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ARMY PRELIMINARY EVALUATION II.
MODEL 200 CEFLY LANCER

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Army Aviation Engineering Flight Activity
Edwards Air Force Base, California

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The United States Army Aviation Engineering Flight Activity conducted a Preliminary Evaluation (APE) II of the Model 200 CEFLY LANCER aircraft, manufactured by Beech Aircraft Corporation, from 27 April to 15 May 1974 at the Beech facility in Wichita, Kansas. During the test program, 7.3 productive hours were flown. Performance, stability and control characteristics, and miscellaneous engineering tests were conducted. During these tests, no additional deficiencies or

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19. Key Words

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20. Abstract

shortcomings were noted. A deficiency determined during APE I and still present during APE II was the loss of power to primary attitude and heading gyros when propeller speed was less than 2000 rpm. It was not possible to duplicate the conditions under which the deficiency of smoke in the cockpit and cabin areas at altitudes above 15,000 feet pressure altitude (determined during APE I) was noted; therefore, it was impossible to determine whether or not the contractor modifications eliminated this deficiency. Four previously determined shortcomings remained uncorrected. The most significant shortcoming was the inadequate single-engine performance under heavy gross weight or high temperature conditions in the single-engine takeoff configuration.

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USAAEFA PROJECT NO. 74-21

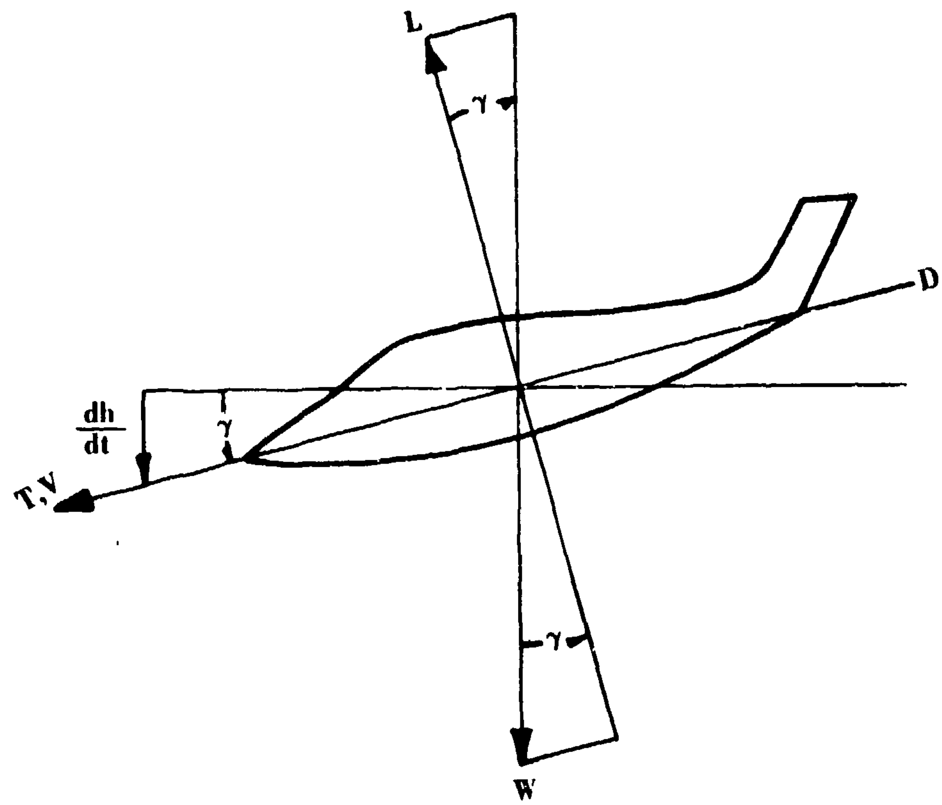
ARMY PRELIMINARY EVALUATION II

MODEL 200 CEFLY LANCER

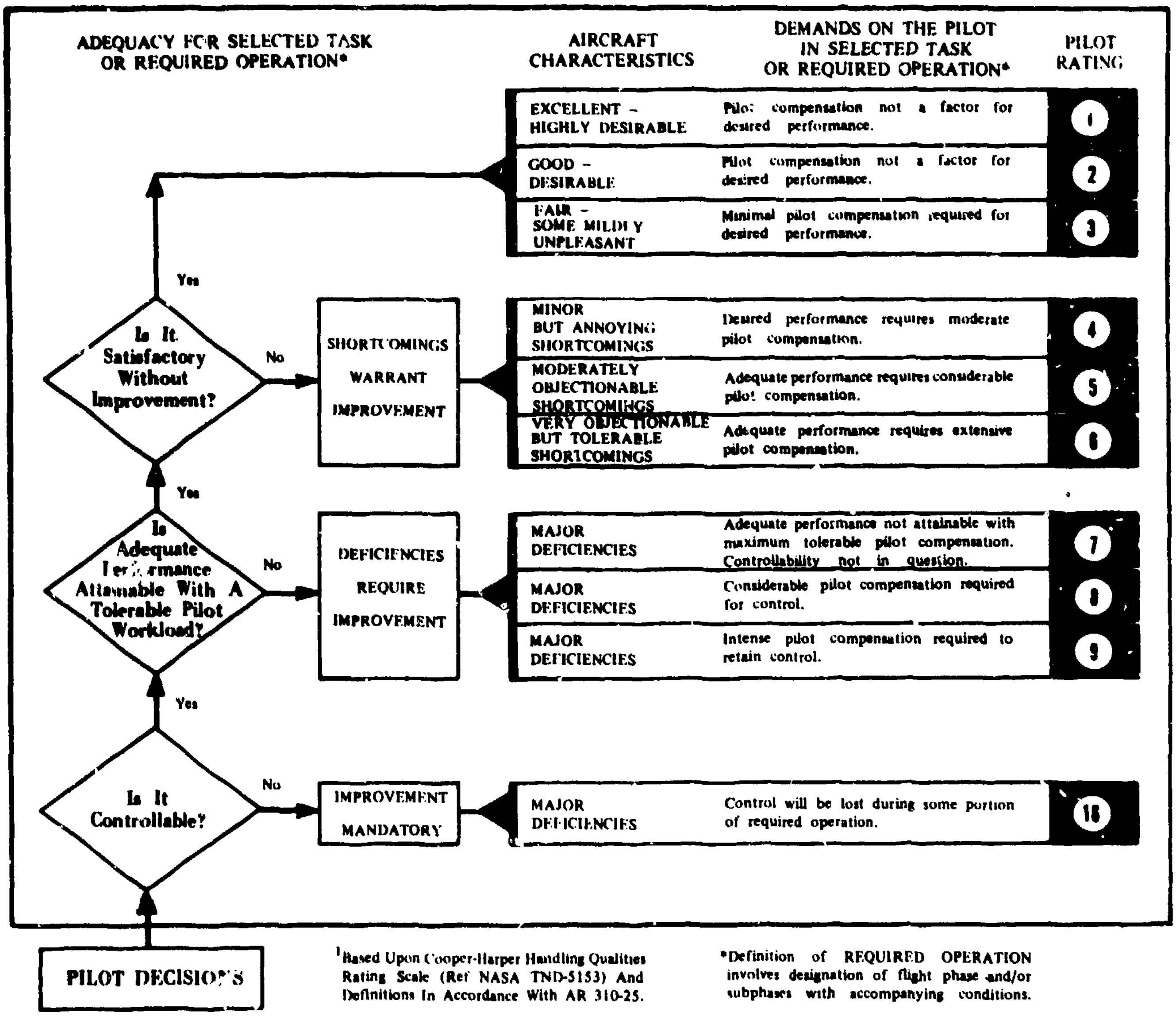
UNITED STATES ARMY AVIATION ENGINEERING TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

1. Page 25, photo C-1: The photograph was reversed in photo reprocessing. Although this photo is not reprinted, readers are advised of this reverse printing.

2. Figure in paragraph 3, page 32: Figure depicted below replaces figure presented on page 32.



3. Appendix D, page 40. Add Figure 1, Handling Qualities Rating Scale.



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

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

Figure 1. Handling Qualities Rating Scale.

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INTRODUCTION

BACKGROUND

1. In June 1971, a contract was initiated by the United States Army Aviation Systems Command (AVSCOM) with Beech Aircraft Corporation (BAC) for the procurement of three modified KingAir Model A100 (U-21F) aircraft. These aircraft were to be utilized by the United States Army Security Agency as research and development test-bed aircraft in support of classified CEFLY LANCER mission requirements. The original contract was modified in June 1973 to permit the procurement of three "T"-tailed Model 200 aircraft in lieu of the modified U-21F. These aircraft are currently being type certificated in the normal category of Federal Air Regulation (FAR) Part 23 (ref 1, app A) of the Federal Aviation Administration (FAA). Within this category the maximum gross weight may not exceed 12,500 pounds. The contractor has performed test and analysis which permits military qualification to extend the maximum gross weight to 15,000 pounds with reduced maneuvering and airspeed limitations. This additional gross weight capability was essential to the inclusion of all desired mission equipment for test-bed purposes. In November 1973, the United States Army Aviation Engineering Flight Activity (USAAEFA) was tasked by an AVSCOM test directive (ref 2) to conduct an Army Preliminary Evaluation (APE) on a prototype BAC Model 200 aircraft. The APE was to be conducted in two phases. The APE I tests were conducted from 20 February to 6 March 1974 at the Beech facility in Wichita, Kansas, using the basic airplane without the external mission equipment installed. The APE II tests were to be conducted with a mission-configured airplane.

TEST OBJECTIVES

2. The objectives of APE II were as follows:

a. Provide quantitative and qualitative engineering flight test data as needed to assist in substantiation of airworthiness at the 15,000-pound gross weight.

b. Verify contract compliance in appropriate areas.

c. Assist in determining the flight envelope to be used for future test-bed flight operations.

d. Provide preliminary aircraft performance data at the military maximum gross weight for operational use.

e. Provide the data required to substantiate a statement of airworthiness qualification of the aircraft after mission provision modification.

DESCRIPTION

3. The test aircraft were two BAC Model 200's, serial numbers (SN) 71-21059 and 71-21060, each powered by two United Aircraft of Canada, Limited (UACL) PT-6A-41 turboprop engines. This aircraft is a prototype military version of the BAC Super KingAir Model 200 pressurized all-weather executive transport. The pilot and copilot are seated side by side with dual flight controls. The tricycle landing gear has dual wheels on each main gear and is retractable. The flight control system is fully reversible. A pneumatic rudder boost is installed to help compensate for asymmetrical thrust and a yaw damper system is provided to improve directional stability. Major differences between the civilian and military versions include the removal of the flight director system, the autopilot, and the weather radar system; the addition of high-flotation landing gear and a fuel dump system; the installation of an 8.5-kilovolt-ampere (KVA) alternating current (AC) generator in a blister on each engine nacelle to provide additional electrical power for classified mission equipment, CEFLY LANCER antenna array, and a 750 volt-ampere (VA) inverter to provide emergency power for the pilot attitude and heading gyro indicators (SN 71-21060 only). A detailed description of the Model 200 aircraft is contained in Beech Prime Item Development Specification BS22296A and Aircraft Procurement Specification, Light Fixed Wing Reconnaissance Aircraft (refs 3 and 4, app A). Appendix B contains a further description of the test aircraft.

TEST SCOPE

4. The APE II tests were conducted at the BAC facility in Wichita, Kansas, from 27 April to 15 May 1974. During the test program, 5 flights were conducted for a total of 9.6 hours, of which 7.3 hours were productive. Aircraft SN 71-21059 was the primary test aircraft utilized for this evaluation. One flight was conducted in aircraft SN 71-21060 to investigate the correction of specific deficiencies which were identified during APE I. The Model 200 aircraft was evaluated to determine performance and handling qualities after the antenna array and mission equipment provisions were installed, and to verify contract compliance. Average test conditions are shown in table 1 and test configurations are shown in table 2. Flight restrictions and operating limitations applicable to this evaluation are contained in the operator's manual (ref 5, app A), as modified by the safety-of-flight releases (refs 6 and 7).

Table 1. Test Conditions.[1]

Test Description	Density Altitude (ft)	Outside Air Temperature (°C)	Gross Weight (lb)	Center-of-Gravity Location (in.)	Trim Calibrated Airspeed (kt)	Configuration
Level flight performance	12,980	5.0	14,500	187.6 (fwd)	[2]110 to 210	Cruise
	7200	13.5	13,920	187.0 (fwd)	[2]110 to 150	Single-engine cruise
Static longitudinal stability	8260	12.0	14,620	187.9 (fwd)	139	Cruise
	7320	14.5	14,780	188.0 (fwd)	170	
	7430	15.5	14,940	188.2 (fwd)	204	
	8260	12.0	14,460	187.7 (fwd)	144	Power-approach
	7320	14.5	14,280	187.5 (fwd)	[3]143	
	7800	8.0	14,360	197.1 (aft)	143	Cruise
	7770	8.5	14,700	197.2 (aft)	168	
	7310	8.5	14,880	197.3 (aft)	202	
	7730	8.0	14,560	197.8 (aft)[4]	170	
	7950	8.0	14,200	197.1 (aft)	142	Power-approach
	7600	9.5	14,040	197.1 (aft)	[3]142	
Static lateral-directional stability	9040	6.0	13,800	197.2 (aft)	145	Cruise
	8860	6.5	13,560	197.4 (aft)	144	Power-approach
Dynamic longitudinal stability	10,760	4.5	14,220	197.1 (aft)	170	Cruise
Dynamic lateral-directional stability	11,020	4.0	14,200	197.1 (aft)	140	Cruise
	11,020	4.0	14,180	197.1 (aft)	170	
	11,020	4.0	14,160	197.1 (aft)	140	Power-approach
Stall characteristics	11,780	0.0	14,640	197.2 (aft)	$1.15V_S$[5]	Cruise
	11,940	-1.0	14,800	197.2 (aft)	$1.15V_S$	Power-approach
	11,960	-1.0	14,900	197.3 (aft)	$1.15V_S$	Glide
	11,940	-1.0	14,880	197.3 (aft)	$1.15V_S$	Landing
	11,660	0.0	14,480	197.1 (aft)	$1.15V_S$	Single-engine cruise
Single-engine characteristics	11,660	0.0	14,480	197.1 (aft)	[6]120	Cruise
	11,640	0.0	14,520	197.1 (aft)	140	
	11,700	0.0	14,440	197.1 (aft)	[6]120	Power-approach

[1] All tests performed at propeller speeds of 2000 rpm.
[2] Approximately 10-knot increments.
[3] Trimmed at 3-degree angle of descent.
[4] Center of gravity to simulate crewmember in latrine.
[5] V_S: Stall airspeed.
[6] Initial airspeed used to begin airspeed-for-minimum-control (V_{MC}) evaluation.

Table 2. Airplane Configurations.

Configuration	Landing Gear Position	Flap Setting (%)	Power Setting
Takeoff	Down	40	Takeoff
Cruise	Up	Zero	Power for level flight
Landing	Down	100	Flight-idle
Power approach	Down	40	Power for level flight
Glide	Up	Zero	Power off, propellers feathered
Single-engine cruise	Up	Zero	Power for level flight on left engine, power off and propeller feathered on right engine
Single-engine takeoff	Down	40	Maximum continuous power (MCP) on left engine, power off and propeller feathered on right engine

TEST METHODOLOGY

5. Established flight test techniques and data reduction procedures were used during this program (refs 8 through 13, app A). The test methods are described briefly in the Results and Discussion section of this report. Flight test data were recorded by hand from test instrumentation in the pilot and copilot panels, and from the photopanel. A detailed list of the test instrumentation is contained in appendix C. Test techniques (other than the standard techniques described in the appropriate references), weight and balance methodology, and data reduction techniques are contained in appendix D. A Handling Qualities Rating Scale (HQRS) (app D) was used to augment pilot comments relative to handling qualities. Airspeed calibrations were obtained from the contractor. Deficiencies and shortcomings are in accordance with the definitions presented in Army Regulation 70-10.

RESULTS AND DISCUSSION

GENERAL

6. Performance and handling qualities of the Model 200 aircraft were evaluated under a limited variety of operating conditions with emphasis on operation in the normal mission configuration (aft center of gravity) (cg) near the military maximum gross weight of 15,000 pounds. The test aircraft was compared to FAR Part 23 (ref 1, app A), BAC Airworthiness Qualification Specification 22301 (ref 14), military specification MIL-F-8785B(ASG) (ref 15), and military specification MIL A-8806A (ref 16), to assist in determining future military applications. No new deficiencies or shortcomings were identified. Deficiencies and shortcomings determined during APE I are contained in USAAEFA Final Report No. 74-21 (ref 17). The deficiency of loss of power to mission equipment and primary attitude and heading gyros when propeller speed was less than 2000 rpm remained uncorrected. It was not possible to duplicate the conditions under which the deficiency of smoke in the cockpit and cabin areas at altitudes above 15,000 feet pressure altitude (Hp) was noted during APE I. Four previously determined shortcomings were uncorrected: inadequate single-engine performance capability under heavy gross weight or high temperature conditions in the single-engine takeoff configuration; inability to maintain a trim airspeed within ±6 knots; fluctuating torque indicator needles; and leaking exhaust gases about the heated engine air inlet lips.

PERFORMANCE

General

7. The performance characteristics of the Model 200 aircraft were evaluated under various operating conditions with emphasis on operation in the normal mission configuration near the military maximum gross weight of 15,000 pounds at the forward cg limit of fuselage station (FS) 188.3. Single-engine performance capability under heavy gross weight or high temperature conditions was inadequate in the power-approach configuration and severely reduced the overall effectiveness of the aircraft. The inadequate single-engine performance of the aircraft under these operating conditions is a shortcoming (ref 18, app A).

Level Flight Performance

8. Level flight performance was evaluated at the conditions shown in table 1 to determine the maximum level flight airspeed, cruise airspeed, range, and endurance capabilities. The zero thrust glide test method was used by the contractor to obtain the base-line drag point for the aircraft. The drag polar of this aircraft with the external mission provision modification was then compared to the drag polar of the clean aircraft used in APE I (fig. 1, app E). This comparison showed

a forty-drag count (ΔC_D) (0.004) increase in parasitic drag over the clean aircraft. The constant pressure-altitude technique for determining the single-engine (propeller feathered) and dual-engine drag polar was then performed to confirm the degradation in performance due to the increase in drag of the antennas. The aircraft was stabilized and trimmed at incremental airspeeds from the maximum airspeed for level flight (V_H) to V_S.

9. Forty counts of drag were added to the drag polar equation of the clean aircraft and were checked against the actual level flight test data from the modified aircraft. With the forty counts of drag confirmed, the drag polar equations of the various configurations tested in APE I were then modified to reflect the drag increase. The aircraft performance was then calculated using the modified drag polars and UACL engine performance data, which included installation and accessories losses. The results of these tests are presented in figures 1 through 5, appendix E. Aircraft specific range, recommended endurance, cruise airspeed, and V_H in level flight for the cruise configuration are summarized in figures 6 through 9. The level flight drag polar equations for the Model 200 aircraft with the antenna array are presented in table 3.

10. At a maximum gross weight of 15,000 pounds at 15,000 feet, standard day, the maximum dual-engine airspeed in level flight using MCP was 251 knots true airspeed (KTAS). The recommended maximum range airspeed was 223 KTAS and the recommended endurance airspeed was 175 KTAS. The maximum single-engine airspeed in level flight (right engine shut down and propeller feathered), using MCP on the left engine at 5000 feet, standard day, was 170 KTAS. The recommended single-engine airspeeds for maximum range and endurance were 167 KTAS and 151 KTAS, respectively. This aircraft is designed to be utilized as a research and development test-bed aircraft, and no specific mission performance profile has been designated.

Climb Performance

Sawtooth Climb:

11. Dual and single-engine climb performance for the aircraft without the antenna array were evaluated at the conditions shown in table 1, using the sawtooth-climb method of test. All dual-engine climb tests were conducted with both engines operating at MCP. All single-engine climb tests were conducted with the left engine operating at flight-idle and the propeller feathered, while the right engine was operating at MCP. Zero sideslip was maintained for all tests. Forty counts of drag were added to the climb drag polar equation of the clean aircraft for reasons stated in paragraphs 8 and 9. The climb drag polar equations for the Model 200 aircraft with the antenna array are presented in table 4. Test results are presented in figures 10 through 17, appendix E.

Table 3. Level Flight Drag Polar Coefficients.[1]

Configuration	Number of Engines Operating	C_{D_o}	$\frac{\Delta C_D}{\Delta C_L^2}$	A	B	C
Cruise	Zero	0.0365	0.04684	Zero	Zero	Zero
	1	0.0365	0.04684	0.857	0.070	-0.004
	2	0.0365	0.04684	Zero	0.140	-0.0055

[1]General drag equation: $C_D = C_{D_o} + \frac{\Delta C_D}{\Delta C_L^2} C_L^2 + AT_C'^2 + BT_C' + C$

Where:

C_D = Coefficient of drag

C_{D_o} = Minimum coefficient of drag of the propeller feathered drag polar

$\frac{\Delta C_D}{\Delta C_L^2}$ = Slope of drag polar

C_L = Coefficient of lift

T_C' = Coefficient of thrust

A, B, C = Constants

Table 4. Climb Drag Polar Coefficients.[1]

Configuration	Number of Engines Operating	C_{D_o}	$\frac{\Delta C_D}{\Delta C_L^2}$	A	B	C
Cruise	Zero	0.0365	0.04684	Zero	Zero	Zero
	1	0.0365	0.04684	0.22917	0.0893	-0.00080
	2	0.0365	0.04684	Zero	0.0893	-0.0018
Takeoff	Zero	0.06918	0.0539	Zero	Zero	Zero
	1	0.06918	0.0539	0.7500	0.0558	-0.01398
	2	0.06918	0.0539	Zero	0.0558	-0.01398

[1]General drag equation: $C_D = C_{D_o} + \frac{\Delta C_D}{\Delta C_L^2} C_L^2 + AT_C'^2 + BT_C' + C$

12. At a representative gross weight of 15,000 pounds, the aircraft has a calculated dual-engine rate of climb of 1735 feet per minute (ft/min) at the recommended best-rate-of-climb airspeed of 133 knots calibrated airspeed (KCAS) in the cruise configuration at sea level on a standard day. At the same conditions in the takeoff configuration, at the recommended best-rate-of-climb airspeed of 120 KCAS, the calculated rate of climb was 1370 ft/min. At a representative gross weight of 15,000 pounds in the cruise configuration, the aircraft has a calculated single-engine rate of climb of 295 ft/min at the recommended best single-engine rate-of-climb airspeed of 127 KCAS at sea level on a standard day (15°C). At the same conditions in the takeoff configuration, at the recommended best single-engine rate-of-climb airspeed of 106 KCAS, the calculated rate of climb was -93 ft/min. Single-engine performance capability under heavy gross weight or high temperature conditions is marginal in the cruise configuration and inadequate in the takeoff configuration. The single-engine performance severely reduces the overall effectiveness of the aircraft under these operating conditions. The inadequate single-engine performance of the Model 200 aircraft in the normal mission takeoff configuration is a shortcoming. An Equipment Performance Report (EPR) concerning this shortcoming was submitted (ref 18, app A).

Continuous Climb:

13. Dual and single-engine continuous climb performance were also calculated as stated in paragraph 11. The dual-engine service ceiling in the cruise configuration was 25,900 feet density altitude (H_D) at a gross weight of 15,000 pounds (fig. 10, app E). The single-engine service ceiling in the cruise configuration was 8190 feet H_D at a gross weight of 15,000 pounds (fig. 14).

HANDLING QUALITIES

General

14. The handling qualities of the Model 200 aircraft were evaluated under a variety of operating conditions, with emphasis on operation in the normal mission configuration near the military maximum gross weight of 15,000 pounds at the aft cg limit of FS 197.4. The test results were compared to MIL-F-8785B(ASG). No new deficiencies or shortcomings were determined. The loss of electrical power to mission equipment and primary attitude and heading gyros when propeller speed was less than 2000 rpm was a deficiency which remained uncorrected from APE I. Inability to maintain a trim airspeed within ±6 knots was a previously determined shortcoming which remained uncorrected.

Control System Characteristics

15. Control system characteristics were evaluated on the ground (longitudinally only) and in flight at the conditions shown in table 1. Control forces were measured on the pilot control wheel and rudder pedals. Breakout forces (including friction) were determined by recording the forces required to obtain initial movement of the controls. There were no significant changes from the results determined and presented in the APE I report. Moderate departures from trim conditions (6 knots) did occur with the controls free, due to the friction band encountered at trim conditions throughout the airspeed envelope. Inability to maintain a trim airspeed within ±6 knots was objectionable and was previously reported as a shortcoming in APE I. An EPR concerning this shortcoming was submitted (ref 19, app A).

Static Longitudinal Stability

16. The static longitudinal stability characteristics were evaluated at the conditions shown in table 1. The aircraft was trimmed in steady-heading, ball-centered level flight at the desired trim airspeed, then stabilized at incremental airspeeds greater than and less than trim airspeeds. The test results are presented in figures 18 through 28, appendix E.

17. The elevator control force gradient, as indicated by the variation in elevator control force with airspeed, was positive for airspeeds above and below trim, indicating stable static longitudinal stability. The elevator control position gradient, as indicated by variation in elevator control position with airspeeds in the forward

cg configuration, was positive, although shallow, for airspeeds above and below trim. However, the gradient in the normal mission configuration (aft cg) was essentially neutral. The neutral elevator control position gradient was not objectionable, due to the strong influence of the positive elevator force gradient. The static longitudinal stability characteristics met the requirements of FAR Part 23. However, these characteristics did not meet the requirements of paragraph 3.2.1.1 of MIL-F-8785B(ASG), in that the elevator control position gradient in the normal mission configuration (aft cg) is essentially neutral. Within the scope of these tests, the static longitudinal stability of this aircraft is satisfactory.

Static Lateral-Directional Stability

18. Static lateral-directional stability characteristics were evaluated at the conditions shown in table 1. The aircraft was initially trimmed for zero sideslip flight at the desired airspeed. The aircraft was then stabilized at incremental sideslip angles left and right, holding airspeed constant until attaining full rudder pedal deflection or until reaching sideslip envelope limits. Test results are presented in figures 29 and 30, appendix E.

19. Static directional stability, as indicated by the variation of sideslip angle with rudder pedal force, was positive for sideslip angles between 10 degrees left and right from trim. A lightening of rudder pedal force occurred at sideslip angles outside this range at airspeeds below 170 KCAS. However, the rudder pedal force never reduced to zero, nor did it result in any unusual flight characteristics or objectionable increases in pilot effort to maintain aircraft control. The variation of sideslip angle with rudder pedal deflection was essentially linear for sideslip angles encountered up to full rudder pedal deflection (170 KCAS and below). The static directional stability failed to meet the requirements of paragraph 23.177(a)(3) of FAR Part 23, in that the rudder pedal force gradient reverses prior to obtaining the full rudder control limit below 170 KCAS. However, the static directional stability did meet the requirements of MIL-F-8785B(ASG) and was satisfactory.

20. Dihedral effect, as indicated by the variation of aileron control displacement with sideslip angle, was positive and essentially linear. Increasing aft displacement of the elevator control was required with increasing sideslip angles in both left and right directions. The corresponding increase in elevator control forces with increased sideslip angles was not objectionable. The side-force characteristics, as indicated by the variation of bank angle with sideslip angle, were positive and essentially linear. The strong side-force characteristics provided excellent cues of out-of-trim flight conditions to the pilot and enhanced coordinated flight while maneuvering (HQRS 2). Within the scope of these tests, the static lateral-directional stability characteristics are satisfactory.

Dynamic Longitudinal Stability

21. The dynamic longitudinal stability characteristics were evaluated at the conditions shown in table 1. The long-term dynamic characteristics were evaluated by slowing the aircraft with aft elevator control to an airspeed 30 knots below trim airspeed and then returning the control to the trim position (stick-fixed) or releasing the control and allowing it to seek the trim position (stick-free). Short-term dynamic characteristics, simulating gust response, were evaluated by rapidly displacing the elevator control 1 inch from trim for a duration of 0.5 second and then returning the control to the trim position. The long-term aircraft response was oscillatory and was lightly damped. The natural frequency was 0.1167 radians per second (rad/sec) and damped natural frequency was 0.1164 rad/sec. The damping ratio was 0.076 and the period was approximately 55 seconds. This weak damping combined with the large friction band contributes to the poor trimmability of the aircraft (para 15). There is no FAA requirement for long-term dynamics, and the long-term characteristics met the requirements of MIL-F-8785B(ASG). Within the scope of this test, the long-term dynamic characteristics are satisfactory.

22. The longitudinal short-term characteristics of the Model 200 aircraft were essentially deadbeat for all test conditions, including flight in turbulent conditions. The short-term characteristics met the requirements of FAR Part 23 and of MIL-F-8785B(ASG). For the conditions investigated, the short-term longitudinal dynamic characteristics are satisfactory.

Dynamic Lateral-Directional Stability

Dutch-Roll Characteristics:

23. The dynamic lateral-directional characteristics were evaluated at the conditions shown in table 1. These tests were conducted by exciting the aircraft from a coordinated level flight trim condition with a rudder pulse and doublet, aileron pulse and doublet, and by release from a steady-heading sideslip. The Dutch-roll oscillations were lightly damped and easily excited with the yaw damper OFF. With the yaw damper ON, all oscillations were damped within four overshoots. The aircraft's lateral-directional response and controllability characteristics were good in atmospheric disturbances with the yaw damper ON. However, considerable pilot compensation was required to overcome the sensitive gust response during turbulent flight conditions at all cg locations with the yaw damper OFF (HQRS 5). The Dutch-roll characteristics met the requirements of FAR Part 23 and MIL-F-8785B(ASG). Within the scope of these tests, the Dutch-roll characteristics of this aircraft with the yaw damper ON were satisfactory.

Spiral Stability Characteristics:

24. The spiral stability characteristics of the Model 200 aircraft were evaluated at the conditions shown in table 1. These tests were conducted by establishing a 20-degree bank (both left and right) from trim conditions (wings-level, zero yaw-rate flight with the controls free) and timing the motion to a 40-degree bank

angle or recording the bank angle achieved after 20 seconds elapsed time. Spiral stability, as indicated by change in bank angle with elapsed time, was essentially neutral for both left and right turns and confirmed the test results reported in APE I. This aircraft possesses the capability of holding lateral trim in hands-off flight for periods of time in excess of 20 seconds. The spiral stability characteristics met the requirements of MIL-F-8785B(ASG) and are satisfactory.

Stall Characteristics

25. Dual and single-engine stall characteristics of the Model 200 aircraft were evaluated at the conditions shown in table 1. These tests were conducted by establishing trim configuration at the desired airspeed and then making a slight pitch attitude increase and decelerating at a rate of approximately 1 knot per second until achieving a stall. Stall warning margins and recovery characteristics were evaluated qualitatively. Test results were essentially identical to the results determined in APE I. Within the scope of these tests. the stall characteristics are satisfactory.

Single-Engine Characteristics

26. The single-engine characteristics of the Model 200 aircraft were evaluated at the conditions shown in table 1. These tests were conducted by establishing trim configuration at the desired airspeed and simulating sudden engine failure by moving the left engine condition lever to the fuel cutoff position, and by establishing single-engine trim conditions at the desired airspeed and slowly decelerating the aircraft to the V_{MC} at which either lateral or directional control could not be maintained. Test results were identical to results determined in APE I. The single-engine stall speeds were also determined to be the V_{MC} for the test configurations of the Model 200 aircraft. The single-engine control characteristics met the requirements of FAR Part 23 and MIL-F-8785B(ASG). Within the scope of these tests, the single-engine characteristics are satisfactory.

Ground Handling Characteristics

27. The ground handling characteristics of the Model 200 aircraft were evaluated throughout the conduct of these tests. The pitch attitude instability evident during loading and ground operations of the aircraft in the normal mission configuration during APE I was not present during this evaluation. The nose gear oleo strut pressure was decreased and the main gear strut pressures were increased to improve the pitch attitude stability on the ground. The requirement to maintain a continuous 2000-rpm propeller speed during ground operations in order to provide power for mission equipment was reevaluated. This requirement was eliminated by incorporating a STANDBY operational mode in which the mission equipment could be placed following warmup. Other ground handling characteristics were identical to those determined in APE I. Within the scope of these tests, the normal ground handling characteristics are satisfactory.

Instrument Flight Capability

28. A limited reevaluation of the instrument flight capability of the Model 200 aircraft (SN 71-21060) was conducted to confirm the proper operation of the 750-VA inverter in the event of the loss of AC generator power to the attitude and heading gyros. During a simulated approach in 120-knot indicated airspeed (KIAS), 700-ft/min descent, the power levers were retarded to the flight-idle position and the propeller speed dropped to 1900 rpm, resulting in the loss of power from the 8.5-KVA AC generators. The 750-VA inverter immediately began supplying power to the primary attitude indicator gyro; however, heading gyro alignment was lost for 90 seconds. This was recorded as a maintenance discrepancy and prevented a further check for correction of the deficiency determined in APE I (loss of the primary attitude and heading gyros when the propeller speed is less than 2000 rpm). Correction of the loss of heading gyro alignment during power source transition should be demonstrated by the contractor prior to Army acceptance flights. An EPR concerning the original deficiency was submitted (ref 20, app A).

Aircraft Systems Failures

Yaw Damper:

29. Yaw damper system failure was simulated by switching the yaw damper system OFF. With the yaw damper system OFF, 1-inch rudder control pulse inputs resulted in 8 to 10 overshoots and long-term residual lateral-directional oscillations. The lightly damped, easily excited residual lateral-directional oscillations which resulted with the yaw damper OFF were objectionable but did not constitute a hazard to safe flight. The yaw damper system is required by the FAA for flights above 17,000 feet, due to the weak lateral-directional damping characteristics discovered in the commercial aircraft.

Rudder Boost:

30. Rudder boost failures were evaluated during the conduct of single-engine tests. Test results were identical to those determined during the conduct of similar tests in APE I.

Alternating Current Generator:

31. The failure of one AC generator was simulated in flight by switching off the left AC generator. Failure of the AC generator was indicated by a flashing MASTER CAUTION light on the glare shield and a steady light in the CAUTION panel. The opposite generator was capable of continuing to supply the necessary AC power requirements. The probability of a dual AC generator failure is remote. However, this condition was artificially induced by retarding the power levers to the flight-idle position at airspeeds below 125 KIAS. This resulted in the immediate loss of electrical power to the primary attitude and heading gyros (para 28), with the same results which were reported in APE I.

750 Volt-Ampere Inverter:

32. The failure of the 750-VA inverter was simulated by switching the inverter OFF. Failure was indicated by a flashing MASTER CAUTION light on the glare shield and a steady light in the CAUTION panel. Loss of this inverter had no effect on the aircraft instruments. However, in the event of the loss of power from both AC generators and the 750-VA inverter, the pilot would lose the use of the primary attitude and heading gyros.

ANTENNA VIBRATION EVALUATION

33. The antenna array installed on the Model 200 aircraft was monitored visually throughout the test program for any vibration tendencies. No antenna vibrations were encountered during large sideslip excursions (plus and minus an estimated 20 degrees) or during dives to achieve the never-exceed airspeed (V_{NE}) of 245 KIAS. Within the scope of these tests, there are no objectionable antenna vibrations.

HUMAN FACTORS

Cockpit Evaluation

34. Results of this evaluation were identical with the results determined in APE I with the exceptions noted below. A capability to adjust each lap belt and keep the shoulder strap attachment point to the lap belts centered in the pilot's lap was provided on the test aircraft. The propeller feather range on the control console was properly marked on the test aircraft. A three-position push-to-talk communications switch was provided for both the pilot and copilot. The copilot was able to monitor very-high-frequency (VHF) communications radios selected on his signal distribution panel while his transmit-select switch was in the intercom position on the aircraft. The communications cord to the pilot control wheel was positioned so that it did not interfere with the ice vane handles during electrical extension.

Noise

35. A limited noise level survey was conducted at various airspeeds using a propeller speed of 2000 rpm in the cruise configuration. Measurements were taken using the General Radio Type 1565-B sound level meter and utilizing the measuring procedures contained in the instrument instruction manual (ref 21, app A). The in-flight measurements are presented in table 5. It was determined that noise levels are acceptable at the pilot/copilot stations.

Table 5. Cockpit Noise Level Measurement.[1]

Trim Indicated Airspeed (kt)	MIL-A-8806A Limit (db)	Test Results (db)		
		Scale A	Scale B	Scale C
140	106	91	95	98
170	106	93	97	100
200	113	98	103	105

[1]Level flight, cruise configuration, 2000-rpm propeller speed.

Toxicity

36. Smoke in the cockpit and cabin areas at altitudes above 15,000 feet Hp was determined to be a deficiency in APE I (ref 22, app A). During the brief evaluation conducted during this evaluation, it was not possible to duplicate the conditions under which smoke was detected during APE I. It was therefore impossible to determine whether or not contractor modifications eliminated the deficiency. Correction of smoke in the cockpit and cabin areas at altitudes above 15,000 feet Hp should be demonstrated by the contractor prior to Army acceptance flights.

RELIABILITY AND MAINTAINABILITY

37. Factors affecting the reliability and maintainability of the Model 200 aircraft were evaluated throughout the conduct of the flight test program. Evaluated characteristics included ground support equipment, accessibility, interchangeability, servicing, fasteners, cables, connectors, and safety. Available contractor technical documents, historical data, and current maintenance procedures were reviewed. A qualitative evaluation was performed because the limited number of program flight hours minimized the opportunity to observe component repair and replacement. Formal removal or replacement tests were not performed. The aircraft was fully instrumented, which resulted in maintenance complications that should not exist on an operational aircraft.

38. The items listed below are shortcomings originally determined during APE I and which were uncorrected during this evaluation. These shortcomings will affect the reliability and maintainability of the Model 200 aircraft. Equipment Performance Reports concerning these shortcomings were submitted (refs 23 and 24, app A).

a. The torque needles continually fluctuated ±25 ft-lb during flight (full scale was 2250 ft-lb).

b. Engine exhaust gases used to continuously heat the engine air inlet lips leaked at random locations about the periphery of these lips, resulting in discolored and blistered paint areas on the inlet cowling.

CONCLUSIONS

GENERAL

39. The following conclusions were reached upon completion of APE II:

a. The Model 200 aircraft with the mission provision modification has reduced performance and essentially the same handling qualities characteristics as the clean aircraft at the same loading and cg conditions.

b. One deficiency and four shortcomings noted during APE I were still present during this evaluation.

DEFICIENCIES AND SHORTCOMINGS

40. The following deficiency remains uncorrected: Loss of the primary attitude and heading gyros when the propeller speed is less than 2000 rpm (para 28).

41. The following shortcomings were still identified:

a. The single-engine performance of the aircraft in the normal mission takeoff configuration was inadequate (para 12).

b. The inability to maintain a trim airspeed within ±6 knots (para 15).

c. The torque needles continually fluctuated ±25 ft-lb during flight (para 38a).

d. Engine exhaust gases used to continuously heat the engine air inlet lips leaked at random locations about the periphery of these lips, resulting in discolored and blistered paint areas on the inlet cowling (para 38b).

SPECIFICATION COMPLIANCE

42. The static longitudinal stability characteristics failed to meet the requirements of paragraph 3.2.1.1 of MIL-F-8785B(ASG), in that the elevator control position gradient in the normal mission configuration (aft cg) is essentially neutral (para 17).

43. The static directional stability failed to meet the requirements of paragraph 23.177(a)(3) of FAR Part 23, in that the rudder pedal force gradient reverses prior to obtaining the full rudder control limit below 170 KCAS (para 19).

RECOMMENDATIONS

44. The deficiency identified during this evaluation must be corrected (para 40).

45. The shortcomings should be corrected (para 41).

46. Correction of the loss of heading gyro alignment during power source transition should be demonstrated by the contractor prior to Army acceptance flights (para 28).

47. Correction of smoke in the cockpit and cabin areas at altitudes above 15,000 feet Hp should be demonstrated by the contractor prior to Army acceptance flights (para 36).

APPENDIX A. REFERENCES

1. Federal Aviation Administration, Federal Air Regulation FAR Part 23, *Airworthiness Standards; Normal, Utility and Aerobatic Category Airplanes*, 13 March 1971.

2. Letter, AVSCOM, AMSAV-EFT, 5 November 1973, subject: Test Directive for CEFLY LANCER Army Preliminary Evaluation.

3. Specification, Beech Aircraft Corporation, BS 22296A, *Beechcraft Model A100-1 Pressurized Turboprop Executive Transport 15,000 pounds Maximum Takeoff Gross Weight*, 23 February 1973, revised 30 April 1973.

4. Procurement Specification, PS 3102, *Aircraft Procurement Specification Light Fixed Wing Reconnaissance Aircraft*, 30 April 1973.

5. Operator's Manual, Beech Aircraft Corporation, *Super King Air Model 200*, 2 November 1973.

6. Message, AVSCOM, AMSAV-EFT, 141645Z, 14 February 1974, subject: Safety-of-Flight Release for CEFLY LANCER, A100-1 Aircr ft for Pilot Training at 12,500 pounds Gross Weight.

7. Message, AVSCOM, AMSAV-EFT, 231735Z, 23 February 1974, subject: Safety-of-Flight Release for CEFLY LANCER, A100-1 Aircraft for Gross Weights up to 15,000 Pounds.

8. Flight Test Manual, Naval Air Test Center, FTM No. 104, *Fixed Wing Performance*, 28 July 1972.

9. Flight Test Manual, Naval Air Test Center, FTM No. 103, *Fixed Wing Stability and Control*, 1 August 1969.

10. Handbook, USAF Aerospace Research Pilot School, FTC-TIH-70-1001, *Performance*, September 1970.

11. Handbook, USAF Aerospace Research Pilot School, FTC-TIH-68-1002, *Stability and Control*, September 1968.

12. Flight Test Manual, Advisory Group for Aeronautical Research and Development, *Volume I, Performance*, Pergamon Press, Los Angeles, California, 1959.

13. Flight Test Manual, Advisory Group for Aeronautical Research and Development, *Volume II, Stability and Control*, Pergamon Press, Los Angeles, California, 1959.

14. Specification, Beech Aircraft Corporation, BS 22301, *Airworthiness Qualification Specification, Beechcraft Model A100-1*, 19 September 1973.

15. Military Specification, MIL-F-8785B(ASG), *Flying Qualities of Piloted Airplanes*, 7 August 1969 with Interim Amendment I, 31 March 1971.

16. Military Specification, MIL-A-8806A, *Acoustical Noise Level in Aircraft; General Specification*, 11 July 1966.

17. Final Report, USAAEFA, Project No. 74-21, *Army Preliminary Evaluation I, Model 200 CEFLY LANCER*, June 1974.

18. Equipment Performance Report, USAASTA, SAVTE-TB, EPR 74-21-21, "Army Preliminary Evaluation I, Model A100-1 CEFLY LANCER," 27 April 1974.

19. EPR, USAASTA, 74-21-18, 12 March 1974.

20. EPR, USAASTA, 74-21-3, 21 March 1974.

21. Manual, General Radio Company, "Instruction Manual, Sound Level Meter Type 1565-B," September 1971.

22. EPR, USAASTA, 74-21-4, 20 March 1974.

23. EPR, USAASTA, 74-21-12, 20 March 1974.

24. EPR, USAASTA, 74-21-22, 21 May 1974.

APPENDIX B. DESCRIPTION

1. The Model 200 aircraft has the general structure and space arrangements of the BAC Super KingAir Model 200 aircraft. The test aircraft is shown in photo 1. General specifications are listed below.

Dimensions

Wing span	54 ft, 6 in.
Horizontal stabilizer span	18 ft, 5 in.
Length	43 ft, 9 in.
Height to top of vertical stabilizer	15 ft
Propeller diameter	8 ft, 2.5 in.
Propeller/fuselage clearance	29.6 in.
Propeller/ground clearance	14.5 in.
Distance between main gear	17 ft, 2 in.
Distance between main and nose gear	15 ft

Cabin Dimensions

Total pressurized length	264 in.
Cabin length, partition to partition	128 in.
Cabin height	57 in.
Cabin width	54 in.
Entrance door	21.5 in. x 26.7 in.

Wing Area and Loading

Wing area	303.0 ft^2
Wing loading	49.5 lb/ft^2
Power loading	8.8 lb/hp

Weights

Maximum takeoff weight	15,000 lb
Maximum ramp weight	15,090 lb
Maximum landing weight	13,500 lb
Maximum zero fuel weight	12,500 lb

Ground Turning Clearance

Radius for inside gear	4 ft
Radius for nose wheel	19 ft, 6 in.
Radius for outside gear	21 ft, 1 in.
Radius for wing tip	39 ft, 10 in.

2. A more detailed description of the test aircraft is presented in references 5 and 17, appendix A.

Photo 1. Model 200 CEFLY LANCER.

APPENDIX C. INSTRUMENTATION

The instrumentation in the BAC Model 200 aircraft, SN 71-21059, was installed, calibrated, and maintained by BAC personnel. In addition to the instrumentation listed, the aircraft was equipped with a pitot-static boom. Photos C-1 and C-2 show the instrument panel and photopanel. A list of test instrumentation showing the manufacturer, calibration range, and parameter accuracies follows.

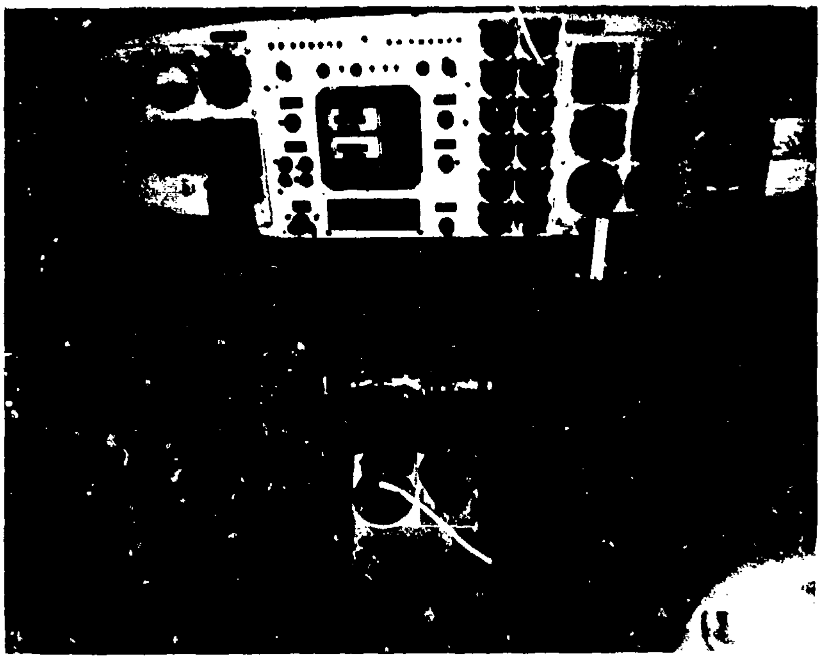


Photo C-1. Pilot Instrument Panel.

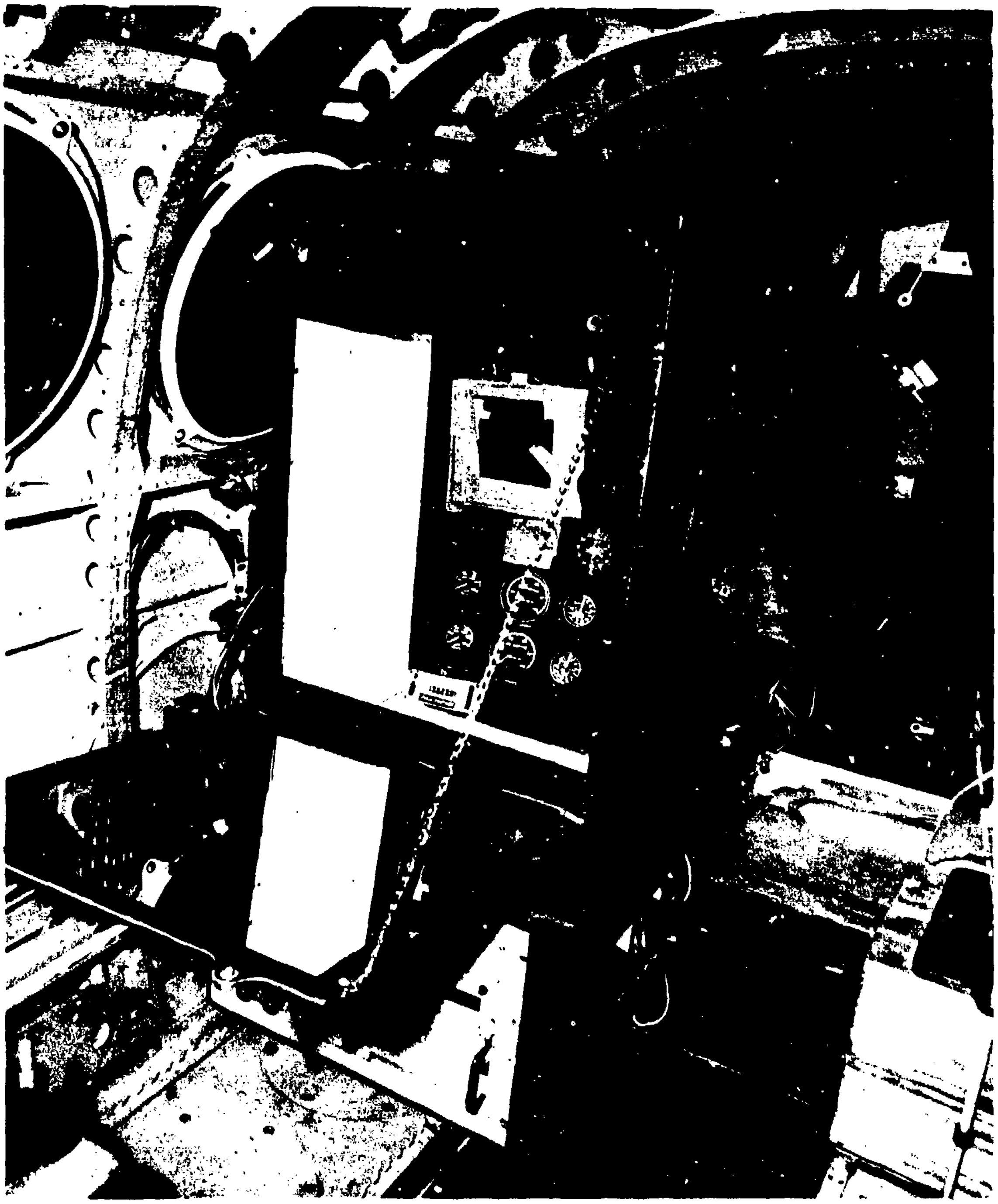


Photo C-2. Photopanel.

Photopanel

Parameter	Manufacturer	S/N	Calibration Range	Estimated Accuracy
Airspeed (left boom)	Kollsman	14569	50 to 290 kt	±1 kt
Airspeed (ship's standard)	Kollsman	105	50 to 290 kt	±1 kt
Altimeter (left boom)	Instruments Inc	2508	Zero to 3500 ft	±20 ft
Altimeter (ship's standard)	Kollsman	2848	Zero to 3500 ft	±20 ft
Air temperature	Rosemount	449	-80 to 160°F	±1/2°F
Fuel totalizer	Foxboro	61126	Zero to 9999 lb	±10 lb
Fuel flow (left engine)	Foxboro	52203	100 to 550 lb/hr	±1 lb/hr
Fuel flow (right engine)	Foxboro	52202	100 to 550 lb/hr	±1 lb/hr
Torque (left engine)	AID[1]	2788	10 to 75 psi	±1/4 psi
Torque (right engine)	AID	2791	10 to 75 psi	±1/4 psi
Gas producer (N_1) tachometer (left engine)	AID	1470	10 to 102%	±1/4%
Gas producer (N_1) tachometer (right engine)	AID	4493	10 to 102%	±1/4%

[1] Aircraft Instrument and Development Inc.

Power turbine (N_2) tachometer (left engine)	AID	5161	500 to 2200 rpm	±1 rpm
Power turbine (N_2) tachometer (right engine)	AID	5369	500 to 2200 rpm	±1 rpm
Inlet turbine temperature (left engine)	Howell	103	400 to 950°C	±1°C
Inlet turbine temperature (right engine)	Howell	105	400 to 950°C	±1°C
		Instrument Panel		
Airspeed (ship's system)	Kollsman	10152	50 to 290 kt	±1 kt
Airspeed (right boom)	Kollsman	5167	50 to 290 kt	±1 kt
Altimeter (ship's system)	Smiths Industries Inc.	18678	Zero to 35,000 ft	±20 ft
Fuel flow (left engine)	Bendix	-	100 to 550 lb/hr	±1 lb/hr
Fuel flow (right engine)	Bendix	-	100 to 550 lb/hr	±1 lb/hr
Torque (left engine)	U.S. Gauge	-	305 to 2292 ft/lb	±5 ft/lb
Torque (right engine)	U.S. Gauge	-	305 to 2292 ft/lb	±5 ft/lb
Gas producer (N_1) tachometer (left engine)	AID	9594	10 to 102%	1/4%

Gas producer (N_1) tachometer (right engine)	AID	9627	10 to 102%	1/4%
Power turbine (N_2) tachometer (left engine)	AID	A122	500 to 2200 rpm	±1 rpm
Power turbine (N_2) tachometer (right engine)	AID	9661	500 to 2200 rpm	±1 rpm
Inlet turbine temperature (left engine)	Howell	397H	400 to 950°C	±1°C
Inlet turbine temperature (right engine)	Howell	383H	400 to 950°C	±1°C
Rudder pedal force	Howell	565A491496	-200 to 200 lb	±1/2 lb
Aileron control force	Howell	565A491496	-100 to 100 lb	±1/2 lb
Elevator control force	Howell	566A491496	-100 to 100 lb	±1/2 lb
Rudder control surface position	Howell	566A491497	26°, right, to 25.75°, left	±0.2°
Aileron control surface position	Howell	566A491497	24.03° left aileron, up, to 17.03° left aileron, down	±0.2°
Elevator control surface position	Howell	566A491497	19.68°, up, to 15.06°, down	±0.2°

Angle of attack	Bendix	–	35°, nose up, to 30°, nose down	±1/4°
Angle of sideslip	Bendix	–	35°, left, to 35°, right	±1/4°
Normal center-of-gravity acceleration	Kollsman	759	-4 to +6g	±0.1G

APPENDIX D. TEST TECHNIQUES AND DATA ANALYSIS METHODS

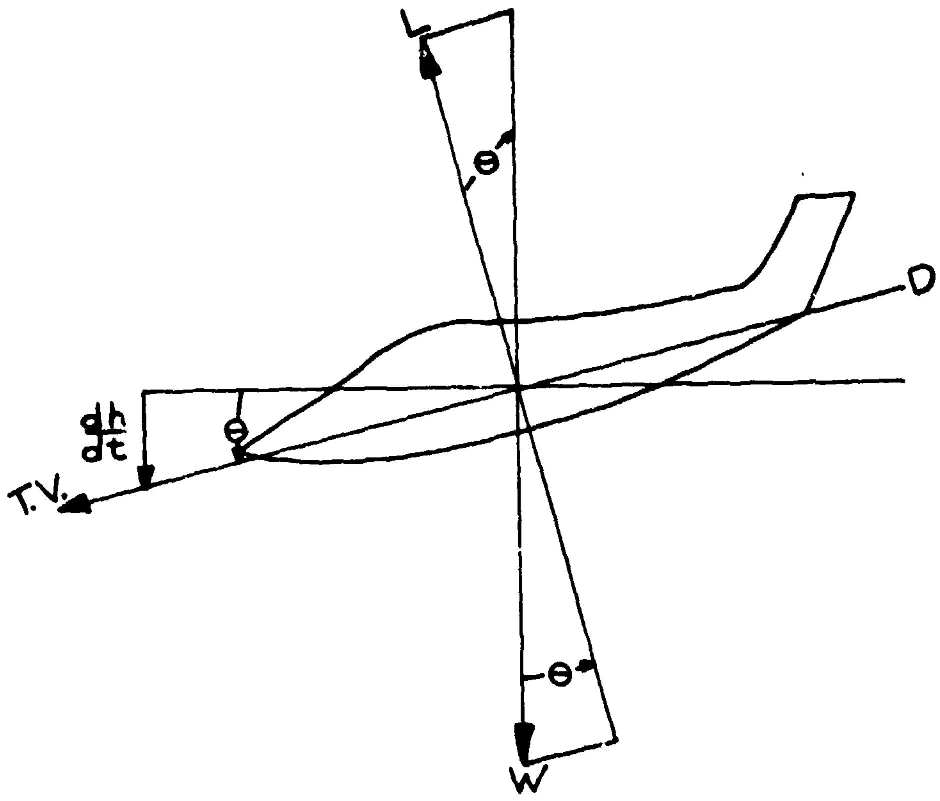
GENERAL

1. This appendix contains some of the data reduction and analysis methods used to evaluate the BAC Model 200 CEFLY LANCER aircraft. Although specific tests were not performed by the test team for propeller feathered glide and sawtooth-climb performance during APE II, the propeller feathered glide technique was used. Beech Aircraft Corporation conducted the above test, including level flight performance, to identify the additional drag caused by the antenna installations. The test team confirmed the drag count by conducting single and dual-engine level flight performance tests. The drag count increment was then added to the base-line drag polar of the clean aircraft (ref 17, app A), and the aircraft performance data at conditions not specifically tested were predicted.

PERFORMANCE

2. Past programs generally developed a drag polar relationship for specific flight conditions. However, the test points showed large deviations from the faired line at extreme altitudes (low versus high). The deviations are attributed to power effects which caused an apparent change in equivalent flat plate area (f) and Oswald's span efficiency factor (e) due to differences in engine thrust at varying altitudes. To eliminate these effects, the propeller feathered glide method was used to evaluate the Model 200 CEFLY LANCER aircraft.

3. The propeller feathered glide technique was used to define the base-line drag polar. The aircraft was stabilized in a descent at a constant airspeed, with both engines inoperative and propellers feathered. Airspeed, pressure altitude, outside air temperature, gross weight, and elapsed time were recorded. The entire airspeed range ($1.1V_S$ to V_{NE}) was investigated for a target altitude band. The following technique was used to develop the base-line drag coefficient equation.



$$L = W \cos \gamma \tag{1}$$

$$D = T + W \sin \gamma \tag{2}$$

$$DV = TV + WV \sin \gamma \tag{3}$$

$$-V \sin \gamma = dh/dt = \frac{TV - DV}{W} \tag{4}$$

Where:

L = Lift force (lb)

W = Aircraft gross weight (lb)

γ = Descent angle (deg)

T = Net thrust (lb) = Zero in a descent

D = Drag force = Level flight drag (lb)
= Net thrust required

V = Aircraft velocity on descent path (ft/min)

dh/dt = Tapeline rate of descent (ft/min)

Considering the drag and lift force equations and applying power-off glide conditions, the following relationship can be developed:

$$C_D = \frac{D}{qs} \tag{5}$$

$$C_D = \frac{W \sin \gamma}{qs} \tag{6}$$

$$C_L = \frac{L}{qs} \tag{7}$$

$$C_L = \frac{W \cos \gamma}{qs} \tag{8}$$

Where:

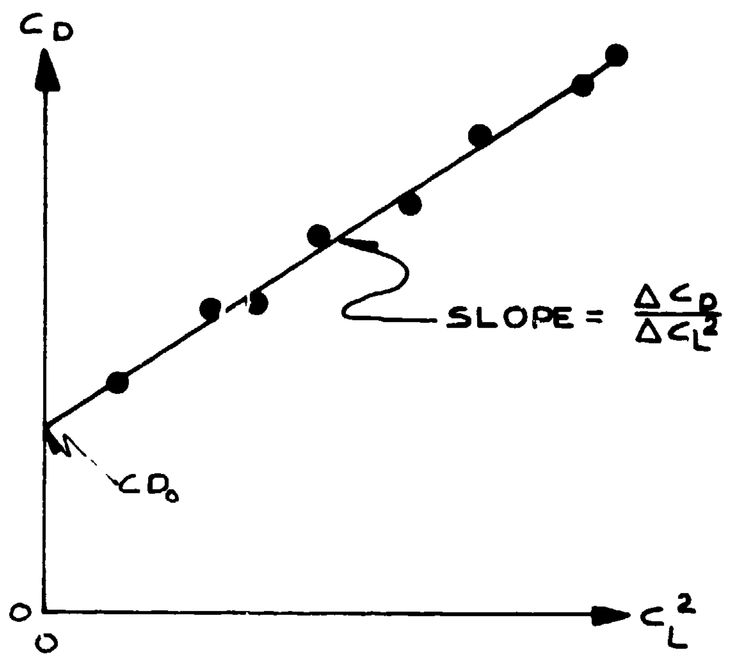
C_D = Coefficient of drag

$q = 1/2 \, \rho \, V^2$ (lb/ft^2) dynamic pressure

s = Wing area (ft^2)

C_L = Coefficient of lift

The base-line coefficient of drag ($C_{D_{BL}}$) was then developed by plotting C_D versus C_L^2 and fitting a first-order equation to the test points.



$$C_{D_{BL}} = C_{D_o} + \frac{\Delta C_D}{\Delta C_L^2} C_L^2 \tag{9}$$

4. During powered flight, the drag of the aircraft increased with thrust. To reflect the change, the basic drag equation was modified.

$$\Delta C_{D_{PF-BL}} = C_{D_{PF}} - C_{D_{BL}} \tag{10}$$

Where:

$\Delta C_{D_{PF-BL}}$ = Increased drag due to thrust effect

$C_{D_{PF}}$ = Total coefficient of drag for powered flight

$C_{D_{BL}}$ = Base-line coefficient of drag

Coefficient of thrust (T_C'), thrust (T), thrust horsepower (THP), and shaft horsepower (SHP) were calculated as follows:

$$T_C' = \frac{2T}{\rho S V_T^2} \tag{11}$$

$$T = \frac{550 \times THP}{V_T} \tag{12}$$

$$THP = \eta_\rho \times SHP + \frac{F_n \times V_T}{550} \tag{13}$$

$$SHP = Q \times N_P \times \frac{2\pi}{33{,}000} \tag{14}$$

Where:

T_C' = Coefficient of thrust

T = Thrust (lb)

ρ = Air density (slug/ft^3)

S = Wing area (ft^2)

V_T = True airspeed (ft/sec)

THP = Thrust horsepower

ηp = Propeller efficiency

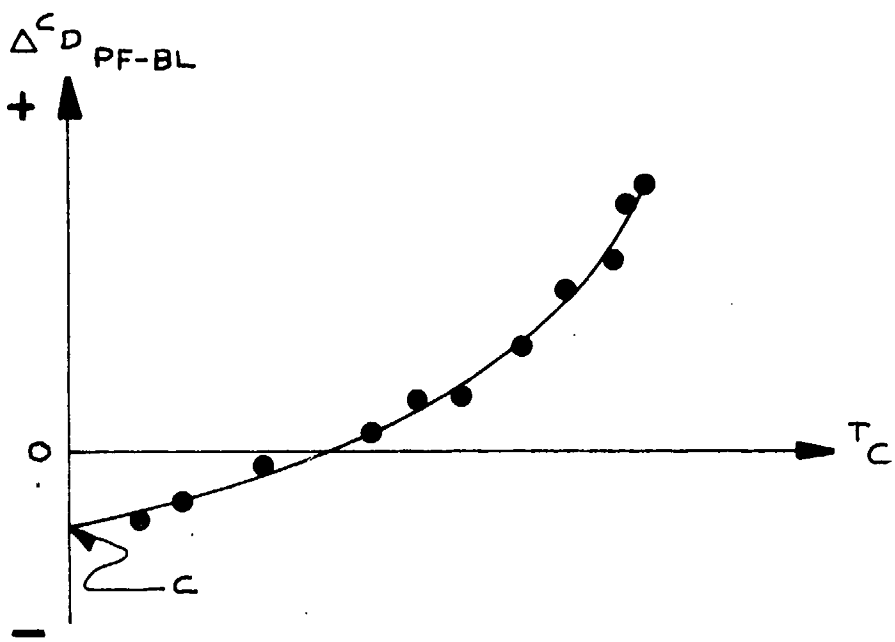
SHP = Shaft horsepower

F_n = Jet thrust (lb)

Q = Engine torque (ft-lb)

Np = Propeller speed (rpm)

The values of $\Delta C_{D_{PF\,-\,BL}}$ and T_C' were then plotted to develop a generalized equation that represented the change in drag due to thrust. A second-order fitting was used.



$$\Delta C_{D_{PF\,-\,BL}} = A\ T_C'^2 + B\ T_C' + C \tag{15}$$

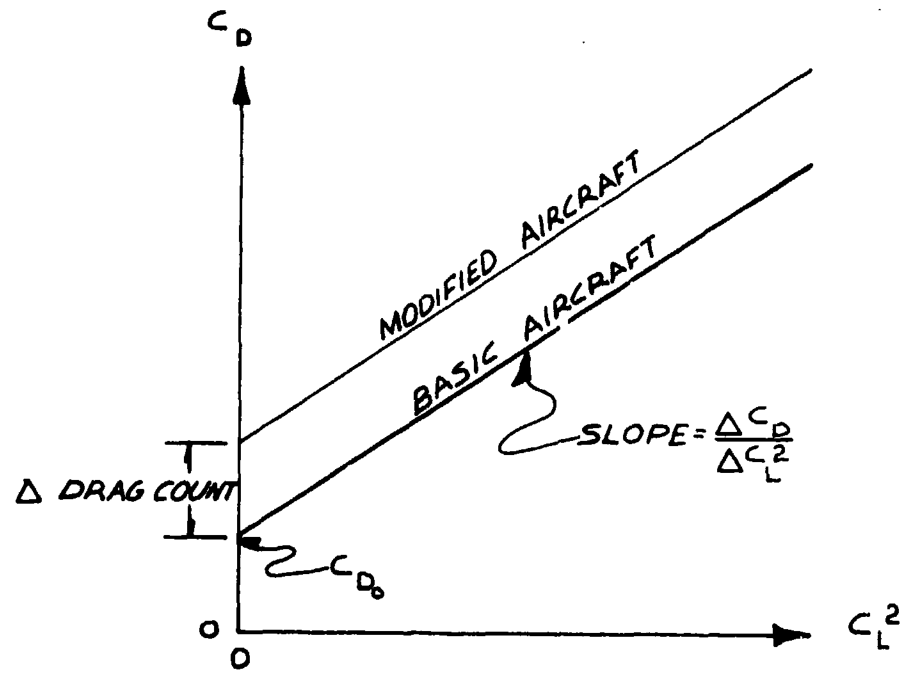
From equation 10,

$$C_{D_{PF}} = C_{D_{BL}} + \Delta C_{D_{PF\,-\,BL}}$$

Or

$$C_{D_{PF}} = C_{D_{BL}} + A\ T_C'^2 + B\ T_C' + C \tag{16}$$

Equation 16 represents the generalized equation for all level flight and climb performance in either single or dual-engine operation. When an external configuration change is made to the aircraft that may affect its performance, the propeller feathered glide test is the only flight required to find the difference in drag count.



The slopes, $\Delta C_D/\Delta C_L{}^2$, for the basic and modified aircraft are identical. The C_{D_0} of the modified aircraft contains the difference in drag which will be reflected in equations 9 and 16.

5. Level flight performance tests (single and dual-engine) were conducted using the constant pressure altitude method. The aircraft was stabilized and trimmed at incremental airspeeds from V_S to V_H while maintaining a constant pressure altitude throughout the entire flight. The coefficient of drag (C_D), lift (C_L), and thrust ($T_C{}'$) were obtained from the recorded test data to determine the coefficients for the generalized equations.

6. The shaft horsepower available, fuel-flow rate, and net thrust of a PT6A-41 specification engine, including all installation losses, were furnished by the airframe manufacturer. The UACL-furnished computer program was used to calculate the performance for an installed specification engine. The computer program is based on the minimum performing engine that has the maximum allowable time before overhaul. For this reason, the calculated aircraft performance data, which are based on a specification engine, were always less than the observed test data. The test engines, SN X70012 and SN X70009, used for this evaluation were uncalibrated experimental engines and the torque conversion factor for each engine was not available. The specification engine torque constant of 30.57 ft-lb per psi was used. The propeller efficiency table was furnished by BAC and is presented in table 1.

3 BLADE HARTZELL PROPELLER

ACTIVITY FACTOR=120

C_p	J																	
	.3	.4	.5	.6	.7	.8	.9	1.0	1.1	1.2	1.3	1.4	1.6	1.8	2.2	2.4	2.6	2.8
.04	.574	.662	.683	.705	.709	.708	.707	.706	.705	.675	.660	.674	.652	.586				
.05	.553	.662	.715	.759	.777	.781	.784	.784	.778	.756	.723	.700	.681	.596				
.06	.540	.650	.725	.766	.794	.808	.814	.816	.812	.810	.786	.758	.701	.627				
.07	.525	.635	.710	.763	.774	.821	.824	.834	.831	.831	.824	.805	.753	.690				
.08	.503	.615	.698	.753	.792	.823	.832	.844	.847	.848	.843	.835	.807	.749				
.09	.481	.538	.681	.740	.784	.820	[illegible]	.845	.852	.853	.852	.848	.826	.786				
.10	.461	.574	.666	.727	.777	.810	[illegible]	.842	.853	.857	.859	.853	.841	.817	.719			
.12	.417	.534	.629	.698	.750	.787	.814	.833	.847	.857	.862	.864	.858	.843	.786	.726		
.14	.385	.436	.580	.665	.723	.770	.797	.821	.837	.850	.859	.864	.864	.855	.819	.779	.729	
.16	.353	.461	.554	.636	.699	.746	.780	.806	.827	.842	.852	.859	.857	.853	.838	.811	.771	.726
.18	.324	.429	.519	.602	.670	.722	.762	.790	.812	.830	.844	.853	.864	.866	.849	.831	.800	.760
.20	.299	.373	.487	.570	.640	.668	.740	.774	.799	.820	.835	.845	.859	.865	.865	.843	.820	.783
.30	.204	.275	.343	.411	.489	.552	.612	.667	.710	.743	.773	.792	.824	.857	.859	.853	.850	.810
.40											.687	.728	.777	.800	[illegible]	.845	.840	.845

$$C_p = \frac{SHP}{2\sigma\left(\frac{N_p}{100}\right)^3\left(\frac{D}{10}\right)^5}$$

$$J = \frac{101.28\, V_T}{N_p D}$$

SHP ~ SHAFT HORSEPOWER

σ ~ ρ/ρ_0

N_p ~ PROPELLER SPEED (RPM)

D ~ PROPELLER DIAMETER (8.208 ft)

V_T ~ TRUE AIRSPEED (KNOTS)

J ~ ADVANCE RATIO

Table 1. Propeller Efficiency Table.

7. Ambient test temperatures (T_a) were obtained by correcting the indicated test temperature (T_i) for instrument error (ΔT_{ic}) and for compressibility (ΔT_c).

$$T_a = T_i + \Delta T_{ic} + \Delta T_c$$

8. Pressure altitudes were obtained by correcting indicated pressure altitudes (H_{Pi}) for instrument error (ΔH_{Pic}).

$$H_P = H_{Pi} + \Delta H_{Pic}$$

9. The density ratio (σ) was determined from the following relationship:

$$\sigma = T_o/T_a \quad P_a/P_o$$

Where:

T_o = Standard-day, sea-level temperature

P_o = Standard-day, sea-level pressure

10. The density altitudes were determined from the test density ratio (σ test) and the US Standard Atmosphere, 1962 tables.

AIRSPEED CALIBRATION

11. The boom and ship's standard pitot-static systems were calibrated by the contractor, using a low-altitude ground speed course to determine the airspeed position error (fig. 3, app E). Calibrated airspeeds (V_{cal}) were obtained by correcting indicated airspeed (V_i) for instrument error (ΔV_{ic}) and position error (ΔV_{pc}).

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

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

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

Where:

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

13. True airspeeds (V_T) were calculated from the equivalent airspeed and density ratio.

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

Where:

σ = Density ratio ($\frac{\rho}{\rho o}$ where ρ is the actual ambient density)

14. Mach numbers (M) were determined from the test altitude absolute temperature (Ta) in degrees Kelvin (K) and the true airspeeds.

$$M = \frac{V_T}{a_t}$$

Where:

a_t = Speed of sound in knots ($38.967\sqrt{Ta}$)

WEIGHT AND BALANCE

15. The aircraft weight and longitudinal cg were determined prior to each weight and/or cg configuration change. Weighing was accomp'ished using electronic scales located under the aircraft jack points with the crew on board at their designated stations.

DYNAMIC STABILITY

16. Dynamic stability characteristics were tested by using the techniques described in references 9, 11, and 13, appendix A. Analyses of the test data were performed to determine the resulting damping ratios (ζ) and damped natural frequencies (ω_d). The damped natural frequencies and the damping ratios were derived by the logarithmic decrement method.

17. The undamped natural frequencies (ω_n) of the motion in radians per second were calculated from the following equation.

$$\omega_n = \frac{\omega_d}{\sqrt{1 - \zeta^2}}$$

APPENDIX E. TEST DATA

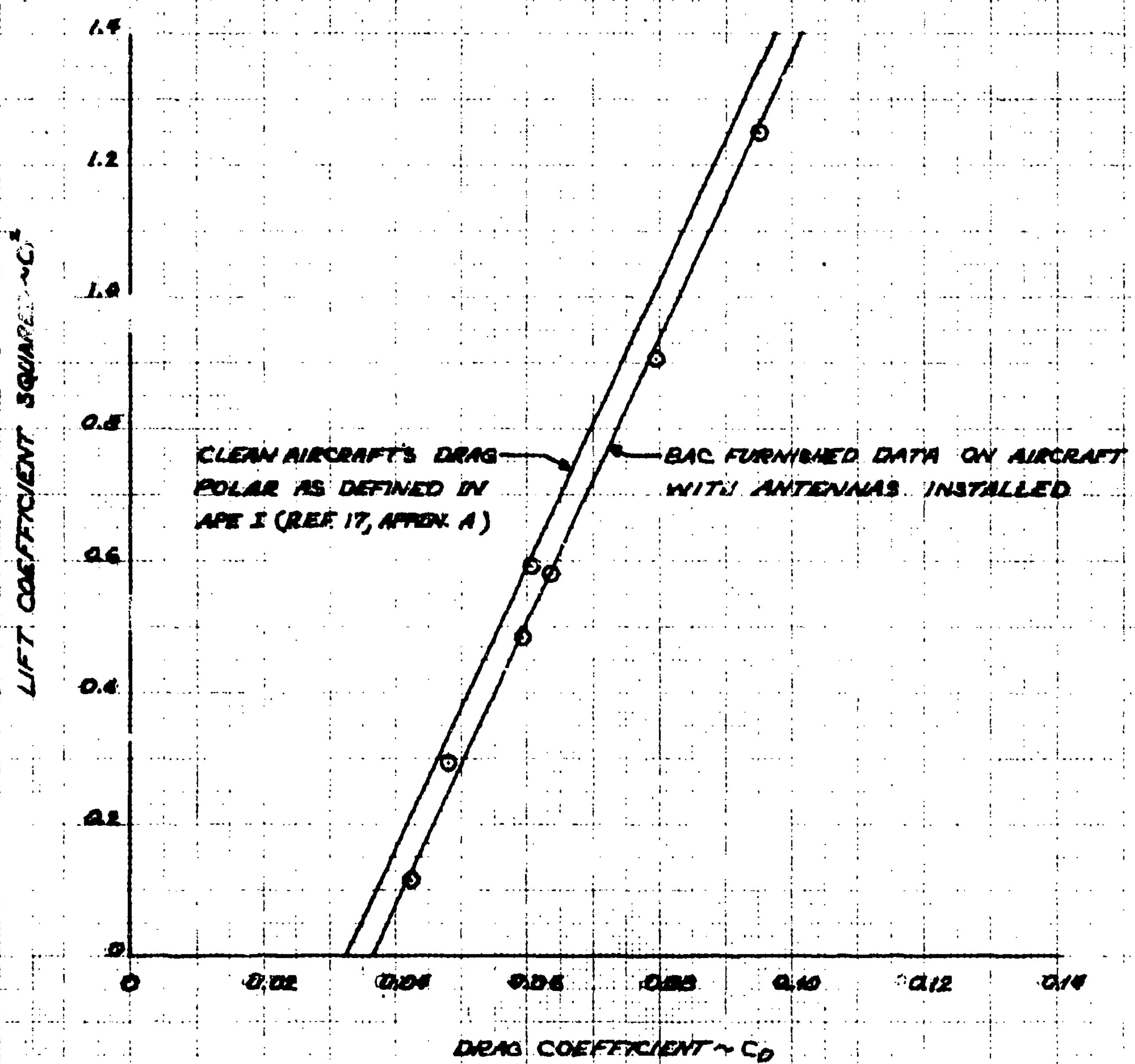
INDEX

Figure	Figure Number
Performance:	
Level Flight	1 through 9
Climb	10 through 17
Handling Qualities:	
Static Longitudinal Stability	18 through 28
Static Lateral-Directional Stability	29 and 30
Airspeed Calibration	31

FIGURE 1

PROPELLER FEATHERED GLIDE DRAG POLAR

NOTE: ENGINES INOPERATIVE AND PROPELLERS FEATHERED



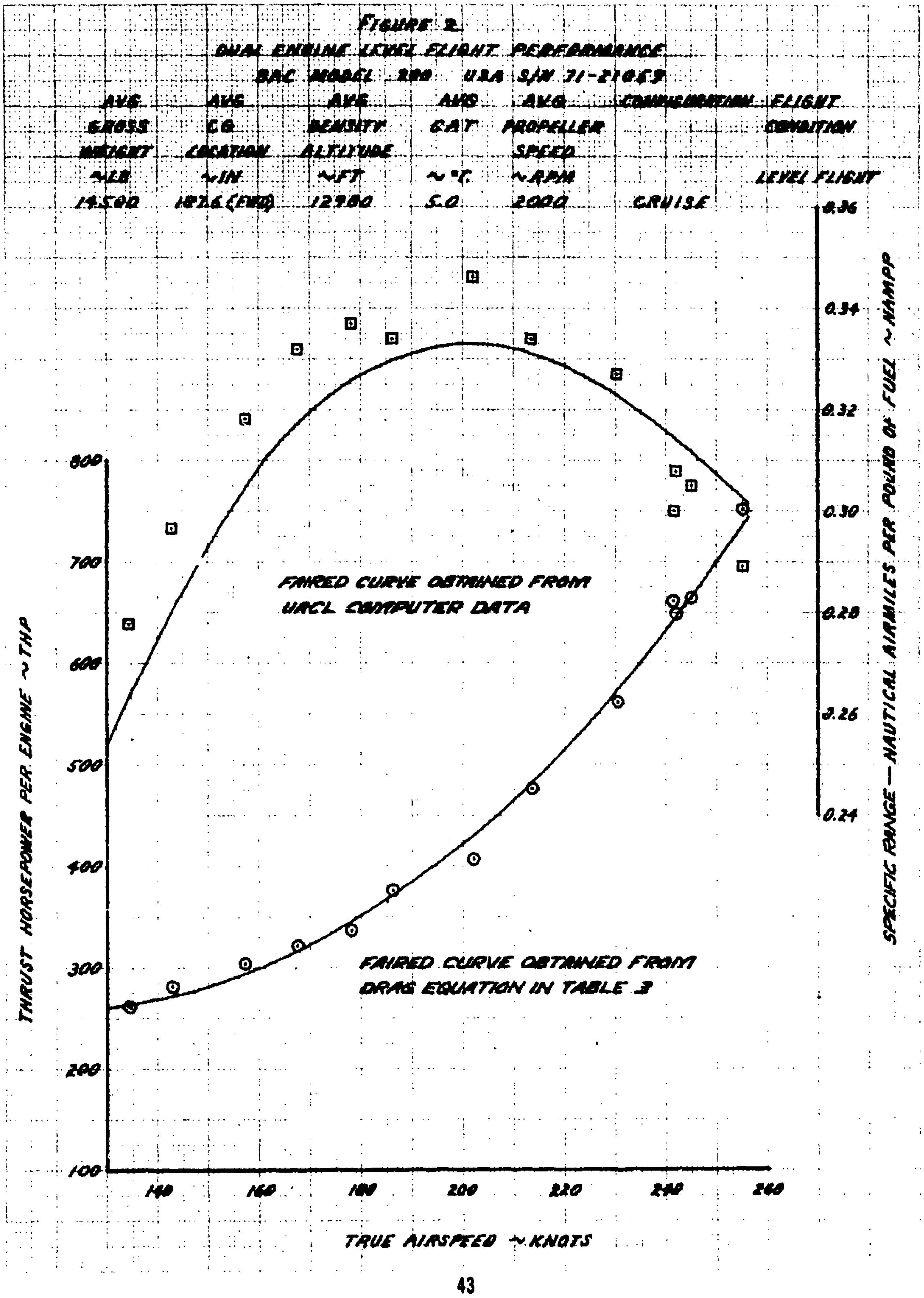


FIGURE 2
DUAL ENGINE LEVEL FLIGHT PERFORMANCE
USA S/N 71-21053
AVG GROSS WEIGHT ~LB 14500
AVG C.G. LOCATION ~IN 187.6 (FWD)
AVG DENSITY ALTITUDE ~FT 12980
AVG OAT ~°C 5.0
AVG PROPELLER SPEED ~RPM 2000
CONFIGURATION CRUISE
FLIGHT CONDITION LEVEL FLIGHT
FAIRED CURVE OBTAINED FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED FROM DRAG EQUATION IN TABLE 3
THRUST HORSEPOWER PER ENGINE ~THP
SPECIFIC RANGE—NAUTICAL AIRMILES PER POUND OF FUEL ~NAMPP
TRUE AIRSPEED ~KNOTS
100
200
300
400
500
600
700
800
140
160
180
200
220
240
260
0.24
0.26
0.28
0.30
0.32
0.34
0.36

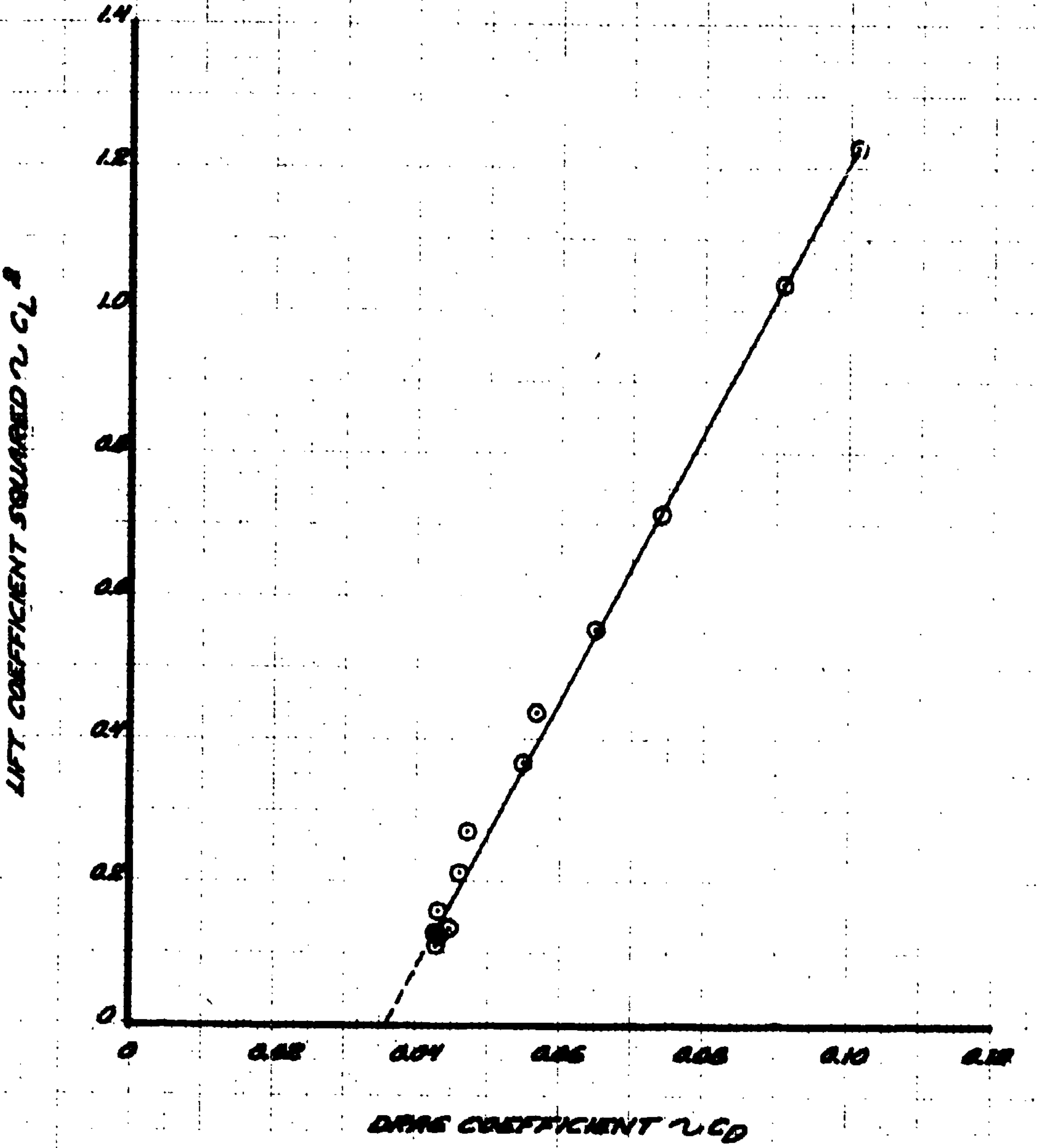
FIGURE 3

DUAL ENGINE LEVEL FLIGHT DRAG POLAR

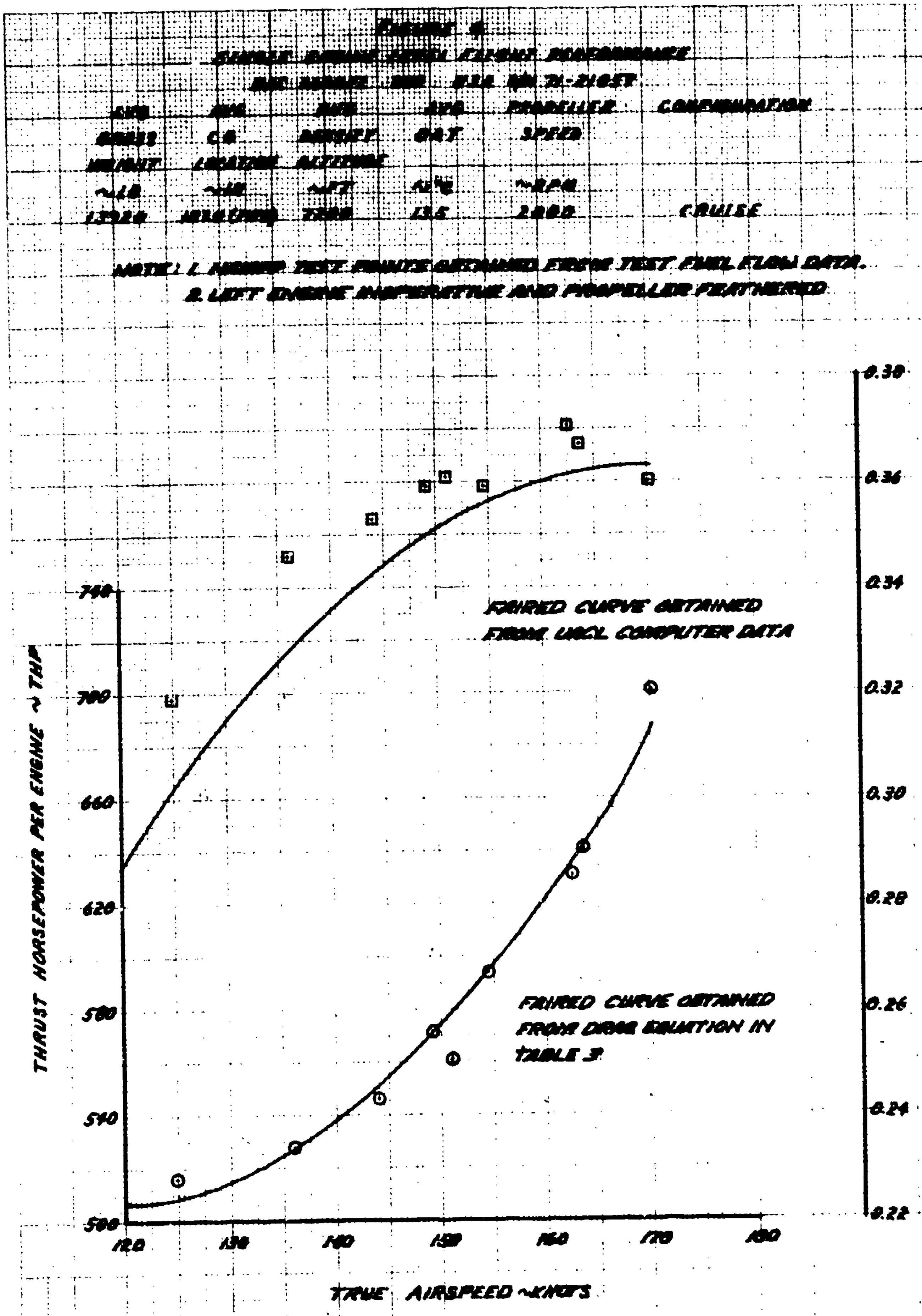
[illegible] MODEL [illegible] USAF S/N 77-[illegible]

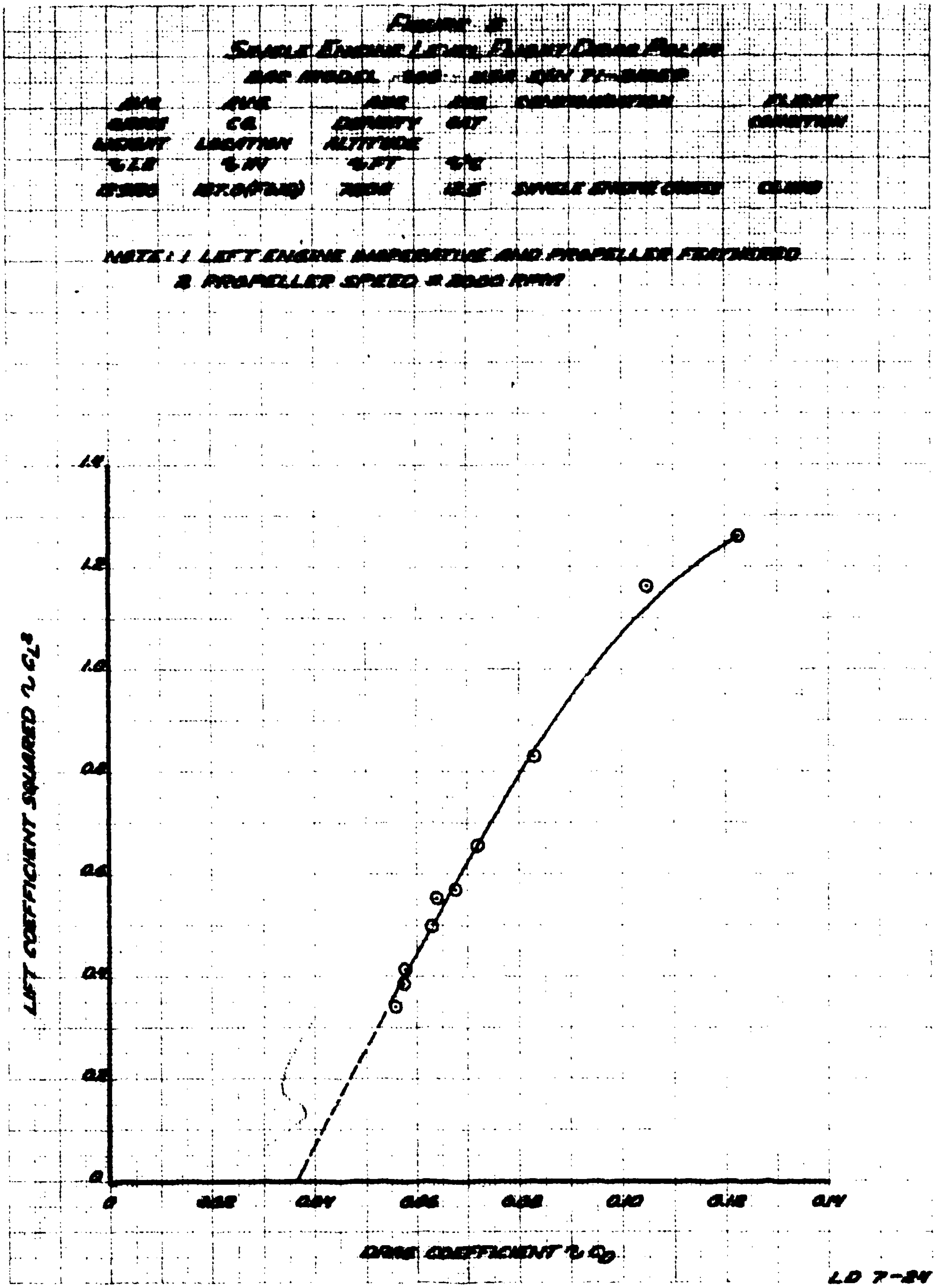
AVG GROSS WEIGHT LBS	AVG C.G. LOCATION FS/IN	AVG DENSITY ALTITUDE FT	AVG OAT °C	CONFIGURATION	FLIGHT CONDITION
[illegible]	[illegible](FWD)	[illegible]	5.0	CRUISE	LEVEL FLIGHT

NOTE: 1 PROPELLER SPEED = 2000 RPM



LD 6-74


AVG
PROPELLER
CONFIGURATION
C.G.
OAT
SPEED
~RPM
13.5
2000
CRUISE
2. LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED
FAIRED CURVE OBTAINED
FROM UACL COMPUTER DATA
FAIRED CURVE OBTAINED
FROM DRAG EQUATION IN
TABLE 3
THRUST HORSEPOWER PER ENGINE ~ THP
SPECIFIC RANGE ~ NAUTICAL AIRMILES PER POUND OF FUEL ~ NAMPP
TRUE AIRSPEED ~ KNOTS
500
540
580
620
660
700
740
120
130
140
150
160
170
180
0.22
0.24
0.26
0.28
0.30
0.32
0.34
0.36
0.38


SINGLE ENGINE LEVEL FLIGHT DRAG POLAR
AVG
AVG
AVG
AVG
CONFIGURATION
FLIGHT
CONDITION
C.G.
DENSITY
OAT
WEIGHT
LOCATION
ALTITUDE
SINGLE ENGINE CRUISE
NOTE: 1 LEFT ENGINE INOPERATIVE AND PROPELLER FEATHERED
2 PROPELLER SPEED
1.4
1.2
1.0
0.8
0.6
0.4
0.2
0
0
0.02
0.04
0.06
0.08
0.10
0.12
0.14
LIFT COEFFICIENT SQUARED ~ C_L^2
DRAG COEFFICIENT ~ C_D

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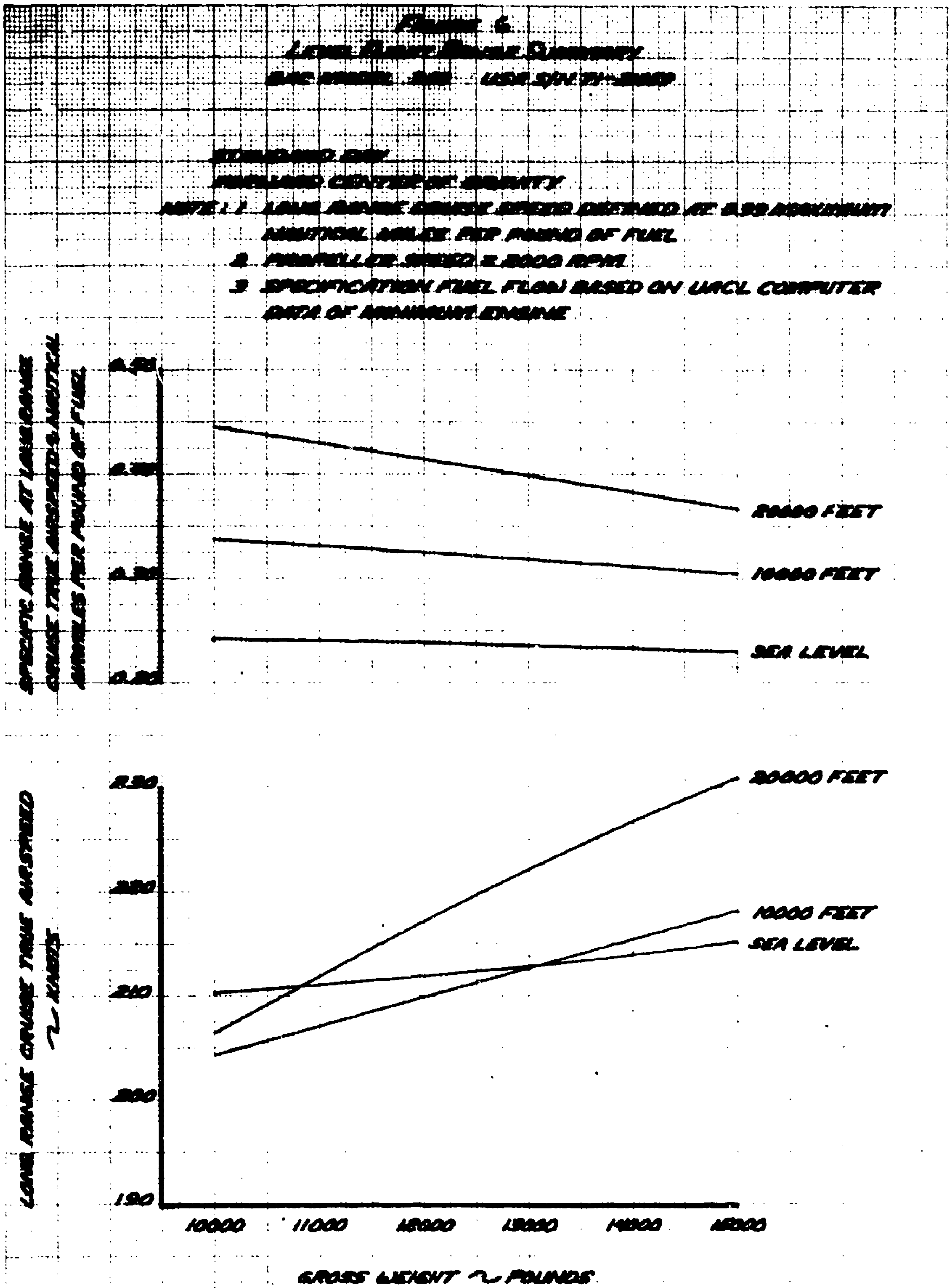


FIGURE 6
LEVEL FLIGHT RANGE SUMMARY
USA S/N 71-
STANDARD DAY
CENTER OF GRAVITY
NOTE: 1. LONG RANGE CRUISE SPEED DEFINED AT 0.99 MAXIMUM NAUTICAL MILES PER POUND OF FUEL
2. PROPELLER SPEED = 2000 RPM
3. SPECIFICATION FUEL FLOW BASED ON UACL COMPUTER DATA OF MINIMUM ENGINE
SPECIFIC RANGE AT LONG RANGE CRUISE TRUE AIRSPEED ~ NAUTICAL MILES PER POUND OF FUEL
0.40
0.35
0.30
0.25
20000 FEET
10000 FEET
SEA LEVEL
LONG RANGE CRUISE TRUE AIRSPEED ~ KNOTS
230
220
210
200
190
20000 FEET
10000 FEET
SEA LEVEL
10000
11000
12000
13000
14000
15000
GROSS WEIGHT ~ POUNDS

LD 7-74

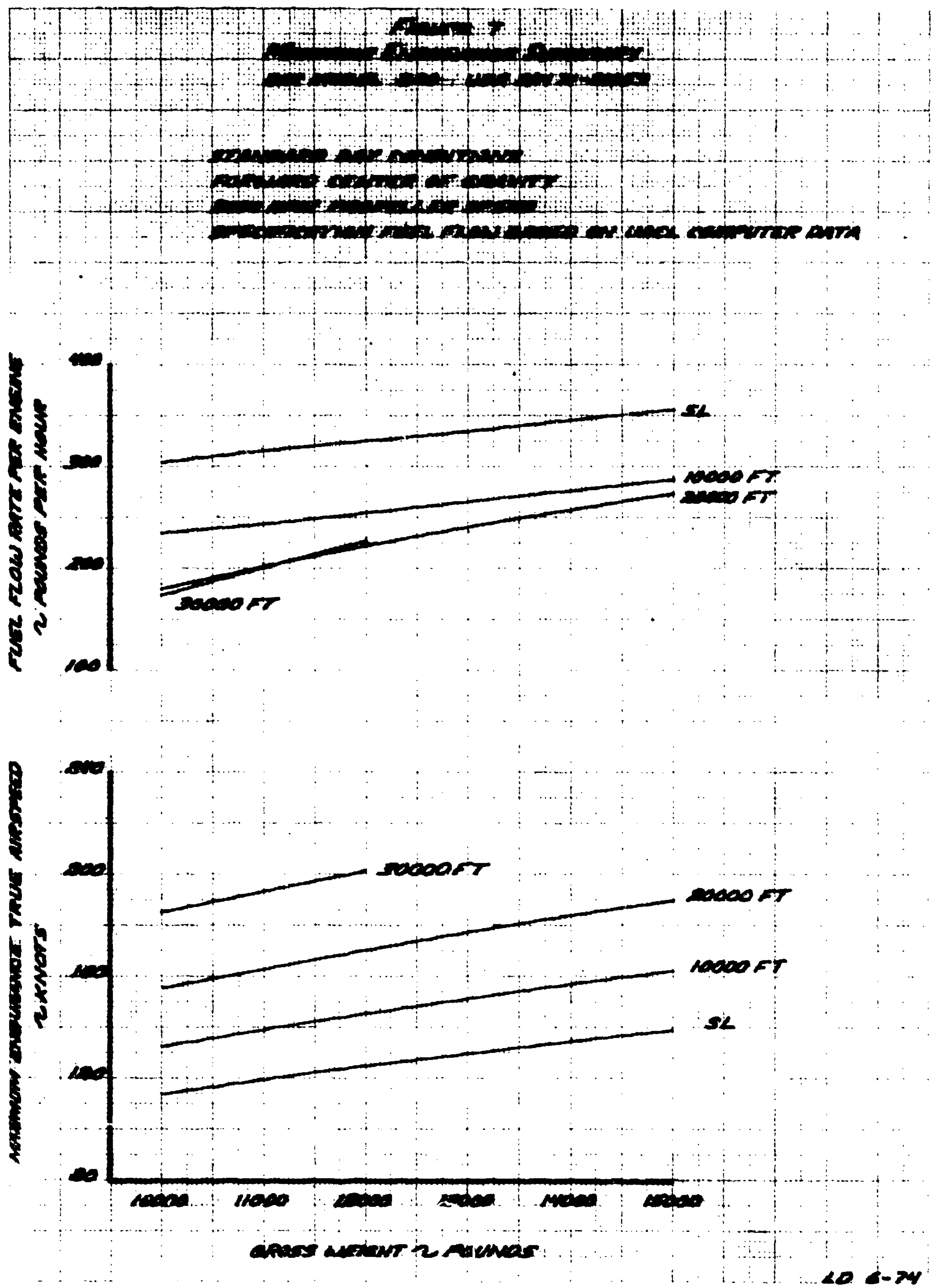


FIGURE 7
STANDARD DAY CONDITIONS
FORWARD CENTER OF GRAVITY
COMPUTER DATA
FUEL FLOW RATE PER ENGINE ~ POUNDS PER HOUR
400
300
200
100
SL
10000 FT
20000 FT
30000 FT
MAXIMUM ENDURANCE TRUE AIRSPEED ~ KNOTS
30000 FT
20000 FT
10000 FT
SL
10000
11000
12000
13000
14000
15000
GROSS WEIGHT ~ POUNDS
LD 6-74

FIGURE 8

MAXIMUM LEVEL FLIGHT AIRSPEED

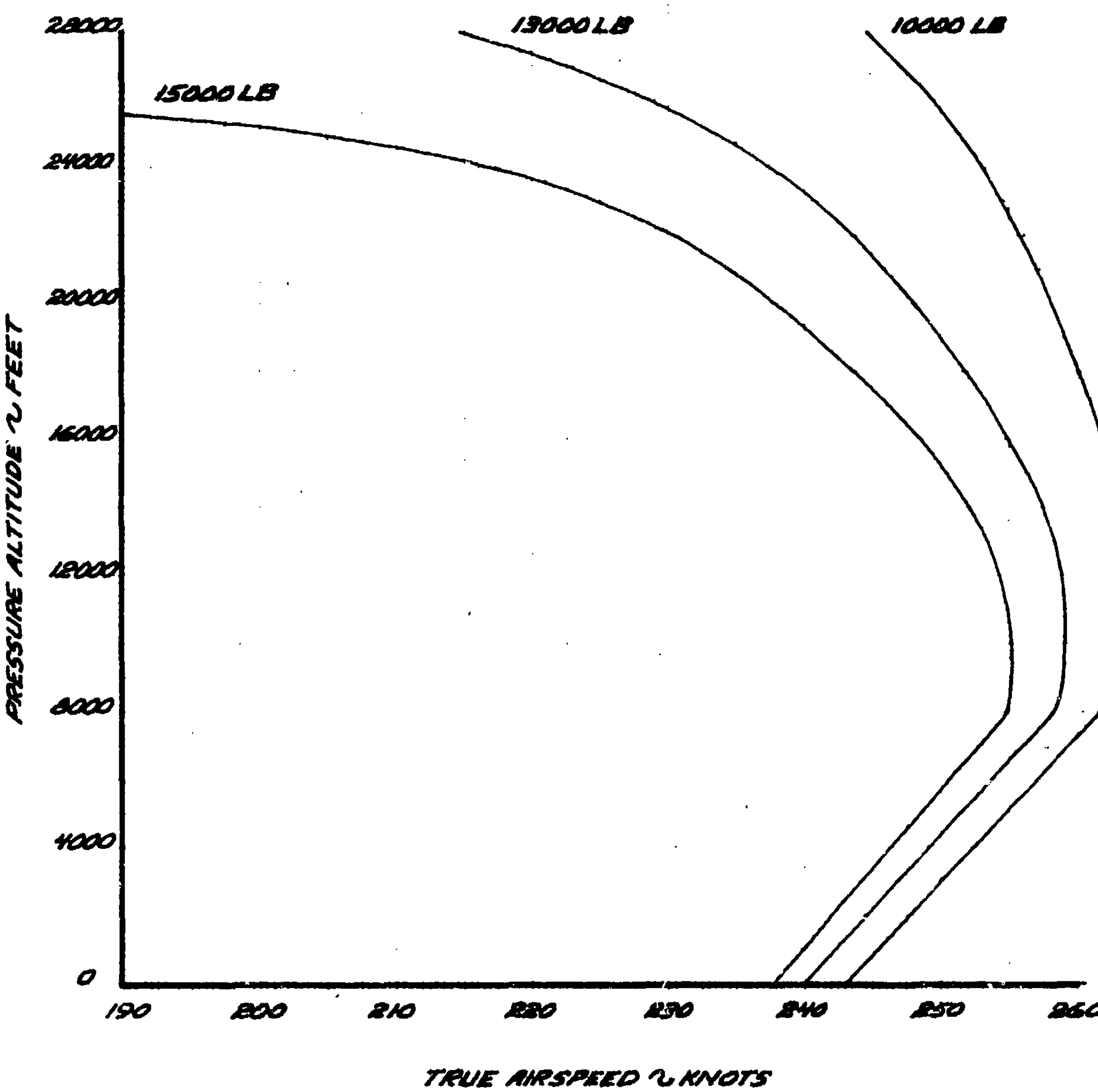
BAC MODEL 200 USA S/N 71-21059

STANDARD DAY CONDITIONS

FORWARD CENTER OF GRAVITY

NOTE: MAXIMUM LEVEL FLIGHT AIRSPEED BASED ON

1. MAXIMUM CRUISE POWER AVAILABLE
2. 5.68 SHP EXTRACTION DUE TO BELT DRIVEN ALTERNATOR
3. ACCESSORY LOSSES: 14.25 SHP BELOW 10°C & 19.29 SHP ABOVE 10°C
4. BLEED AIR: 6.1 LB/MIN. BELOW −17°C & 4.9 LB/MIN. ABOVE −17°C
5. ENGINE INLET EFFICIENCY CURVE REFERENCE 17 APPENDIX A
6. PROPELLER SPEED = 2000 RPM



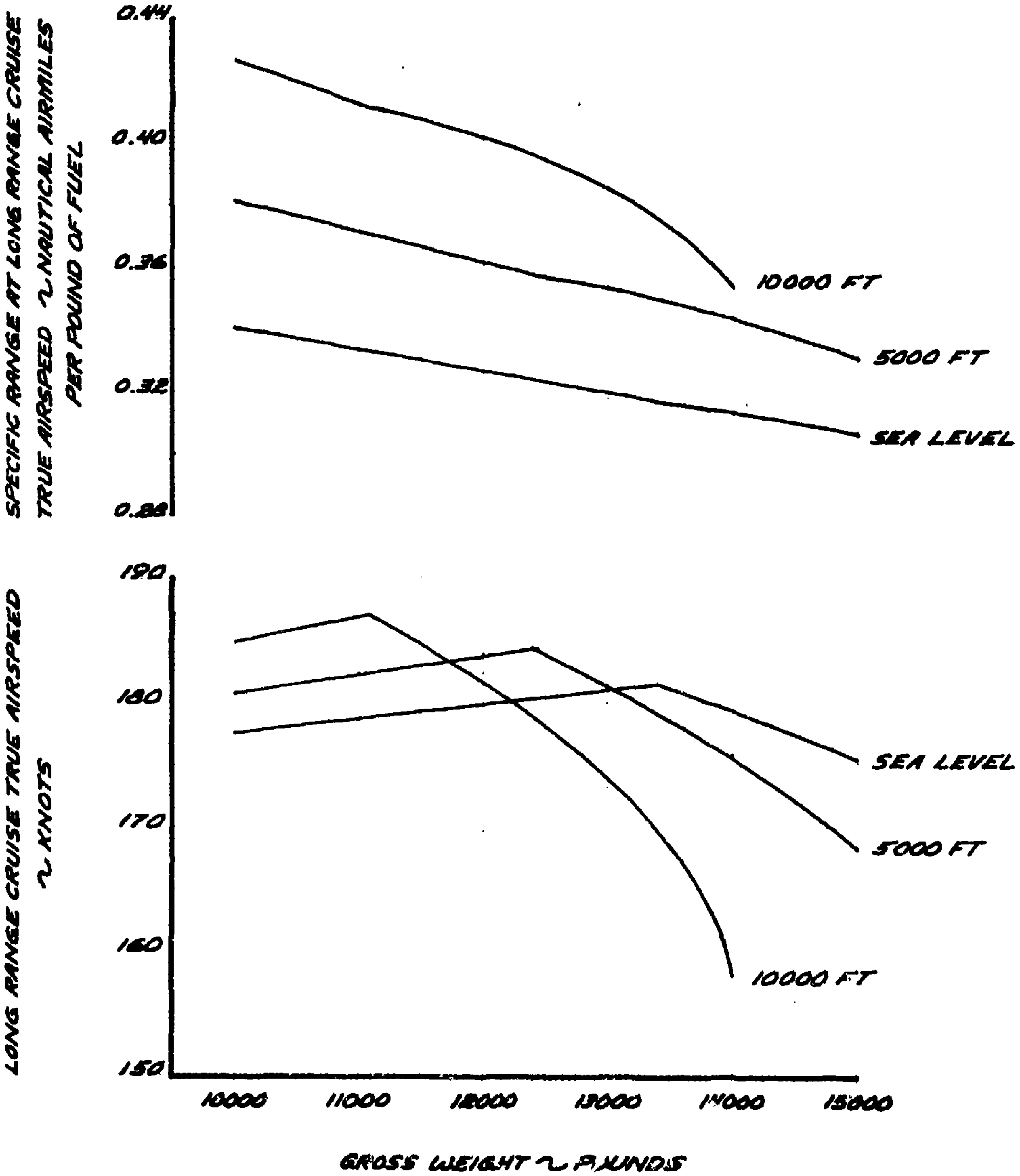
LD 6-74

FIGURE 9

SINGLE ENGINE LEVEL FLIGHT RANGE SUMMARY

BAC MODEL 280 - USA S/N 71-21059

CRUISE CONFIGURATION
STANDARD DAY
FORWARD CENTER OF GRAVITY
2000 RPM PROPELLER SPEED
MAXIMUM CONTINUOUS POWER AVAILABLE



LD G-74

FIGURE 10

DUAL ENGINE CLIMB PERFORMANCE

BAC MODEL 260 : USA S/N 71-21059

STANDARD DAY

BEST RATE OF CLIMB AIRSPEED

FORWARD CENTER OF GRAVITY

15000 POUNDS GROSS WEIGHT

CRUISE CONFIGURATION

MAXIMUM CONTINUOUS SHAFT HORSEPOWER

RATE OF CLIMB CALCULATED USING DRAG POLAR EQUATION AND UACL ENGINE DATA

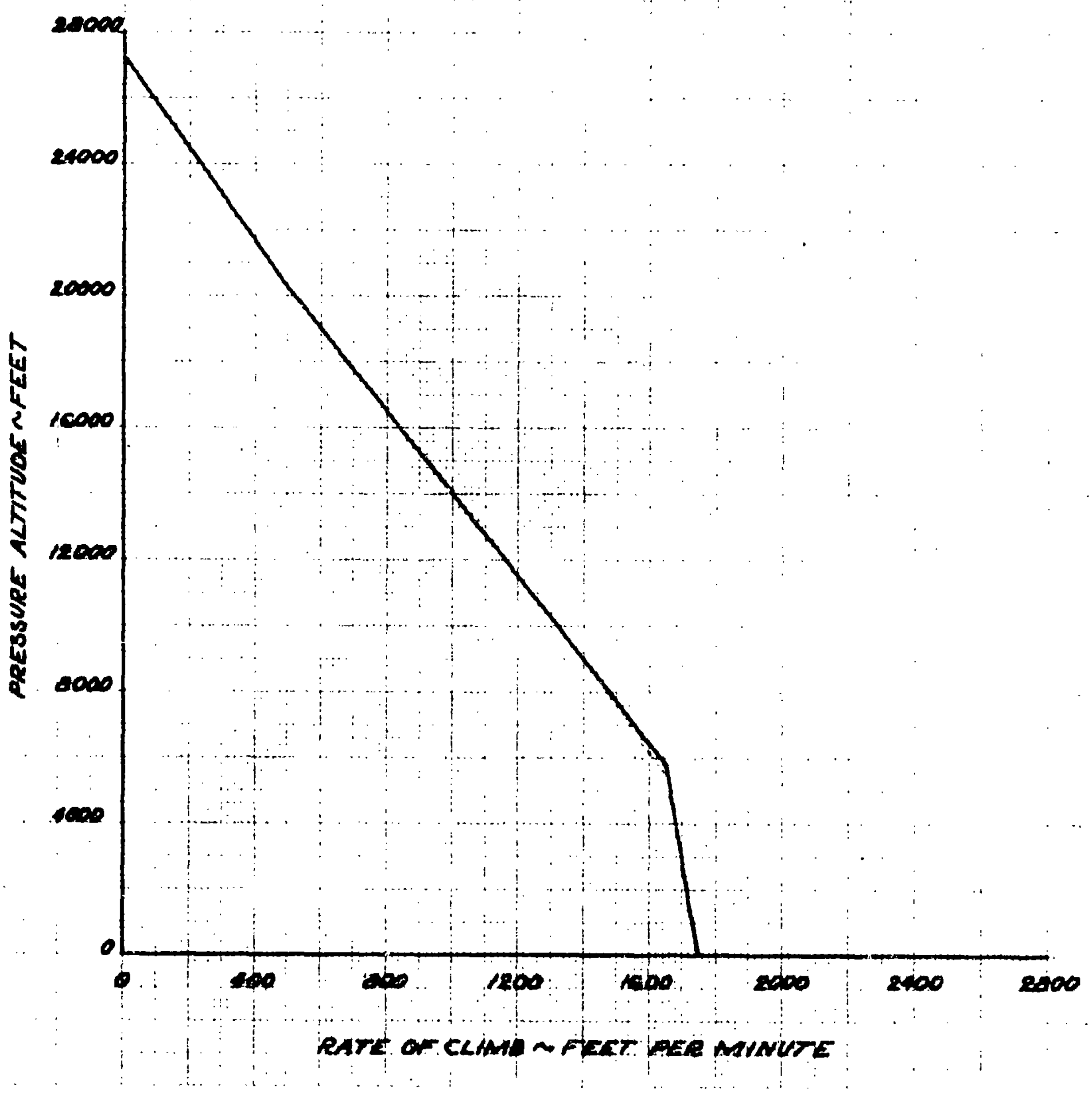
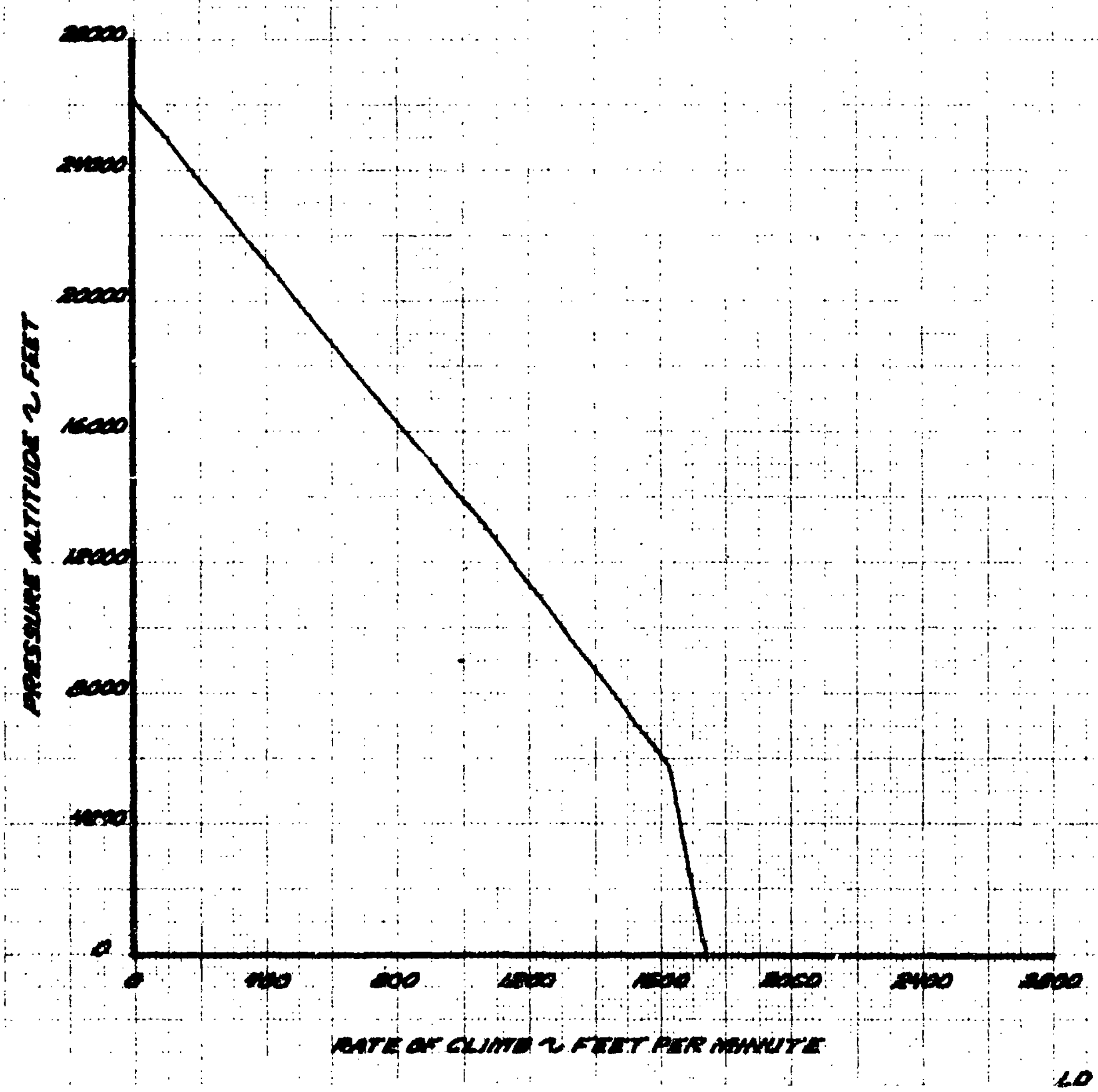


FIGURE 11
DUAL ENGINE CLIMB PERFORMANCE
[illegible] MODEL [illegible] - USA S/N 71-[illegible]

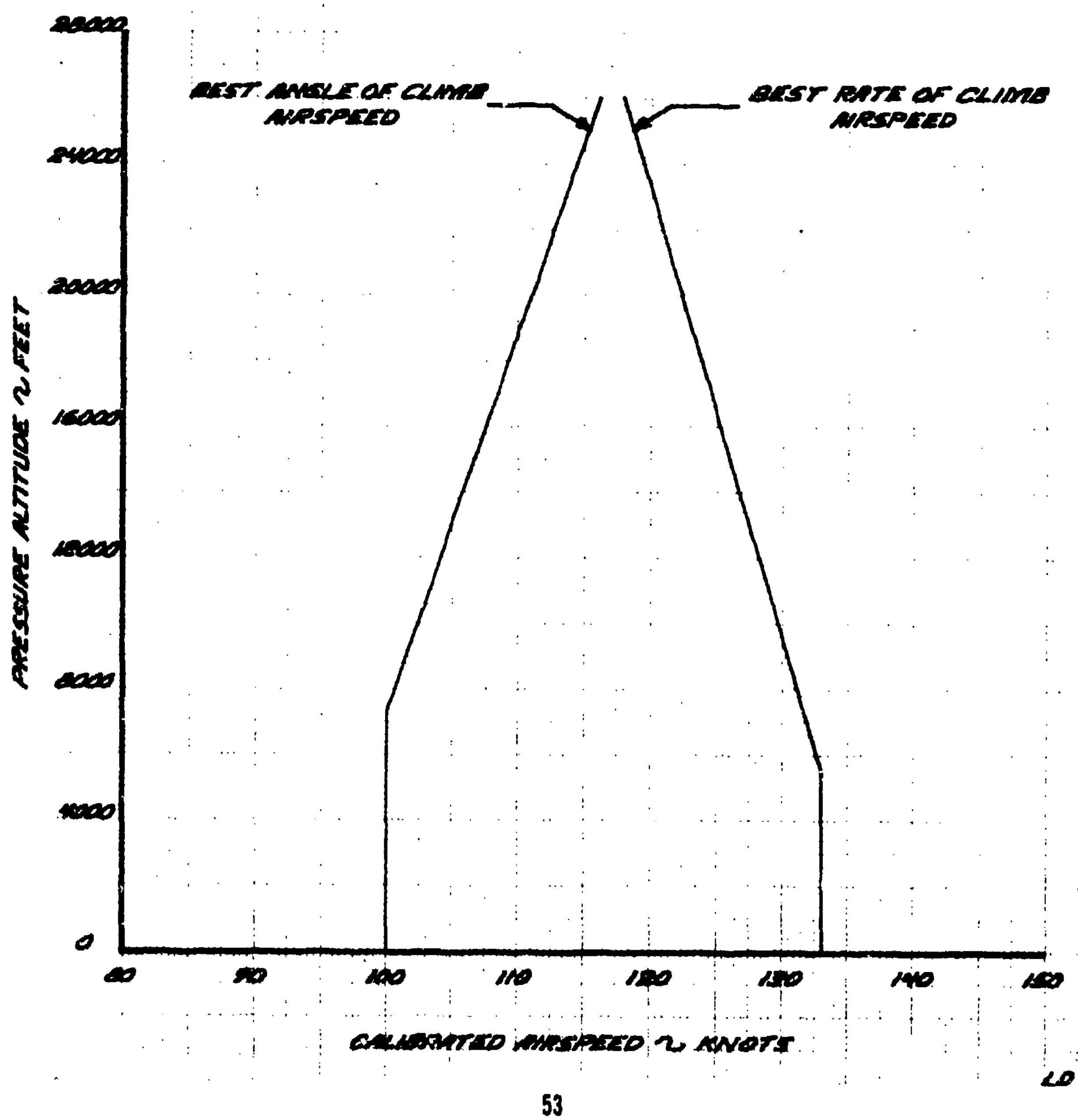
STANDARD DAY
133 KNOTS CALIBRATED AIRSPEED
FORWARD CENTER OF GRAVITY
15000 POUNDS GROSS WEIGHT
CRUISE CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
RATE OF CLIMB CALCULATED USING DRAG POLAR EQUATION
AND UACL ENGINE DATA



LD 6-74

FIGURE 12
DUAL ENGINE CLIMB PERFORMANCE
BHC MODEL [illegible] USA S/N 71-[illegible]

STANDARD DAY
FORWARD CENTER OF GRAVITY
[illegible] POUNDS GROSS WEIGHT
CRUISE CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
AIRSPEEDS CALCULATED USING DRAG POLAR EQUATION
AND UACL ENGINE DATA



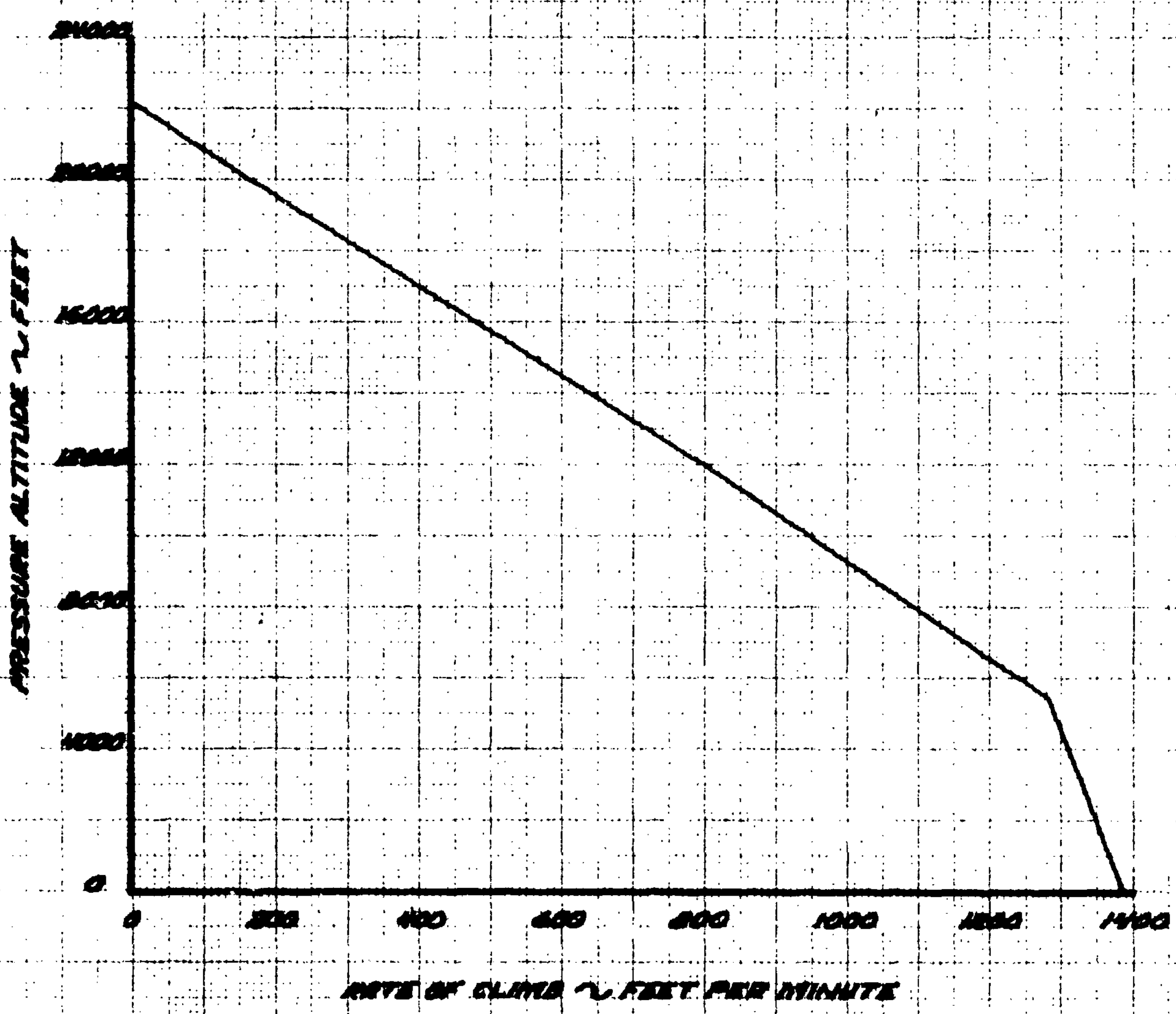
LD 6-74

FIGURE [illegible]

DUAL ENGINE CLIMB PERFORMANCE

[illegible]

STANDARD DAY
[illegible] KNOTS CALIBRATED AIRSPEED
TAKEOFF CONFIGURATION
FORWARD CENTER OF GRAVITY
[illegible] POUNDS GROSS WEIGHT
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
RATE OF CLIMB CALCULATED USING DRAG POLAR EQUATION
AND UACL ENGINE DATA



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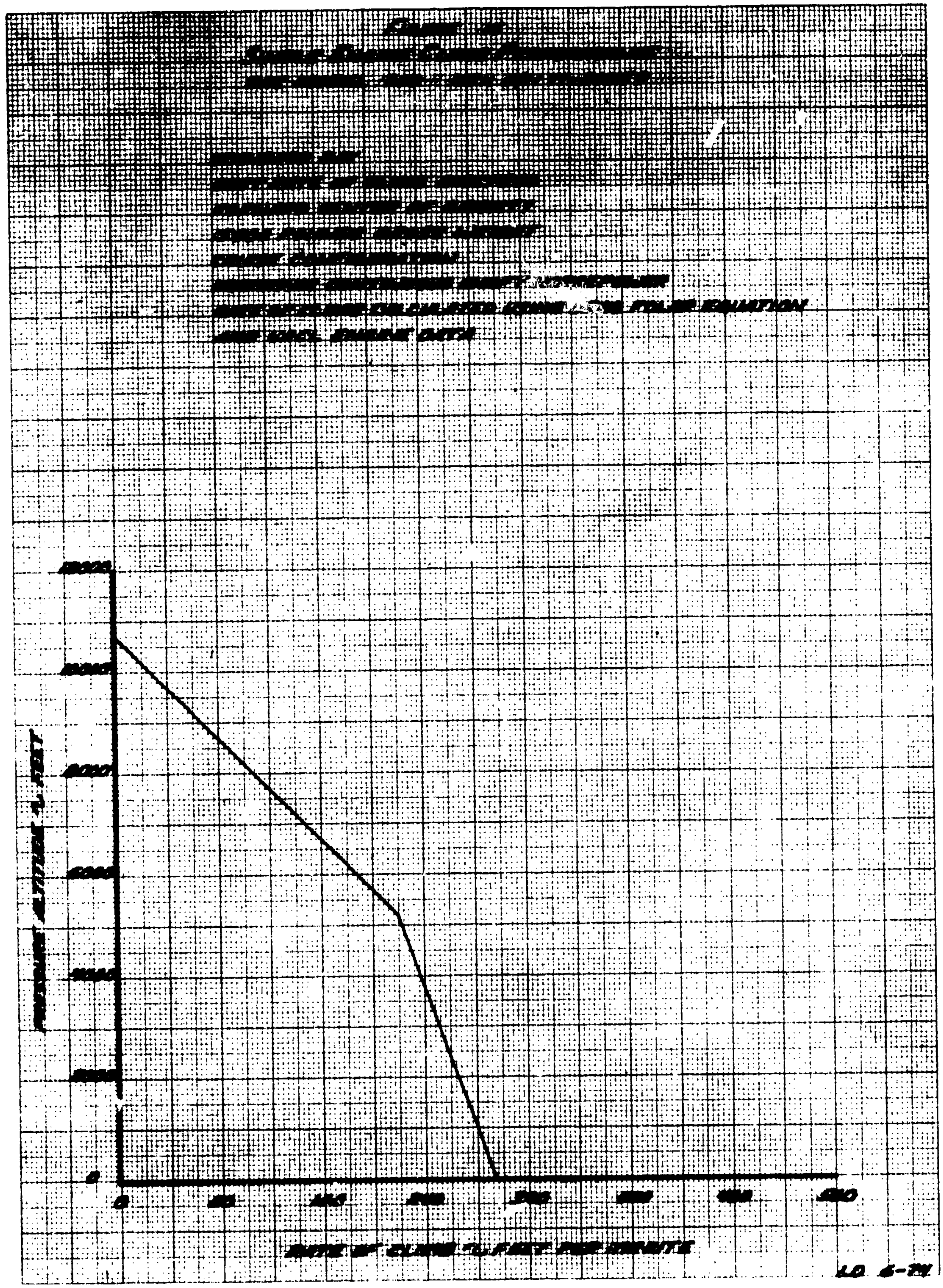


FIGURE 15

SINGLE ENGINE CLIMB PERFORMANCE

BHC MODEL 300 - USA S/N 71-21059

STANDARD DAY

124 KNOTS CALIBRATED AIRSPEED

FORWARD CENTER OF GRAVITY

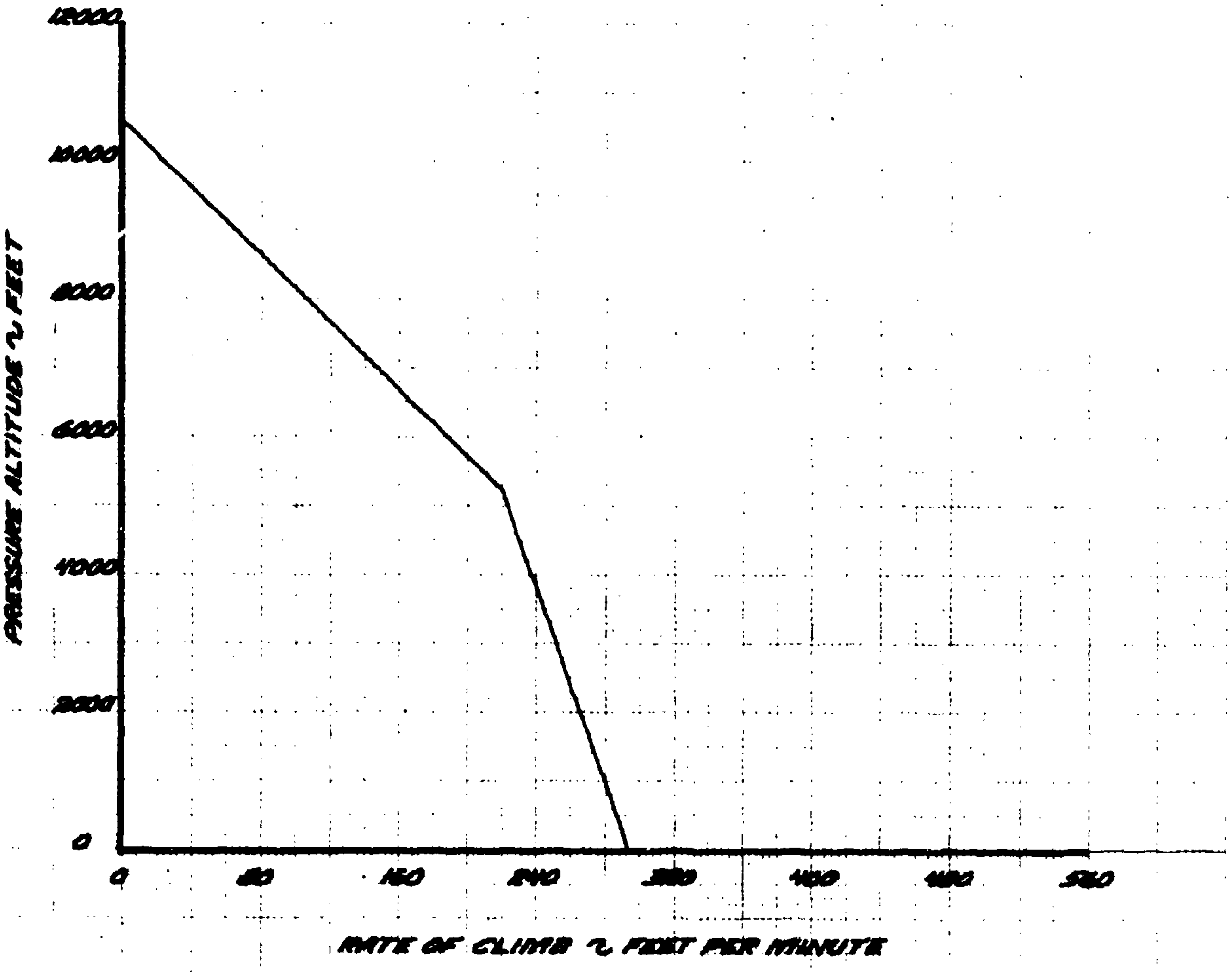
15000 POUNDS GROSS WEIGHT

CRUISE CONFIGURATION

MAXIMUM CONTINUOUS SHAFT HORSEPOWER

RATE OF CLIMB CALCULATED USING DRAG POLAR EQUATION

AND UACL ENGINE DATE



LD 6-74

FIGURE 16

SINGLE ENGINE CLIMB PERFORMANCE

BAC MODEL 200 USA S/N 71-[illegible]

STANDARD DAY

FORWARD CENTER OF GRAVITY

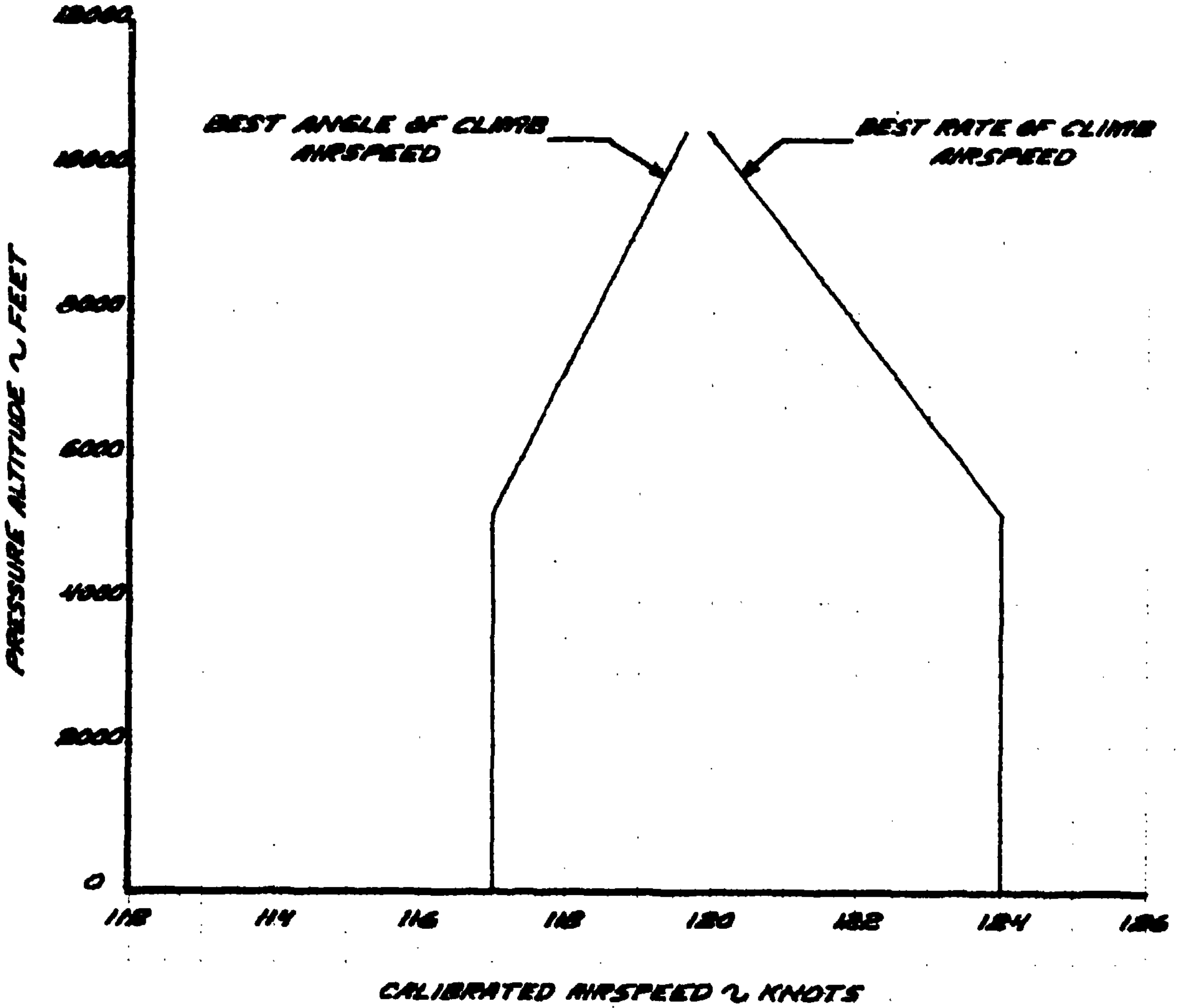
[illegible] POUNDS GROSS WEIGHT

CRUISE CONFIGURATION

MAXIMUM CONTINUOUS SHAFT HORSEPOWER

AIRSPEEDS CALCULATED USING DRAG POLAR EQUATION

AND UACL ENGINE DATA



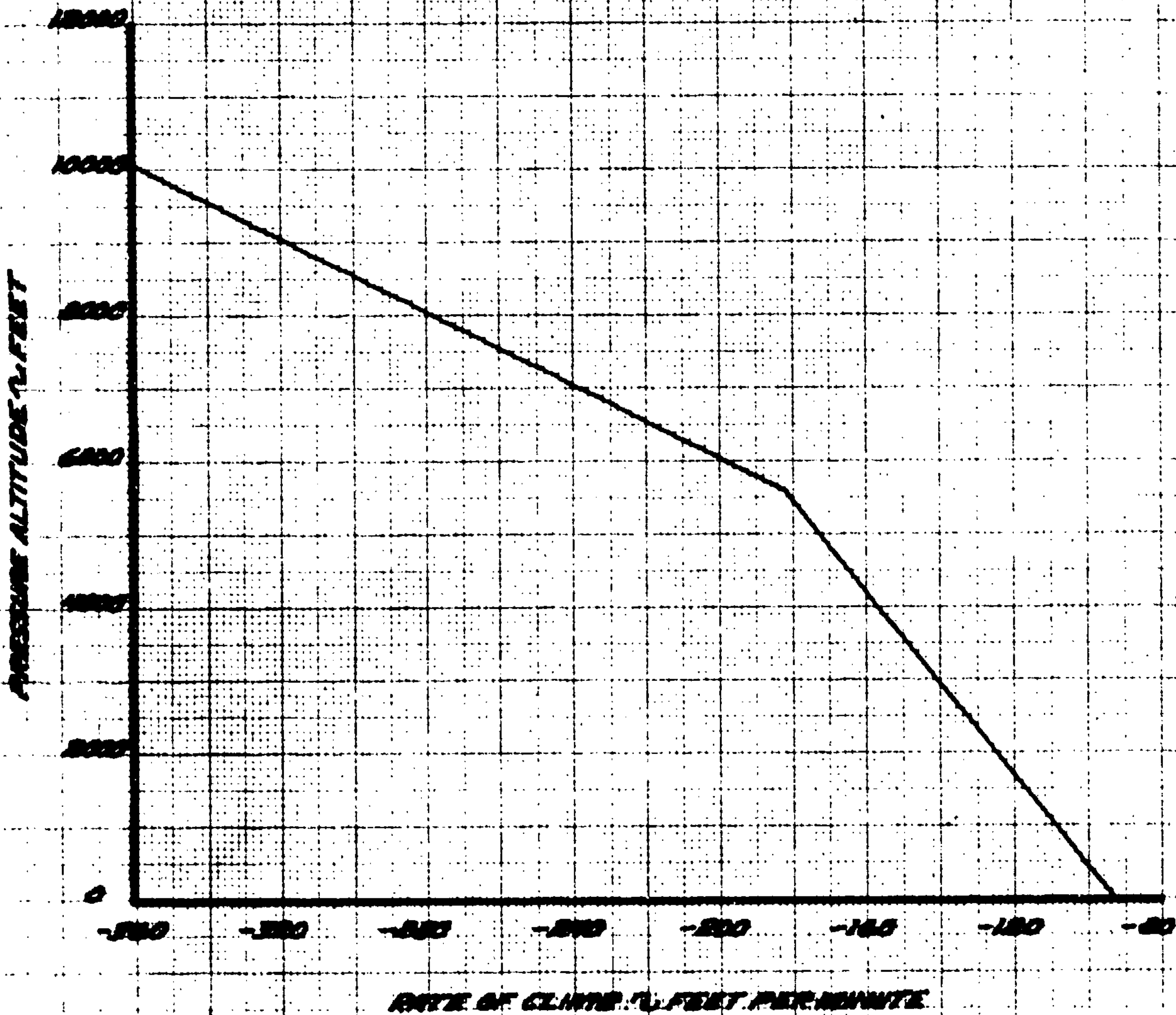
LD 6-74

FIGURE 17

SINGLE ENGINE CLIMB PERFORMANCE

[illegible] MODEL [illegible] USA S/N 71-[illegible]

STANDARD DAY
[illegible] KNOTS CALIBRATED AIRSPEED
FORWARD CENTER OF GRAVITY
[illegible] POUNDS GROSS WEIGHT
TAKEOFF CONFIGURATION
MAXIMUM CONTINUOUS SHAFT HORSEPOWER
RATE OF CLIMB CALCULATED USING DRAG POLAR EQUATION
AND [illegible] ENGINE DATA



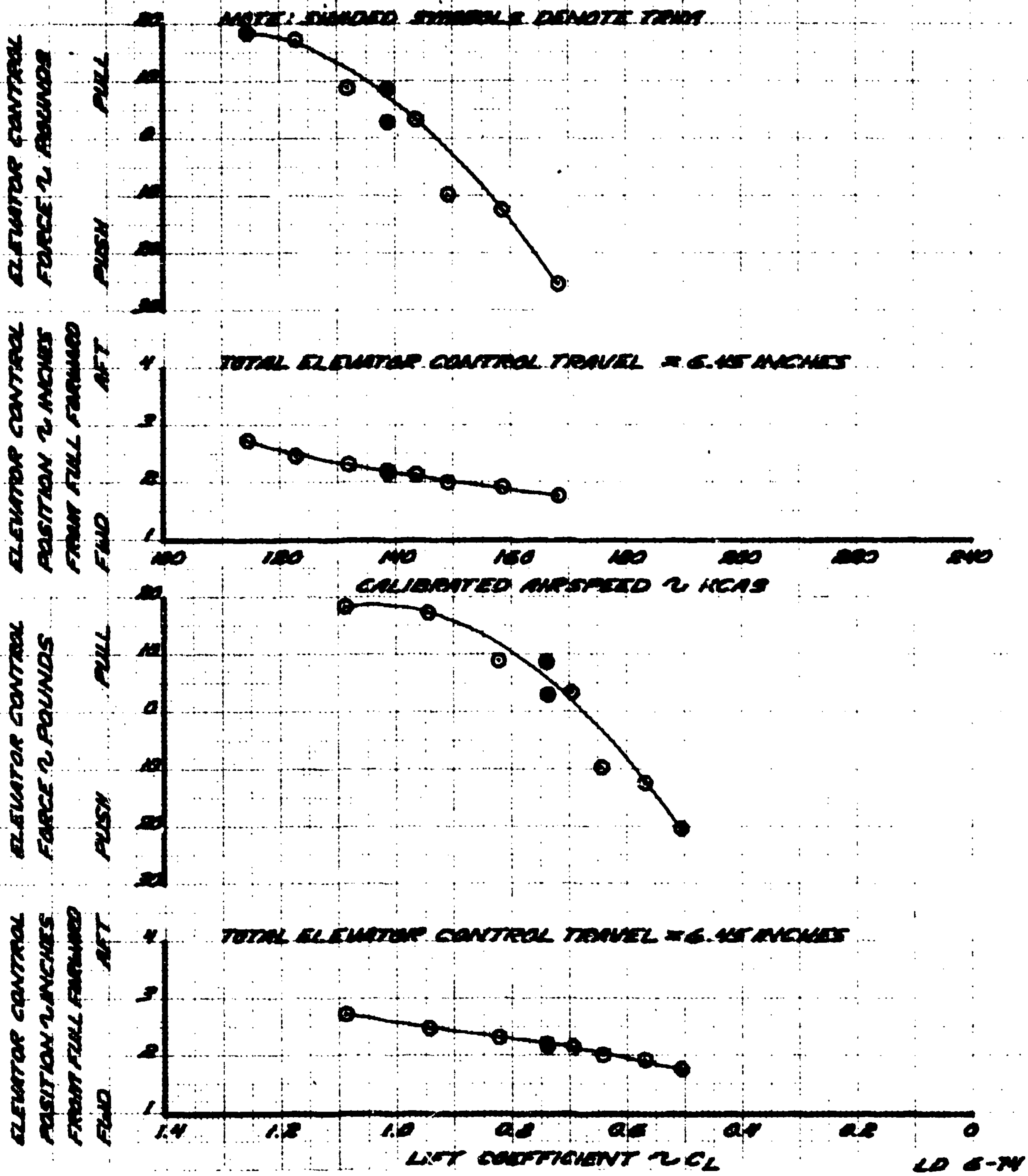
I.D. 6-74

FIGURE 10

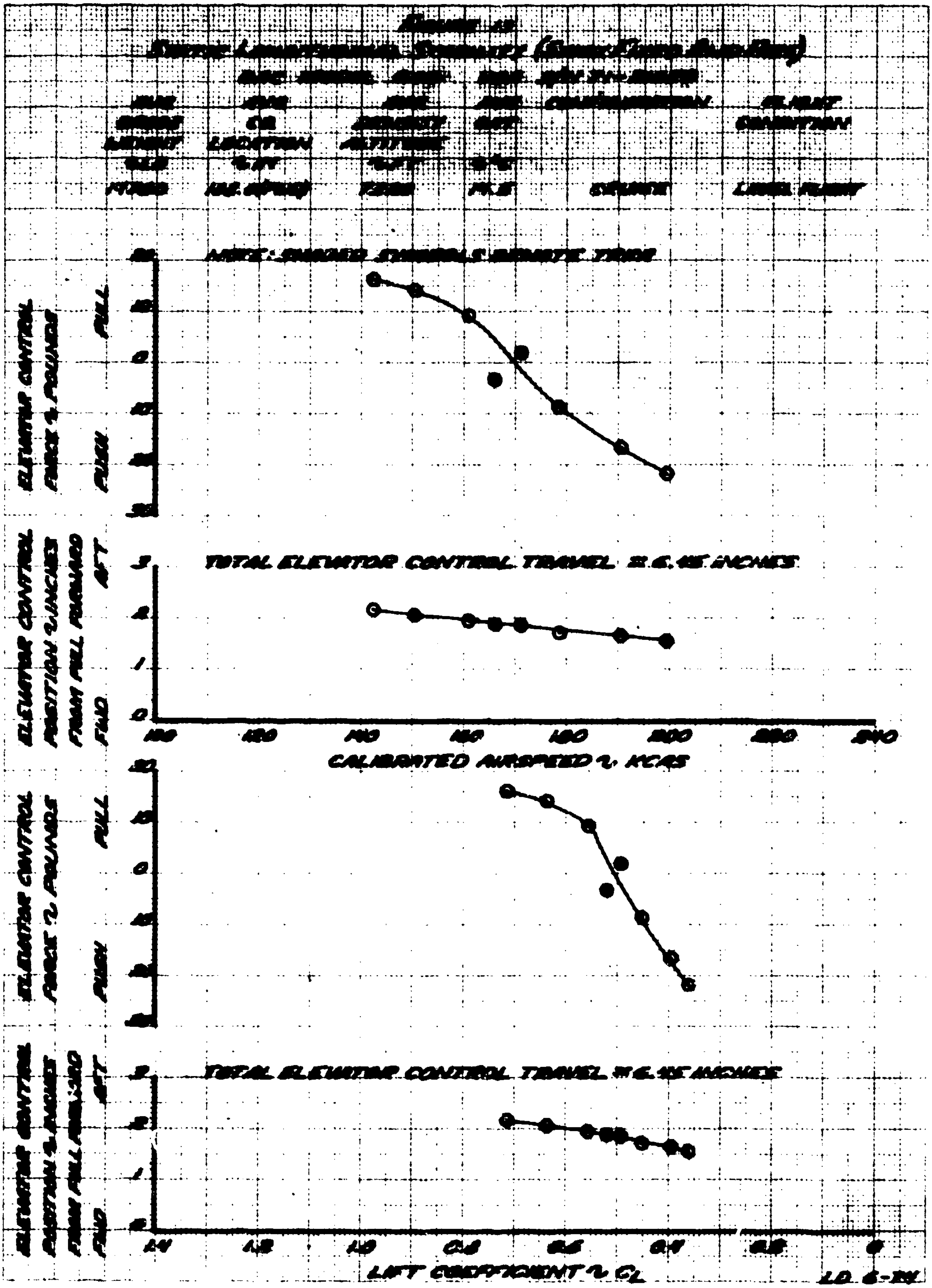
STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

[illegible] MODEL [illegible] USA S/N [illegible]

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
[illegible]	137.5(FWD)	[illegible]	12.0	CRUISE	LEVEL FLIGHT



LD 6-74


NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE ~ POUNDS
PUSH
PULL
TOTAL ELEVATOR CONTROL TRAVEL = 6.45 INCHES
ELEVATOR CONTROL POSITION ~ INCHES FROM FULL FORWARD
FWD
AFT
CALIBRATED AIRSPEED ~ KCAS
LIFT COEFFICIENT ~ C_L

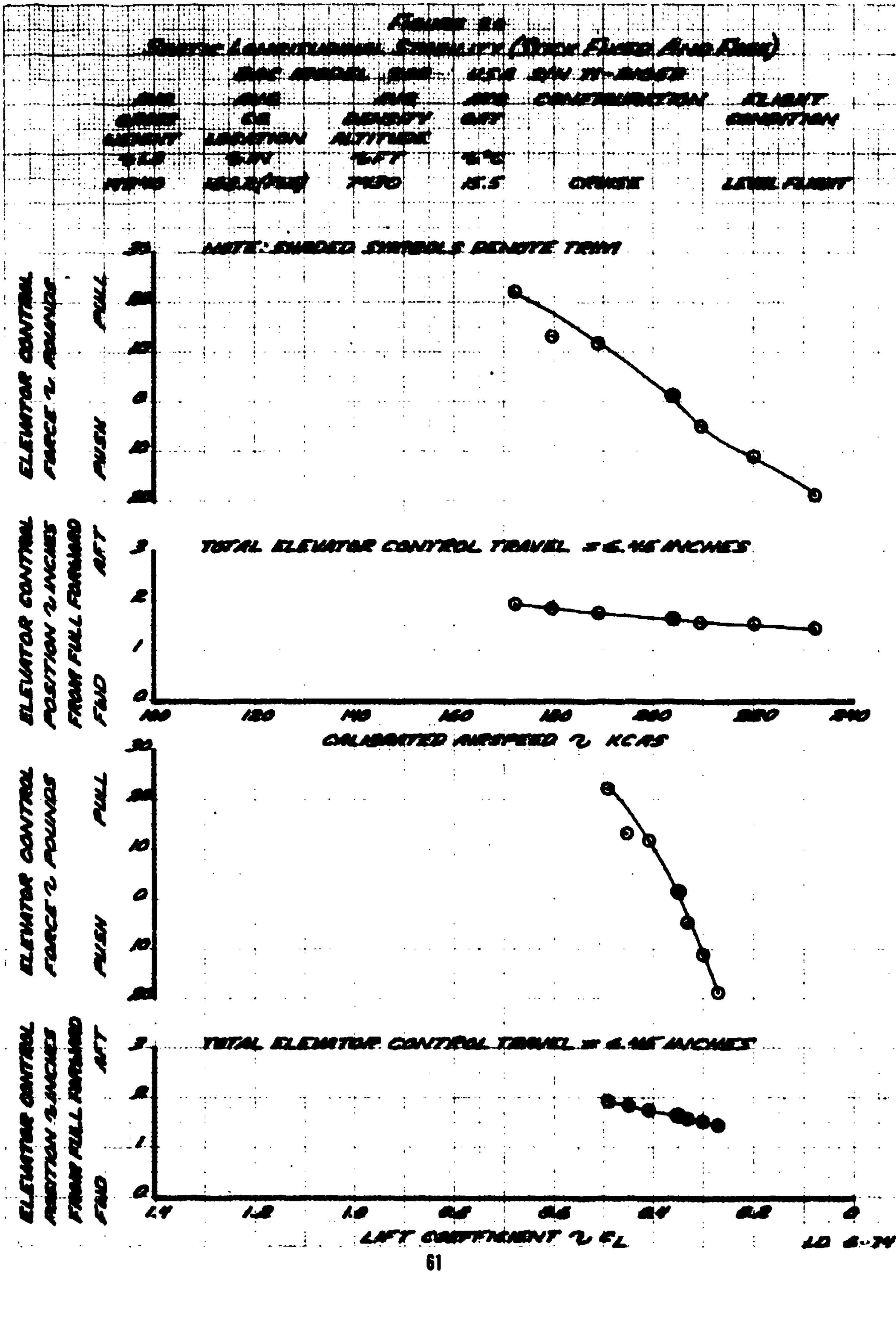
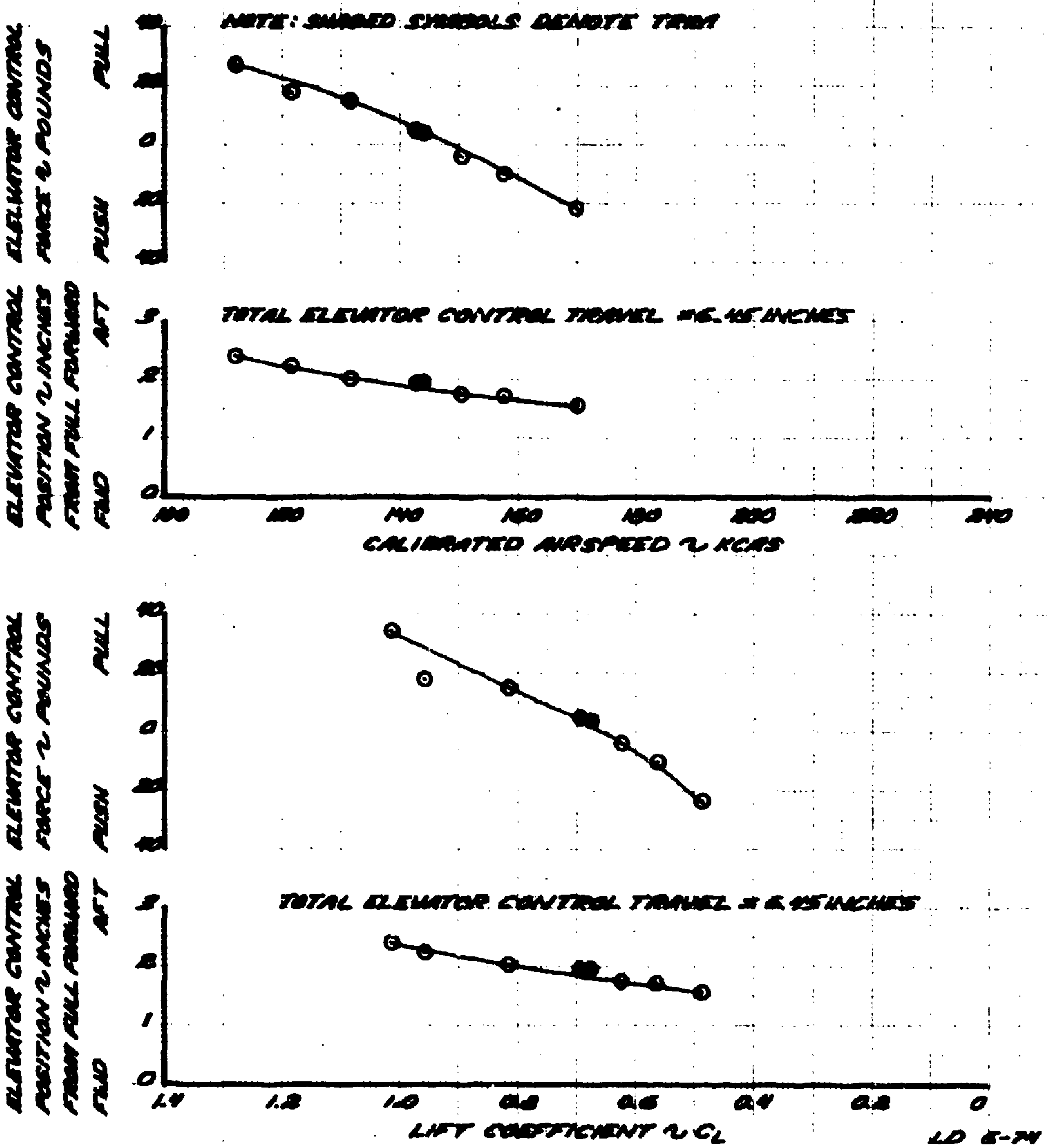

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)
CONFIGURATION
FLIGHT CONDITION
DENSITY ALTITUDE
CRUISE
LEVEL FLIGHT
NOTE: SHADED SYMBOLS DENOTE TRIM
ELEVATOR CONTROL FORCE ~ POUNDS
PULL
PUSH
TOTAL ELEVATOR CONTROL TRAVEL = 6.45 INCHES
ELEVATOR CONTROL POSITION ~ INCHES FROM FULL FORWARD
AFT
FWD
CALIBRATED AIRSPEED ~ KCAS
LIFT COEFFICIENT ~ C_L

FIGURE 21

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

[illegible] MODEL [illegible] USA S/N [illegible]

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
[illegible]	[illegible]	[illegible]	[illegible]	POWER APPROACH	LEVEL FLIGHT



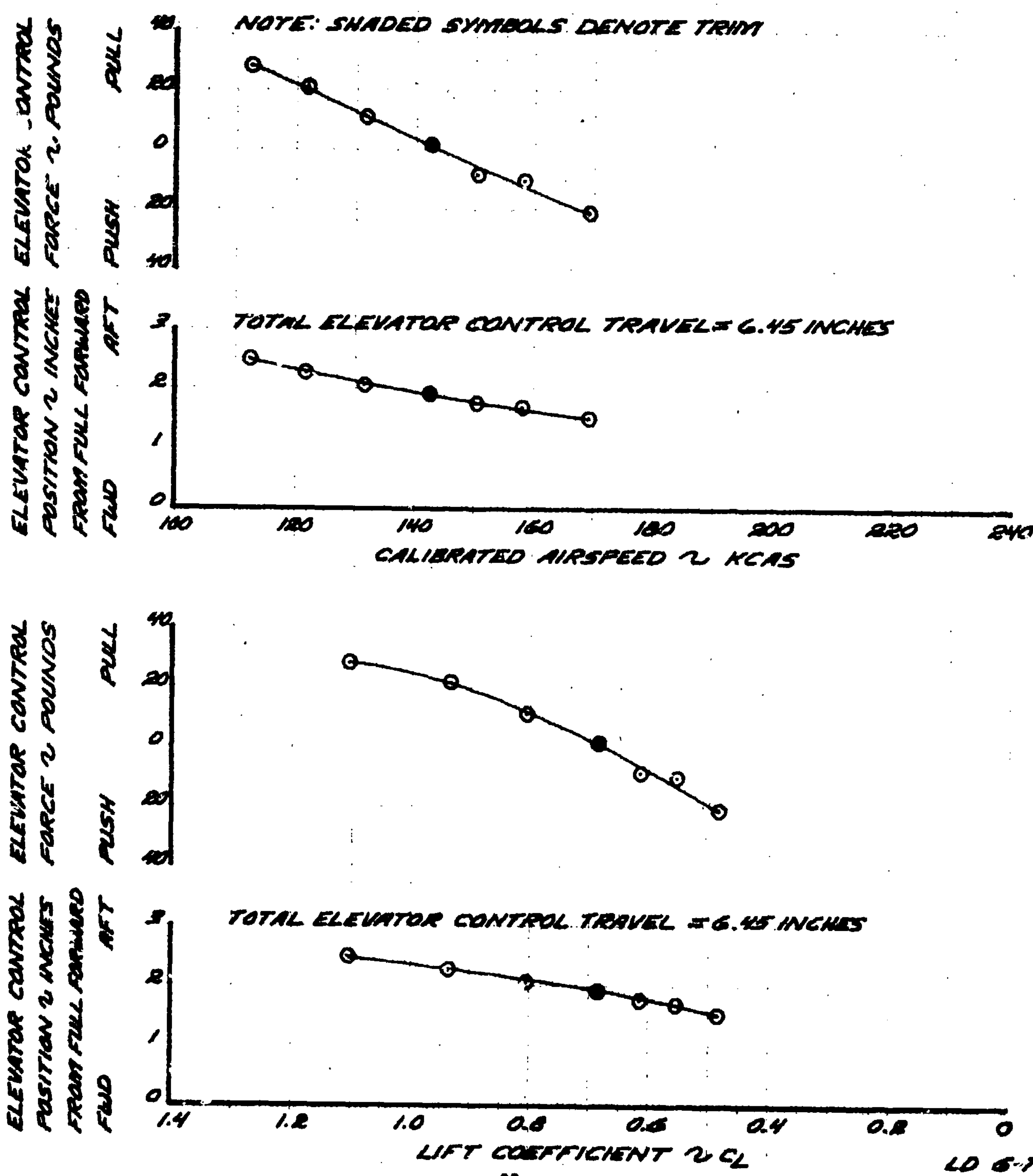
LD 6-74

FIGURE 22

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL 200 USA S/N 71-21059

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14880	187.5(FWD)	7320	14.5	POWER APPROACH	DESCENT



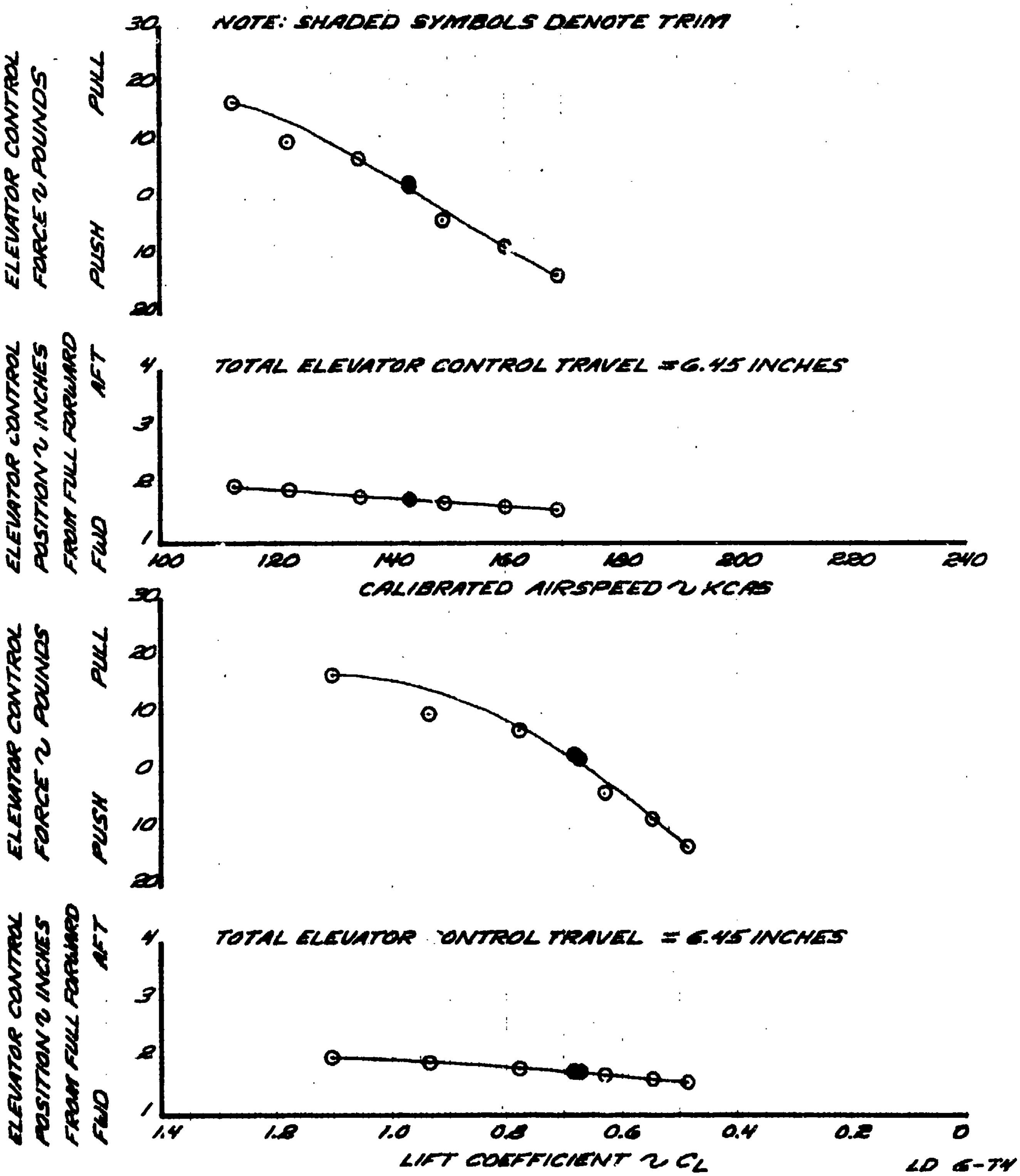
LD 6-74

FIGURE 23

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL 200 USA S/N 71-21059

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14360	197.1 (AFT)	7800	8.0	CRUISE	LEVEL FLIGHT



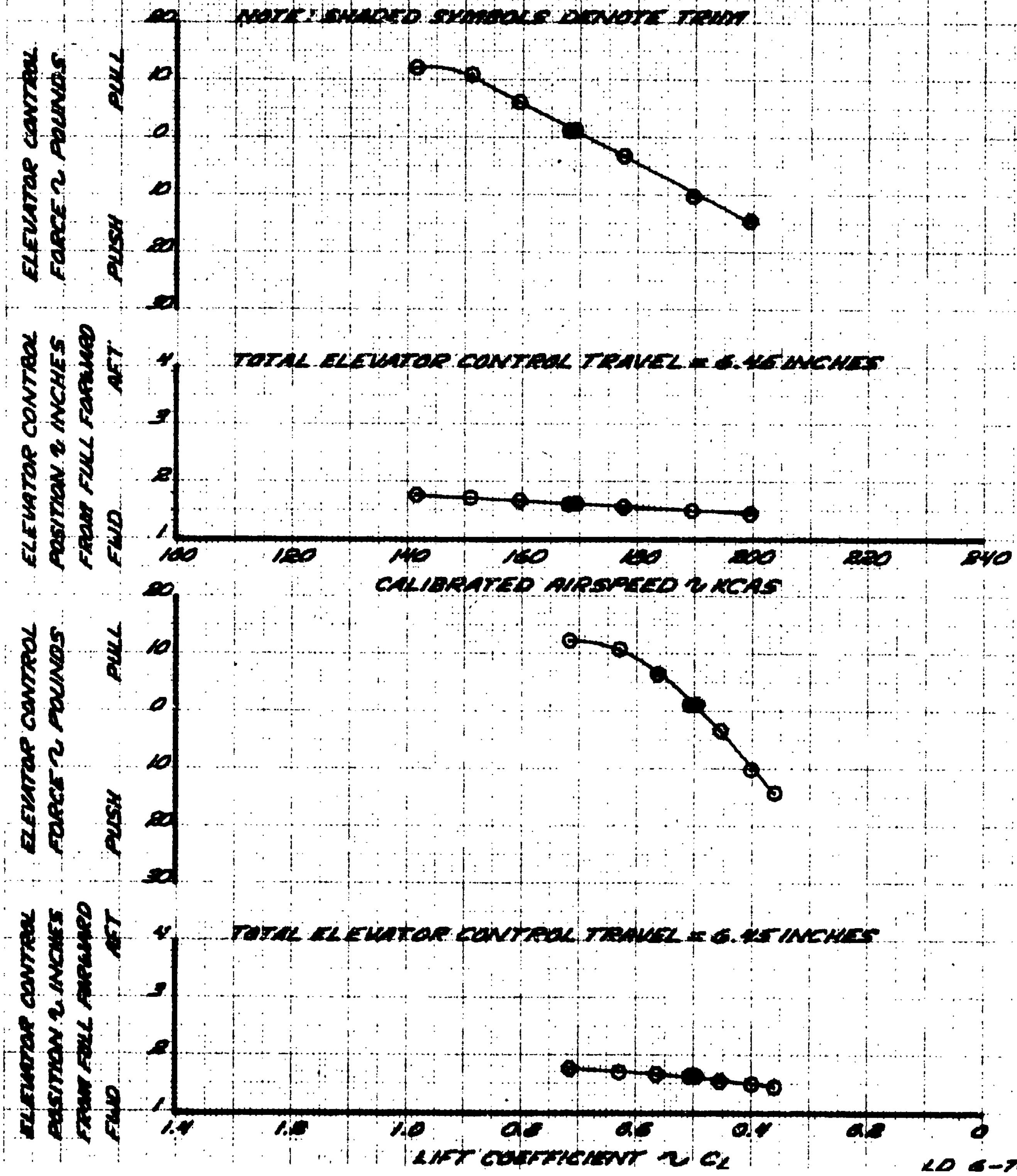
LD 6-74

FIGURE [illegible]

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

[illegible]

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
[illegible]	157.3(AFT)	7770	3.5	CRUISE	LEVEL FLIGHT

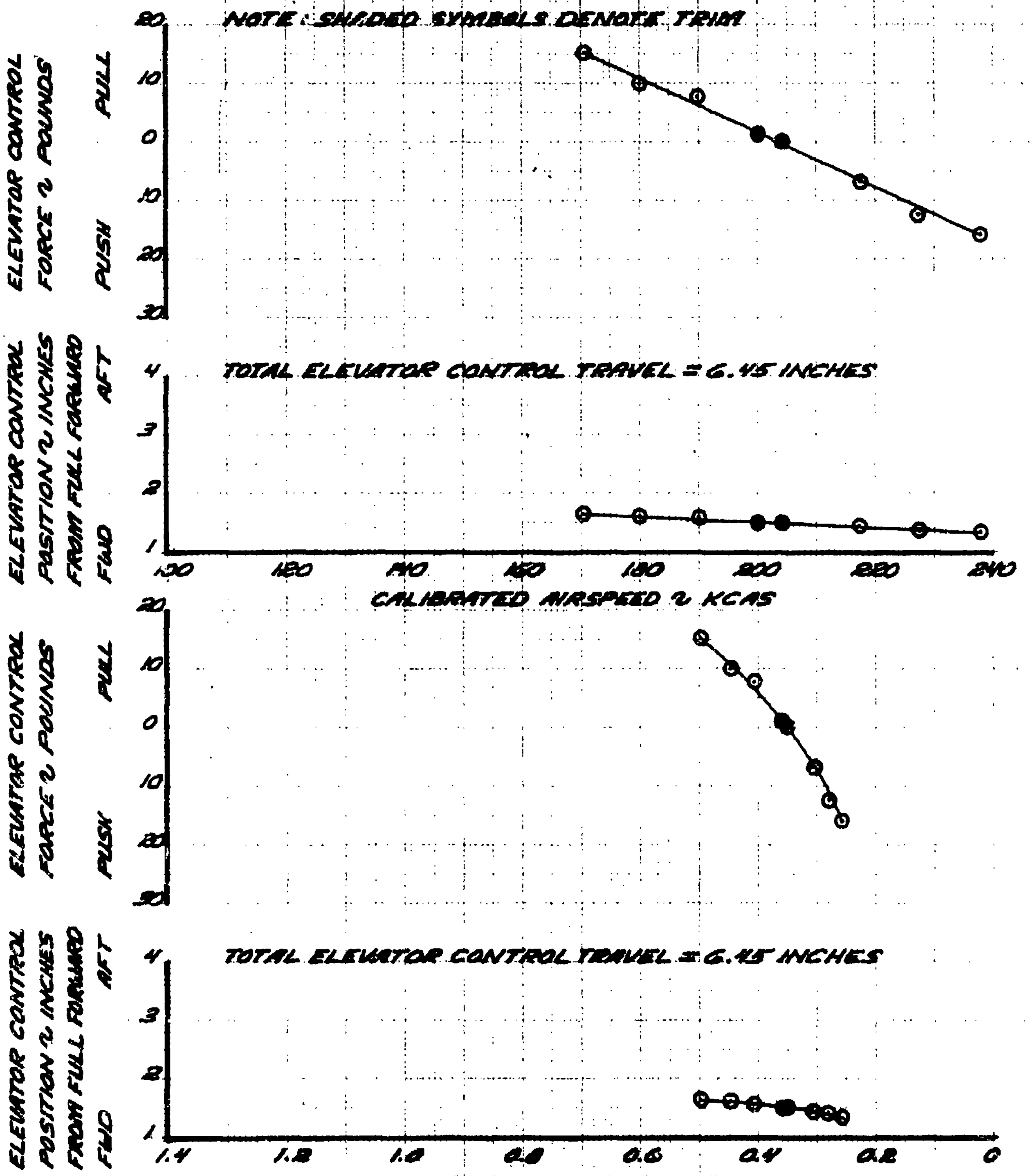


LD 6-74

FIGURE 25

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

AVG GROSS WEIGHT ~ LB	AVG CG LOCATION ~ IN	AVG DENSITY ALTITUDE ~ FT	AVG OAT ~ °C	CONFIGURATION	FLIGHT CONDITION
14880	197.3 (AFT)	7310	8.5	CRUISE	LEVEL FLIGHT



LD 6-74

FIGURE 26

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL 200 USA S/N 71-21059

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14560	197.8(AFT)	7730	8.0	CRUISE	LEVEL FLIGHT

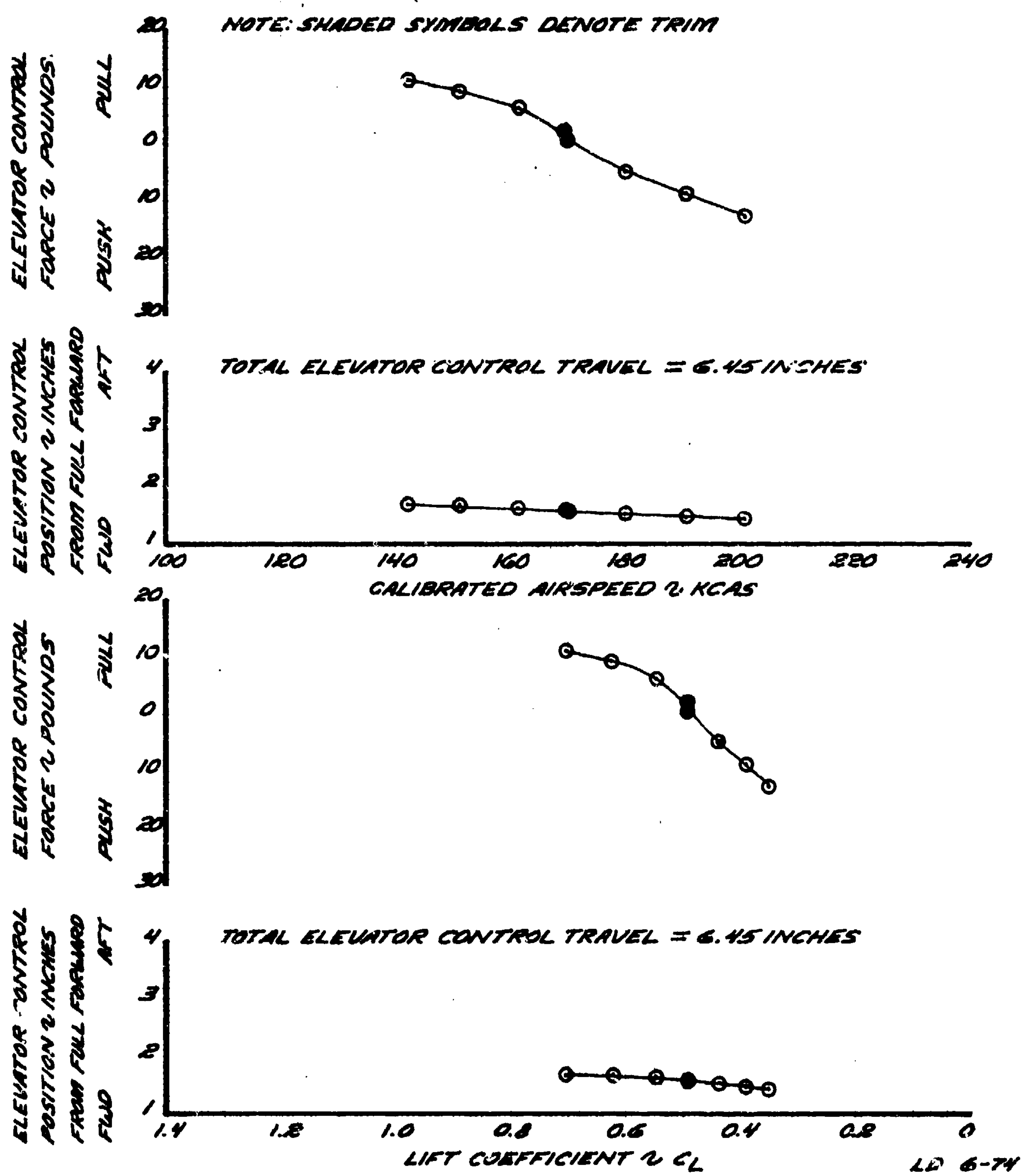
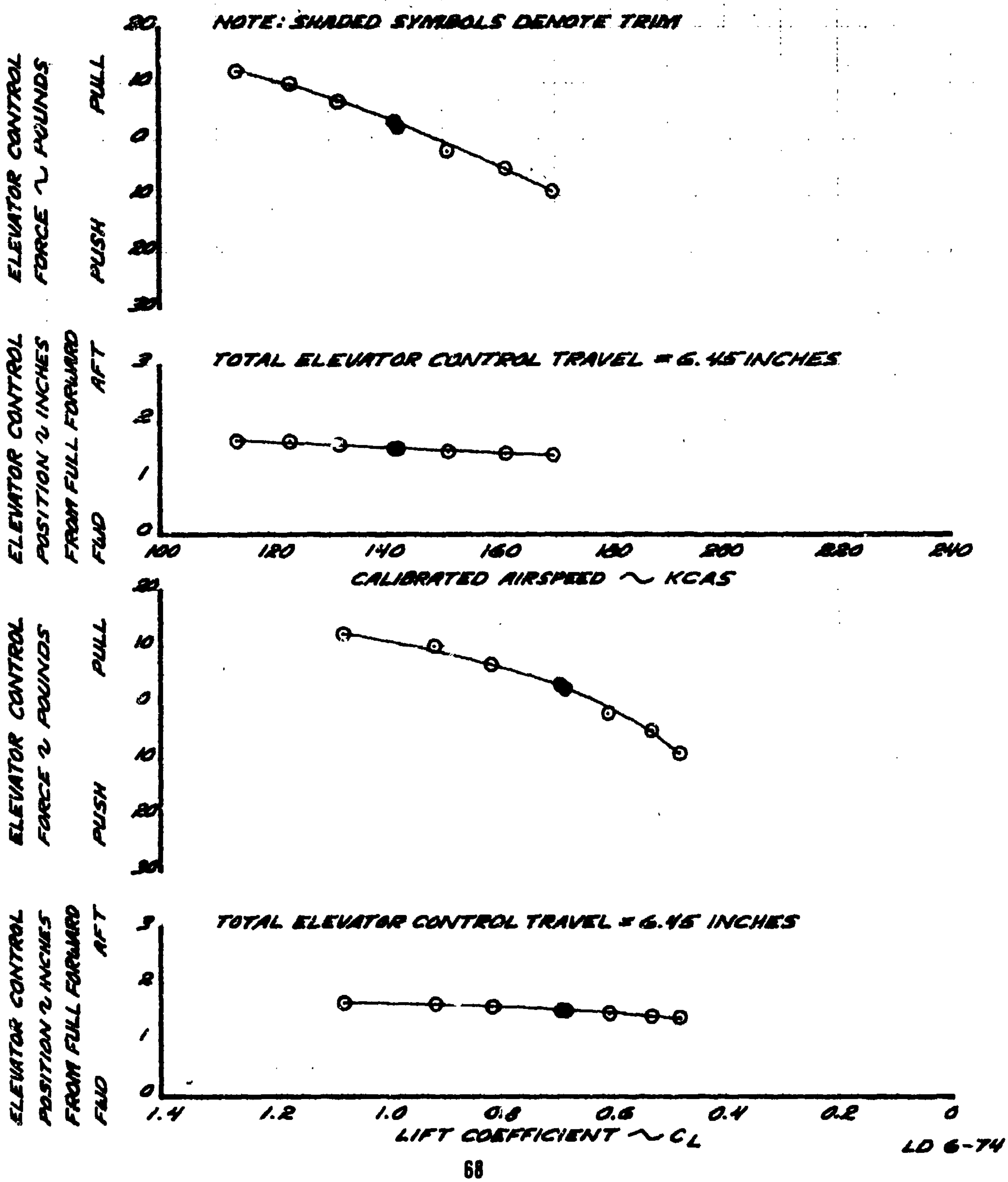


FIGURE 27

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL 200 USA S/N 71-21059

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14200	197.1 (AFT)	7950	8.0	POWER APPROACH	LEVEL FLIGHT



LD 6-74

FIGURE 28

STATIC LONGITUDINAL STABILITY (STICK FIXED AND FREE)

BAC MODEL 200 USA S/N 71-21058

AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	CONFIGURATION	FLIGHT CONDITION
14040	197.1(AFT)	7600	9.5	POWER APPROACH	DESCENT

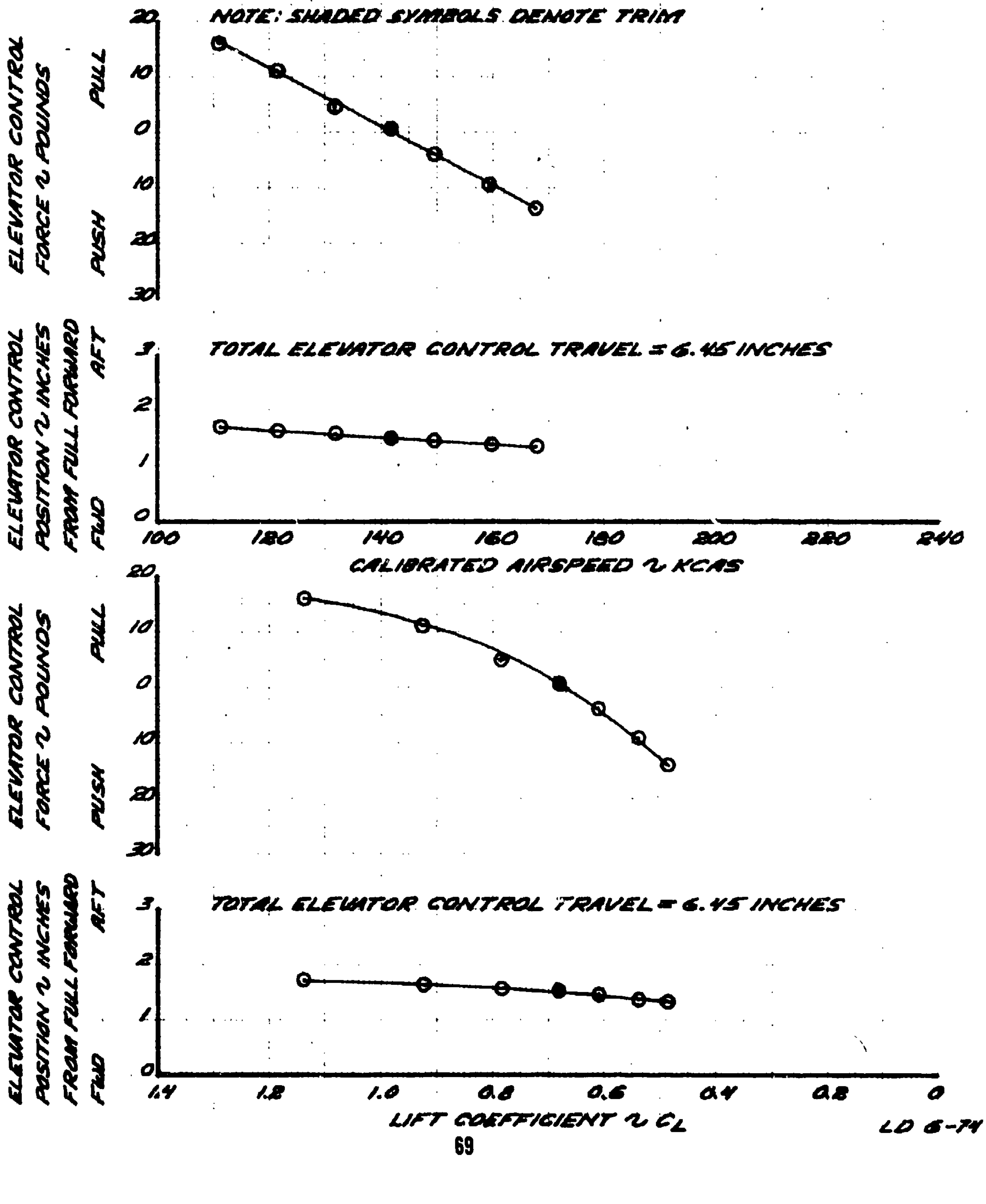
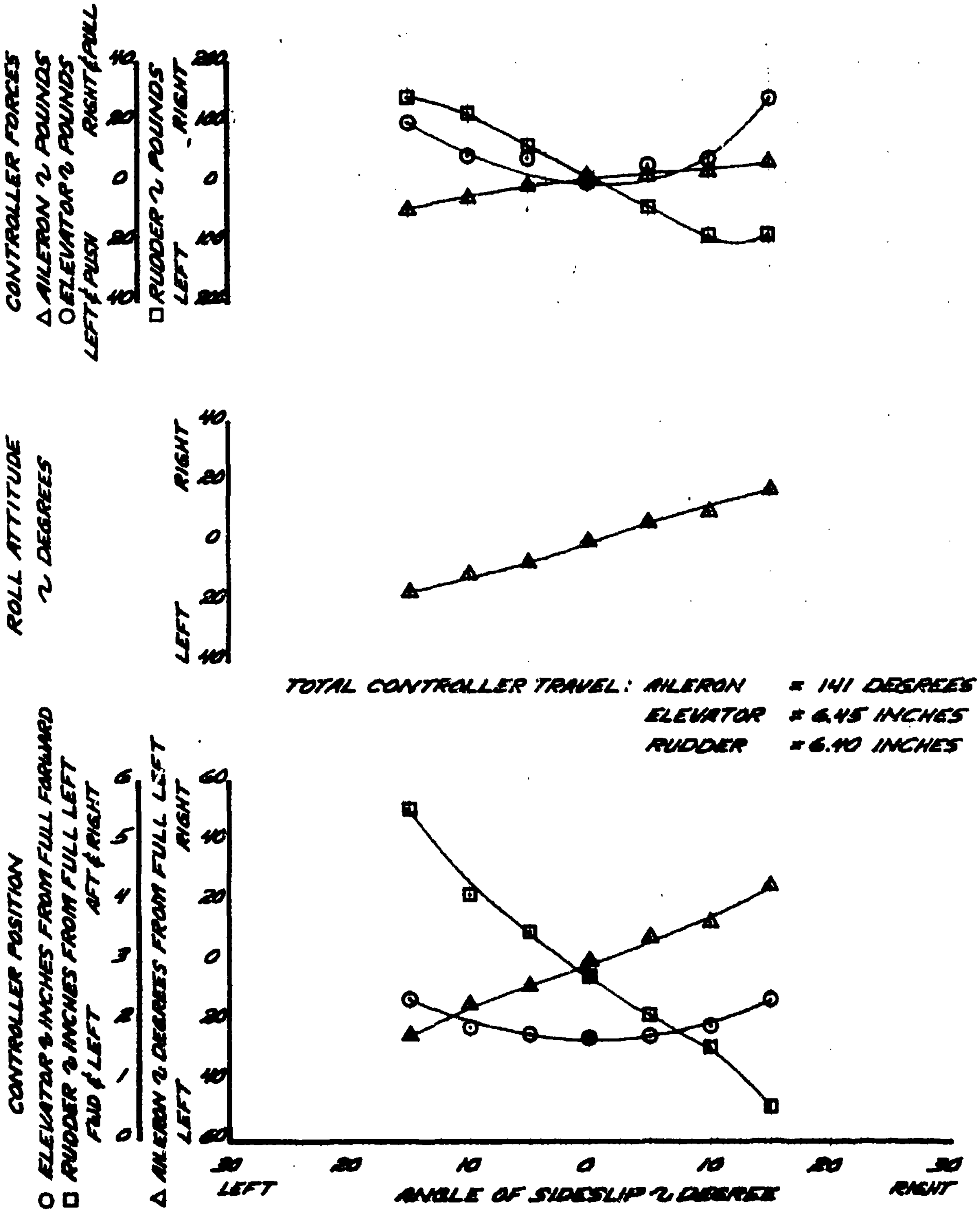


FIGURE 23

STATIC LATERAL - DIRECTIONAL STABILITY

BAC MODEL 200 USA S/N 71-[illegible]

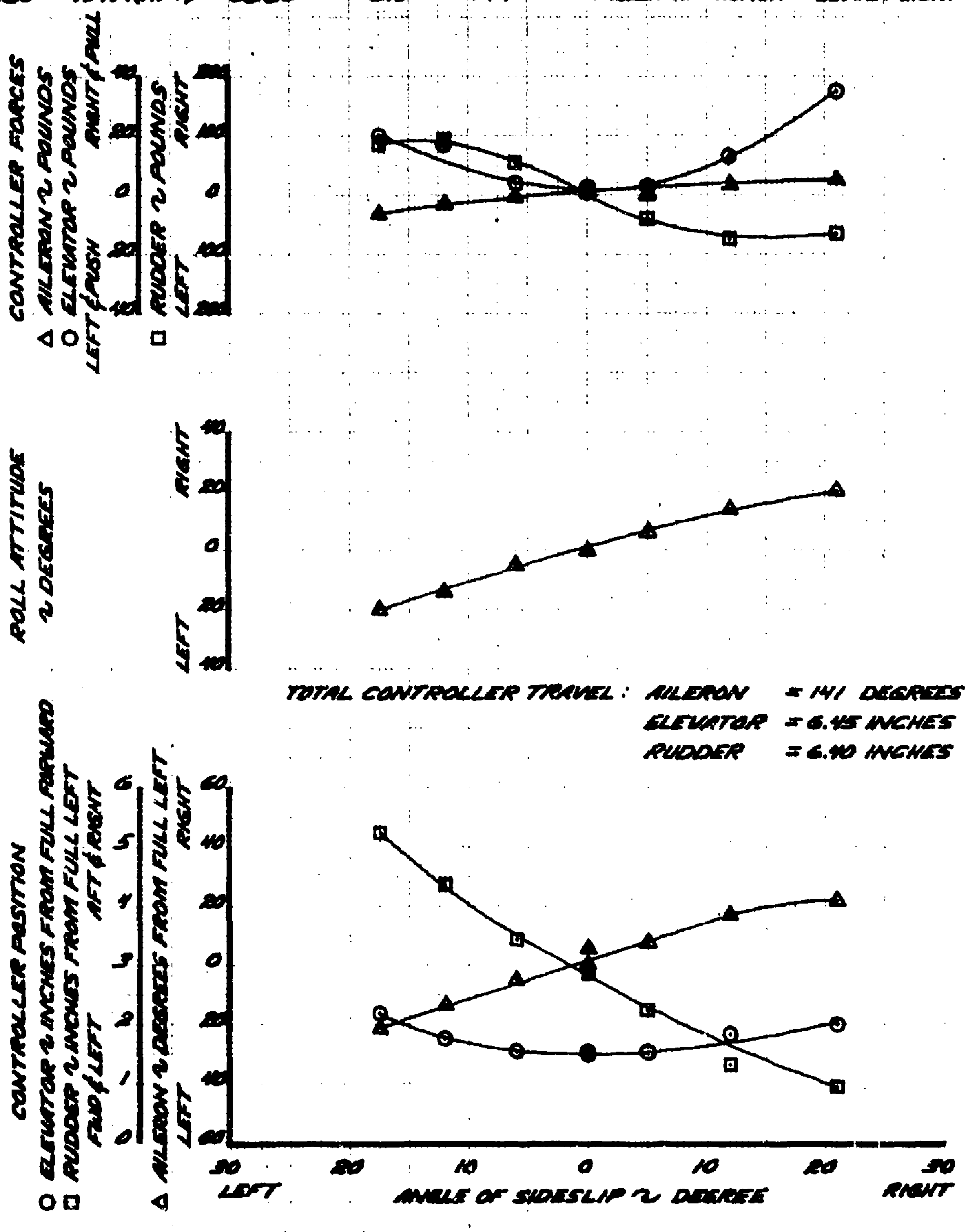
AVG GROSS WEIGHT ~LB	AVG CG LOCATION ~IN	AVG DENSITY ALTITUDE ~FT	AVG OAT ~°C	TRIM AIRSPEED ~KCAS	CONFIGURATION	FLIGHT CONDITION
13800	197.8(AFT)	9040	6.0	145	CRUISE	LEVEL FLIGHT



LD 6-74

FIGURE 30
STATIC LATERAL-DIRECTIONAL STABILITY
BHC MODEL 209 - USA S/N 71-21053

AVG. GROSS WEIGHT ~ LB	AVG. CG LOCATION ~ IN	AVG. DENSITY ALTITUDE ~ FT	AVG. OAT °C	TRIM AIRSPEED ~ KCAS	CONFIGURATION	FLIGHT CONDITION
13560	197.4(AFT)	8860	6.5	144	POWER APPROACH	LEVEL FLIGHT



LD 6-74

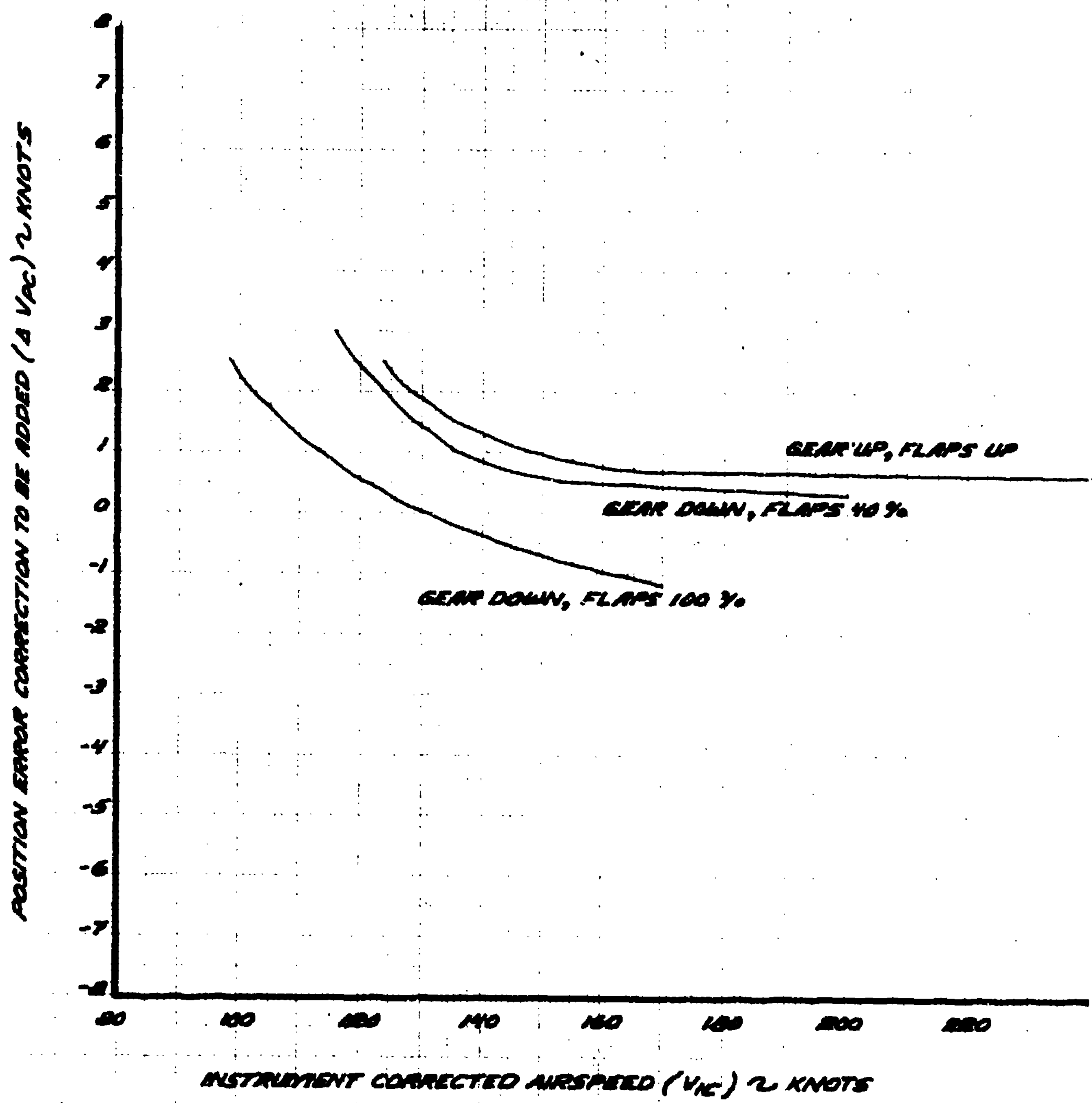
FIGURE 31

PITOT STATIC SYSTEM CALIBRATION

BAC MODEL 380 S/N 71-21059

PILOT'S STANDARD SYSTEM

NOTE: DATA FURNISHED BY BAC



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