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USAAVSCOM PROJECT NO. 68-51
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AIRWORTHINESS AND FLIGHT QUALIFICATION TESTS
RU-8D AIRPLANE
WINEBOTTLE CONFIGURATION

PHASE II

FINAL REPORT

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OCTOBER 1970

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

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ABSTRACT

The limited airworthiness and flight qualification test (Phase D) evaluation of the RU-8D airplane Winebottle configuration was conducted to obtain quantitative handbook data for accurate and safe mission planning. The tests included level flight, landing and takeoff performance; stalls and single-engine characteristics; and longitudinal and lateral-directional handling qualities. Forty-eight test flights were flown for a total of 51 productive flight test hours. Two shortcomings were noted for which correction is desirable to improve mission effectiveness: poor sensitivity of the aileron trim and the masking of the longitudinal control force gradient by the breakout forces. Within the scope of this test, the performance capabilities and the handling qualities of the RU-8D are satisfactory for the reconnaissance mission.

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INTRODUCTION

BACKGROUND

1. The RU-8D airplane, produced by Beech Aircraft Corporation, is currently in Army use as a reconnaissance vehicle. Ling-Temco-Vought Corporation has designed an airfoil-type antenna installation to be incorporated on Army Security Agency RU-8D airplanes. The US Army Aviation Systems Command (USAAVSCOM) directed the US Army Aviation Systems Test Activity (USAASTA) to conduct airworthiness qualification tests on the RU-8D with this antenna system installed. The airworthiness tests were to be conducted in two phases. The Phase I tests consisted of a qualitative evaluation conducted at Lakehurst Naval Air Station, New Jersey, and were completed on 21 November 1968 (ref 1, app I). The Phase II tests were conducted to acquire quantitative flight test data to complete the airworthiness qualification of the system (ref 2).

TEST OBJECTIVES

2. The objectives of this test were to evaluate the airplane's performance and stability and control characteristics and to obtain quantitative flight test data. Areas of investigation included, but were not limited to:

a. Power on and power off stall speeds.

b. Single-engine minimum control speeds.

c. Level flight performance.

d. Takeoff and landing performance to clear a 50-foot obstacle.

e. Stability and control characteristics in the landing and takeoff configurations.

DESCRIPTION

3. The RU-8D airplane is an all-metal, low-wing monoplane powered by two supercharged Lycoming O-480-1 engines. Side-by-side seating and dual-flight controls and instruments are provided. Distinguishable features of the airplane are square-tipped wings and tail surfaces, three-bladed propellers and a retractable tricycle

landing gear. The RU-8D also has wing tip extensions that increase the wing span to 50.2 feet versus 45.3 feet for the U-8D. The antenna system consists of two dipoles mounted vertically near mid-chord of each wing approximately 2½ feet inboard of the wing tips. The antennae have an airfoil cross-section design and are approximately 1 inch thick, 6 inches wide and 7 feet long. A spoiler is flush mounted on the leading edge of each antenna to eliminate antenna vibration. The RU-8D with this antenna system and electronic components installed has been designated as the Winebottle configuration. The mission of the RU-8D Winebottle configuration is classified.

SCOPE OF TEST

4. The RU-8D airplane was evaluated as a reconnaissance airplane with an instrument flight capability. Where applicable, the airplane's handling qualities were qualitatively and quantitatively evaluated as a Class I airplane against the requirements of reference 3, appendix I (hereafter referred to as the specification). Forty-eight test flights were conducted for a total of 51 productive flight test hours. The tests were conducted at Edwards Air Force Base (AFB), Bakersfield, and Bishop, California. The flight restrictions and operating limitations contained in the operator's manual (ref 4) were observed during the tests. The airplane test conditions and configurations are presented in appendixes II and III, respectively.

METHOD OF TEST

5. The engineering flight test methods used for these tests are contained in references 5 and 6, appendix I, and are described briefly in the Results and Discussion section of this report. Appendix IV contains a list of the test instrumentation. The test engines used in this program were certified engines, and power-required and fuel-flow data were derived from the engine model specification (ref 7, app I). Takeoff and landing data were obtained using a Fairchild flight analyzer camera. Qualitative ratings of handling qualities were based on the Handling Qualities Rating Scale (HQRS) presented as appendix V.

CHRONOLOGY

6. The chronology of the RU-8D test program is as follows:

Test directive received	20	December	1968
Test aircraft received	6	April	1969
Test instrumentation completed	28	May	1969
Flight test commenced	3	June	1969
Flight test completed	6	August	1969
Draft report submitted	12	March	1970

RESULTS AND DISCUSSION

PERFORMANCE

General

7. Performance tests were conducted on the RU-8D airplane in the Winebottle configuration to obtain quantitative data for inclusion in the operator's manual. The test program encompassed an evaluation of the airplane's maximum performance takeoff, level flight and maximum performance landing capabilities. Maximum performance takeoff tests were performed to determine the optimum flap setting and control technique yielding the shortest takeoff distance over a 50-foot obstacle. The shortest distances were 1310 and 1480 feet at density altitudes of sea level (SL) and 4000 feet. Level-flight performance tests were conducted to define the cruise speed, range and endurance characteristics of the airplane. The maximum endurance airspeeds determined for the airplane in the Winebottle configuration should be included in the operator's manual. The manual should also be revised to include the best-range data. The landing tests were performed to define the maximum landing performance of the airplane. The shortest landing distances over a 50-foot obstacle were 1710 and 1845 feet at density altitudes of SL and 4000 feet, respectively.

Takeoff Performance

8. Maximum performance takeoff tests were conducted under the conditions listed in appendix II. The tests were performed on dry, hard-surfaced runways to obtain the curves of true airspeed at liftoff versus ground roll distance and the true airspeed at a height of 50 feet versus the total horizontal distance required to attain this height above the runway. Each curve was developed by varying the rotation airspeed for each takeoff. After rotation and liftoff, pitch attitude was adjusted to maintain rotation airspeed through a height of 50 feet. All tests were performed using takeoff power, and brakes were released when the engine manifold pressures reached 40 inches Hg. Different flap settings were investigated to determine the optimum flap setting yielding the shortest takeoff distance. The landing gear was not retracted until after the airplane reached a height of 50 feet above the runway. During each takeoff series, ballast was added as fuel was consumed to maintain the test gross weight (grwt) and center of gravity (cg).

9. Test results are presented in figures 1 and 2, appendix VI. Test data disclosed no difference in takeoff distance for 10- and 20-degree flap settings at a given airspeed; however, lower rotation and climb speeds were attained with the 20-degree flap setting and resulted in a shorter takeoff distance. Takeoff distances were greatest with the flaps set at 30 degrees for a given airspeed. To achieve maximum takeoff performance, a rotation and climb airspeed of 75 knots indicated airspeed (KIAS) with a 20-degree flap setting is recommended. Slower rotation and climb airspeeds did not allow a sufficient margin for accelerated stall. The technique used during this test agrees with the technique recommended in the operator's manual. The operator's manual does not include maximum takeoff performance data; consequently, no comparison can be made from the results of tests. Table 1 is a summary of the maximum takeoff performance data.

Table 1. Maximum Takeoff Performance Summary.

Density Altitude (ft)	Gross Weight (lb)	Flap Setting (deg)	Indicated Airspeed at Liftoff (kt)	Indicated Airspeed at 50-Foot Height (kt)	Ground Roll (ft)	Total Distance at 50-Foot Height (ft)
SL	7350	20	75	75	990	1310
4000	7350	20	75	75	1215	1480

Level Flight Performance

10. Level-flight performance tests were conducted under the conditions listed in appendix II to determine the cruise speed, range and endurance capabilities. Tests were conducted for two-engine operation only. The test data were acquired using the pressure altitude test technique (ref 5, app I), and the data were reduced by the methods presented in appendix VII. The power-required data include installation power losses and the power required to drive engine accessories. Specific range data do not include the 5-percent increase in fuel-flow per Military Specification MIL-C-5011A (ref 8, app I). Results of these tests are presented in figures 3 through 10, appendix VI.

11. Table 2 shows a comparison of the maximum endurance airspeeds as presented in the operator's manual for the RU-8D airplane and as determined from the evaluation of the Winebottle configured

airplane. It is recommended that the operator's manual be revised to indicate the maximum endurance airspeeds for the RU-8D Winebottle configured airplane.

Table 2. Maximum Endurance Airspeed Summary.

Altitude (ft)	6500-Pound Gross Weight		7000-Pound Gross Weight	
	RU-8D Operator's Manual KIAS (kt)	RU-8D Winebottle Test Result KIAS (kt)	RU-8D Operator's Manual KIAS (kt)	RU-8D Winebottle Test Result KIAS (kt)
SL	78	85	80	88
5,000	78	85	80	88
10,000	79	85	81	88

12. All range data included in this report are based on the engine model specification fuel-flow data. While the operator's manual does not specify recommended best-range data for the RU-8D aircraft, the data shown in column V of the flight operations instruction charts, TM 55-1510-201-10/4 (ref 4, app I), were compared with the Winebottle configuration test results for similar operating conditions of gross weight, altitude and true airspeed. For a gross weight of 7100 pounds, test results indicated a 7.7 to 10.5 percent less range capability than that presented in the operator's manual; the specific range obtained during tests for a gross weight of 6600 pounds was less by 5 to 7.2 percent. The recommended best-range cruise airspeeds (the true airspeed corresponding to 0.99 maximum specific range) as determined from these tests are summarized in table 3. Test results disclosed no difference in recommended best-range cruise airspeed with variation in aircraft gross weight from 6600 to 7100 pounds at a given altitude.

Table 3. Summary of Recommended Best-Range Airspeed at Gross Weights of 6600 and 7100 Pounds.

Altitude (ft)	True Airspeed[1] (kt)	Calibrated Airspeed[1] (kt)
1,000	127	125
5,000	133	123
10,000	140	120

[1] Airspeed corresponding to 0.99 maximum specific range.

Landing Performance

13. Maximum performance landing tests were conducted under the conditions specified in appendix II. Tests were performed to obtain data to depict true airspeed at touchdown versus ground roll, and true airspeed at a 50-foot height versus the total horizontal distance required to bring the airplane to a full stop. Curves were developed by conducting a series of landings using various approach speeds. Sufficient power was maintained during each approach to hold a 400- to 600-foot-per-minute (fpm) rate of descent (R/D). Power was reduced to idle during the flare phase of the higher airspeed approaches and at touchdown during the lower airspeed approaches. In each landing test, an attempt was made to apply the maximum braking possible without skidding the tires. Ballast was added as fuel was consumed to maintain the test grwt and cg for each series of landings.

14. The results of the landing tests are presented in figures 11 and 12, appendix VI. Based on the data obtained during these tests, the optimum technique for maximum performance landings at a 6600-pound grwt is as follows:

a. Set flaps at 30 degrees.

b. Maintain approach speed of 75 KIAS. (Slower approach airspeeds did not allow an adequate margin for controllability or accelerated stall.)

c. Set power to maintain an approximate 500-fpm R/D.

d. Reduce power to idle immediately after touchdown.

e. Apply maximum braking without skidding the tires.

15. Table 4 is a summary of the maximum performance landing data. It is recommended that these data, plus the landing technique described in paragraph 14, be include in the operator's manual.

Table 4. Maximum Performance Landing Summary.

Density Altitude (ft)	Gross Weight (lb)	Flap Setting (deg)	Indicated Airspeed at 50-Foot Height (kt)	Indicated Airspeed at Touchdown (kt)	Ground Roll (ft)	Total Distance From 50-Foot Height (ft)
SL	6600	30	75	75	960	1710
4000	6600	30	75	75	1080	1845

STABILITY AND CONTROL

General

16. Stability and control tests were conducted to define the stall and single-engine characteristics and also the longitudinal and lateral-directional handling qualities. The handling qualities of the test airplane were satisfactory for the reconnaissance mission. However, two shortcomings were noted for which correction is desirable to improve mission effectiveness: the poor sensitivity of the aileron trim and masking of the longitudinal control force gradient by breakout forces.

Control System Characteristics

17. Control system free play was measured in flight and was negligible in the longitudinal, lateral and directional controls. The response of control surfaces following rapid inputs to the wheel or rudder pedals was essentially deadbeat. The longitudinal and lateral controls exhibited positive centering in that the controls returned to the trim condition when released following a rapid displacement. The directional control did not exhibit positive centering, but this characteristic was not objectionable. Longitudinal breakout forces, including friction, were measured from oscillograph records and by a hand-held force gage. The results are presented

in table 5 and are within the ½- to 4-pound limits of the specification. The longitudinal friction forces were essentially zero. The control system characteristics evaluated during this test were satisfactory (HQRS 3) and met the requirements of the specification.

Table 5. Summary of Longitudinal Breakout Forces.[1]

Configuration	Calibrated Airspeed (kt)	Pull Force (lb)	Push Force (lb)
Cruise	87	3.7	2.0
Cruise	107	4.0	2.5
Power approach (no flaps)	120	3.0	1.9
Power approach (30-degree flaps)	97	3.1	2.7

[1] Including friction.

Trimmability

18. The trimmability characteristics of the airplane were evaluated about all axes during each flight test. The rates of operation and sensitivity of the longitudinal and directional trim controls were satisfactory (HQRS 2). The trim speed band measured during the static longitudinal stability tests was approximately 1 knot. The rate of operation of the aileron trim control was satisfactory, but the sensitivity was poor and moderate pilot effort was required to trim the airplane laterally (HQRS 4). Increased sensitivity of the aileron trim is desirable for increased mission effectiveness. The trimmability characteristics met the requirements of the specification and, except for the poor sensitivity of the aileron trim, are satisfactory for the airplane's mission.

Static Longitudinal Stability

19. Static longitudinal stability tests were conducted under the conditions listed in appendix II, and the results are presented in figures 13 through 22, appendix VI. The data show that the longitudinal control force stability was positive (stable) for all conditions tested. In the cruise configuration, the control force

gradient increased positively as the trim speed was decreased; and there was essentially no change in the gradients between the two centers of gravity and gross weights tested. For all configurations tested, the control force gradients close to the trim airspeed were masked by the breakout forces; and, in some cases, a control force reversal accompanied a change in airspeed. This characteristic resulted in elimination of control force as a cue for accurate small airspeed changes. This characteristic is objectional, particularly for instrument approaches (HQRS 4), and improvement is desirable for increased mission effectiveness. Elevator position stability was slightly positive to neutral under all conditions tested; however, this characteristic was not objectionable in itself because of the positive control force gradient. The longitudinal control force stability met the requirements of the specification. With the exception of the masking of the longitudinal control force gradient by the breakout forces, the static longitudinal stability characteristics are satisfactory for mission accomplishment.

Dynamic Longitudinal Stability

20. The short-period characteristics of the airplane were investigated under the conditions listed in appendix II. The airplane was initially trimmed at the desired test airspeed. Airspeed was decreased by increasing pitch attitude, then increased by applying forward longitudinal control and allowing the airplane to dive. A pull-up was performed from the dive, and an approximate 2g normal acceleration was reached as the airplane approached trim airspeed and altitude. When the pitch attitude reached trim, the longitudinal control was returned to the trim position and released. The resultant airplane motion was recorded on an oscillograph. Under all conditions tested, the short-period response was essentially deadbeat. The long-period characteristics were also evaluated under the conditions listed in appendix II. A representative time history of a long period is presented in figure 23, appendix VI. During cruise flight at high airspeeds, mild atmospheric disturbances did not tend to excite the long period. When flying at endurance speeds, minimal pilot compensation was required to maintain an exact airspeed when the airplane was disturbed by a wind gust (HQRS 3). The long- and short-period characteristics met the requirements of the specification and are satisfactory for the airplane's mission.

Static Lateral-Directional Stability

21. Static lateral-directional stability was evaluated by performing steady-heading sideslips under the conditions listed in appendix II. The test results are presented graphically in figures 24 through 32, appendix VI. Positive static directional stability was indicated by the variations of rudder position and force with

sideslip. The directional control position and force gradients were essentially linear and positive and were satisfactory. The variations in aileron position and lateral control force with sideslip were also essentially linear and positive and indicated positive dihedral effect. The side-force characteristics as indicated by the variation in bank angle with sideslip were positive and satisfactory. The data reflect an increase in directional stability, dihedral effect and side-force characteristics with increasing airspeed. The static lateral-directional characteristics met the requirements of the specification and are acceptable for the airplane's mission (HQRS 3).

Dynamic Lateral-Directional Stability

22. The dynamic lateral-directional stability characteristics were evaluated by releasing the airplane from a steady-heading sideslip, then neutralizing and holding the controls fixed while recording the resultant motion. The test was conducted under the conditions listed in appendix II. Under all conditions, damping of the lateral-directional oscillation was satisfactory and met the requirements of the specification. A representative time history of the lateral-directional oscillation following release from steady-heading sideslip is presented in figure 33, appendix VI. The lateral-directional mode was easily excited while flying in turbulent air. The resultant oscillation was primarily about the yaw axis and would normally be described qualitatively as a "snaking motion." Although damping was satisfactory, the oscillation was annoying and distracting, particularly during night or instrument flights (HQRS 4).

Spiral Stability

23. Spiral stability was investigated in the cruise and power approach configurations under the conditions listed in appendix II. Assymmetric power was used to initially establish a small bank angle; then the power settings were rematched, and the resultant lateral motion was recorded. Under all conditions tested, the spiral mode was neutral or slightly convergent (HQRS 3). A representative time history is presented as figure 34, appendix VI. The spiral stability characteristics in the cruise and power approach configurations met the requirements of the specification and are satisfactory for the airplane's mission.

Single-Engine Characteristics

24. Single-engine tests were performed to determine the minimum control airspeeds (static and dynamic) and the minimum trim airspeed. The test results are presented in table 6.

Table 6. Single-Engine Characteristics.

Test	Gear Position	Flap Position (deg)	Power Setting (rpm)	Power Setting (in. Hg)	Test Results[1]
Static V_{MC}[2]	Down	0 and 20	[3]3400	46.5	85 KCAS (limited by directional control)
Dynamic V_{MC}	Down	0 and 20	[3]3400	46.5	90 KCAS
Minimum trim speed	Up	0	[4]2900	36.3	101 KCAS (limited by directional trim)
Minimum trim speed	Down	0	[4]2750	38	100 KCAS (limited by directional trim)

[1] Average test gross weight of 6700 pounds and density altitude of 5000 feet.
[2] Minimum control airspeed.
[3] Left engine propeller windmilling.
[4] Left engine propeller feathered.

25. It should be noted that the static and dynamic minimum control airspeeds are well above the 75-KCAS maximum performance takeoff airspeed recommended in paragraph 9. However, the design mission of the RU-8D normally does not require short takeoff and landing (STOL) performance. Further, the airplane is usually operated from airfields of sufficient size to effect a takeoff within the single-engine operating envelope. Because of these considerations, the minimum control speeds are acceptable for mission accomplishment.

26. The V_{MC} for the static condition was determined with the critical (left) engine shut down and unfeathered by decreasing the airspeed at a rate of approximately 1 knot per second until a lack of control was experienced. Bank angle toward the operating engine was 5 degrees or less. Static V_{MC} was 85 KCAS with full right directional control applied. A further decrease in airspeed resulted in an uncontrollable left yaw. The airplane failed to meet the requirements of paragraph 3.4.12 of the specification in that directional control could not be maintained for all airspeeds above $1.2V_{STO}$ (stall takeoff).

27. Dynamic responses to sudden engine failure were evaluated by stabilizing the airplane in steady-heading balanced flight, then failing the left engine by fully retarding the mixture control.

All flight controls were held fixed for 1 second before initiating recovery to the original heading and airspeed. The airplane's response was a rapid left yaw followed by a left roll. Recovery was effected using rudder and aileron controls. Dynamic V_{MC} was qualitatively determined to be 90 KCAS. In the determination, the ease of regaining and maintaining control of the airplane was taken into account, and an adequate safety margin was allowed for average pilot skill and proficiency.

28. Minimum trim airspeeds were determined with the left engine shut down and the propeller feathered. The minimum trim airspeed was the slowest airspeed in stabilized, wings-level, steady-heading flight where all control forces could be trimmed to zero. In both configurations evaluated, the directional axis was the limiting trim axis.

Stall Characteristics

29. Stall tests were performed to determine stall airspeeds and to evaluate the airplane's handling qualities associated with the stall. The test conditions are listed in appendix II. The test was conducted by stabilizing the airplane in balanced flight at the desired trim airspeed, then reducing airspeed at a rate of approximately 1 knot per second until stall occurred. The airspeeds for stall warning horn actuation, airframe buffet and stall are presented in tables 7 and 8.

Table 7. Stall Airspeeds of the RU-8D, S/N 57-6063, at a Gross Weight of 6620 Pounds.

Configuration	Roll Angle (deg)	Trim KCAS (kt)	Warning KCAS (kt)	Buffet KCAS (kt)	Stall KCAS (kt)
Takeoff	0 30 L 30 R	-- Note[1]	61 61 63	57 64 62	52.5 58 57
Cruise	0 30 L 30 R	90 90 90	79 89.5 87	82 90 88	70 78 76
Cruise	0 30 L 30 R	124 124 124	78 87.5 85	79 88 85	69 76 73
Cruise	0 30 L 30 R	146 146 146	74.5 83 82	71 80.5 81.5	66 74 71
Power approach (no flaps)	0 30 L 30 R	120 120 120	74 82 83	67.5 80 80	64.5 73 71
Power approach	0 30 L 30 R	97 97 97	58.5 65.5 66	56 60 60.5	52 60 57
Landing	0 30 L 30 R	99 99 99	72 75.5 77	73.5 79 80	66 72 72

[1] Trim controls set at zero for aileron, rudder and elevator.

Table 8. Stall Airspeeds of the RU-8D, S/N 57-6063, at a Gross Weight of 7120 Pounds.

Configuration	Roll Angle (deg)	Trim KCAS (kt)	Warning KCAS (kt)	Buffet KCAS (kt)	Stall KCAS (kt)
Takeoff	0	--	64	56.5	52.5
	30 L	Note[1]	67	66	64
	30 R	--	74	66	58
Cruise	0	90	84	82	72
	30 L	90	89	89	79
	30 R	90	89	86	79.5
Cruise	0	125	79	79	70
	30 L	125	88	88	81
	30 R	125	92.5	84	78
Cruise	0	146	80	74	68
	30 L	146	88	83	72
	30 R	146	87.5	85	73
Power approach	0	119	79	70	65
	30 L	119	85	81	78
	30 R	119	85	--	72
Landing	0	82	73	73	67
	30 L	82	76	79	68
	30 R	82	79	80	71

[1] Trim controls set at zero for aileron, rudder and elevator.

30. The approach to the stall was characterized by the following effects:

a. Activation of the stall warning horn.

b. Slight to moderate airframe buffet.

c. Increased longitudinal control forces and decreased aileron effectiveness as airspeed decreased.

d. A mild porpoising motion with 2 to 4 degrees of pitch amplitude between buffet airspeed and the stall.

31. For all configurations, the stall was characterized by a wing roll-off (normally to the left), a nose-down pitch of approximately 5 degrees and a lessening of the longitudinal control forces. Rudder effectiveness in the deep stall was good, but the aileron controls were minutely effective (HQRS 4). Attempts to fly the airplane in a deep stall usually required full aileron control to bring up the low wing and resulted in a wing roll-off in the opposite direction. Roll excursions were approximately 45 degrees to either side. One exception was noted: due to the lack of elevator control, the classic stall was not attainable in the landing configuration at a forward cg. With full UP elevator control, the airplane exhibited a moderate buffet, a porpoising motion with ±5 degrees in pitch amplitude and an increase in rate of descent from 500 to 600 fpm. The effectiveness of both aileron and rudder were good during this condition.

32. Stall recovery was initiated by relaxing longitudinal control force, leveling the wings and increasing power to cruise setting (if applicable). All controls were effective throughout the recovery (HQRS 3). Progressive stall tendencies were noted only when aft longitudinal control was applied too early during the recovery.

33. When attempting to attain a deep stall in the cruise configuration, the 30-degree right bank entry resulted in a left wing roll-off rate of approximately 90 degrees per second and a nose-down pitch attitude. Because the maneuvers closely resembled a spin entry, power was immediately reduced to effect recovery. The roll was arrested after 90 degrees with the airplane in an approximate 15-degree nose-down attitude and a 60-degree left bank angle. Further stall recovery was normal. This was repeatable and was the most adverse handling quality noted (HQRS 5) but occurred only when attempting a deep stall penetration. Furthermore, the airplane's attitude at entry was a 30-degree right bank angle and an approximate 20-degree nose-up pitch. Since this attitude would be highly unusual for the airplane's mission, this characteristic is acceptable. The stall characteristics noted during this evaluation are satisfactory for mission accomplishment and met the requirements of the specification.

AIRSPEED CALIBRATION

34. An airspeed calibration was performed using both pacer airplane and ground speed course methods to determine the position error of the test boom and standard airspeed systems. The results are presented in figures 35 and 36, appendix VI.

CONCLUSIONS

GENERAL

35. Within the scope of this test, the RU-8D's performance capabilities and handling qualities, in the Winebottle configuration, are satisfactory for its intended mission.

36. Adequate quantitative flight test data were obtained to permit accurate and safe mission planning.

SPECIFIC

37. Correction of the following shortcomings is desirable for improved mission effectiveness:

a. Poor sensitivity of the aileron trim (para 18).

b. Masking of the longitudinal control force gradient by the breakout forces (para 19).

SPECIFICATION COMPLIANCE

38. With the exception of paragraph 3.4.12, the RU-8D airplane met all of the requirements of MIL-F-8785, against which it was tested. This exception is the inability to maintain directional control at all speeds above $1.2V_{STO}$ during assymmetrically powered flight (para 26).

RECOMMENDATIONS

39. The shortcomings, correction of which is desirable, should be corrected at the earliest practical date.

40. The data in this report should be included in the RU-8D operator's manual.

APPENDIX I. REFERENCES

1. Letter Report, USAASTA, Project No. 68-51, *Airworthiness Qualification Test, RU-8D Airplane (Winebottle Configuration)*, July 1969.

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3. Military Specification, MIL-F-8785(ASG), *Flying Qualities of Piloted Airplanes*, 1 September 1954.

4. Technical Manual, TM 55-1510-201-10/4, *Operator's Manual, Army Models U-8D and U-8G Aircraft (Beech)*, February 1969.

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6. Technical Report, US Air Force, No. 6273, *Flight Test Engineering Handbook*, revised June 1964.

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8. Military Specification, MIL-C-5011A, *Charts, Standard Aircraft Characteristics and Performance, Piloted Aircraft*, 5 November 1951.

9. Test Plan, USAASTA, Project No. 68-51, *Engineering Flight Test of the RU-8D Airplane (Winebottle Configuration)*, February 1969.

10. Message, USAAVSCOM, AMSAV-R-F, 05-012, subject: Safety-of-Flight, RU-8D (Winebottle), 14 May 1969.

11. Letter, USAAVSCOM, AMSAV-R-F, subject: RU-8D (Winebottle), Project No. 68-51, 5 July 1969.

APPENDIX II. TEST CONDITIONS

PERFORMANCE

Performance test conditions with the data corrected to standard day conditions and the specified gross weight are as shown in table A.

Table A. Performance Test Conditions.

Test	Configuration	Gross Weight (lb)	Center of Gravity Fuselage Station (in.)	Altitude (ft)
Level flight power required	Cruise	6600	122.3	SL 5,000 10,000
Level flight power required	Cruise	7100	123.9	SL 5,000 10,000
Takeoff performance	Takeoff (flaps 10, 20, 30 degrees)	7350	124.1	SL 4,000
Landing performance	Power approach, landing	6600	122.5	SL 4,000

STABILITY AND CONTROL

Stability and control test conditions for all tests were conducted at a 5000-foot density altitude (±500 feet) with a cg location between 122.2 and 124.1 inches. These test conditions are as shown in table B.

Table B. Stability and Control Test Conditions.

Test	Configuration	Average Gross Weight (lb)	Trim KCAS (kt)
Static and dynamic longitudinal stability	Cruise	6620 7120	88, 107, 125 86 107, 124
Static and dynamic longitudinal stability	Power approach (flaps up)	6620 7120	125 120
Static and dynamic longitudinal stability	Power approach	6620 7120	97.5 97.5
Static and dynamic lateral-directional	Cruise	6620 7120	90, 110, 125 90, 110, 125
Static and dynamic lateral-directional	Power approach (flaps up)	6620 7120	120 120
Stalls	Power approach	6620	100
Stalls	Cruise	6620 7120	90, 124, 146 90, 124, 146
Stalls	Takeoff	6620 7120	Neutral trim Neutral trim
Stalls	Power approach (flaps up)	6620 7120	120 119
Stalls	Power approach	6620	97
Stalls	Landing	6620 7120	90 90

APPENDIX III. AIRPLANE CONFIGURATION DESCRIPTIONS

CRUISE

Power for level flight at trim airspeed, gear up, flaps up.

LANDING

Idle power, gear down, flaps at 30 degrees.

POWER APPROACH

Gear down, power for level flight at trim airspeed, flaps at 30 degrees or zero degrees, as stated.

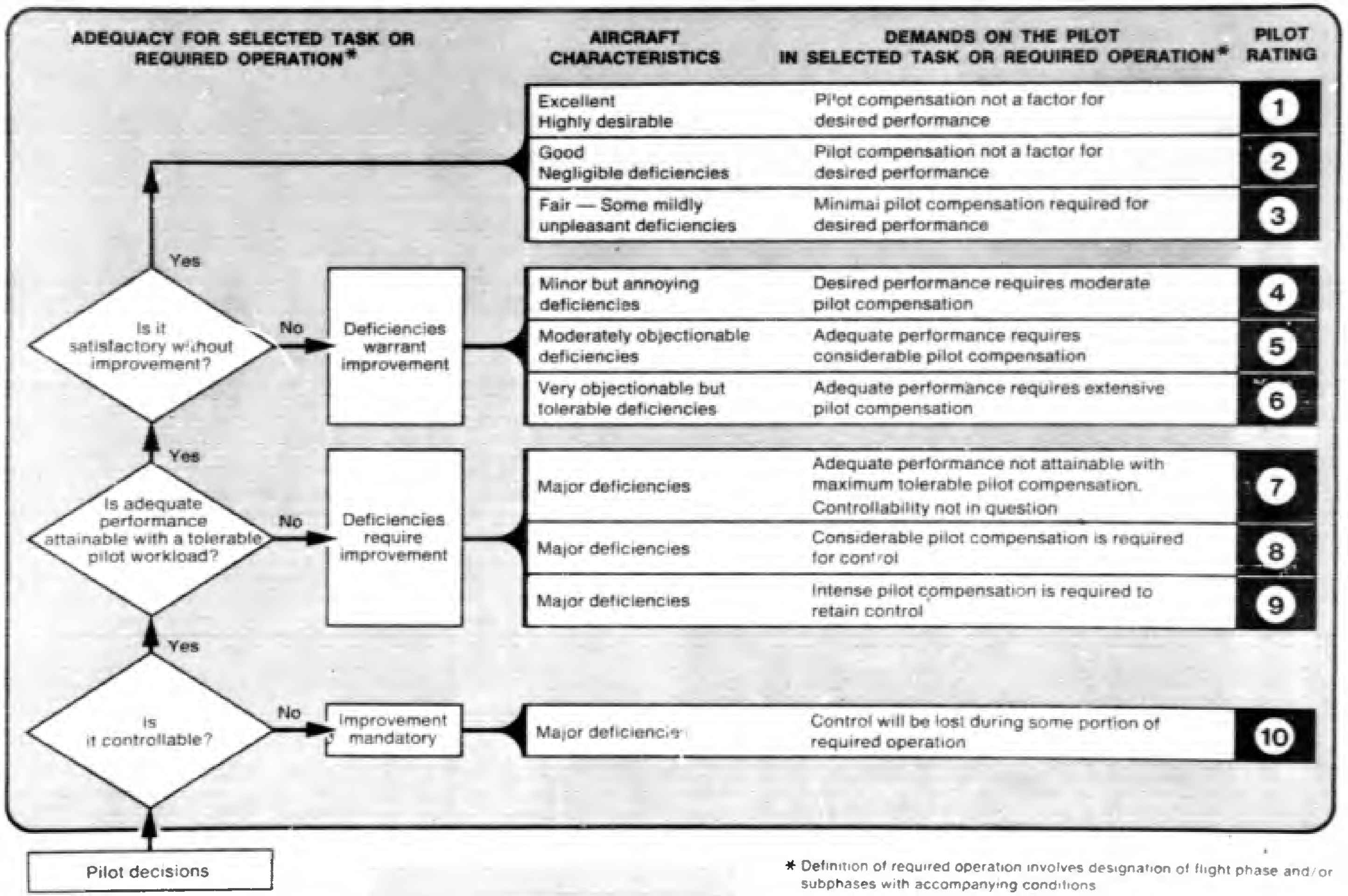
TAKEOFF

Gear down, takeoff power, flaps at 20 degrees unless otherwise stated.

APPENDIX IV. TEST INSTRUMENTATION

Description	Photopanel	Oscillograph	Cockpit
Engine manifold pressure (2)	X		X
Carburetor inlet temperature (2)	X		
Carburetor inlet pressure (2)	X		
Engine rpm (2)	X		X
Fuel counter (2)	X		X
Indicated airspeed (airplane)	X		
Indicated airspeed (boom)	X		X
Altitude (airplane)	X		
Altitude (boom)	X		X
Time (clock)	X		X
Hayden timer	X		
Camera frame number	X	X	X
Oscillograph burst number	X	X	X
Camera ON light			X
Oscillograph ON light			X
Outside air temperature (boom)	X		X
Rudder position		X	X
Aileron position		X	X
Elevator position		X	X
Yaw rate gyro		X	
Roll rate gyro		X	
Pitch rate gyro		X	
Yaw attitude		X	
Pitch attitude		X	
Roll attitude		X	
Longitudinal control force		X	
Lateral control force		X	
Rudder pedal force		X	
CG vertical acceleration		X	
Event marker	X	X	
Angle of attack		X	X
Sideslip angle		X	X

APPENDIX V. HANDLING QUALITIES RATING SCALE



AIRCRAFT CHARACTERISTICS	DEMANDS ON THE PILOT IN SELECTED TASK OR REQUIRED OPERATION*	PILOT RATING
Excellent Highly desirable	Pilot compensation not a factor for desired performance	1
Good Negligible deficiencies	Pilot compensation not a factor for desired performance	2
Fair — Some mildly unpleasant deficiencies	Minimal pilot compensation required for desired performance	3
Minor but annoying deficiencies	Desired performance requires moderate pilot compensation	4
Moderately objectionable deficiencies	Adequate performance requires considerable pilot compensation	5
Very objectionable but tolerable deficiencies	Adequate performance requires extensive pilot compensation	6
Major deficiencies	Adequate performance not attainable with maximum tolerable pilot compensation. Controllability not in question	7
Major deficiencies	Considerable pilot compensation is required for control	8
Major deficiencies	Intense pilot compensation is required to retain control	9
Major deficiencies	Control will be lost during some portion of required operation	10

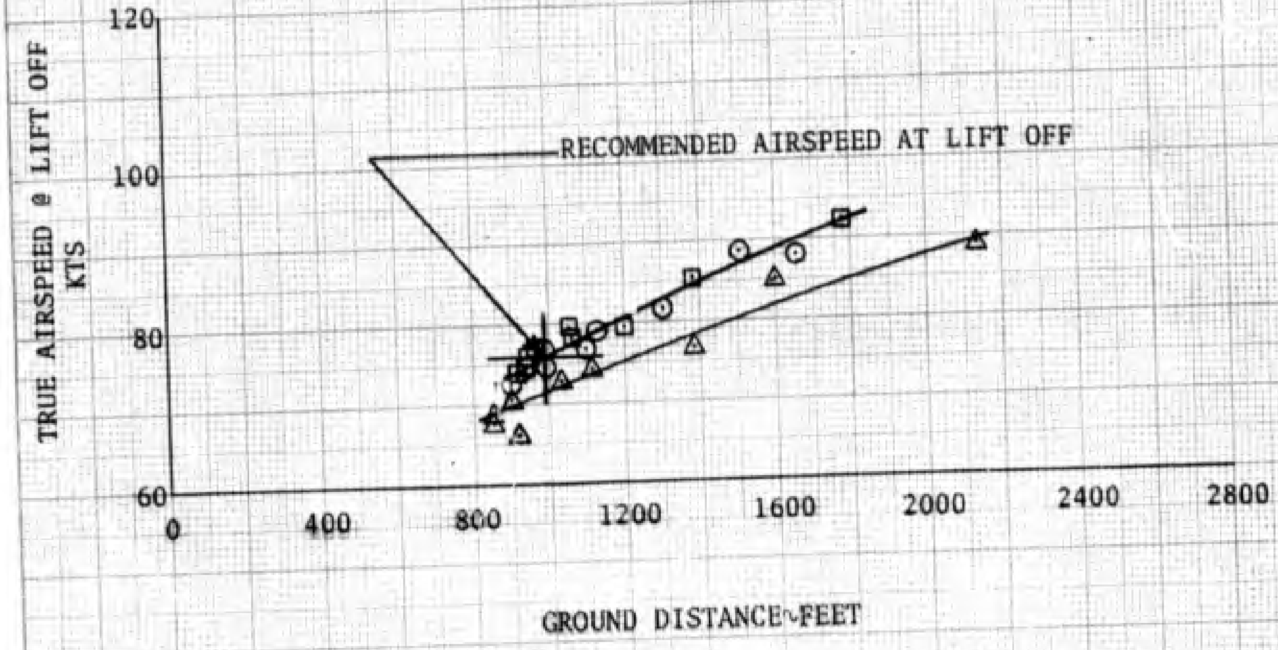
* Definition of required operation involves designation of flight phase and/or subphases with accompanying conditions

APPENDIX VI. TEST DATA

Figure No. 1
TAKEOFF PERFORMANCE
RU-8D S/N 57-6063

SYM	GROSS WEIGHT ~ LB	CG STATION ~ INCH	DENSITY ALT ~ FT	CONFIGURATION
○	7350	124.1	SL	10° FLAPS
□	7350	124.1	SL	20° FLAPS
△	7350	124.1	SL	30° FLAPS

NOTE: DATA CORRECTED TO
1. ZERO WIND
2. US STANDARD DAY



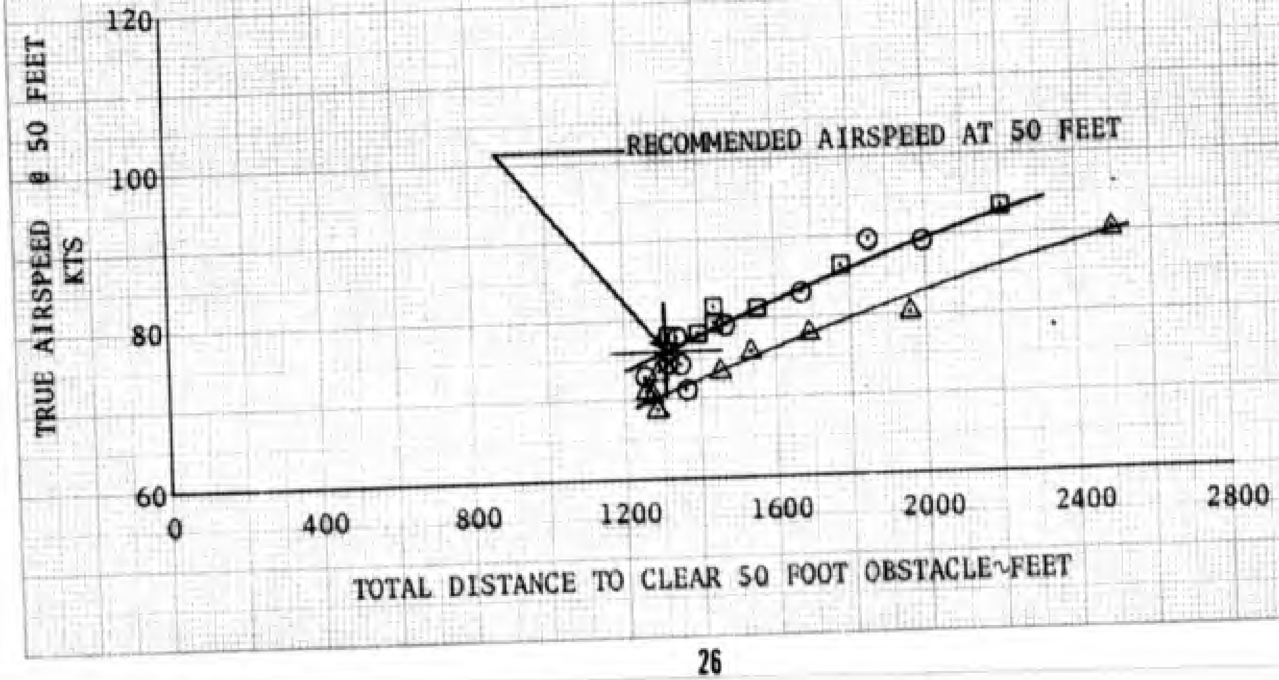


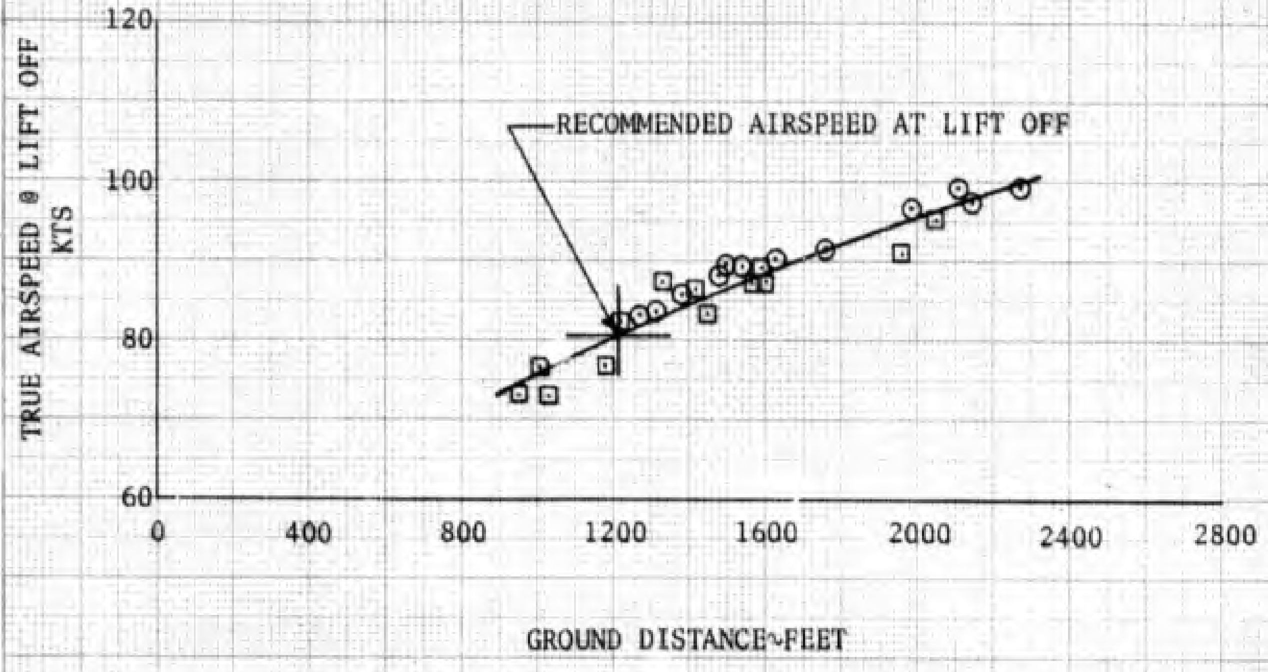
Figure No. 2

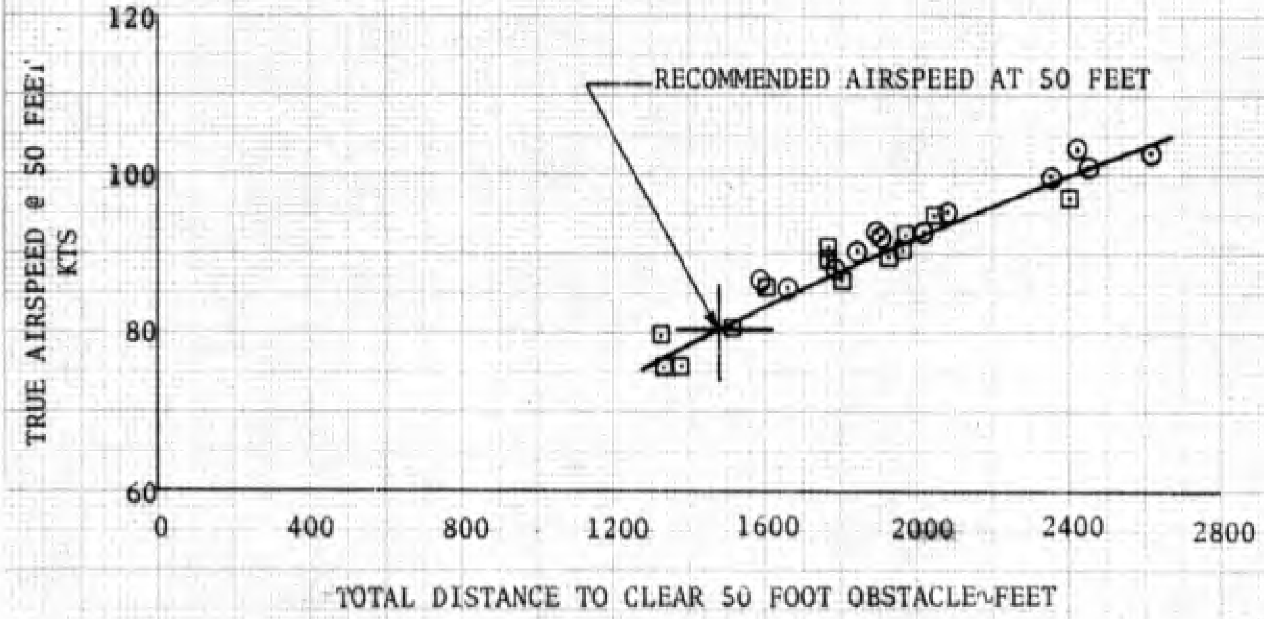
TAKEOFF PERFORMANCE

RU-8D S/N 57-6063

SYM	GROSS WEIGHT ~ LB	CG STATION ~ INCH	DENSITY ALT ~ FT	CONFIGURATION
○	7350	124.1	4000	10° FLAPS
□	7350	124.1	4000	20° FLAPS

NOTE: DATA CORRECTED TO
1. ZERO WIND
2. US STANDARD DAY





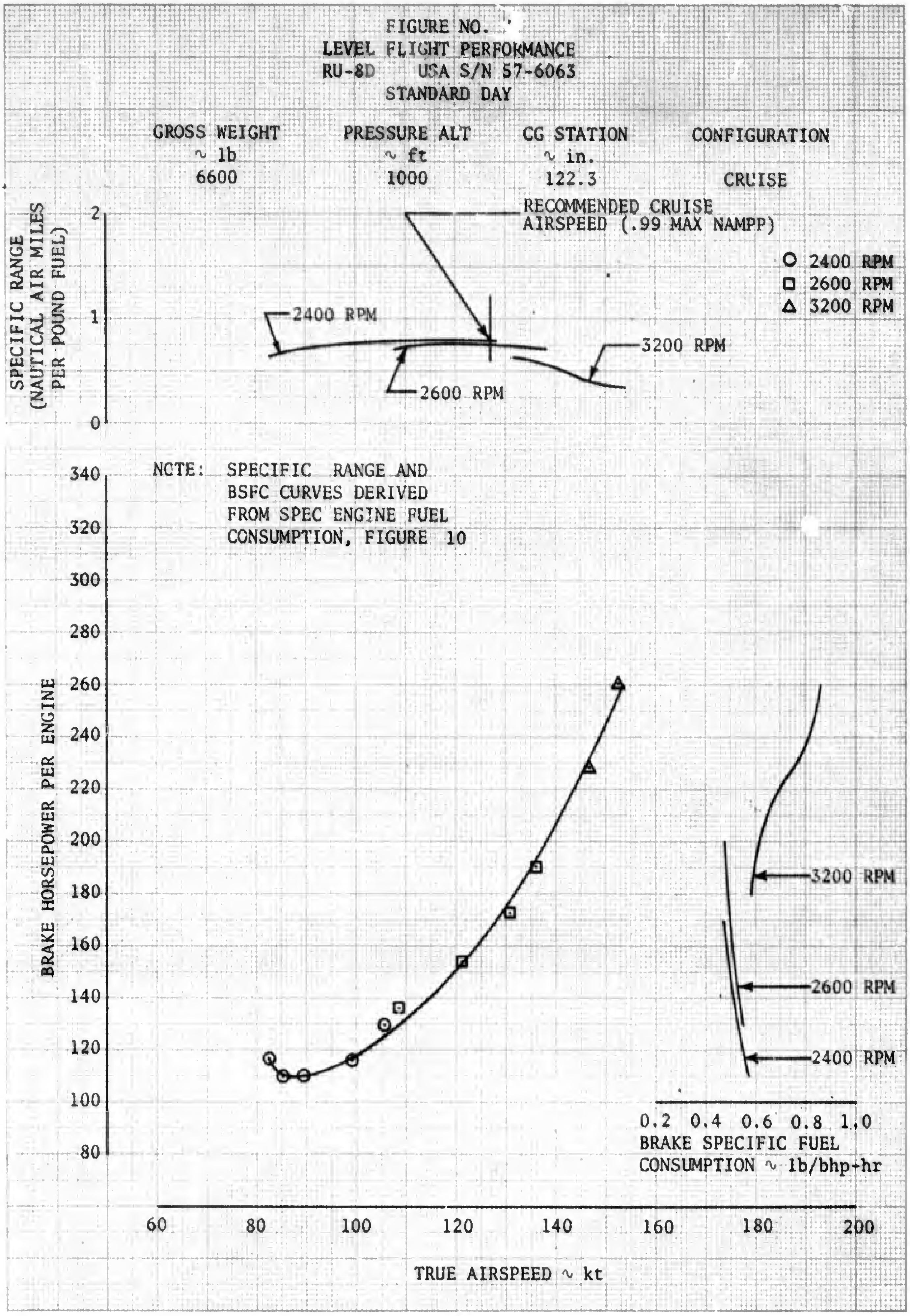


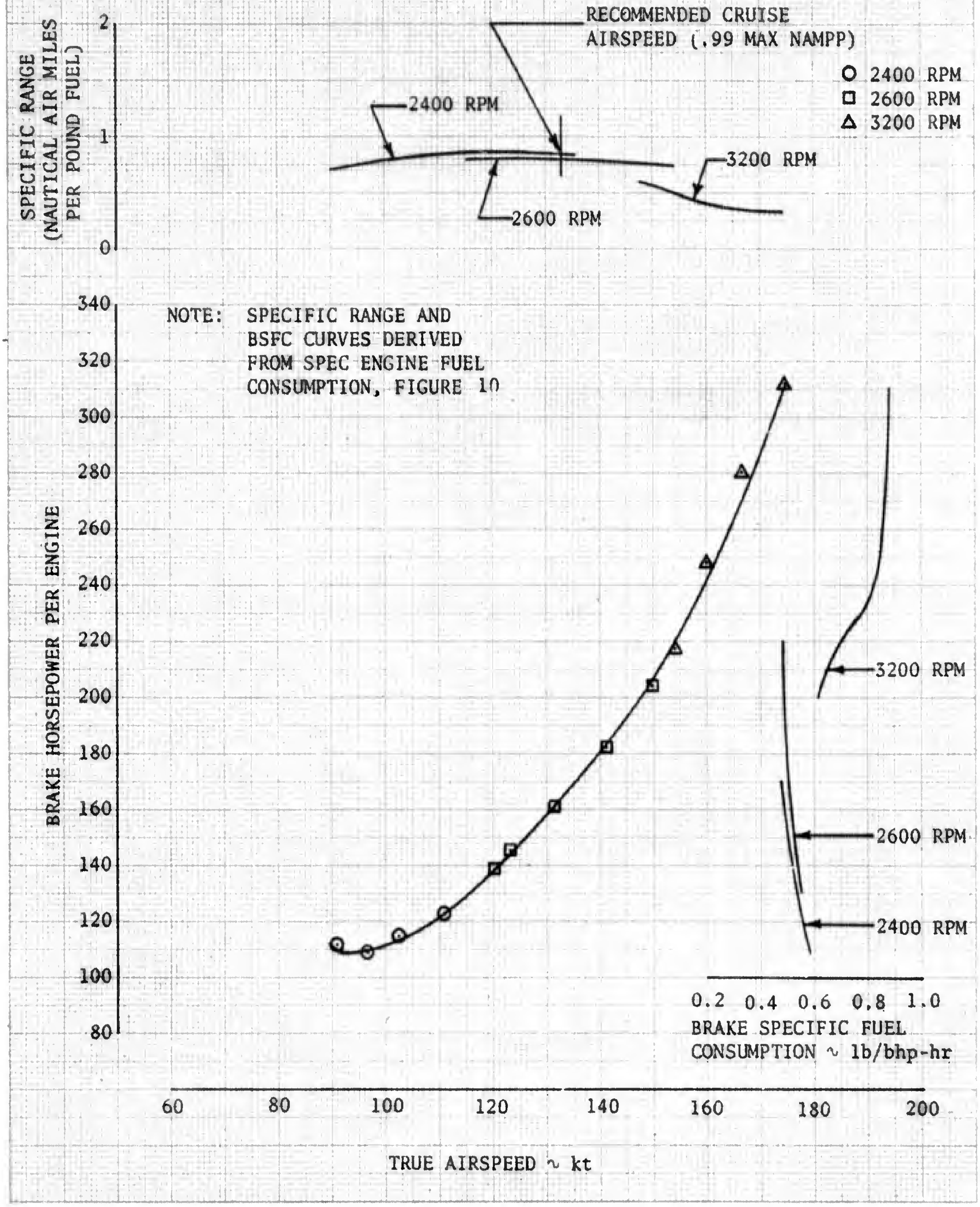
FIGURE NO.
LEVEL FLIGHT PERFORMANCE
RU-8D USA S/N 57-6063
STANDARD DAY

GROSS WEIGHT ~ lb	PRESSURE ALT ~ ft	CG STATION ~ in.	CONFIGURATION
6600	1000	122.3	CRUISE

NOTE: SPECIFIC RANGE AND BSFC CURVES DERIVED FROM SPEC ENGINE FUEL CONSUMPTION, FIGURE 10

FIGURE NO. 4
LEVEL FLIGHT PERFORMANCE
RU-8D USA S/N 57-6063
STANDARD DAY

GROSS WEIGHT ~ lb	PRESSURE ALT ~ ft	CG STATION ~ in.	CONFIGURATION
6600	5000	122.3	CRUISE



NOTE: SPECIFIC RANGE AND BSFC CURVES DERIVED FROM SPEC ENGINE FUEL CONSUMPTION, FIGURE 10

FIGURE NO. 5
LEVEL FLIGHT PERFORMANCE
RU-8D USA S/N 57-6063
STANDARD DAY

GROSS WEIGHT ~ lb	PRESSURE ALT ~ ft	CG STATION ~ in.	CONFIGURATION
6600	10,000	122.3	CRUISE

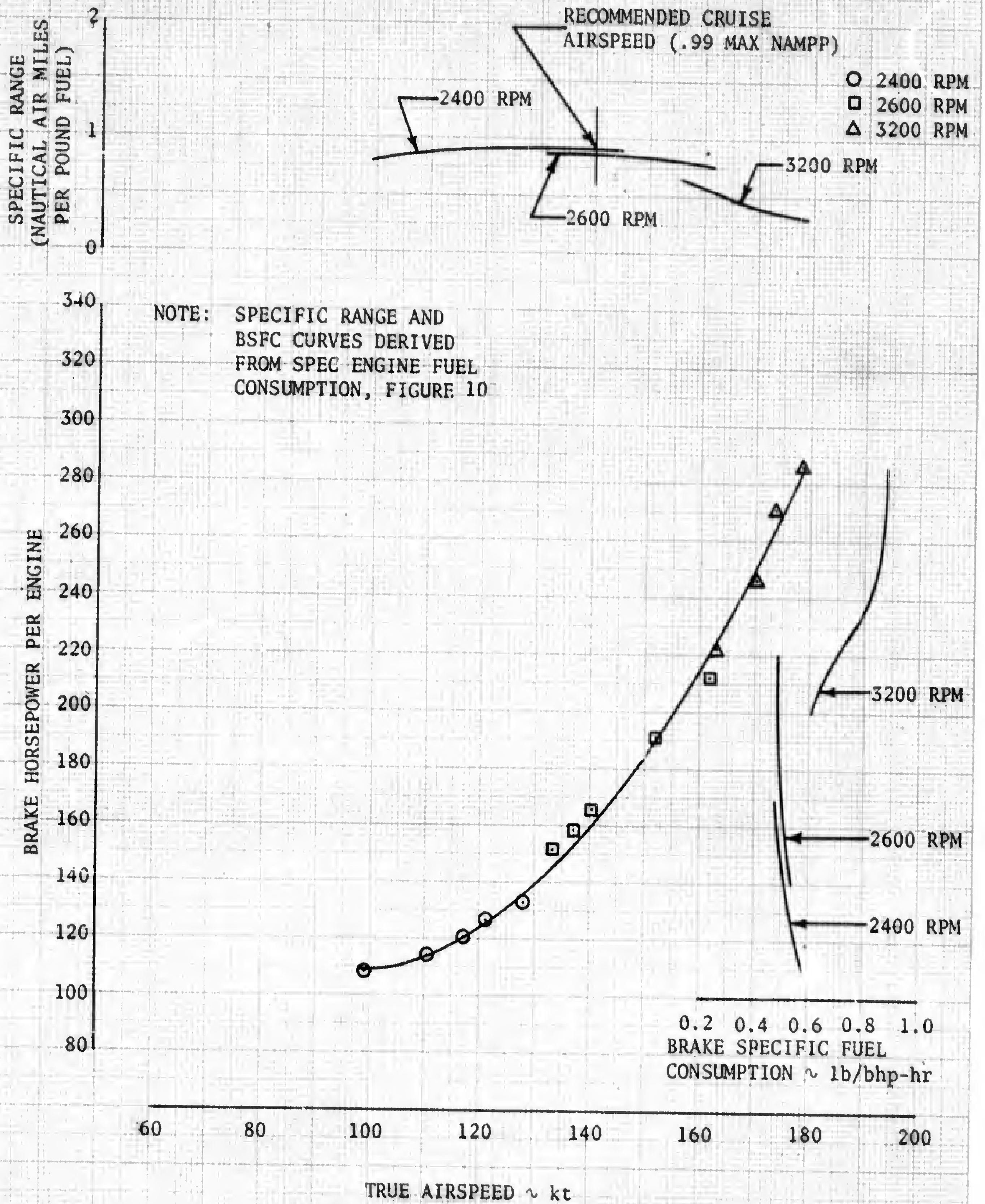
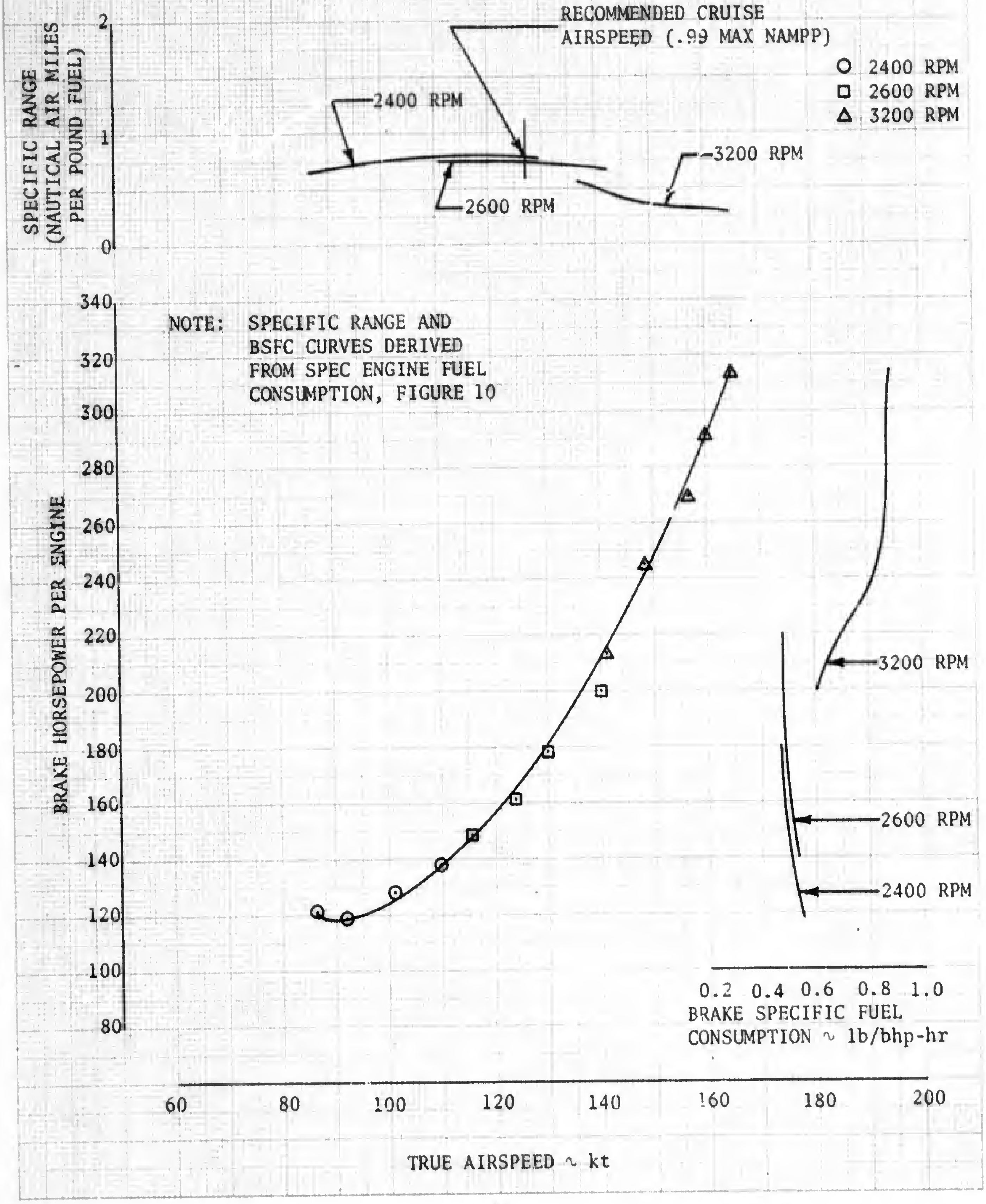


FIGURE NO. 6
LEVEL FLIGHT PERFORMANCE
RU-8D USA S/N 57-6063
STANDARD DAY

GROSS WEIGHT ~ lb	PRESSURE ALT ~ ft	CG STATION ~ in.	CONFIGURATION
7100	1000	123.9	CRUISE



NOTE: SPECIFIC RANGE AND BSFC CURVES DERIVED FROM SPEC ENGINE FUEL CONSUMPTION, FIGURE 10

FIGURE NO. 7
LEVEL FLIGHT PERFORMANCE
RU-8D USA S/N 57-6063
STANDARD DAY

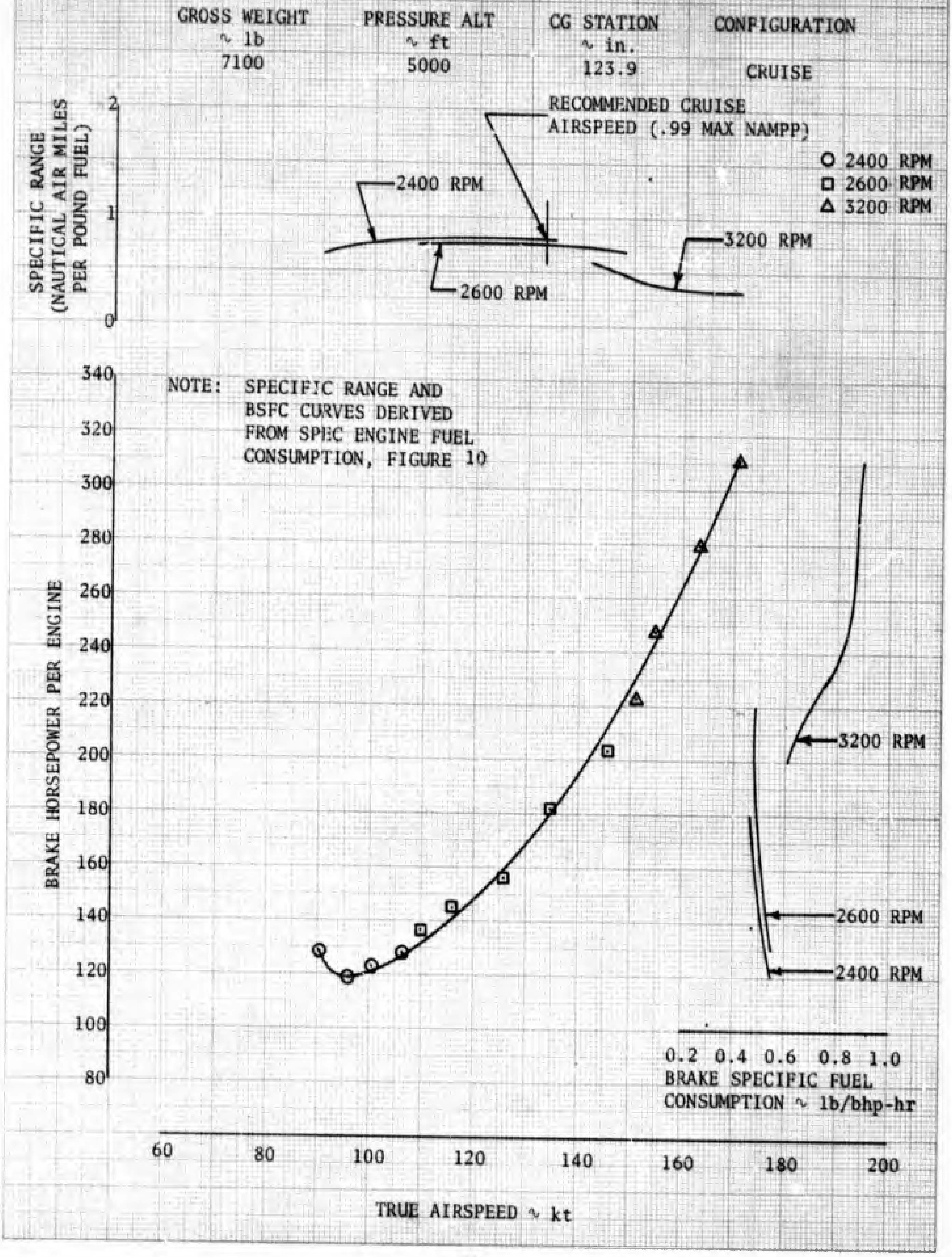
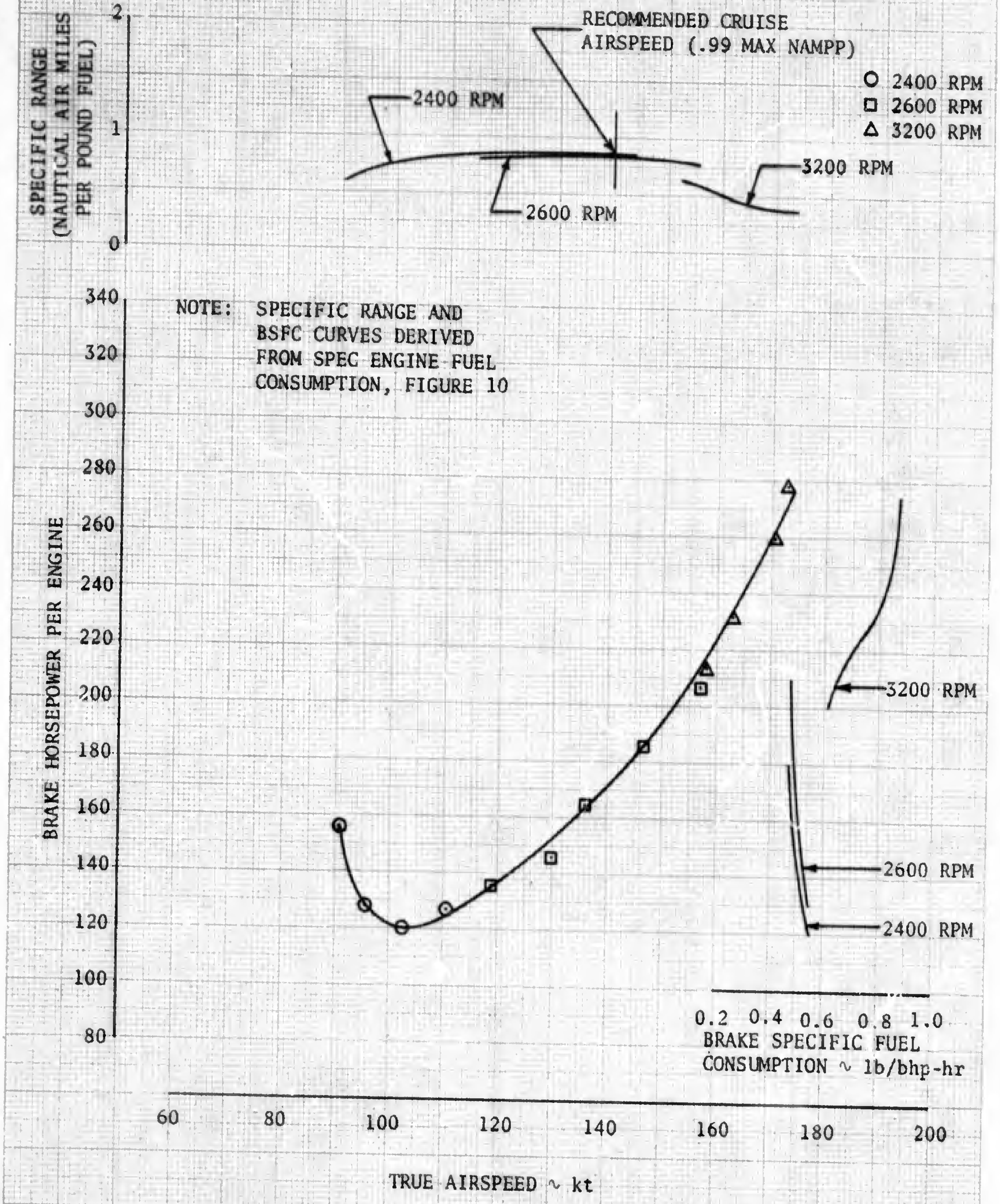


FIGURE NO. 8
LEVEL FLIGHT PERFORMANCE
RU-8D USA S/N 57-6063
STANDARD DAY

GROSS WEIGHT ~ lb	PRESSURE ALT ~ ft	CG STATION ~ in.	CONFIGURATION
7100	10,000	123.9	CRUISE



NOTE: SPECIFIC RANGE AND BSFC CURVES DERIVED FROM SPEC ENGINE FUEL CONSUMPTION, FIGURE 10

FIGURE NO. 9
GENERALIZED LEVEL FLIGHT PERFORMANCE
RU-8D S/N 57-6063

CRUISE CONFIGURATION
STANDARD WEIGHT 7350 LB

SYMBOL	ALTITUDE ~ FT	GROSS WEIGHT ~ LB
☆	1000	7090
○	1000	6650
⬡	5000	7120
□	5000	6580
◇	10,000	7080
△	10,000	6500

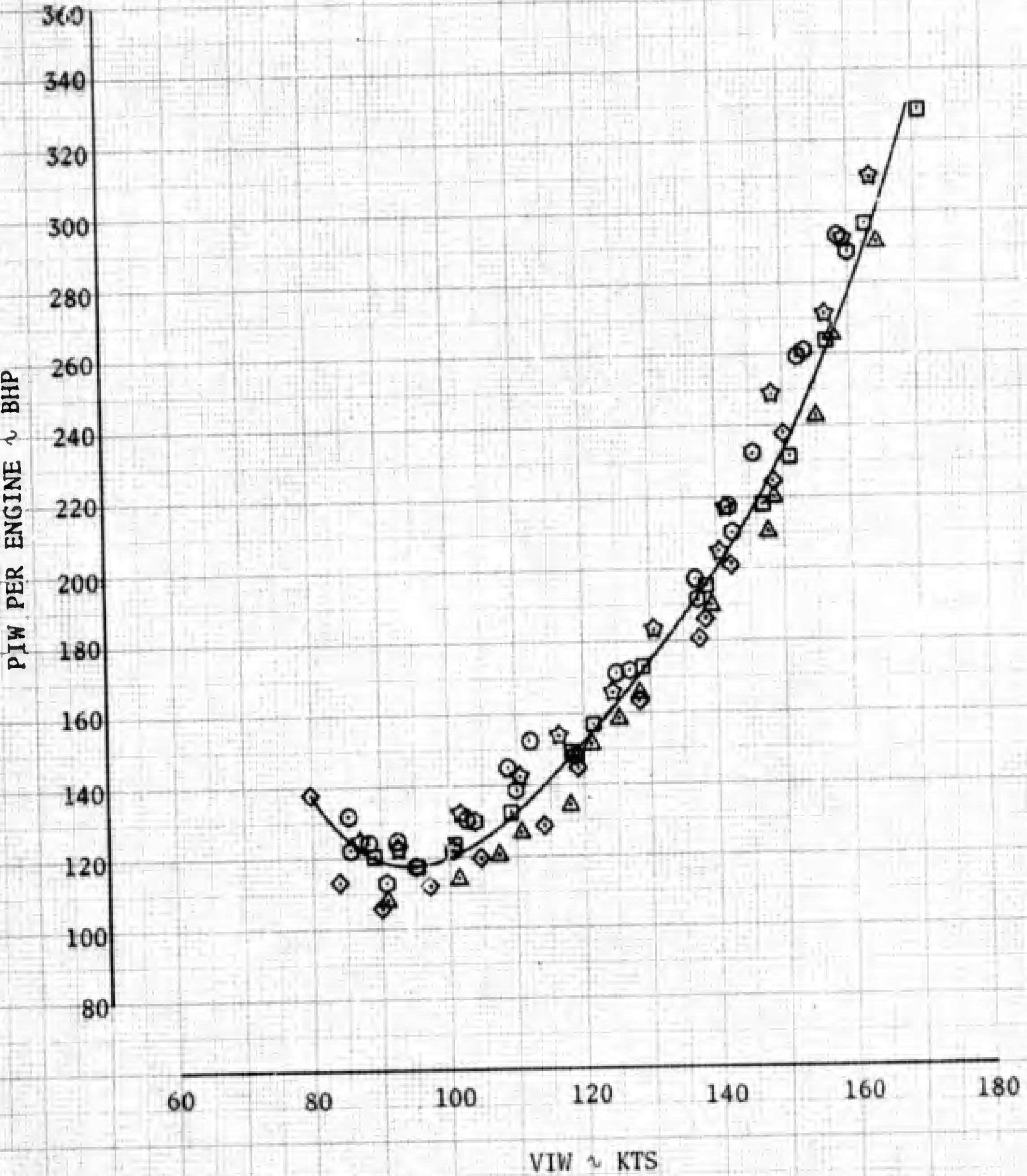


FIGURE NO 10
SPEC ENGINE FUEL CONSUMPTION
RU-8D S/N 57-6063

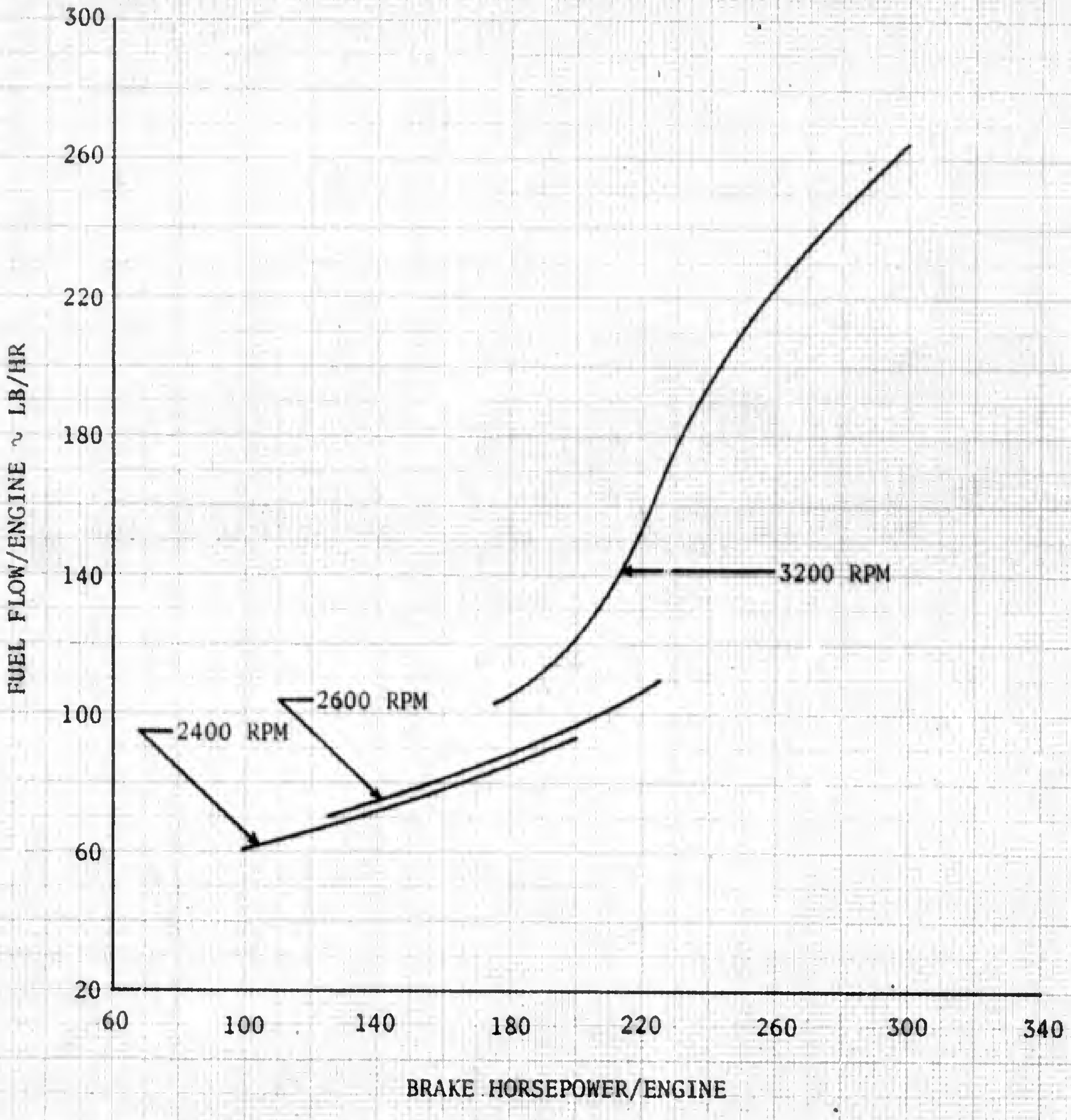


Figure No. 11
LANDING PERFORMANCE
RU-8D S/N 57-6063

SYM	GROSS WEIGHT ~ LB	CG STATION ~ INCH	DENSITY ALT ~ FT	CONFIGURATION POWER APPROACH
○	6600	122.5	S.L.	GEAR DOWN 30° FLAPS

NOTE: DATA CORRECTED TO ZERO WIND

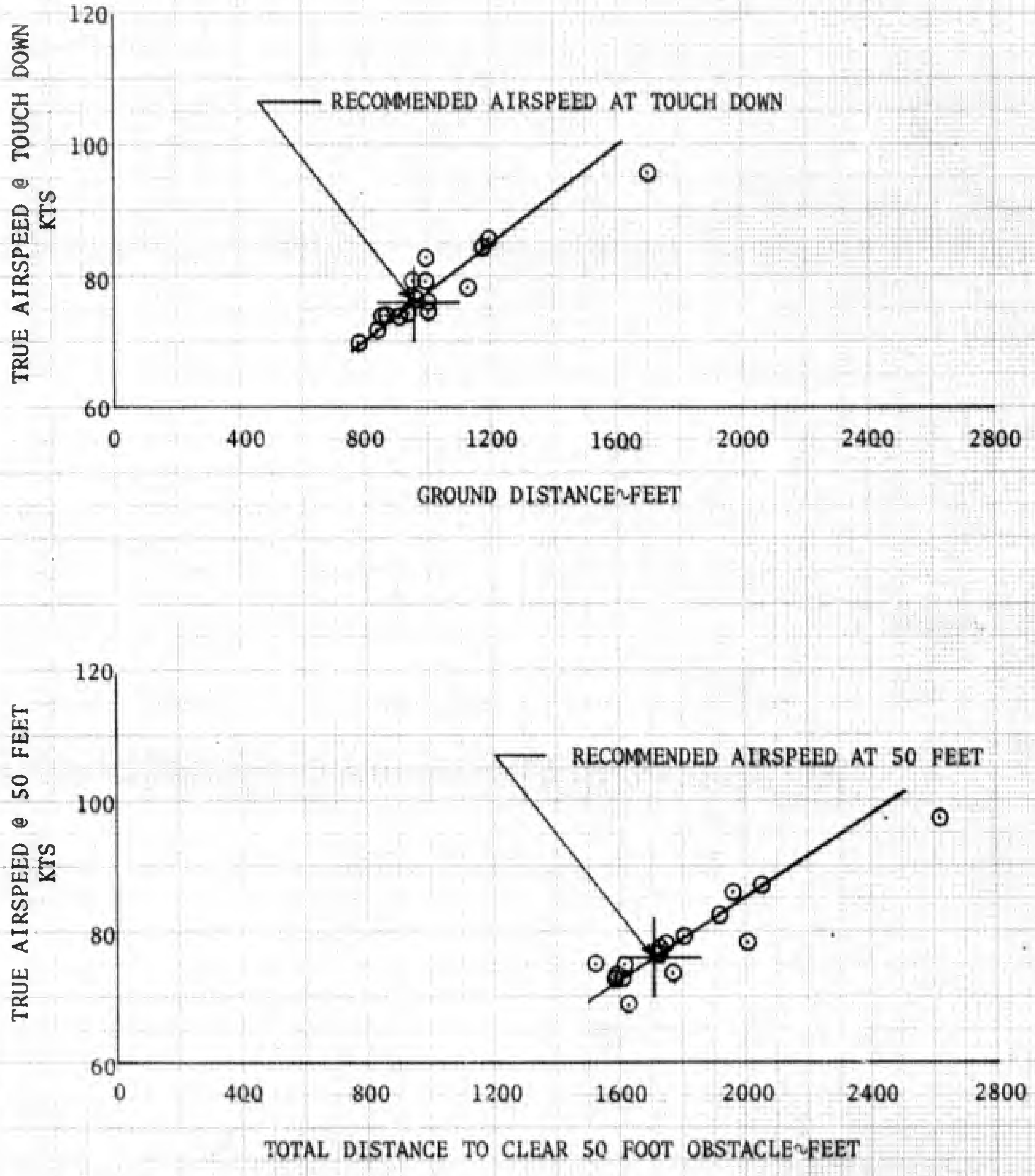
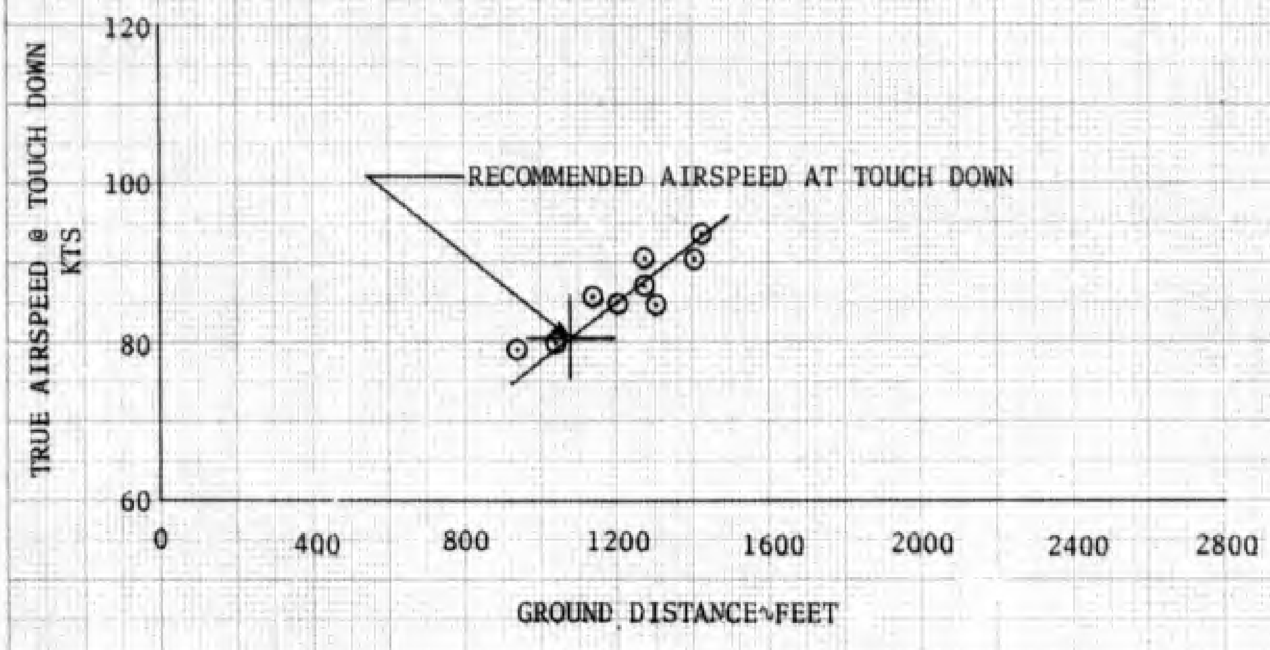


Figure No. 12
LANDING PERFORMANCE
RU-8D S/N 57-6063

SYM	GROSS WEIGHT ~ LB	CG STATION ~ INCH	DENSITY ALT ~ FT	CONFIGURATION
○	6600	122.5	4000	POWER APPROACH GEAR DOWN 30° FLAPS

NOTE: DATA CORRECTED TO ZERO WIND.



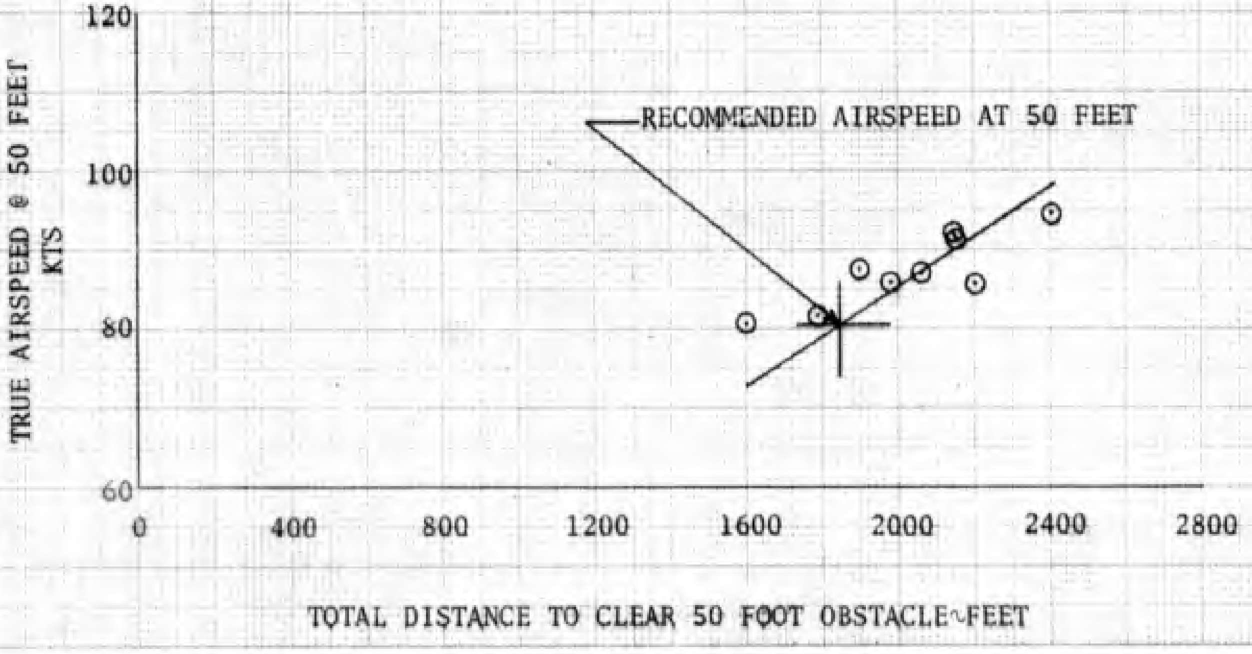


Figure No. 13
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	6600 LB	122.3 IN.	4830 FT	CRUISE

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE
TRIM POINTS

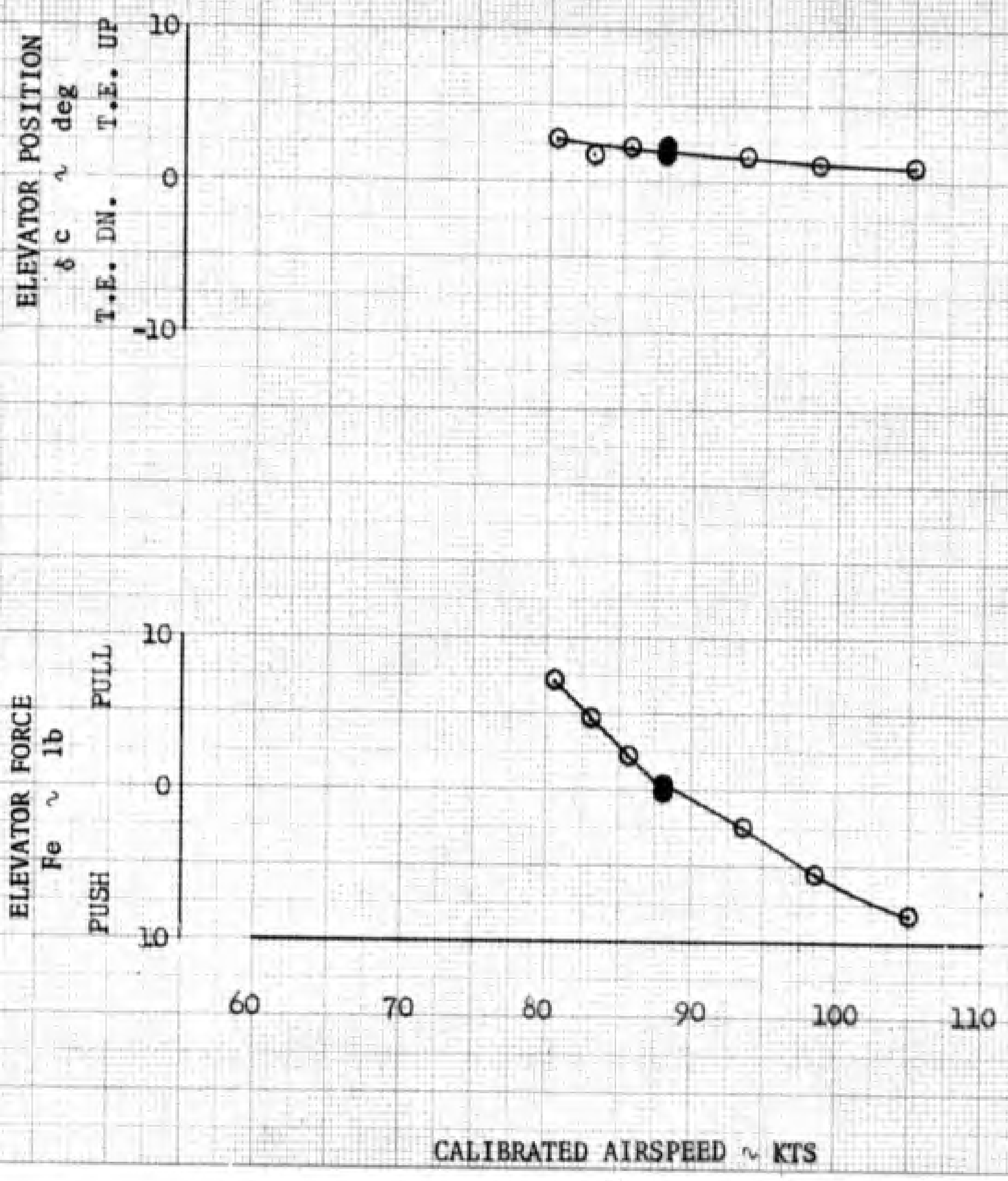


Figure No. 14
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	6640 LB	122.3 IN.	5070 FT	CRUISE

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE TRIM POINTS

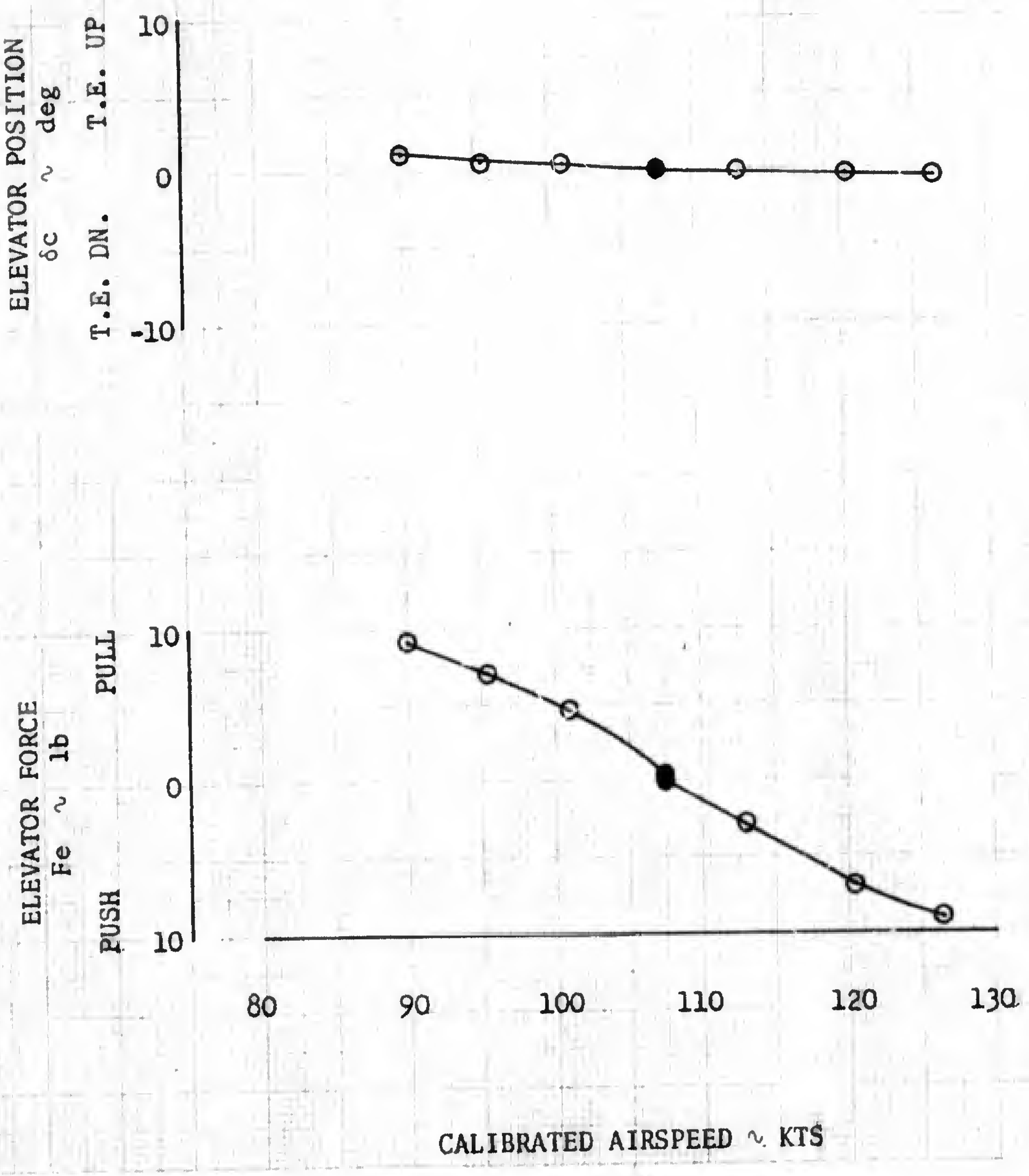


Figure No. 15
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	6740 LB	122.3 IN.	5330 FT	CRUISE

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE
TRIM POINTS

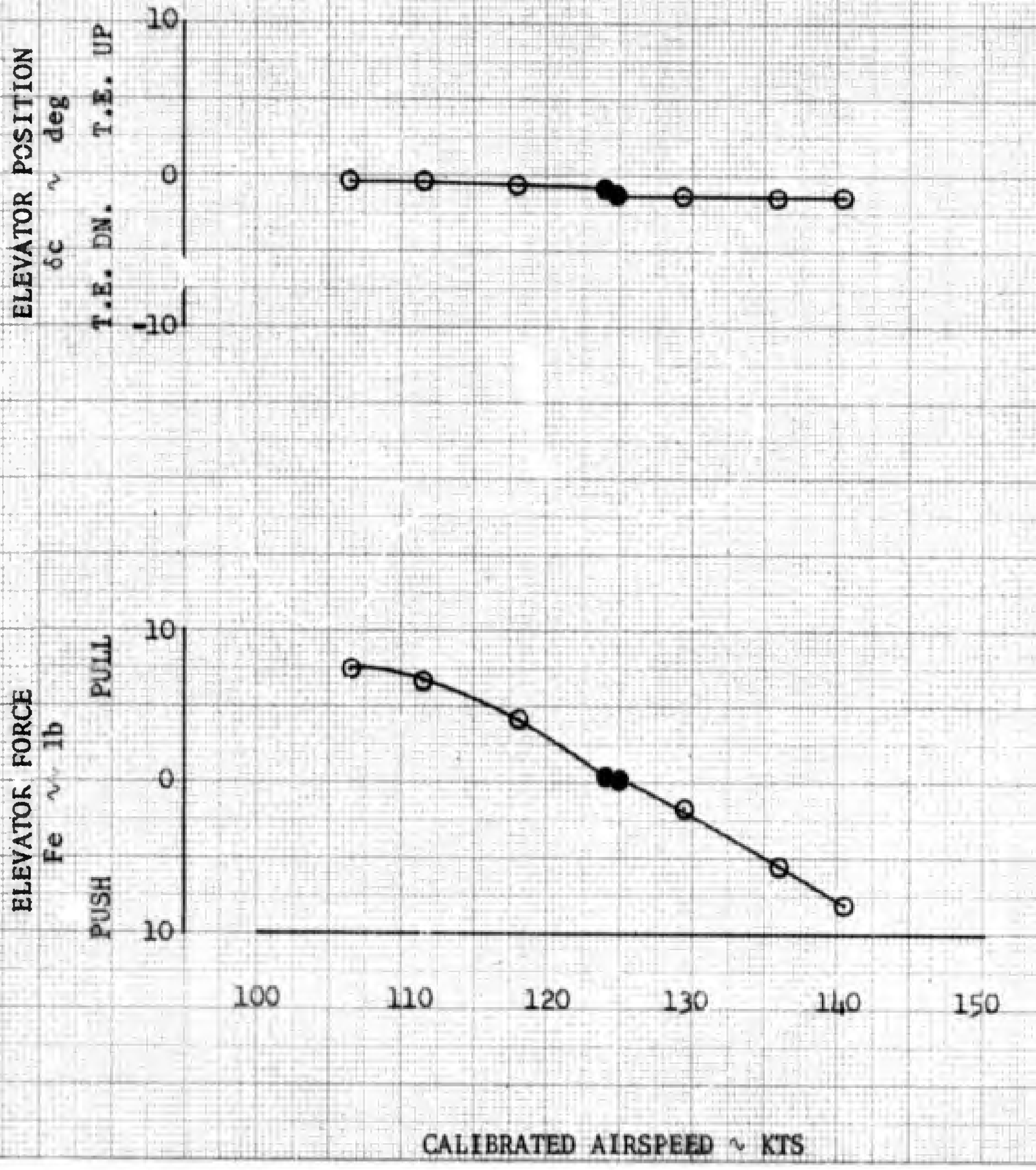


Figure No. 16

STATIC LONGITUDINAL STABILITY

RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	6570 LB	122.5 IN.	4630 FT	POWER APPROACH GEAR DOWN

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE TRIM POINTS

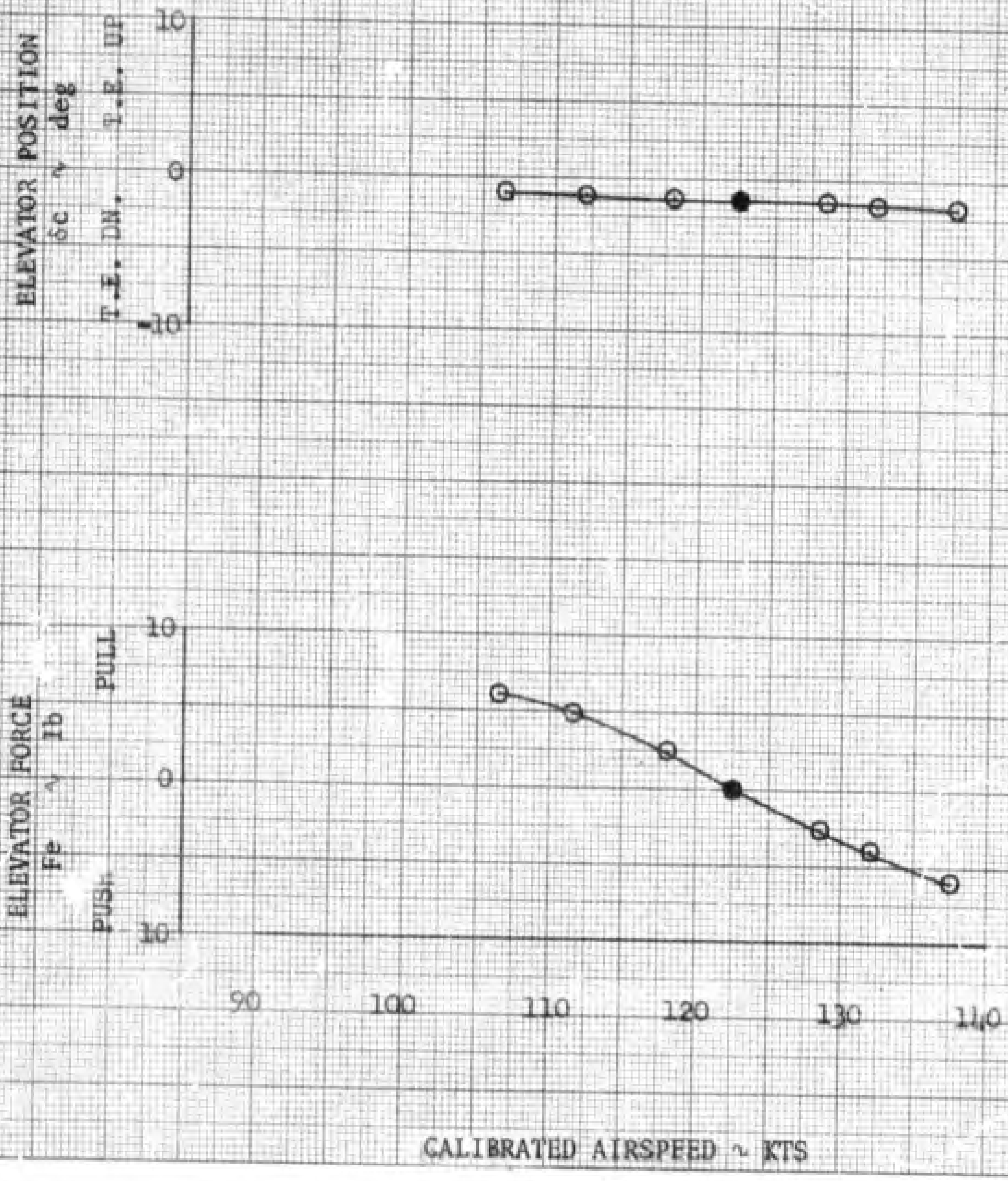


Figure No. 17
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	6540 LB	122.5 IN.	5480 FT	POWER APPROACH GEAR DOWN 30° FLAPS

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG.
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE TRIM POINTS

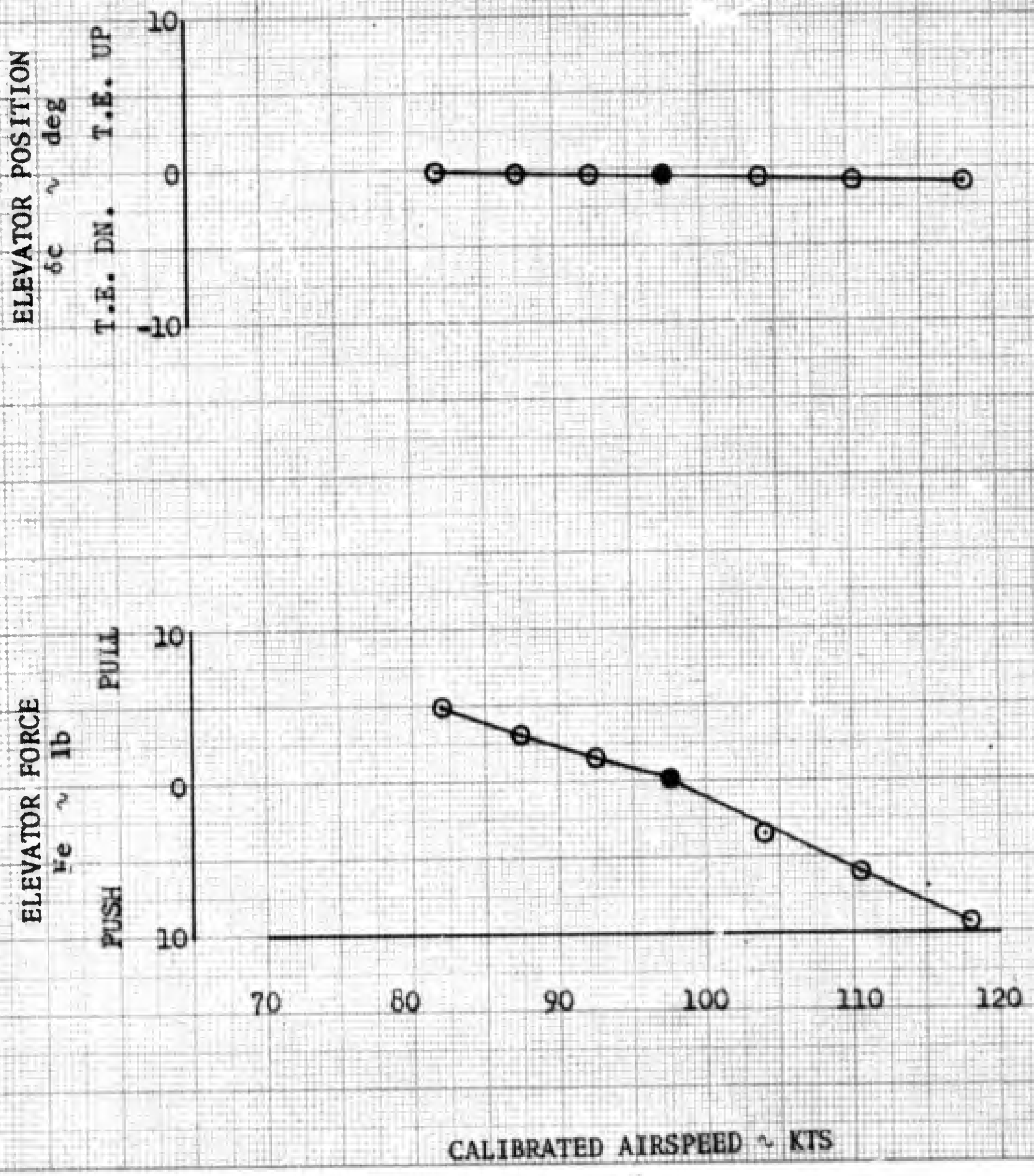


Figure No. 18
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	7160 LB	123.9 IN.	4920 FT	CRUISE

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE
TRIM POINTS

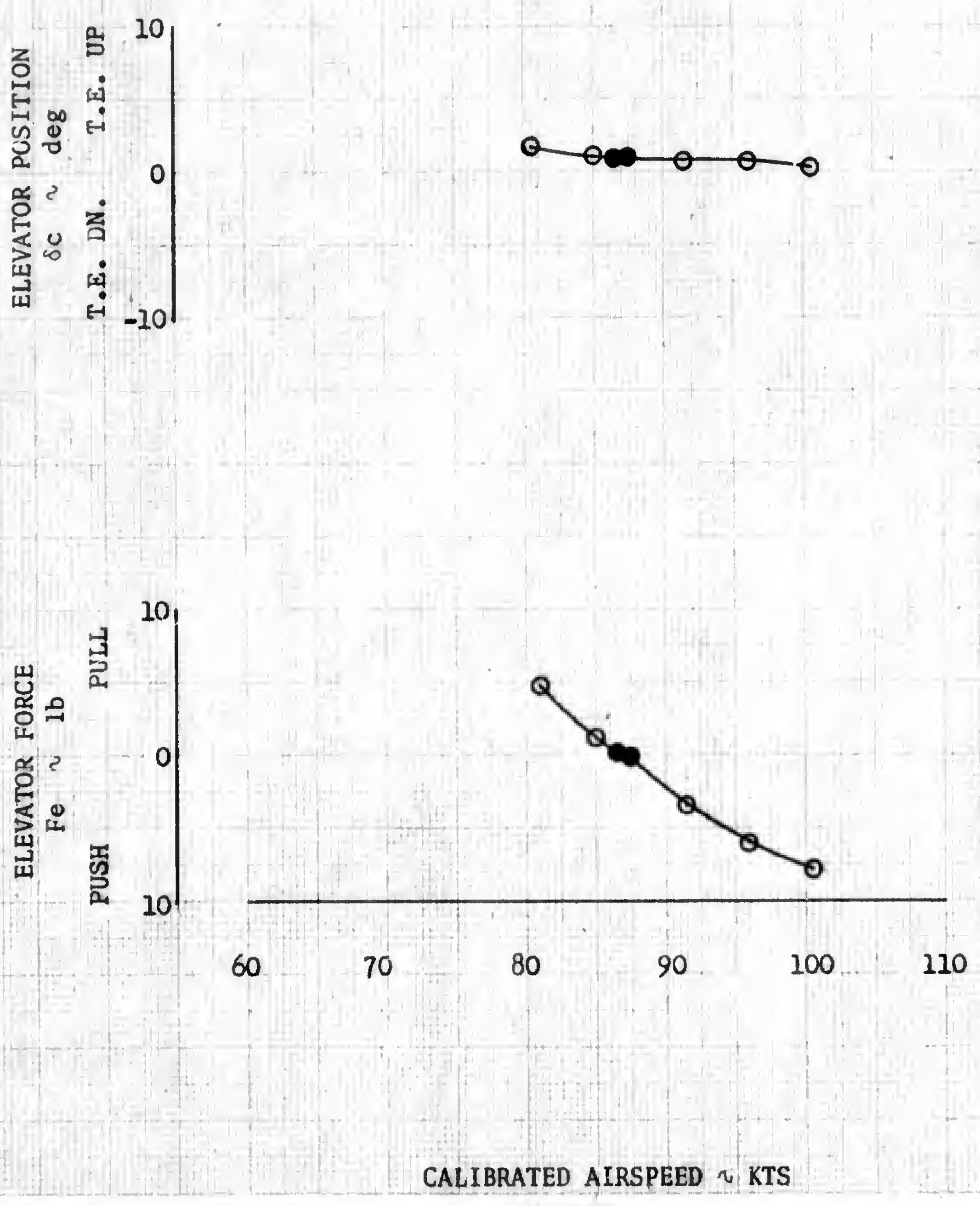


Figure No. 19
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	7190 LB	123.9 IN.	4880 FT	CRUISE

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE
TRIM POINTS

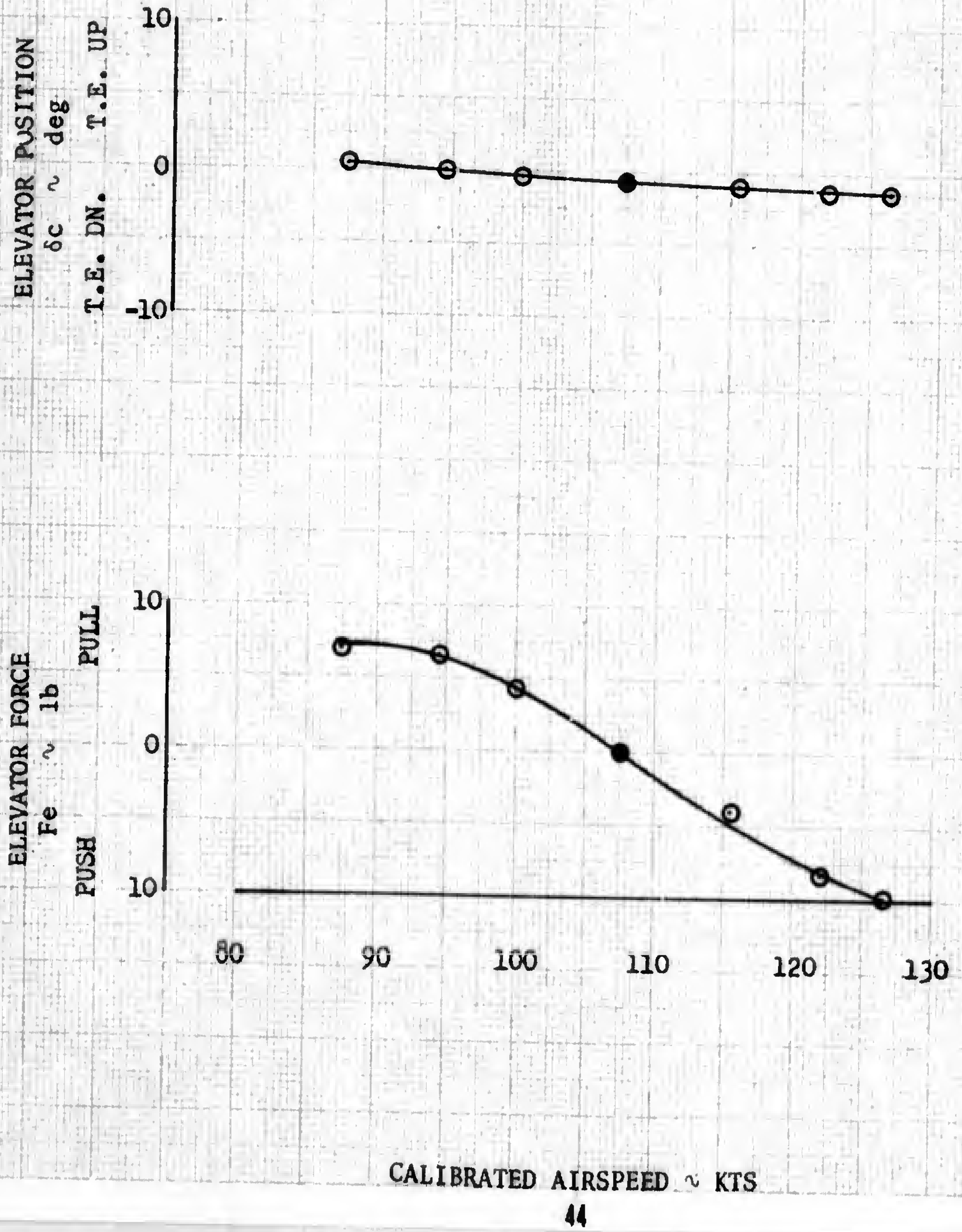


Figure No. 20
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	7260 LB	123.9 IN.	4870 FT	CRUISE

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE
TRIM POINTS

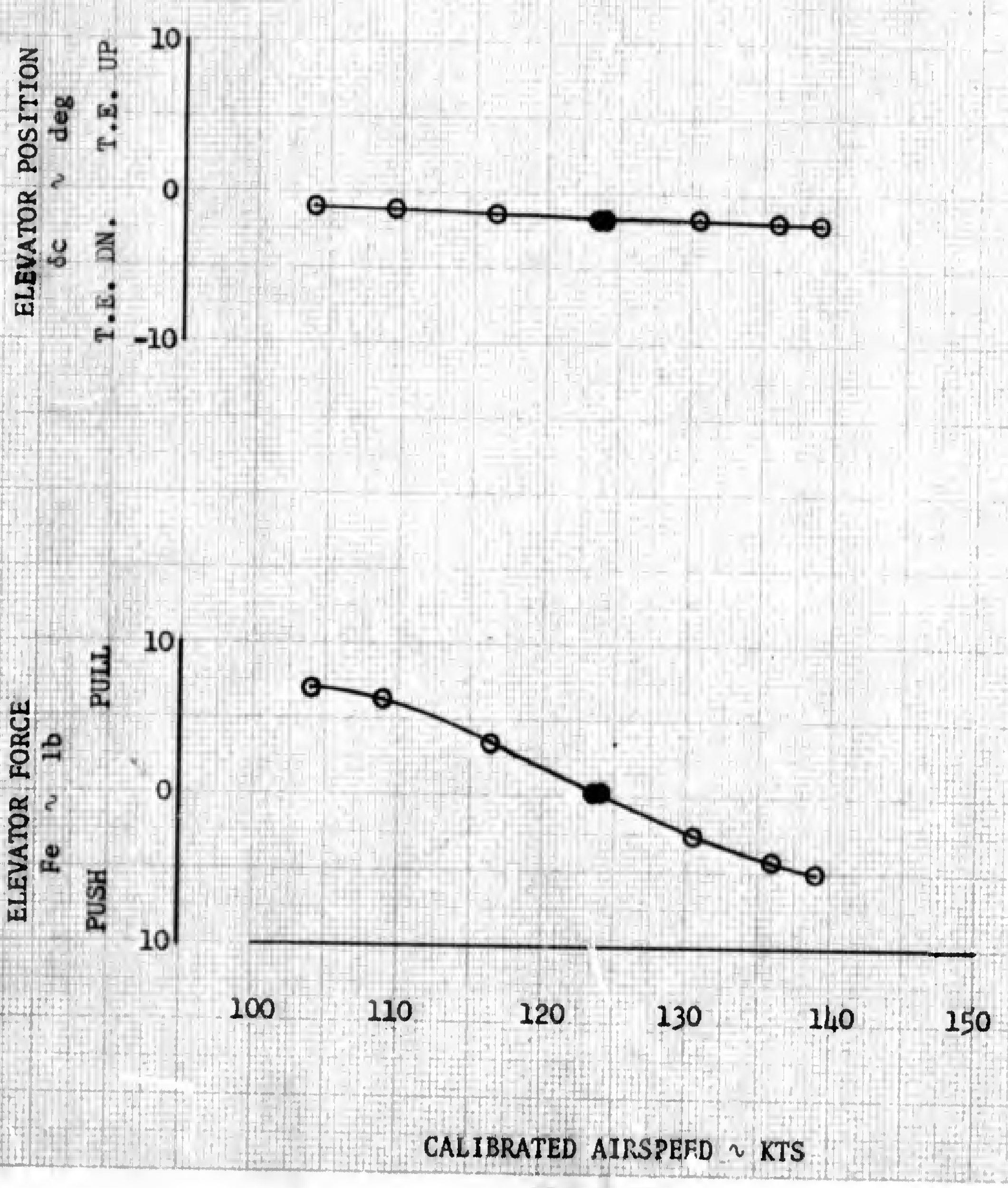


Figure No. 21
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
○	7110 LB	124.1 IN.	4960 FT	POWER APPROACH GEAR DOWN

NOTE: 1. FULL ELEVATOR TRAVEL
UP = 25.17 DEG
DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE
TRIM POINTS

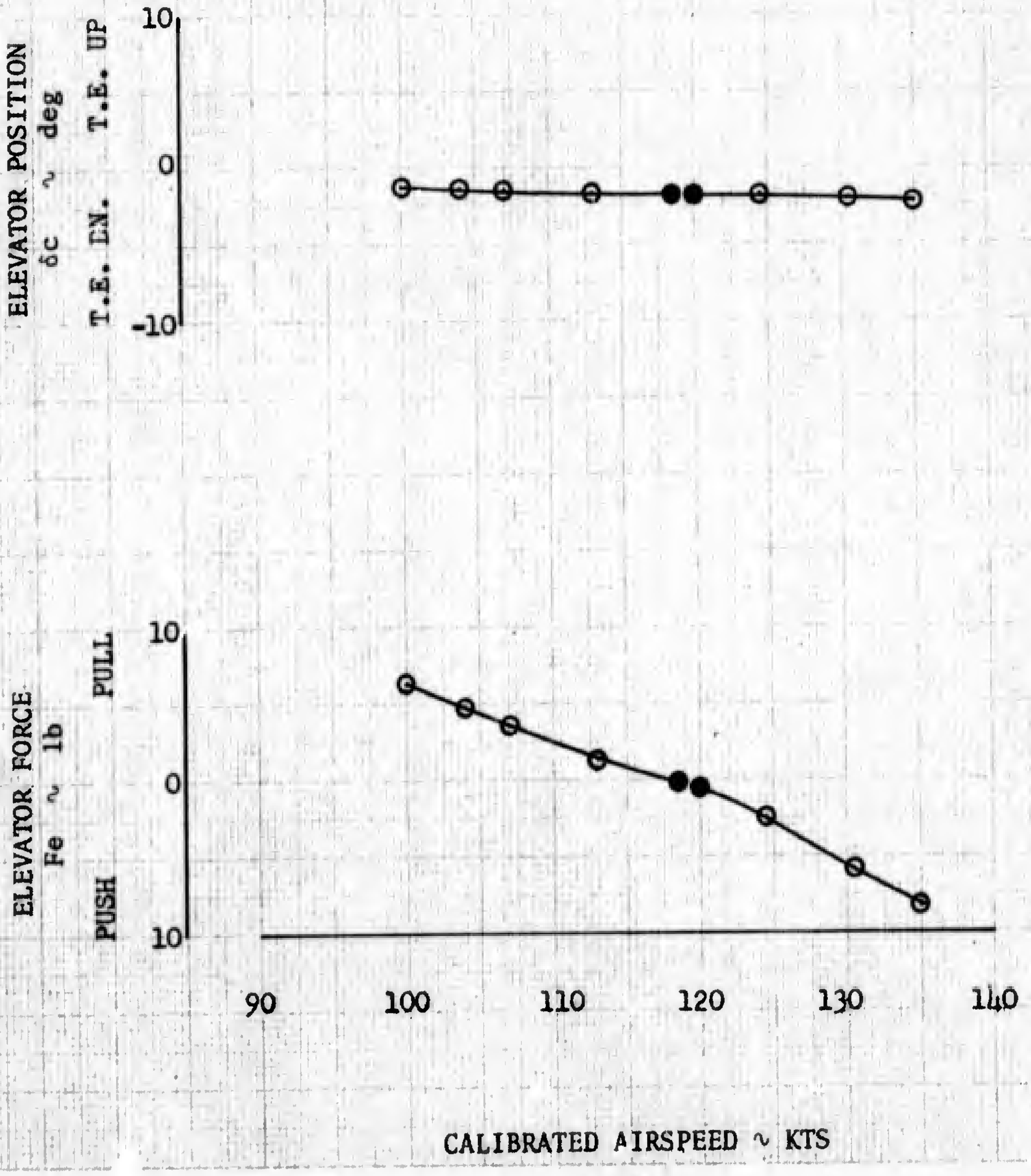


Figure No. 22
STATIC LONGITUDINAL STABILITY
RU-8D S/N 57-6063

SYM	GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
O	7060 LB	124.1 IN.	4810 FT	POWER APPROACH GEAR DOWN 30° FLAPS

NOTE:
1. FULL ELEVATOR TRAVEL
   UP = 25.17 DEG
   DOWN = 14.67 DEG
2. SOLID SYMBOLS DENOTE TRIM POINTS

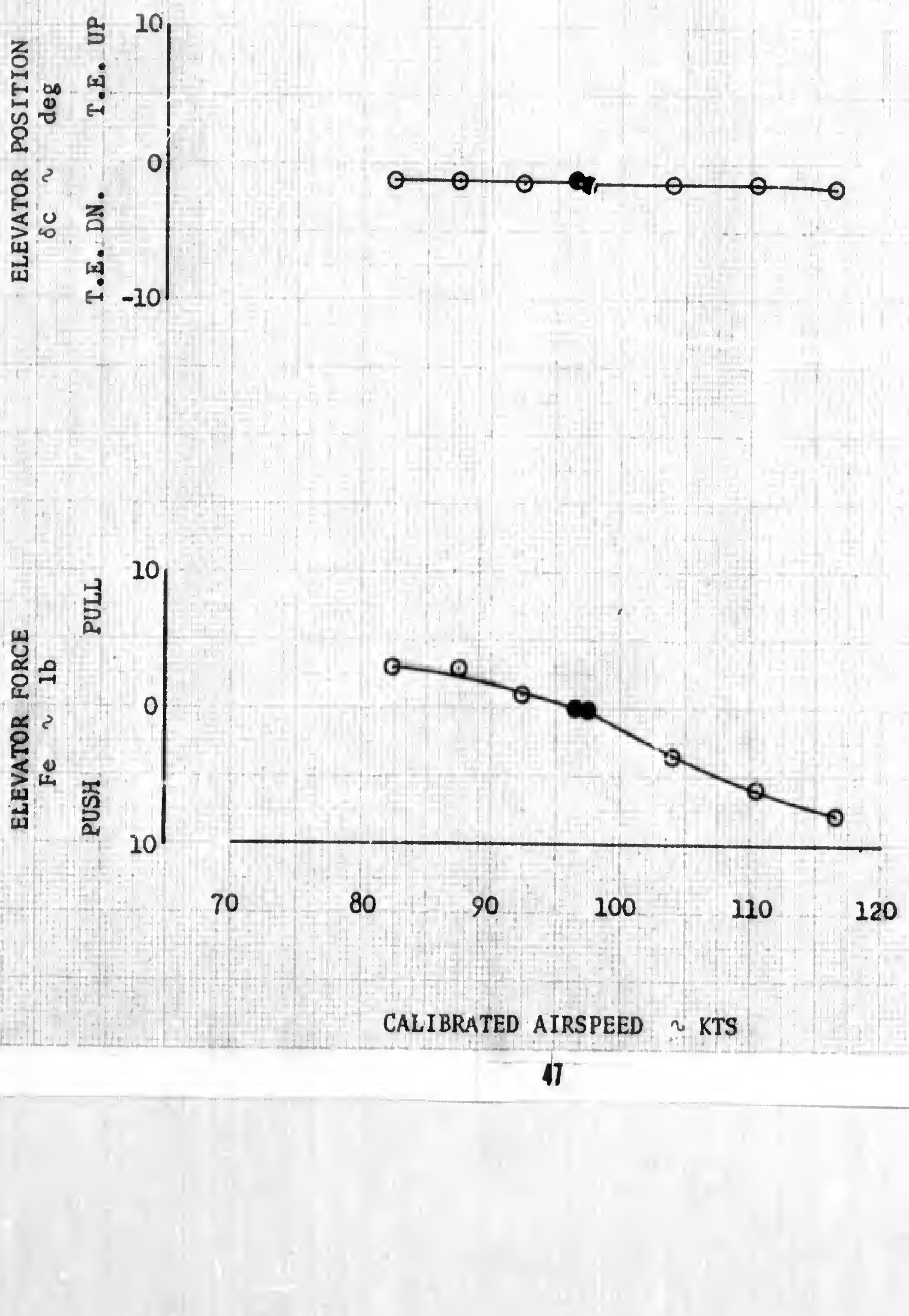
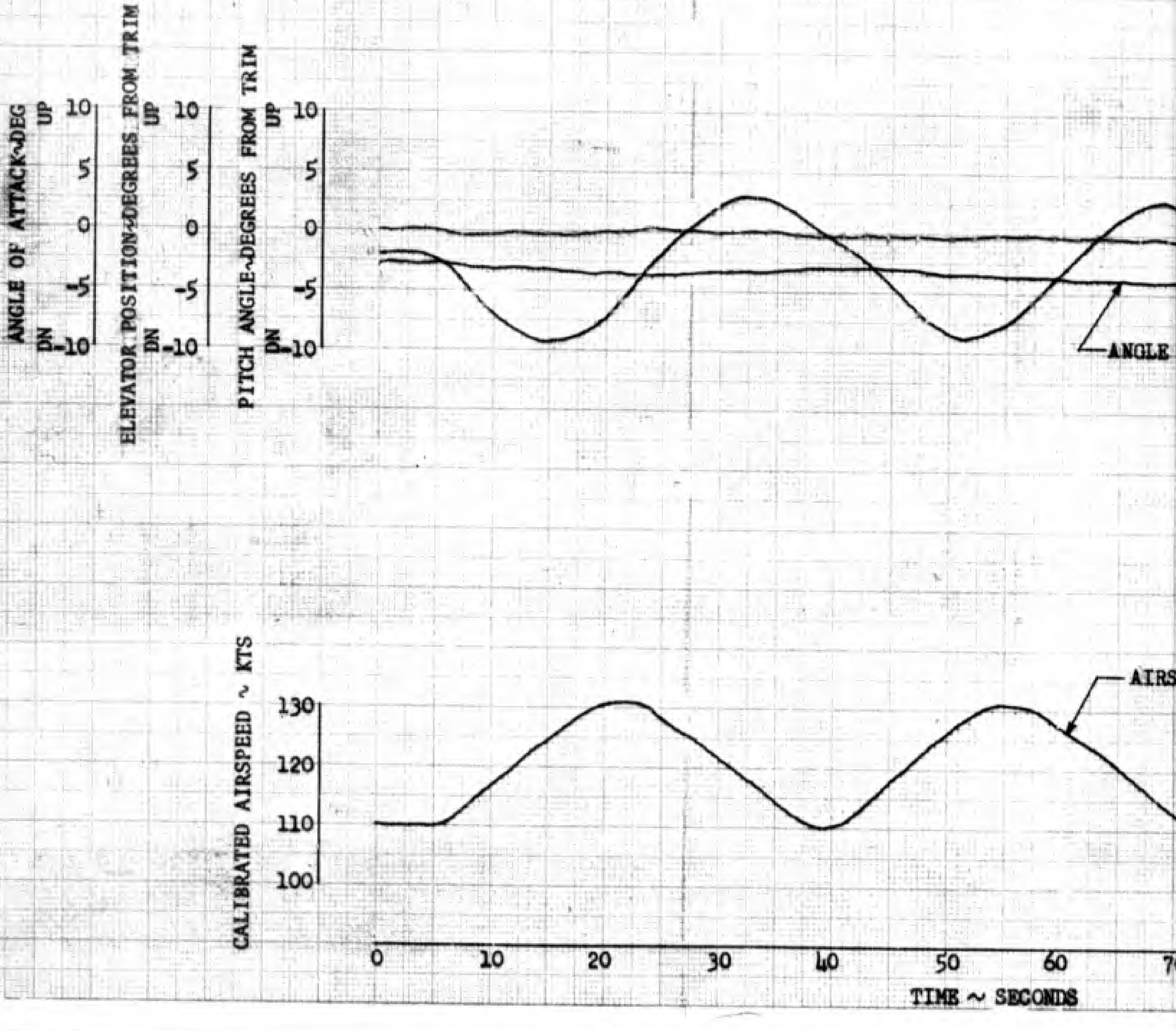


FIGURE NO. 23

LONG PERIOD OSCILLATION

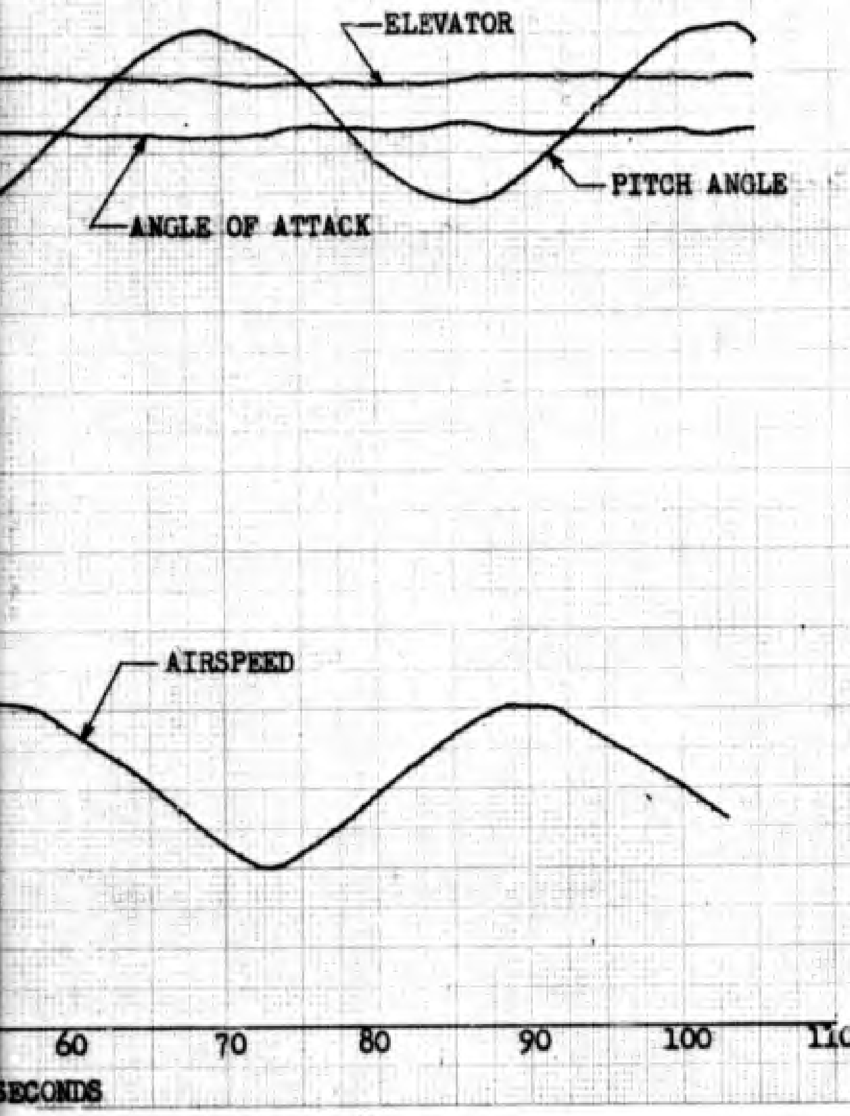
RU-8D S/N 57-6067

GROSS WEIGHT ~ LB	C.G. STATION ~ INCH	DENSITY ALTITUDE ~ FT	CONFIGURATION
6780	122.5	4980	CRUISE



ONFIGURATION

CRUISE



BINDING EDGE

2

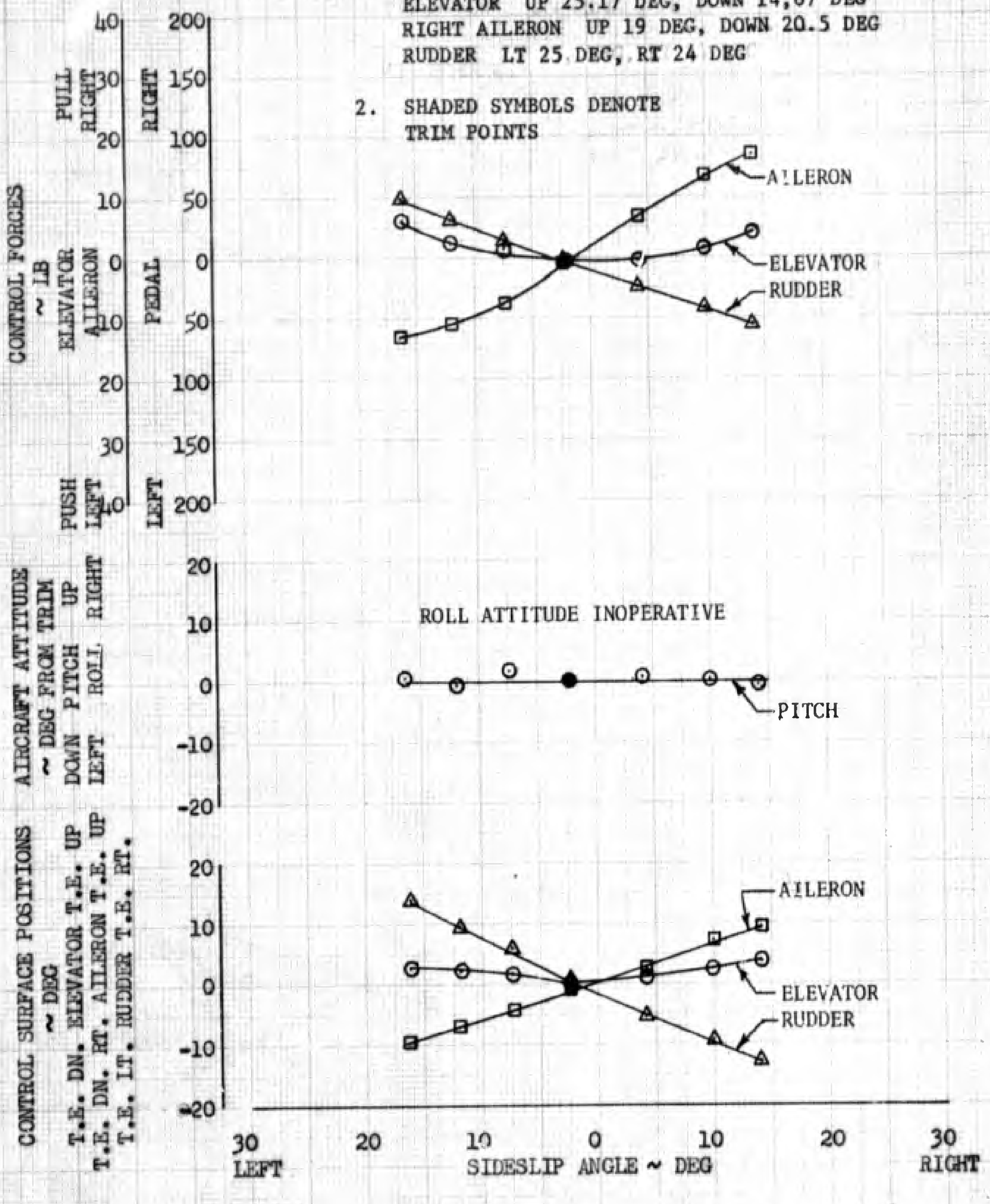
Figure No. 24

STATIC LATERAL-DIRECTIONAL STABILITY

RU-8D S/N 57-6063

90 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
6640 LB	122.2 IN.	5120 FT	CRUISE



NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14,67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE
TRIM POINTS

ROLL ATTITUDE INOPERATIVE

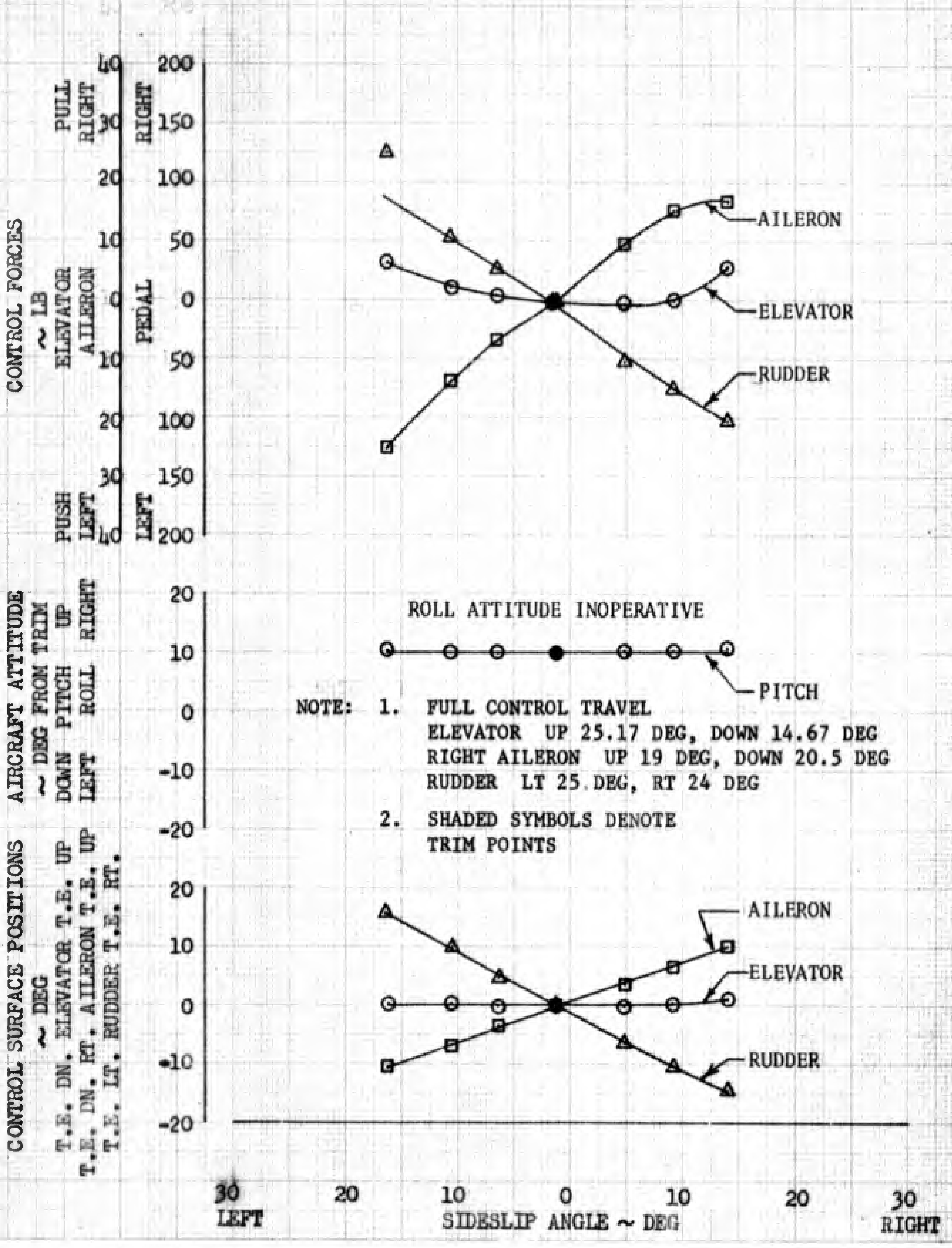
Figure No. 25

STATIC LATERAL-DIRECTIONAL STABILITY

RU-8D S/N 57-6063

110 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
6700 LB	122.2 IN.	4990 FT	CRUISE



NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE
TRIM POINTS

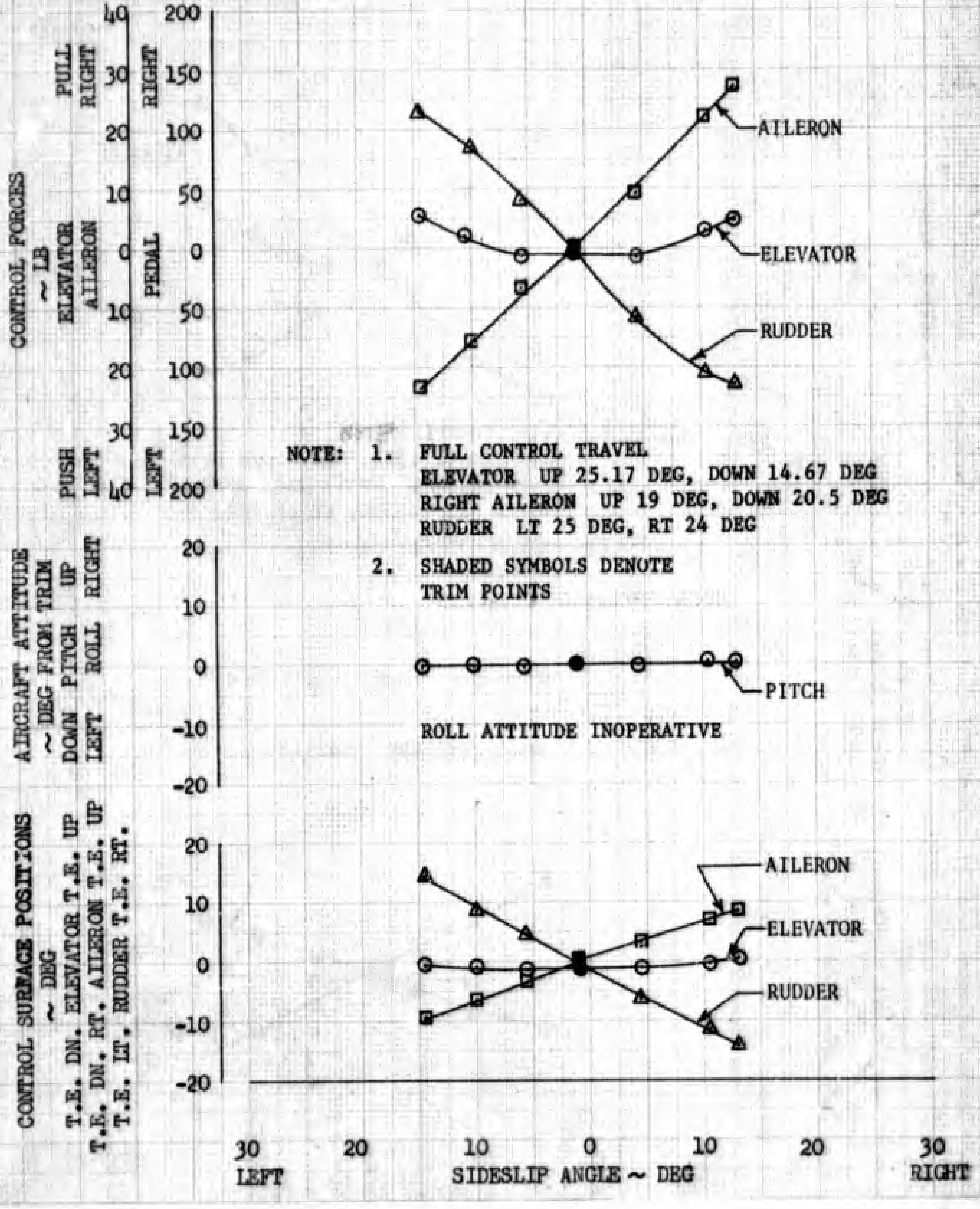
Figure No. 26

STATIC LATERAL-DIRECTIONAL STABILITY

RU-8D S/N 57-6063

125 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
6760 LB	122.2 IN.	4660 FT	CRUISE



NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE TRIM POINTS

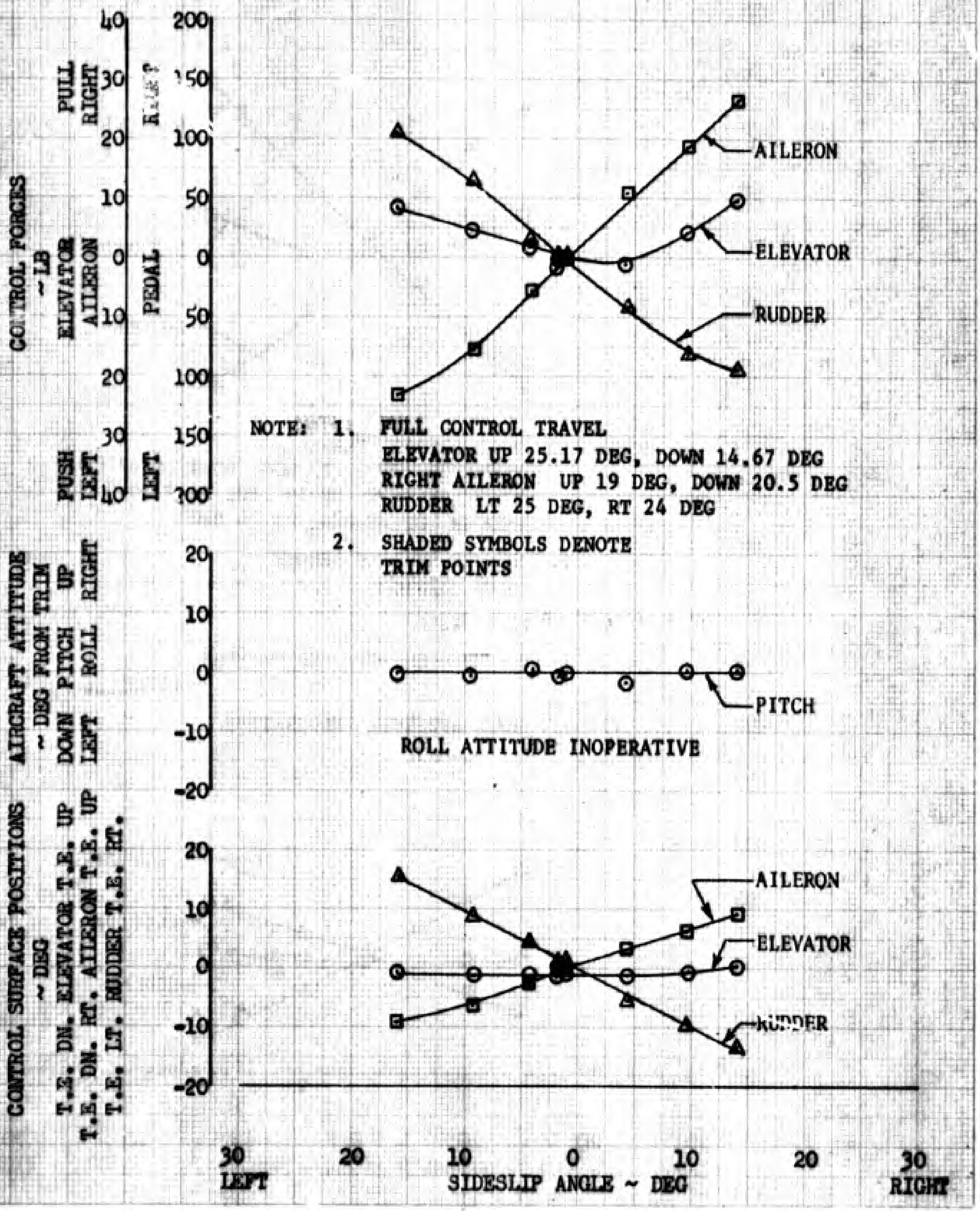
ROLL ATTITUDE INOPERATIVE

Figure No. 27

STATIC LATERAL-DIRECTIONAL STABILITY
RU-8D S/N 57-6063

120 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
6780 LB	122.5 IN.	5140 FT	POWER APPROACH GEAR DOWN



NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE TRIM POINTS

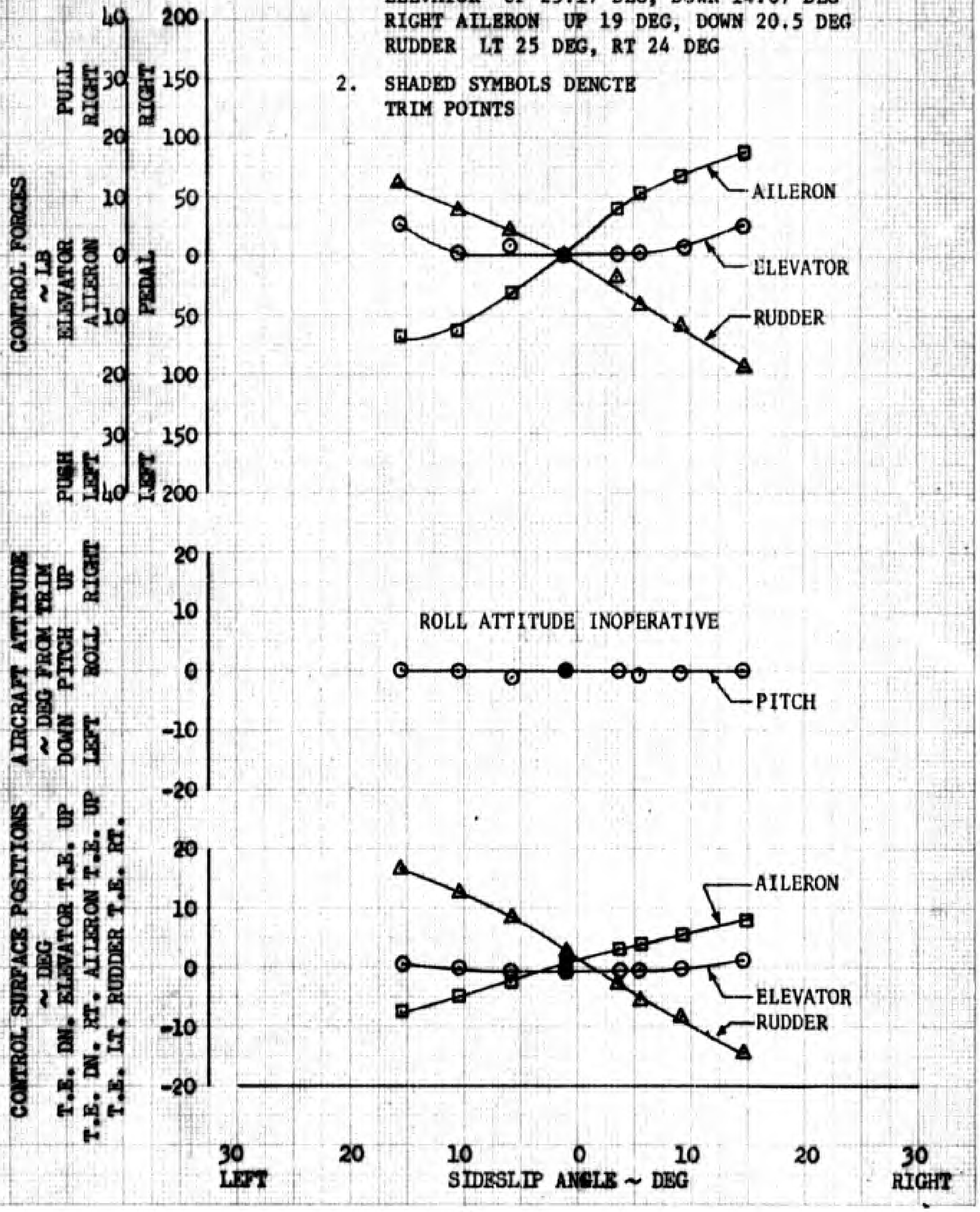
ROLL ATTITUDE INOPERATIVE

Figure No. 28

STATIC LATERAL-DIRECTIONAL STABILITY
RU-8D S/N 57-6063

100 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
6710 LB	122.5 IN.	5300 FT	POWER APPROACH GEAR DOWN 30° FLAPS



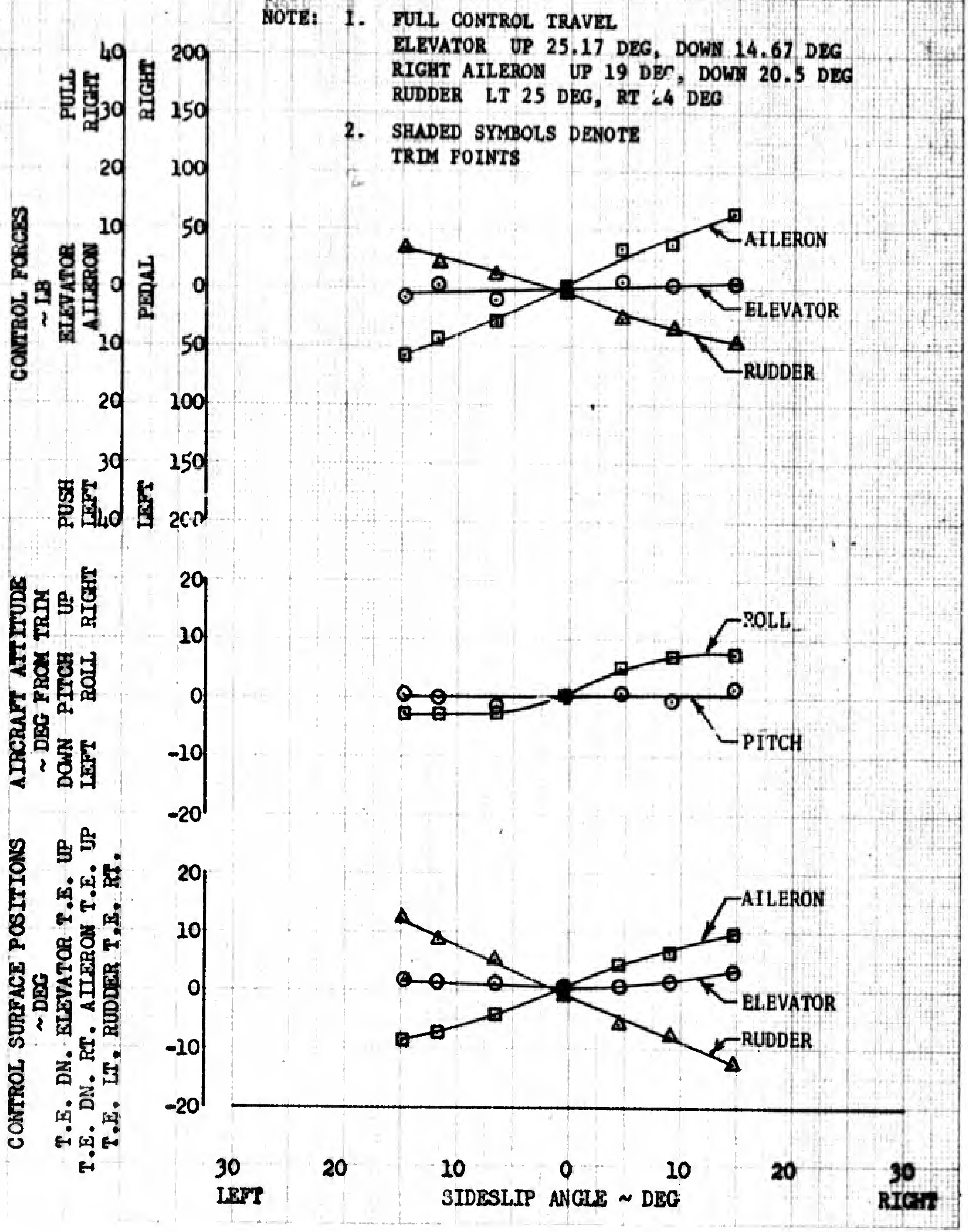
NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE TRIM POINTS

Figure No. 29
STATIC LATERAL-DIRECTIONAL STABILITY
RU-8D S/N 57-6063

90 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
7170 LB	123.9 IN.	4920 FT	CRUISE



NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE
TRIM POINTS

Figure No. 30

STATIC LATERAL-DIRECTIONAL STABILITY

RU-8D S/N 57-6063

110 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
7120 LB	123.9 IN.	4820 FT	CRUISE

NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE TRIM POINTS

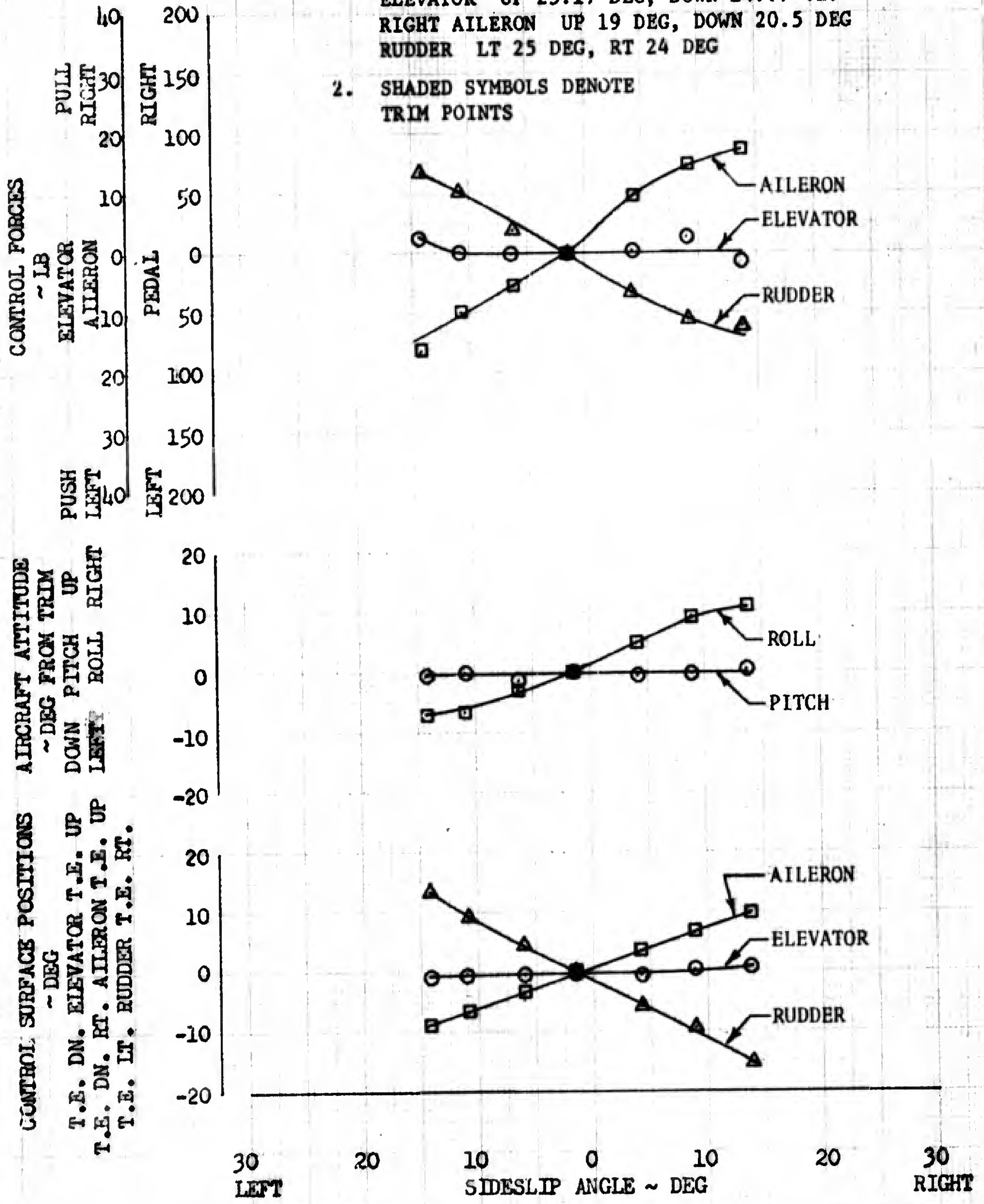


Figure No. 31
STATIC LATERAL-DIRECTIONAL STABILITY

125 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
7240 LB.	123.9 IN.	4790 FT	CRUISE

NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE TRIM POINTS

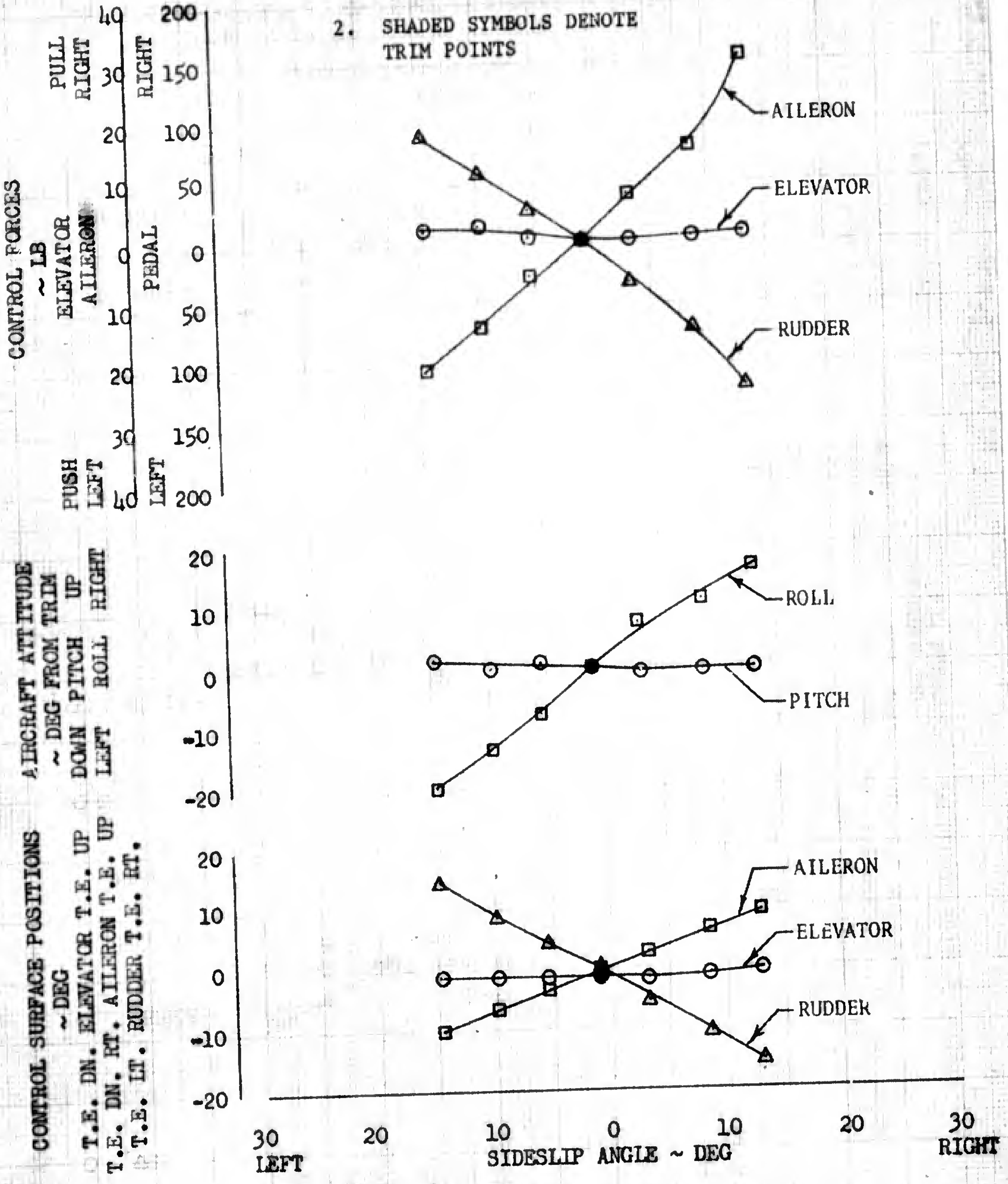


Figure No. 32

STATIC LATERAL-DIRECTIONAL STABILITY

RU-8D S/N 57-6063

120 KNOTS CALIBRATED AIRSPEED

GROSS WT	C.G. STATION	DENSITY ALT	CONFIGURATION
7060 LB	124.1 IN.	4940 FT	POWER APPROACH GEAR DOWN

NOTE: 1. FULL CONTROL TRAVEL
ELEVATOR UP 25.17 DEG, DOWN 14.67 DEG
RIGHT AILERON UP 19 DEG, DOWN 20.5 DEG
RUDDER LT 25 DEG, RT 24 DEG

2. SHADED SYMBOLS DENOTE
TRIM POINTS

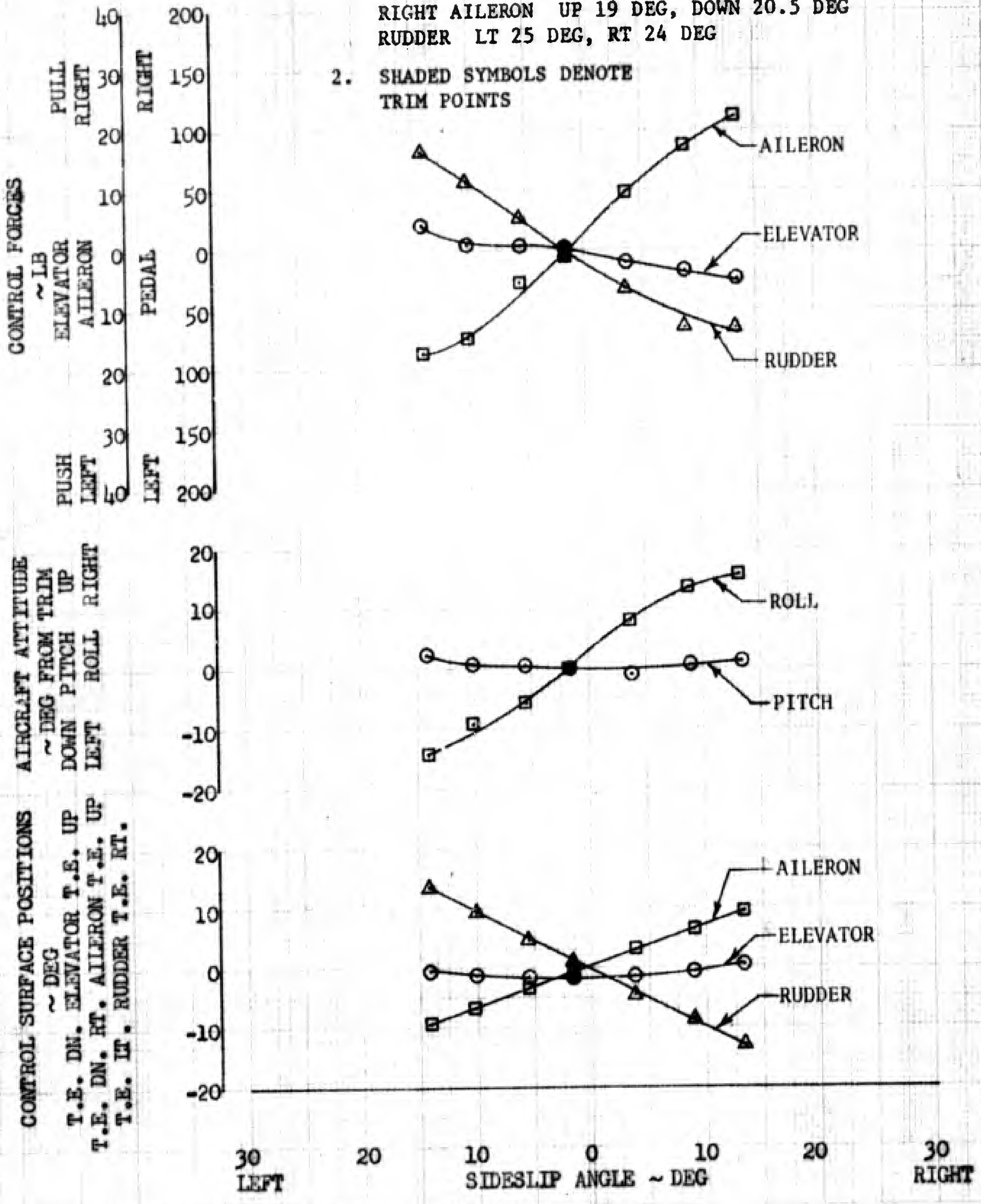
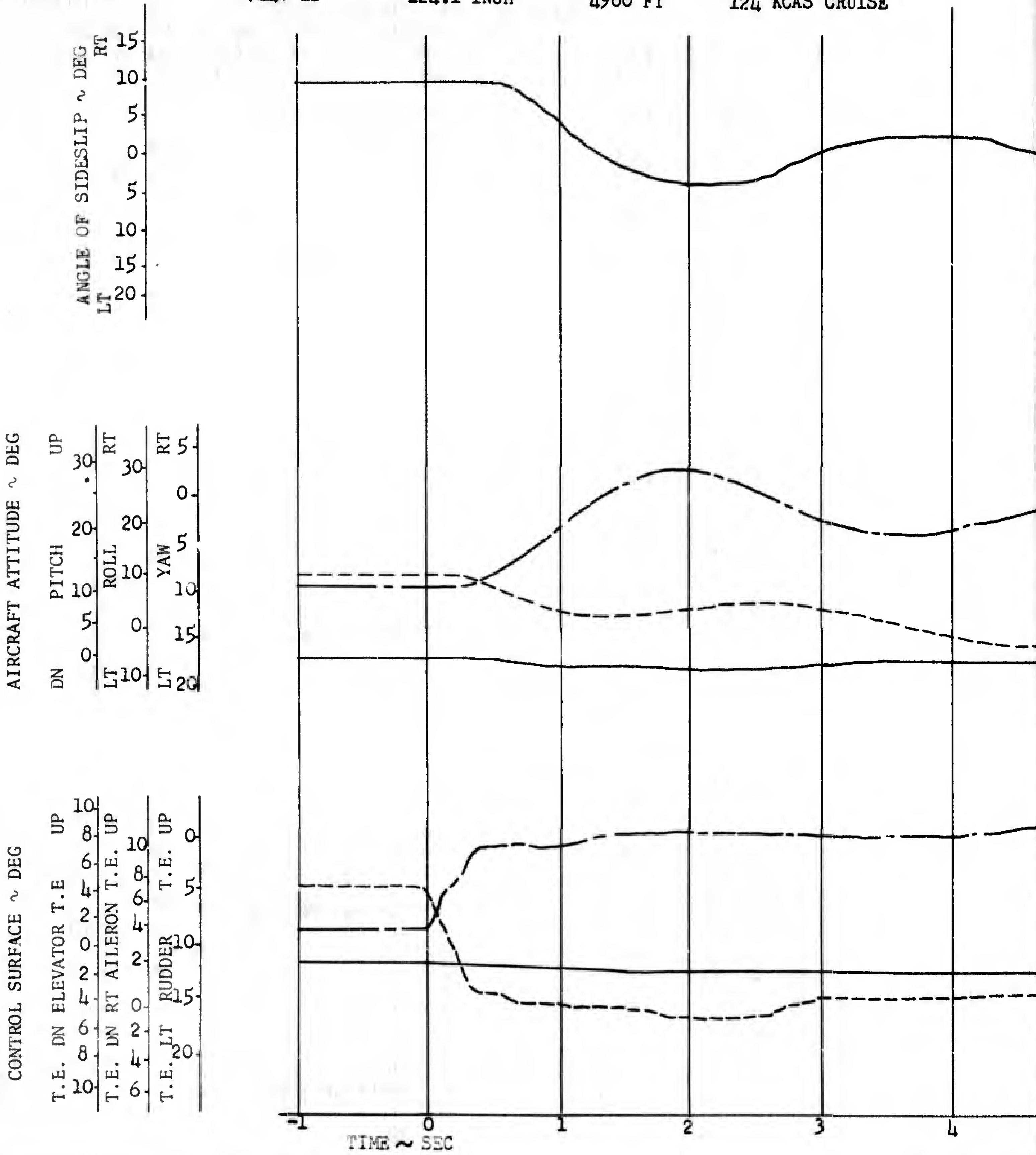
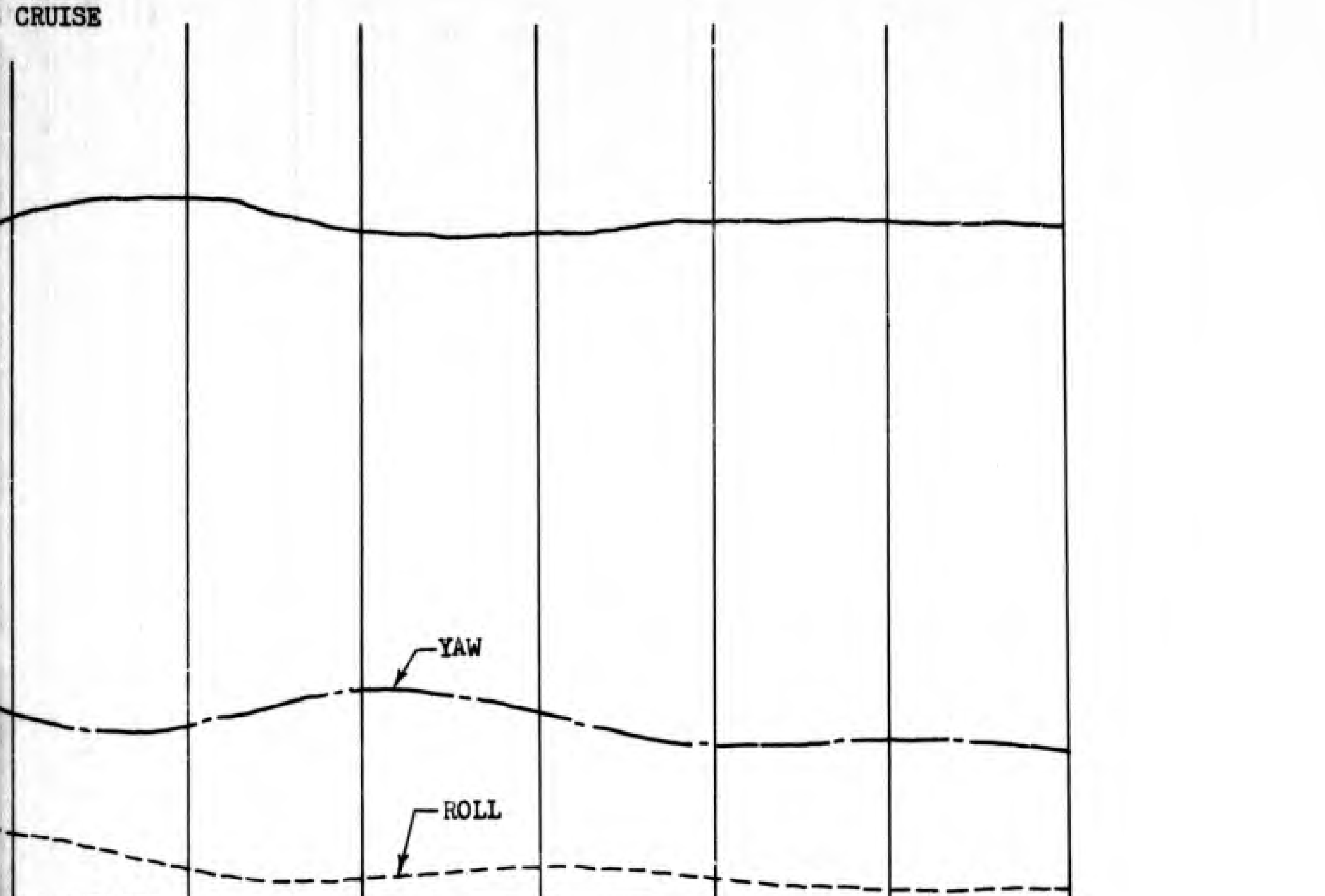


Figure No. 33
LATERAL DIRECTIONAL OSCILLATION
RU-8D S/N 57-6063

GROSS WT	C.G. POSITION	DENSITY ALT	CONFIGURATION
7240 LB	124.1 INCH	4980 FT	124 KCAS CRUISE



RATION
CRUISE



2

FIGURE NO. 34
SPIRAL STABILITY
RU-8D S/N 57-6063

GROSS WEIGHT ~ LB	CG POSITION ~ IN.	DENSITY ALT ~ FT	CONFIGURATION
7070	123.6	5220	90 KCAS CRUISE

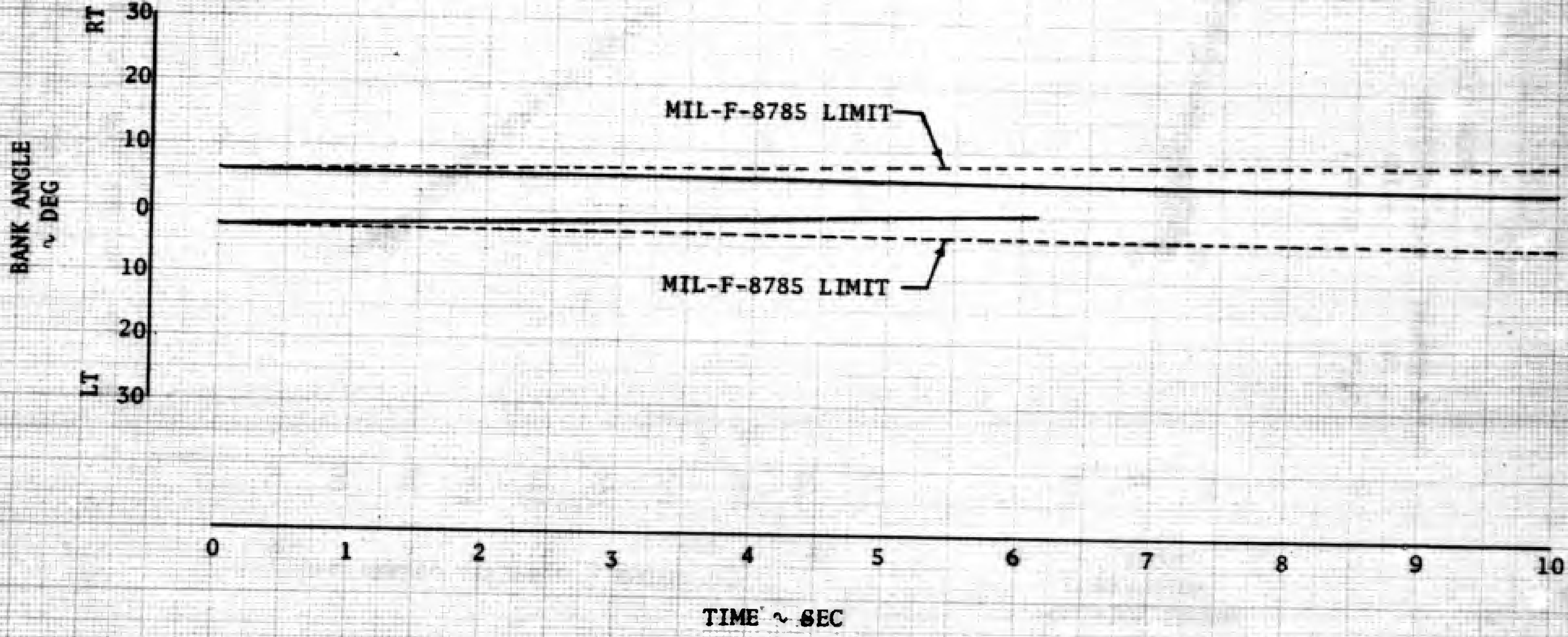


Figure No. 35

AIRSPEED CALIBRATION

RU-8D S/N 57-6063

BOOM SYSTEM

GROSS WEIGHT ~ lb	C.G. STATION ~ in.	DENSITY ALTITUDE ~ ft	CONFIGURATION
7150	124.1	4200	○ CRUISE □ POWER APPROACH GEAR DOWN 30° FLAPS

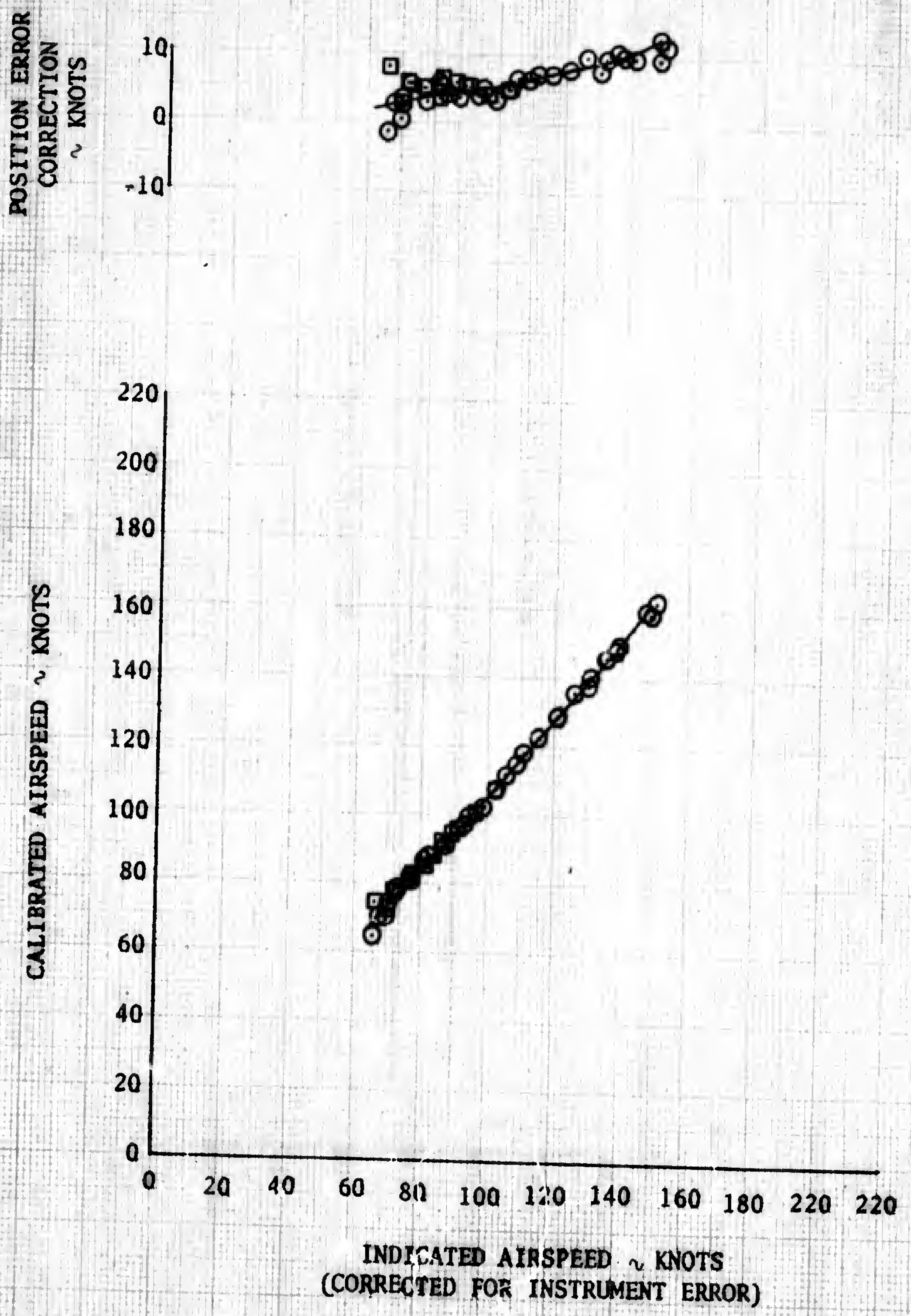


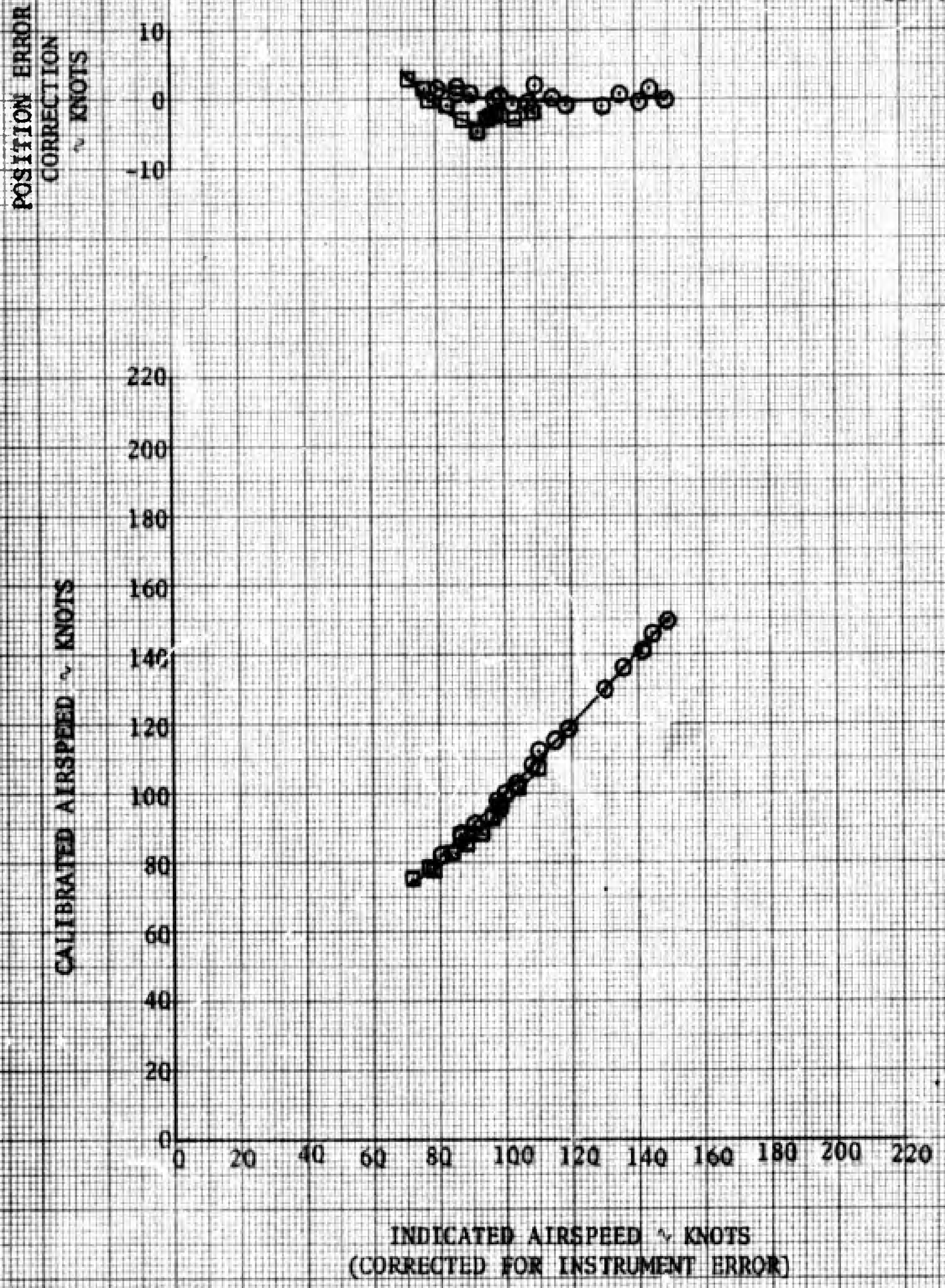
Figure No. 36

AIRSPEED CALIBRATION

RU-8D S/N 57-6063

STANDARD SYSTEM

GROSS WEIGHT ~ lb	C.G. STATION ~ in.	DENSITY ALTITUDE ~ ft	CONFIGURATION
7150	124.1	4200	○ CRUISE □ POWER APPROACH GEAR DOWN 30° FLAPS



APPENDIX VII. DATA REDUCTION METHODS

AIRSPEED DETERMINATION

1. Test calibrated airspeeds (V_{cal}) were obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}) and position error (ΔV_{pc}).

$$V_{cal} = V_i + \Delta V_{ic} + \Delta V_{pc} \tag{1}$$

2. Test true airspeeds (V_{t_t}) were determined from the test day air density ratio (σ) and calibrated airspeed as follows:

$$V_{t_t} = V_{cal}/\sigma^{\frac{1}{2}}_{test} \tag{2}$$

3. Test true airspeeds were corrected to standard true airspeeds (V_{t_s}) according to the equation:

$$V_{t_s} = V_{t_t} (T_{a_s}/T_{a_t})^{\frac{1}{2}} \tag{3}$$

where: T_{a_s} = Standard ambient temperature

T_{a_t} = Test free air temperature

AMBIENT AIR TEST PARAMETERS

4. Pressure altitudes (H_p) were obtained by correcting indicated pressure altitude (H_{p_i}) for instrument error ($\Delta H_{p_{ic}}$).

$$H_p = H_{p_i} + \Delta H_{p_{ic}} \tag{4}$$

5. Ambient test pressures (P_{a_t}) were determined from pressure altitudes and *US Standard Atmosphere, 1962* tables.

6. Ambient test temperatures (T_{a_t}) were obtained by correcting the indicated test temperature ($T_{a_{ti}}$) for instrument error.

$$T_{a_t} = T_{a_{ti}} + \Delta T_{ic} \tag{5}$$

7. The test density ratio (σ_{test}) was determined from the following relationship:

$$\sigma_{test} = (T_o/T_{a_t})(P_{a_t}/P_o) \tag{6}$$

where: T_o = Standard day, sea level temperature

P_o = Standard day, sea level pressure

8. The density altitudes (H_D) were determined from the test density ratios (σ_{test}) and *US Standard Atmosphere, 1962* tables.

GROSS WEIGHT DETERMINATION

9. Airplane test gross weights (W_t) were calculated as follows:

$$W_t = W_{es} - (FC)(k)(\text{fuel density}) \tag{7}$$

where: W_{es} = Gross weight of the aircraft at the time engines were started

FC = Fuel counter reading

k = Constant to convert fuel counter reading into the amount of fuel used (in gallons)

POWER-REQUIRED DETERMINATION

10. The engines used for this test program were certified by Columbia Aircraft Services, Bloomsburg, Pennsylvania. The certified engine test cell data were corrected to standard day, SL conditions and compared with the specification power chart data (ref 7, app I). The certified engine test cell data were found to be within one-half

of 1 percent of the specification power chart data; therefore, the specification power chart was used to derive the standard day, power-required data, in accordance with the method presented in paragraphs 11 through 14 (ref 6, app I).

LEVEL FLIGHT PERFORMANCE

11. Test day power required (BHP_t) was calculated from:

$$BHP_t = BHP_c \, (DT_{a_s}/CAT_t)^{1/2} \tag{8}$$

where: BHP_c = Brake horsepower determined from the specification power chart under existing test conditions of engine speed (N_e), engine manifold pressure (MAP) and pressure altitude (H_p)

CAT_t = Instrument corrected carburetor air temperature

DT_{a_s} = Standard temperature corresponding to the instrument-corrected carburetor deck pressure

12. Test day power required was corrected to standard day temperature and standard weight (W_s) by the following equation:

$$BHP_s' = BHP_t \, (T_{a_s}/T_{a_t})^{1/2} + \Delta BHP_w \tag{9}$$

where: BHP_s' = Standard day, weight-corrected power required; uncorrected for carburetor air temperature and manifold pressure

ΔBHP_w = Weight correction term

The second term of equation 9 is further defined as:

$$\Delta BHP_w = \frac{0.288(W_s^2 - W_t^2)(T_{a_s}/T_{a_t})^{1/2}}{eb^2 \, \sigma_t \, V_{t_t} \, N_p} \tag{10}$$

where: e = Airplane efficiency factor

b = Wing span

N_p = Propeller efficiency

13. For partial throttle operation, a carburetor air temperature correction was applied to BHP'_s to obtain the standard brake horsepower required (BHP_s).

$$BHP_s = BHP'_s \, (CAT_t/CAT_s)^{\frac{1}{2}} \qquad (11)$$

where: CAT_s = Standard carburetor air temperature

14. For full throttle operation, it was necessary to apply a manifold pressure correction, as well as the carburetor air temperature correction, to BHP'_s in order to obtain the standard brake horsepower.

$$BHP_s = BHP'_s \, (MAP_s/MAP_t)(CAT_t/CAT_s)^{\frac{1}{2}} \qquad (12)$$

where: MAP_t = Instrument corrected test manifold pressure

MAP_s = Standard manifold pressure

15. The standard brake horsepower required per engine (from equation 11 for partial throttle and equation 12 for full throttle) was plotted against standard true airspeed for each altitude and gross weight configuration (figs. 3 through 8, app VI).

16. The power-required data obtained for each pressure altitude and gross weight were generalized (fig. 9, app VI) according to the following equations:

$$P_{IW} = BHP_t \, \sigma^{\frac{1}{2}} \, (W_s/W_t)^{1\frac{1}{2}} \qquad (13)$$

$$V_{IW} = V_t \, (\sigma W_s/W_t)^{\frac{1}{2}} \qquad (14)$$

where: P_{IW} = Generalized power parameter

V_{IW} = Generalized airspeed parameter

W_s = Standard airplane gross weight (7350 pounds)

W_t = Test airplane gross weight

V_{t_t} = Test true airspeed

17. Engine fuel-flow (W_f) data were obtained from the engine model specification (ref 7, app I). For each test gross weight and pressure altitude, the specific range (SR_g in nautical air miles per pound of fuel (NAMPP)) was computed.

$$SR_g = V_{t_s}/W_f \tag{15}$$

18. The specific range was plotted against true airspeed for each pressure altitude and gross weight (figs. 3 through 8, app VI), and the recommended cruise airspeed was indicated on each plot (0.99 $SR_{g_{max}}$).

19. Maximum endurance airspeeds were obtained from each speed-power plot as those airspeeds corresponding to the minimum power required.

20. Brake specific fuel consumption (BSFC) was computed for each test gross weight and pressure altitude from the relationship:

$$BSFC = W_f/BHP_s \tag{16}$$

21. The brake specific fuel consumption was plotted against the brake horsepower per engine for each test gross weight and pressure altitude (figs. 3 through 8, app VI).

TAKEOFF PERFORMANCE

22. The total horizontal takeoff distance (S) required for the airplane to clear a 50-foot obstacle was determined from the corrected observed ground roll distance (S_g) and the corrected test airborne horizontal distance (S_a).

$$S = S_g + S_a \qquad (17)$$

23. Observed values of ground roll distance ($S_{g_{t_w}}$) and the airborne horizontal distance ($S_{a_{t_w}}$) were obtained using a Fairchild flight analyzer camera.

24. The test ground roll distance was corrected for wind using the empirical equation:

$$S_{g_t} = S_{g_{t_w}} \left[(V_{to} + V_w)/V_{to}\right]^{1.85} \qquad (18)$$

where: S_{g_t} = Observed ground roll distance corrected for wind

$S_{g_{t_w}}$ = Test ground roll distance

V_{to} = True ground speed at liftoff (from Fairchild flight analyzer camera)

V_w = Velocity of the wind component along the runway (+ for headwind; - for tail wind)

25. The wind-corrected ground roll distance was then corrected for runway slope.

$$S'_{g_t} = S_{g_t}/(1 + 2gS_{g_t} \operatorname{Sin}\theta/V_{to}^2) \qquad (19)$$

where: S'_{g_t} = Observed ground roll distance corrected for wind and runway slope

g = Acceleration due to gravity

θ = Slope of runway in degrees

26. The ground roll distance was next corrected for variation in the weight, density, propeller speed and engine power parameters from standard by the relationship:

$$\Delta S'_{g_t} = S'_{g_t} (2.6\Delta W/W_t - 1.7\Delta\sigma/\sigma_t - 0.9\Delta P/P_t - 0.7\Delta N/N_t) \qquad (20)$$

where: ΔW = Standard gross weight minus test gross weight

W_t = Test gross weight

$\Delta\sigma$ = Standard density ratio at field elevation minus test density ratio

σ_t = Test density ratio

ΔP = Standard power minus test power (standard power was taken as the maximum allowed by the operator's manual)

P_t = Test power

ΔN = Standard propeller speed minus test propeller speed

N_t = Test propeller speed

27. The corrected ground roll distance was obtained from:

$$S_g = S'_{g_t} + \Delta S'_{g_t} \qquad (21)$$

28. Values of the test airborne horizontal air distance were corrected for wind using the relationship:

$$S_{a_t} = S_{a_{t_w}} + V_w t \qquad (22)$$

where: S_{a_t} = Observed airborne distance corrected for wind

$S_{a_{t_w}}$ = Test airborne distance

V_w = Velocity of the wind component along the runway (+ for headwind; - for tail wind)

t = Time from liftoff to the height of 50 feet

29. Wind corrected values of the airborne horizontal air distance were then corrected for variations in the weight, density, propeller speed and engine power parameters from standard according to the equation:

$$\Delta S_{a_t} = S_{a_t} (2.3\Delta W/W_t - 1.2\Delta\sigma/\sigma_t - 0.8\Delta N/N_t - 1.1\Delta P/P_t) \quad (23)$$

30. The corrected airborne distances were then determined:

$$S_a + S_{a_t} + \Delta S_{a_t} \quad (24)$$

LANDING PERFORMANCE

31. The total horizontal landing distance (S) required for the airplane to land over a 50-foot obstacle and come to a complete stop was determined from the corrected test airborne horizontal distance (S_a) and the corrected observed ground roll (S_g).

$$S = S_g + S_a \quad (25)$$

32. Observed values of airborne horizontal distance ($S_{a_{t_w}}$) and ground roll distance ($S_{g_{t_w}}$) were obtained using a Fairchild flight analyzer camera.

33. The test airborne horizontal distance was corrected for wind according to the equation:

$$S_a = S_{a_{t_w}} + V_w t \quad (26)$$

where: S_a = Observed airborne landing distance corrected for wind

$S_{a_{t_w}}$ = Test airborne landing distance

V_w = Velocity of the wind component along the runway (+ for headwind; - for tail wind)

t = Time from a height of 50 feet to touchdown

34. Observed values of the landing ground roll distance were corrected for wind and for variations in the weight and density parameters from standard using the following equation:

$$S_g' = S_{g_{t_w}} [(V_{td} + V_w)/V_{td}]^{1.85} (W_s/W_t)^2 (\sigma_t/\sigma_s) \quad (27)$$

where: S_g' = Landing ground distance uncorrected for runway slope

$S_{g_{t_w}}$ = Test landing ground distance

V_{td} = True ground speed at touchdown

V_w = Velocity of the wind component along the runway (+ for headwind; - for tail wind)

W_s = Standard airplane weight

W_t = Test airplane weight

σ_t = Test density ratio

σ_s = Standard density ratio at field elevation

35. Standard landing ground roll distance (S_g) was then obtained by correcting for the runway slope (θ).

$$S_g = S_g'/(1 + 2gS_g' \sin\theta/V_{td}^2) \tag{28}$$

36. True airspeeds at liftoff, touchdown and at a height of 50 feet were determined by correcting the true ground speeds obtained from the Fairchild flight analyzer for the wind component along the runway for takeoffs and landings.

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13. ABSTRACT

The limited airworthiness and flight qualification test (Phase D) evaluation of the RU-8D airplane Winebottle configuration was conducted to obtain quantitative handbook data for accurate and safe mission planning. The tests included level flight, landing and takeoff performance; stalls and single-engine characteristics; and longitudinal and lateral-directional handling qualities. Forty-eight test flights were flown for a total of 51 productive flight test hours. Two shortcomings were noted for which correction is desirable to improve mission effectiveness: poor sensitivity of the aileron trim and the masking of the longitudinal control force gradient by the breakout forces. Within the scope of this test, the performance capabilities and the handling qualities of the RU-8D are satisfactory for the reconnaissance mission.

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RU-8D airplane						
Winebottle configuration						
Aileron trim sensitivity						
Longitudinal control force gradient						