINED AFTER ONE SECOND.
2. CURVE DERIVED FROM FIGURES 76 THROUGH 80
3. TOTAL DIRECTIONAL CONTROL TRAVEL = 6.25 INCHES.
4. SHADED SYMBOLS DENOTE A/C 74-22246.
5. FLAGGED SYMBOLS DENOTE LEFT INPUT.
6. 8-TOW CONFIGURATION.
7. A/C 74-22246 DIRECTIONAL CONTROL TRAVEL = 6.6 INCHES.

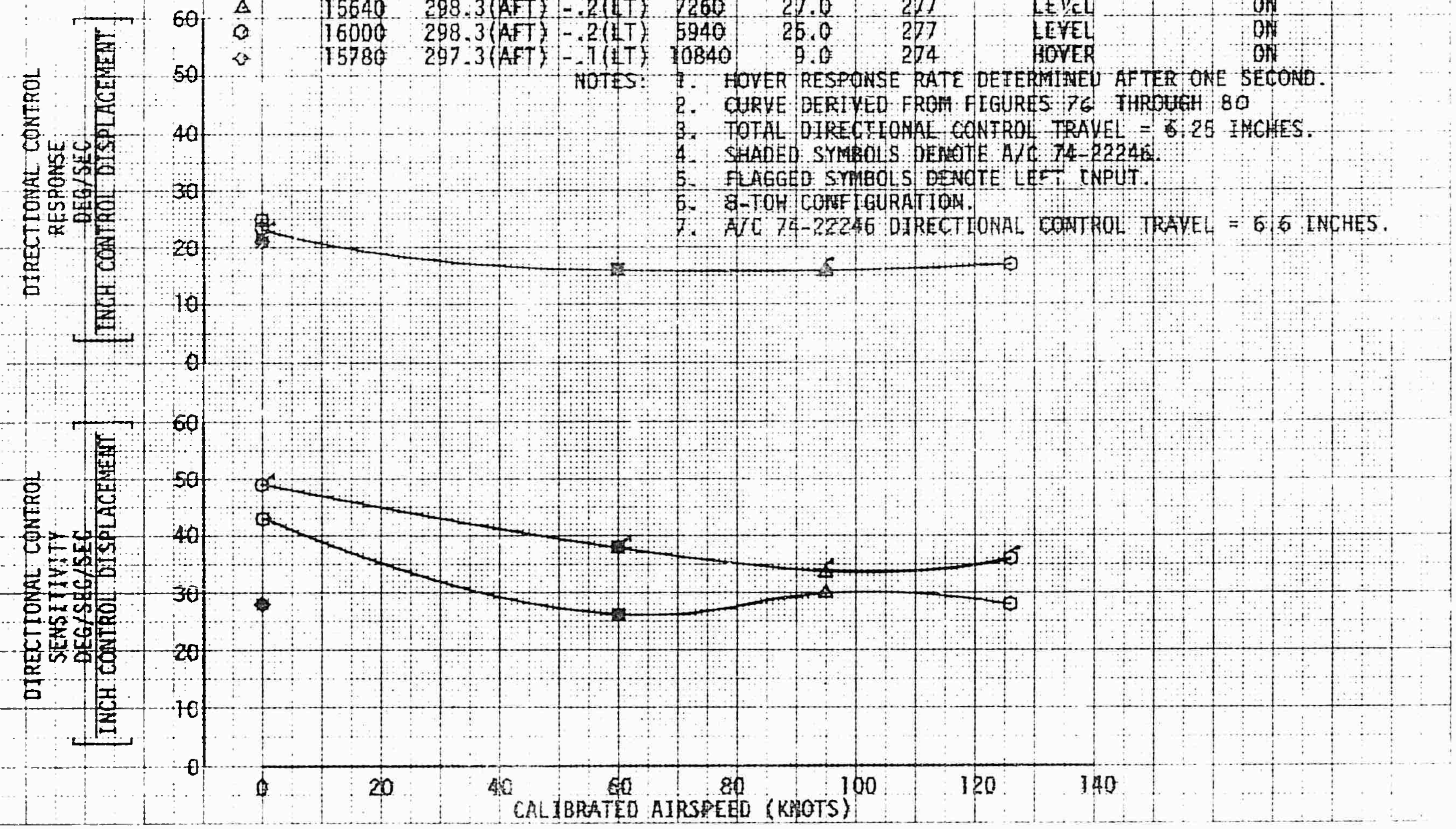

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FIGURE 76
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION	SCAS CONDITION
15840	298.4(AFT)	-.2(LT)	3520	21.5	276	0	HOVER	ON

NOTE: 8 TOW CONFIGURATION.

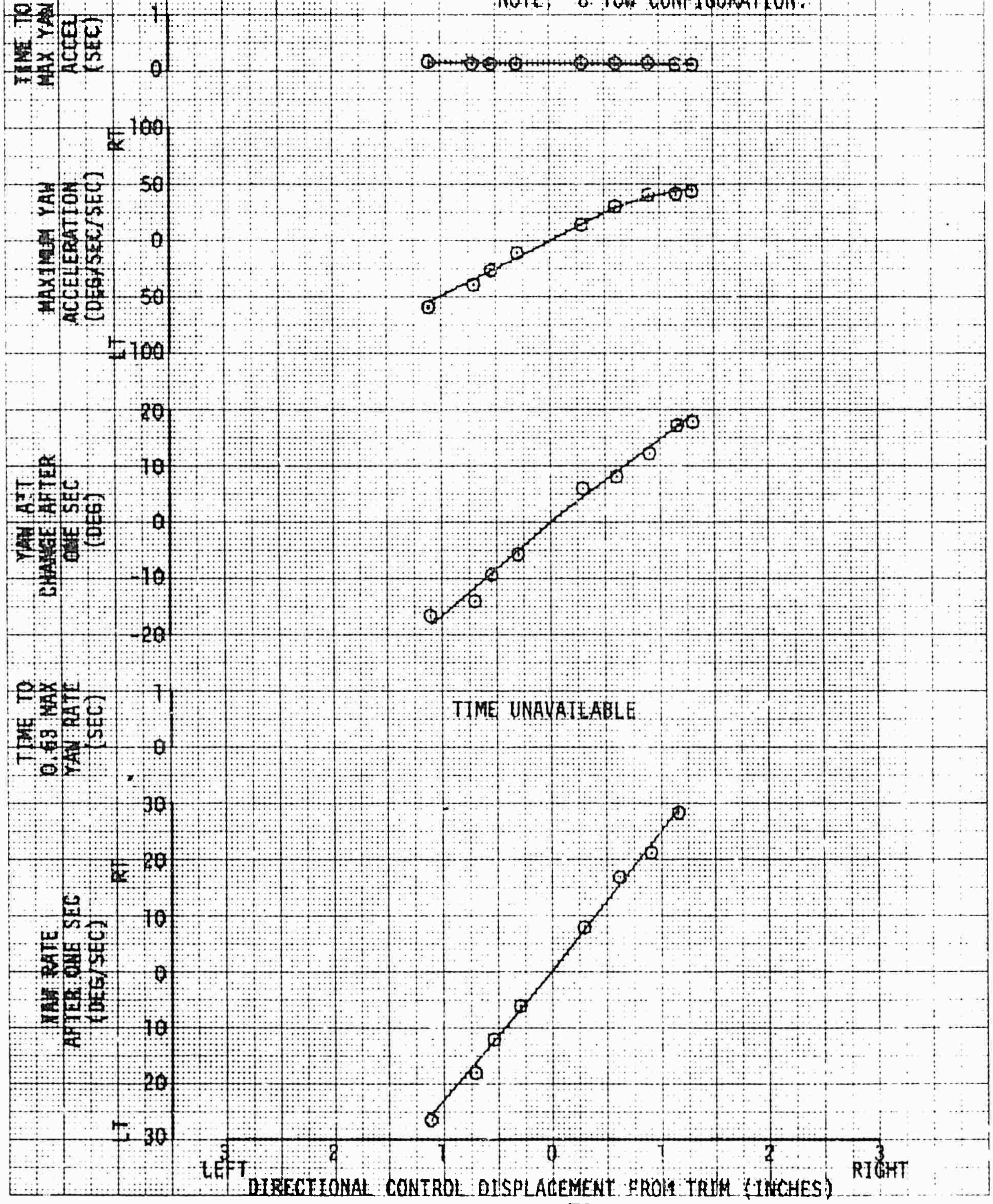

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FIGURE 77

DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION	SCAS CONDITION
15740	297.3(AFT)	0.1(LT)	10880	9.0	274	0	HOVER	ON

NOTE: 8 TOW CONFIGURATION

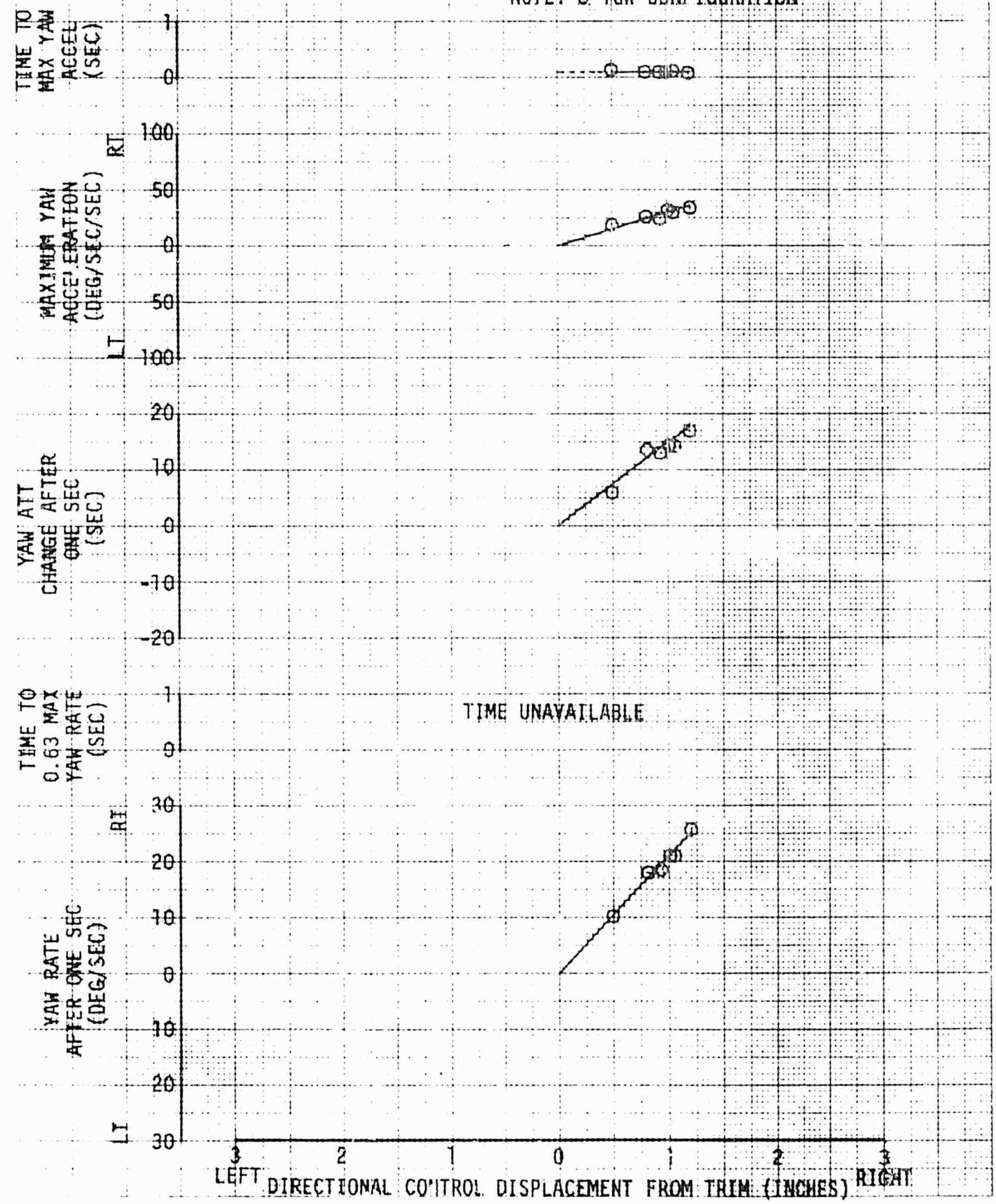

FIGURE 78

DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION	SCAS CONDITION
16260	298.5(AFT)	-.1(LT)	5320	11.5	276	60	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

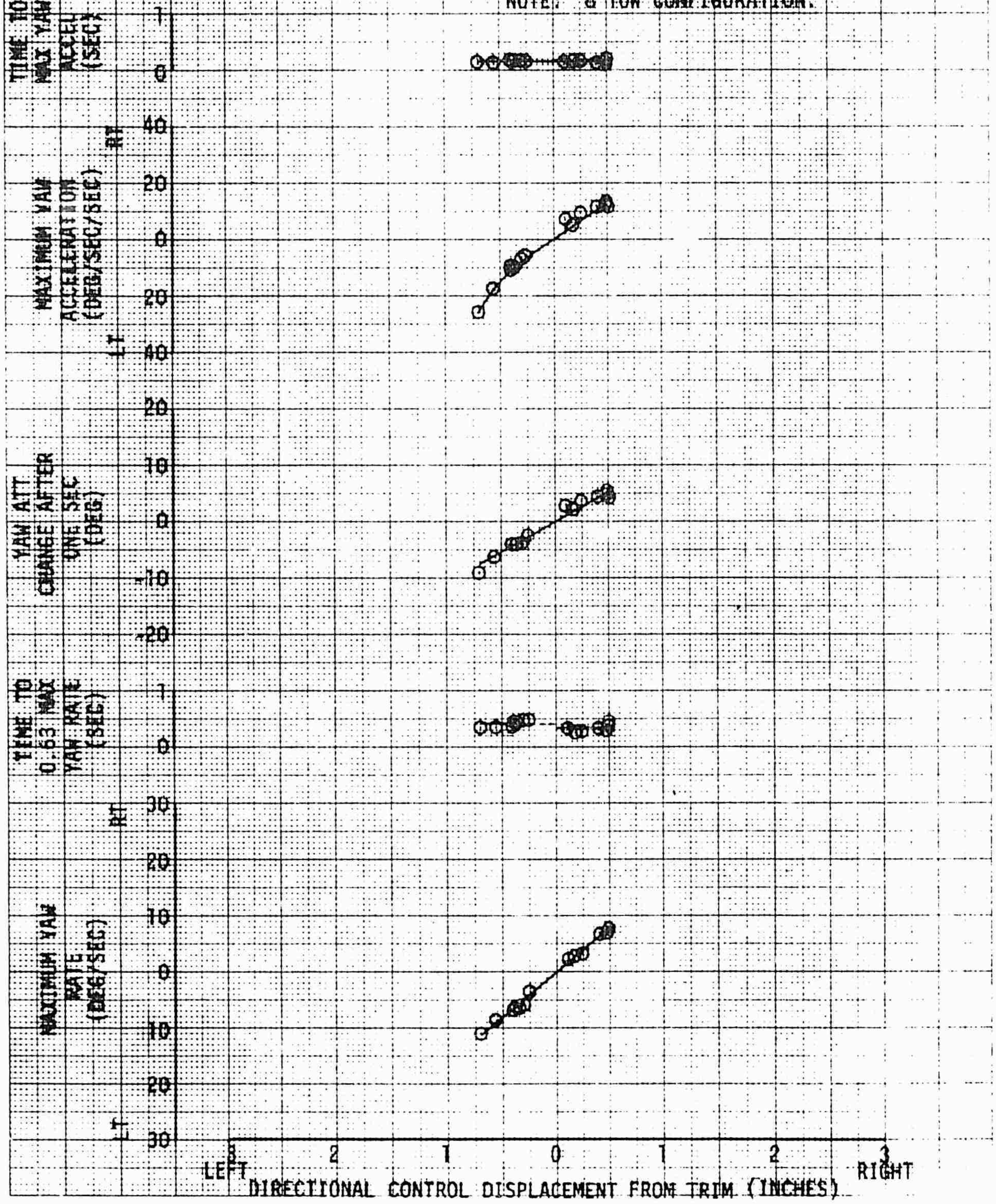

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FIGURE 79

DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY

YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION	SCAS CONDITION
15640	298.3(AFT)	-.2(LT)	7260	27.0	277	95	LEVEL	ON

NOTE: 8 TOW CONFIGURATION.

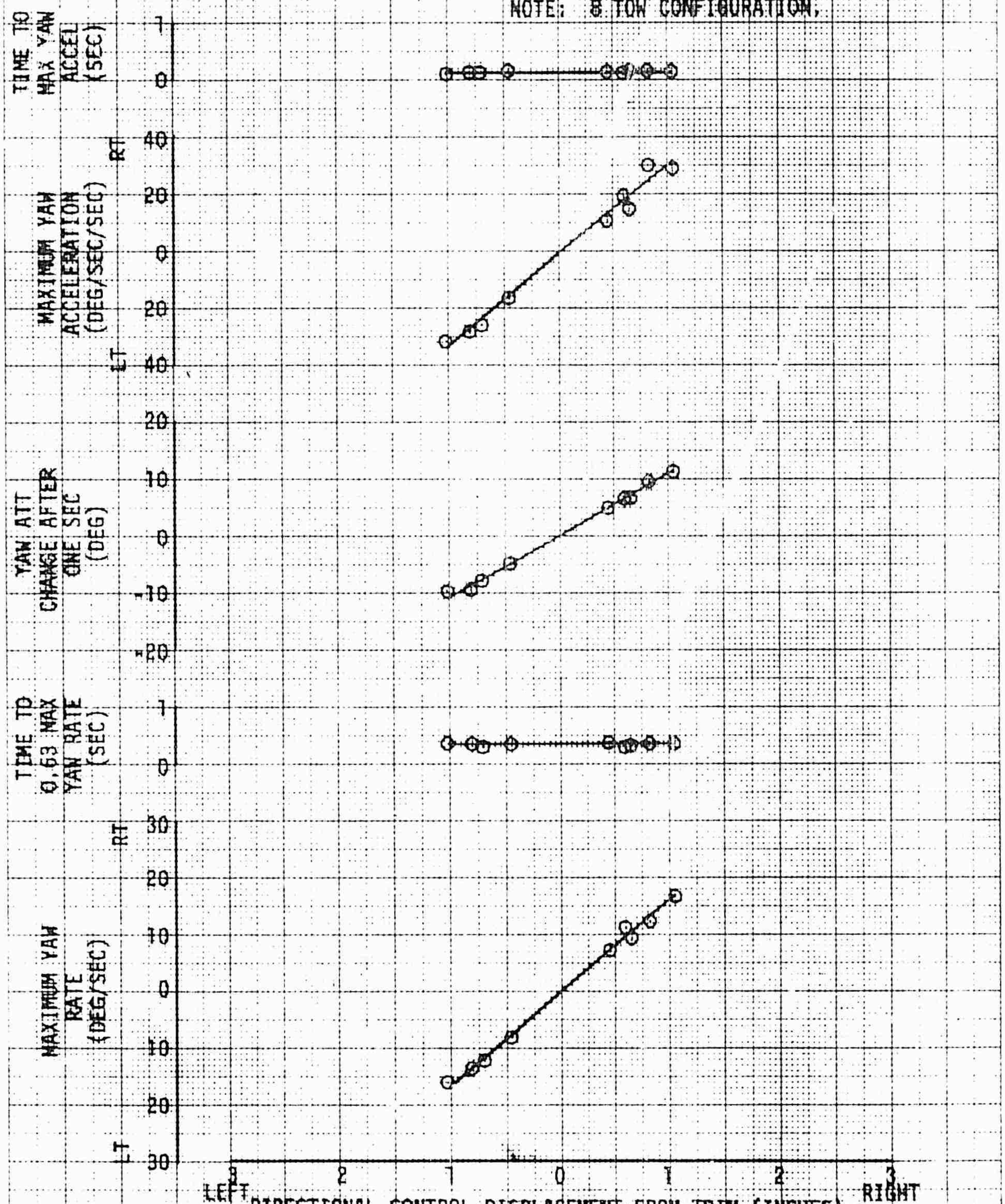

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FIGURE 80
DIRECTIONAL CONTROL RESPONSE AND SENSITIVITY
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	TRIM CALIB AIRSPEED (KT)	FLIGHT CONDITION	SCAS CONDITION
16000	299.3(AFT)	-.2(LT)	5940	25.0	277	126	LEVEL	ON

NOTE: 8 TOW CONFIGURATION

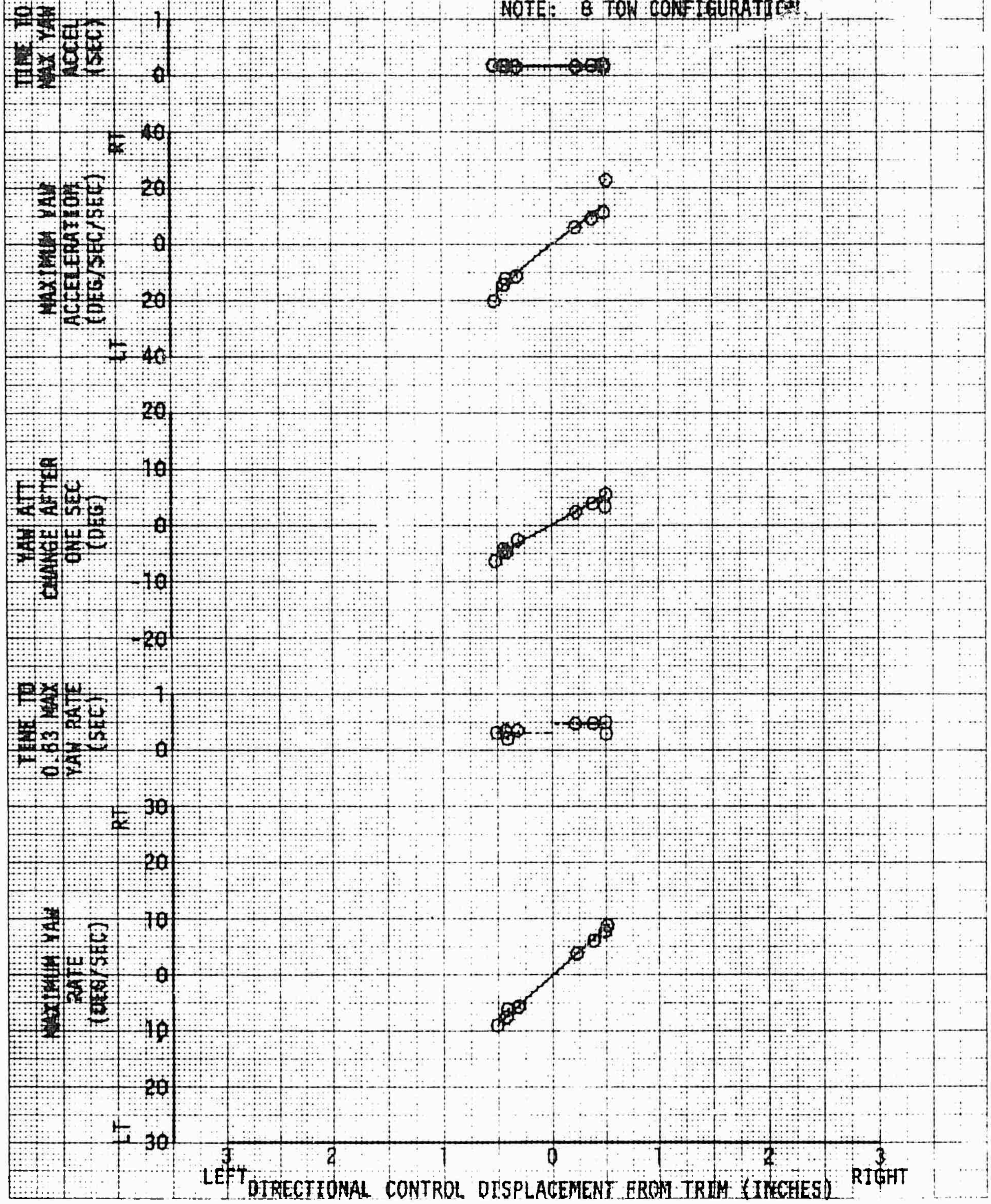

FIGURE 81
SIDEWARD FLIGHT
YAH-63 USA S/N 74-22247

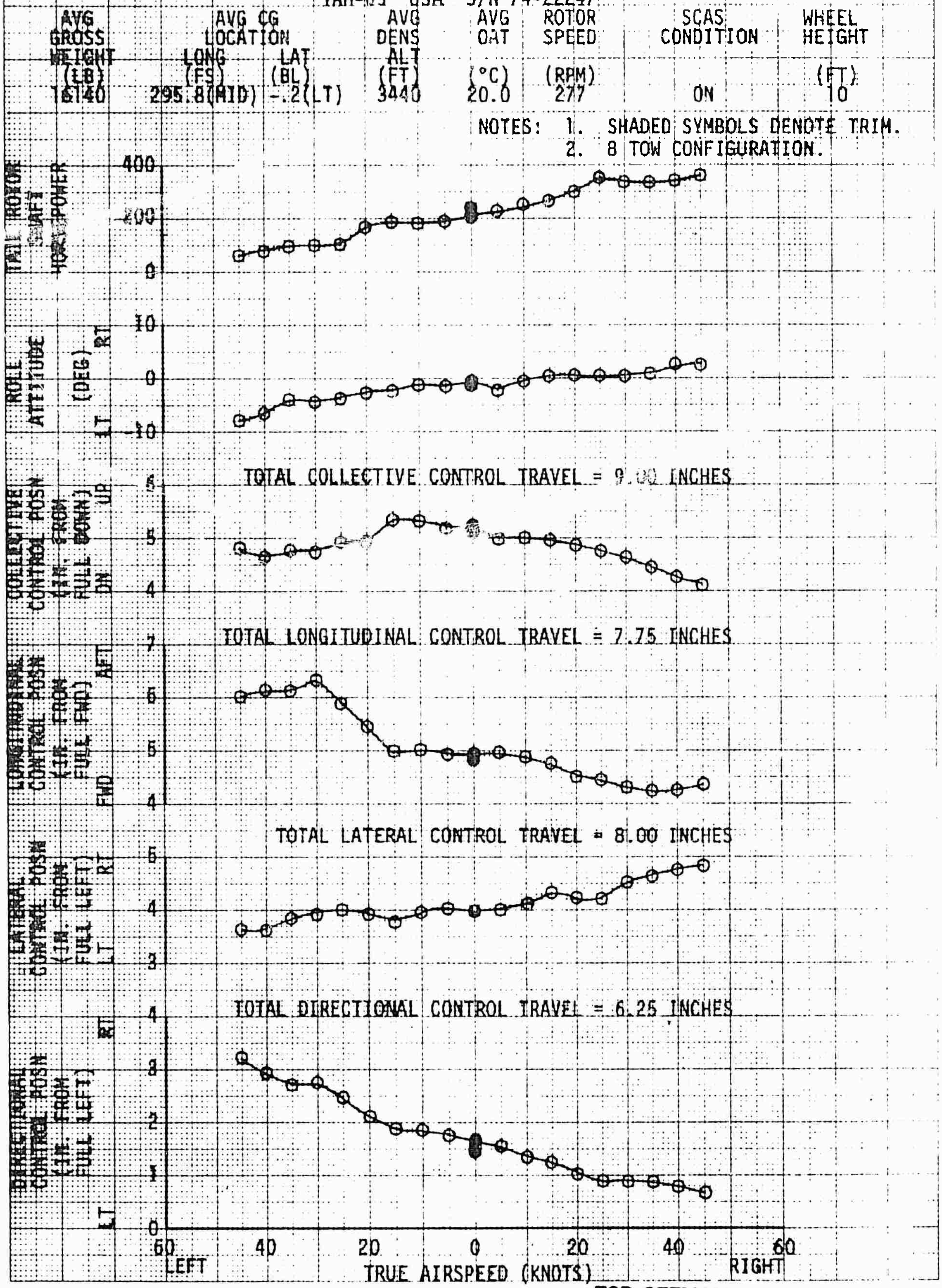

FIGURE 82
SIDEWARD FLIGHT
YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	SCAS CONDITION	WHEEL HEIGHT (FT)
15760	294.9(MID)	-.5(LT)	10760	7.5	275	ON	10

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION.

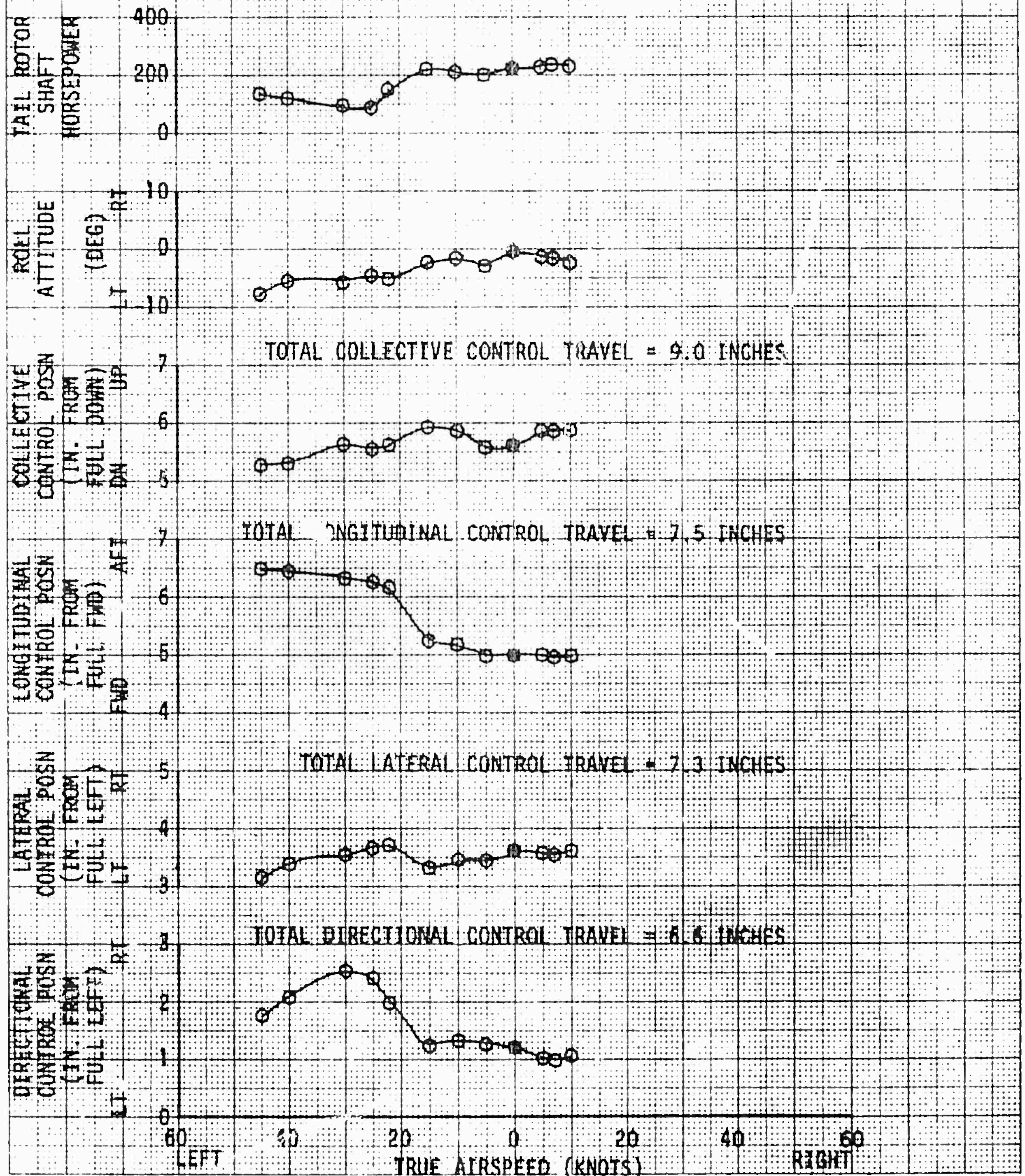

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FIGURE 83
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-63 USA S/N 74-22247

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	SCAS CONDITION	WHEEL HEIGHT (FT)
16120	295.8(MID)	-.2(LT)	4440	30.0	277	ON	10

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION.

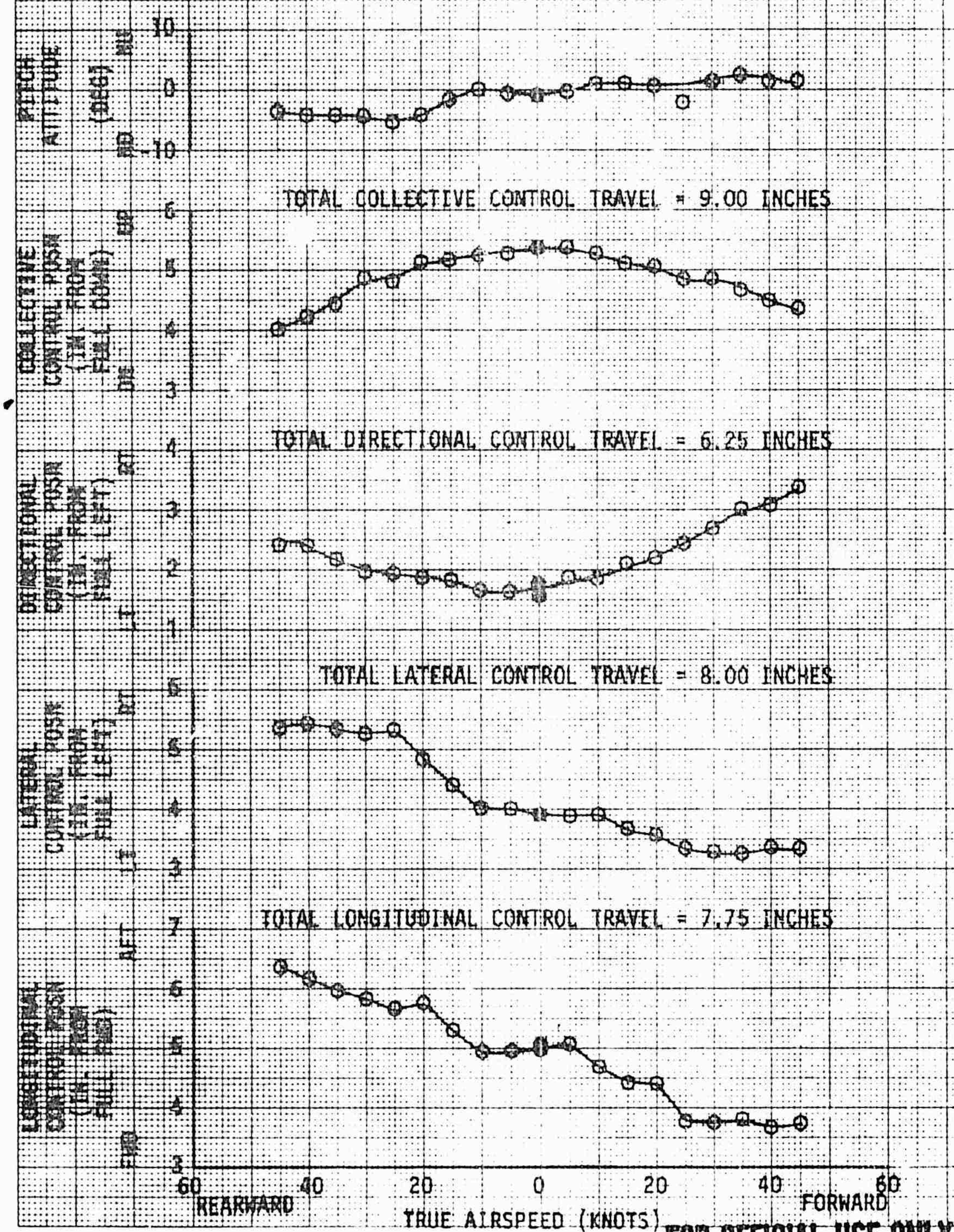

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FIGURE 84
LOW-SPEED FORWARD AND REARWARD FLIGHT
YAH-63 USA S/N 74-22246

SYM	AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	SCAS CONDITION	WHEEL HEIGHT (FT)
O	16260	296.3(MID)	-.5(LT)	11620	14.0	276	ON	10
Δ	15420	294.5(MID)	-.5(LT)	11400	10.5	276	ON	10

NOTES: 1. SHADED SYMBOLS DENOTE TRIM.
2. 8 TOW CONFIGURATION.

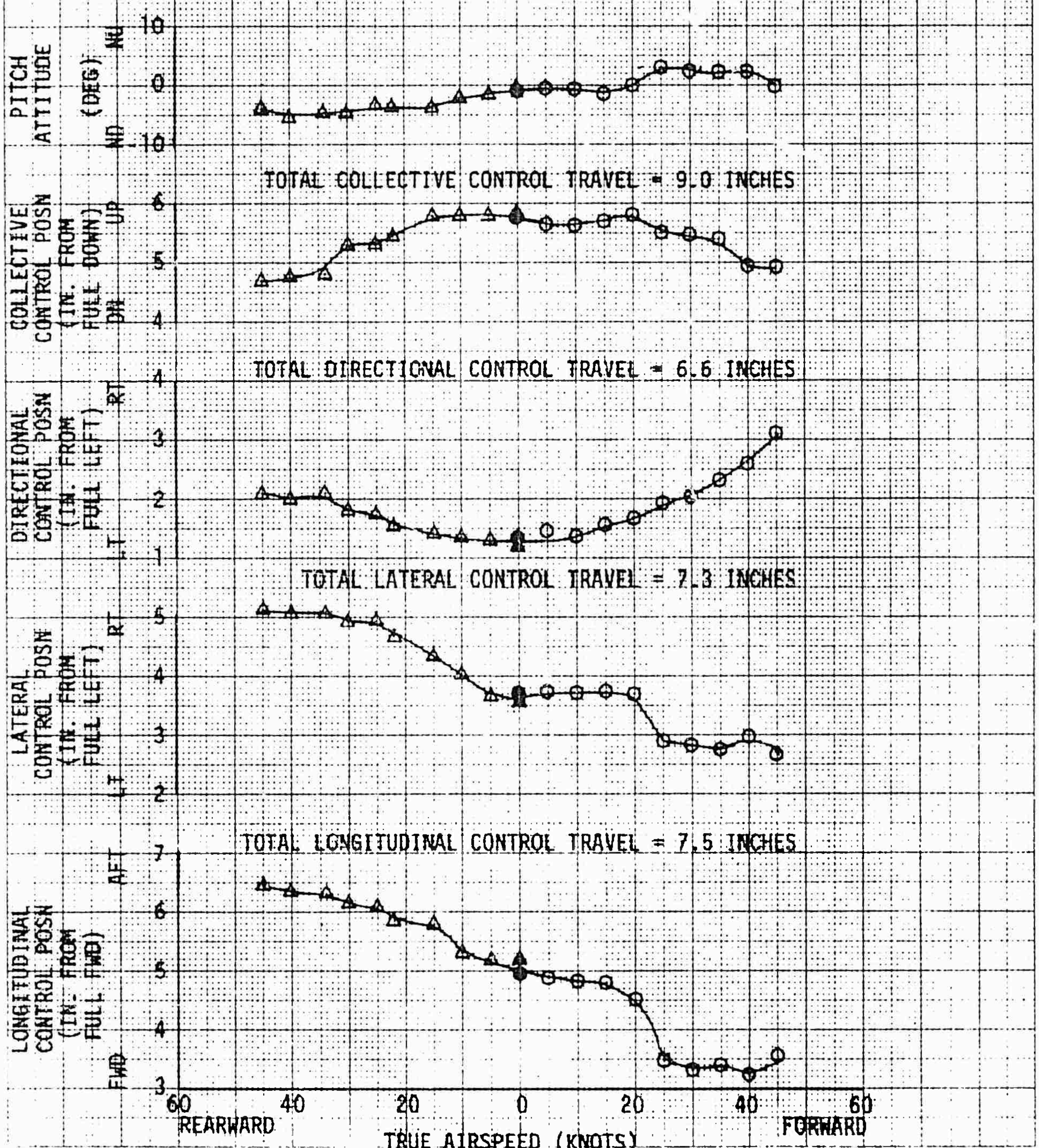

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FIGURE 85

LONGITUDINAL SCAS HARDOVER

YAH-63 USA S/N 74-22246

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
15450	299.3 (AFT)	-.1 (LT)	6060	17.5	278	115	ON

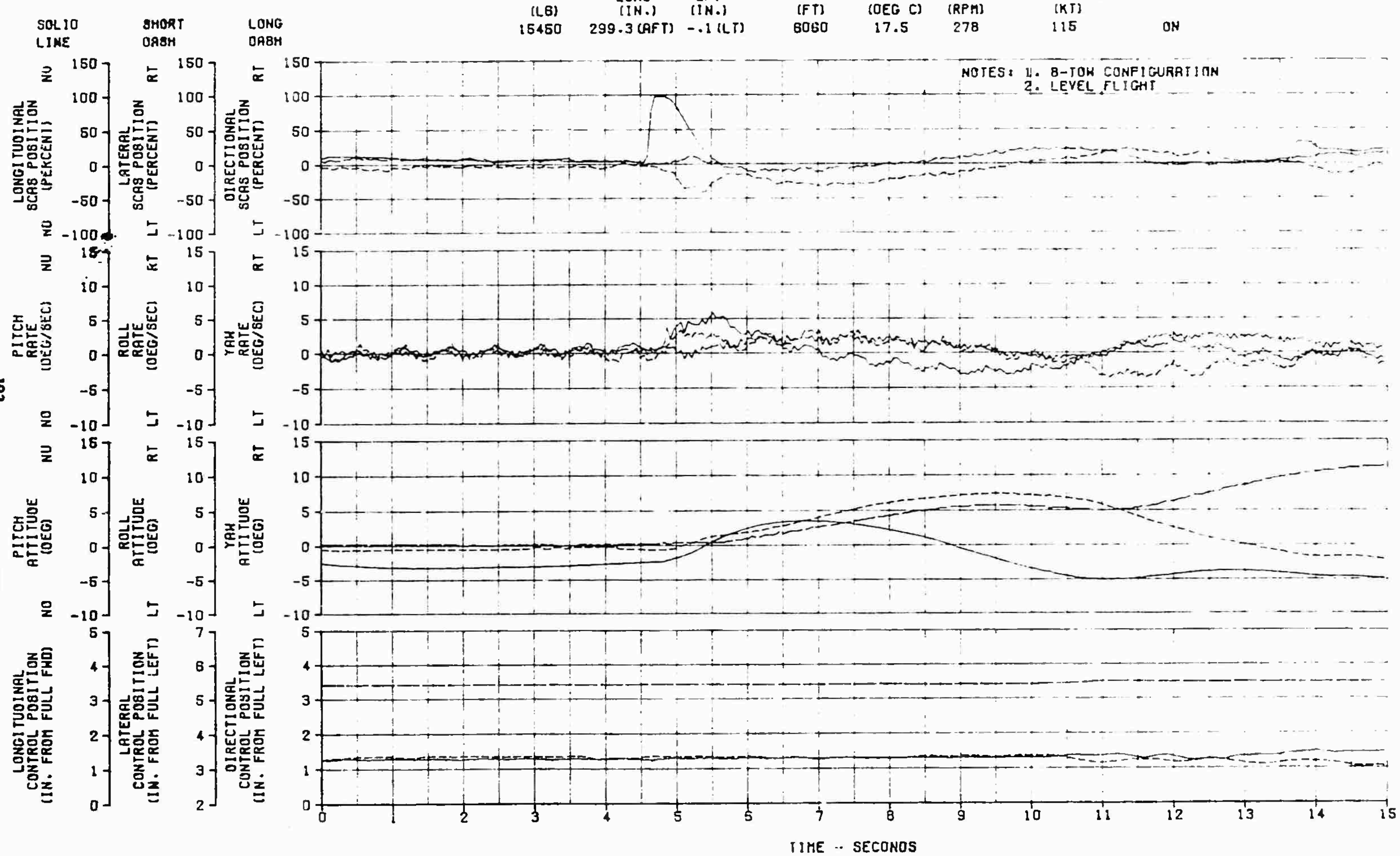


FIGURE 96

LATERAL SCAS HARDOVER

YAH-63 USA S/N 74-22248

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
16440	299.3 (AFT)	-.1 (LT)	6080	17.5	276	119	ON

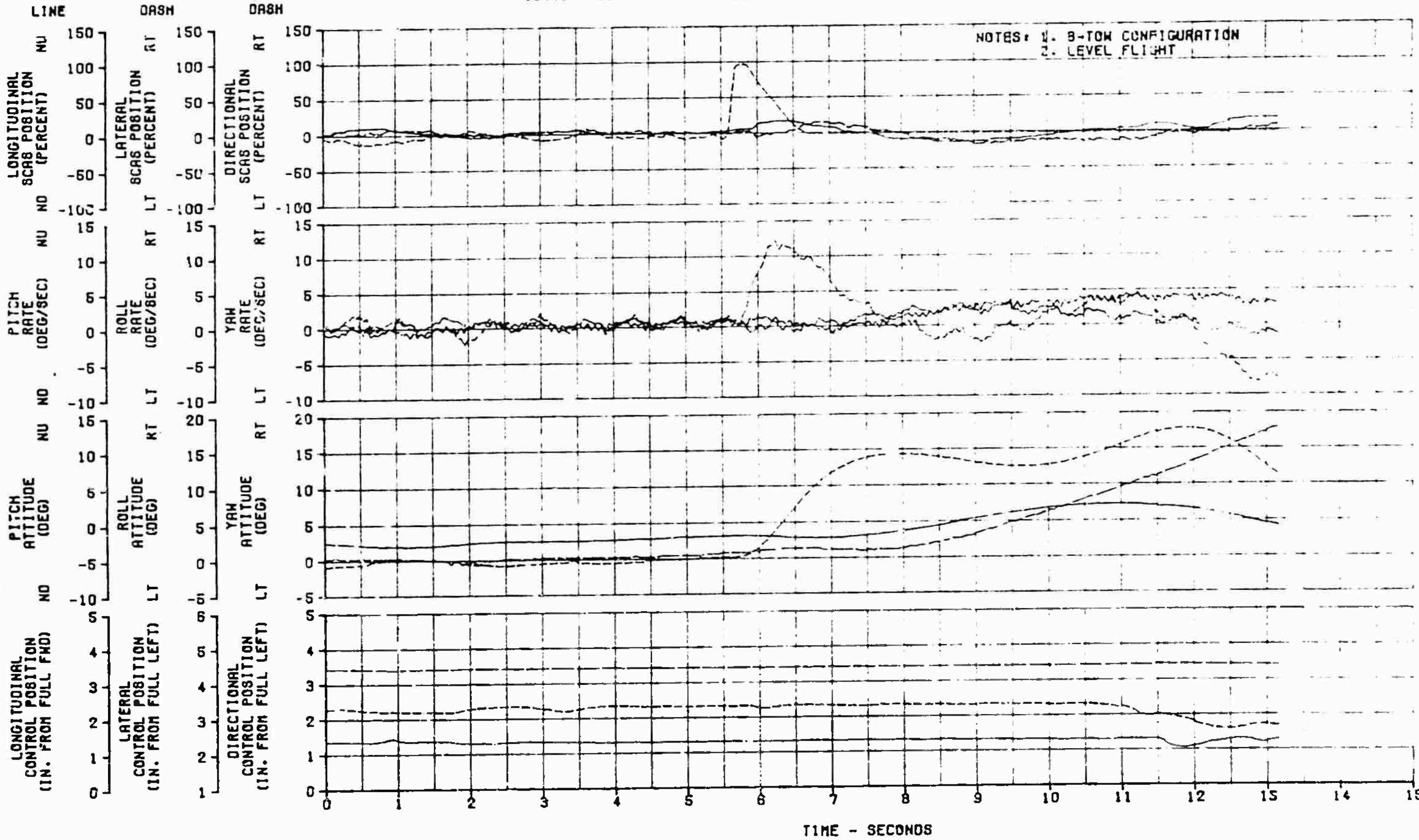
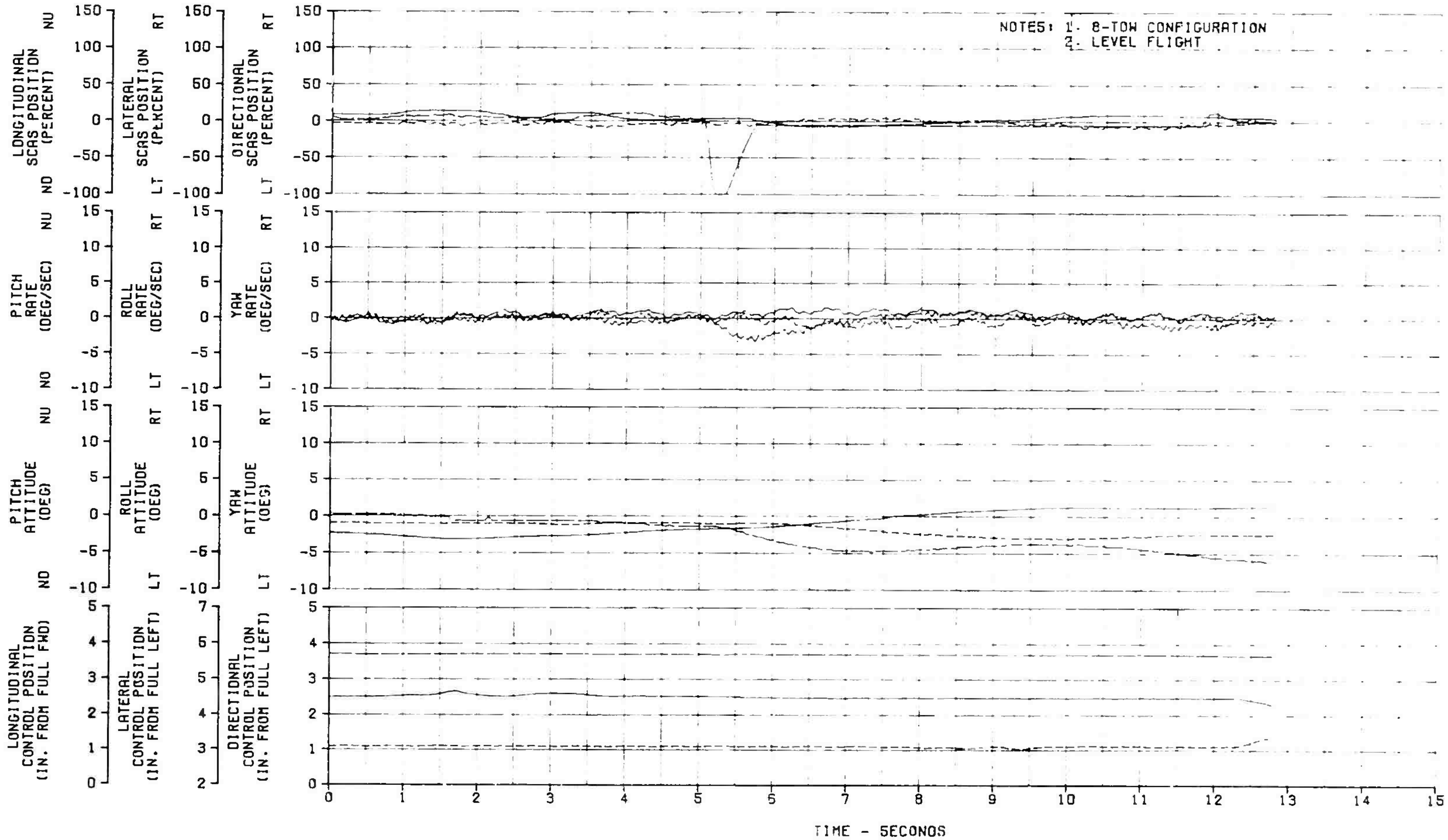


FIGURE 97

DIRECTIONAL SCAS HARDOVER

YAH-64 USA S/N 74-22246

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	SCAS CONDITION
16500	199.4 (AFT)	-.1 (LT)	5200	18.0	276	82	ON



FIGURE

ENGINE FAILURE

YAH-63 USA S/N 74-22247

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
16120	298.7 (AFT)	.2 (LT)	8290	10.5	275	80	MAXIMUM POWER CLIMB

NOTES: 1. SIMULATED NO 2 ENGINE FAILURE
2. B TOW CONFIGURATION

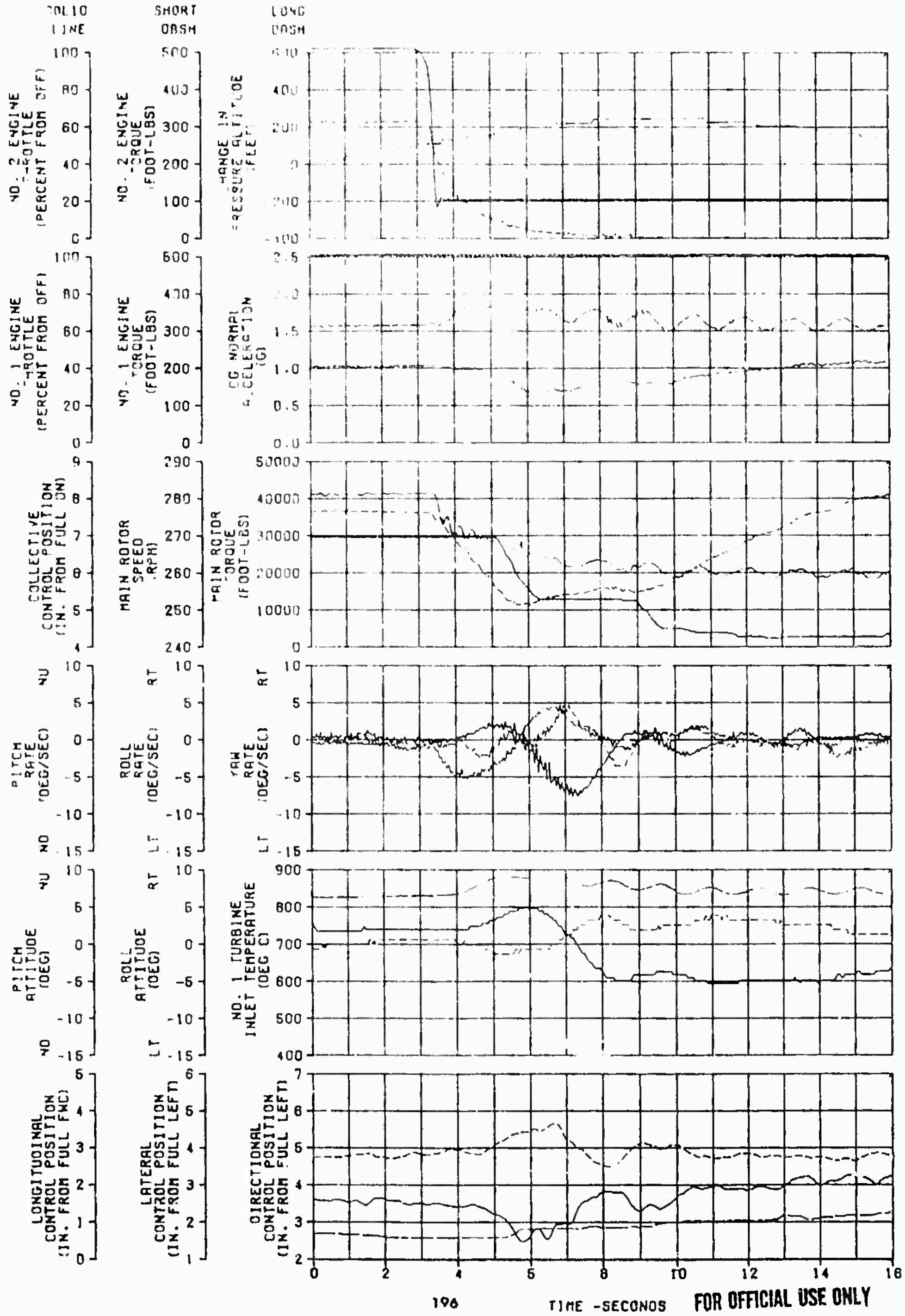


FIGURE 89

ENGINE FAILURE

YAH-63 USA S/N 74-22247

GROSS WEIGHT (LB)	C.G. LOCATION LONG. (IN.)	C.G. LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM ROTOR SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
15510	298.2 (AFT)	-.2 (LT)	5190	9.0	275	75	LEVEL FLIGHT

NOTES: 1. SIMULATED NO 2 ENGINE FAILURE FROM SINGLE ENGINE FLIGHT

2. R-TOW CONFIGURATION

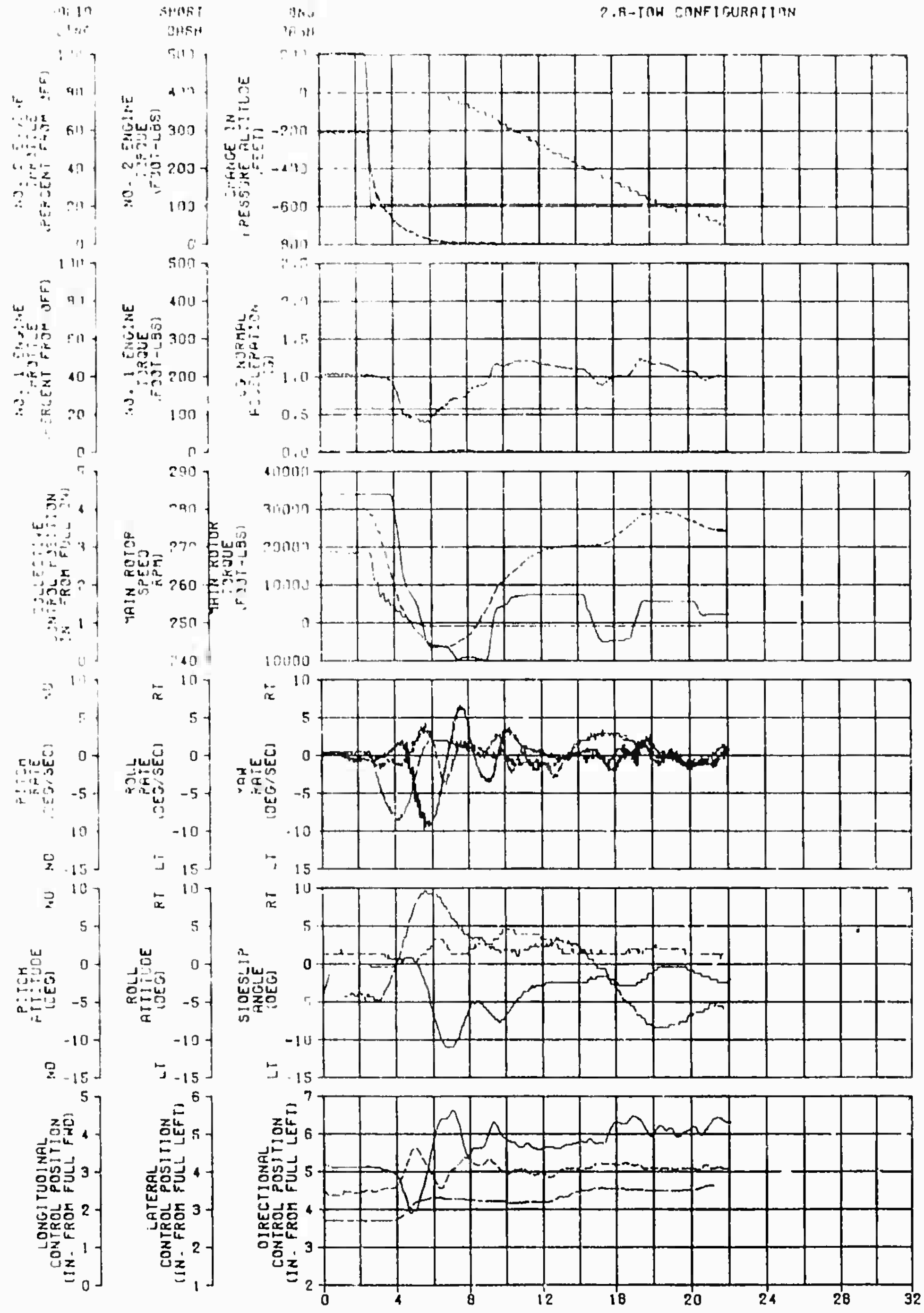


FIGURE 90

ENGINE FAILURE

YAH-63 USA S/N 74-22247

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	TRIM SPEED (RPM)	TRIM CALIBRATED AIRSPEED (KT)	FLIGHT CONDITION
14640	298.5 (AFT)	-.2 (LT)	9000	14.0	274	73	LEVEL FLIGHT

NOTES: 1. SIMULATED NO 2 ENGINE FAILURE FROM SINGLE ENGINE FLIGHT

2.8-TOW CONFIGURATION

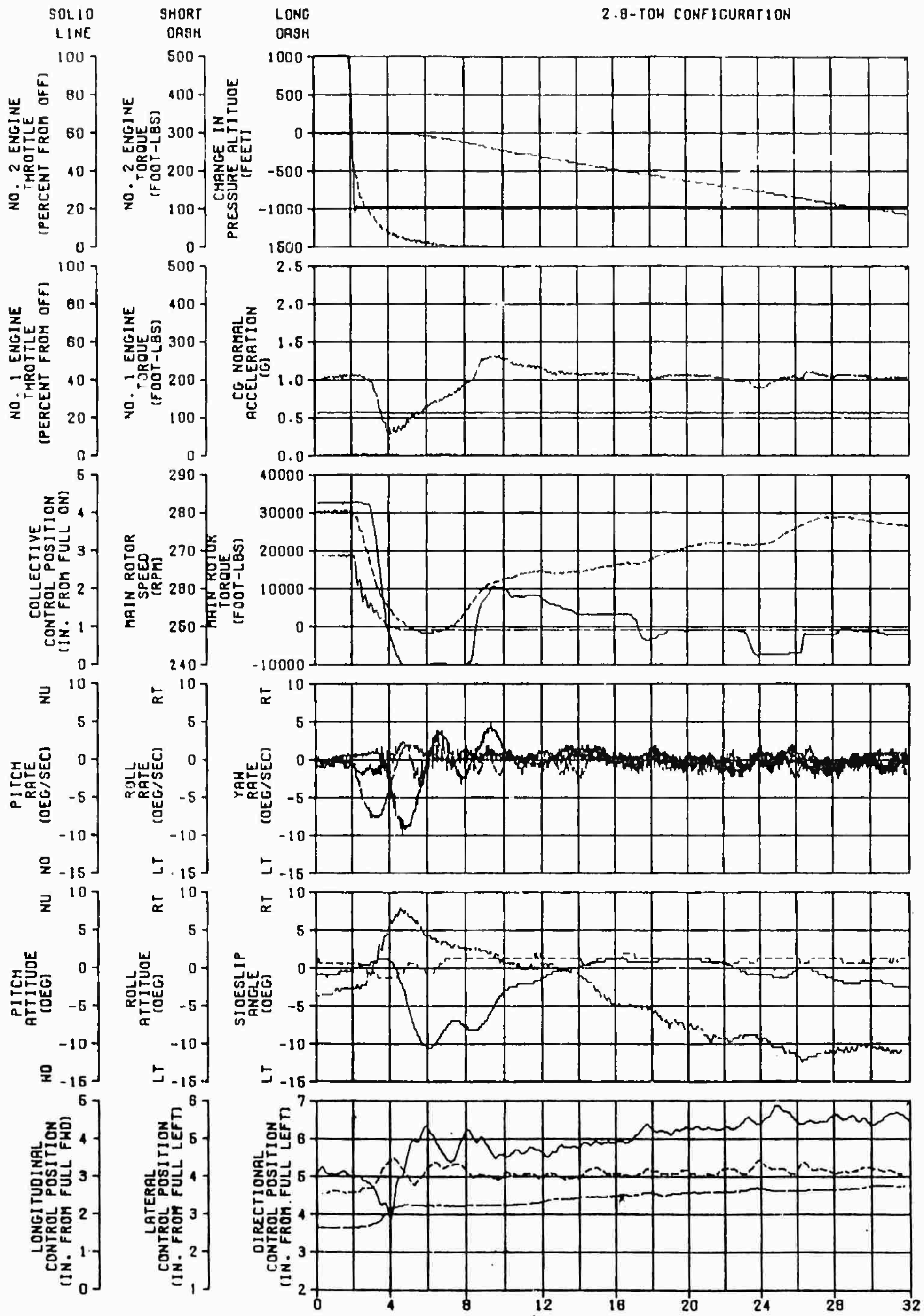


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FIGURE 91
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

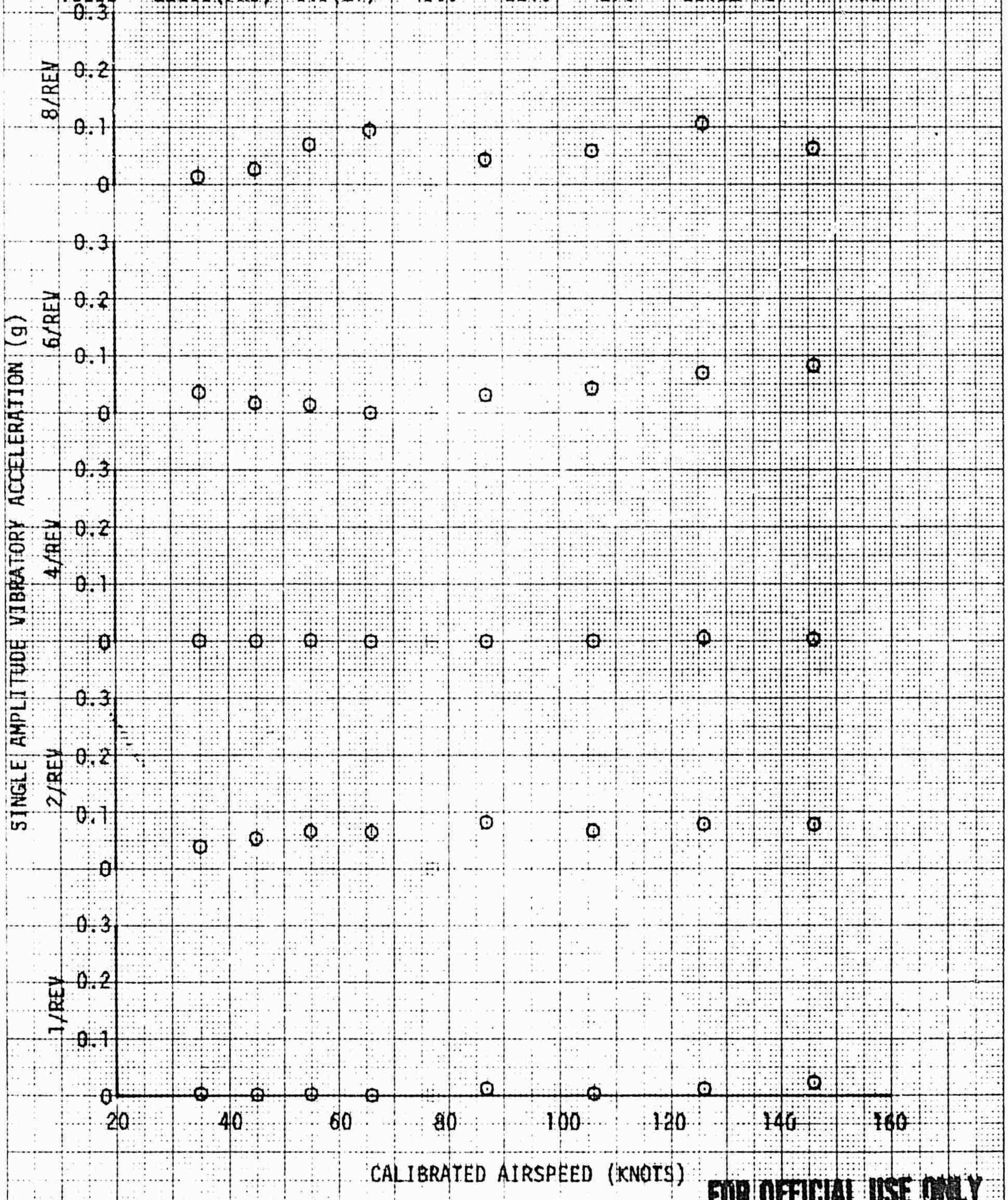

FIGURE 92
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT SEAT LATERAL

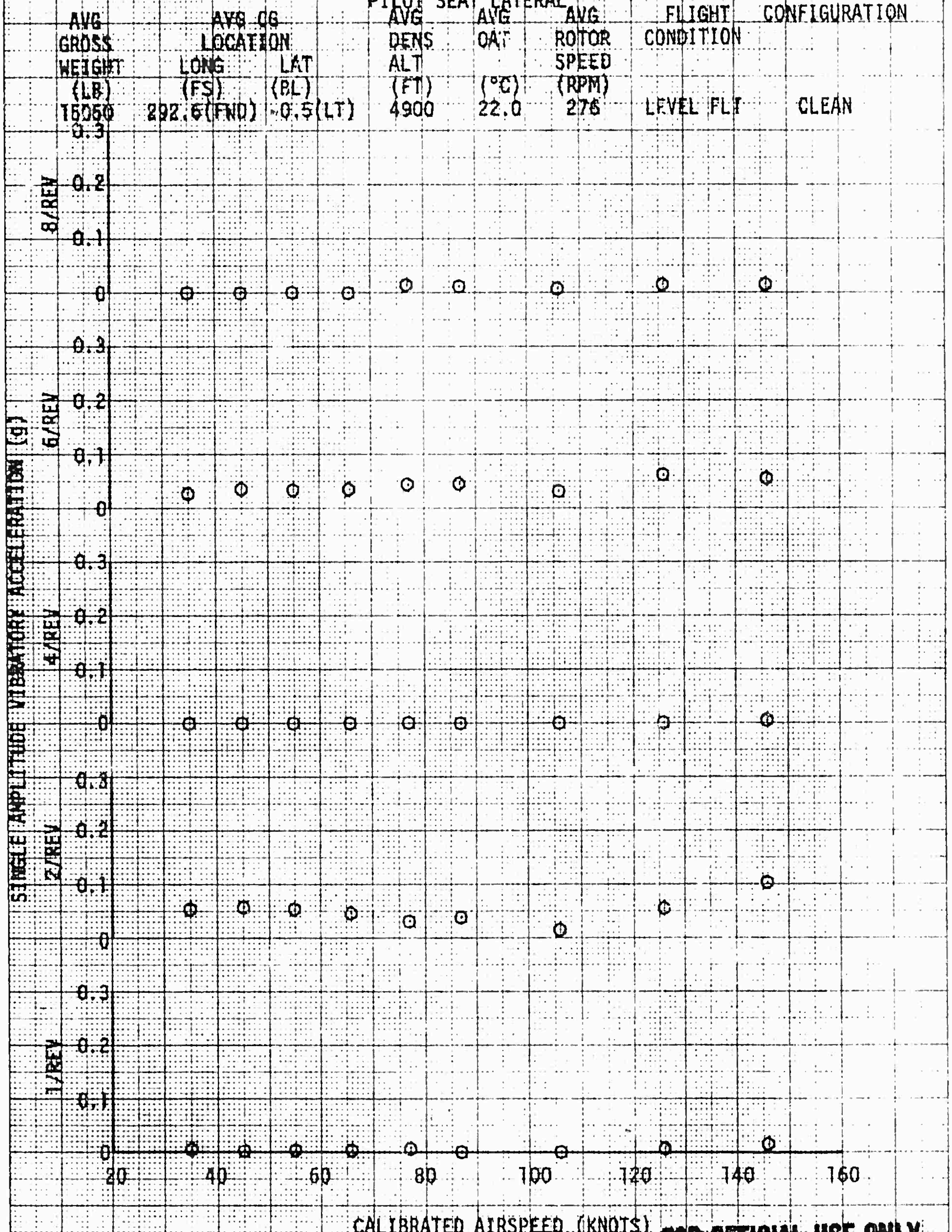

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FIGURE 93
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT SEAT LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.5(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

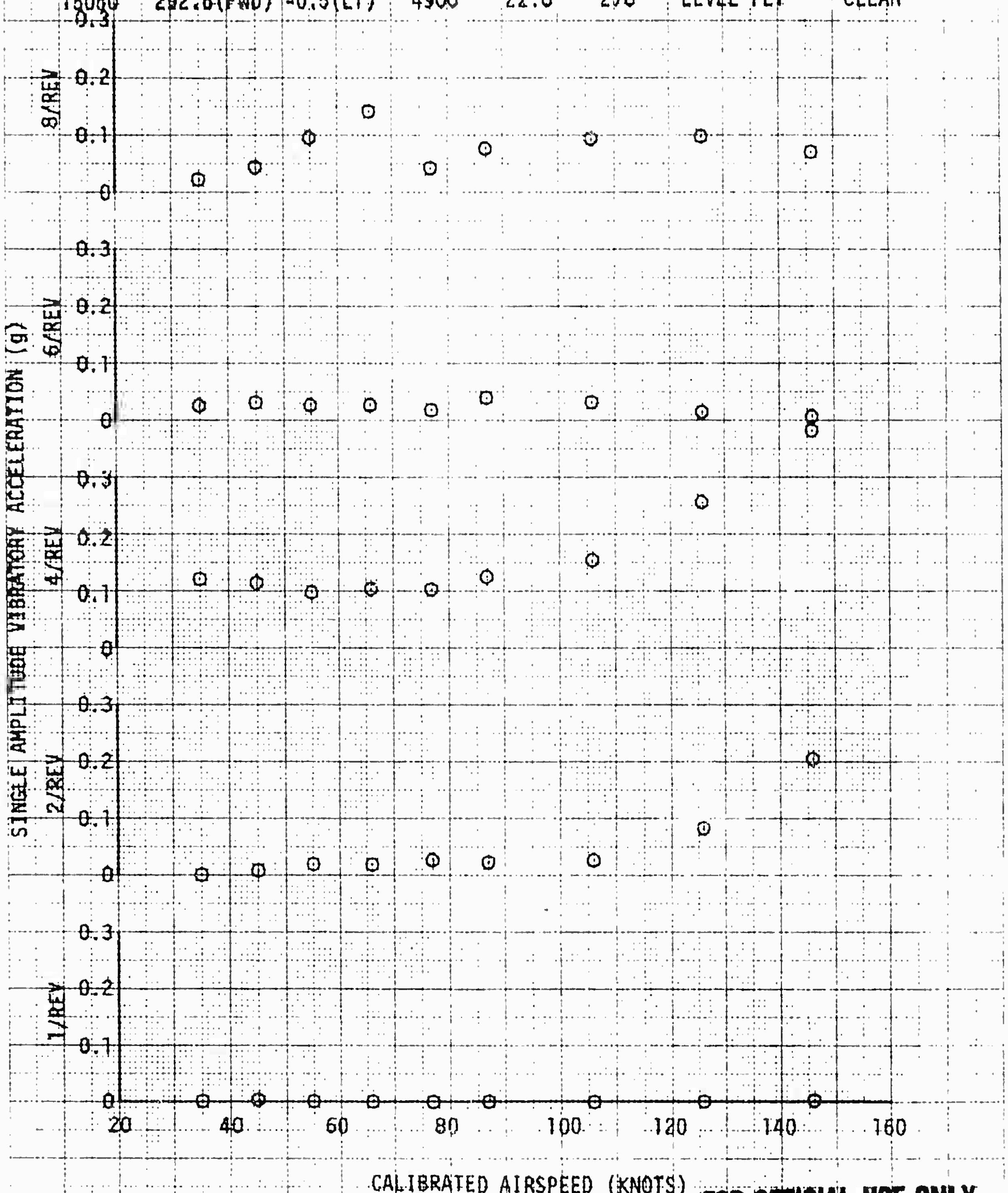

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FIGURE 94
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
COPILOT SEAT VERTICAL

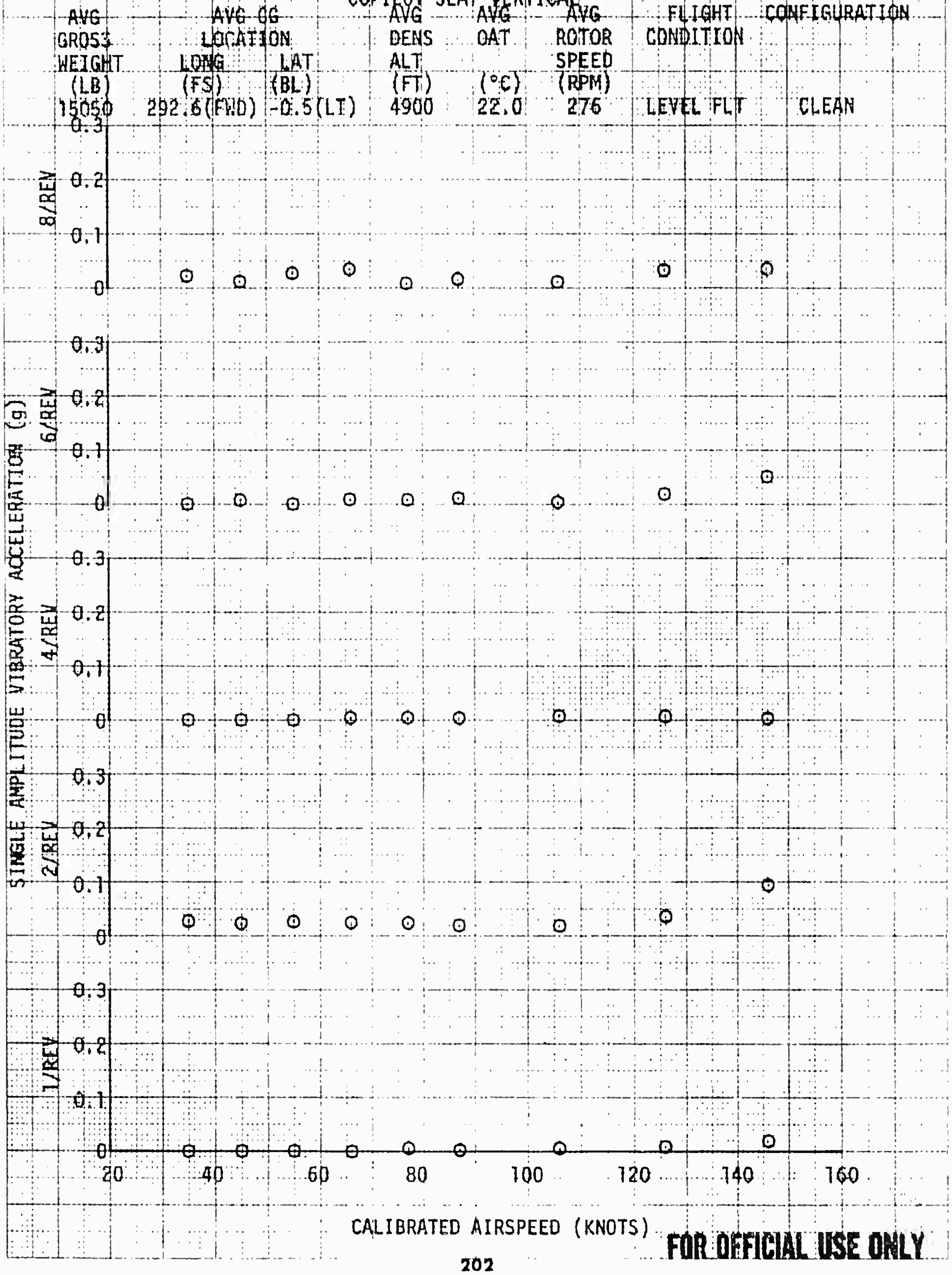

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FIGURE 95
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
COPILOT SEAT LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15080	292.6 (FWD)	-0.5 (LT)	4900	22.0	276	LEVEL FLT	CLEAN

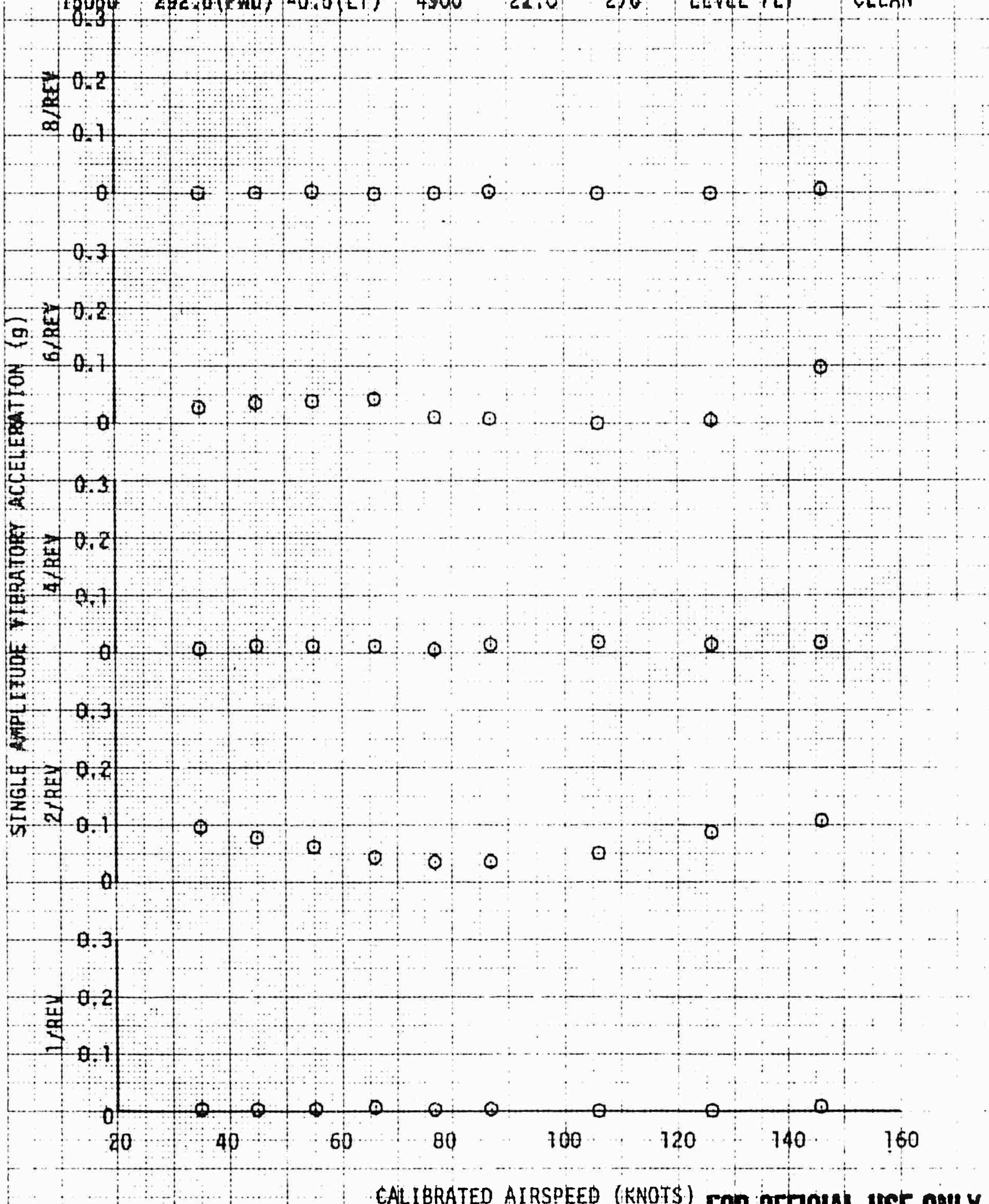

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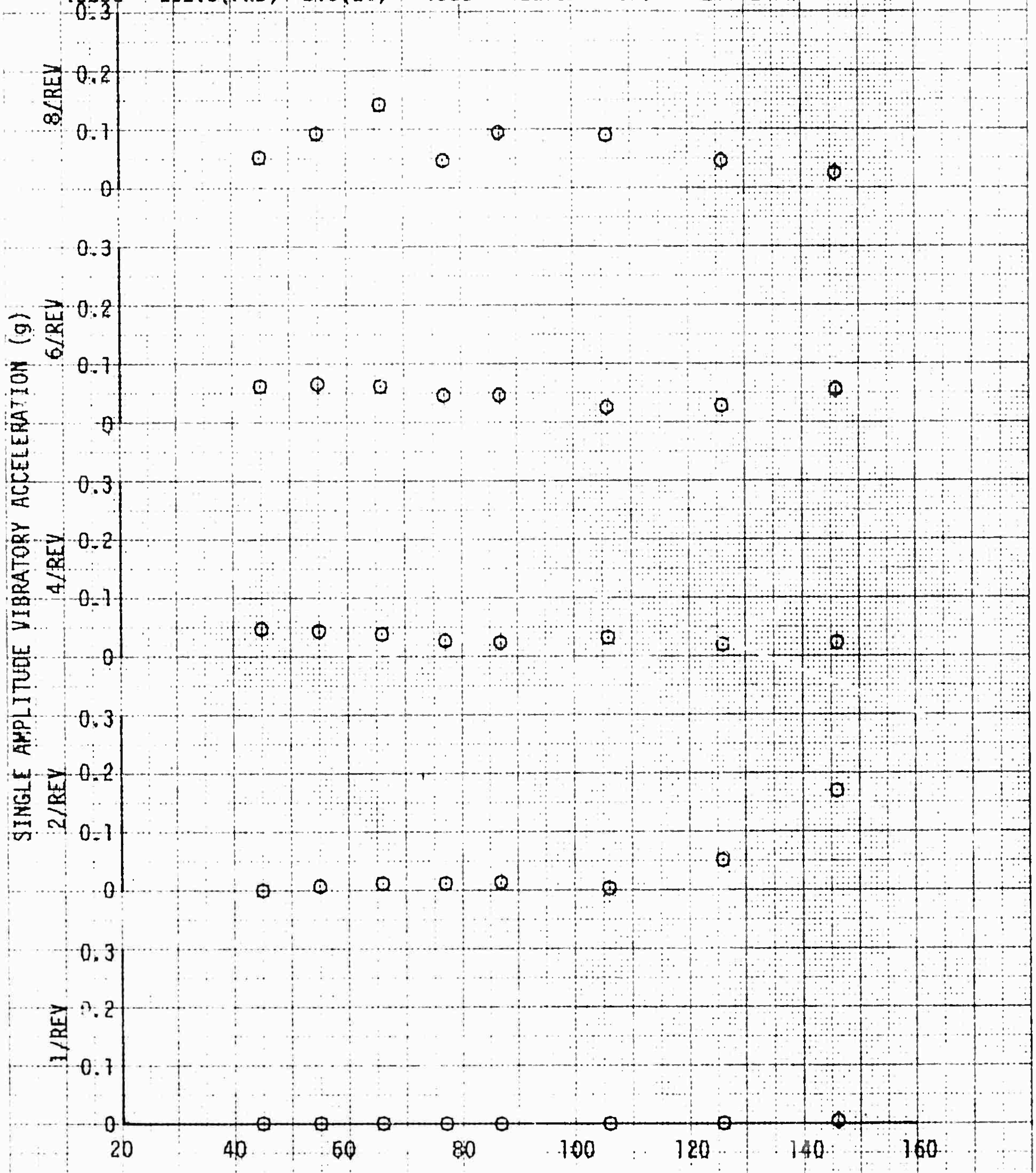
FIGURE 96
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
COPILOT SEAT LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN



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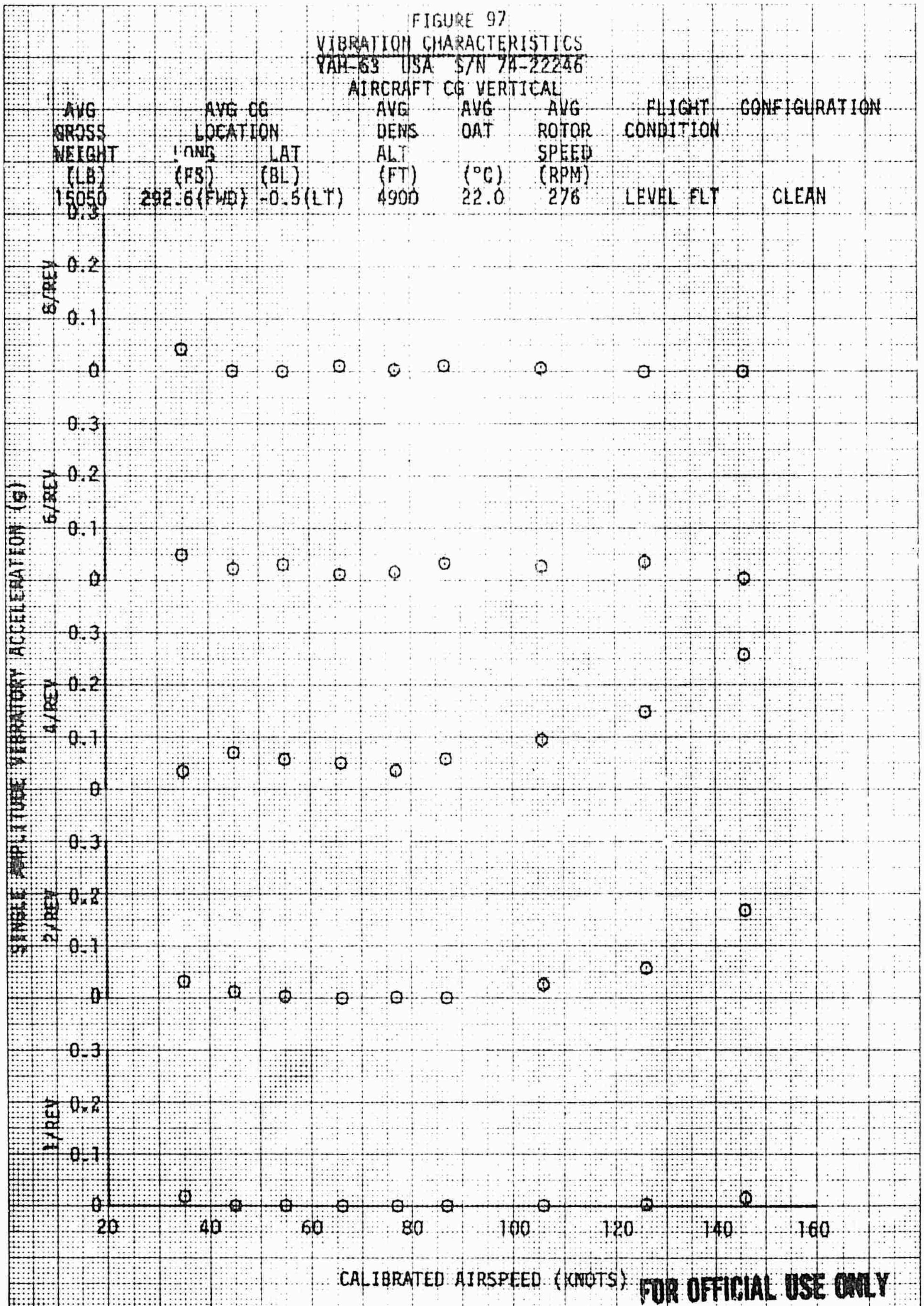

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FIGURE 98
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
AIRCRAFT CG LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

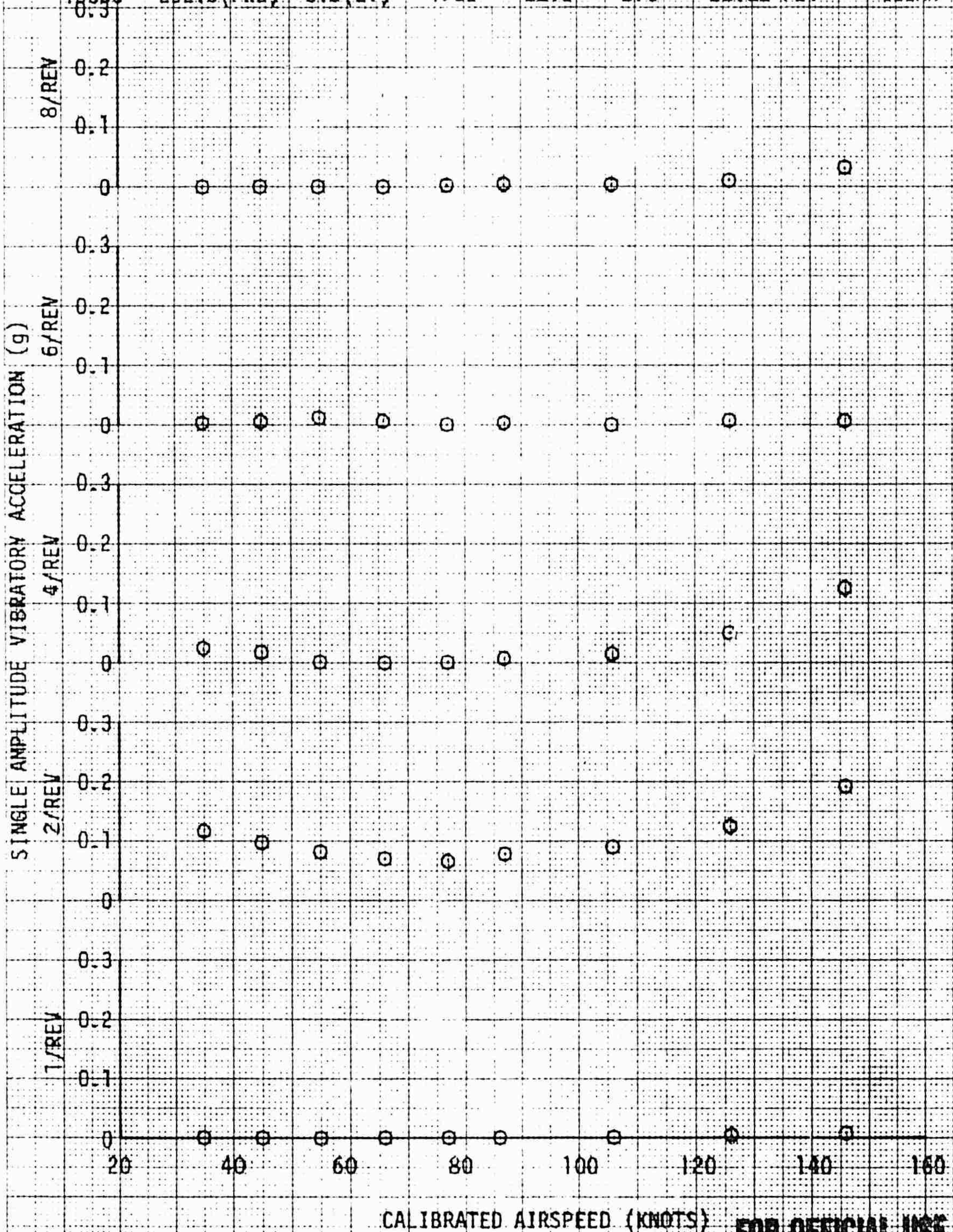

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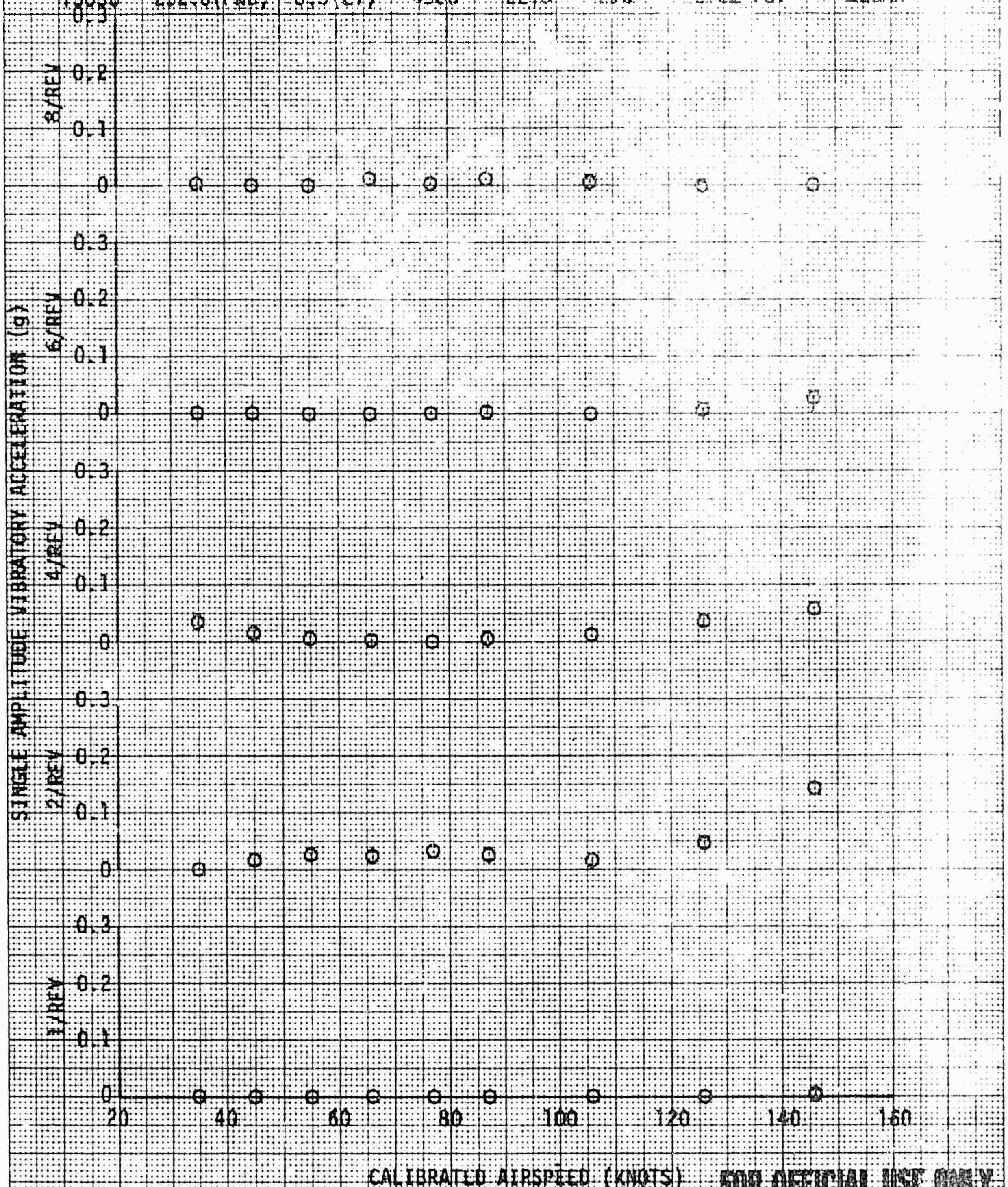
FIGURE 99
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
AIRCRAFT CG LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN



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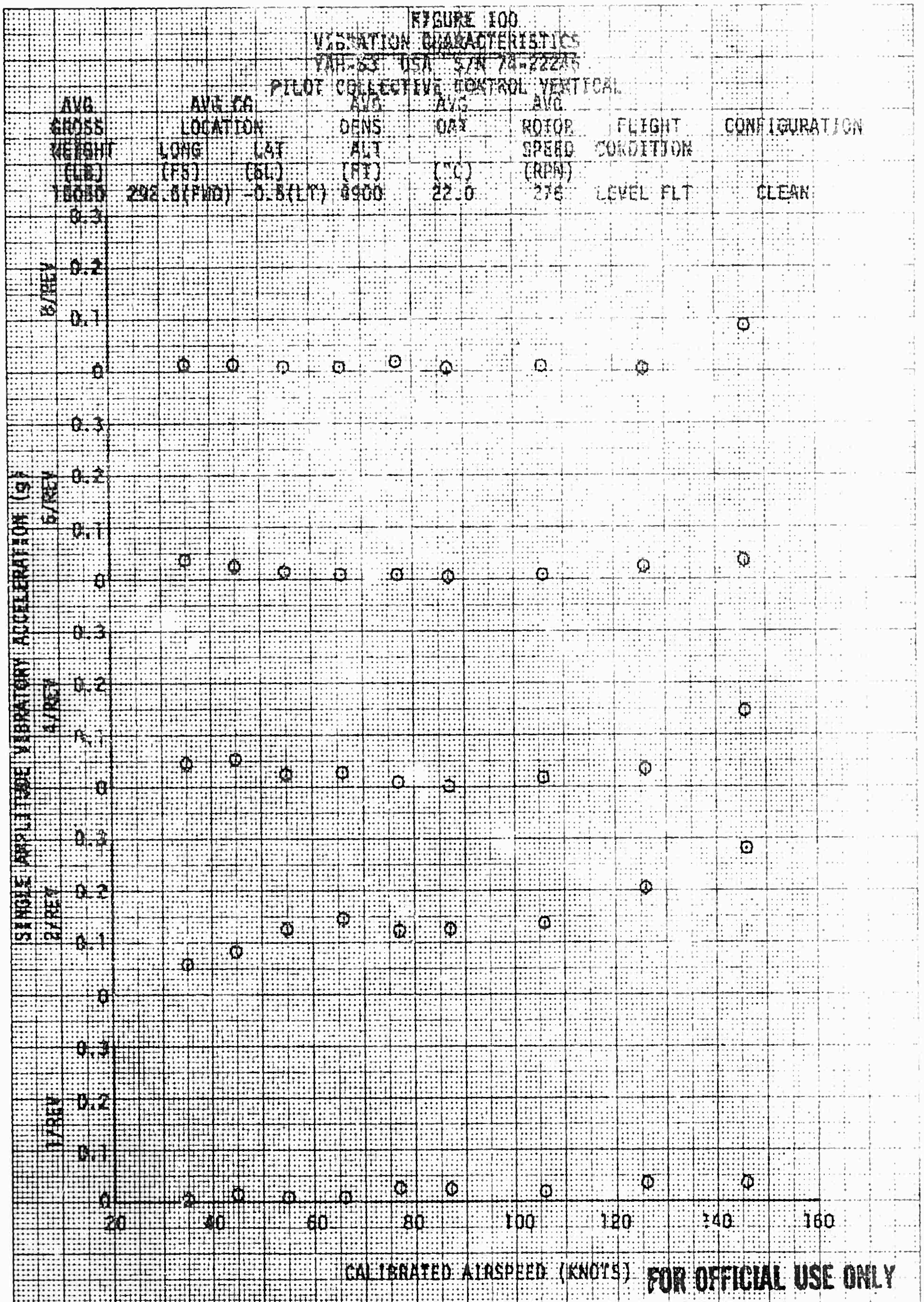

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FIGURE 101
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT COLLECTIVE CONTROL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

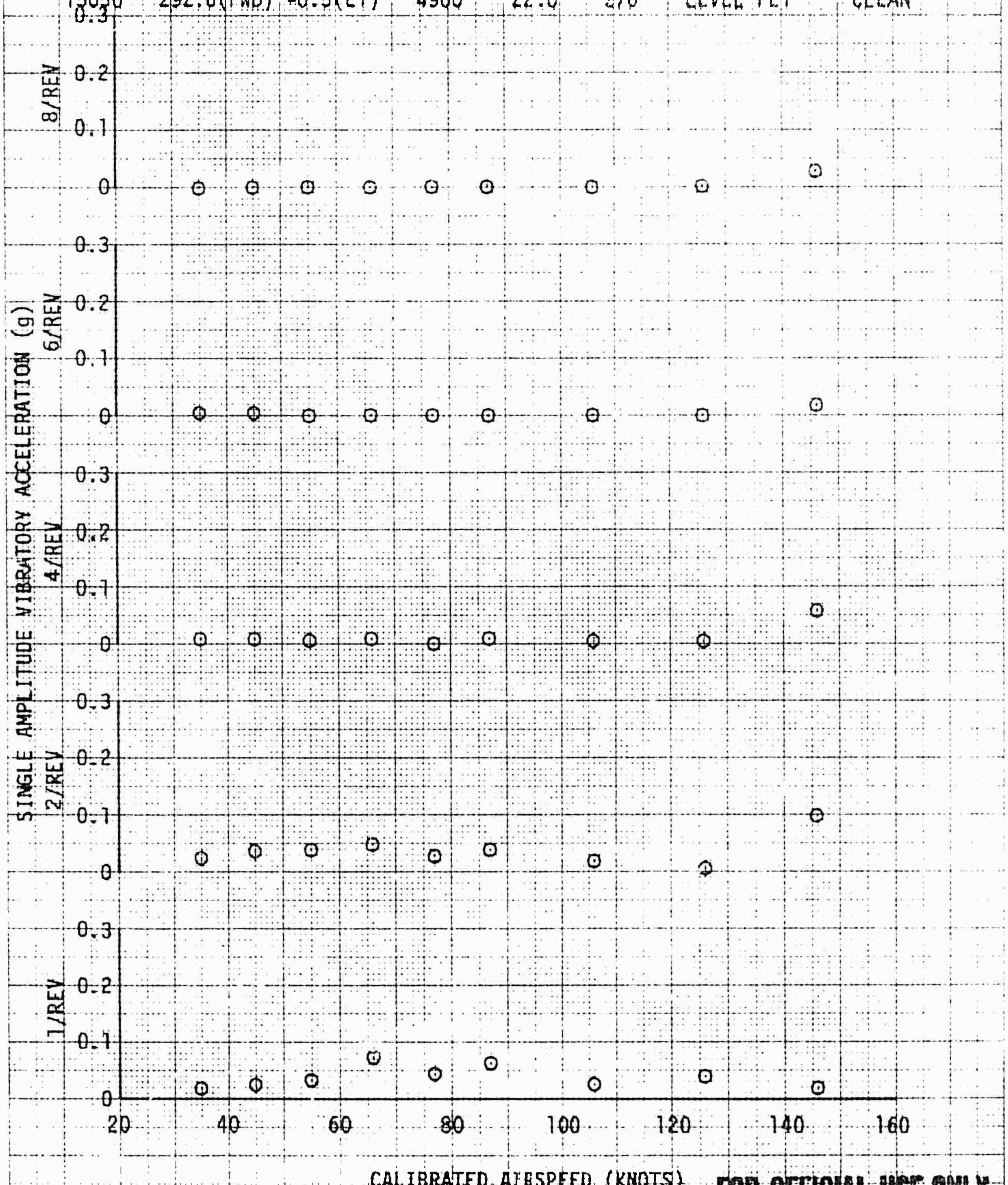

FIGURE 102
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT HEEL REST VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
16050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

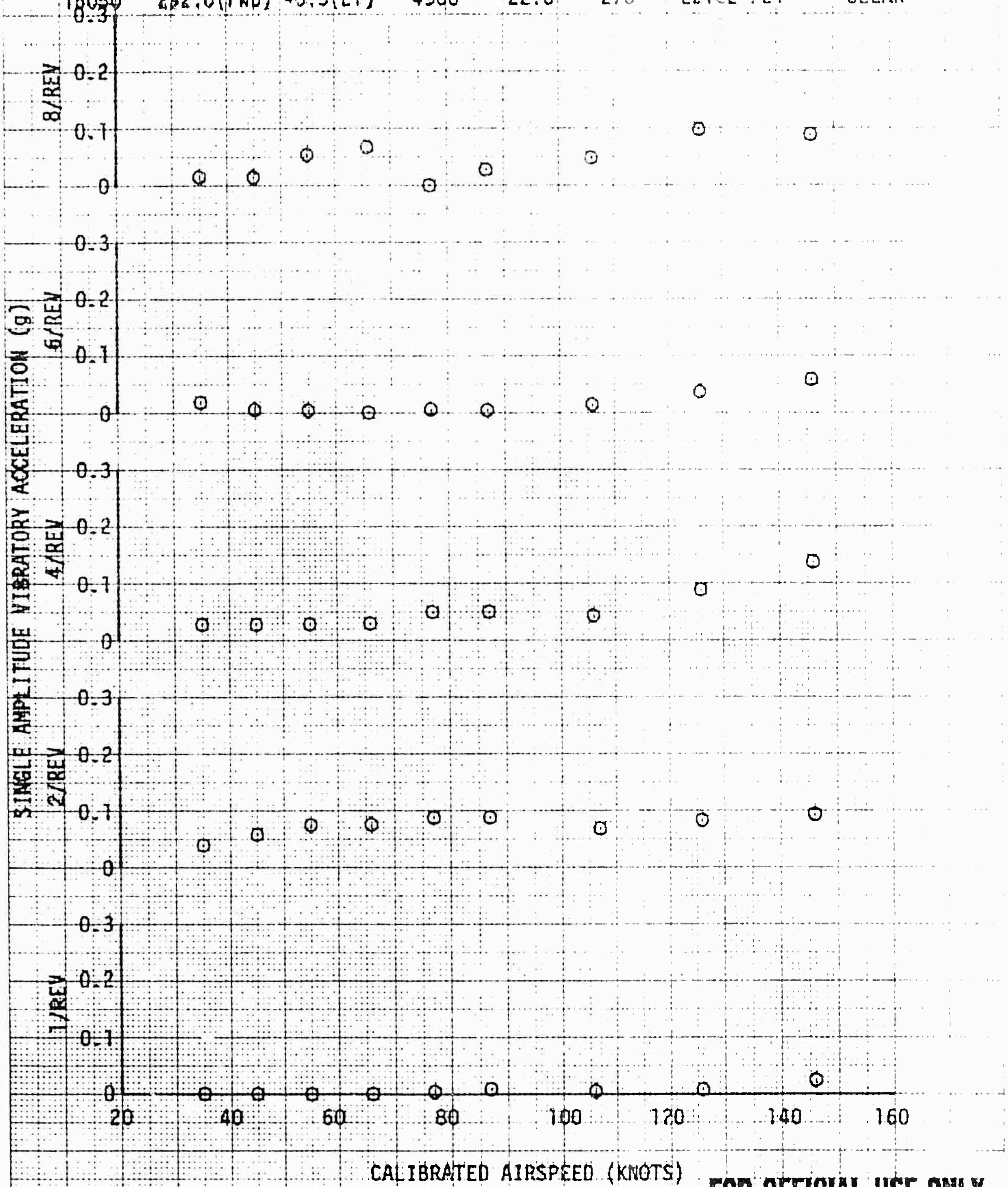

FIGURE 108
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT HEEL REST LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

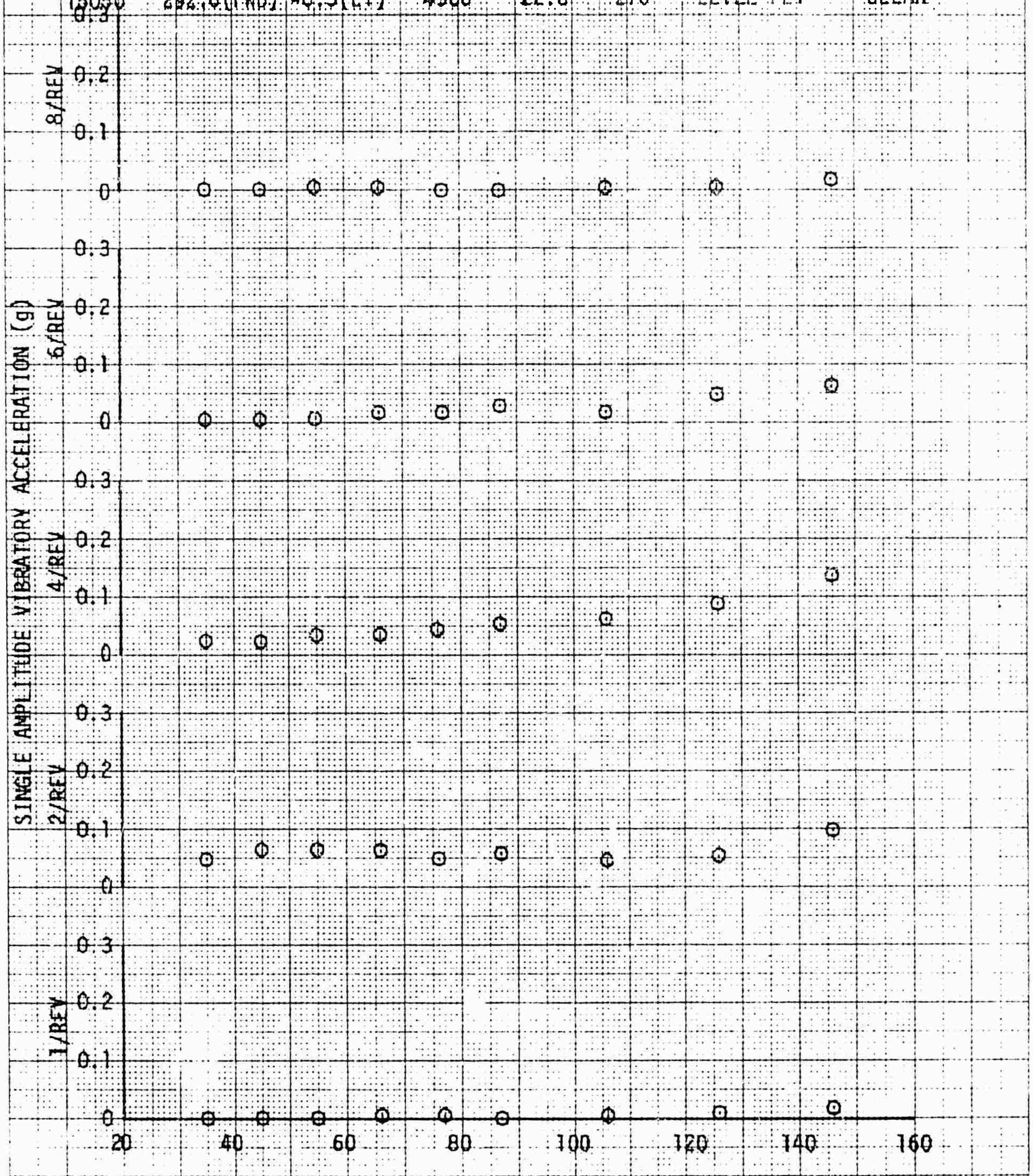

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FIGURE 104
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
LEFT PILOT INSTRUMENT PANEL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
16050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

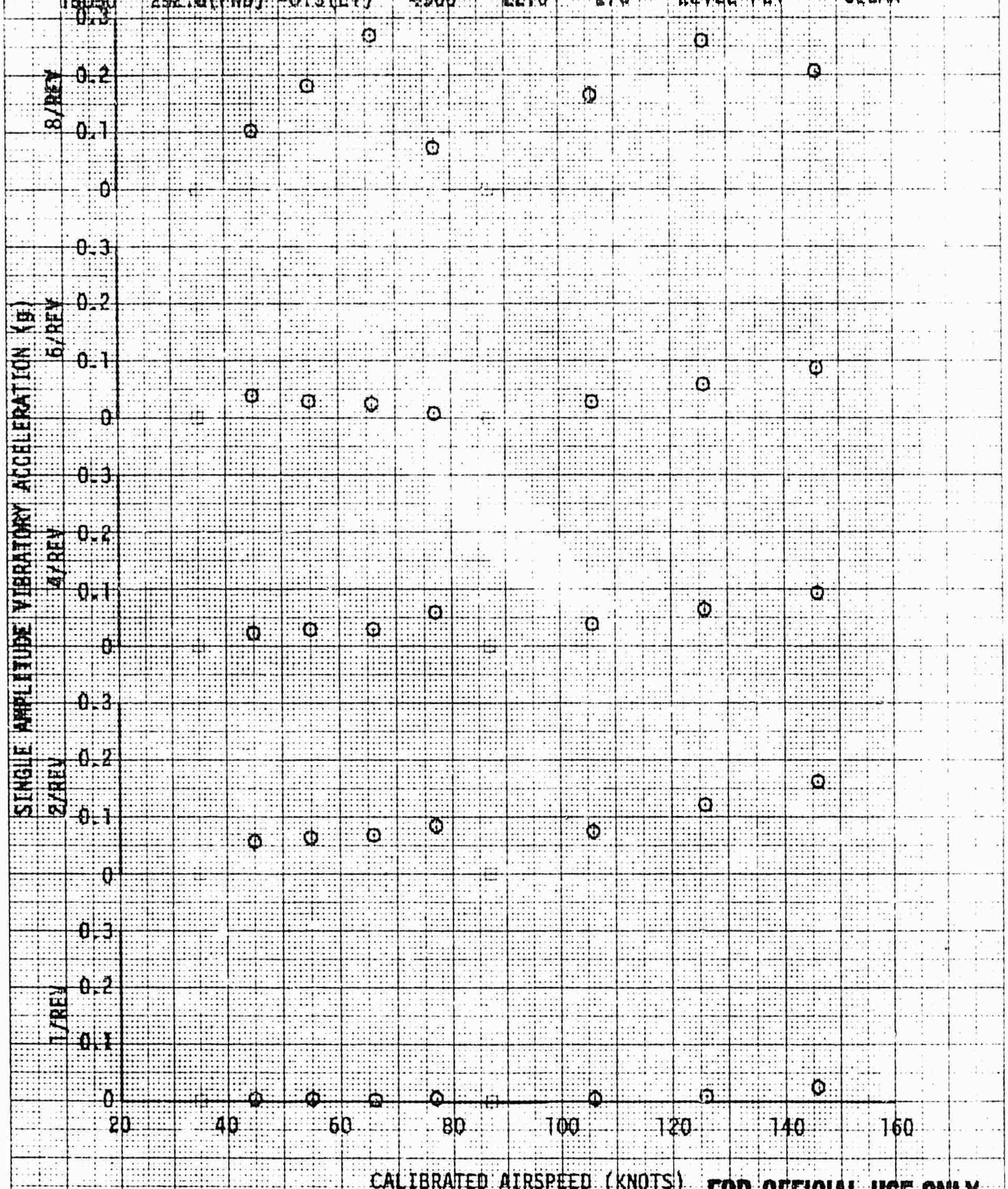

FIGURE 105
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
LEFT PILOT INSTRUMENT PANEL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15060	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

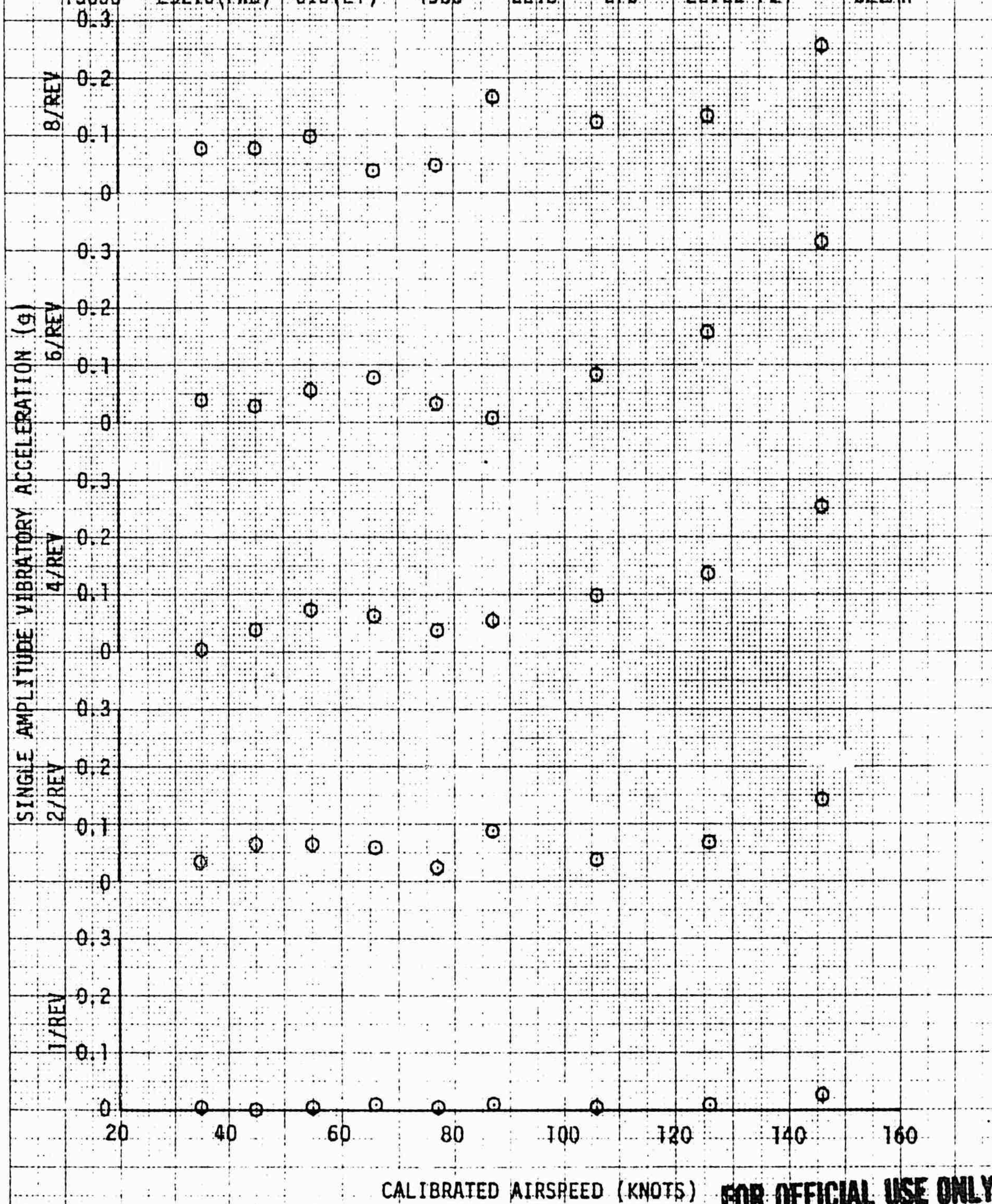

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FIGURE 106
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
LEFT PILOT INSTRUMENT PANEL LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

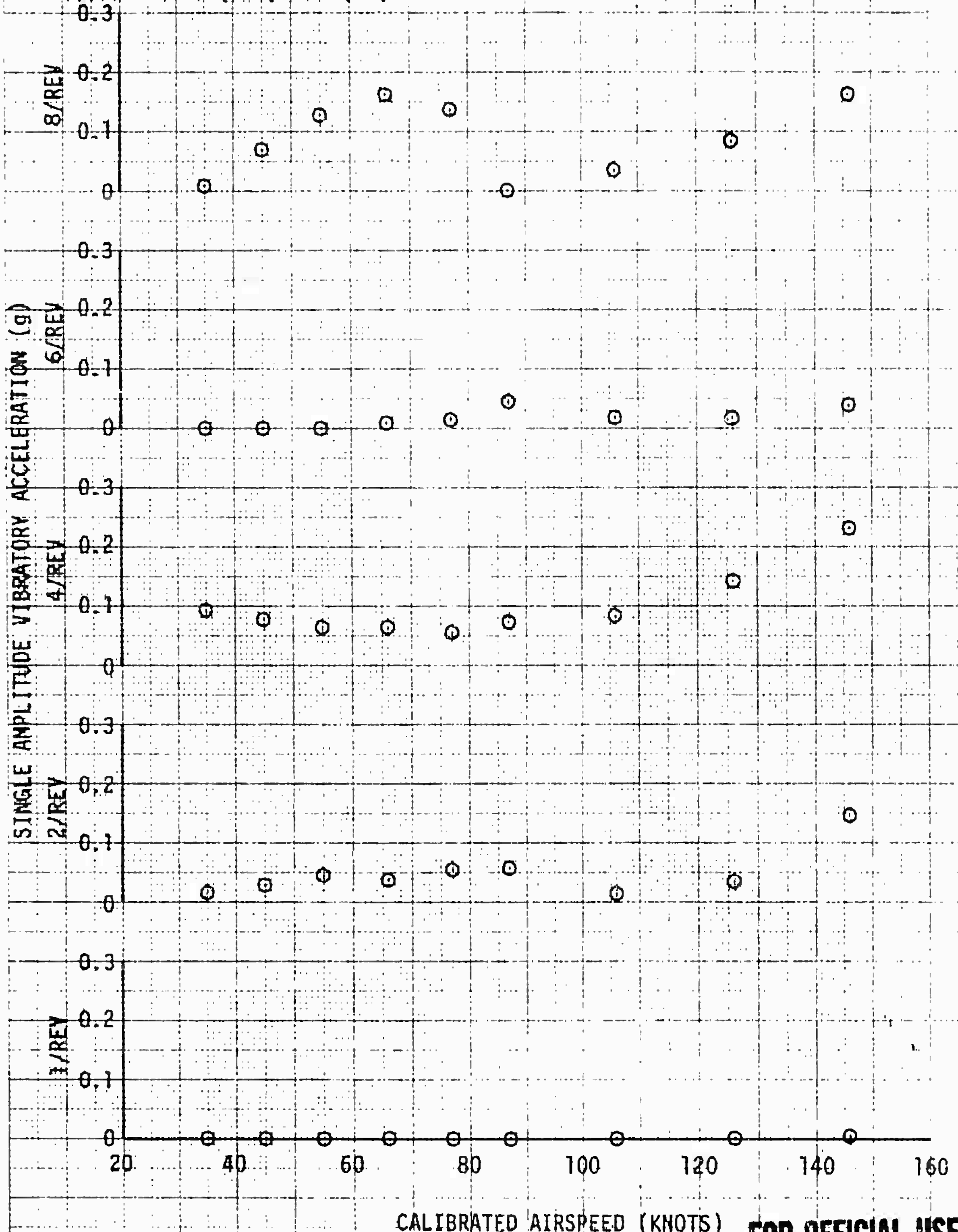

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FIGURE 107
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
RIGHT PILOT INSTRUMENT PANEL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

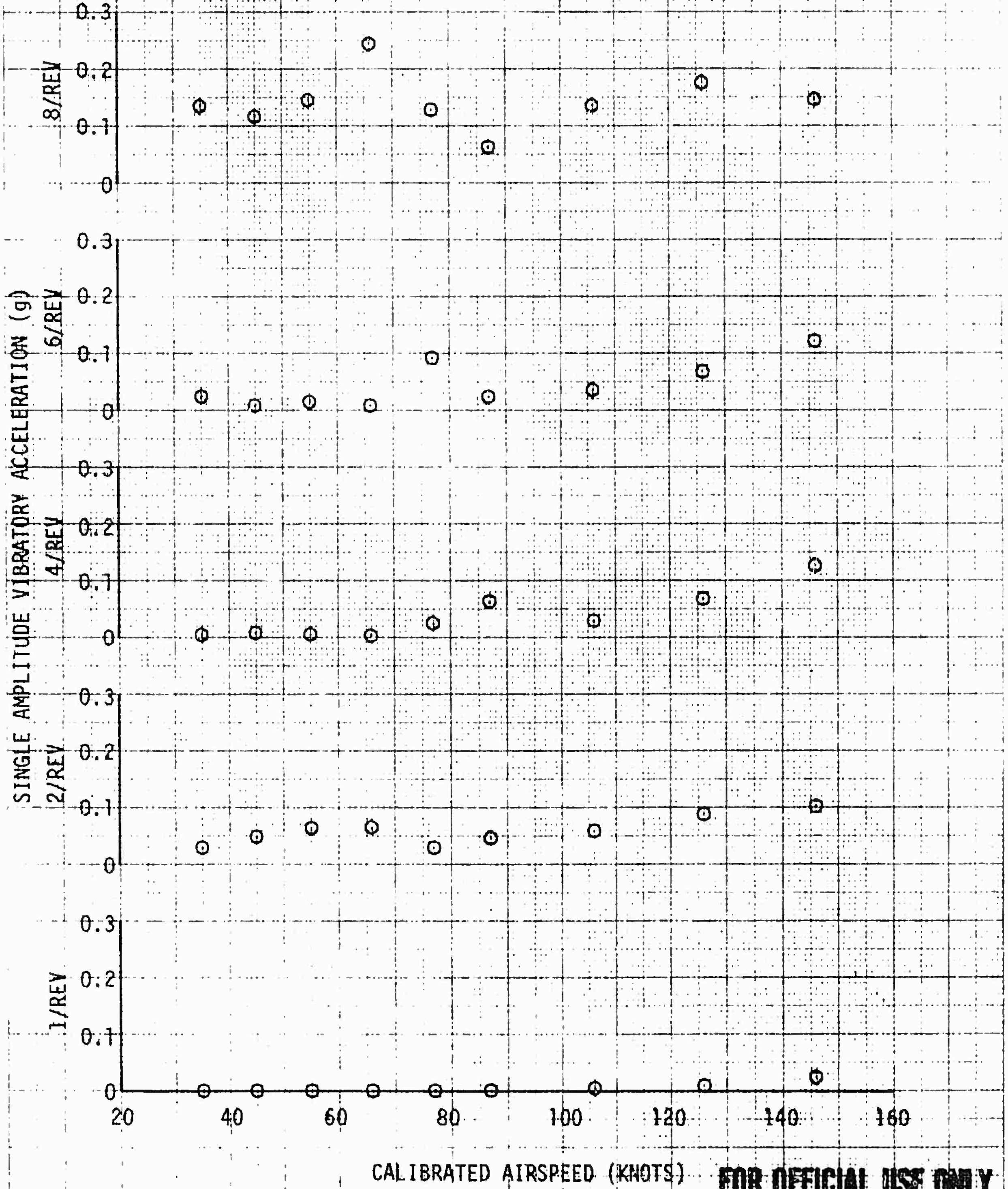

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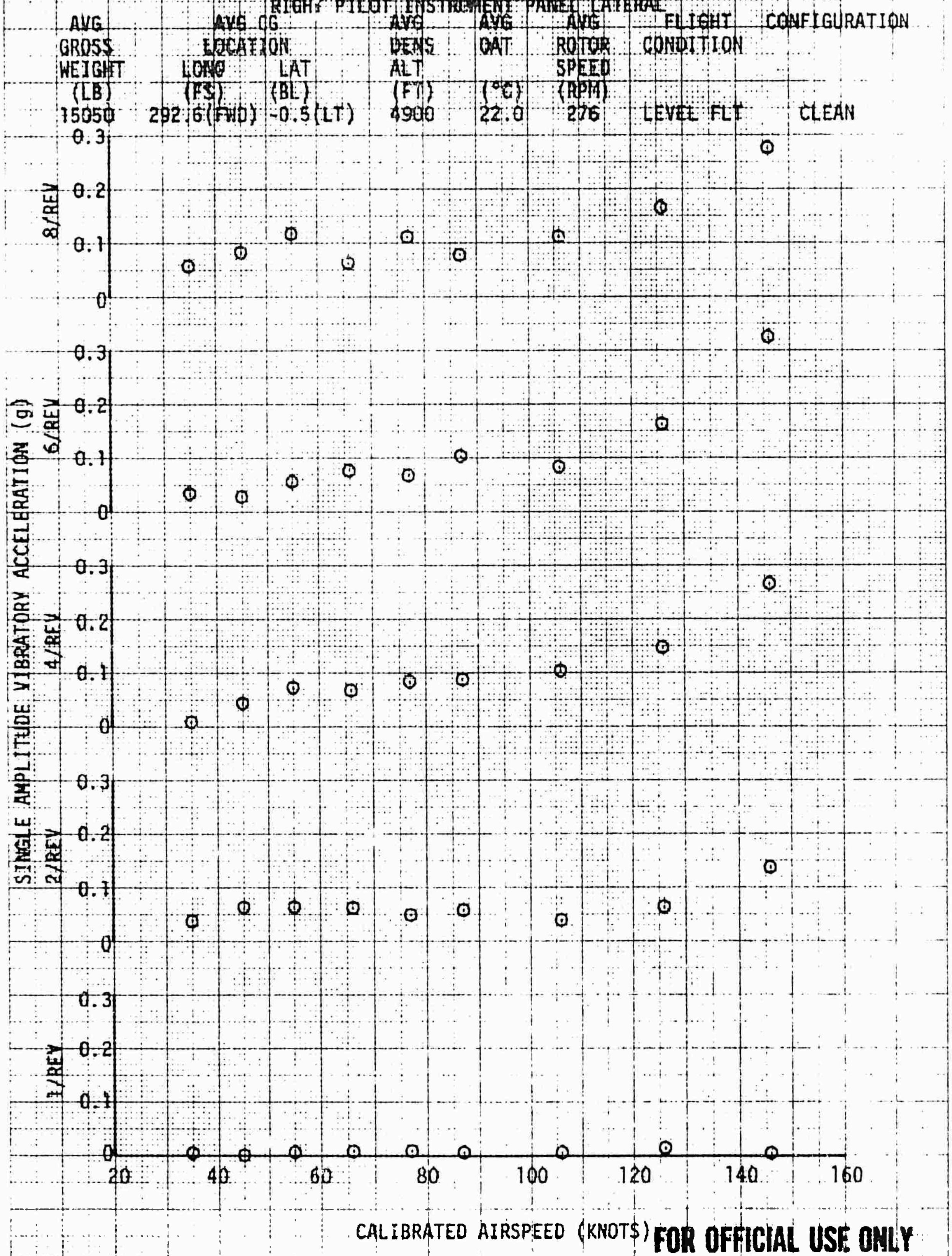
FIGURE 108
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
RIGHT PILOT INSTRUMENT PANEL LATERAL



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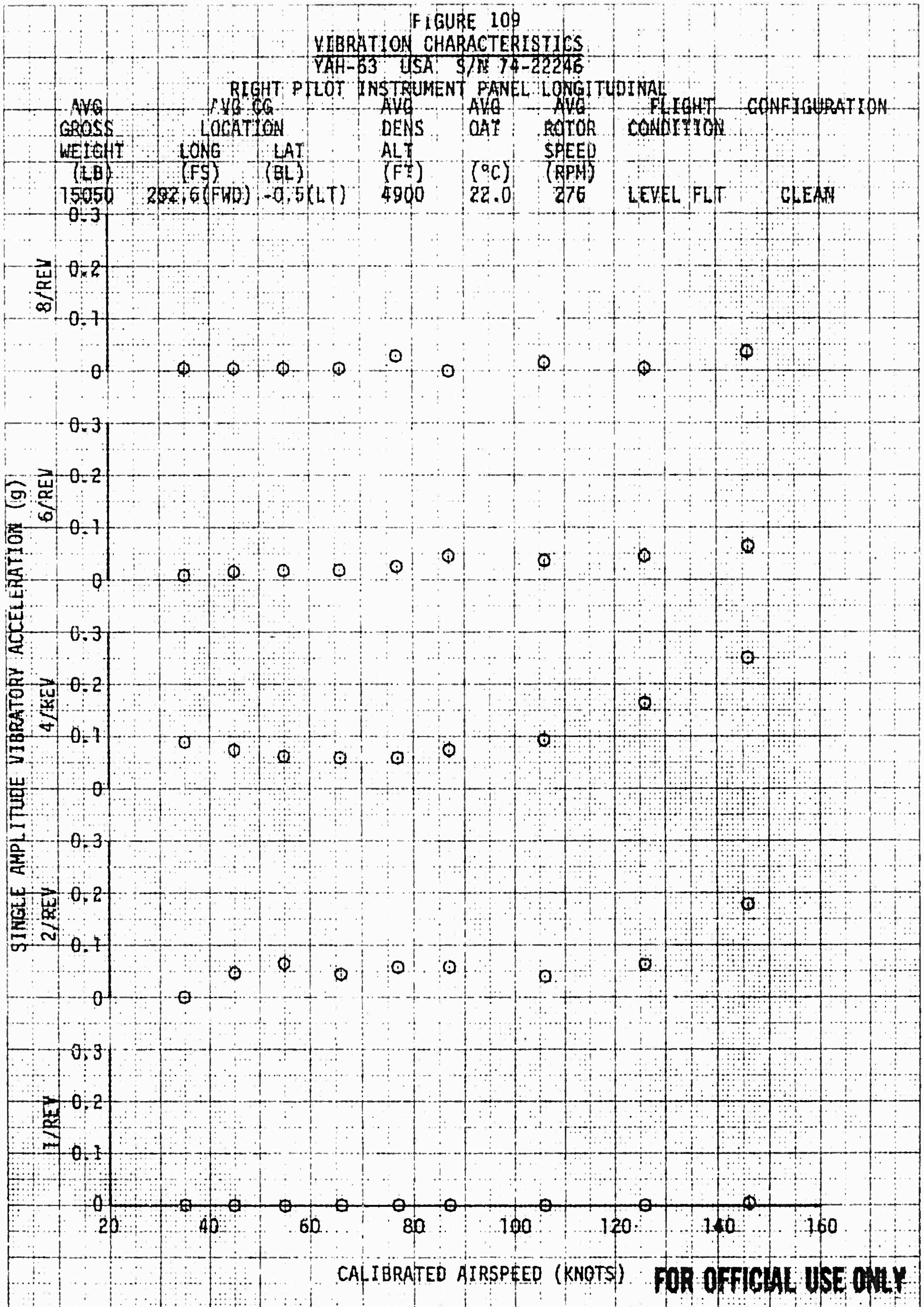

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FIGURE 110
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
COPILOT INSTRUMENT PANEL VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

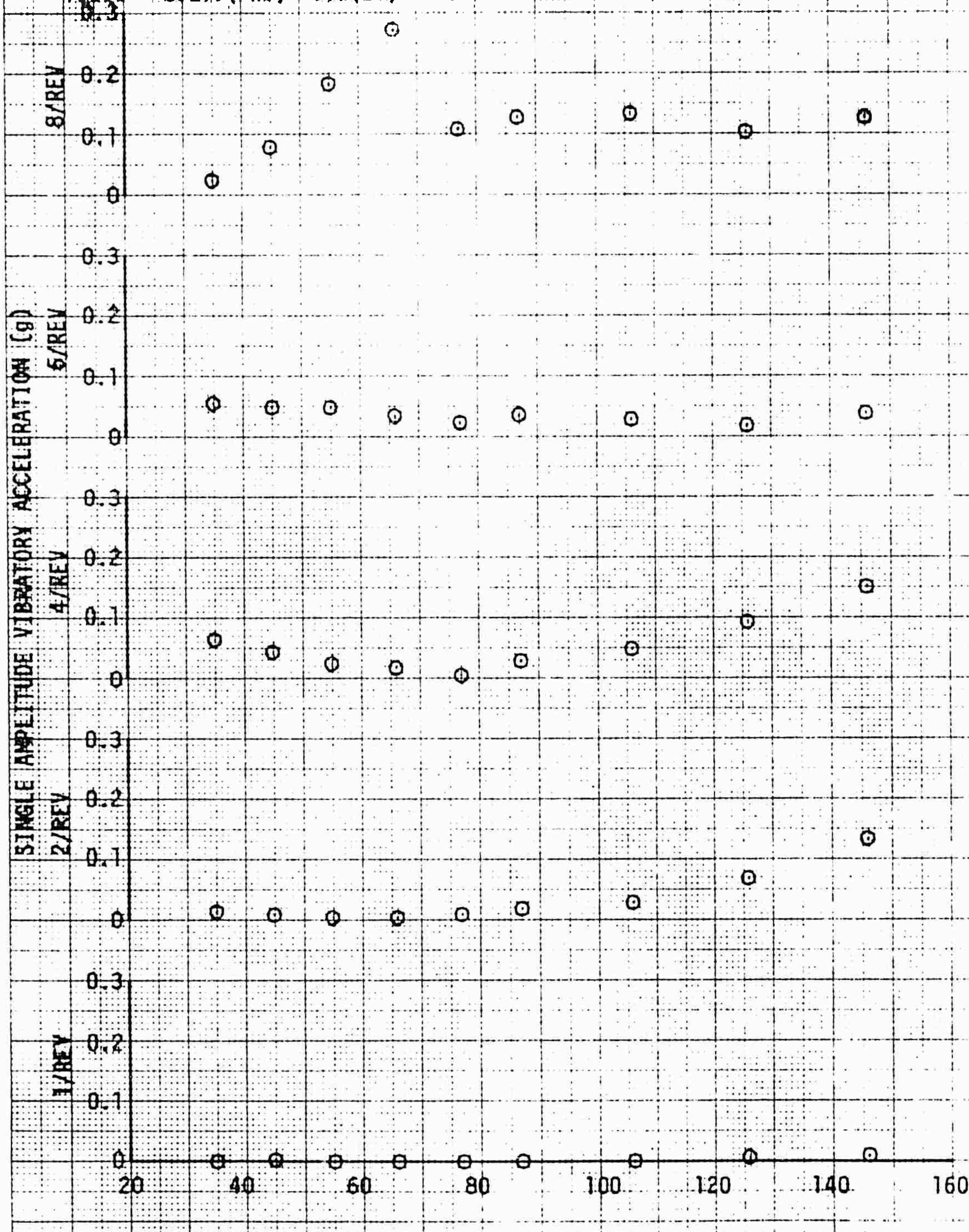

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FIGURE 111
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
COPILOT INSTRUMENT PANEL LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

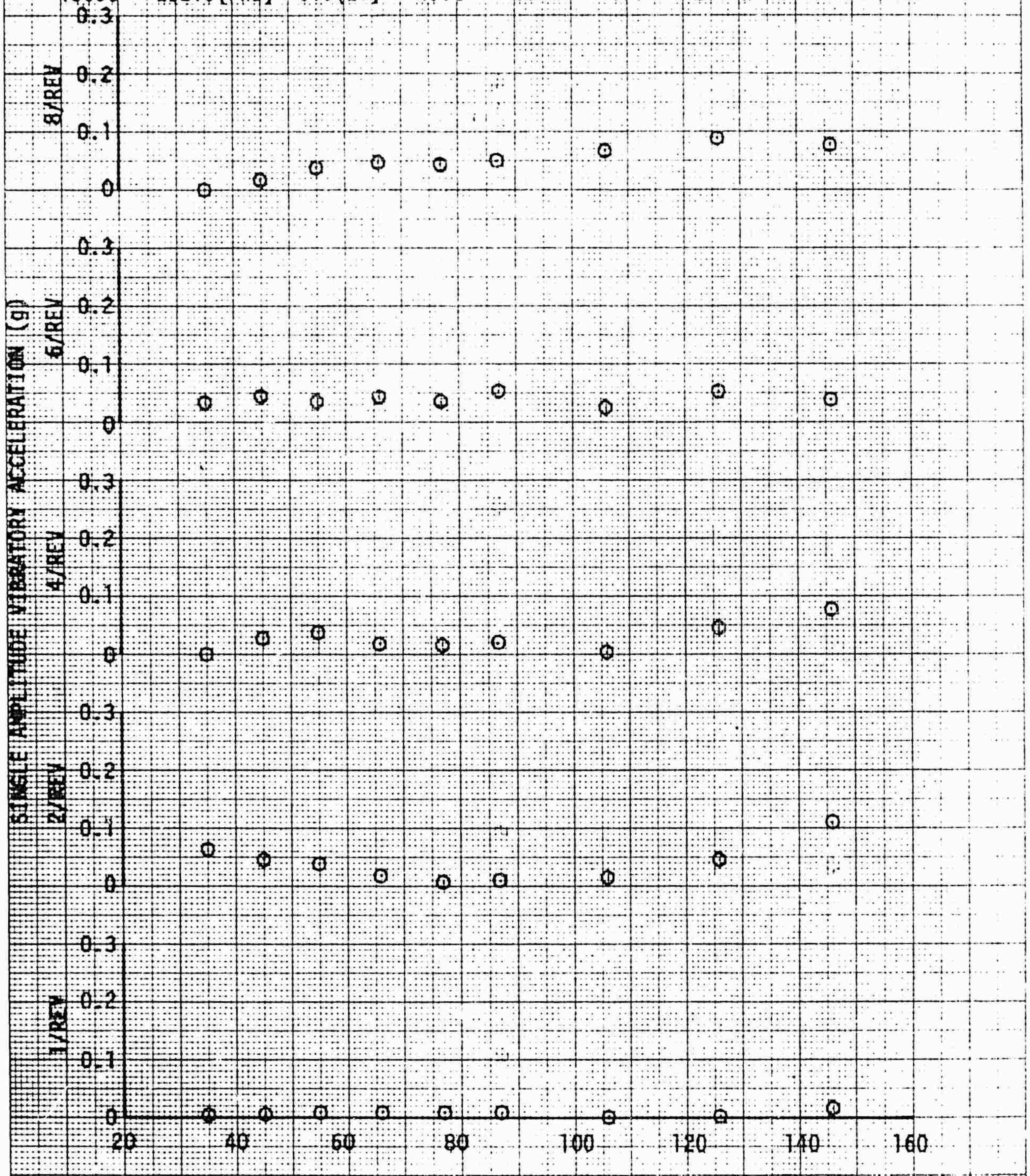

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FIGURE 112
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
COPILOT INSTRUMENT PANEL LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

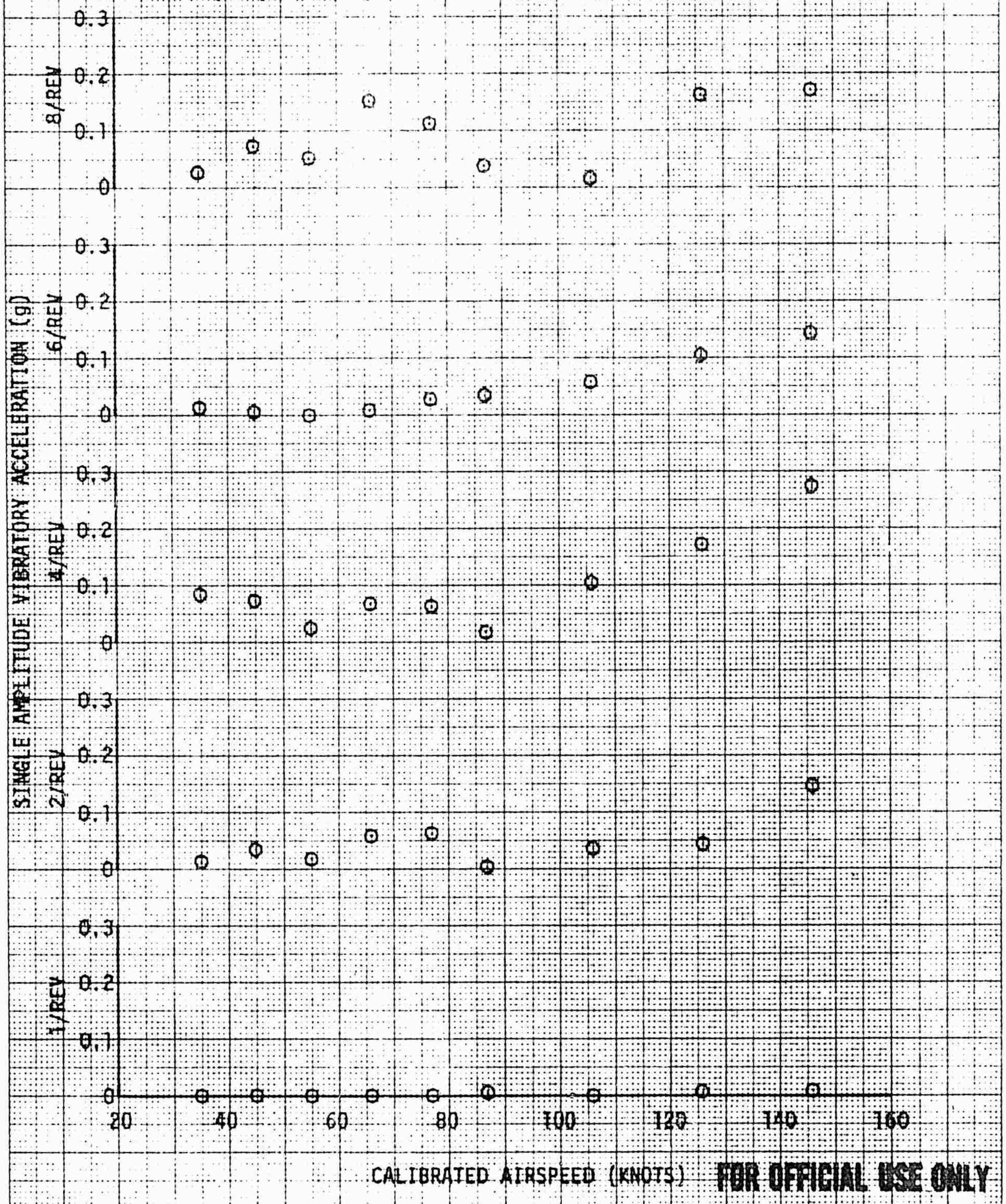

FIGURE 113
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
RIGHT AVIONICS BAY VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

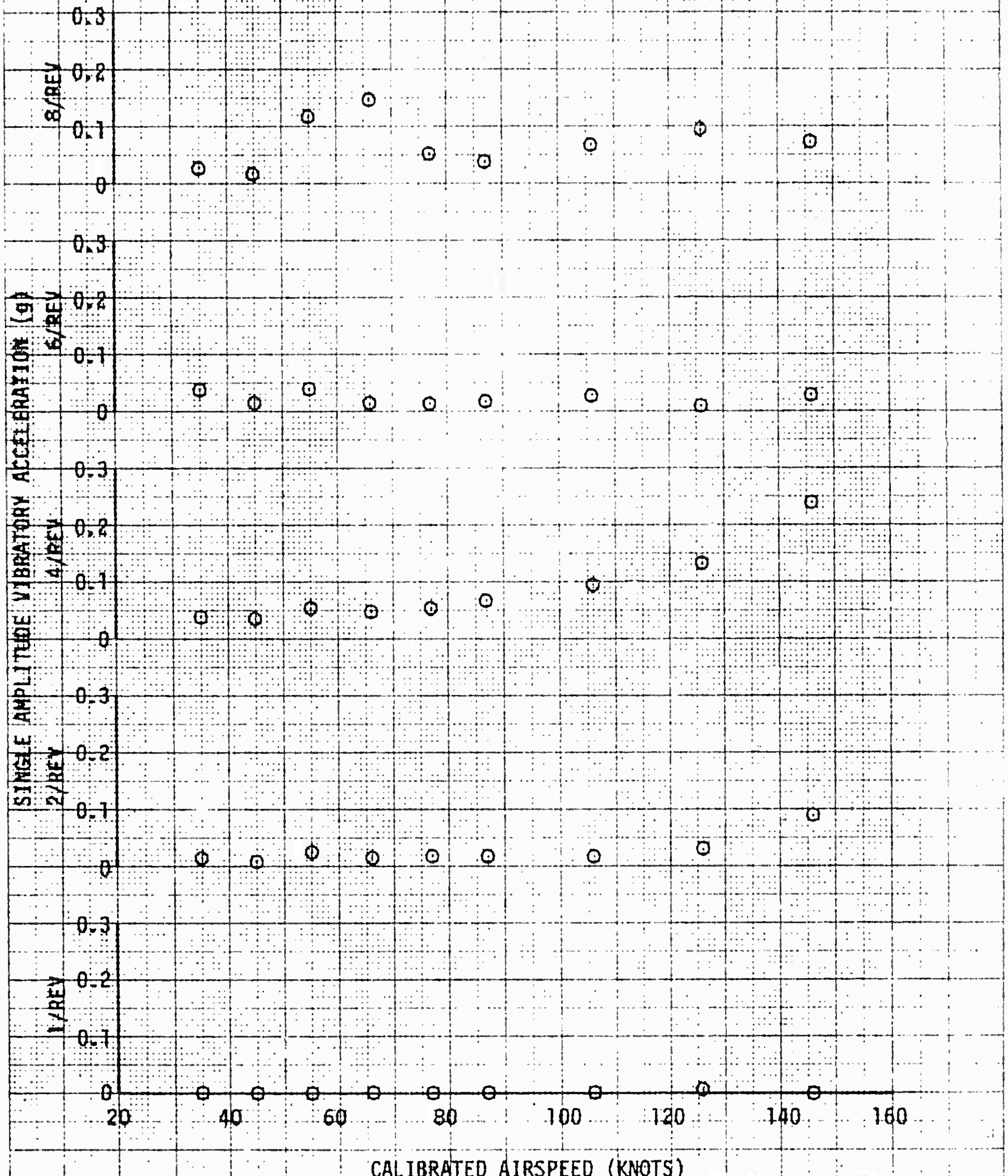

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FIGURE 114
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
RIGHT AVIONICS BAY LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.0(FWD)	+0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

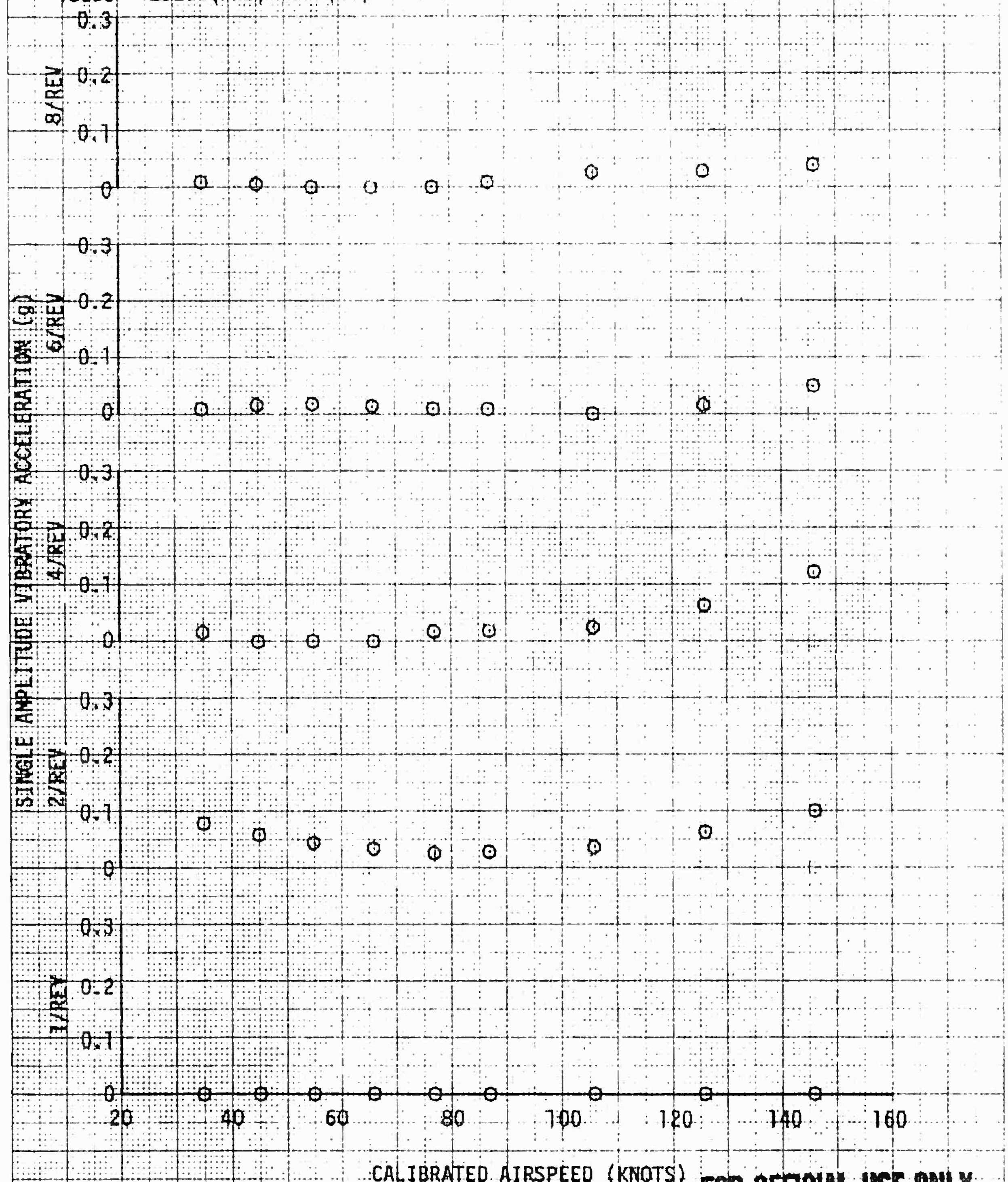

FIGURE 115
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
RIGHT AVIONICS BAY LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

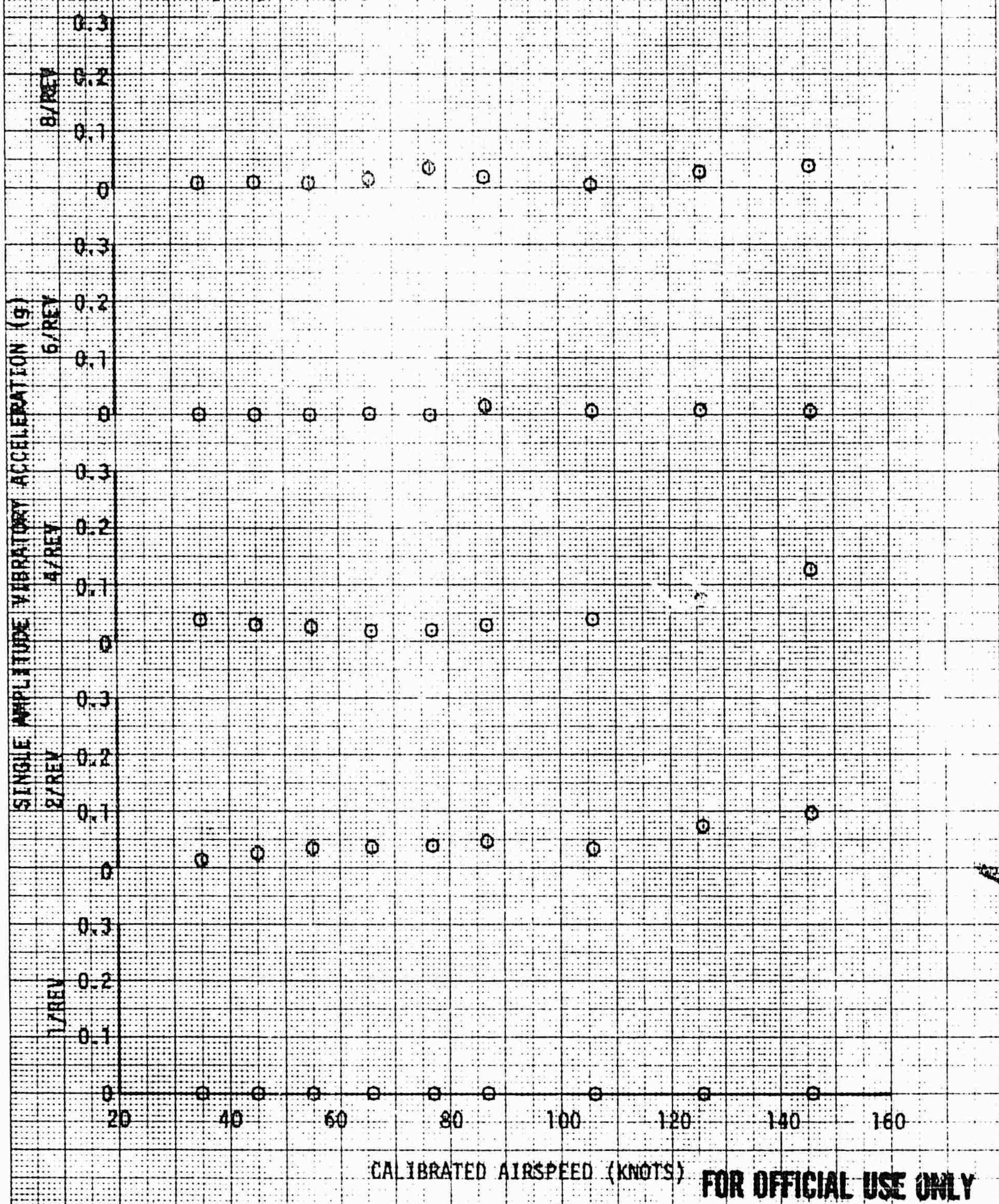

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FIGURE 115

VIBRATION CHARACTERISTICS

YAH-63 USA S/N 73-22246

LEFT AVIONICS BAY VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	+0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

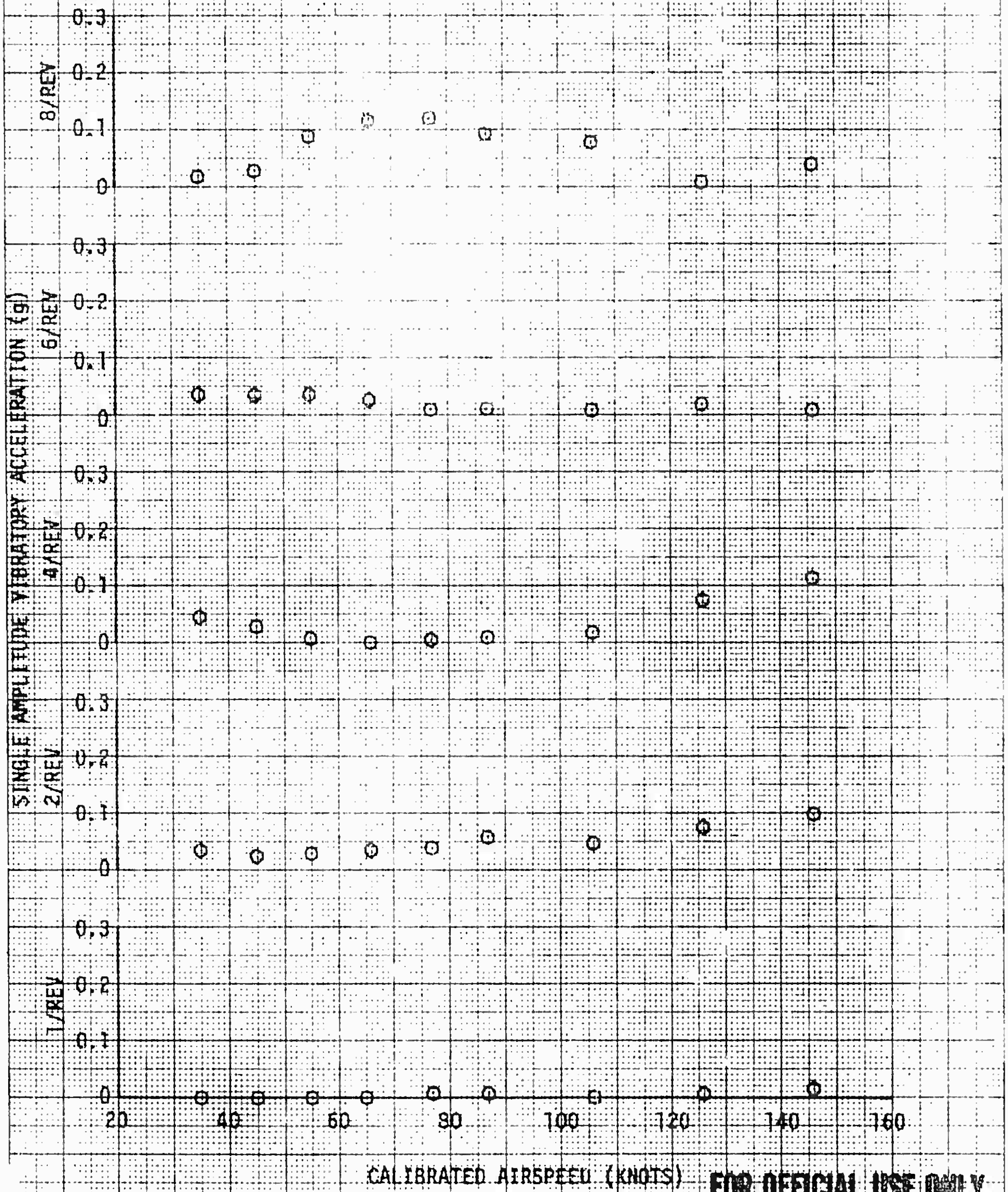

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FIGURE 117
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
LEFT AVIONICS BAY LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

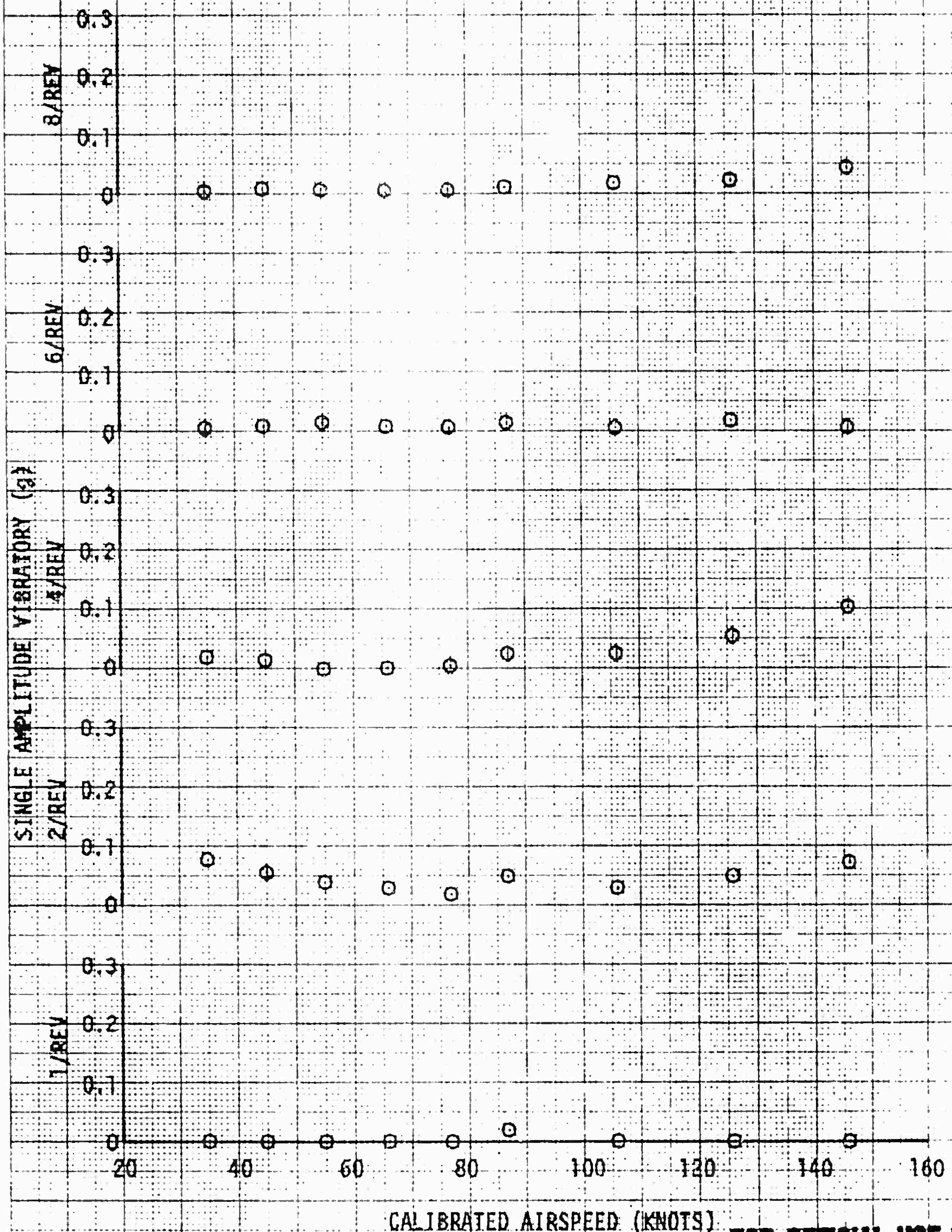

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FIGURE 118

VIBRATION CHARACTERISTICS

YAH-63 USA S/N 74-22246

LEFT AVIONICS BAY LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15050	292.6(FWD)	0.5(LT)	4500	22.0	276	LEVEL FLT	CLEAN

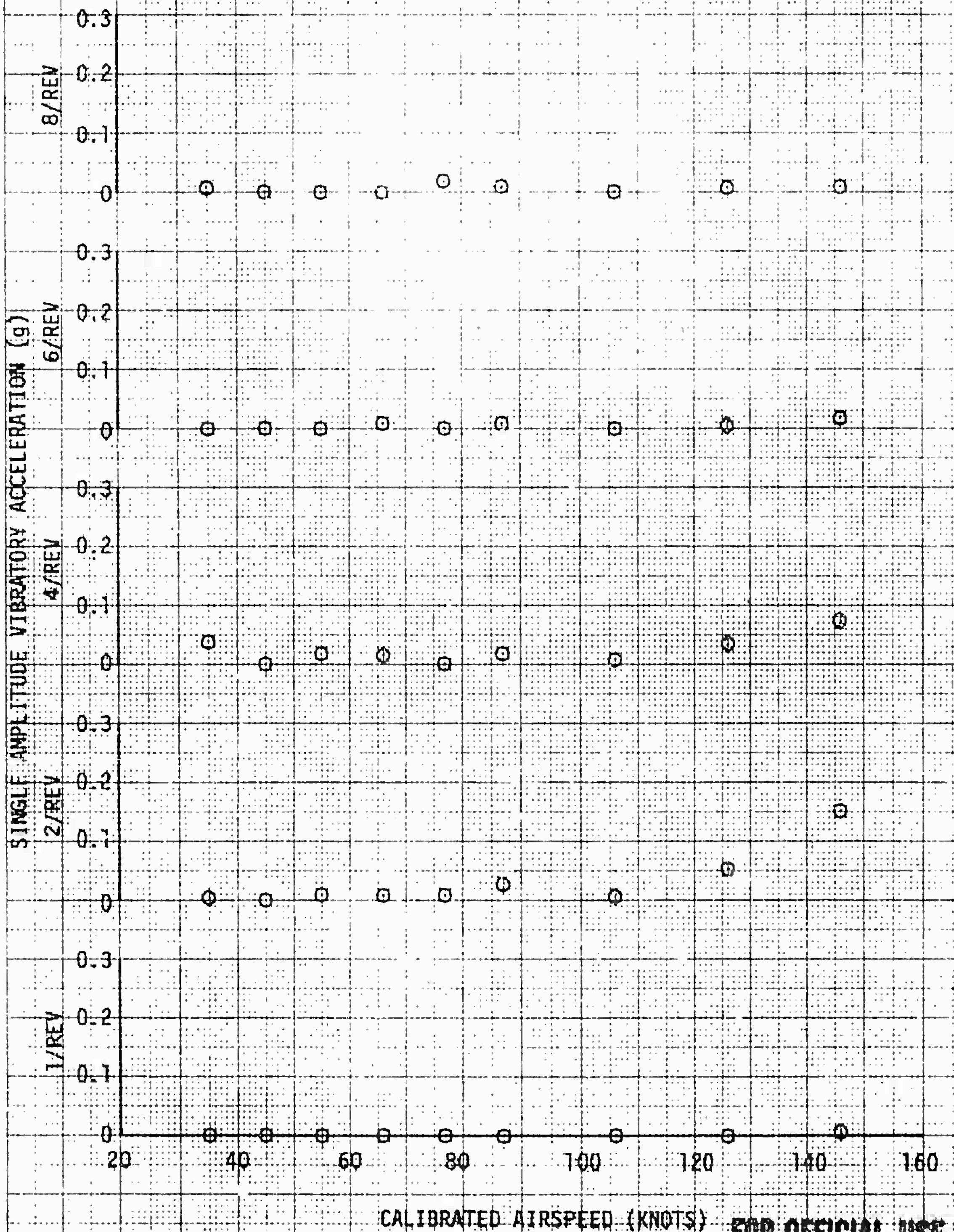

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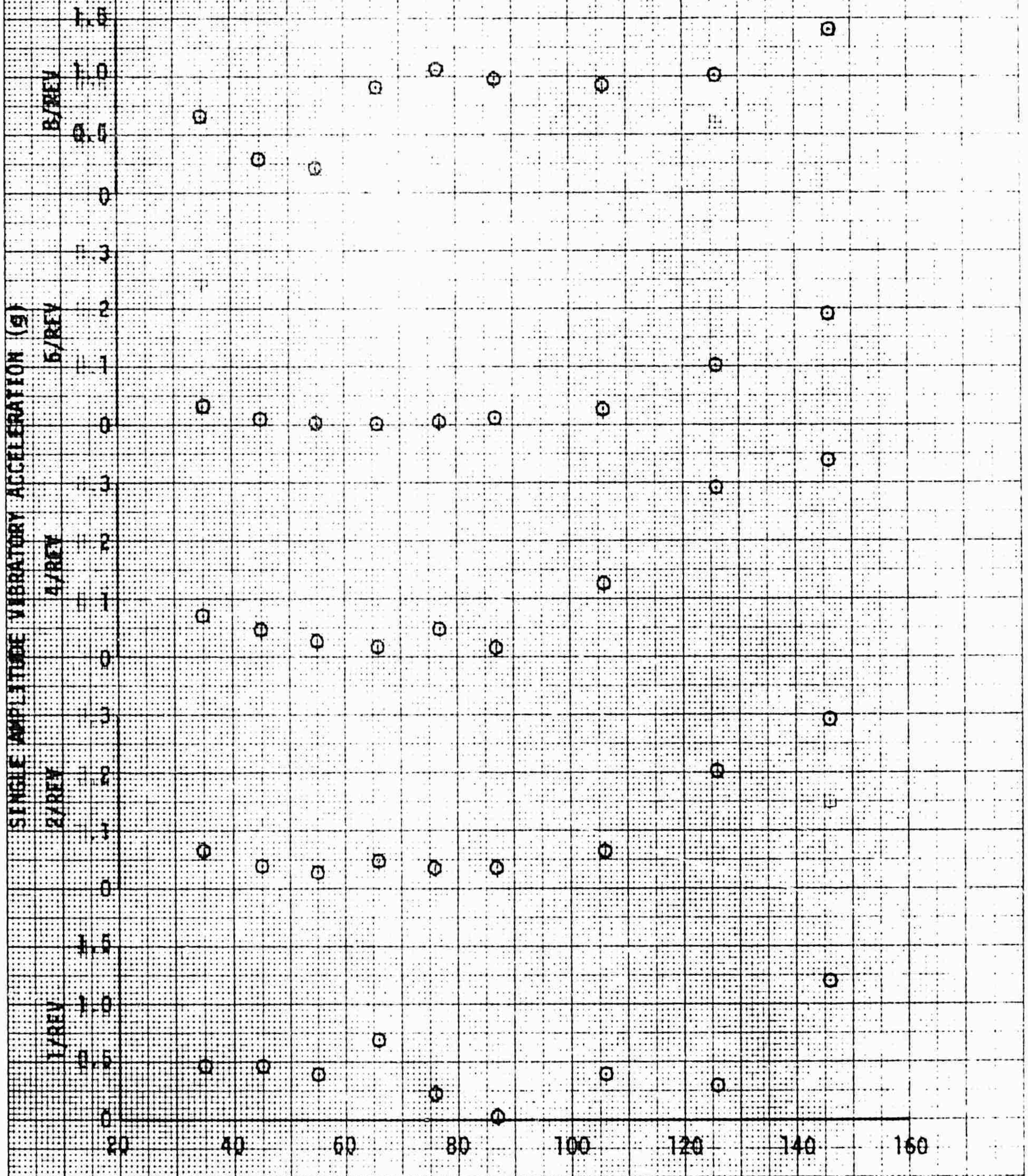
FIGURE 119
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
LEFT WING VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
16050	292.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN



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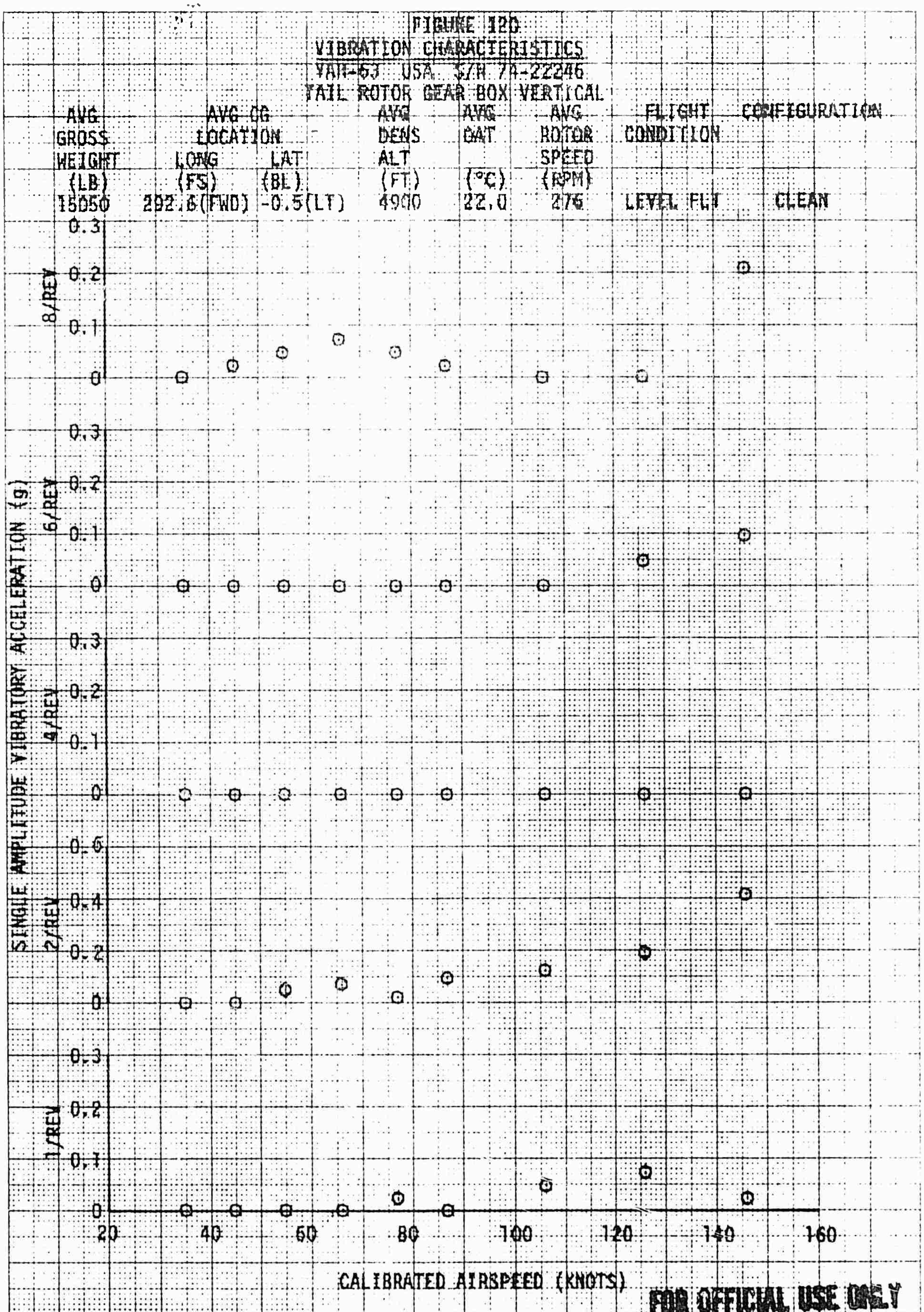

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FIGURE 121
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
MAIN TRANSMISSION VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIGURATION
15080	202.6(FWD)	-0.5(LT)	4900	22.0	276	LEVEL FLT	CLEAN

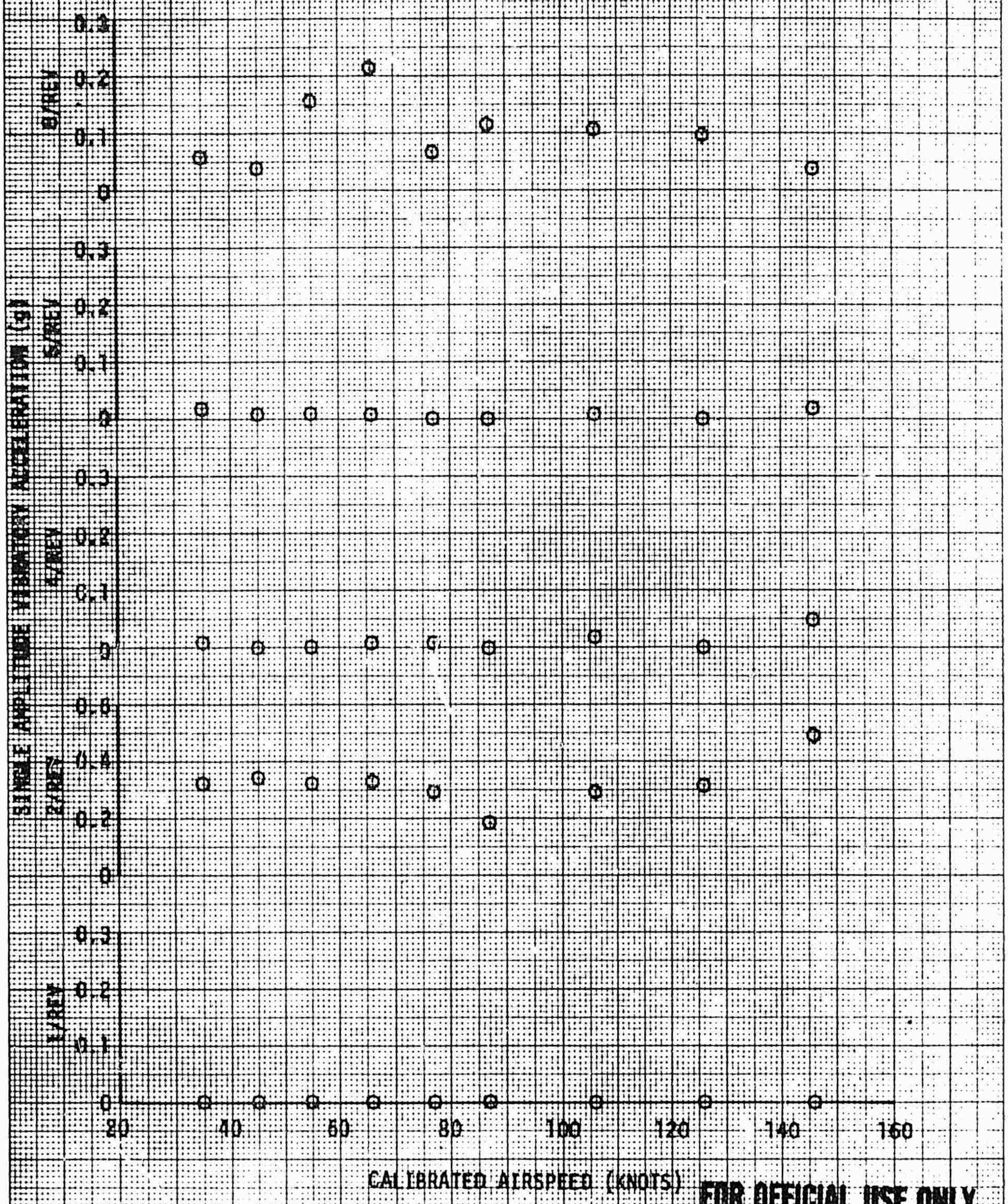

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FIGURE 122
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT SEAT VERTICAL

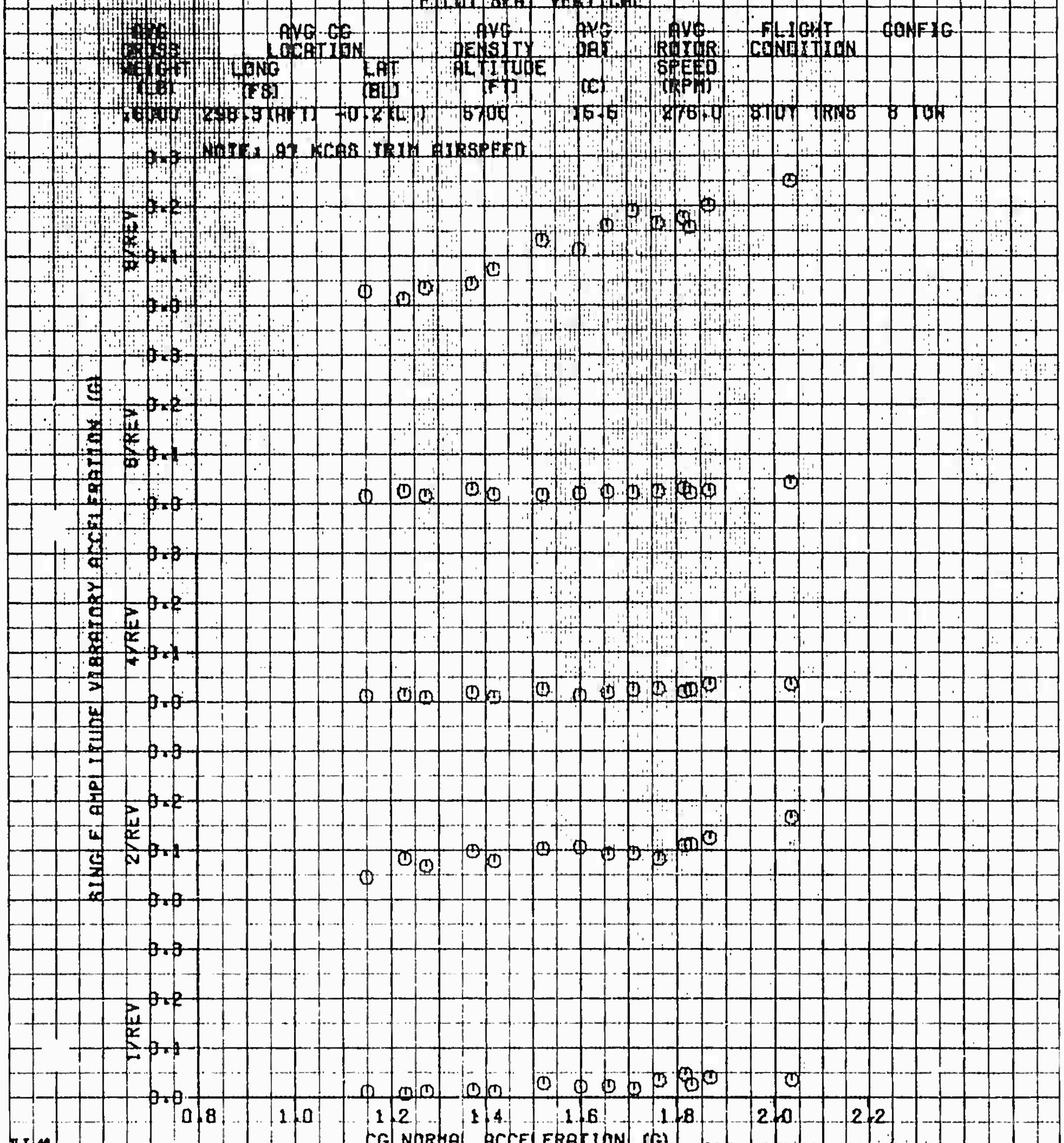
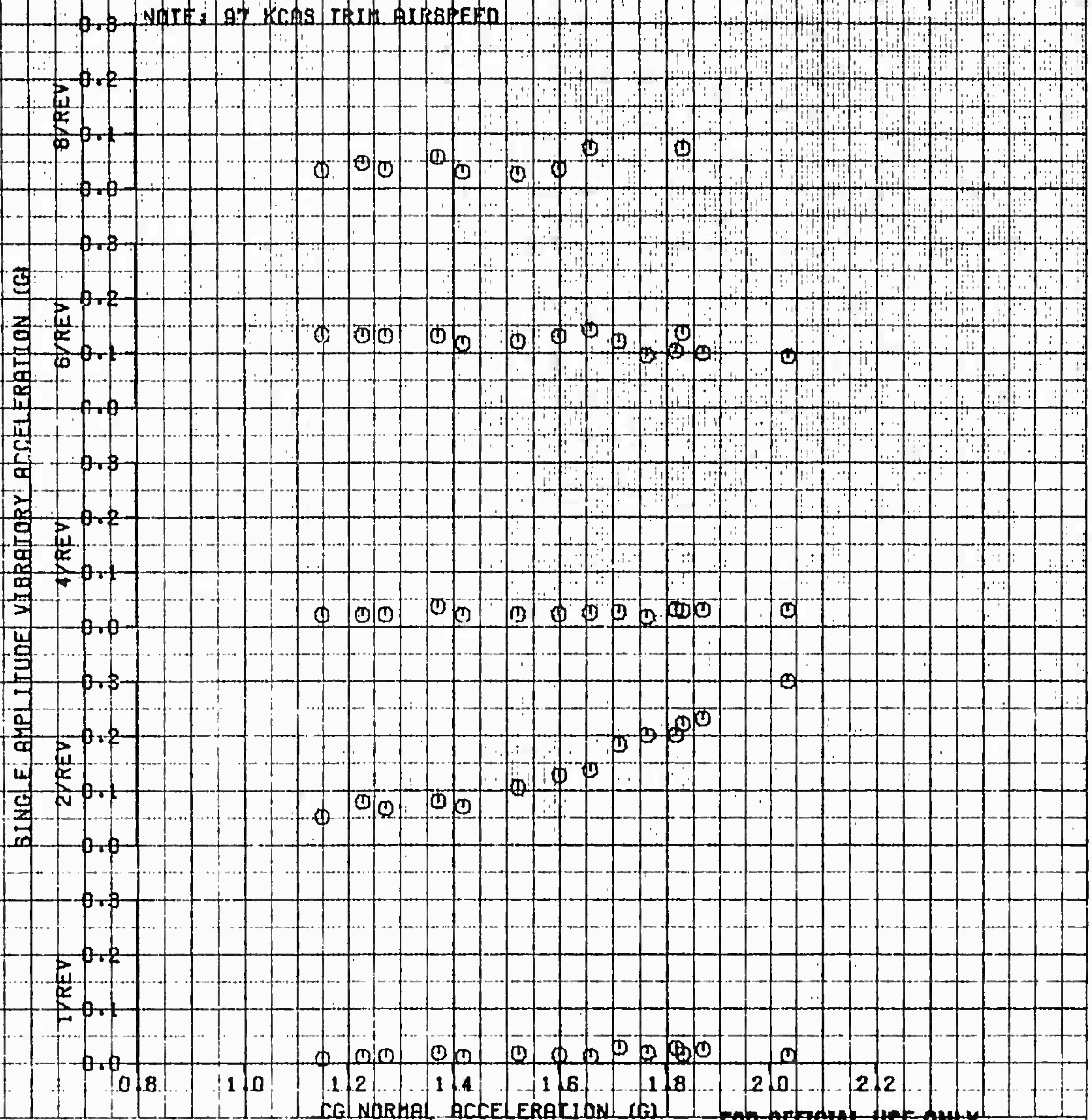


FIGURE 123
VIBRATION CHARACTERISTICS
YAH-63 USA S/N 74-22246
PILOT SEAT LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
16000	298.3(AFT)	-0.2(LT)	5700	16.5	276.0	STDY TRNS	8 TOW

NOTE: 97 KCAS TRIM AIRSPEED



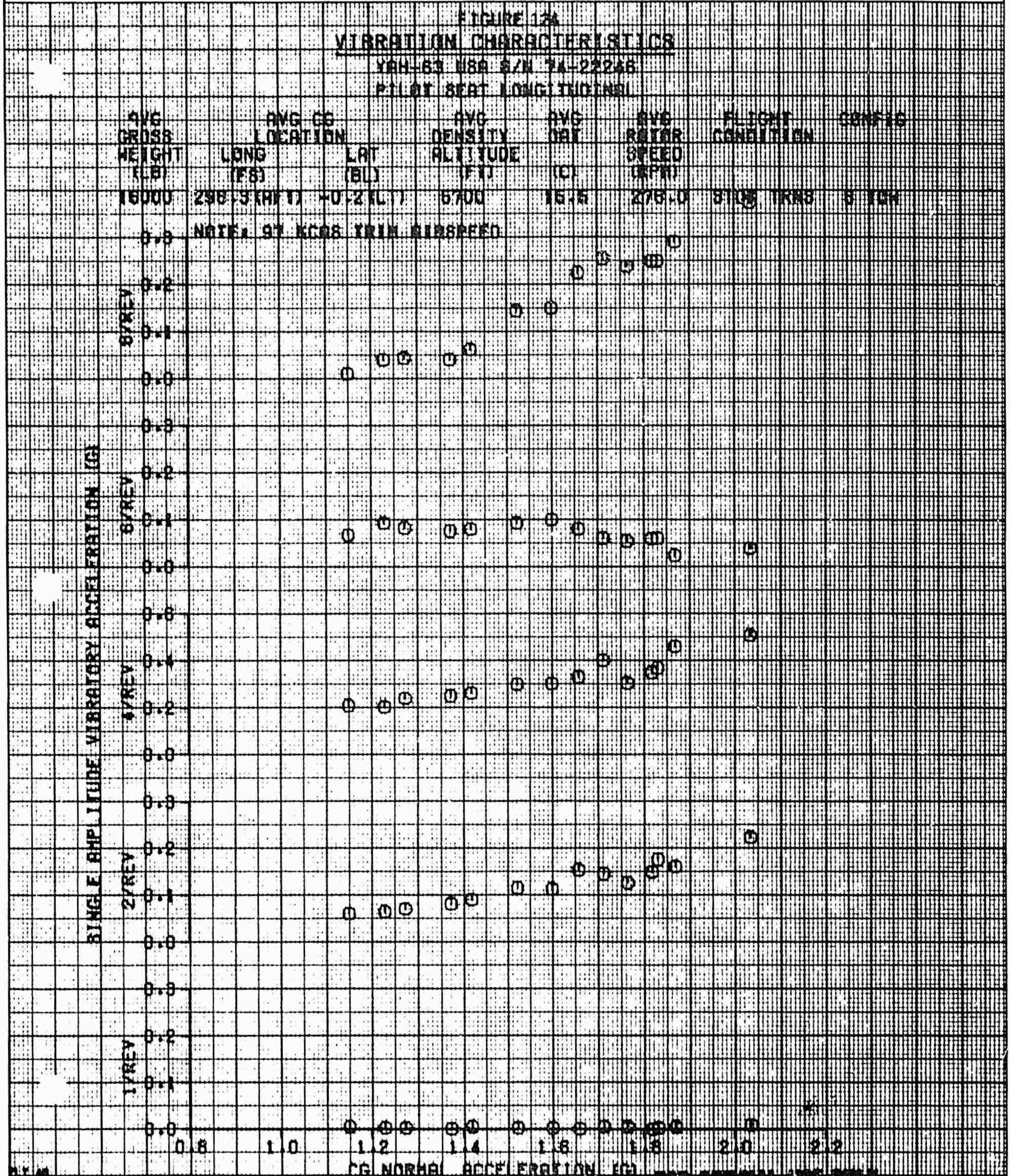


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FIGURE 125

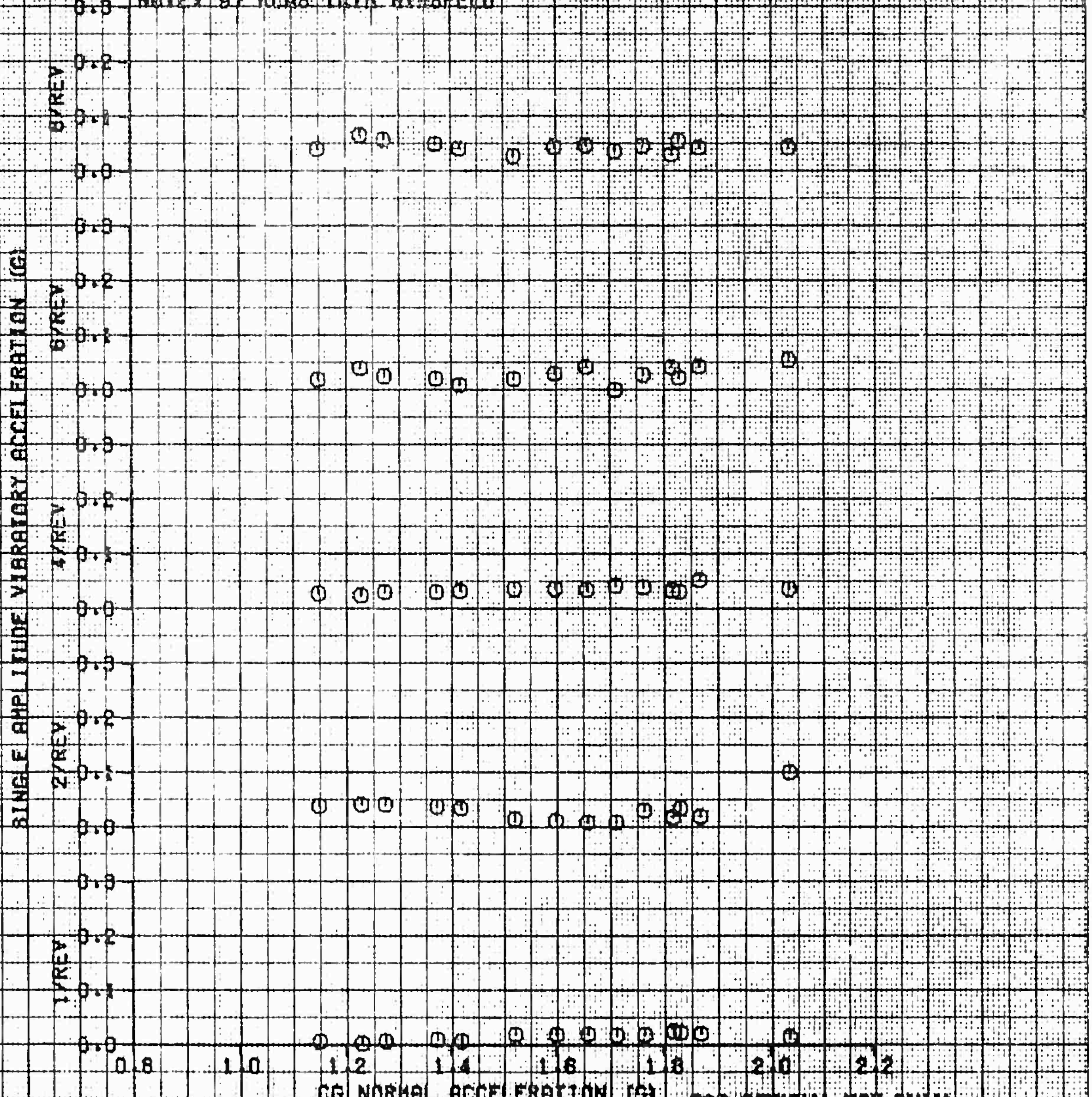
VIBRATION CHARACTERISTICS

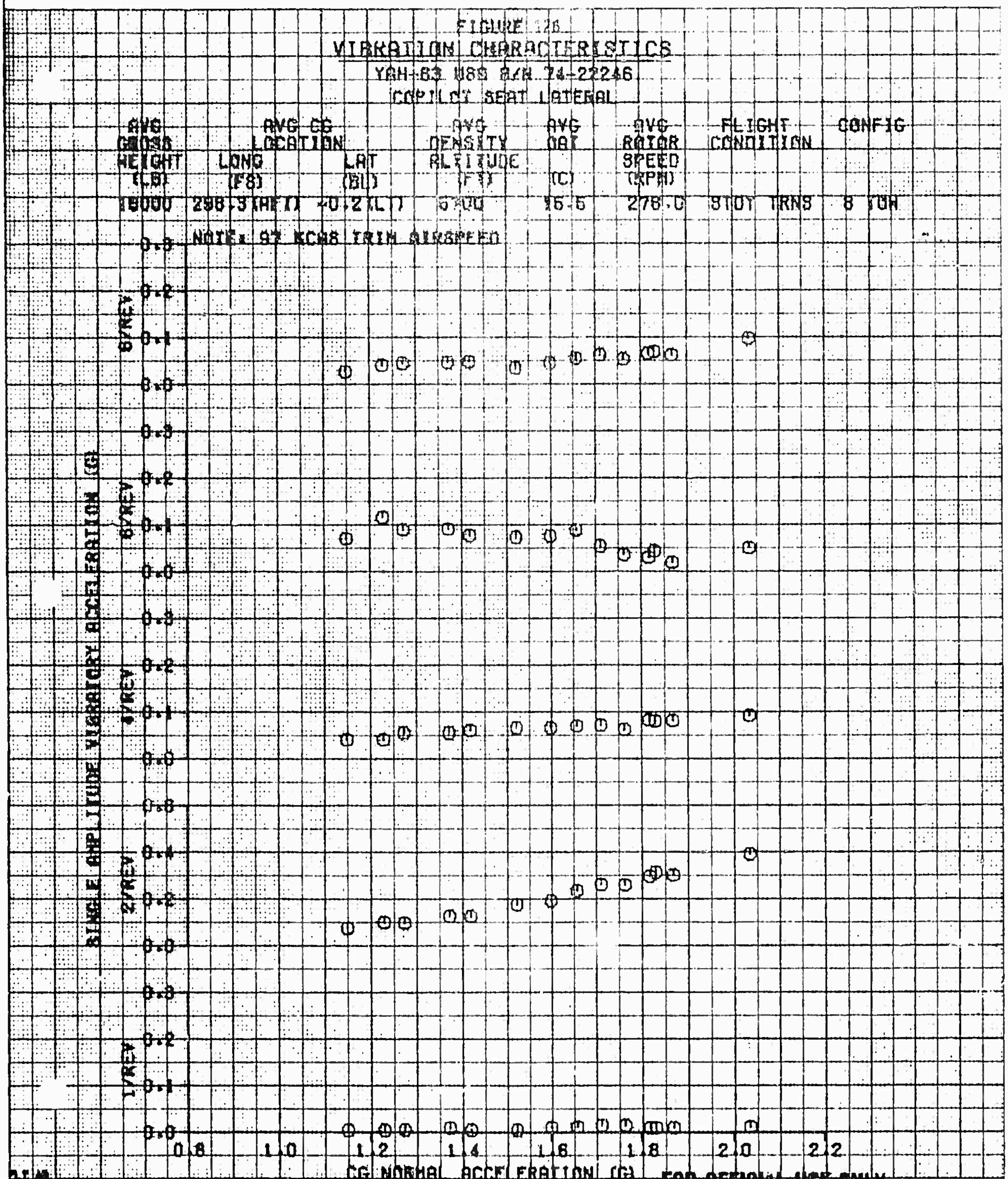
YAH-63 USA S/N 74-22246

COPILOT SEAT VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
18000	298.9(AFT)	-0.2(LT)	6700	15.5	275.0	STDY TRNS	3 TUR

NOTE: 97 KCAS TRIM AIRSPEED





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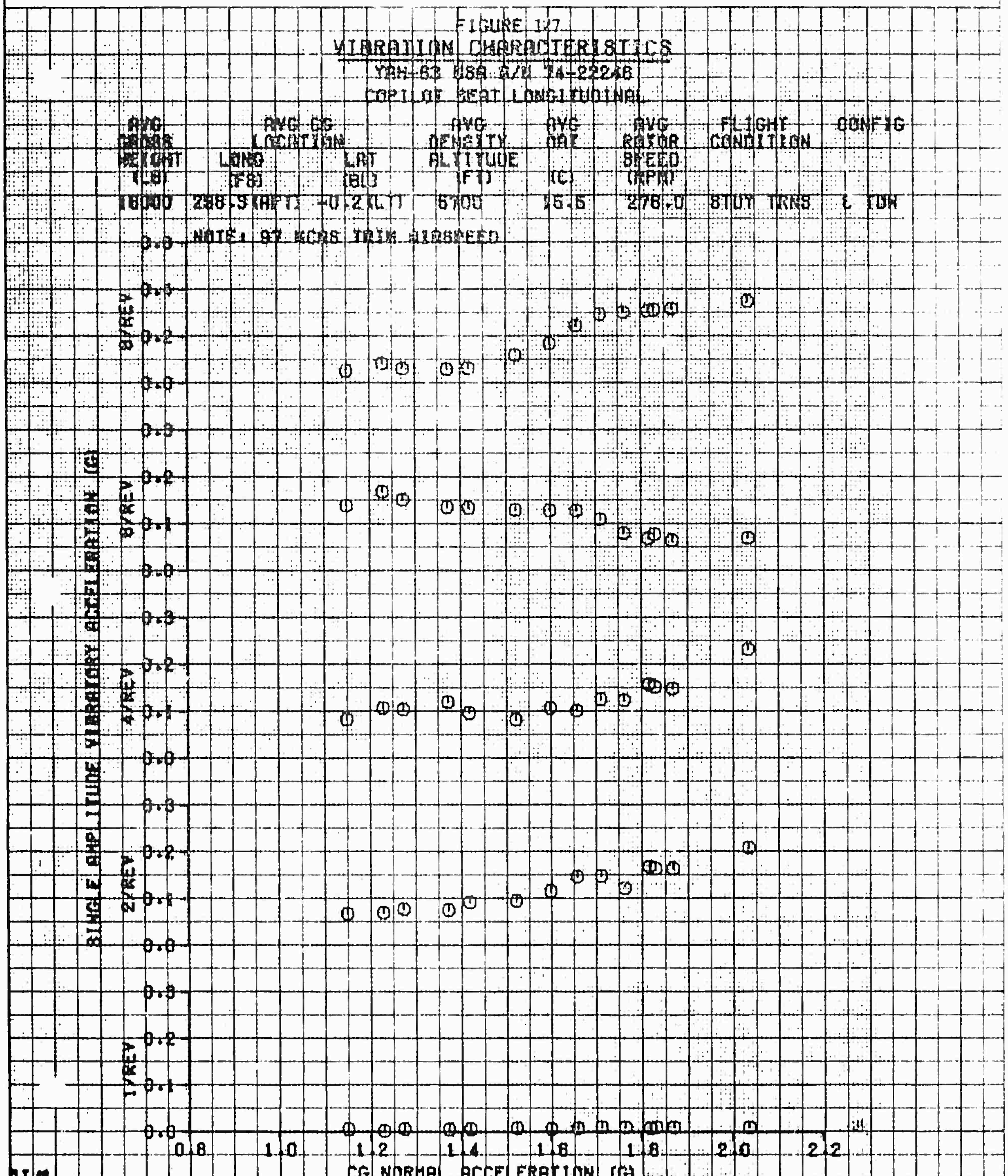

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FIGURE 120

VIBRATION CHARACTERISTICS

YAH-63 USA S/N 74-22248

C.G. VERTICAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
16000	298.3(AFT)	+0.2(LT)	5700	15.5	278.0	STDY TRNS	8 TOW

NOTE: 97 KCAS TRIM AIRSPEED

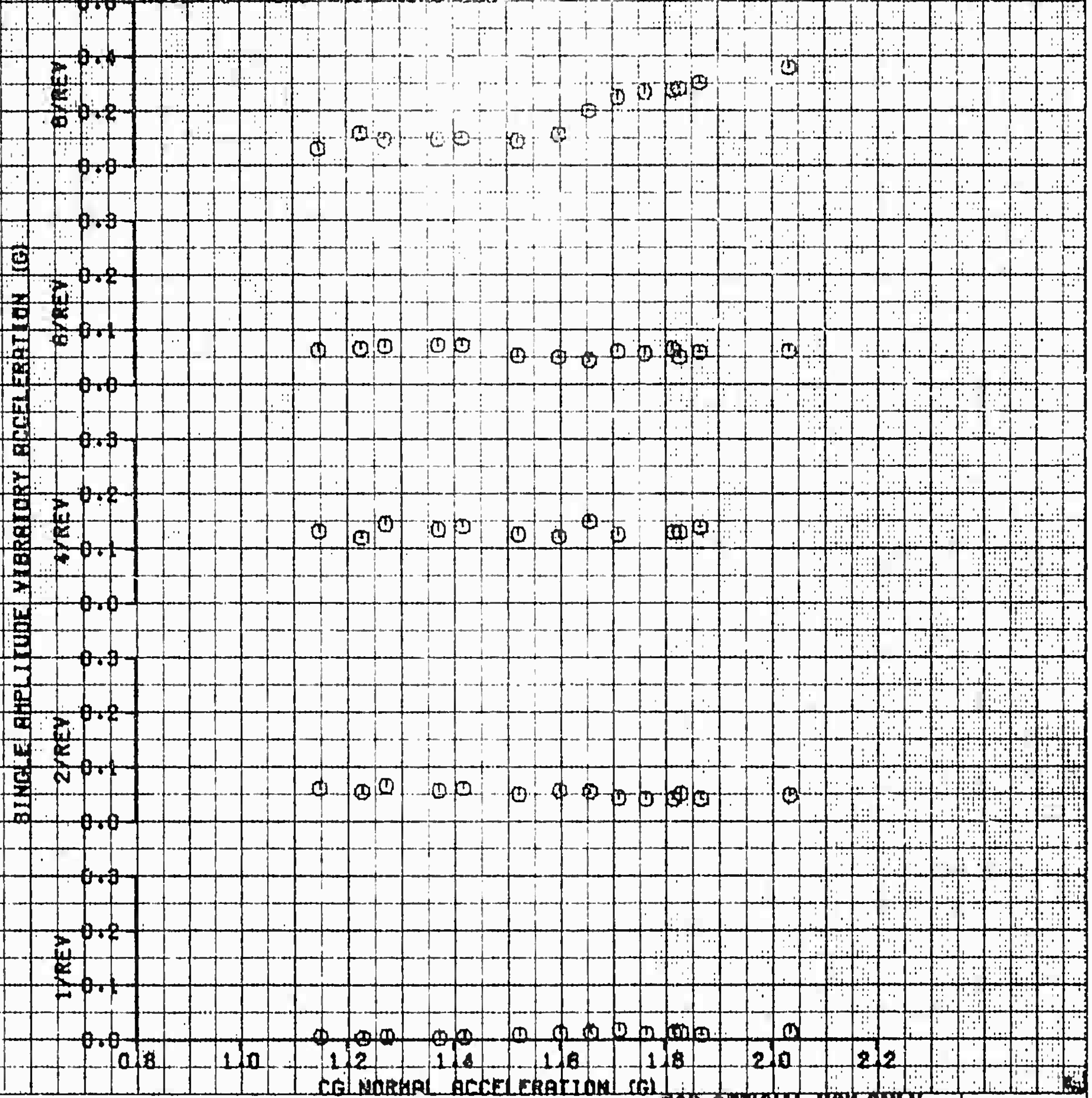

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FIGURE 129

VIBRATION CHARACTERISTICS

YAH-63 USA S/N 74-22246

C.G. LATERAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
16000	298.9(AFT)	-0.2(C)	6700	16.5	278.0	STDY TRNS	8 TOW

NOTE: 97 KCAS TRIM AIRSPEED

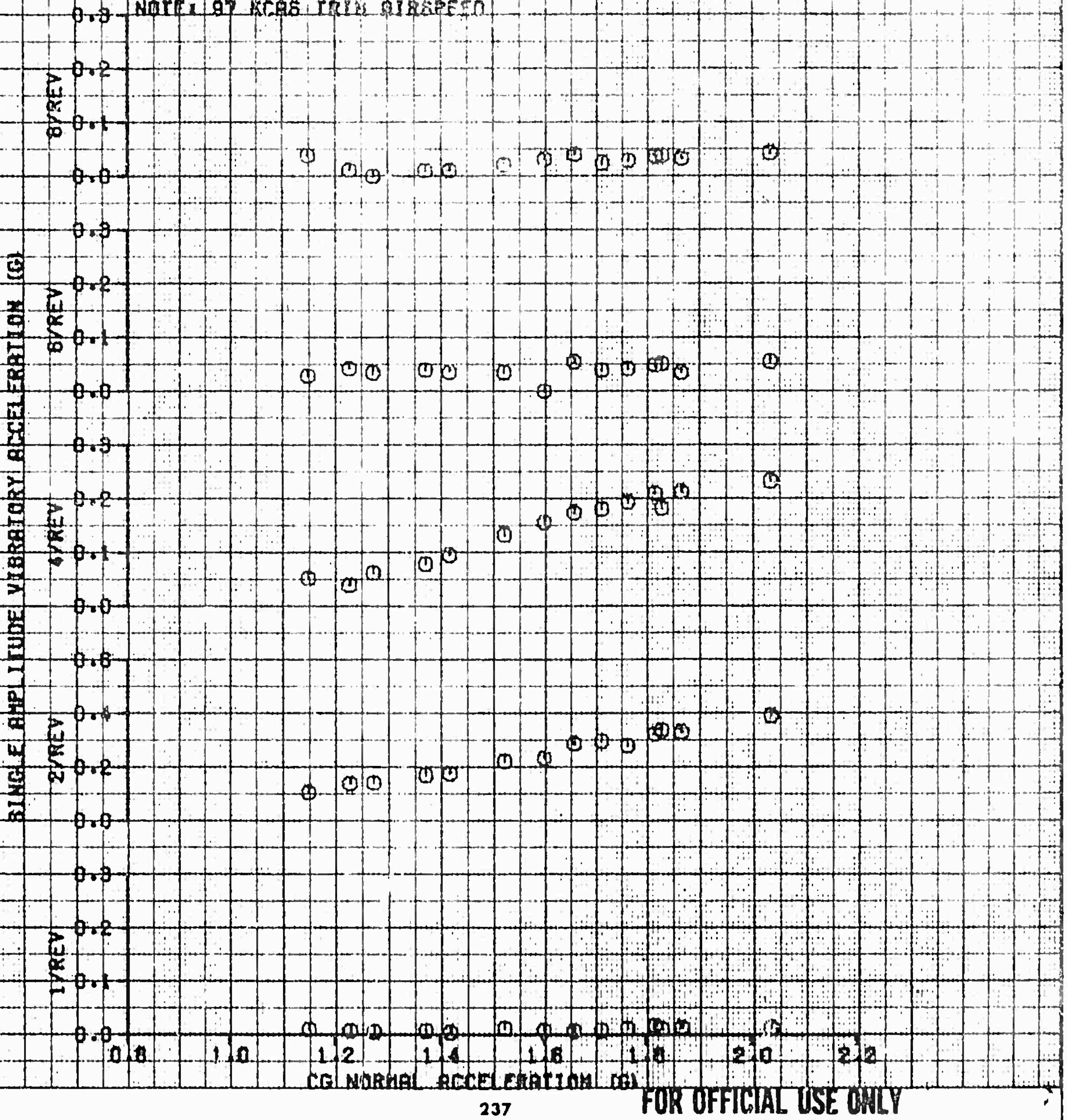

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FIGURE 130

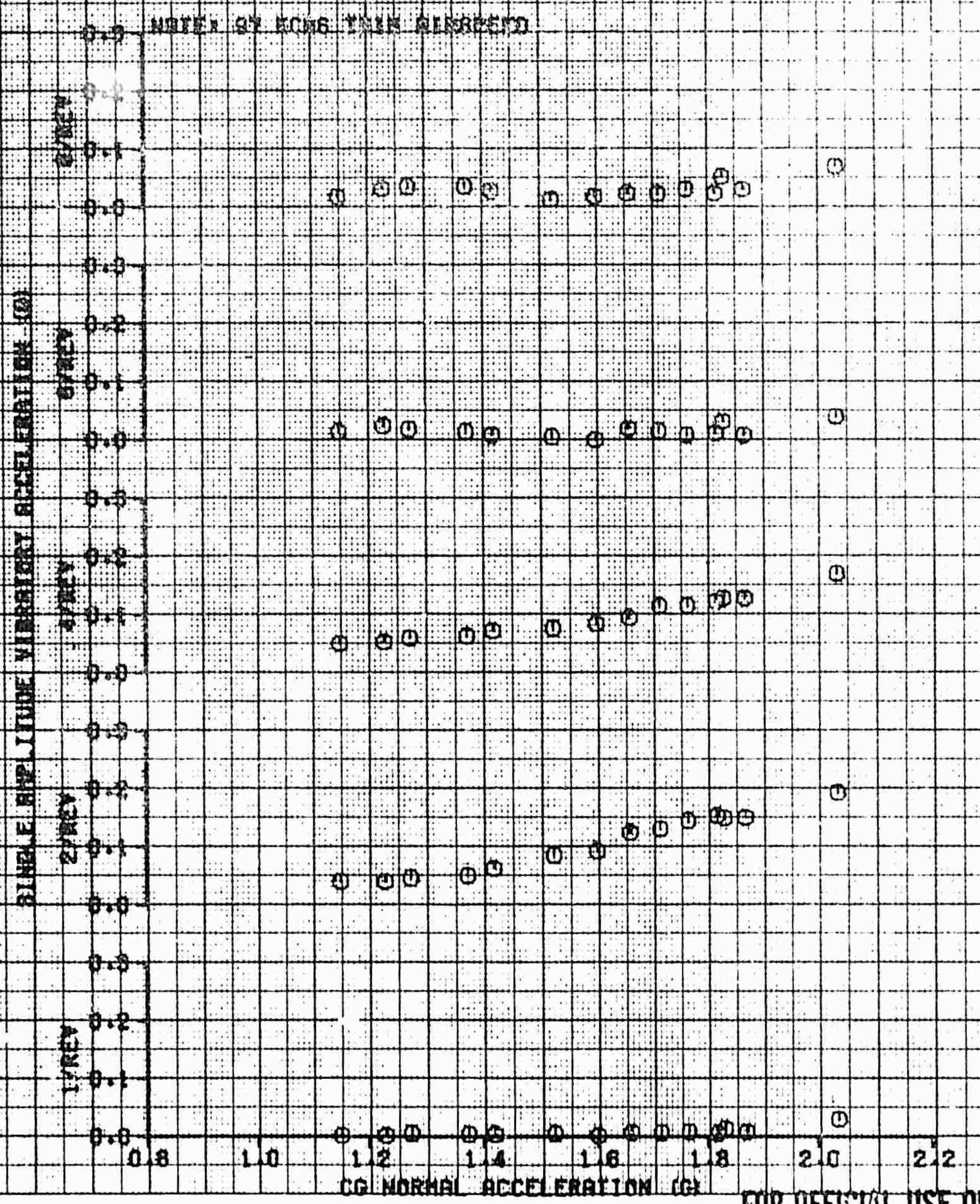
VIBRATION CHARACTERISTICS

YAH-63 USA S/N 74-22246

C G LONGITUDINAL

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENSITY ALTITUDE (FT)	AVG OAT (C)	AVG ROTOR SPEED (RPM)	FLIGHT CONDITION	CONFIG
16000	288.3(AFT)	+0.2(LT)	5700	15.6	276.0	STDY TRNS	C TOW

NOTE: 97 KCAS TRIM AIRSPEED



FOR OFFICIAL USE ONLY

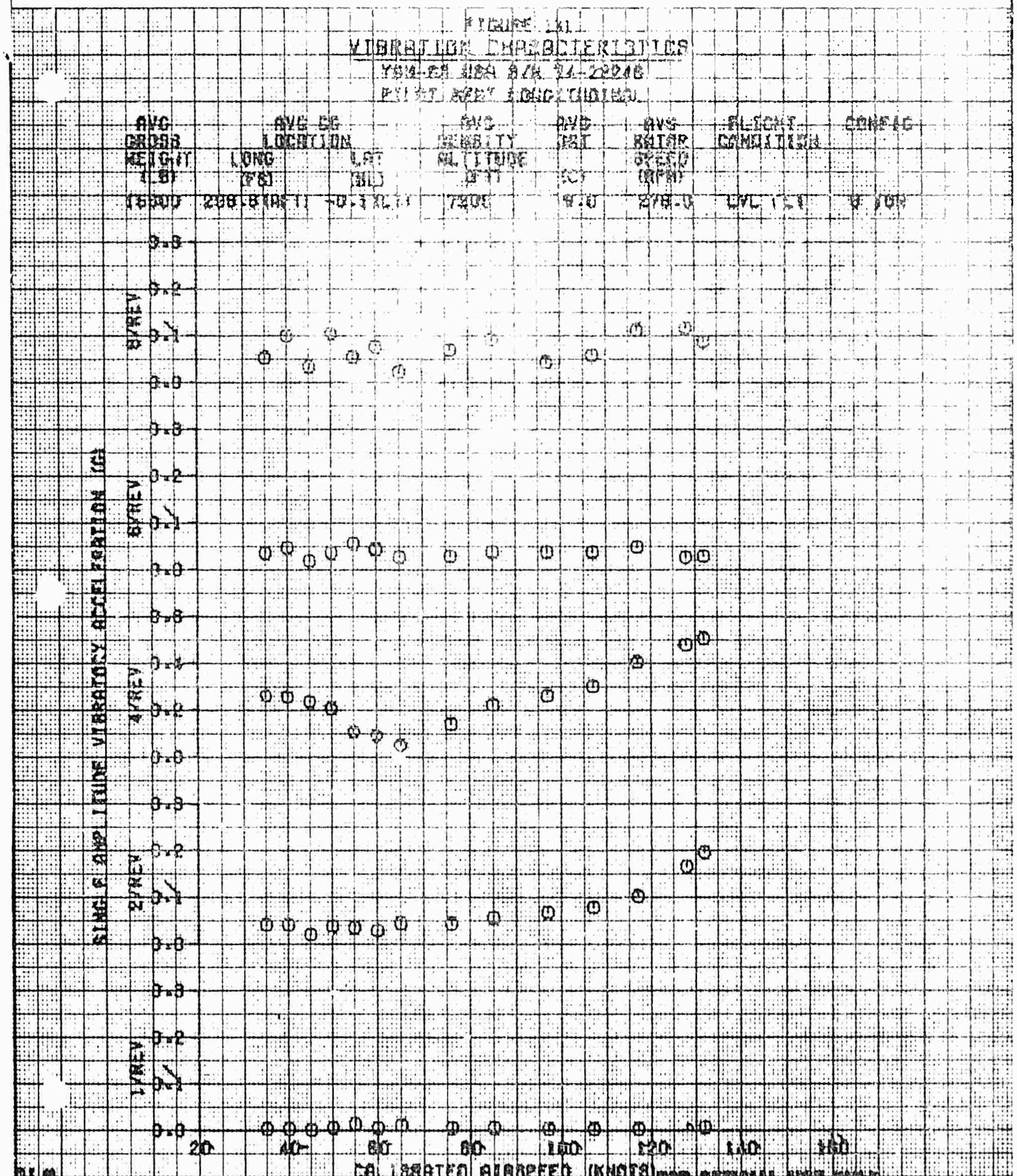


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FIGURE 182

VIBRATION CHACTERISTICS

YAH-63 USA S/N 74-22246

LOCATION PILOT INSTRUMENT PANEL | FUNDAMENTAL FREQUENCY IS 4.60 HZ

AXIS LONGITUDINAL RT

GROSS WEIGHT -LB	LONG CG FS -IN.	LAT CG BL -IN.	DENSITY ALTITUDE -FT	OUTSIDE AIR TEMPERATURE -DEG C	ROTOR SPEED -RPM	TRIM CALIB. AIRSPEED -KTS	FLIGHT CONDITION	CONFIGURATION
15000	298.0	-0.5	6000	31.0	276	40	LEVEL	76 ROCKET

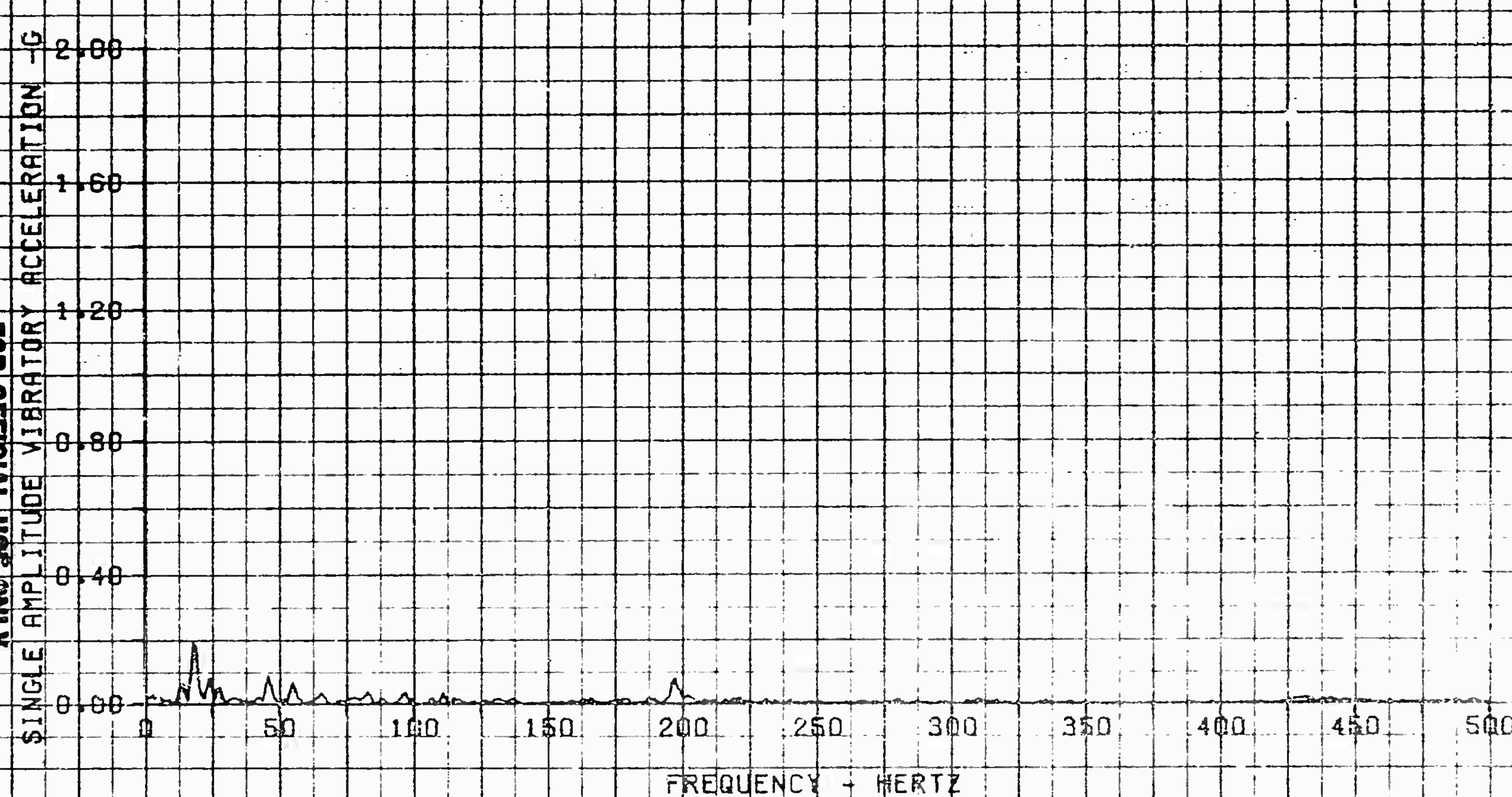

FIGURE 183

VIBRATION CHACTERISTICS

YAH-63 USA S/N 74-22246

LOCATION PILOT INSTRUMENT PANEL — FUNDAMENTAL FREQUENCY IS 4.60 HZ

AXIS LONGITUDINAL RT

GROSS WEIGHT -LB	LONG CG FS -IN.	LAT CG BL -IN.	DENSITY ALTITUDE -FT	OUTSIDE AIR TEMPERATURE -DEG C	ROTOR SPEED -RPM	TRIM CALIB. AIRSPEED -KTS	FLIGHT CONDITION	CONFIGURATION
15000	298.0	-0.5	8000	31.0	276	40	LEVEL	76 ROCKET

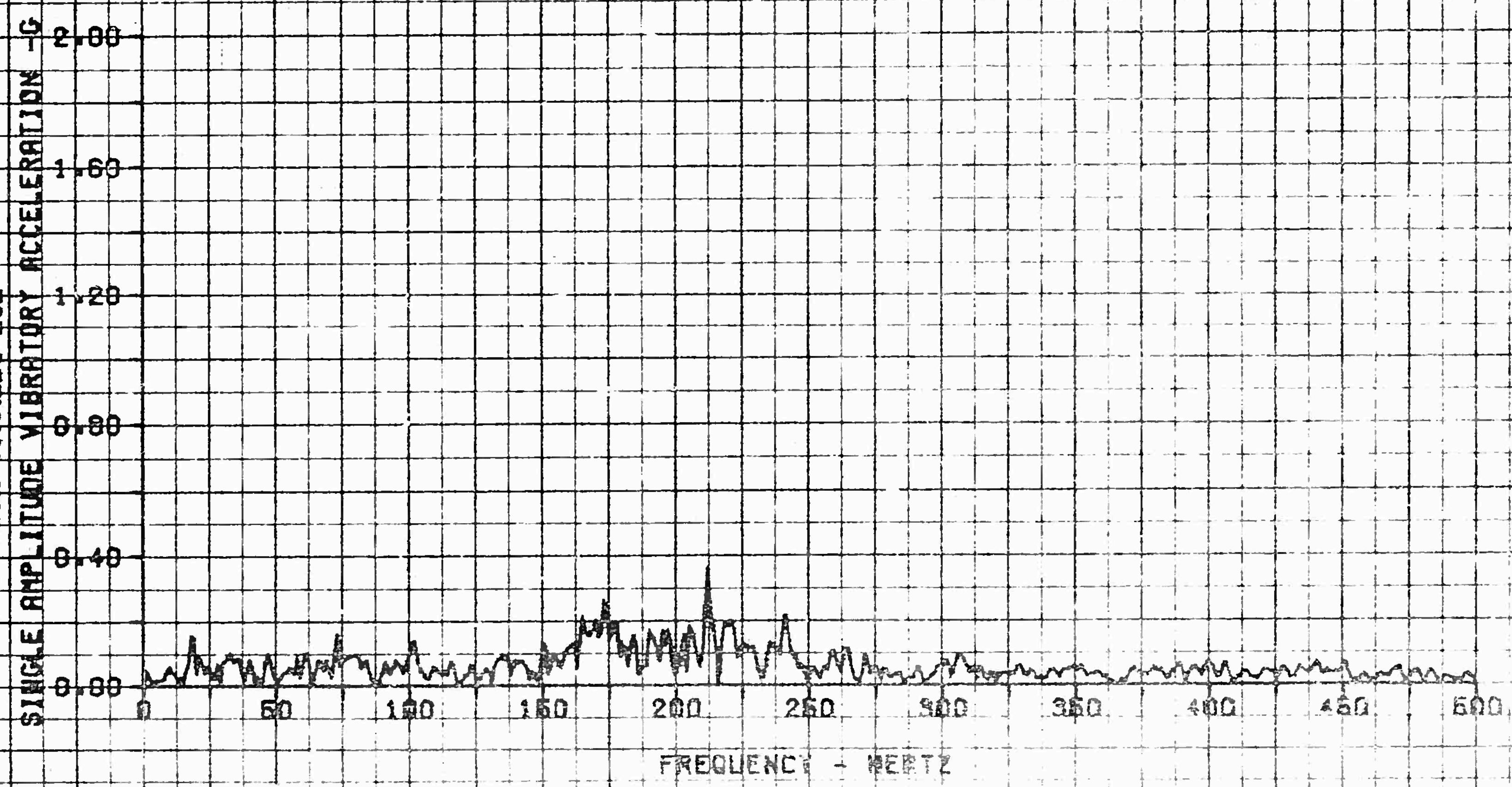

FIGURE 134

INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YAH-63, USA S/N 74-22246 T700-GE-700 LEFT ENGINE

20,046 OUTPUT SHAFT (276 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 = 1°C
2. Engine inlet pressure ratio = .998
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characteristics as shown on figure 148
7. Out of ground effect

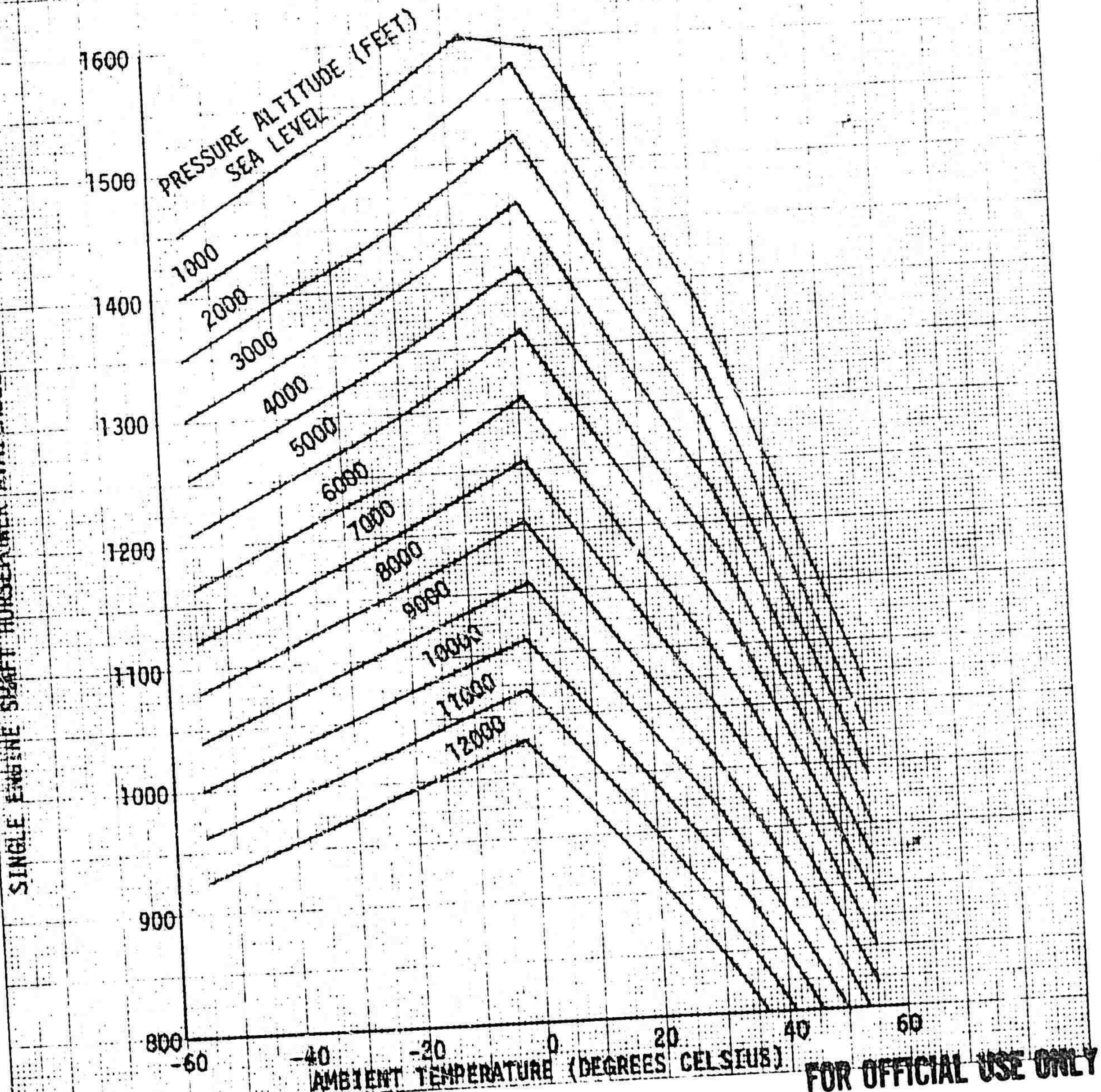

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FIGURE 135

INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YAH-63 USA S/N 74-22246 T700-GE-700 RIGHT ENGINE

20,046 OUTPUT SHAFT (276 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 = 2°C
2. Engine inlet pressure ratio = 1.000
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characteristics as shown on figure 160
7. Out of ground effect

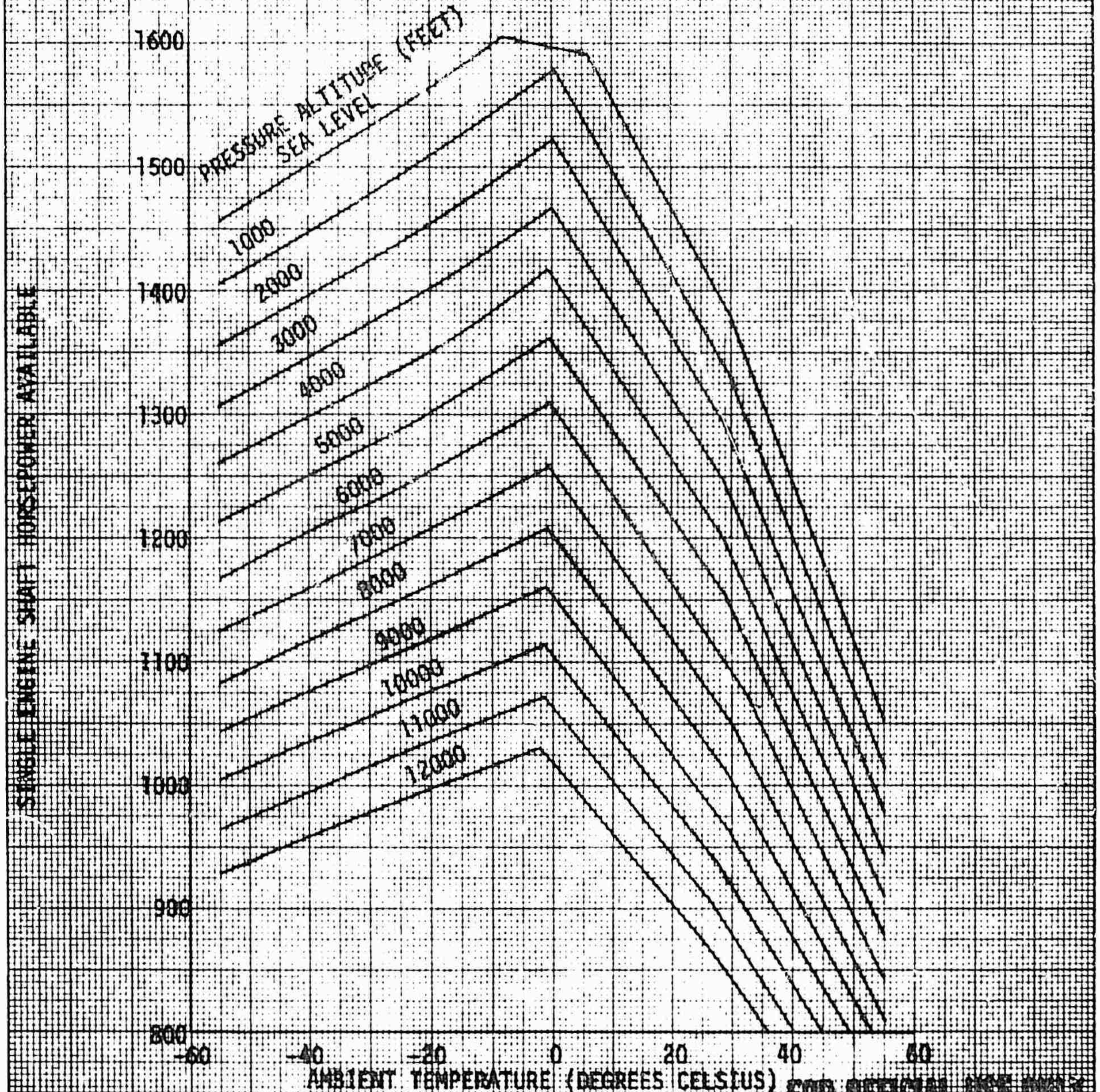

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FIGURE 135

POWER AVAILABLE VERSUS AIRSPEED

YAH-63 USA S/N 74-22246 T-700-GE-700 BOTH ENGINES
20,046 OUTPUT SHAFT (276 ROTOR) RPM AMBIENT TEMPERATURE 15°C 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 characteristics obtained from figures 142 and 145
2. Engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb
5. Exhaust system characteristics as shown in figures 148 and 149
6. Out of ground effect

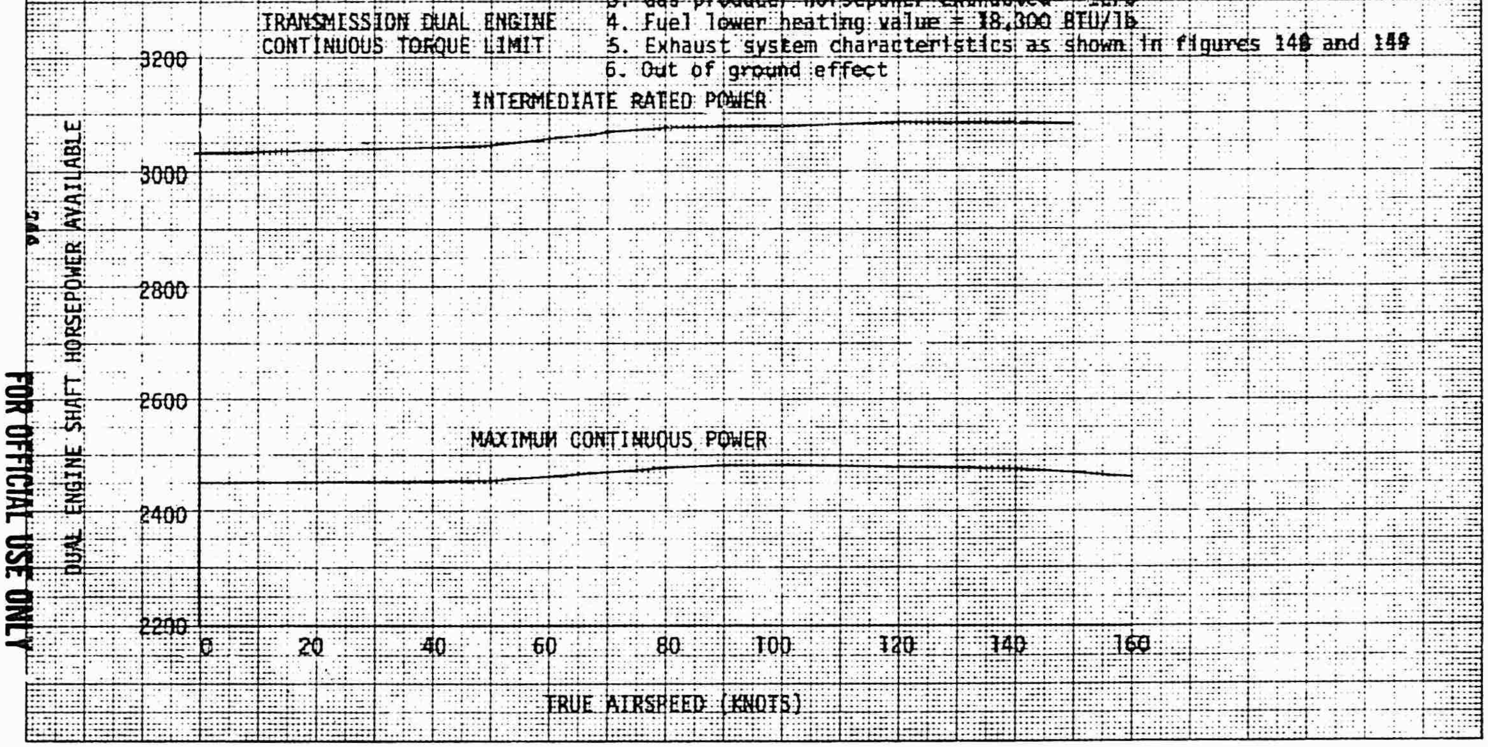

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FIGURE 137

POWER AVAILABLE VERSUS AIRSPEED

YAH-63 USA S/N 74-22246 T700-GE-700 BOTH ENGINES

20,046 OUTPUT SHAFT (276 ROTOR) RPM AMBIENT TEMPERATURE 35 C PRESSURE ALTITUDE (FEET) 4000

NOTE: Based on T700-GE-700 PID Specification AMC-CP-2222-02000, dated 2 Feb 73, corrected for the following installation conditions:

1. Inlet characteristics obtained from figures 142 and 145
2. Engine anti-ice off
3. Gas producer horsepower extracted = zero
4. Fuel lower heating value = 18,300 BTU/lb
5. Exhaust system characteristics as shown on figures 148 and 149
6. Out of ground effect

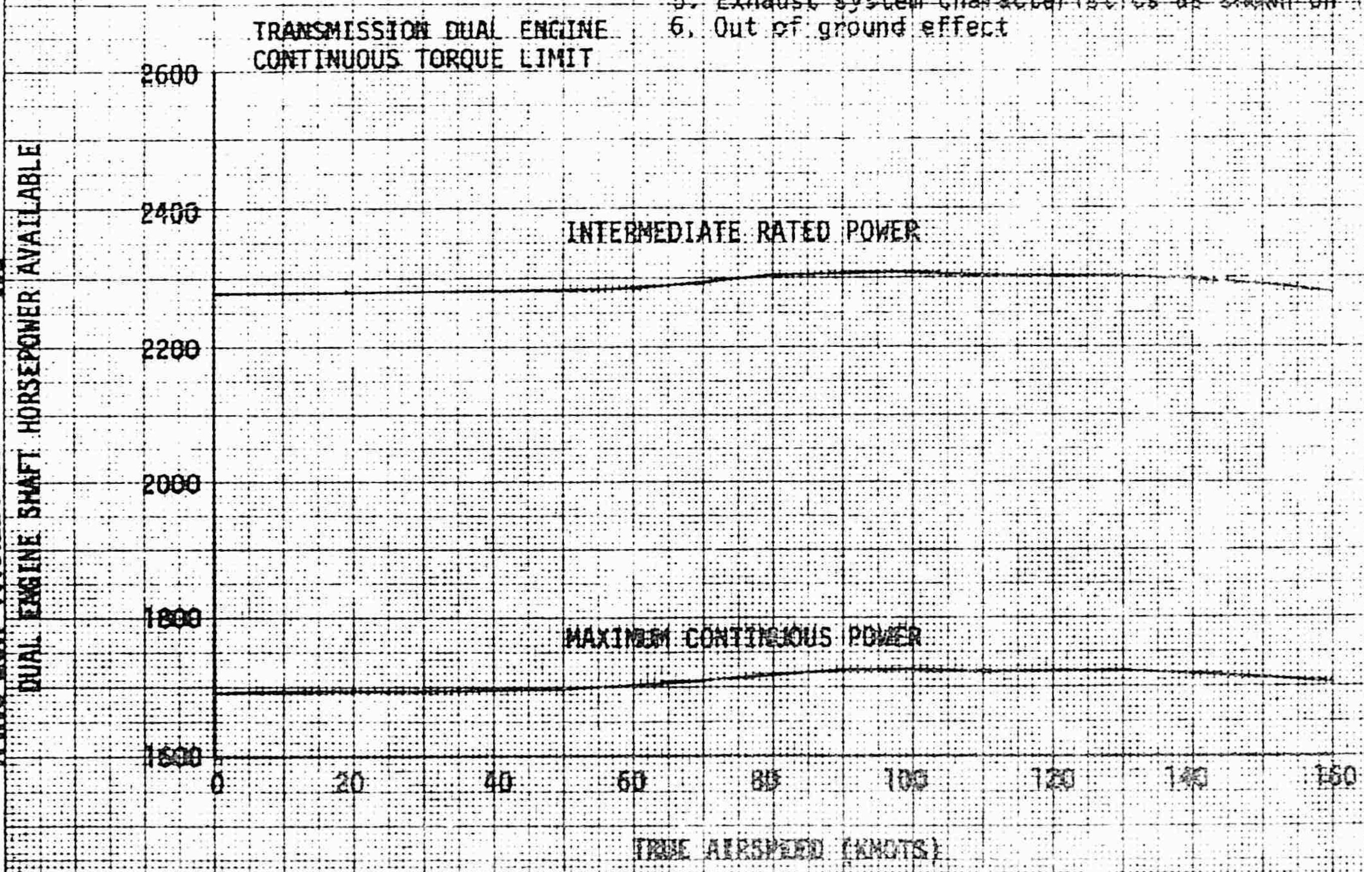

FIGURE 138

INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YAH-63 USA S/N 74-22246 T700-GE-700 LEFT ENGINE
20,046 OUTPUT SHAFT (276 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 = 1°C
2. Engine inlet pressure ratio = .998
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characterization as shown on figure 149
7. Out of ground effect

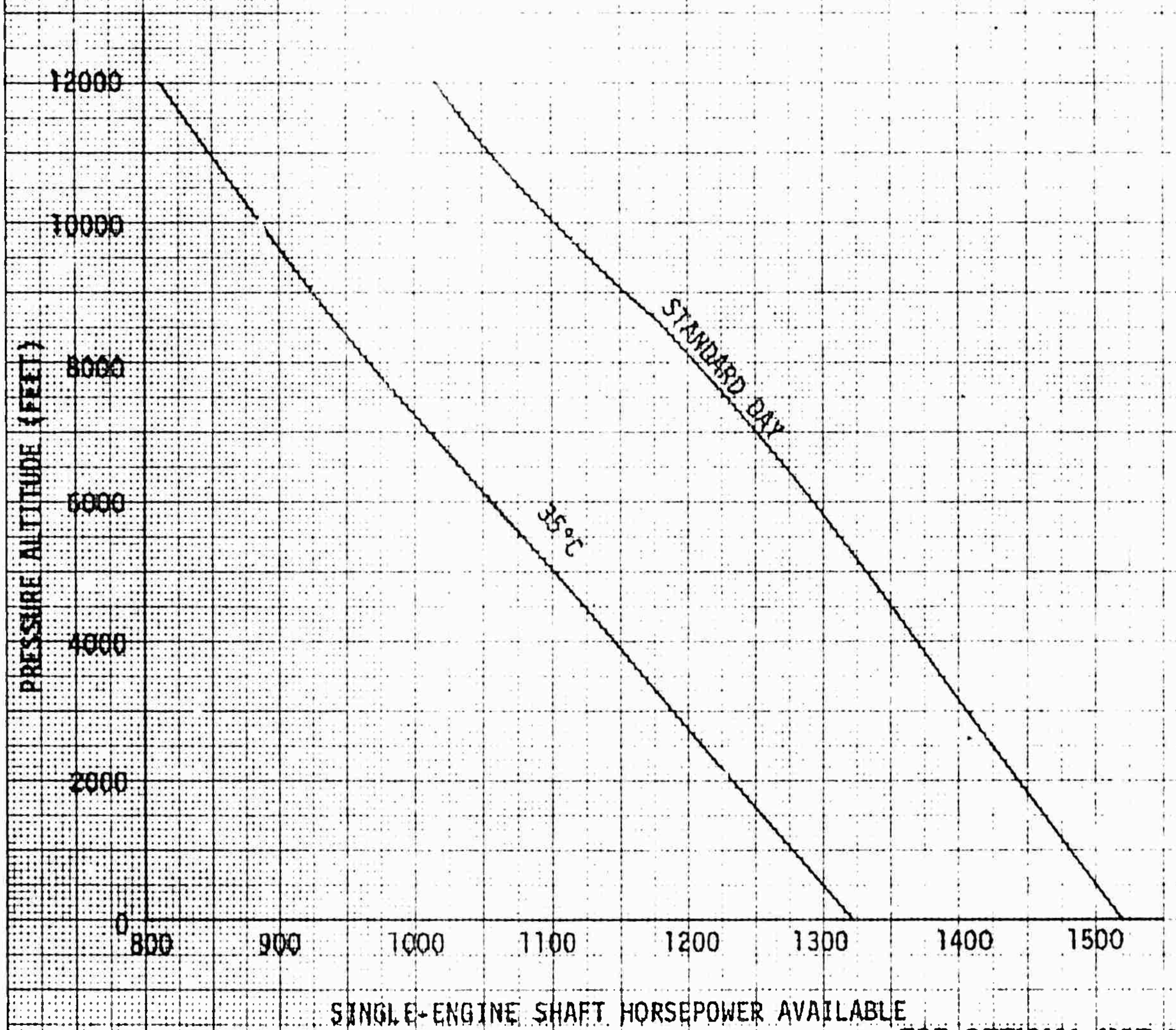

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FIGURE 139

INTERMEDIATE (30-MINUTE LIMIT) POWER AVAILABLE

YAH-63 USA S/N 74-22246 T700-GE-700 RIGHT ENGINE

20,046 OUTPUT SHAFT (276 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 = 2°C
2. Engine inlet pressure ratio = .998
3. Customer bleed air = zero
4. Engine anti-ice off
5. Fuel lower heating value = 18,300 BTU/lb
6. Exhaust system characterization as shown on figure 138
7. Out of ground effect

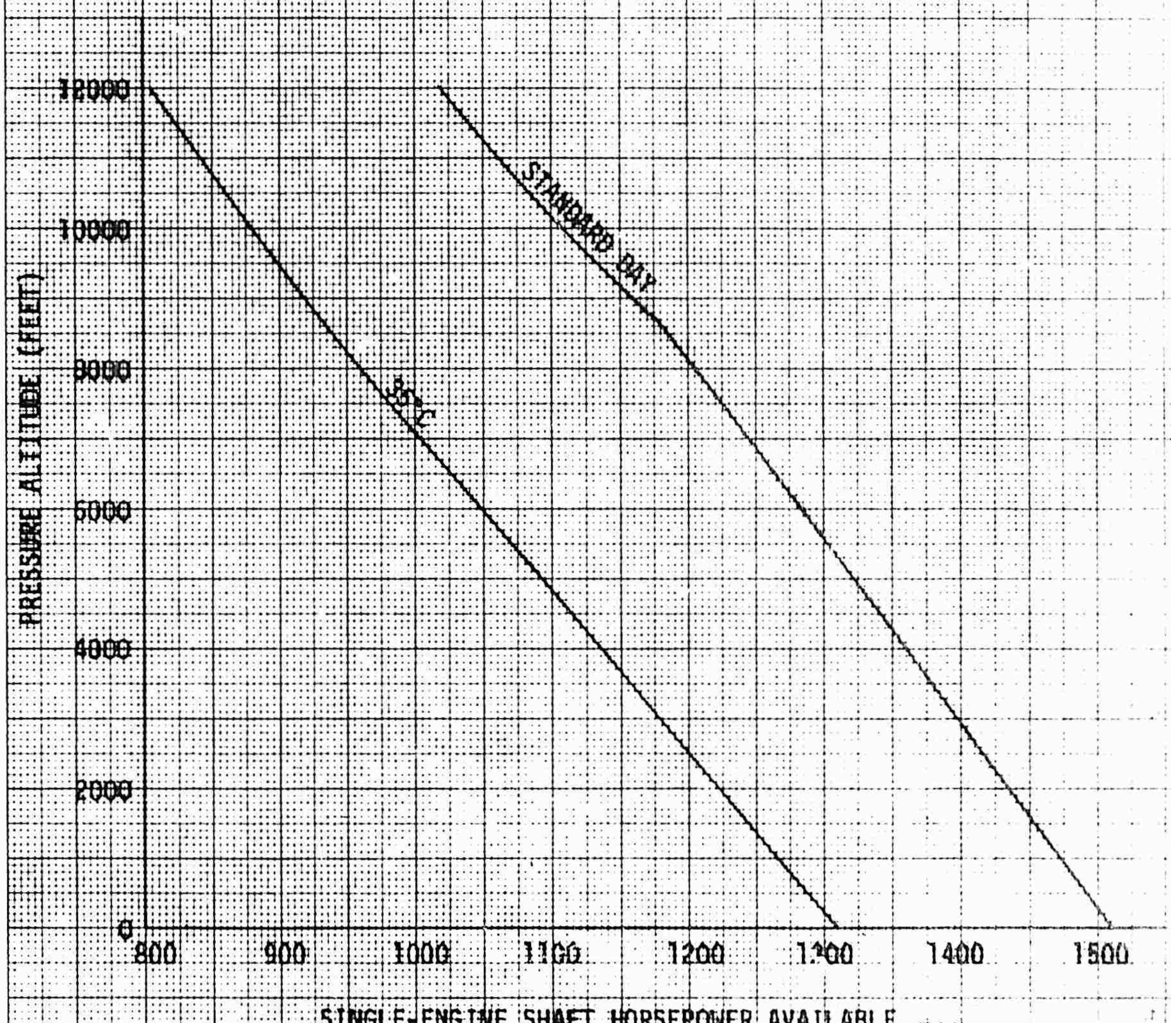

FIGURE 140

DUAL ENGINE SPECIFICATION FUEL FLOW

YUH-63 USA S/N 74-22246 T700-GE-700 BOTH ENGINES

20,046 OUTPUT SHAFT (276 ROTOR) RPM AMBIENT TEMPERATURE = 35°C PRESSURE ALTITUDE = 4000 FEET

NOTE: Based on T-700-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 142, 145, 148 and 149
2. Customer bleed air and engine anti-ice off
3. Fuel lower heating value = 18,300 BTU/lb
4. Out of ground effect

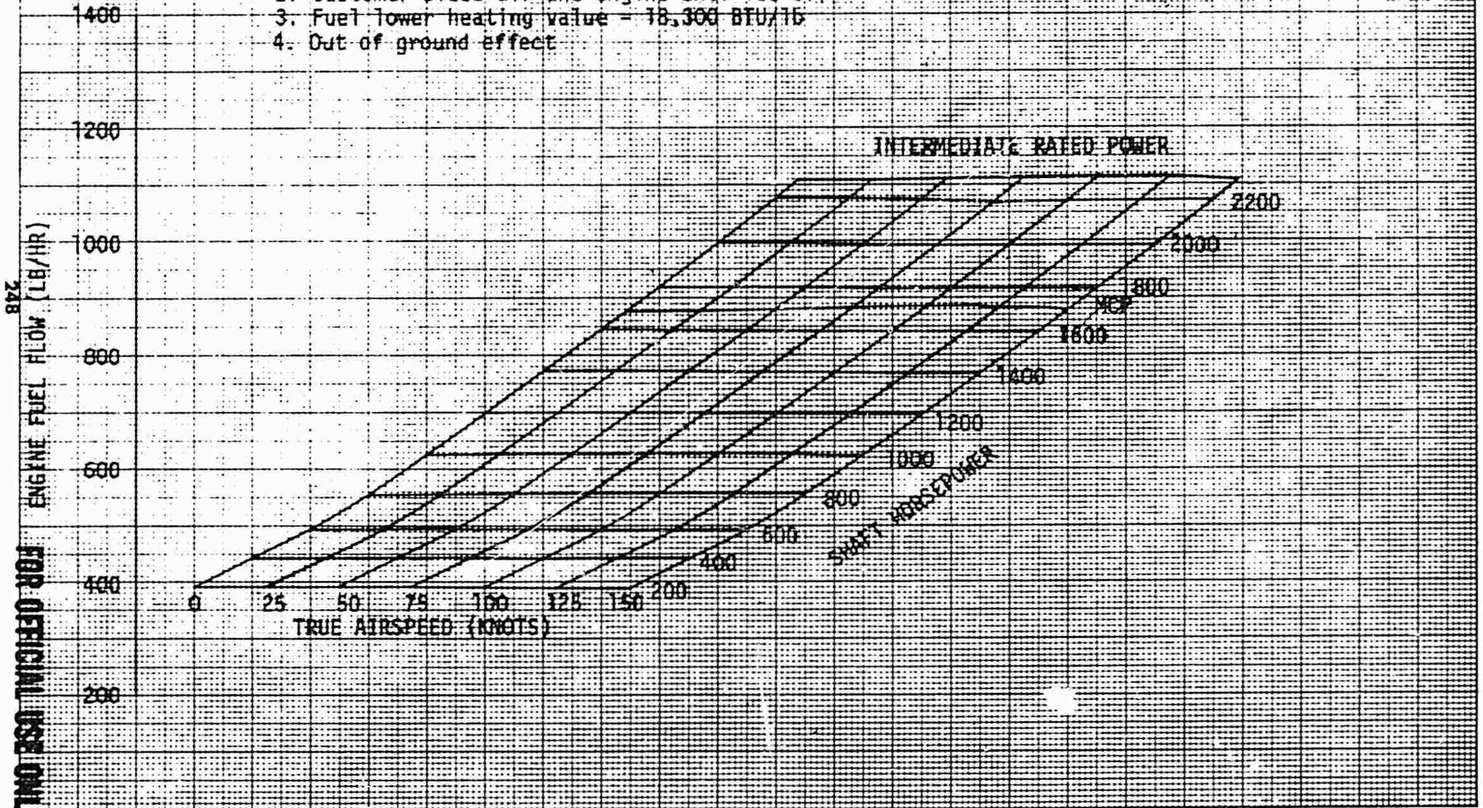

FIGURE 141

DUAL ENGINE SPECIFICATION FUEL FLOW

YAH-63 USA S/N 74-22246 T700-GE-700 BOTH ENGINES

20,046 OUTPUT SHAFT (276 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 142, 145, 148 and 149
2. Customer bleed air and engine anti-ice off
3. Fuel lower heating value = 18,300 BTU/lb
4. Out of ground effect

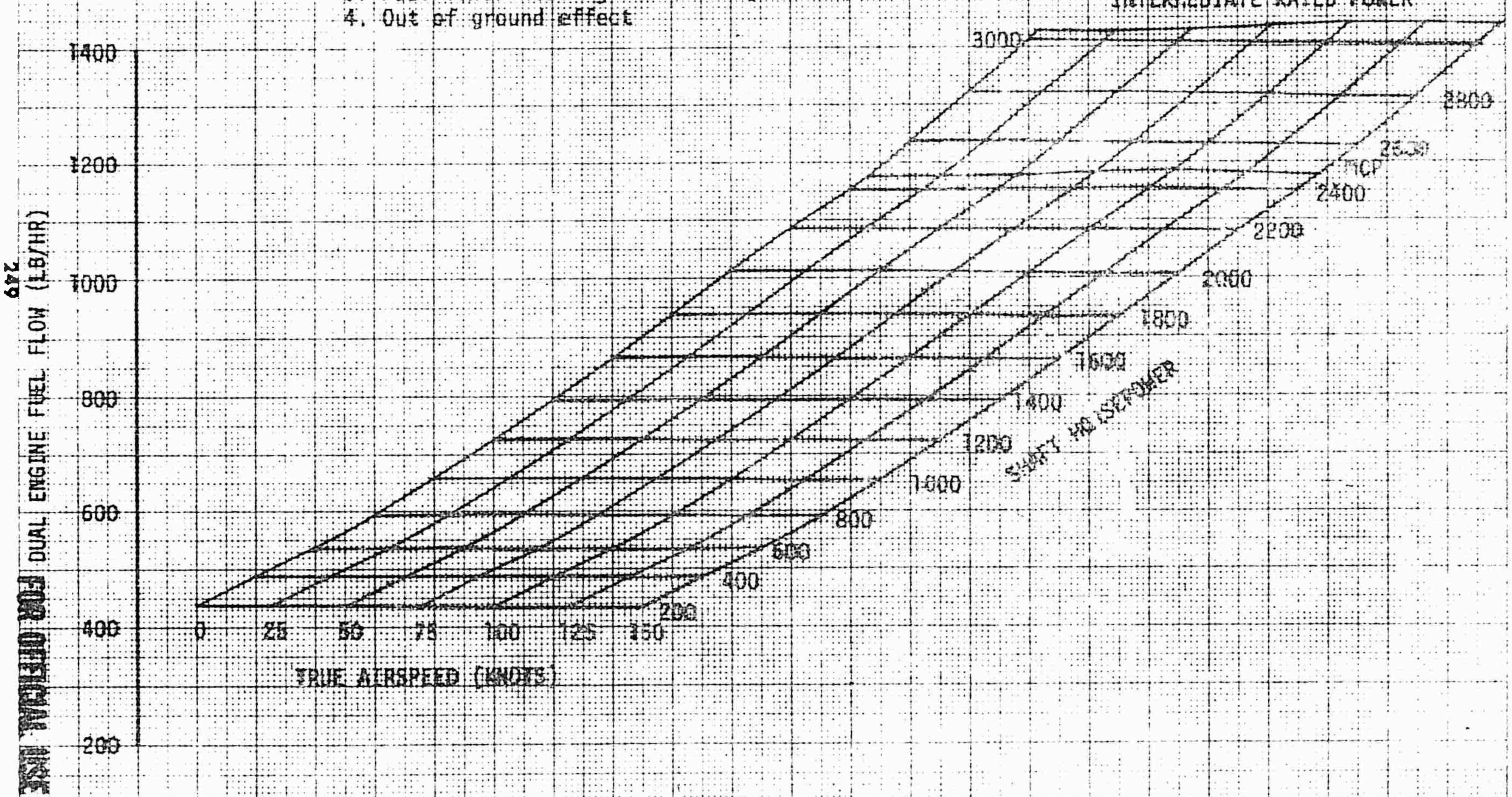

FIGURE 142
ENGINE INLET TEMPERATURE RISE
AIRCRAFT YAH-63 USA S/N 74-22246

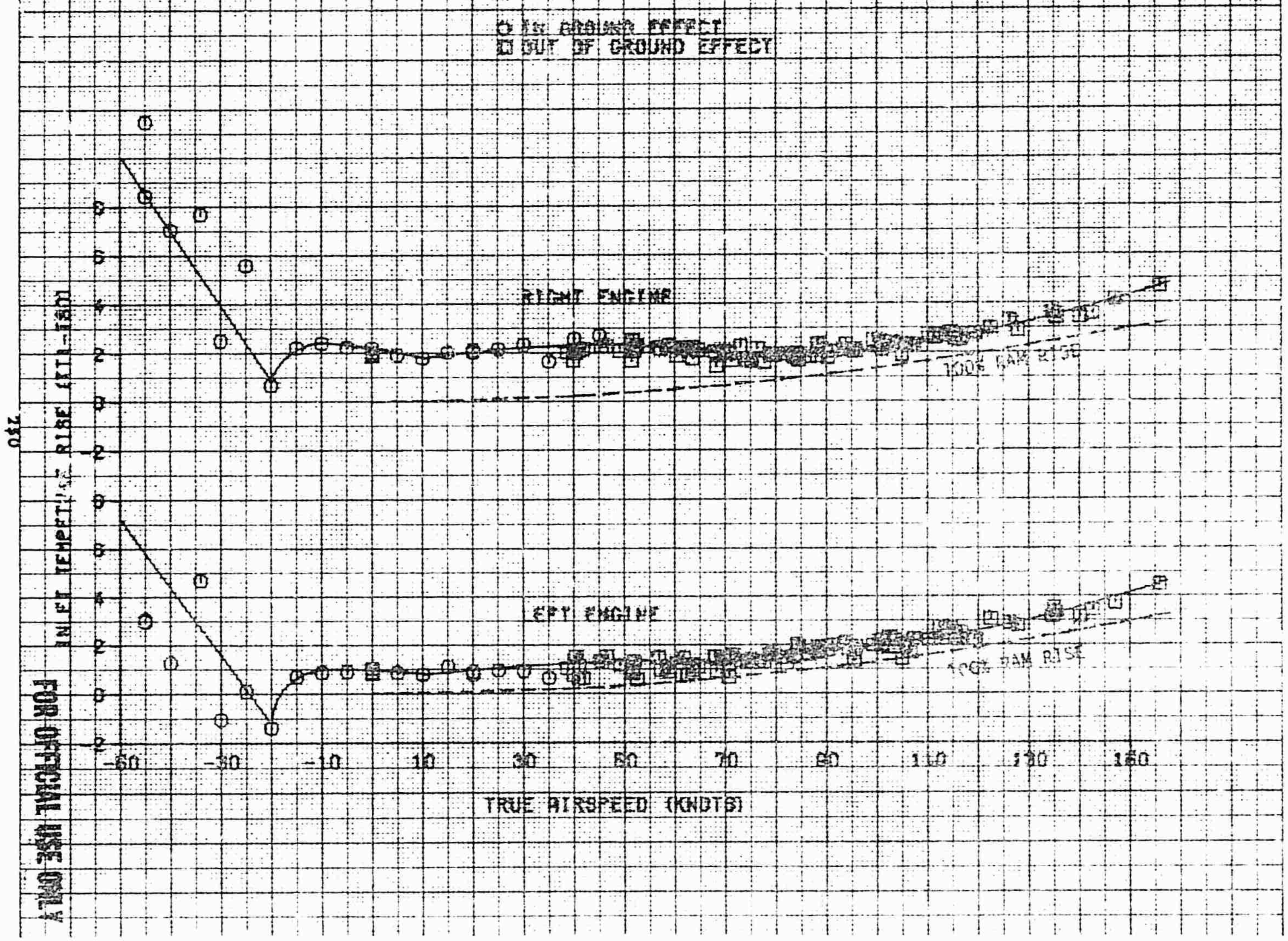

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FIGURE 143

ENGINE INLET TEMPERATURE RISE IN SIDEWARD FLIGHT

AIRCRAFT YAH-63 USA S/N 73-22246

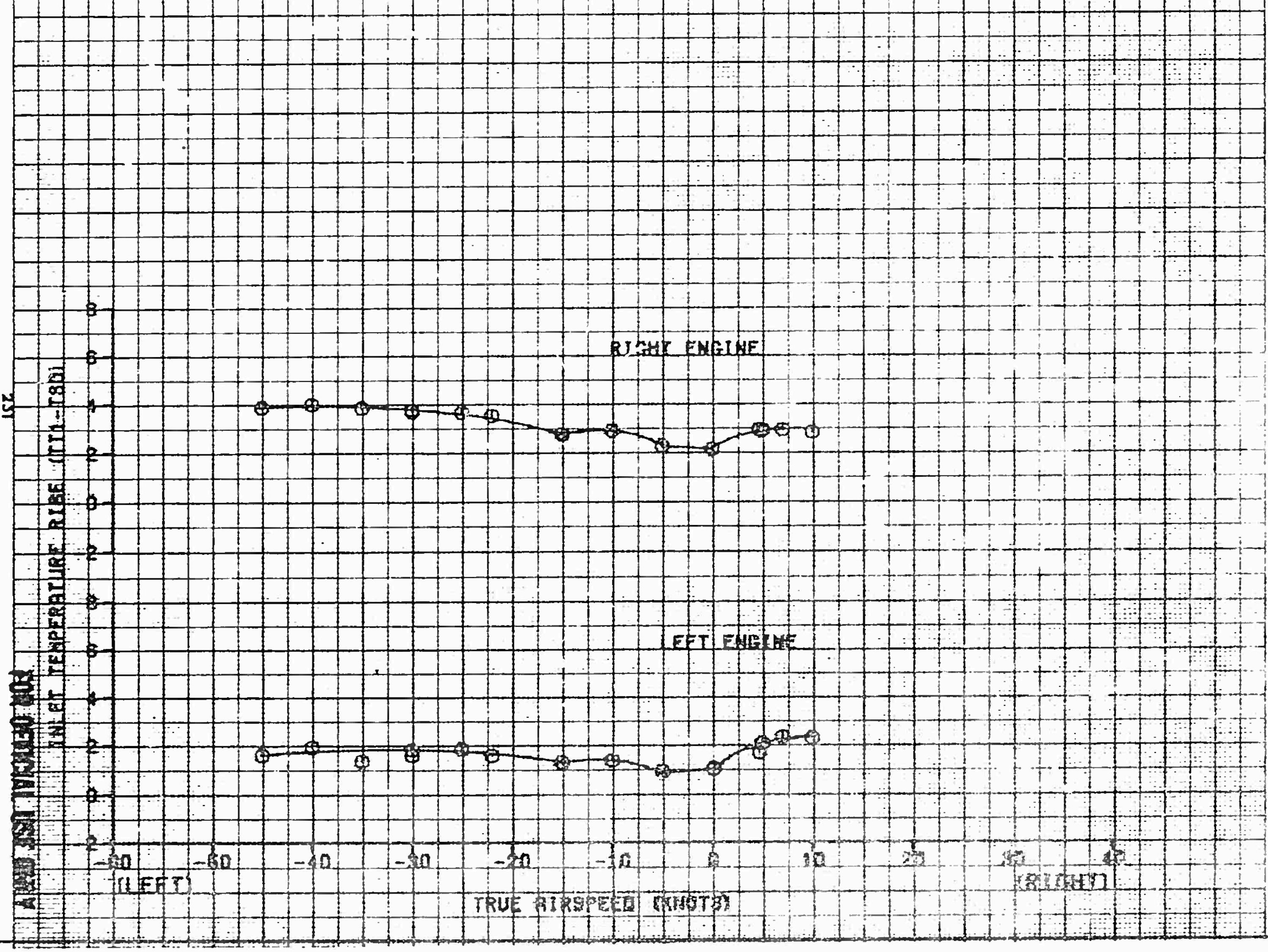

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FIGURE 134

ENGINE INLET TEMPERATURE RISE VERSUS SIDESLIP ANGLE

AIRCRAFT YAH-63 USA S/N 73-22246

TRIM TRUE AIRSPEED = 110 KNOTS

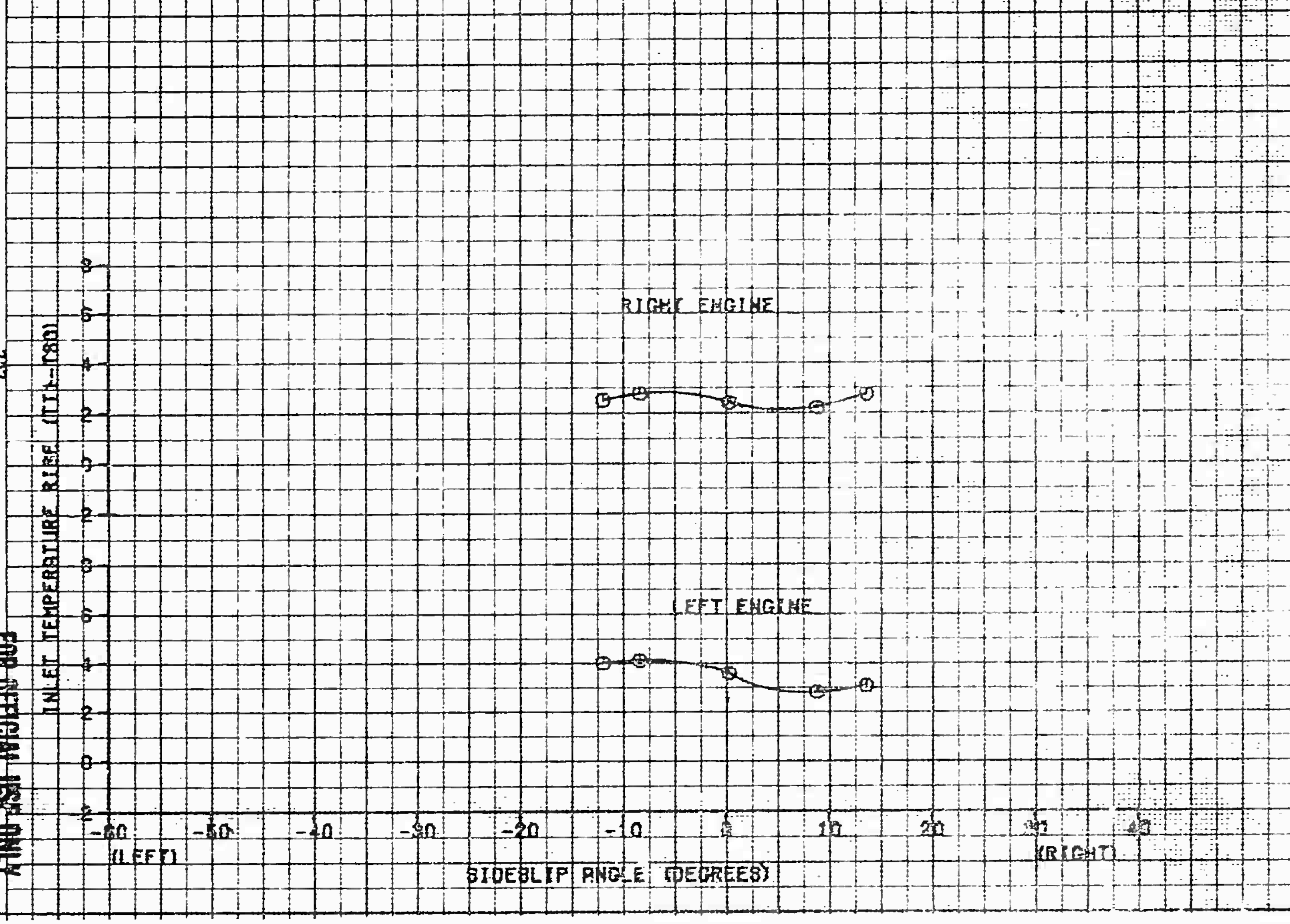


FIGURE 145
ENGINE INLET PRESSURE RATIOS
AIRCRAFT YAH-63 USA S/N 7[illegible]-22246

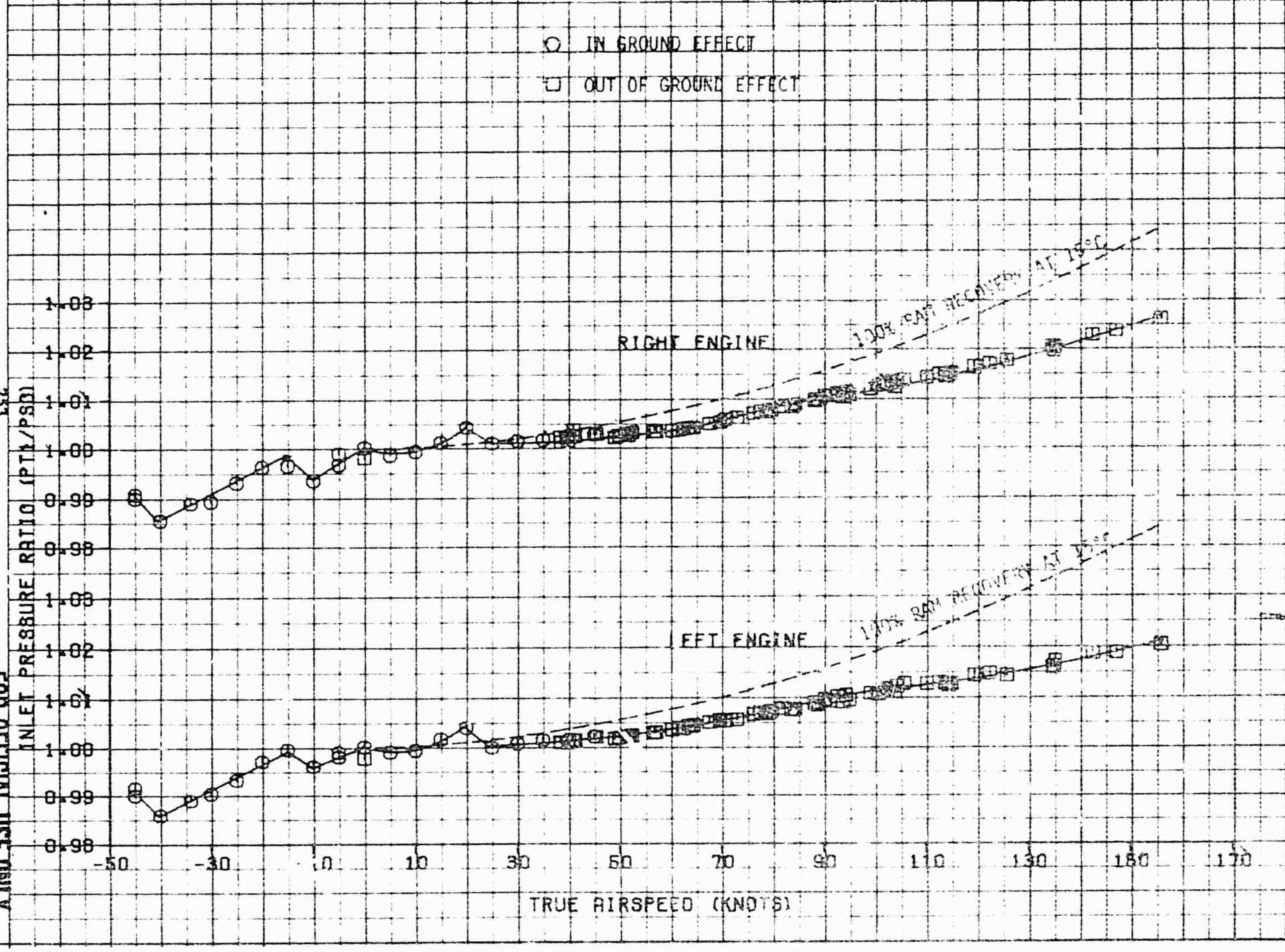


FIGURE 146

ENGINE INLET PRESSURE RATIOS IN SIDEWARD FLIGHT

AIRCRAFT YAH-63 USA S/N 73-22246

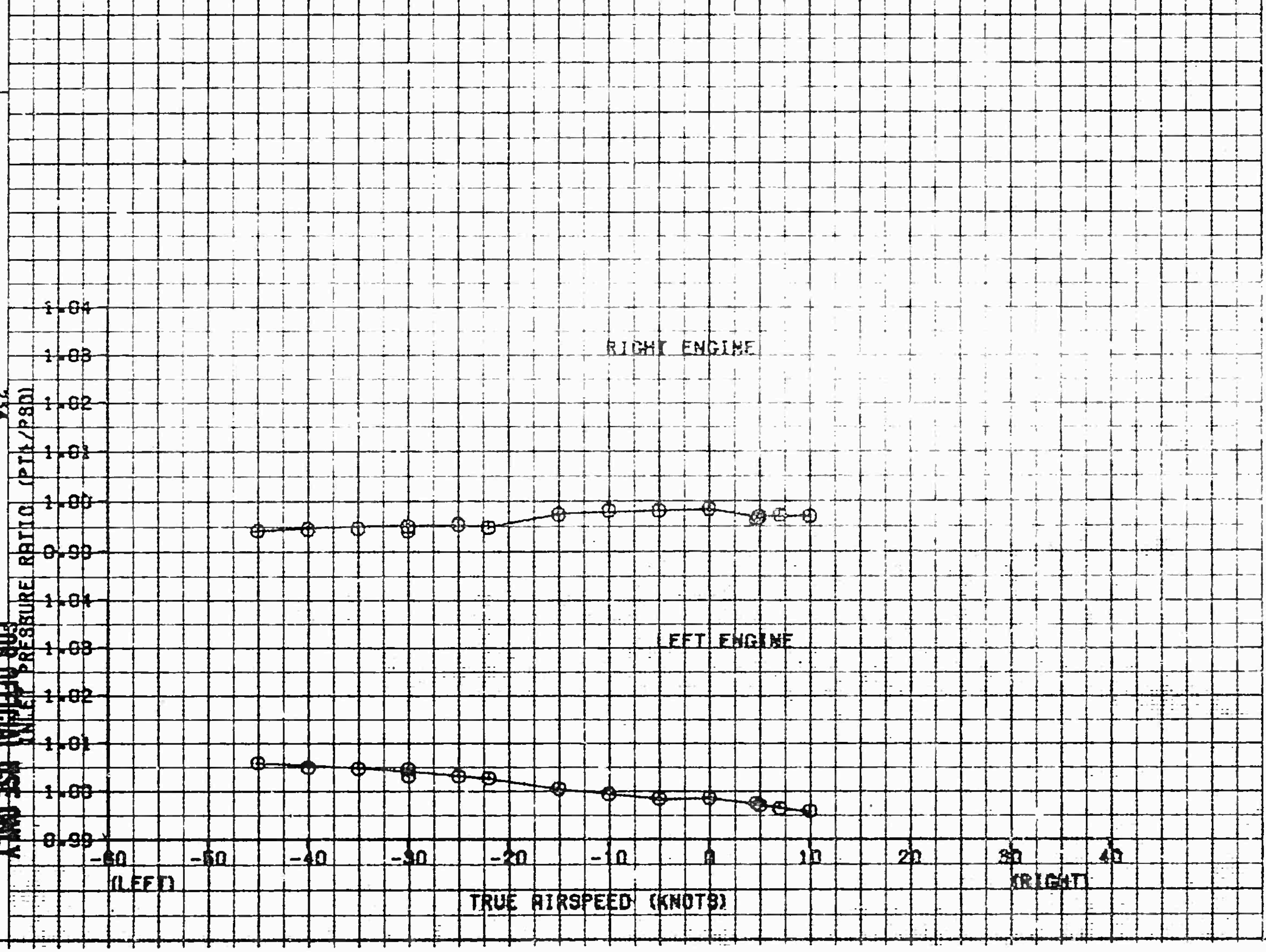


FIGURE 147

ENGINE INLET PRESSURE RATIOS VERSUS SIDESLIP ANGLE

AIRCRAFT YAH-63 USA S/N 73-22246

TRIM TRUE AIRSPEED = 110 KNOTS

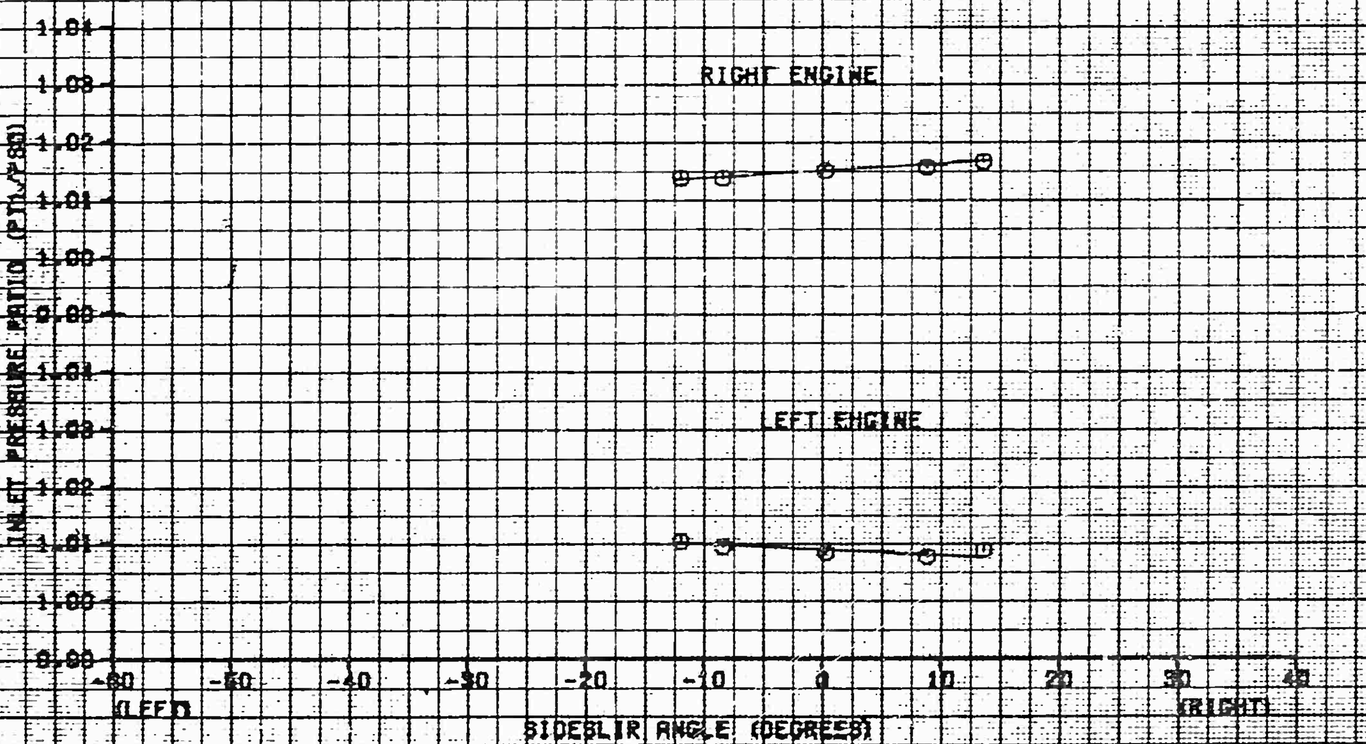


FIGURE 148

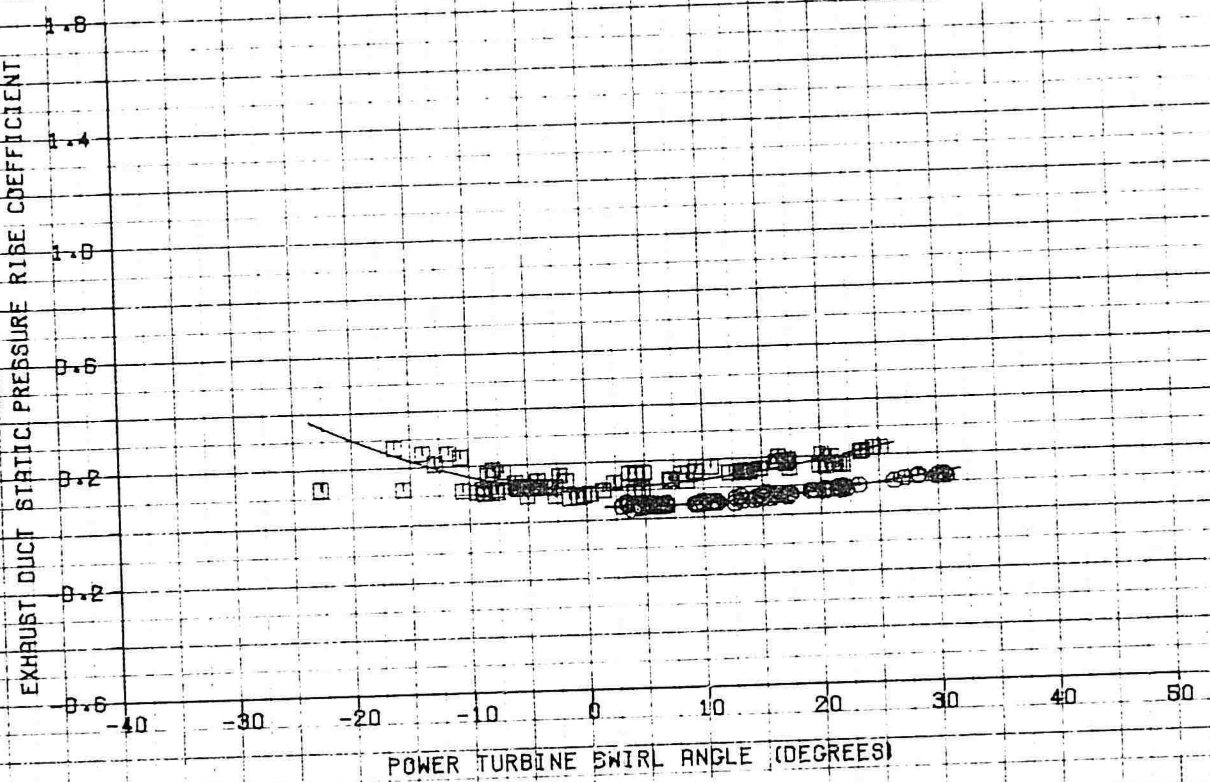
EXHAUST SYSTEM CHARACTERISTICS

AIRCRAFT YAH-63 USA S/N 74-22246

RIGHT ENGINE YT 700-GE-700 S/N 207260

○ IN GROUND EFFECT

□ OUT OF GROUND EFFECT



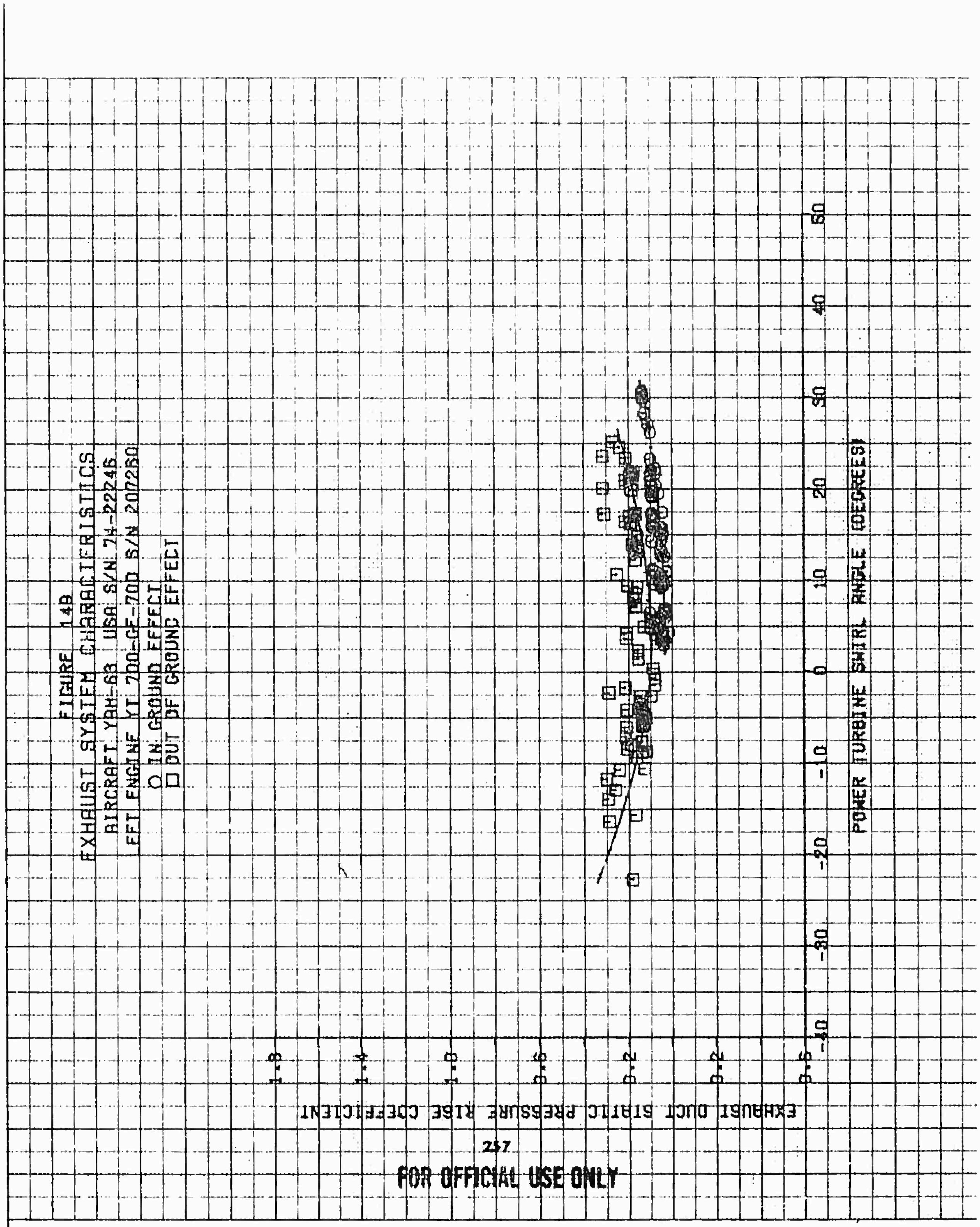


FIGURE 14B
EXHAUST SYSTEM CHARACTERISTICS
AIRCRAFT YAH-63 USA S/N 74-22246
EFT ENGINE YT 700-GE-700 S/N 207260
○ IN GROUND EFFECT
□ OUT OF GROUND EFFECT
POWER TURBINE SWIRL ANGLE (DEGREES)
EXHAUST DUCT STATIC PRESSURE RISE COEFFICIENT
-40
-30
-20
-10
0
10
20
30
40
50
1.8
1.4
1.0
0.6
0.2
-0.2
-0.6

FIGURE 160
ROTOR AND ENGINE POWER COMPARISON
YAH-63 USA S/N 73-22248
YT700-GE-700 ENGINE LEFT S/N 207260 RIGHT S/N 207279

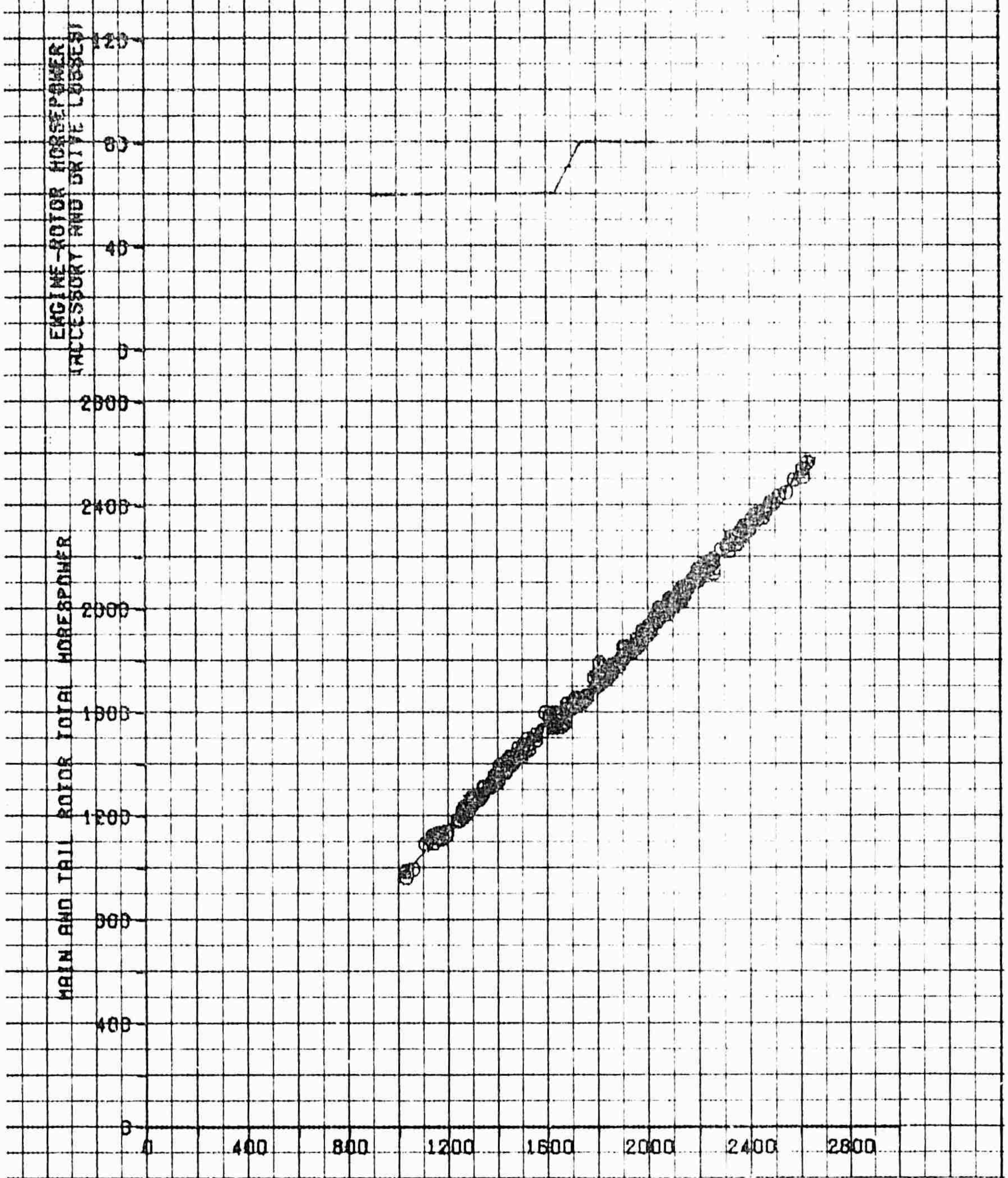


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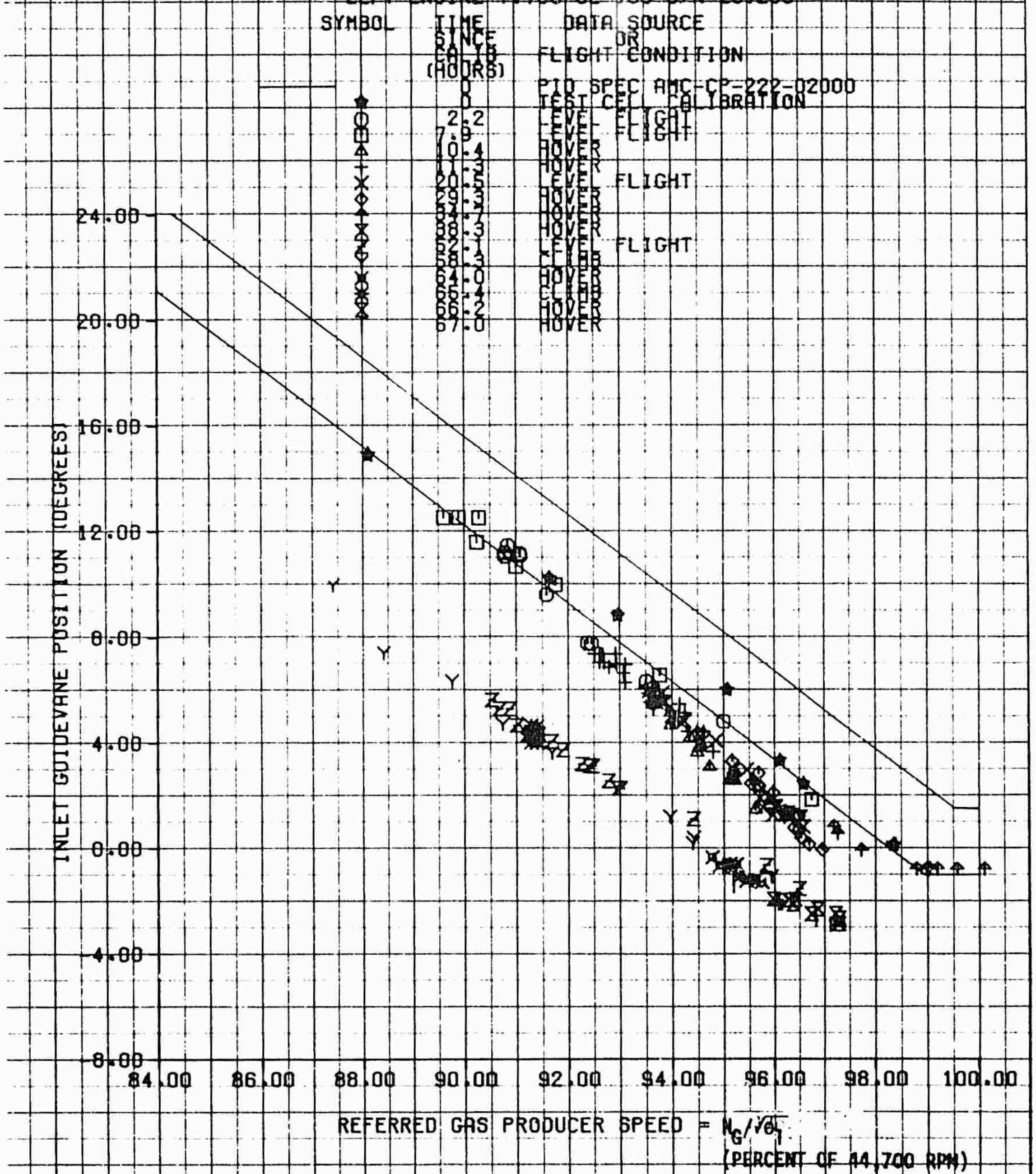
FIGURE 151

INLET GUIDE VANE AND REFERRED GAS PRODUCER

AIRCRAFT YAH-63 USA S/N 74-22246

LEFT ENGINE YT700-GE-700 S/N 207260



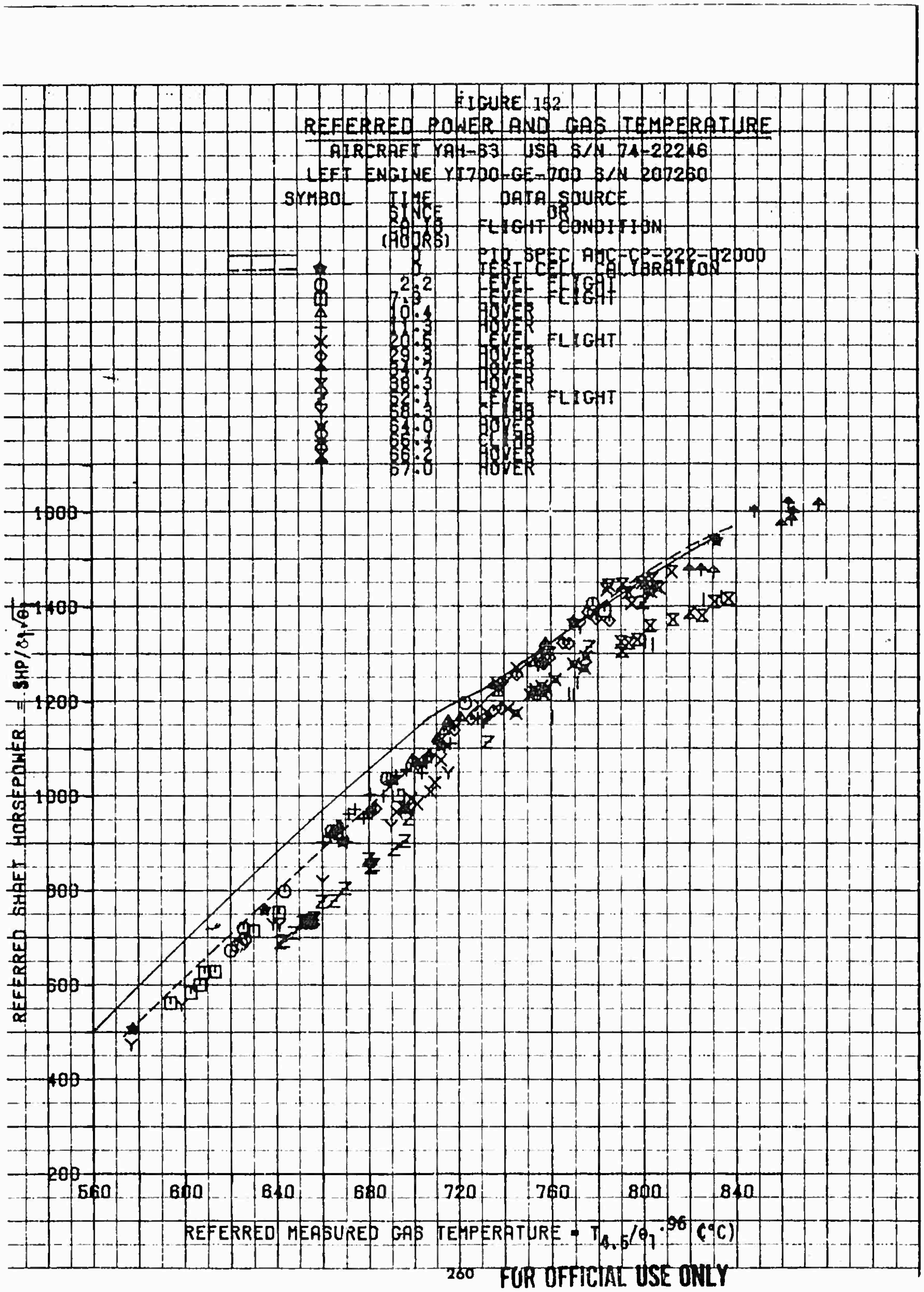


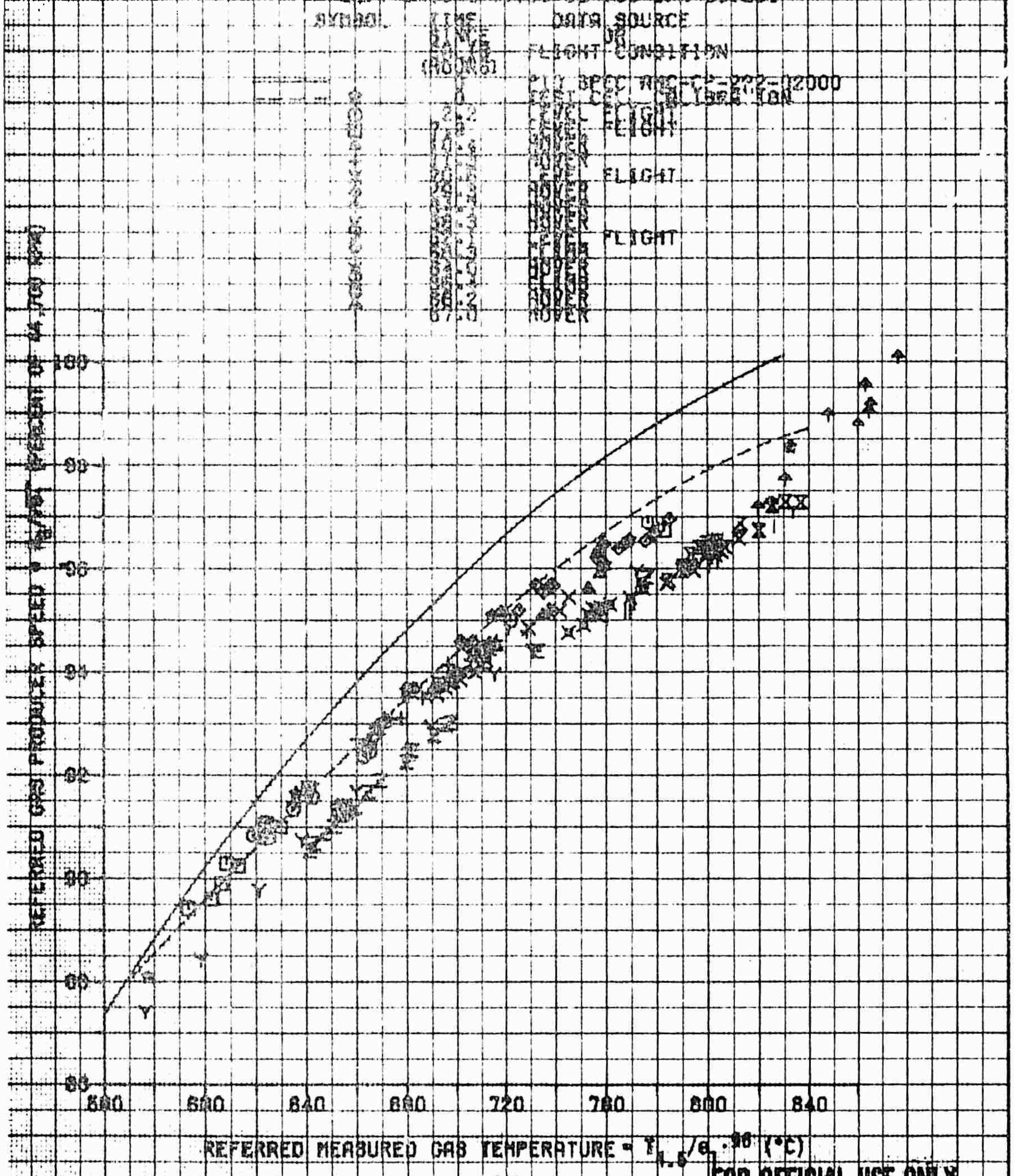
FIGURE 152
REFERRED POWER AND GAS TEMPERATURE

FIGURE 153

REFERRED GAS PRODUCER SPEED AND GAS TEMPERATURE

AIRCRAFT YAH-63 USA S/N 74-22248

LEFT ENGINE YT700-GE-700 S/N 207260



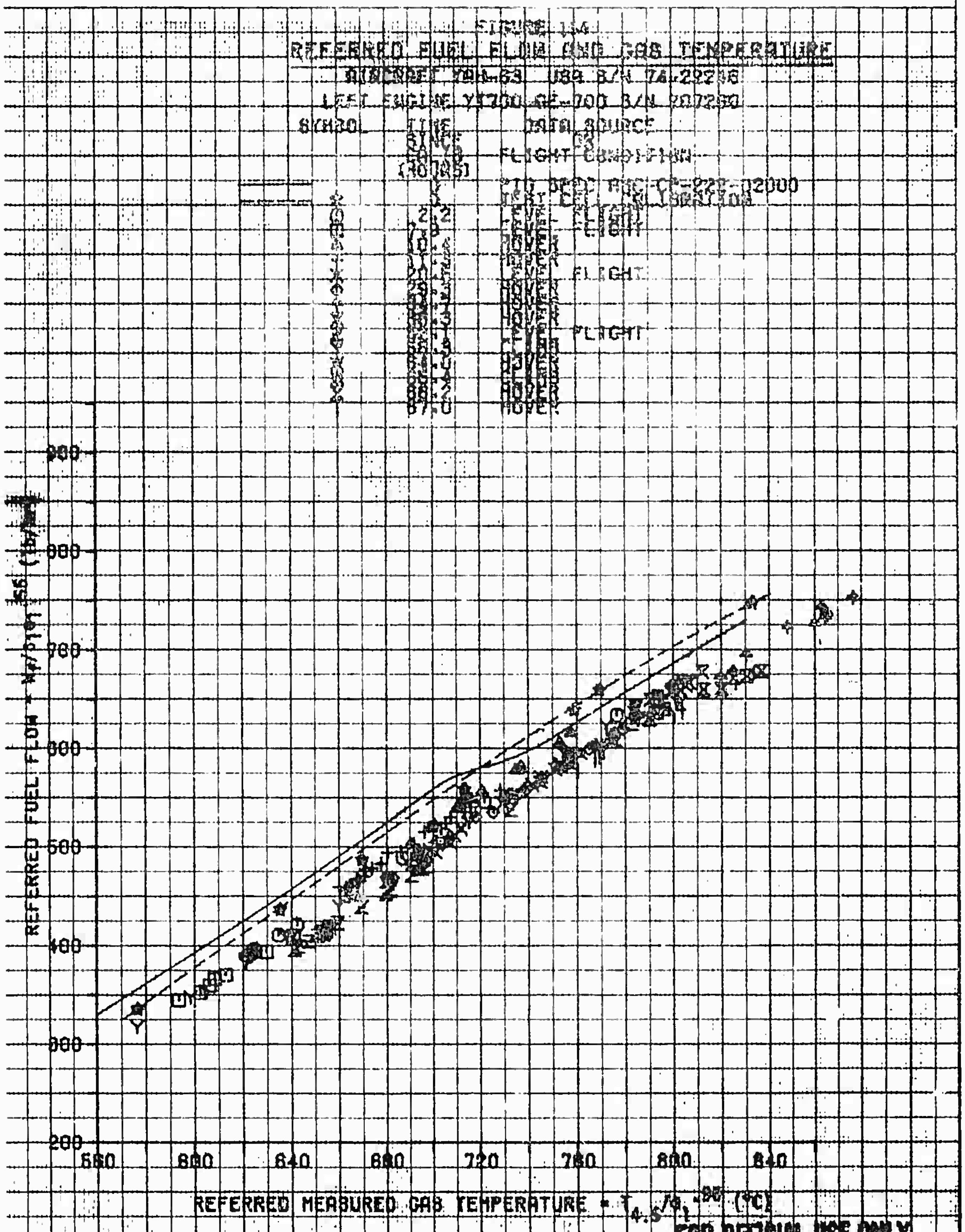

REFERRED FUEL FLOW AND GAS TEMPERATURE
SYMBOL
DATA SOURCE
FLIGHT CONDITION
LEVEL FLIGHT
HOVER
REFERRED FUEL FLOW
REFERRED MEASURED GAS TEMPERATURE
200
300
400
500
600
700
800
900
560
600
640
680
720
760
800
840

FIGURE 155

INLET GUIDE VANE AND REFERRED GAS PRODUCER

AIRCRAFT YAH-63 USA S/N 74-22246

RIGHT ENGINE YT700-GE-700 S/N 207211

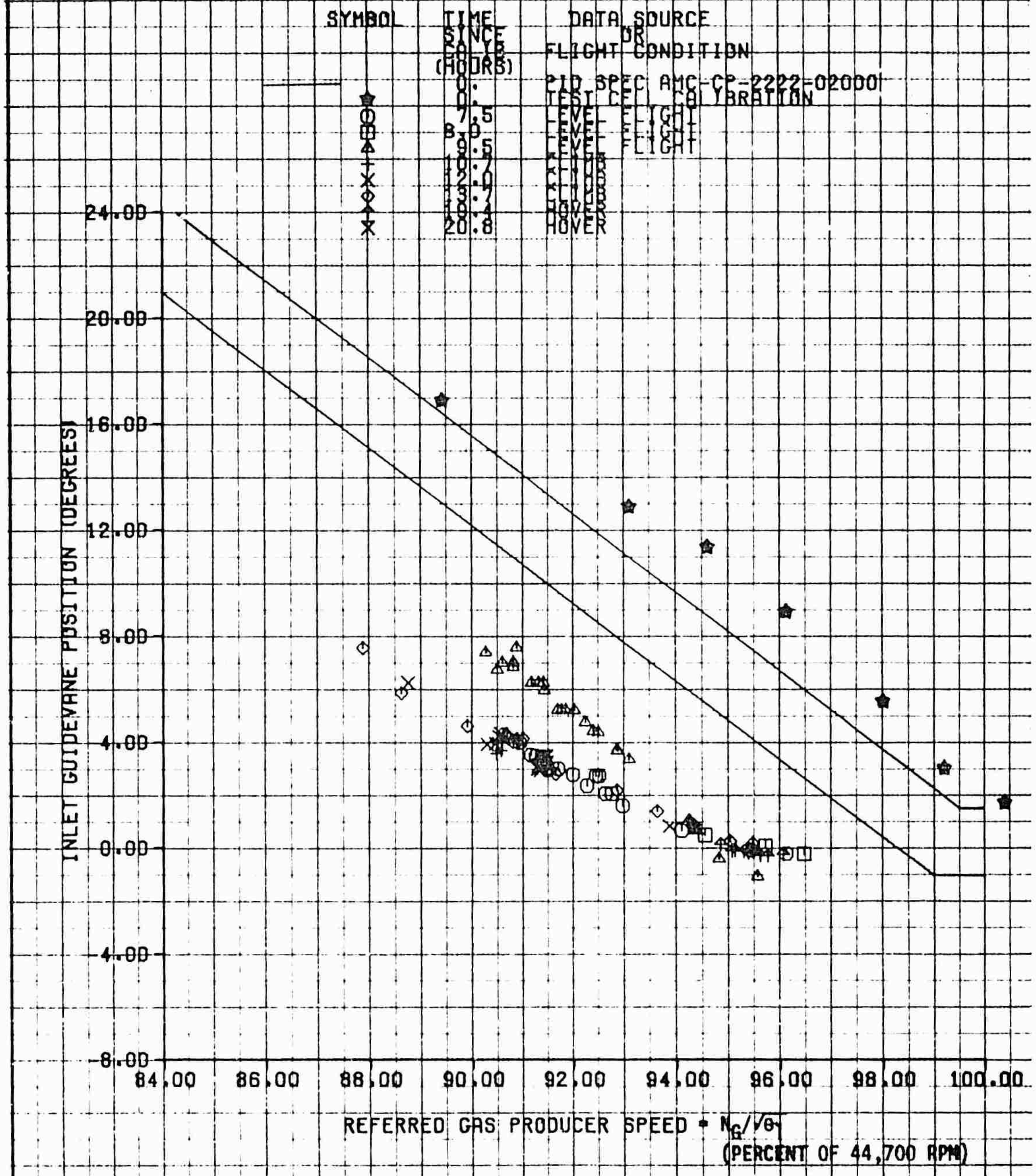


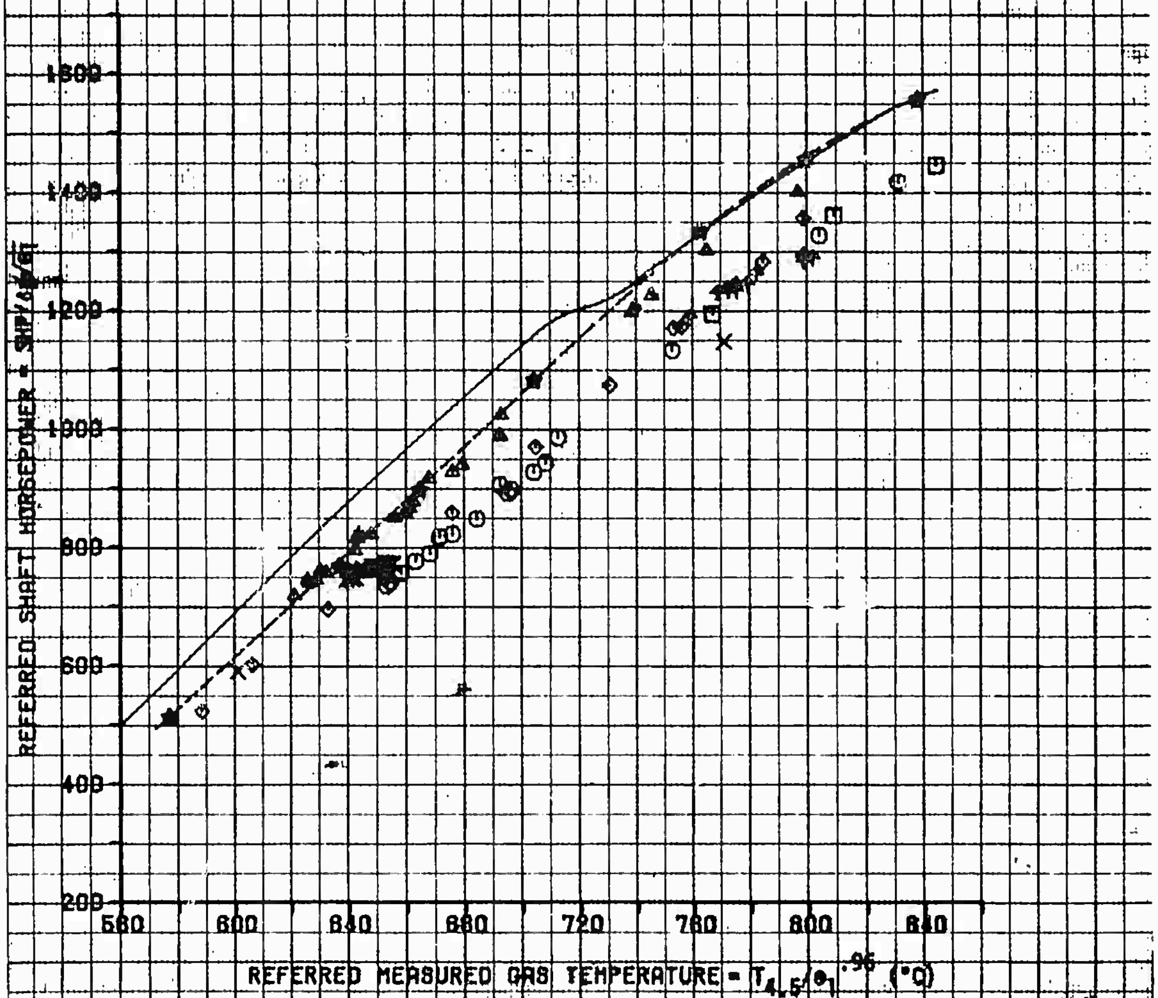
FIGURE 166

REFERRED POWER AND GAS TEMPERATURE

AIRCRAFT YAH-63 USA S/N 74-22248

RIGHT ENGINE YT700-GE-700 S/N 207211

SYMBOL	TIME SINCE RAT/N (HOURS)	DATA SOURCE OR FLIGHT CONDITION
———	0.	PID SPEC AVC-CP-2222-02000
— — —	0.	TEST CELL CALIBRATION
☆	7.5	LEVEL FLIGHT
▢	8.0	LEVEL FLIGHT
◯	9.5	LEVEL FLIGHT
△	10.7	CLIMB
+	12.0	CLIMB
×	13.7	CLIMB
◇	18.4	HOVER
★	20.8	HOVER



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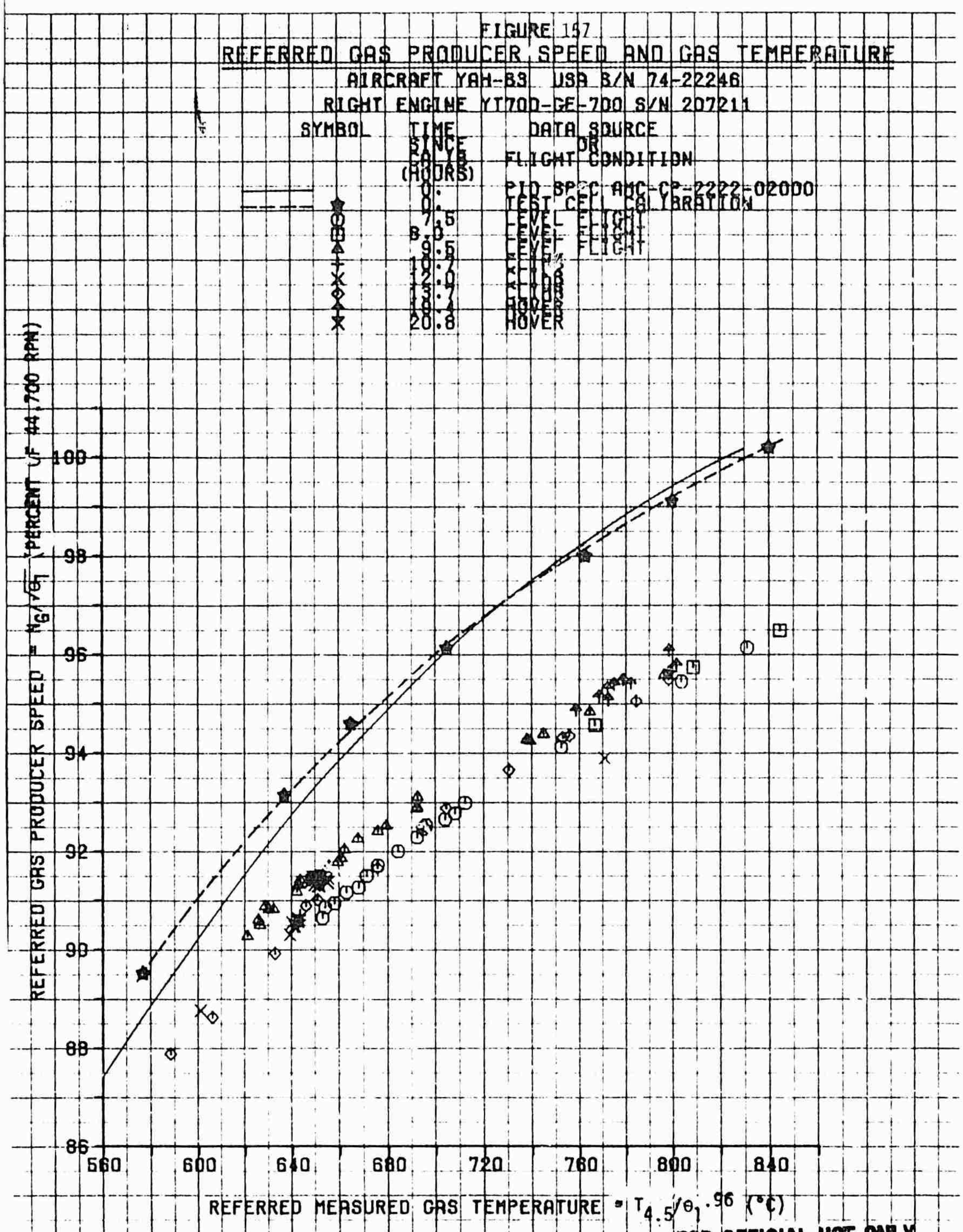

FOR OFFICIAL USE ONLY

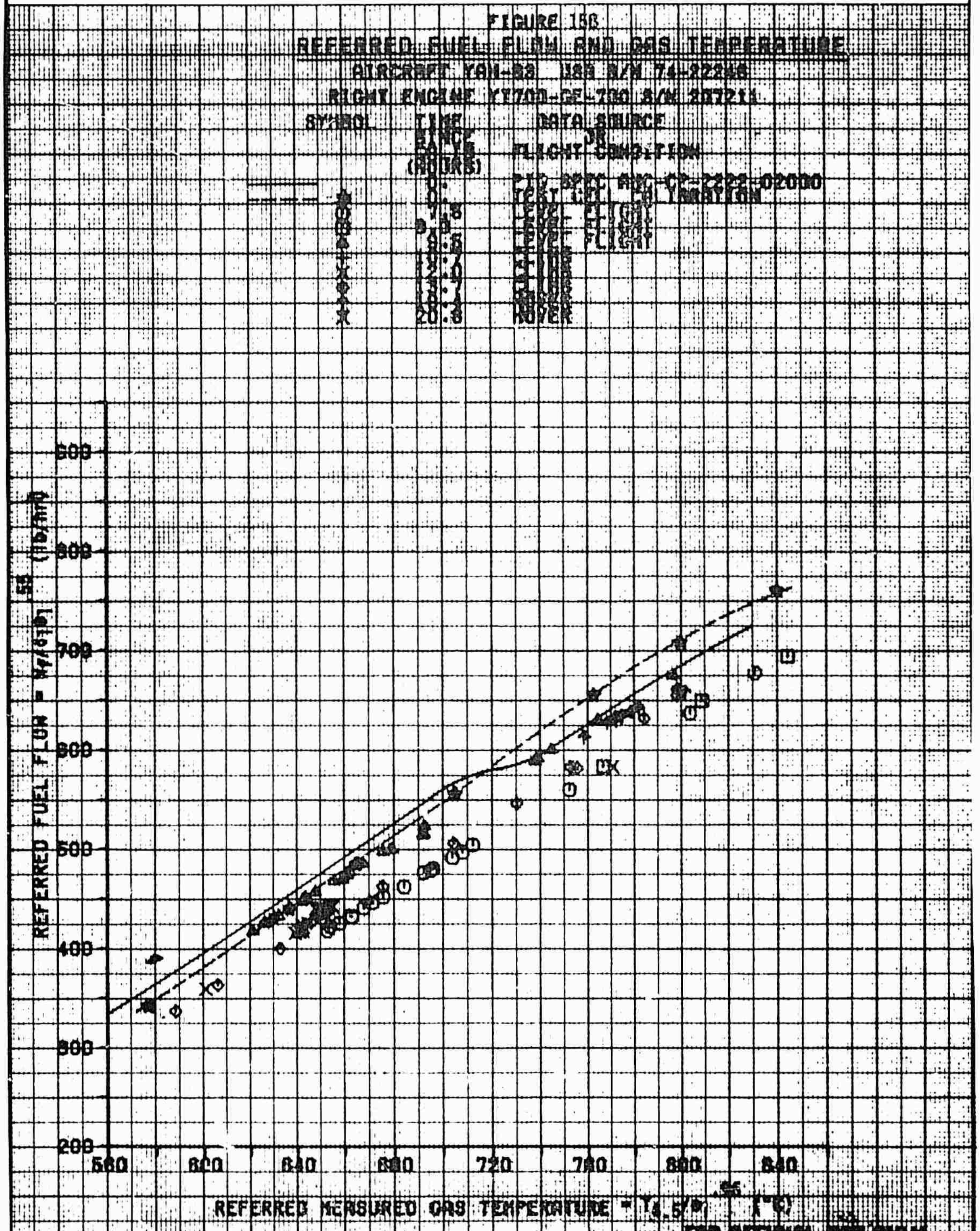
FIGURE 158
REFERRED FUEL FLOW AND GAS TEMPERATURE
SYMBOL
DATA SOURCE
20.8
HOVER
REFERRED FUEL FLOW
(lb/hr)
300
400
500
600
700
200
580
620
640
680
720
760
800
840
REFERRED MEASURED GAS TEMPERATURE
(°C)

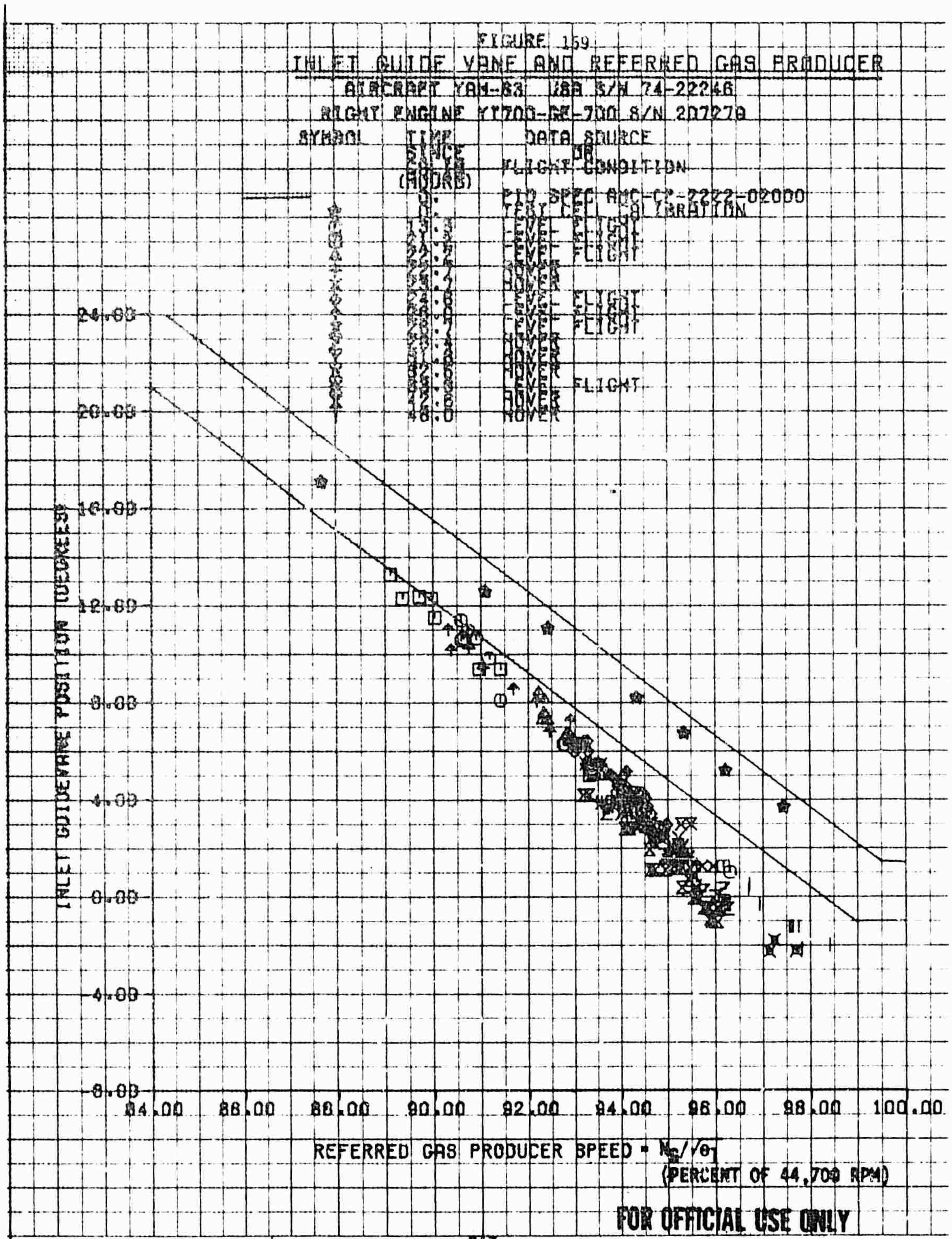
FIGURE 159
INLET GUIDE VANE AND REFERRED GAS PRODUCER
AIRCRAFT YAH-63 USA S/N 74-22246
RIGHT ENGINE YT700-GE-700 S/N 207270
SYMBOL
TIME SINCE (HOURS)
DATA SOURCE OR FLIGHT CONDITION
LEVEL FLIGHT
HOVER
FLIGHT
24.00
20.00
16.00
12.00
8.00
4.00
0.00
-4.00
-8.00
84.00
86.00
88.00
90.00
92.00
94.00
96.00
98.00
100.00
INLET GUIDEVANE POSITION (DEGREES)
REFERRED GAS PRODUCER SPEED
(PERCENT OF 44,700 RPM)

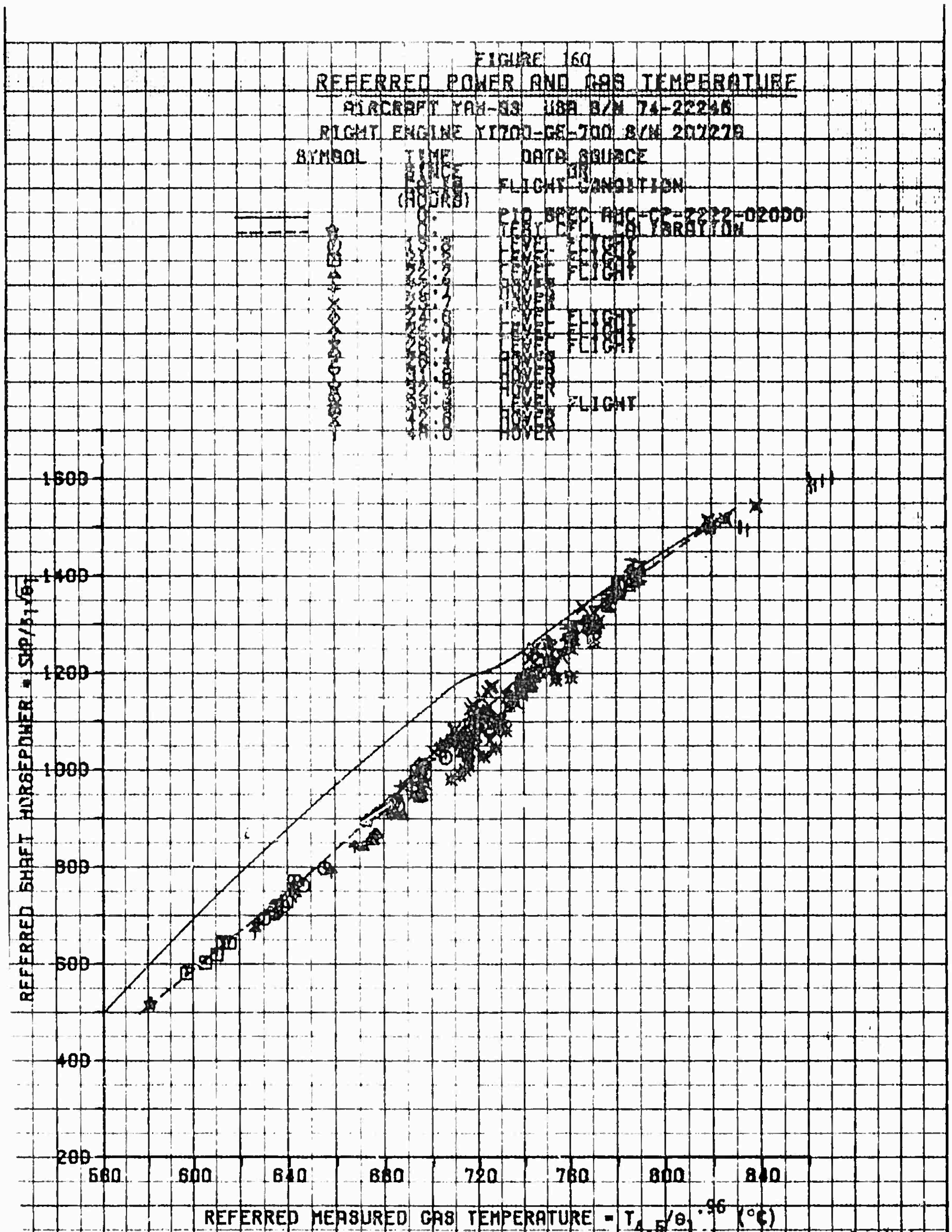

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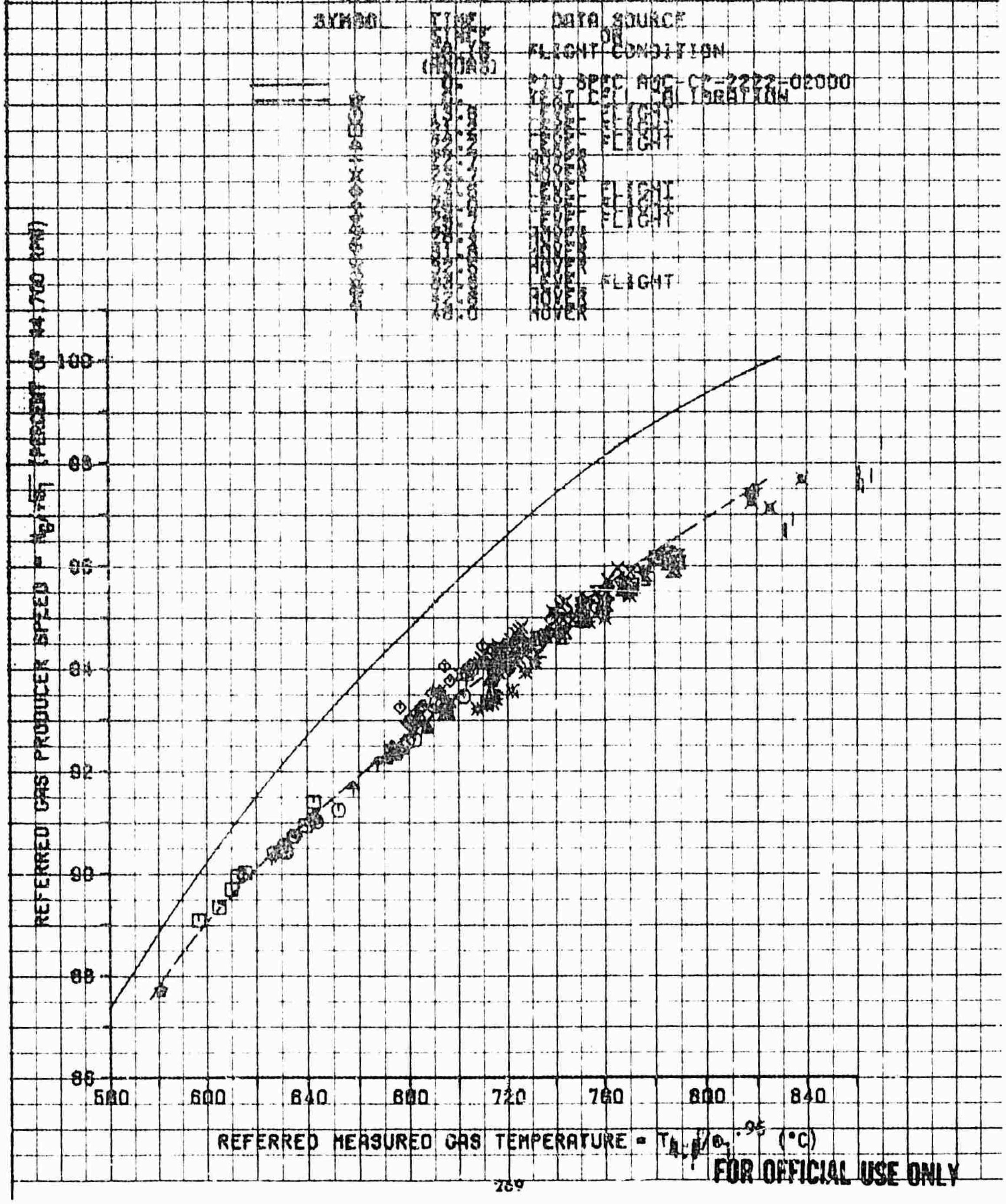
FIGURE 161

REFERRED GAS PRODUCER SPEED AND GAS TEMPERATURE

AIRCRAFT YAH-63 USA S/N 74-22248

RIGHT ENGINE YT700-GE-700 S/N 207278



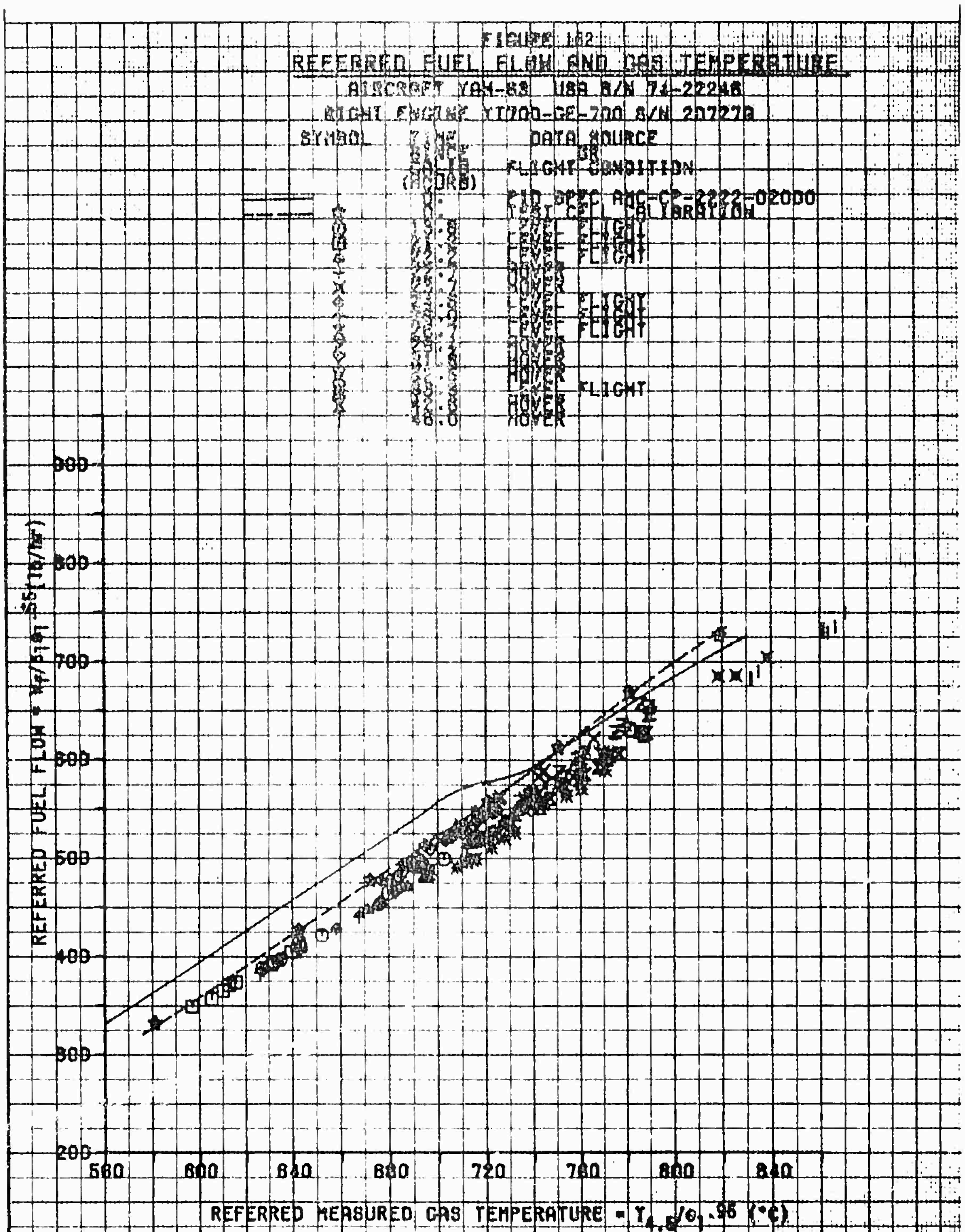


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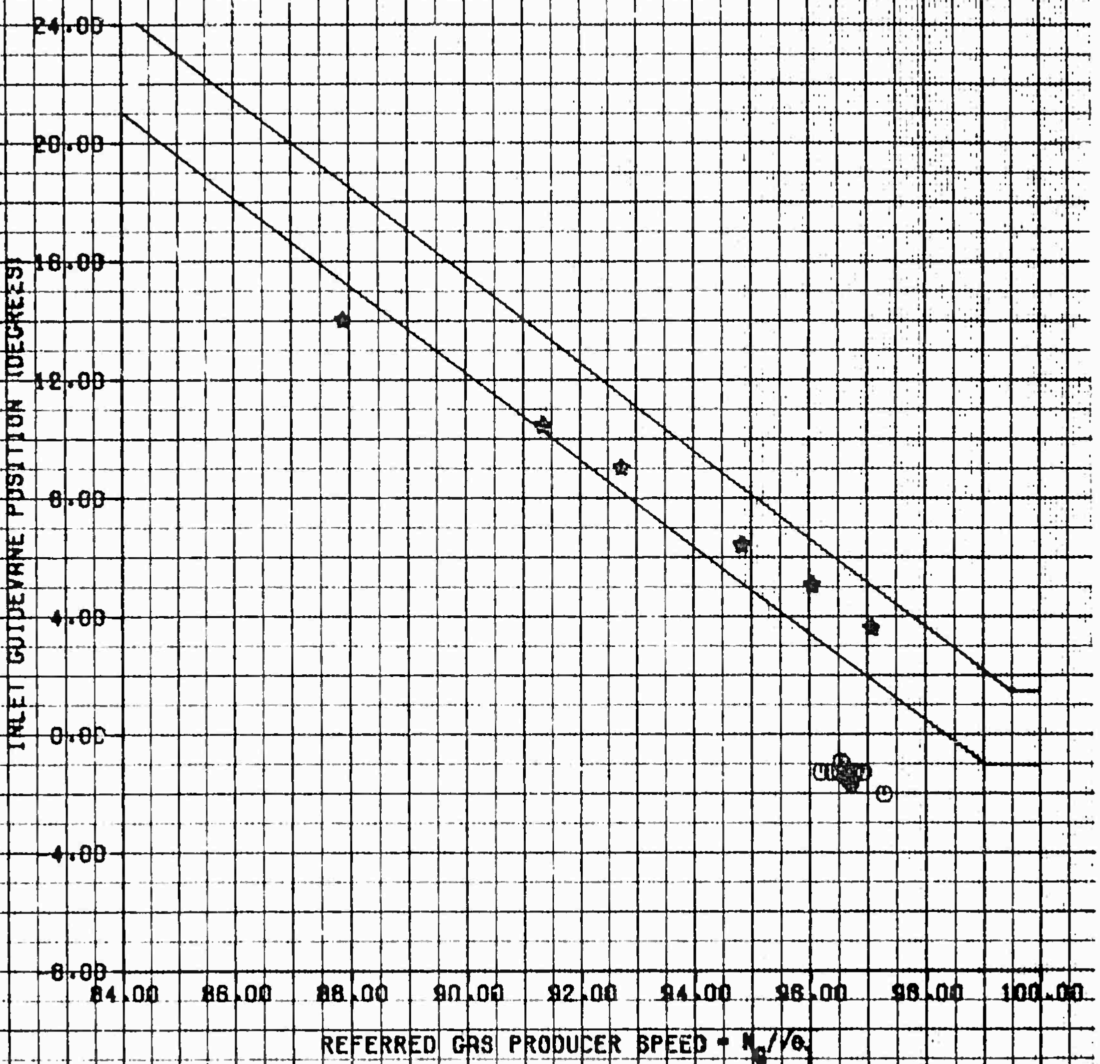
FIGURE 153

INLET GUIDE VANE AND REFERRED GAS PRODUCER

AIRCRAFT YAH-63 USA S/N 74-22248

RIGHT ENGINE YT700-GE-700 S/N 207243

SYMBOL	TIME SINCE CAL IB (HOURS)	DATA SOURCE OR FLIGHT CONDITION
——	0.	PID SPEC AGC-CP-2222-02000
☆	0.	TEST CELL CALIBRATION
⊙	3.8	HOVER



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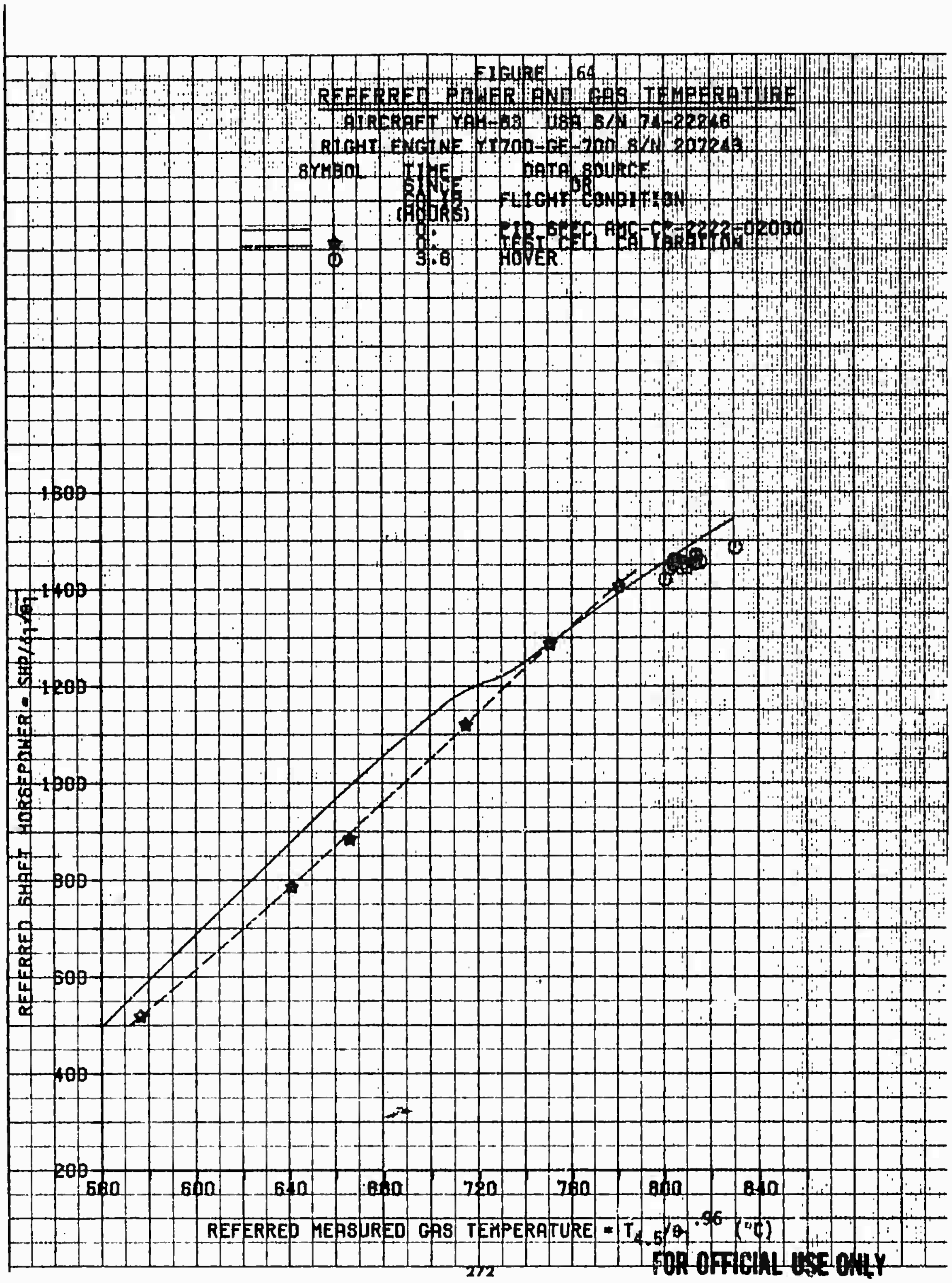

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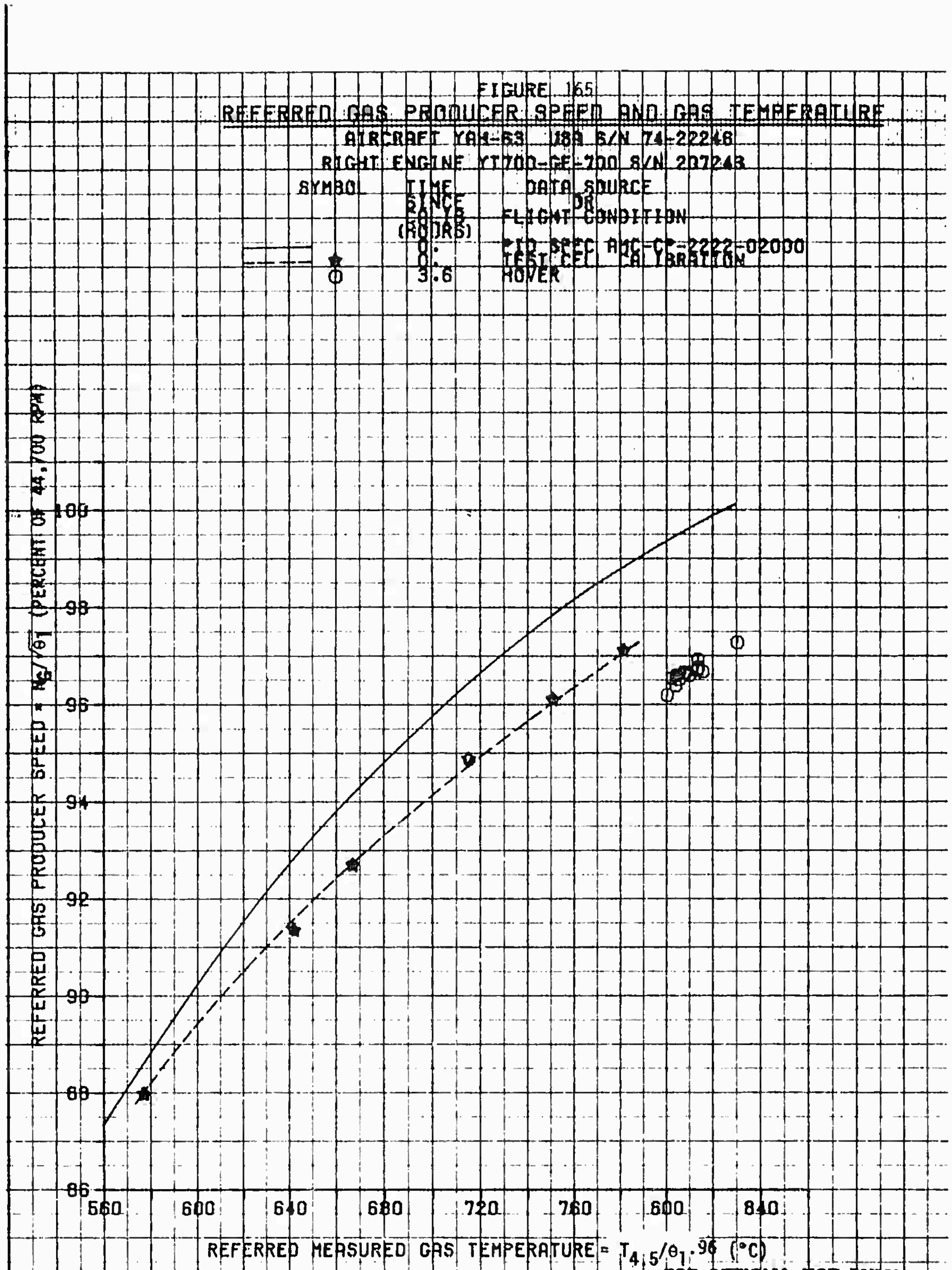

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FIGURE 180

REFERRED FUEL FLOW AND GAS TEMPERATURE

AIRCRAFT YAH-63 USA S/N 74-22248

RIGHT ENGINE YT700-GE-700 S/N 207248

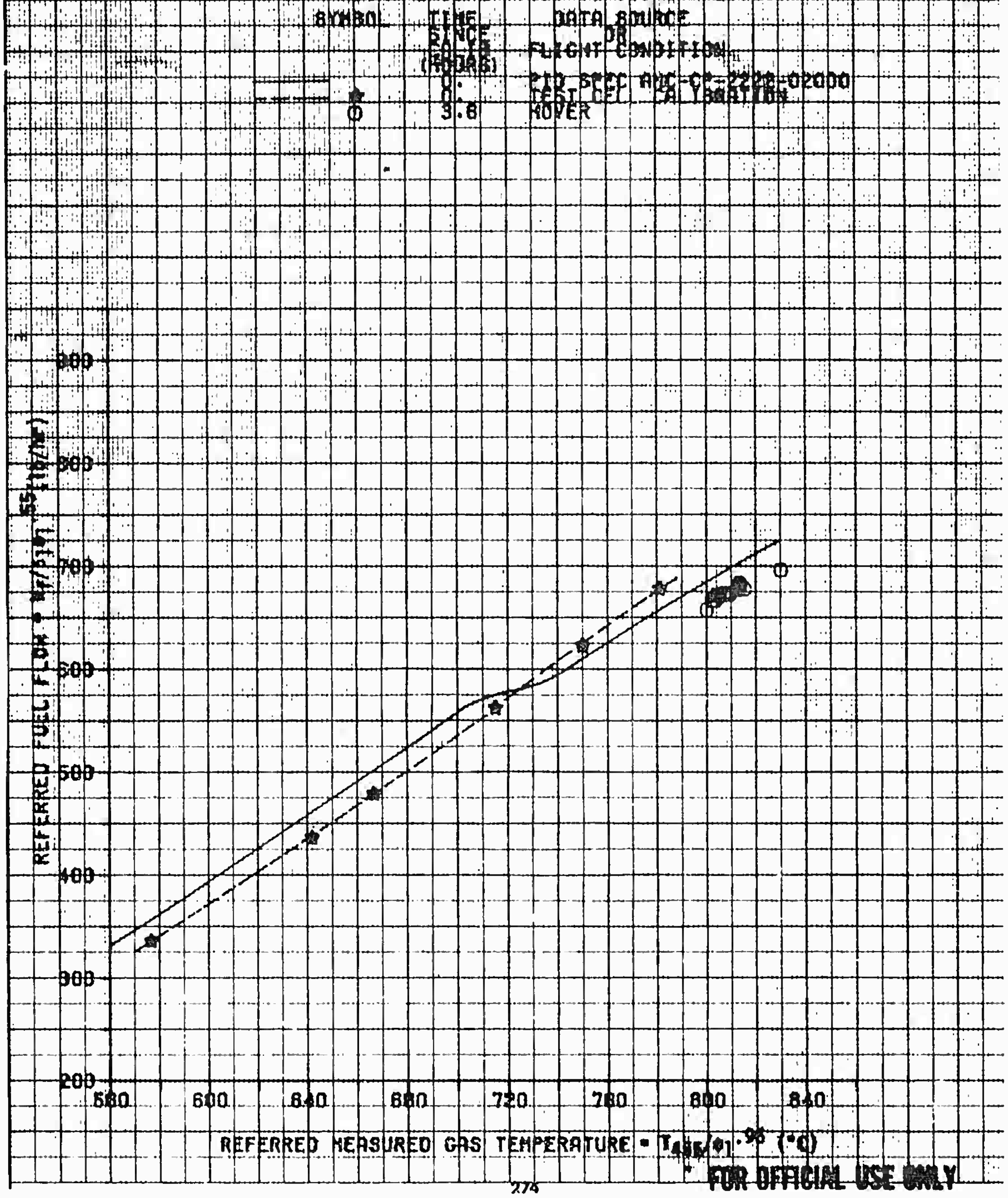

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FIGURE 167

QUICK STOP

YAH-63 USA S/N 74-22246

GROSS WEIGHT (LB)	CG LOCATION LONG (IN.)	CG LOCATION LAT (IN.)	DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY ROTOR SPEED (RPM)	ENTRY CALIBRATED AIRSPEED (KT)	SCAS CONDITION
15120	295.5 (FWD)	-.5 (LT)	3500	20.5	275	119	ON

NOTE: CLEAN CONFIGURATION

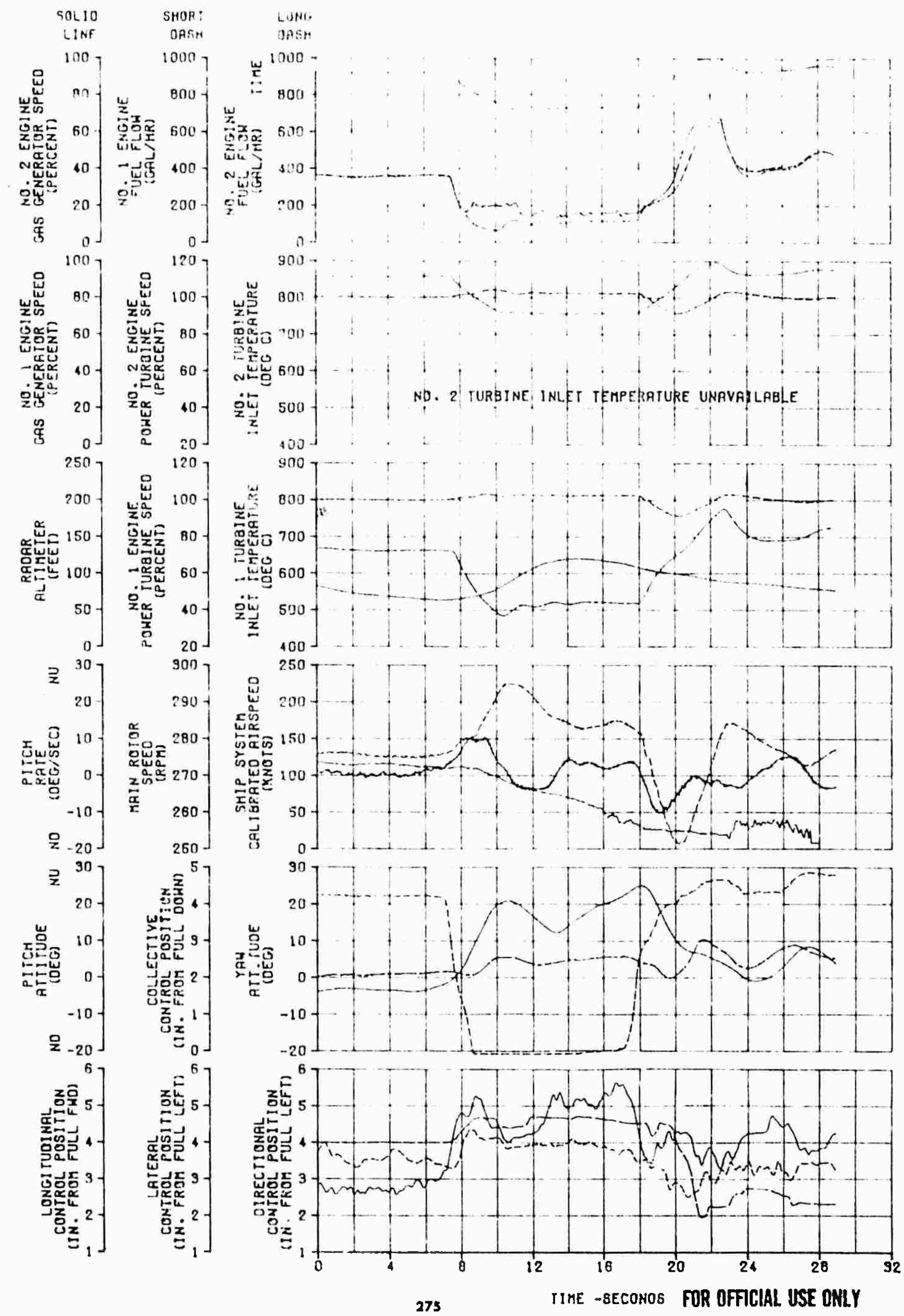

FIGURE 160

ELECTRICAL CONTROL UNIT LOCKOUT CHARACTERISTICS

YAH-63 USA S/N 74-22246

DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY ROTOR SPEED (RPM)
3400	20.0	274

NOTE: GROUND RUN

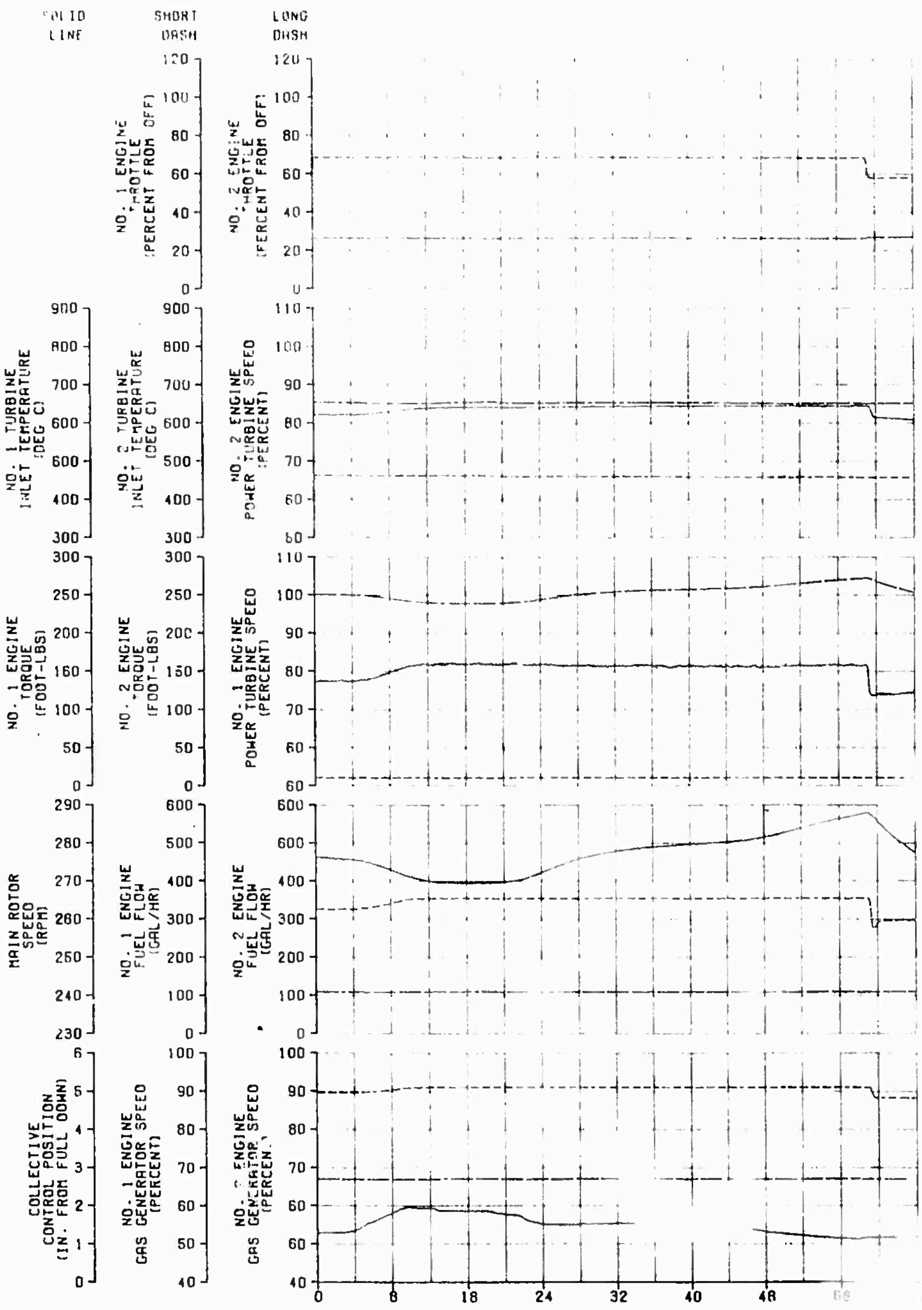

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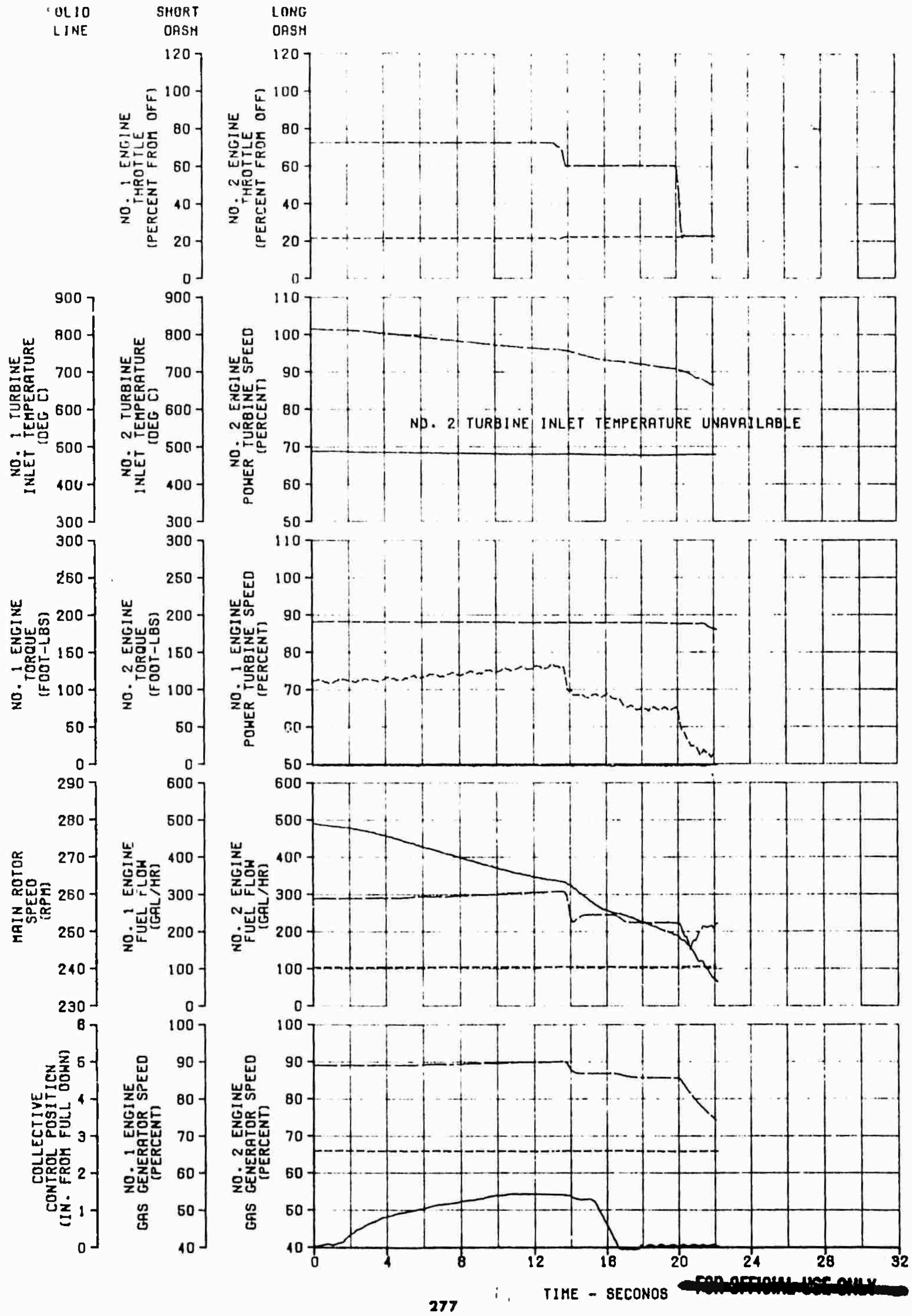
FIGURE 169

ELECTRICAL CONTROL UNIT LOCKOUT CHARACTERISTICS

YAH-63 USA S/N 74-22246

DENSITY ALTITUDE (FT)	OAT (DEG C)	ENTRY ROTOR SPEED (RPM)
3100	18.5	279

NOTE: GROUND RUN



[redacted]

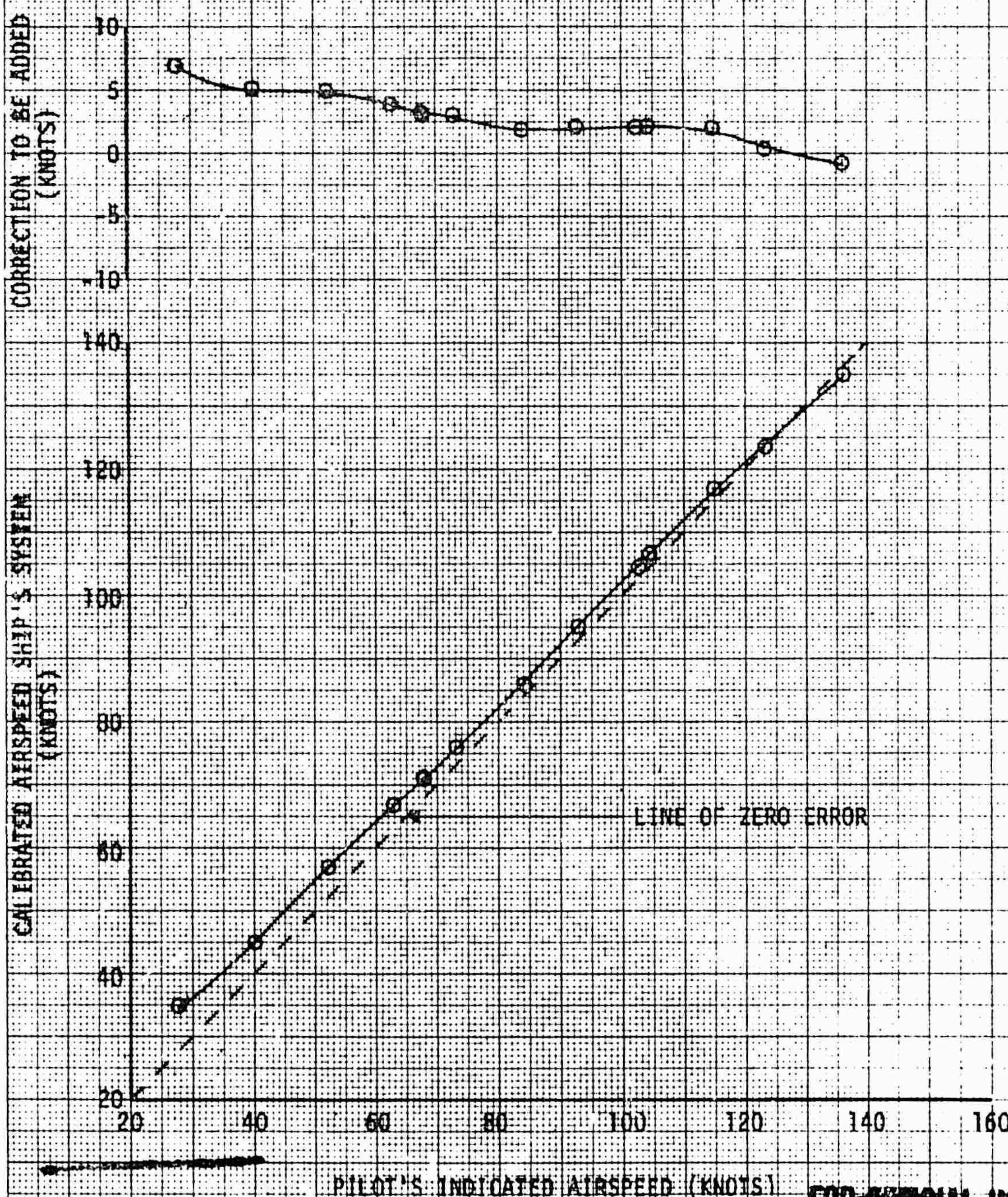
FIGURE 170

AIRSPEED CALIBRATIONS SHIP'S SYSTEM

YAH-63 USA S/N 74-22246

AVG GROSS WEIGHT (LB)	AVG CG LOCATION LONG (FS)	AVG CG LOCATION LAT (BL)	AVG DENS ALT (FT)	AVG OAT (°C)	ROTOR SPEED (RPM)	SCAS CONDITION
16480	293.0(FWD)	+.6(LT)	6480	23.0	276	ON

NOTE: CLEAN CONFIGURATION.



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APPENDIX H. EQUIPMENT PERFORMANCE REPORTS

The following EPR's were submitted during the YAH-63 DT I evaluation:

EPR Number	Date Submitted	Descriptive Title
74-07-1-1	20 Jul 1976	Performance deterioration of YT700-GE-700 engine
74-07-1-2	21 Jul 1976	Malfunction of attitude indicator
74-07-1-3	17 Aug 1976	Excessive wear of main rotor pitch link rod end bearings
74-07-1-4	18 Aug 1976	Suspected malfunction of engine electrical control unit
74-07-1-5	27 Aug 1976	Fuel shut-off valve failed to open
74-07-1-6	27 Aug 1976	Auxiliary power unit failed to start
74-07-1-7	9 Sep 1976	Severe engine compressor stalls
74-07-1-8	9 Sep 1976	Engine compressor seizure
74-07-1-9	9 Sep 1976	Structural failure of number one swashplate input control link
74-07-1-10	22 Sep 1976	Debonding of main rotor trunion elastomeric bearing
74-07-1-11	22 Sep 1976	Malfunction of engine starter sequence valve

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