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ARMY PRELIMINARY EVALUATION JOH-58A HELICOPTER WITH LOW REFLECTIVE PAINT AND INFRARED COUNTERMEASURE EXHAUST SYSTEM

ARMY AVIATION ENGINEERING FLIGHT ACTIVITY

DECEMBER 1975

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ARMY PRELIMINARY EVALUATION JOH-58A HELICOPTER WITH LOW REFLECTIVE PAINT AND INFRARED COUNTERMEASURE EXHAUST SYSTEM

FINAL REPORT

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

The United States Army Aviation Engineering Flight Activity conducted an Army Preliminary Evaluation of a JOH-58A helicopter painted with a low reflective paint and equipped with a prototype infrared countermeasure exhaust system from 1 July through 14 October 1975. The testing was conducted at Edwards Air Force Base and at Point Mugu Naval Air Station, California. Results of these tests were compared with test results from earlier evaluations of the OH-58A performance

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20. Abstract

and handling qualities characteristics and with preliminary base-line testing performed during this evaluation. One deficiency and two shortcomings were noted during this evaluation. Although not classified as a deficiency or shortcoming, the most significant finding in this evaluation was the serious degradation in hover and level flight performance when the rotor blades are painted with low reflective paint.

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INTRODUCTION

BACKGROUND

1. The Aircraft Survivability Equipment (ASE) Product Manager is currently attempting to provide the OH-58A helicopter with increased survivability from the enemy infrared (IR) and heat-seeking missile threat. The United States Army Aviation Engineering Flight Activity (USAAEFA) was tasked by the United States Army Aviation Systems Command (AVSCOM) (ref 1, app A) to evaluate the effect of a low reflective (LR) paint and an IR countermeasure (IRCM) exhaust system on the performance and handling qualities of an OH-58A helicopter.

TEST OBJECTIVES

2. The test objectives were as follows:

a. To verify the airworthiness of the helicopter with LR paint and the IRCM exhaust stacks installed.

b. To determine the changes in the OH-58A handling qualities and performance characteristics with LR paint and the IRCM exhaust stacks installed.

DESCRIPTION

3. The OH-58A is a single-main-rotor turbine-powered light observation helicopter manufactured by Bell Helicopter Company (BHC), Fort Worth, Texas. The main rotor is a two-bladed, semirigid, teetering type and the tail rotor is a two-bladed, semirigid, delta-three hinge type. The cockpit configuration is two-place (pilot and copilot/observer) and the cargo compartment has provisions for two passengers. All primary flight controls are hydraulically boosted and irreversible, while the directional control on the standard OH-58A aircraft is unboosted and reversible. The main landing gear is a fixed, energy-absorbing skid type. The helicopter is powered by an Allison T63-A-700 free gas turbine engine with a takeoff power rating of 317 shaft horsepower (shp) under sea-level, standard-day uninstalled conditions. The main transmission has a rating of 270 shp (maximum continuous) with a takeoff power limit of 317 shp (5-minute rating). A more detailed description of the OH-58A helicopter is contained in the operator's manual (ref 2, app A). The modifications incorporated in the test helicopter which resulted in the "J" designation were installation of a BHC electronic 3-axis stability and control augmentation system (SCAS) which incorporates a hydraulically boosted and irreversible directional control and a Sperry helicopter command information system (HCIS). The SCAS was not used during handling qualities testing. A more

detailed description of the SCAS and HCIS is contained in reference 3. The aircraft was further modified by the installation of IRCM exhaust stacks and application of LR paint, as described in paragraphs 4 and 5, respectively.

4. The prototype exhaust stacks used in this evaluation were designed to reduce the IR signature of the aircraft. They were designed to use the exhaust nozzle as an aspirating pump to draw cool air through an annular opening in the stack. The cooling air helps to protect the shield section of the stack from hot exhaust gases. The shield is also cooled by the flow of ambient air over the cooling fins. The exhaust stacks were manufactured by the Hughes Helicopter Division of Summa Corporation, Culver City, California. Photographs of the exhaust stacks are shown in appendix B. A more detailed description is contained in appendix C.

5. The paint used on the test aircraft is a low reflective olive drab acrylic lacquer (FSN 8010-083-6588) specified by military specification MIL-L-46159. It is formulated to reduce IR solar reflections in the spectral band pass of all currently identified IR-seeking missiles. It is also designed to have a low visual gloss to aid in visual contrast reduction. The exterior surfaces were painted with the LR paint with the exception of high conspicuity and/or safety markings, main rotor mast, tail rotor drive, tail rotor blades, and main rotor blade leading edges (ref 10, app A). Painting of the main rotor blades in accordance with reference 10 resulted in the rotor blade having 76 percent of chord (measured from the trailing edge) covered with the LR paint.

TEST SCOPE

6. This APE provided a limited evaluation of performance, handling qualities, and engine characteristics of the JOH-58A helicopter equipped with LR paint and IRCM exhaust stacks. The configurations tested are describe in paragraph 9 and table 1. Testing was performed at Edwards Air Force Base (2302 feet field elevation) and at the Pacific Missile Test Center (13 feet field elevation), Point Mugu, California, from 1 July through 14 October 1975. Forty-eight test flights for a total of 41 productive flight test hours were flown. The test aircraft was evaluated against the requirements of military specification MIL-H-8501A (ref 4, app A), with the deviations contained in reference 12.

7. Test results were compared to those previously reported for the OH-58A helicopter with standard paint and exhaust stacks (refs 5 and 6, app A). Flight limitations contained in the operator's manual and the safety-of-flight release (ref 7) were observed during the tests, with the exception of minimum transient rotor speed which was violated inadvertently during the simulated sudden engine failure tests.

TEST METHODOLOGY

8. Standard flight test techniques were used for the handling qualities and performance testing (refs 8, 9, and 13, app A). Test methods are described briefly in the Results and Discussion section of this report. A detailed description of the test instrumentation is contained in appendix F. A description of the data reduction procedures is contained in appendix E. Pilot comments were used to aid in the analysis of data and to determine the overall qualitative assessment of the flying qualities of the JOH-58A helicopter with LR paint and IRCM exhaust system installed. The Handling Qualities Rating Scale (HQRS) used to augment pilot qualitative comments is included as figure 1, appendix E.

AIRCRAFT TEST CONFIGURATIONS

9. Initial testing for base-line data was performed with the test helicopter in the standard configuration (configuration A). After the initial testing had been completed, the aircraft was equipped with the IRCM exhaust stacks (configuration B) and 2.7 hours of testing were completed. After testing in configuration B, the helicopter was delivered to Sharpe Army Depot, California, where it was painted with LR paint (ref 10, app A). The main rotor blades were painted the entire length of the blade from the trailing edge forward to the point of maximum airfoil thickness (configuration C). Nineteen hours of testing were completed in this configuration. Additional testing was then conducted with the same paint as configuration C but with standard exhaust stacks (configuration D). After testing with configurations A, B, C, and D, it was thought that the small ridge of paint which existed where the normal acrylic lacquer, used from the leading edge to a point of maximum airfoil thickness met the LR paint might be a significant performance factor. The ridge was removed by sanding the area of transition between the normal acrylic lacquer on the leading edge and the LR paint on the remainder of the rotor blade (configuration E) and further testing was conducted. Additional testing was accomplished with the IRCM exhaust stacks and a new set of rotor blades with standard lacquer paint (configuration F) in an attempt to determine the amount of degradation in performance attributable to the painted rotor blades. The remaining tests were conducted with standard exhaust stacks and standard rotor blades (configuration G). An abbreviated description of the aircraft configurations is contained in table 1.

Table 1. Aircraft Test Configurations.

Configuration Designation	Configuration Description
A	Std paint, std stacks, std rotors (basic aircraft)
B	Std paint, IRCM stacks[1], std rotors
C	LR paint[2], IRCM stacks, LR rotors[3]
D	LR paint, std stacks, LR rotors
E	LR paint, IRCM stacks, LR rotor (ridge removed)[4]
F	LR paint, IRCM stacks, std rotor
G	LR paint, std stacks, std rotor

[1]IRCM stacks: Hughes IRCM exhaust suppressor stacks installed.
[2]LR paint: Fuselage painted with LR acrylic lacquer.
[3]LR rotors: Rotor blades painted with LR acrylic lacquer.
[4]LR rotor (ridge removed): LR rotor that was sanded to remove the paint ridge formed between paint applied to leading edge and the LR paint on the remainder of the rotor blade.

RESULTS AND DISCUSSION

GENERAL

10. An APE of the OH-58A helicopter was performed to determine any changes in the OH-58A performance and handling qualities characteristics when painted with LR paint and equipped with IRCM exhaust stacks. Results of these tests were compared with results from earlier evaluations of the OH-58A performance and handling qualities characteristics and with preliminary base-line testing performed during this evaluation. One deficiency and two shortcomings were noted. Although not classified as a deficiency or shortcoming, the most significant finding in this evaluation was the serious degradation in hover and level flight performance when the rotor blades are painted with LR paint.

PERFORMANCE

Hover

11. Hover performance capabilities of the OH-58A helicopter were evaluated in ground effect (IGE) and out of ground effect (OGE) at various skid heights under the conditions shown in table 2. The free flight hover method was used to determine hover performance capability. Skid height was measured by visual reference to a measured weighted cord attached to the left skid. Incremental amounts of ballast were added to the helicopter until the gross weight was such that either the engine temperature or transmission power limit was reached.

12. The hover performance capabilities are summarized and presented in figures 1 through 11, appendix G. Nondimensional plots of power coefficient (C_P) versus thrust coefficient (C_T) are presented for configuration C (LR-painted fuselage and rotor and IRCM exhaust stacks) in figures 1 through 7. A comparison of the standard-day and 35°C hot-day foot IGE and OGE hover ceiling is presented in figures A and B for configurations A and C. Depicted in table 3 is the loss in gross weight hover capability incurred with LR paint and/or installation of the IRCM exhaust stacks.

13. The standard-day OGE hover ceiling decreased from 5100 feet for the standard aircraft (configuration A) to 2850 feet for configuration C at 3000 pounds gross weight (fig. B). The 35°C hot-day 4-foot IGE hover ceiling decreased from 2200 feet (configuration A) to a point that the aircraft will only hover at 2980 pounds gross weight at sea level, as shown in figure A.

Table 2. Hover Performance.[1]

Configuration	Skid Height[2,3] (ft)	Gross Weight (lb)	Density Altitude (ft)	Outside Air Temperature (°C)
A	2, 4, 10, 20, and 50	2750 to 3250	4040	24.5
B	Not tested			
C	2, 4, 10, 20, and 50	2315 to 3500	Sea level to 4150	15.5 to 27.0
D	50	2450 to 3010	340	15.5
E	50	2530 to 3040	2860	16.0
F	50	2450 to 3110	2820	15.5
G	2 and 50	2090 to 3030	2820	14.5

[1]Rotor speed: 347 to 354 rpm.
[2]Free flight hover technique used.
[3]2, 4, 10, and 20 feet are IGE hover points, and 50 feet is OGE hover.

Table 3. OGE Hover Weight Capability.[1]

Configuration	Gross Weight (lb)	Decrease in Lift Capability (lb)
A	2930	---
C	2680	250
D	2790	140
E	2740	190
F	2820	110
G	2930	---

[1]Sea level, 35°C hot day.

FIGURE [illegible]

SUMMARY HOVER PERFORMANCE COMPARISON

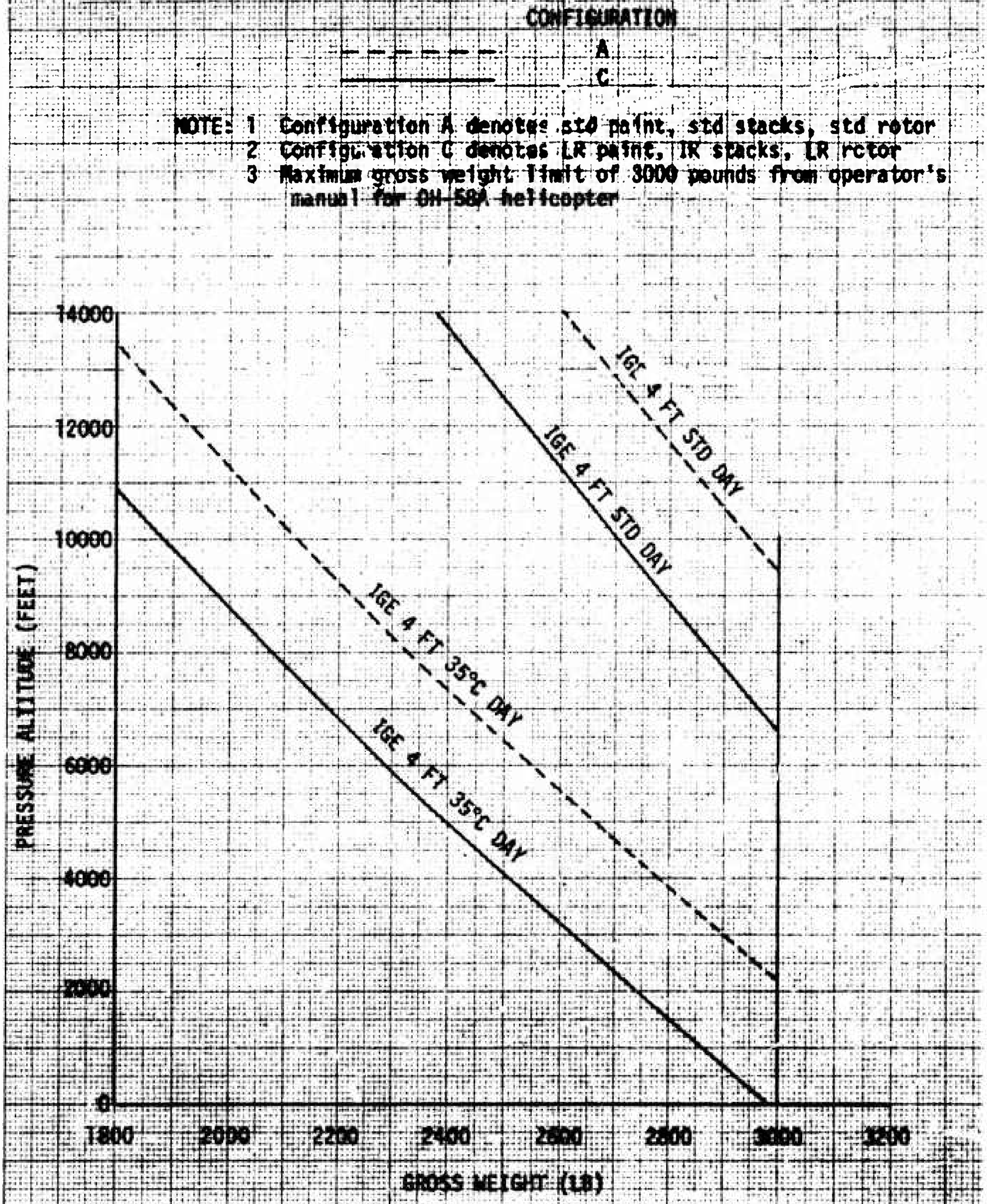
JOH-58A

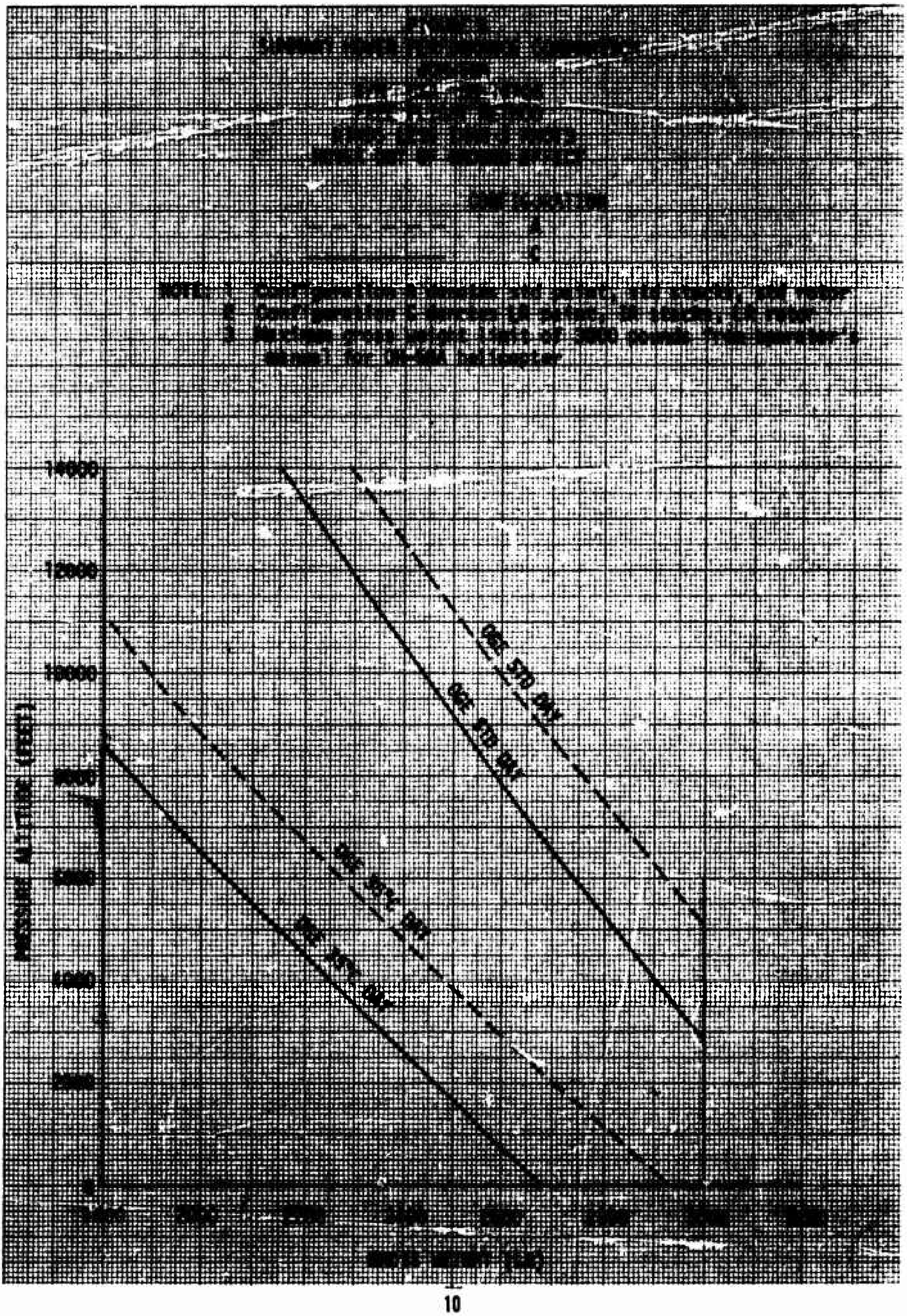
S/N USA 68-16706

FREE FLIGHT METHOD

WINDS LESS THAN 3 KNOTS

HOVER IN 4 FOOT GROUND EFFECT




CONFIGURATION
A
C
14000
12000
10000
OGE STD DAY
IGE STD DAY
OGE 35°C DAY
IGE 35°C DAY

14. With the paint ridge removed (configuration E), hover performance was slightly improved over configuration C, as shown in figure 9, appendix G. The OGE hover ceiling for configuration E was 3600 feet, standard day, at 3000 pounds gross weight and for a 35°C hot day, maximum gross weight for sea-level hover (OGE) was 2740 pounds, as computed from figures 9 and 60.

15. As depicted in figure 11, appendix G, the OH-58A hover performance is not degraded by applying LR paint to the fuselage. However, painting the rotor blades with LR paint greatly reduced the capability of the OH-58A helicopter to perform mission maneuvers, in that OGE hover lift capability was reduced approximately 5 percent.

16. As shown in figures 8 and 10, appendix G, the OH-58A hover capability was significantly reduced by the addition of the IRCM exhaust stacks. As depicted, the addition of the stacks resulted in an approximate 4 percent loss in lift capability in an OGE hover.

17. The engine inlet air temperature (T_2) was monitored throughout the hover evaluation and within the scope of this test, the aircraft did not ingest exhaust gases.

Level Flight

18. Tests were conducted to determine airspeed, fuel flow, and power-required relationships to define the OH-58A level flight performance. Conditions, and configurations tested are presented in table 4. All flights were conducted at zero sideslip in stabilized unaccelerated level flight. Each speed power test was conducted at a constant weight-to-density ratio (W/σ). This procedure necessitated an increase in altitude for successive data points as fuel was consumed.

19. Test results, to include nondimensional summary plots, specific range, various individual configuration power polars, and endurance summaries, are presented in figures 12 through 26, appendix G. The nondimensional summary plots, specific range, endurance summaries, and the individual power polars are presented in figures 12 through 23 for configuration F. Test results are summarized in table 5.

20. The standard aircraft with IRCM exhaust stacks installed (configuration B) had an increase in parasite drag of 2.0 square feet of equivalent flat plate area (Δf_e) over the standard aircraft (configuration A), as shown in figure 24, appendix G. An increase of 18 shp was required at 100 knots true airspeed (KTAS), which was attributed to the increase in f_e and upward thrust component of the engine exhaust.

Table 4. Level Flight Performance.[1]

Configuration	Average Gross Weight (lb)	Average Density Altitude (ft)	Outside Air Temperature (°C)	Thrust Coefficient	True Airspeed Range (kt)
A	2820	7200	17.0	0.003498	35 to 113
B	2820	7240	24.0	0.003503	35 to 105
C	3780	7880	18.0	0.003522	35 to 102
	2520	4080	27.0	0.002843	35 to 109
D	No flights in this configuration				
E	2460	4420	19.5	0.002804	34 to 115
	2860	10,940	10.5	0.003987	32 to 102
	2820	5240	20.5	0.003295	24 to 110
F	2620	9620	12.0	0.003515	36 to 110
	2880	9100	12.5	0.003790	25 to 102
	2560	3320	24.0	0.002815	23 to 115

[1]Rotor speed for all tests: 354 rpm.
Bleed air OFF, all doors ON, unarmed.

Table 5. Level Flight Performance Comparison.[1]

Configuration	Power Required at 100 KTAS (shp)	Increase in Power Required Over Standard Aircraft (Δshp)	Specific Range (NAMPP)	Reduction in Specific Range From Configuration A (%)
A	206	--	0.625	--
C	244	38	0.541	13
E	231	25	0.568	9
F	223	17	0.588	6

[1]At 2800 pounds gross weight under standard-day, sea-level conditions.

21. Configuration C (LR-painted fuselage and rotor and IRCM exhaust stacks) had a large increase in power required over configuration A. As depicted in figure 25, appendix G, 20 more shp (9 percent of available) was required for 100 KTAS for configuration C. The large increase in power required over configuration A can be attributed to the increase in f_e due to the IRCM exhaust stacks and an increase in induced and profile drag from the LR paint.

22. As indicated in figure 26, appendix G, configuration E (LR-painted fuselage, modified LR-painted rotor, and IRCM exhaust stacks) showed an improvement in level flight performance over configuration C but still required a large power increase over the standard aircraft. For 100 KTAS an increase of 19 shp was required for configuration E, also attributed to the increase in f_e and the LR paint.

23. Configuration F (LR-painted fuselage, standard rotor, and IRCM exhaust stacks) (figs. 19 through 23, app G; summarized in figs. 12 through 14) shows some decrease in power required when compared to configurations D and E; however, a significant increase still exists in power required over the standard aircraft. The increase in power required can be attributed to the parasitic drag increase of the IRCM exhaust stacks and the LR-painted fuselage. [redacted] Configuration F decreased the specific range by 14 percent under standard-day, sea-level conditions from that of the standard aircraft, as shown in figure 15.

24. Level flight performance was degraded by the addition of LR paint and/or IRCM exhaust stacks. Painting the rotor blades with LR paint degraded the level flight performance to an extent that the aircraft's operational capability is seriously reduced.

HANDLING QUALITIES

Control Positions in Trimmed Forward Flight

25. The trimmed control positions were determined in forward flight in configuration A (basic aircraft), configuration B (basic aircraft with IRCM exhaust stacks), and configuration E (LR-painted fuselage, modified rotor, and IRCM exhaust stacks). The tests were conducted with the helicopter trimmed in zero sideslip steady-heading level flight in 10-knot increments from 25 knots calibrated airspeed (KCAS) to 105 KCAS. As depicted in figure 27, appendix G, there was essentially no difference in control positions (less than 1 inch) between the configurations tested. All control margins were adequate and the requirements of MIL-H-8501A concerned with control margin were met. Within the scope of this test, the LR paint on the fuselage and rotor blades and/or the addition of the IRCM exhaust stacks had no significant effect on control positions in trimmed forward flight.

Static Longitudinal Stability

26. Static longitudinal stability of the OH-58A helicopter was evaluated by varying the airspeed in 5-knot increments using the longitudinal cyclic control while maintaining the collective fixed at the initial trim position. Control positions were recorded while the helicopter was stabilized at each incremental airspeed above and below the trim airspeed. Test results are presented in figures 28 through 31, appendix G.

27. For all configurations tested (A, B, and C), the aircraft possessed positive static longitudinal stability as evidenced by the forward longitudinal control displacement for increased airspeed and aft longitudinal control displacement for decreased airspeed from trim. In configuration C (LR-painted fuselage and rotor blades and IRCM exhaust stacks) more forward longitudinal control was required at trim than in the other configurations, with no discernible change in control position gradient; however, the maximum difference in control positions between configurations was less than 1 inch and was not considered significant.

28. There was less than 1 inch of lateral cyclic and tail rotor control (pedal) variation between configurations (figs. 28 through 31, app G). Variations in the lateral cyclic movement and pedal travel with changes in airspeed between configurations were insignificant. The requirements of MIL-H-8501A for static longitudinal stability were met for all conditions tested.

Static Lateral-Directional Stability

29. Static lateral-directional stability was evaluated under the flight conditions detailed in table 6. The tests were conducted by establishing a trim airspeed with zero sideslip and incrementally varying the sideslip angle while maintaining constant airspeed, collective control, and steady heading. Control positions were recorded at each stabilized steady-heading sideslip angle. The results are presented in figures 32 through 34, appendix G. As shown, there is no significant difference between configurations.

30. Static directional stability was positive for all airspeeds and configurations, as indicated by the directional control position versus sideslip gradient. However, the shallow gradient near trim (zero sideslip) at low airspeeds causes poor directional stability. The poor directional stability permits a residual directional oscillation noted during flight at low airspeeds. This characteristic is common to the OH-58A helicopter and is not attributable to the LR paint or IRCM exhaust system.

31. Side force, as indicated by the bank angle versus sideslip gradients (figs. 32 through 34, app G), increased with an increase in airspeed. At low airspeed, the side forces were light and did not provide the pilot adequate cues to small sideslip excursions. At higher airspeeds, side force was sufficient to prevent the pilot from inadvertently flying in a sideslip. Although side-force cues were insufficient at low airspeeds, the allowable sideslip limit would not normally be exceeded. Within the scope of this test, side-force characteristics are satisfactory.

Table 6. Static Lateral-Directional Stability.[1]

Configuration	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)
A	9060	18.5	2840	107.8 (fwd)	85
A	8400	22.0	2780	107.6 (fwd)	40
B	7060	21.5	2780	107.6 (fwd)	86
C	7740	15.0	2820	107.1 (fwd)	42
C	6660	15.5	2860	107.2 (fwd)	80

[1]Rotor speed: 354 rpm.
Doors-on, unarmed configuration.

32. Effective dihedral was weak in all configurations; however, it was not objectionable at the airspeeds and sideslips tested. Longitudinal control displacement changed only slightly throughout the sideslip range tested. There was no significant change between configurations.

Maneuvering Stability

33. The maneuvering stability characteristics of the OH-58A helicopter were tested in configuration A (basic aircraft), configuration B (IRCM exhaust stacks), and configuration C (LR paint and IRCM exhaust stacks). The symmetrical pull-up was the test technique used during this test. The helicopter was trimmed in level flight at the desired airspeed, after which a cyclic pull-up to a slightly higher altitude was initiated. A dive was then established and the helicopter was accelerated to slightly higher than trim airspeed. A symmetrical pull-up was executed so as to pass through the required trim airspeed, altitude, a nose-level pitch attitude, and the desired load factor simultaneously. Longitudinal stick force, stick position, and normal acceleration were recorded at each test point; the results are presented in figure 35, appendix G, and summarized in figure C.

34. The boost-on stick force gradient relative to the acceleration (FS/g) was positive for all configurations tested, *ie*, a greater aft cyclic force was required for an increased load factor. Considerably less stick force and travel were required with configuration C for the same amount of g force than with the basic aircraft. Although stick-force-per-g gradients were light in configuration C, it is not likely that the helicopter operational flight envelope will be exceeded during normal operation because of the cues provided by the 2-per-rotor-revolution (2/rev) vertical vibration (impending blade stall) near 2g and the large pitch attitudes required. The OH-58A maneuvering stability characteristics are satisfactory for all configurations tested.

Dynamic Stability

35. Longitudinal, lateral, and directional dynamic stability characteristics of the OH-58A helicopter were tested under the conditions listed in table 7. Representative results are presented as time histories in figures 36 through 51, appendix G.

36. The longitudinal long-period stability characteristics were evaluated by stabilizing the helicopter at trim airspeed and then increasing or decreasing the airspeed in desired increments, using only the cyclic control. The cyclic was then returned to the trim position and the helicopter response was recorded on magnetic tape. The long-period oscillatory mode was convergent for all level flight conditions (figs. 36 through 38, app G) except at low airspeeds (40 KTAS trim), where a neutral long-period oscillation was noted (figs. 39 and 40). The period of oscillation was approximately 23 seconds. The oscillation was easily excited, resulting in moderate pilot compensation required to maintain 40 knots indicated airspeed (KIAS) (HQRS 4). The easily excitable longitudinal long-term oscillation at low airspeeds (approximately 40 KIAS) occurring in all configurations tested

FIGURE C
MANEUVERING STABILITY
JOH-58A
S/N USA 68-16706

NOTE: 1 Configuration A denotes std paint, std stacks, std rotor
2 Configuration B denotes std paint, IR stacks, std rotor
3 Configuration C denotes IR paint, IR stacks, IR rotor

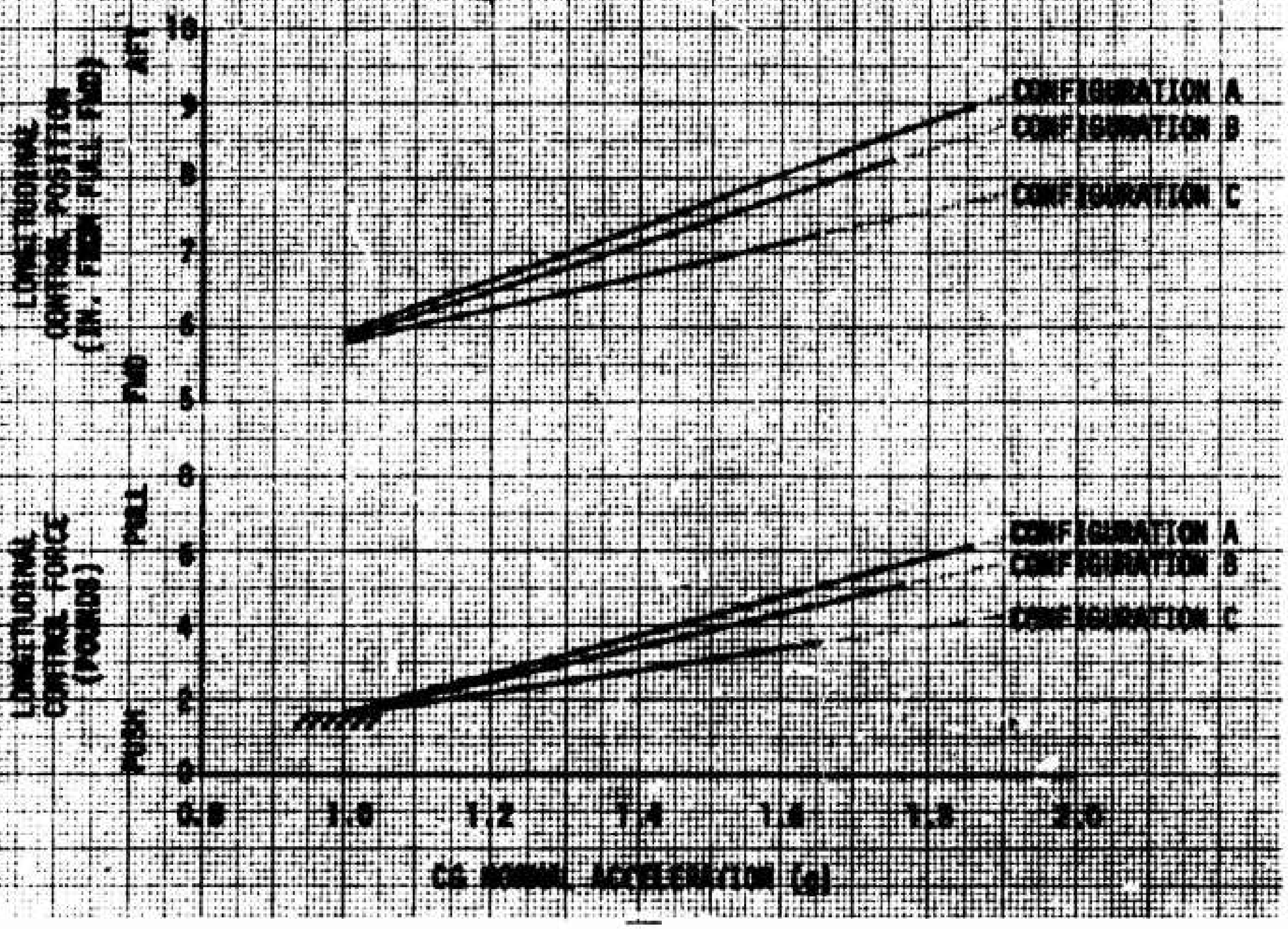


Table 7. Dynamic Stability.[1]

Test Description	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)	Configuration
Longitudinal long-term response	7680	24.0	2800	107.6 (fwd)	52	A
	7480	23.5	2780	107.6 (fwd)	88	A
	5860	17.0	2920	107.7 (fwd)	41	C
	7780	14.0	2720	107.5 (fwd)	42	C
Longitudinal pulse	7200	14.0	2780	107.5 (fwd)	88	C
Release from steady sideslip	8340	42.5	2740	107.6 (fwd)	42	A
	8320	42.0	2720	107.6 (fwd)	85	A
	6580	20.5	2760	107.6 (fwd)	83	B
	8100	14.0	2800	107.6 (fwd)	42	C
	8340	14.0	2800	107.6 (fwd)	85	C
	8400	22.0	2720	107.6 (fwd)	87	A
	6260	16.0	2840	107.6 (fwd)	87	C
	7220	14.0	2800	107.6 (fwd)	87	C
	6940	14.5	2800	107.6 (fwd)	89	C
	6280	15.0	2820	107.6 (fwd)	91	C

[1]Rotor speed for all tests: 354 rpm.
Aircraft flown with doors on in the unarmed configuration.

is a shortcoming of the basic OH-58A which is unaffected by the addition of LR paint or IRCM exhaust stacks. Within the scope of this test, there was no discernible difference between configurations tested.

37. The longitudinal short-period characteristics of the OH-58A helicopter were tested by applying cyclic pulse inputs to excite the aircraft short-period mode. As indicated in figure 41, appendix G, the short-period mode was essentially deadbeat and representative of all configurations tested.

38. The dynamic lateral-directional stability characteristics of the OH-58A helicopter were tested in level flight under the conditions shown in table 7. Gust response tests were conducted by applying 1.5-inch lateral and directional control inputs (doublets) to the controls independently in both directions for 0.5 second duration. Additionally, releases from steady-heading sideslip were made to excite the Dutch-roll characteristics.

39. In all configurations the OH-58A had a lightly damped lateral-directional oscillation (Dutch-roll) (figs. 42 through 46, app G) and a neutral spiral stability (figs. 47 through 51). The roll-to-yaw ratio was approximately 0.5 (ϕ/β = 0.5) and the period was approximately 4 seconds. The inherent lateral-directional oscillations were annoying and required a moderate amount of pilot compensation (HQRS 4) to maintain a constant heading. Although a shortcoming, the lateral-directional oscillation characteristics were not altered by the addition of IRCM exhaust stacks or LR paint.

40. Flight was made into light-to-moderate turbulence of varying degrees in configurations A and C at 80 KCAS (operator's manual recommended turbulence penetration airspeed). Considerable pilot effort (HQRS 5) was required to maintain straight and level flight as airspeed varied 10 knots, pitch attitude changed ±3 degrees, roll ±4 degrees, and sideslip varied ±5 degrees. The constant attitude changes resulted in considerable pilot compensation and made flying in such conditions difficult. The amount of attitude and airspeed excursions did not perceptibly change between configurations. However, the neutral long-term response discussed in paragraph 36 will increase pilot workload should the flight be made in turbulence at low airspeeds (40 to 50 KCAS).

Simulated Sudden Engine Failure

41. Simulated sudden engine failure tests (throttle chops) were conducted under the conditions detailed in table 8. To simulate a sudden engine failure, the helicopter was trimmed at a given flight condition and the throttle abruptly closed to the flight-idle position. The flight controls were held fixed until either a minimum transient rotor speed of 304 rpm was reached or until 2 seconds had elapsed. The delay in moving the controls was to simulate the normal delay in pilot reaction time following an actual engine failure. The resultant pitch, roll, and yaw excursions were recorded, as were rotor speed decay rates. The results of these tests, including time history plots, are presented in figures 52 through 55, appendix G.

Table 8. Simulated Sudden Engine Failure.[1]

Configuration	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)
A	8300	21.0	2720	107.5 (fwd)	52
A	7660	21.0	2740	107.5 (fwd)	97
C	5960	16.5	2880	107.5 (fwd)	51
C	6200	16.0	2880	107.5 (fwd)	51
C	5660	17.0	2860	107.5 (fwd)	52
C	6020	16.0	2860	107.5 (fwd)	96
C	5940	16.0	2860	107.5 (fwd)	96

[1]Rotor speed: 354 rpm.
Door-on, unarmed configuration.

42. Aircraft response varied between configurations. The most notable difference was rotor speed decay rate between configuration A (basic aircraft) and configuration C (LR paint and IRCM exhaust stacks). For configuration A the rate of decay was 16.0 rpm/sec and for configuration C 29.0 rpm/sec. In configuration A after lowering the collective, rotor speed decayed an additional 18 rpm (fig. 52, app G) and in configuration C an additional 20 rpm (fig. 54).

43. Should a sudden engine failure occur in configuration C with a 2-second delay as specified in MIL-H-8501A, the rotor speed can be expected to reach a minimum of 282 rpm, which is 22 rpm below the minimum allowable as specified in reference 12, appendix A. A rotor speed decay of such magnitude (72 rpm) may prevent the pilot from performing a safe autorotational entry and landing while observing the existing operator's manual height-velocity diagram. If the OH-58A rotor blades are to be painted with LR paint, additional autorotational testing should be conducted, to include height-velocity testing. The autorotational rotor speed decay rate and the minimum transient rotor speed encountered prevent the OH-58A from meeting the requirements of paragraph 3.5.5 of MIL-H-8501A. Although the standard OH-58A has reportedly failed to meet the specification requirements in previous tests, nonconformance with the specification is increased with the addition of LR paint. The excessive rotor speed decay rate with LR-painted rotor blades is a deficiency.

SUBSYSTEM TESTS

Engine Characteristics

44. A limited evaluation of the effects of IRCM exhaust stacks on engine performance was conducted and the results are shown in figure 56, appendix G. A single static probe was located 2 inches downstream from the marmon clamp on the inside of the left exhaust stack, as shown in photo C, appendix B. The high-pressure readings probably resulted from interference caused by the close proximity of the tap to the clamp. The pressure measured is approximately equal to the dynamic pressure ($Pv^2/2g$) and is probably not representative of the average static pressure at other circumferential locations. Meaningful data were difficult to obtain because of the close proximity of the tap to the flange clamp joint and the tangential velocities induced by the gas turbine. Further testing should be performed on a more fully instrumented exhaust installation to ascertain the true exhaust static pressure and its effect on engine performance.

45. Referred engine parameters were used to compare the test engine with the standard BHC exhaust extension against the same engine with the Hughes IRCM exhaust stacks installed. Data on referred gas producer speed, shp, turbine outlet temperature, and fuel flow are presented in figures 57 through 59, appendix G.

46. The referred shp versus referred gas producer speed shows that the gas producer speed of the engine equipped with the IRCM exhaust extension was increased over that of the standard ducts. At a gas producer speed equal to 100 percent of rpm, the engine equipped with IRCM exhaust stacks produced 3 shp less than the engine equipped with the standard ducts (fig. 57, app G). Referred shp versus referred turbine outlet temperature shows that the engine runs 6.5°C hotter (at the same power setting) when equipped with the IRCM exhaust stacks than with the standard ducts. At the turbine outlet temperature for normal rated power of 693°C, the engine when equipped with IRCM exhaust stacks produced 6.5 shp less (6.5 pounds more of fuel per hour at the same power setting) than the engine equipped with standard ducts (figs. 58 and 59).

47. The Hughes IRCM exhaust stacks have degraded the engine performance by approximately 2 percent. The engine model specification power and fuel flow (ref 11, app A) for the T63-A-700 engine were modified to reflect the installation of the IRCM exhaust stacks as presented in figures 60 through 62, appendix G. These corrected powers and fuel flows were used to calculate the hover performance summaries and the level flight range data presented in this report.

Engine Compartment Temperature Survey

48. A limited engine compartment temperature survey was performed with the Hughes IRCM exhaust stacks installed. Temperatures were recorded at locations specified in Allison Installation Design Manual (ref 11, app A). The temperature probes were located at the following locations: compressor section, top engine mount pad surface, ignition harness, thermocouple harness, and oil heat exchanger. Temperature data for all conditions tested are presented in figure 63, appendix G. When the component/fluid temperature data are corrected to the Army's design requirement maximum ambient temperature of 125°F (52°C), no overtemperature condition is indicated, as is shown in figure 65, appendix G. The temperature data were corrected by the following equation:

$$T = T_{measured} \frac{52 + 273}{T_{ambient} + 273}$$

(all temperatures are in degrees Centigrade)

The effect of altitude-temperature variation was not determined. Within the scope of this evaluation, the engine compartment and oil heat exchanger cooling were satisfactory.

CONCLUSIONS

GENERAL

49. The following general conclusions were reached upon completion of testing:

a. Painting the rotor blades with LR paint degraded the hover and level flight performance to an extent that the aircraft's operational capability is seriously reduced.

b. The configuration of LR-painted fuselage, standard rotor, and IRCM exhaust stacks resulted in a significant level flight performance reduction.

c. One deficiency and two shortcomings were identified during the evaluation.

DEFICIENCY AND SHORTCOMINGS

50. The following deficiency was identified: the excessive rotor speed decay rate with LR paint on the rotor blades (para 43).

51. The following shortcomings were identified:

a. Easily excited longitudinal long-term oscillation at low airspeeds (approximately 40 KIAS) unaffected by the addition of LR paint or IRCM exhaust stacks (para 36).

b. Inherent lateral-directional oscillations requiring moderate pilot compensation to maintain a constant heading were unaffected by the addition of LR paint or IRCM exhaust stacks (para 39).

SPECIFICATION COMPLIANCE

52. Within the scope of this test, the OH-58A helicopter failed to meet the requirements of paragraph 3.3.5 of MIL-H-8501A, in that the 2-second time delay following a simulated sudden engine failure was not achievable in a maximum power climb at 51 KCAS for all configurations tested (para 43).

RECOMMENDATIONS

53. If the OH-58A rotor blades are to be painted with LR paint, additional autorotational evaluations should be conducted, to include height-velocity testing (para 43).

54. Further testing should be performed on a more fully instrumented exhaust installation to ascertain the true exhaust static pressure and its effect on engine performance (para 44).

APPENDIX A. REFERENCES

1. Letter, AVSCOM, AMSAV-EFT, 5 November 1974, subject: OH-58A Suppressor DT II Army Preliminary Evaluation, as amended by telephone message to include LR paint.

2. Technical Manual, TM 55-1520-288-10, *Operator's Manual, Army Model OH-58A Helicopter*, July 1969.

3. Final Report, United States Army Aviation Systems Test Activity, Project No. 72-20, *Handling Qualities Evaluation of the OH-58A Helicopter Incorporating the Model 570B Three-Axis Stability and Control Augmentation System*, February 1973.

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

5. Final Report, United States Army Aviation Systems Test Activity, Project No. 68-30, *Airworthiness and Flight Characteristics Evaluation, Engineering Flight Test of a Production OH-58A Helicopter Unarmed and Armed with XM27E1 Weapon Subsystem, Performance*, September 1970.

6. Final Report, United States Army Aviation Systems Test Activity, Project No. 72-01, *Instrument Flight Evaluation, OH-58A Helicopter*, September 1972.

7. Message, AVSCOM, AMSAV-EQI, 3 June 1975, subject: Safety of Flight Release, Project No. 75-11.

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

9. Engineering Design Handbook, Army Materiel Command Pamphlet No. 706-204, *Helicopter Performance Testing*, August 1974.

10. Letter, AVSCOM, AMCPM-ASE-TM, 3 June 1975, subject: IR Low Reflective Paint for OH-58 Helicopter.

11. Model Specification, Allison Division of General Motors Corporation, No. 803, "Allison Model T63-A-700," 19 July 1968.

12. Detail Specification, Bell Helicopter Company, No. 206-947-031, "Model 206A (Mod)," 16 October 1967.

13. Flight Test Manual, USAF Aerospace Research Pilot School, FTC-TIH-68-1002, *Stability and Control*, September 1968.

APPENDIX B. PHOTOGRAPHS

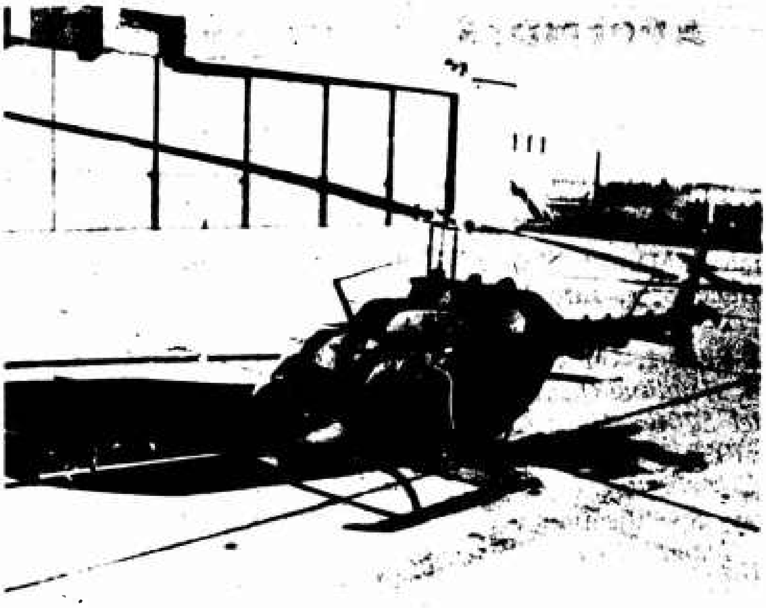


Photo A. Aircraft in Configuration A.

Photo B. Aircraft with IRCM Exhaust Stacks Installed.

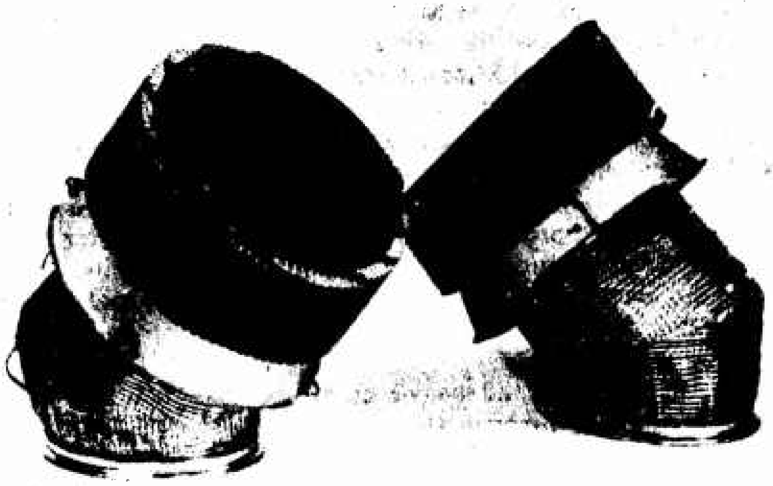


Photo C. Hughes IRCM Exhaust Stacks.

APPENDIX C. DESCRIPTION

INFRARED COUNTERMEASURE EXHAUST STACKS

1. The prototype exhaust stacks are interchangeable with the standard exhaust stacks (P/N 206-061-300) on the aircraft. No modification to the airframe is required to receive the IRCM exhaust stacks.

2. The stacks are constructed in two sections: the nozzle portion is constructed of stainless steel and insulated with 0.25 inch of "Low Q" insulation, and the shielding section is made of aluminum with crimped fins brazed to the exterior. The shielding section is approximately 2 inches larger in diameter than the nozzle section and is mounted on the nozzle by four heat-resistant bolts. The two sections are concentrically located relative to each other and the nozzle protrudes about 0.75 inch into the shield.

3. This configuration uses the nozzle as an exhaust pipe and as an aspirating pump to draw cooling air through the annular opening between the two sections. The cooling air acts as a boundary layer, helping to protect the shield section from the hot exhaust gases. The shield is also cooled by the flow of ambient air over the cooling fins. The nozzle has a 40-degree bend as viewed from the front of the aircraft, allowing a transition from the axis of the engine outlet to a vertical axis for countermeasure purposes.

4. The system was designed to the following parameters:

a. Weight: 3.85 pounds per duct, which is an increase of 2.58 pounds over the standard exhaust stack.

b. Temperature: The average temperatures of exposed metal surfaces are not to exceed 300°F at maximum power, 95°F ambient, hovering out of ground effect.

c. Structure: The system is a highly damped structure which will not be subject to thermal shock under any mode of aircraft operation.

d. Engine compartment temperatures: Installation of the improved exhaust stacks shall not increase engine compartment temperatures from that of the standard configuration.

LOW REFLECTIVE PAINT

5. The paint supplied for these tests is a low reflective olive-drab acrylic lacquer (FSN 8010-083-6588) specified by MIL-L-46159. It is specially formulated to reduce IR solar reflections in the spectral band pass of all currently identified IR-seeking missiles. It is also designed to have a low visual gloss to aid in visual contrast reduction. This paint is a substitute for standard Army olive-drab paint and can be applied directly over standard paint. The LR paint added 19 pounds to the gross weight of the aircraft.

APPENDIX D. GENERAL TEST CONDITIONS

Configuration	Test	Average Gross Weight (lb)	Average Density Altitude (ft)	Average Outside Air Temperature (°C)	Average True Airspeed (kt)
A	Hover performance	2750 to 3250	4040	24.5	Zero
C		2315 to 3500	Sea level to 4150	15.5 to 27.0	
D		2450 to 3010	340	15.5	
E		2530 to 3040	2860	16.0	
F		2450 to 3110	2820	15.5	
G		2690 to 3030	2820	14.5	
A	Level flight performance	2820	7880	17.0	35 to 113
B		2820	7240	24.0	35 to 105
C		2780	2880	18.0	35 to 102
E		2460	4420	19.5	34 to 115
F		2650 to 2880	3320 to 10,940	10.5 to 24.0	23 to 115
A	Control positions in trimmed forward flight	2820	7880	17.0	35 to 113
B		2820	7240	24.0	35 to 105
C		2460	4420	19.5	23 to 115
A	Static longitudinal stability	2860 to 2940	6600 to 7520	23.0	86 and 52
B		2900 to 2940	5360 to 8260	18.0 to 23.0	52 and 92
C		2920	5940 and 6540	15.0 and 26.0	53 and 82
A	Static lateral-directional stability	2780 and 2840	8400 and 9060	18.5 and 22.0	40 and 85
B		2780	2060	21.5	86
C		2870 and 2860	7740 and 6660	15.0	42 and 80
A	Maneuvering stability	2780	8960	18.5	80
B		2760	8120	22.0	90
C		2760	8220	14.5	90
A	Longitudinal dynamic response	2800 and 2780	7680 and 7480	24.0 and 23.5	52 and 88
C		2920, 2720, and 2780	5860, 7780, 7200	17.0, 14.0, and 14.0	41, 42, and 88
A	Release from steady-heading sideslip	2720 to 2740	8340 to 8400	42.5	42 to 87
B		2760	6580	22.0	83
C		2800	6260 to 8340	14.0 to 16.0	42 to 91
A	Simulated sudden engine failure	2720	7660 to 8300	21.0	52 and 91
C		2860	6000	16.0	51 and 96

APPENDIX E. DATA ANALYSIS METHODS

1. This appendix presents the data reduction and analysis methods used to evaluate the OH-58A performance capabilities. These equations correct test-day conditions to standard-day conditions and also provide the necessary tools with which to predict helicopter performance for various atmospheric conditions. The topics discussed include level flight, hover, and specific range.

2. Level flight performance was defined by measuring the shp required to maintain level unaccelerated flight throughout the airspeed range of the helicopter. The results of each level flight test are presented as engine shp corrected to the average test conditions for each flight, tip Mach number, and specific range as a function of airspeed.

3. Helicopter performance test data were generalized through the use of nondimensional coefficients. The purpose is to accurately define performance at conditions not specifically tested. The following nondimensional coefficients were used to generalize the level flight and hover test results obtained during this flight test program.

a. Coefficient of power (C_P):

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

b. Coefficient of thrust (C_T):

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

c. Advance ratio (μ)

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

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

$$M_{tip} = \frac{1.6889\ V_T + \Omega R}{a} \qquad (4)$$

Where:

SHP = Engine output shp

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

ρ = Air density (lb-sec^2/ft^4)

A = Main rotor disc area (ft^2)

Ω = Main rotor angular velocity (radian/sec)

R = Main rotor radius (ft)

W = Gross weight (lb)

1.6889 = Conversion factor (ft/sec/kt)

V_T = True airspeed (kt)

a = Speed of sound (ft/sec)

4. Engine output shp was determined from the engine torque pressure. Torque pressure as a function of the power output of the engine was obtained from ARADMAC engine acceptance test cell data. Horsepower transmitted by a rotating shaft was determined by the following equation:

$$SHP = \frac{2\pi \times Kt \times GR \times Nr \times Q}{33{,}000}$$

Where:

SHP = Shaft horsepower

Kt = Conversion factor to change measured engine torque pressure (psi) to ft-lb (from engine acceptance data)

GR = Gear ratio of the output shaft rotational speed to the main rotor rotational speed

Nr = Main rotor speed (rpm)

Q = Engine torque pressure (psi)

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

5. 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_s are identical to C_{P_t}, C_{T_t}, and μ_t, respectively. From the definition of equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t}$$

Where:

SHP = Engine output shp

ρ = Air density (slug/ft^3)

t = Test day

s = Standard day

6. Specific range was calculated using the level flight performance curve and the specification installed engine fuel-flow characteristics:

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

Where:

$NAMPP$ = Nautical air miles per pound of fuel (naut mi/lb)

V_T = True airspeed (kt)

W_f = Fuel flow rate (lb/hr)

7. Changes in the equivalent flat plate area (f_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)

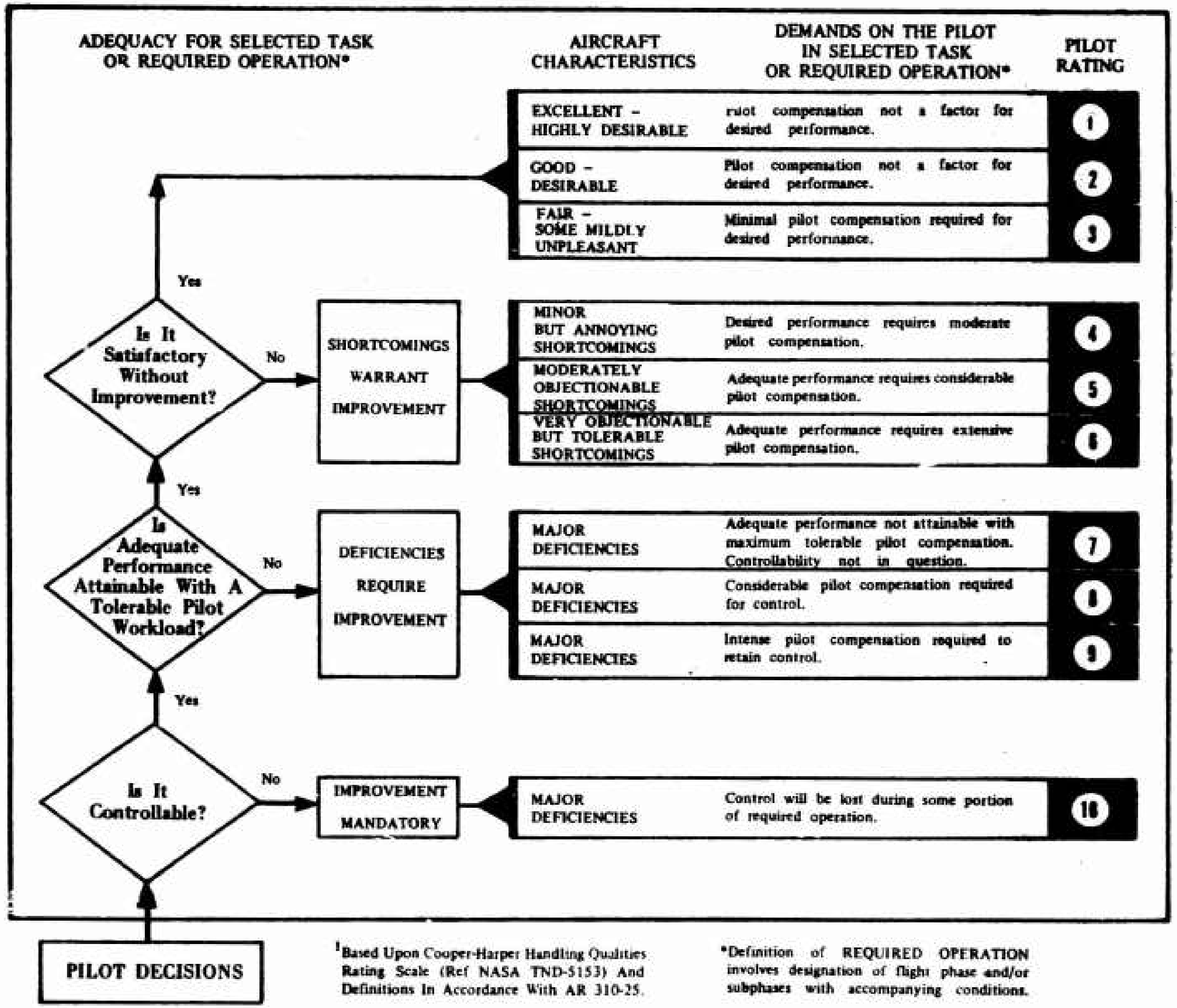
ΔC_P = Change in coefficient of power

A = Main rotor disc area (ft^2)

μ = Advance ratio

8. Hover performance was determined IGE and OGE by the free flight hover technique. Equations 1 and 2 were used to define the hover capability. A plot of C_P versus C_T was constructed for each skid height tested. Hover performance characteristics may be extracted from these curves in preparing labels or curves for flight manuals for any combination of conditions.

9. The HQRS presented as figure 1 was used to augment pilot comments relative to handling qualities.



[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.

APPENDIX F. INSTRUMENTATION

Instrumentation was installed in the test helicopter by USAAEFA personnel prior to the start of the test program. One magnetic tape recorder was installed in the aft cabin compartment for the majority of all testing. The recorder was removed for the final 6 hours of testing so that light gross weights could be attained. All instrumentation was calibrated and maintained by USAAEFA personnel. The following parameters were recorded:

<u>Pilot/Engineer Panel</u>

Airspeed (boom)
Altitude (boom)
Angle of sideslip
Rotor speed
Center-of-gravity normal acceleration
Free air temperature
Total fuel used
Turbine outlet temperature
Engine torque
Gas producer speed
Fuel flow rate
Fuel totalizer
Engineer event marker
Record counter
Engine compartment temperatures
Engine fuel nozzle pressure

<u>Magnetic Tape Recorder</u>

Control position and force:
- Longitudinal
- Lateral
- Directional
- Collective

Attitude and rate:
- Pitch
- Roll
- Yaw

Airspeed (boom)
Altitude (boom)
Altitude (radar)
Free air temperature
Angle of sideslip
Center-of-gravity normal acceleration
Rotor speed
Throttle position
Engine torque
Gas producer speed

APPENDIX G. TEST DATA

INDEX

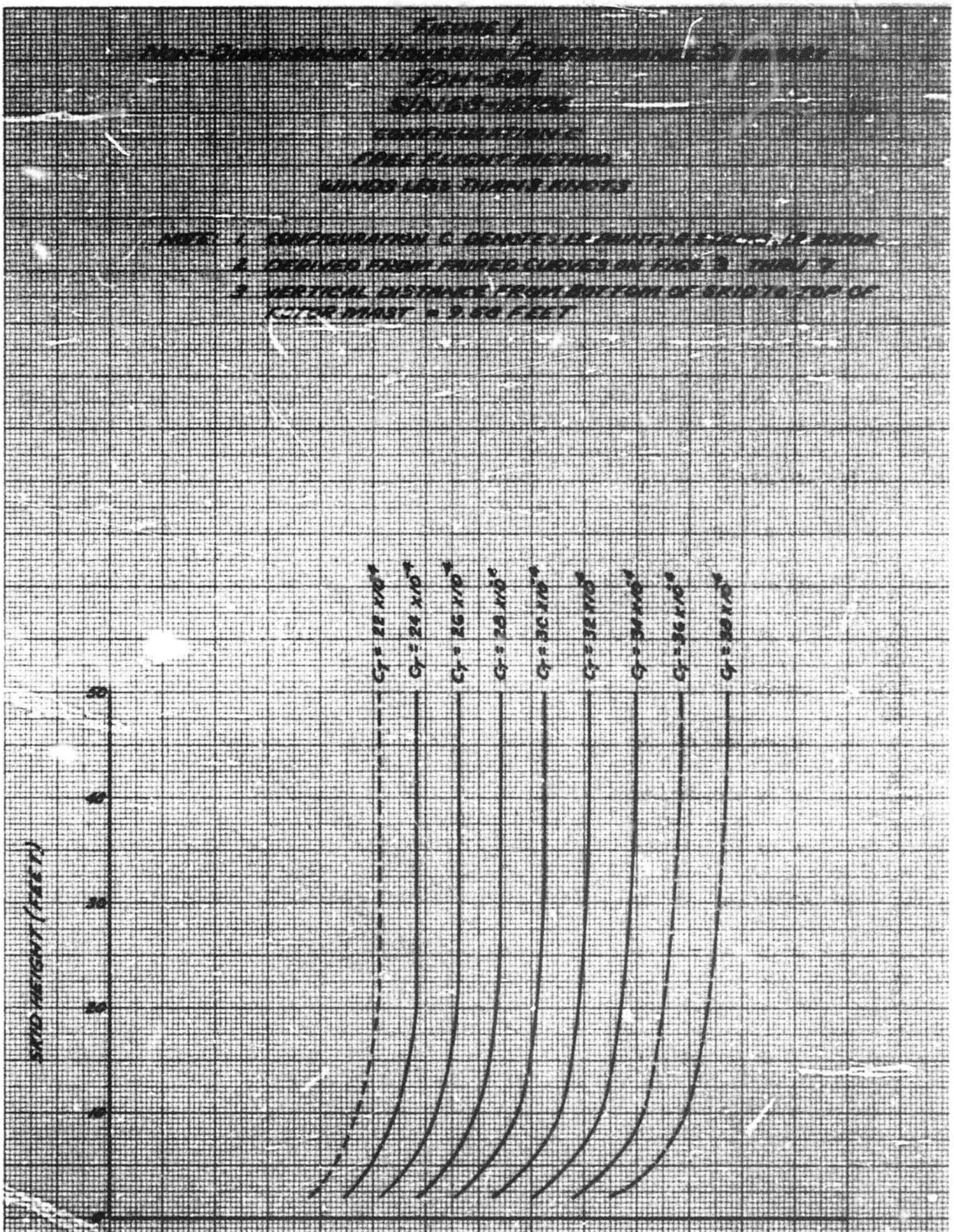
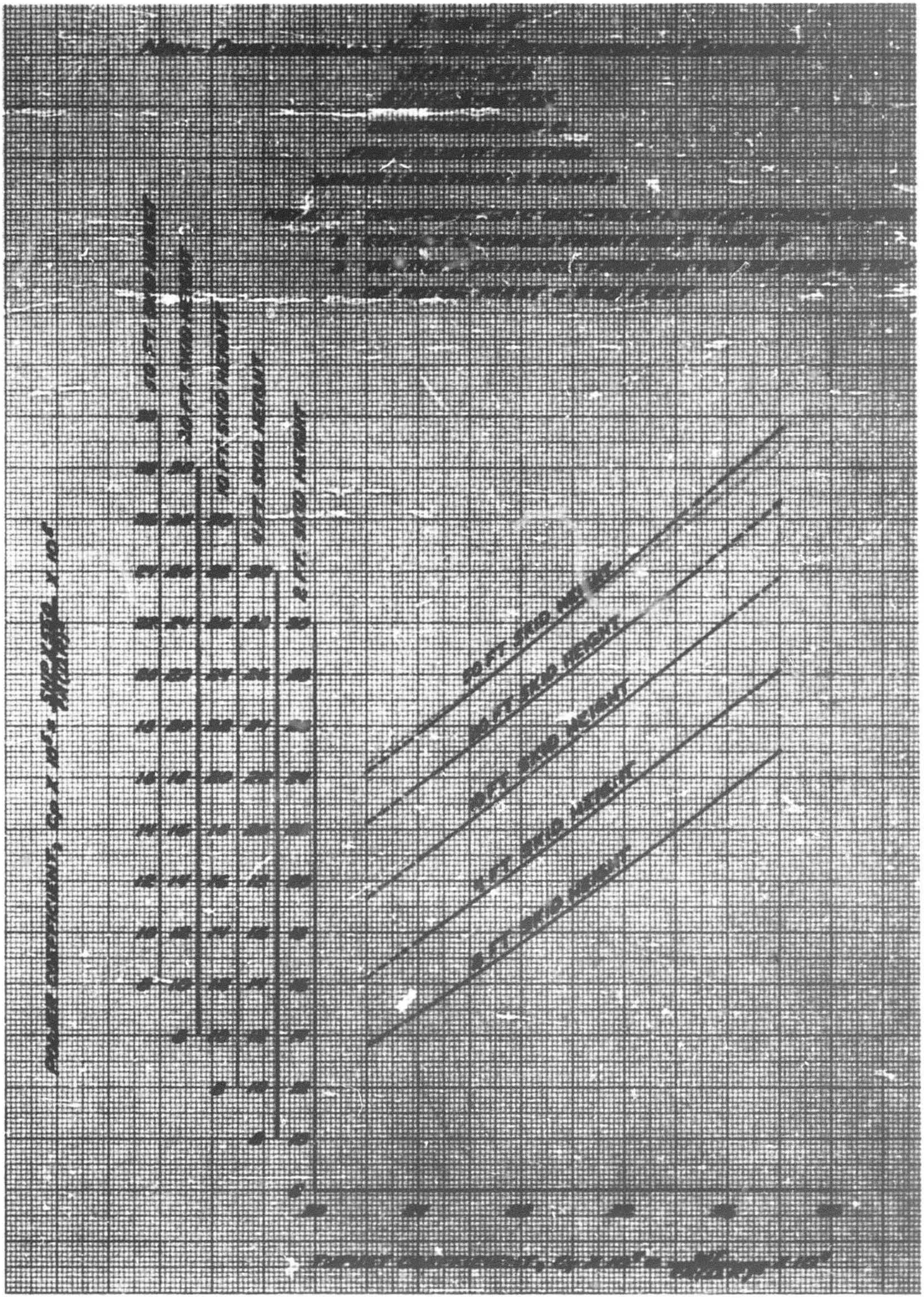
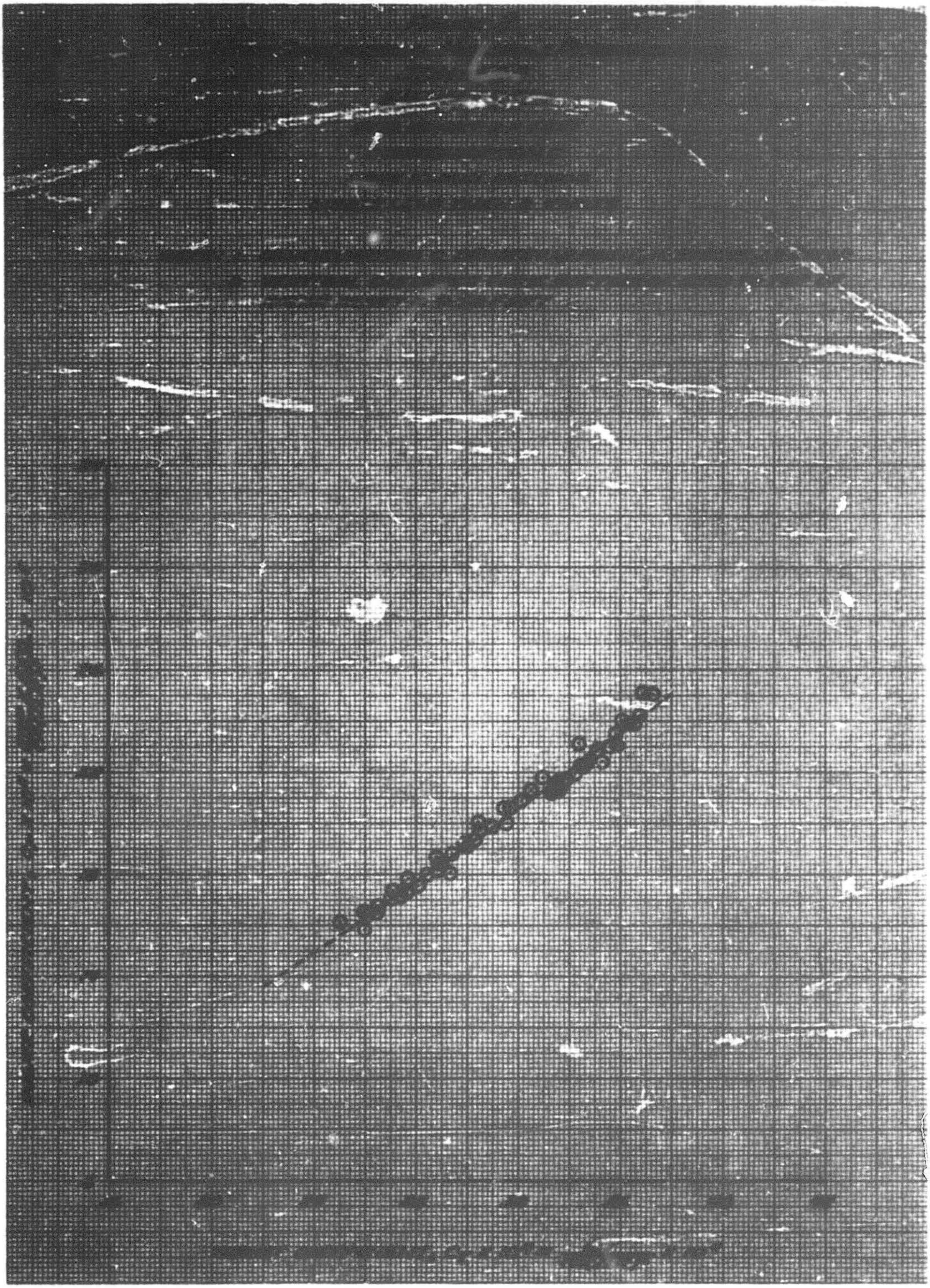
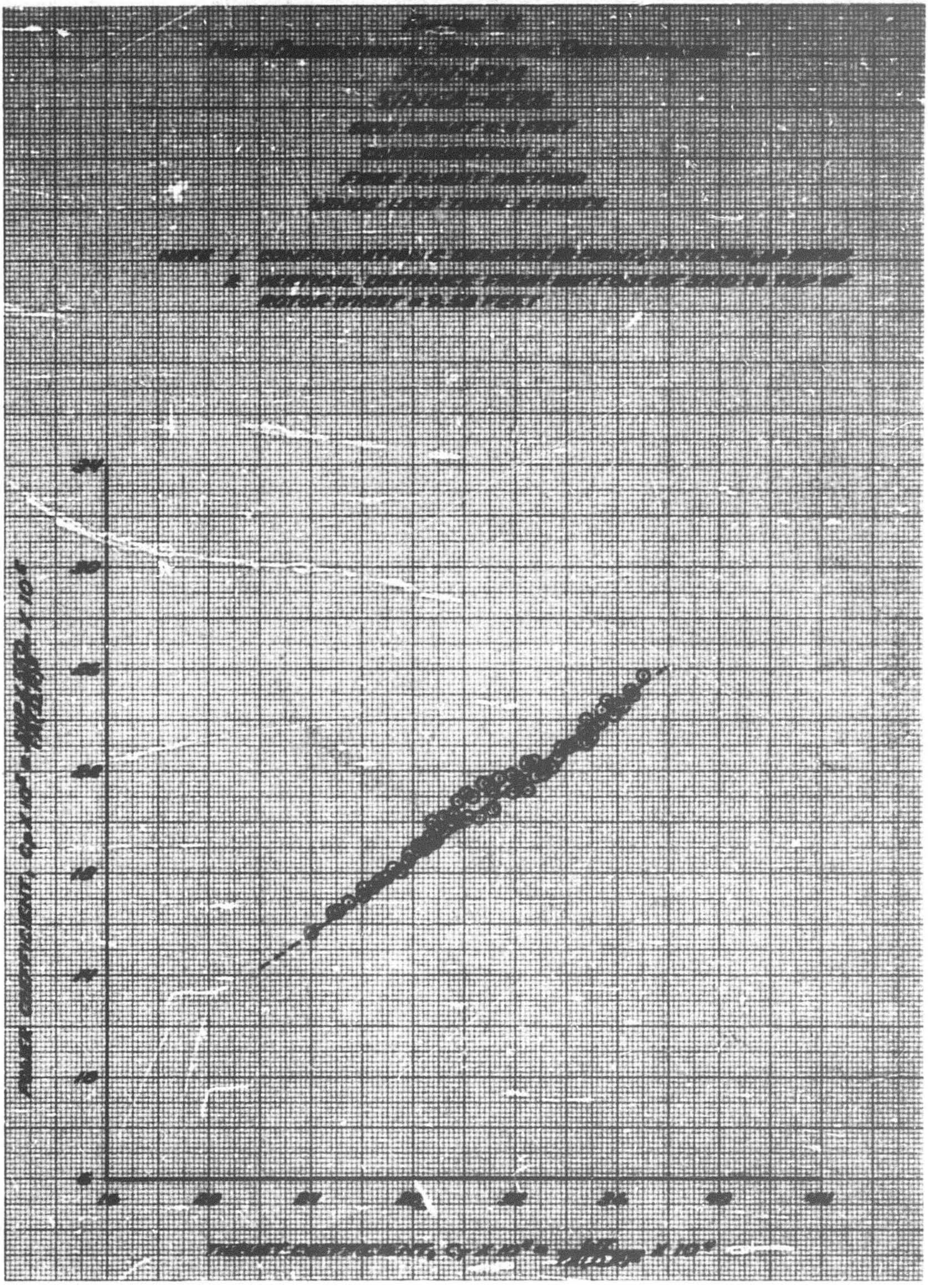
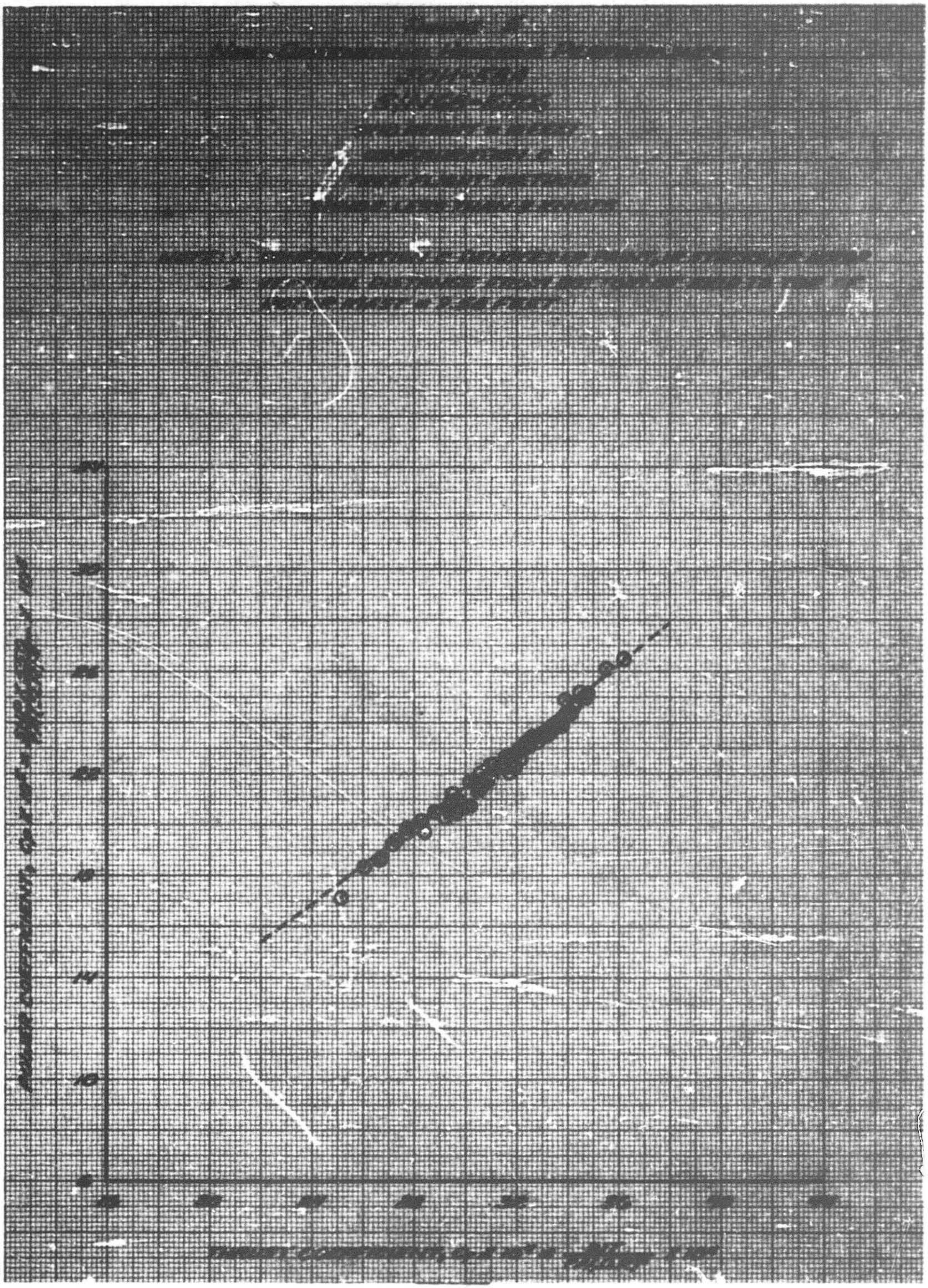


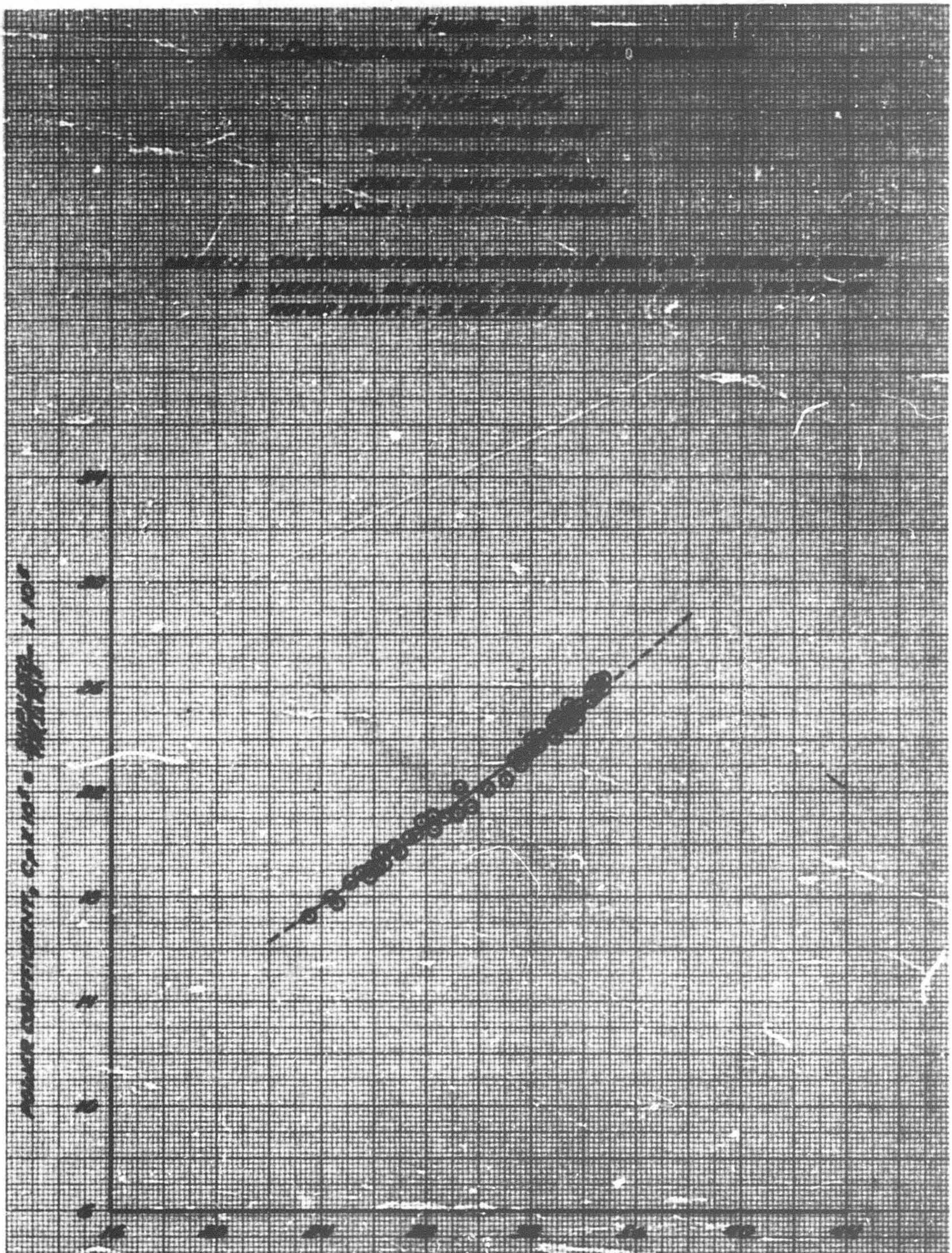
FIGURE 1
NON-DIMENSIONAL HOVERING PERFORMANCE SUMMARY
OH-58A
CONFIGURATION C
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS
NOTE: 1. CONFIGURATION C DENOTES
2. DERIVED FROM FAIRED CURVES ON FIGS 3 THRU 7
3. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.66 FEET
CT = 22 X 10-4
CT = 24 X 10-4
CT = 26 X 10-4
CT = 28 X 10-4
CT = 30 X 10-4
CT = 32 X 10-4
CT = 34 X 10-4
CT = 36 X 10-4
CT = 38 X 10-4
SKID HEIGHT (FEET)
0
10
20
30
40
50
10
14
18
22
26
30
34
POWER COEFFICIENT, CP X 10^5 = SHP X 550 / ρA(ΩR)^3 X 10^5

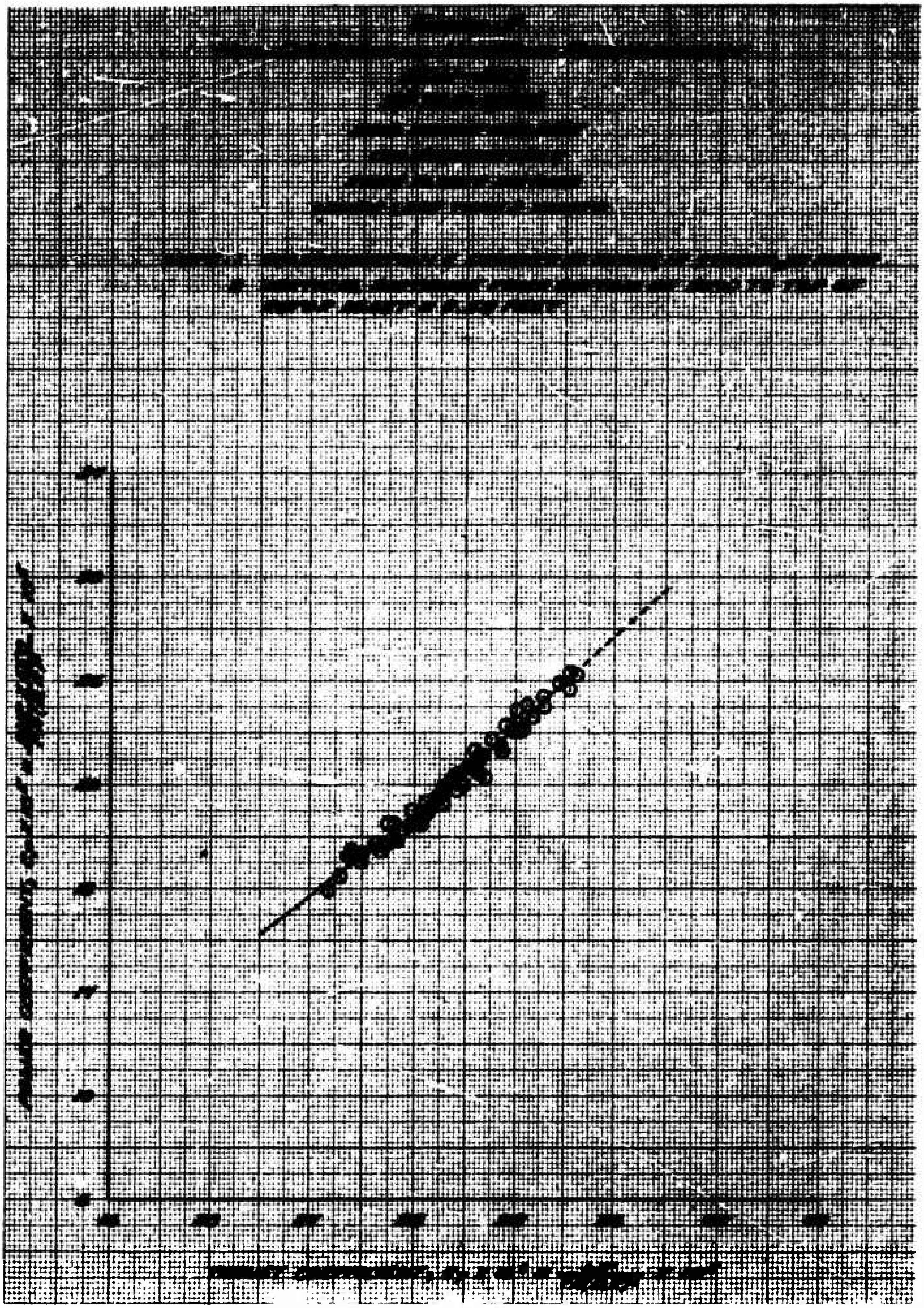


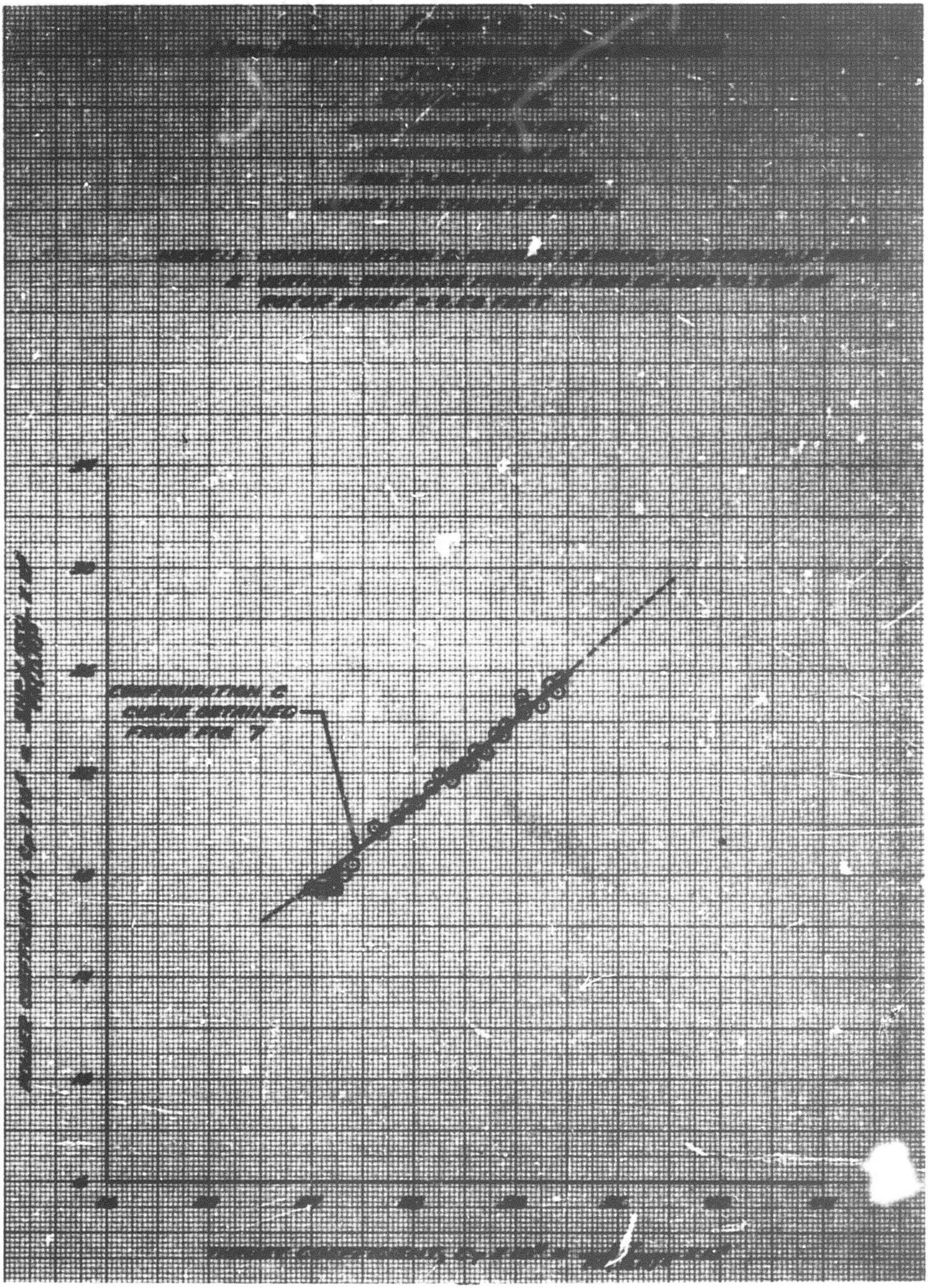


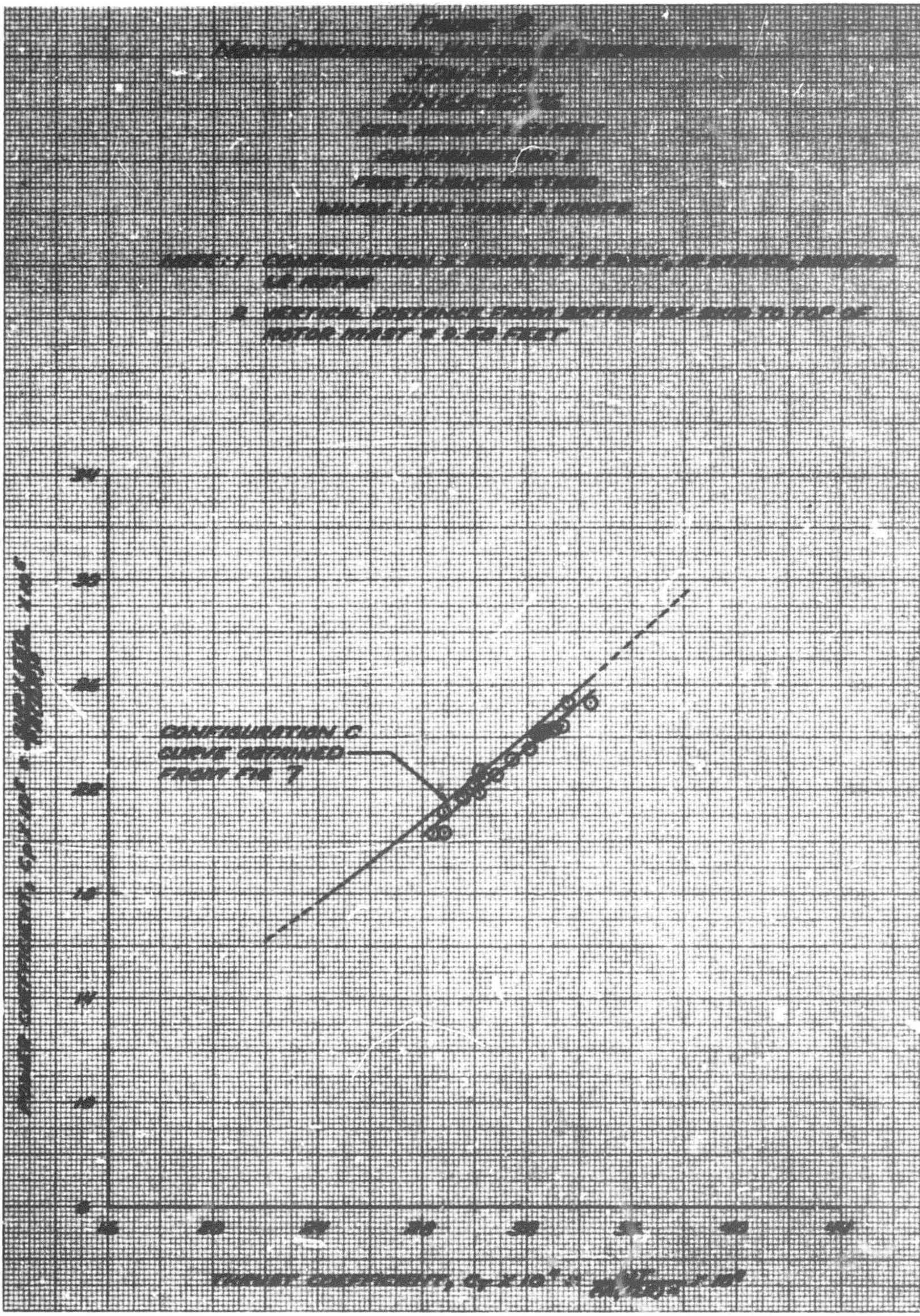


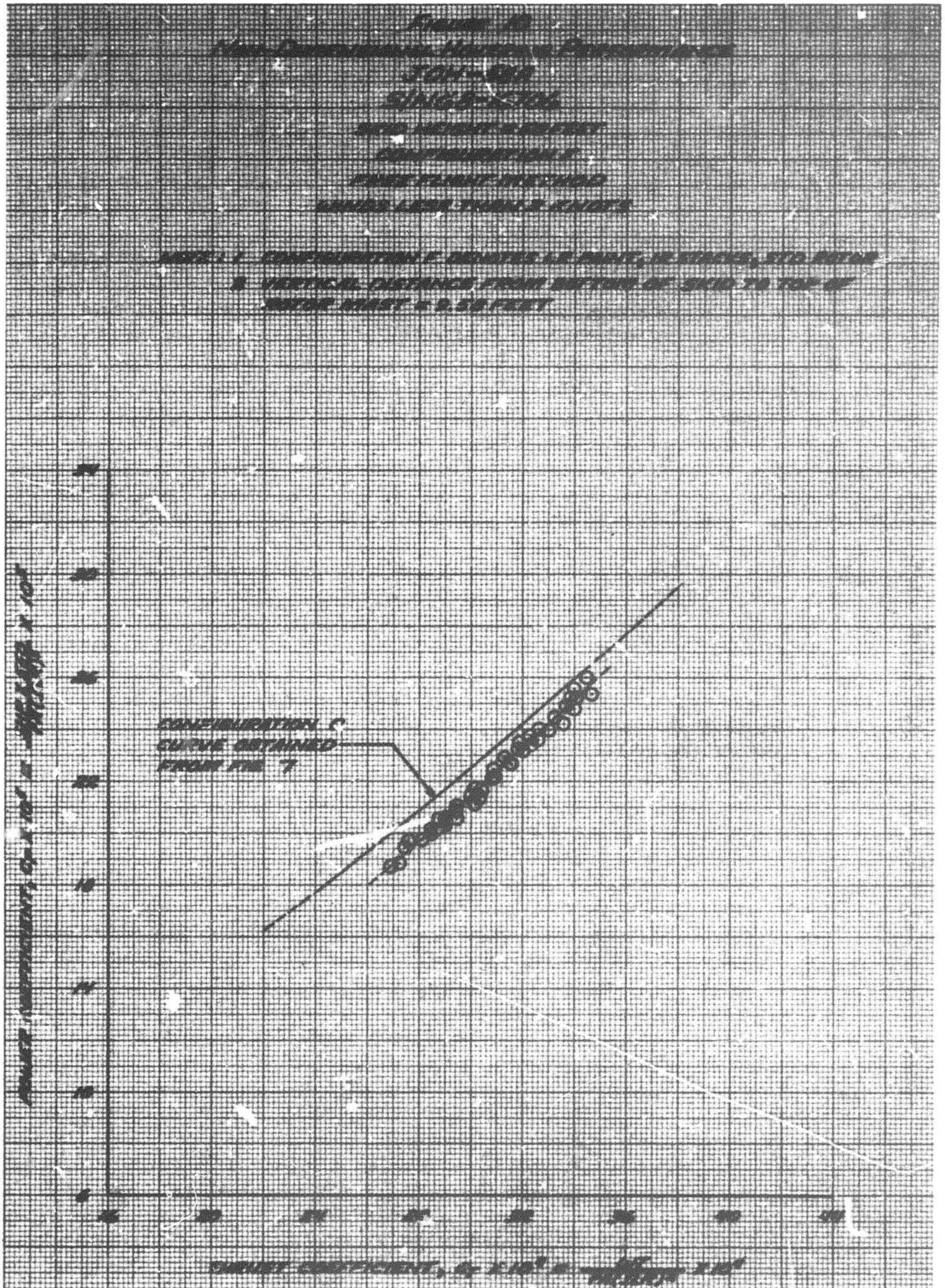


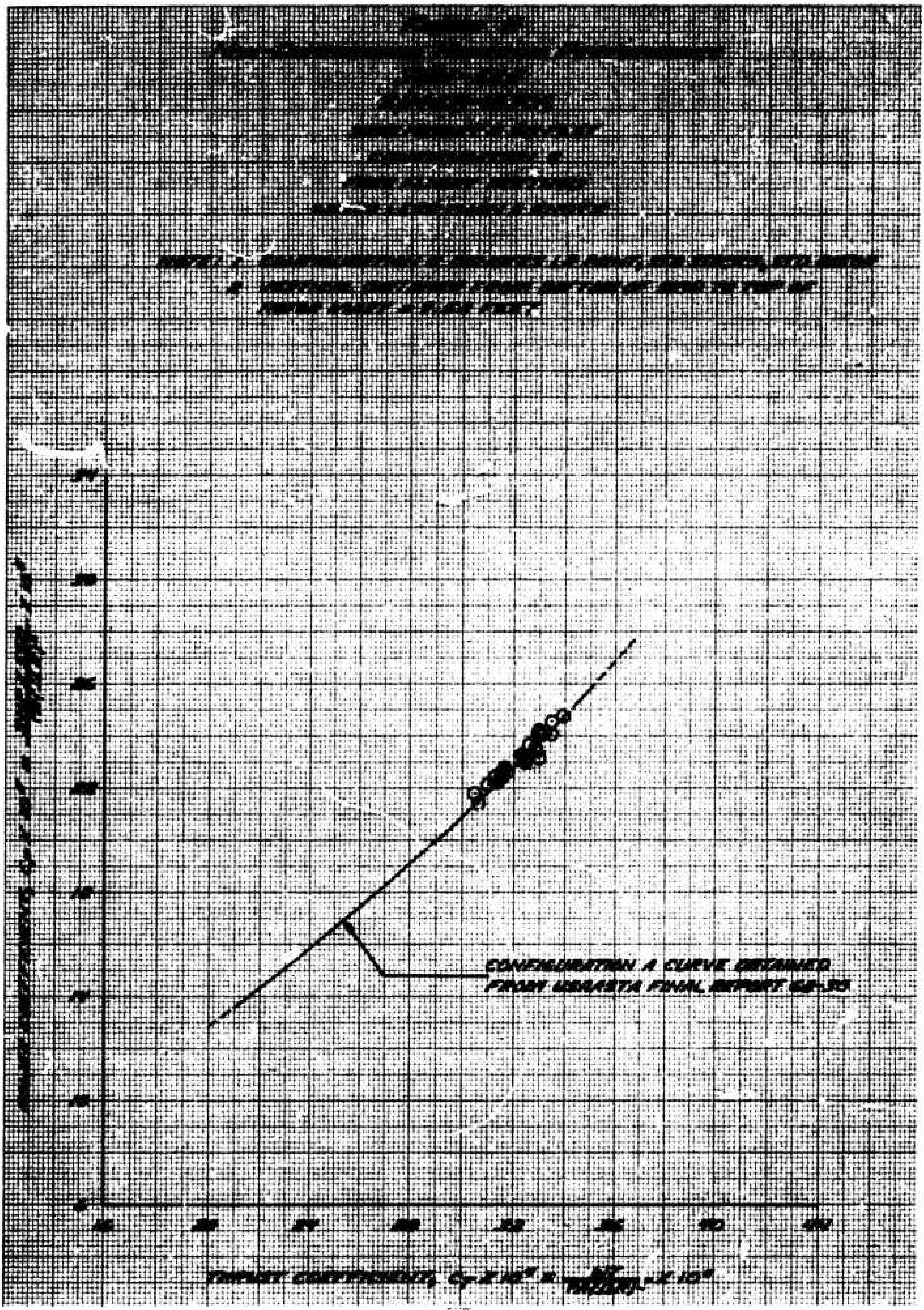





CONFIGURATION C
CURVE OBTAINED
FROM FIG 7


CONFIGURATION C
CURVE OBTAINED
FROM FIG 7


CONFIGURATION F
FREE FLIGHT METHOD
WINDS LESS THAN 3 KNOTS
2. VERTICAL DISTANCE FROM BOTTOM OF SKID TO TOP OF ROTOR MAST = 9.58 FEET
CONFIGURATION C
CURVE OBTAINED
FROM FIG. 7


CONFIGURATION A CURVE OBTAINED
FROM [illegible] FINAL REPORT [illegible]

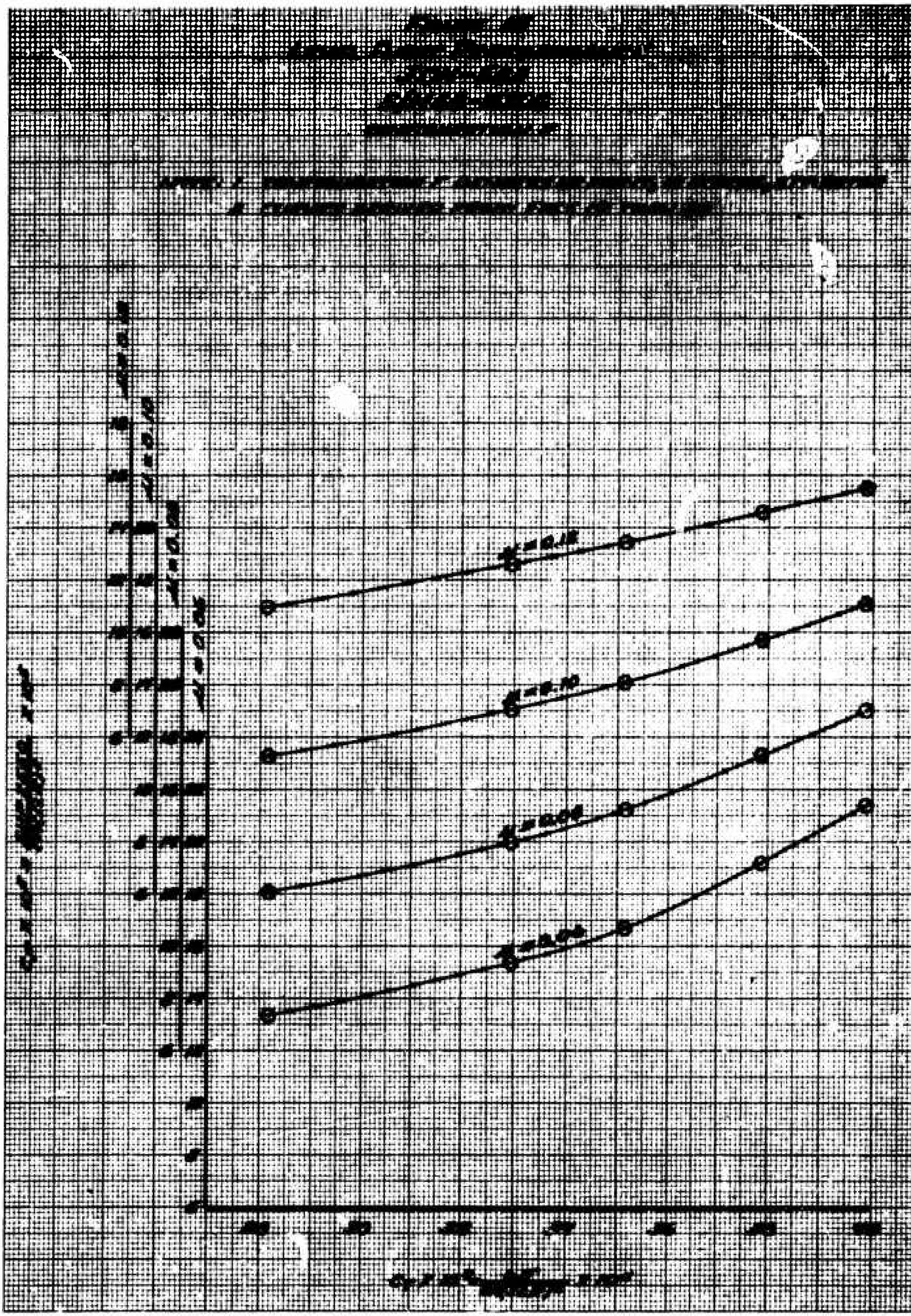
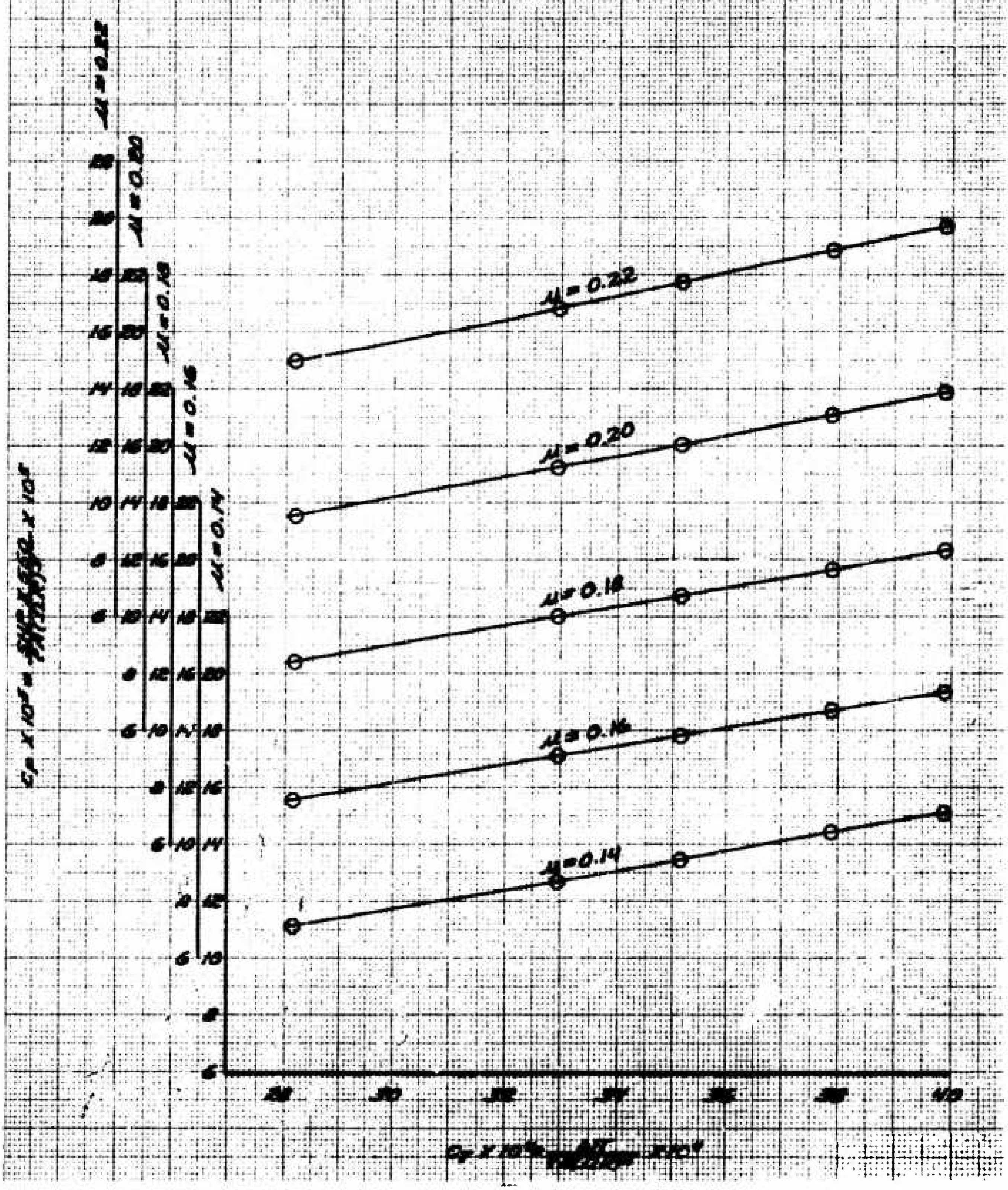
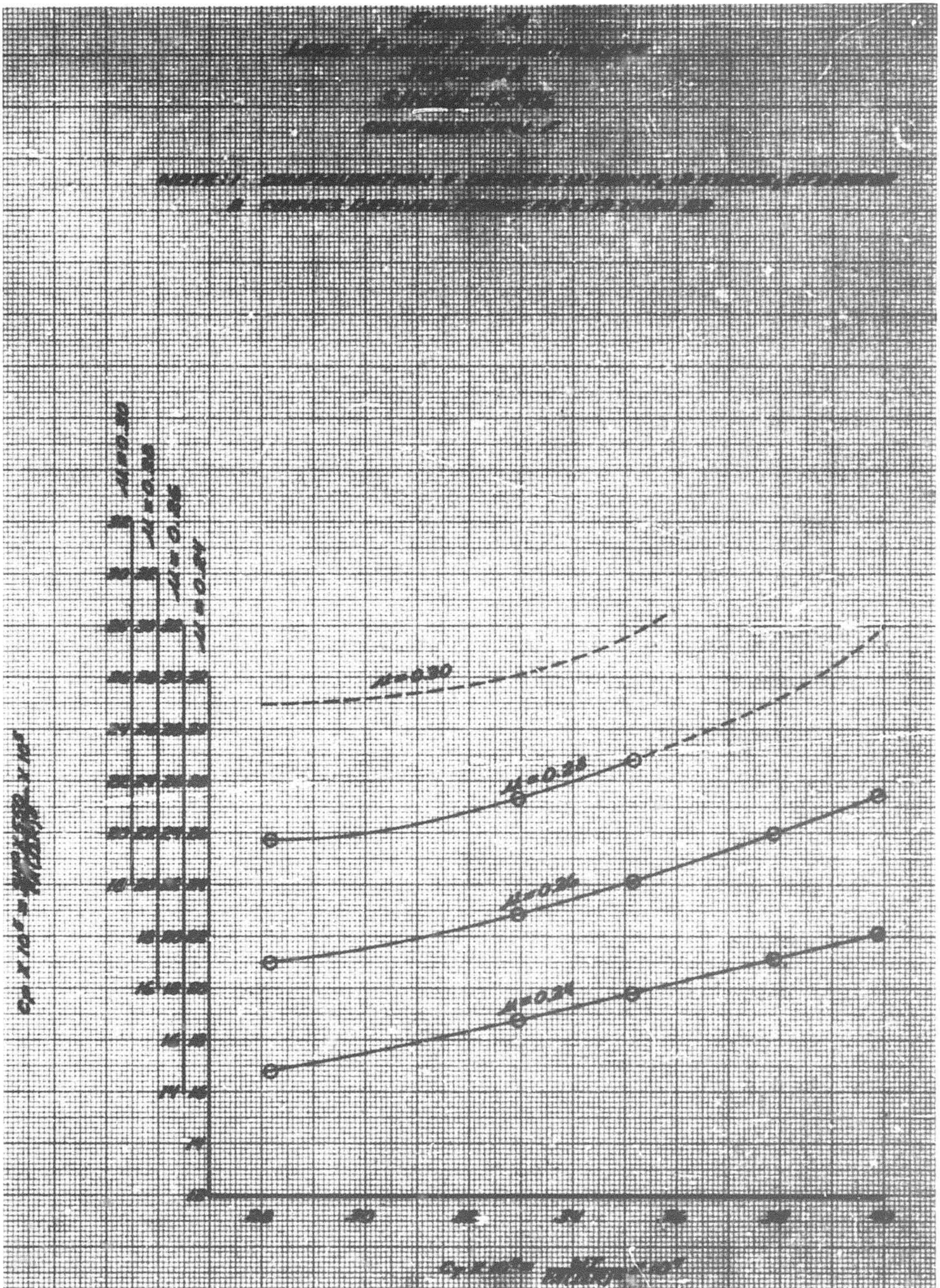


FIGURE 13
LEVEL FLIGHT PERFORMANCE
JOH-58A
CONFIGURATION F

NOTE: 1 CONFIGURATION F DENOTES IR PAINT, IR STACKS, STD ROTOR
2 CURVES DERIVED FROM FIGS

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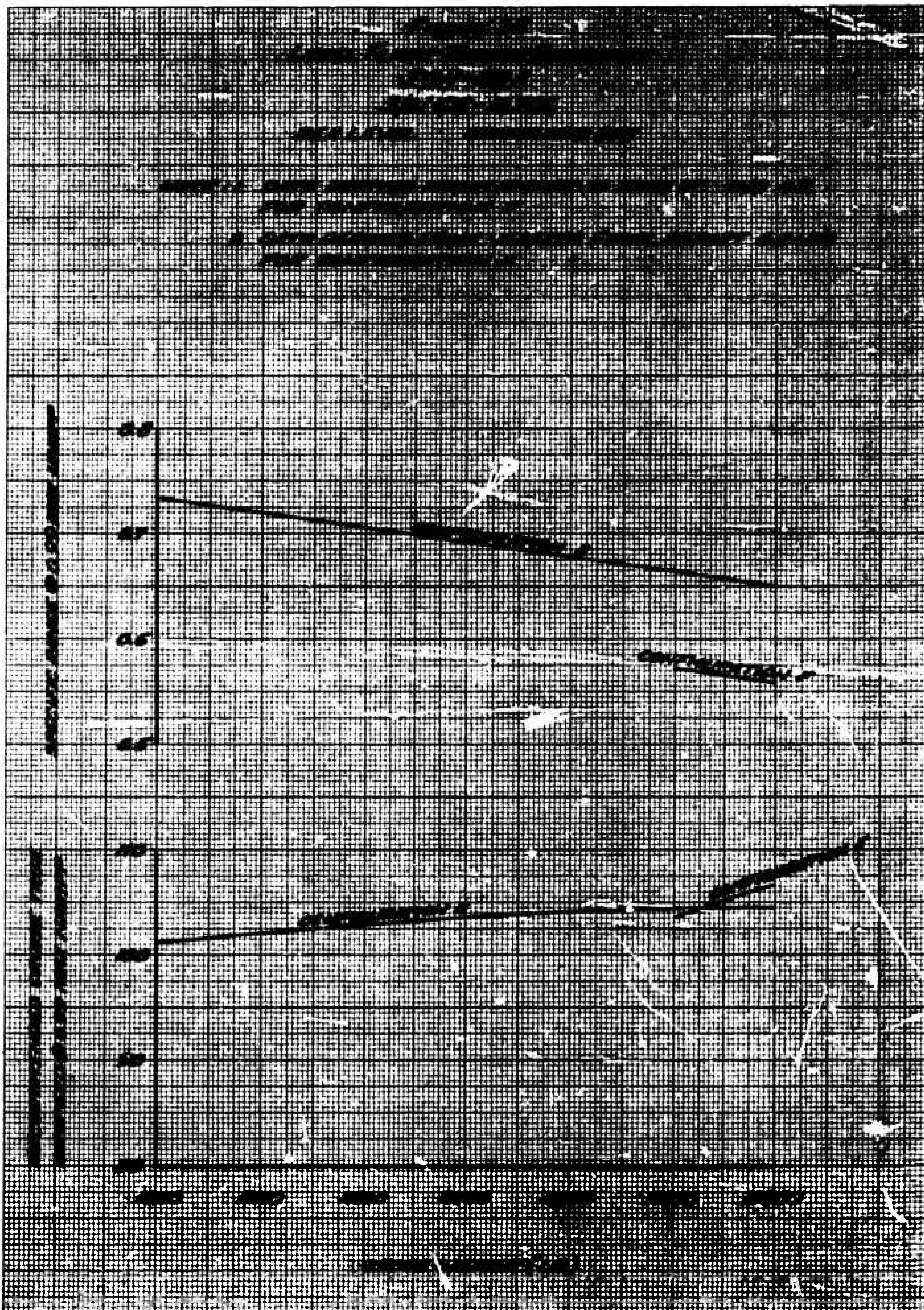
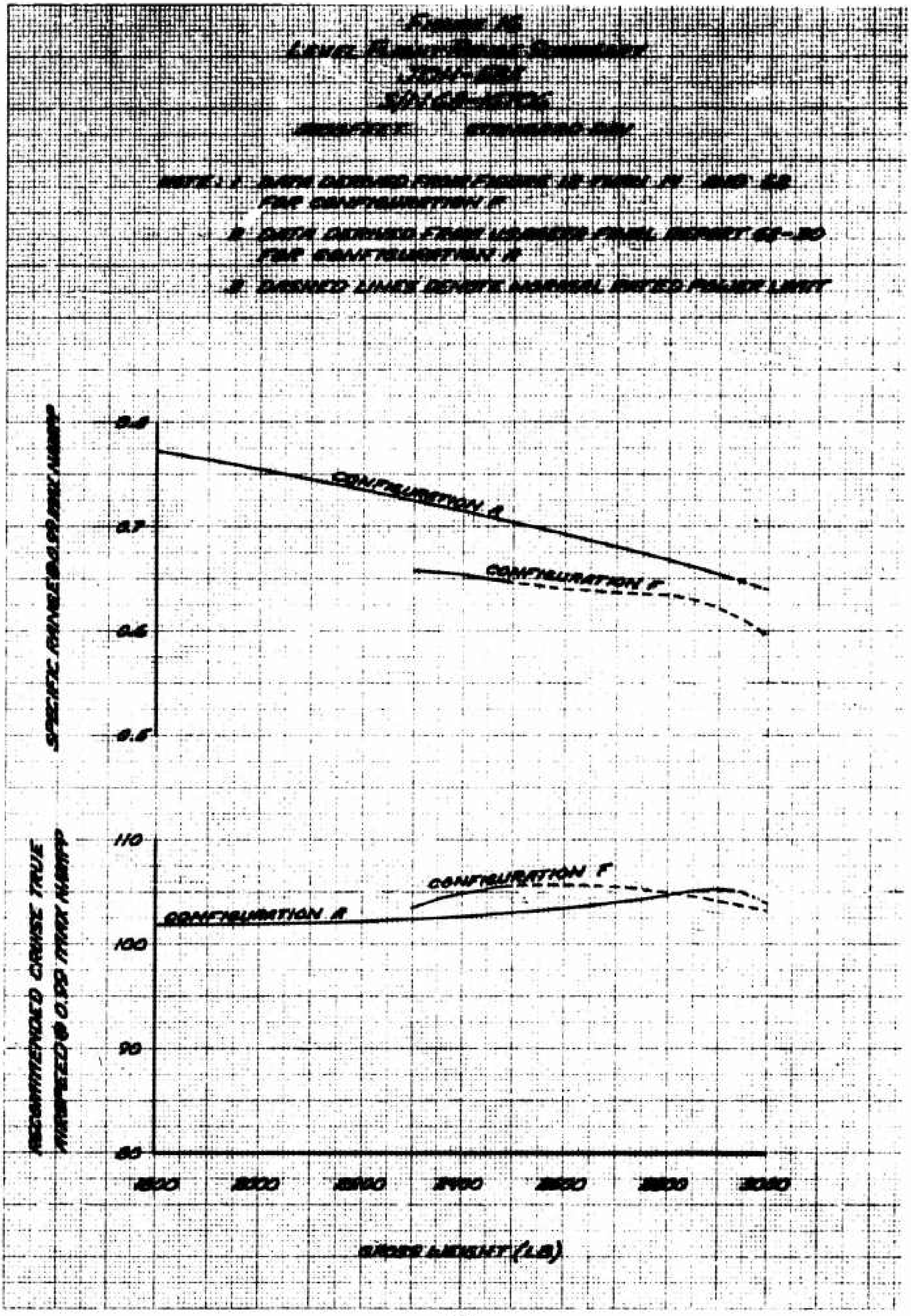
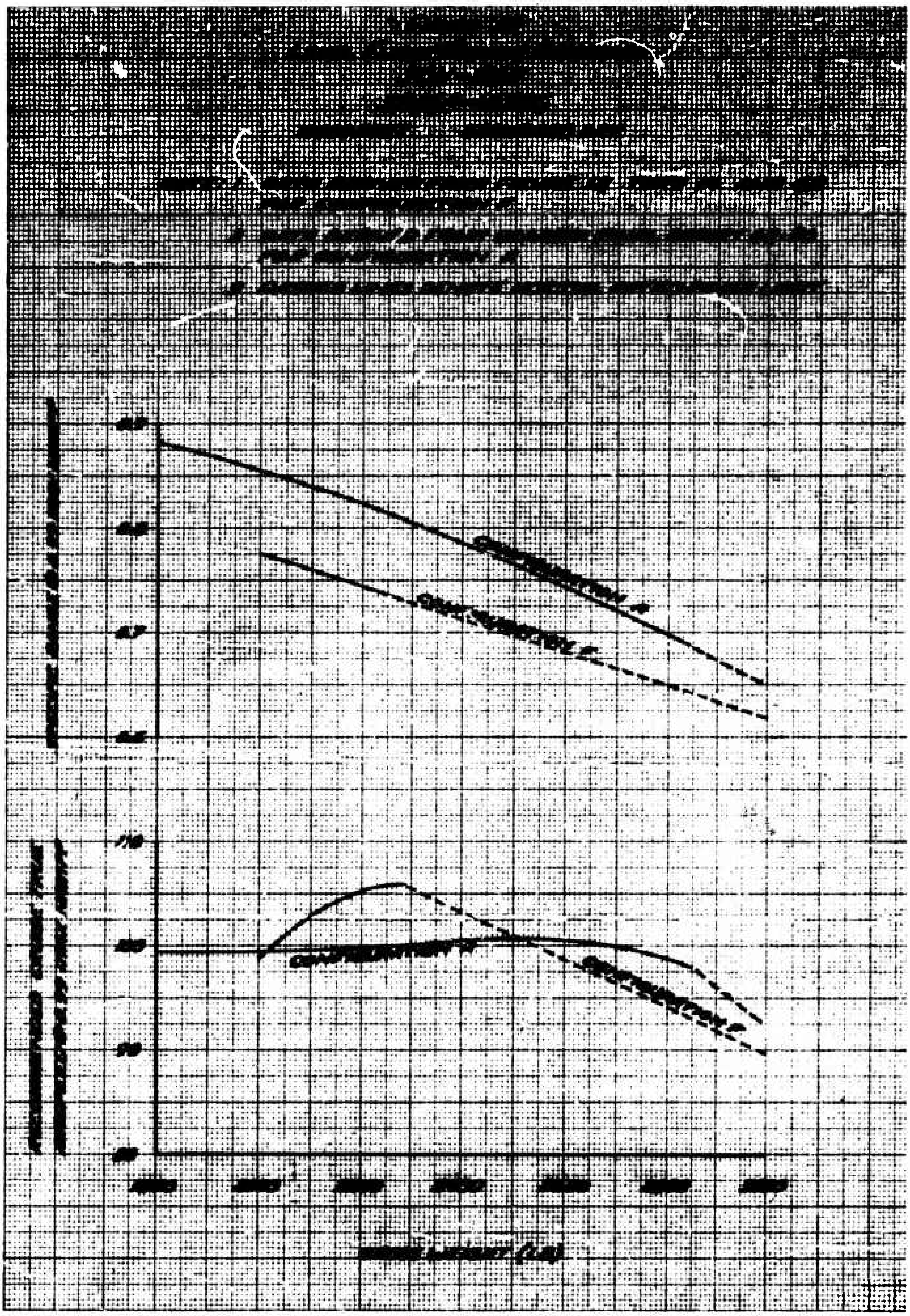
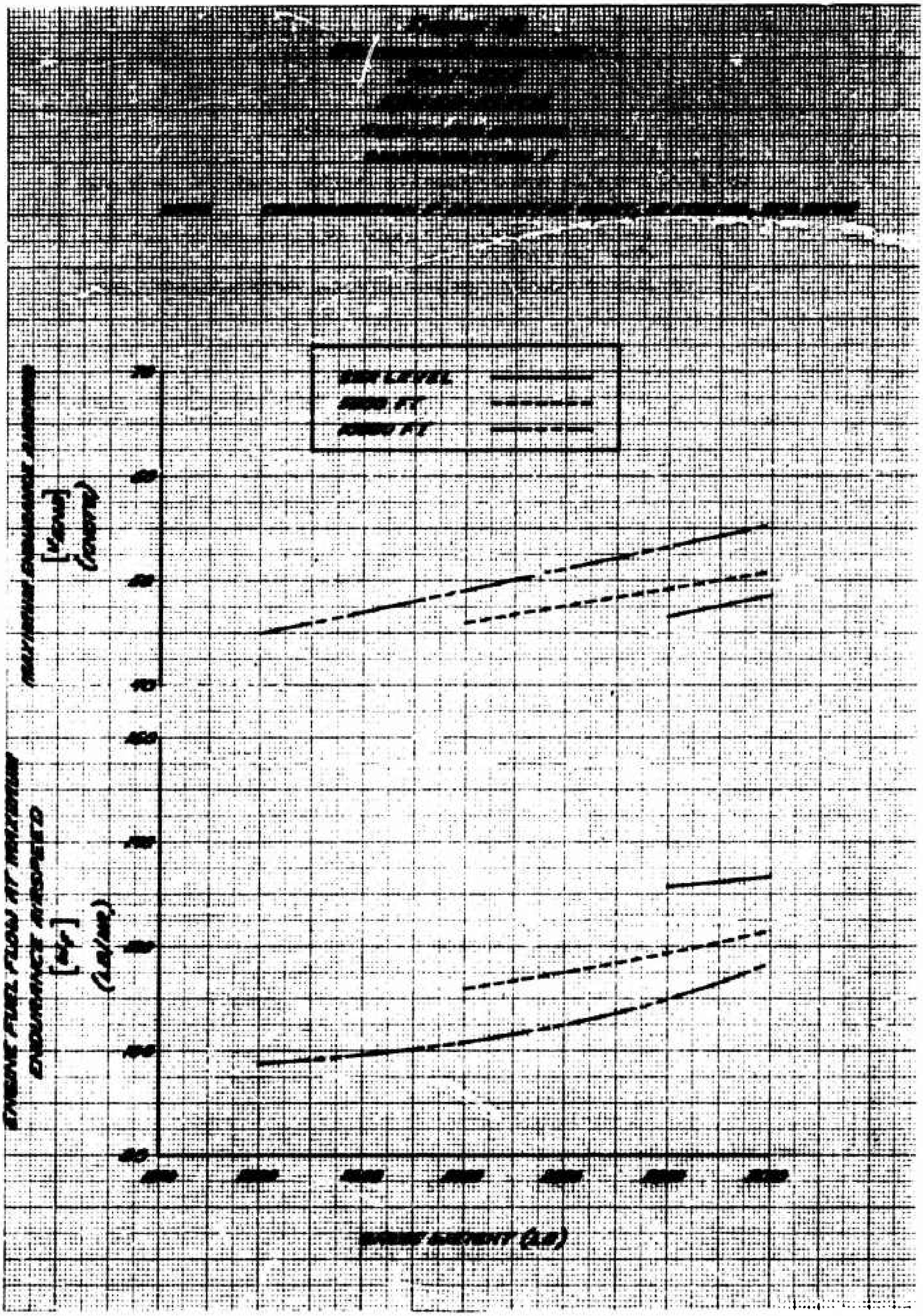
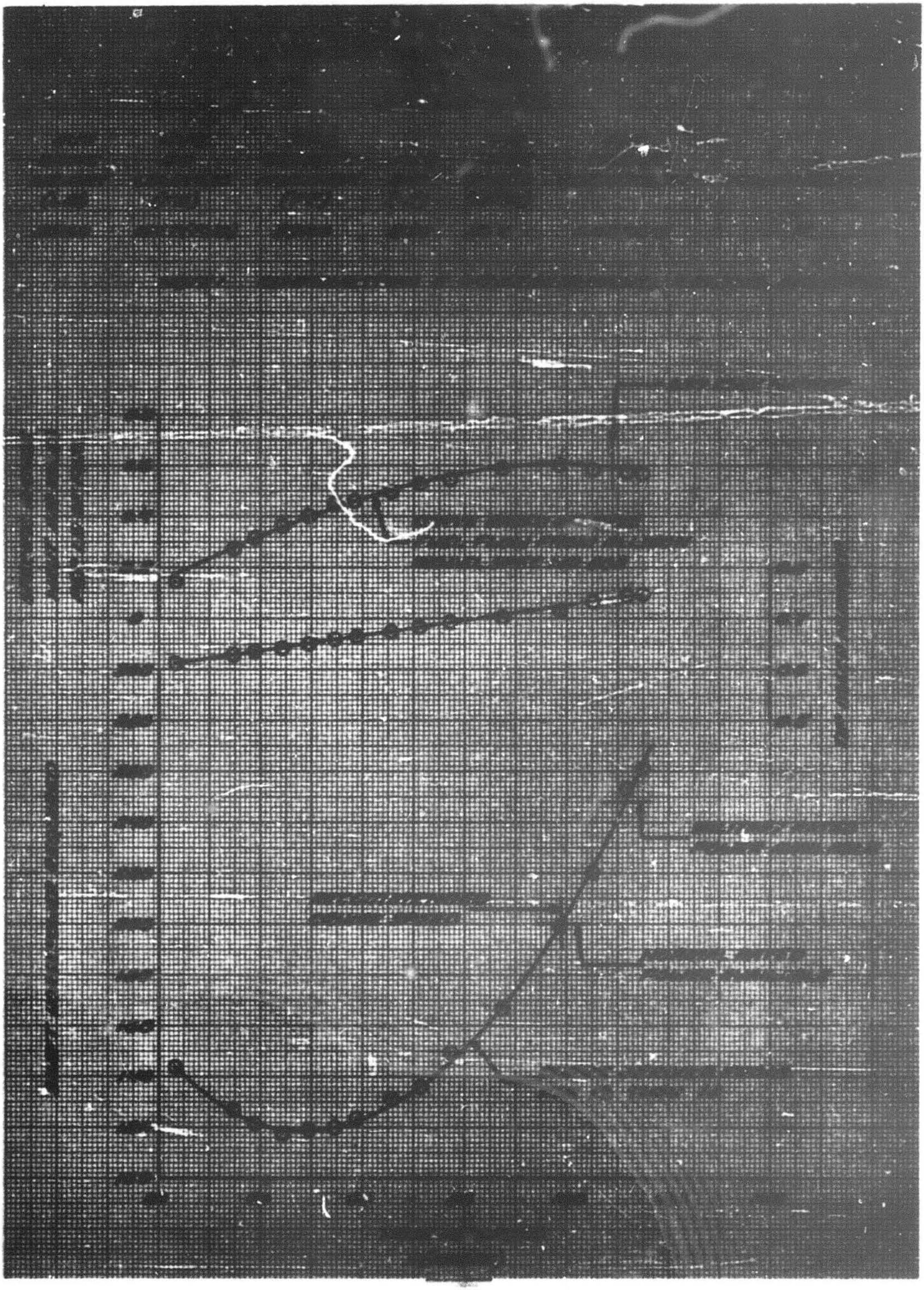
μ = 0.28
μ = 0.26
μ = 0.24

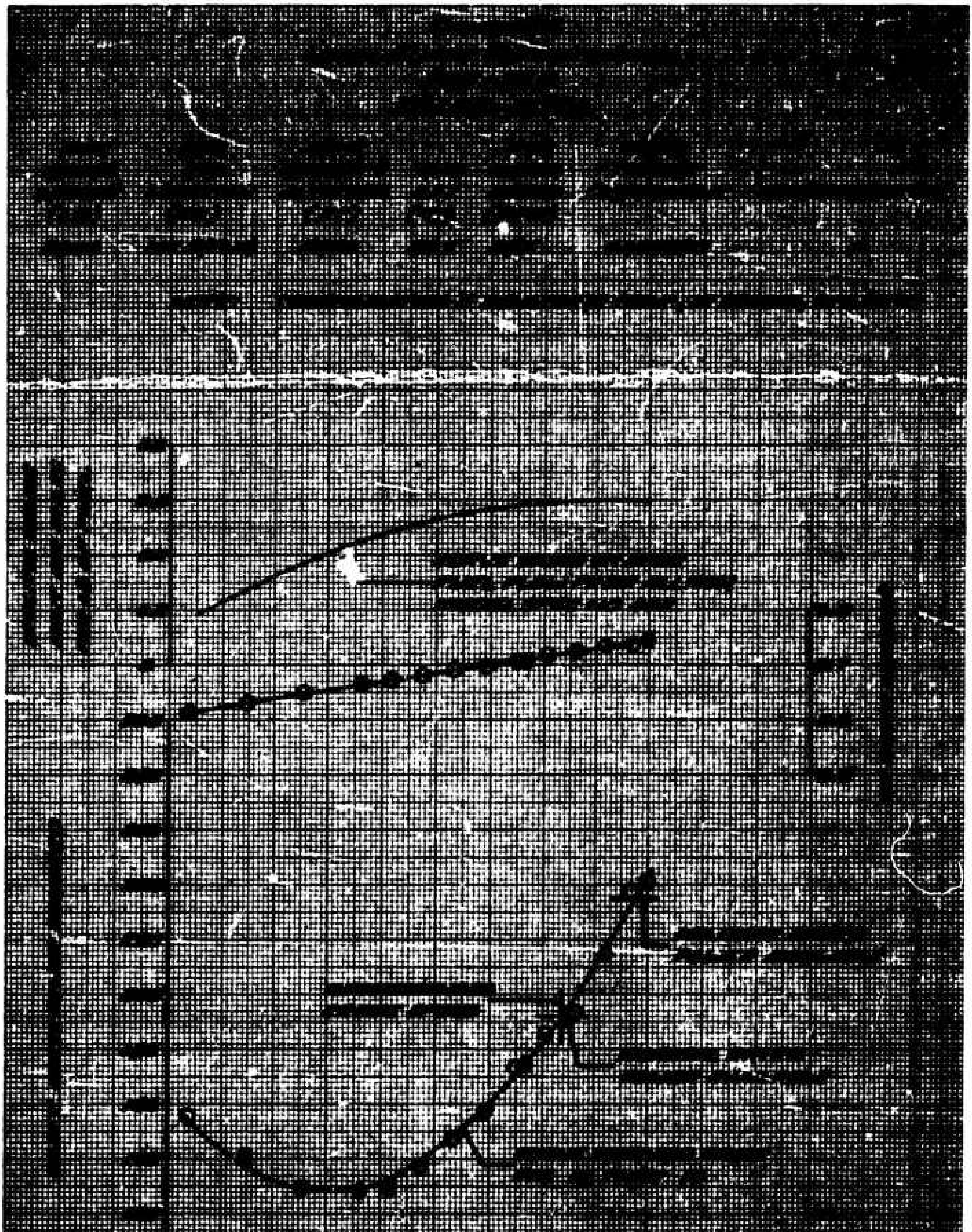


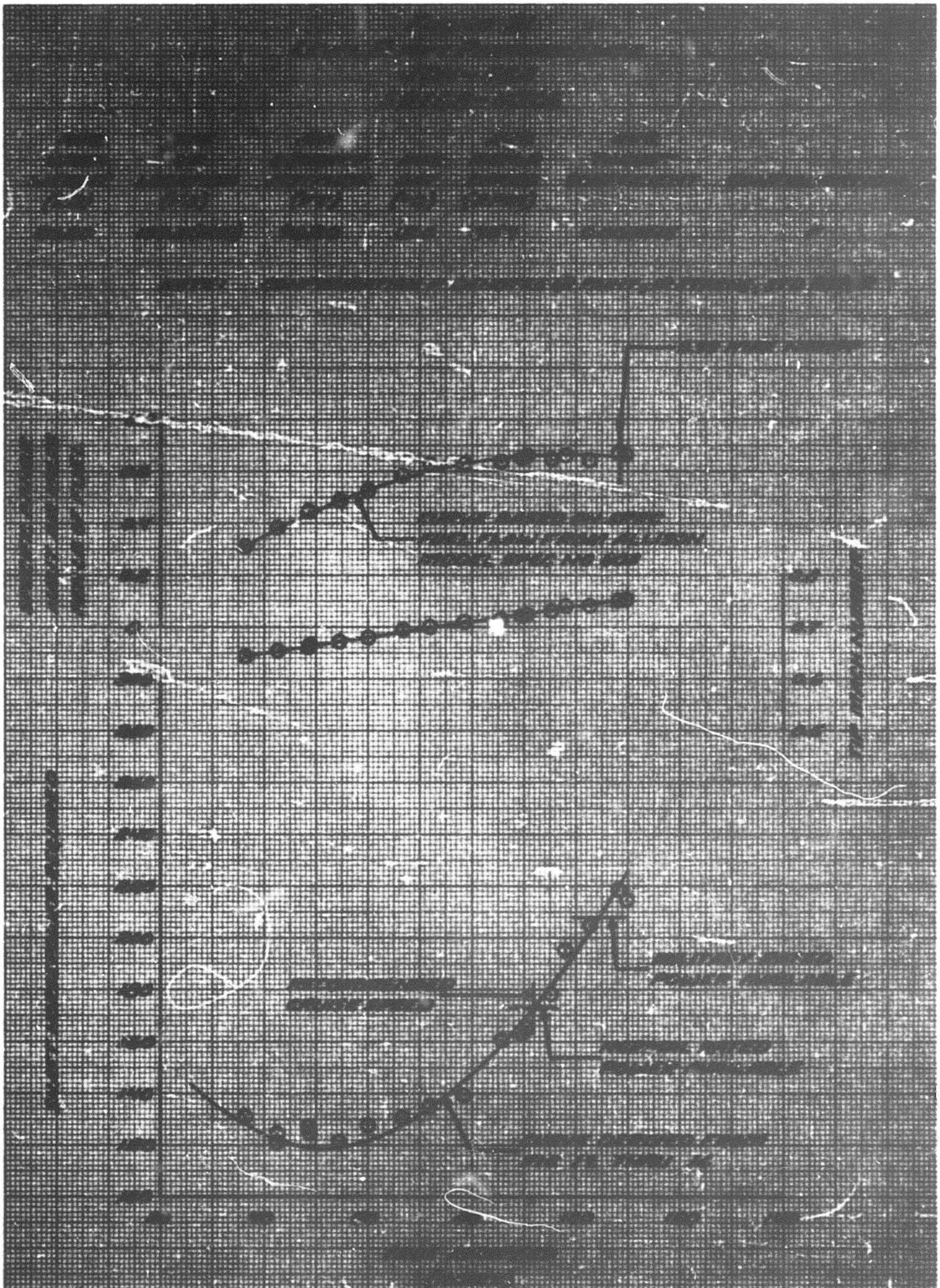

3. DASHED LINES DENOTE NORMAL RATED POWER LIMIT
CONFIGURATION A
CONFIGURATION F
0.7
0.6
110
100
90
RECOMMENDED CRUISE TRUE

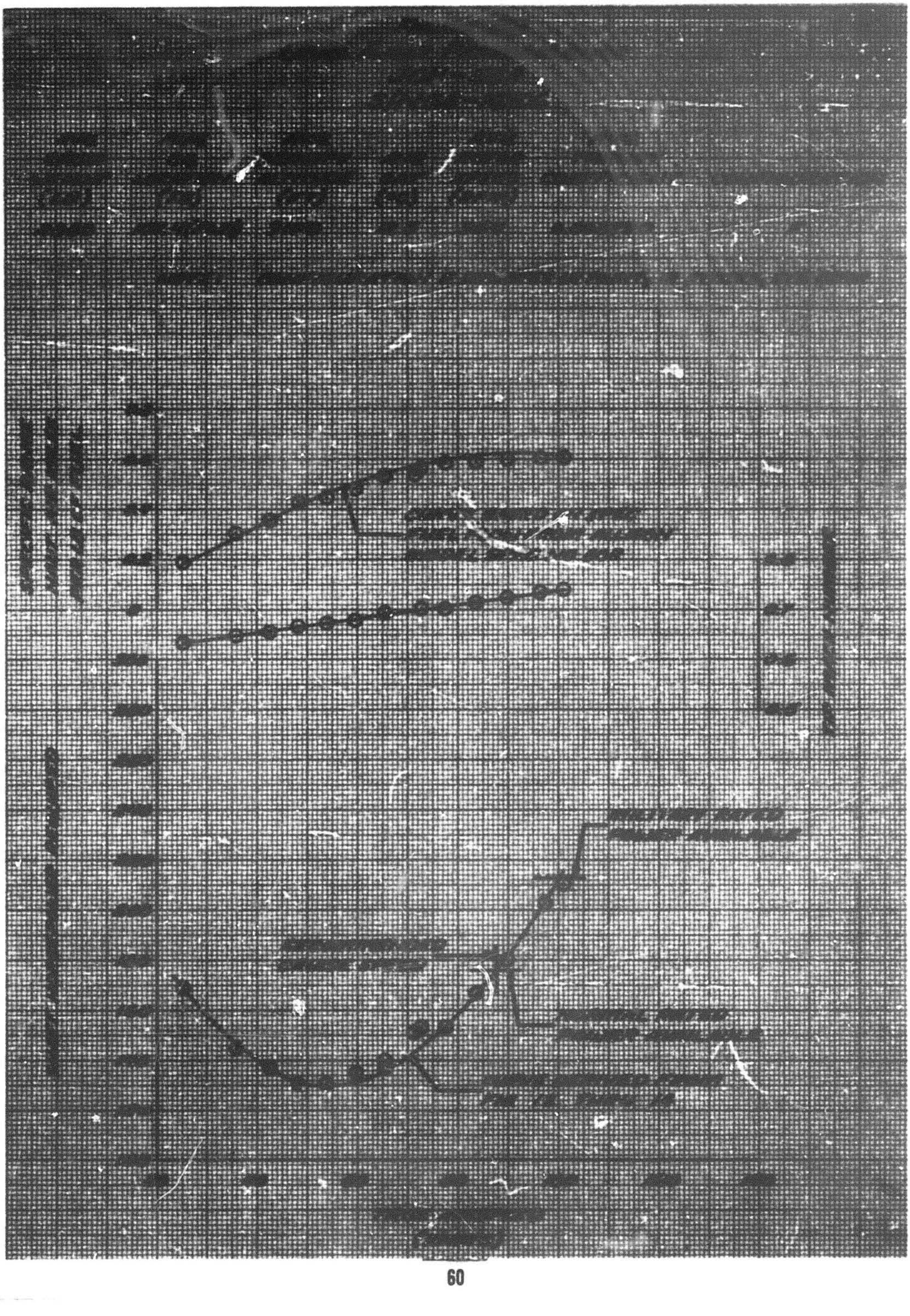


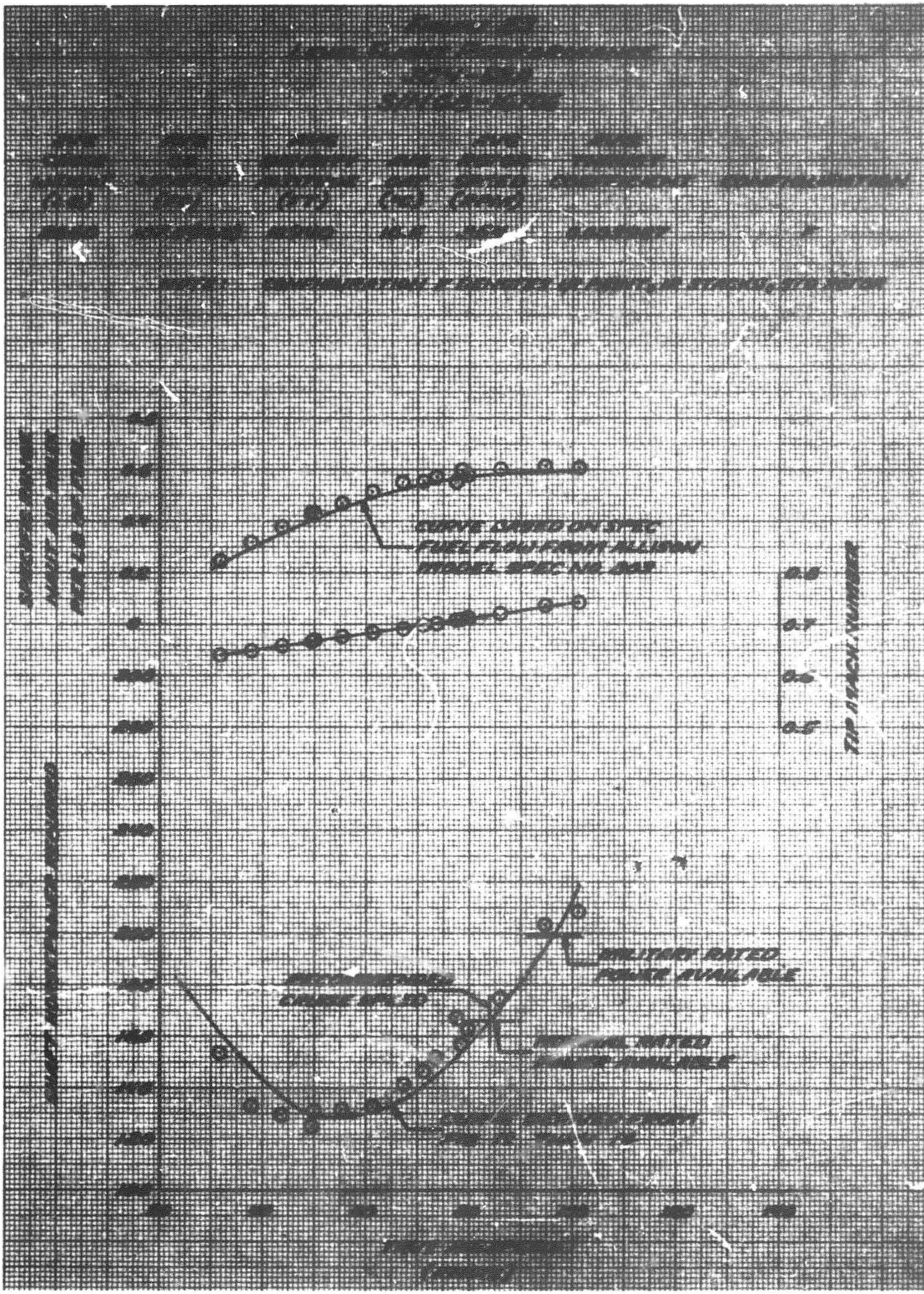




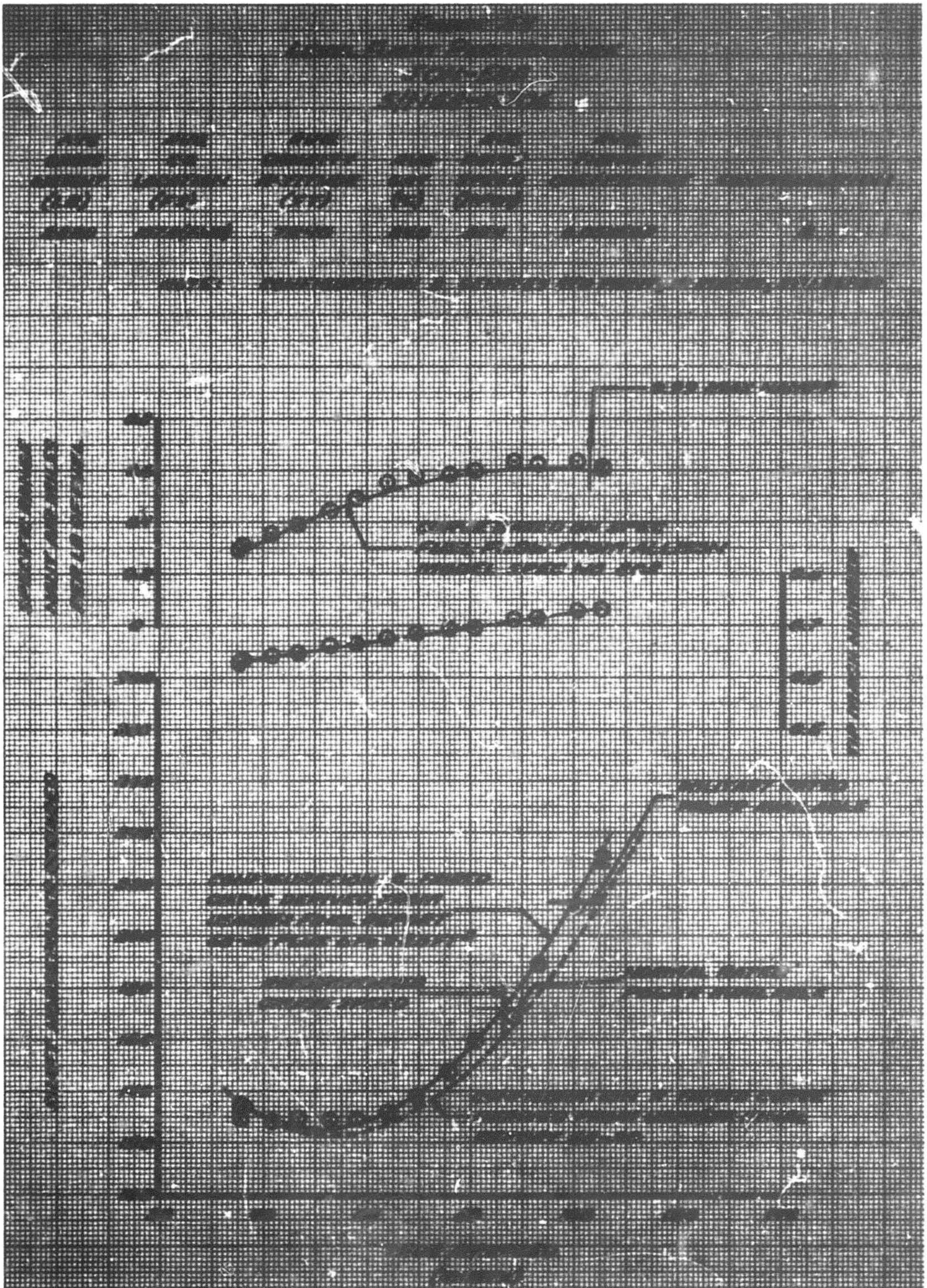


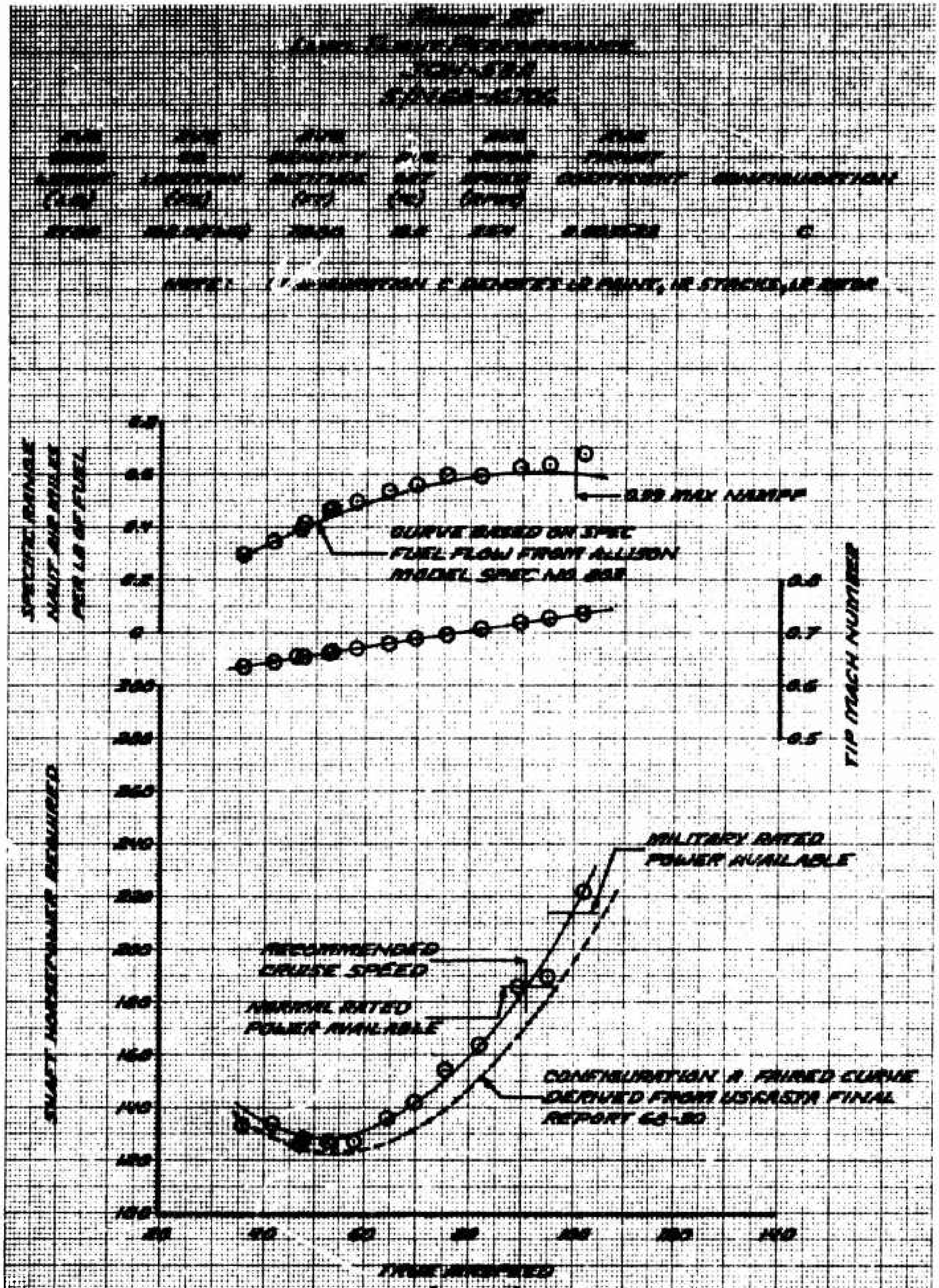


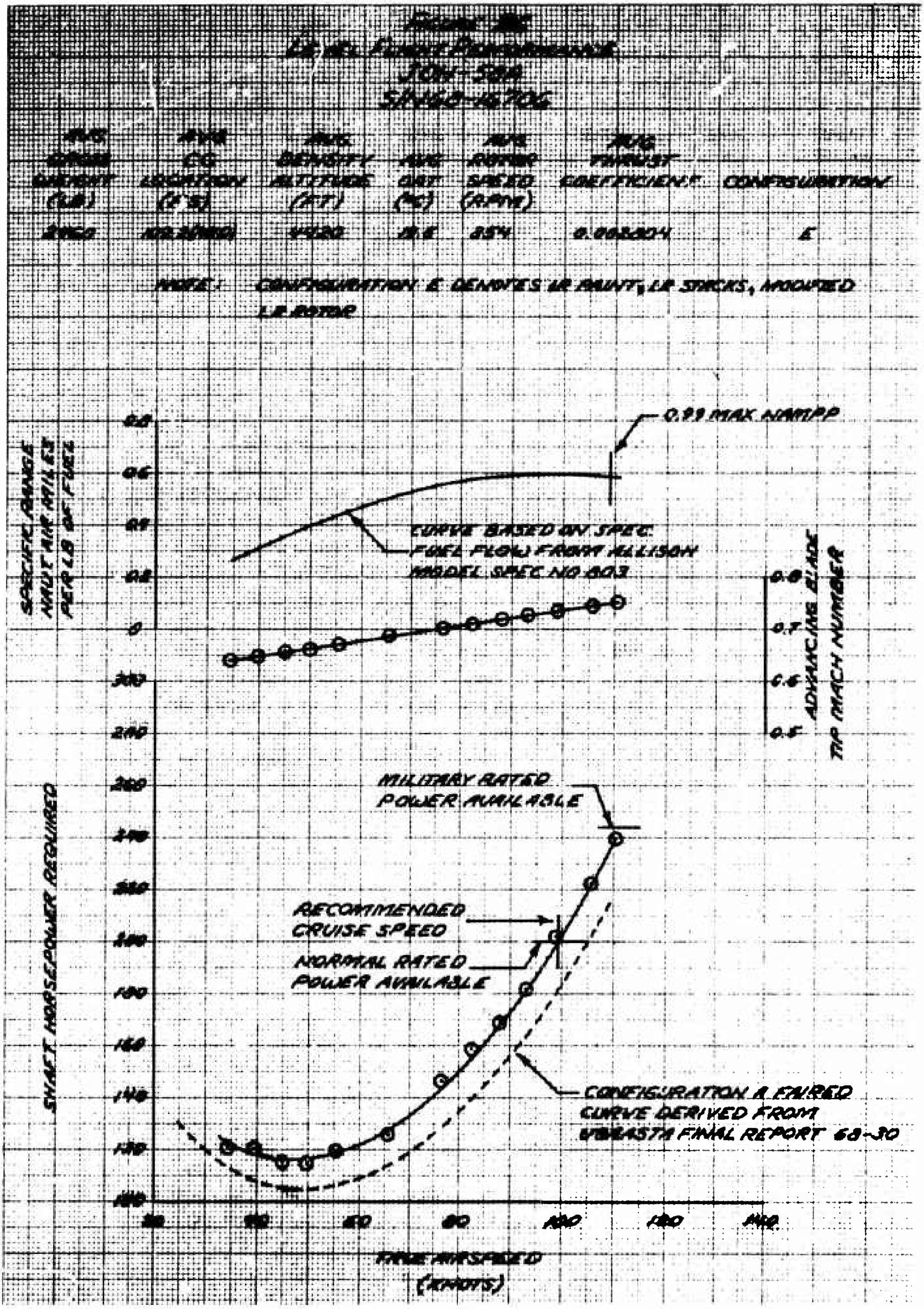


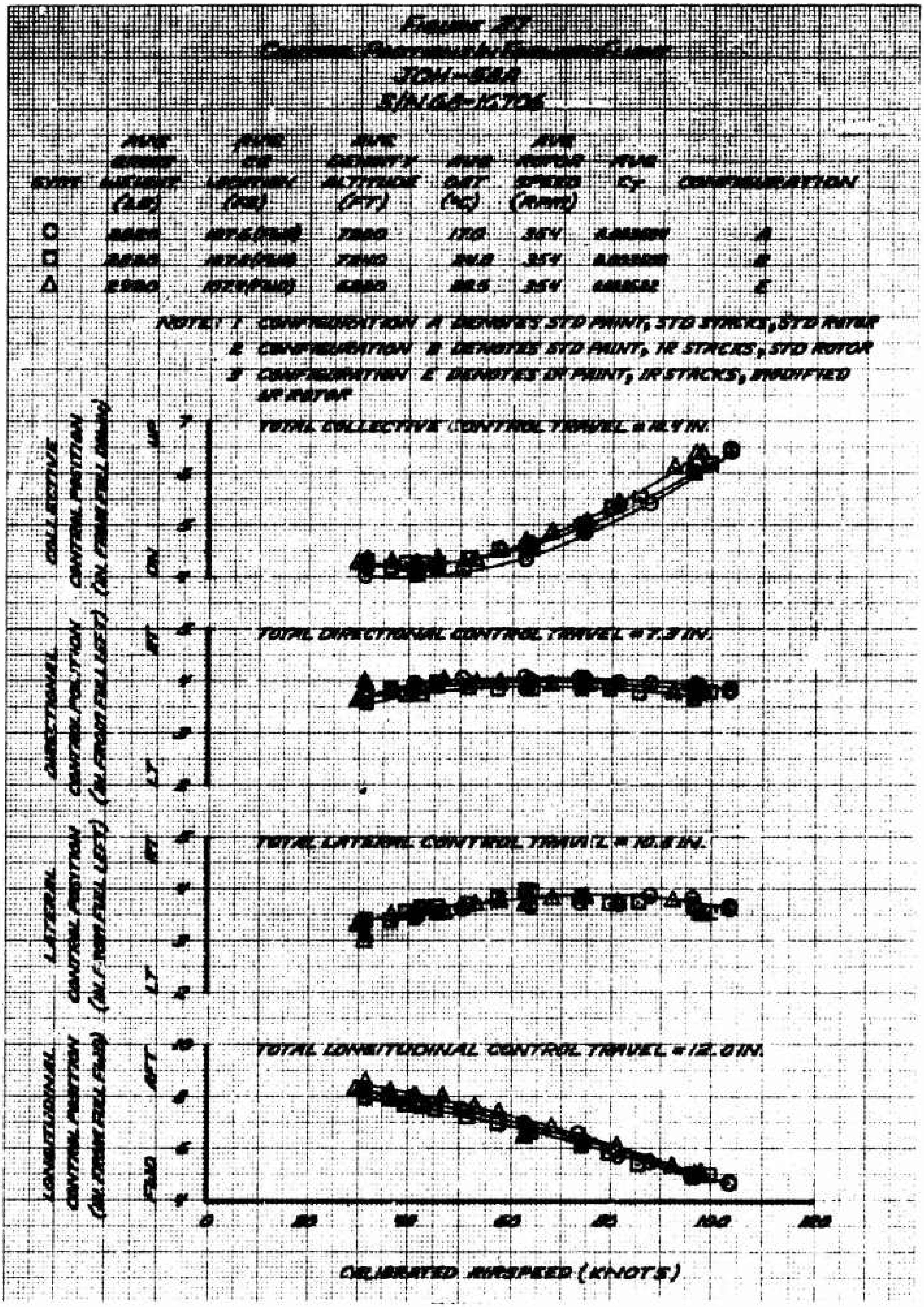


CURVE BASED ON SPEC
FUEL FLOW FROM ALLISON
TIP MACH NUMBER
MILITARY RATED
POWER AVAILABLE
CRUISE SPEED

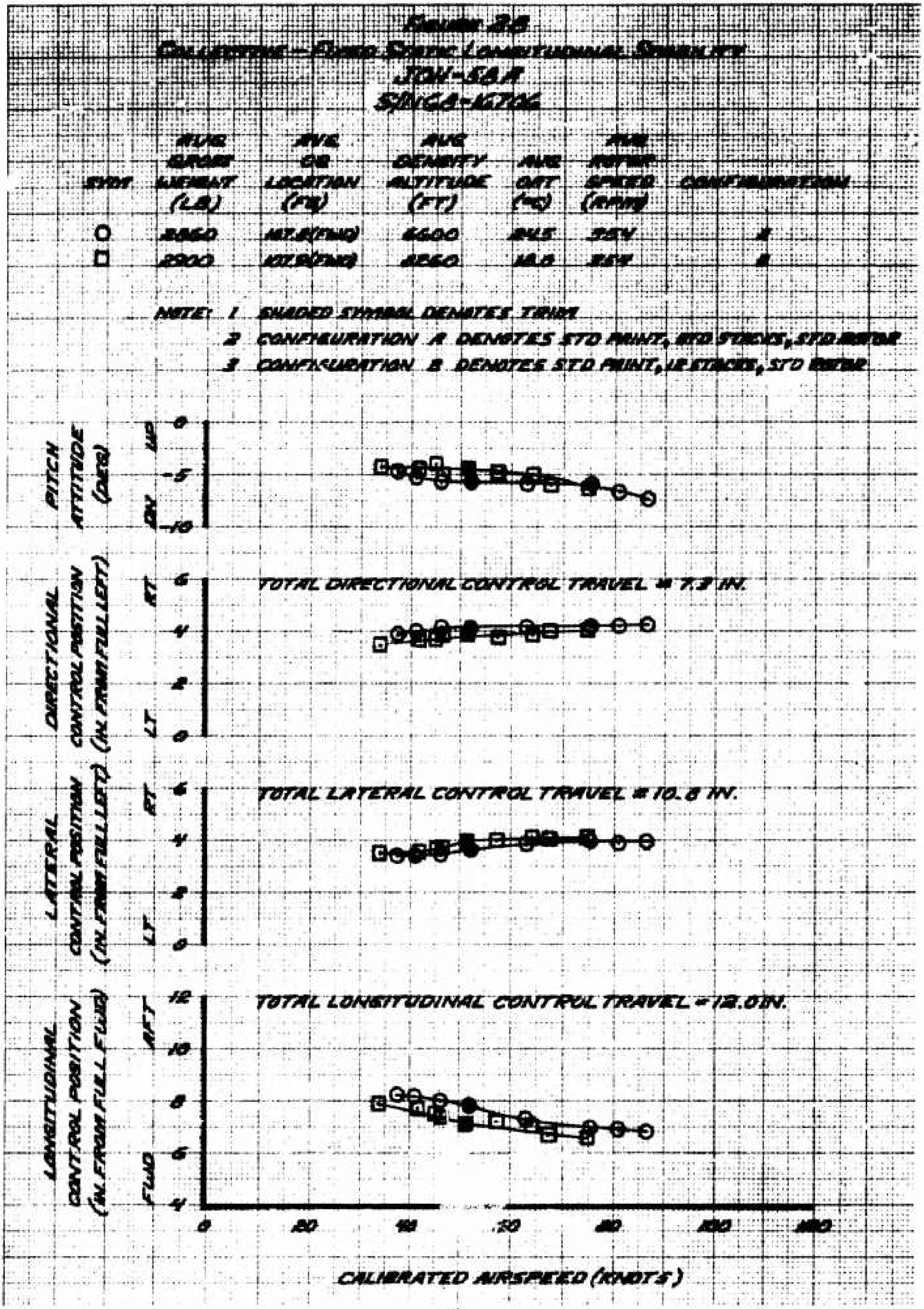



CURVE BASED ON SPEC
FUEL FLOW FROM ALLISON
MODEL SPEC
0.8
0.7
0.6
0.5
TIP MACH NUMBER
MILITARY RATED
POWER AVAILABLE
RECOMMENDED
CRUISE SPEED
NORMAL RATED
POWER AVAILABLE
CONFIGURATION A FAIRED CURVE
DERIVED FROM USAASTA FINAL
REPORT 68-30
TRUE AIRSPEED
(KNOTS)

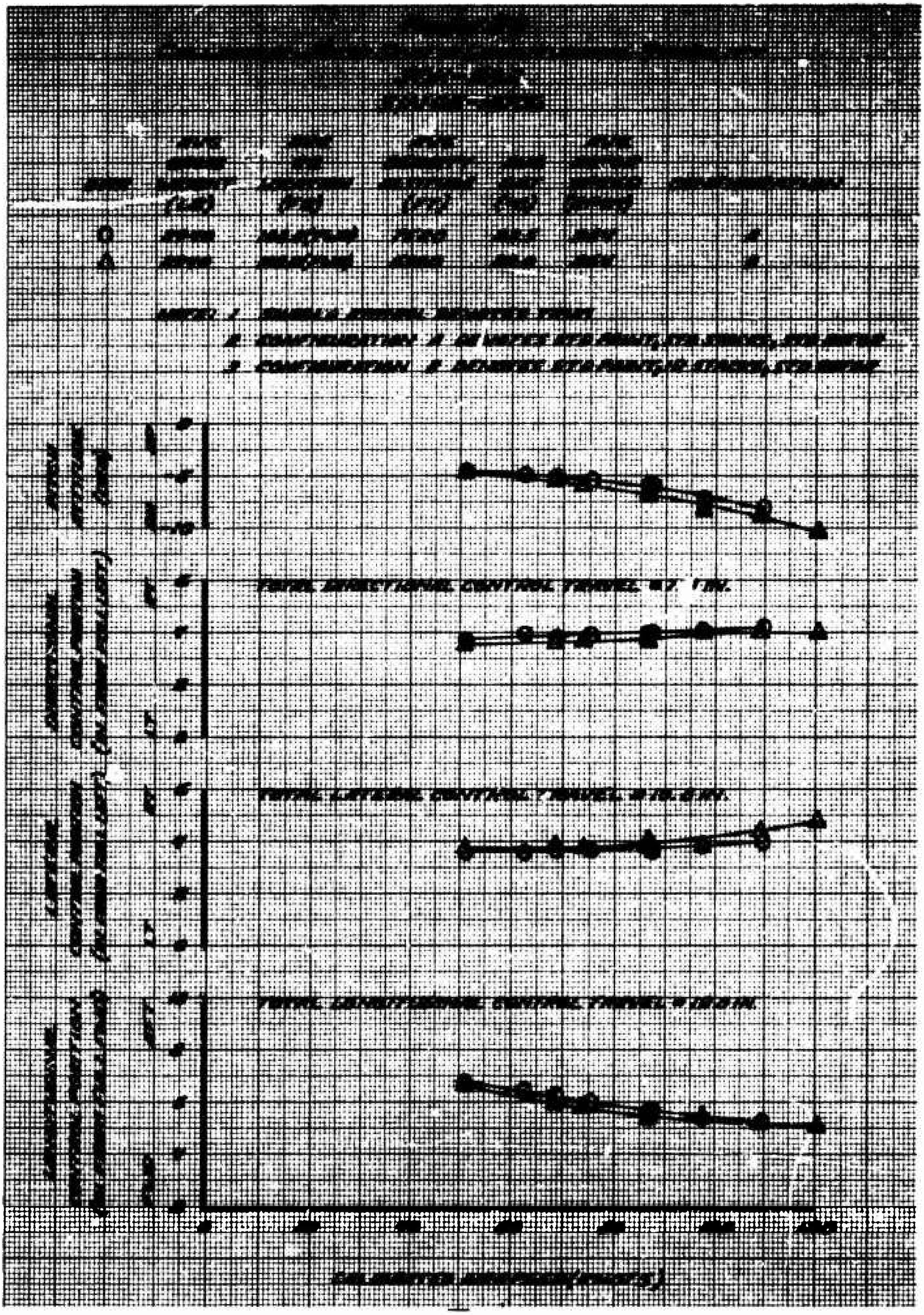

CONFIGURATION
CURVE BASED ON SPEC. FUEL FLOW FROM ALLISON MODEL SPEC NO 803
SPECIFIC RANGE NAUT AIR MILES PER LB OF FUEL
ADVANCING BLADE TIP MACH NUMBER
MILITARY RATED POWER AVAILABLE
RECOMMENDED CRUISE SPEED
NORMAL RATED POWER AVAILABLE
CONFIGURATION A FAIRED CURVE DERIVED FROM USAASTA FINAL REPORT 68-30
SHAFT HORSEPOWER REQUIRED
TRUE AIRSPEED
(KNOTS)

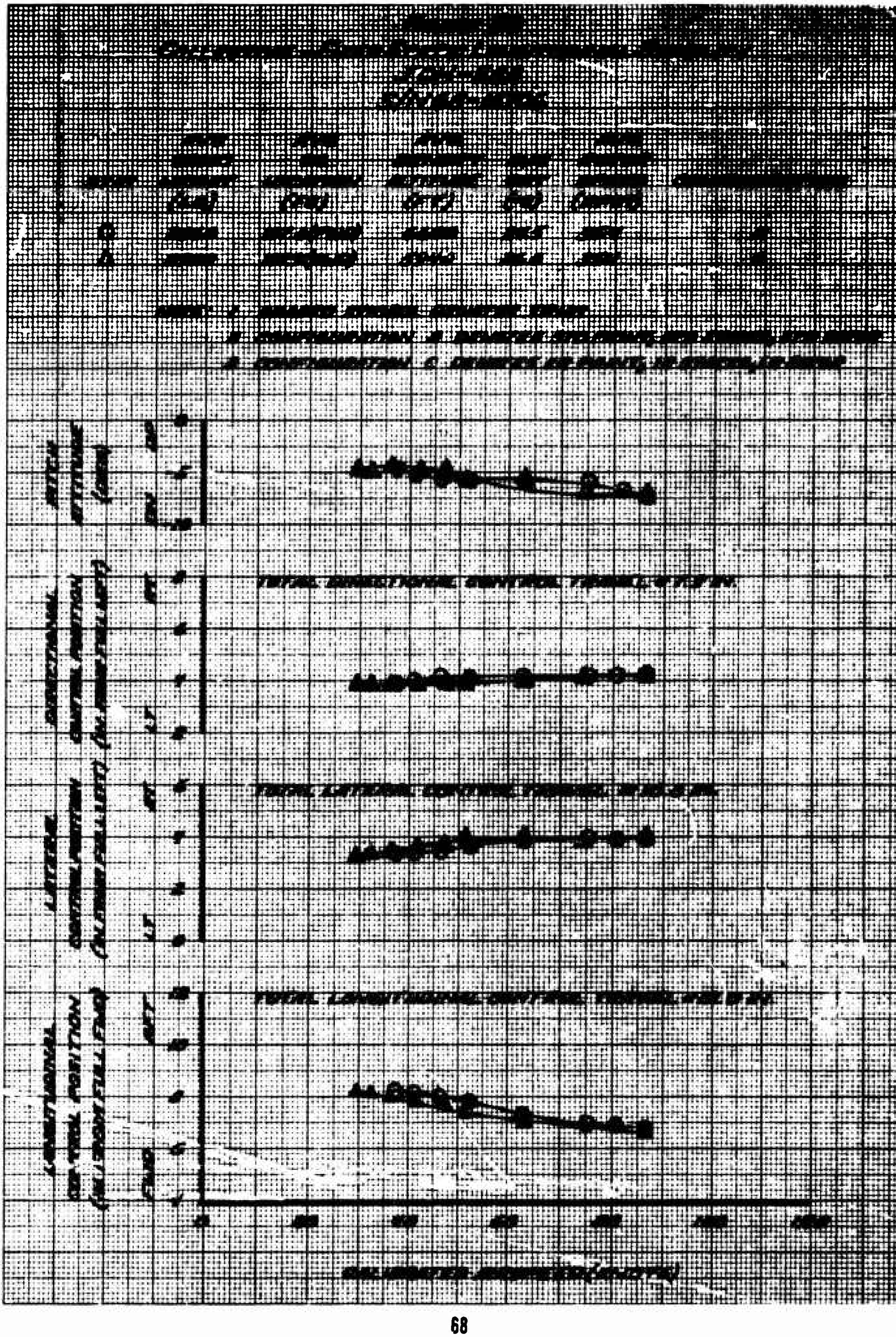


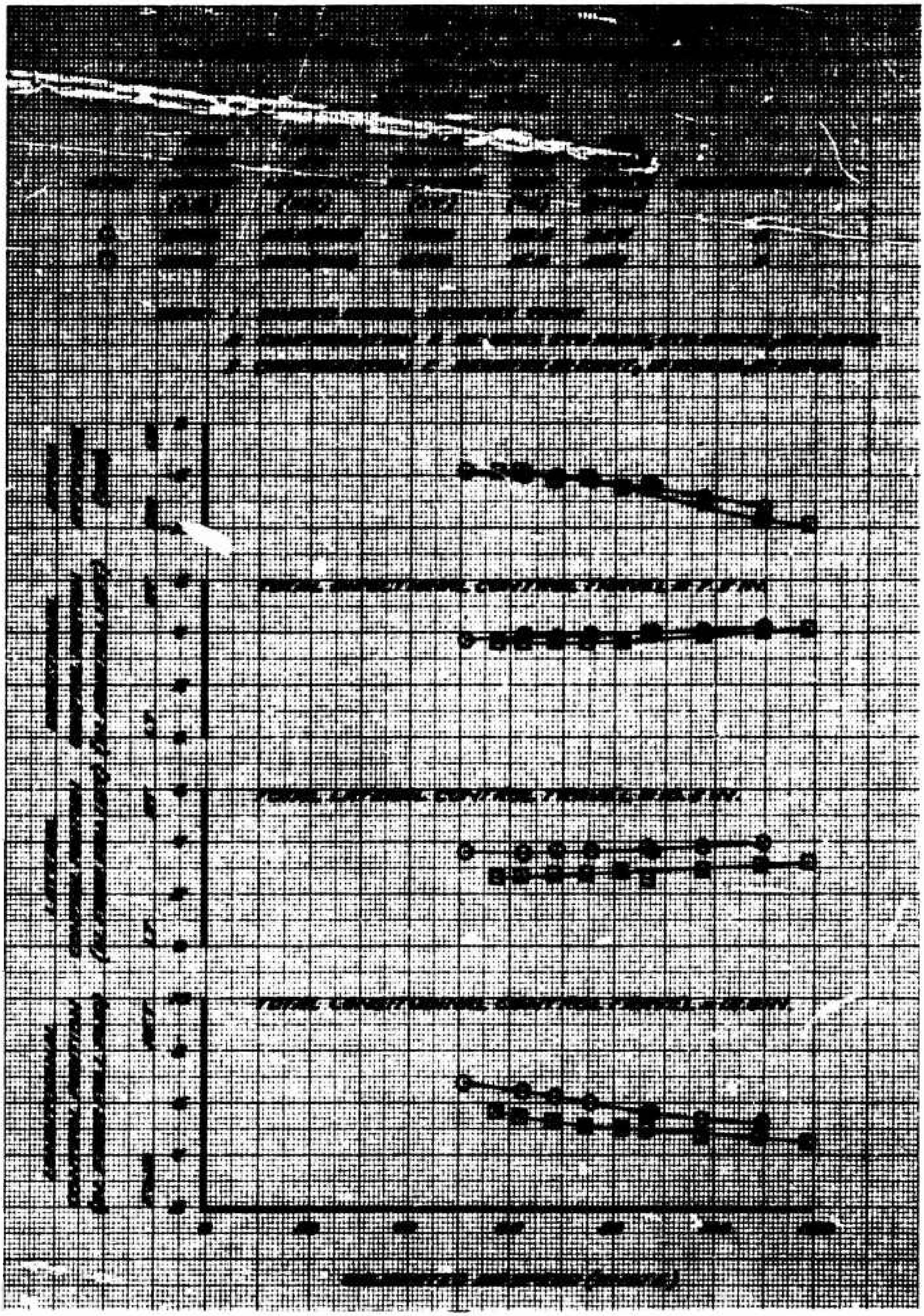
NOTE: 1 CONFIGURATION A DENOTES STD PAINT, STD STACKS, STD ROTOR
2 CONFIGURATION B DENOTES STD PAINT, IR STACKS, STD ROTOR
3 CONFIGURATION C DENOTES IR PAINT, IR STACKS, MODIFIED IR ROTOR
TOTAL DIRECTIONAL CONTROL TRAVEL = 7.3 IN.
TOTAL LATERAL CONTROL TRAVEL = 10.3 IN.
TOTAL LONGITUDINAL CONTROL TRAVEL = 12.0 IN.
CALIBRATED AIRSPEED (KNOTS)

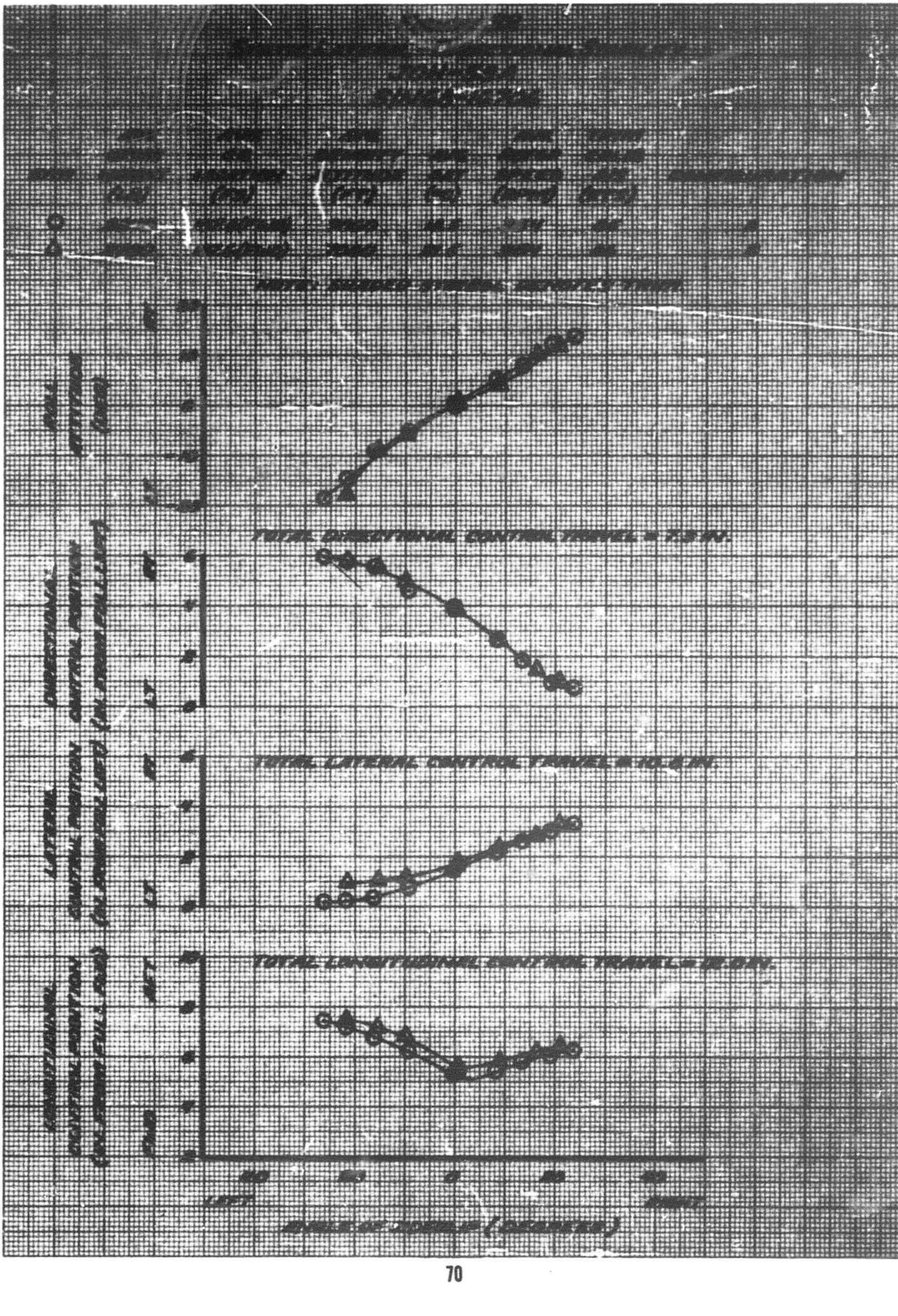


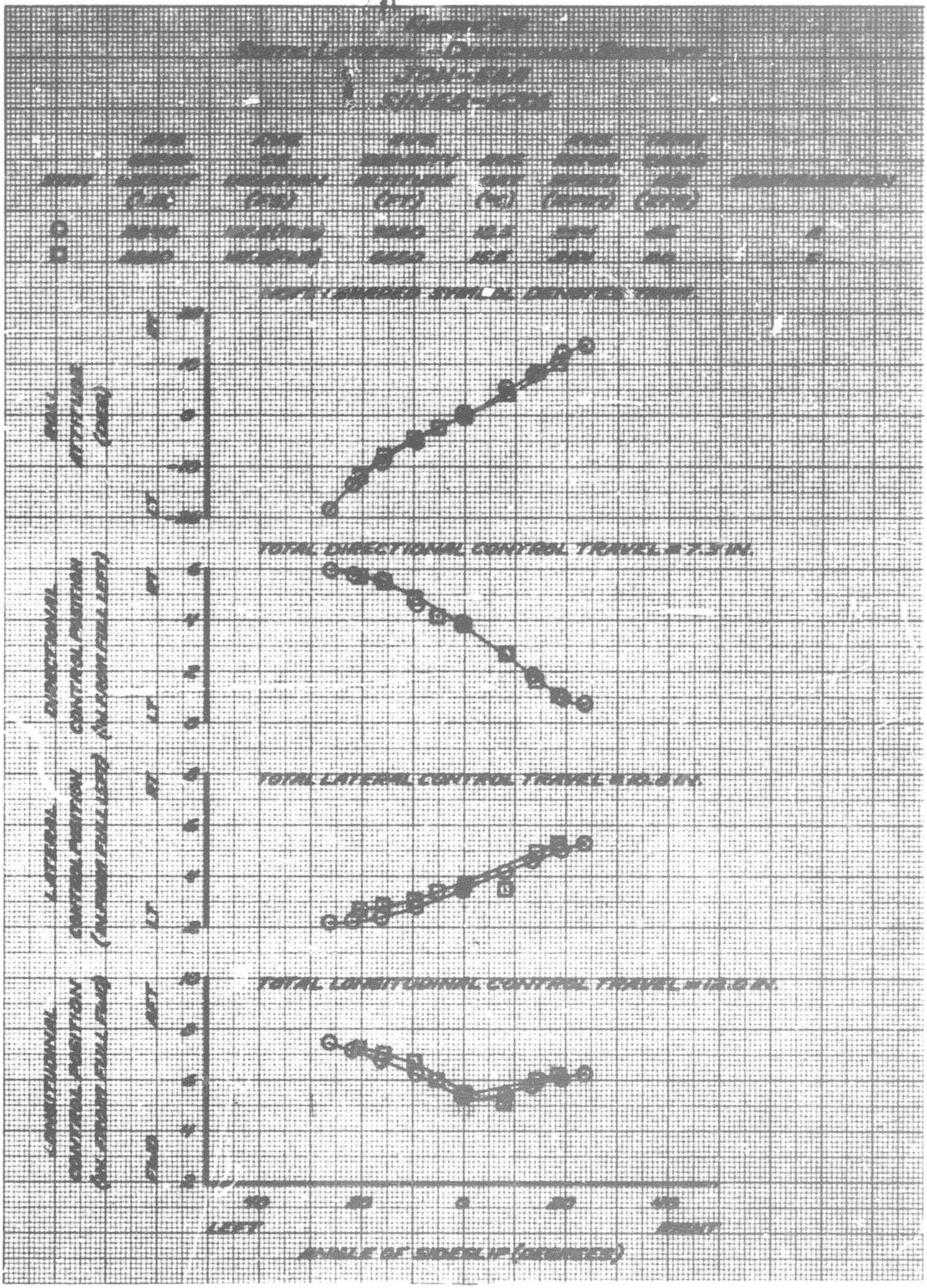
LOCATION
ALTITUDE
CONFIGURATION
NOTE: 1 SHADED SYMBOL DENOTES TRIM
2 CONFIGURATION A DENOTES STD PAINT, STD STACKS, STD ROTOR
3 CONFIGURATION B DENOTES STD PAINT, IR STACKS, STD ROTOR
PITCH ATTITUDE (DEG)
UP
DN
DIRECTIONAL CONTROL POSITION (IN. FROM FULL LEFT)
TOTAL DIRECTIONAL CONTROL TRAVEL = 7.3 IN.
LATERAL CONTROL POSITION (IN. FROM FULL LEFT)
TOTAL LATERAL CONTROL TRAVEL = 10.8 IN.
RT
LT
LONGITUDINAL CONTROL POSITION (IN. FROM FULL FWD)
TOTAL LONGITUDINAL CONTROL TRAVEL = 12.0 IN.
AFT
FWD
CALIBRATED AIRSPEED (KNOTS)








TOTAL DIRECTIONAL CONTROL TRAVEL = 7.3 IN.
TOTAL LATERAL CONTROL TRAVEL
TOTAL LONGITUDINAL CONTROL TRAVEL
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)


TOTAL DIRECTIONAL CONTROL TRAVEL = 7.3 IN.
TOTAL LATERAL CONTROL TRAVEL
TOTAL LONGITUDINAL CONTROL TRAVEL
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)

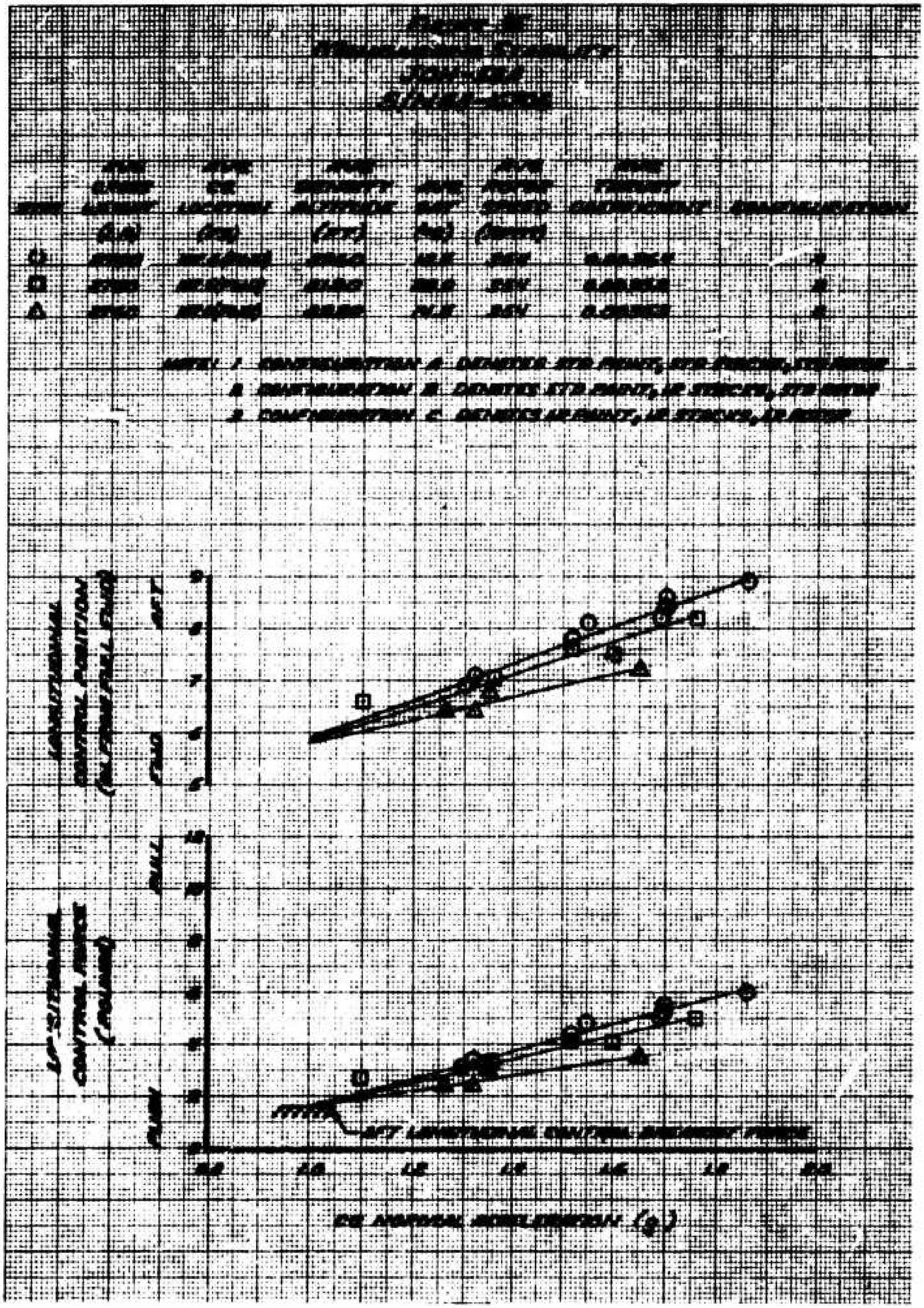


FIGURE
LONGITUDINAL LONG TERM RESPONSE
JOH-58A S/N 68-16706
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION
2800	107.8 (FWD)	7680	24.0	354	62	0.00362	CONFIG. A

NOTE: SCAS OFF

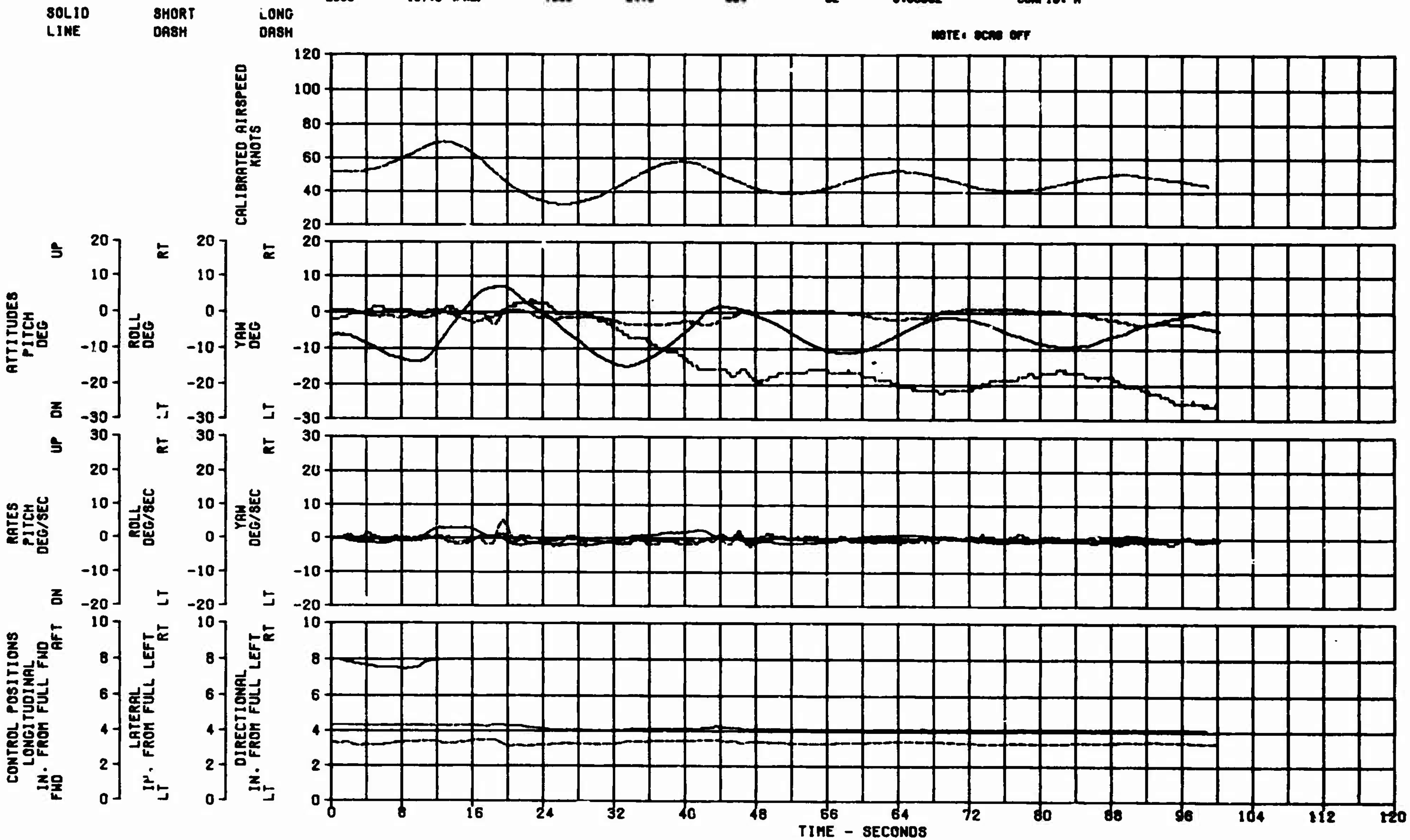


FIGURE 37
LONGITUDINAL LONG TERM RESPONSE
JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION
2800	107.8 (FWD)	7600	24.0	354	62	0.00352	CONFIG. A

NOTE: SCAS OFF

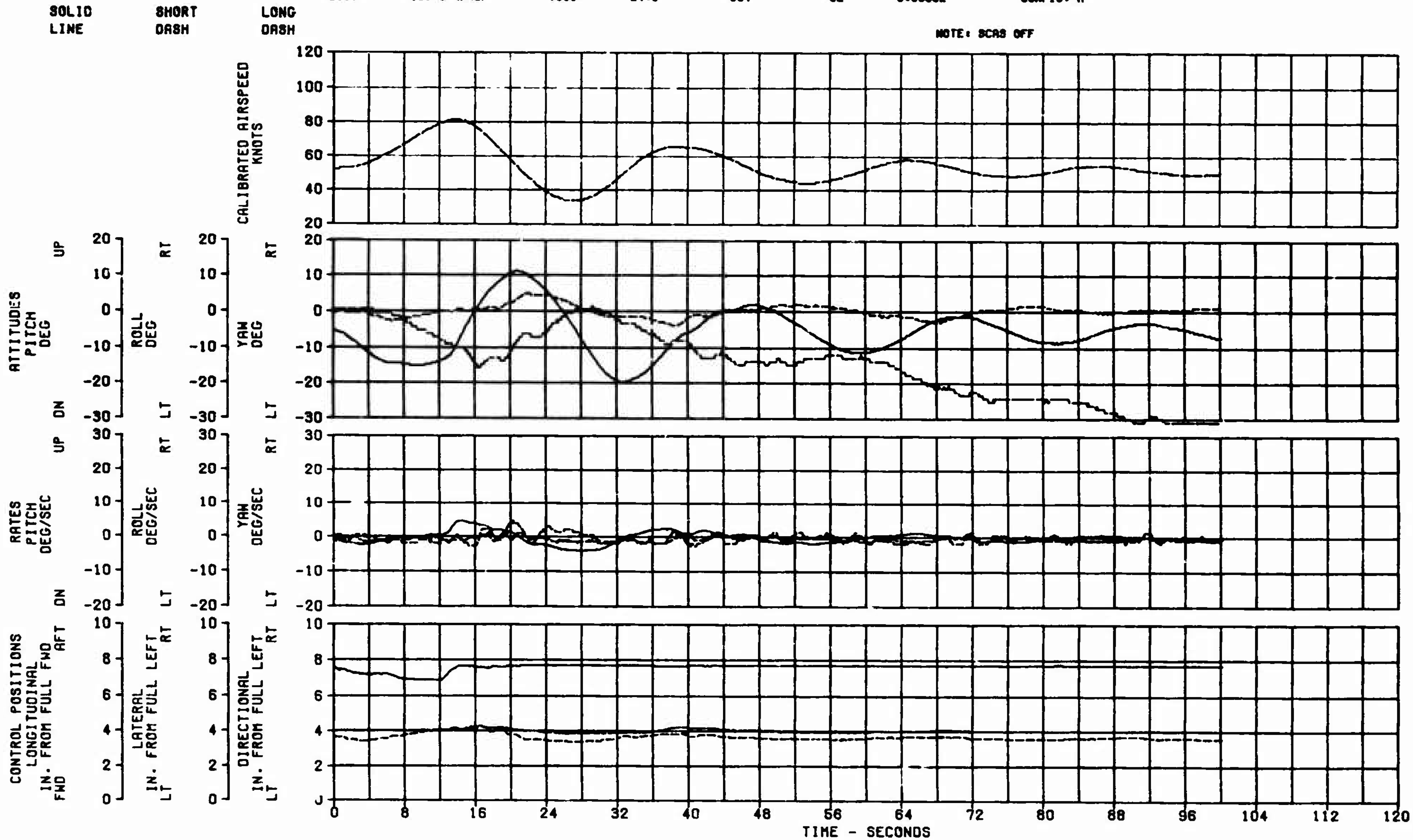


Figure 38

LONGITUDINAL LONG TERM RESPONSE

MODEL T63-A-700 ENGINE

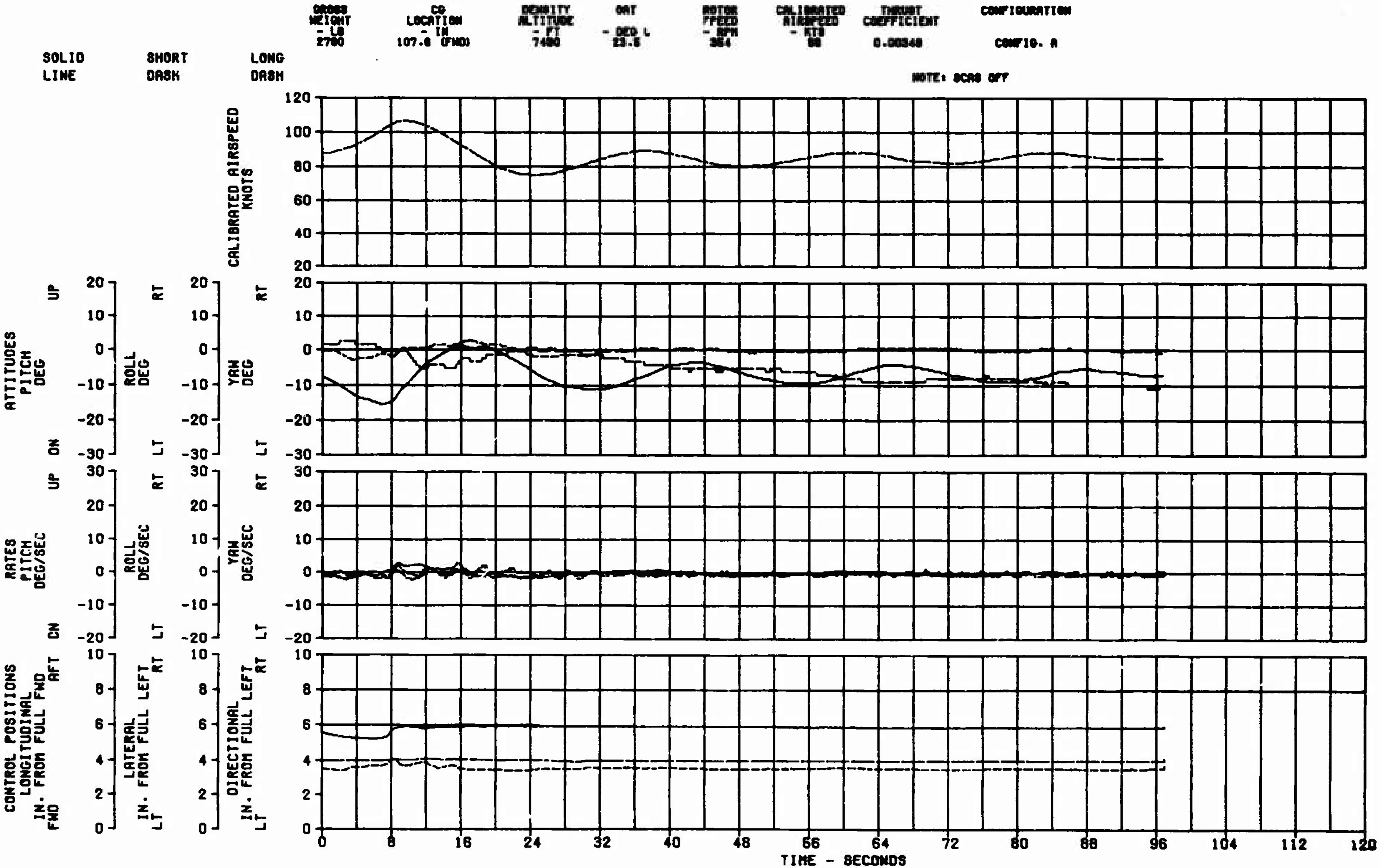


FIGURE 39
LONGITUDINAL LONG TERM RESPONSE
JOH-58A S/N 68-16706
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION
2920	107.7 (FWD)	5860	17.0	354	41	0.00348	CONFIG. C

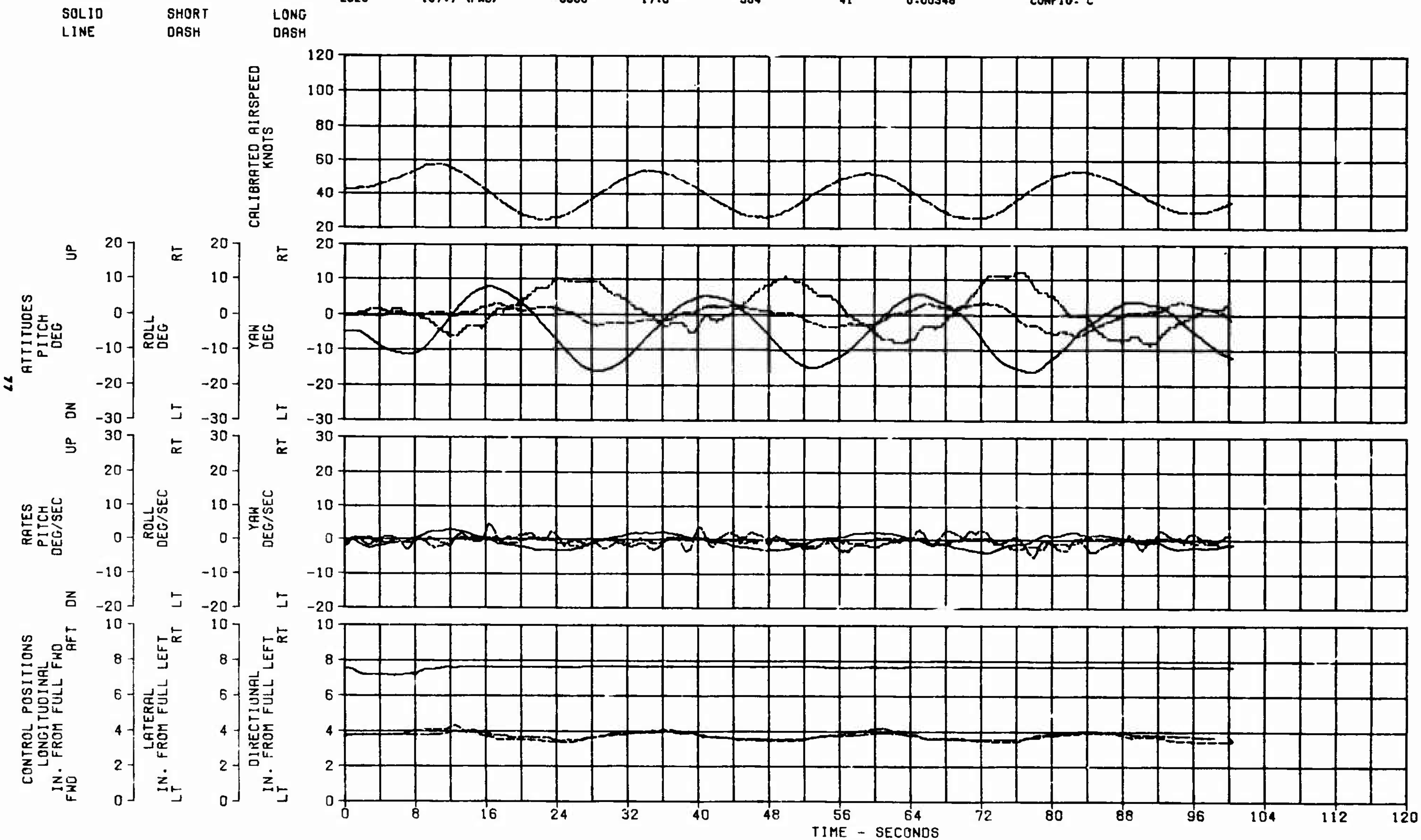


FIGURE 40
LONGITUDINAL LONG TERM RESPONSE
JOH-58A S/N 68-16706
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION
2720	107.5 (FWD)	7780	14.0	354	42	0.00344	CONFIG. C

NOTE: SCAS OFF

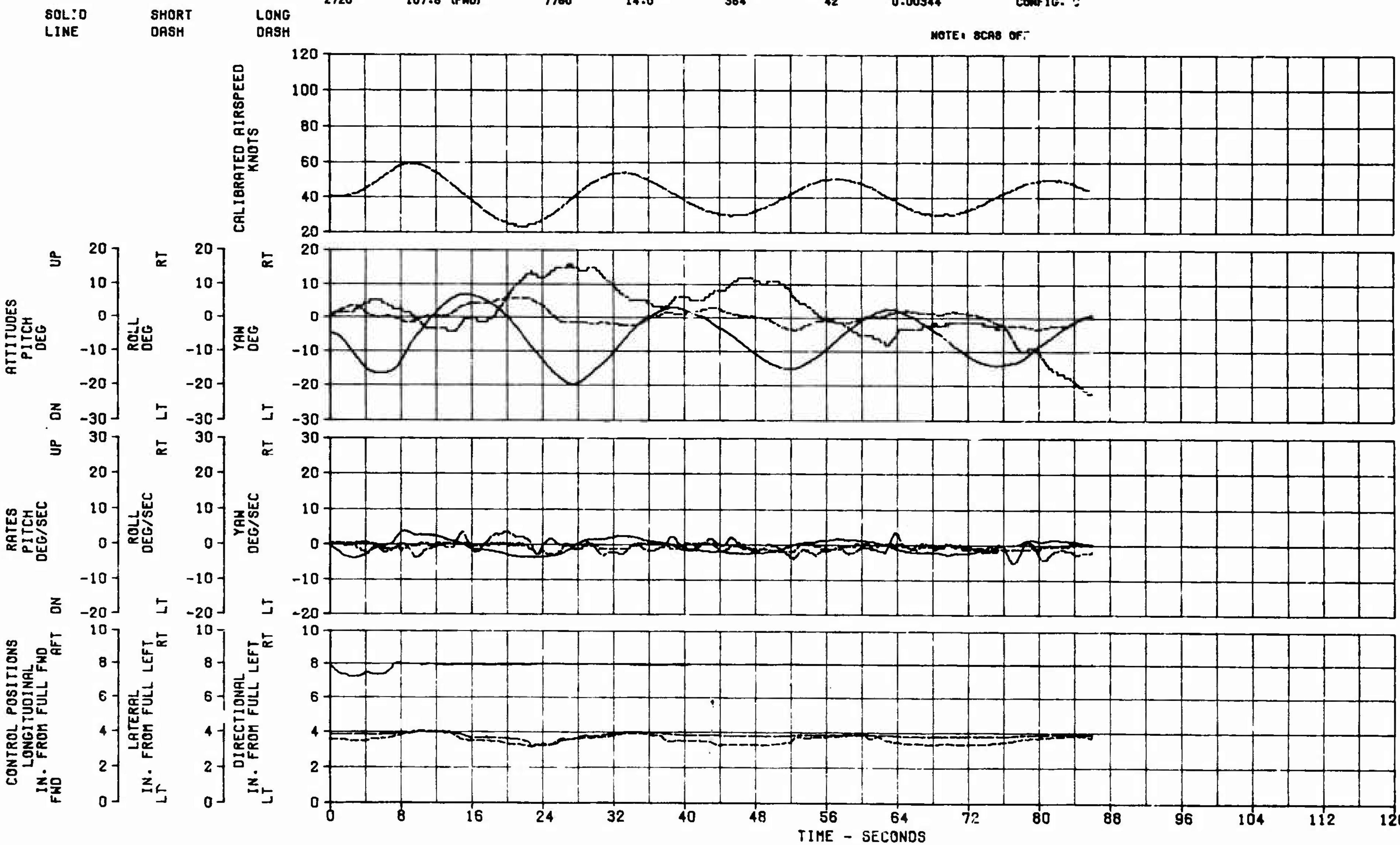


FIGURE 41

LONGITUDINAL PULSE

JOH-58A S/N 69-16706
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION -IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2780	107.5 (FWD)	7200	14.0	354	[illegible]	0.00345	CONFIG. C	LEVEL

NOTE: SCAS OFF

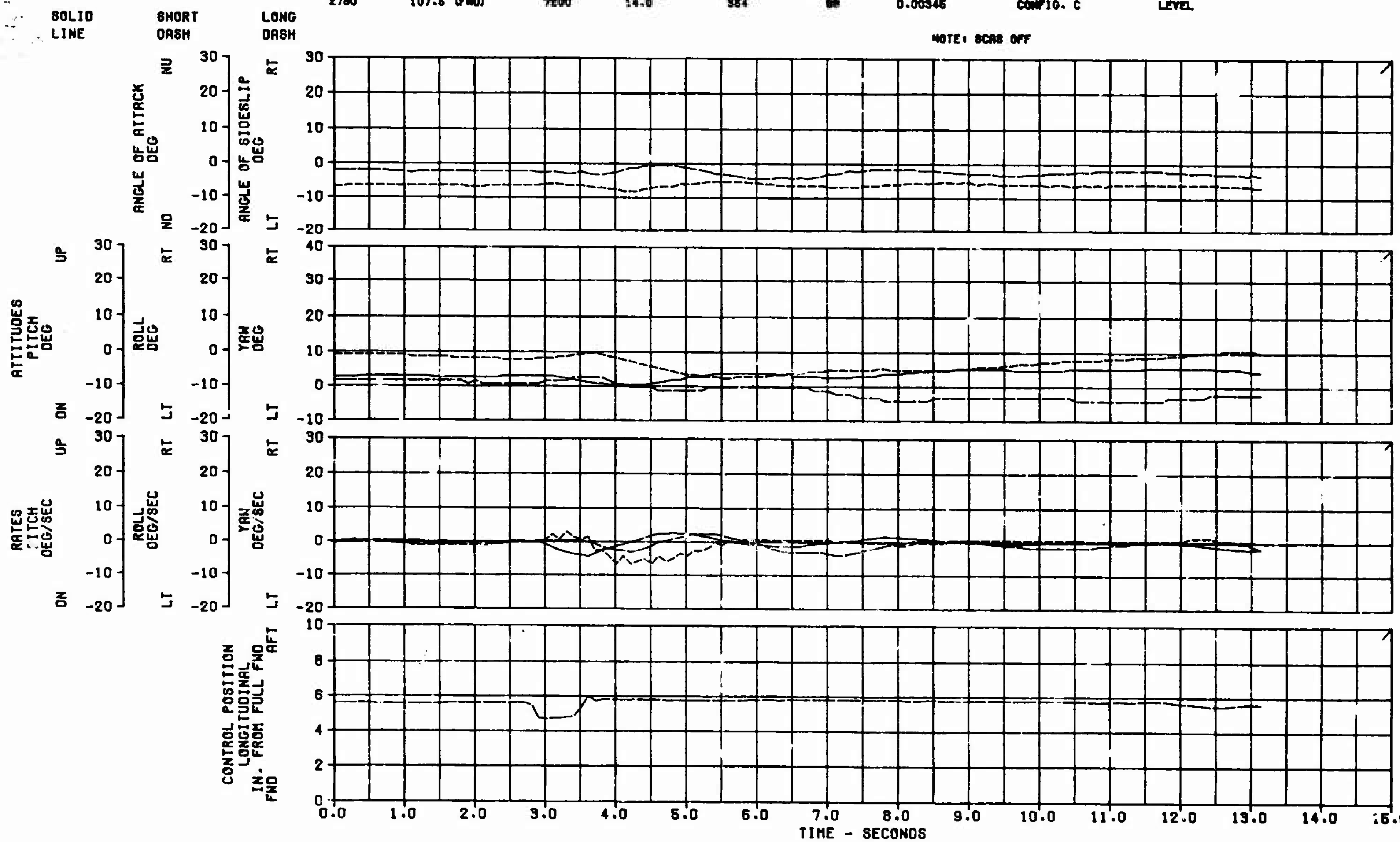


FIGURE 42

RELEASE FROM STEADY SIDESLIP

JOH-58A S/N 53-16706
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2740	107.6 (FWD)	8340	42.5	354	42	0.00362	CONFIG. A	LEVEL

NOTE: SCAS OFF

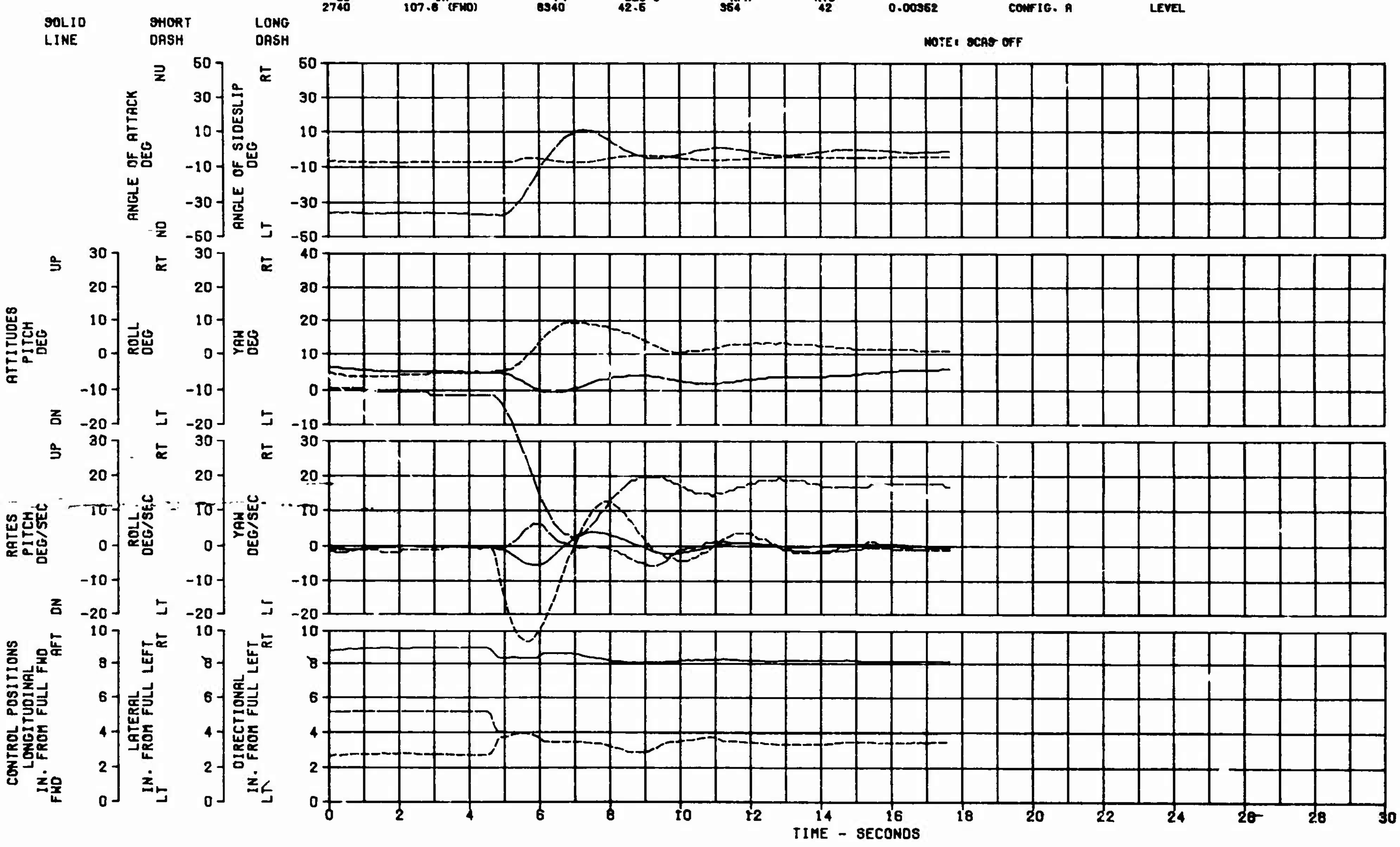


FIGURE 43
RELEASE FROM STEADY SIDESLIP
JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2720	107.6 (FWD)	8320	42.0	364	86	0.00349	CONFIG. A	LEVEL

NOTE: SCAS OFF

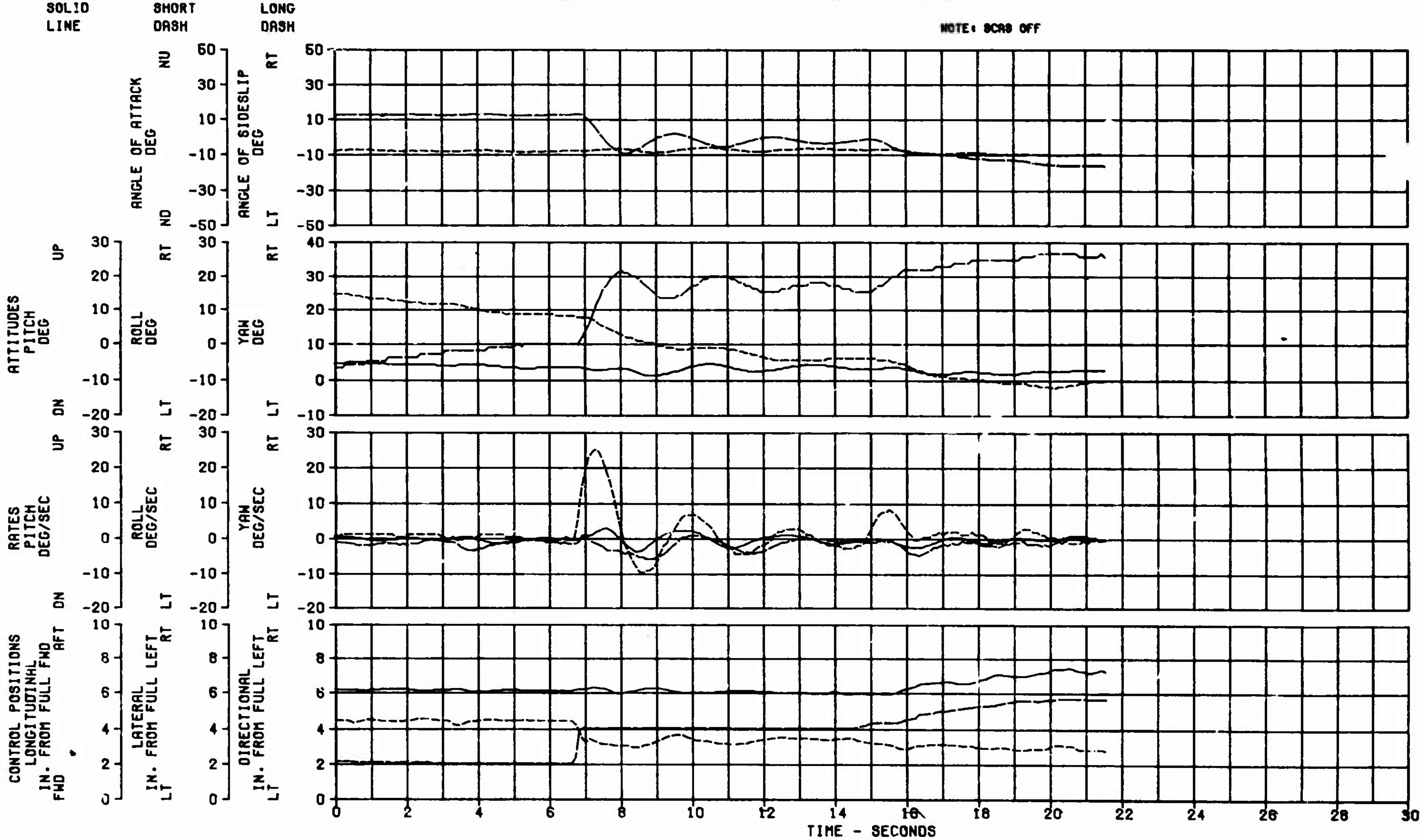


FIGURE 44
RELEASE FROM STEADY SIDESLIP
JOH-58A S/N 68-16709
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2760	107.6 (FWD)	6580	20.5	354	83	0.00336	CONFIG. B	LEVEL

NOTE: SCAS OFF

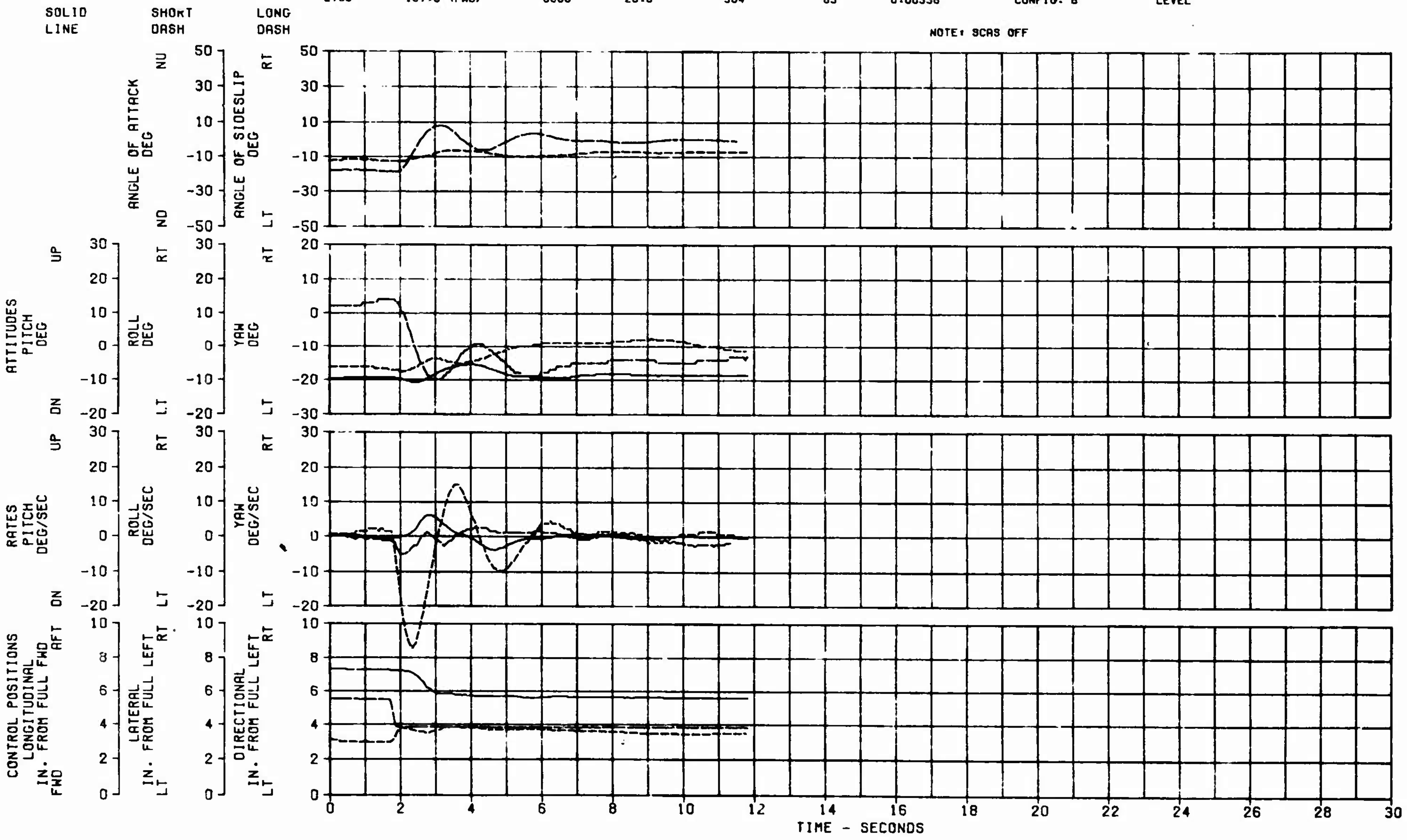


FIGURE 45
RELEASE FROM STEADY SIDESLIP
MODEL T63-A-700 ENGINE

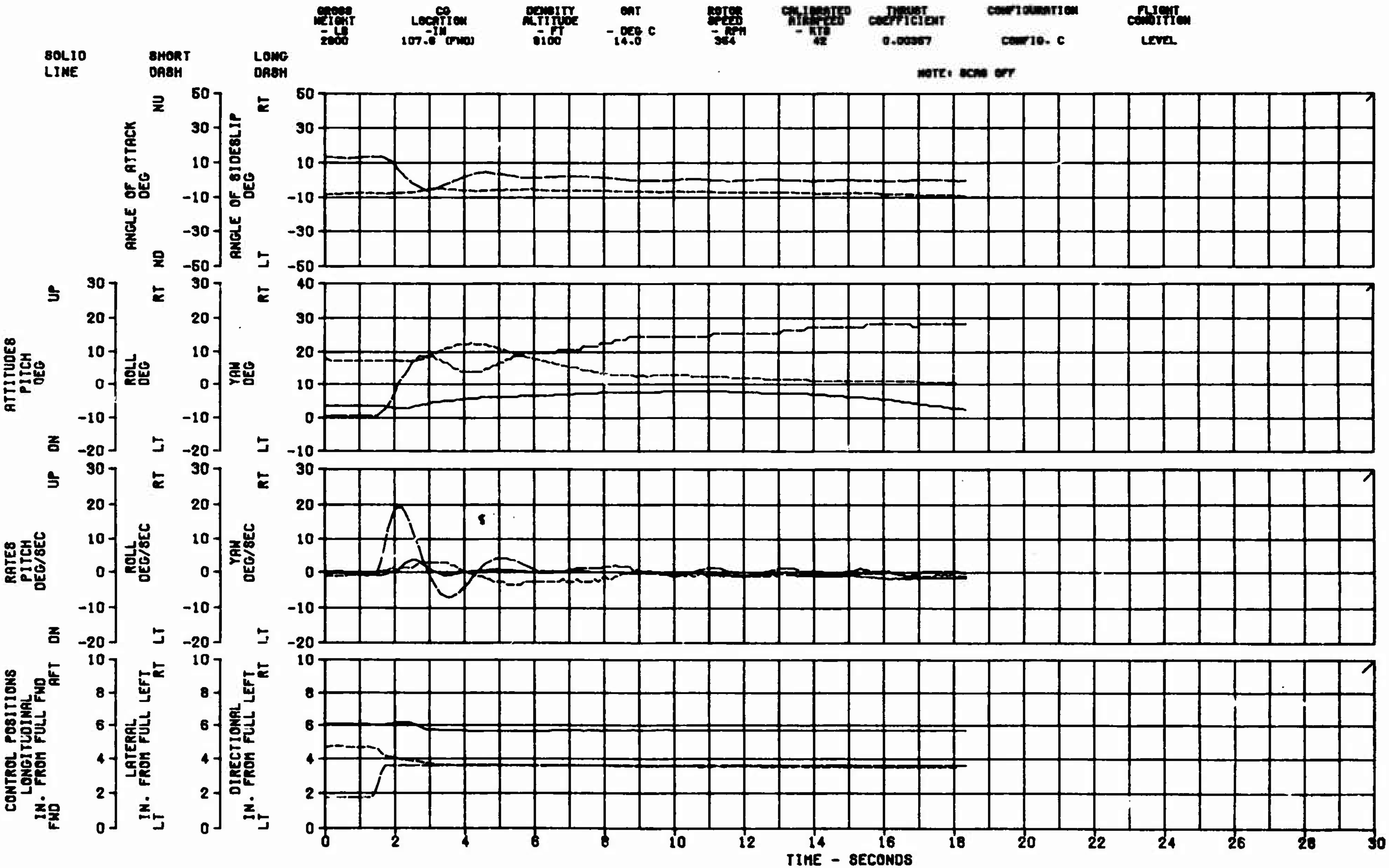


FIGURE 46
RELEASE FROM STEADY SIDESLIP
OH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION -IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2800	107.8 (FWD)	8340	14.0	354	86	0.00360	CONFIG. C	LEVEL

NOTE: SCAS OFF

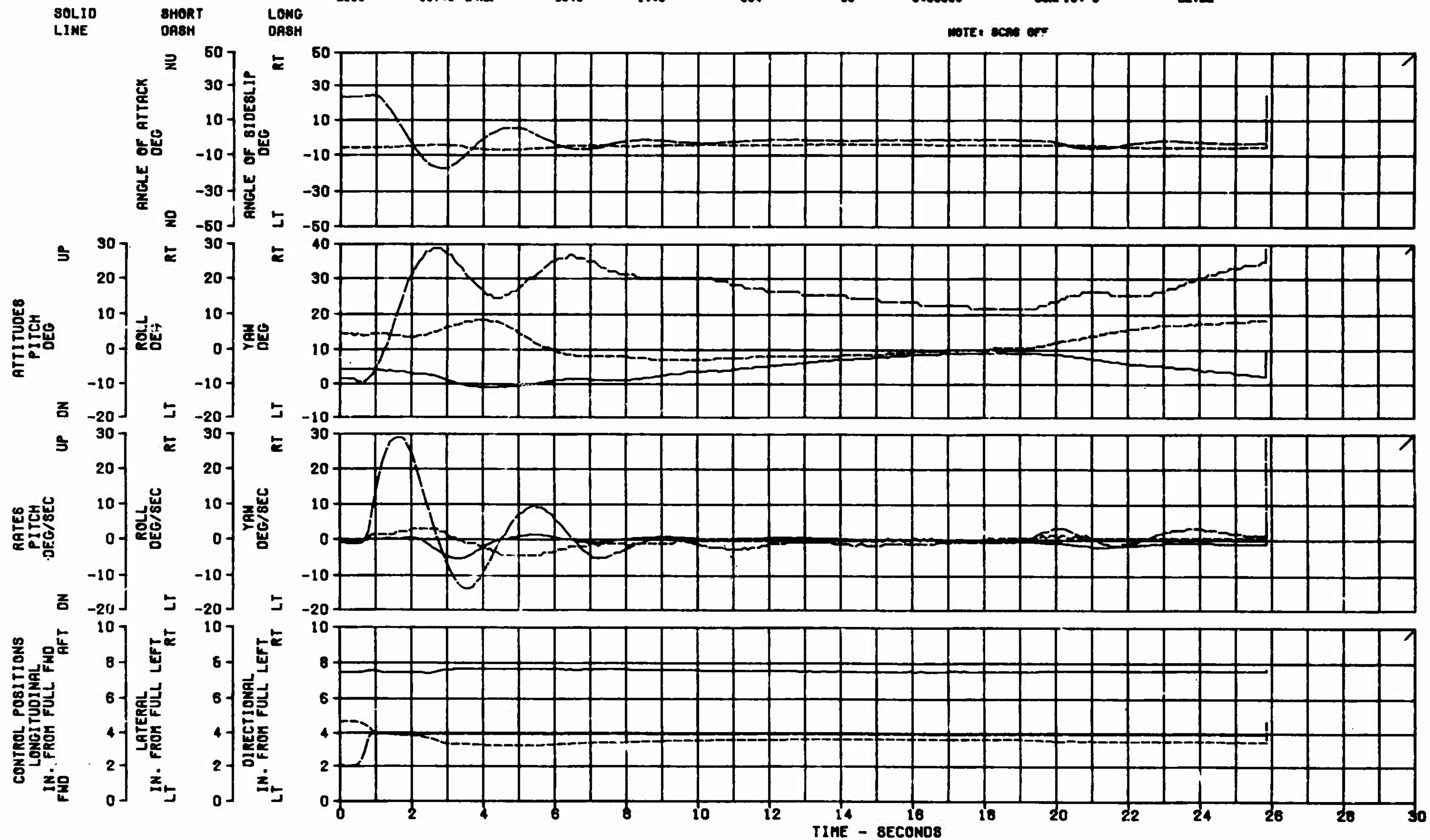


FIGURE 47
SPIRAL STABLITY
JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2720	107.5 (FWD)	8400	22.0	354	87	0.00350	CONFIG. A	LEVEL

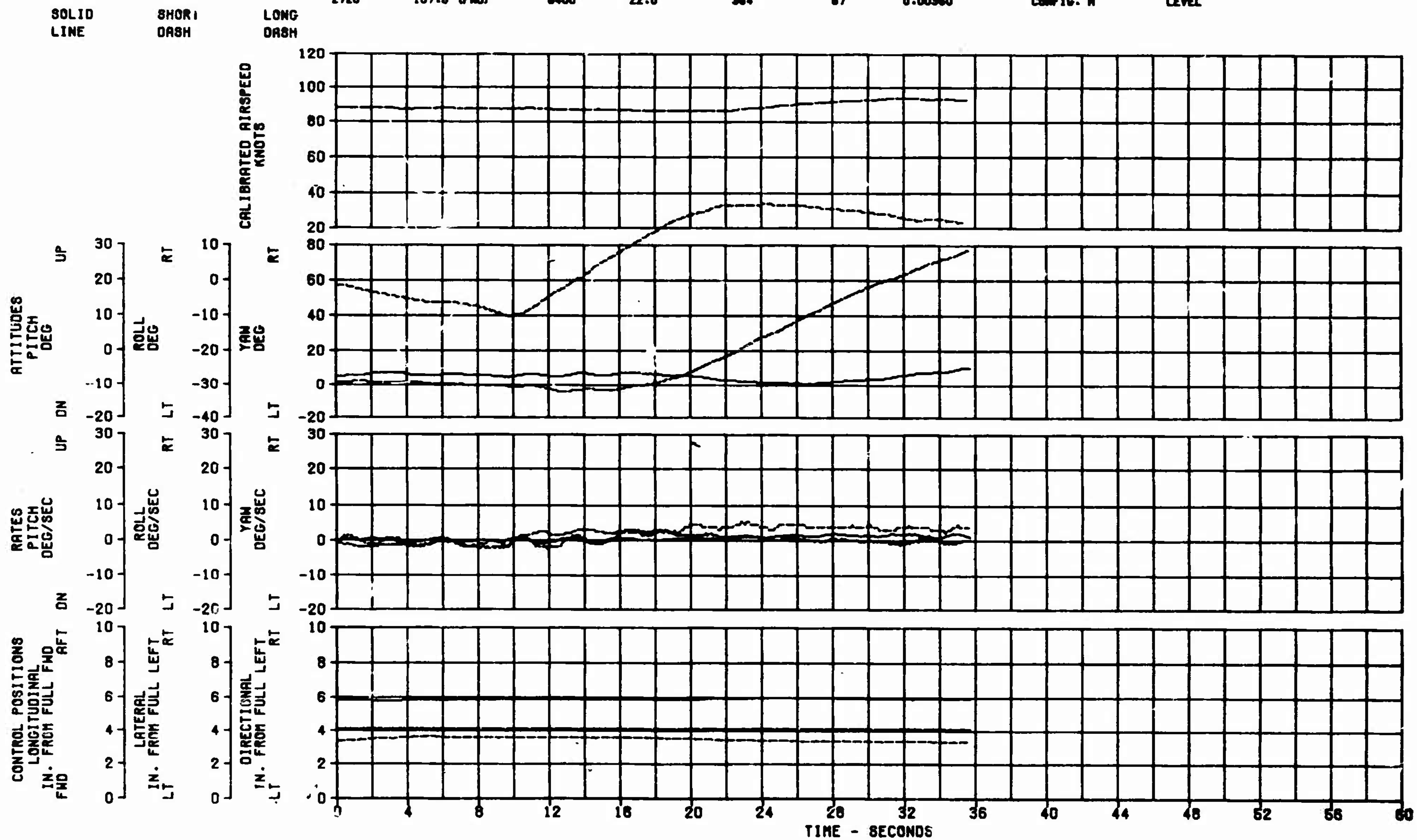


FIGURE 48
SPIRAL STABLITY
JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2840	107.6 (FWD)	6260	16.0	364	87	0.00342	CONFIG. C	LEVEL

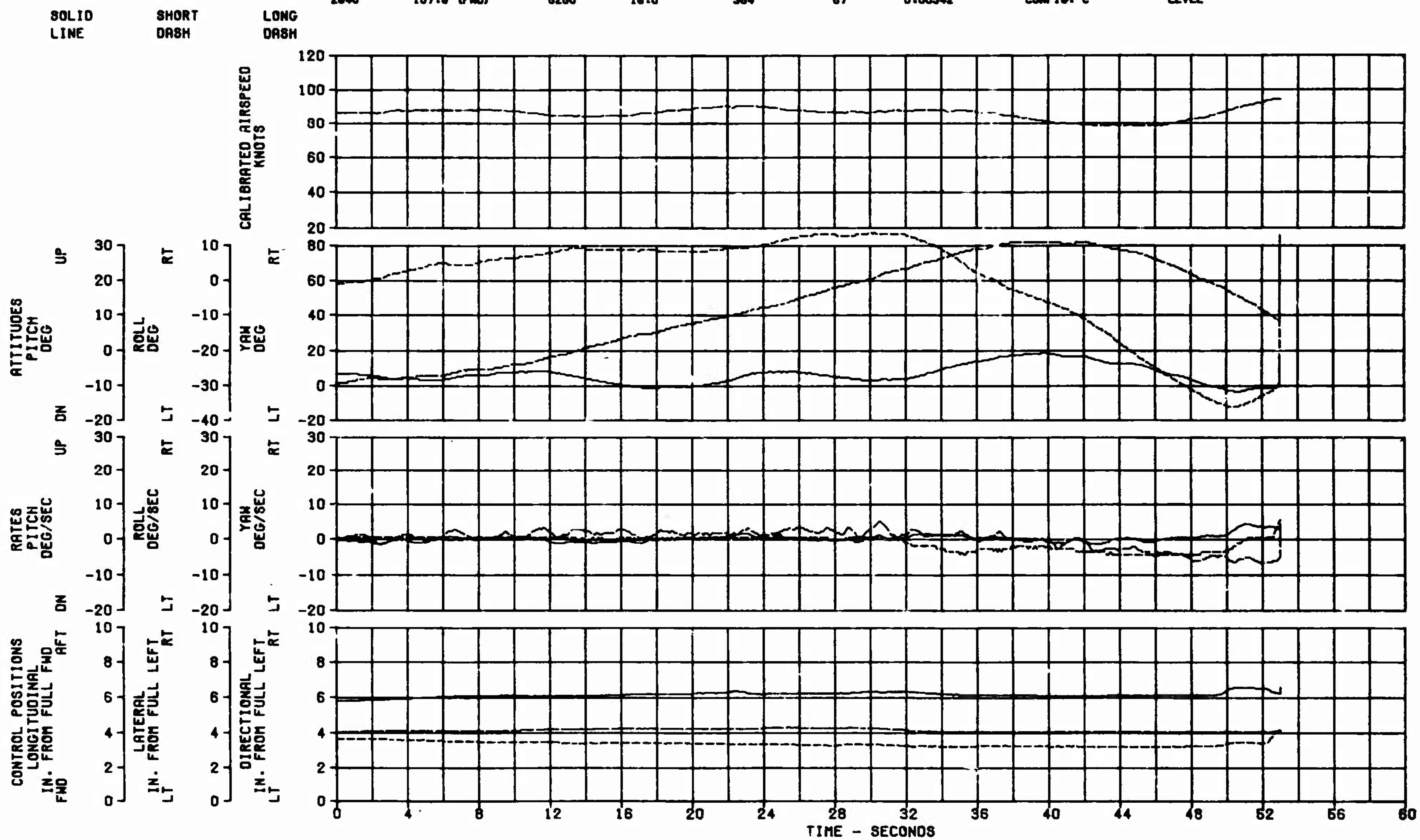


FIGURE 49
SPIRAL STABLITY
JOH-58A S/N 69-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2800	107.8 (FWD)	7220	14.0	354	87	0.00348	CONFIG. C	LEVEL

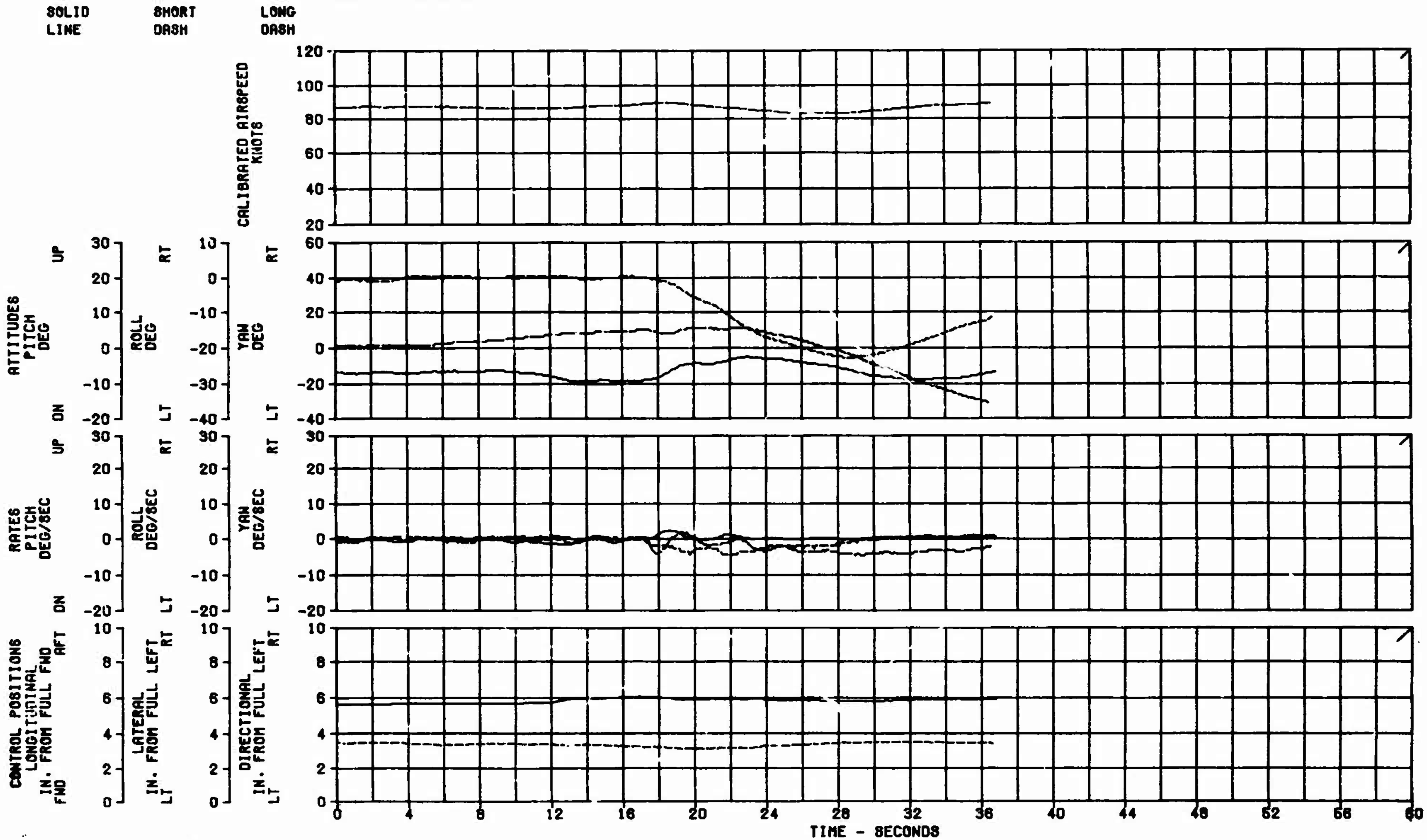


FIGURE 50

SPIRAL STABLITY

[illegible]

MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
[illegible]	107.6 (FWD)	[illegible]	14.5	[illegible]	[illegible]	0.00345	CONFIG. C	LEVEL

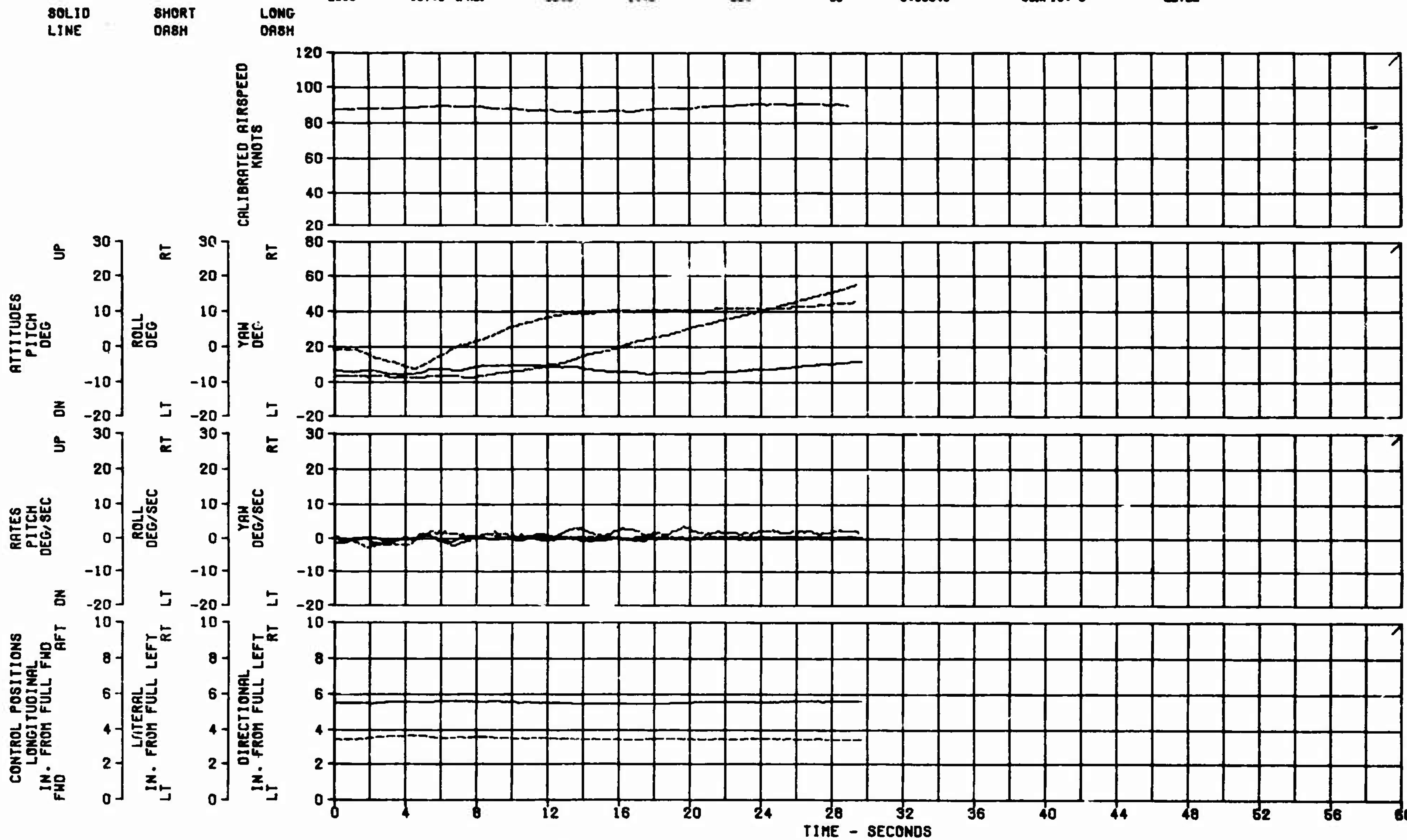


FIGURE 51
SPIRAL STABLITY
JOH-58A S/N 69-18708
MODEL T63-A-700 ENGINE

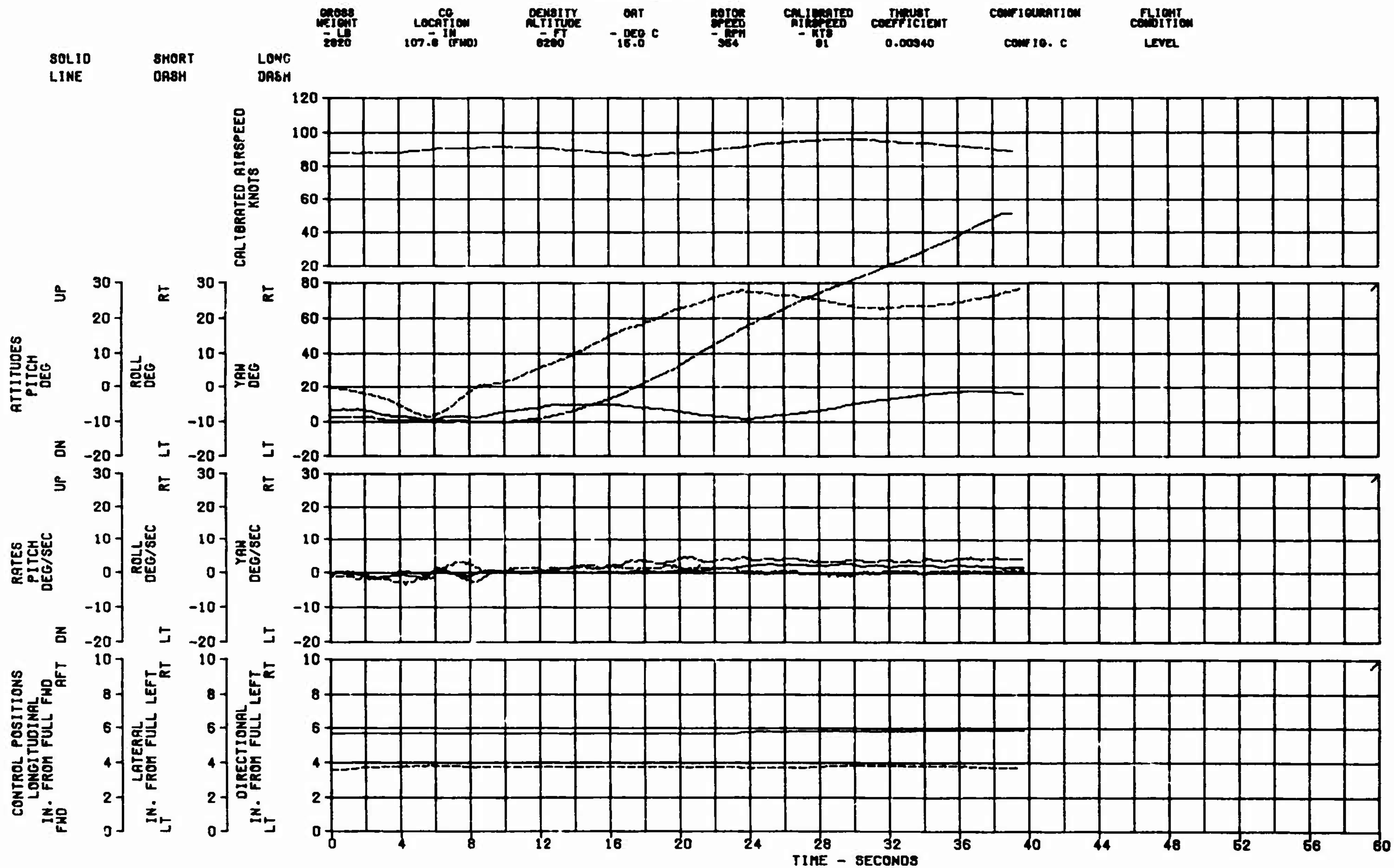


FIGURE 52

AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE

JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

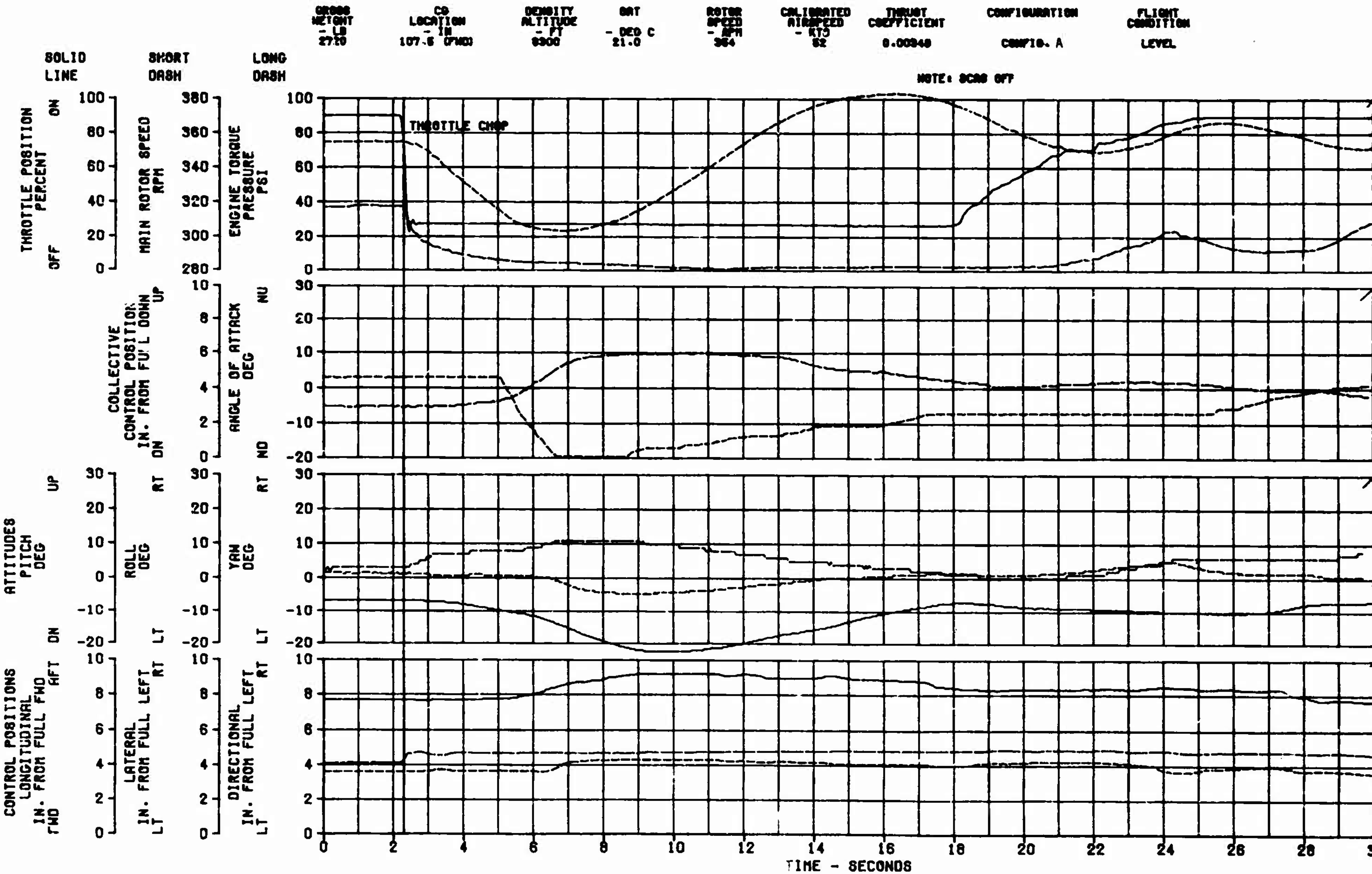


FIGURE 53
AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE
OH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

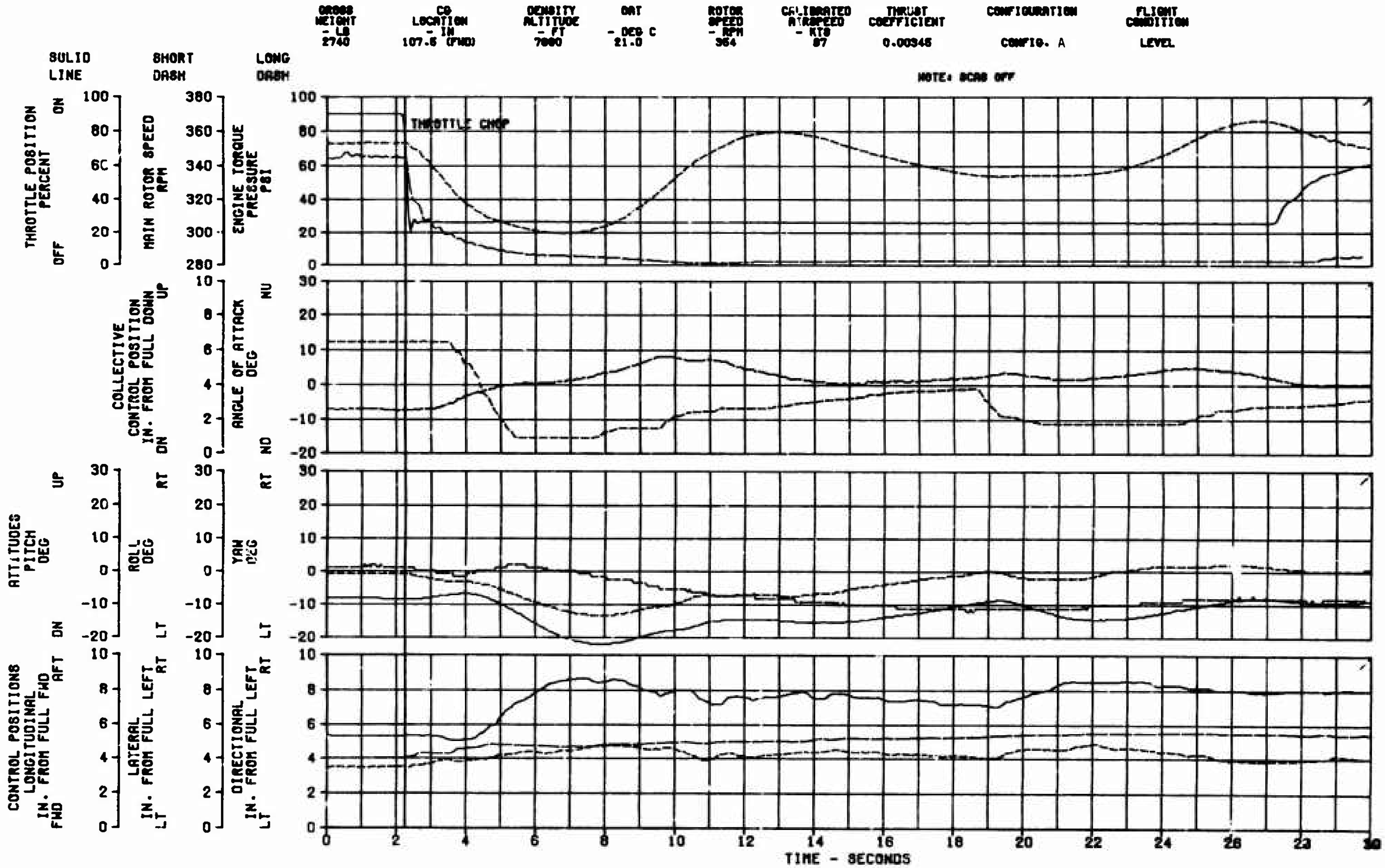


FIGURE 54
AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE
JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2860	107.5 (FWD)	5840	16.0	354	96	0.00342	CONFIG. C	LEVEL

NOTE: SCAS OFF

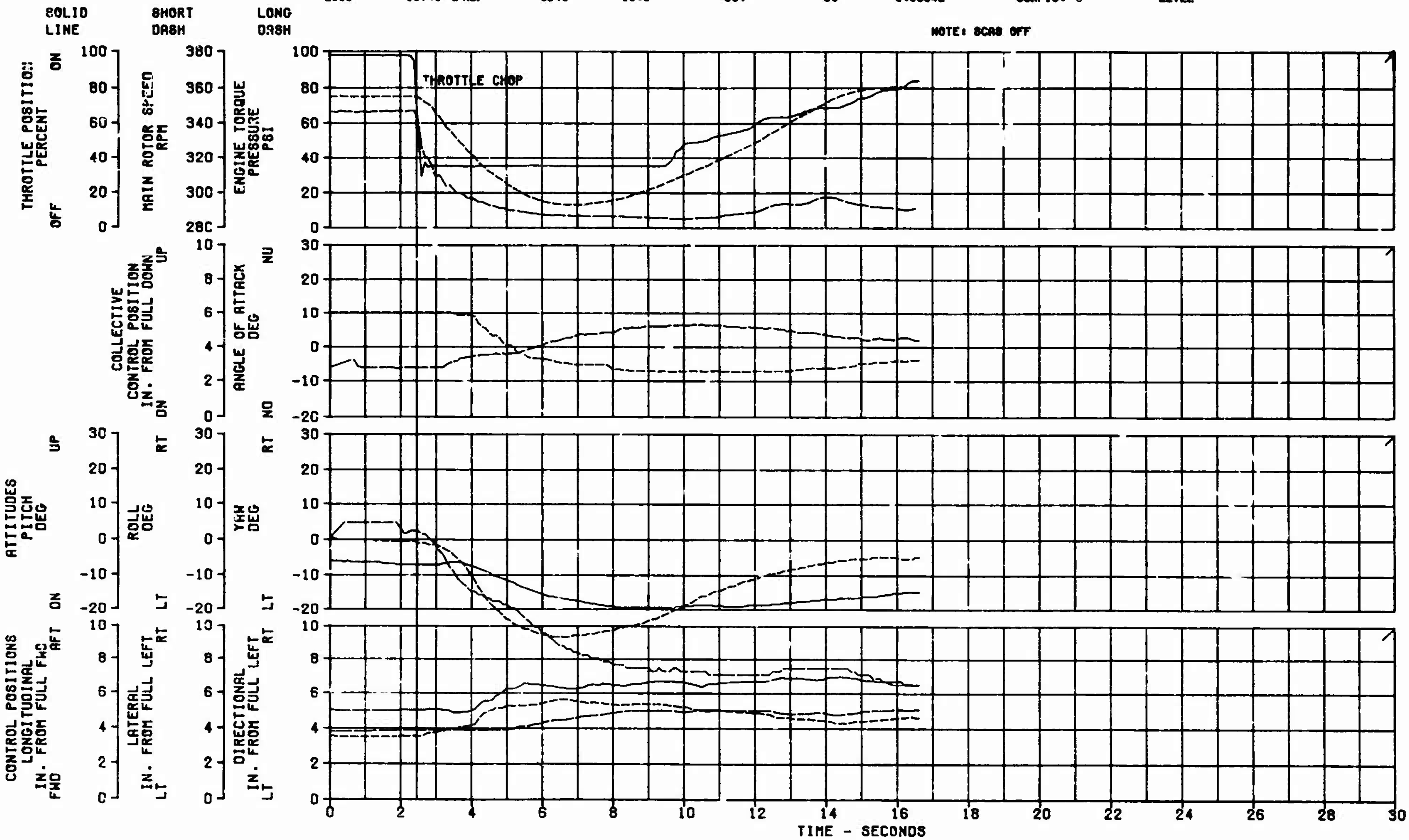


FIGURE 55

AIRCRAFT REACTION TO SIMULATED SUDDEN ENGINE FAILURE

JOH-58A S/N 68-16708
MODEL T63-A-700 ENGINE

GROSS WEIGHT - LB	CG LOCATION - IN	DENSITY ALTITUDE - FT	OAT - DEG C	ROTOR SPEED - RPM	CALIBRATED AIRSPEED - KTS	THRUST COEFFICIENT	CONFIGURATION	FLIGHT CONDITION
2960	107.6 (FWD)	5960	17.0	354	62	0.00339	CONFIG. C	CLIMB

NOTE: SCAS OFF

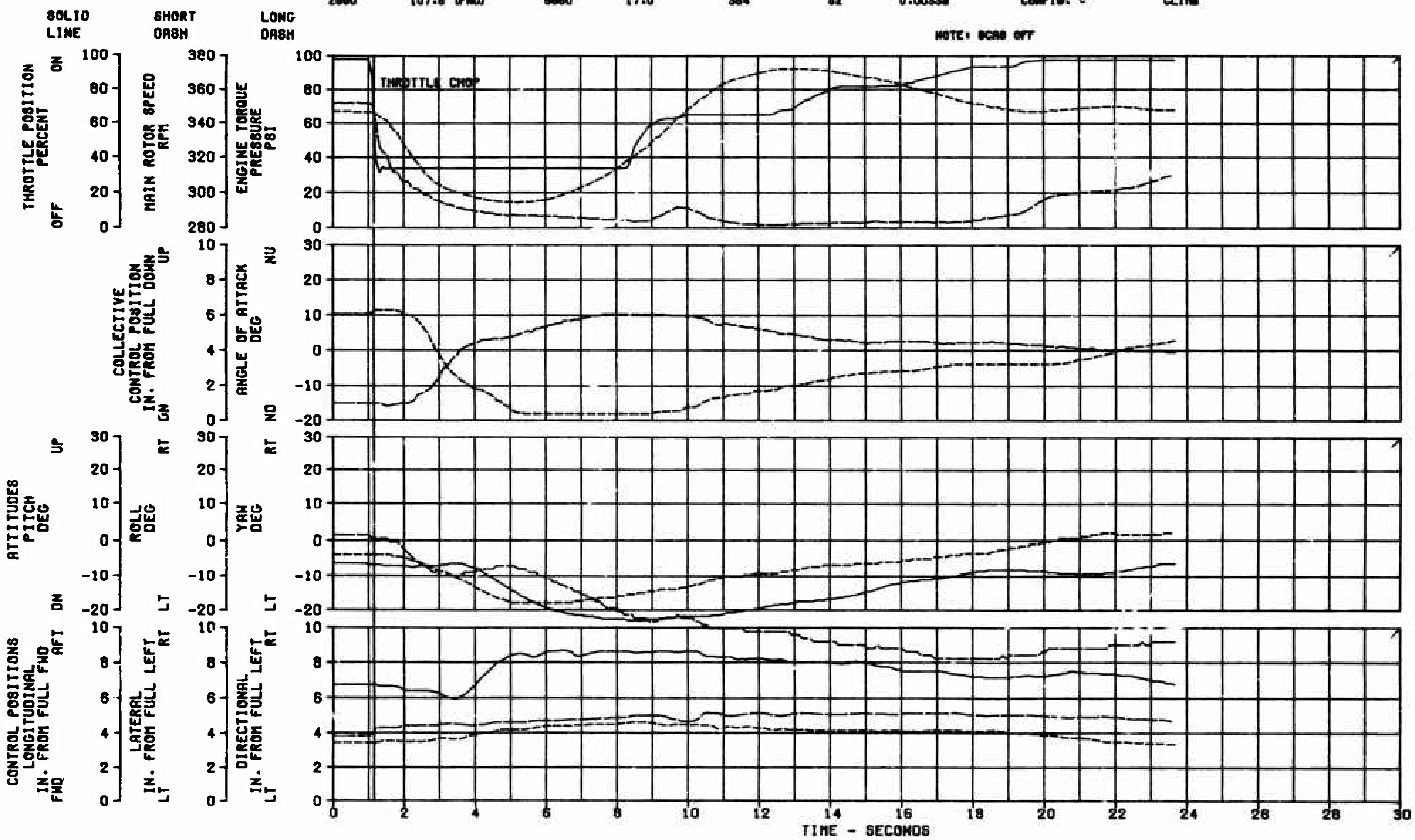
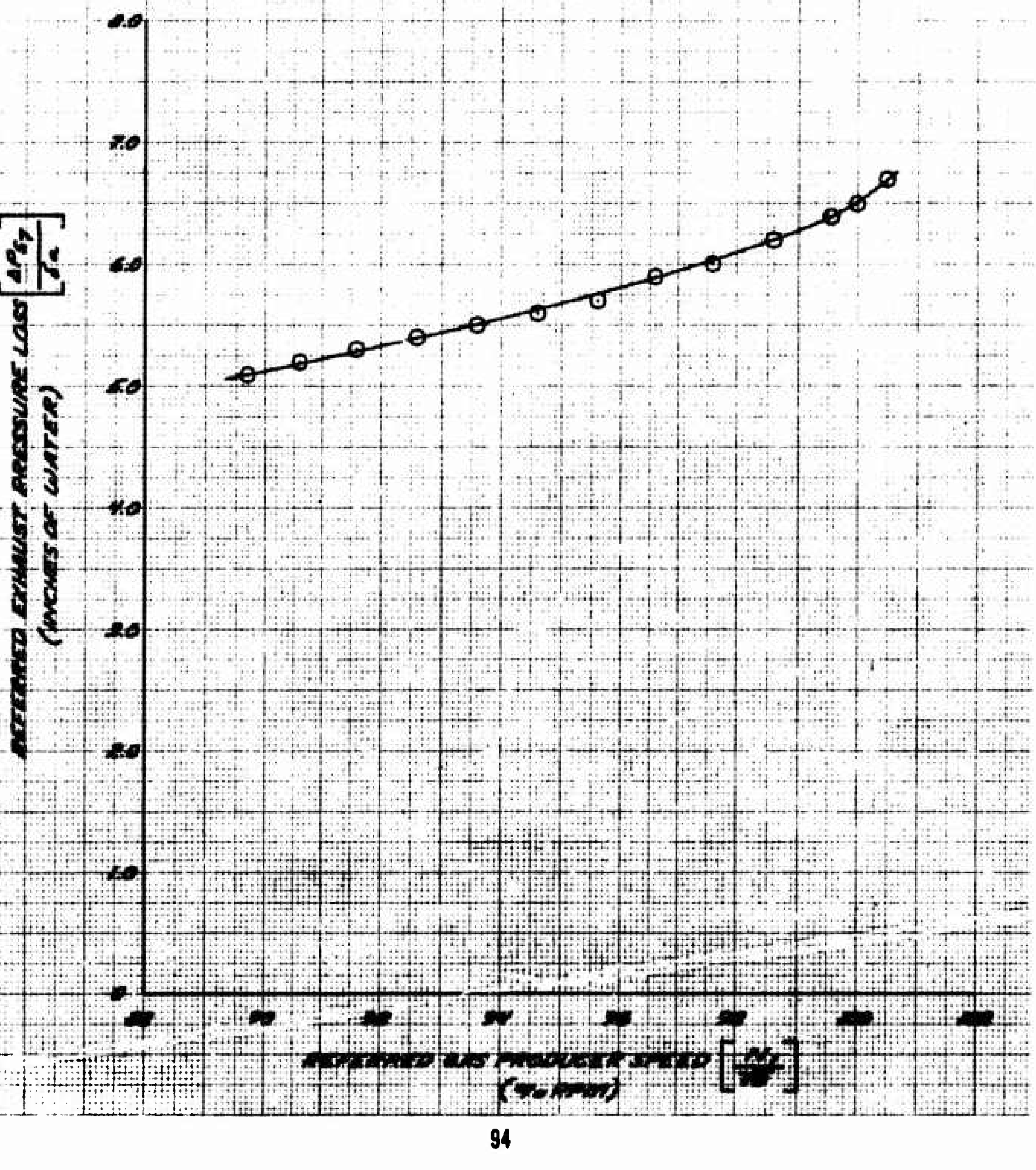
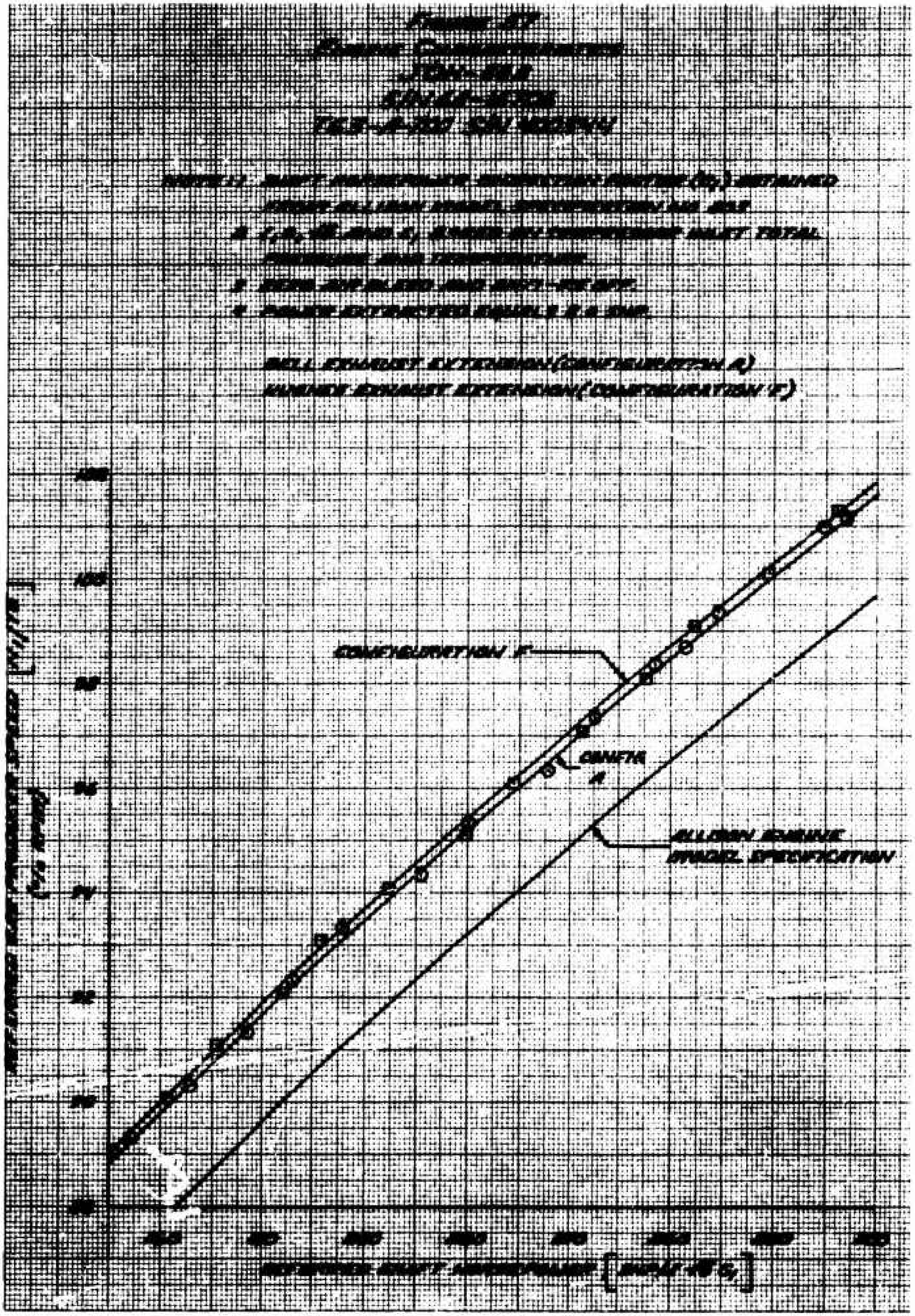
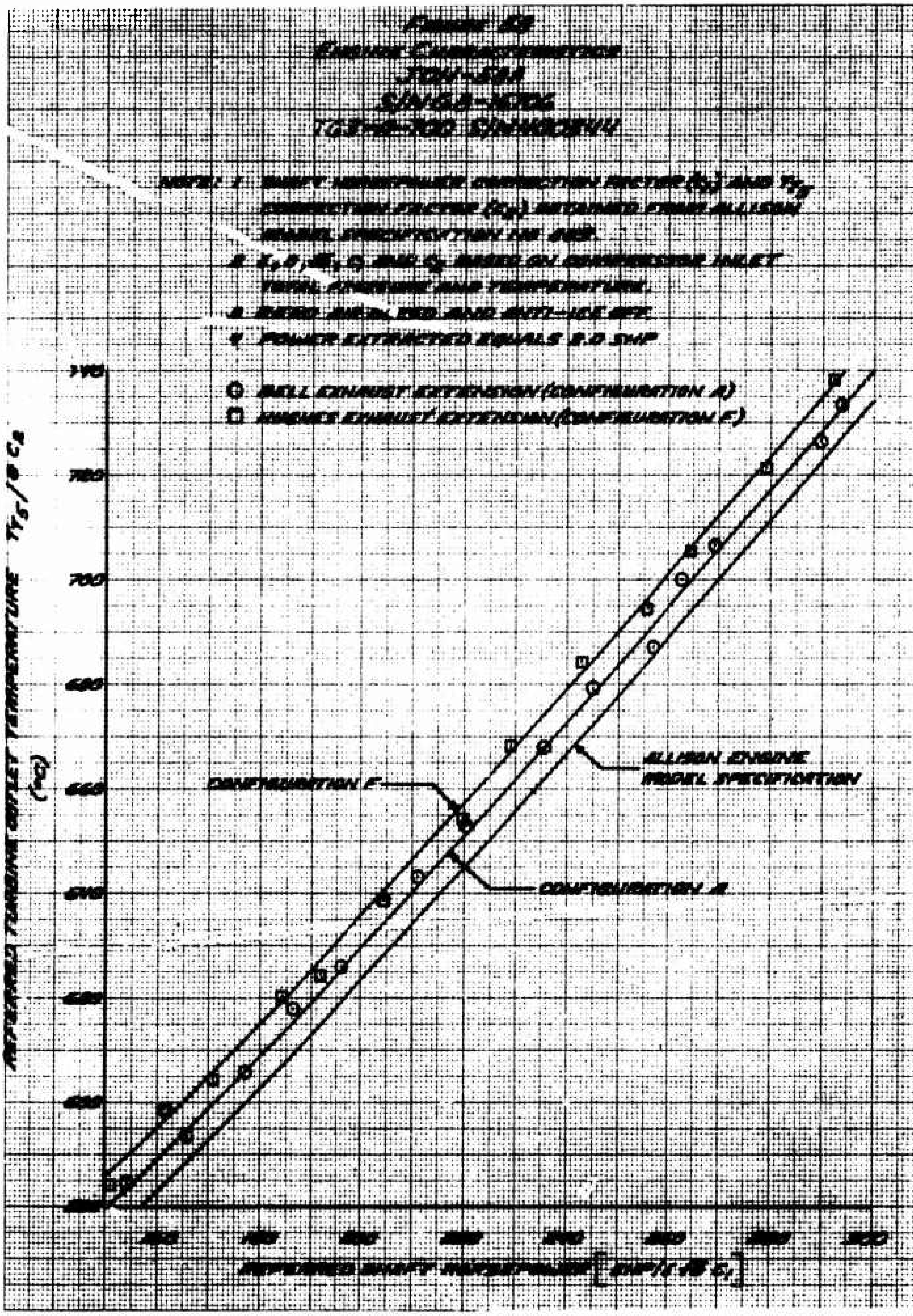
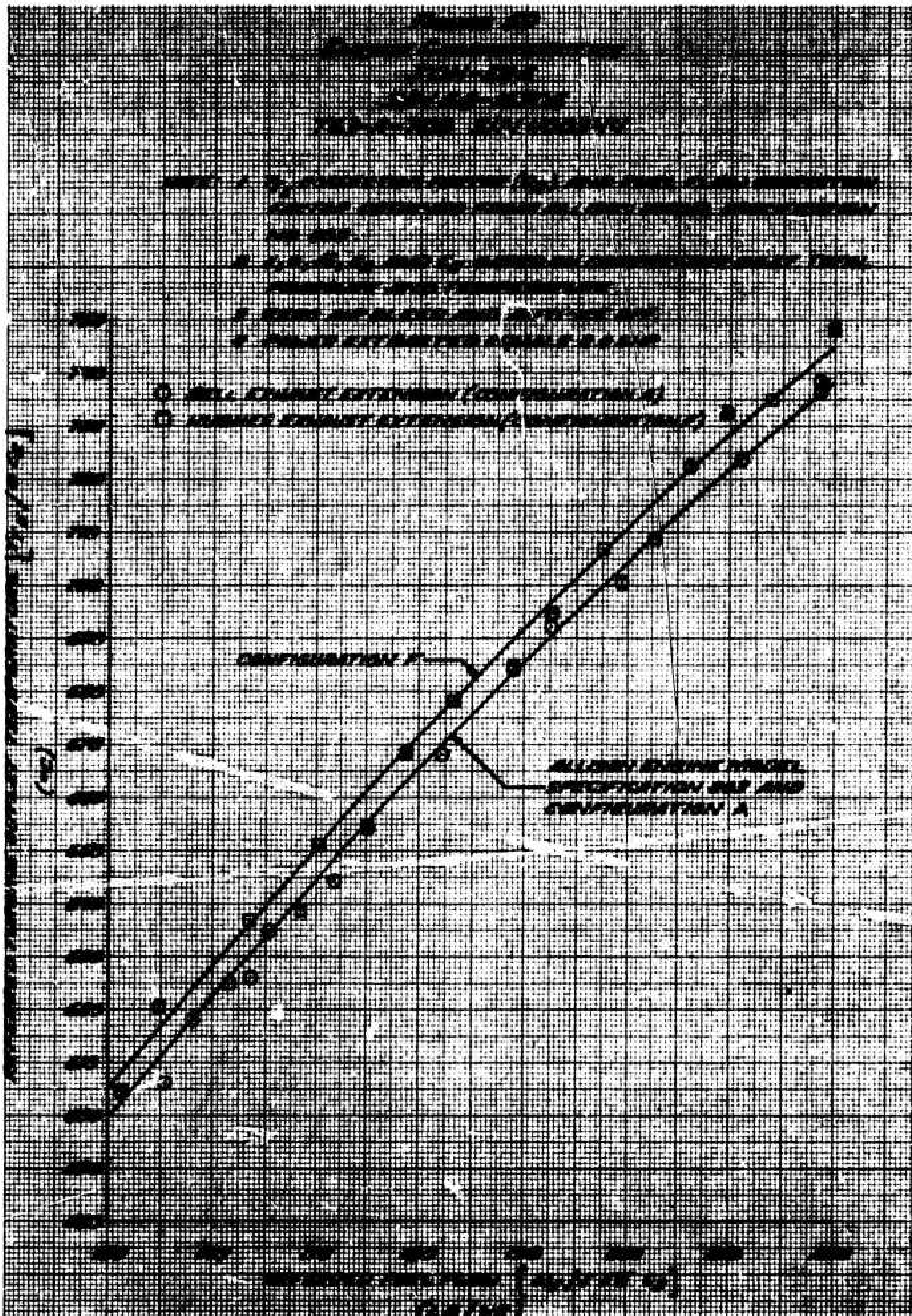


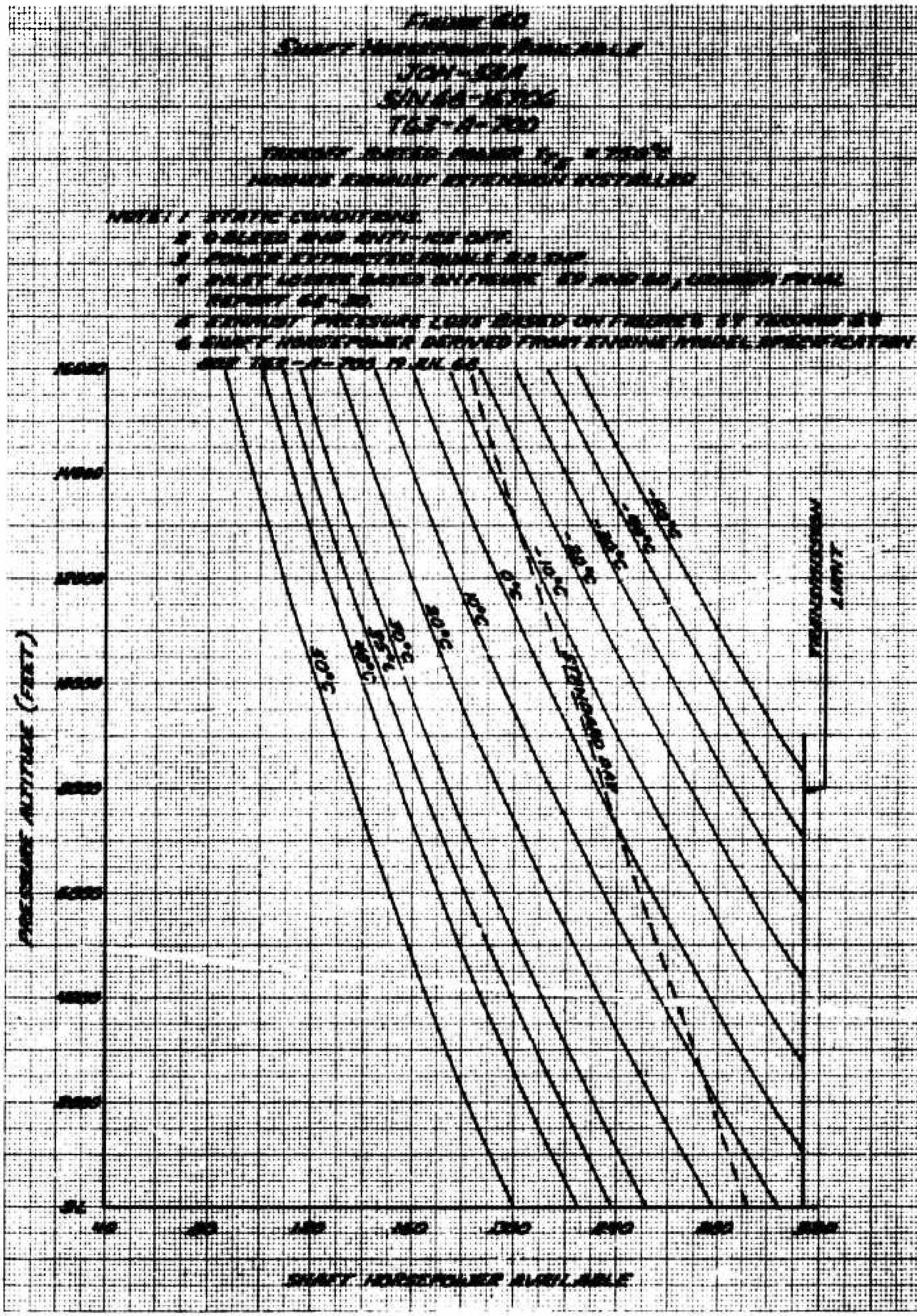
FIGURE 56
EFFECT OF HUGHES'S EXHAUST EXTENSION
ON T63-A-700 EXHAUST STATIC PRESSURE
JOH-58A
S/N 68-16706

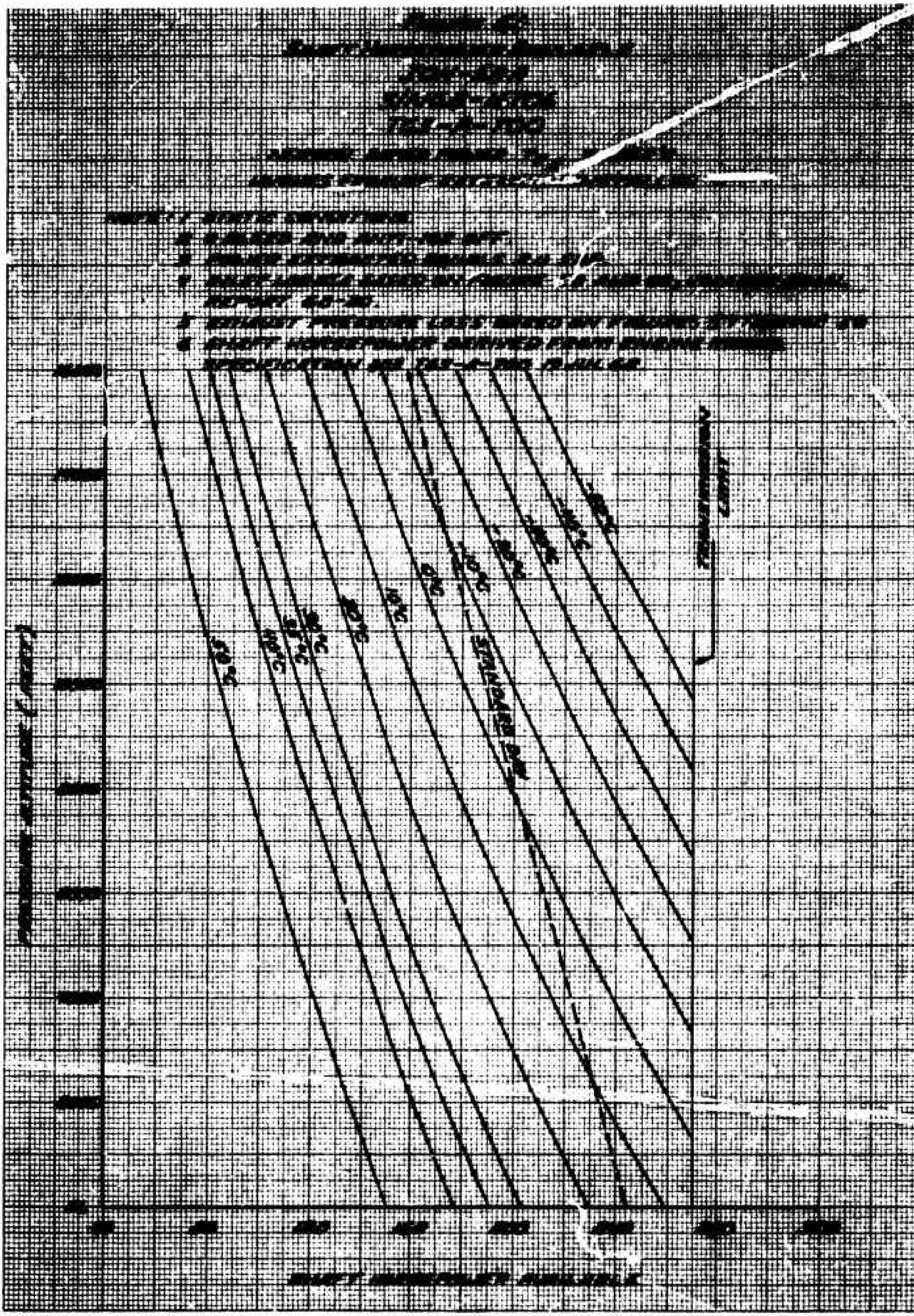



CONFIGURATION F
CONFIG A
ALLISON ENGINE MODEL SPECIFICATION


CONFIGURATION F
ALLISON ENGINE MODEL SPECIFICATION
CONFIGURATION A




T63-A-700
STATIC CONDITIONS
PRESSURE ALTITUDE (FEET)
SHAFT HORSEPOWER AVAILABLE
TEMPERATURE LIMIT
50°C
30°C
20°C
10°C
0°C
-10°C
-20°C
-30°C



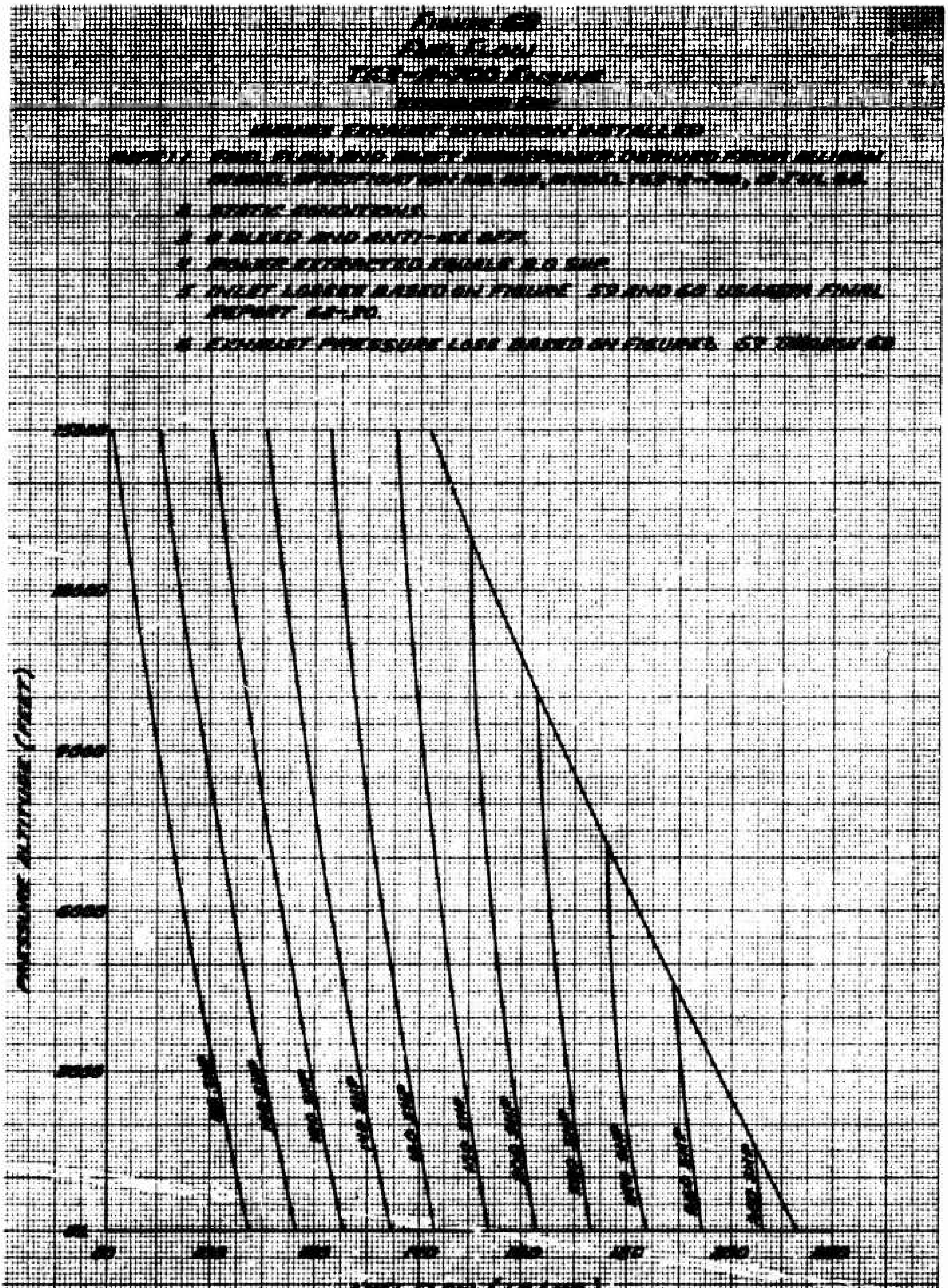

STATIC CONDITIONS.
INLET LOSSES BASED ON FIGURE
EXHAUST PRESSURE LOSS BASED ON FIGURES
PRESSURE ALTITUDE (FEET)
FUEL FLOW (LB/HR)

FIGURE 63
T63-A-700 ENGINE TEMPERATURE SURVEY (°C) WITH HUGHES EXHAUST EXTENSIONS INSTALLED

TEMPERATURE PROBE LOCATION	FLIGHT CONDITION							MAXIMUM PERMISSIBLE OPERATING TEMPERATURE
	MAXIMUM POWER CLIMB			LEVEL FLIGHT			500 FPM DESCENT	
---	51 (KCAS)	51 (KCAS)	51 (KCAS)	52 (KCAS)	52 (KCAS)	87 (KCAS)	49 (KCAS)	---
Compressor section	76 (84)	70 (79)	69 (79)	76 (85)	75 (85)	61 (68)	28 (32)	135
Gearbox section	42 (46)	40 (45)	38 (44)	35 (39)	36 (41)	39 (44)	46 (53)	121
Turbine and combustor section	78 (86)	71 (80)	78 (89)	80 (90)	78 (88)	81 (91)	72 (83)	232
Top engine mount pad surface	95 (104)	85 (96)	90 (103)	85 (95)	86 (95)	90 (101)	71 (81)	160
Ignition harness surface	79 (87)	58 (65)	59 (68)	70 (78)	61 (69)	56 (63)	58 (66)	232
Thermocouple harness surface	127 (140)	112 (126)	112 (128)	120 (134)	111 (126)	118 (132)	100 (115)	315
Oil cooler temperature (Exit)	76 (84)	76 (86)	76 (87)	75 (84)	75 (85)	75 (84)	72 (83)	107
Oil cooler temperature (Inlet)	84 (92)	80 (90)	78 (89)	79 (88)	77 (87)	78 (87)	73 (84)	
Maximum ambient temperature	22.5	15.0	10.5	17.5	13.5	17	10.5	52
Pressure altitude (feet)	2520	5840	7960	4320	6320	4280	7780	---

NOTE: Values in parentheses are correct.