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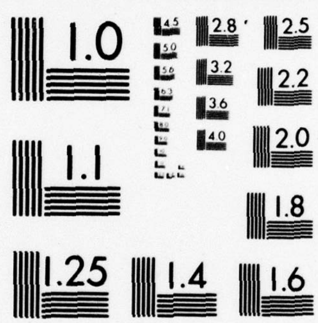

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PRELIMINARY AIRWORTHINESS EVALUATION AH-1S HELICOPTER EQUIPPED WITH A GARRETT INFRARED RADIATION SUPPRESSOR AND AN/ALQ-144 JAMMER

FINAL REPORT

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MAY 1978

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Production AH-1S helicopter
Garrett AiResearch infrared radiation suppressor
Limited performance and handling qualities tests

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

The United States Army Aviation Engineering Flight Activity conducted a Preliminary Airworthiness Evaluation of a production AH-1S helicopter equipped with a Garrett AiResearch Manufacturing Company infrared radiation suppressor (IRS). The evaluation consisted of limited performance and handling qualities tests and was conducted between 9 and 30 March at the Garrett AiResearch Manufacturing Company facility at Sky Harbor Airport, Phoenix, Arizona. The → over

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20. ABSTRACT

tests were conducted to determine the effects of installation of the IRS and the AN/ALQ-144 infrared jammer. There was no apparent effect on power required to hover and only a small increase in power required to maintain level flight. Because of a power-available degradation, however, there was a loss of 220 pounds payload in an out-of-ground-effect hover at 4000 feet, 35°C. Additionally, at 2000 feet, 25°C, and 9300 pounds gross weight, there was a reduction in maximum level flight airspeed of 3.5 knots true airspeed in the clean wing configuration. No degradation in handling qualities was caused by IRS or infrared jammer installation. However, one deficiency and six shortcomings inherent in the basic AH-1S were noted. The nonadjustable ventilation system is a deficiency because it blows dirt and dust into the pilot's eyes during operations from unprepared surfaces.

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

DRDAV-EQ

21 DEC 1978

SUBJECT: USAAEFA Project No. 77-33, Preliminary Airworthiness Evaluation, Production AH-1S Helicopter Equipped with the Garrett Hot Metal Plus Plume IR Suppressor System (P/N 19140FSCM70210) and the AN/ALQ-144 IR Jammer, May 1978

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1. The purpose of this letter is to present the Directorate for Development and Engineering position on the subject report. In general, we agree with the report and specifically the performance penalties with the Infrared Suppressor installed. For the purpose of steady state performance measurements, the data acquisition system used is considered adequate (paragraph 7). If it had not been, the Contractor would not have been allowed to proceed with test efforts. Extensive time histories of flying qualities type data could not be obtained with this system; however, since the static stability and control characteristics with the IR Suppressor installed were unchanged from other AH-1S data, extensive quantitative dynamic stability testing was not needed or intended. It must further be pointed out that the incorporation of the IR Suppressor in the AH-1S permits the removal of 20 pounds of ballast, which is not credited in the AEFA analysis of a 220 pound out of ground effect hover performance penalty. Considering the ballast removed, the hover out of ground effect payload penalty is 200 pounds which meets the 203 pound specification value (Reference 4).

2. The deficiency listed in paragraph 40 has been corrected by the addition of adjustable air vents starting with the #67 Production AH-1S, S/N 77-22729.

3. No action is planned concerning the shortcomings listed in paragraph 41. They are not unique to the IR Suppressor installation and have not been considered as significant problems with AH-1S aircraft already deployed.

4. During user type tests of the IR Suppressor numerous but minor cracks (not safety related) have occurred resulting in the Aircraft Survivability Equipment (ASE) Project Manager extending the development contract in order to provide the field with more durable items. Since planned improvements will not effect the exterior configuration or engine power losses, additional testing by AEFA is not being considered. While specific Infrared radiation

DRDAV-EQ
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specifications were not a part of this test and their actual values are classified, it is important to take note that the suppression objectives of this program have been met.

FOR THE COMMANDER:

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

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INTRODUCTION

BACKGROUND

1. The Aircraft Survivability Equipment (ASE) Project Manager's Office has contracted with the Garrett AiResearch Manufacturing Company (GAMC) to design, develop, and test an infrared radiation suppressor (IRS) for the production AH-1S helicopter. The United States Army Aviation Research and Development Command (AVRADCOM) directed the United States Army Aviation Engineering Flight Activity (USAAEFA) to conduct a Preliminary Airworthiness Evaluation (PAE) of the AH-1S with the IRS installed, (ref 1, app A) in accordance with the approved test plan (ref 2).

TEST OBJECTIVES

2. The objectives of the PAE were as follows:

a. Determine any degradation in aircraft performance and handling qualities caused by the IRS and the AN/ALQ-144 infrared (IR) jammer.

b. Provide data for revisions to the AH-1S operator's manual (ref 3, app A) as necessary.

DESCRIPTION

3. The AH-1S is a tandem-seat, two-place helicopter with a two-bladed teetering main rotor and a two-bladed tractor tail rotor. The helicopter is powered by a Lycoming T53-L-703 turboshaft engine derated from 1800 shaft horsepower (shp) at sea-level, standard-day conditions, to the main transmission torque limit of 1290 shp. Distinctive features of the helicopter include the narrow fuselage, stub wings with four stores stations, a flat-plate canopy, and an IRS exhaust system. A more detailed description of the AH-1S is presented in appendix B and in the operator's manual.

4. The IRS system consists of three major components: the suppressor, an engine exhaust duct assembly, and an aft engine cowl. The IRS is supported by four hollow struts inside the engine exhaust pipe and prevents line-of-sight visibility of hot engine parts. The engine exhaust duct is of much larger diameter than the standard duct to accommodate the IRS. This larger diameter necessitates replacement of the standard AH-1S aft engine cowling. The GAMC IRS system provides accommodations for mounting an AN/ALQ-144 IR jammer. Further description of the IRS system is available in the Critical Item Development Specification (CIDS) (ref 4, app A) and in appendix B.

TEST SCOPE

5. The PAE was conducted between 9 and 30 March 1978 at the GAMC facility at Sky Harbor Airport, Phoenix, Arizona, using a production AH-1S (S/N 76-22573). Twenty-six test flights were conducted for a total flight time of 30 hours, of which 21.4 hours were productive. The aircraft and instrumentation system were maintained by the contractor during this evaluation. The aircraft was tested for compliance with military specification MIL-H-8501A (ref 5, app A), the detail specification (ref 6), and the IR suppressor CIDS. Flight restrictions of the operator's manual and the airworthiness release (ref 7) were observed throughout the test program.

6. The level flight performance tests were flown with no wing stores installed and at a forward center-of-gravity (cg) location. All the handling qualities tests, except dynamic stability, were flown in the 8-TOW configuration. All handling qualities tests except dynamic stability and low-speed flight characteristics were flown at an aft cg location. All tests were flown with the IRS installed except for one dynamic stability flight. Performance data were gathered by the contractor on the AH-1S with the standard exhaust system installed. These data were used as a base line for comparison where possible. Table 1 presents the general test conditions.

7. The data generated during this PAE (and during the contractor base-line tests which preceded it) are of questionable validity because of limitations of the airborne data acquisition system. The system provided was not well suited to the type of experimental measurements required in this program. The primary shortcoming of the system was an inadequate data sampling rate. A more complete description of the data acquisition system is presented in appendix C.

TEST METHODOLOGY

8. The flight test techniques and data analysis methods are described in appendix D. A digital instrumentation system was used to measure and record the parameters listed in appendix C. A display in the front cockpit allowed the flight test engineer to display any eight parameters at one time. A data sample rate of two measurements per second was used for hover performance and low-speed flight characteristics tests because of the oscillatory nature of some parameters during these tests. However, at this sample rate engine static pressures could not be sampled. Therefore, a lower sample rate was used during level flight performance tests. During the level flight test, three samples were taken (approximately 30 seconds apart) at each flight condition. These three samples were averaged to define one data point.

Table 1. General Test Conditions.[1]

Test	Density Altitude (ft)	Gross Weight (lb)	Center-of-Gravity Location (FS)	Configuration	Trim Calibrated Airspeed (kt)
Hover performance[2]	1200	Low	Mid	Clean	Zero
Level flight performance[3]	2000 to 12,000	Low to high	Fwd	Clean	40 to V_H[4]
Static longitudinal stability[5]	6000	High	Aft	8-TOW	60, 120, 140
Static lateral-directional stability	6000	High	Fwd	Clean	120[6]
			Aft	8-TOW	60,[7] 120
Dynamic stability[8]	6000	High	Fwd	Clean	60 to 140
Engine failures[9]	5000	High	Aft	8-TOW	60 and 120
Low-speed flight characteristics	1860	High	Fwd	8-TOW	Zero to 35[10]

[1]All tests were conducted with IRS installed except for one dynamic stability flight.
[2]Tests were conducted at a 15-foot skid height.
[3]Five values of thrust coefficient were flown with the IRS alone, and three with the IRS and the AN/ALQ-144.
[4]V_H: Maximum airspeed for level flight.
[5]Tests were conducted with the IRS installed and with both the IRS and the AN/ALQ-144 installed.
[6]Tests conducted with IRS installed and with standard tail pipe installed.
[7]Tests were conducted in level flight and autorotation.
[8]Tests were conducted with stability and control augmentation system (SCAS) ON and OFF.
[9]Level flight at both airspeeds, climb at 60 knots.
[10]Knots true airspeed (KTAS).

RESULTS AND DISCUSSION

GENERAL

9. Limited performance and handling qualities tests were performed on an AH-1S helicopter with a GAMC IRS installed. Some tests were flown with and without an AN/ALQ-144 IR jammer installed. The tests were conducted to determine the effects, if any, of installation of the IRS and IR jammer on the AH-1S. There was no apparent effect on power required to hover and only a small increase in power required to maintain level flight. Because of a power available degradation, however, there was a reduction of 220 pounds payload in an out-of-ground effect (OGE) hover at 4000 feet, 35°C. Additionally, at 2000 feet, 25°C, and 9300 pounds gross weight, there was a reduction in maximum level flight airspeed of 3.5 KTAS in the clean wing configuration. No degradation in handling qualities was caused by IRS or IR jammer installation. However, one deficiency and six shortcomings inherent to the basic AH-1S were noted.

PERFORMANCE

Hover Performance

10. The hover performance of the IRS equipped AH-1S helicopter was evaluated at the general conditions listed in table 1. A 15-foot skid height was used during the tethered hover tests. The test techniques and data analysis methods used are described in appendix D.

11. Nondimensional hover performance of the AH-1S with the IRS installed is presented in figure 1, appendix E. This performance is almost identical to the performance of the AH-1G aircraft presented in reference 8, appendix A. It also agrees quite well with the performance presented in the AH-1S operator's manual. Data gathered by GAMC with the standard tail pipe installed indicated much better hover performance capability than any previous AH-1 tests. Therefore, the GAMC data are assumed to be in error and have not been used for computing performance degradation. Because the AH-1S/IRS data agree with previous AH-1 data with the standard tail pipe, it is assumed that installation of the IRS causes no increase in power required to hover.

12. The CIDS specifies a maximum payload loss caused by IRS installation of 203.4 pounds for OGE hover at a pressure altitude of 4000 feet and ambient air temperature of 35°C. Power available at these ambient conditions is 1133 shp with the IRS and 1163 shp with the standard tail pipe (figures provided by AVRADCOM, ref 9, app A). Based on these two figures and the OGE performance presented in reference 8, the lifting capability of the AH-1S is reduced 169 pounds, from 9045 to 8876 pounds by installation of the IRS. The payload is further reduced by the aircraft empty weight increase caused by the IRS (51 pounds).

Therefore, the total payload penalty is 220 pounds. At a 15-foot hover height and the same ambient conditions, the total payload loss is 229 pounds.

Level Flight Performance

13. The level flight performance of the IRS equipped AH-1S helicopter was evaluated at the general conditions listed in table 1. Eight level flight performance flights were made with the IRS installed and on three of these flights the AN/ALQ-144 IR jammer was installed. The data were compared to contractor-furnished data obtained with the standard AH-1S tail pipe installed.

14. All level flight performance was flown at zero sideslip, while maintaining constant ratios of gross weight to air pressure ratio (W/δ) and main rotor speed to the square root of temperature ratio ($N/\sqrt{\theta}$). The results of these tests are presented in appendix E, nondimensionally in figures 2 through 4 and dimensionally in figures 6 through 14.

15. The change in equivalent flat plate area (Δf_e) between the standard tail pipe and the IRS was 1/2 square foot (propulsive efficiency of one assumed). Figure 5, appendix E, shows standard tail pipe data fit with IRS curves corrected for Δf_e. The data at a thrust coefficient of 52.8×10^{-4} appeared inconsistent with the rest of the data set and were not used when determining Δf_e.

16. Figure 6, appendix E, presents level flight performance of the AH-1S with and without the IRS, at the conditions specified in the CIDS. At these conditions, V_H was reduced from 142.5 to 139 KTAS by the IRS installation. Power-available data were provided by AVRADCOM (ref 9, app A). This 3.5 knot reduction in V_H equals the 3.5 knot reduction allowed by the CIDS.

17. Figures 12 through 14, appendix E, present data obtained with both the IRS and the AN/ALQ-144 IR jammer installed. No measurable Δf_e was found with installation of the AN/ALQ-144.

HANDLING QUALITIES

Control System Characteristics

18. The longitudinal cyclic mechanical characteristics of the AH-1S were evaluated on the ground with the rotors stopped using ground hydraulic power. A plot of force versus displacement is presented in figure 15, appendix E. A wide trim control displacement band was measured during the ground tests and qualitatively noted in flight. This band made it difficult to return to an original longitudinal cyclic trim control position once the control was displaced from trim. Lack of absolute longitudinal cyclic control centering is a shortcoming. The longitudinal cyclic breakout characteristics failed to meet the provisions of the detail specification in that breakout force in the aft direction was 2.5 pounds

(1/4 pound greater than the permissible maximum value). Breakout in the forward direction was only 1 pound. This asymmetry in breakout was observed in flight and was objectionable. The longitudinal cyclic control failed to meet the limit force provision of paragraph 3.2.6 of MIL-H-8501A in that the limit force was 14 pounds (6 pounds greater than the permissible maximum value).

19. The lateral cyclic mechanical characteristics of the AH-1S were evaluated on the ground partially with ground hydraulic power and partially with rotors turning, engine at flight idle. The change in means of furnishing hydraulic power was necessitated by failure of the ground hydraulic power unit. A plot of force versus displacement is presented in figure 16 appendix E. A wide trim control displacement band of 0.72 inch was noted on the ground. This band caused lateral cyclic trim difficulties in flight. When the cyclic was displaced laterally from trim it would not return to the original trim position. Lack of precise centering is a shortcoming. The lateral cyclic breakout characteristics failed to meet the provisions of the detail specification in that the breakout force to the right was 3.3 pounds (1.05 pounds greater than the permissible maximum value). The lateral cyclic control failed to meet the limit force provision of paragraph 3.2.6 of MIL-H-8501A, in that the limit force was 19 pounds (8 pounds permissible).

Control Positions in Trimmed Forward Flight

20. Control positions in trimmed forward flight were evaluated in conjunction with level flight performance tests at the general conditions listed in table 1. Test data are presented in figures 17 and 18, appendix E, for both the jammer-off and jammer-on configurations. The trimmed control position characteristics of the AH-1S equipped with the IRS in both the jammer-off and jammer-on configurations are satisfactory.

Static Longitudinal Stability

21. The static longitudinal stability characteristics of the AH-1S were evaluated at the conditions shown in table 1 for both jammer-on and jammer-off configurations. Test techniques are described in appendix D. The variation of control position with airspeed is presented in figures 19 and 20, appendix E. The only apparent change in static longitudinal stability caused by the IR jammer can be seen in figure 20. At airspeeds above the trim airspeed of 119 knots calibrated airspeed (KCAS), stability was nearly neutral with the jammer installed, but the pilot noticed no change in stability. Qualitatively, the static longitudinal stability of the AH-1S was not significantly affected by installation of the IRS or IR jammer.

Static Lateral-Directional Stability

22. The static lateral-directional stability of the AH-1S was evaluated at the conditions listed in table 1, using the techniques described in appendix D. The variation of control position with sideslip angle is presented in figures 21 through 24, appendix E.

23. Figures 23 and 24, appendix E, present data gathered at a trim airspeed of 119 KCAS, in four different aircraft configurations. The data indicate that no significant change in lateral-directional characteristics was caused by aircraft cg location, wing configuration, engine exhaust configuration (IRS or standard), or by installation of the IR jammer. Data in figures 21 and 22 indicate that installation of the IR jammer caused no degradation in lateral-directional stability at a nominal 65 KCAS in level flight or autorotation. Pilot comments substantiated these conclusions. The static lateral-directional stability of the AH-1S was unaffected by installation of the IRS or IR jammer.

Dynamic Stability

24. Throughout this evaluation, an annoying dutch-roll oscillation was observed. Therefore, an investigation to determine the cause of the oscillation was conducted at the conditions listed in table 1. The test was conducted by making directional doublet inputs at incrementally increasing airspeeds in level flight (SCAS ON and OFF) and dives (SCAS ON). The stability was evaluated by counting the number of cycles to damp to a specified fraction of the initial amplitude. The data system limitations precluded a quantitative definition of the damping.

25. The dutch-roll oscillation observed was characterized by an approximate 3-second period and a roll-to-yaw ratio of 1/3. The oscillation was observed during level flight performance testing with clean wing, forward cg location, and with jammer ON and OFF. It was also observed during handling qualities tests with an 8-TOW configuration at an aft cg location. Therefore, only the effects of aircraft gross weight and airspeed and of IRS installation were evaluated.

26. At light gross weight (8460 pounds) with IRS installed and SCAS ON, damping of the dutch roll was acceptable at all airspeeds tested. With SCAS OFF, the oscillation became neutrally damped at approximately 95 KCAS and was divergent at higher airspeeds. In level flight at approximatley 9440 pounds with SCAS ON and IRS installed, the dutch roll was unacceptable at airspeeds greater than 110 KCAS. At 130 KCAS the oscillation was self-excited and undamped. At 70 KCAS, SCAS OFF, the oscillation became neutrally damped, indicating that damping decreases with increasing gross weight. One flight was made at heavy gross weight with the standard AH-1 tail pipe installed. The dutch-roll characteristics were essentially unchanged from those observed with the IRS installed. During diving simulated gun runs at airspeeds up to 180 KCAS, dutch-roll oscillations were not apparent.

27. The dutch-roll characteristics at heavy gross weights above 110 KCAS level flight required continuous corrective pilot inputs to hold heading and aircraft attitude constant. This oscillation has been previously observed on the AH-1G (ref 10, app A) but is more pronounced at the heavier gross weights associated with AH-1S operation. The dutch-roll characteristics at heavy gross weights significantly degrade flight characteristics at cruise airspeeds, degrade the

effectiveness of the AH-1S as a weapons platform, and are a shortcoming. The requirements of paragraph 3.2.11 of MIL-H-8501A were not met, in that damping to one-half amplitude did not occur within four cycles, and a tendency existed for small amplitude undamped oscillations to persist.

Low-Speed Flight Characteristics

28. The low-speed flight characteristics of the AH-1S in the 8-TOW configuration were evaluated at the general conditions listed in table 1. A pace vehicle was employed for stabilized airspeed reference. The test techniques and data analysis methods used are described in appendix D. The results of the test are presented as figures 25 and 26, appendix E. Figure 26 shows that at the right sideward flight limit of 20 KTAS (gross weight 9800 pounds), only 6 percent directional control margin remained. The lack of adequate control margin at the operator's manual specified limit airspeed is a shortcoming.

Autorotational Entries

29. The aircraft response to sudden engine failure was evaluated in the 8-TOW, jammer-installed configuration at the general conditions listed in table 1. The engine failures were evaluated at 60 KCAS in level flight, and 60 KCAS in an 1100-foot per minute climb (85 percent torque). Qualitatively, the aircraft response to sudden engine failure was unaffected by the addition of the IRS and the IR jammer.

HUMAN FACTORS

Preflight Inspection

30. Preflight inspections were performed prior to each flight in accordance with the operator's manual. The sight glass on the 90-degree gearbox was not readily visible, due to its recessed location and the small cowl opening. The location of the sight glass on the AH-1S is different from the AH-1G. The oil level could not be seen without a flashlight or a ladder (photo A). This item is a mandatory flight inspection item. The difficulty in checking the 90 degree gearbox oil level due to cowling design is a shortcoming.

Cockpit Evaluation

31. The cockpit was evaluated throughout the test as a normal portion of each flight. During engine start, difficulties were encountered in visually locating the battery, generator, and inverter switches. These switches were located directly below the collective lever control head and were obscured by it (photo B). These switch locations are different from the AH-1G switch locations. The pilot could not see these switches without releasing his harness lock and leaning far forward, a position from which he could not fly the aircraft. Emergency procedures such as engine

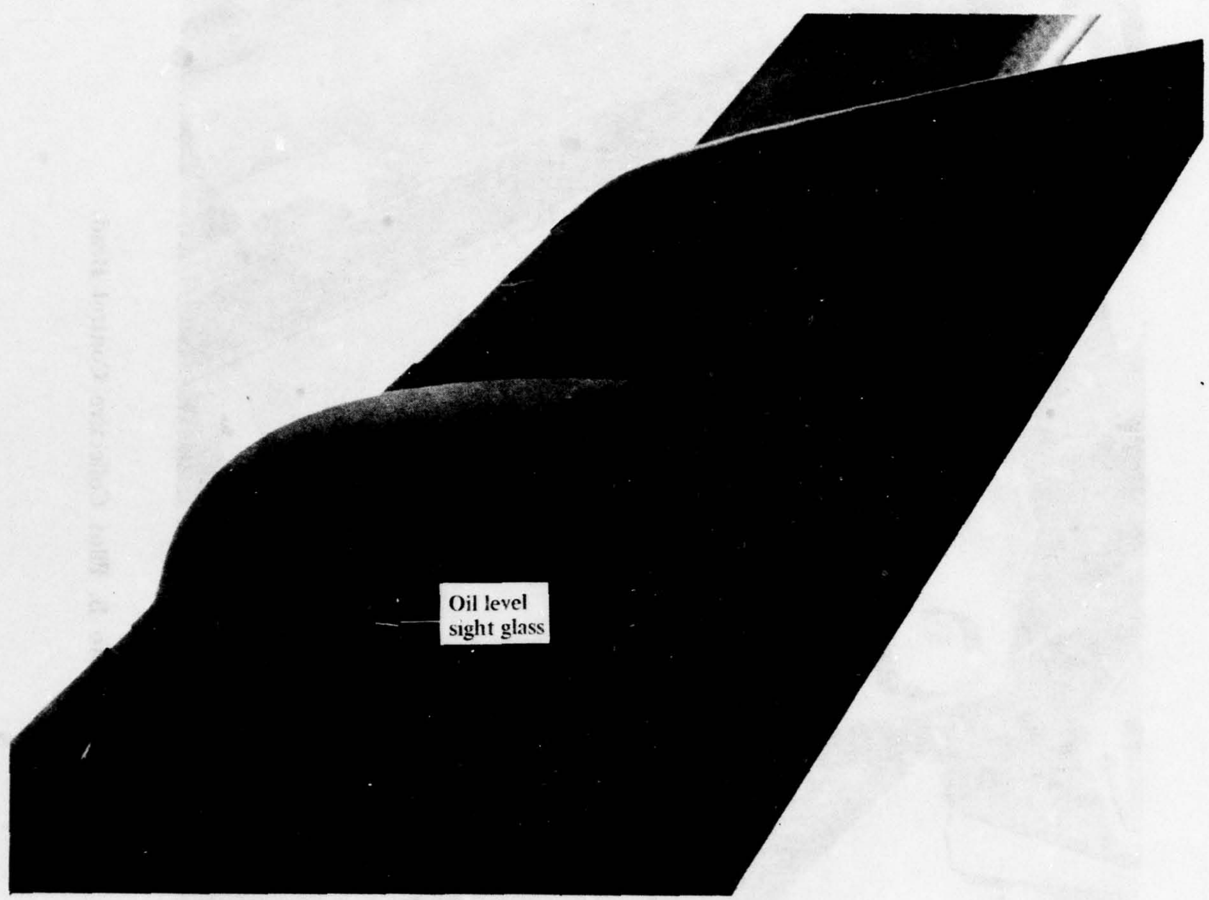


Photo A. 90-Degree Gearbox Cowl.

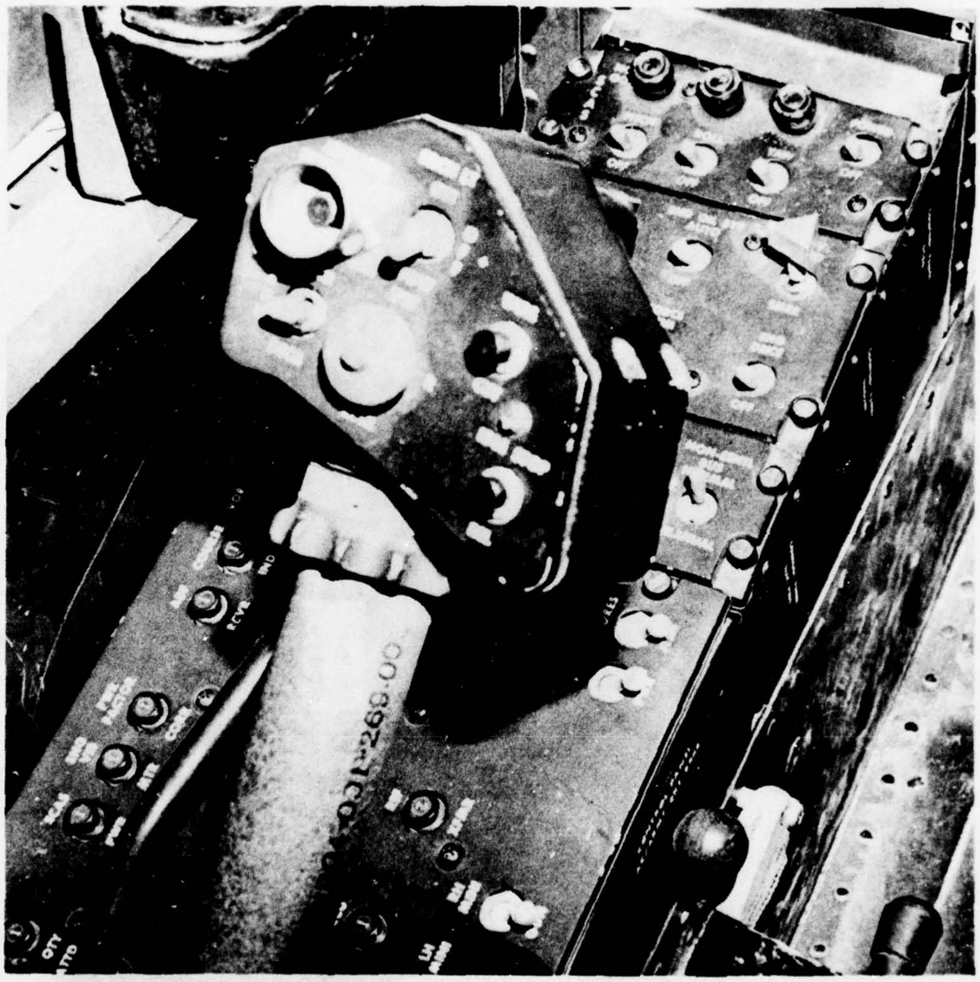


Photo B. Pilot Collective Control Head.

failure, fuselage fire, electrical fire, generator failure, and inverter failure require correct and rapid identification and actuation of the hidden switches. The inability to see the battery, generator and inverter switches from the pilot station is a shortcoming.

32. The pilot station ventilation system on the production AH-1S consists of a main vent control knob and two rectangular vent outlets located on the left and right sides of the instrument console. This ventilation system is different from the AH-1G system. Vent outlets are nonadjustable, having fixed vanes which direct airflow to the pilot's face and chest. Seperate on/off vent controls are provided at each outlet, but are poorly designed and easily damaged, as evidenced by the broken right side control on the test aircraft (photo C). When hovering over unprepared surfaces, the dust and dirt raised by the rotor downwash is readily ingested into the aircraft vent system and forcibly ejected onto the pilot's face and chest. Even with a properly fitted helmet and with the visor full down, sufficient dust and dirt can be blown into the pilot's eyes to cause temporary blindness and eye damage with the potential for loss of visual reference and aircraft damage. Operation from unprepared surfaces is an integral portion of the AH-1S mission. The present nonadjustable ventilation system is a deficiency and should be modified as soon as possible.

SUBSYSTEM TESTS

Engine Performance

33. No attempt was made to experimentally determine engine performance degradation caused by IRS installation. Such a determination has been made by GAMC using an engine test stand. However, exhaust gas static pressure (PS7) data were gathered during the level flight performance test and are presented in figure 27, appendix E. The data are presented in response to an AVRADCOM request. No attempt has been made to analyze the data.

Pitot-Static System

34. The pitot-static system of the AH-1S was evaluated in level and climbing flight at the general conditions listed in table 1, using the calibrated pace aircraft method. The test techniques and data reduction methods are presented in appendix D. Figure 28, appendix E, presents the corrections for level flight, which are consistent with airspeed corrections listed in the operator's manual.

35. During climbing flight at 80 KIAS and at 1000 feet per minute rate of climb (85 percent torque), airspeed errors of 20 knots were noted on the AH-1S ship's system. Although the test (boom-mounted) airspeed system and the pace aircraft airspeed system were not calibrated in climbs, both systems showed excellent agreement and indicated that the AH-1S ship's system was reading 20 KIAS high. Upon power reduction to level flight power, the ship's system rapidly dropped

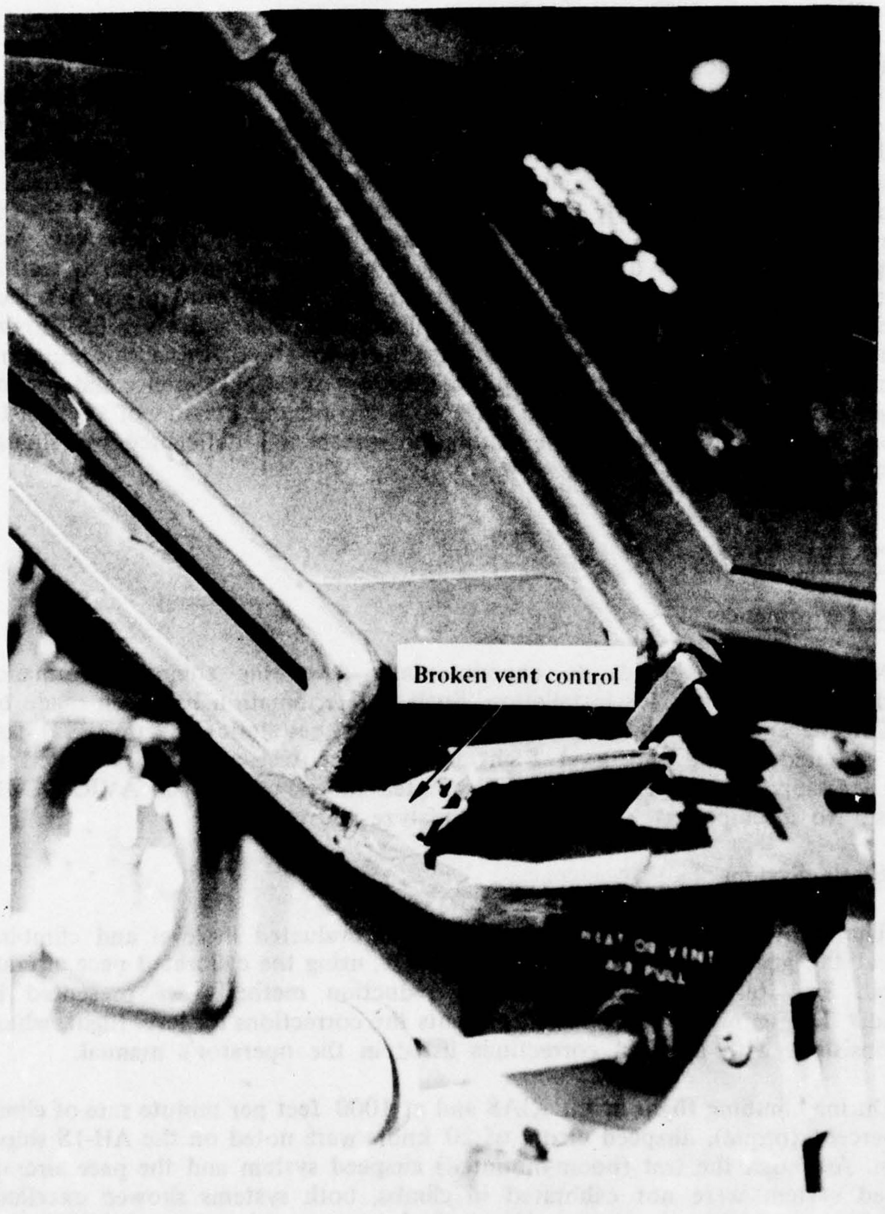


Photo C. Pilot Right-Hand Vent Control.

20 knots to indicate correctly while both the test airspeed system and the pace airspeed remained constant. The operator's manual indicates that the position error in climbs at 80 KIAS is 4 knots. The excessive airspeed indicating error in climbing flight can adversely affect aircraft performance and is a shortcoming.

CONCLUSIONS

GENERAL

36. Installation of the IRS and the IR jammer on the AH-1S caused little degradation in power required. The reduction in power available was not determined during these tests but was supplied by AVRADCOM. Additionally, no significant changes in handling qualities were found as a result of the IRS or IR jammer. One deficiency and six shortcomings inherent in the AH-1S aircraft were identified.

SPECIFIC

37. Out-of-ground-effect hover payload capability of the AH-1S with the IRS installed was reduced 220 pounds at 4000 feet, 35°C ambient air conditions (para 12).

38. Maximum level flight airspeed at 9300 pounds, 2000 feet, and 25°C was reduced 3.5 KTAS with installation of the IRS (para 16).

39. The AN/ALQ-144 IR jammer caused no measurable increase in the drag of the AH-1S (para 17).

DEFICIENCY

40. The nonadjustable ventilation system of the AH-1S is a deficiency (para 32).

SHORTCOMINGS

41. The following AH-1S shortcomings were identified and are listed in decreasing order of relative importance:

a. The lightly damped dutch-roll oscillations above 110 KCAS at heavy gross weight with SCAS ON or OFF (para 27).

b. The excessive airspeed position error in climbing flight (para 35).

c. The lack of adequate directional control margin at 20 KTAS right sideward flight (operator's manual limit) (para 28).

d. The inability to see the battery, generator, and inverter switches from the pilot station (para 31).

e. The difficulty in checking the 90 degree gearbox oil level (para 30).

f. The lack of precise cyclic control centering (paras 18 and 19).

SPECIFICATION COMPLIANCE

42. The AH-1S failed to meet the following requirements of the detail specification: The breakout force for the pilot's cyclic shall be 2.0 ± 0.25 pounds. The longitudinal and lateral breakout forces were 2.5 pounds (aft) and 3.3 pounds (right), respectively (paras 18 and 19).

43. The AH-1S failed to meet the following requirements of MIL-H-8501A:

a. Paragraph 3.2.6. - Longitudinal and lateral limit control forces of 14 pounds and 19 pounds, respectively, failed to meet the 8 pound maximum specified (paras 18 and 19).

b. Paragraph 3.2.11. - The dutch-roll oscillations at heavy gross weights above 110 KCAS with SCAS ON failed to meet the damping requirements (para 27).

RECOMMENDATIONS

44. The deficiency and shortcomings in paragraphs 40 and 41 should be corrected as soon as possible to increase the safety and operational capability of the AH-1S helicopter.

APPENDIX A. REFERENCES

1. Letter, AVRADCOM, DRDAV-EQI, 2 November 1977, subject: Preliminary Airworthiness Evaluation, AH-1S Garrett IR Exhaust Suppression System.

2. Test Plan, USAAEFA, Project No. 77-33, *Preliminary Airworthiness Evaluation, AH-1S Helicopter with Infrared Suppression System*, January 1978.

3. Technical Manual, TM 55-1520-236-10, *Operator's Manual, Army Model AH-1S (Prod) Helicopter*, 29 April 1977.

4. Critical Item Development Specification for AH-1S Infrared Suppression System, Garrett AiResearch Manufacturing Company of California, 24 March 1977.

5. Military Specification, MIL-H-8501A, *Helicopter Flying and Ground Handling Qualities; General Requirements For*, 7 September 1961.

6. Detail Specification, Bell Helicopter Company, Report No. 209-947-265, 13 November 1975.

7. Letter, AVRADCOM, DRDAV-EQI, 6 March 1978, Airworthiness Release for AEFA to Conduct a PAE of JAH-1S Helicopter, S/N 76-22573, Equipped with a Garrett AiResearch Infrared Suppressor (IRS) System and an AN/ALQ-144 Infrared Jammer (nonoperative), Project 77-33.

8. Final Report, US Army Aviation Systems Test Activity, Project No. 66-06, *Engineering Flight Test, AH-1G Helicopter (Huey Cobra), Phase D, Part 2, Performance*, April 1970.

9. Letter, AVRADCOM, DRDAV-EQI, 17 August 1978, subject: Review of Preliminary Airworthiness, AH-1S Helicopter Equipped with an Infrared Radiation Suppressor, Final Report USAAEFA Project No. 77-33.

10. Final Report, USAAEFA, Project No. 72-29, *Instrument Flight Evaluation, AH-1G Helicopter*, July 1975.

APPENDIX B. DESCRIPTION

GENERAL

1. The test helicopter, S/N 76-22573, was a production AH-1S modified to accommodate the IR suppressor. The principal structural modification was the redesign of the cowling which provided support for the IR suppressor and the AN/ALQ-144 IR jammer. Photos 1 through 5 show the test aircraft, IRS, and IR jammer.

IR SUPPRESSOR SYSTEM

2. The IR suppressor system consisted of three major elements: The cowling assembly, upon which the AN/ALQ-144 jammer was mounted, the exhaust duct and the IR suppressor. The cowling assembly was redesigned from that on a production AH-1S aircraft. The IR suppressor was a plug-type suppressor which used the size and shape of the plug to hide the hot engine parts. The suppressor also had circumferentially oriented vents to act as an ejector to entrain compartment and ambient air to mix with the engine exhaust, thereby reducing exhaust gas temperature. Airflow through the engine was extended aft and upward by an exhaust duct and the IR suppressor. The exhaust duct was covered by an insulated blanket.

3. The weight of the installation was approximately 103.1 pounds and the weight of the original aircraft components replaced by the suppressor and mounting structure was approximately 32.1 pounds for a net weight increase of 71 pounds (contractor-furnished data). A change in aircraft cg location was also associated with the IRS installation. This change allowed removal of 20 pounds of factory-installed ballast in the aircraft tail. Therefore, the total aircraft empty weight increase was 51 pounds. Aircraft empty weight and cg location (with digital instrumentation installed) was 6751 pounds at FS 201.43.

ENGINE

4. The T53-L-703 turboshaft engine is installed in the AH-1S helicopter. This engine employs a two-stage, axial-flow free power turbine; a two-stage, axial-flow turbine driving a five-stage axial and one-stage centrifugal compressor; variable inlet guide vanes; and an external annular combustor. A 3.2105:1 reduction gear located in the air inlet housing reduces power turbine speed to a nominal output shaft speed of 6600 rpm at 100 percent N_2. The engine reduction gearbox is limited to 1175 foot-pounds (ft-lb) torque for 30 minutes and to 1110 ft-lb torque for continuous operation. A T_7 interstage turbine temperature sensor harness measures interstage turbine temperatures and displays this information in the cockpit as TGT on the cockpit instruments.

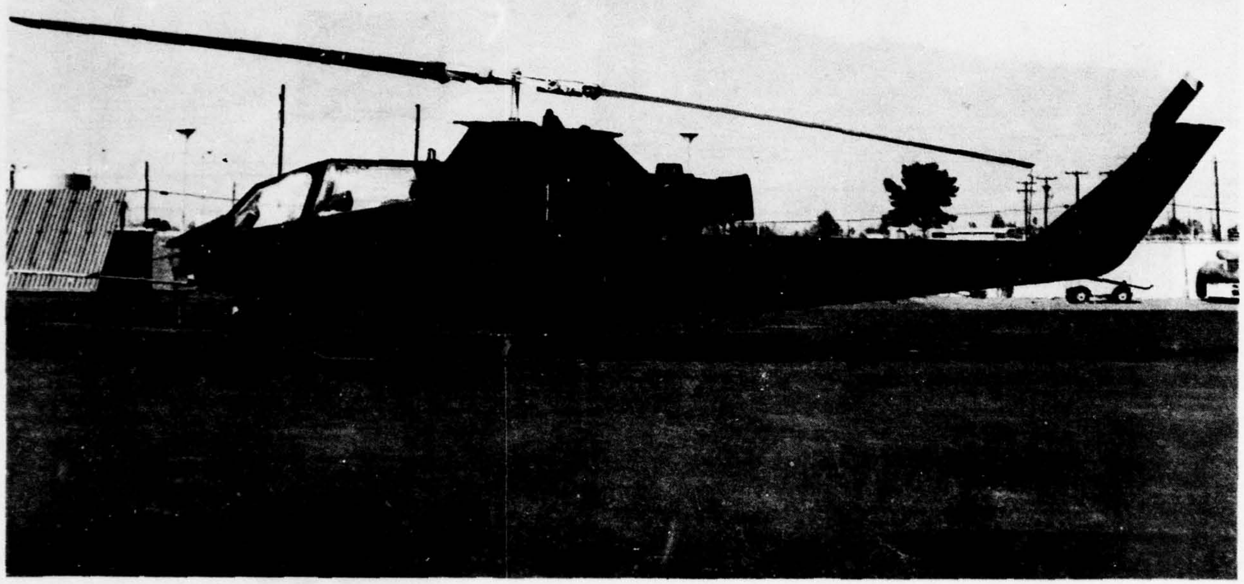


Photo 1. AH-1S Helicopter with IRS Installed.

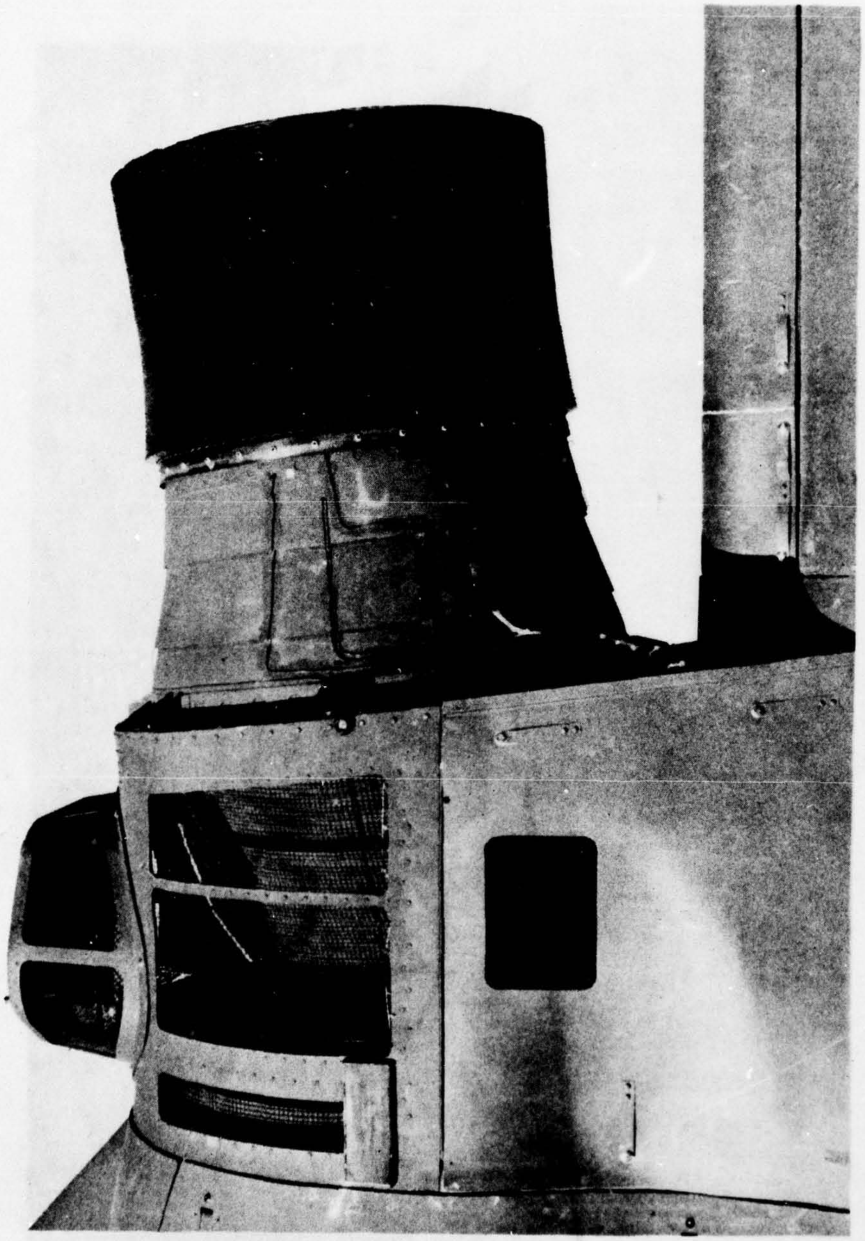


Photo 2. Close-Up View of IRS Installation.

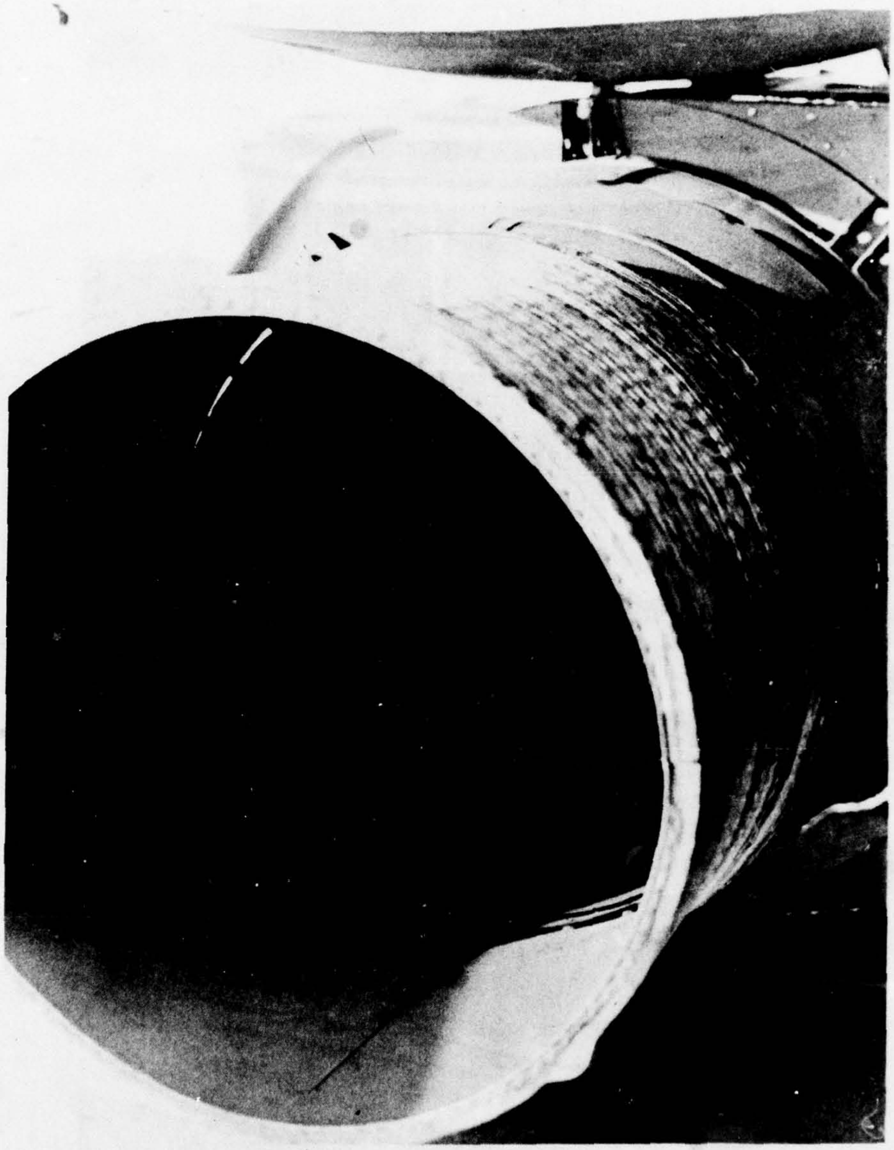


Photo 3. IR Suppressor System.

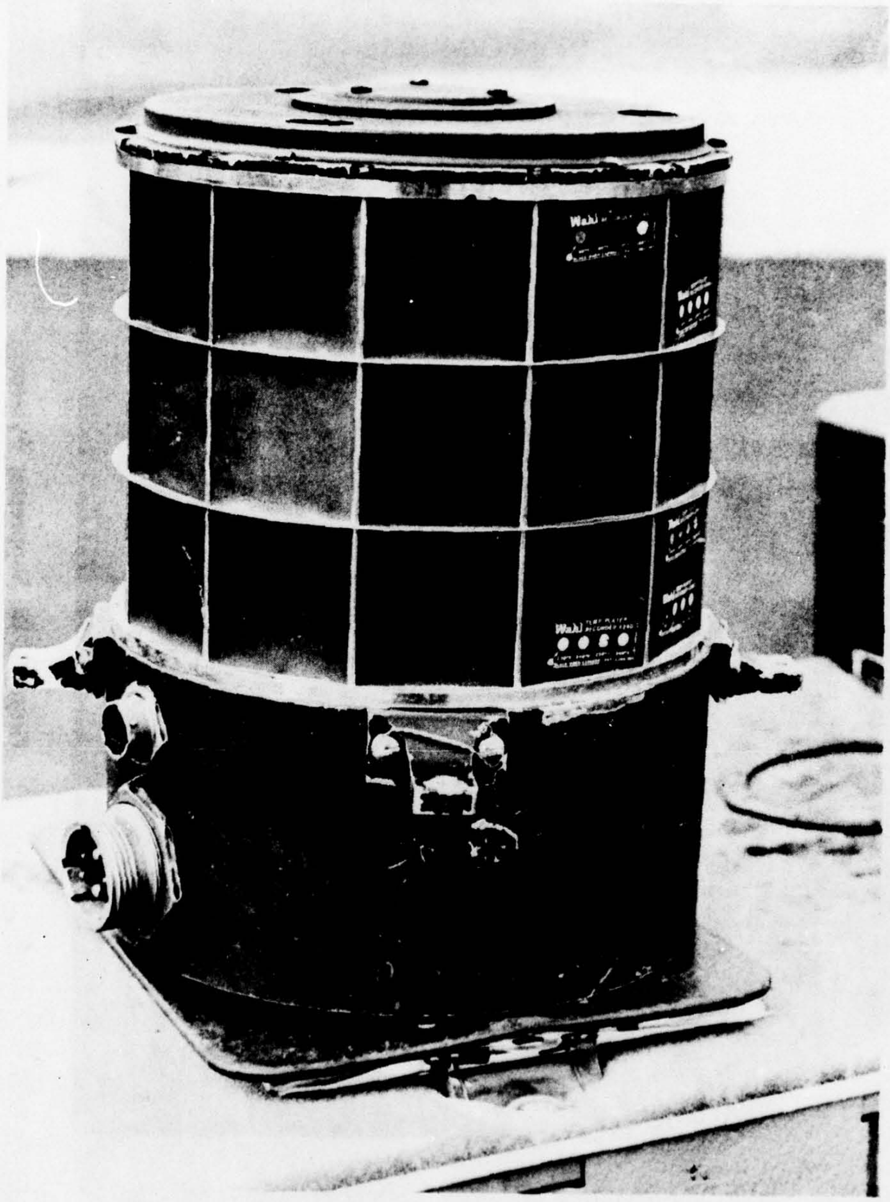


Photo 4. AN/ALQ-144 Jammer.

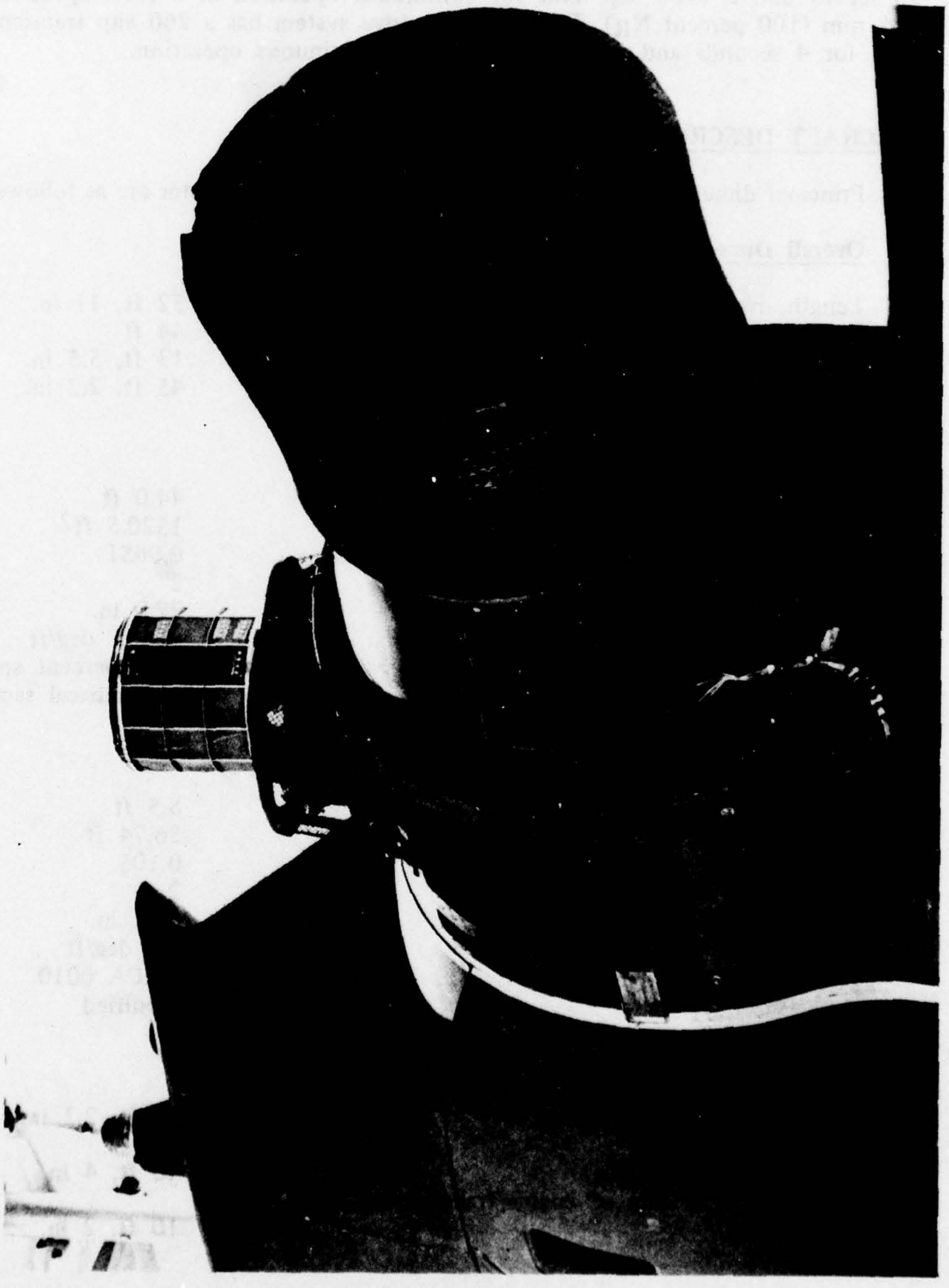


Photo 5. IRS and AN/ALQ-144 Jammer Installation.

TRANSMISSION AND TAIL ROTOR DRIVE

5. The main transmission has a 1290 shp limit for 30 minutes at airspeeds below 90 KCAS and a 1134 shp limit for continuous operation at a rotor speed of 324 rpm (100 percent N_R). The tail rotor drive system has a 260 shp transient limit for 4 seconds and a 187 shp limit for continuous operation.

AIRCRAFT DESCRIPTION

6. Principal dimensions and general data of the AH-1S helicopter are as follows:

Overall Dimensions

Length, rotors turning	52 ft, 11 in.
Width, rotors turning	44 ft
Height, highest point	13 ft, 5.5 in.
Length, rotors removed	45 ft, 2.2 in.

Main Rotor

Diameter	44.0 ft
Disc area	1520.5 ft^2
Solidity	0.0651
Number of blades	2
Blade chord	27.0 in.
Blade twist	-0.455 deg/ft
Airfoil	9.33 percent special symmetrical section

Tail Rotor

Diameter	8.5 ft
Disc area	56.74 ft^2
Solidity	0.105
Number of blades	2
Blade chord	8.41 in.
Blade twist	0.0 deg/ft
Airfoil section	NACA 0010 modified

Fuselage

Length, rotors removed	45 ft, 2.2 in.
Height:	
To tip of tail fin	10 ft, 4 in.
Ground to top of engine/transmission fairing	10 ft, 2 in.

Width:	
Fuselage only	3 ft
Wing tip to wing tip	10 ft, 8.24 in.
Engine cowling	3 ft, 6 in.
Skid gear tread	7 ft, 4 in.
Elevator:	
Span, tip to tip	6 ft, 2 in.
Area	25.2 ft^2
Airfoil	Inverted Clark Y
Vertical fin:	
Area	18.5 ft^2
Airfoil	Special camber
Height	5 ft, 6 in.
Wing:	
Span, tip to tip	10 ft, 8.24 in.
Area	27.8 ft^2
Incidence	14 deg
Airfoil (root)	NACA 0030
Airfoil (tip)	NACA 0024

FLIGHT ENVELOPE

7. The AH-1S with the IRS and IR jammer installed was cleared for flight within the flight envelope in the operator's manual with the additional load factor and sideslip limits presented in figures 1 and 2. These additional limits were imposed by the airworthiness release.

FIGURE 1
AIRSPEED-LOAD FACTOR LIMITATIONS
AH-1S

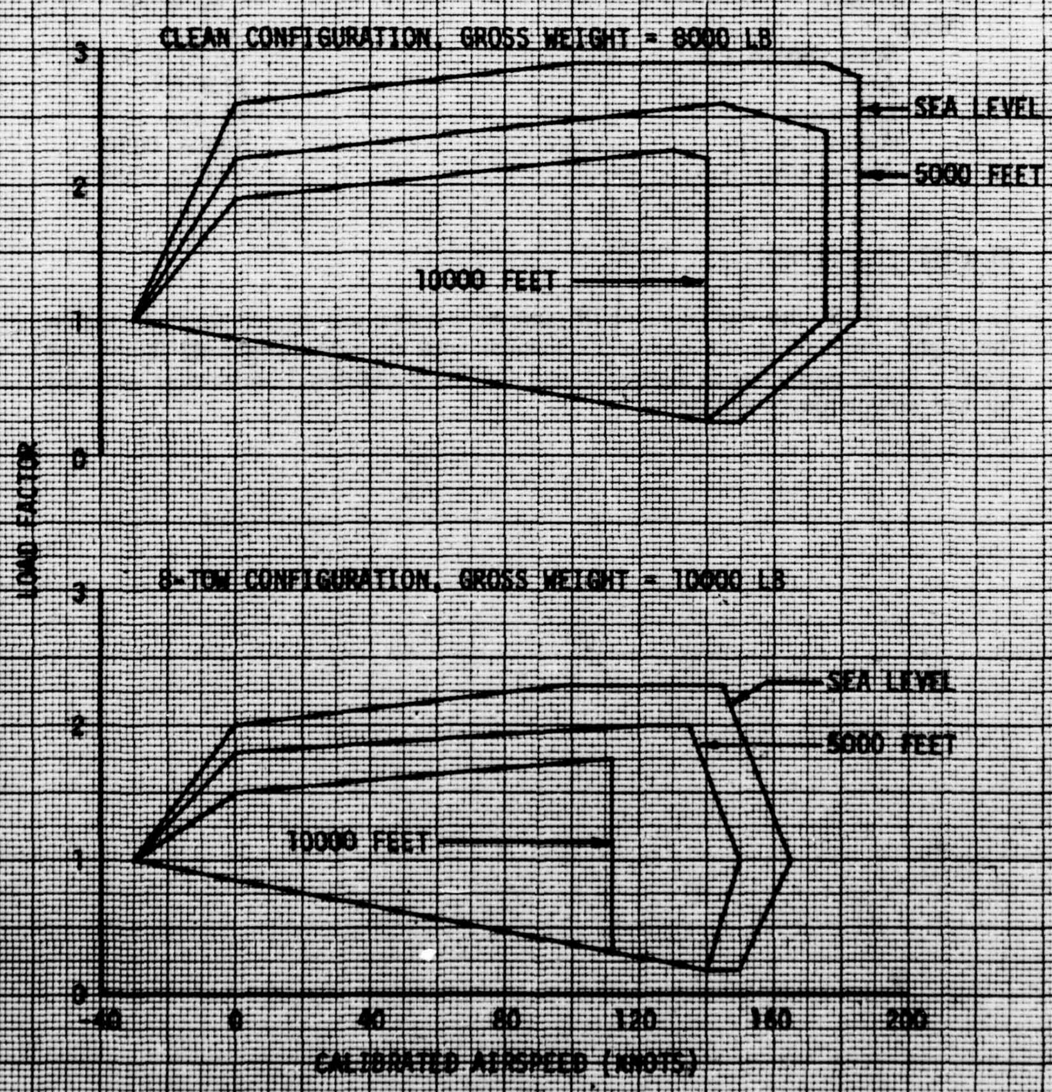
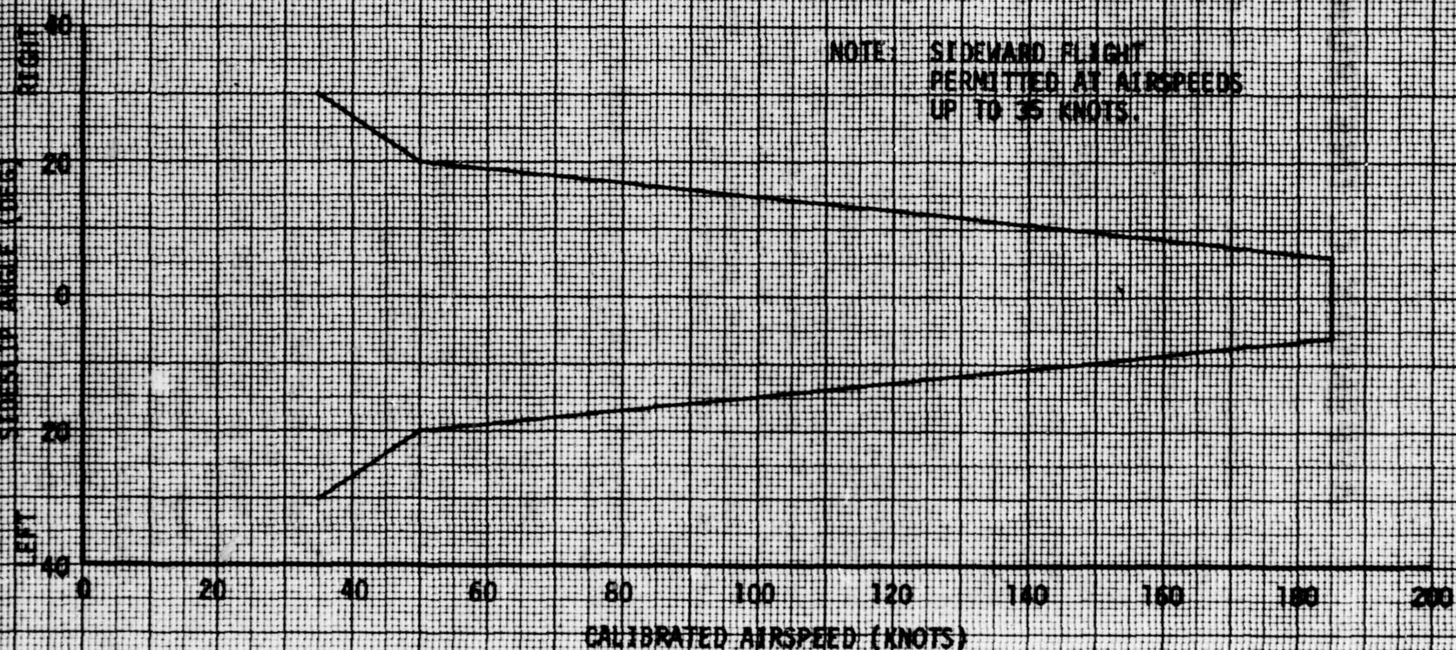


FIGURE 2
AIRSPEED-SIDESLIP LIMITATIONS
AH-1S



APPENDIX C. INSTRUMENTATION

1. The cassette magnetic tape system used as the primary means of obtaining engineering flight test data was designed, installed, and maintained by Garrett AiResearch at Phoenix, Arizona. The main instrumentation package (photo 1) was located in the ammunition compartment area (FS 115) and the visual display was located at the engineer flight station (FS 60). The visual display was designed to allow parameter selection by the flight engineer. The up date period of the visual display could also be selected and ranged from 0.5 to 7.0 seconds. A pitot-static boom incorporating angle-of-attack and angle-of-sideslip vanes was mounted on the aircraft nose.

2. The data generated during this PAE (and during the contractor tests which preceded it) are of questionable validity because of the data acquisition system. The system is not well suited to the type of experimental measurements required in this program. Accurate measurement requires that each parameter be sampled several times per second and that an average of these samples over several seconds be taken. To avoid aliasing errors the minimum number of samples per second should equal five times the maximum frequency of parameter oscillations. Even with one-hertz, low-pass, presampling filters the minimum data sampling rate required is 5 samples per second. The maximum available was 2 samples per second.

3. Parameters recorded on tape and displayed at the flight engineer station are listed below:

- Airspeed (boom)
- Altitude (boom)
- Main rotor speed
- Engine inlet turbine temperature
- Gas generator speed
- Engine output shaft speed
- Engine fuel flow
- Fuel used
- Engine torque oil pressure
- Hover thrust load cell
- Exhaust gas pressure (10 locations)
- Outside air temperature
- Angle of attack
- Angle of sideslip
- Control position:
  - Longitudinal
  - Lateral
  - Directional
  - Collective
- Throttle position

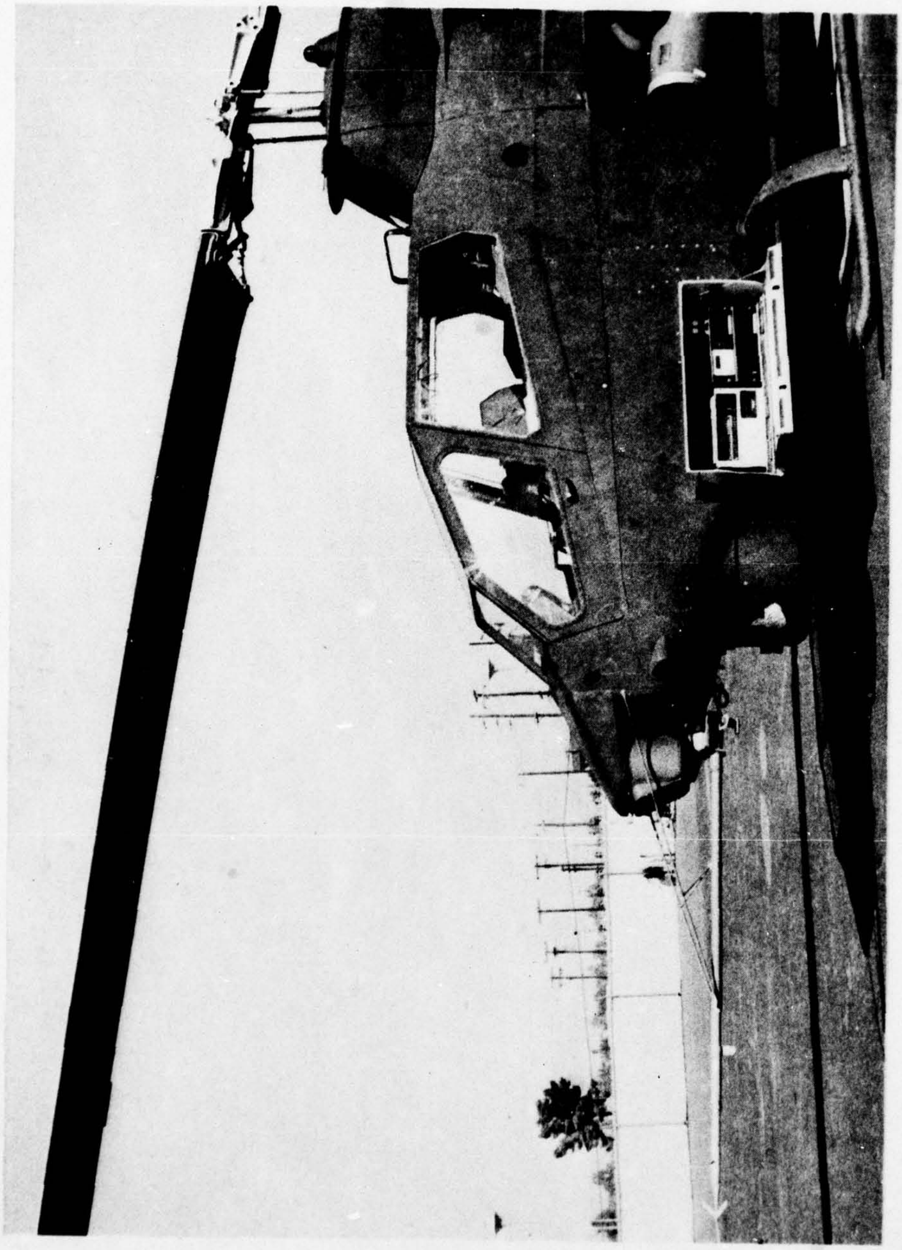


Photo 1. Digital Instrumentation Package in Ammunition Bay.

4. Sensitive indicators located at the pilot station were:

Airspeed (boom)
Altitude (boom)
Angle of sideslip

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

GENERAL

1. **Helicopter performance test data were generalized by use of nondimentional coefficients. The purpose of this generalization was to accurately predict performance at aircraft gross weight/ambient air condition combinations not specifically tested. The following coefficients were used:**

a. **Coefficient of power (C_P):**

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

b. **Coefficient of thrust (C_T):**

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

c. **Advance ratio (μ):**

$$\mu = \frac{1.6878 \times V_T}{\Omega R}$$

d. **Advancing tip mach number (M_{TIP}):**

$$M_{TIP} = \frac{1.6878\ V_T + \Omega R}{a}$$

Where:

SHP = Engine output shaft horsepower

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 (rad/sec)

R = Main rotor radius

GW = Gross weight (lb)

1.6878 = Conversion factor (ft/sec/kt)

V_T = True airspeed (kt)

a = Speed of sound (ft/sec)

2. 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 the engine manufacturer's test cell calibration. Horsepower was determined by the following equation:

$$SHP = \frac{2\pi \times N_e \times T_q}{33{,}000}$$

Where:

N_e = Engine output shaft speed (rpm)

T_q = Engine output shaft torque (ft-lb)

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

SHP = Shaft horsepower

HOVER PERFORMANCE

3. The tethered method of hover performance testing was used. This method required that the aircraft be at a very light gross weight, that it be tied to the ground by a cable which would allow only a 15-foot skid height, and that a load cell be used to measure cable tension. During the test the cable was kept taut and vertical at all times. To get a maximum variation of C_T (equation 2) rotor speed and cable tension were varied during the test. The technique used to vary cable tension was to set various torque settings from minimum required to hover at 15 feet to the maximum allowed at test conditions. Cable angle was relayed to the pilot by radio from two ground observers in order to maintain the aircraft directly over the ground tie-down point.

4. The data were plotted as C_P versus C_T using equations 1 and 2. The gross weight in equation 2 was determined by adding cable tension and the weight of the cable and load cell to the engine start gross weight, and then subtracting the weight of the fuel burned prior to each data point. The data points were then curve fit using a multiple linear regression program. The equation of the resulting line is:

$$C_P \times 10^5 = 2.689747625 + 0.121185269(C_T \times 10^4)^{3/2} - 0.000014595(C_T \times 10^4)^3.$$

The equation is valid only for the range of C_T's actually tested and should not be used to extrapolate to higher or lower values of C_T.

5. A data sample rate of 2 per second was used and all the data samples for a given test point were averaged together to obtain one C_P vs C_T data point on figure 1, appendix E. This was done to try to minimize aliasing errors caused by limited sampling of a fluctuating parameter (cable tension). Some error probably remains in the data.

LEVEL FLIGHT PERFORMANCE

6. Each level flight performance flight was designed to obtain one curve of C_P versus μ at a constant value of C_T. The flight technique was to stabilize at zero sideslip at incremental airspeeds from approximately 40 KIAS to the maximum attainable. At each airspeed, torque, altitude, airspeed, and rotor speed were held constant for at least 1 minute prior to recording data. Altitude was increased between data points as a function of fuel burnoff in order to maintain a constant ratio of gross weight to air pressure ratio (GW/δ). Also, rotor speed (N) was varied as a function of ambient air temperature in order to maintain a constant ratio of rotor speed to square root of the air temperature ratio ($N/\sqrt{\theta}$). By rearranging equation 2 as follows:

$$C_T = \frac{GW/\delta}{\rho_o A \left(\frac{2\pi R}{60}\right)^2 \left(\frac{N}{\sqrt{\theta}}\right)^2}$$

it can be seen that C_T will also be constant if GW/δ and $N/\sqrt{\theta}$ are constant. During these tests, the target GW/δ was different for each flight in a given aircraft configuration, but the target $N/\sqrt{\theta}$ was 320 rpm for all flights. Because of varying differential between the actual rotor speed (recorded on magnetic tape) and the cockpit displayed rotor speed, $N/\sqrt{\theta}$ varied slightly from flight to flight. The reason for maintaining constant $N/\sqrt{\theta}$ was to minimize the difference in compressibility effects between flights.

7. The C_P versus μ curves were cross plotted as C_P versus C_T with lines of constant μ. From these curves (figs. 2 through 4, app E) level flight performance at any combination of gross weight, rotor speed, pressure altitude, and air temperature can be determined.

Test-day level flight power was corrected to standard-day conditions (as shown in figs. 7 throught 14, app E) by assuming that the test-day dimensionless parameters, C_{P_t}, C_{T_t}, and μ_t 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 the definition of equation 1, the following relationship can be derived:

$$SHP_s = SHP_t \times \frac{\rho_s}{\rho_t} \times \left(\frac{\Omega_s}{\Omega_t}\right)^3$$

Where:

SHP = Engine output shaft horsepower

ρ = Air density (slug/ft^3)

t = Test day

s = Standard day

Ω = Main rotor angular velocity (rad/sec)

A similar correction for V_T could be derived from the definition of μ (equation 3). This correction was insignificantly small and therefore not made.

9. Specific range was calculated using measured values of V_T and fuel flow as follows:

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

Where:

NAMPP = Specific range (nautical air miles per pound of fuel)

V_T = True airspeed (kt)

W_f = Fuel flow (lb/hr)

10. The change in drag between IRS installed and uninstalled configurations was determined in terms of a change in equivelent flat plate area (Δf_e). The equation for Δf_e is as follows:

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

Where:

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

ΔC_P = Difference in power coefficient required between two configurations at the same advance ratio and thrust coefficient

A = Main rotor disc area (ft^2)

μ = Advance ratio

11. Three neat instantaneous samples of each parameter were taken at each test condition and averaged together to make one data point. The three samples were approximately 30 seconds apart. With this instrumentation system this was the fastest sample rate which would provide valid engine exhaust static pressures. Parameter fluctuations during this test are usually small (unlike cable tension during hover performance tests) and therefore aliasing errors at this sample rate are probably small.

CONTROL SYSTEM CHARACTERISTICS

12. These tests were conducted on the ground with hydraulic and electrical power provided by ground power units. A hand-held force gage was used to measure the force required to move the cyclic control in incremental displacements to the limits of travel in four directions. Hysteresis was checked by taking measurements in the increasing and decreasing force directions.

CONTROL POSITIONS IN TRIMMED FORWARD FLIGHT

13. Data for this evaluation were a by-product of the level flight performance tests. No special test techniques or data analysis methods were required.

STATIC LONGITUDINAL STABILITY

14. These tests were accomplished by establishing a trim condition (airspeed/power combination) with zero control forces. Without changing the collective positions, trim setting or rotor speed, the helicopter was stabilized at incremental airspeeds both faster and slower than the trim airspeed, using cyclic only. To speed the data reduction, only one data sample was taken at each stabilized speed.

STATIC LATERAL-DIRECTIONAL STABILITY

15. These tests were conducted by first establishing a zero sideslip trim condition, in level and autorotational flight and then incrementally varying sideslip angle (both

left and right) until the limits of the sideslip envelope or a full control deflection were reached. Each test was conducted with collective and airspeed maintained at the trim values.

DYANAMIC STABILITY

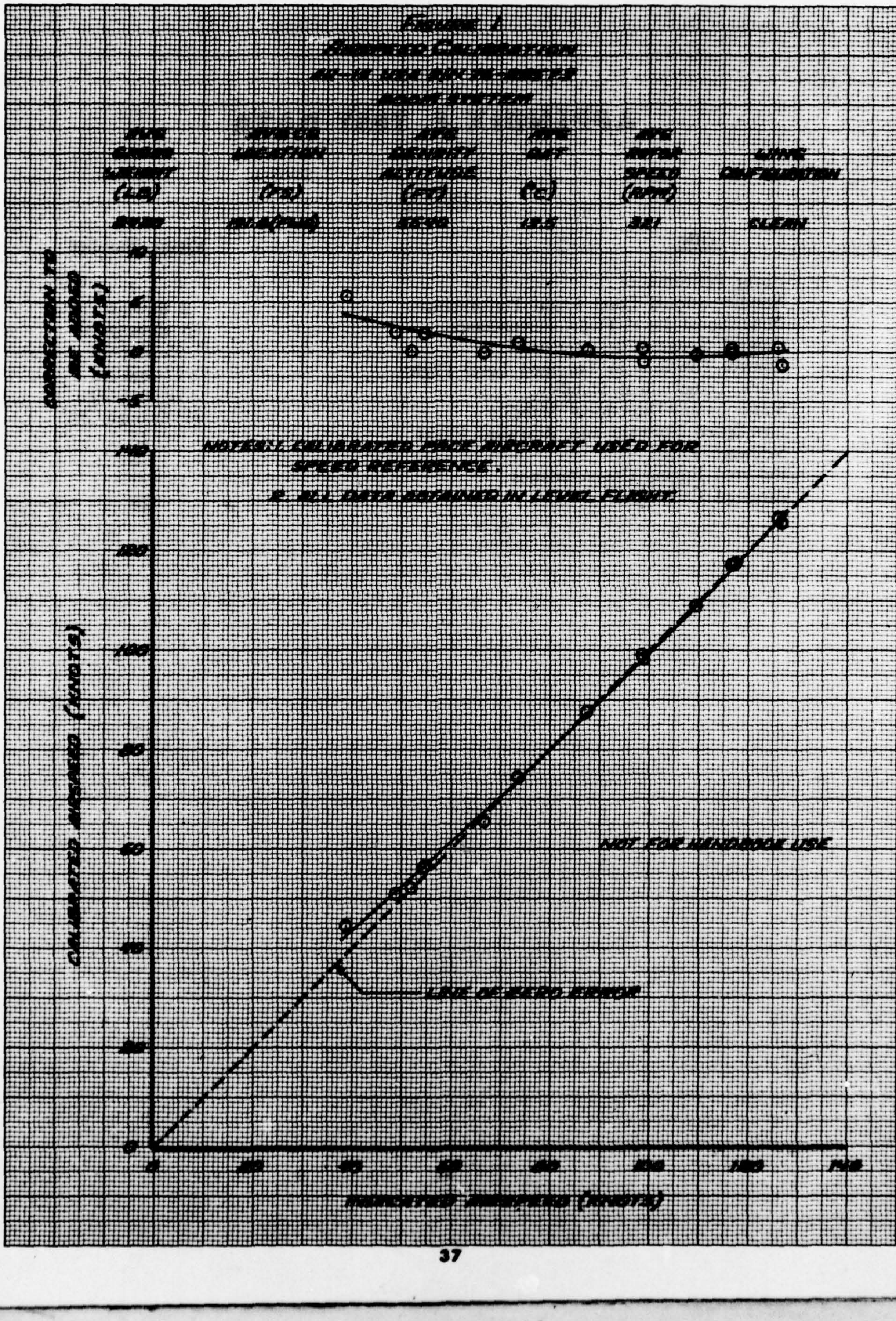
16. These tests consisted of a dutch roll investigation only. The test was conducted by first establishing a trim condition at a desired airspeed, making a pedal doublet input, and then returning all controls to trim. Damping of the resultant oscillation was evaluated by counting the number of times the aircraft passed through the trim condition (trim overshoots) before it restabilized on trim. The number of cycles to damp is equal to half the number of trim overshoots.

LOW-SPEED FLIGHT CHARACTERISTICS

17. Testing was accomplished using the ground pace vehicle method at a constant skid height of 10 feet in winds of 5 knots or less. The method consisted of stabilizing the test aircraft on a ground pace vehicle having a calibrated speed system. Tests were flown in 5 knot increments from a hover to 40 knots forward, 30 knots rearward, 30 knots left sideward, and 20 knots right sideward (operator's manual limit). A data sample rate of 2 per second was used and the data were averaged over each test point.

PITOT-STATIC SYSTEM

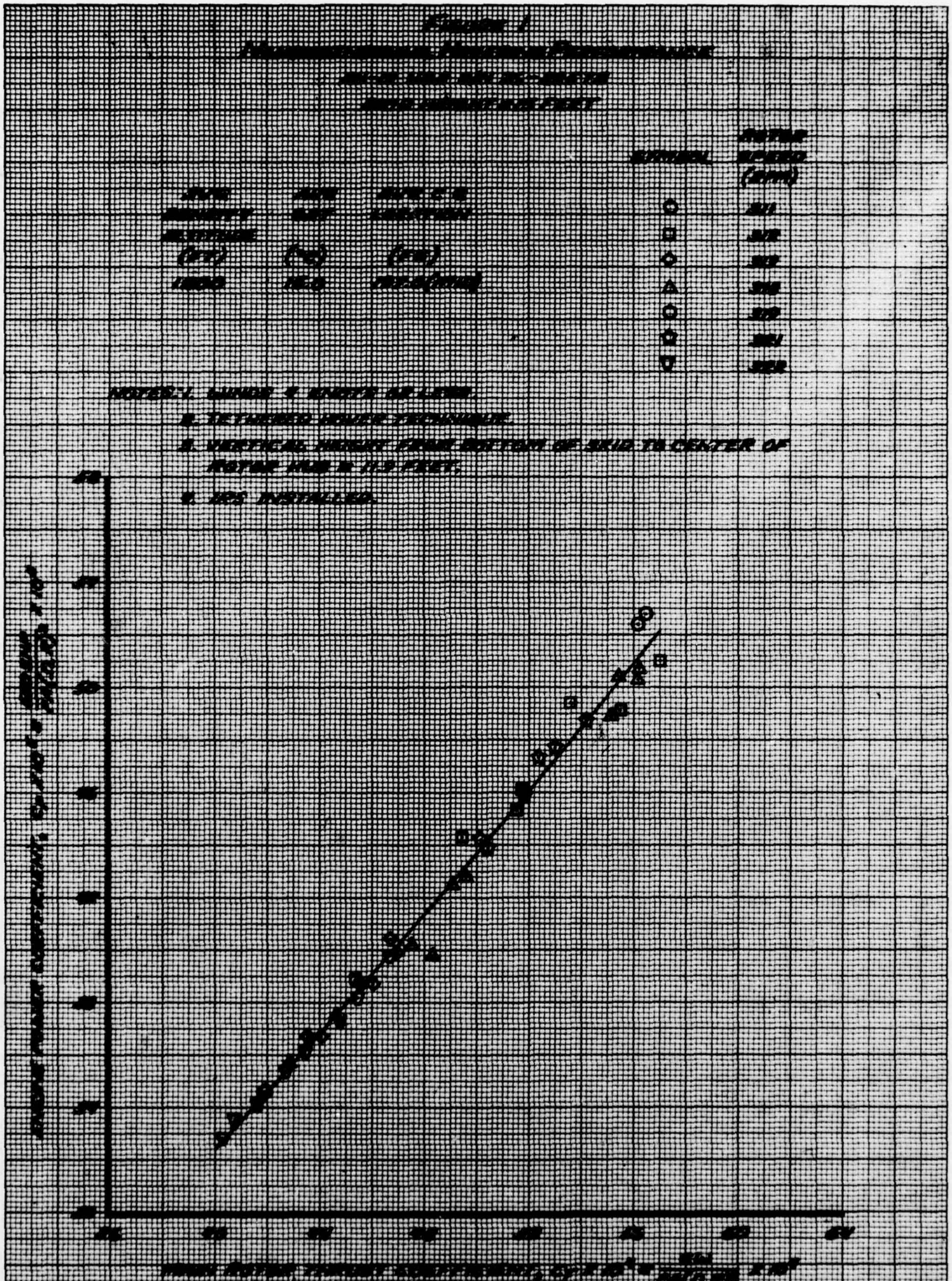
18. Airspeed position errors of the boom and ship's systems were determined by pacing the test aircraft with an AH-1G with a calibrated airspeed system. The test aircraft boom system was used as a reference for all data reduction during this evaluation. The position error of this system is presented in figure 1.

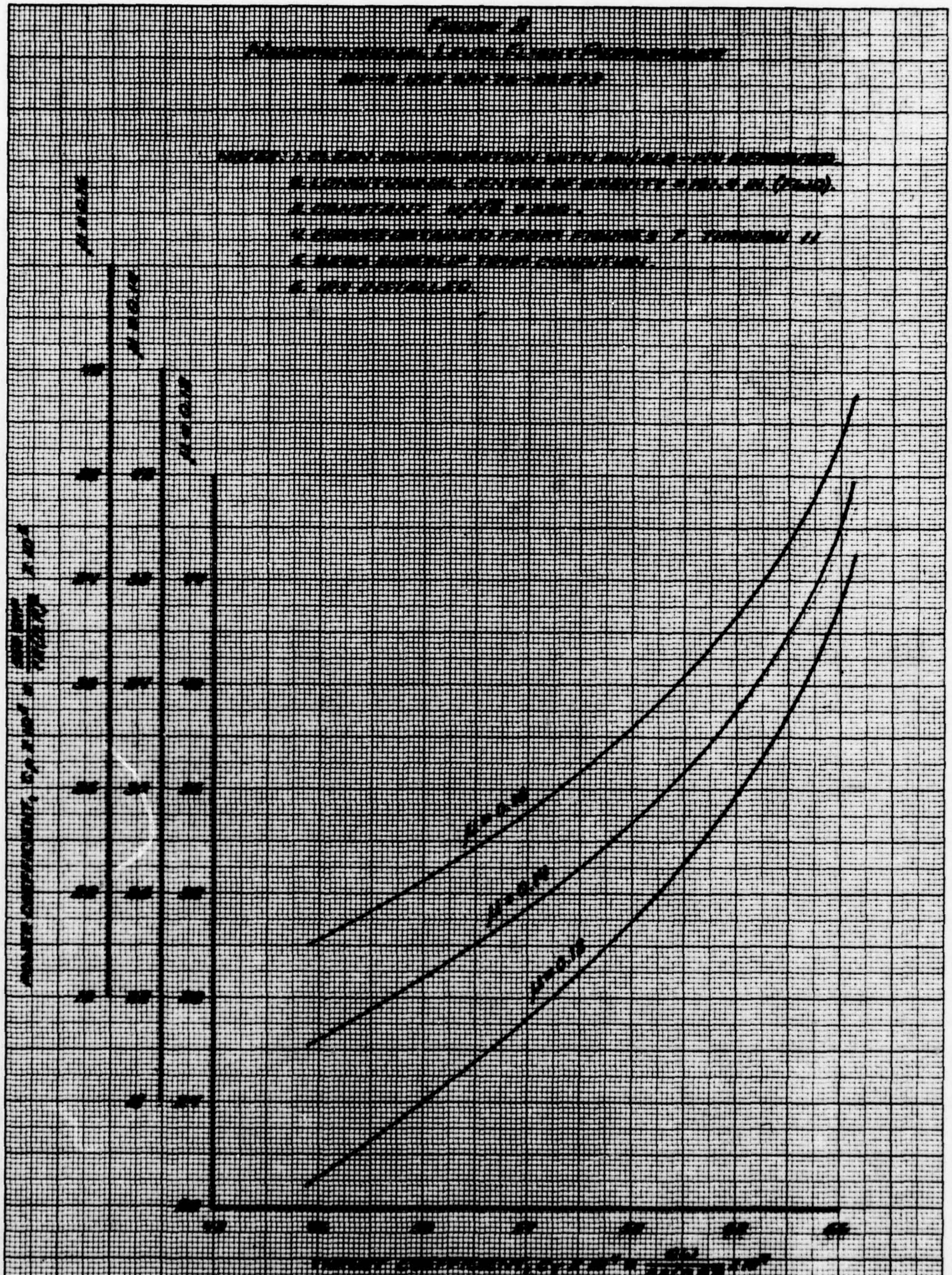


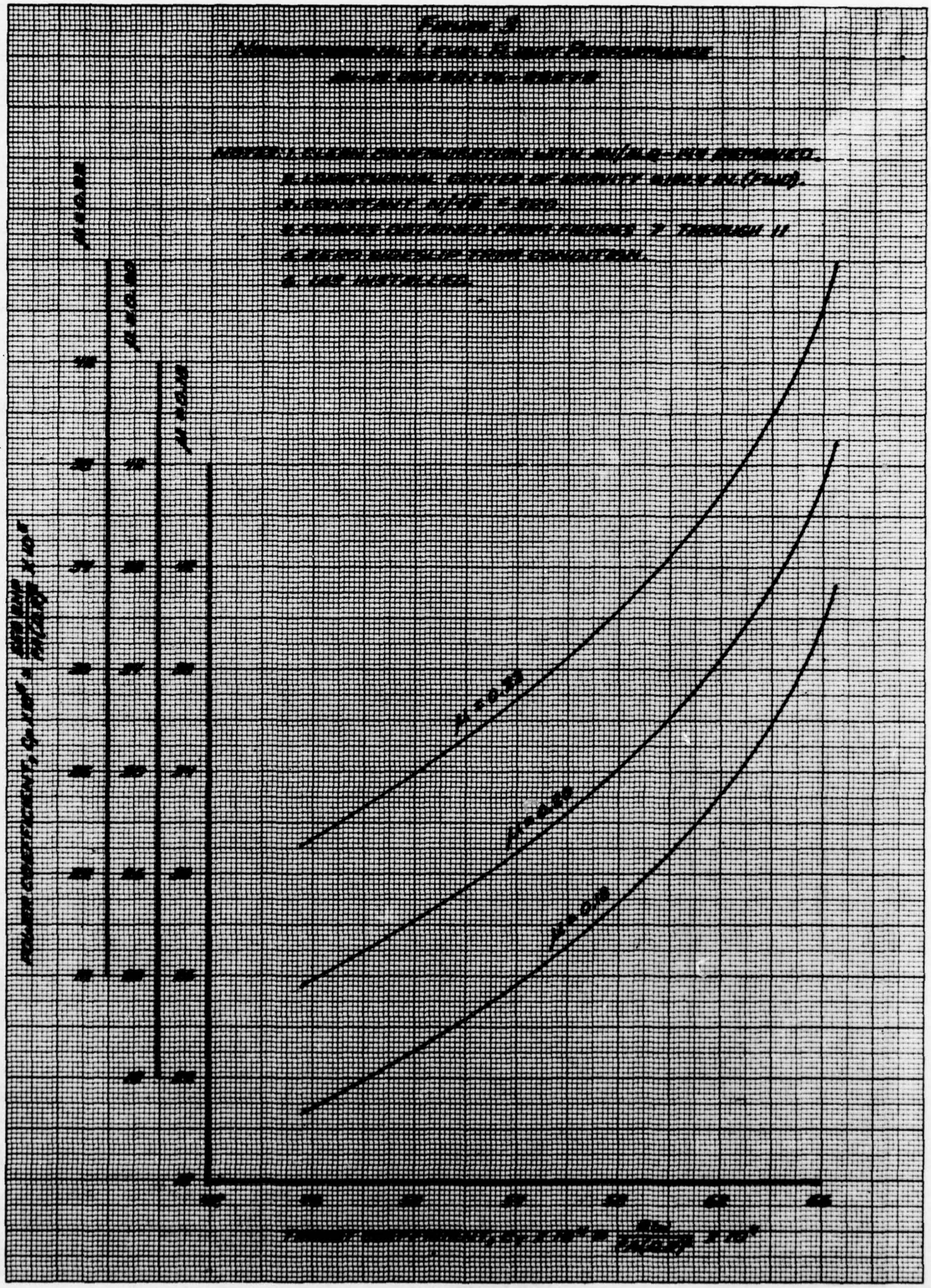
WING CONFIGURATION
CLEAN
NOTES: 1. CALIBRATED PACE AIRCRAFT USED FOR SPEED REFERENCE.
2. ALL DATA OBTAINED IN LEVEL FLIGHT.
NOT FOR HANDBOOK USE
LINE OF ZERO ERROR
CALIBRATED AIRSPEED (KNOTS)
INDICATED AIRSPEED (KNOTS)
0
20
40
60
80
100
120
140

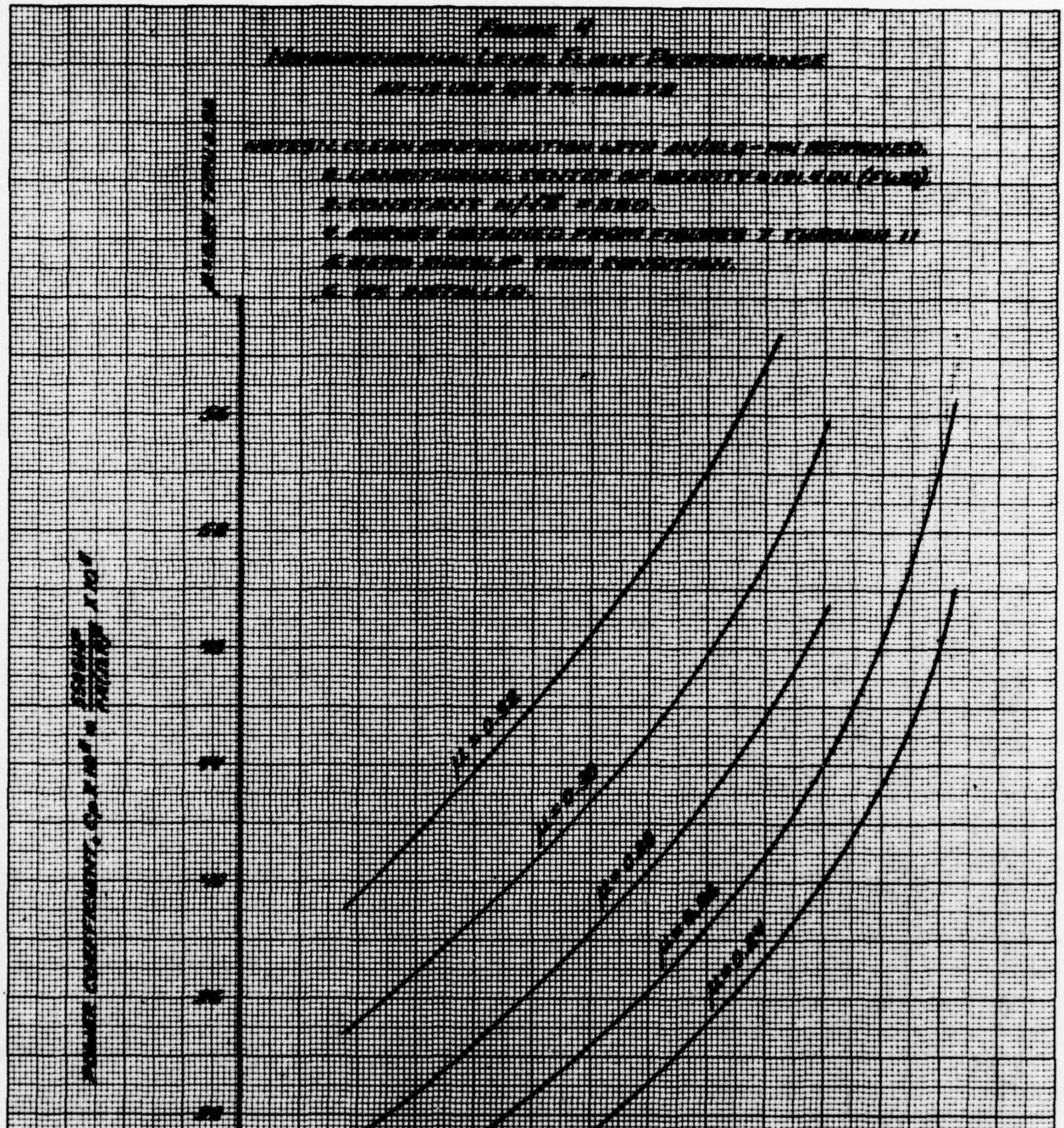
APPENDIX E. TEST DATA

INDEX


SYMBOL
ROTOR SPEED (RPM)
2. TETHERED HOVER TECHNIQUE.
3. VERTICAL HEIGHT FROM BOTTOM OF SKID TO CENTER OF ROTOR HUB IS 11.9 FEET.
4. IRS INSTALLED.







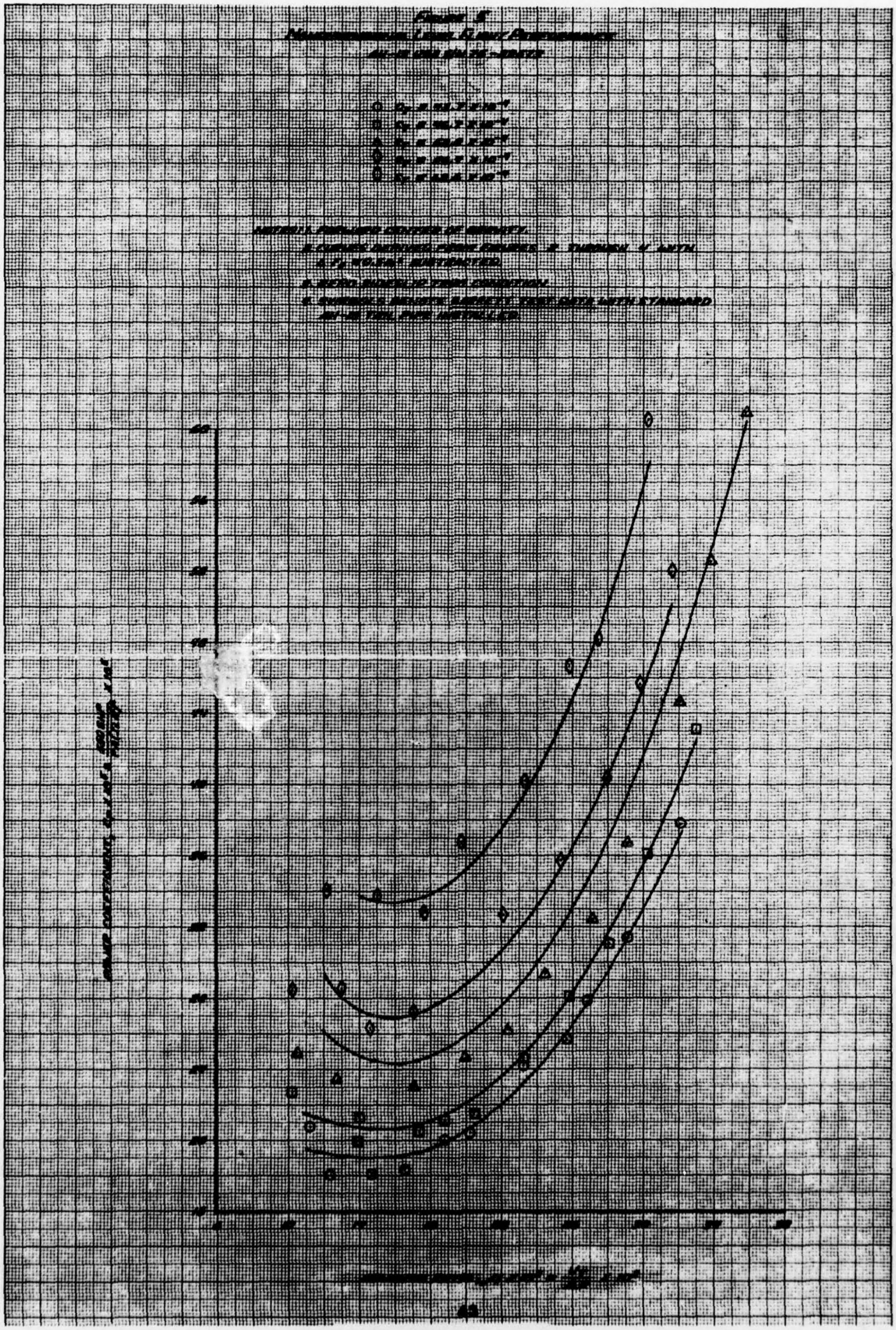
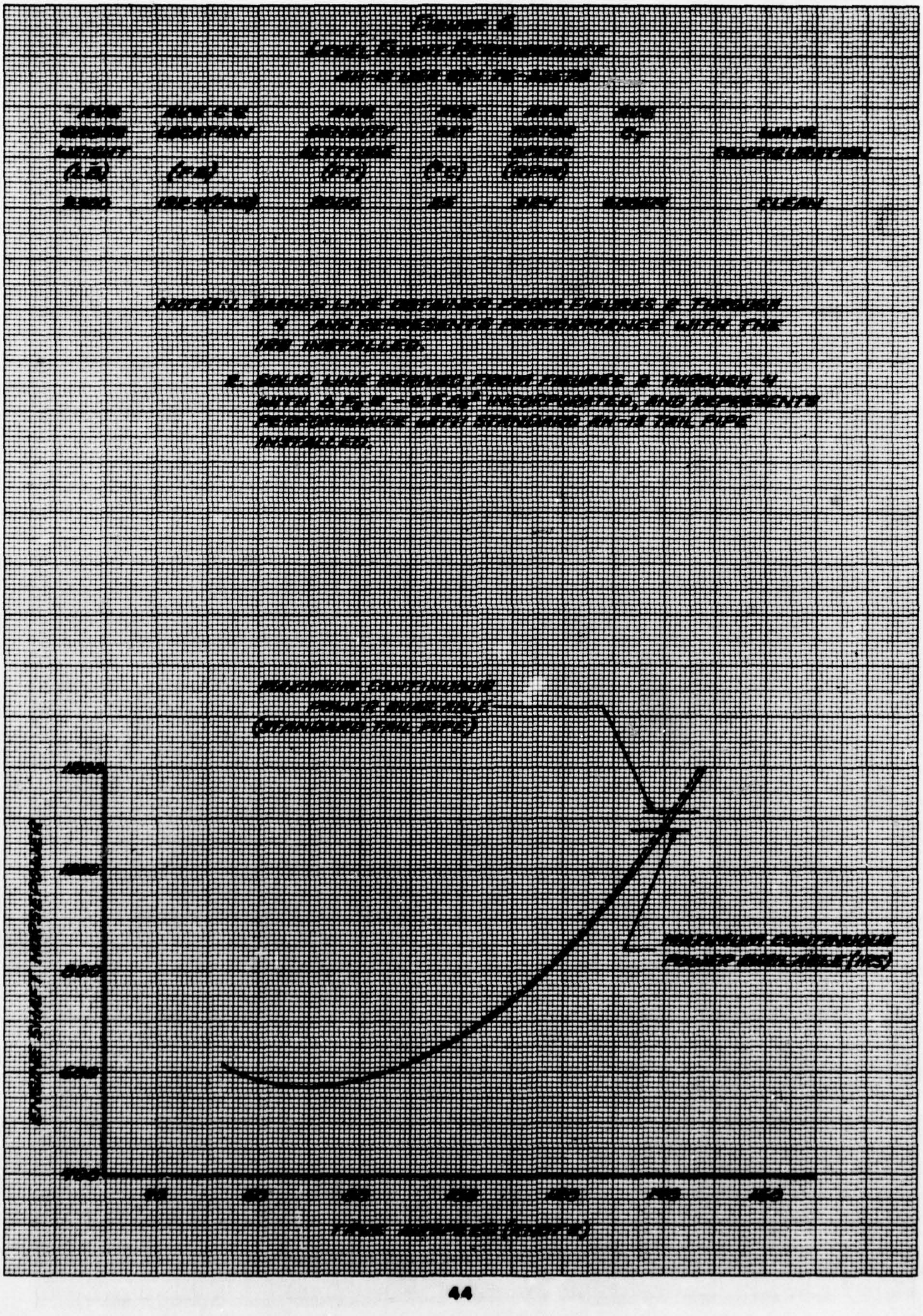
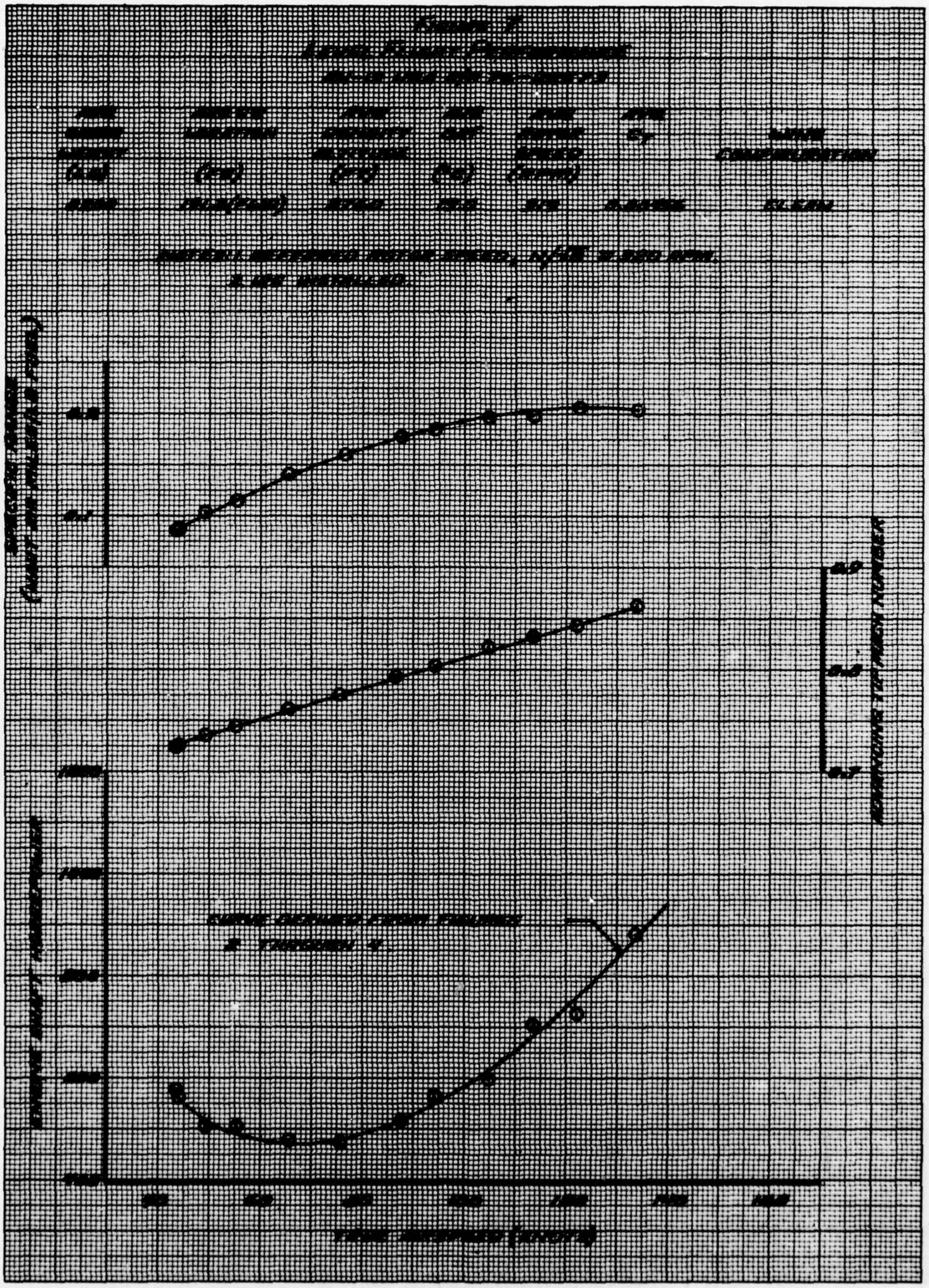
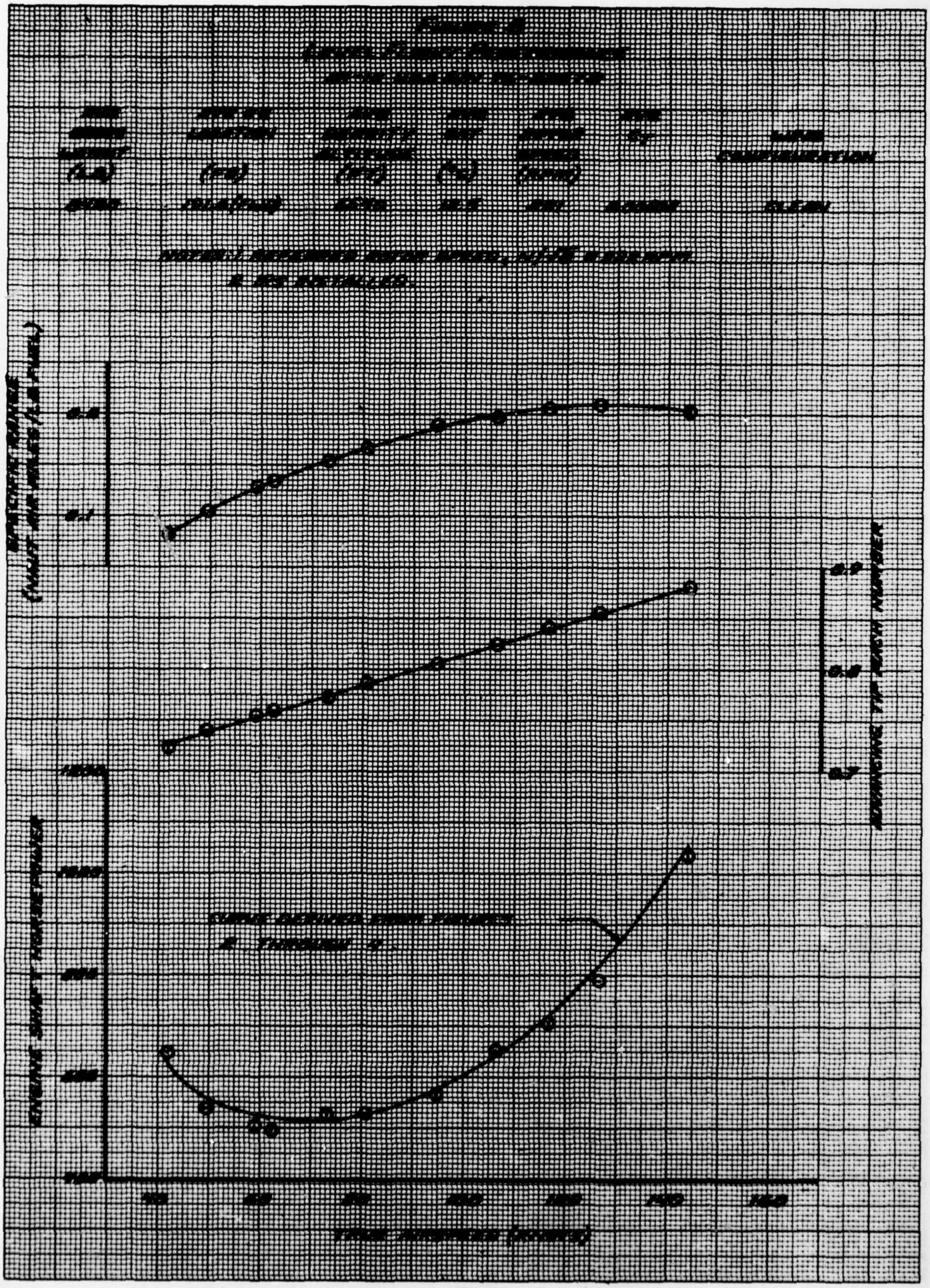
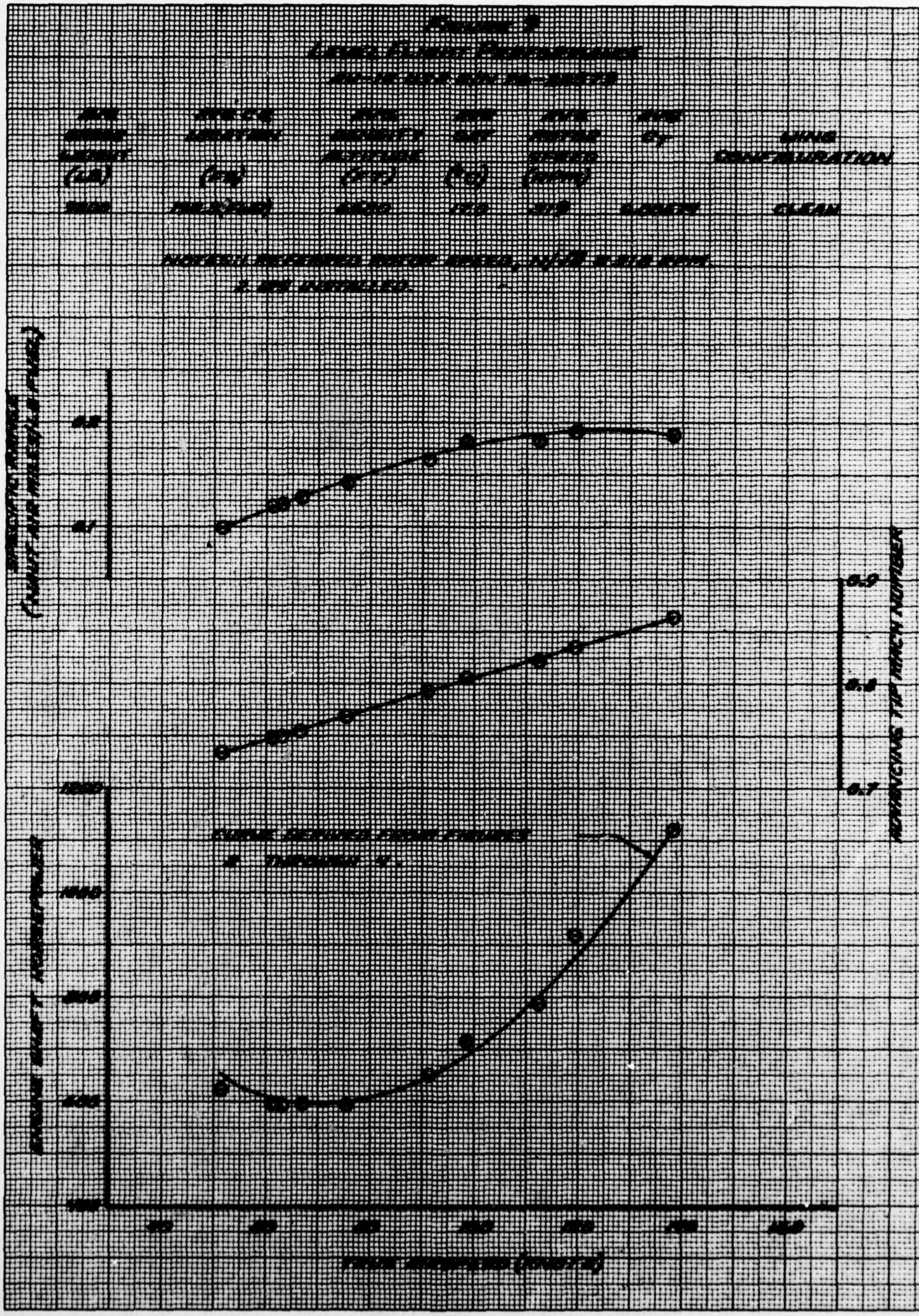


FIGURE 5


MAXIMUM CONTINUOUS
POWER AVAILABLE
(STANDARD TAIL PIPE)
MAXIMUM CONTINUOUS
POWER AVAILABLE (IRS)
ENGINE SHAFT HORSEPOWER
TRUE AIRSPEED (KNOTS)






WING
CONFIGURATION
CLEAN
NONDIMENSIONAL TIP MACH NUMBER
0.9
0.8
0.7

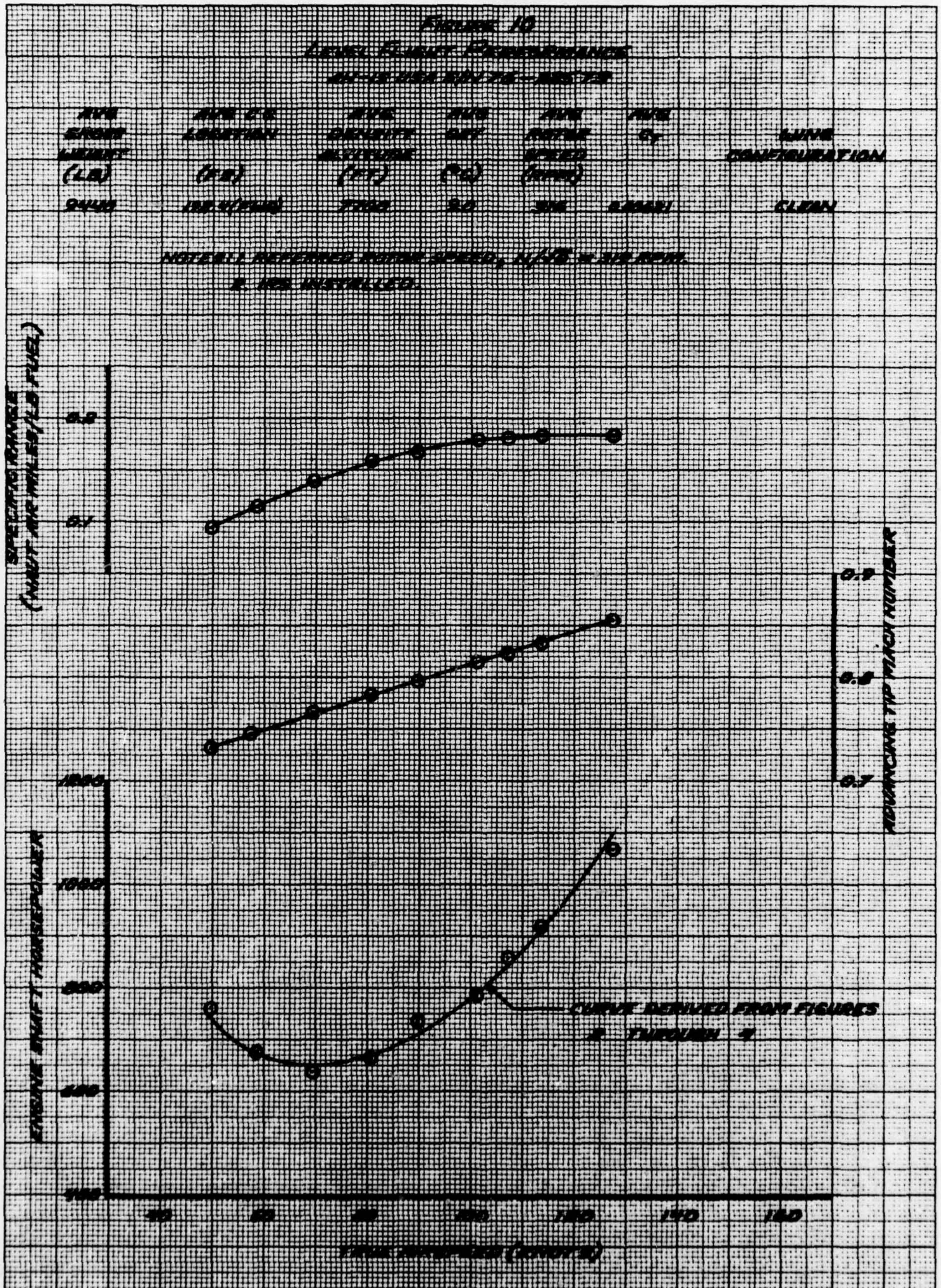
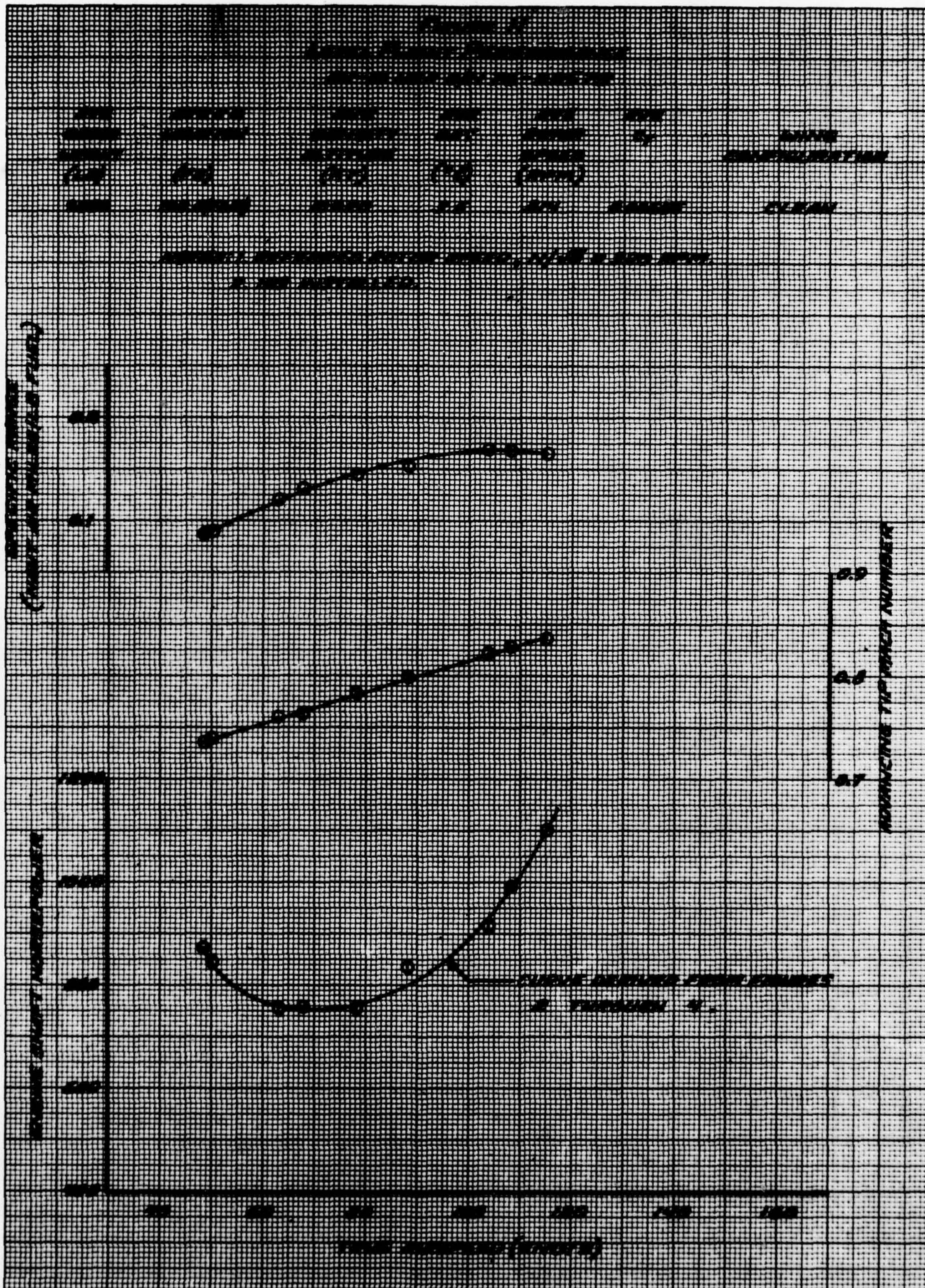
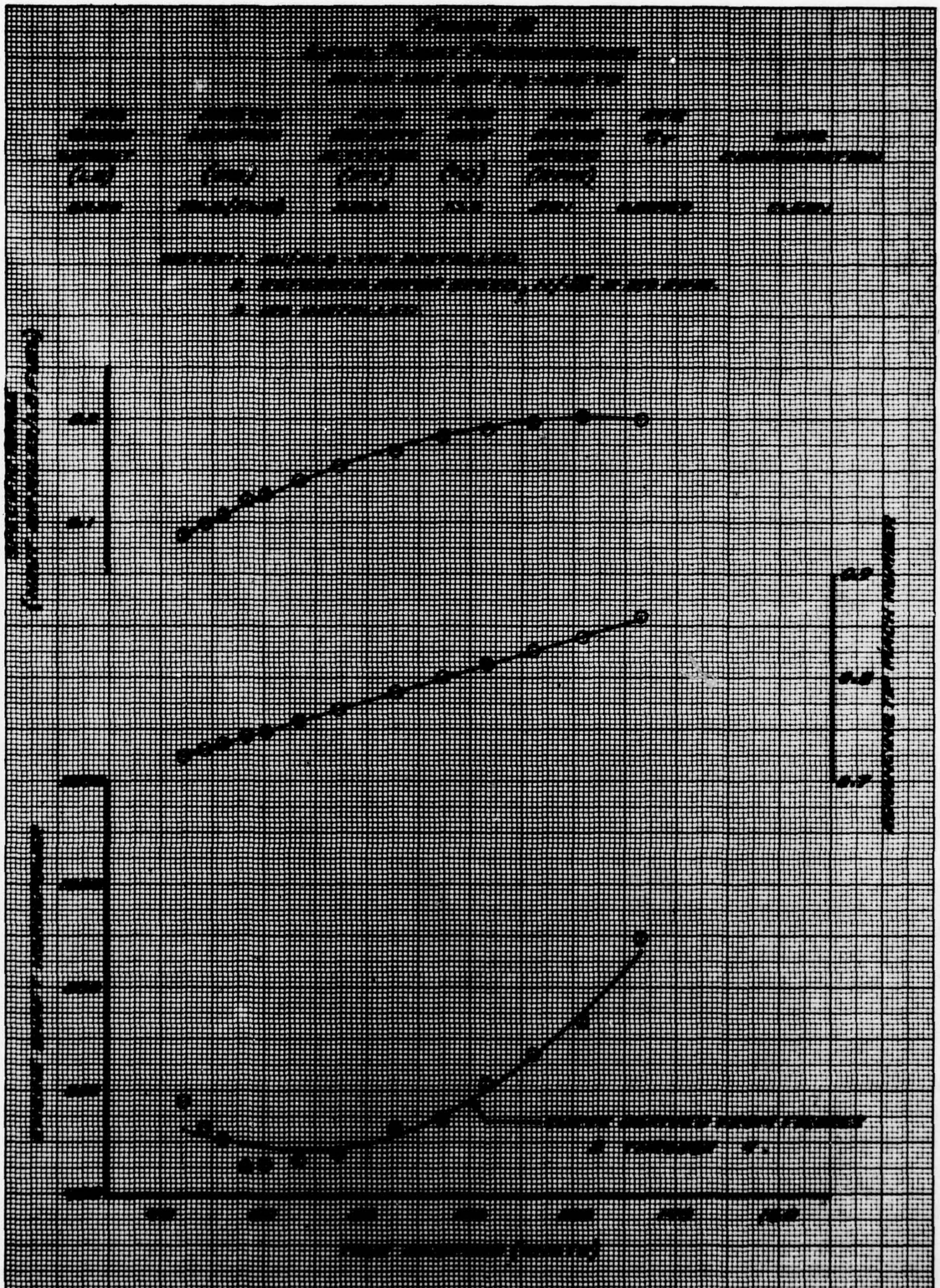
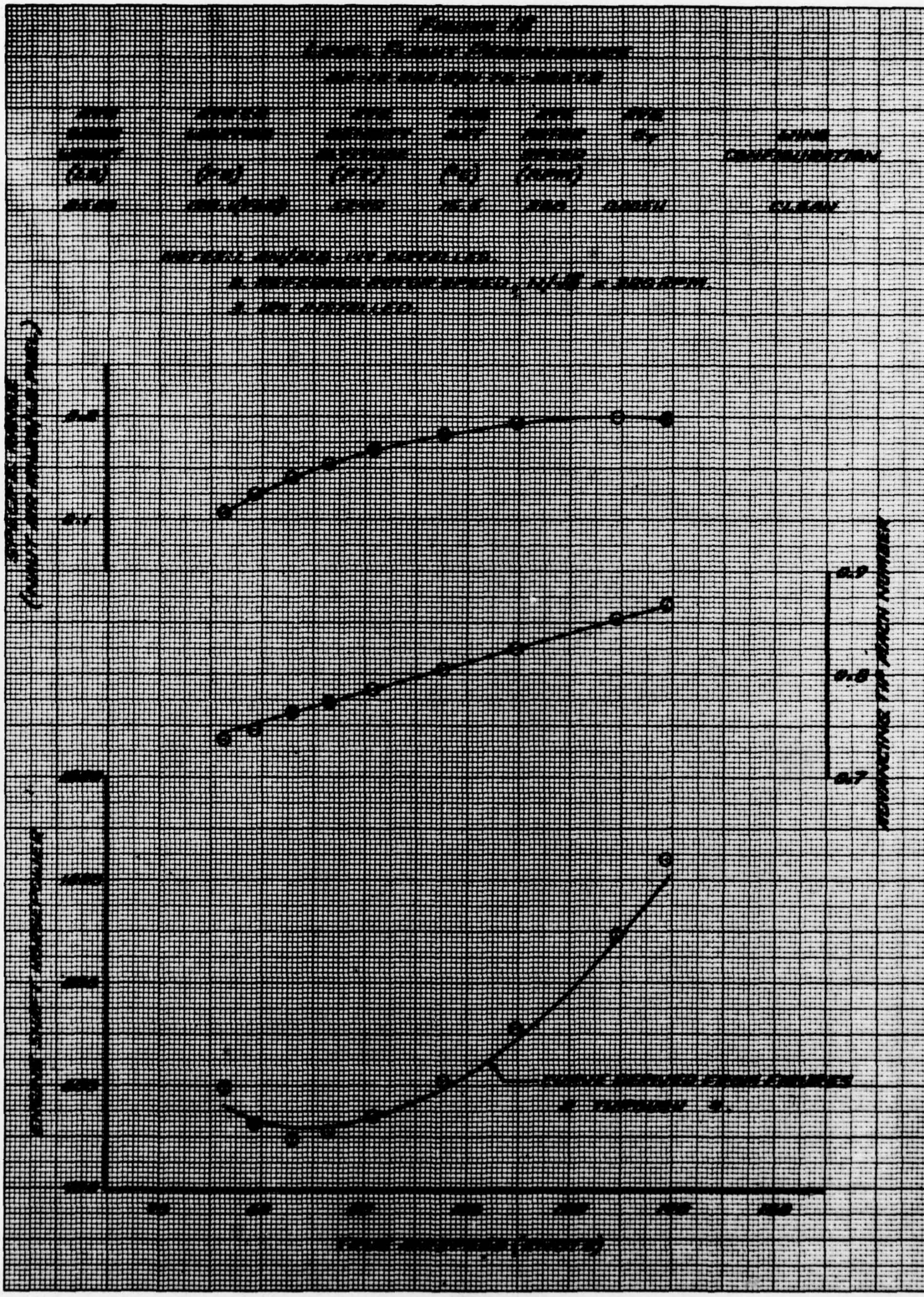


FIGURE 10
LEVEL FLIGHT PERFORMANCE


CLEAN
0.9
0.8
0.7
CURVE DERIVED FROM FIGURES
2 THROUGH 4.




CLEAN
ADVANCING TIP MACH NUMBER
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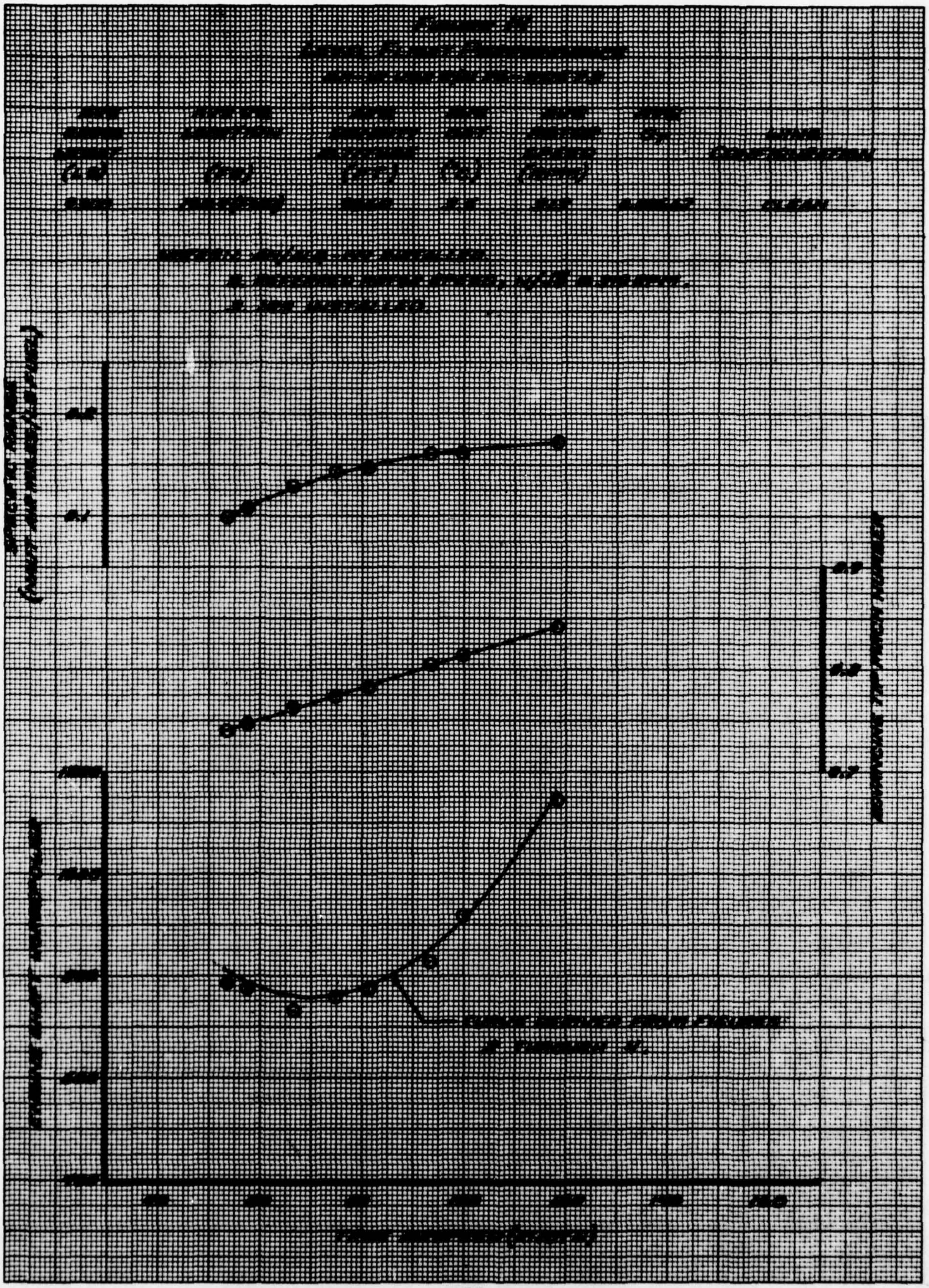
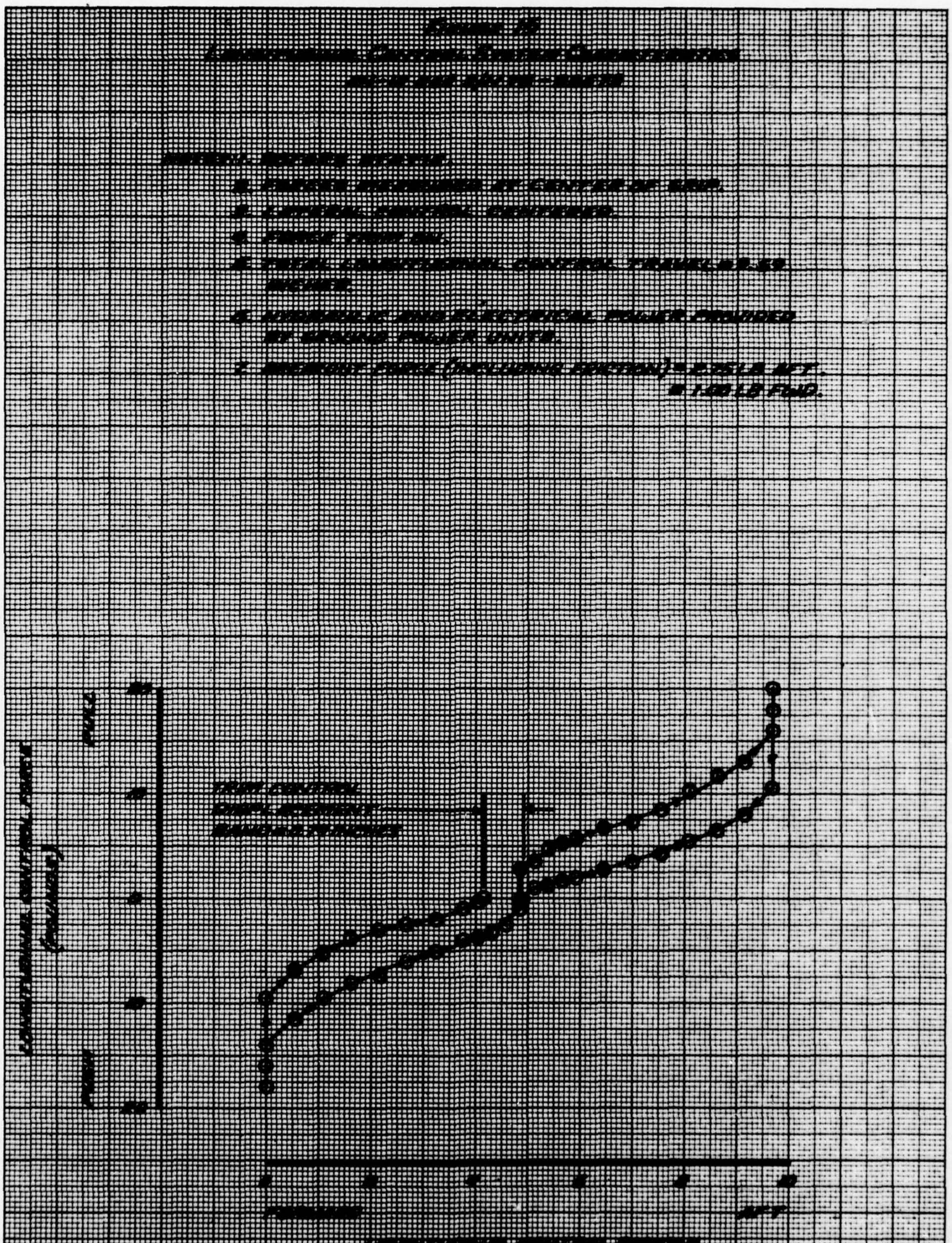
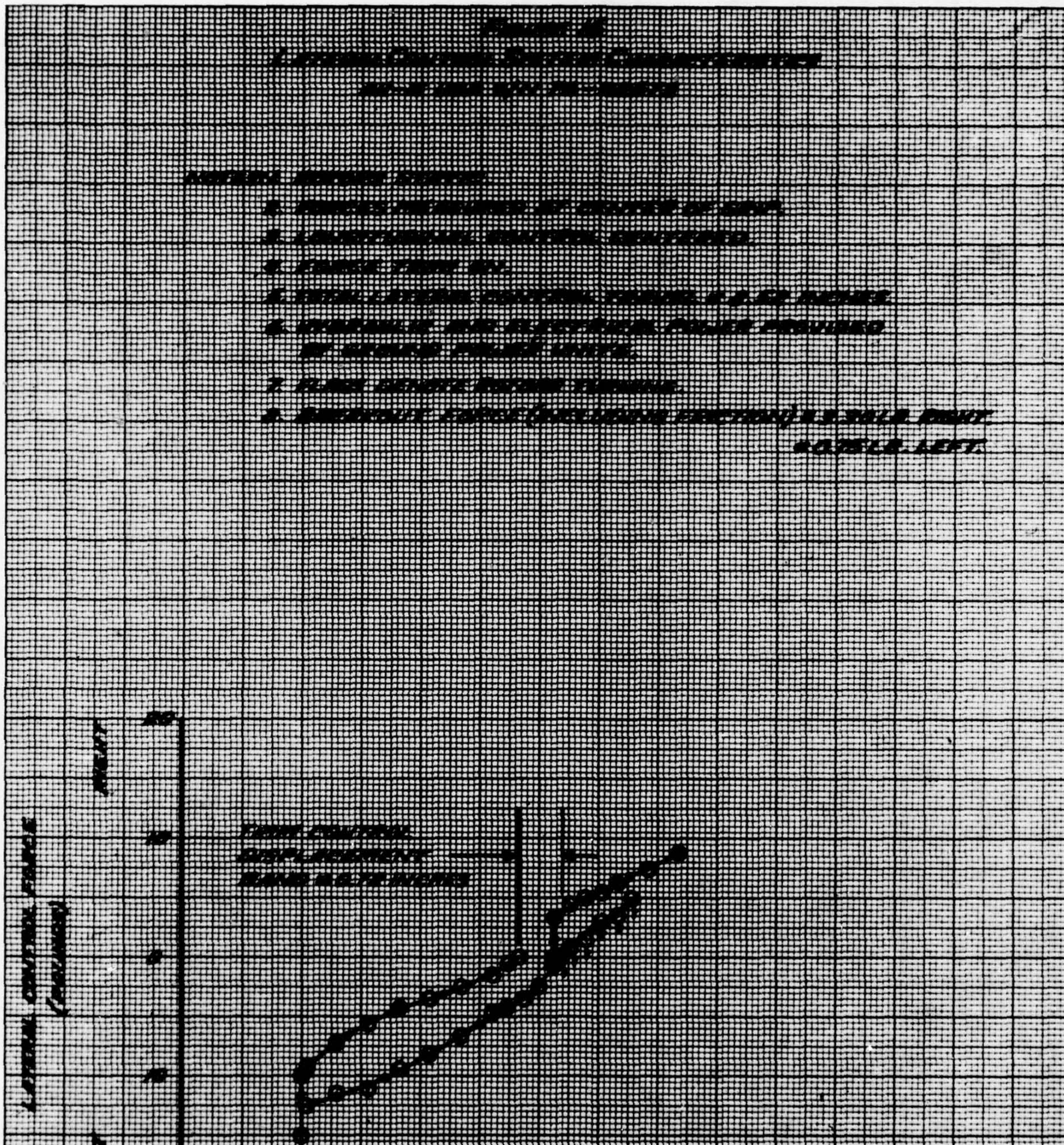
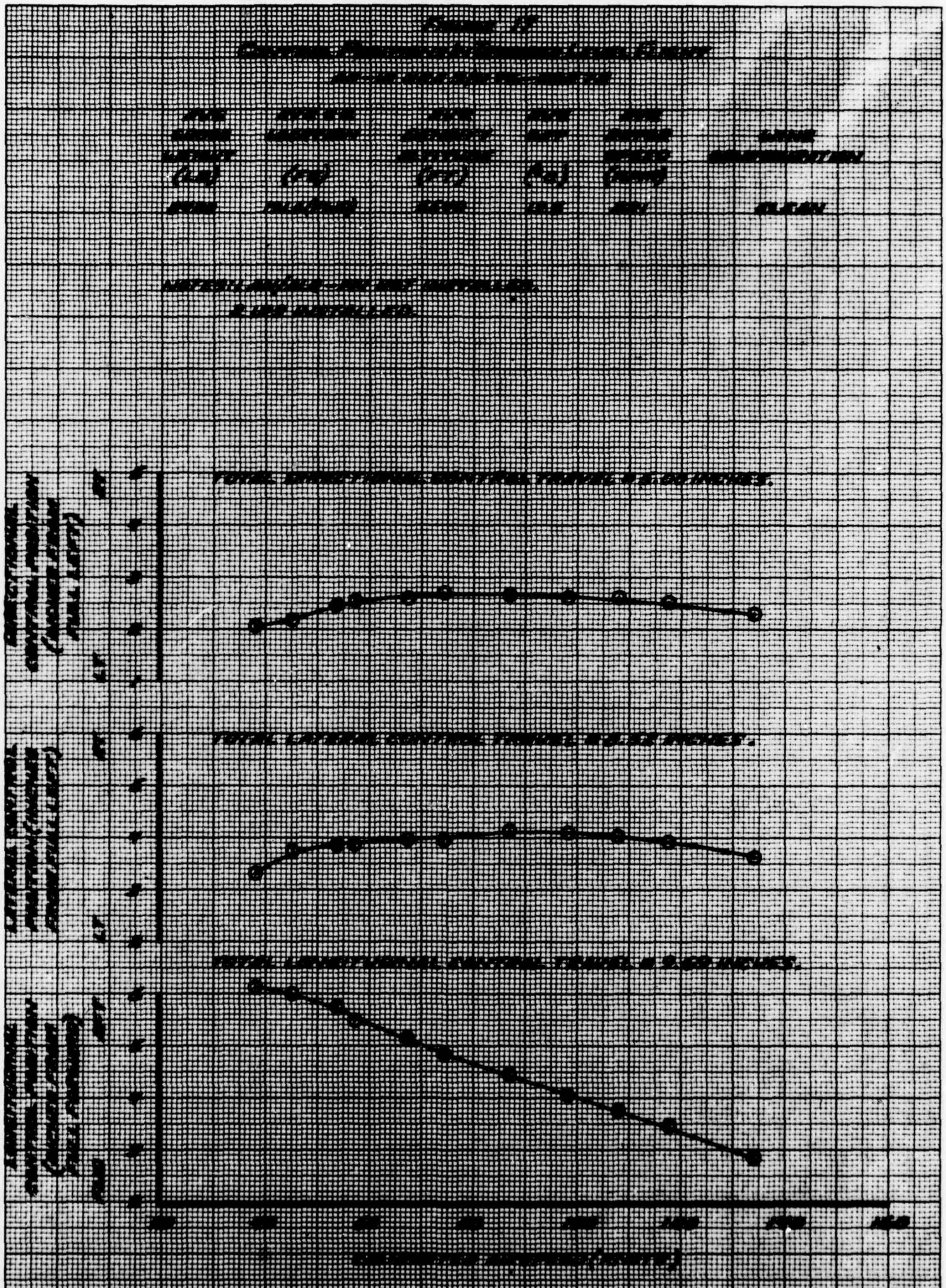
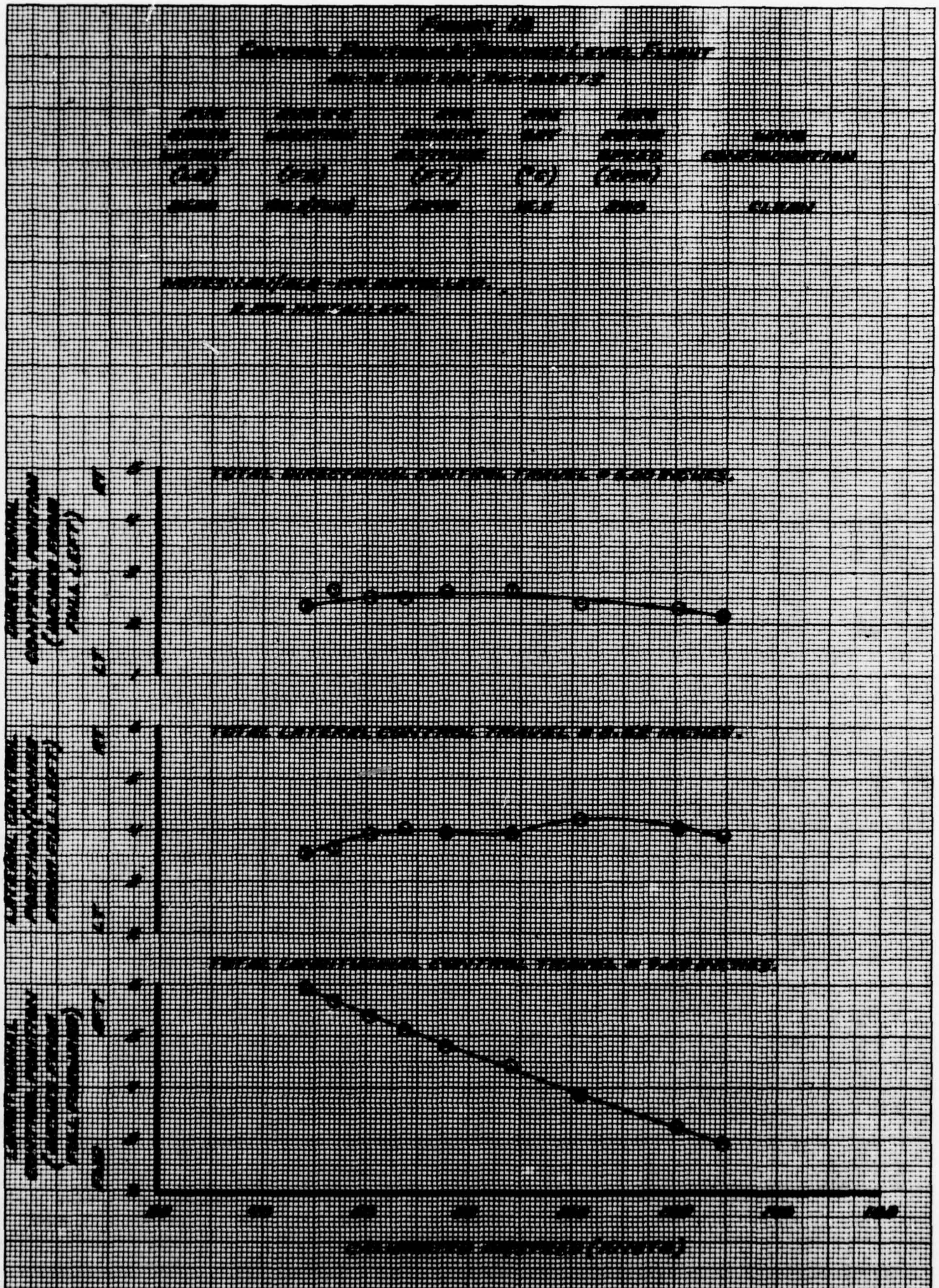
0.7

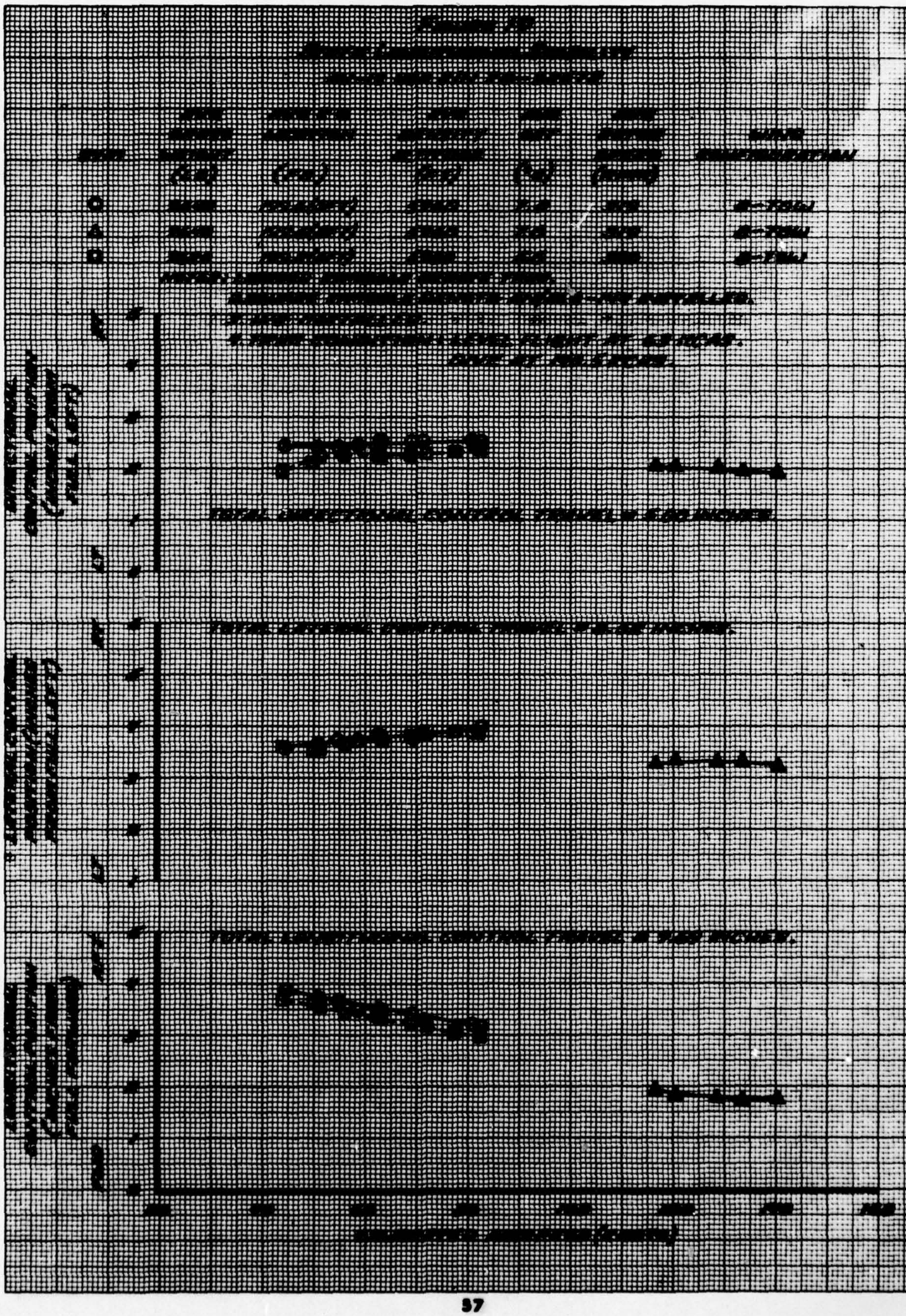


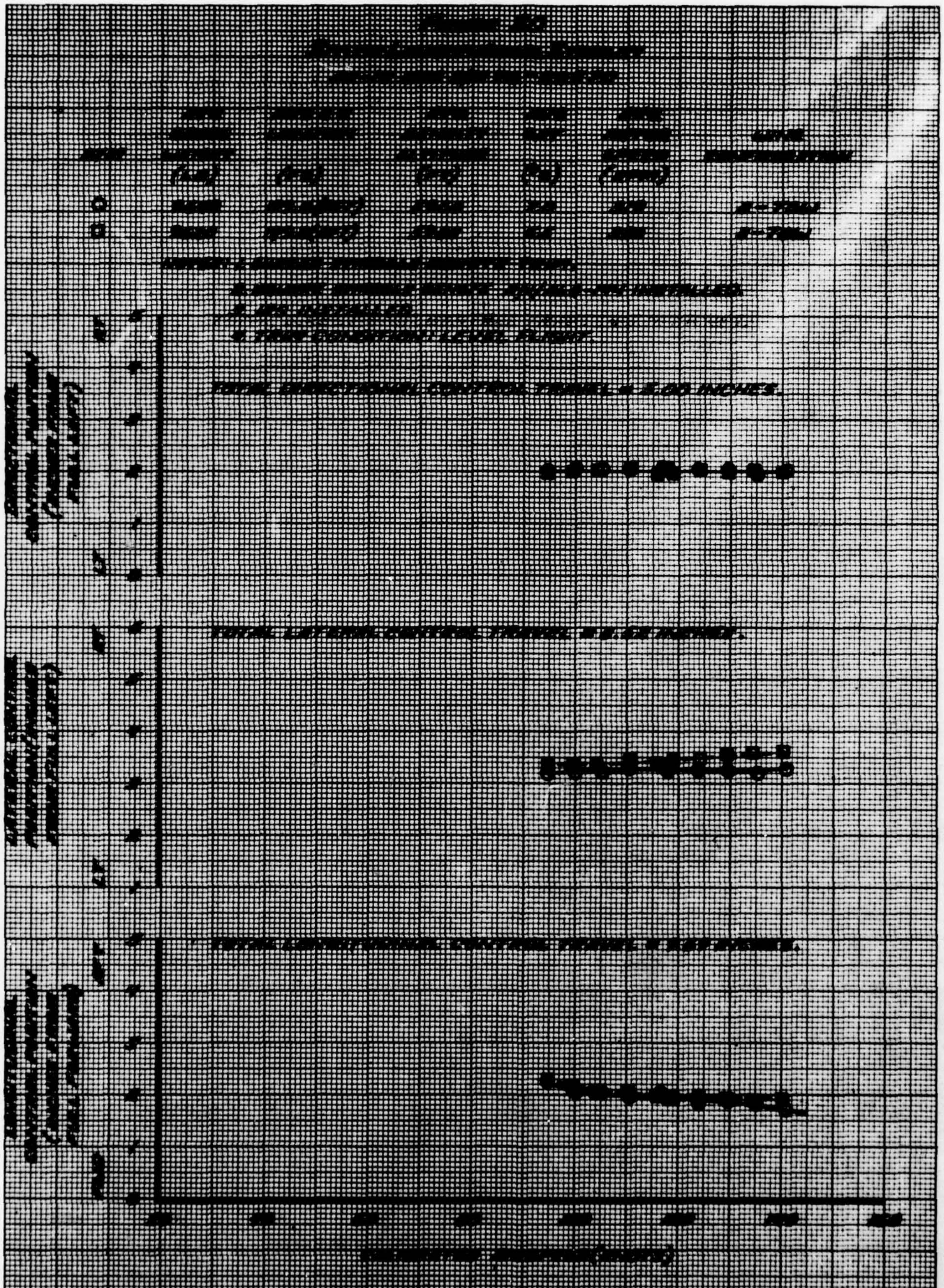


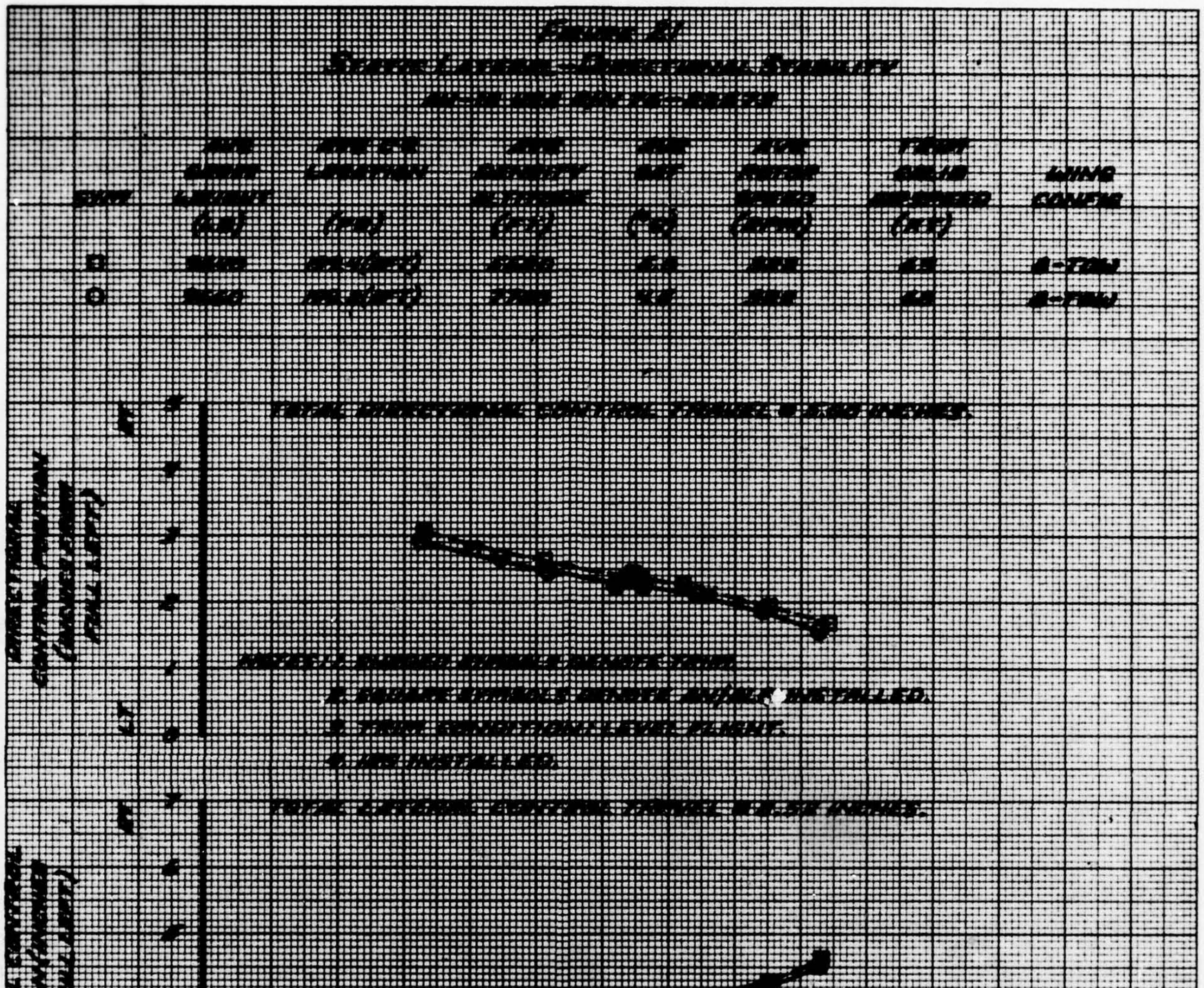

LEFT
RIGHT

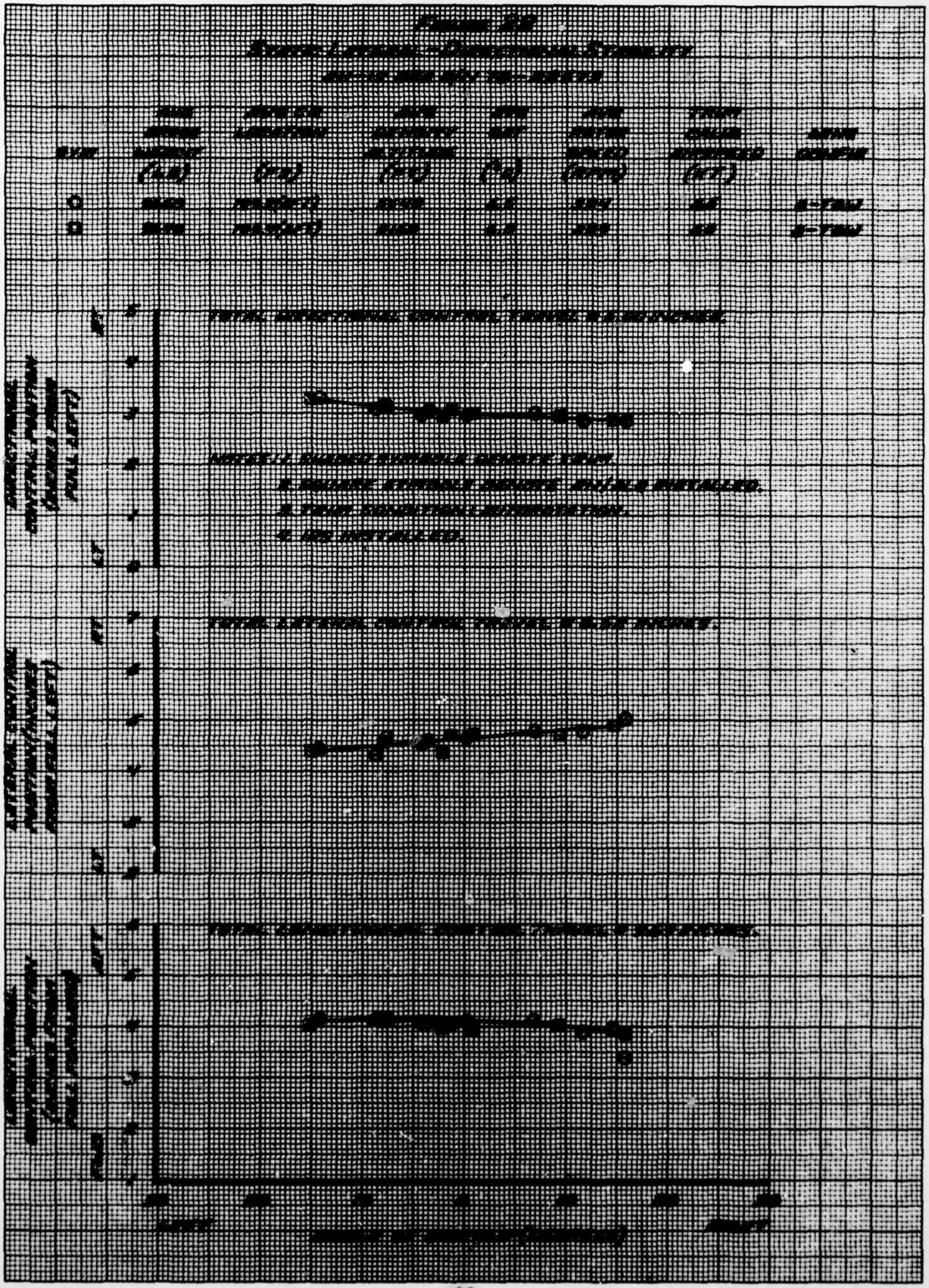


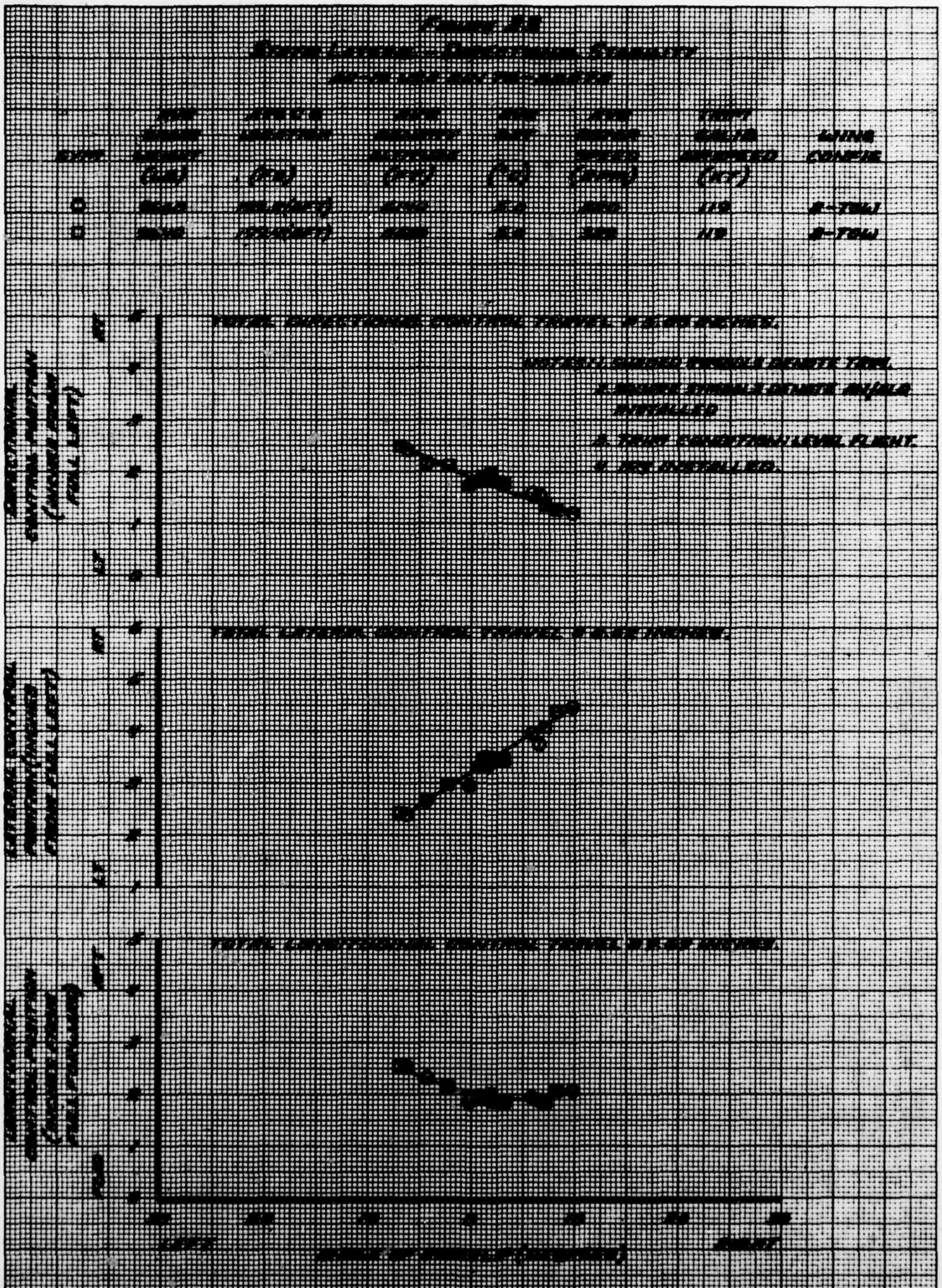


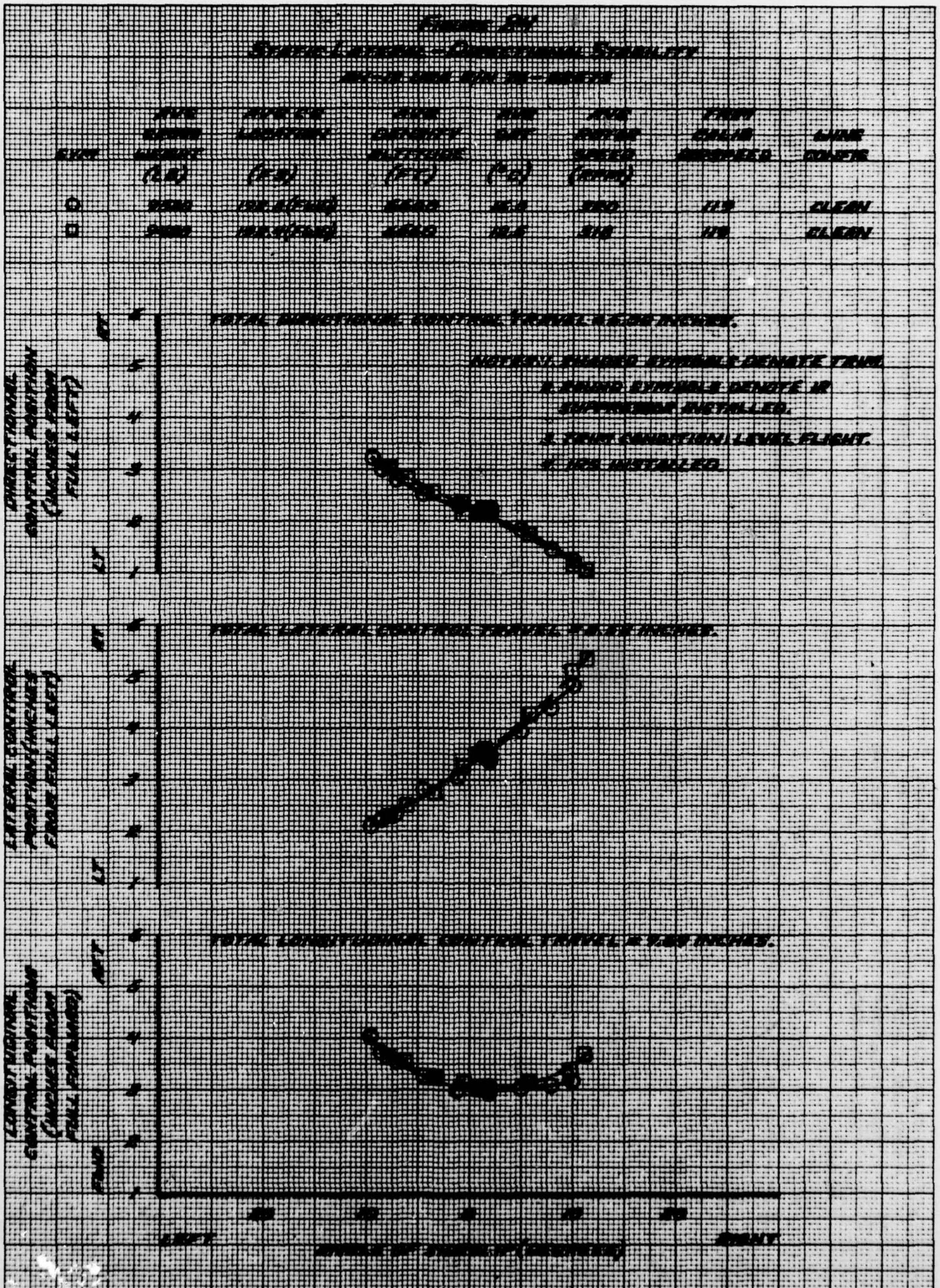


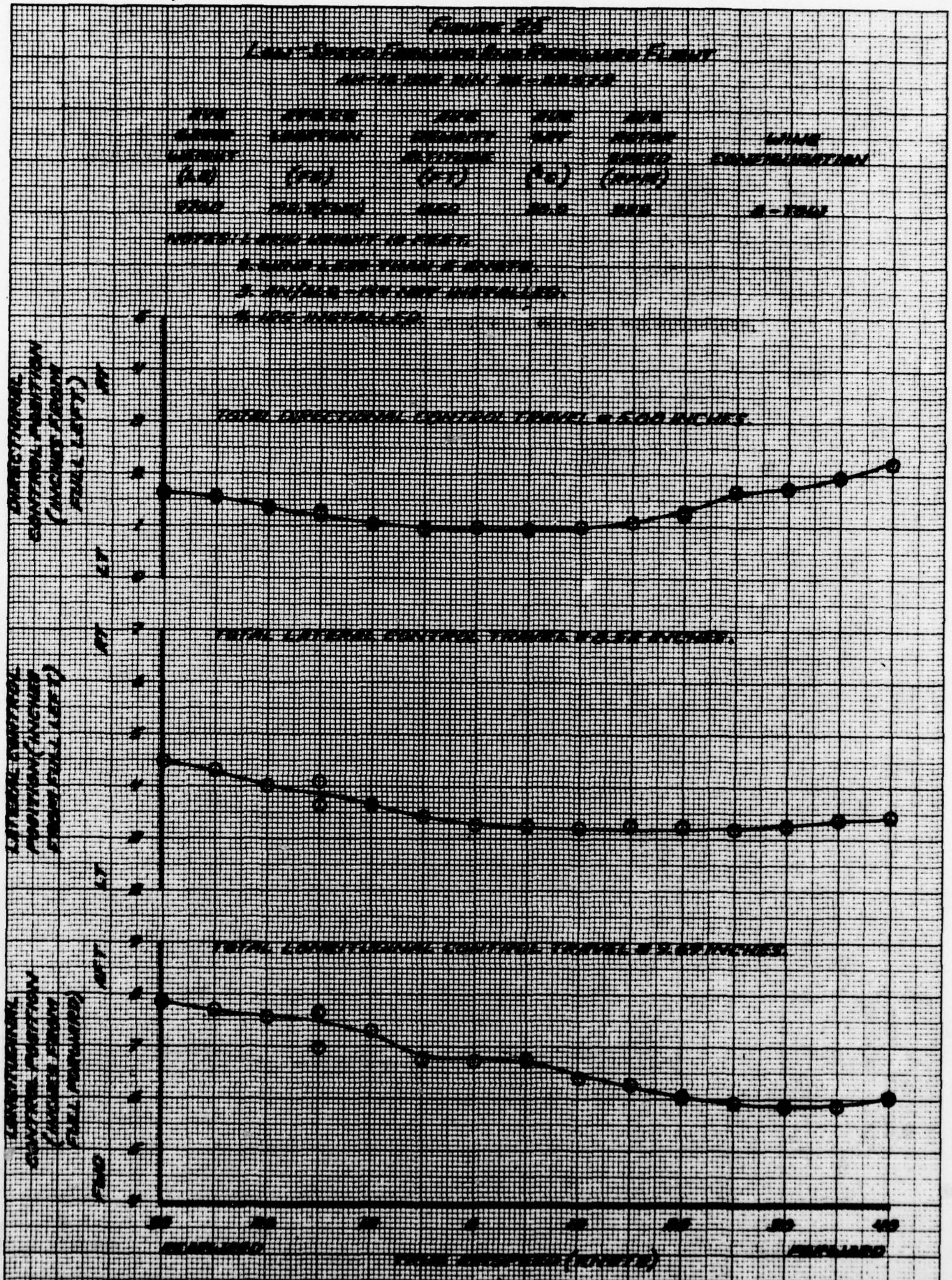


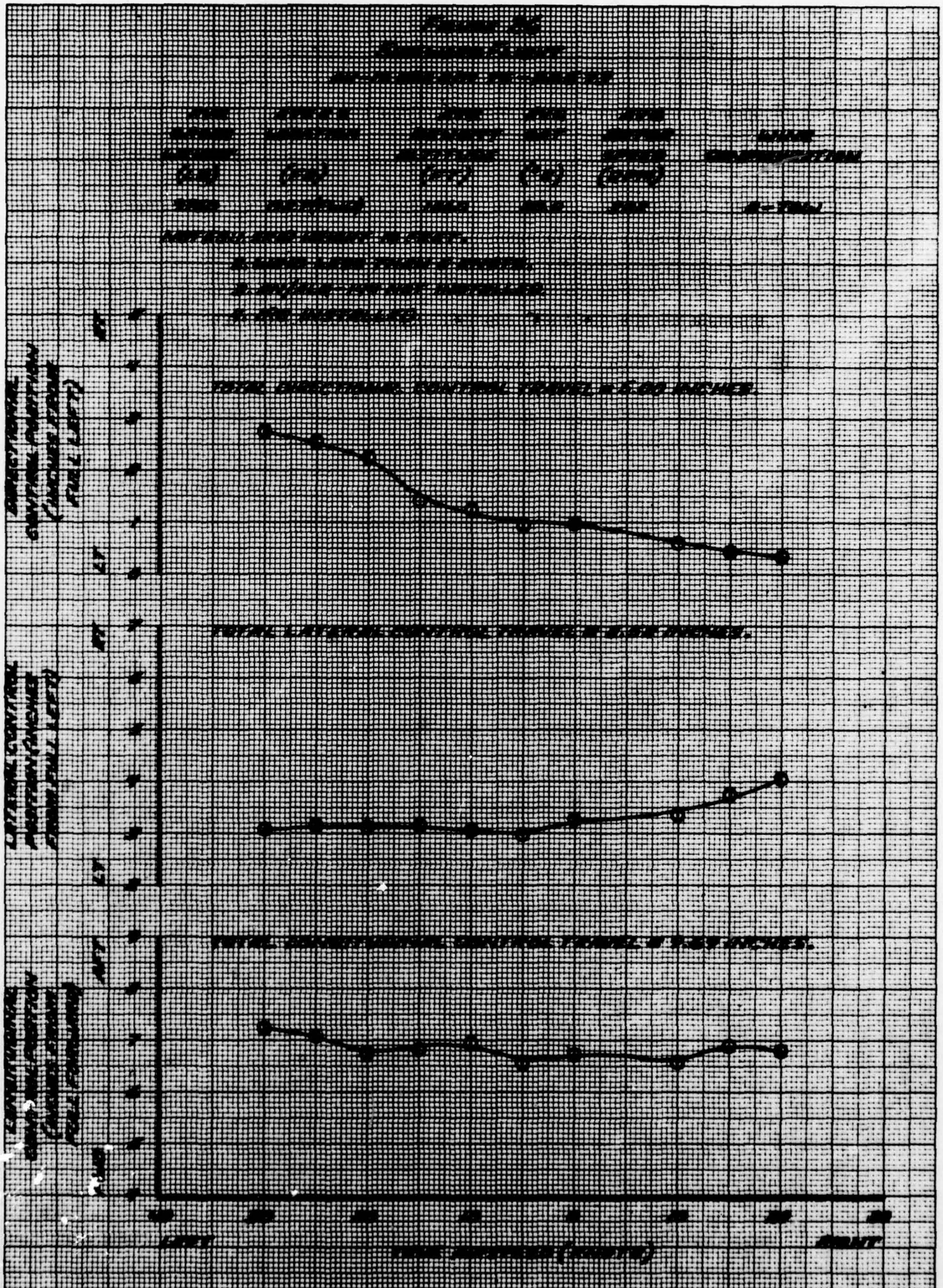





WING
CONFIG
(KT)
LEFT
RIGHT


STATIC LATERAL-DIRECTIONAL STABILITY
CLEAN
CLEAN
TOTAL DIRECTIONAL CONTROL TRAVEL
NOTES:
1. SHADED SYMBOLS DENOTE TRIM.
INSTALLED.
3. TRIM CONDITION: LEVEL FLIGHT.
INSTALLED.
TOTAL LATERAL CONTROL TRAVEL
TOTAL LONGITUDINAL CONTROL TRAVEL
LEFT
RIGHT
ANGLE OF SIDESLIP (DEGREES)





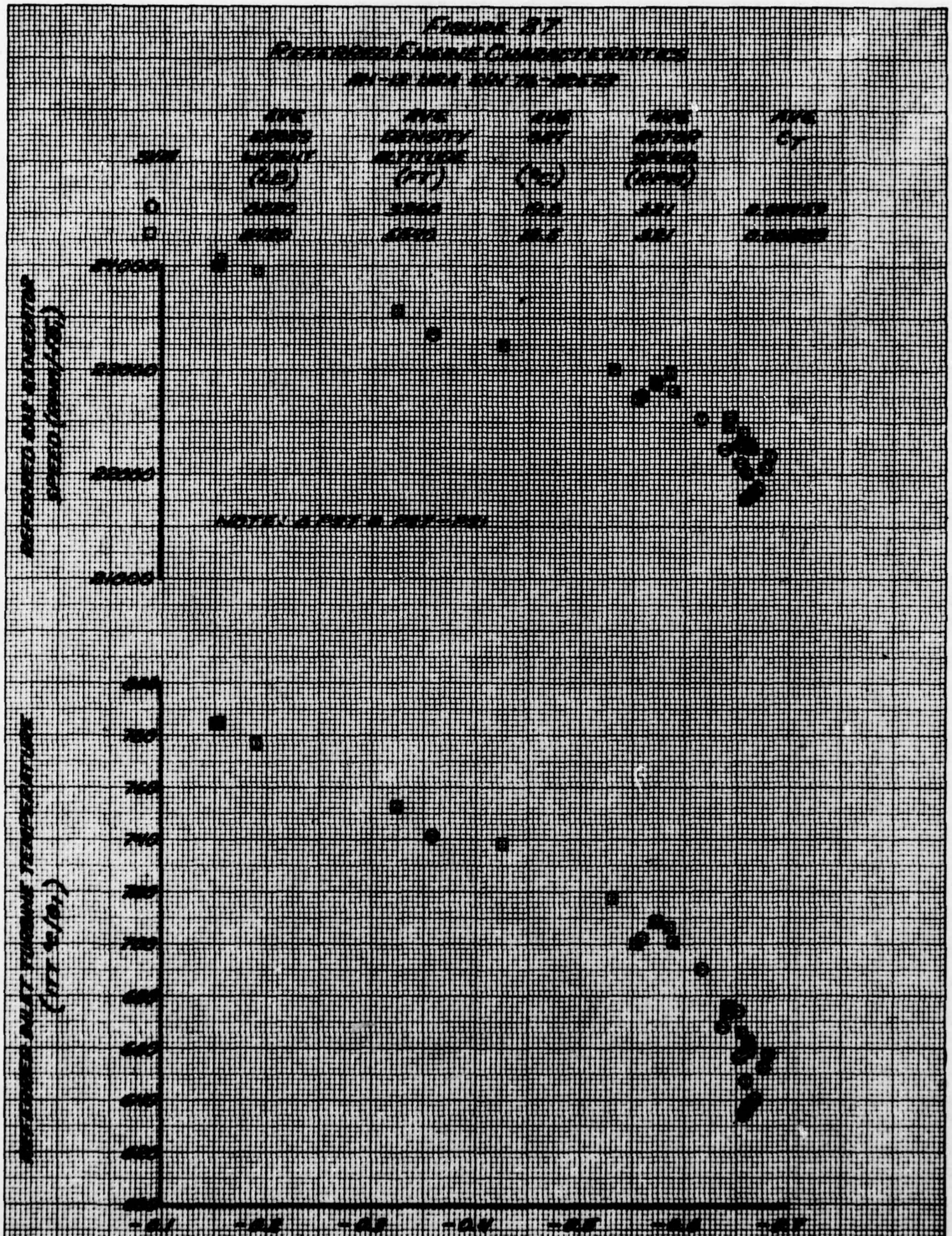
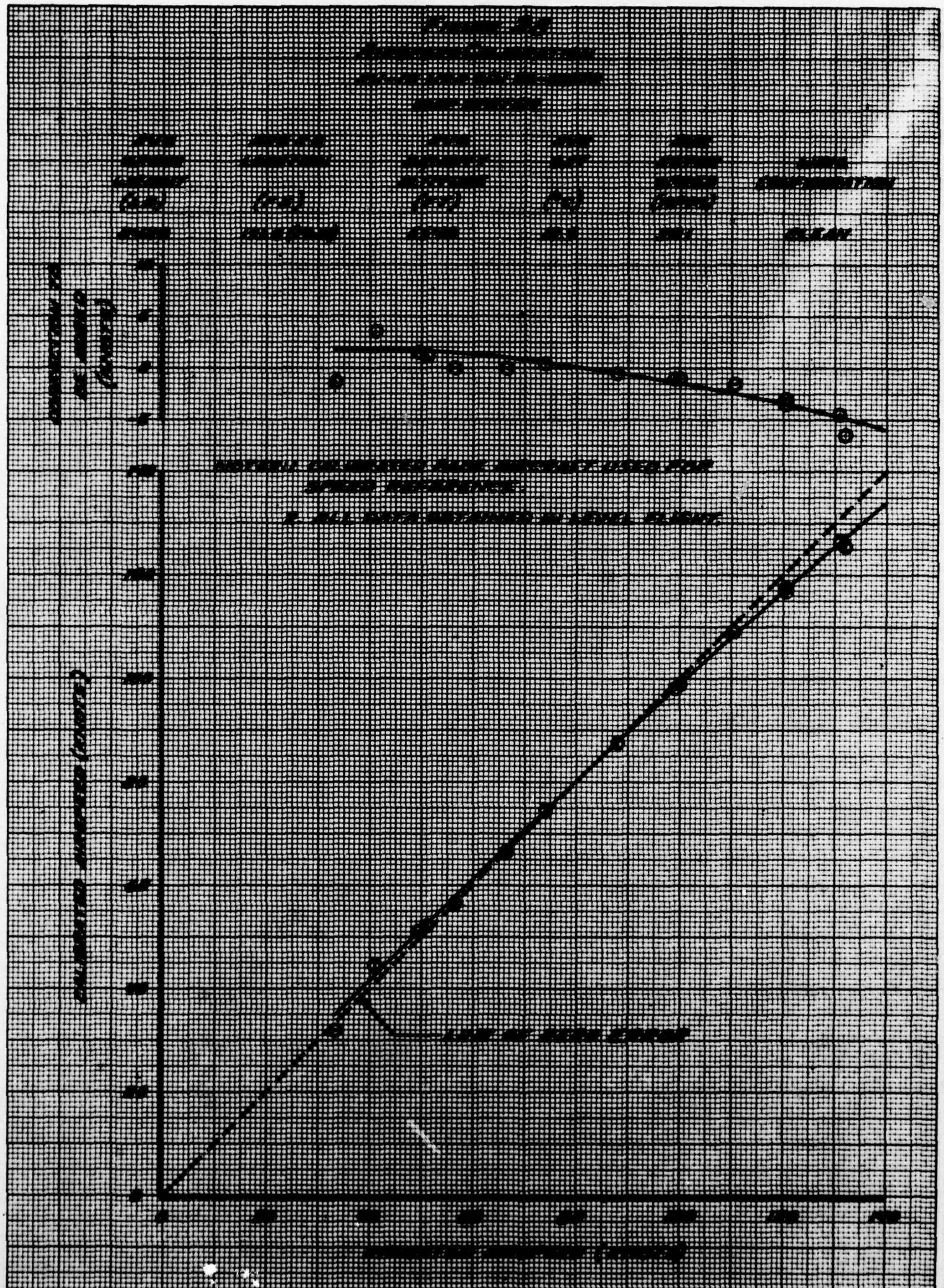


FIGURE 27
REFERENCE ENGINE CHARACTERISTICS



APPENDIX F. EQUIPMENT PERFORMANCE REPORTS

EQUIPMENT PERFORMANCE REPORT
(AVSCOM Reg 70-12)

DATE: 28 March 1978

OFFICE SYMBOL: DAVTE-TA

TO:
COMMANDER
US ARMY AVIATION SYSTEMS COMMAND
ATTN DRSAV-EQ
PO BOX 209
ST. LOUIS, MISSOURI 63166

FROM:
Commander
US Army Aviation Engineering Flt Acty
ATTN: DAVTE-TA
Edwards AFB, CA 93523

1. EPR NUMBER	2. PROJECT NUMBER	3. TEST TITLE
77-33-02	77-33	PAE of AH-1S with Infrared Suppression System

SECTION A - MAJOR ITEM DATA

4. MODEL: AH-1S	5. SERIAL NO: 76-22573
6. QUANTITY: 1 each	7. MANUFACTURER: Bell Helicopter Textron

SECTION B - PART DATA

8. NOMENCLATURE/DESCRIPTION: SCAS System AH-1S (Pitch Card - SCAS Amplifier)	9. NSN:
10. MFR PART NO:	11. MANUFACTURER: Bell Helicopter Textron
12. QUANTITY:	13. NEXT ASSEMBLY:

SECTION C - INCIDENT DATA

14. OBSERVED DURING	15. TEST ENVIRONMENT	16. INCIDENT CLASS	17. ACTION TAKEN
X a. OPERATION	Flight in calm air above 100 kts	a. DEFICIENCY	X Reinstalled
b. MAINTENANCE		b. SHORTCOMING	REPAIRED
c.		c. SUGGESTED IMPROVEMENT	ADJUSTED
		X d. OTHER Observation/ Maintenance procedure	DISCONNECTED
			X REMOVED, Cleaned
			NONE

18. DATE AND HOUR OF INCIDENT

SECTION D - INCIDENT DESCRIPTION

19. DESCRIBE INCIDENT FULLY (Deficiencies and Shortcomings are subject to reclassification)

On the first flight of the day, while accelerating through 100 KIAS, a moderate vertical vibration was observed. The frequency was approximately 1/2 the blade passage frequency (1/2 per. rev.). The vibration had not been observed on previous test flights. The cause of the vibration was isolated to the pitch channel of the SCAS system. Upon turning off the pitch channel of SCAS the vertical vibration ceased immediately. When the pitch channel was re-engaged, the vertical vibrations would build over a period of approximately 30 seconds to the original magnitude. The vertical vibrations were not apparent at airspeeds below 100 KIAS. After termination of the flight, the matter was discussed with the contractor (AiResearch Inc.), who informed us that they had had a similar problem during their flight program the preceding month, and that cleaning the contacts of the pitch card in the SCAS amplifier had alleviated the problem. The contacts were cleaned and the vertical vibrations have not reappeared.

20. DEFECTIVE MATERIAL SENT TO

21. NAME, TITLE, TEL EXT OF PREPARER
SHERWOOD C. SPRING,
Project Pilot, Project 77-33 76336

SIGNATURE: JCW

SAV Form 1002
1 May 76

Edition of 1 Aug 74 may be used

EQUIPMENT PERFORMANCE REPORT
(AVSCOM Reg 70-12)

DATE: 28 March 1978

OFFICE SYMBOL: DAVTE-TA

TO:	FROM:
COMMANDER US ARMY AVIATION SYSTEMS COMMAND ATTN: DRSAV-EQ PO BOX 209 ST. LOUIS, MISSOURI 63166	Commander US Army Aviation Engineering Flt Acty ATTN: DAVTE-TA Edwards AFB, CA 93523

1. EPR NUMBER	2. PROJECT NUMBER	3. TEST TITLE
77-33-01	77-33	PAE of AH-1S with Infrared Suppression System

SECTION A - MAJOR ITEM DATA

4. MODEL: AH-1S	5. SERIAL NO: 76-22573
6. QUANTITY: 1 each	7. MANUFACTURER: Bell Helicopter Textron

SECTION B - PART DATA

8. NOMENCLATURE/DESCRIPTION: Cockpit Ventilation System	9. NSN:
10. MFR PART NO:	11. MANUFACTURER: Bell Helicopter Textron
12. QUANTITY:	13. NEXT ASSEMBLY:

SECTION C - INCIDENT DATA

14. OBSERVED DURING	15. TEST ENVIRONMENT	16. INCIDENT CLASS	17. ACTION TAKEN
X a. OPERATION	Hover on unprepared surface (dust/dirt environment)	X a. DEFICIENCY	REPLACED
b. MAINTENANCE		b. SHORTCOMING	REPAIRED
c.		c. SUGGESTED IMPROVEMENT	ADJUSTED
		d. OTHER	DISCONNECTED
			REMOVED
			X NONE

18. DATE AND HOUR OF INCIDENT: Flight Test Program, March 1978

SECTION D - INCIDENT DESCRIPTION

19. DESCRIBE INCIDENT FULLY *(Deficiencies and Shortcomings are subject to reclassification)*

The pilot station ventilation system on the production AH-1S consists of a main vent control knob and two rectangular vent outlets located on the left and right sides of the instrument console. Vent outlets are non-adjustable, having fixed vanes which direct airflow to the pilot's face and chest. Separate on/off vent controls are provided at each outlet, but are poorly designed and easily damaged as evidenced by the broken right side control on the test aircraft. Discussion with AH-1S Instructor Pilots from Ft. Rucker revealed that broken vent controls are common on the AH-1S helicopter. When hovering over unprepared surfaces, the dust and dirt raised by the rotor downwash is readily ingested into the aircraft vent system and forcibly ejected onto the pilot's face and chest. Even with a properly fitted helmet and with the visor full down, sufficient dust and dirt can be blown into the pilot's eyes to cause temporary blindness and eye damage with the obvious potential for loss of visual reference and aircraft damage. Operation from unprepared surfaces is an integral portion of the AH-1S mission, and as such, the present non-adjustable ventilation system is a potential hazard and should be modified as soon as possible.

20. DEFECTIVE MATERIAL SENT TO:

21. NAME, TITLE, TEL EXT OF PREPARER	SIGNATURE
SHERWOOD C. SPRING Project Pilot, Proj. 77-33 76336	

AV Form 1002, 1 May 76 — Edition of 1 Aug 74, may be used.

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US Army Aviation Research and Development Command (DRDAV-EQ)	12
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